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Technology Requirements for Future Earth-to-Geosynchronous Orbit Transportation Systems

Volume II - Technical Results

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Technology Requirements for Future Earth-to-Geosynchronous Orbit Transportation Systems

Volume II - Technical Results

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FINAL REPORT OUTLINE

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LIST OF SYMBOLS AND ACRONYMS

ACEE	Aircraft energy efficiency
ALRC	Aerojet Liquid Rocket Company
APU	Auxiliary power unit
APS	Auxiliary power system
ASE	Advanced space engine
BLOW	Booster lift-off weight
CCV	Control configured vehicle
CV/FBW	Control configured vehicle/fly-by-wire
Cg, c.g.	Center of gravity
CH_4	Hydrocarbon fuel
cm	Centimeter
D&D	Design and development
DDT&E	Design, development, test and evaluation
DF/DE	Dual fuel/dual expander
DOE	Department of Energy
ECS	Environmental control system
EIS	Estimating information system
ELI	Extra low interstitial
EPS	Electrical power system
ET	External tank
EVA	Extra vehicular activity
FAB	Fabrication
FBW	Fly-by-wire
FPR	Flight performance reserve
FLT	Flight
FSTSA	Future Space Transportation Systems Analysis

G	Acceleration of gravity
GaAs	Galium arsenide solar cell
GCH_4	Gaseous methane
GEO	Geosynchronous earth orbit
GLOW	Gross lift off weight
GO_2	Gaseous oxygen
GSE	Ground support equipment
HL	Horizontal landing
HLLV	Heavy lift launch vehicle
IOC	Initial operating capability
I_{SP}	Specific impulse
I_{SP}S.L.	Specific impulse at sea level
I_{SP}VAC	Specific impulse at vacuum
IUS	Inertial upper stage
JPL	Jet Propulsion Laboratories
kg	Kilogram
km	Kilometer
KSC	Kennedy Space Center
LCC	Life cycle costs
LCOTV	Large cargo orbit transfer vehicle
LCH_4	Liquid methane
L/D	Length over diameter ratio
LE	Leading edge
LEO	Low earth orbit
LO	Lift-off
LO_2	Liquid oxygen
LH_2	Liquid hydrogen
LO_2/LH_2	Liquid oxygen and liquid hydrogen
LO_2/LCH_4	Liquid oxygen and liquid methane
LRC	Langley Research Center
LRSI	Low temperature reusable surface insulation

M	Mach number
m	meter
Max	Maximum
MER	Manhour estimating relationships
MICM	Mature Industry Costing Methodology
Min	Minimum
MLI	Multi layer insulation
MR	Mass ratio
M Ton, MT	Metric Ton, 1000 kg
MPS	Main propulsion system
MSFC	Marshall Space Flight Center
NASA	National Aeronautics and Space Administration
nm	nautical mile
NPSH	Net position suction head
O/F	Oxidizer to fuel ratio
OLOW	Orbiter lift off weight
OMS	Orbital maneuver system
OTV	Orbit transfer vehicle
P_c	Chamber pressure
P/L	Payload
PCM	Parametric cost model
POP	Perpendicular to orbit plane
POST	Program to Optimize Simulated Trajectories
POTV	Priority orbit transfer vehicle
psi	Pounds per square inch
RCS	Reaction control system
R&D	Research and development
RP-1	Hydrocarbon fuel type RF-1
RSI	Reuseable surface insulation
SEI	System engineering and integration
SHAG	Small hole accelerator grid

SL	Sea level
SLH_2	Slush hydrogen
SPS	Solar power satellite
SSME	Space shuttle main engine
SSTO	Single-stage-to-orbit
STE	Special test equipment
STS	Space transportation system
S	Wing area
TBO	Time between overhaul
t/c	Thickness ratio, wing thickness (t)/wing chord (c)
TFU	Theoretical first unit
THI	Tank head idle
TPI	Terminal phase initiation
TPS	Thermal protection system
TVC	Thrust vector control
T/W	Thrust overweight ratio
V.A.C.	Volts alternating current
Vac	Vacuum
VTO	Vertical take-off
WBO	Burnout weight
WBS	Work breakdown structure
WER	Weight extimating relationship
WP	Ascent propellant weight
W/S	Wing loading, weight (w)/wing area (s)
λ'	Mass fraction $= \dfrac{\text{Propellant Weight}}{\text{Gross Lift Off Weight less Payload}}$
ε	Nozzle expansion ratio, emissivity or strain
ΔV	Velocity change

1.0 INTRODUCTION

Volume II of the final report is the technical summary of the work performed under Contract No. NAS1-15301 for NASA Langley Research Center (LRC). This final report documents the technology requirements for future earth-to-geosynchronous-orbit transportation systems as required in the Exhibit A, NASA Statement of Work 1-16-6450.0039A dated February 8, 1978.

Use of trade names or names of manufacturers in this report does not constitute an official endorsement of such products or manufacturers, either expressed or implied, by the National Aeronautics and Space Administration.

1.1 DISCUSSION OF MISSION MODEL, REQUIREMENTS GROUND RULES AND DESIGN CRITERIA

The impact of technology advancement on future space transportation elements has been the subject of a series of studies as addressed in references (1), (2) and (20) at the Langley Research Center and by contractors under its sponsorship. The focus of these studies has been single-stage-to-orbit (SSTO) vehicles in the years 1990-2000 time frame. These studies consistently produced evidence of the cost effectiveness of selected technology advances, assuming a reasonable traffic model existed. The purpose of the study, which is summarized in this report, was to bring a unique perspective to these evaluations. Technology advancement effectiveness was to be measured in a total transportation system context. The system was to include both priority and cargo vehicles for missions from Earth to geosynchronous orbit. Priority cargo vehicles are the orbit transfer vehicle (OTV) and SSTO manned winged vehicles for the delivery of total entity types of payloads such as crew transfers, resupply of geosynchronous space depots, automated free-flying satellites, and refurbishment of geosynchronous automated satellites as opposed to large cargo OTV (LCOTV) of unmanned cargo and the heavy lift launch vehicle (HLLV) of LCOTV payloads, POTV delivery and refurbishment, POTV propellant, and heavy lift to LEO. A space base or depot at 500 km altitude was an integral part of the system although not defined in this study.

To balance the broadness of the vehicle family under study, a set of constraining input data and ground rules was specified. These constraints were necessary to maintain a focus on the objective of the study. Although the vehicles were to be innovative and represent a fresh look, there was no requirement to optimize them. Their general concepts were specified as follows: 1) the priority Earth-to-LEO (low earth orbit) vehicle was to be a vertical takeoff single-stage-to-orbit vehicle operating in a dual fuel mode (a hydrocarbon and a high specific impulse fuel are both burned initially, and the high specific impulse fuel is used alone during the later period of the ascent trajectory); 2) the priority orbit transfer vehicle (POTV) was to be space based with all its implications as noted in reference (26); 3) the cargo launch vehicle, or heavy lift launch vehicle (HLLV) was to be a vertical takeoff, two-stage parallel burn configuration with propellant transfer from booster to orbiter. The booster was to stage at a "heat sink" velocity and return to base using an air-breathing flyback system; and 4) the cargo OTV was to use a solar electric propulsion system which will be space based. This vehicle in fact must be assembled in space.

A mission model was specified. This model represented a space industrialization scenario leading to the deployment of a solar power satellite (SPS). It spanned 15 years from 1990 to 2005. The mission model included two basic categories of missions: priority and cargo. The former included the missions of the SSTO and POTV, which consisted of the priority launch rate (flights per year) requirements shown in Figure 1-1 and the priority payload delivery (metric tons of payload per year) requirement shown in Figure 1-2. The cost optimum payload capability of the SSTO-POTV combination was to be determined during the study, considering both the launch and payload requirements. Large cargoes will be delivered to LEO by the HLLV and from LEO to GEO by the LCOTV. The payload size of 227 metric ton (m-ton) was a study ground rule. The HLLV payload delivery requirement is shown in Figure 1-3. In addition, the HLLV had to deliver POTV's and LCOTV's, spares and fuel. Design criteria established in previous studies, reference 4, were ground ruled. These criteria included thrust over weight (T/W), fuel splits for the dual fuel vehicles, landing speeds, reentry trim corridors, etc. Key study ground rules follow:

- initial operating capability (IOC) 1990
- all elements reusable
- space base depot at 500 km
- KSC launch site: LEO base at 28.5^{o}, GEO destination - equatorial
- heavy lift payload = 227 metric tons
- priority cargo payload size(s) to be cost optimized
- payload density = 100 kg/m^3
- return payloads: SSTO - 100%; HLLV - 10%; POTV - 75%; LCOTV - none
- all winged vehicles vertical take-off - 165 knots landing speed
- re-entry trim corriders: SSTO 30^{o} - 60^{o}; HHLV 35^{o} - 60^{o}
- SSTO - 2000 km cross range
- SSTO OMS sized for 93 x 186 km insertion
- T/W at liftoff = 1.3, maximum acceleration = 3g
- CH_4 = hydrocarbon fuel

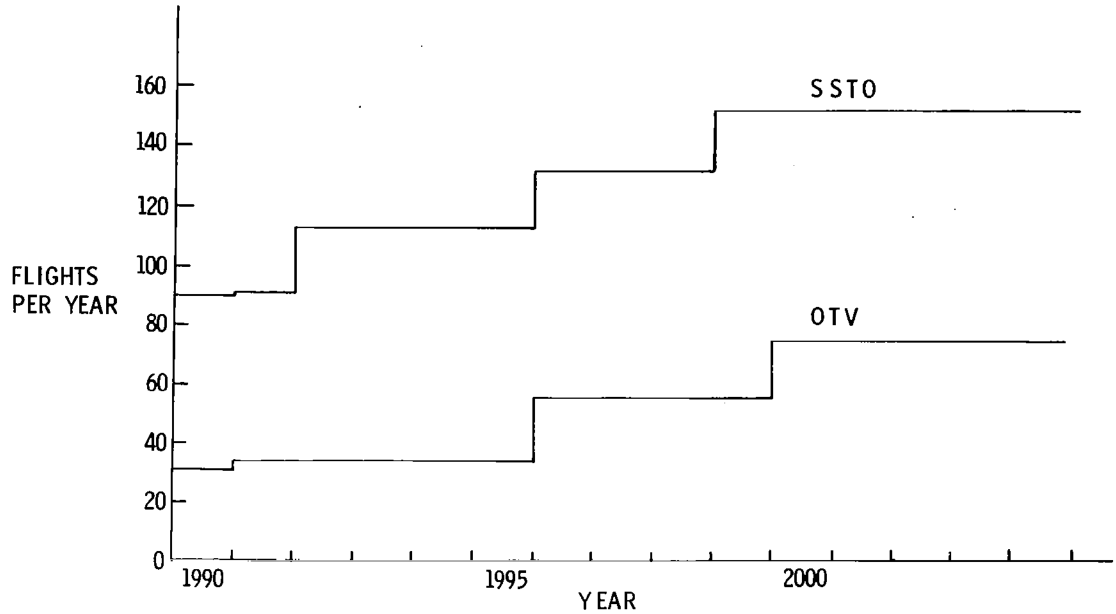


Figure 1-1.—Priority Flight Requirements (Exclusive of Payload Delivery Requirements)

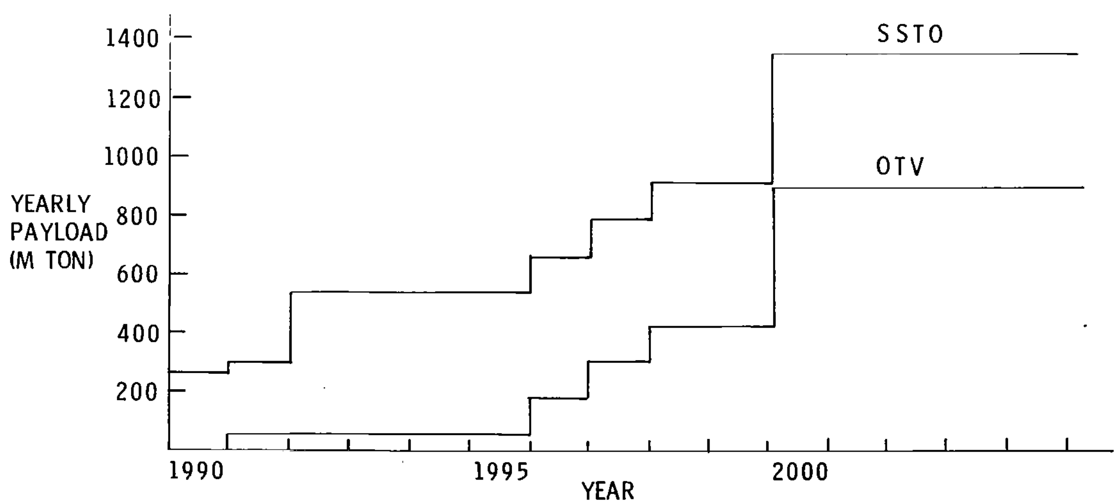


Figure 1-2.—Priority Payload Delivery Requirement

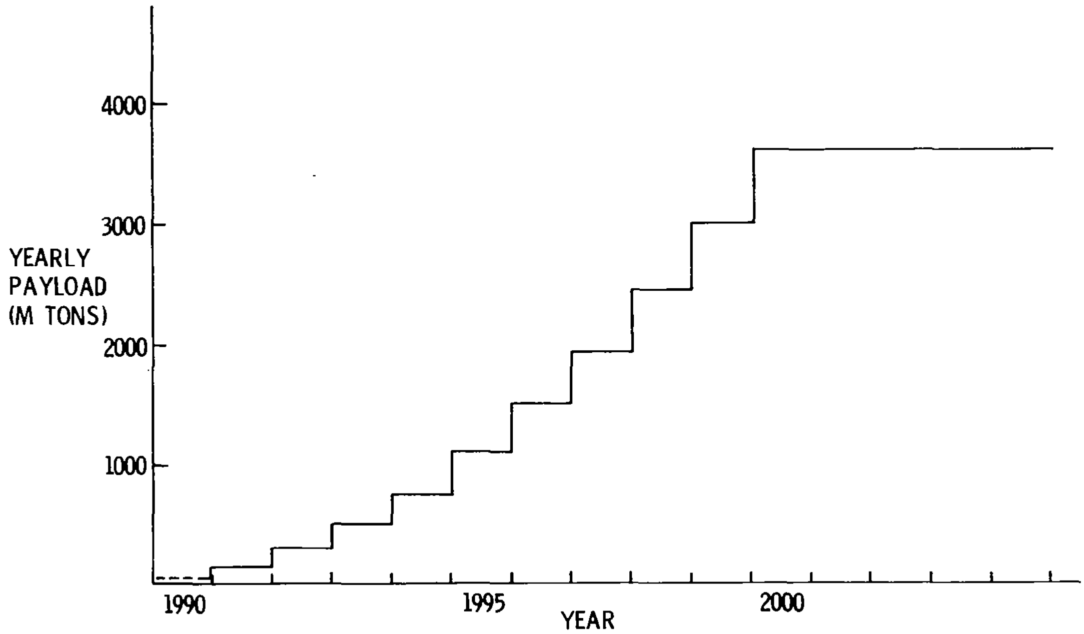


Figure 1-3.—HLLV Payload Delivery Requirement

1.2 APPROACH

The study was divided into two logical phases as shown in Figure 1-4. The first phase was to produce an evaluation yardstick. This yardstick would be the life cycle cost (LCC) of the transportation system. The system would be based on normal growth technology or technology which through current or anticipated research and development (R&D) funding would have reached a degree of maturity necessary for a reasonable risk commitment to design, development, test and evaluation (DDT&E). The technology readiness date for a 1990 initial operating capability (IOC) is approximately 1985.

During the second phase of the study an assessment was made to determine the impacts of accelerated technology on the transportation system. Accelerated technology levels were established and applied to the vehicles in the transportation system to define those technologies which were either critical to a vehicle concept or offered significant benefit (or yield) when applied to the transportation system. The vehicle concepts were updated to incorporate the high yield or critical technologies and the life cycle cost improvements resulting from the incorporation of the technology advancements were determined.

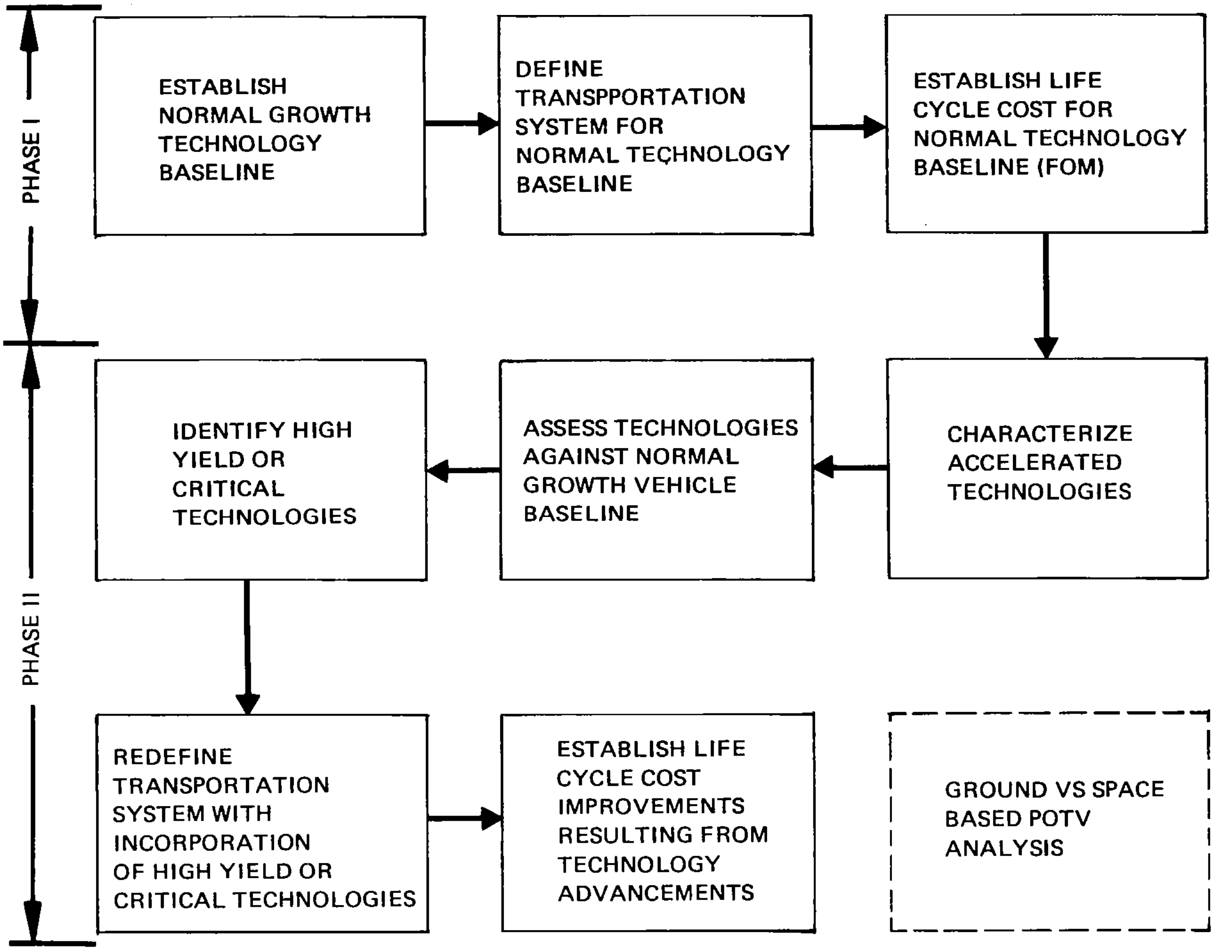


Figure 1-4.—Study Approach

An analysis of ground versus space basing of the POTV was accomplished under an extension to the basic study contract. This analysis included the development of a design concept for a ground based POTV requiring a single HLLV launch, establishing operations requirements for ground and space based OTV concepts, and the comparison of these operations requirements to identify enhancing or enabling technologies.

The remainder of this report is organized to respond to the two phases of the basic study and the ground versus space based POTV analysis task. Section 2.0 discusses the normal technology forecasts, normal technology vehicle concepts, and their life cycle costs. Section 3.0 addresses the accelerated technology portion of the study including accelerated technology forecasting, benefits, and accelerated technology vehicle design concepts. Section 4.0 summarizes the findings and conclusions of the basic study. Section 5.0 contains the analysis of ground versus space based POTV vehicle options.

2.0 NORMAL GROWTH TECHNOLOGY BASELINE

2.1 TECHNOLOGY FORECAST & CHARACTERISTICS

The normal growth technology forecast provides the fundamental basis for the design of the reference vehicle system. With an IOC of 1990 for the system an approximate readiness date of 1985 was assumed for the various technologies. Normal growth in the context of this forecast means that funds are either being expended or are planned to bring the technical risk down to a reasonable level for initiation of design, development, test and evaluation (DDT&E) by this readiness date. All sources of research and development (R&D) are considered including the NASA, the military services, Department of Energy (DOE) and other branches of Government, the academic community and industry.

Input data to this task included the previous SSTO technology requirement studies, reference (1) and (2) and the JPL technology forecast reference (3). A current Boeing proprietary forecast reference (4) was also used as a data base. Individuals and organizations throughout the various Boeing Companies were consulted. Various NASA and other sources were in turn consulted by the various Boeing specialists or by the study participants.

The approach to the technology forecasting task is summarized as follows: The subsections, which follow, discuss the forecasts for each of the technology areas:

- o Previous SSTO studies and JPL forecast form data base

Need:

- o Updating
- o Normalization
- o Expansion - to include new vehicles/requirements

OTV's	Chemical
	Electrical
HLLV	Orbiter
	Flyback Booster
Space Basing	

Methodology:

- o Understand Existing Data Base
- o In house consultation
- o Outside contacts in selected areas

2.1.1 Structures and Materials

Structures and materials considerations were focused on improved reusable surface insulation (RSI) and metallic multi-wall thermal protection systems (TPS) systems, advanced composite structures, and titanium honeycomb main hydrogen tanks.

Improved RSI - NASA/Ames is developing a new generation of RSI materials offering much better mechanical properties than the current (Shuttle Orbiter) generation, with some improvement in temperature capability. Progress is rapid now but is expected to taper off in the early 1980's. Anticipated characteristics of RSI, normal growth technology, are presented as follows:

- o Improved RSI
  - o 1649°C reuse limit @ 320 Kg/m^3 (20 lb/ft^3)
    (Current Li 2200, redline @ 1482°C & weighs 352.4 Kg/m^3 (22 lb/ft^3)
    (Leading edges, etc)
  - o 1260°C reuse limit @ 120 Kg/m^3 (7.5 lb/ft^3)
    (Current Li 900, 1260°C @ 144 Kg/m^3 (9 lb/ft^3)
  - o 1093°C reuse limit @ 120 Kg/m^3 (7.5 lb/ft^3)
    (Current coating limit @ 649°C)
    (Flexible Blanket)
- o Tile RSI - $200-400/Tile (Current - $400-700/tile + $50 installation)
- o Flexible RSI (Current - $50/ft^2)

The increase in redline temperature from 1482°C to 1649°C will allow for weight saving RSI application in those regions (wing leading edges, body nose) currently utilizing RCC.

Metallic TPS - NASA/Langley efforts in this area were reviewed. The design as currently envisioned is not compatible with the high temperature heating environment of the lower surface. With respect to upper surface application, it does not offer a weight reduction relative to RSI. For these reasons, metallic multi-wall TPS was not used. (It was later considered as an accelerated technology item but only with respect to its possible impact on cost and maintenance. See section 3.1.8).

Composites - A review of advanced composite structures was undertaken. The review focused on the Boeing, Lockheed, and McDonnell Douglas efforts on the Aircraft Energy Efficiency (ACEE) Composite Structures Program sponsored by NASA/Langley, Boeing 727 wing box section demonstration hardware, a Boeing study, reference (5), for NASA/Langley for a composite wing study program, a Boeing study, reference (4), on composite structures application to an advanced bomber, and the extent of composite structures application on the F-18 aircraft, and numerous technical papers. As a result of this review, a normal growth technology weight factor (composites versus aluminum) of 70% was established, as was a utilization factor of 100%. The 70% weight factor represents a "composite substitution" approach rather than the more efficient "composite design" approach which is considered as accelerated technology.

Titanium Honeycomb Hydrogen Tanks - A major consideration in the selection of titanium honeycomb for main hydrogen tankage on launch vehicles was the absence of any ongoing development programs for reusable internal foam insulation. Once the decision to use titanium had been made, the subject of hydrogen embrittlement was addressed. The extensive research and test efforts of the Aerojet Liquid Rocket Company, performed in support of their SSME effort, reference (6), were reviewed. Key factors in the avoidance of hydrogen embrittlement are low hydrogen gas pressure, the limiting of hydrogen gas temperature to a value no greater than room temperature, and the use of titanium alloys with extra low interstitial (ELI) characteristics. The launch vehicles use 6AL-4V (ELI) titanium honeycomb tanks in which the pressure and temperature of the hydrogen bleed gas pressurant are limited to 159 kPa (23 psia) and 189 K (-120°F), respectively. A more conservative approach of using helium gas pressurant was rejected because of weight considerations. While the use of titanium honeycomb hydrogen tanks appears feasible, further work is necessary to establish the quantitative effects of stress state, gas pressure and purity, and alloy microstructure upon hydrogen embrittlement susceptibility.

2.1.2 Propulsion

The propulsion systems features of the normal growth technology forecast are listed as follows:

- SSME - 2 position nozzle
  ε = 50/1 50 I_{sp} = 450 sec/464 sec
  ΔCost = \$50M DDT&E ΔWeight = 173.7 kg (383 lb)

- Hydrocarbon booster engine

- "ASE" type engine
  I_{sp} = 473 sec @ ε= 400 thrust = 88,964N (20 klb)
  Weight = 206.4 kg (455 lb)
  Cost = \$250M DDT&E

- LOX/LH_2 RCS
  I_{sp} = 427 sec
  Cost = \$46M DDT&E

- 50 CM ION thruster

- Zero NPSH boost pumps for ASE, 18 kg (40 lb)/engine

The following paragraphs discuss these features in more detail.

<u>SSME</u>

A slightly modified space shuttle main engine (SSME) was selected as the upper stage engine for the baseline transportation system. The only change was the replacement of the standard ε = 71 nozzle with a two position nozzle ε = 50/150. This increases the specific impulse from 363 to 390 secs at sea level and from 455 to 464 in vacuum for a mass increase of only 173 kg (383 lb). Previous SSTO technology requirement studies, references (1) and (2), have considered this engine a normal growth feature. It is a relatively simple modification of the standard SSME, with an estimated DDT&E bill of \$50 million which was charged to the SSTO vehicle.

For the normal growth baseline transportation system the 50/150 expansion ratio SSME is used on the HLLV Orbiter and the SSTO. The characteristics of this extendable nozzle SSME are tabulated as follows:

- o LO_2/LH_2 stage combustion cycle MR = 6:1
  - o P_c = 20 490 kPa (2970 psia)
  - o ϵ = 50/150 Exit diameter (ϵ = 150) = 4.47m (176 in)
  - o Isp S.L. = 390.4 sec
  - o Isp VAC. = 450 sec @ ϵ = 50
    464 sec @ ϵ = 150
  - o Thrust = 1.79×10^6N (403 100 lbf) S.L.
    214×10^6N (480 100 lbf) VAC. @ ϵ = 150
  - o Dry mass = 3222 kg (7105 lbm)
  - o Length = 4.06m (160 in) stowed; 5.59m (220 in) extended
  - o DDT&E = $50M

Hydrocarbon Booster Engine - The development of a new high performance booster engine is generally accepted as a key requirement for any new launch system. Studies have been undertaken and are continuing to define and select the engine cycle. In addition, test hardware development has been initiated recently by MSFC.

The engine selected as the normal growth baseline is an expander bleed cycle, LO_2/LCH_4 engine concept developed by the Aerojet Liquid Rocket Company (ALRC) from work performed under contract reference (7) to LRC. Its characteristics of the normal growth technology hydrocarbon engine are summarized in following tabulation. Appendix A, volume III describes this engine in greater detail.

- o LO_2/LCH_4 expander bleed cycle MR = 3.5:1
  - o P_C = 29 325 kPa (4250 psia)
  - o ϵ = 50 o Exit diameter = 2.21 m (87 in.)
  - o I_{SP} S.L. = 328 sec
  - o I_{SP} VAC. = 361 sec
  - o Thrust = 3.86×10^6N (867 700 lb) S.L.
    4.25×16^6N (955 000 lb) VAC
  - o Dry mass = 3031 kg (6680 lb) o T/W = 143
  - o Length = 4.32 m (170 in.) power head diam. = 2.49 m (98 in.)

- TBO = 250 starts (50% refurbishment)
- Hydrogen cooling & turbo pump drive (5% of LCH_4 mass flow)
- DDT&E = $570M

The LO_2/LCH_4 engine provides main propulsion for the HLLV booster and mode I propulsion on the SSTO (which has a dual mode propulsion system).

ASE TYPE ENGINE

An advanced space engine is presently being studied by several engine manufacturers under contract to LRC. Full scale components have been tested to verify performance. The proposed engine uses a LO_2/LH_2 stage combustion cycle at a mixture ratio (MR) of 6:1. At an expansion ratio of 400 the engine has a specific impulse of 473 sec. It has a thrust level of 88 960 N (20 000 lbf) and a dry weight of 206 kg (455 lbm). This engine has an overhaul interval of 60 starts and a total life of 240 starts.

This engine provides main propulsion on the POTV and OMS propulsion on the SSTO and HLLV orbiter.

LO_2/LH_2 RCS

LO_2/LH_2 RCS was selected as a normal growth technology for all the vehicles in the system. This selection was based on extensive tests of a flight weight 6670 N (1500 lbf) LO_2/LH_2 thruster developed by ALRC under contract reference (8) to LRC.

This thruster operates at a chamber pressure of 2 070 kPa (300 psia) and a specific impulse of 427 sec. Both performance and long life (65 000 firings) have been demonstrated. The LO_2/LH_2 RCS thruster is described further in Appendix A.

50 cm Ion Thruster - A 50 cm argon ion thruster was selected as the normal growth baseline for the solar electric propulsion LCOTV. This thruster was considered a normal growth technology because it is the largest size thruster that can be extrapolated from existing single cathode mercury thrusters. Characteristics of the normal growth technology ion thruster are summarized as follows:

- 50 cm/shag optics - largest single cathode design extrapolation of 30 cm thruster technology
  (shag - single hole accelerator grid)

- Argon propellant

Thrust	=	.7 newtons (.16 lb)
Isp	=	10 000 sec.
Input power	=	46 kw @ 2513 beam voltage
Efficiency	=	82%
Life	=	6000 hrs @ beam current = 16 amps
Weight	=	34 kg
Refurbishment requirements	=	50% of initial cost (10 rebuilds max)
Cost	=	$25M DDT&E

2.1.3 Power Conversion and Distribution

The features of the normal growth technology forecast in the power conversion and distribution technology are are as follows:

- APU driven generators/pumps
  - Eliminate launch vehicle fuel cells

- 8 000 psi hydraulics
  - 30% weight reduction

- High voltage/solid state electrical power system
  - 40% weight reduction

- Silicon solar arrays

Each of these technologies are discussed below.

APU Driven Generators/Pumps - This technology feature was based on the work described in reference (1). The main advantage of this arrangement is the multi-purpose usage of the auxiliary power unit (APU). Weight savings are achieved by eliminating fuel cells and their associated subsystems for the prime electrical power.

8 000 psi Hydraulics - Application of this very high pressure technology will reduce system weight by 30% when compared to a 20 700 kPa (3 000 psi) baseline. Previous studies had forecast growth to 34 500 kPa (5 000 psi) however confirming R&D by the Navy reference (9) on a 55 200 kPa (8000 psi) system allows the selection of this technology.

High Voltage/Solid State Electrical Power System - This is another area being actively pursued by the Naval Air System Command reference (10). The objectives of this work are to provide high quality power, efficiently controlled and distributed, using solid state, digital and programmable components design for flexibility and growth. Overall effectiveness will result in a 40% reduction in weight and volume, an 8:1 increase in reliability, a 5:1 increase in maintainability and 1000 to 1 decrease in control power consumption.

Solar Arrays - The silicon array forecast for normal growth is based on the work performed by Boeing reference (1) in its solar power satellite definition studies. The array consists of a parallel-series matrix of 50 micron radiation resistant cells, electrostatically bonded between two large sheets of 50 micron "microsheet" glass. This array is compatible with the temperatures associated with cell annealing to remove radiation damage. Specific weight of the array is 427 grams per square meter and specific power (beginning of life) is 197.7 watts per square meter. The power level used for sizing the array was 138 watts per square meter which is an array efficiency of 10% and includes the effects of radiation degradation.

2.1.4 Subsystems

A summary of the subsystems normal growth is listed as follows and discussed below.

- Avionics
  - Lower weight, less power, same cost
  - IUS reference for OTV's, shuttle reference for launch vehicles

- Landing gear
  - New shock/strut design using composites, improved tires - 2.8% landing weight

- Crew accommodations/ECS
  - 2nd generation shuttle - lower weight

- Flight control system
  - CCV capability - shuttle based weights

Avionics - Avionic forecasts were baselined from two reference vehicles. For launch vehicles a shuttle reference was used. For the OTV's an IUS reference was used. The shuttle reference was considerably reduced by eliminating the additional crew and mission requirements beyond the transportation function. Both references were reviewed in detail and adjusted for normal growth. Adjustment were weighed based on the type of equipment (communications were not treated the same as computers). The average reductions in both power and weight were between 15 and 25%. Costs were not changed for normal growth as compared to current costs.

Landing Gear - The normal growth landing gear was based on the Boeing SSTO studies reference (1). It is a new shock/strut design using composites (boron-aluminum) and improved tires. These features result in a landing gear weighing 2.8% of vehicle landing weight. This weight fraction was verified by detailed analysis. Work is in progress relative to the composite strut design by the Naval Air System Command.

Crew Accommodations/ECS - Weights for this area were based on shuttle with appropriate adjustments for the difference in requirements. Since a study groundrule required a 10% weight margin, normal growth was pegged at a 10% weight reduction in these areas. The result is weights identical to the adjusted Shuttle reference.

2.2 BASELINE TRANSPORTATION SYSTEM

2.2.1 Priority Cargo Launch Vehicle (SSTO)

2.2.1.1 Design Requirements and Configuration Issues

The primary SSTO design requirements, ground rules and guidelines are summarized as follows:

Requirements:

- o 100 KG/M^3 payload densiity - 100% return - size to be cost optimized
- o OMS sized for 500 KM from 93 x 186 KM insertion
- o vertical takeoff @ T/W = 1.3 - horizontal landing @ 165 knots
- o single stage - dual fuel (methane engine propellant fraction = .6)
- o 2000KM cross-range (reentry trim 30^{o} - 60^{o})
- o minimum orbit stay time

Issues:

- o payload size
- o payload bay configuration & access
- o aero configuration/cg
- o structural simplicity & high volumetric efficiency
- o engine size selection

These top level requirements are straightforward for this type of vehicle and its mission. Previous studies (references 1 and 2) addressed similar requirements except for the circularization requirement at 500 Km altitude. This requirement has a significant sizing impact on this performance sensitive vehicle. The payload size, optimized for the mission model which required a considerable number of off-loaded or priority flights was found to be 13 605 kg (30 000 lbs).

Since this vehicle was groundruled to a vertical takeoff, the primary configuration criteria was an optimum ascent configuration. Structural simplicity and high volumetric efficiency were sought after. The wing was kept to an absolute minimum to meet the landing requirement and therefore reentry planform loadings were such that the thermal environment required a reusable surface insulation (RSI). High volumetric efficiency is essential as the weights of the propellent tanks and the weights of the thermal propulsion system are a direct function of this criteria.

High volumetric efficiency for the body requires large diameter single lobe tanks which are not compatible with design concepts employing long payload bay arrangements. A nose located payload bay is not practical because of the large cg shift between the "payload in" and "payload out" reentry conditions. A mid body location was selected as the best compromise for the vehicle. This location allowed a more than adequate bay without signficantly extending the body length. This is a characteristic of an SSTO vehicle whose payload is small compared to its gross weight.

As vehicle sizing trends were established and the vehicle grew, its number of engines also grew. Figure 2.2.1-1 illustrates the impact of this growth in terms of cg shift and subsequent necessary aerodynamic tailoring of the vehicle. Although the trim cg requirement and actual cg did not converge, their relationship was considered close enough to be considered acceptable based on LaRC conducted reentry trim studies.

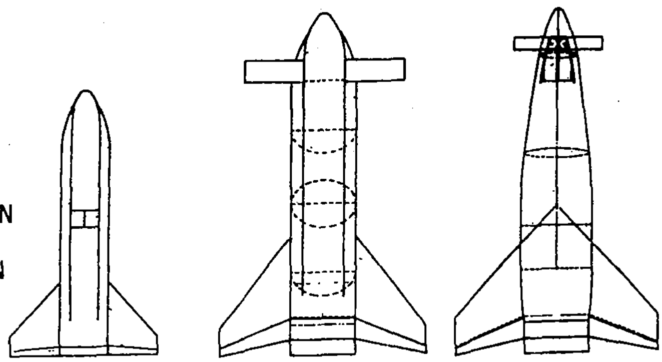
The new booster engine, a high pressure hydrocarbon design, was sized to match an SSME while meeting the hydrocarbon propellent fraction of .6 which previous studies(references 12, 13, and 14) had established as optimum or near optimum for single stage launch vehicles operating in a dual fuel mode. Dual fuel propulsion, where a high density fuel (hydrocarbon) and a high specific-impulse fuel are both burned initially and the high specific-impulse fuel is used alone during the later period of the ascent trajectory, is critical to this type of vehicle. Reference 12 includes an indepth discussion of this area. To further enhance the high specific impulse period, the hydrogen fueled SSME is equipped with a two position nozzle. The nozzle extends to its altitude position right after the booster engine shutdown.

Figure 2.2.1-2 shows the vehicle configuration concept as evaluated and calls out its main features.

2.2.1.2 Performance and Sizing

The SSTO payload size was not groundruled at the beginning of this study. The optimum payload weight was determined from the mission model requirements based on minimum life cycle cost. This evaluation included the POTV, which was sized to deliver the SSTO payload from LEO to GEO. In order to develop a minimum LCC SSTO payload size, parametrics were developed from point designs and a relationship between payload size and LCC was established.

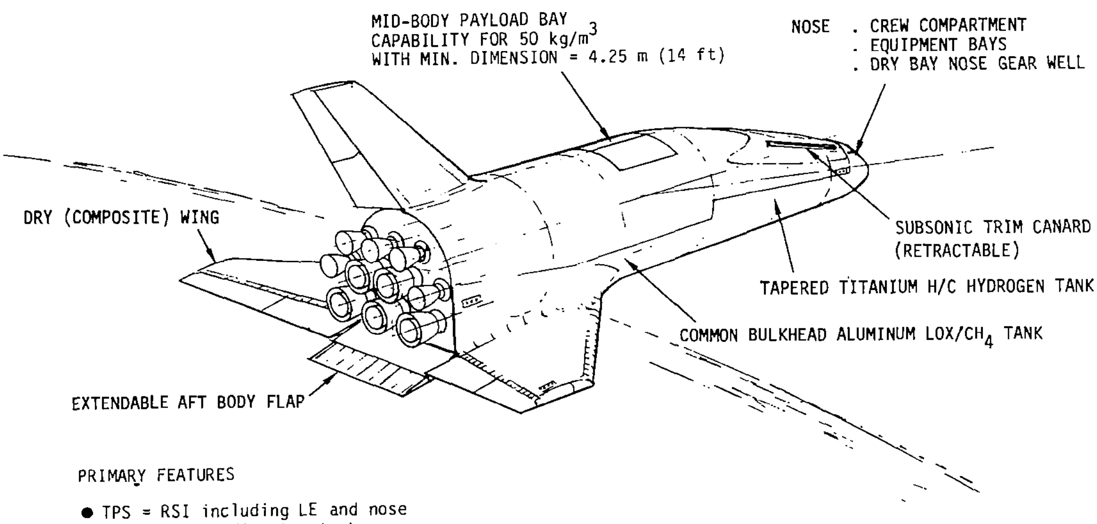
- HYPERSONIC
- BODY FLAP & ELEVON DEFLECTIONS NOT EXCEEDING 5° DOWN



GLOW Kg X 10^{-6} (LB X 10^{-6})	.894 (1.97)	2.195 (4.84)	2.195 (4.84)
LDG WEIGHT Kg (LBS)	111,130 (245,000)	224,075 (494,000)	224,075 (494,000)
W/S kPa (PSF)	3.70 (77.3)	3.73 (78)	3.73 (78)
MOST AFT* ENTRY CG	72.3	74.8	76.4
AERO* CAPABILITY (Trimmable CG)	66	70	75

*Percent Body Length

Figure 2.2.1-1. SSTO Aero Configuration Development



PRIMARY FEATURES

- TPS = RSI including LE and nose
- Simple large diameter tanks
- T/W = 1.3
- Dual fuel using CH_4 ENG's and SSME's
- 500 Flight life criteria
- LOX/LH_2 RCS and OMS
- Wing/canard sized for landing

Figure 2.2.1-2. - SSTO Design Concept

Parametric Sizing - An SSTO configuration based on preliminary data was provided by LRC as a starting point. The initial configuration data is listed below:

GLOW (gross lift off weight)	878 200 kg	(1 936,000 lbs)
Propellant	778 400 kg	(1 716,000 lbs)
Payload	10 000 kg	(22 000 lbs)

With this data a point design configuration was developed using the characteristics and groundrules discussed in the design requirements and configuration issues section. The vehicle was sized to match an engine installation built up around two SSME's. The LO_2/LCH_4 thrust fraction was 2.15 times as large as the SSME fraction, providing the optimum thrust split previously groundruled. For this design point an LO_2/LCH_4 engine twice as large as the baseline LO_2/LCH_4 engine (described in section 2.1.2) was used because it allowed a better engine arrangement.

Performance characteristics were calculated using the program to optimize simulated trajectories (POST) (reference 15) program. The mass ratio required for insertion into a 93x186 km orbit via a 12.57 kPa (850 PSF) max dynamic pressure lifting trajectory was 8.811 (includes flight performance reserve (FPR) in ascent propellant weight P.) Transfer to a 500 km circular orbit was accomplished with the OMS.

Detailed mass properties analysis yielded a dry mass 25 000 kg (55 100 lbs) greater than the initial estimate. This resulted in a negative payload of 10 800 kg (23 900 lbs). A substantially larger vehicle would be required to achieve any positive payload at all.

The GLOW of the SSTO was increased until it matched a 3 SSME engine installation. At this point the LO_2/LCH_4 main engine was reduced in thrust to the level that became the baseline engine, 4.25×10^6N (955 000 lbf). Three engines of this size provided the required thrust balance for the 3 SSME's. This configuration, design point 2 was sized to the following characteristics:

GLOW	1 384 500 kg	(3 052 400 lbm)
Propellant	1 226 800 kg	(2 704 700 lbm)
Payload	10 000 kg	(22 000 lbm)

Design point 2 represented an improvement in performance, but still failed to generate a positive payload. The dry mass was 14 500 kg (23 000 lbm) greater than the initial estimate, causing a negative payload of 2 400 kg (5 300 lbm).

The first two design points provided trending data from which a size estimate could be made for an SSTO capable of delivering the 10 m ton payload. The vehicle size required exceeded the capability of 4 SSME - LO_2/LCH_4 engine pair propulsion system. A configuration matched to 5 engine pairs, with a GLOW of 2 205 800 kg (4 862 900 lbm), was developed. This third design point had a mass ratio of 8.767 and delivered a payload of 11 100 kg (24 500 lbm).

Payload Optimization - With the parametric sizing and performance model developed and with parametric costing relationships derived from design points 1 and 3, an SSTO payload optimization was made. This optimization also included parametric size and cost data for the POTV. The objective was to select an SSTO and POTV combination with the lowest LCC for the same mission requirements.

The results of the optimization are shown in figure 2.2.1-3. The curve of LCC versus payload size is relatively flat (± 1% of LCC) between 11 Mton (25 000 lbm) and 22 Mton (50 000 lbm). SSTO LCC's are the determining factor, the POTV being insensitive to payload size due to its ability to perform well while offloaded. Since design point 3 was close to the optimum and had been configured, it was selected as the normal growth baseline SSTO. The optimum payload range was strongly influenced by the mission model which included a large percentage of launches that were not sensitive to payload mass. Increasing or decreasing the payload delivery requirement of the mission model would directly increase or decrease the optimum payload size.

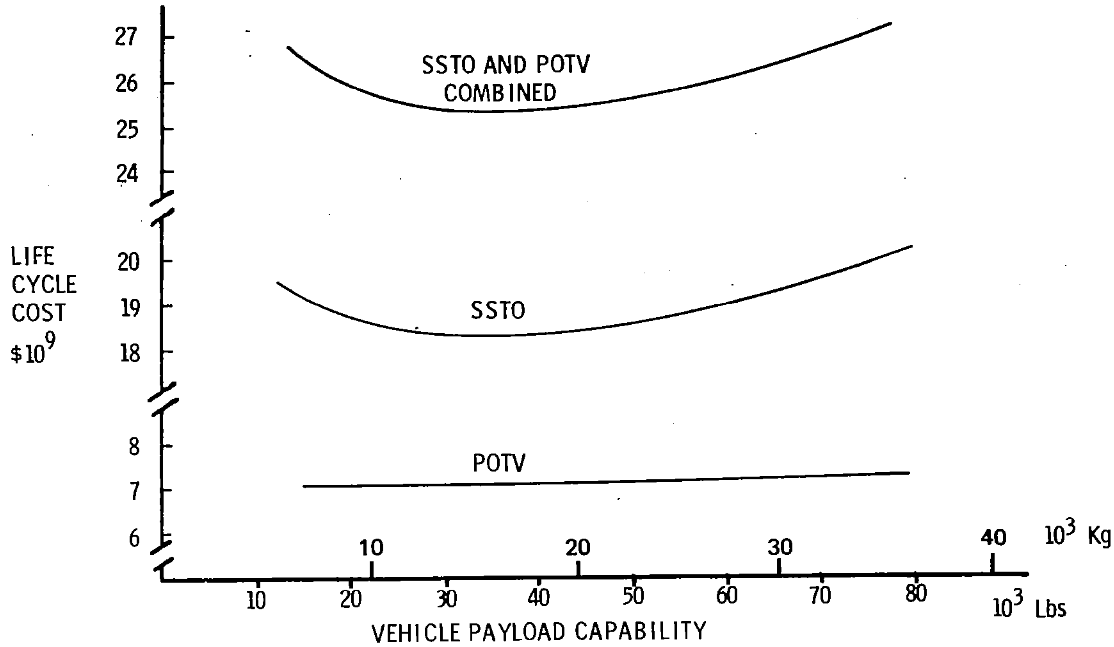


Figure 2.2.1-3. - Priority Cargo, Payload Size Cost Optimization

<u>Reference Trajectory</u> - The trajectory for the baseline SSTO is essentially the same as that of design point 3. the SSTO flies a vertical takeoff, lifting trajectory utilizing dual mode propulsion. The key trajectory characteristics are listed below:

T/W @ liftoff:	1.306	
Max Dynamic Pressure:	40.7 kpa	(850 PSF)
Max Acceleration:	3.00 g	
Mode I Burn Time:	220.5 sec	
Altitude @ Transition:	76,080 m	(249,600 ft)
Rel. Velocity @ Transition	3961 m/sec	(12,995 ft/sec)
Total Burn Time:	357.2 sec	
Insertion Orbit:	93x186 km	(50x100 n mi.)
Trajectory Mass Ratio:	8.6503	(8.7642 including FPR)

At liftoff, the vehicle operates on Mode I propulsion, both sets of engines firing. The SSME's run with their nozzles in the stowed position, ϵ =50, to give higher sea level thrust. When all the LCH_4 has been burned the vehicle transitions from Mode I to Mode II propulsion. The LO_2/LCH_4 engines are shut down and the SSME nozzles are extended to the ϵ=150 position to maximize I_{SP}.

The SSME's continue to operate until insertion, which occurs at an altitude of 92.60 km (303 800 ft) and a relative velocity of 7 463 m/sec (24 484 ft/sec). After coasting to an apogee of 185 km (100 n. mi.), the OMS engines circularize the vehicle with a burn of 28 m/sec (91 ft/sec). The SSTO remains in this orbit until phasing is complete, at which point the OMS provides a transfer out to 500 km (270 n. mi.) of 91 m/sec (297 ft/sec). Circularization at 500 km is accomplished by the OMS with an 89 m/sec (294 ft/sec).

The only significant change between this trajectory and that of design point 3 is the handling of residuals. In the earlier trajectories all main engine and feed line residuals were dumped after insertion into the 93x185 km orbit. For this trajectory the LO_2/LCH_4 engine and associated feedline residuals are dumped at the transition from Mode I to Mode II. This increases the baseline SSTO payload to 13 600 kg (30 000 lbm) from the 11 100 kg (24 500 lbm) of design point 3.

Ascent trajectory parameters for the baseline SSTO are plotted as a function of time in figure 2.2.1-4.

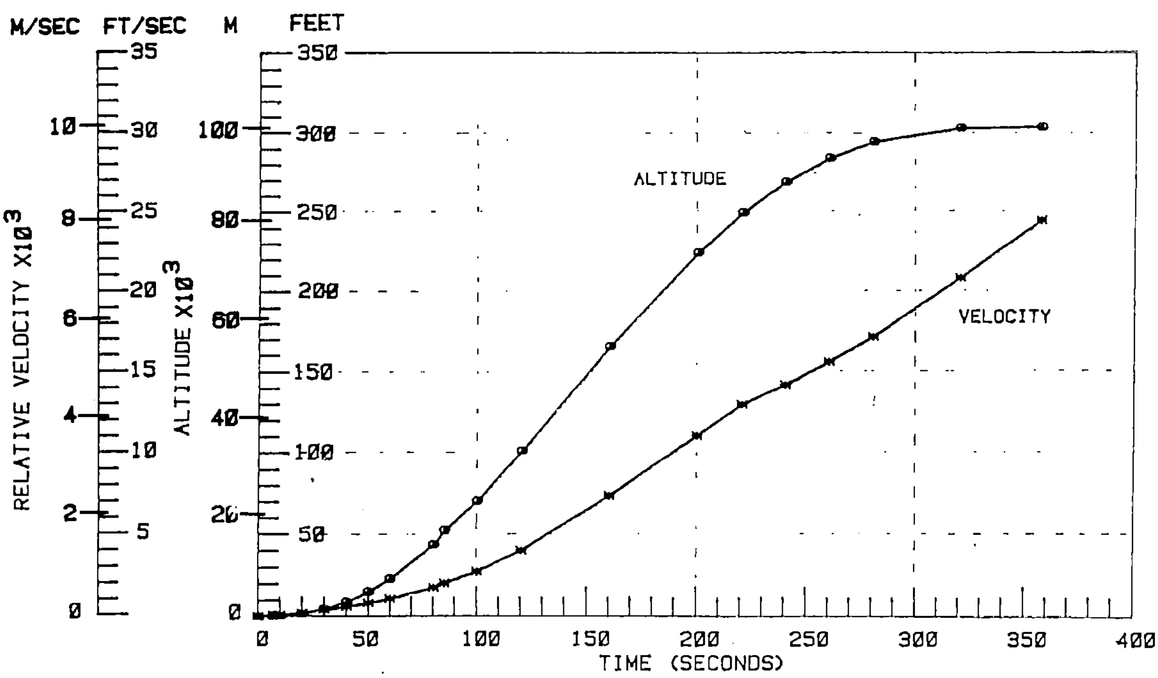


Figure 2.2.1-4a. - Normal Growth Baseline SSTO Ascent Performance

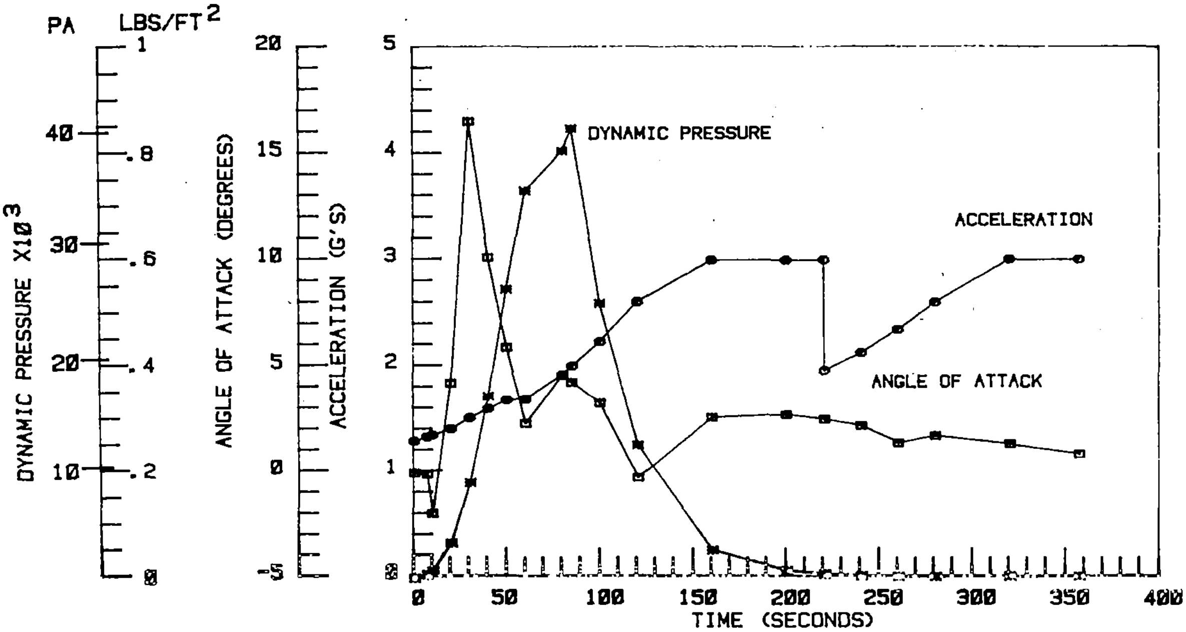
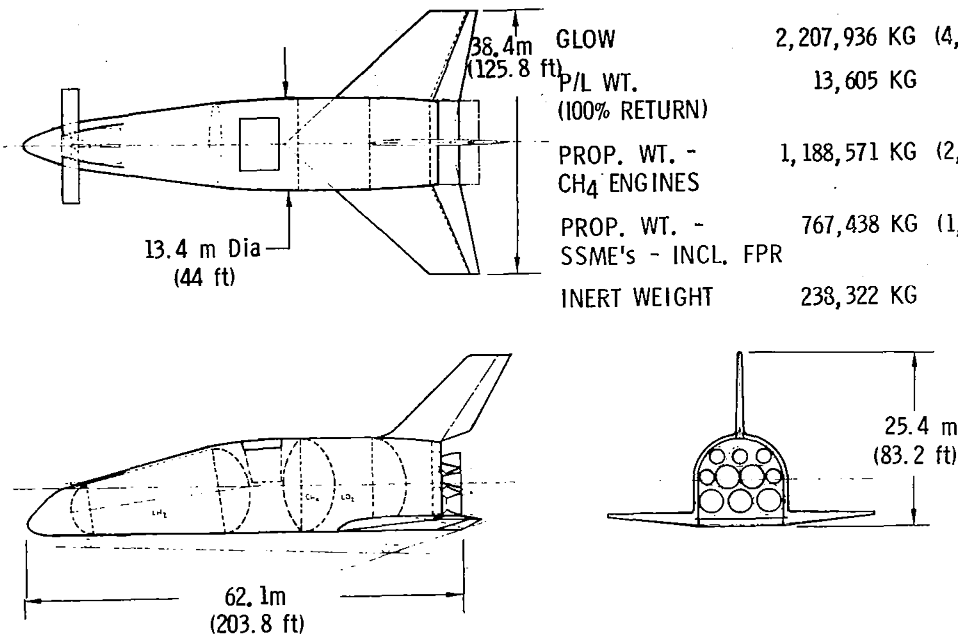


Figure 2.2.1-4b. - Normal Growth Baseline SSTO Ascent Performance

2.2.1.3 Configuration Description

The configuration of the SSTO is presented in Figure 2.2.1-5 with overall geometry and mass data noted. Roundtrip payload capability is 13 605 kg (30 000 lb). The SSTO has a 3-man flight crew. Ascent thrust is provided by five LO_2/LCH_4 engines and five SSME's. The LO_2/LCH_4 engines employ an expander bleed cycle and utilize hydrogen for cooling and turbopump drive. Each LO_2/LCH_4 engine provides a sea level thrust of 3.86 X 10^6N (867 7000 lbf) and each SSME provides a sea level thrust of 1.79 X 10^6N (403 100 lbf). The LO_2/LCH_4 engines consume 60% of the total propellant load prior to being shutdown at a velocity of approximately 3960 m/s (13 000 ft/sec). The airframe structure consists of unpressurized structure and integral main propellant tanks. The unpressurized structures are composite design and are protected from entry heating by advanced RSI. Main tankage structures are 6AL-4V(ELI) titanium sandwich for the LH_2 tank and 2219-T87 aluminum for the LO_2/LCH_4 tank. The tank sidewalls are protected from entry heating by advanced RSI over composite design standoff panels and fairing structures. A summary mass and balance statement is presented in table 2.2.1-1.

Each of the items in the summary mass and balance statement, exclusive of payload and ascent propellant, is discussed in the following paragraphs including definition of rationale for mass estimates.



GLOW	2,207,936 KG (4,868,500 LB)
P/L WT. (100% RETURN)	13,605 KG (30,000 LB)
PROP. WT. - CH_4 ENGINES	1,188,571 KG (2,620,800 LB)
PROP. WT. - SSME's - INCL. FPR	767,438 KG (1,692,200 LB)
INERT WEIGHT	238,322 KG (525,500 LB)

Figure 2.2.1-5. - SSTO Configuration, Normal Growth Technology Baseline

Wing - The wing is a composite design with a mass of 20 771 kg (45 800 lb) and a unit mass of 35.3 kg/m^2 (7.23 lb/ft^2) based on its reference area of 589 m^2 (6337 ft^2). The wing mass is estimated at 70% of that of an aluminum design wing sized for a 2.5g subsonic maneuver at the landing wing loading (reference wing + canard) of 360 kg/m^2 (73.8 lb/ft^2), a 406 K (270^{o}F) surface temperature during the maneuver, and a 1000 hour design accumulated time at surface temperatures above 394 K (250^{o}F). The wing mass includes the box body-carry-thru section and main gear installation provisions. A constant thickness/chord ratio (t/c) of 10% was used. The ratio of elevon area to wing reference area is 11.5%.

Vertical Tail - The vertical tail, which incorporates a rudder/speed brake, is a composite design with a unit mass of 26.0 kg/m^2 (5.32 lb/ft^2) of exposed planform area. The tail unit mass is 70% of the unit mass of the aluminum design tail on the Shuttle Orbiter. The ratio of rudder/speed brake area to tail area is 27%. Tail mass is 3084 kg (6800 lb).

TABLE 2.2.1-1. - SSTO Summary Mass and Balance Statement
Normal Growth Technology Baseline

GROUP/ITEM	MASS (kg)	XCG (m)	WEIGHT (lb)	XCG (in.)
WING	20,771	55.88	45,800	2200
VERTICAL TAIL	3,084	60.20	6,800	2370
CANARD	1,315	7.62	2,900	300
BODY	66,485	40.92	146,600	1611
INDUCED ENVIRONMENTAL PROTECTION	24,490	42.54	54,000	1675
LANDING AND AUXILIARY SYSTEMS	6,258	44.63	13,800	1757
PROPULSION–ASCENT	47,165	56.97	104,000	2243
ROCKET ENGINES	31,270	58.67	68,950	2310
ENGINE ACCESSORIES	2,553	58.67	5,630	2310
PROPELLANT SYSTEM	13,342	53.09	29,420	2090
PROPULSION–RCS	1,496	12.70	3,300	500
PROPULSION–OMS	2,495	41.91	5,500	1650
PRIME POWER	1,152	34.29	2,540	1350
ELECTRICAL CONVERSION & DISTR.	2,372	19.56	5,230	770
HYDRAULIC CONVERSION & DISTR.	1,737	40.64	3,830	1600
SURFACE CONTROLS	3,656	59.87	8,060	2357
AVIONICS	1,977	5.08	4,360	200
ENVIRONMENTAL CONTROL	789	5.08	1,740	200
PERSONNEL PROVISIONS	272	4.57	600	180
PAYLOAD PROVISIONS	154	33.78	340	1330
MARGIN	17,234	45.21	38,000	1780
(DRY CONDITION)	(202,902)	(46.28)	(447,400)	(1822)
PERSONNEL	667	4.57	1,470	180
PAYLOAD–ASCENT (100% RETURN)	13,605	33.78	30,000	1330
RESIDUALS–SYSTEMS/RCS/OMS/ASCENT PROPULSION	5,229	42.87	11,530	1688
RESERVES–SYSTEMS/RCS/OMS	1,560	27.66	3,440	1089
INFLIGHT LOSSES–SYSTEMS	1,061	33.05	2,340	1301
INFLIGHT LOSSES–ASCENT PROPULSION	8,322	55.30	18,350	2177
PROPELLANT–NOMINAL RCS	1,701	5.08	3,750	200
PROPELLANT–NOMINAL OMS	16,880	34.29	37,220	1350
(INERT CONDITION, WITH P/L)	(251,927)	(44.45)	(555,500)	(1750)
ASCENT PROPELLANT–INCL FPR	1,956,009	45.47	4,313,000	1790
LO_2	1,572,091	48.26	3,466,460	1900
LH_2	122,694	21.59	270,540	850
LCH_4	261,224	39.88	576,000	1570
(LIFT OFF CONDITION, WITH P/L)	(2,207,936)	(45.34)	(4,868,500)	(1785)

$$\text{MASS FRACTION} = \frac{\text{ASCENT PROPELLANT MASS}}{\text{LIFTOFF MASS, LESS P/L}} = 0.891$$

Canard - The subsonic trim canard group consists of an aerosurface, hinge/attachment fittings and deployment/latch mechanisms. The aerosurface is a composite design with a unit mass of 20.5 kg/m^2 (4.2 lb/ft^2) based on its exposed planform area. The aerosurface unit mass is estimated at 70% of that of an aluminum design. Fittings and mechanisms unit mass is estimated at 22.0 kg/m^2 (4.1 lb/ft^2) of aerosurface area. The ratio of canard exposed planform area to wing reference area is 5.5%. Total mass is 1315 kg (2900 lb).

Body - The body group consists of the following: nose section, crew module, LH_2 tank, payload bay section, payload bay doors, LO_2/LCH_4 tank, aft body section, thrust structure, body flap, and fairing structures. Total body mass is 66 485 kg (146 600 lb).

- Nose Section-The nose section consists of a semimonocoque shell structure, nose gear installation provisions and crew module peculiar items. The shell structure is of composite design with a unit mass of 11.2 kg/m^2 (2.3 lb/ft^2). This unit mass is 70% of the unit mass of the aluminum design shell structure on the Shuttle Orbiter. The nose gear installation provisions consist of the wheel well, door and mechanisms, and support struts, and are of composite design with a unit mass of 0.25% of design landing weight. This unit mass is 70% of the unit mass of the aluminum design installation provisions on the Shuttle Orbiter. Crew module peculiar items consist of a thermal windshield, observation windows, star tracker door and mechanisms, and miscellaneous access panels, and have a mass of 254 kg (560 pounds) based on the Shuttle Orbiter. Nose section total mass is 2431 kg (5360 lb).

- Crew Module-The crew module is similar to that of the Shuttle Orbiter but is smaller in volume. It is an all-welded 2219-T87 aluminum pressure vessel which incorporates a windshield, observation windows, and side hatch and mechanisms. (Because the payload bay is not adjacent to the crew module, items such as aft observation windows, rear hatch and airlock, were not provided.) The basic pressure vessel mass is estimated at 75% of its Shuttle Orbiter counterpart. The mass of the window and hatch installations is 630 kg (1390 lb) based on Shuttle Orbiter. Crew module total mass is 2104 kg (4640 lb).

- LH_2 Tank-The tank containing the liquid hydrogen is an all-welded 6AL-4V(ELI) titanium honey comb sandwich pressure vessel with a ring stiffened sidewall. The tank is designed by cryo temperature proof test conditions corresponding to 3g boost onset with maximum ullage pressure of 159 kPa (23 psia). To satisfy a 500 mission requirement with minimal weight impact, the tank is designed for 200 cycles to failure and, following the initial proof test, is reproofed every 100 missions. The inner face of the sandwich is sized to carry 100% of proof test pressure. The outer face is minimum gage at 0.030 cm (0.012 inch), including the sidewall regions to which TPS subpanels and fairing structures are attached. Tank mass is 8526 kg (18 800 lb), 6.95% of the mass of the liquid hydrogen.

- Payload Bay Section-This section is a semimonocoque shell structure of composite design. It has a unit mass of 19.5 kg/m^2 (4.0 lb/ft^2) exclusive of the payload bay door area. This unit mass is estimated at 70% of an aluminum design shell structure. The payload bay section mass is 8580 kg (18 920 lb).

- Payload Bay Doors-Approximately 30.7 m^2 (330 ft^2) of door area is provided. The doors are of composite design and have a unit mass of 15.1 kg/m^2 (3.1 lb/ft^2) based on Shuttle Orbiter. Door mass is 463 kg (1020 lb).

- LO_2/LCH_4 Tank-The tank containing the liquid oxygen and liquid methane is an all-welded 2219-T87 aluminum pressure vessel with a common dome of sandwich construction and a stiffened sidewall in the oxygen compartment. The tank is designed by room temperature proof test conditions corresponding to 3g boost onset with maximum ullage pressure of 159 kPa (23 psia). To satisfy a 500 mission requirement with minimal mass impact, the tank is designed for 200 cycles to failure and, following the initial proof test, is reproofed every 100 missions. The mass of the liquid oxygen compartment, exclusive of the common dome, is 16 358 kg (36 070 lb), 1.04% of the mass of the liquid oxygen. The mass of the liquid methane compartment, inclusive of the common dome, is 5955 kg (13 130 lb), 2.28% of the mass of the liquid methane. Both compartments have slosh baffles. Total tank mass is 29 488 kg (49 200 lb).

- Aft Body Section-This section consists of a semimonocoque shell structure extending from the LO_2/LCH_4 tank to the engine support plane, a 1.3m (4.2 ft) long fairing structure located aft of the engine support plane, and a base heat shield support structure installation located at the engine support plane. The structures are all of composite design with unit masses of 24.4 kg/m^2 (5.0 lb/ft^2), and 12.2 kg/m (2.5 lb/ft^2), respectively. These unit masses are estimated at 70% of those of aluminum design structures. Aft body section total mass is 11 900 kg (26 240 lb).

- Thrust Structure-The thrust structure is a beam system of composite design which transmits thrust loads from the five LO_2/LCH_4 engines and five SSME's to the aft body section. Maximum vacuum thrust of the ten engines is 31.9×10^6N (7.175×10^6 lbf). The thrust structure has a mass of 5732 kg (12 640 lb) which is 70% of that of an aluminum design. Its unit mass is 0.00018 kg/N (0.00178 lb/lbf).

- Body Flap-The constant chord body flap provides for pitch trim control and thermally shields the main engines during entry. The flap has a translating trailing edge section which, when extended, increases flap area by 75%. It is of composite design with a unit mass of 15.9 kg/m^2 (3.25 lb/ft^2) of total extended area. This unit mass is 70% of the unit mass of the aluminum design body flap on the Shuttle Orbiter, increased by a 1.5 factor for the translation feature. In addition to the translation mechanism, the body flap mass includes the hinge line support fittings/closeout provisions. Total body flap mass is 1914 kg (4 220 lb).

- Fairing Structures-This group consists of a wing-to-body fairing, a canard fairing (forward of payload bay doors), and a LH_2 main delivery line fairing (aft of payload bay doors). The fairings are of composite design and include approximately 251 m^2 (2 700 ft^2) of fixed fairing at 6.1 kg/m^2 (1.25 lb/ft^2) and approximately 123 m^2 (1325 ft^2) of removeable fairing at 8.1 lb/ft^2 (1.65 lb/ft^2). The unit masses are estimated at 70% of those of aluminum design fairings. Total fairing structure mass is 2 522 kg (5 560 lb).

<u>**Induced Environmental Protection**</u> - This group consists of the external TPS system plus the internal provisions for thermal control, purge/vent/drain, and window conditioning. The external TPS system utilizes advanced RSI on all areas including body nose and wing leading edge. The density of the advanced RSI is 120 kg/m^3 (7.5 lb/ft^3) except in the body nose and wing leading edge areas where the density is 320 kg/m^3 (20.0 lb/ft^3). Based on an entry planform loading of 228 kg/m^2 (46.8 lb/ft^2) with flap retracted, and a 2,000 km crossrange, the TPS unit masses are: outboard wing panels, 18.8 kg/m^2 (3.86 lb/ft^2) of wing exposed planform area; body, including base region, 14.0 kg/m^2 (2.86 lb/ft^2) of body planform area with flap extended; vertical tail, 12.2 kg/m^2 (2.50 lb/ft^2) of tail side planform area. In addition, for the body, the main tankage sidewall regions not covered by fairing structures are covered by composite design standoff subpanels having an estimated average unit mass of 5.9 kg/m^2 (1.2 lb/ft^2). The internal provisions have an estimated mass of 1 859 kg (4 100 lb). The induced environmental protection total mass is 24 490 kg (54 000 lb).

Landing and Auxiliary Systems - This group consists only of landing gear. The landing gear design incorporates the use of high strength tires, servo-design shock struts with extensive use of boron-aluminum, and 55 200 kPa (8 000 psi) hydraulic operated mechanisms. The unit mass of the landing gear is estimated at 2.8% of design landing mass based on modified Shuttle Orbiter data. Total mass is 6 258 kg (13 800 lb).

Ascent Propulsion - The ascent propulsion group consists of rocket engines, engine accessories, and the propellant system. Total mass is 47 165 kg (104 000 lb).

- Rocket Engines-Ascent thrust is provided by five SSME's and five LO_2/LCH_4 engines. Total liftoff thrust is 28.3 x 10^6N (6 354 000 lbf). Pertinent engine characteristics are presented in Section 2.1.2. Total dry engine mass is 31 270 kg (68 950 lb).

- Engine Accessories-Accessories for gimballed engines consist of a gimbal system, hydraulic supply, powerhead heat shield, and nitrogen purge provisions. Fixed engines have only the powerhead heat shield and nitrogen purge provisions. Based on the use of 55 200 kPa (8 000 psi) hydraulic components and modest improvements in heat shield design, the engine accessory masses for a SSME are estimated at 376 kg (830 lb) and 127 kg (280 lb) for gimballed engine and a fixed engine, respectively. The engine accessory masses for a LO_2/LCH_4 engine are estimated at 13% and 4.2% of dry engine weight for a gimballed engine and a fixed engine, respectively, based on modified accessories mass data for a standard SSME. Total accessories mass for three gimballed SSME's, two fixed SSME's, two fixed SSME's two gimballed LO_2/LCH_4 engines, and three fixed LO_2/LCH_4 engines is 2 553 kg (5 630 lb).

- Propellant System-This subgroup consists of the propellant delivery system, tank pressurization (autogenous) and vent subsystems, umbilical hardware, and associated installation hardware. Individual SSME feed line diameters are 0.305 m (12 in) for both LO_2 and LH_2. Individual LO_2/LCH_4 engine feed line diameters are 0.462 m (18.2 in) for LO_2, 0.290 m (11.4 in) for LCH_4, and 0.147 m (5.8 in) for LH_2. The LO_2 lines to each engine are direct tank-to-engine lines. The LCH_4 feed lines are fed by a single main line of 0.531 m (20.9 in) diameter extending through the LO_2 tank. The LH_2 feed lines are fed by two main lines, each 0.437 m (17.2 in) in diameter. Using the Shuttle Orbiter propellant system as a data base, and allowing for a 20% mass reduction relative to current state-of-the-art, the unit mass of the propellant system components located in the aft body region is 934 kg (2 060 lb) per SSME and 1 161 kg (2 560 lb) per LO_2/LCH_4 engine. Using the external tank (ET) propellant system as a data base, and allowing for a 20% mass reduction relative to current state-of-the-art, the mass of the propellant system components located forward of the aft body region is 2 866 kg (6 320 lb). Total propellant system mass is 13 342 kg (29 420 lb).

RCS Propulsion - A LO_2/LH_2 reaction control system provides for vehicle orientation prior to entry and for control during entry. Dry system mass is 1 496 kg (3 300 lb) and was estimated at 73% of total usable (nominal + reserve) RCS propellant mass.

OMS Propulsion - The orbital maneuver system consists of engines and accessories, and associated tank pressurization and propellant delivery and storage elements. A LO_2/LH_2 system was utilized. Dry system mass is 2 495 kg (5 500 lb) and was estimated at 14% of total usable (nominal + reserve) OMS propellant mass.

Prime Power - Power source elements consist of three independent N_2H_4 powered APU's, auxiliary batteries, and alternators. The APU's provide for hydraulic power and for 50 kW(67 hp) (nominal) of electric power. Total system dry mass including the reactant system, lube oil coolant system and exhaust system is estimated at 1 152 kg (2 540 lb).

Electrical Conversion and Distribution - This group consists of the power conversion, conditioning, and cabling elements for a 270 Vdc electrical system. Estimated system mass is 2 372 kg (5 230 lb).

Hydraulic Conversion and Distribution - This group consists of the hydraulic power supply and distribution equipment, including a water boiler temperature control system, for a 55 200 kPa (8 000 psi) hydraulic system. Estimated system mass including hydraulic fluid is 1 737 kg (3 830 lb).

Surface Control - The actuation systems for the aerodynamic control surfaces are included in this group, as are the cockpit controls. The actuation system unit masses, based on modest improvements in systems design relative to Shuttle Orbiter are: elevons, 11.7 kg/m^2 (2.4 lb/ft^2); split rudders/speed brakes, 57.1 kg/m^2 (11.7 lb/ft^2); body flaps, 12.2 kg/m^2 (2.5 lb/ft^2). Total surface controls mass, including 27 kg (60 lb) of cockpit controls, is 3 656 kg (8 060 lb).

Avionics - The avionics group includes elements for guidance, navigation and control, communications and tracking, displays and controls, instrumentation, and data processing and software. Total mass is 1 977 kg (4 360 lb) or 68% of Shuttle Orbiter avionics mass, based on use of advanced avionics and reduced power requirements.

Environmental Control - The environmental control group maintains a habitable environment and provides basic life support functions for the crew, and maintains a conditioned thermal environment for the avionics. Estimated system mass including closed loop fluids is 789 kg (1 740 lb).

Personnel Provisions - This group consists of the fixed life support system, personnel accommodations (seats), and furnishings for the 3-man flight crew. Total mass is 272 kg (600 lb).

Payload Provisions - This group consists of fixed scar items and removable provisions with unit masses of 0.30% and 0.85%, respectively, of maximum ascent payload mass, based on Shuttle Orbiter. Total mass is 154 kg (340 lb).

Margin - A margin allowance of approximately 10% of subsystems dry mass exclusive of SSME's has been incorporated. Total mass is 17 234 kg (38 000 lb).

Personnel - This group consists of the 3-man flight crew and their accessories. Unit masses for a crewman and his accessories are 86 kg (190 lb) and 136 kg (300 lb), respectively, based on Shuttle Orbiter. Total mass is 667 kg (1 470 lb).

Nominal RCS Propellant - The RCS delta-V budget is 32.5 m/sec (105 ft/sec) and consists of the following:

Trim Burns - Post Ascent	1.5 m/s (5 ft/sec)
Trim Burns - 500 km Orbit	17 m/s (55 ft/sec)
Trim Burns - Entry	14 m/s (45 ft/sec)

Based on a specific impulse of 420 sec for the LO_2/LH_2 thrusters, and consideration of vehicle sequential mass, the total nominal propellant mass is 1 701 kg/(3 750 lb).

Nominal OMS Propellant - The estimated OMS delta-V budget is 336 m/s (1 103 ft/sec) and consists of the following:

Circularization into 186 km Parking Orbit from 93 x 186 km Launch Orbit	28 m/s (91 ft/sec)
Insertion into 186 x 500 km Transfer Orbit from 186 km Parking Orbit	90 m/s (297 ft/sec)
Circularization into 500 km Orbit from 186 x 500 km Transfer Orbit	90 m/s (294 ft/sec)
Trim Burns in 500 km Orbit	11 m/s (37 ft/sec)
Deorbit from 500 km Orbit	117 m/s (384 ft/sec)

Based on a specific impulse of 476 sec for the LO_2/LH_2 OMS engines, and consideration of vehicle sequential mass, the total nominal propellant mass is 16 880 kg (37 220 lb).

Systems Inflight Losses - This group consists of the nominal usage of the following: N_2H_4 for hydraulic and electric power; GO_2, GN_2, H_20, and NH_3 for crew module environmental control and crew life support; and H_20 for hydraulic cooling. Total estimated mass is 1 061 kg (2 340 lb).

Ascent Propulsion Inflight Losses - All ascent propulsion liquid residuals are jettisonable and are included in this group. Total mass is 8 322 kg (18 350 lb). Pertinent line size data is included under ascent propulsion-propellant system.

- Propellant in Engines at Cutoff-Trapped propellant mass is 236 kg (520 lb) per SSME and is estimated at 204 kg (450 lb) per LO_2/LCH_4 engine. Total mass is 2,200 kg (4 850 lb).

- Oxidizer in Engine Feed Lines at Cutoff-The LO_2 lines to each SSME and each LO_2/LCH_4 engine are direct tank-to-engine lines. The lines average 4.0 m (13 ft) in length and contain an average 3.0m (10 ft) of LO_2 at cutoff. The LO_2 unit masses are 254 kg (560 lb) per SSME and 585 kg (1 290 lb) per LO_2/LCH_4 engine. Total mass is 4 195 kg (9 250 lb).

- Fuel in Engine Feed Lines at Cutoff-An average 3.7 m (12 ft) of LH_2 line and 4.9 m (16 ft) of LCH_4 line are located between the main delivery line(s) and each engine.

These lines are full at engine cutoff. The LH_2 unit mass is 18 kg (40 lb) per SSME and 4.5 kg (10 lb) per LO_2/LCH_4 engine. The LCH_4 unit mass is 131.5 kg (290 lb) per LO_2/LCH_4 engine. Total mass is 770m kg (1 700 lb).

o Fuel in Main Delivery Lines at Cutoff-There are two main LH_2 lines and one main LCH_4 line. At engine cutoff under nominal conditions, these lines contain 3.0 m (10 ft) of fuel, exclusive of bias fuel. The LH_2 unit masses is 10.9 kg (24 lb) per SSME and 2.7 kg (6 lb) per LO_2/LCH_4 engine. The LCH_4 unit mass is 54.5 kg (120 lb) per LO_2/LCH_4 engine. Total mass is 340 kg (750 lb).

o Bias Fuel-A single-engine reference bias fuel allowance of 0.50% was used. For multiple engines, this allowance was factored by the inverse of the square root of the number of engines. The resulting LH_2 bias for the five SSME's and the LCH_4 bias for the five LO_2/LCH_4 engines are both 0.22%. The LH_2 mass is 240 kg (530 lb) and the LCH_4 mass is 576 kg (1 270 lb). Total mass is 816 kg (1 800 lb).

With respect to the above, the liquid residuals associated with the LO_2/LCH_4 engines (with the exception of the LH_2 in the main delivery lines) are jettisoned at LO_2/LCH_4 engines cutoff which occurs at a velocity of approximately 3 960 m/s (13 000 ft/sec). This early jettisoning (of approximately 66% of the inflight losses) is necessary to maximize payload.

Reserves - This group consists of reserve allowances for the RCS, OMS, and systems. (The flight performance reserve for ascent is 0.85% of the total ideal velocity change and is included in ascent propellant mass.) Reserve allowances for RCS and OMS are 20% and 5%, respectively, of nominal propellant usage. The systems reserve allowances are estimated at 376 kg (830 lb). Total reserves mass is 1 560 kg (3 440 lb).

Residuals - This group consists of the residual fluids and gases in the ascent propulsion, RCS, OMS, and systems at time of landing. Total mass is 5 229 kg (11 530 lb).

o Ascent Propulsion Residuals-Only main tank pressurants are onboard at landing. The pressurant masses are based on maximum ullage pressures of 159 kPa (23 psia) and mean temperatures for the GO_2, GH_2, and GCH_4 of 264 K (475^oR), 189 K)340^oR), and 220 K (400^oR), respectively. The masses of the GO_2, GH_2, and GCH_4 are 3,320 kg (7 320 lb), 372 kg (820 lb), and 893 kg (1 970 lb), respectively. Total pressurant mass is 4 585 kg (10 110 lb).

- o RCS, OMS, and Systems Residuals-Residual allowances for both RCS and OMS were estimated at 2% of usable propellant (nominal + reserve). The systems residuals were estimated at 218 kg (480 lb). Total mass is 644 kg (1 420 lb).

2.2.1.4 Life Cycle Costs

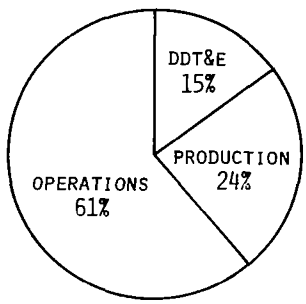
The life cycle costs (LCC) for the baseline priority cargo launch vehicle (SSTO) have been separated into three phases:

1) DDT&E
2) Production
3) Operations

A breakdown of the SSTO LCC by phase and subphase is shown in figure 2.2.1-6. The costs associated with each of the three phases are discussed in the following paragraphs. All costs are in constant year 1977 dollars except where noted. The cost methodology and groundrules used in this study are presented in Appendix B.

DDT&E Phase - The total DDT&E phase cost for the baseline SSTO is $2714.5 million. The DDT&E cost breakdown is summarized in table 2.2.1-2. There are four subphases in the DDT&E phase: (1) program management, (2) engineering, (3) manufacturing, and (4) test.

	$ MILLIONS		
TOTAL PROGRAM	17,912.90		
DDT&E		2,714.47	
PROGRAM MANAGEMENT			86.98
ENGINEERING			833.50
MANUFACTURING			1,373.58
TEST			420.41
PRODUCTION		4,245.06	
PROGRAM MANAGEMENT			377.48
SYSTEM ENGINEERING			87.27
MANUFACTURING			3,780.32
OPERATIONS		10,953.37	
OPERATIONS SUPPORT			6,700.19
LAUNCH SUPPORT			4,253,18



● COST/FLT = $3.79 x 10^6 ● COST = $278/Kg

Figure 2.2.1-6. - SSTO Cost Summary

TABLE 2.2.1-2—BASELINE SSTO DDT&E COSTS

DDT&E	2714.5			
PROGRAM MANAGEMENT		87.0		
ENGINEERING		833.5		
SYSTEMS ENG. & INT.			47.3	
SOFTWARE ENGINEERING			17.2	
DESIGN & DEVELOPMENT			769.0	
STRUCTURE				272.4
TPS				87.3
LANDING & AUX. SYSTEMS				53.0
ASCENT PROPULSION				42.3
AUX. PROPULSION				12.9
PRIME POWER				2.4
ELECT. CONV. & DIST.				17.8
HYDRAULIC CONV. & DIST.				11.7
SURFACE CONTROLS				6.8
AVIONICS				87.8
ECS				8.8
PERSONNEL PROVISION				3.3
GSE				162.4
MANUFACTURING		1373.6		
DEVELOPMENTAL TOOLING & S.T.E.			153.2	
TEST HDWE & SPARES			1220.3	
TEST		420.4		
SYSTEMS TEST OPERATIONS			390.1	
FLIGHT TEST OPERATIONS			30.3	

Engineering - There are several SSTO subsystems that are used in common with other vehicles in the transportation system. The design & development (D&D) costs of these subsystems have been included in the LCC's of these other vehicles as follows:

(1) LO_2/LCH_4 main engine; cost included in HLLV LCC

(2) SSME extendable nozzle; cost included in HLLV LCC

(3) LO_2/LH_2 RCS thruster; cost included in HLLV LCC

(4) LO_2/LH_2 OMS engine; cost included in POTV LCC

In addition the following subsystems are derived from or use components from STS Orbiter subsystems, and as a result have reduced D&D costs:

- o Avionics
- o Prime power
- o Thermal protection system

Manufacturing - This DDT&E subphase contains the costs of producing the hardware required for the SSTO test program. This includes: ground and flight test hardware, test program spares, developmental tooling required to produce the test hardware, and special test equipment (STE).

The test hardware consists of a structural test article, a propulsion/dynamic test vehicle a flight test vehicle and one shipset of GSE. The propulsion/dynamic test vehicle is used initially for dynamic tests and then refurbished and set up for main propulsion tests. Both it and the flight test vehicle are refurbished at the completion of the test program and included in the operational fleet. The test hardware and spares cost is divided as follows:

- o Structural test article - $85.7 million
- o Propulsion/dynamic test vehicle - $562.6 million (includes $187.5 million for refurbishment between dynamic testing and main propulsion tests)
- o Flight test vehicle - $375.1 million
- o Test program GSE and spares - $197.0 million

Test - The Test subphase contains two cost elements: (1) System test operations and (2) Flight test operations. The first element includes the manpower cost of all component, subsystem and system ground tests. In the second element are included all the operations costs assocated with the flight test program, except the cost of the flight test hardware.

Production Phase - The Production phase of the baseline SSTO LCC includes all the costs related to production of the operating fleet. The cost of this phase is $4245.1 million. Not included are the costs of operational spares and overhauls. A breakdown of the production phase costs are shown in table 2.2.1-3.

TABLE 2.2.1-3—PRODUCTION PHASE COST

PROGRAM ELEMENT	COST ($ MILLION)		
PRODUCTION PHASE	4245.1		
PROGRAM MANAGEMENT		377.5	
SUSTAINING ENGINEERING		87.3	
MANUFACTURING		3780.3	
PRODUCTION TOOLING			1382.9
FLIGHT HARDWARE & SPARES			2397.4

Manufacturing is the major cost element in the production phase. It includes the fleet hardware and initial spares and the production tooling. The fleet hardware costs are developed from the fleet size determined in the operations phase analysis and the vehicle, engine and GSE Theoretical First Unit (TFU) costs. The TFU costs are:

- Vehicle - $420.86 million
- LO_2/LCH_4 Main Engine - $9.26 million
- SSME - $15.06 million
- OMS Engine - $1.83 million
- GSE - $86.03 million

The fleet of 8 vehicles includes two vehicles from the test program refurbished at a cost of $210.4 million.

Operations Phase - The distribution of the operations phase costs, totalling $10,953.4 million, is shown in table 2.2.1-3. The spares cost includes the cost of per flight replenishment spares, periodic overhauls and any hardware required to replace limited life components. The spares criteria used are listed as follows:

- launch vehicle airframes overhauled every 100 flights at 30% of unit cost
- POTV airframes overhauled every 50 flights at 30% of unit cost
- CH_4 engines overhauled every 250 flights at 50% of unit cost
- SSME's overhauled every 55 flights at 50% of unit cost

TABLE 2.2.1-4.–OPERATIONS PHASE COST

PROGRAM ELEMENT	$ MILLION		
OPERATIONS PHASE	10953.4		
OPERATIONS SUPPORT		6700.2	
PROGRAM SUPPORT			1576.8
SPARES PROCUREMENT			5123.4
LAUNCH SUPPORT		4253.2	
OPERATIONS			3177.2
PROPELLANT			1076.0

- ASE's overhauled every 60 starts (15 PTOV flights, 20 SSTO flights or 30 HLLV flights) at 90% of unit cost
- airframe replenishment @ .18% of unit cost
- engine replenishment @ .5% of unit cost
- ion thruster refurbishment/flight @ 50% of unit cost per flight
- LCOTV airframe refurbishment spares @ 10% of unit cost per flight
- LCOTV solar array annealed after each flight

The major spares cost is the refurbishment and replacement of SSME's at approximately $2.7 billion.

The SSTO Operations phase spans a 15 year period with 2888 total flights. Of these, 1951 are priority launches that were not given specific payloads in the mission model. For the purpose of analysis, these flights were treated as full payload flights, i.e., no adjustment was made for possible propellant offloading.

The SSTO fleet size was determined by the peak flight rate of 252 launches per year, using the following groundrules:

- baseline 2 shifts/day, 5 days/week, 50 weeks/yr = 4000 hr/yr
- 3 shifts/day available on temporary basis
- max. vehicle flight rate/yr = 4000/vehicle turnaround time
- fleet sized to meet maximum yearly flight rate with available vehicles (excludes vehicles being overhauled)
- fleet also sized to meet maximum flight rate with one vehicle undergoing unscheduled maintenance for up to 3 months.

A turnaround time of 115 hours was estimated by adjusting HLLV turnaround times from previous studies (references 16 and 17).

2.2.2 Heavy Lift Launch Vehicle (HLLV)

2.2.2.1 Design Requirements and Configuration Issues

The HLLV provides the capability to deliver large payloads to LEO at low cost. The primary goal of the HLLV is to achieve a low cost per kg delivered to LEO. Normal growth HLLV design requirements and guidelines groundruled at the beginning of this study follow:

- 227 M.T. to 500 km & 28.5^{o} inclination - 10% return
- 100 KG/M^3 payload density
- two stage-parallel burn with x-feed (LH_2 & LOX)
- vertical takeoff @ T/W = 1.3 - horizontal landing @ 165 knots
- heat sink booster - airbreathing flyback system
- unmanned vehicles
- 500 mission life
- minimum orbit stay time

They are based on previous studies of this type of launch vehicle. Figure 2.2.2-1 presents the selected design HLLV concept and its key features. Figure 2.2.2-2 shows the Orbiter design key features.

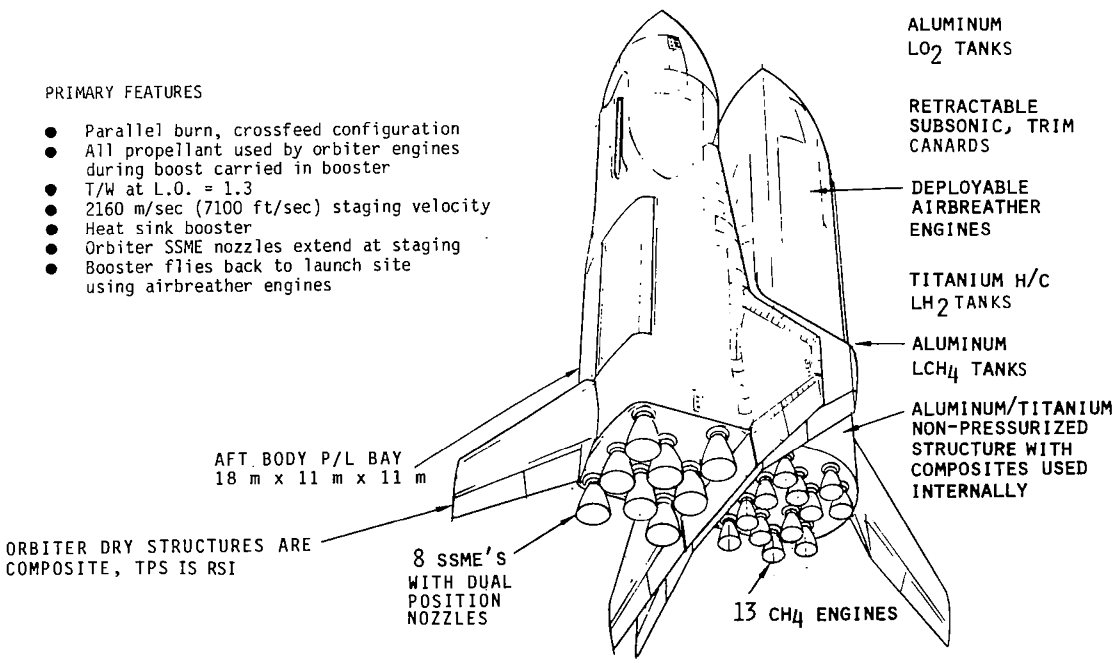


Figure 2.2.2-1. - HLLV Design Concept

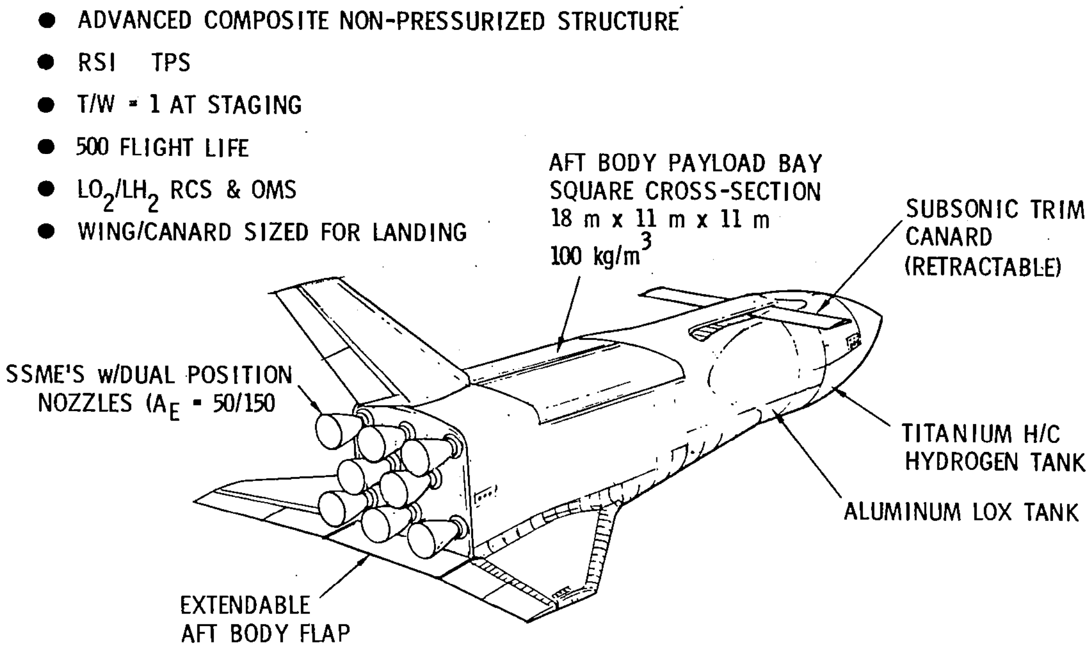


Figure 2.2.2-2. - Orbiter Design Concept

The vehicle is a winged two stage, parallel burn crossfeed configuration. In the parallel burn crossfeed arrangement the Orbiter engines are used during first stage operations, reducing the booster total thrust requirement by 22%. The propellant is carried in the booster and crossfed to the Orbiter during boost, leaving the Orbiter fully fueled at staging.

The major design issues involved in defining the HLLV concerned the Orbiter payload bay the mated vehicle ascent configuration, and the aerodynamic characteristics of the stages. Just prior to staging almost all the vehicle mass is located in the fully fueled Orbiter. However most of the thrust is produced by the nearly empty booster. This causes the HLLV to assume a negative angle of attack, although the engines remain aligned with the flight path. The location of the Orbiter cg determines the size of the angle of attack. Based on Orbiter considerations alone, a LO_2 tank aft arrangement is preferable, to reduce the axial loads in the remainder of the vehicle. However this causes a 22^o negative angle of attack, with resultant excessive body bending loads, engine TVC requirements and top side heating. A midbody LO_2 location alleviates this problem.

The angle of attack at staging is reduced to 14^{o} and the bending loads reduced by 30%. Moving the LO_2 tank into the nose would reduce the problem further, but is not possible because the LO_2 tank lacks the volume to fill the nose. The Orbiter top side heating does not add to the TPS requirement. The TPS required for re-entry is sufficient.

The location and shape of the payload bay is the other major design issue. A comparison was made between a payload bay located in the nose and one located in the body aft of the LO_2 tank. The nose payload bay consisted of a cylindrical section with an ogive nose cap. The aft body payload had a rectangular cross section. This was selected instead of a cylindrical bay because it had a higher ratio of usable to wetted volume and was shorter. It also provided superior bending strength because the door cutout was smaller than that of the cylindrical bay.

The rectangular aft body payload bay was selected over the nose bay for the following reasons:

- It required 15% less wetted volume.
- It resulted in a 12% shorter stage.
- A 5 000 kg reduction in mass was achieved because of the decrease in length and payload bay surface area.
- The nose payload bay had additional mass penalties due to the load carrying doors and the TPS sealing problem.

The aerodynamic configuration of the normal growth HLLV was determined primarily from the results of the SSTO aero analysis. Although the HLLV stage cg's were slightly further aft, their re-entry trim requirements were less stringent due to the absence of any cross range requirement. In addition small discrepancies can be accommodated by using ballast, with slight impact on performance.

2.2.2.2 Performance and Sizing

The normal growth HLLV delivers a 227 M ton payload to a 28.5^{o} inclination, 500 km circular orbit from a 28.5^{o} latitude launch site. The vehicle flies a typical 2 stage non-lifting trajectory. It lifts off at a 1.30 T/W, pitches over 6^{o} and flies a ballistic trajectory until staging. After separation the Orbiter extends the nozzles on its SSME's to the =150 position and flies an angle of attack controlled trajectory to insertion into a 110 x 500 km orbit. Circularization at 500 km is accomplished with 112 m/sec (367 ft/sec) V burn by the OMS engines. The staging velocity was chosen to take maximum advantage of the high performance booster engines, but still allow use of heat sink TPS on the booster. The booster re-enters following separation at a high angle of attack that generates maximum drag with a minimum total heat load. It flies back to the launch site subsonically using airbreather engines for propulsion. The Orbiter de-orbits after delivering its payload and re-enters using an STS Orbiter type of re-entry trajectory.

A summary of the HLLV ascent trajectory characteristics is presented below and the trajectory parameters as a function of time are shown in figure 2.2.2-3.

T/W @ Liftoff:	1.30	
Max. Dynamic Pressure:	32.65 kPa	(682.0 psf)
Max. Acceleration (both stages):	3.00 g	

Conditions @ Staging:

Time:	159.0 sec	
Altitude:	56 499m	(185 364 ft)
Velocity (Rel.):	2 150 m/sec c)	(7 055 ft/sec)
Dynamic Pressure:	1.17 kPa	(24.5 psf)
Flight Path Angle:	16.6°	
2nd Stage T/W:	1.04	

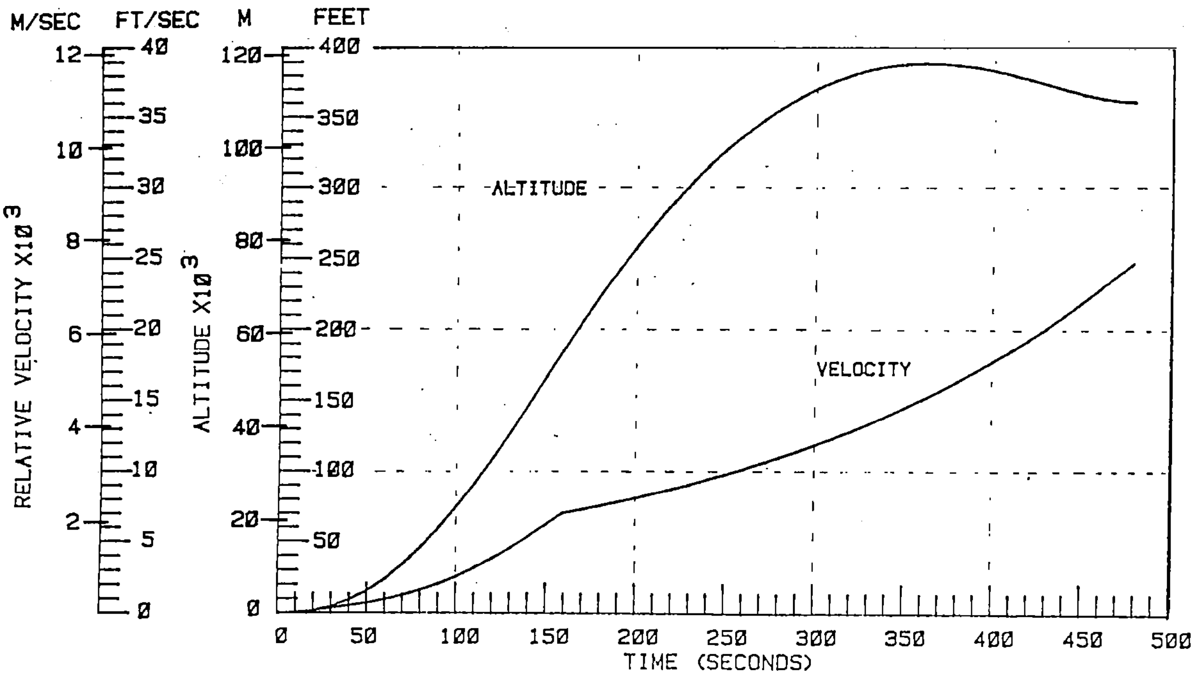


Figure 2.2.2-3a. - Normal Growth Baseline HLLV Ascent Performance

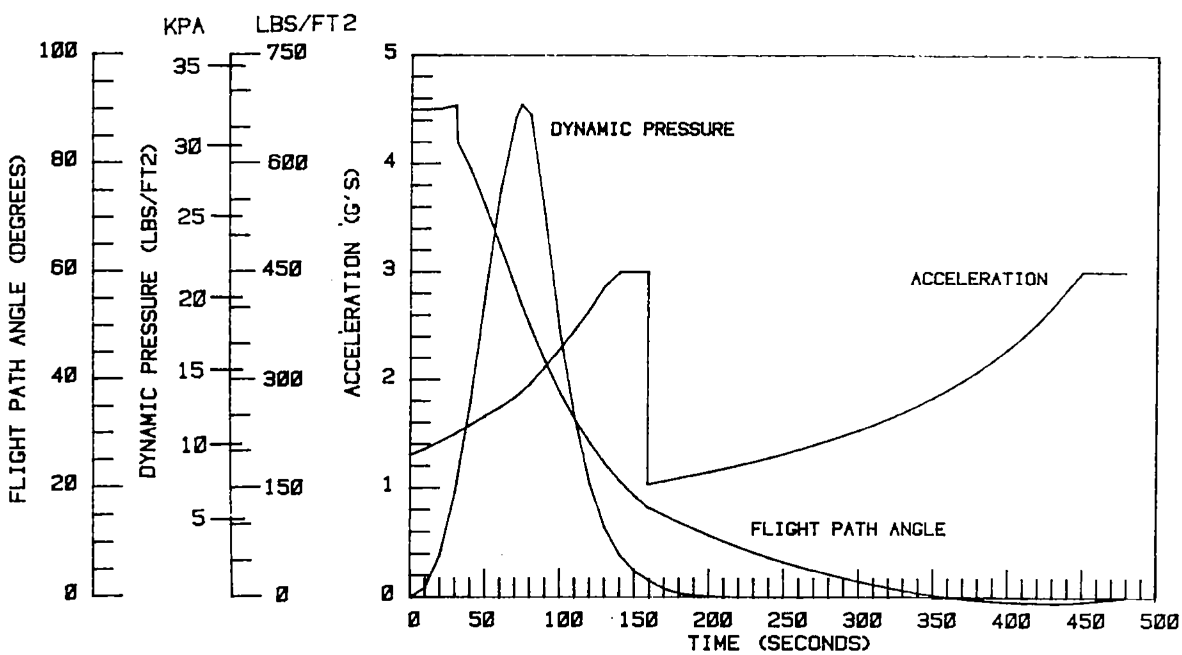


Figure 2.2.2-3b. - Normal Growth HLLV Baseline Ascent Performance

2.2.2.3 Orbiter Configuration Description

The configuration of the HLLV Orbiter is presented in Figure 2.2.2-4 with the overall geometry noted. Net delivered payload capability is 226 757 kg (500 000 lb) with 10% net return payload capability. The Orbiter is unmanned. Ascent thrust is provided by eight SSME's, each having a sea level thrust of 1.79×10^6N (403 100 lbf) and a vacuum thrust of 2.14×10^6N (480 000 lbf). The airframe structure consists of unpressurized structures, an integral main LH_2 tank and a nonintegral LO_2 tank. The unpressurized structures are composite design and are protected from entry heating by advanced RSI. Main LH_2 tank structure is 6AL-4V(ELI) titanium sandwich, the sidewall of which is protected from entry heating by advanced RSI over composite design standoff panels and fairing structures. Main LO_2 tank structure is 2219-T87 aluminum. A summary mass and balance statement is presented in Table 2.2.2-1.

Each of the items in the summary mass and balance statement, exclusive of payload and ascent propellant, is discussed in the following paragraphs including definition of rationale for mass estimates.

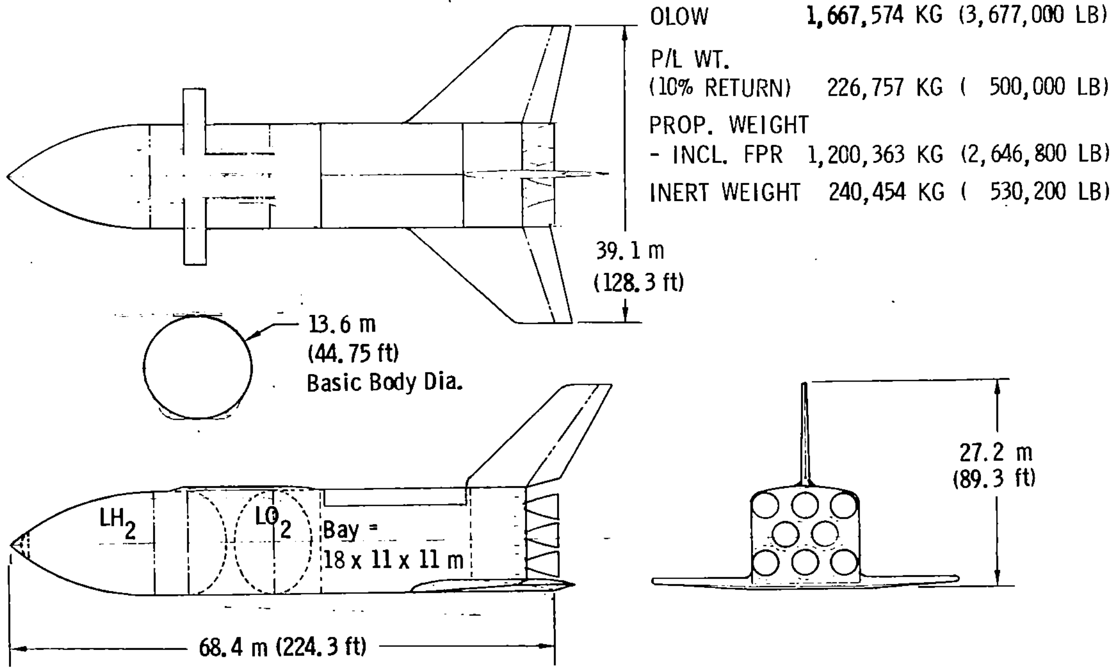


Figure 2.2.2-4. - HLLV Orbiter Configuration, Normal Growth Technology Baseline

TABLE 2.2.2-1 Summary Mass and Balance Statement
Normal Growth Technology Baseline

GROUP/ITEM	MASS (kg)		XCG (m)		WEIGHT (lb)		XCG (in.)	
WING		20,771		62.23		45,800		2450
VERTICAL TAIL		3,084		66.04		6,800		2600
CANARD		2,177		25.40		4,800		1000
BODY		68,163		41.71		150,300		1642
INDUCED ENVIRONMENTAL PROTECTION		26,122		44.60		57,600		1756
LANDING AND AUXILIARY SYSTEMS		7,982		50.98		17,600		2007
PROPULSION–ASCENT		39,592		62.20		87,300		2449
ROCKET ENGINES	25,778		65.02		56,840		2560	
ENGINE ACCESSORIES	3,011		65.02		6,640		2560	
PROPELLANT SYSTEM	10,803		54.69		23,820		2153	
PROPULSION–RCS		2,494		33.02		5,500		1300
PROPULSION–OMS		2,676		40.64		5,900		1600
PRIME POWER		1,293		37.59		2,850		1480
ELECTRICAL CONVERSION & DISTR.		2,993		39.37		6,600		1550
HYDRAULIC CONVERSION & DISTR.		2,630		62.23		5,800		2450
SURFACE CONTROLS		4,308		65.15		9,500		2565
AVIONICS		1,224		27.94		2,700		1100
ENVIRONMENTAL CONTROL		227		27.94		500		1100
PAYLOAD PROVISIONS		2,608		48.36		5,750		1904
MARGIN		16,010		47.55		35,300		1872
(DRY CONDITION)		(204,354)		(49.71)		(450,600)		(1957)
PAYLOAD–ASCENT (10% RETURN)		226,757		48.36		500,000		1904
RESIDUALS–SYSTEMS/RCS/OMS/ASCENT PROPULSION		3,401		31.37		7,500		1235
RESERVES–SYSTEMS/RCS/OMS		2,059		30.58		4,540		1204
INFLIGHT LOSSES–SYSTEMS		957		44.70		2,110		1760
INFLIGHT LOSSES–ASCENT PROPULSION		8,980		61.92		19,800		2438
PROPELLANT–NOMINAL RCS		2,844		27.94		6,270		1100
PROPELLANT–NOMINAL OMS		17,859		27.94		39,380		1100
(INERT CONDITION, WITH P/L)		(467,211)		(48.11)		(1,030,200)		(1894)
ASCENT PROPELLANT–INCL. FPR		1,200,363		30.89		2,646,800		1216
LO_2	1,028,889		33.02		2,268,700		1300	
LH_2	171,474		18.03		378,100		710	
(LIFT OFF CONDITION, WITH P/L)		(1,667,574)		(35.69)		(3,677,000)		(1405)

$$\text{MASS FRACTION} = \frac{\text{ASCENT PROPELLANT MASS}}{\text{LIFTOFF MASS, LESS P/L}} = 0.833$$

Wing - The wing is a composite design with a mass of 20 771 kg/m^2 (45 800 lb) and a unit mass of 33.9 kg/m^2 (6.95 lb/ft^2) based on its reference area of 612 m^2 (6 588 ft2). The wing mass is estimated at 70% of that of an aluminum design wing sized for a 2.5g subsonic maneuver at the landing wing loading (reference wing + canard) of 352 kg/m^2 (72.0 lb/ft^2), a 406 K (270°F) surface temperature during the maneuver, and a 1 000-hour design accummulated time at surface temperatures above 394 K (250°F). The wing mass includes the box body-carry-through section and main gear installation provisions. A constant t/c of 10% was used. The ratio of elevon area to wing reference area is 13.5%.

Vertical Tail - The vertical tail, which incorporates a rudder/speed brake, is a composite design with a unit mass of 26.0 kg/m^2 (5.32 lb/ft^2) of exposed planform area. The tail unit mass is 70% of the aluminum design tail on the Shuttle Orbiter. The ratio of rudder/speed brake area to tail area is 29%. Tail mass is 3 084 kg (6 800 lb).

Canard - The subsonic trim canard group consists of an aerosurface, hinge/attachment fittings and deployment/latch mechanisms. The aerosurface is a composite design with a unit mass of 20.5 kg/m^2 (4.2 lb/ft^2) based on its exposed planform area. The aerosurface unit mass is estimated at 70% of that of an aluminum design. Fittings and mechanisms unit mass is estimated at 23.9 kg/m^2 (4.9 lb/ft^2) of aerosurface area. The ratio of canard exposed planform area to wing reference are is 8.0%. Total mass is 2 177 kg (4 800 lb).

Body - The body group consists of the following: nose cap, LH_2 tank, intertank section, LO_2 tank, payload bay section, payload bay doors, aft body section, thrust structure, body flap, and fairing structures. Total body mass is 68 163 kg (150 300 lb).

o Nose Cap-The nose cap is a semimonocoque shell structure of composite design with a unit mass of 11.2 kg/m^2 (2.3 lb/ft^2). This unit mass is estimated at 70% of that of an aluminum design nose cap. Nose cap mass is 181 kg (400 lb).

o LH_2 Tank-The tank containing the liquid hydrogen is an all-welded 6AL-4V(ELI) titanium sandwich pressure vessel with a ring stiffened sidewall. The tank is designed by cryo temperature proof test conditions corresponding to 3g boost with maximum ullage pressure of 159 kPa (23 psia). To satisfy a 500-mission requirement with minimal weight impact, the tank is designed for 200 cycles to failure and, following the initial proof test, is reproofed every 100 missions. The inner face of the sandwich is sized to carry 100% of proof test pressure. The outer face is minimum gage at 0.030 cm (0.012 in), including the sidewall regions to which TPS subpanels and fairing structures are attached. Tank mass is 9 206 kg (20 300 lb), 5.38% of the mass of the liquid hydrogen.

o Intertank Section-The intertank section consists of a semimonocoque shell structure and incorporates nose gear installation provisions. The shell structure is of composite design with a unit mass of 11.2 kg/m^2 (2.3 lb/ft^2). This unit mass is estimated at 70% of an aluminum design shell structure. The nose gear installation provisions consist of the wheel well, door and mechanisms, and support struts, and are of composite design with a unit mass of 0.25% of design landing mass. This unit mass is 70% of the unit mass of the aluminum design installation provisions on the Shuttle Orbiter. Intertank section total mass is 5 669 kg (12 500 lb).

o LO_2 Tank-The tank containing the liquid oxygen is an all-welded 2219-T87 aluminum pressure vessel. The tank is designed by room temperature proof test conditions corresponding to 3g boost with maximum ullage pressure of 159 kPa (23 psia). To satisfy a 500-mission requirement with minimal mass impact, the tank is designed for 200 cycles to failure and, following the initial proof test, is reproofed every 100 missions. The tank mass, including slosh baffles, is 12 426 kg (27 400 lb), 1.21% of the mass of the liquid oxygen.

o Payload Bay Section-This section is a semimonocoque shell structure of composite design. It has a unit mass of 19.5 kg/m^2 (4.0 lb/ft^2) exclusive of the payload bay door area. This unit mass is estimated at 70% of that of an aluminum design shell structure. The payload bay section mass is 18 594 kg (41 000 lb).

o Payload Bay Doors-Approximately 273 m^2 (2 935 ft^2) of door area is provided. The doors are of composite design and have a unit mass of 15.1 kg/m^2 (3.1 lb/ft^2) based on Shuttle Orbiter. Door mass is 4 127 kg (9 100 lb).

o Aft Body Section-This section consists of a semimonocoque shell structure extending from the payload bay aft bulkhead to the engine support plane, a 1.3 m (4.2 ft) long fairing structure located aft of the engine support plane, and a base heat shield support structure installation located at the engine support plane. The structures are all of composite design with unit masses of 24.4 kg/m^2 (5.0 lb/ft^2), 12.2 kg (2.5 lb/ft^2), and 12.2 kg/m^2 (2.5 lb/ft^2), respectively. These unit masses are estimated at 70% of those of aluminum design structures. Aft body section total mass is 10 113 kg (22 300 lb).

- Thrust Structure-The thrust structure is a beam system of composite design which transmits thrust loads from the eight SSME's to the aft body section. Maximum vacuum thrust of the eight engines is 17.1 x 10^6N (3.840 x 10^6 lbf). The thrust structure has a mass of 4 127 kg (9 100 lb) which is 70% of that of an aluminum design. Its unit mass is 0.00024 kg/N (0.00237 lb/lbf).

- Body Flap-The constant chord body flap provides for pitch trim control and thermally shields the main engines during entry. The flap has a translating trailing edge section which, when extended, increases flap area by 75%. It is of composite design with a unit mass of 15.9 kg/m^2 (3.25 lb/ft^2) of total extended area. This unit mass is 70% of the unit mass of the aluminum design body flap on the Shuttle Orbiter, increased by a 1.5 factor for the translation feature. In addition to the translation mechanism, the body flap mass includes the hinge line support fittings/closeout provisions. Total body flap mass is 2 358 kg (5 200 lb).

- Fairing Structures-This group consists of a small wing leading edge-to-body fairing, a canard fairing and, in the LO2 tank upper and lower check regions, body shape transition fairings. The fairings are of composite design and include approximately 88 m^2 (950 ft^2) of fixed fairing at 6.1 kg/m^2 (1.25 lb/ft^2) and approximately 102 m^2 (1 100 ft^2) of removable fairing at 8.1 kg/m^2 (1.65 lb/ft^2). The unit masses are estimated at 70% of those of aluminum design fairings. Total fairing structure mass is 1 360 kg (3 000 lb).

<u>Induced Environmental Protection</u> - This group consists of the external TPS system plus the internal provisions for thermal control and purge/vent/drain. The external TPS system utilizes advanced RSI on all areas including body nose and wing leading edge. The density of the advanced RSI is 120 kg/m^3 (7.5 lb/ft^3) except in the body nose and wing leading edge areas where the density is 320 kg/m^3 (20.0 lb/ft^3). Based on an entry planform loading of 199 kg/m^2 (40.7 lb/ft^2) with flap retracted, no crossrange, the TPS

unit masses based on modified Shuttle Oribter data are: outboard wing panels, 17.6 kg/m^2 (3.60 lb/ft^2) of wing exposed planform area; body, including base region, 14.5 kg/m^2 (2.97 lb/ft^2) of body planform area with flap extended; vertical tail, 12.2 kg/m^2 (2.5 lb/ft^2) of tail side planform area. In addition, for the body, the main tankage sidewall regions not covered by fairing structures are covered by composite design standoff subpanels having an estimated average unit mass of 5.9 kg/m^2 (1.2 lb/ft^2). The internal provisions have an estimated mass of 1 814 kg (4 000 lb). The induced environmental protection total mass is 26 122 kg (57 600 lb).

Landing and Auxiliary Systems - In addition to landing gear, this group includes auxiliary systems for Orbiter-booster separation. The landing gear design incorporates the use of high strength cord tires, servo-design shock struts with extensive use of boron-aluminum, and 55 200 kPa (8 000 psi) hydraulic operated mechanisms. The landing gear unit mass is estimated at 2.8% of design landing mass based on modified Shuttle Orbiter data. Separation systems mass is estimated at 1 451 kg (3 200 lb). Total mass is 7 982 kg (17 600 lb).

Ascent Propulsion - The ascent propulsion group consists of rocket engines, engine accessories, and the propellant system. Total mass is 39 592 kg (87 300 lb).

- Rocket Engines-Ascent thrust is provided by eight SSME's. Total vacuum thrust is 17.1×10^6 N (3 840 000 lbf). Pertinent engine characteristics are presented in Section 2.1.2. Total dry engine mass is 25 778 kg (56 840 lb).

- Engine Accessories-Accessories for each of the eight SSME's consist of a gimbal system, hydraulic supply, powerhead heat shield, and nitrogen purge provisions. Based on the use of 55 200 kPa (8 000 psi) hydraulic components and modest improvements in heat shield design, the engine accessories mass for a gimballed SSME is estimated at 376 kg (830 lb). Total accessories mass is 3 011 kg (6 640 lb).

- Propellant System-This subgroup consists of the propellant delivery system, tank pressurization (autogenous) and vent systems, umbilical hardware, and associated installation hardware. Individual SSME feed line diameters are 0.305 m (12 in) for both LO_2 and LH_2. The LO_2 feed lines are fed by a single main line of 0.704 m (27.7 in) diameter. The LH_2 feed lines are fed by two main lines, each 0.498 m (19.6 in) in diameter. Using the Shuttle Orbiter propellant system as a data base, and allowing for 20% mass reduction relative to current state-of-the-art, the unit mass of the propellant system components located in the aft body region is 934 kg (2 060 lb) per SSME. Using the ET tank propellant system as a data base, and allowing for a 20% mass reduction relative to current state-of-the-art, the mass of the propellant system components located forward of the aft body region is 3 329 kg (7 340 lb). Total propellant system mass is 10 803 kg (23 820 lb).

RCS Propulsion - A LO_2/LH_2 reaction control system provides for vehicle orientation assist during orbiter-booster separation, for vehicle orientation prior to entry, and for control during entry. Dry system mass is 2 494 kg (5 500 lb) and was estimated at 73% of total usable (nominal + reserve) RCS propellant mass.

OMS Propulsion - The orbital maneuver system consists of engines and accessories, and associated tank pressurization and propellant delivery and storage elements. A LO_2/LH_2 system was utilized. Dry system mass is 2 676 kg (5 900 lb) and was estimated at 14% of total usable (nominal + reserve) OMS propellant mass.

Prime Power - Power source elements consist of threee independent N_2H_4 powered APU's, auxiliary batteries, and alternators. The APU's provide for hydraulic power and for 50 kW (67 hp) (nominal) of electrical power. Total system dry mass including the reactant system, lube oil coolant system and exhaust system is estimated at 1 293 kg (2 850 lb).

Electrical Conversion and Distribution - This group consists of the power conversion, conditioning, and cabling elements for a 270 Vdc electrical system. Estimated system mass is 2 993 kg (6 600 lb).

Hydraulic Conversion and Distribution - This group consists of the hydraulic power supply and distribution equipment, including a water boiler temperature control system, for a 55 200 kPa (8 000 psi) hydraulic system. Estimated system mass including hydraulic fluid is 2 630 kg (5 800 lb).

Surface Controls - The actuation systems for the aerodynamic control surfaces are included in this group. The actuation system unit masses, based on modest improvements in systems design relative to Shuttle Oribter, are: elevons, 11.7 kg/m^2 (2.4 lb/ft^2); split rudders/speed brakes, 57.1 kg/m^2 (11.7 lb/ft^2); body flap, 12.2 kg/m^2 (2.5 lb/ft^2). Total surface controls mass is 4 308 kg (9 500 lb).

Avionics - The avionics group includes elements for guidance, navigation and control, tracking, instrumentation, and data processing and software. Total mass is 1 224 kg (2 700 lb) or 42% of Shuttle Orbiter avionics mass, based on use of advanced avionics, reduced functional requirements, and reduced power requirements.

Environmental Control - The environmental control group maintains a conditioned thermal environment for the avionics. Estimated system mass including closed loop fluids is 227 kg (500 lb).

Payload Provisions - This group consists of fixed scar items and removable provisions with unit masses of 0.30% and 0.85%, respectively, of maximum ascent payload mass, based on Shuttle Orbiter. Total mass is 2 608 kg (5 750 lb).

Margin - A margin allowance of approxiamtely 10% of subsystems dry mass exclusive of SSME's has been incorporated. Total mass is 16 009 kg (35 300 lb).

Nominal RCS Propellant - The estimated RCS delta-V budget is 33 m/s (107 ft/sec) and consists of the following:

Trim Burns - Ascent (Separation)	0.5 m/s (2 ft/sec)
Trim Burns - Post Ascent	1.5 m/s (5 ft/sec)
Trim Burns - 500 KM Orbit	17 m/s (55 ft/sec)
Trim Burns - Entry	14 m/s (45 ft/sec)

Based on specific impulse 420 sec for the LO_2/LH_2 thrusters, and consideration of vehicle sequential mass, the total nominal propellant mass is 2 844 kg (6 270 lb).

Nominal OMS Propellant - The OMS delta-V budget is 240 m/s (788 ft/sec) and consists of the following:

Circularization into 500 KM Orbit from 110 x 500 KM Launch Orbit	112 m/s (367 ft/sec)

Trim Burns in 500 KM Orbit	11 m/s (37 ft/sec)
Deorbit from 500 KM Orbit	117 m/s (384 ft/sec)

Based on a specific impulse of 476 sec for the LO_2/LH_2 OMS engines, and consideration of vehicle sequential mass, the total nominal propellant mass is 17 859 kg (39 380 lb).

<u>Systems Inflight Losses</u> - This group consists of the nominal usage of N_2H_4 for hydraulic and electric power and of H_20 for hydraulic cooling. Total estimated mass is 957 kg (2 110 lb).

<u>Ascent Propulsion Inflight Losses</u> - All ascent propulsion liquid residuals are jettisonable and are included in this group. Total mass is 8 980 kg (19 800 lb). Pertinent line size data is included under ascent propulsion-propellant system.

- o Propellant in Engines at Cutoff-Trapped propellant mass is 236 kg (520 lb) per SSME. Total mass is 1 887 kg (4 160 lb).

- o Propellant in Feed Lines at Cutoff-An average 4.0 m (13 ft) of LO_2 line and 4.9 m (16 ft) of LH_2 line is located between the main delivery line(s) and each SSME. These lines are full at engine cutoff. Approximately 4.9 m (16 ft) of main delivery line(s) for both LO_2 and LH_2 is located between the junction with the engine feed lines and junction with the transfer line(s). These lines are full at engine cutoff as are the 0.6 m (2 ft) long transfer lines. In addition, at engine cutoff under nominal conditions, the main delivery line(s) upstream of the transfer line junction contain 3.0 m (10 ft) of fluid, exclusive of bias fuel. The LH_2 and LO_2 unit masses are 798 kg (1 760 lb) and 50 kg (110 lb), respectively, per SSME. Total mass is 6 785 kg (14 960 lb).

- o Bias Fuel-A single-engine reference bias fuel allowance of 0.50% was used. For multiple engines, this allowance was factored by the inverse of th square root of the number of engines. The resulting LH_2 bias for the eight SSME's is 0.18%. Total LH_2 mass is 308 kg (680 lb).

<u>Reserves</u> - This group consists of reserve allowances for the RCS, OMS, and systems. (The flight performance reserve for ascent is 0.85% of total ideal velocity change and is included in ascent propellant mass.) Reserve allowances for RCS and OMS are 20% and

5%, respectively, of nominal propellant usage. The systems reserve allowances are estimated at 322 kg (710 lb). Total reserves mass is 2 059 kg (4 540 lb).

Residuals - This group consists of the residual fluids and gases in the ascent propulsion, RCS, OMS, and systems at time of landing. Total mass is 3 401 kg (7 500 lb).

- Ascent Propulsion Residual-Only main tank pressurants are onboard at landing. The pressurant masses are based on maximum ullage pressures of 159 kPa (23 psia) and mean temperatures for the GO_2 and GH_2 of 264 K (475°R) and 189 K (340°R), respectively. The masses of the GO_2 and GH_2 are 2 200 kg (4 850 lb) and 517 kg (1 140 lb), respectively. Total pressurant mass is 2 717 kg (5 990 lb).

- RCS, OMS, and Systems Residuals-Residual allowances for both RCS and OMS were estimated at 2% of usable propellant (nominal + reserve). The systems residuals were estimated at 209 kg (460 lb). Total mass is 685 kg (1 510 lb).

2.2.2.4 Booster Configuration Description

The configuration of the HLLV Booster is presented in figure 2.2.2-5 with the overall geometry noted. The booster is unmanned. Ascent thrust is provided by thirteen LO_2/LCH_4 engines, each having a sea level thrust of 3.86×10^6N (867 700 lbf). The engines employ an expander bleed cycle and utilize hydrogen for cooling and turbopump drive. Of the total propellant load, approximately 19.4% is crossfed to the SSME's in the orbiter, prior to booster-orbiter staging at a velocity of approximately 2 195 m/s (7 200 ft/sec). The airframe structure consists of unpressurized structures and integral main propellant tanks. The unpressurized structures are aluminum/titanium/composites design (composites used internally only) and are protected from entry heating by increasing the thickness of the outer surface material to provide heat-sink capability. Main tankage structures ar 6AL-4V(ELI) titanium sandwich for the LH_2 tank and 2219-T87 aluminum for the LO_2 and LCH_4 tanks. The tank sidewalls are protected from entry heating by increasing the skin thickness to provide heat-sink capability. Flyback thrust is provided by thirteen JTF22A-Mod III engines, each having a sea level static thrust of approximately 133 000 N (30 000 lbf). Flyback fuel is RP-1. A summary mass and balance statement is presented in Table 2.2.2-2.

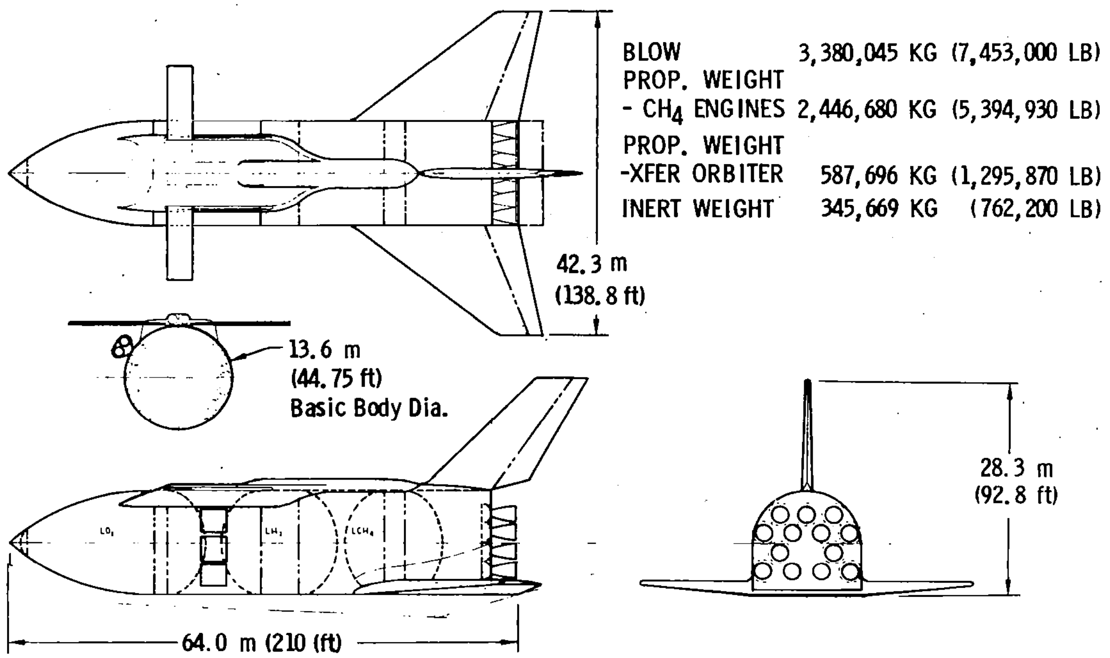


Figure 2.2.2-5. - HLLV Booster Configuration, Normal Growth Technology Baseline

Each of the items in the summary mass and balance statement, exclusive of payload and ascent propellant, is discussed in the following paragraphs including definition of rationale for mass estimates.

Wing–The wing is an aluminum/titanium/composite design with a mass of 31 474 kg (69 400 lb) and a unit mass of 43.9 kg/m^2 (9.00 lb/ft^2) based on its reference area of 716 m^2 (7 706 ft^2). The design features an aluminum box and intermediate sections and titanium leading edges, trailing edges, and elevons, all of which incorporate extensive use of composite internal members. The wing mass is estimated at 85% of that of an aluminum design wing sized for a 2.5g subsonic maneuver at the start flyback wing loading (reference wing + canard) of 362 kg/m^2 (74.2 lb/ft^2), a 406 K (270^{o}F) surface temperature during the maneuver, and a 1,000 hour design accumulated time at surface temperatures above 394 K (250^{o}F). The wing mass includes the box-body-carry-through section and main gear installation provisions. A constant t/c of 10% was used. The ratio of elevon area to wing reference area is 12.0%.

TABLE 2.2.2-2. HLLV Booster Summary Mass and Balance Statement Normal Growth Technology Baseline

GROUP/ITEM	MASS (kg)	XCG (m)	WEIGHT (lb)	XCG (in.)
WING	31,474	57.15	69,400	2250
VERTICAL TAIL	4,444	62.23	9,800	2450
CANARD	2,585	22.86	5,700	900
BODY	77,347	39.45	170,550	1553
INDUCED ENVIRONMENTAL PROTECTION	12,562	52.04	27,700	2049
LANDING AND AUXILIARY SYSTEMS	9,297	46.53	20,500	1832
PROPULSION–ASCENT	63,447	58.01	139,900	2284
ROCKET ENGINES	39,410	60.45	86,900	2380
ENGINE ACCESSORIES	5,125	60.45	11,300	2380
PROPELLANT SYSTEM	18,912	52.30	41,700	2059
PROPULSION–RCS	861	40.64	1,900	1600
PROPULSION–FLYBACK	23,492	25.40	51,800	1000
PRIME POWER	1,451	40.64	3,200	1600
ELECTRICAL CONVERSION & DISTR.	3,605	40.64	7,950	1600
HYDRAULIC CONVERSION & DISTR.	3,288	55.88	7,250	2200
SURFACE CONTROLS	4,694	60.96	10,350	2400
AVIONICS	1,225	40.64	2,700	1600
ENVIRONMENTAL CONTROL	227	40.64	500	1600
MARGIN	22,359	48.56	49,300	1912
(DRY CONDITION)	(262,358)	(47.32)	(578,500)	(1863)
RESIDUALS–SYSTEMS/RCS/FLYBACK/ASCENT PROPULSION	7,946	25.40	17,520	1000
RESERVES–SYSTEMS/RCS/FLYBACK	9,850	26.29	21,720	1035
INFLIGHT LOSSES–SYSTEMS	889	44.91	1,960	1768
INFLIGHT LOSSES–ASCENT PROPULSION	29,796	57.33	65,700	2257
PROPELLANT–NOMINAL RCS	998	40.64	2,200	1600
FUEL–NO HEADWIND FLYBACK	33,832	25.40	74,600	1000
(INERT CONDITION)	(345,669)	(44.91)	(762,200)	(1768)
BOOST PROPELLANT	3,034,376	22.10	6,690,800	870
LO_2	2,385,805	15.75	5,260,700	620
LH_2	110,839	32.51	244,400	1280
LCH_4	537,732	48.16	1,185,700	1896
(LIFT OFF CONDITION)	(3,380,045)	(24.43)	(7,453,000)	(962)

$$\text{MASS FRACTION} = \frac{\textbf{ASCENT PROPELLANT MASS}}{\text{LIFTOFF MASS}} = 0.898$$

Vertical Tail–The vertical tail, which incorporates a rudder/speed brake, is an aluminum/titanium/composite design with a unit mass of 31.6 kg/m^2 (6.46 lb/ft^2) of exposed planform area. The tail unit mass is 85% of the unit mass of the aluminum design tail on the space Shuttle. The ratio of rudder/speed brake area to tail area is 29%. Tail mass is 4 444 kg (9 800 lb).

Canard–The subsonic trim canard group consists of an aerosurface, hinge/attachment fittings and deployment/latch mechanisms. The aerosurface is a composite design with a unit mass of 20.5 kg/m^2 (4.2 lb/ft^2) based on its exposed planform area. The aerosurface unit mass is estimated at 70% of that of an aluminum design. Fittings and mechanisms unit mass is estimated at 24.4 kg/m^2 (5.1 lb/ft^2) of aerosurface area. The ratio of canard exposed planform area to wing reference area is 8.0%. Total mass is 2 585 kg (5 700 lb).

Body–The body group consists of the following: nose cap, LO2 tank, forward intertank section, LH_2 tank, aft intertank section, LCH_4 tank, aft body section, thrust structure, body flap, and fairing structures. Total body mass is 77 347 kg (170 550 lb).

- Nose Cap–The nose cap is a semimonocoque shell structure of titanium/composite design with a unit mass of 13.2 kg/m^2 (2.7 lb/ft^2). This unit mass is estimated at 85% of that of an aluminum design nose cap. Nose cap mass is 204 kg (450 lb).

- LO_2 Tank–The tank containing the liquid oxygen is an all-welded 2219-T87 aluminum pressure vessel. The tank is designed by room temperature proof test conditions corresponding to 3g boost onset with maximum ullage pressure of 159 kpa (23 psia). To satisfy a 500-mission requirement with minimal mass impact, the tank is designed for 200 cycles to failure and, following the initial proof test, is reproofed every 100 missions. The tank mass, including slosh baffles, is 17 415 kg (38 400 lb), 0.73% of the mass of the liquid oxygen.

- Forward Intertank Section–This intertank, which supports the flyback propulsion system, consists of a semimonocoque shell structure and nose gear installation provisions. The shell structure is of aluminum/composite design with a unit mass of 13.2 kg/m^2 (2.7 lb/ft^2). This unit mass is 85% of that of an aluminum design shell structure. The nose gear installation provisions consist of the wheel well, door and mass of 0.28% of design landing weight. This unit mass is 80% of the unit mass of the aluminum design installation provisions of the Shuttle Orbiter. Forward intertank section total mass is 7 392 kg (16 300 lb).

- LH_2 Tank—The tank containing the liquid hydrogen is an all-welded 6AL-4V(ELI) titanium sandwich pressure vessel with a ring stiffened sidewall. The tank is designed by cryo temperature proof test conditions corresponding to 3g boost onset with maximum ullage pressure of 159 kpa (23 psia). To satisfy a 500 mission requirement with minimal weight impact, the tank is designed for 200 cycles to failure and, following initial proof test, is reproofed every 100 missions. The inner face of the sandwich is sized to carry 100% of proof test pressure. The outer face is minimum gage at 0.030 cm (0.012 in), including the sidewall regions to which fairing structures are attached. Tank mass is 7 710 kg (17 000 lb), 6.95% of the mass of the liquid hydrogen.

- Aft Intertank Section—This intertank is a semimonocoque shell structure of aluminum/composite design. It has a unit mass of 14.6 kg/m^2 (3.0 lb/ft^2). This unit mass is estimated at 85% of that of an aluminum design shell structure. The aft intertank mass is 6 667 kg (14 700 lb).

- LCH_4 Tank—The tank containing the liquid methane is an all-welded 2219-T87 aluminum pressure vessel with a stiffened sidewall. The tank is designed by room temperature proof test conditions corresponding to 3g boost onset with maximum ullage pressure of 159 kPa (23 psia). To satisfy a 500 mission requirement with minimal mass impact, the tank is designed for 200 cycles to failure and, following the initial proof test, is reproofed every 100 missions. The tank mass, including slosh baffles, is 7 755 kg (17 100 lb), 1.44% of the mass of the liquid methane.

- Aft Body Section—This section consists of a semimonocoque shell structure extending from the LCH_4 tank to the engine support plane, a 1.3 m (4.2 ft) long fairing structure located aft of the engine support plane, and a base heat shield support structure installation located at the engine support plane. The shell structure and fairing structure are of aluminum/composite design. The heat shield support structure is of composite design. Unit masses of the foregoing structures are 29.8 kg/m^2 (6.1 lb/ft^2) 14.6 kg/m^2 (3.0 lb/ft^2), and 12.2 kg/m^2 (2.5 lb/ft^2), respectively. These unit masses are estimated at 85%, 85%, and 70%, respectively, of those of aluminum design structures. Aft body section total weight is 14 875 kg (32 600 lb).

- Thrust Structure—The thrust structure is a beam system of composite design which transmits thrust loads from the thirteen LO_2/LCH_4 engines to the aft body section. Maximum vacuum thrust of the thirteen engines is 55.2 x 10^6N (12.415 x 10^6 lbf). The thrust structure has a mass of 9 614 kg (21 200 lb) which is 70% of that of an aluminum design. Its unit mass is 0.00017 kg/N (0.00171 lb/lbf).

- Body Flap—The constant chord body flap provides for pitch trim control and thermally shields the main engines during entry. The flap has a translating trailing edge section which, when extended, increases flap area by 75%. It is of titanium/composite design with a unit mass of 19.3 kg/m^2 (3.95 lb/ft^2) of total extended area. This unit mass is 85% of the unit mass of the aluminum design body flap in the Shuttle Orbiter, adjusted by a 1.4 factor for the translation feature. In addition to the translation mechanism, the body flap mass includes the hinge line support fittings/closeout provisions. Total body flap mass is 2 721 kg (6 000 lb).

- Fairing Structures—This group consists of a wing-to-body fairing, a canard fairing, and a LO_2 main delivery line fairing. The fairings are of aluminum/titanium/ composite design and include approximately 260 m^2 (2 800 ft^2) of fixed fairing at 7.3 kg/m^2 (1.5 lb/ft^2) and approximately 121 m^2 (1 300 ft^2) of removable fairing at 9.8 kg/m^2 (2.0 lb/ft^2). The unit masses are estimated at 85% of those of aluminum design fairings. Total fairing structure mass is 3 084 kg (6 800 lb).

Induced Environmental Protection—This group consists of the external TPS system plus the internal provisions for thermal control and purge/vent/drain. The external TPS system consists of the heat sink additions required to maintain the airframe out skin within acceptable temperature limits, and an advanced RSI base heat shield. Based on a booster-orbiter staging velocity of approximately 2 195 m/s (7 200 ft/sec), a 4g entry, and an entry planform loading of 269 kg/m^2 (55.1 lb/ft^2) with flap retracted, the heat sink addition unit masses based on heat sink booster studies are: wing including body flap, 10.0 kg/m^2 (2.05 lb/ft^2) of wing reference area; body, 1.5 kg/m^2 (0.30 lb/ft^2) of body planform area forward of the wing reference area; vertical tail, 11.3 kg/m^2 (2.32 lb/ft^2) of tail side planform area. In addition, for the body, the unit mass of the RSI base heat shield is 5.2 kg/m^2 (1.07 lb/ft^2) of body base area. The internal provisions have an estimated mass of 1 814 kg (4 000 lb). The induced environmental protection total mass is 12 562 kg (27 700 lb).

Landing and Auxiliary Systems–In addition to landing gear, this group includes auxiliary systems for orbiter-booster separation. The landing gear design incorporates the use of high strength tires, servo-design shock struts with extensive use of boron-aluminum, and 55 200 kpa (8 000 psi) hydraulic operated mechanisms. The landing gear unit mass is estimated at 2.8% of design landing mass based on modified Shuttle Orbiter data. Separation systems mass is estimated at 1 451 kg (3 200 lb). Total mass is 9 297 kg (20 500 lb).

Ascent Propulsion–The ascent propulsion group consists of rocket engines, engine accessories, and the propellant system. Total mass is 63 447 kg (139 000 lb).

- Rocket Engines–Ascent thrust is provided by thirteen LO_2/LCH_4 engines. Total liftoff thrust is 50.2×10^6N (11 280 100 lbf). Pertinent engine characteristics are presented in Section 2.1.2. Total dry engine mass is 39 410 kg (86 900 lb).

- Engine Accessories–Accessories for each of the thirteen LO_2/LCH_4 engines consist of a gimbal system, hydraulic supply, powerhead heat shield, and nitrogen purge provisions. Based on the use of 55 200 kpa (8 000 psi) hydraulic components and modest improvements in heat shield design, the engine accessories mass for a gimballed LO_2/LCH_4 engine is estimated at 13% of dry engine mass based on modified accessories mass data for a standard SSME. Total accessories mass is 5 125 kg (11 300 lb).

- Propellant System–This subgroup consists of the propellant delivery system, tank pressurization (autogenous) and vent provisions, umbilical hardware, and associated installation hardware. Individual LO_2/LCH_4 engine feed lines diameters are 0.462m (18.2 inch) for LO_2, 0.290m (11.4 inch) for LCH_4, and 0.147m (5.8 inch) for LH_2. The lines which cross feed LO_2 and LH_2 to the eight SSME's in the orbiter consist of two LO_2 lines of 0.498m (19.6 inch) diameter (with common junction at interface) and two LH_2 lines of 0.498m (19.6 inch) diameter. The LO_2 feed lines and transfer lines are fed by two main lines, each 1.125m (44.3 inch) in diameter. The LH_2 feed lines and transfer lines are fed by a single main line of 0.932m (36.7 inch) diameter. The LCH_4 lines to each engine are direct. Using the Shuttle Orbiter propellant system as a data base, and allowing for a 20% mass reduction relative to current state-of-the-art, the unit mass of the propellant system components located in the aft body region is 1 161 kg (2 560 lb) per LO_2/LCH_4 engine. Using the ET tank propellant system as a data base, and allowing for a 20% mass reduction relative to current state-of-the-art, the mass of the propellant system components located forward of the aft body region is 3 810 kg (8 400 lb). Total propellant system mass if 18 912 kg (41 700 lb).

RCS Propulsion–A LO_2/LH_2 reaction control system provides for vehicle orientation assist during orbiter-booster separation, for vehicle orientation prior to entry, and for control during entry. Dry system mass is 861 kg (1 900 lb) and was estimated at 73% of total usuable (nominal+reserve) RCS propellant mass, based on previous studies.

Flyback Propulsion–The flyback propulsion group consists of the airbreathing engines, engine accessories, engine oil, fuel system, fuel tank, and engine installation nacelles, deployment structures, mechanisms, and doors. Flyback thrust is provided by thirteen JTF22A-Mod III turbojet engines, each having a sea level static thrust of approximately 133 000N (30 000 lbf). Flyback fuel is RP-1. The mass of the turbojet engines is 13 678 kg (30 160 lb). Total dry system mass, including engine oil, is 23 492 kg (51 800 lb).

Prime Power–Power source elements consist of three independent N_2H_4 powered APU's, auxiliary batteries, and alternators. The APU's provide for hydraulic power and for 50 KW (67 hp) of electrical power. Total system dry mass including dry mass including the reactant system, lube oil coolant system and exhaust system is estaimated at 1 451 kg (3 200 lb).

Electrical Conversion and Distribution–This group consists of the power conversion, conditioning, and cabling elements for a 270 Vdc electrical system. Estimated system mass is 3 605 kg (7 950 lb).

Hydraulic Conversion and Distribution–This group consists of the hycraulic power supply and distribution equipment, including a water boiler temperature control system, for a 55 200 kPa (8 000 psi) hydraulic system. Estimated system mass including hydraulic fluid is 3 288 kg (7 250 lb).

Surface Controls—The actuation systems for the aerodynamic control surfaces are included in this group. The actuation system unit masses, based on modest improvements in systems design relative to Shuttle Orbiter, are: elevons, 11.7 kg/m^2 (2.4 lb/ft^2); split rudders/speed brakes, 57.1 kg/m^2 (11.7 lb/ft^2); body flap, 12.1 kb/m^2 (21.5 lb/ft^2). Total surface controls mass is 4 694 kg (10 350 lb).

Avionics—The avionics group includes elements for guidance, navigation and control, tracking, instrumentation, and data processing and software. Total mass is 1225 kg (2 700 lb) or 42% of Shuttle Orbiter avionics mass, based on use of advanced avionics, reduced functional requirements, and reduced power requirements.

Environmental Control—The environmental control group maintains a conditioned thermal environment for the avionics. Estimated system mass including closed loop fluids is 227 kg (500 lb).

Margin—A margin allowance of approximately 10% of subsystems dry mass exclusive of JTF22A-Mod III airbreather engines has been incorporated. Total mass is 22 359 kg (49 300 lb).

Nominal RCS Propellant—The estimated RCS delta-V budget is 12.5 m/s 42 ft/sec) and consists of the following:

Trim Burns - Ascent (Separation)	2.0 m/s (7 ft/sec)
Trim Burns - Post Ascent	1.5 m/s (5 ft/sec)
Trim Burns - Entry	9 m/s (30ft/sec)

Based on a specific impulse of 420 sec for the LO_2/LH_2 thrusters, and consideration of vehicle sequential mass, the total nominal propellant mass is 998 kg (2 200 lb).

Nominal (No Headwind) Flyback Fuel—Booster-orbiter staging occurs at a velocity of approximately 2 195 m/s (7 200 ft/sec) following an easterly launch. Subsequent to the transition maneuver at end of entry, the booster performs a long glide prior to starting its powered flyback of approximately 212 nm range. Consideration of 3 km (10 000 ft) cruise ceiling, a 1.5 km (5 000 ft) minimum cruise altitude, a maximum L/D of approximately 5, and the use of thirteen JTF 22A-Mod III engines in a deployable installation results in a total nominal (no headwind) flyback fuel mass of 33 832 kg (74 600 lb).

Systems Inflight Losses–This group consists of the nominal usage of N_2H_4 for hydraulic and electric power and of H_20 for hydraulic cooling. Total estimated mass is 889 kg (1 960 lb).

Ascent Propulsion Inflight Losses–All ascent propulsion liquid residuals are jettisonable and are included in this group. Total mass is 19 796 kg (65 700 lb). Pertinent line size data is included under ascent propulsion-propellant system.

- Propellant in Engines at Cutoff–Trapped propellant mass is estimated at 204 kg (450 lb) per LO_2/LCH_4 engine. Total mass is 2 653 kg (5 850 lb).

- LCH_4 in Engine Feed Lines at Cutoff–The LCH_4 lines to each LO_2/LCH_4 engine are direct tank-to-engine lines. The lines average 5.2m (17 ft.) in length and contain an average 3.0m (10 ft.) of LCH_4 at nominal engine cutoff, exclusive of bias fuel. The LCH_4 unit mass is 86 kg (190 lb) per engine. Total mass is 1 120 kg (2 470 lb).

- LO_2 and LH_2 in Feed Lines at Cutoff–An average 6.1m (20 ft.) of both LO_2 line and LH_2 line are located between the main delivery line(s) and each engine. These lines are full at engine cutoff. Approximately 6.7m (22 ft) of LO_2 transfer line and 6.1m (20 ft) of LH_2 transfer line are located between the main delivery line(s) and the transfer interface. These lines are full at engine cutoff. In addition, at engine cutoff under nominal conditions, the main delivery line(s) upstream of the common junction with the engine feed lines and transfer line(s) contains 3.0m (10 ft) of fluid exclusive of bias fuel. The LO_2 unit masses are 1 567 kg (3 456 lb) per booster engine plus 544 kg (1 200 lb) per SSME in the orbiter. The LH_2 unit masses are 10 kg (22 lb) per booster engine plus 32 kg (70 lb) per SSME in the orbiter. Total mass is 25 116 kg (55 380 lb).

- Bias Fuel–A single-engine reference bias fuel allowance of 0.50% was used. For multiple engines, this allowance was factored by the inverse of the square root of the number of engines. The resulting LCH_4 bias for the thirteen $L0_2/LCH_4$ booster engines and the LH_2 bias for the eight SSME's in the orbiter are 0.14% and 0.18%, respectively. The LCH_4 bias is 757 kg (1 670 lb) and the LH_2 bias is 150 kg (330 lb) total mass is 907 kg (2 000 lb).

Reserves–This group consists of reserve allowance for the RCS, flyback propulsion, and systems. The reserve allowance for RCS is 20% of nominal propellant usage. Reserve allowances for flyback are for headwinds and go-around. The headwind reserve is estimated at 11% of the nominal (no headwind) fuel. The go-around reserve is estimated at 299 kg (660 lb). Total reserves mass is 9850 kg (21 720 lb).

Residuals–This group consists of the residual fluids and gases in the ascent propulsion, RCS, flyback propulsion, and systems at time of landing. Total mass is 7 946 kg (17 520 lb).

- o Ascent Propulsion Residuals–Only main tank pressurants are onboard at landing. The pressurant masses are based on maximum ullage pressures of 159 kPa (23 psia) and mean temperatures for the GO_2, GH_2, and GCH_4 of 264 K (475^{o}R), 189 K (340^{o}R), and 222 K (400^{o}R), respectively. The masses of the GO_2, GH_2, and GCH_4 are 5 125 kg (11 300 lb), 336 kg (740 lb), and 1 841 kg (4 060 lb), respectively. Total pressurant mass is 7 302 kg (16 100 lb).

- o RCS, Flyback Propulsion, and Systems Residuals–Resident allowances for RCS and for flyback propulsion were estimated at 2% of usable propellant (nominal + reserve) and 1% of usable fuel (nominal + reserve), respectively. The systems residuals were estimated at 186 kg (410 lb). Total mass is 644 kg (1 420 lb).

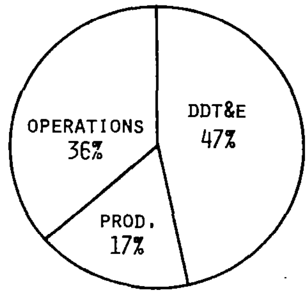
2.2.2.5 HLLV Life Cycle Costs

The baseline heavy lift launch vehicle (HLLV) life cycle costs (LCC) are separated into three phases:

(1) DDT&E
(2) Production
(3) Operations

Figure 2.2.2-6 presents a breakdown of the HLLV LCC by phase and subphase. Each of three phases will be discussed in the following paragraphs. All costs are in constant year 1977 dollars except where noted.

	$ MILLIONS
TOTAL PROGRAM	13,346.41
DDT&E	6,243.87
PROGRAM MANAGEMENT	172.63
ENGINEERING	2,411.08
MANUFACTURING	2,825.81
TEST	834.35
PRODUCTION	2,252.21
PROGRAM MANAGEMENT	183.17
SYSTEM ENGINEERING	51.92
MANUFACTURING	2,017.13
OPERATIONS	4,850.33
OPERATIONS SUPPORT	2,968.43
LAUNCH SUPPORT	1,881.90



- COST/FLT = $\$7.96 \times 10^6$
- COST = $ 35.12/Kg

Figure 2.2.2-6. - HLLV Cost Summary

DDT&E Phase

The total DDT&E phase cost for the baseline HLLV is $6243.9 million. The breakdown between orbiter and booster is:

- o Orbiter: $2915.5 million
- o Booster: $3328.4 million

DDT&E phase costs are summarized in tables 2.2.2-3 and 2.2.2-4.

Engineering—The major cost element in the engineering subphase is the vehicle D&D at $902.4 million for the Orbiter and $1357.4 million for the Booster. The Orbiter has a low D&D cost for propulsion because of its use of existing SSME's, with only the addition of an extendible nozzle at $50 million D&D cost. Included in the Auxilary propulsion D&D is $45.8 million for development of a LO_2/LH_2 RCS thruster. The booster D&D is subsequentially larger due to the additional cost of the LO_2/LCH_4 main engine at $568.7 million, and the flyback airbreather engine at $103.7 million.

TABLE 2.2.2-3. BASELINE HLLV ORBITER DDT&E COSTS

DDT&E	2915.5			
PROGRAM MANAGEMENT		91.3		
ENGINEERING		993.0		
SYSTEMS ENGINEERING & INTEGRATION (SE&I)			72.7[1]	
SOFTWARE ENGINEERING			17.9	
DESIGN & DEVELOPMENT			902.4	
STRUCTURE				260.1
TPS				108.9
LANDING & AUX. SYSTEMS				105.7
ASCENT PROPULSION				94.0
AUX. PROPULSION				63.4
PRIME POWER				2.8
ELECT. CONV. & DIST.				20.9
HYDRAULIC CONVE. & DIST.				16.1
SURFACE CONTROLS				8.1
AVIONICS				47.4
ECS				2.7
PERSONNEL PROVISION				0.0
GSE				172.3
MANUFACTURING		1353.1		
DEVELOPMENTAL TOOLING & S.T.E.			152.6	
TEST HDWE & SPARES			1200.5	
TEST		478.1		
SYSTEMS TEST OPERATIONS			414.4	
FLIGHT TEST OEPRATIONS			63.66[2]	

1 INCLUDES $23.4 M FOR INTEGRATED VEHICLE SE&I

2 FLIGHT TEST OPERATIONS FOR INTEGRATED VEHICLE

TABLE 2.2.2-4. BASELINE HLLV BOOSTER DDT&E COSTS

DDT&E	3328.4			
PROGRAM MANAGEMENT		81.3		
ENGINEERING		1418.1		
SYSTEMS ENG. & INT.			44.4	
SOFTWARE ENGINEERING			16.2	
DESIGN & DEVELOPMENT			1357.4	
STRUCTURE				285.3
TPS				17.0
LANDING & AUX. SYSTEMS				83.1
ASCENT PROPULSION				583.2
AUX. PROPULSION				135.3
PRIME POWER				3.1
ELECT. CONV. & DIST.				23.8
HYDRAULIC CONV. & DIST.				19.1
SURFACE CONTROLS				8.7
AVIONICS				47.4
ECS				2.7
PERSONNEL PROVISION				0.0
GSE				148.7
MANUFACTURING		1472.7		
DEVELOPMENTAL TOOLING & S.T.E.			166.1	
TEST HDWE & SPARES			1306.6	
TEST		356.3		
SYSTEMS TEST OEPRATIONS			356.3	
FLIGHT TEST OPERATIONS			0.0	

The following HLLV systems have reduced D&D costs because of technology or components inherited from the Space Shuttle:

- o Avionics
- o Power system
- o Hydraulic system and surface controls
- o Thermal protection system on Orbiter

Manufacturing–The major costs in the manufacturing subphase are those of the test hardware. They consist of:

- o Structural test article - $174.7 million
- o Propulsion/Dynamic test vehicle - $1156.7
- o Flight test vehicle - $771.1

The propulsion/dynamic test vehicle cost includes $385.6 million for refurbishment between the dynamic and main propulsion tests. Both test vehicles are refurbished after the test program and included in the operational fleet.

Test–The test subphase has a cost of $834.4 M, with a breakdown of:

- o $414.4 million for Orbiter system tests
- o $356.3 million for booster system tests
- o $63.7 million for flight tests of the integrated vehicle

Production Phase

A breakdown of the production phase cost, totalling $2252.2 million is shown in table 2.2.2-5. The major cost elements are production tooling and flight hardware.

TABLE 2.2.2-5. PRODUCTION PHASE COSTS

PROGRAM ELEMENT	COST ($ MILLION)		
PRODUCTION PHASE	2252.2		
PROGRAM MANAGEMENT		183.2	
SUSTAINING ENGINEERING		51.9	
MANUFACTURING		2017.1	
PROD. TOOLING & S.T.E			590.8
FLT. HARDWARE & SPARES			1426.3

The production tooling, at a cost of $590.8 million, includes the tooling required to produce the initial HLLV fleet and the spares for the operational phase.

In order to meet the mission requirements, a fleet of 3 HLLV's is required. The flight hardware cost, including initial spares, is $1426.3 million. Of this amount, $433.5 million is the cost of refurbishing the two test program vehicles for use in the operational fleet. The TFU costs used in pricing the production hardware are:

o	Orbiter airframe	$302.91 million
o	Booster airframe	$323.12 million
o	LO_2/LCH_4 Main engine	$9.26 million
o	SSME	$15.06 million
o	GSE	$176.66 million
o	Complete vehicle	$867.06 million

Operations Phase

The cost of the HLLV operations phase is $4850.3 million. Table 2.2.2-6 shows the distribution of costs between the subphases.

TABLE 2.2.2-6. OPERATIONS PHASE COSTS

PROGRAM ELEMENT	COST ($ MILLION)		
OPERATIONS PHASE	4850.3		
OPERATIONS SUPPORT		2968.4	
PROGRAM SUPPORT			442.3
SPARES PROCUREMENT			2526.1
LAUNCH SUPPORT		1181.9	
OPERATIONS			1378.8
PROPELLANT			503.1

Analysis of HLLV operations was based on a mission model of 609 flights spanning 15 years. The fleet size was set by a peak flight rate of 74 per year. The groundrules used in determining the fleet size and the operational spares criteria are the same as those of the SSTO. A turnaround time of 133 hours was used for the operations analysis. This estimate is based on turnaround timelines for HLLV's developed in the "Systems Concepts for STS - Derived HLLV's Study" and in the "Solar Power Satellite Systems Definition Study" references 16 and 19.

The most expensive element in the operations phase is spares procurement at $2968.4 million. Of this amount, approximately $1000 million is the cost of SSME refurbishment and replacement. The remainder is attributable to per flight spares and major overhauls of the other vehicle systems.

2.2.3 Priority Cargo Orbit Transfer Vehicle (POTV)

2.2.3.1 Design Requirements and Configuration Issues

The primary POTV groundrules and guidelines are summarized as follows:

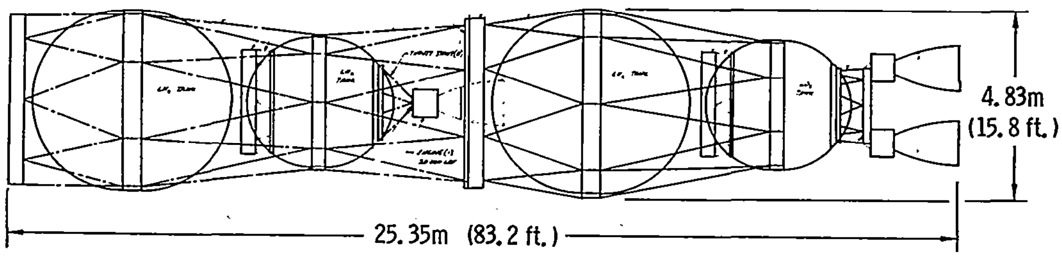
- o Space based @ 500 km
- o GEO destination–equatorial
- o 75% return payload
- o Trip time-T/W suitable for manned mission
- o 50 mission vehicle life/10 mission engine life

These requirements are consistent with a 30 day manned sortie to visit and repair satellites at GEO, however the propellant requirements for additional maneuvers at GEO are included in payload as are any boil off losses. The payload size (12 400 kg) for this vehicle was selected in the combined optimization as discussed in section 2.2.1.2.

The key design issue for this vehicle was the advantages accrued by space basing. A key element of the space based scenario is the space base. Although undefined, it was counted on to provide key services such as assuming the burden for payload handling and the vehicle interface as well as propellant conditioning, storage and transfer. The OTV is delivered empty to the space base by the HLLV. It is subsequently fueled and serviced on orbit until ready for overhaul when it is returned to earth. Since the optimum thrust-to-weight for this class of vehicle is near .2 g, it is designed to a far less severe environment than a ground based version.

Since the propellant tanks are pressure stabilized thin wall tanks, the propellant temperature was assumed to be reduced by the space base such that the propellant vapor pressures would be 10 psi. It was also assumed that the space base would maintain tank pressure between missions so that the tanks would never see a full pressure cycle. Zero NPSH boost pumps were also employed to reduce the tank structural design criteria. As a result, the tanks are minimum gauge over almost their entire surface. Since axial loads are low, a composite strut intertie structural system is employed. These features result in a very high structural efficiency. Figure 2.2.3-1 illustrates the single vs. common two stage trade which provided the rationale for a single stage concept selection. The high ' allows the vehicle to be offloaded to handle lower payload missions while maintaining high performance.

Figure 2.2.3-2 illustrates the vehicle concept and its main features.



	TWO STAGE	SINGLE STAGE
o WEIGHTS		
GROSS WEIGHT-INCL.10K Kg,75%F RETURN P/L	84,807 KG (187,000 LB)	87,846 KG (193,700 LB)
MAINSTAGE PROP. WEIGHT	68,027 KG (150,000 LB)	72,562 KG (160,000 LB)
STAGE MASS FRACTION	0.915 (Stg. 1) 0.903 (Stg. 2)	0.932
o ENGINES	3	2
o AVIONICS	2	1
o OPERATIONAL COMPLEXITY		√

Figure 2.2.3-1. - POTV, Single vs. Two Stage Trade Summary

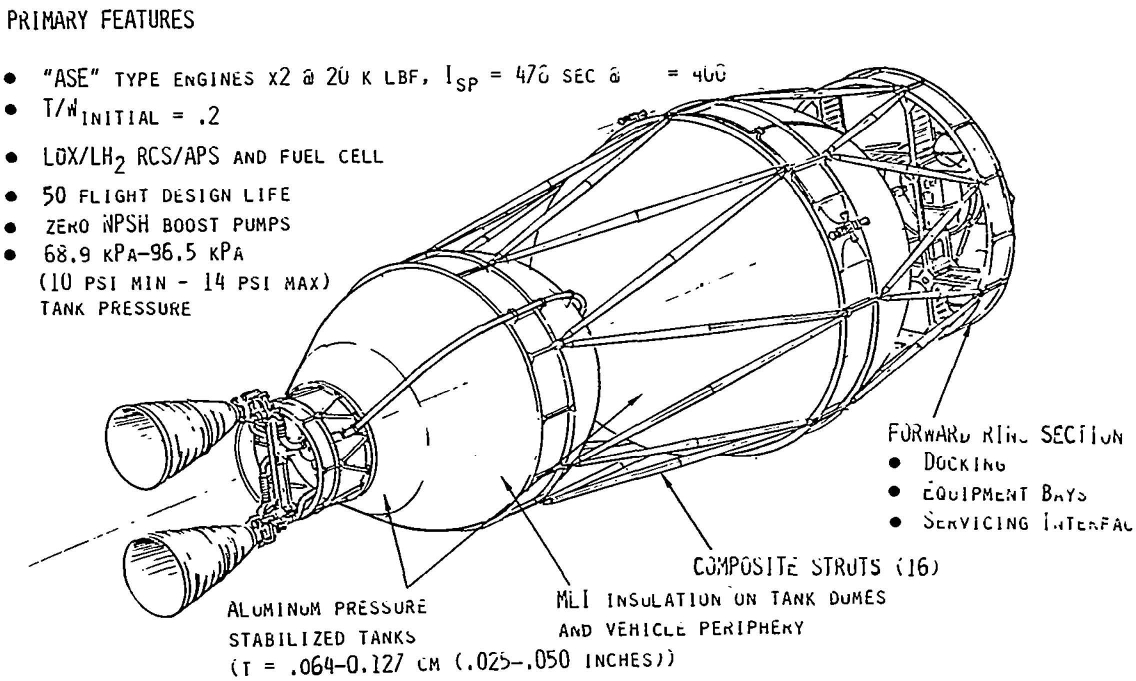


Figure 2.2.3-2. - POTV Design Concept

2.2.3.2 Performance and Sizing

Orbit transfer vehicle performance requirements were approximated by impulse maneuver delta-V's with additions for finite burn losses, phasing maneuvers, and course corrections. The impulse delta-V's assumed 2.5° of plane change for the LEO burns and 26.0° of plane change for the GEO burns. Circular orbit velocities at LEO (500 km) and GEO were computed at 7613 m/s (24 977 ft/sec) and 3075 m/s (10 089 ft/sec) , respectively. Perigee and apogee velocities in the transfer ellipse were computed at 9983 m/s (32 753 ft/sec) and 1628 m/s (5341 ft/sec) , respectively. Transfer delta-V's were evaluated as 2400 m/s (7874 ft/sec) and 1762 m/s (5781 ft/sec) for perigee and appogee burns, respectively.

Orbit transfer timelines include phasing operations as defined in the earlier Future Space Transportation Systems Analysis Study (FSTSA) studies (reference 18). The LEO orbit nodal period is such that the longitudinal shift per rev is approximately 24°. Thus, GEO longitudinal destinations for transfer opportunities, which occur at every nodal crossing, are spaced at approximately 24° intervals. Waiting in LEO for the best transfer opportunity will permit arrival at GEO within approximately $\pm 24^{\circ}$ of the desired longitude. The wait period will not exceed 24 hours; 12 hours is a representative value. Upon arrival at GEO, an elliptical coplanar phasing orbit is used with period up to 1.6 hours less than the GEO orbit period of 24 hours. The phasing orbit period should always be less than the GEO period; the GEO circularization then occurs in two burns that ideally sum to the delta-V required for a single burn injection. When departing GEO, a wait period is required to permit the return transfer to always be coplanar with the staging base orbit since the GEO mission will ordinarily not be synchronized with the staging base orbit period.

A summary of representative mission events, delta-V's, and times is given in table 2.2.3-1.

Table 2.2.3-1. Mission Profile for LO_2/LH_2 POTV

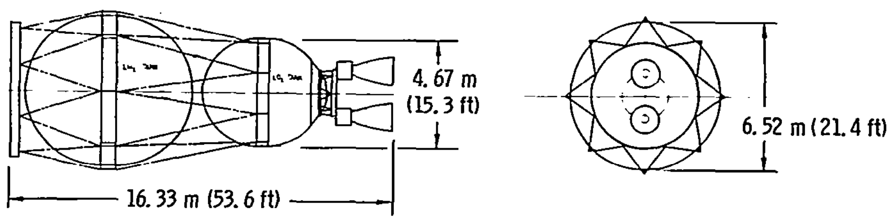
MISSION EVENT	REQUIRED TIME (hr)	DELTA m/sec	PROPULSION (MAIN OR AUXILIARY)	REMARKS
		MISSION		
STANDOFF	0	3	A	PROVIDES SAFE SEPARATION DISTANCE BETWEEN FACILITY AND VEHICLE
PHASE	12.0	3	A	ΔV IS ATTITUDE CONTROL
BOOST	0.6	2460	M	INCLUDES 60 m/s ACCUMULATED FINITE-BURN LOSS
COAST	5.4	3	A	TRANSFER TO GEO
PHASE INJECT	0.1	1722	M	REPRESENTATIVE FOR 15° PHASING
PHASE	23.0	3	A	
TPI (TERMINAL PHASE INITIATION)	0.1	55	A	INCLUDES 15 m/s OVER IDEAL TO ALLOW FOR CORRECTIONS
RENDEZVOUS	2.0	10	A	TPI ASSUMED TO OCCUR WITHIN 50 km OF TARGET
DOCK	1.0	10	A	
WAIT	8.0	0	–	ASSUMED DOCKED
STANDOFF	0.1	3	A	
DEORBIT	0.1	1762	M	
COAST	5.4	10	A	TRANSFER TO LEO
PHASE INJECT	0.1	2350	M	
PHASE	12.0	3	A	ORBIT PERIGEE AT STAGING BASE ALTITUDE
TPI	0.1	50	A	
RENDEZVOUS	2.0	10	A	
DOCK	1.0	10	A	
RESERVE	–	166	M	2% OF STAGE MAIN PROPULSION ΔV BUDGET

2.2.3.3 Configuration Description

The configuration of the POTV is presented in figure 2.2.3-3 with the overall geometry noted. Main propulsive thrust is provided by two ASE-type engines, each having a vacuum thrust of 89 000N (20 000 lbf). These engines provide thrust for the orbit transfer maneuvers exclusive of the GEO and LEO terminal phase initiation (TPI) maneuvers. Thrust for the TPI's is provided by the LO_2/LH_2 auxiliary propulsion system. The spacecraft structure consists primarily of truss assemblies and semi-integral main propellant tanks. The truss assemblies are of composite design. Main propellant tanks are of 2219-T87 aluminum. A multi-layer insulation blanket provides for propellant boiloff control and spacecraft meteoroid protection. Electrical power is provided by O_2/H_2 fuel cells.

A summary mass statement is presented in table 2.2.3-2. Note that the resulting mass fraction, in which the mainstage propellant excludes the propellant for TPI's, is 0.928.

Each of the items in the summary mass statement, exclusive of payload and mainstage propellant, is discussed in the following paragraphs including definition of rationale for mass estimates.



GROSS WEIGHT	96,839 KG	(213,530 LB)
P/L WEIGHT (75% RETURN)	12,381 KG	(27,300 LB)
MAINSTAGE PROPELLANT - MPS	78,413 KG	(172,900 LB)
INERT WEIGHT	6,045 KG	(13,330 LB)

T/W @ STARTBURN = 0.187

Figure 2.2.3-3. - POTV Configuration

TABLE 2.2.3-2. - POTV SUMMARY MASS STATEMENT

GROUP	MASS KG	WEIGHT (LB)
STRUCTURES & MECHANISMS	1,297	(2,860)
THERMAL CONTROL	490	(1,080)
MAIN PROPULSION SYSTEM (MPS)	753	(1,660)
AUX. PROPULSION SYSTEM (APS)	422	(930)
ELECT. POWER SYSTEM (EPS)	59	(130)
AVIONICS	213	(470)
MARGIN	324	(715)
DRY	3,558	(7,845)
PAYLOAD (75% RETURN)	12,381	(27,300)
RESIDUAL FLUIDS & GASES	878	(1,935)
RESERVES - EPS & APS	131	(290)
INFLIGHT LOSSES	195	(430)
PROPELLANT-APS	1,283	(2,830)
PROP.-MAINSTAGE-INCL. FPR	78,413	(172,900)
GROSS	96,839	(213,530)

$$\text{MASS FRACTION} = \frac{\text{MAINSTAGE PROP. (INCL FPR)}}{\text{GROSS, LESS PAYLOAD}} = 0.928$$

Structures and Mechanisms–This group consists of the following: LH_2 and LO_2 tanks, primary trusses, docking/service/equipment/avionics section, thrust structure, and secondary structures. Total mass is 1297 kg (2 860 lb).

- LH_2 and LO_2 Tanks - The tanks containing the liquid hydrogen and liquid oxygen are all-welded 2219-T87 aluminum pressure vessels. Both tanks have stiffened belly-band sections. In addition, the aft dome of the LO_2 tank incorporates a short stiffened cone frustrum section to allow for distribution of main engine thrust loads. The tank pressure shells are designed by room temperature proof test conditions corresponding to 0.2g initial T/W with maximum ullage pressure of 97 kPa (14 psia). To satisfy a 50 mission service life requirement with very high assurance of no-leak with minimal mass impact, the tanks are designed using conservative fracture mechanics design data (i.e., lower boundary data in lieu of best fit data) and are not pressure cycled between missions. A minimum shell thickness of 0.064 cm (0.025 inch) was incorporated. The LH_2 tank mass is 456 kg (1005 lb), 4.1% of the liquid hydrogen capacity. The LO_2 tank mass, including slosh baffles, is 347 kg (765 lb), 0.52% of the liquid oxygen capacity.

- Primary Trusses—The primary trusses consist of the intertank truss assembly and the forward truss assembly. These truss assemblies incorporate composite tube struts with aluminum end fittings and attachment fittings. Each truss assembly is designed to carry an axial load equal to the total thrust of the two main engines as adjusted by a 1.4 ultimate factor of safety. The masses of the truss assemblies are 48 kg (105 lb) and 27 kg (60 lb) for the intertank assembly and forward assembly, respectively.

- Docking/Service/Equipment/Avionics Section—This structural/mechanical assembly consists of a 0.4m (16 inch) high by 5.8m (230 inch) outer diameter toroidal framework which incorporates the following: a universal docking system; a peripheral latch/release system for payload accommodation; service connector panels for fluids, gases, and electric power; and support assemblies for major equipment and avionic components. Extensive use is made of composite structural members. Total estimated mass is 349 kg (770 lb).

- Thrust Structure—The thrust structure transmits loads from the two ASE-type engines, each having a vacuum thrust of 89 000N (20 000 lbf), to the aft dome of the LO_2 tank. The structural assembly consists of a 2.0m (80 inch) diameter engine mounting ring, sixteen struts, and the fittings for attaching the struts to a support ring in the aft dome. The basic tube struts and the basic I-beam section of the engine mounting ring are composite structures. Total estimated mass is 38 kg (85 lb).

- Secondary Structures—An allowance of 32 kg (70 lb) has been incorporated for secondary structures.

Thermal Control—Thermal control elements consist of MLI blankets, temperature control provisions for the fuel cells and equipment/avionics, and heat protection provisions for the vehicle base region. MLI blankets enclose the LH_2 tank, the LO_2 tank, and the spacecraft structural frame. These blankets consist of thirteen layers of 0.15 mil* mylar radiation shields aluminized on both sides, nylon net spaces, and protective cover sheets of 3-mil teflon. The teflon sheet on the outer surface of the spacecraft blanket is aluminized on its backside. The combination of radiation shields and protective covers provides an effective meteroid protection system for the propellant tanks. The installed mass of the MLI blankets is 408 kg (900 lb). Total group mass is 490 kg (1 080 lb).

*(1 mil = 0.001 in. = 0.0254 mm.)

Main Propulsion System (MPS)—The main propulsion system consists of rocket engines and accessories, and the propellant system. Total mass is 753 kg (1 660 lb).

- Rocket Engines and Accessories—Main propulsive thrust is provided by two ASE-type engines, each having a vacuum thrust of 80 000N (20 000 lbf). Pertinent engine characteristics are presented in Section 2.1.2. The dry mass of the two engines including accessories is 413 kg (910 lb).

- Propellant System—This subgroup consists of zero-NPSH boost pumps and electric drive motors, propellant feed/fill/drain system, vent/relief system, helium pneumatic system, and a propellant loading/monitoring system. There is no active pressurization system. Engine start is satisfied by the combination of propellant acquisition devices, zero-NPSH boost pumps, and main engine tank head idle (THI). Total subgroup mass is estimated at 340 kg (750 lb).

Auxiliary Propulsion System (APS)—A LO_2/LH_2 auxiliary propulsion system provides for vehicle orientation and small delta-V maneuvers and, in addition, provides for GEO and LEO terminal phase initiation maneuvers. The ratio of system dry mass to usable propellant mass was estimated at 0.30 based on previous studies. Total dry mass is 422 kg (930 lb).

Electrical Power System (EPS)—This group consists of O_2/H_2 fuel cells and the storage and feed provisions for the reactant. The O_2 and H_2 are stored in the supercritical condition. System dry mass is estimated at 59 kg (130 lb).

Avionics—The avionics group includes elements for guidance, navigation and control, communications, data management, rendezvous and docking, data measuring, and power conversion and distribution. Total mass is 213 kg (470 lb), based on a combination of IUS and space tug data as modified by consideration of advanced avionics and reduced power requirements.

Margin—A margin allowance of 10% of subsystems dry weight has been incorporated. Total mass is 324 kg (715 lb).

Nominal APS Propellant—The APS delta-V budget for vehicle orientation and small delta-V maneuvers is 78 m/s (258 ft/sec). An additional 105 m/s (344 ft/sec) is provided for GEO and LEO terminal phase initiation maneuvers. Based on a specific impulse of 4187N-sec/kg (427 sec) for the LO_2/LH_2 thrusters, and vehicle sequential mass, the total nominal propellant mass is 1283 kg (2 380 lb).

Inflight Losses–This group consists of the following: fuel cell nominal reactant usage for the 3-day mission; main propellant boiloff for the 3-day mission; and mass propellant start-stop losses associated with four firings of each of the two main rocket engines. Total mass is 195 kg (430 lb).

Reserves, APS and EPS–Reserve allowances for APS and EPS are 10% and 33% of nominal requirements, respectively. Total mass is 131 kg (290 lb). (The reserve allowance for MPS is 2% of total ideal delta-V and is included with the mainstage propellant mass.)

Residual Fluids and Gases–This group consists of the fluids and gases, onboard at end of mission under nominal conditions, for MPS, APS, and EPS. Total mass is 878 kg (1 935 lb).

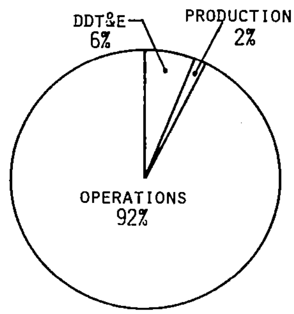
- MPS Residuals–This subgroup consists of the following: propellant trapped in engines, propellant lines, boost pumps, and tank bottom sumps/propellant acquisition devices; bias fuel; and gases in the empty tanks. Trapped propellant mass is estimated at 254 kg (560 lb). A single-engine reference bias fuel allowance of 0.50% was used. For multiple engines, this allowance was factored by the inverse of the square root of the number of engines. The resulting LH_2 bias allowance for the two main rocket engines is 0.35%. Bias fuel mass is 68 kg (150 lb). The gases in the empty propellant tanks are at the maximum ullage pressure of 97 kpa (14 psia) and mean temperatures for the GO_2 and GH_2 of 90 K (162^{o}R) and 20 K (36^{o}R), respectively. Gas masses are 268 kg (590 lb) of GO_2 and 213 kg (470 lb) of GH_2. Total mass of MPS residuals is 803 kg (1 770 lb).

- APS and EPS Residuals–The residuals for the APS and EPS are estimated at 75 kg (165 lb).

2.2.3.4 POTV Life Cycle Costs

The total POTV life cycle cost is $7101.3 million. A summary of the LCC is presented in figure 2.2.3-4. There are three phases in the LCC: DDT&E, production and operations. All costs presented here are in constant 1977 dollars.

	$ MILLIONS
TOTAL PROGRAM	7,101.32
DDT&E	437.26
PROGRAM MANAGEMENT	5.67
ENGINEERING	313.52
MANUFACTURING	61.71
TEST	56.36
PRODUCTION	114.66
PROGRAM MANAGEMENT	15.58
SYSTEM ENGINEERING	2.34
MANUFACTURING	96.74
OPERATIONS	6,549.45
OPERATIONS SUPPORT	288.16
LAUNCH SUPPORT	6,261.29



- COST/FLT = $\$4.97 \times 10^6$
- COST/KG (LEO TO GEO) = $401/KG (75% Return)

Figure 2.2.3-4. - POTV Cost Summary

DDT&E Phase

The DDT&E phase cost is $437.3 million. The development cost of the advanced space engine at $250 million dominates the DDT&E cost. Use of STS components or technology resulted in reduced DDT&E costs for the avionics and electrical power systems.

A propulsion/dynamic test vehicle and a flight test vehicle are included in the manufacturing subphase cost. These vehicles are refurbished in the production phase for use in the operational fleet.

Production Phase

The cost of manufacturing the POTV fleet of 5 vehicles and the cost of production tooling for the fleet and the operational spares are contained in the Production phase cost of $114.7 million.

This cost is based on the following TFU costs:

o POTV airframe	$16.76 million
o ASE	$1.83 million
o Complete POTV	$20.33 million

The manufacturing subphase cost of $96.7 million includes $10.2 million for refurbishment of the 2 test vehicles for operational use.

Operations Phase

The POTV has an operations phase cost of $6549.5 million. A breakdown is shown in table 2.2.3-3.

TABLE 2.2.3-3. POTV OPERATIONS PHASE COSTS

PROGRAM ELEMENT	COST ($ MILLION)		
OPERATIONS PHASE	6549.5		
OPERATIONS SUPPORT		288.2	
PROGRAM SUPPORT			44.4
SPARES PROCUREMENT			243.8
LAUNCH SUPPORT		6261.3	
OPERATIONS			213.8
PROPELLANT			6047.5

The total cost is dominated by propellant at $6047.5 million. Of this amount, $6019.7 million is the cost of delivering the propellant to LEO by the HLLV. LEO delivery costs for spares and stages are included in the program support cost.

Within the launch support subphase, the operations cost of $213.8 million provides for the rotation and re-supply of a 12 man crew at the LEO base.

2.2.4 Large Cargo Orbit Transfer Vehicle (LCOTV)

2.2.4.1 Design Requirements and Configuration Issues

Top level LCOTV design requirements are summarized as follows:

- o Payload to match HLLV
- o No return payload
- o No trip time constraint
- o Space assembled/space based

Since there were few previous studies of this type and size of solar-electric vehicle, the requirements and concept issues are not as well defined as for the other vehicles in the system. A data base did exist in the studies of solar power satellites (reference 19) and it was used whenever possible.

Trip time, thruster characteristics and power requirements are issues which must be addressed for this type of vehicle. Figure 2.2.4-1 summarizes these issues and shows the design point selected and rationale.

The configuration concept is shown in figure 2.2.4-2. This vehicle is fabricated in space at the space base and is maintained in space for its entire 15 mission life. Normal maintenance includes solar array annealing to remove radiation damage, refurbishment of power processing equipment and removal and replacement of the thruster panels.

- TRIP TIME ↔ THRUSTER LIFE MATCHUP
- HIGH BEAM CURRENT & HIGH I_{SP} = LESS THRUSTERS
- LOWER TRIP TIME = MORE POWER/LARGER VEHICLE
- COMPROMISE BETWEEN # THRUSTERS & AMOUNT OF ARRAY

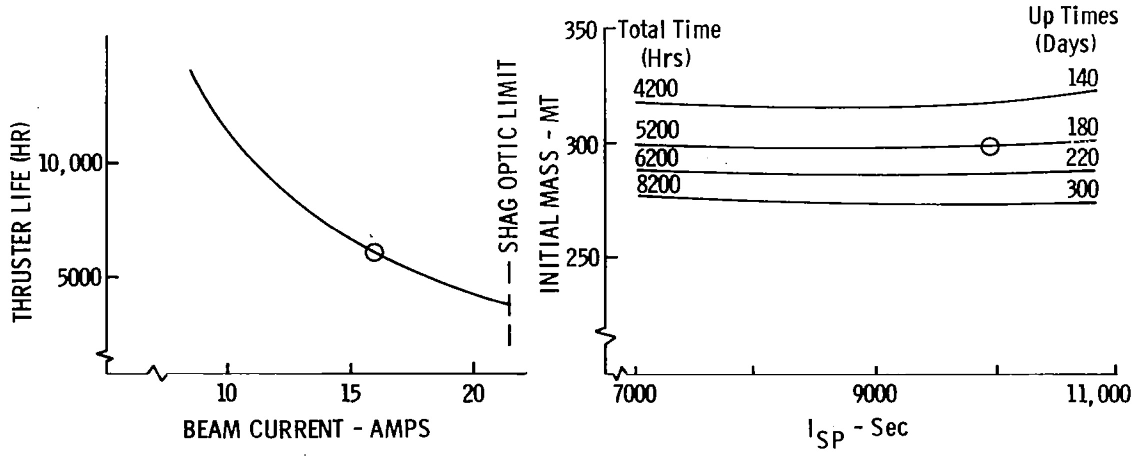


Figure 2.2.4-1. - LCOTV Reference Configuration Selection Criteria

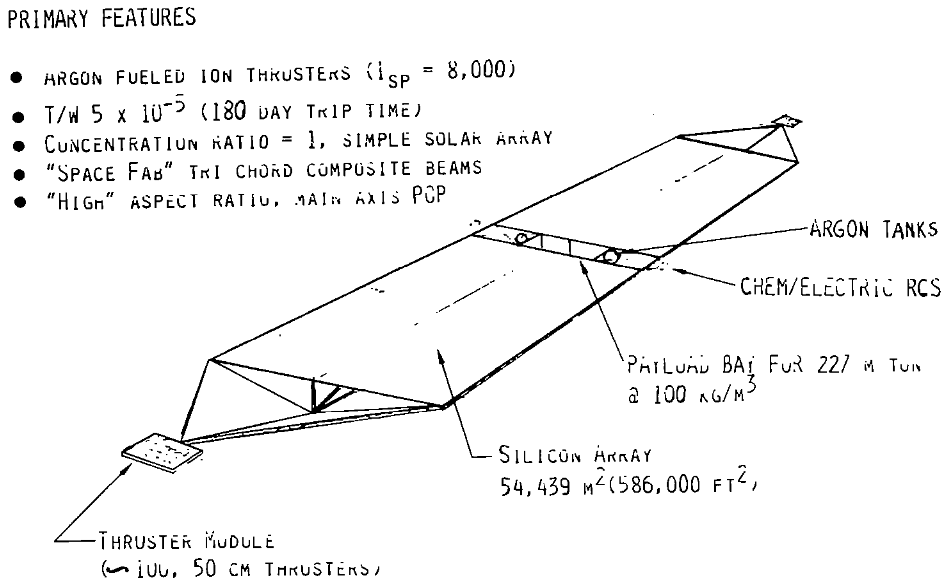


Figure 2.2.4-2. - LCOTV Design Concept

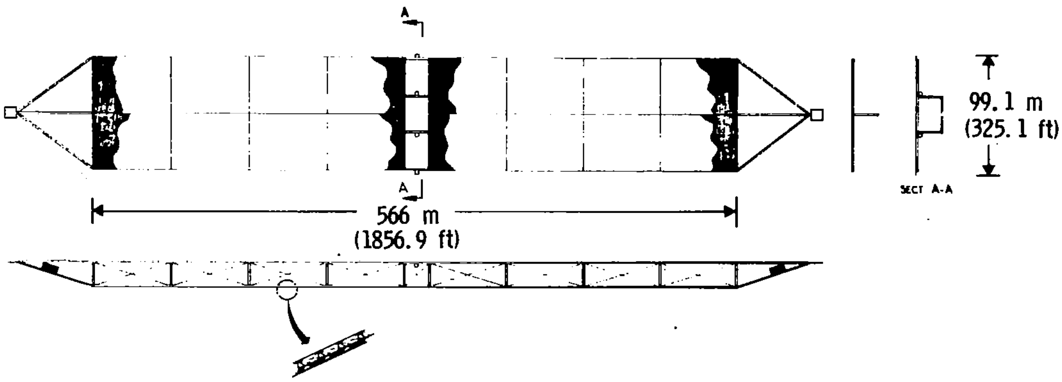
2.2.4.2 Configuration Description (LCOTV)

The configuration of the LCOTV is shown in figure 2.2.4.-3. The main propulsion modules are mounted on the centerline beams at each end of the vehicle. Auxiliary propulsion modules are located at the vehicle center on the lateral axis. Both of these modules include electric and chemical thrusters. The chemical thrusters are utilized during occulted periods of the transfer trajectory.

A summary mass statement is shown in table 2.2.4-1. As noted, the groundruled 10% weight growth margin was increased to 25% in those areas with a commensurate degree of uncertainty.

Each of the items in the mass statement, exclusive of payload and mainstage propellant, is discussed in the following paragraphs.

Structure--This item consists of the tri-chord beams which make up the vehicle basic framework including the solar array section subframes and the thruster module structural panels.



ARRAY AREA = 54,416 m^2
NUMBER OF THRUSTERS = 206
TOTAL THRUST = 145 Newtons

Figure 2.2.4-3. - LCOTV Configuration Normal Accelerated Technology

TABLE 2.2.4-1. - LCOTV Characteristics & Weights

WEIGHTS	KG	(LB)
STRUCTURE	4,057	(8,946)
POWER GEN.	26,831	(59,163)
PROPULSION	11,671	(25,735)
PROPELLANT SYSTEM	2,217	(4,888)
THERMAL CONTROL	377	(831)
AVIONICS	520	(1,147)
GROWTH*	9,339	(20,593)
DRY	55,012	(121,301)
PROPELLANT	29,744	(65,586)
RESERVES	892	(1,967)
PAYLOAD (0% RETURN)	227,000	(500,535)
GROSS WEIGHT - START BURN	312,648	(689,389)

*Minimum 10% - Maximum 25%

The thruster panel and gimbal masses were established using a weight estimating relationship (WER) of 15% of the thruster mass. This WER is based on previous work in support of self powered transfer of solar power satellites.

The structural beams consist of three continuous cap members, equally spaced upright cross members, and continuous diagonal cord cross-bracing. The chords are arranged into an equilateral triangle cross section with a typical leg length of 1.4 meters. This type beam is consistent with an automated beam building concept. Beam sizing was checked for the static loads and found to have sufficient margin within the minimum fabrication gauges used for weights calculations.

Power Generation–This item consists of the solar array including its installation provisions and the power distribution system.

The solar array, which easily makes up the single largest mass, is based on detailed analysis performed in the solar power satellite studies, (reference 9). The specific weight of the array, including a 15% factor for installation, is 427 kg/m^2.

The power distribution system consists of aluminum sheet conductors or power busses serving each of the 8 vehicle power sectors. Current density in the conductors was maintained at 988 watts/m^2 in order to limit conductor temperature to 100°C. The conductors were sized at a thickness of 5 mm and carried a current of 1150 amperes.

Propulsion–This item consists of thrusters, power processors and the auxiliary propulsion system.

The thruster mass was estimated at 34 kg. This estimate was based on 30 cm thruster technology utilizing small hole accelerator grid (SHAG) optics. The 50 cm thruster is the largest extrapolation of this single cathode technology.

Power processor masses were estimated at .945 kg/kW which is representative of dc-to-dc conversion utilizing advanced motor-generator technology.

A 1000 kg mass allocation was made to cover the auxiliary propulsion system.

Propellant System–The propellant system includes the tanks and feed system.

The argon tanks are fabricated of 2219 aluminum alloy. A minimum gauge of 0.064 cm (.025 in) was used and is considerably thicker than required by operating pressure. The tanks were insulated with 70 layers of MLI with appropriate factors for covers and installation.

Feed lines were sized at 5 mm (.196 in)diameter and represent a minor percentage of the system weight.

Thermal Control–This item consists entirely of a radiator system for the power processors unit (PPU). They were sized to radiate the PPU waste heat load, 5% of total power, and included pumps, manifolds and fin and pipe radiator panels. A radiator system was installed at each end of the vehicle on the thruster panel support tripod.

Avionics–The avionics included a full suit of the POTV avionics with an allowance for radiation shielding.

2.2.4.3 LCOTV Life Cycle Costs

The Normal Growth Technology LCOTV Life Cycle Costs of $3276.1 million is distributed by phase and subphase as shown in figure 2.2.4-4. The following paragraphs discuss the three phases: DDT&E, production and operations. All costs reported in this section are constant 1977 dollars.

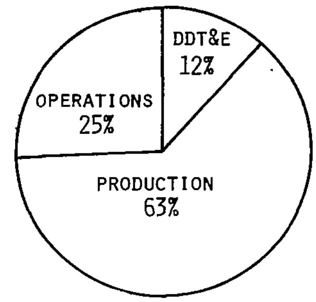
DDT&E Phase

The LCOTV DDT&E phase has a cost of 387.9 million. The breakdown of DDT&E costs is shown in table 2.2.4-2. Included in the manufacturing subphase are the developmental tooling costs and the cost of a test vehicle.

Production Phase

The major portion of the LCOTV LCC is in the production phase. This occurs because the LCOTV is a hardware intensive vehicle with small propellant and manpower costs. The long trip time and high annual rate at the end of the mission model require a large fleet, 13 vehicles.

	$ MILLIONS	
TOTAL PROGRAM	3,276.06	
DDT&E	387.93	
PROGRAM MANAGEMENT		12.54
ENGINEERING		150.86
MANUFACTURING		128.18
TEST		96.35
PRODUCTION	1,930.12	
PROGRAM MANAGEMENT		70.74
SYSTEM ENGINEERING		51.81
MANUFACTURING		1,807.57
OPERATIONS	958.01	
OPERATIONS SUPPORT		574.78
LAUNCH SUPPORT		383.23



- COST/FLT = $\$17.11 \times 10^6$
- COST/KG (LEO TO GEO) = $75/KG

Figure 2.2.4-4. - LCOTV Cost Summary

TABLE 2.2.4-2. NORMAL GROWTH TECHNOLOGY LCOTV DDT&E COSTS

Program Element	Cost ($Million)			
DDT&E Phase	387.93			
Program Management		12.54		
Engineering		150.86		
Systems Eng. & Int.			11.78	
Software Engineering			4.29	
Design & Development			134.79	
Structure				31.81
Solar Array				30.00
Power Distribution				3.96
Thrusters				25.00
PPU's				8.16
RCS				5.41
Tanks & Feed Sys.				10.76
Radiators				2.15
Avionics				13.24
Manufacturing		128.18		
Developmental Tooling & S.T.E.			8.97	
Test Hardware & Spares			119.21	
Test		96.35		
System Test Operations			51.06	
Flight Test Operations			45.29	

Solar array and electric thruster unit costs were estimated using the "Mature Industry" costing methodology developed during solar power satellite studies and discussed in Volume III. This method can estimate costs when production quantities are high enough to justify dedicated facilities. These facilities are designed around a certain production rate for their lifespan. The unit costs assigned to the solar array and the thrusters included both material costs and the capital costs of the dedicated production facilities. Using mature industry costing for an array production rate of 52 000 m^2/year and a thruster production rate of 215 units/year the respective unit costs are \$1510/m^2 and \$1180/kg (25 kg thrusters).

The remainder of the LCOTV hardware is not produced in sufficient quantity to justify mature industry production. These items are costed using traditional aerospace costing methods. The TFU cost of the LCOTV, less array and thrusters, is \$58.0 million. Included in the manufacturing subphase cost of \$1807.6 million is \$194.5 million for production tooling of the LCOTV less array and thrusters.

Operations Phase

The Operations Phase costs are broken down as follows:

Program Element	Cost ($ Million)		
Operations Phase	958.0		
Operations Support		574.8	
Program Support			122.2
Spares Procurement			452.6
Launch Support		383.2	
Operations			285.0
Propellant			98.2

Program support includes the management cost of the operations phase and the delivery cost of spares.

The Spares Procurement cost is based on the following ground rules:

- o 50% Thruster refurbishment per flight; 10 flight life
- o Solar array annealed after each flight, no spares required.
- o Remainder of LCOTV has 10% refurbishment per flight.

LCOTV operations during the mission and during the turnaround between flights is performed by a 12 man crew at the LEO base. The total rotation and resupply cost for this crew is $285.0 million.

The $98.2 million propellant cost is primarily the cost of delivering argon to LEO.

3.0 ACCELERATED TECHNOLOGY

3.1 FORECAST & ASSESSMENT

The technology requirements or areas chosen for assessment as accelerated technology (additional funding required to bring this capability to the reasonable risk level) were those which had the potential for greatest system impact. These items were suggested by various NASA organizations as well as the participants and were reduced to a manageable number by agreement between LRC and Boeing.

Each of these technology advances were individually characterized and applied to the vehicle family. The life cycle cost impacts were calculated and used to assess each advancement.

The following tabulation summarizes the areas investigated and the vehicles to which they were applied.

ACCELERATED TECHNOLOGY ITEMS	SSTO	HLLV	POTV	LCOTV
IMPROVED SSME	X	X		
DUAL FUEL/DUAL EXPANDER ENGINE	X	X		
ALTERNATIVE OTV ENGINE			X	
SLUSH PROPELLANTS	X	X		
INTEGRATED LOX/LH_2 SUBSYSTEMS	X	X	X	
IMPROVED AVIONICS	X	X	X	
COMPOSITE STRUCTURES	X	X		
METALLIC TPS	X	X		
CCV CONFIGURATIONS	X	X		
SILICON VS GaAs SOLAR ARRAY				X
ION THRUSTER OPTIONS				X
DIRECT POWER PROCESSING				X

Each of the following paragraphs discusses the individual assessments.

3.1.1 Composite Structures

For normal growth technology, a weight factor (composites versus aluminum) of 70% was used. This 70% weight factor represented a "composite substitution" approach. For accelerated technology, the more efficient "composite design" approach applies, and the corresponding weight factor was estimated at 60%. In applying this 60% weight factor, no consideration was given to possible secondary weight reductions due to improved material thermal properties. In addition, for the accelerated technology launch vehicles, composites application was extended to the propellant lines in the ascent propulsion system.

Applying the 60% weight factor to the normal growth technology launch vehicles, and resizing the vehicles downward, the reduction in life cycle costs are: SSTO, $3 112 million; HLLV, $986 million. In deriving these cost benefits, the fabrication and DDT&E costs were estimated as equal in $/lb to conventional metallic construction. Overall, based on cost benefits, the acceleration of composite structures design is the most important technology area.

3.1.2 Dual Fuel - Dual Expander Engine

The Dual Fuel - Dual Expander (DF/DE) engine was developed by ALRC to provide improved performance for dual mode SSTO's. It is a tri-propellant engine burning LO_2 as the oxidizer with LCH_4 and LH_2 as fuels. There are two combustion chambers, a central one burning LO_2/LCH_4 at 41 370 kPa(6000 psia) and an annular outer chamber burning LO_2/LH_2 at 20 690 kPa(3000 psia). A schematic diagram of a DF/DE engine cycle RP-1 instead of LCH_4, but otherwise identical, is shown in figure 3.1.2-1.

The engine operates in two modes:

(1) During Mode I both chambers are operating to give high thrust and good Isp at a moderate expansion ratio

(2) During Mode II the central LO_2/LCH_4 chamber is shut down and the engine operates as a high expansion ratio LO_2/LH_2 engine at high Isp.

The characteristics of the DF/DE engine selected for the advanced technology launch vehicles SSTO and HLLV are summarized as follows:

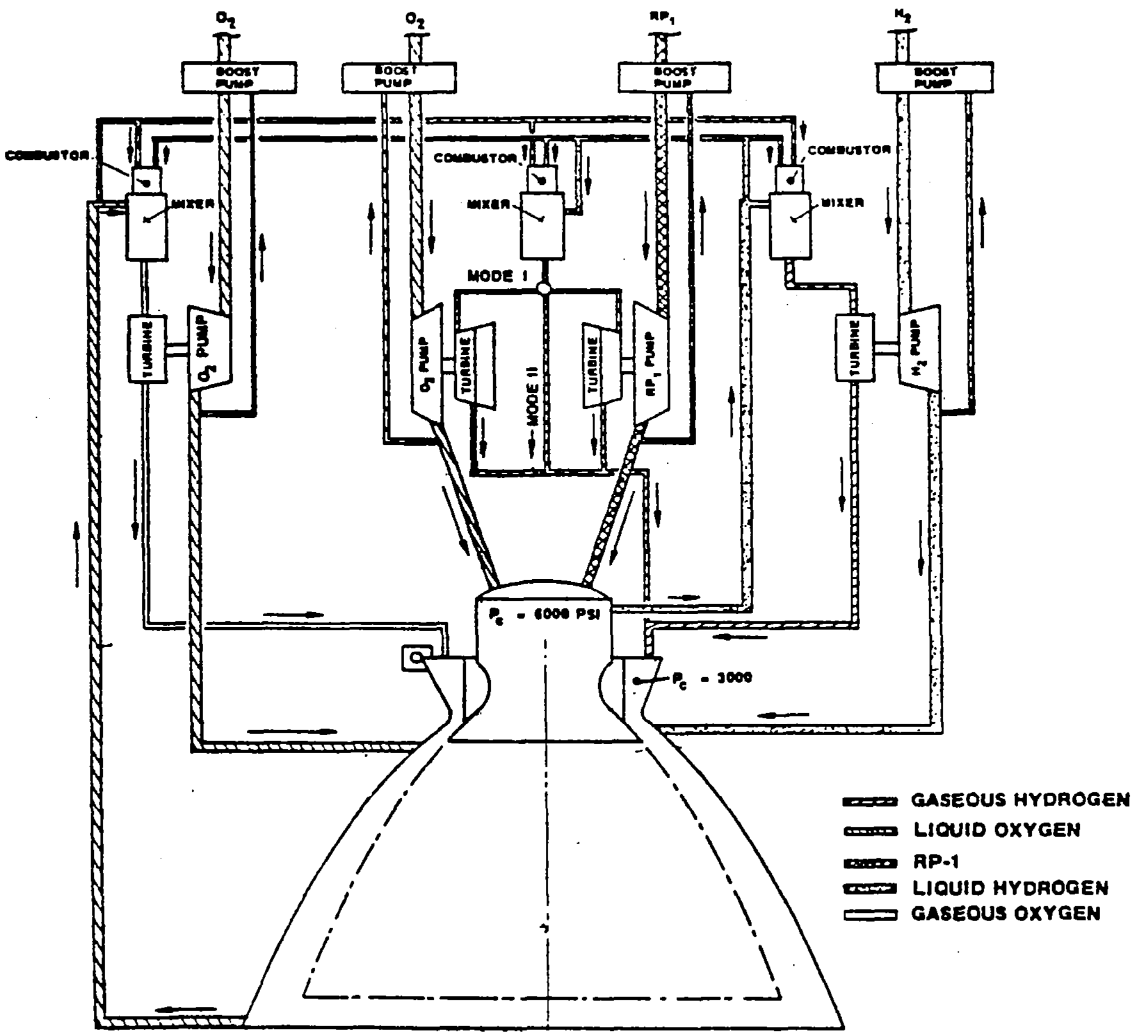


Figure 3.1.2-1. - Tripropellant Dual-Expander Engine

- o 70% - 30% S.L. Thrust Split between LO_2/LCH_4 and LO_2/LH_2
- o Two Concentric Combustion: Chambers
  - o INNER: LO_2/LCH_4 MR = 3.6:1 Pc = 41 400 kPa (6000 psia)
  - o OUTER: LO_2/LH_2 MR = 7.0:1 Pc = 20 700 kPa (3000 psia)
- o MODE I: Both chambers

  Thrust: 5.12×10^6N (1.150,000 lbf) S.L. / 5.67×10^6N (1 275 200 lbf) vac.

  Isp: 352.2 sec S.L. / 390.5 sec vac.

  Combined MR: 4.21:1 Fuel Split: 82.25% LCH_4 , 17.75% LH_2

  Area Ratio: 60.5 : 1

- o MODE II: Outer Chamber Only

  Thrust: 1.82×10^6N (409 850 lbf) vac.

  Isp: 459.8 sec vac.

  MR.: 7.0:1

  Area Ratio: 127:1

- o Overall Length: 3.25 m (128 in)

  Nozzle Exit Diameter: 2.67 m (105 in)

  Dry Mass: 4458 kg (9830 lbm)

- o Life: 250 Starts between Overhaul

- o Costs: DDT&E $1219 million

  TFU $18.5 million

DF/DE Engine on SSTO

Initially the engine proposed by ALRC had a 60%/40% split in sea level thrust between the LO_2/LCH_4 and the LO_2/LH_2 stream tubes. However, when this engine was evaluated on the normal growth dual mode SSTO it caused several problems. In order to achieve the optimum propellant split, wherein 60% of the total propellant is burned in the LO_2/LCH_4 chamber, the engine did not transition from Mode I to Mode II until just prior to insertion. Without extensive throttling the resultant g levels exceeded the 3g maximum. In addition, Mode II Isp decreases relative to an engine with a smaller LO_2/LH_2 thrust fraction but equivalent Mode I S.L. thrust. This occurs because the LO_2/LH_2 throat area increases but the exit area remains essentially constant, resulting in a lower Mode II expansion ratio.

A 70%/30% thrust split was selected to correct these problems. This matches closely the thrust and flow rate split of the SSME - LO_2/LCH_4 engine pair, which were initially selected to produce an optimum trajectory.

When the 70%/30% split DF/DE engine was evaluated on the baseline SSTO it had a dramatic impact. The trajectory mass ratio decreased from 9.007 to 8.692 and the vehicle inert mass was reduced by 9 390 kg (20 700 lbm). These produced an increase in payload from 13 630 kg (30 050 lbm) to 20 870 kg (46 000 lbm).

The vehicle was then resized parametrically to match the payload and mission model of the baseline SSTO. GLOW was reduced from 2 208 300 kg (4 868 500 lbm) to 1 861 900 kg (4 104 700 lmb) and the propellant load from 1 956 300 kg (4 313 000 lbm) to 1 653 600 kg (3 645 500 lbm).

The LCC for the resized SSTO was estimated to determine the cost benefits of the Dual Fuel/Dual Expander engine. The net reduction in LCC, including the increased DDT&E for the engine, was $2118 million. The cost per flight decreased from $3.79 million/flight to $3.08 million/flight.

DF/DE Engine on HLLV

Both HLLV and SSTO use a common hydrocarbon fueled main engine in this study in order to reduce total system DDT&E costs as well as procurement and spares costs. Since the DF/DE engine showed significant value when applied to the SSTO, it was evaluated on the HLLV to determine if engine commonality could be maintained between the two launch vehicles.

Although the DF/DE engine has a delta sea level Isp advantage of 24.2 sec, it has a lower sea level T/W, 115.5 vs 129.9, compared to the LO_2/LCH_4 engine. When compared to an SSME it is inferior in both categories, with 4.2 sec less Isp and a decrease in T/W from 56.7 to 37.3. Based on these factors, the DF/DE engine was only evaluated on the booster stage of the HLLV.

The HLLV showed moderate performance improvement with the DF/DE engine, payload increasing from 226 800 kg (500 000 lbm) to 251 300 kg (554 100 lbm). This increased payload decreased the total number of flights in the mission model from 609 to 549. The HLLV LCC decreased by $613 million.

However, most of this resulted from elimination of the DDT&E cost of the LO_2/LCH_4 engine. If the HLLV were charged for the DF/DE engine development, the LCC would instead increase by $686 million.

The results of the evaluations indicate that the Dual Fuel/Dual Expander engine is a decisive winner for the SSTO, but not cost effective as an exclusive HLLV development. The benefits of the accelerated technology DF/DE engine are summarized as follows.

- o KEY ADVANTAGE IS THRUST-TO-WEIGHT RATIO & COSTS FOR DUAL FUEL PROPULSION

		DF/DE	CH_4/SSME
T/W	MODE 1 (S.L)*	115.5	92.2
	MODE 2 (VAC)	41.2	34.8
	COSTS (TFU)	$20.1 Million	$24.3 Million

*Sizing point ~20% improvement in engine weight for equivalent S.L. thrust (However, in a one-on-one comparison with CH_4 engine - 10% disadvantage)

- o COST BENEFITS

SSTO	$2118 Million (Net, includes DDT&E)
HLLV	$ 673 Million*

*Net loser by $686 million if DDT&E included.

- o FINDINGS
  - o BEATS A PAIR OF ENGINES (CH_4 + SSME) ON ALL COUNTS.
    PERFORMANCE - COST - WEIGHT
  - o LOSES ON A ENGINE-TO-ENGINE BASIS ON ALL COUNTS
    -NOT JUSTIFIABLE FOR TWO STAGE SYSTEM

*Evaluation based on DF/DE of 5.653×10^6N (1 270 800 LBF).

3.1.3 Control Configured Vehicle (CCV)

Control configured vehicle technology has been the subject of numerous studies. Reusable launch vehicles are ideal applications of this technology because of their wide operating envelope and aft center-of-gravities.

There has been a continuing program (references 20 and 21) at LRC over the past few years to study the impact of relaxing the longitudinal and lateral-directional stability requirements of this class of vehicle.

Data from these studies was used to make this trade study of removing the centerline vertical tail and adding RCS provisions, aero trim devices and side force generators.

The following data summarizes this trade on the SSTO vehicle.

- o CONFIGURATION REVISIONS (BASED ON LRC STUDIES)

	NORMAL GROWTH SSTO WGT Δs (lb)	Kg
o Remove vertical tail, associated flt. controls and body tie in.	(-15,115)	-6856
o Add wing tip aero trim surfaces	(+ 1,600)	+726
o Add subsonic forward yaw surface (retractable)	(+ 450	+204
o Adjust RCS for additional duty cycle & altitude requirement	(+ 500)	+227
o Add landing chute or alternate speed brake	(+ 1,600)	+726
o Adjust wing sweep for favorable cg impact	(-1,374)	-623
o Adjust Growth	(- 1,234)	-560
TOTAL	(-13,570)	-6,156

- o COST BENEFITS:

SSTO - $1,652 Million

HLLV - 412 Million

The weight savings are significant and in addition are significant contributions to the cg location. As a result of the forward cg shift of approximately 1% of body length, significant secondary savings of almost 635 kg (1400) lbs can be made in the wing by reducing the sweep required for hypersonic trim.

The cost benefits shown are significant by themselves and are augmented by the additional benefits called out under "findings". In addition to the impact on ground facilities, an equivalent impact on the ease of docking to the space base or safe retreival of payloads can also be assumed.

This technology area is obviously a key area and in fact may become critical to the successful operation of a vehicle family as characterized in this study.

3.1.4 Extended Life SSME

The LCC estimates for the normal growth technology transportation system showed that SSME replacement and refurbishment were the major hardware costs in the launch vehicle Operations phases, accounting for $4 Billion (10% of total system LCC). This indicated that there was a potential for major savings if SSME life could be extended to the level of the other engines in the system, 250 starts between overhauls.

The estimated DDT&E cost of this improvement is $370 million. The LCC's for the HLLV and the SSTO were evaluated with the extended life SSME. The number of SSME overhauls decreased from 262 to 57 for the SSTO and from 88 to 19 for the HLLV, resulting in savings of $1261 and $474 million for the SSTO and HLLV respectively. The net change in LCC for the total system was a reduction of $1365 million. The results of this accelerated technology long life SSME evaluation are summarized as follows.

- Life Improved to 250 Cycles
- No Performance Improvement
- No impact on weight/geometry
- DDT&E = $370 M
- Savings
    - SSTO - $1261 M
    - HLLV - $ 474 M
- Net total system LCC reduction - $1365 Million

They might indicate that the Extended Life SSME is a major advantage for the SSTO unless a DF/DE engine is used.

3.1.5 Integrated O_2/H_2 Subsystems

Appendix D of Volume III presents summary data from the study effort to evaluate and compare weight data for dedicated and integrated O_2/H_2 subsystems. The vehicles and subsystems are: SSTO and HLLV Orbiter -- OMS, RCS, and APS; HLLV Booster -- RCS and APS; POTV --MPS, RCS, and EPS. The summary data consists of simplified schematics, power system requirements, primary weight estimating criteria, detailed weight statements, and (for the launch vehicles) weight scaling equations for the integrated subsystems.

Design considerations, weight and cost impact, and major findings associated with integrating the O_2/H_2 subsystems are summarized as follows.

- o Normal Growth = Independent Oxygen/Hydrogen Fueled Subsystems
- o Accelerated Technology = Synergistic Integration of These Subsystems
  - o Common Storage Tanks
  - o Common Service Interface
  - o Common Reserves

(design price = long feed lines, different conditioning requirements, conflicting duty cycles, cascading failure modes, interrelated development planning/risks)

- o Impact

		Dry Wgt		Propellant		Cost
		Kg	(lb)	kg	(lb)	$M
SST		-949	(-2090)	-421	(-928)	-434
HLLV	Booster	-546	(-1205)	-86	(-189)	-30
	Orbiter	-1059	(-1205)	-578	(-1269)	
POTV		-54	(-119)	-67	(-147)	-102

- o Findings
  - o Moderate payoff based on cost benefits
  - o Significant operational advantage to space-based POTV-single point refueling.

The weight impact data applies to the integration of the O_2/H_2 substystems on the normal growth technology vehicles. It is first order data. The cost data, on the other hand, includes the secondary effects of sizing down the vehicles to maintain constant performance.

3.1.6 Slush Propellant

Some previous technology advancement studies have indicated that triple point or partially solidified "slush" propellants might have a beneficial effect. However there was no data that could be used to determine the impact on the type of launch vehicles used in this study. The major source of data was a recently completed study on the ground operations aspects of densified propellants produced under contract to LRC (reference 22).

The LRC study recommended the use of "slush" LH_2 (SLH_2), a 50/50 mixture of solid and liquid H_2 at triple point conditions. Slush is superior to triple point hydrogen for two reasons:

- o Hydrogen density increases substantially from liquid to solid
- o The heat of fusion of the solid H_2 greatly increases the total heat capacity of the propellant, reducing the boiloff rate.

Since the emphasis in past studies had been directed towards densified LH_2, the initial evaluation on the SSTO was limited to this propellant. The primary reason for this emphasis is the impact of LH_2 tank volume on vehicle overall size. LH_2 tanks tend to be size by ullage pressure and their mass is a function of volume rather than propellant mass. Any reduction in volume would produce a proportional drop in tank surface area and mass. LO_2 tanks, on the other hand, have less of an affect on vehicle size because an increase in density and a corresponding reduction in volume have less impact on LO_2 tank mass. The mass of an LO_2 tank tends to be driven by the head pressures which result from boost acceleration. These head pressures are a function of the density multiplied by the tank length. An increase in LO_2 density does decrease the tank length, but the reduction is directly proportional to the density increase, resulting in the same head pressure. The net effect is that the mass reduction would be much less than in a LH_2 tank. Based on these results and the previous discussion of potential LO_2 benefits, no attempt was made to use densified LO_2 on the launch vehicles.

Slush LH_2 Evaluation

SLH_2 has a density of 81.7 kg/m^3(5.1 lbm/ft^3) as opposed to 70.8 kg(m^3 (4.42 lbm/ft^3) for conventional LH_2. It is produced at the triple point conditions of 13.9 K (25.1°R) and 7.17 kPa (1.04 psia). During fueling and prior to launch, the SLH_2 tanks must be pressurized to 1 atm to avoid collapsing the tanks. A high recirculation rate must be maintained to counteract the loss in SLH_2 quality due to the GH_2 pressurant heat input and heat leak through the tank walls.

The normal growth baseline SSTO was the initial vehicle evaluated with SLH_2. The hydrogen tank volume decreased by 13% and its length decreased 6 m (236 in). There was a corresponding decrease in TPS and subpanel mass because of the decrease in LH_2 tank sidewall area and vehicle planform area. On the negative side the core of the titanium honeycomb tank was thickened to increase insulation quality. The SSTO nose also increased in mass because the forward diameter of the tapered LH_2 tank increased.

The impact on SSTO inert mass of these changes was:

- o Tank mass = - 521 kg (1148 lbs)
- o Subpanel mass = - 903 kg (1990 lbs)
- o TPS mass = - 619 kg (1365 lbs)
- o Nose mass = + 340 kg (750 lbs)
- o GH_2 Pressurant mass = - 49 kg (108 lbs)

The total change was a net reduction in inert mass of 1751 kg (3861 lbs). There was an equivalent increase in payload mass, from 13 630 kg (30 050 lbs) to 15 380 kg (33910 lbs). Although this amount is a significant fraction of the payload, 13%, it represents a decrease in inert mass of less than 1%. The LCC of the baseline SSTO was reduced $300 million by using SLH_2. This LCC includes an $50 million DDT&E increase for the additional complexity associated with SLH_2. It also reflects the increased cost of SLH_2 ($.368/kg ($.811/lb) from $.332/kg ($.731/lb) for LH_2) and the cost of solidifying the SLH_2 that melts prior to launch (2/3 of the total). The net effect is to increase LH_2 costs by 25%.

Because the impact of SLH_2 on the SSTO was small compared to estimates of previous studies, a parametric analysis was made of different tank geometries. Several different LH_2 tanks were examined to determine the decrease in mass attributable to SLH_2. Included in the analysis were the effects of reduced tank volume, additional tank insulation and reduced subpanel and TPS area. After the results had been examined a correlation was observed between tank "volumetric efficiency" and reduction in mass. Tank volumetric efficiency, L*, is defined as:

$$L^* = \frac{\text{(Volume/Surface Area) of tank}}{\text{(Volume/Surface Area) of sphere of equal volume}}$$

The effect of L* on the decrease in tank mass fraction (tank mass fraction = tank mass/LH_2 mass) is shown in figure 3.1.6-1. Points 1 and 5 represent the two extremes, a sphere and a cylinder with an L/D of 5, respectively. Points 2 and 3 are LH_2 tanks from the HLLV Orbiter and booster and point 4 is the SSTO. As tanks become more volumetrically efficient, the benefits of using SLH_2 diminish until the increased insulation outweighs the reduced tank wall and TPS mass in the case of the sphere. It apears that SLH_2 results in significant mass reduction only for vehicles with large LH_2 tanks of low volumetric efficiency, as occurs when tank configuration is driven by aero requirements.

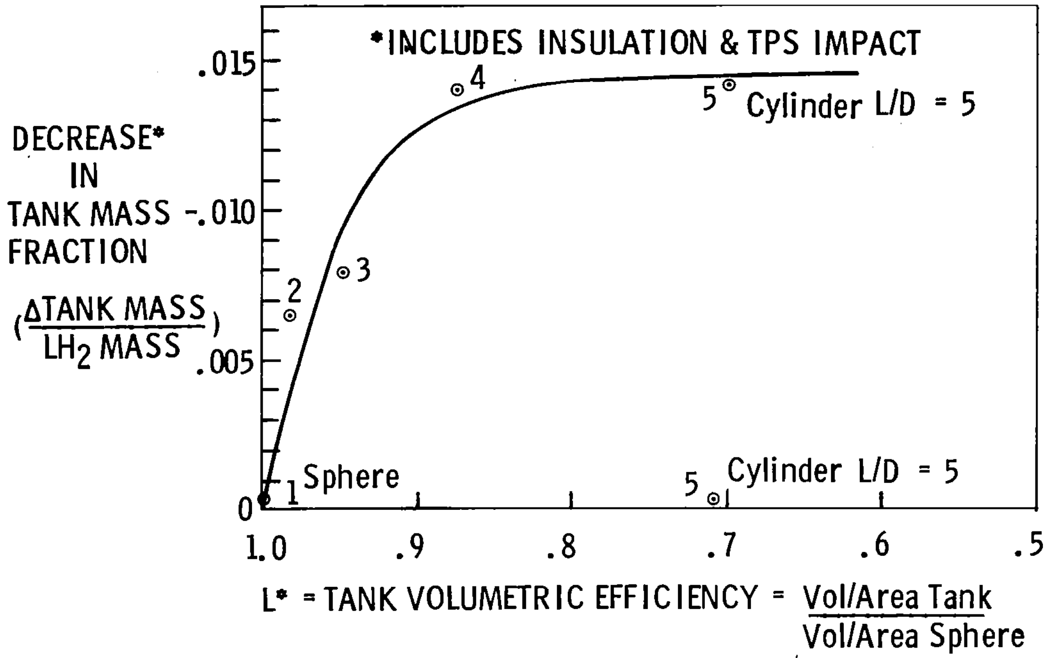


Figure 3.1.6-1. - Accelerated Technology Slush Propellants

An evaluation of the impact of SLH_2 on the HLLV was also performed. Because the HLLV tanks are much more volumetrically efficient the gains were less. The HLLV had a reduction in inert mass of 1504 kg (3316 lbs) for the Orbiter and only 72 kg (160 lbs) for the booster. The booster was essentially unchanged because it is a heat sink vehicle with no subpanels or RSI. Thus the effect on mass was limited to the decreased tank surface area and the increased titanium honeycomb thickness.

The HLLV LCC showed a net increase in LCC of $68 million because the increased propellant cost outweighed the improvement in payload.

The findings of this anlaysis of slush LH_2 are summarized below.

- o SLUSH HYDROGEN ONLY:
  - o "Slush" lox would require insulated tank
  - o All lox tanks are small and have high volumetric efficiency
  - o Lox tank weights more sensitive to propellant mass (instead of density) since inertial loads rather than burnout pressure size tank walls.
  - o Significant ground facility impact for "slush" lox as compared to slush LH_2

o SLUSH HYDROGEN IMPACT:

		MASS	LENGTH
SSTO:		-1750 kg (-3860 lb)	-5.59 m (-220 in.)
HLLV:	Orbiter	-1504 kg (-3316 lb)	-2.39 m (-94 in.)
	Booster	-73 kg (-160 lb)	-1.55 m (-61 in.)

3.1.7 Avionics

Accelerated technology avionics produces the following weight and power reductions. The reductions in weight and power can easily be justified based on any of a number of forecasts.

o CHARACTERISTICS:

	% REDUCTION WEIGHT	% REDUCTION POWER
POTV	20%	17%
SSTO, HLLV	25%	15%

NO COST IMPROVEMENT/NO DDT&E PENALTY

NO RELIABILITY IMPROVEMENT

No improvements were assumed in production costs. A cost reduction could be justified but is an unlikely occurrence. To compensate no DDT&E penalty was charged for the advanced avionics.

The net cost benefits shown below are marginal except for the SSTO.

o COST BENEFITS:

POTV	$18.5 Million
SSTO	$411.0 Million
HLLV	$9.2 Million

This vehicle's performance sensitivity results in sufficient overall vehicle sizing impact to show a significant cost impact.

Two findings are significant:

- o IMPROVEMENT ABOVE NORMAL GROWTH MARGINAL EXCEPT FOR SSTO
- o RELIABILITY IMPROVEMENT COULD BE MORE SIGNIFICANT

This is a marginal area on a weight/power basis beyond normal growth. Costs could be significantly impacted, however, if a reliability improvement were forecast.

3.1.8 Metallic TPS

Several metallic thermal protection system (TPS) concepts have been pursued by NASA/LARC. All of these systems have limited application to vehicles in this study because of the design philosophy associated with the vertical take-off operating mode. Minimum wing area (and therefore weight) is desirable but results in high reentry planform loadings and commensurate temperatures for a given entry profile. For these vehicles, metallic systems are generally limited to upper surface, low temperature applications. Reference (23) describes the metallic system characterized in this assessment. This multiwall TPS consists of lightweight, load-bearing insulation, fabricated as panels and installed as tiles. Each panel consists of multiple layers of dimpled and plain foils joined at dimple crests to form a strong load-bearing insulation. Thermal stresses are minimized by using simple support that is also slip jointed. Figure 3.1.8-1 depicts the panel and its installation.

Characterization of this system, its operational and cost treatment and the assessment of its value in this system are as follows.

o CHARACTERISTICS

Temp limits of 871°C (1600°F)–982°C (1800°F) (LRSI substitute)
300 entry life
No weight reduction assumed
Installed unit cost up by a factor of 2 over RSI costs

o OPERATIONAL ASSUMPTIONS

Post flight TPS inspection reduced by 50%
Per flight TPS spares and turnaround repair reduced by 40%
Reduced overhaul costs
Total turnaround flow time not affected

o COST BENEFIT

SSTO	$76 Million	No DDT&E Delta $
HLLV	$36 Million	

This technology, within the constraints of this assessment, is marginal at best. These factors must be considered together with the fact that no DDT&E penalty was assessed for a dual TPS design & development.

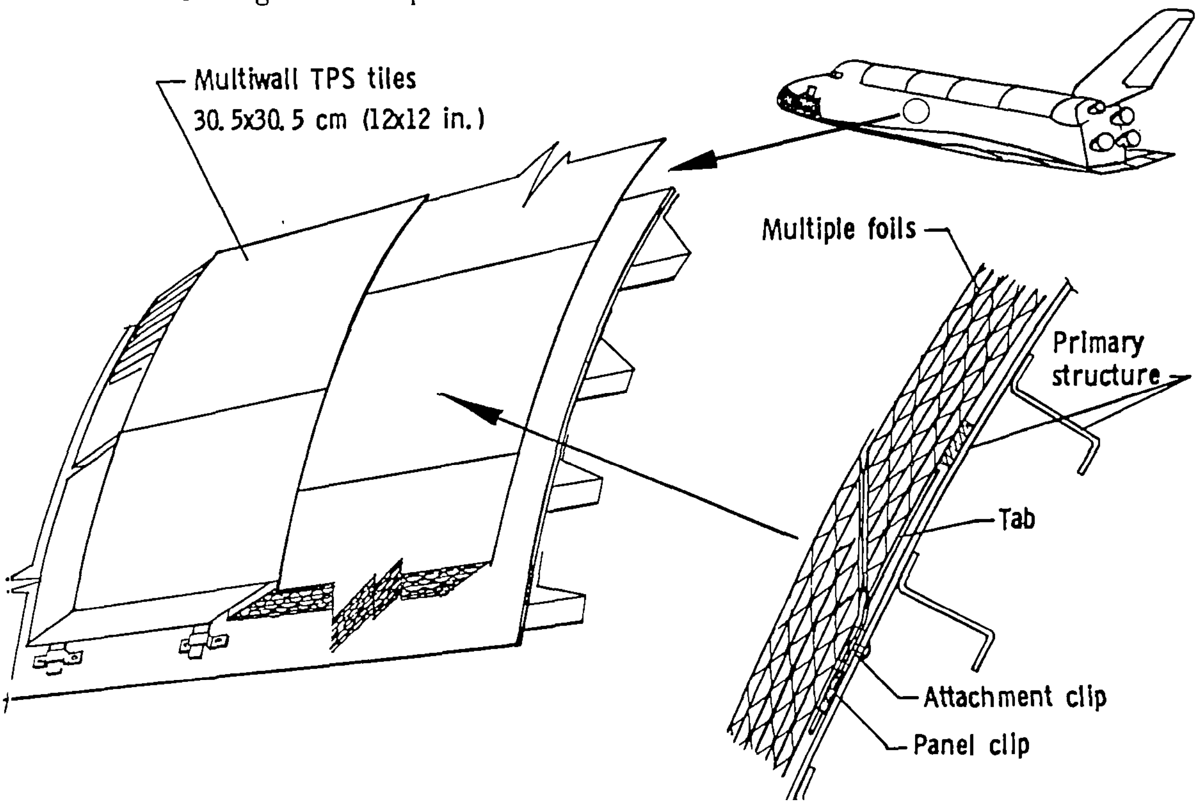


Figure 3.1.8-1. - Multiwall Thermal Protection System

3.1.9 Alternate OTV Engine

A potential OTV engine was developed by ALRC on contract to LRC, reference (24). It is a "plug cluster concept", consisting of a number of small low pressure combustion chambers mounted around the perimeter of plug extending from the aft end of the OTV. The interaction of the chambers and the plug gives a high effective expansion ratio and good Isp. The thrust chambers are the LO_2/LH_2 thrusters used for RCS on the launch vehicles. These have the advantage of simplicity and very long life. As a whole the plug cluster engine has a substantial life and cost advantage over the ASE type engine. It is also significantly shorter although for space based OTV's this is not an important factor.

The plug cluster engine was evaluated on the normal growth baseline POTV. There was an increase in engine mass of 78 kg (173 lbs) over the ASE installation and the Isp decreased to 466.9 sec. from 473 sec.

As a result the payload decreased by 590 kg (1310 lbs). However engine life increased to 1200 cycles from the ASE's 250, and the unit and DDT&E costs decreased from $1.83 to $.8 million and $250 to $125 million respectively. The net impact on the POTV LCC was negligible, an increase of $2 million on a total of $7.1 billion. The plug cluster engine is illustrated in figure 3.1.9-1 along with characteristics.

- PLUG CLUSTER CONCEPT
  - THRUST — 1.78×10^5 N (40,000 lbs)
  - Isp — 466.9 sec
  - WEIGHT — 575 kg (1267 lb)
  - INSTALLED WEIGHT DELTA +78 kg (+173 lb)
  - CHAMBER PRESSURE — 3,447 kPa (500 psi)
  - ε — 600
  - LIFE — 1200 cycles
  - TFU COST — $.8 Million
  - DDT&E COST — 125 Million

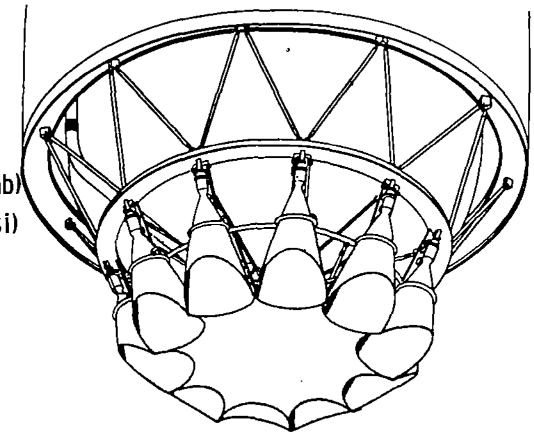


Figure 3.1.9-1. - Accelerated Technology Alternate OTV Engine

3.1.10 Solar–Electric Technologies

The three key technologies evaluated relative to the solar-electric LCOTV were: a) a GaAs solar array; b) a series of thruster improvements; and c) power processing. The following is a summary of these technologies:

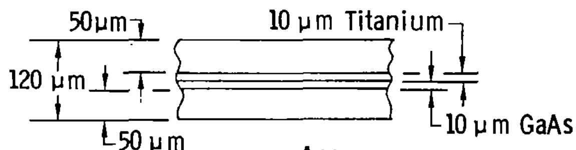
- o GaAs = accelerated technology solar array
  - o Thin film, 2 mil covers, 384 g/m^2, 234.1 W/m^2
  - o Cost up by 50%

- o Thruster options
  - o 100 cm (39 kg), 80 amp beam current
  - o 100 cm with mission life (50 kg), 150 amp beam current
  - o 100 cm with mission life and self-regulating characteristics (50 kg), 150 amp beam per unit

- o Power processing option
  - o Direct - Open Loop - Power from array

Fig. 3.1.10-1 summarizes the GaAs accelerated technology array characteristics and performance. A key assumption in its evaluation was the cost of this array relative to a silicon array. Many projections with substantial variation in costs are available. A 50% surcharge relative to silicon was used. This surcharge offsets the reduction in array due to the higher performance of the GaAs. Since approximately half of the production costs of the vehicle are array costs this offset dominates the trade. This bias toward production costs in fact dominates all the trades as shown below.

- ANNEALABLE, THIN FILM, GaAs BLANKET



COMPONENT	MATERIAL	$g/m^2/mil$ *	Thickness (mil)	Approx Area Factor	Mass (g/m^2)
COVER	BORO SILICATE MICROSHEET	55	2.0	1.0	110.0
SOLAR CELL	GaAs	135.13	0.4	0.96	51.9
SUBSTRATE	TITANIUM	114.3	0.4	0.96	43.9
INTERCONNECTS	COPPER	227.1	0.4	0.20	18.2
BACKING/SHIELDING	MICROSHEET	55	2.0	1.0	110.0
		SUBTOTAL			334.0
		TOLERANCES & INSTALLATION (15%)			50.1
		TOTAL			384.1

	EFFICIENCY	POWER
BASIC CELL EFFICIENCY [AM-0, 25°C]	20.0%	270.6 w/m^2
BLANKET PERFORMANCE FACTORS	18.2%	246.0 w/m^2
THERMAL DEGRADATION	17.3%	234.1 w/m^2
RADIATION DEGRADATION	13.8%	186.7 w/m^2

$$\text{MASS-TO-POWER RATIO} \quad \frac{384.1\ g/m^2}{186.7\ w/m^2} = 2.06\ g/w \text{ or } kg/kw$$

*(1 mil = 0.001 in. = 0.0254 mm.)

Figure 3.1.10-1. - Accelerated Technology Ga As Solar Array o COST SUMMARY ($ Millions)

	New Baseline	GaAs Replaces Silicon	Baseline with 100 cm Thrusters	Baseline w/100 cm Thruster w/Long Life	GaAs Array 100 cm Long Life Thruster Direct Power Processing
TOTAL LCC	3 276	3 334	3 492	3 348	3 236
DDT&E	388	394	463	617	596
PRODUCTION	1 930	2 009	1 964	1 963	1 911
OPERATIONS	958	930	1 063	767	728
PROPELLANT	98	92	97	96	88

o GaAs—reduced overall size offset by higher array cost.

o Thruster Improvements - improvements in thrust density/efficiency offset by increased DDT&E and higher production costs due to production rate sensitivity except for:

self-regulating feature which allows open loop direct solar array power supply

o Findings - This vehicle requires production oriented technology—not performance.

Thrusters and Power Processing

Thruster improvements started with a 100 cm resize whose efficiency and thrust density were both improved and progressed through further beam current increases with extended mission life. An initial improvement postulated was the capability of the thrusters to operate with a direct array power supply. This open loop operations mode requires a self-regulating thruster characteristic noted above. These improvements led to a reduction in operating costs. However, the increased DDT&E more than offset this reduction. Since production costs are rate sensitive the potential costs improvements due to reduced thruster requirements were lost in the increased production charges commensurate with the decreased production rates. As a result, the thruster improvements did not result in an overall costs reduction with the exception of the combined elimination of the power processing equipment and the self-regulating thruster option.

3.2 ADVANCED TRANSPORTATION SYSTEM

This section addresses the gains to the SSTO, HLLV, POTV, and LCOTV vehicles resulting from the combined accelerated technologies.

3.2.1 Priority Cargo Launch Vehicle (SSTO)

3.2.1.1 Technology Impact

As discussed in section 3.1, all technology advancements assessed individually paid off on this vehicle. The SSTO described in this section incorporates all of these items except for the long life SSME which was precluded by the dual fuel, dual expander (DF/DE) engine. The sygnergistic impact of these technology advancements was dramatic. The change in the vehicle is graphically shown in figure 3.2.1-1. This vehicle, marginal with normal growth technology, is now a realistic concept.

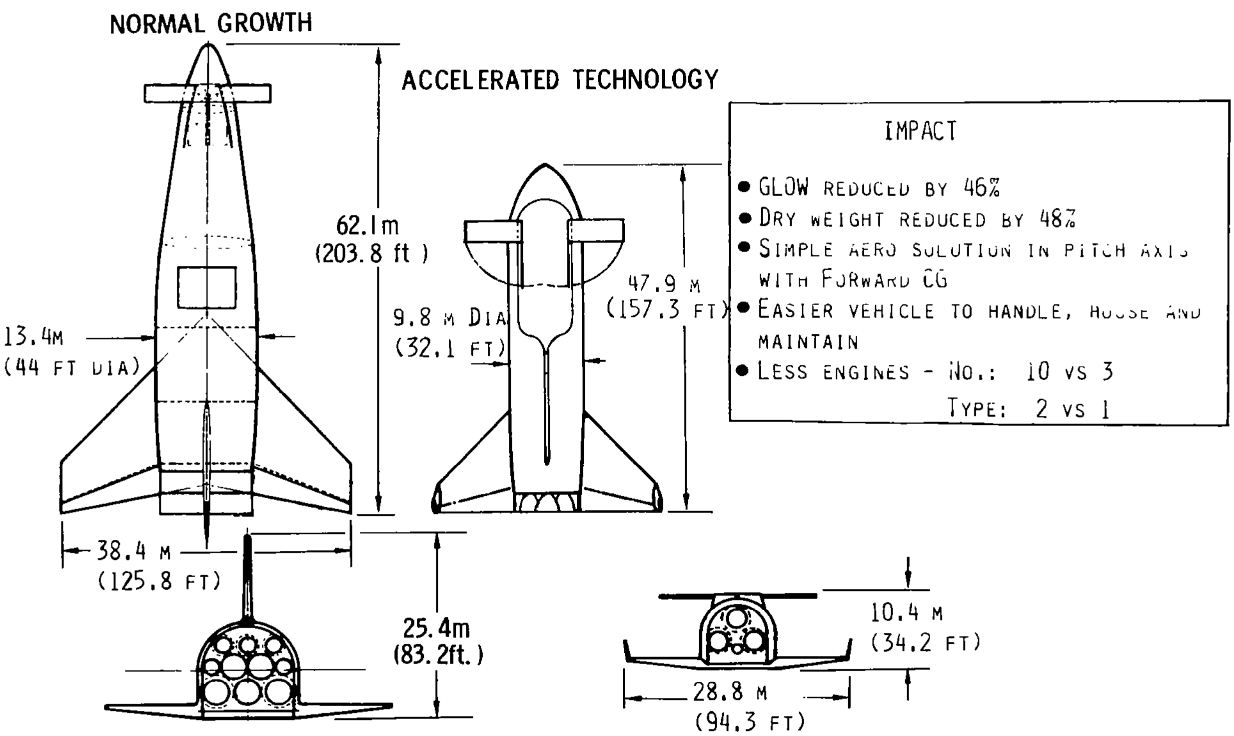
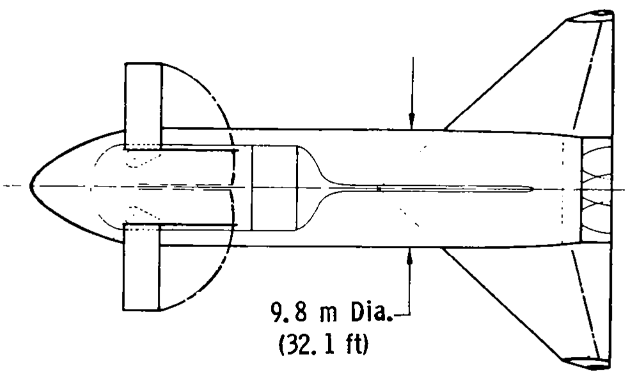


Figure 3.2.1-1. - Accelerated Technology Impact on SSTO

With the considerable reduction in GLOW, the number of DF/DE engines required dropped to only 3. This reduced number of engines coupled with the weight reduction from removal of the vertical tail resulted in a forward CG shift of 10% of body length. The aero solution at this forward cg is relatively straightforward and did not require the extendable body flap. The wing configuration is a simple delta without trailing edge sweep. The subsonic trim canard did have to increase in size. This canard growth is a result of the combined cg shift and shortened body. It is probable that the canard could be reduced in size if an optimum configuration were developed. It is also probable that slush hydrogen would not be incorporated. This is based on the reduced propellent requirements and revised tank configuration (no taper) for this advanced vehicle.

3.2.1.2. Configuration Description

The configuration of the SSTO is presented in figure 3.2.1-2 with the overall geometry noted. Roundtrip payload capability is 13 605 kg (30 000 lb). The SSTO has a 3-man flight crew. Ascent thrust is provided by three DF/DE engines, each having a Mode 1 sea level thrust of 5.12×10^6N (1 150 000 lbf) and a Mode 2 vacuum thrust of 1.82×10^6N (409 900 lbf). The airframe structure consists of unpressurized structures and integral



GLOW	1,203,628 KG	(2,654,000 LB)
P/L WT. (100% RETURN)	13,605 KG	(30,000 LB)
PROP. WT - INCL FPR	1,065,216 KG	(2,348,800 LB)
INERT WEIGHT	124,807 KG	(275,200 LB)

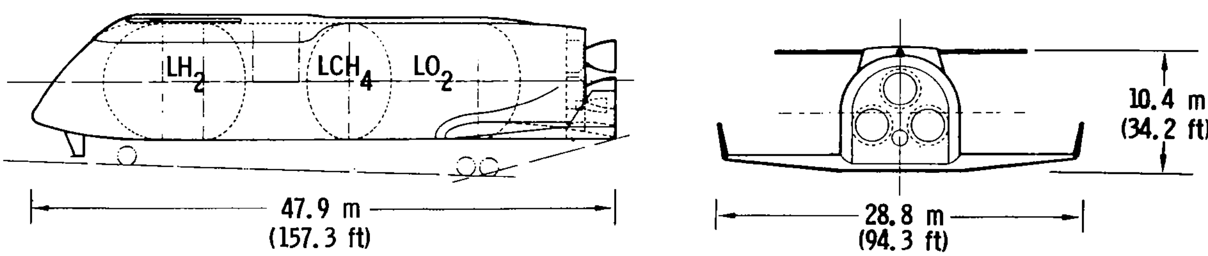


Figure 3.2.1-2. - SSTO Configuration, Accelerated Technology

main propellant tanks. The unpressurized structures are composite design and are protected from entry heating by advanced RSI. Main tankage structures are 6AL-4V(ELI) titanium sandwich for the LH_2 tank and 2219-T87 aluminum for the LO_2/LCH_4 tank. The tank sidewalls are protected from entry heating by advanced RSI over composite design standoff panels and fairing structures.

The advanced technology SSTO has the same type of ascent trajectory as the normal growth vehicle, a vertical take off, dual fuel mode using aerodynamic lifting. The SSTO lifts off at a T/W of 1.3 with both chambers of the DF/DE engines operating. Transition from Mode I to Mode II propulsion occurs when 60% of the total propellant has been burned by the LO_2/LCH_4 chambers of the engines. 83.6 percent of the total propellant is burned during the Mode I portion of the flight. Ascent performances is summarized in the table below:

T/W @ liftoff:	1.30	
Max Dynamic Pressure:	40.7 kPa	(850 psf)
Max Acceleration:	3.00 g	
Mode I Burn Time:	219.9 sec.	
Altitude @ Transition:	73 765 m	(242 011 ft)
Rel. Velocity @ Transition:	3 906 m/sec	(12 815 ft/sec)
Total Burn Time:	366.7 sec	
Insertion Orbit:	93x185 km	(50x100 n.mi.)

Trajectory parameters for the advanced technology SSTO are plotted as a function of time in figure 3.2.1-3.

A summary mass and balance statement is presented in table 3.2.1-1.

Each of the items in the summary mass and balance statement, exclusive of payload and ascent propellant, is discussed in the following paragraphs including definition of rationale for mass estimates.

Wing—The wing is a composite design with a mass of 7 465 kg (16 460 lb) and a unit mass of 22.6 kg/m^2 (4.62 lb/ft^2) based on its reference area of 331 m^2 (3 562 ft^2). The wing mass is estimated at 60% of that of an aluminum design wing sized for a 2.5g subsonic maneuver at the landing wing loading (reference wing + canard) of 331 kg/m^2 (67.8 lb/ft^2),

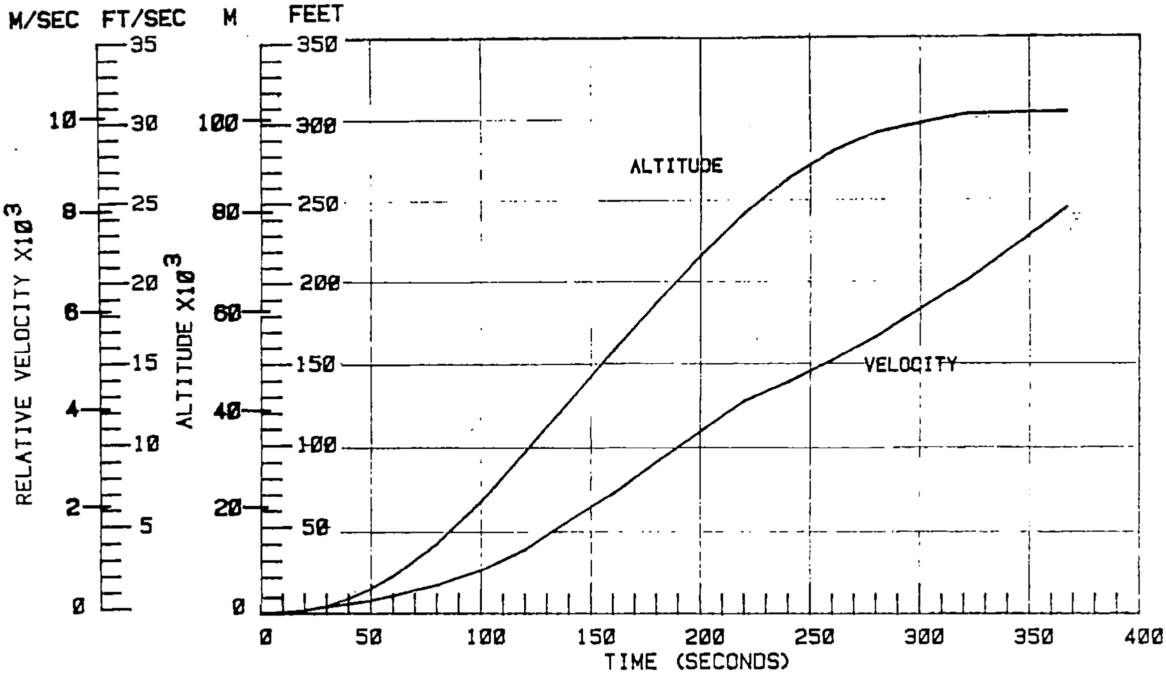


Figure 3.2.1-3a. - Advanced Technology SSTO Ascent Performance

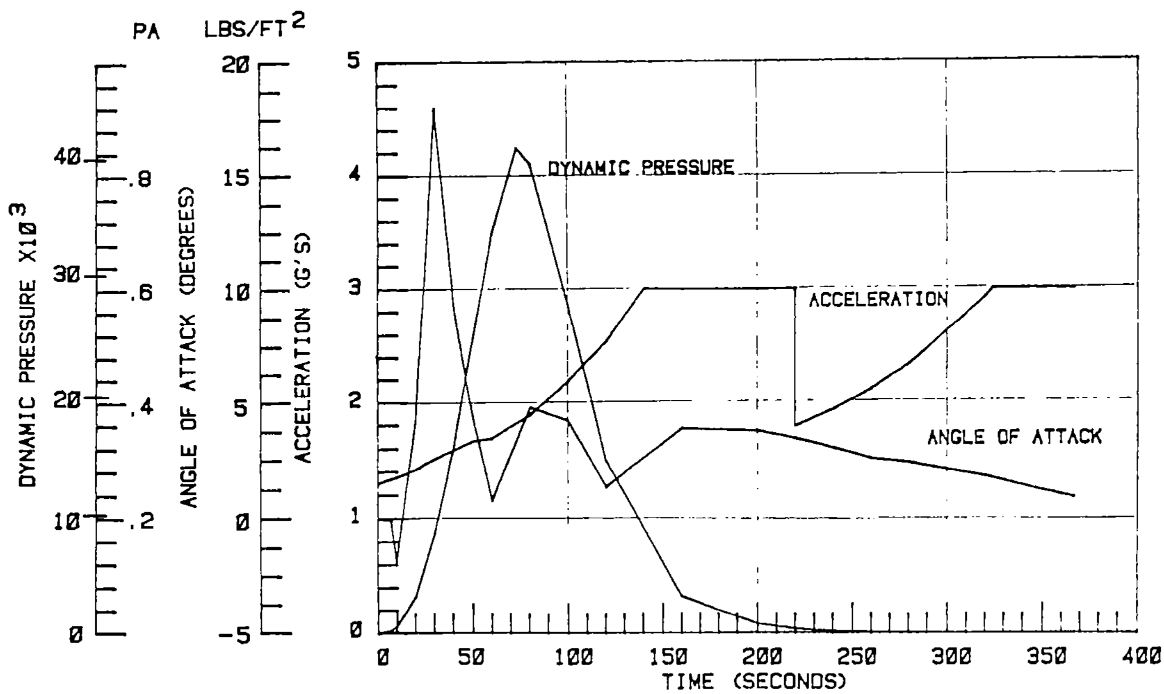


Figure 3.2.1-3b. - Advanced Technology SSTO Ascent Performance

TABLE 3.2.1-1. - SSTO Summary Mass and Balance Statement Accelerated Technology

GROUP/ITEM	MASS (kg)		XCG (m)		WEIGHT (lb)		XCG (in.)	
WING		7,465		42.42		16,460		1670
CANARD		1,560		9.65		3,440		380
WING TIPLETS		390		46.23		860		1820
YAW VENTRAL		181		4.32		400		170
BODY		37,406		26.70		82,480		1051
INDUCED ENVIRONMENTAL PROTECTION		14,875		29.85		32,800		1175
LANDING AND AUXILIARY SYSTEMS		3,864		34.57		8,520		1361
PROPULSION–ASCENT		20,132		43.26		44,390		1703
ROCKET ENGINES	13,374		44.70		29,490		1760	
ENGINE ACCESSORIES	1,742		44.70		3,840		1760	
PROPELLANT SYSTEM	5,016		38.94		11,060		1533	
PROPULSION–OMS } INTEGRATED		1,247		26.67		2,750		1050
PROPULSION–RCS } INTEGRATED		1,002		25.40		2,210		1000
PRIME POWER } INTEGRATED		381		20.07		840		790
ELECTRICAL CONVERSION & DISTR.		2,086		22.68		4,600		893
HYDRAULIC CONVERSION & DISTR.		726		38.10		1,600		1500
SURFACE CONTROLS		1,043		39.93		2,300		1572
AVIONICS		1,483		3.81		3,270		150
ENVIRONMENTAL CONTROL		789		3.81		1,740		150
PERSONNEL PROVISIONS		272		3.81		600		150
PAYLOAD PROVISIONS		159		20.07		350		790
MARGIN		10,540		31.50		23,240		1240
(DRY CONDITION)		(105,601)		(31.50)		(232,850)		(1240)
PERSONNEL		667		3.81		1,470		150
PAYLOAD–ASCENT (100% RETURN)		13,605		20.07		30,000		790
RESIDUALS–SYSTEMS/RCS/OMS/ASCENT PROPULSION		2,966		30.76		6,540		1211
RESERVES–SYSTEMS/RCS/OMS		558		20.17		1,230		794
INFLIGHT LOSSES–SYSTEMS		281		23.09		620		909
INFLIGHT LOSSES–ASCENT PROPULSION		4,245		41.30		9,360		1626
PROPELLANT–NOMINAL RCS		1,124		20.07		2,480		790
PROPELLANT–NOMINAL OMS		9,365		20.07		20,650		790
(INERT CONDITION, WITH P/L)		(138,412)		(29.59)		(305,200)		(1165)
ASCENT PROPELLANT–INCL FPR		1,065,216		32.31		2,348,800		1272
LO_2	872,227		35.54		1,923,260		1360	
LH_2–LOADED AS SLUSH	52,195		11.68		115,090		460	
LCH_4	140,794		26.03		310,450		1025	
(LIFTOFF CONDITION, WITH P/L)		**(1,203,628)**		(32.00)		(2,654,000)		(1260)

$$\text{MASS FRACTION} = \frac{\text{ASCENT PROPELLANT MASS}}{\text{LIFTOFF MASS, LESS P/L}} = 0.895$$

a 406 K (270°F) surface temperature during the maneuver, and a 1,000 hour design accumulated time at surface temperatures above 394 K (250°F). The wing mass includes the box body-carry-through section and main gear installation provisons. A constant t/c of 10% was used. The ratio of elevon area to wing reference area is 11.9%.

Canard—The subsonic trim canard group consists of an aerosurface, hinge/attachment fittings and deployment/latch mechanisms. The aerosurface is a composite design with a unit mass of 17.6 kg/m^2 (3.6 lb/ft^2) based on its exposed planform area. The aerosurface unit mass is estimated at 60% of that of an aluminum design. Fittings and mechanisms unit mass is estimated at 19.5 kg/m^2 (4.0 lb/ft^2) of aerosurface area. The ratio of canard exposed planform area to wing reference area is 12.6%. Total mass is 1 560 kg (3 440 lb).

Wing Tiplets—This group consists of the wing tiplets and associated wing tip reinforcement. The wing tiplets, which are 30% rudder, are of composite design with a unit mass of 14.6 kg/m^2 (3.0 lb/ft^2). The tiplets unit mass is estimated at 60% of that of an aluminum design. Wing tip reinforcement unit mass is estimated at 7.3 kg/m^2 (1.5 lb/ft^2) of tiplet area. The ratio of tiplet area to wing reference area is 2.67% (per tiplet). Total mass is 390 kg (860 lb).

Yaw Ventral—The subsonic yaw ventral group consists of a deployable all-movable aerosurface and its installation provisions. The installation provisions include the aerosurface torque tube, tube support fittings, rails, rail support structures, deployment/retract mechanisms, and a door installation. The aerosurface is a composite design with a unit mass of 14.6 kg/m^2 (3.0 lb/ft^2) based on its exposed planform area. The aerosurface unit mass is estimated at 60% of that of an aluminum design. Installation provisions unit mass is estimated at 53.7 kg/m^2 (11.0 lb/ft^2) of aerosurface area. The ratio of yaw ventral exposed area to wing reference area is 0.66%. Total mass is 101 kg (400 lb).

Body—The body group consists of the following: nose section, crew module, LH_2 tank, payload bay section, payload bay doors, LO_2/LCH_4 tank, aft body section thrust structure, body flap, and fairing structures. Total body mass is 37 406 kg (82 480 lb).

- Nose Section—The nose section consists of a semimonocoque shell structure, nose gear installation provisions, and crew module peculiar items. The shell structure is of composite design with a unit mass of 9.8 kg/m^2 (2.0 lb/ft^2). This unit mass is 60% of the unit mass of the aluminum design shell structure on the Shuttle Orbiter.

The nose gear installation provisions consist of the wheel well, door and mechanisms, and support struts, and are of composite design with a unit mass of 0.21% of design landing weight. This unit mass is 60% of the unit mass of the aluminum design installation provisions on the Shuttle Orbiter. Crew module peculiar items consist of a thermal windshield, observation windows, star tracker door and mechanisms, and miscellaneous access panels, and have a mass 254 kg (560 lb) based on Shuttle Orbiter. Nose section total mass is 2 771 kg (6 110 lb).

- Crew Module—The crew module is similiar to that of the Shuttle Orbiter but is smaller in volume. It is an all-welded 2219-T87 aluminum pressure vessel which incorporates a windshield, observation windows, and side hatch and mechanisms. (Because the payload bay is not adjacent to the crew module, items such as aft observation windows, rear hatch and airlock, were not provided.) The basic pressure vessel mass is estimated at 75% of its Shuttle Orbiter counterpart. The mass of the window and hatch installations is 630 kg (1 390 lb) based on Shuttle Orbiter. Crew module total mass is 2 104 kg (4 640 lb).

- LH_2 Tank—The tank containing the slush hydrogen is an all-welded 6AL-4V(ELI) titanium sandwich pressure vessel with a ring stiffened sidewall. The tank is designed by cryo temperature proof test conditions corresponding to 3g boost onset with maximum ullage pressure of 159 kPa (23 psia). To satisfy a 500 mission requirement with minimal weight impact, the tank is designed for 200 cycles to failure and, following the intial proof test, is reproofed every 100 missions. The inner face of the sandwich is sized to carry 100% of proof test pressure. The outer face is minimum gage at 0.030 cm (0.012 in), including sidewall regions to which TPS subpanels and fairing structures are attached. Tank mass is 4 281 kg (9 440 lb), 8.20% of the mass of the slush hydrogen.

- Payload Bay Section—This section is a semimonocoque shell structure of composite design. It has a unit mass of 16.6 kg/m^2 (3.4 lb/ft^2) exclusive of the payload bay door area. This unit mass is estimated at 60% of that of an aluminum design shell structure. The payload bay section mass is 5 755 kg (12 690 lb).

- Payload Bay Doors—Approximately 21.4 m^2 (230 ft^2) of door area is provided. The doors are of composite design and have a unit mass of 15.1 kg/m^2 (3.1 lb/ft^2) based on Shuttle Orbiter. Door mass is 322 kg (710 lb).

- LO_2/LCH_4 Tank—The tank containing the liquid oxygen and liquid methane is an all-welded 2219-T87 aluminum pressure vessel with a common dome of sandwich construction and a stiffened sidewall in the oxygen compartment. The tank is designed by room temperature proof test conditions corresponding to 3g boost onset with maximum ullage pressure of 159 kPa (23 psia). To satisfy a 500 mission requirement with minimal mass impact, the tank is designed for 200 cycles to failure and, following the initial proof test, is reproofed every 100 missions. The mass of the liquid oxygen compartment, exclusive of the common dome, is 9 070 kg (20 000 lb), 1.04% of the mass of the liquid oxygen. The mass of the liquid methane compartment, inclusive of the common dome, is 3 170 kg (6 990 lb), 2.25% of the mass of the liquid methane. Both compartments have slosh baffles. Total tank mass is 12 240 kg (26 990 lb).

- Aft Body Section—This section consists of a semimonocoque shell structure extending from the LO_2/LH_4 tank to the engine support plane, a 1.5 m (5 ft) long fairing structure located aft of the engine support plane, and a base heat shield support structure installation located at the engine support plane. The structures are all of composite design with unit masses of 20.0 kg/m^2 (4.1 lb/ft^2), 10.7 kg/m^2 (2.2 lb/ft^2), and 10.3 kg/m^2 (2.1 lb/ft^2), respectively. These unit masses are estimated at 60% of those of aluminum design structures. In addition, the unit mass of the shell structure reflects a 5% reduction due to the absence of a vertical tail. Aft body section total mass is 5 696 kg (12 560 lb).

- Thrust Structure—The thrust structure is a beam system of composite design which transmits thrust loads from the three DF/DE engines to the aft body section. Maximum vacuum thrust of the three engines is 17.0×10^6N (3.826×10^6 lbf). The thrust structure has a mass of 2 268 kg (5 000 lb) which is 60% of that of an aluminum design. Its units mass is 0.00013 kg/N (0.00131 lb/lbf).

- Body Flap—The constant chord body flap provides for pitch trim control and thermally shields the main engines during entry. The flap group consists of the basic flap and the hinge line support fittings/closeout provisions. Unit masses of the foregoing items are 12.2 kg/m^2 (2.5 lb/ft^2) and 5.8 kg/m^2 (1.2 lb/ft^2), respectively. The basic flap is of composite design and its unit mass is 76% of the unit mass of the aluminum design basic flap on the Shuttle Orbiter. The unit weight of the hinge line support fittings/closeout provisions is based on Shuttle Orbiter. Total body flap mass is 576 kg (1 270 lb).

- Fairing Structures—This group consists of a wing-to-body fairing, a canard fairing (forward of payload bay doors), and a LH_2 main delivery line fairing (aft of payload bay doors). The fairings are of composite design and include approximately 179 m^2 (1 930 ft^2) of fixed fairing at 5.2 kg/m^2 (1.07 lb/ft^2) and approximately 66 m^2 (710 ft) of removable fairing at 6.9 kg/m^2 (1.41 lb/ft^2). The unit masses are estimated at 60% of those of aluminum design fairings. Total fairing structure mass is 1 392 kg (3 070 lb).

Induced Environmental Protection—This group consists of the external TPS system plus the internal provisions for thermal control, purge/vent/drain, and window conditioning. The external TPS system utilizes advanced RSI on all areas including body chine and wing leading edge. The density of the RSI is 120 kg/m^3 (7.5 lb/ft^3) except in the body chine and wing leading edge areas where the density is 320 kg/m^3 (20.0 lb/ft^3). Based on an entry planform loading of 205 kg/m^2 (42.0 lb/ft^2) and a 2,000 km crossrange, the TPS unit masses based on modified Shuttle Orbiter data are: outboard wing panels, 18.6 kg/m^2 (3.82 lb/ft^2) of wing exposed planform area; body, including base region, 13.9 kg/m^2 (2.84 lb/ft^2) of body planform area; wing tiplets, 14.6 kg/m^2 (3.0 lb/ft^2) of tiplet side planform area. In addition, for the body, the main tankage sidewall regions not covered by fairing structures are covered by composite design standoff subpanels having an estimated average unit mass of 5.9 kg/m^2 (1.2 lb/ft^2). The internal provisions have an estimated mass of 1 134 kg (2 500 lb). The induced environmental protection total mass is 14 875 kg (32 800 lb).

Landing and Auxiliary Systems—In addition to landing gear, this group includes a landing drag device. The landing gear design incorporates the use of B-1 cord tires, servo-design shock struts with extensive use of boron-aluminum, and 55 200 kPa (8 000 psi) hydraulic operated mechanisms. The unit masses of the landing gear and drag service are estimated at 2.8% and 0.33%, respectively, of design landing mass based on modified Shuttle Orbiter data and past studies.

Ascent Propulsion—The ascent propulsion group consists of rocket engines, engine accessories, and the propellant system. Total mass is 20 132 kg (44 390 lb).

- Rocket Engines—Ascent thrust is provided by three DF/DE engines. Total liftoff thrust is 15.3×10^6N (3 450 000 lb). Pertinent engine characteristics are presented in Section 2.1.2. Total dry engine mass is 13 374 kg (29 490 lb).

- Engine Accessories—Accessories for each of the three DF/DE engines consist of a gimbal system, hydraulic supply, powerhead heat shield, and nitrogen purge provisions. Based on the use of 55 200 kPa (8 000 psi) hydraulic components and modest improvements in heat shield design, the engine accessories mass for a gimballed DF/DE engine is estimated at 13% of dry engine mass based on modified accessories mass data for a standard SSME. Total accessories mass is 1 742 kg (3 840 lb).

- Propellant System—This subgroup consists of the propellant delivery system, tank pressurization (autogenous) and vent systems, umbilical hardware, and associated installation hardware. Individual DF/DE engine feed line diameters are 0.526 m (20.7 in) for LO_2, 0.274 m (10.8 in) for LCH_4, and 0.267 m (10.5 in) for LH_2. The LO_2 lines to each engine are direct tank-to-engine lines. The LCH_4 feed lines are fed by a single main line of 0.386 m (15.2 in) diameter extending through the LO_2 tank. The LH_2 feed lines are fed by a single main line of 0.376 m (14.8 in) diameter. Using the Shuttle Orbiter propellant system as a data base, and allowing for a 30% mass reduction relative to current state-of-the-art (including composite overwrapped lines), the unit mass of the propellant system components located in the aft body region is 1 391 kg (3 067 lb) per DF/DE engine. Using the ET tank propellant system as a data base, and allowing for a 40% mass reduction relative to current state-of-the-art (including composite overwrapped lines), the mass of the propellant system components located forward of the aft body region is 844 kg (1 860 lb). Total propellant system mass is 5 016 kg (11 060 lb).

OMS Propulsion—The orbital maneuver system consists of engines and accessories, and associated propellant delivery and storage elements. The LO_2/LH_2 OMS is integral with the reaction control system, (RCS) and auxiliary power system (APS) in that the RCS propellant and APS reactant are stored in the OMS tanks. A simplified schematic, pertinent characteristics, discussion data, and weight equations are presented in Appendix D of Volume III. Based on an OMS total installed vacuum thrust of 89 000 N (20 000 lbf), nominal propellant requirements of 9 365 kg (20 650 lb) for OMS, 1 125 kg (2 480 lb) for RCS, and 159 kg (350 lb) for APS, plus an OMS/RCS/APS root-sum-square reserve propellant requirement of 531 kg (1 170 lb), the OMS dry mass is 1 247 kg (2 750 lb).

RCS Propulsion—A LO_2/LH_2 reaction control system provides for vehicle orientation prior to entry and for control during entry. The RCS is integral with the OMS and APS in that the RCS propellant and APS reactant are stored in the OMS tanks and are routed to the RCS thrusters and APS auxiliary power units, respectively, via the bellows accumulators in the RCS. Based on a RCS total installed vacuum thrust of 211 000 N (47 360 lbf) and nominal propellant requirements of 1 125 kg (2 480 lb) for RCS and 159 kg (350 lb) for APS, the RCS dry mass is 1 002 kg (2 210 lb).

Prime Power (Auxiliary Power System)—Power source elements consist of three independent LO_2/LH_2 powered APU's, auxiliary batteries, and alternators. The APS is integral with the OMS and RCS in that its propulsive grade reactant is stored in the OMS tanks and is routed to the APU's via bellows accumulators in the RCS. Based on a peak power requirement of 231 kW (310 hp) hydraulic plus 15 kW (20 hp) electric, the APS dry mass including reactant feed system and exhaust system is 381 kg (840 lb).

Electrical Conversion and Distribution—This group consists of the power conversion, conditioning, and cabling elements for a 270 Vdc electrical system. Estimated system mass is 2 086 kg (4 600 lb).

Hydraulic Conversion and Distribution—This group consists of the hydraulic power supply and distribution equipment, including a water boiler temperature control system, for a 55 200 kPa (8 000 psi) hydraulic system. Estimated system mass including hydraulic fluid is 726 kg (1 600 lb).

Surface Controls—The actuation systems for the aerodynamic control surfaces are included in this group, as are the cockpit controls. The actuation system unit masses, based on modest improvements in systems design relative to Shuttle Orbiter are: elevons, 11.7 kg/m^2 (2.4 lb/ft^2); body flap, 12.2 kg/m^2 (2.5 lb/ft^2); wing tiplet rudders, 11.7 kg/m^2 (2.4 lb/ft^2); and yaw ventral, 13.2 kg/m^2 (2.7 lb/ft^2). Total surface controls mass, including 27 kg (60 lg) of cockpit controls, is 1 043 kg (2 300 lb).

Avionics—The avionics group includes elements for guidance, navigation and control, communications and tracking, displays and control, instrumentation, and data processing and software. Total mass is 1 483 kg (3 270 lb) or 51% of Shuttle Orbiter avionics mass, based on use of advanced avionics and reduced power requirements.

Environmental Control—The environmental control group maintains a habitable environment and provides basic life support functions for the crew, and maintains a conditioned thermal envionment for the avionics. Estimated system mass including closed loop fluids is 789 kg (1 740 lb).

Personnel Provisions—This group consists of the fixed life support system, personnel accommodations (seats), and furnishings for the 3-man flight crew. Total mass is 272 kg (600 lb).

Payload Provisions—This group consists of fixed scar items and removable provisions with unit masses of 0.30% and 0.85%, respectively, of maximum ascent payload mass, based on Shuttle Orbiter. Total mass is 159 kg (350 lb).

Margin—A margin allowance of approximately 10% of subsystems dry mass has been incorporated. Total mass is 10 540 kg (23 140 lb).

Personnel—This group consists of the 3-man flight crew and their accessories. Unit masses for a crewman and his accessories are 86 kg (190 lb) and 136 kg (300 lb), respectively, based on Shuttle Orbiter. Total mass is 667 kg (1 470 lb).

Nominal RCS Propellant—The RCS delta-V budget is 38.5 m/s (125 ft/sec) and consists of the following:

Trim Burns - Post Ascent	1.5 m/s (5 ft/sec)
Trim Burns - 500 km Orbit	17 m/s (55 ft/sec)
Trim Burns - Entry	20 m/s (65 ft/sec)

The entry trim burns delta-V includes an allowance of 6 m/s (20 ft/sec) due to the absence of a vertical tail. Based on a specific impulse of 420 sec for the LO_2/LH_2 thrusters, and consideration of vehicle sequential mass, the total nominal propellant mass is 1 124 kg (2 480 lb).

Nominal OMS Propellant—The OMS delta-V budget is 336 m/s (1 103 ft/sec) and consists of the following:

Circularization into 185 km Parking Orbit from 93 x 185 km Launch Orbit	28 m/s (91 ft/sec)
Insertion into 185 x 500 km Transfer Orbit from 185 x km Parking Orbit	90 m/s (297 ft/sec)
Circularization into 500 km Orbit from 185 x 500 km Transfer Orbit	90 m/s (294 ft/sec)
Trim Burns in 500 km Orbit	11 m/s (37 ft/sec)
Deorbit from 500 km Orbit	117 m/s (384 ft/sec)

Based on a specific impulse of 473 sec for the LO_2/LH_2 OMS engine, and consideration of vehicle sequential mass, the total nominal propellant mass is 9 365 kg (20 650 lb).

Systems Inflight Losses—This group consists of the following nominal usages: O_2/H_2 for hydraulic and electric power, 159 kg (350 lb); GO_2, GN_2, H_2O, and NH_3 for crew module environmental control and crew life support, 45 kg (100 lb); and H_2O for hydraulic cooling, 77 kg (170 lb). Total estimated mass is 281 kg (620 lb).

Ascent Propulsion Inflight Losses—All ascent propulsion liquid residuals are jettisonable and are included in this group. Total mass is 4 245 kg (9 360 lb). Pertinent line size data is included under ascent propulsion-propellant system.

- Propellant in Engines at Cutoff—Trapped propellant mass is estimated at 295 kg (650 lb) per DF/DE engine. Total mass is 885 kg (1 950 lb).

- Oxidizer in Engine Feed Lines at Cutoff—The LO_2 lines to each DF/DE engine are direct tank-to-engine lines. The lines average 4.0 m (13 ft) in length and contain approximately 3.0 m (10 ft) of LO_2 at nominal engine cutoff. The LO_2 unit mass is 742 kg (1 637 lb) per DF/DE engine. Total mass is 2 226 kg (4.910 lb).

- Fuel in Engine Feed Lines at Cutoff—An average 3.7 m (14 ft) of LH_2 line and 4.9 m (15 ft) of LCH_4 line are located between the main delivery lines and each engine. These lines are full at engine cutoff. The LH_2 and LCH_4 unit masses are 17 kg (37 lb) and 116 kg (256 lb), respectively, per DF/DE engine. Total mass is 399 kg (880 lb).

- Fuel in Main Delivery Lines at Cutoff—There is one main LH_2 line and one main LCH_4 line. At engine cutoff under nominal conditions, these lines contain approximately 3.0 m (10 ft) of fluid, exclusive of bias fuel. The LH_2 and LCH_4 masses are 25 kg (55 lb) and 152 kg (335 lb), respectively. Total mass is 177 kg (390 lb).

- Bias Fuel—A single-engine reference bias fuel allowance of 0.50% was used. For multiple engines, this allowance was factored by the inverse of the square root of the number of engines. The resulting LH_2 and LCH_4 bias allowances for the three DF/DE engines are 0.29%. The LH_2 and LCH_4 masses are 150 kg (330 lb) and 408 kg (900 lb), respectively. Total mass is 558 kg (1 230 lb).

Reserves—This group of reserve allowances for RCS, OMS, hydraulic and electric power, crew module environmental control and life support, and hydraulic cooling. (The flight performance reserve is 9.85% of total ideal velocity change and is included in ascent propellant weight.) The RCS, OMS, and power system are integrated LO_2/LH_2 systems in which the oxidizer/fuel ratio (O/F) mixture ratio of the nominal propellants is 4:1, 6:1, and 1:1, respectively. Because these systems have common propellant storage tanks, the reserve allowances peculiar to dedicated systems were combined on a root-sum-square basis with consideration of the above O/F mixture ratios. (The dedicated systems reserve allowances for RCS, OMS, and prime power are 20%, 5%, and 50%, respectively.) The resulting integrated systems reserve allowance is 531 kg (1 170 lb) with an O/F mixture ratio of 4.9:1. Reserve allowances for environmental control/life support and hydraulic cooling are 30% and 20%, respectively. Total reserves mass is 558 kg (1 230 lb).

Residuals—This group consists of the fluids and gases, onboard at landing, for ascent propulsion, RCS, OMS, hydraulic and electric power, crew module environmental control/life support, and hydraulic cooling. Total mass is 2 966 kg (6 540 lb).

- Ascent Propulsion Residuals—Only main tank pressurants are onboard at landing. The pressurant masses are based on maximum ullage pressures of 159 kPa (23 psia) and mean temperatures for the GO_2, GH_2, and GCH_4 of 264 K (475°R), 189 K (340°R), and 222 K (400°R), respectively. The masses of the GO_2, GH_2, and GCH_4 are 1 868 kg (4 120 lb), 136 kg (300 lb), and 481 kg (1 060 lb), respectively. Total pressurant mass is 2 485 kg (5 480 lb).

- Other Residuals—The residuals mass for the integrated RCS/OMS/prime power is 453 kg (1 000 lb). Residual allowances for crew module environmental control/life support and hydraulic cooling are 25% and 15%, resectively, of usable (nominal+reserve) quantities. Total mass for the subgroup is 481 kg (1 060 lb).

3.2.1.3 Life Cycle Costs

The advanced technology SSTO has a LCC of $12650.7 million, distributed by phase and subphase as shown in table 3.2.1-2. All costs reported in this section are in constant 1977 dollars.

DDT&E Phase

A breakdown of the SSTO DDT&E Phase cost, totalling $2980.8 million, is presented in table 3.2.1-3. The cost has increased from the baseline SSTO due to the inclusion in the Engineering subphase of the D&D costs for the Dual Fuel/Dual Expander engine.

TABLE 3.2.1-2. - SSTO - COST SUMMARY COMPARISON ($ IN MILLIONS)

	NORMAL TECH.	ADV. TECH.
TOTAL PROGRAM	17,912. 90	12,650.65
DDT&E	2,714.47	2,980.77
PROGRAM MANAGEMENT	86.98	65.51
ENGINEERING	833.50	1,848.44
MANUFACTURING	1,373.58	757.16
TEST	420.41	309.66
PRODUCTION	4,245.06	2,229.73
PROGRAM MANAGEMENT	377.48	183.88
SUSTAINING ENGINEERING	87.27	51.01
MANUFACTURING	3,780.32	1,994.85
PROD. TOOLING & S.T.E.	1,382.87	593.53
FLT. HARDWARE & SPARES	2,397.45	1,401.32
OPERATIONS	10,953.37	7,440.15
OPERATIONS SUPPORT	6,700.19	3,860.45
PROGRAM SUPPORT	1,576.75	1,577.20
SPARES PROCUREMENT	5,123.44	2,283.25
LAUNCH SUPPORT	4,253.18	3,579.70
OPERATIONS	3,177.18	2,998.87
PROPELLANT	1,076.00	580.83
• COST/FLT	$3.79M	$2.58M
• COST/KG	$278.	$189.
• ADVANCED TECHNOLOGY SAVINGS:	$5,262.25M	

TABLE 3.2.1-3. - ADVANCED TECHNOLOGY SSTO DDT&E COSTS

DDT&E	2980.8			
PROGRAM MANAGEMENT		65.5		
ENGINEERING		1848.4		
SYSTEMS ENGINEERING & INTEGRATION			38.5	
SOFTWARE ENGINEERING			14.1	
DESIGN & DEVELOPMENT			1795.8	
STRUCTURE				120.3
TPS				64.1
ASCENT PROPULSION				38.6
LANDING & AUX. SYSTEMS				1236.6
AUX. PROPULSION				9.6
PRIME POWER				3.1
ELECTRICAL CONVERSION & DISTRIBUTION				16.3
HYDRAULIC CONVERSION & DISTRIBUTION				11.8
SURFACE CONTROLS				9.2
AVIONICS				151.2
ECS				8.8
PERSONNEL PROVISIONS				5.0
GSE				121.2
MANUFACTURING		757.2		
DEVELOPMENTAL TOOLING & S.T.E.			78.5	
TEST HDWE & SPARES			678.7	
TEST		309.7		
SYSTEMS TEST OPERATIONS			289.3	
FLIGHT TEST OPERATIONS			20.4	

The D&D costs for the LO_2/LH_2 RCS thruster and the OMS engine have been charged to the HLLV and POTV LCC's, respectively. All other systems have the full D&D costs associated with unique development. There are no other applications of inherited technologies or components.

The manufacturing cost of two vehicles used for propulsion/dynamic tests and flight tests is included in the manufacturing subphase of DDT&E. These vehicles are refurbished during the production phase for use in the operational fleet.

Production Phase

The total production phase cost for the advanced technology SSTO is $2229.7 million. This is the amount required to produce a fleet of 8 vehicles and the necessary tooling for the fleet and the operational spares.

The following TFU costs where used in estimating the production phase cost:

- Vehicle - $237.29 million
- Dual Fuel/Dual Expander engine - $18.49 million
- OMS Engine - $1.83 million
- GSE - $54.62 million

In the flight hardware and spares cost is included $118.7 million for refurbishment of the two test vehicles.

Operations Phase

The advanced technology SSTO has the same mission model as the baseline SSTO, resulting in a fleet size of 8 vehicles and 2889 total flights. The operations phase cost for the 15 year flight program is $7440.2 million.

Hardware costs for this phase are based on the same groundrules used for the Baseline launch vehicles shown in 2.2.1.4 LCC, operations phase. The Dual Fuel/Dual Expander engine has the same maintenance characteristics as the LO_2/LCH_4 main engine.

3.2.2 Heavy Lift Launch Vehicle (HLLV)

3.2.2.1 Technology Impact

This vehicle, although considerably improved, did not show the dramatic changes the SSTO did. Figure 3.2.2-1 summarizes the key changes and graphically depicts the size change.

Although the total number of engines was reduced from 21 to 15, the vehicle CG shifts were only 1% on the booster and 4% of body length on the second stage. The boosters CG was forward of the orbiter as a result of carrying flyback fuel forward and therefore was impacted to a lesser degree. The aero solution for these vehicles includes wing trailing edge sweep, an extendable body flap and, a cambered nose to reduce its hypersonic effectiveness. This last feature resulted in a revised body configuration for these

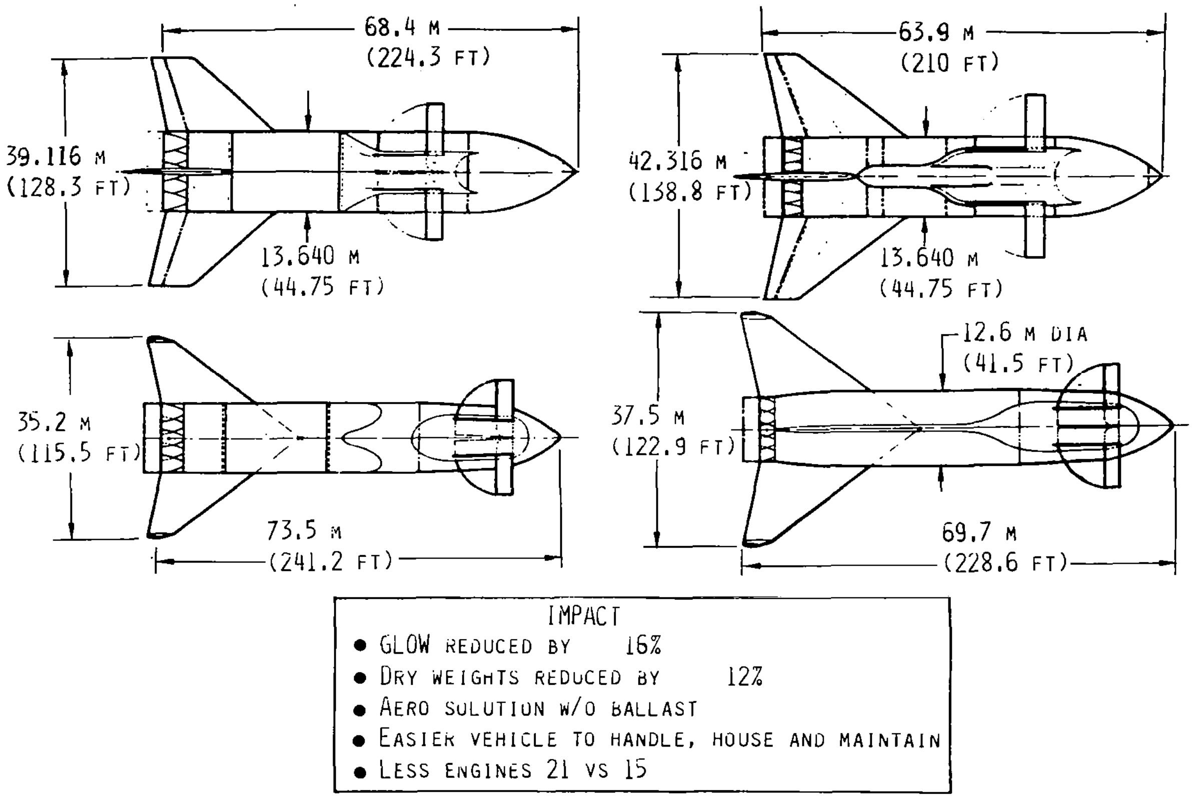


Figure 3.2.2-1 - Accelerated Technology Impact on HLLV

vehicles. The ogive nose tanks gave way to conventional tanks with a slight taper and dry bay nose airing. Compensating weight changes were accrued by moving the canard forward, at a reduced size, and reducing the amount of body fairing.

The overall size of the normal growth and accelerated technology vehicles is still very similar because of a change in body fineness ratio and a revised propellent split. The revised propellent split added fuel to the booster to accommodate the requirements of the dual fuel, dual expander engine. This engine was used since its DDT&E bill was paid by the SSTO and it does offer a performance gain. Contributing to the fineness ratio change were the reduced diameter tanks (due to reduced propellent quantity) and the dry bay nose configuration.

The advanced technology HLLV mission, delivery of large payloads to a 28.5^{o} 500 km orbit, is unchanged from the normal growth baseline. It has the same type of ascent trajectory: initial pitchover and ballistic trajectory during boost and an angle of attack controlled trajectory until insertion into the 110 x 50 km beam launch orbit. There is a slight change in the trajectory due to the replacement of the LO_2/LCH_4 booster engines with DF/DE engines. The following table lists the key ascent performance characteristics:

T/W @ Liftoff:	1.290
Max. Dynamic Pressure:	30.94 kPa (646.2 psf)
Max. Acceleration (both stages:	3.00 g
Total Burn Time:	486.0 sec

Conditions @ Staging

Time:	164.7 sec
Altitude:	58 510 m (191 962 ft.)
Velocity (Rel.):	2 178 m/sec (7 145 ft/sec)
Dynamic Pressure:	.94 kPa (19.7 psf)
Flt. Path Angle:	16.06°
2nd Stage T/W	1.03

Figure 3.2.2-2 plots HLLV ascent trajectory parameters as a function of time.

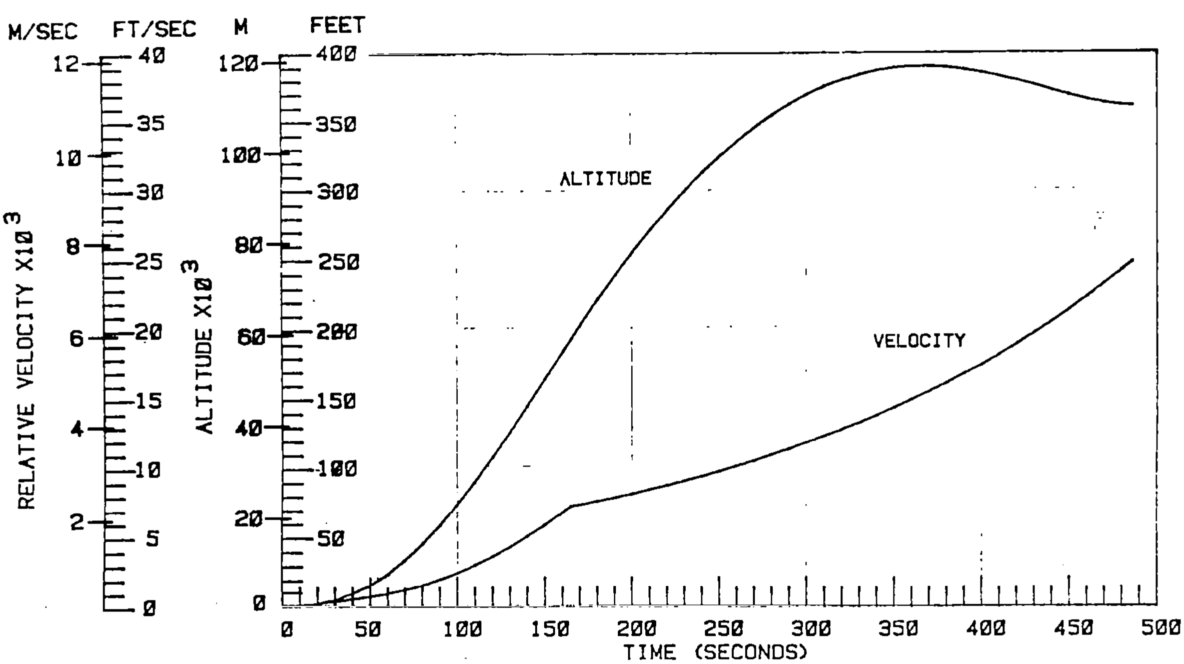


Figure 3.2.2-2. - Advanced Technology HLLV Ascent Performance

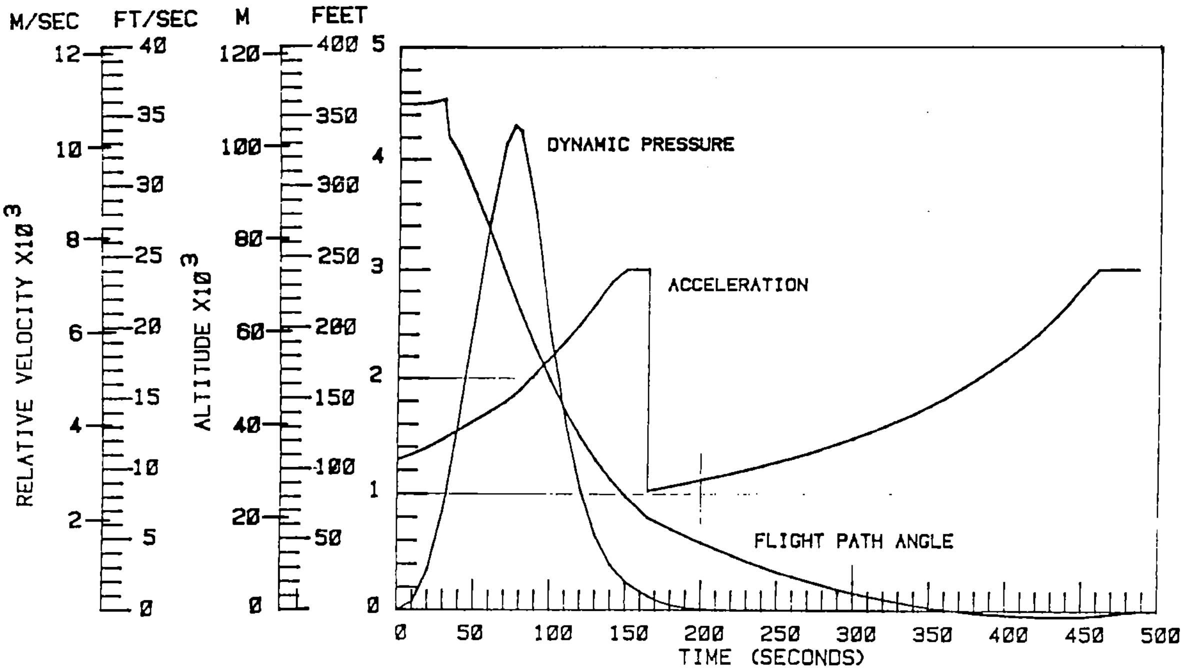


Figure 3.2.2-2b. - Advanced Technology HLLV Ascent Performance

3.2.2.2 Orbiter Configuration Description

The configuration of the HLLV orbiter is presented in figure 3.2.2-3 with the overall geometry noted. Net delivered payload capability is 222 222 kg (490 000 lb) with 10% net return payload capability. The orbiter is unmanned. Ascent thrust is provided by seven SSME's, each having a sea level thrust of 1.79 x 10^6N (403 100 lbf) and a vacuum thrust of 2.14 x 16^6N (480 000 lbf). The airframe structure consists of unpressurized structures, an integral main LH_2 tank and a non-integral LO_2 tank. The unpressurized structures are composite design and are protected from entry heating by advanced RSI. Main LH_2 tank structure is 6AL-4V (ELI) titanium sandwich, the sidewall of which is protected from entry heating by advanced RSI over composite design standoff panels and fairing structures. Main LO_2 tank structure is 2219-T87 aluminum. A summary mass and balance statement is presented in table 3.2.2-1.

Each of the items in the summary mass and balance statement, exclusive of payload and ascent propellants is discussed in the following paragraphs including definition of rationale for mass estimates.

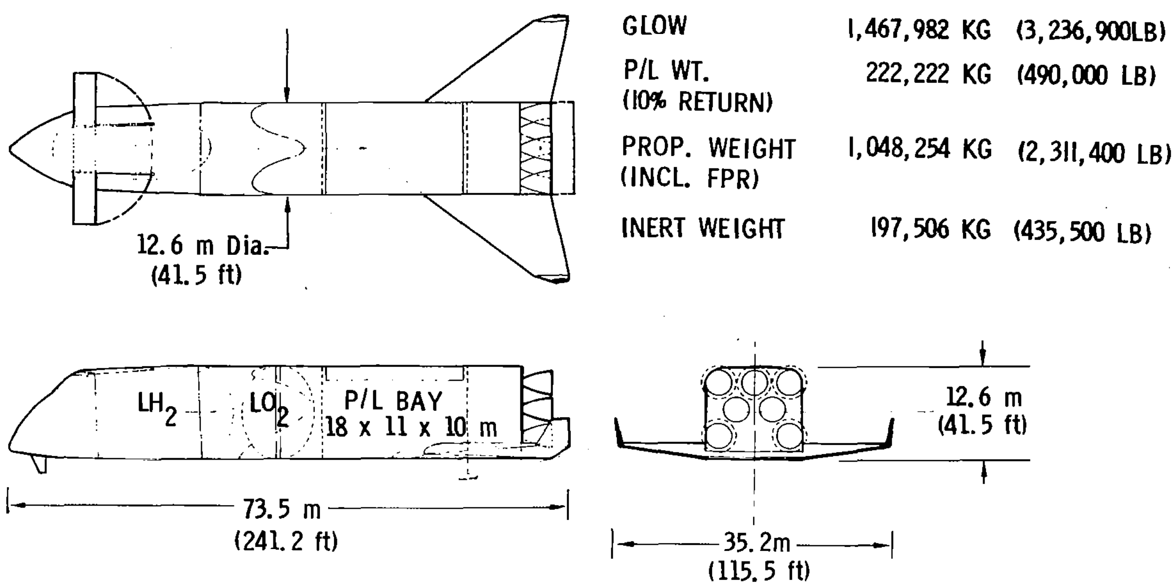


Figure 3.2.2-3. - HLLV Orbiter Configuration, Accelerated Technology

Wing–The wing is a composite design with a mass of 12 880 kg (28 400 lb) and a unit mass of 26.0 kg/m^2 (5.33 lb/ft^2) based on its reference area of 495 m^2 (5327 ft^2). The wing mass is estimated at 60% of that of an aluminum design wing sized for a 2.5g subsonic maneuver at the landing wing loading (reference wing + canard) of 360 kg/m^2 (73.8 lb/ft^2), a 406 K (270^{o}F) surface temperature during the maneuver, and a 1000 hour design accumulated time at surface temperatures above 394 K (250^{o}F). The wing mass includes the box body-carry-through section and main gear installation provisions. A constant t/c of 10% was used. The ratio of elevon area to wing reference area is 12.4%

Canard–The subsonic trim canard group consists of an aerosurface, hinge/attachment fittings and deployment/latch mechanisms. The aerosurface is a composite design with a unit weight of 17.6 kg/m^2 (3.6 lb/ft^2) based on its exposed planform area. The aerosurface unit mass is estimated at 60% of that of an aluminum design. Fittings and mechanisms unit mass is estimated at 19.5 kg/m^2 (4.0 lb/ft^2) of aerosurface area. The ratio of canard exposed planform area to wing reference area is 8.4%. Total mass is 1560 kg (3440 lb).

TABLE 3.2.2-1. - HLLV Orbiter Summary Mass and Balance Statement Accelerated Technology

GROUP/ITEM	MASS (kg)	XCG (m)	WEIGHT (lb)	XCG (in.)
WING	12,880	65.28	28,400	2570
CANARD	1,560	9.91	3,440	390
WING TIPLETS	580	71.12	1,280	2800
YAW VENTRAL	223	4.06	490	160
BODY	62,131	40.59	137,000	1598
INDUCED ENVIRONMENTAL PROTECTION	21,088	42.16	46,500	1660
LANDING AND AUXILIARY SYSTEMS	7,002	52.63	15,440	2072
PROPULSION—ASCENT	32,998	63.93	72,760	2517
ROCKET ENGINES	22,558	66.55	49,740	2620
ENGINE ACCESSORIES	1,891	66.55	4,170	2620
PROPELLANT SYSTEM	8,549	56.44	18,850	2222
PROPULSION—OMS (INTEGRATED)	2,118	43.18	4,670	1700
PROPULSION—RCS (INTEGRATED)	1,542	40.64	3,400	1600
PRIME POWER (INTEGRATED)	649	30.07	1,430	1184
ELECTRICAL CONVERSION & DISTR.	2,531	36.20	5,580	1425
HYDRAULIC CONVERSION & DISTR.	1,837	60.96	4,050	2400
SURFACE CONTROLS	2,068	62.84	4,560	2474
AVIONICS	916	5.08	2,020	200
ENVIRONMENTAL CONTROL	226	5.08	500	200
PAYLOAD PROVISIONS	2,608	50.60	5,750	1992
MARGIN	13,610	45.47	30,010	1790
(DRY CONDITION)	(166,567)	(48.31)	(367,280)	(1902)
PAYLOAD—ASCENT (10% RETURN)	222,222	50.60	490,000	1992
RESIDUALS—SYSTEMS/RCS/OMS/ASCENT PROPULSION	3,233	35.10	7,130	1382
RESERVES—SYSTEMS/RCS/OMS	1,007	30.99	2,220	1220
INFLIGHT LOSSES—SYSTEMS	417	42.37	920	1668
INFLIGHT LOSSES—ASCENT PROPULSION	7,869	62.03	17,350	2442
PROPELLANT—NOMINAL RCS	2,766	30.07	6,100	1184
PROPELLANT—NOMINAL OMS	15,647	30.07	34,500	1184
(INERT CONDITION, WITH P/L)	(419,728)	(48.82)	(925,500)	(1972)
ASCENT PROPELLANT—INCL FPR	1,048,254	33.07	2,311,400	1302
LO_2	898,503	35.33	1,981,200	1391
LH_2	149,751	19.55	330,200	770
(LIFTOFF CONDITION, WITH P/L)	(1,467,982)	(37.57)	(3,236,900)	(1479)

$$\text{MASS FRACTION} = \frac{\text{ASCENT PROPELLANT MASS}}{\text{LIFTOFF MASS, LESS P/L}} = 0.841$$

Wing Tiplets—This group consists of the wing tiplets and associated wing tip reinforcement. The wing tiplets, which are 30% rudder, are of composite design with a unit mass of 14.6 kg/m^2 (3.0 lb/ft^2). The tiplets unit mass is estimated at 60% of that of an aluminum design. Wing tip reinforcement unit mass is estimated at 7.3 kg/m^2 (1.5 lb/ft^2) of tiplet area. The ratio of tiplet area to wing reference area is 2.67% (per tiplet). Total mass is 580 kg (1280 lb).

Yaw Ventral-The subsonic trim yaw ventral group consists of a deployable all-moveable aerosurface and its installation provisions. The installation provisions include the aerosurface torque tube, tube support fittings, rails, rail support structures, deployment/retract mechanisms, and a door installation. The aerosurface is a composite design with a unit mass of 14.6 kg/m^2 (3.0 lb/ft^2) based on its exposed planform area. The aerosurface unit mass is estimated at 60% of that of an aluminum design. Installation provisions unit mass is estimated at 537 kg/m^2 (11.0 lb/ft^2) of aerosurface area. The ratio of yaw ventral exposed area to wing reference area is 0.66%. Total mass is 223 kg (490 lb).

Body-The body group consists of the following: nose section, LH_2 tank, intertank section, LO_2 tank, payload bay section, payload bay doors, aft body section, thrust structure, body flap, and fairing structures. Total body mass is 62 131 kg (137 000 lb).

- Nose Section—The nose section consists of a semimonocoque shell structure and incorporates nose gear installation provisions. The shell structure is of composite design with a unit mass of 9.8 kg/m^2 (2.0 lb/ft^2). This unit mass is 60% of the unit mass of the aluminum design shell structure on the Shuttle Orbiter. The nose gear installation provisions consist of the wheel well, door and mechanisms, and support structures, and are of composite design with a unit mass of 0.21% of design landing mass. This unit mass is 60% of the unit mass of the aluminum design installation provisions on the Shuttle Orbiter. Nose section total mass is 3214 kg (7090 lb).

- LH_2 Tank—The tank containing the liquid hydrogen is an all-welded 6AL-4V(ELI) titanium sandwich pressure vessel with a ring stiffened sidewall. The tank is designed by cryo temperature proof test conditions corresponding to 3g boost with maximum ullage pressure of 159 kPa (23 psia). To satisfy a 500 mission requirement with minimal weight impact, the tank is designed for 200 cycles to failure, and following the initial proof test, is reproofed every 100 missions. The inner face of the sandwich is sized to carry 100% of proof test pressure. The outer face is minimum gage at 0.030 cm (0.012 inch), including the sidewall regions to which TPS subpanels and fairing structures are attached. Tank mass is 10 136 kg (22 350 lb), 6.77% of the mass of the liquid hydrogen.

- Intertank Section—The intertank is a semimonocoque shell structure of composite design. It has a unit mass of 11.7 kg/m^2 (2.4 lb/ft^2). This unit mass is estimated at 60% of that of an aluminum design shell structure. The intertank mass is 4635 kg (10 220 lb).

- LO_2 Tank—The tank containing the liquid oxygen is an all-welded 2219-T87 aluminum pressure vessel. The tank is designed by room temperature poof test conditions corresponding to 3g boost with maximum ullage pressure of 159 kPa (23 psia). To satisfy a 500 mission requirement with minimal mass impact, the tank is designed for 200 cycles to failure and, following the initial proof test, is reproofed every 100 missions. The tank mass, including slosh baffles, is 10 871 kg (23 970 lb), 1.21% of the mass of the liquid oxygen.

- Payload Bay Section—This section is a semimonocoque shell structure of composite design. It has a unit mass of 16.6 kg/m^2 (3.4 lb/ft^2) exclusive of the payload bay door area. This unit mass is estimated at 60% of that of an aluminum design shell structure. The payload bay section mass is 14 404 kg (31 760 lb).

- Payload Bay Doors—Approximately 251 m^2 (2700 ft^2) of door area is provided. The doors are of composite design and have a unit mass of 15.1 kg/m^2 (3.1 lb/ft^2) based on Shuttle Orbiter door mass is 3791 kg (8360 lb).

- Aft Body Section—This section consists of a semimonocoque shell structure extending from the payload bay aft bulkhead to the engine support plane, a 1.5 m (5 ft) long fairing structure located aft of the engine support plane, and a base heat shield support structure installation located at the engine support plane. The structures are all of composite design with unit masses of 20.0 kg/m^2 (4.1 lb/ft^2), 10.7 kg (2.2 lb/ft^2), and 10.3 kg/m (3.1 lb/ft^2), respectively. These unit masses are estimated at 60% of those of aluminum design structures. In addition, the unit mass of the shell structure reflects a 5% reduction due to the absence of a vertical tail. Aft body section total mass is 7533 kg (16 610 lb).

- Thrust Structure—The thrust structure is a beam system of composite design which transmits thrust loads from the seven SSME's to the aft body section. Maximum vacuum thrust of the seven engines is 14.9×10^6N (3.360×10^6 lbf). The thrust structure has a mass of 3 061 kg (6 750 lb) which is 60% of that of an aluminum design. Its unit mass is 0.00021 kg/N (0.00201 lb/lbf).

- Body Flap—The constant chord body flap provides for pitch trim control and thermal shields the main engines during entry. The flap has a translating trailing edge section which, when extended, increases flap area by 70%. The flap group consists of the basic flap, the trailing edge translation mechanisms, and the hinge line support fittings/closeout provisons. Unit masses of the foregoing items are 12.2 kg/m^2 (2.5 lb/ft^2), 7.3 kg/m^2 (1.5 lb/ft^2), and 5.8 kg/m^2 (1.2 lb/ft^2), respectively, of total extended area. The basic flap is of composite design and its unit mass is 76% of the unit mass of the aluminum design basic flap on the Shuttle Orbiter. The unit mass of the translation mechanism is an estimate and the unit weight of the hinge line support fittings/closeout provisions is based on Shuttle Orbiter. Total body flap mass is 2327 kg (5130 lb).

- Fairing Structures—This group consists of a small wing leading edge-to-body fairing, a canard fairing and, in the LO_2 tank upper and lower cheek regions, body shape transition fairings. The fairings are of composite design and include approximately 168 m^2 (1810 ft^2) of fixed fairing at 5.2 kg/m^2 (1.07 lb/ft^2) and approximately 186 m^2 (2000 ft^2) of removable fairing at 6.9 kg/m^2 (1.41 lb/ft^2). The unit masses are estimated at 60% of those of aluminum design fairings. Total fairing structure mass is 2159 kg (4760 lb).

Induced Environmental Protection—This group consists of the external TPS system plus the internal provisions for thermal control and purge/vent/drain. The external TPS system utilizes advanced RSI on all areas including body chin and wing leading edge. The density of the RSI is 120 kg/m^3 (7.5 lb/ft^3) except in the body chin and wing leading edge areas where the density is 320 kg/m^3 (20.0 lb/ft^3). Based on an entry planform loading of 182 kg/m^2 (37.2 lb/ft^2) with flap retracted, and no crossrange, the TPS unit masses based on modified Shuttle Orbiter data are: outboard wing panels, 17.1 kg/m^2 (3.50 lb/ft^2) of wing exposed planform area; body, including base region, 14.2 kg/m^2 (2.91 lb/ft^2) of body planform area with flap extended; wing tiplets, 14.6 kg/m^2 (3.0 lb/ft^2) of tiplet side planform area. In addition, for the body, the main tankage sidewall regions not covered by fairing structures are covered by composite design standoff subpanels having an estimated average unit mass of 5.9 kg/m^2 (1.2 lb/ft^2). The internal provisions have an estimated mass of 1723 kg (3800 lb). The induced environmental protection total mass is 21 088 kg (46 500 lb).

Landing and Auxiliary Systems—In addition to landing gear, this group includes a landing drag device and auxiliary systems for orbiter-booster separation. The landing gear design incorporates the use of B-1 cord tires, servo-design shock struts with extensive use of boron-aluminum, and 55 200 kPa (8000 psi) hydraulic operated mechanisms. The unit masses of the landing gear and drag device are estimated at 2.8% and 0.33%, respectively, of design landing mass based on modified Shuttle Orbiter data and past studies. Separation systems mass is estimated at 1088 kg (2400 lb).

Ascent Propulsion—The ascent propulsion group consists of rocket engines, engine accessories, and the propellant system. Total mass is 32 998 kg (72 760 lb).

- Rocket Engines—Ascent thrust is provided by seven SSME's. Total vacuum thrust is 14.9×10^6N (3 360 000 lbf). Pertinent engine characteristics are presented in Section 2.1.2. Total dry engine mass is 22 558 kg (49 740 lb).

- Engine Accessories—Engine accessories for gimballed engines consist of a gimbal system, hydraulic supply, powerhead heat shield, and nitrogen purge provisions. Fixed engines have only the powerhead heat shield and nitrogen purge provisions. Based on the use of 55 200 kPa (8000 psi) hydraulic components and modest improvements in head shield design, the engine accessory masses for a SSME are estimated at 376 kg (830 lb) and 127 kg (280 lb) for a gimballed engine and a fixed engine, respectively. Total accessories mass for four gimballed SSME's and three fixed SSME's is 1891 kg (4170 lb).

- Propellant System—This subgroup consists of the propellant delivery system, tank pressurization (autogenous) and vent system, umbilical hardware, and associated installation hardware. Individual SSME feed line diameters are 0.305m (12 inch) for both LO_2 and LH_2. The LO_2 feed lines are fed by a single main line of 0.658m (25.9 inch) diameter. The LH_2 feed lines are fed by two main lines, each 0.465m (18.3 in.) in diameter. Using the Shuttle Orbiter propellant system as a data base, and allowing for a 30% mass reduction relative to current state-of-the-art (including composite overwrapped lines), the unit mass of the propellant system components located in the aft body region is 818 kg (1803 lb) per SSME. Using the ET tank propellant system as a data base, and allowing for a 40% mass reduction relative to current state-of-the-art (including composite overwrapped lines), the mass of the propellant system components located forward of the aft body region is 2825 kg (6230 lb). Total propellant system mass is 8549 kg (18 850 lb).

OMS Propulsion—The orbital maneuver system consists of engines and accessories, and associated propellant delivery and storage elements. The LO_2/LH_2 OMS is integral with the reaction control system (RCS) and auxiliary power system (APS) in that the RCS propellant and APS reactant are stored in the OMS tanks. Based on a OMS total installed vacuum thrust of 178 000N (40 000 lbf), nominal propellant requirements of 15 646 kg (34 500 lb) for OMS 2766 kg (6100 lb) for RCS, and 254 kg (560 lb) for APS, plus an OMS/RCS/APS root-sum-square reserve propellant requirement of 977 kg (2155 lb), the OMS dry mass is 2118 kg (4670 lb).

RCS Propulsion—A LO_2/LH_2 reaction control system provides for vehicle orientation assist during orbiter-booster separation, for vehicle orientation prior to entry, and for control during entry. The RCS is integral with the OMS and APS in that the RCS propellant and APS reactant are sorted in the OMS tanks, are routed to the RCS thrusters and APS auxiliary power units, respectively, via the bellows accumulators in the RCS. Based on a RCS total installed vacuum thrust of 323 000N (72 540 lbf) and nominal propellant requirements of 2766 kg (6100 lb) for RCS and 254 kg (560 lb) for APS, the RCS dry mass is 1542 kg (3400 lb).

Prime Power (Auxiliary Power System)-Power source elements consist of three independent LO_2/LH_2 powered APU's, auxiliary batteries, and alternators. The APS is integral with the OMS and RCS in that its propulsive grade reactant is stored in the OMS tanks and is routed to the APU's via bellows accumulators in the RCS. Based on a peak power requirement of 522 kW (700 hp) hydraulic plus 15 kW (20 hp) electric, the APS dry mass including reactant feed system and exhaust system is 649 kg (1430 lb).

Electrical Conversion and Distribution—This group consists of the power conversion, conditioning, and cabling elements for a 270 Vdc electrical system. Estimated system mass is 2531 kg (5580 lb).

Hydraulic Conversion and Distribution—This group consists of the hydraulic power supply and distribution equipment, including a water boiler temperature control system, for a 55 200 kPa (8000 psi) hydraulic system. Estimated system mass including hydraulic fluid is 1837 kg (4050 lb).

Surface Controls—The actuation systems for the aerodynamic control surfaces are included in this group. The actuation system unit masses, based on modest improvements in systems design relative to Shuttle Orbiter, are: elevons, 11.7 kg/m^2 (2.4 lb/ft^2); body flap, 12.2 kg/m^2 (2.5 lb/ft^2); wing tiplet rudders, 11.7 kg/m^2 (2.4 lb/ft^2); and yaw ventral, 13.2 kg/m^2 (2.7 lb/ft^2). Total surface controls mass is 2068 kg (4560 lb).

Avionics—The avionics group includes elements for guidance, navigation and control, tracking, instrumentation, and data processing and software. Total mass is 916 kg (2020 lb) or 32% of Shuttle Orbiter avionics mass, based on use of advanced avionics, reduced functional requirements, and reduced power requirements.

Environmental Control—The environmental control group maintains a conditioned thermal environment for the avionics. Estimated system mass including closed loop fluids is 226 kg (500 lb).

Payload Provisions—This group consists of fixed scar items and removable provisions with unit masses of 0.30% and 0.85%, respectively, of maximum ascent payload mass, based on Shuttle Orbiter. Total group mass is 2608 kg (5750 lb).

Margin—A margin allowance of approximately 10% of subsystems dry mass exclusive of SSME's has been incorporated. Total mass is 13 610 kg (30 010 lb).

Nominal RCS Propellant—The RCS delta-V budget is 39 m/s (127 ft/sec) and consists of the following:

Trim Burns - Ascent (Separation)	0.5 m/s (2 ft/sec)
Trim Burns - Post Ascent	1.5 m/s (5 ft/sec)
Trim Burns - 500 km Orbit	17.0 m/s (55 ft/sec)
Trim Burns - Entry	20.0 m/s (65 ft/sec)

The entry trim burns delta-V includes an allowance of 6 m/s (20 ft/sec) due to the absence of a vertical tail. Based on a specific impulse of 2119 N-sec/kg (420 sec) for the LO_2/LH_2 thrusters, and consideration of vehicle sequential mass, the total nominal propellant mass is 2766 kg (6100 lb).

Nominal OMS Propellant—The OMS delta-V budget is 240 m/s (788 ft/sec) and consists of the following:

Circularization into 500 km Orbit from 110 x 500 km Launch Orbit	112 m/s (367 ft/sec)
Trim Burns in 500 km Orbit	11 m/s (37 ft/sec)
Deorbit from 500 km Orbit	117 m/s (384 ft/sec)

Based on a specific impulse of 473 sec for the LO_2/LH_2 OMS engines, and consideration of vehicle sequential mass, the total nominal propellant mass is 15 647 kg (34 500 lb).

Systems Inflight Losses—This group consists of the following nominal usages: O_2/H_2 for hydraulic and electrical power, 254 kg (560 lb); and H_2O for hydraulic cooling, 163 kg (360 lb). Total estimated mass is 417 kg (920 lb).

Ascent Propulsion Inflight Losses—All ascent propulsion liquid residuals are jettisonable and are included in this group. Total mass is 7869 kg (17 350 lb). Pertinent line size data is included under ascent propulsion-propellant system.

- Propellant in Engines at Cutoff-Trapped propellant mass is 236 kg (520 lb) per SSME. Total mass is 1651 kg (3640 lb).

- Propellant in Lines at Cutoff—An average 4.0m (13 ft) of LO_2 line and 4.9m (16 ft) of LH_2 line is located between the main delivery line(s) and each SSME. These lines are full at engine cutoff. Approximately 4.9M (16 ft) of main delivery line(s) for both LO_2 and LH_2 is located between the junction with the engine feed lines and the junction with the transfer line(s). These lines are full at engine cutoff as are the 0.6m (2 ft) long transfer lines. In addition, at engine cutoff under nominal conditions, the main delivery line(s) upstream of the transfer line junction contain approximately 3.0m (10 ft) of fluid, exclusive of bias fuel. The LH_2 and LO_2 unit masses are 798 kg (1760 lb) and 50 kg (110 lb), respectively, per SSME. Total mass is 5937 kg (13 090 lb).

- Bias Fuel—A single-engine reference bias fuel allowance of 0.50% was used. For multiple engines, this allowance was factored by the inverse of the square root of the number of engines. The resulting LH_2 bias allowance for the seven SSME's is 0.19%. Total LH_2 mass 281 kg (620 lb).

Reserves—This group consists of reserve allowances for RCS, OMS, hydraulic and electric power, and hydraulic cooling. (The flight performance reserve for ascent is 0.85% of total ideal velocity change and is included in ascent propellant weight.) The RCS, OMS, and power system are integrated LO_2/LH_2 systems in which the O/F mixture ratio of the nominal propellants if 4:1, 6:2, and 1:1, respectively. Because these systems have common propellant storage tanks, the reserve allowances peculiar to dedicated systems were combined on a root-sum-square basis with consideration of the above O/F mixture ratios (the dedicated systems reserve allowances for RCS, OMS, and prime power are 20%, 5%, and 50%, respectively). The resulting integrated systems reserve allowance is 975 kg (2150 lb) with an O/F mixture ratio of 4.7:1. The reserve allowance for hydraulic cooling is 20%. Total reserves mass is 1007 kg (2220 lb).

Residuals-This group consists of the fluids and gases, onboard at landing, for ascent propulsion, RCS, OMS, hydraulic and electric power, and hydraulic cooling. Total mass is 3233 kg (7120 lb).

- Ascent Propulsion Residuals—Only main tank pressurants are onboard at landing. The pressurant masses are based on maximum ullage pressures of 159 kPa (23 psia) and mean temperatures for the GO_2 and GH_2 of 264 K (475^oR) and 189 K (340^oR), respectively. The masses of the GO_2 and GH_2 are 1923 kg (4240 lb), and 453 kg (1000 lb), respectively. Total pressurant mass is 2376 kg (5240 lb).

- Other Residuals—The residuals mass for the integrated RCS/OMS/prime power is 830 kg (1830 lb). The residuals allowance for hydraulic cooling is 15% of usage (nominal + reserve) water. Total mass for this subgroup is 857 kg (1890 lb).

3.2.2.3 Booster Configuration Description

The configuration of the HLLV booster is presented in figure 3.2.2-4 with the overall geometry noted. The booster is unmanned. Ascent thrust is provided by eight DF/DE engines, each having a sea level thrust of 5.12×10^6N (1 150 000 lbf). Of the total propellant load, approximately 78% is consumed by the booster engines while approximately 18% is crossfed to the SSME's in the orbiter, prior to booster-orbiter staging at a velocity of approximately 2 195 m/s (7 200 ft/sec). The airframe structure consists of unpressurized structures and integral main propellant tanks. The unpressurized structures are aluminum/titanium/composites design (composites used internally only) and are

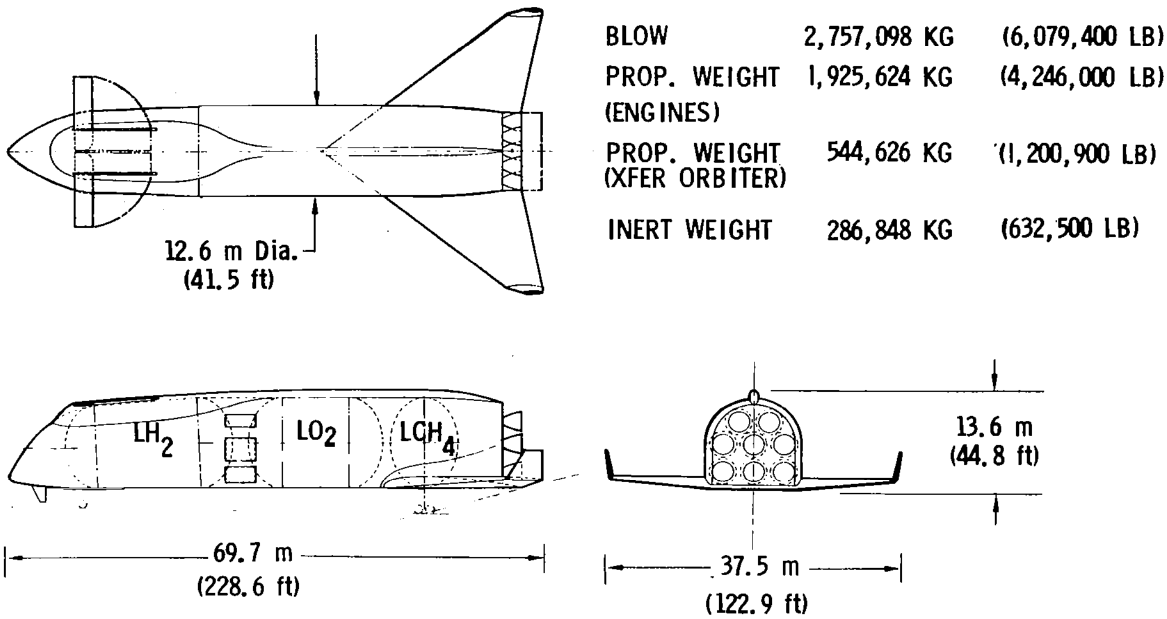


Figure 3.2.2-4. - HLLV Booster Configuration Accelerated Technology

protected from entry heating by increasing the thickness of the outer surface material to provide heat-sink capability. Main tankage structures are 6AL-4V(ELI) titanium sandwich for the LH_2 tank and 2219-T87 aluminum for the LO_2 and LCH_4 tanks. The tank sidewalls are protected from entry heating by increasing the skin thickness to provide heat-sink capability. Flyback thrust is provided by ten JTF22A-Mod III engines, each having a sea level static thrust of approximately 133 000N (30 000 lbf). Flyback fuel is RP-1. A summary mass and balance statement is presented in table 3.2.2-2.

Each of the items in the summary mass and balance statement, exclusive of ascent propellant, is discussed in the following paragraphs including definition of rationale for mass estimates.

Wing—The wing is an aluminum/titanium/composite design with a mass of 21 859 kg (48 200 lb) and a unit mass of 38.9 kg/m^2 (7.97 lb/ft^2) based on its reference area of 562 m^2 (6 046 ft^2). The design features an aluminum box and intermediate sections and titanium leading edges, trailing edges, and elevons, all of which incorporate extensive use of composite internal members. The wing mass is estimated at 80% of that of an

TABLE 3.2.2-2. - HLLV Booster Summary Mass and Balance Statement Accelerated Technology

GROUP/ITEM	MASS (kg)	XCG (m)	WEIGHT (lb)	XCG (in.)
WING	21,859	61.72	48,200	2430
CANARD	1,388	9.91	3,060	390
WING TIPLETS	871	68.07	1,920	2680
YAW VENTRAL	254	4.06	560	160
BODY	75,011	41.45	165,400	1632
INDUCED ENVIRONMENTAL PROTECTION	9,043	54.91	19,940	2162
LANDING AND AUXILIARY SYSTEMS	8,331	48.84	18,370	1923
PROPULSION–ASCENT	56,010	61.39	123,500	2417
ROCKET ENGINES	35,665	63.50	78,640	2500
ENGINE ACCESSORIES	4,626	63.50	10,200	2500
PROPELLANT SYSTEM	15,719	55.96	34,660	2203
PROPULSION–FLYBACK	18,231	29.79	40,200	1173
PROPULSION–RCS } INTEGRATED	2,195	35.56	4,840	1400
PRIME POWER } INTEGRATED	703	29.79	1,550	1173
ELECTRICAL CONVERSION & DISTR.	2,667	33.25	5,880	1309
HYDRAULIC CONVERSION & DISTR.	2,073	50.80	4,570	2000
SURFACE CONTROLS	1,909	60.30	4,210	2374
AVIONICS	916	5.08	2,020	200
ENVIRONMENTAL CONTROL	226	5.08	500	200
MARGIN	15,025	49.78	33,130	1960
(DRY CONDITION)	(216,712)	(48.82)	(477,850)	(1922)
RESIDUALS–SYSTEMS/RCS/FLYBACK/ASCENT PROPULSION	6,590	39.39	14,530	1551
RESERVES–SYSTEMS/RCS/FLYBACK	8,222	29.92	18,130	1178
INFLIGHT LOSSES–SYSTEMS	413	43.00	910	1693
INFLIGHT LOSSES–ASCENT PROPULSION	24,235	58.90	53,440	2319
PROPELLANT–NOMINAL RCS	1,220	21.79	2,690	1173
FUEL–NO HEADWIND FLYBACK	29,456	21.79	64,950	1173
(INERT CONDITION)	(286,848)	(46.86)	(632,500)	(1845)
ASCENT PROPELLANT	2,470,250	40.11	5,446,900	1579
LO_2	2,022,403	39.65	4,459,400	1561
LH_2	143,537	17.27	316,500	680
LCH_4	304,308	53.98	671,000	2125
(LIFTOFF CONDITION)	(2,757,098)	(40.82)	(6,079,400)	(1607)

$$\text{MASS FRACTION} = \frac{\text{ASCENT PROPELLANT MASS}}{\text{LIFTOFF MASS}} = 0.896$$

aluminum design wing sized for a 2.5g subsonic maneuver at the start of flyback wing loading (reference wing + canard) of 362 kg/m^2 (74.2 lb/ft^2), a 406 K (270°F) surface temperature during the maneuver, and a 1,000 hour design accumulated time at surface temperatures above 394 K (250°F). The wing mass includes the box body-carry-through section and main gear installation provisions. A constant t/c of 10% was used. The ratio of elevon area to wing reference area is 12.0%.

Canard—The subsonic trim canard group consists of an aerosurface, hinge/attachment fittings and deployment/latch mechanisms. The aerosurface is a composite design with a unit mass of 17.6 kg/m^2 (3.6 lb/ft^2) based on its exposed planform area. The aerosurface unit mass is estimated at 60% of that of an aluminum design. Fittings and mechanisms unit mass is estimated at 19.5 kg/m^2 (4.0 lb/ft^2) of aerosurface area. The ratio of canard exposed planform area to wing reference area is 6.6%. Total mass is 1 388 kg (3 060 lb).

Wing Tiplets—This group consists of the wing tiplets and associated wing tip reinforcement. The wing tiplets, which are 30% rudder, are of aluminum/titanium/composite design with a unit mass of 19.5 kg/m^2 (4.0 lb/ft^2). The tiplets unit mass is estimated at 80% of that of an aluminum design. Wing tip reinforcement unit mass is estimated at 9.8 kg/m^2 (2.0 lb/ft^2 of tiplet area. The ratio of tiplet area to wing reference area is 2.67% (per tiplet). Total mass is 871 kg (1920 lb).

Yaw Ventral—The subsonic trim yaw ventral group consists of a deployable all-movable aerosurface and its installation provisions. The installation provisions include the aerosurface torque tube, tube support fittings, rails, rail support structures, deployment/retract mechanisms, and a door installation. The aerosurface is a composite design with a unit mass of 14.6 kg/m^2 (3.0 lb/ft^2) based on its exposed planform area. The aerosurface unit mass is estimated at 60% of that of an aluminum design. Installation provisions unit mass is estimated at 53.7 kg/m^2 (11.0 lb/ft^2) of aerosurface area. The ratio of yaw ventral exposed area to wing reference area is 0.66%. Total mass is 254 kg (560 lb).

Body—The body group consists of the following: nose section, LH_2 tank, forward intertank section, LO_2 tank, aft intertank section, LCH_4 tank, aft body section, thrust structure, body flap, and fairing structures. Total body mass is 75 011 kg (165 400 lb).

- Nose Section—The nose section consists of a semimonocoque shell structure and incorporates nose gear installation provisions. The shell structure is of aluminum/titanium/composite design with a unit mass of 12.7 kg/m^2 (2.6 lb/ft^2). This unit mass is 80% of the unit mass of the aluminum design shell structure on the Shuttle Orbiter. The nose gear installation provisions consist of the wheel well, door and mechanisms, and support struts, and are of aluminum/composite design with a unit mass of 0.235% of design landing weight. This unit mass is 67% of the unit mass of the aluminum design installation provisions on the Shuttle Orbiter. Nose section total mass is 4172 kg (9200 lb).

- LH_2 Tank—The tank containing the liquid hydrogen is an all-welded 6AL-4V(ELI) titanium sandwich pressure vessel with a ring stiffened sidewall. The tank is designed by cryo temperature proof test conditions corresponding to 3g boost onset with maximum ullage pressure of 159 kPa (23 psia). To satisfy a 500 mission requirement with minimal weight impact, the tank is designed for 200 cycles to failure and, following the initial proof test, is reproofed every 100 missions. The inner face of the sandwich is sized to carry 100% of proof test pressure. The outer face is minimum gage at 0.030 cm (0.012 in), including the sidewall regions to which fairing structures are attached. Tank mass is 9 478 kg (20 900 lb), 6.60% of the mass of the liquid hydrogen.

- Forward Intertank Section—This intertank, which supports the flyback propulsion system, is a semimonocoque shell structure of aluminum/composite design. It has a unit mass of 15.6 kg/m^2 (3.2 lb/ft^2). This unit mass is estimated at 80% of that of an aluminum design shell structure. The forward intertank mass is 6803 kg (15 000 lb).

- LO_2 Tank—The tank containing the liquid oxygen is an all-welded 2219-T87 aluminum pressure vessel with a stiffened sidewall. The tank is designed by room temperature proof test conditions corresponding to 3g boost onset with maximum ullage pressure of 159 kPa (23 psia). To satisfy a 500 mission requirement with minimal mass impact, the tank is designed for 200 cycles to failure and, following the initial proof test, is reproofed every 100 missions. The tank mass, including slosh baffles, is 16 236 kg (35 800 lb), 0.80% of the mass of the liquid oxygen.

- Aft Intertank Section—This intertank is a semimonocoque shell structure of aluminum/composite design. It has a unit mass of 23.4 kg/m^2 (4.8 lb/ft^2). This unit mass is estimated at 80% of that of an aluminum design shell structure. The aft intertank mass is 9252 kg (20 400 lb).

- LCH_4 Tank—The tank containing the liquid methane is an all-welded 2219-T87 aluminum pressure vessel. The tank is designed by room temperature proof test conditions corresponding to 3g boost onset with maximum ullage pressure of 159 kPa (23 psia). To satisfy a 500 mission requirement with minimal mass impact, the tank is designed for 200 cycles to failure and, following the initial proof test, is reproofed every 100 missions. The tank mass, including slosh baffles, is 5 215 kg (11 500 lb), 1.71% of the mass of the liquid methane.

- Aft Body Section—This section consists of a semimonocoque shell structure extending from the LCH_4 tank to the engine support plane, a 1.5m (5 ft) long fairing structure located aft of the engine support plane, and a base heat shield support structure installation located at the engine support plane. The shell structure and fairing structure are of aluminum/composite design. The heat shield support structure is of composite design. Unit masses of the foregoing structures are 26.4 kg/m^2 (5.4 lb/ft^2), 14.2 kg/m^2 (2.9 lb/ft^2), and 10.3 kg/m^2 (2.1 lb/ft^2), respectively. These unit masses are estimated at 80%, 80%, and 60%, respectively, of those of aluminum design structures. In addition, the unit mass of the shell structure reflects a 5% reduction due to the absence of a vertical tail. Aft body section total weight is 10 930 kg (24 100 lb).

- Thrust Structure—The thrust structure is a beam system of composite design which transmits thrust loads from the eight DF/DE engines to the aft body section. Maximum vacuum thrust of the eight engines is 45.4×10^6N (10.202×10^6 lbf). The thrust structure has a mass of 6 803 kg (15 000 lb) which is 60% of that of an aluminum design. Its units mass is 0.00015 kg/N (0.00147 lb/lbf).

- Body Flap—The constant chord body flap provides for pitch trim control and thermally shields the main engines during entry. The flap has a translating trailing edge section which, when extended, increases flap area by 70%. The flap group consists of the basic flap, the trailing edge translation mechanism, and the hinge line support fittings/closeout provisions. Unit masses of the foregoing items are 12.7 kg/m^2 (2.6 lb/ft^2), 7.3 kg/m^2 (1.5 lb/ft^2), and 5.8 kg/m^2 (1.2 lb/ft^2), respectively, of total extended area. The basic flap is of titanium/composite design and its unit mass is 80% of the unit mass of the aluminum design basic flap on the Shuttle Orbiter. The unit mass of the translation mechanisms is an estimate and the unit weight of the hinge line support fittings/closeout provisions is based on Shuttle Orbiter. Total body flap mass is 1 859 kg (4 100 lb).

- Fairing Structures—This group consists of a wing-to-body fairing, a canard fairing, and a LH_2 main delivery line fairing. The fairings are of aluminum/titanium/composite design and include approximately 251 m^2 (2 700 ft^2) of fixed fairings at 6.9 kg/m^2 (1.42 lb/ft^2) and approximately 275 m^2 (2 960 ft^2) of removable fairing at 9.2 kg/m^2 (1.88 lb/ft^2). The unit masses are estimated at 80% of those of aluminum design fairings. Total fairing structure mass is 4 263 kg (9 400 lb).

Induced Environmental Protection—This group consists of the external TPS system plus the internal provisions for thermal control and purge/vent/drain. The external TPS system consists of the heat sink additions required to maintain the airframe outer skin within acceptable temperature limits, and an advanced RSI base heat shield. Based on a booster-orbiter staging velocity of aproximately 2195 m/s (7200 ft/sec), a 4g entry, and an entry planform loading of 252 kg/m^2 (51.7 lb/ft) with flap retracted, the heat sink addition unit masses based on heat sink booster studies are: wing including body flap, 9.9 kg/m2 (2.03 lb/ft^2) of wing reference area; body, 1.4 kg/m^2 (0.29 lb/ft^2) of body planform area forward of the wing reference area; wing tiplets, 14.6 kg/m^2 (3.0 lb/ft^2) of tiplet side planform area. In addition, for the body, the unit mass of the RSI base heat shield is 5.2 kg/m^2 (1.07 lb/ft^2) of body base area. The internal provisions have an estimated mass of 1 723 kg (3 800 lb). The induced environmental proteciton total mass is 9 043 kg (19 940 lb).

Landing and Auxiliary Systems—In addition to landing gear, this group includes a landing drag device and auxiliary systems for orbiter-booster separation. The landing gear design incorporates the use of B-1 cord tires, servo-design shock struts with extensive use of boron-aluminum, and 55 200 kPa (8000 psi) hydraulic operated mechanisms. The unit masses of the landing gear and drag device are estimated at 2.8% and 0.33%, respectively, of design landing mass based on modified Shuttle Orbiter data and past studies. Separation systems mass is estimated at 1 088 kg (2 400 lb). Total mass is 8 331 kg (18 370 lb).

Ascent Propulsion—The ascent propulsion group consists of rocket engines, engine accessories, and the propellant system. Total mass is 56 010 kg (123 500 lb).

- Rocket Engines—Ascent thrust is provided by eight DF/DE engines. Total liftoff thrust is 40.9×10^6N (9 2000 000 lbf). Pertinent engine characteristics are presented in Section 2.1.2. Total dry engine mass is 35 665 kg (78 640 lb).

- Engine Accessories—Accessories for each of the eight DF/DE engines consists of a gimbal system, hydraulic supply, powerhead heat shield, and nitrogen purge provisions. Based on the use of 55 200 kPa (8000 psi) hydraulic components and modest improvements in heat shield design, the engine accessories mass for a gimballed DF/DE engine is estimated at 13% of dry engine mass based on modified accessories mass data for a standard SSME. Total accessories mass is 4 626 kg (10 200 lb).

- Propellant System—This subgroup consists of the propellant delivery system, tank pressurization (autogenous) and vent provisions, umbilical hardware, and associated installation hardware. Individual DF/DE engine feed line diameters are 0.526 m (20.7 inch) for LO_2, 0274m (10.8 inch) for LCH_4, and 0.267m (10.5 inch) for LH_2. The lines which cross feed LO_2 and LH_2 to the seven SSME's in the orbiter consist of two LO_2 lines of 0.465m (18.3 inch) diameter (with common junction at interface) and two LH_2 lines of 0.465m (18.3 inch) diameter. The LO_2 feed lines and transfer lines are feed by two main lines, each 0.978m (38.5 inch) in diameter. The LH_2 feed lines and transfer lines are fed by a single main line of 0.899m (35.4 inch) diameter. The LCH_4 lines to each engine are direct tank-to-engine lines. Using the Shuttle Orbiter propellant system as a data base, and allowing for a 30% mass reduction relative to current state-of-the-art, (including composite overwrapped lines), the unit mass of the propellant system components located forward of the aft body region is 4 594 kg (10 130 lb). Total propellant system mass is 15 719 kg (34 660 lb).

Flyback Propulsion—The flyback propulsion group consists of the airbreathing engines, engine accessories, engine oil, fuel system, fuel tank, and engine installation nacelles, deployment structures, mechanisms, and doors. Flyback thrust is provided by ten TJF22A-Mod III turbojet engines, each having a sea level static thrust of approximately 133 000N (30 000 lbf). Flyback fuel is RP-1. The mass of the turbojet engines is 10 535 kg (23 230 lb). Total dry system mass, including engine oil, is 18 231 kg (40 200 lb).

RCS Propulsion—A LO_2/LH_2 reaction control system provides for vehicle orientation assist during orbiter-booster separation, for vehicle orientation prior to entry, and for control during entry. The RCS is integral with the auxiliary propulsion system (APS) in that the APS reactant is stored in the RCS main tanks and is routed to the APS auxiliary power units via the bellows accumulators in the RCS. A simplified schematic, pertinent characteristics, discussion data, and weight equations are presented in Appendix E. Based on a RCS total installed vacuum thrust of 445 000N (100 050 lbf) and nominal propellant requirements of 1 220 kg (2 690 lb) for RCS and 222 kg (490 lb) for APS, plus a RCS/APS root-sum-square reserve propellant requirement of 279 kg (615 lb), the RCS dry mass is 2 195 kg (4 480 lb).

Prime Power (Auxiliary Power System)—Power source elements consist of three independent LO_2/LH_2 powered APU's, auxiliary batteries, and alternators. The APS Is integral with the RCS in that its propulsive grade reactant is stored in the RCS main tanks and is routed to the APU's via the RCS bellows accumulators. Based on a peak power requirement of 578 kW (775 hp) hydraulic plus 15 kW (20 hp) electric, the APS dry mass including reactant feed system and exhaust system is 703 kg (1 550 lb).

Electrical Conversion and Distribution—This group consists of the power conversion, conditioning, and cabling elements for a 270 Vdc electrical system. Estimated system mass is 2 667 kg (5 880 lb).

Hydraulic Conversion and Distribution—This group consists of the hydraulic power supply and distribution equipment, including a water boiler temperature control system, for a 55 200 kPa (8000 psi) hydraulic system. Estimated system mass including hydraulic fluid is 2 073 kg (4 570 lb).

Surface Controls—The actuation systems for the aerodynamic control surfaces are included in this group. The actuation system unit masses, based on modes improvements in systems design relative to Shuttle Orbiter, are: elevons, 11.7 kg/m^2 (2.4 lb/ft^2); body flap, 12.2. kg/m^2 (2.5 lb/ft^2); wing tiplet rudders, 11.7 kg/m^2 (2.4 lb/ft^2); and yaw ventral, 13.2 kg/m^2 (2.7 lb/ft^2). Total surface controls mass is 1 909 kg (4 210 lb).

Avionics—The avionics group includes elements for guidance, navigation and control, tracking, instrumentation, and data processing and software. total mass is 916 kg (2 020 lb) or 32% of Shuttle Orbiter avionics mass, based on use of advanced avionics, reduced functional requirements, and reduced power requirements.

Environmental Control—The environmental control group maintains a conditioned thermal environment for the avionics. Estimated system mass including closed loop fluids is 226 kg (500 lb).

Margin—A margin allowance of approximately 10% of subsystems dry mass exclusive of JTF22A-Mod III airbreather engines has been incorporated. Total mass is 15 025 kg (33 130 lb).

Nominal RCS Propellant—The RCS delta-V budget is 18.5 m/s (62 ft/sec) and consists of the following:

Trim Burns - Ascent (Separation)	2.0 m/s (7 ft/sec)
Trim Burns - Post Ascent	1.5 m/s (5 ft/sec)
Trim Burns - Entry	15 m/s (50 ft/sec)

The entry trim burns delta-V includes an allowance of 6 m/s (20 ft/sec) due to the absence of a vertical tail. Based on a specific impulse of 420 sec for the LO_2/LH_2 thrusters, and consideration of vehicle sequential mass, the total nominal propellant mass is 1 220 kg (2 690 lb).

Nominal (No Headwind) Flyback Fuel—Booster-orbiter staging occurs at a velocity of approximately 2 195 m/s (7 200 ft/sec) following an easterly launch. Subsequent to the transition maneuver at end of entry, the booster performs a long glide prior to starting its powered flyback of approximately 212 nm range. Consideration of a 3 km (10 000 ft) cruise ceiling, a 1.5 km (5 000 ft) minimum cruise altitude, a maximum L/D of approximately 5, and the use of ten JTF22A-Mod III engines in a deployable installation results in a total nominal (no headwind) flyback fuel mass of 29 456 kg (64 950 lb).

Systems Inflight Losses—This group consists of the following nominal usages: O_2/H_2 for hydraulic and electric power, 222 kg (490 lb); and H20 for hydraulic cooling, 191 kg (420 lb). Total estimated mass is 413 kg (910 lb).

Ascent Propulsion Inflight Losses—All ascent propulsion liquid residuals are jettisonable and are included in this group. Total mass is 24 235 kg (53 440 lb). Pertinent line size data is included under ascent propulsion-propellant system.

- Propellant in engines at cutoff—Trapped propellant mass is estimated at 2295 kg (650 lb) per DF/DE engine. Total mass is 2360 kg (5200 lb).

- LCH_4 in Engine Feed Lines at Cutoff—The LCH_4 lines to each DF/DE engine are direct tank-to-engine lines. The lines average 5.2m (17 ft) in length and contain an average 3.0m (10 ft) of LCH_4 at nominal engine cutoff exclusive of bias fuel. The LCH_4 unit mass is 75 kg (165 lb) per DF/DE engine. Total mass is 600 kg (1320 lb).

- LO_2 and LH_2 in Feed Lines at Cutoff—An average 6.1m (20 ft) of both LO_2 line and LH_2 line are located between the main delivery line(s) and each engine. These lines are full at engine cutoff. Approximately 6.7m (22 ft) of LH_2 transfer line and 6.1m (20 ft) of LH_2 transfer line are located between the main delivery line(s) and the transfer interface. These lines are full at engine cutoff. In addition, at engine cutoff under nominal conditions, the main delivery line(s) upstream of the common junction with the engine feed lines and transfer line(s) contains 3.0m (10 ft) of fluid exclusive of bias fuel. The LO_2 unit masses are 2018 kg (4450 lb) per booster engine plus 544 kg (1200 lb) per SSME In the orbiter. The LH_2 unit masses are 38 kg (83 lb) per booster engine plus 32 kg (70 lb) per SSME In the orbiter. Total mass is 20 476 kg (45 150 lb).

- Bias Fuel—A single-engine reference bias fuel allowance of 0.50% was used. For multiple engines, this allowance was factored by the inverse of the square root of the number of engines. The resulting LCH_4 and LH_2 bias allowances for the eight DF/DE booster engines are 0.18%. The additional LH_2 bias allowance for the transfer fuel to the seven SSME's in the orbiter is 0.19%. The LCH_4 bias is 540 kg (1190 lb) and the net LH_2 bias is 263 kg (580 lb). Total mass is 803 kg (1770 lb).

Reserves—This group consists of reserve allowances for the RCS, hydraulic and electric power, flyback, and hydraulic cooling. The RCS and power system are integrated LO_2/LH_2 systems in which the O/F mixture ratio of the nominal propellants is 4:1 and 1:1, respectively. Because these systems have common propellant storage tanks, the reserve allowances peculiar to dedicated systems were combined on a root-sum-square basis with consideration of the above O/F mixture ratios. (The dedicated systems reserve allowances for RCS and prime power are 20% and 50%, respectively.) The resulting integrated systems reserve allowance s 279 kg (615 lb) with an O/F mixture ratio of 2.7:1. Reserve allowances for flyback are for headwinds and go-around. The headwind reserve is estimated at 11% of the nominal (no headwind) fuel. The go-around reserve is estimated at 2% of design landing weight. Flyback reserves total 7905 kg (17 430 lb). The hydraulic cooling water reserve is 38 kg (85 lb) based on a 20% reserve allowance. Total reserves mass is 8222 kg (18 130 lb).

Residuals—This group consists of the fluids and gases, onboard at landing, for ascent propulsion, RCS, hydraulic and electric power, flyback propulsion, and hydraulic cooling. Total mass is 6590 kg (14 530 lb).

- Ascent Propulsion Residuals—Only main tank pressurants are onboard at landing. The pressurant masses are based on maximum ullage pressures of 159 kPa (23 psia) and mean temperatures for the GO_2, GH_2, and GCH_4 of 264 K (475^{o}R), 189 K (340^{o}R), and 222 K (400^{o}R), respectively. The masses of the GO_2, GH_2, and GCH_4 are 4326 kg (9 540 lb), 435 kg (960 lb), and 1039 kg (2290 lb), respectively. Total pressurant mass is 5801 kg (12 790 lb).

- Other Residuals—The residuals mass for the integrated RCS/prime power is 383 kg (845 lb). Residual allowances for flyback and hydraulic cooling are 1% and 15%, respectively, of usable (nominal + reserve) quantities. Total mass for this subgroup is 789m kg (1740 lb).

3.2.2.4 Advanced Technology HLLV Life Cycle Costs

Table 3.2.2-3 presents a summary of the advanced technology HLLV LCC, along with the baseline HLLV LCC for comparison. The total LCC is $11360.3 million. All costs presented in this section are in constant 1977 dollars.

DDT&E Phase

The total HLLV DDT&E phase cost of $49 860.0 million has the following split between stages:

- Orbiter - $2509.2 million
- Booster - $2476.8 million

DDT&E cost breakdowns for the Orbiter and Booster are presented in tables 3.2.2-4 and 3.2.2-5, respectively.

TABLE 3.2.2-3. - HLLV - COST SUMMARY ($ IN MILLIONS)

	NORMAL TECH.	ADV. TECH.
TOTAL PROGRAM	13,346.41	11,360.28
DDT&E	6,243.87	4,986.93
PROGRAM MANAGEMENT	172.63	154.19
ENGINEERING	2,411.08	1,557.89
MANUFACTURING	2,825.81	2,412.35
TEST	834.35	743.49
PRODUCTION	2,252.21	1,937.59
PROGRAM MANAGEMENT	183.17	161.45
SUSTAINING ENGINEERING	51.92	44.49
MANUFACTURING	2,017.13	1,731.64
PROD. TOOLING & S.T.E.	590.84	509.35
FLT. HARDWARE & SPARES	1,426.28	1,222.29
OPERATIONS	4,850.33	4,436.66
OPERATIONS SUPPORT	2,968.43	2,705.12
PROGRAM SUPPORT	442.30	447.63
SPARES PROCUREMENT	2,526.13	2,257.49
LAUNCH SUPPORT	1,881.90	1,731.54
OPERATIONS	1,378.78	1,281.03
PROPELLANT	503.11	450.51
• COST/FLT =	$7.96 M	$7.18 M
• COST/KG =	$35.	$32.
• ADVANCED TECHNOLOGY SAVINGS: $2,353.66 M		

Main propulsion D&D costs for both stages are small due to use of SSME's on the Orbiter and Dual Fuel/Dual Expander engines on the booster. The only D&D costs associated with the SSME is the addition of an extendable nozzle at $50 million. The Dual Fuel/Dual Expander engine D&D is charged to the SSTO.

The orbiter auxillary propulsion D&D cost includes $45.9 million for development of LO_2/LH_2 RCS thrusters. The SSTO and HLLV booster share this subsystem. The booster has $103.7 million for D&D of the flyback airbreather engine.

TABLE 3.2.2-4. - ADV. TECH. HLLV ORBITER DDT&E COSTS

DDT&E	2509.2			
PROGRAM MANAGEMENT		81.6		
ENGINEERING		892.7		
SYSTEMS ENGINEERING & INTEGRATION			45.5	
SOFTWARE ENGINEERING			16.6	
DESIGN & DEVELOPMENT			830.7	
STRUCTURE				173.6
TPS				96.6
LANDING & AUXILIARY SYSTEMS				94.8
ASCENT PROPULSION				91.9
AUXILIARY PROPULSION				68.5
PRIME POWER				5.0
ELECTRICAL CONVERSION & DISTRIBUTION				24.0
SURFACE CONTROLS				15.4
AVIONICS				86.1
ECS				2.7
PERSONNEL PROVISION				0.0
GSE				153.6
MANUFACTURING		1109.1		
DEVELOPMENTAL TOOLING & S.T.E.			116.7	
TEST HDWE & SPARES			992.4	
TEST		425.8		
SYSTEMS TEST OPERATIONS			368.4	
FLIGHT TEST OPERATIONS			57.4	

TABLE 3.2.2-5. - ADV. TECH. HLLV BOOSTER DDT&E COSTS

DDT&E	2476.8			
PROGRAM MANAGEMENT		72.6		
ENGINEERING		783.3		
SYSTEMS ENG. & INT.			41.1	
SOFTWARE ENGINEERING			15.0	
DESIGN & DEVELOPMENT			727.2	
STRUCTURE				210.5
TPS				15.1
LANDING & AUXILIARY SYSTEMS				69.7
ASCENT PROPULSION				18.4
AUXILIARY PROPULSION				127.2
PRIME POWER				5.3
ELECTRICAL CONVERSION & DISTRIBUTION				18.4
HYDRAULIC CONVERSION & DISTRIBUTION				26.3
SURFACE CONTROLS				14.5
AVIONICS				86.1
ECS				2.7
PERSONNEL PROVISION				0.0
GSE				132.9
MANUFACTURING		1303.3		
DEVELOPMENTAL TOOLING & S.T.E.			123.9	
TEST HDWE & SPARES			1179.4	
TEST		317.7		
SYSTEMS TEST OPERATIONS			317.7	
FLIGHT TEST OPERATIONS			0.0	

The manufacturing subphase of DDT&E includes the cost of the following test hardware:

o	Structural test article -	$126.27 million
o	Propulsion/dynamic test vehicle - (includes $340.41 million refurbishment between tests)	$1021.24 million
o	Flight test vehicle -	$680.83 million

The two test vehicles are refurbished in the production phase for use in the operational fleet.

Production Phase

The production phase cost for the fleet of 3 HLLV's is $1937.6 million. Only one vehicle is actually manufactured during the production phase, the other two are refurbished test vehicles. The cost of refurbishing the two test vehicles is $378.5 million.

The following TFU costs were used in estimating the advanced technology HLLV production costs:

o	Orbiter airframe	$246.76 million
o	Booster airframe	$258.41 million
o	SSME	$15.06 million
o	Dual Fuel/Dual Expander engine	$18.49 million
o	GSE	$145.98 million
o	Complete vehicle	$756.97 million

Operations Phase

The advanced technology HLLV has an operations phase cost of $4436.7 million. This cost is based on the same mission model as the Baseline HLLV. However the payload capability for the advanced technology HLLV is slightly less, resulting in 618 launches over a 15-year span. The maximum launch rate has increased to 75 per year. Using the same turnaround time as the baseline HLLV, 133 hours, and the groundrules of 2.2.1 results in a fleet size of 3 vehicles.

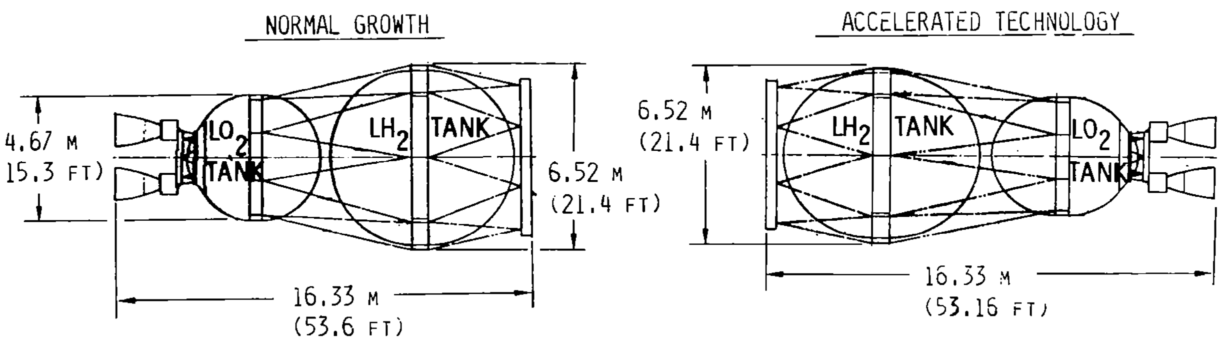
3.2.3 Priority Orbit Transfer Vehicle (POTV)

3.2.3.1 Technology Impact

As shown in figure 3.2.3-1, this vehicle is virtually unchanged. It does have slightly more payload and is an easier vehicle to service due to its integrated Lox/LH_2 subsystems. Within the confines of this space based concept there was little room for technology impact. Tanks were essentially minimum guage, there is little weight tied up in the truss intertia system, and at this size range the subsystem weights are less significant. The resulting λ' coupled to the high I_{SP} of the ASE-type engine leaves little room for improvement.

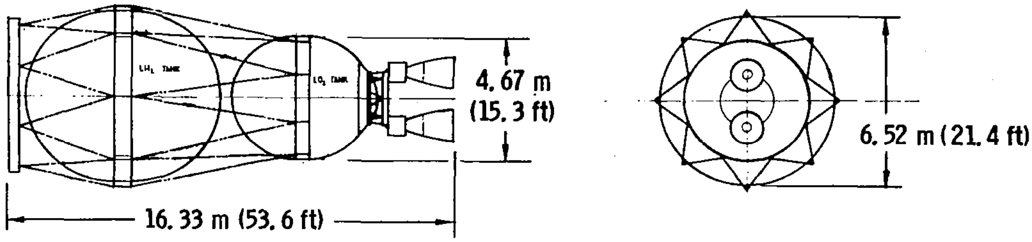
3.2.3.2 Configuration Description

The configuration of the POTV is presented in figure 3.2.3-2 with the overall geometry noted. This configuration is almost identical to the normal growth technology baseline. Main propulsive thrust is provided by two ASE-type engines, each having a vacuum thrust of 89 000N (20 000 lbf). These engines provide for the orbit transfer maneuvers including GEO and LEO terminal phase initiation (TPI) maneuvers. The spacecraft structure consists primarily of truss assemblies and semi-integral main propellant tanks. The truss assemblies are of composite design. Main propellant tanks are of 2219-T87 aluminum. A multi-layer insulation blanket provides for propellant boiloff control and spacecraft meteoroid protection. Electrical power is provided by O_2/H_2 fuel cells.



IMPACT
o DRY WEIGHT DOWN BY 2%*
o DELIVERY PAYLAOD (WITH 75% RETURN) BASICALLY UNCHANGED

Figure 3.2.3-1. - Accelerated Technology Impact on POTV



GROSS WEIGHT	96,579 KG	(212,957 LB)
P/L WEIGHT (75% RETURN)	12,322 KG	(27,170 LB)
RESERVES - MPS/APS/EPS	500 KG	(1,102 LB)
MAINSTAGE PROPELLANT - MPS	78,530 KG	(173,160 LB)
INERT WEIGHT	5,227 KG	(11,525 LB)
T/W @ STARTBURN = 0.188		

Figure 3.2.3-2. - POTV Configuration, Accelerated Technology

A summary mass statement is presented in table 3.2.3-1. Note that the resulting mass fraction, in which the mainstage propellant includes the propellant for TPI's, is 0.938.

Each of the items in the summary mass statement, exclusive of payload and mainstage propellants, is discussed in the following paragraphs including definition of rationale for mass estimates.

Structures and Mechanisms–This group consists of the following: LH_2 and LO_2 tanks, primary trusses, docking/service/equipment/avionics section, thrust structure, and secondary structures. Total mass is 1315 kg (2 899 lb).

o LH_2 and LO_2 Tanks—The tanks containing the liquid hydrogen and liquid oxygen are all-welded 2219-T87 aluminum pressure vessels. Both tanks have stiffened belly-band sections. In addition, the aft dome of the LO_2 tank incorporates a short stiffened cone frustrum section to allow for distribution of main engine thrust loads. The tank pressure shells are designed by room temperature proof test conditions corresponding to 0.2g initial T/W with maximum ullage pressure of 97 kPa (14 psia). To satisfy a 50 mission service life requirement with very high assurance of no-leak with minimal mass impact, the tanks are designed using conservative fracture

TABLE 3.2.3-1. - POTV SUMMARY MASS STATEMENT
ACCELERATED TECHNOLOGY

GROUP	MASS KG	WEIGHT (LB)
STRUCTURES & MECHANISMS	1,315	(2,899)
THERMAL CONTROL	490	(1,080)
MAIN PROPULSION SYSTEM (MPS)	753	(1,660)
AUX. PROPULSION SYSTEM (APS)	389	(857)
ELECT. POWER SYSTEM (EPS)	60	(132)
AVIONICS	183	(403)
MARGIN	318	(703)
DRY	3,508	(7,734)
PAYLOAD (75% RETURN)	12,322	(27,170)
RESIDUAL FLUIDS & GASES	874	(1,928)
RESERVES - MPS/APS/EPS	500	(1,102)
INFLIGHT LOSSES	238	(524)
NOMINAL PROP. - EPS	20	(44)
NOMINAL PROP. - APS	587	(1,295)
MAINSTAGE PROP. - MPS	78,530	(173,160)
GROSS	96,579	(212,957)

$$\text{MASS FRACTION} = \frac{\text{MAINSTAGE PROP (INCL FPR)}}{\text{GROSS, LESS PAYLOAD}} = 0.938$$

mechanics design data (i.e., lower boundary data in lieu of best fit data) and are not pressure cycled between missions. A minimum shell thickness of 0.064 cm (0.025 in) was incorporated. The LH_2 tank mass is 464 kg (1 023 lb), 4.1% of the liquid hydrogen capacity. The LO_2 tank mass, including slosh baffles, is 353 kg (777 lb), 0.52% of the liquid oxygen capacity.

o Primary Trusses—The primary trusses consist of the intertank truss assembly and the forward truss assembly. These truss assemblies incorporate composite tube struts with aluminum end fittings and attachment fittings. Each truss assembly is designed to carry an axial load equal to the total thrust of the two main engines as adjusted by a 1.4 ultimate factor of safety. The masses of the truss assemblies are 50 kg (111 lb) and 29 kg (63 lb) for the intertank assembly and forward assembly, respectively.

- Docking/Service/Equipment/Avionics Section—This structural/mechanical assembly consists of a 0.4m (16 in) high by 5.8m (230 in) outer diameter toroidal framework which incorporates the following: a universal docking system; a peripheral latch/release system for payload accommodation; service connector panels for fluids, gases, and electric power; and support assemblies for major equipment and avionic components. Extensive use is made of composite structural members. Total estimated mass is 349 kg (770 lb).

- Thrust Structure—The thrust structure transmits loads from the two ASE-type engines, each having a vacuum thrust of 89 000N (20 000 lbf), to the aft dome of the LO_2 tank. The structural assembly consists of a 2.0m (80 in) diameter engine mounting ring, sixteen struts, and the fittings for attaching the struts to a support ring in the aft dome. The basic tube struts and the basic I-beam section of the engine mounting ring are composite structures. Total estimated mass is 38 kg (85 lb).

- Secondary Structures—An allowance of 32 kg (70 lb) has been incorporated for secondary structures.

Thermal Control—Thermal control elements consist of MLI blankets, temperature control provisions for the fuel cells and equipment/avionics, and heat protection provisions for the vehicle base region. MLI blankets enclose the LH_2 tank, the LO_2 tank, and the spacecraft structural frame. These blankets consist of thirteen layers of 0.15 mil mylar radiation shields aluminized on both sides, nylon net spaces, and protective cover sheets of 3-mil teflon. The teflon sheet on the outer surface of the spacecraft blanket is aluminized on its backside. The combination of radiation shields and protective covers provides an effective meteoroid protection system for the propellant tanks. The installed mass of the MLI blankets is 415 kg (914 lb.). Total group mass is 490 kg (1080 lb.).

Main Propulsion System (MPS)—The main propulsion system consists of rocket engines and accessories, and the propellant system. Total mass is 753 kg (1 660 lb).

- Rocket Engines and Accessories—Main propulsive thrust is provided by two ASE-type engines, each having a vacuum thrust of 89 000N (20 000 lbf). Pertinent engine characteristics are presented in Section 2.1.2. The dry mass of the two engines including accessories is 413 kg (910 lb).

- Propellant System—This subgroup consists of zero-NPSH boost pumps and electric drive motors, propellant feed/fill/drain system, vent/relief system, helium pneumatic system, and a propellant loading/monitoring system. There is no active pressurziation system. Engine start is satisfied by the combination of propellant aquisition provisions, zero-NPSH boost pumps, and main engine tank head idle (THI). Total subgroup mass is estimated at 340 kg (750 lb).

Auxiliary Propulsion System (APS)–A LO_2/LH_2 auxiliary propulsion system provides for vehicle orientation and small delta-V maneuvers. The APS is integral with the MPS and EPS in that the APS propellant and the EPS reactant are stored in the MPS main tanks and the EPS reactant is routed to the fuel cells via the bellows accumulators in the APS. Based on an APS total installed vacuum thrust of 10 676N (16 x 150 lbf) and LH_2 and LO_2 bellows accumulator capacities of 22 kg (48 lb) of LH_2 and 88 kg (194 lb) of LO_2 (sufficient to allow for GEO docking without recharging), the APS dry mass is 389 kg (857 lb).

Electrical Power System (EPS)–Power source elements consist of two redundent O_2/H_2 fuel cells and a NiH_2 auxiliary battery. The EPS is integral wih the MPS and APS in that its propulsive grade reactant is stored in the MPS main tanks and is routed to the fuel cells via the APS bellows accumulators. Based on a fuel cell rating of 1.3 kW average/2.1 kW peak, and a battery rating of 13 amp-hr/28 volt, the EPS dry mass is 60 kg (132 lb).

Avionics–The avionics group includes elements for guidance, navigation and control, communications, data management, rendezvous and docking, data measuring, and power conversion and distribution. Total mass is 183 kg (403 lb), based on a combination of IUS and space tug data, and consideration of advanced avionics and reduced power requirements.

Margin–A margin allowance of 10% of subsystems dry mass has been incorporated. Total mass is 318 kg (703 lb).

Nominal APS Propellant–The APS delta-V budget for vehicle orientation and small delta-V maneuvers is 78 m/s (253 ft/sec). Based on a specific impulse of 427 sec for the LO_2/LH_2 thrusters, and consideration of vehicle sequential mass, the total nominal APS propellant mass is 587 kg (1 295 lb).

Nominal EPS Propellant—The fuel cell reactant for the integrated MPS/APS/EPS is propulsive grade LO_2/LH_2. The nominal power requirement for the 3-day mission is 53.3 kW-hr. Based on a reactant power density of 2.71 kW-hr/kg (1.23 kW-hr/lb), the EPS propellant mass is 20 kg (44 lb).

Inflight Losses—This group consists of main propellant boiloff for the 3-day mission and main propellant start-stop losses associated with six firing of each of the two main rocket engines. Based on tank geometry, tank size, and MLI blanket definition, the boiloff rate is approximately 1.4 kg/hr (3.1 lb/hr). Propellant start-stop losses per engine per firing are estimated at 11 kg (25 lb). Total mass loss is 238 kg (524 lb).

Reserves, MSP/APS/EPS—The MPS, APS, and EPS are integrated LO_2/LH_2 systems in which the O/F mixture ratio of the nominal propellants is 6:1, 4:1, and 1:1, respectively. Because these systems have common propellant storage tanks, the reserve allowances peculiar to dedicated systems were combined on a root-sum-square basis with consideration of the above O/F mixture ratios. (The dedicated systems reserve allowances are: MPS, 2% of total ideal delta-V; APS and EPS, 10% and 33% of nominal requirements, respectively.) The resulting integrated systems reserve allowance is 500 kg (1 102 lb) with an O/F mixture ratio of 5.97:1.

Residual Fluids and Gases—This group consists of the fluids and gases, onboard at end of mission under nominal conditions, for MPS, APS, and EPS. Total mass is 874 kg (1 928 lb).

- MPS Residuals—This subgroup consists of the following: propellant trapped in engines, propellant lines, boost pumps, and tank bottom sumps/propellant acquisition devices; bias fuel; and gases in the empty tanks. Trapped propellant mass is estimated at 254 kg (560 lb). A single-engine reference bias fuel allowance of 0.50% was used. For multiple engines, this allowance was factored by the inverse of the square root of the number of engines. The resulting LH_2 bias allowance for the two main rocket engines in 0.35%. Bias fuel mass is 68 kg (151 lb). The gases in the empty propellant tanks are at the maximum ullage pressure of 97 kPa (14 psia) and mean temperatures for the GO_2 and GH_2 of 90 K (162°R) and 20 K (36°R), respectively. Gas masses are 271 kg (598 lb) of GO_2 and 217 kg (478 lb) of GH_2. Total mass of MPS residuals is 810 kg (1 787 lb).

- APS and EPS Residuals—The residuals for the APS and EPS portions of the integrated MPS/APS/EPS total 64 kg (141 lb).

3.2.3.3 Advanced Technology POTV Life Cycle Cost

The advanced technology POTV has LCC of \$6442.5 million, a reduction of \$658.9 million from the baseline POTV. The advanced technology POTV LCC is summarized in table 3.2.3-2, along with the baseline LCC for comparison. All costs are in constant 1977 dollars.

DDT&E Phase

The major cost element in the DDT&E Phase cost of \$437.8 million is the D&D cost of the ASE, \$250 million. All subsystems have full D&D costs, except for the LO_2/LH_2 RCS thrusters which are developed by the HLLV.

The manufacturing subphase includes the cost of a structural test article, a propulsion/dynamic test vehicle and a flight test vehicle. The last two are refurbished during the production phase and included in the operational fleet.

TABLE 3.2.3-2. - POTV - COST SUMMARY, ADVANCED TECHNOLOGY ($ IN MILLIONS)

	NORMAL TECH.	ADV. TECH.
TOTAL PROGRAM	7,101.37	6,442.46
DDT&E	437.26	437.78
PROGRAM MANAGEMENT	5.67	6.44
ENGINEERING	313.52	322.17
MANUFACTURING	61.71	55.30
TEST	56.36	53.88
PRODUCTION	114.66	104.58
PROGRAM MANAGEMENT	15.58	14.59
SUSTAINING ENGINEERING	2.34	2.10
MANUFACTURING	96.74	87.89
PROD. TOOLING & S.T.E.	32.57	30.17
FLT. HARDWARE & SPARES	64.17	57.73
OPERATIONS	6,549.45	5,900.10
OPERATIONS SUPPORT	288.16	271.96
PROGRAM SUPPORT	44.40	41.59
SPARES PROCUREMENT	243.76	230.37
LAUNCH SUPPORT	6,261.29	5,628.14
OPERATIONS	213.75	213.75
PROPELLANT	6,047.54	5,414.39
• COST/FLT =	\$4.97M	\$4.29M
• COST/KG (LEO TO GEO) =	\$401.00	\$348.00
• ADVANCED TECHNOLOGY SAVINGS:	\$658.91M	

Production Phase

The advanced technology POTV has the same fleet size as the baseline vehicle. For the 5 vehicle fleet the production phase cost is $104.6 million. The TFU costs used in estimating this cost are:

o	POTV Airframe	$14.67 million
o	ASE	$1.83 million
o	Complete vehicle	$18.24 million

Refurbishment of the two test vehicles, $9.1 million, is included in the flight hardware element of the manufacturing subphase.

Operations Phase

The operations phase cost of $5900.1 million is based on the same costing groundrules as the baseline vehicle.

Program support includes the LEO delivery costs of spares and stages. The operations cost element within the launch support subphase is the support, rotation and re-supply costs of a 12-man crew at the LEO base, dedicated to POTV operations while the propellant costs include delivery to LEO.

3.2.4 Large Cargo Orbit Transfer Vehicle (LCOTV)

3.2.4.1 Technology Impact

Although there would appear to be some significant changes in this vehicle as shown in figure 3.2.4-1, the vehicles basic problems remain unchanged. It is still a very large and expensive vehicle which must be assembled on orbit. It still requires a large fleet size to meet the mission requirements. Thirteen vehicles are produced to fly 56 missions. The fleet production costs cannot be amortized and therefore technology impacts are masked.

The significant changes in array area and dry weight are not reflected in the thruster requirements since glow remains at approximately the same level. This is a result of the high I_{SP} of this propulsion system, which creates a very high gross weight to inert weight relationship.

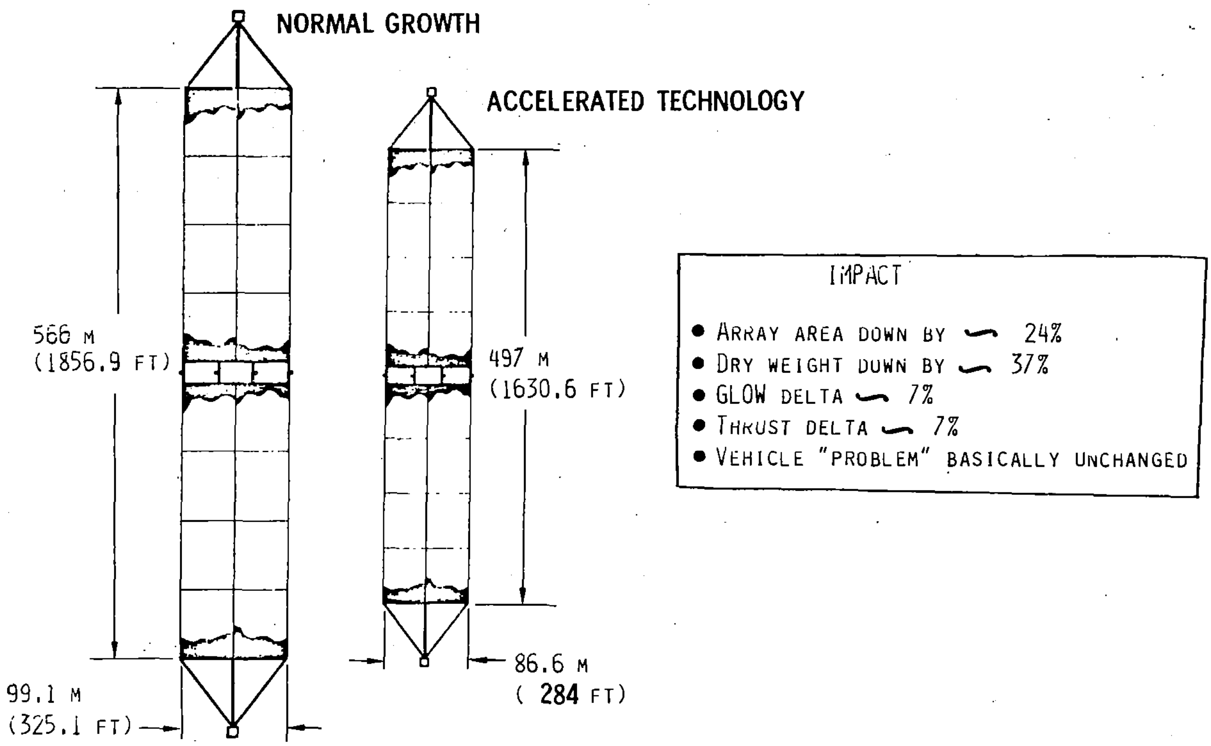


Figure 3.2.4-1. - Accelerated Technology Impact on LCOTV

The most important technology impact may be the overall simplification of the propulsion system as a result of the reduced number of thrusters running off their own dedicated array sections.

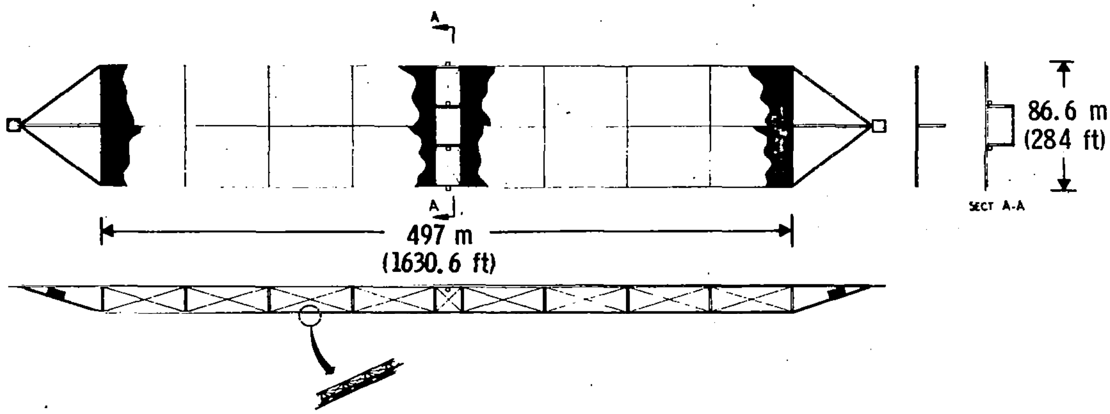
3.2.4.2 Configuration Description (Accelerated Technology LCOTV)

The configuration of the LCOTV utilizing accelerated technology is shown in figure 3.2.4-2 This configuration is similiar to the normal growth vehicle in almost all aspects with the exception of the overall size and the details of some subsystems.

The summary mass statement for this vehicle is shown in table 3.2.4-1. Each of the items is discussed below.

Structures–This item consists of the tri-chord beams which make up the vehicles basic framework including the solar array section subframes and the thruster module standard panels.

Since the beams were minimum gauge and the thruster power structure bond on the WER there are no changes to the normal growth baseline rationale in this area.



ARRAY AREA	=	54,416 m^2
NUMBER OF THRUSTERS	=	206
TOTAL THRUST	=	145 Newtons

Figure 3.2.4-.2 - LCOTV Configuration, Accelerated Technology

TABLE 3.2.4-1. - LCOTV Characteristics and Weights

WEIGHTS	NORMAL GROWTH KG	NORMAL GROWTH (LB)	ACCELERATED TECHNOLOGY KG	ACCELERATED TECHNOLOGY (LB)
STRUCTURE	4,057	(8,946)	2,880	(6,350)
POWER GEN.	26,831	(59,162)	22,212	(48,977)
PROPULSION	11,671	(25,735)	1,979	(4,364)
PROPELLANT SYSTEM	2,217	(4,888)	2,005	(4,421)
THERMAL CONTROL	377	(831)	68	(150)
AVIONICS	520	(1,147)	520	(1,147)
GROWTH*	9,339	(20,593)	5,075	(11,190)
DRY	55,012	(121,301)	34,739	(76,600)
PROPELLANT	29,744	(65,586)	26,901	(59,317)
RESERVES	892	(1,967)	784	(1,729)
PAYLOAD	227,000	(500,535)	227,000	(500,535)
START BURN	312,648	(689,389)	289,424	(638,180)
ARRAY AREA	54,416 m^2		41,495 m^2	
NO. OF THRUSTERS	206		26	
Isp	8,000		8,000	
TOTAL THRUST	145 N		135 N	

*Minimum 10% - Maximum 25%

Power Generation–This area was substantially affected by the basic change in array from silicon to GaAs and the impact on the power distribution system due to direct power usage of the thrusters.

The solar array unit weight including installation is shown in table 3.2.4-1. At 384.1 grams/m^2 it is significantly lighter than the normal growth array. In addition, this superior output efficiency results in a reduced array requirement. The power distribution system is primarily affected by the increased number of individual power sections served. Each of the 206 thrusters has its own power sections.

Propulsion–This item consists of the thrusters, power provisions and the auxiliary propulsion system. The significant decrease in mass results from the reduction in number of thrusters from 206 to 26 and the elimination of the power processing equipment the mass allocation for the APS was unchanged.

Propellant System–This system includes tanks and feed systems. The mass of this system is not a significant factor. No fundamental changes were made other than to accommodate the change in propellantload and feed system geometry.

Thermal Control–Since this item consists entirely of a radiation system for the power processors it was significantly reduced. No technology changes were incorporated from the normal growth baseline.

Avionics–The avionics includes a full suit of the POTV avionics with an allowance for radiation shielding. No changes were made in this area.

3.2.4.3 Advanced Technology LCOTV Life Cycle Costs

The advanced technology LCOTV has an LCC of $3236.1 million, distributed as shown in table 3.2.4-2. The following paragraphs discuss the LCC breakdown. All costs are in 1977 constant dollars.

<u>DDT&E</u>

The advanced technolocy LCOTV has a DDT&E phase cost of $596.4 million. The increase over the normal technology LCOTV is due to the D&D costs of the advanced long life thruster $166 million as opposed to $25 million for the baseline thruster. There is also an increase in systems test costs associated with the higher cost thrusters and GaAs solar array.

TABLE 3.2.4-2. - LCOTV - COST SUMMARY
($ IN MILLIONS)

	NORMAL TECH.	ADV. TECH.
TOTAL PROGRAM	3,276.06	3,236.08
DDT&E	387.93	596.42
PROGRAM MANAGEMENT	12.54	23.59
ENGINEERING	150.86	292.07
MANUFACTURING	128.18	130.38
TEST	96.35	150.38
PRODUCTION	1,930.12	1,911.38
PROGRAM MANAGEMENT	70.74	84.48
SUSTAINING ENGINEERING	51.81	61.50
MANUFACTURING	1,807.57	1,765.35
PROD. TOOLING & S.T.E.	194.51	164.20
FLT. HARDWARE & SPARES	1,613.06	1,601.15
OPERATIONS	958.01	728.33
OPERATIONS SUPPORT	574.78	355.53
PROGRAM SUPPORT	122.20	90.11
SPARES PROCUREMENT	452.58	265.42
LAUNCH SUPPORT	383.23	372.80
OPERATIONS	285.00	285.00
PROPELLANT	98.23	87.80
• COST/FLT	$17.11 M	$13.01 M
• COST/KG (LEO TO GEO)	$75	$57
• ADVANCED TECHNOLOGY SAVES -	$39.98 M	

Production

The production phase cost of $1911.4 million dominates the LCOTV LCC. This results from the large fleet size of 13 vehicles required to meet the mission model. Mature industry costing methodology was used to estimate GaAs array costs. The array unit cost was $1520/m^2 for a production rate of 44 000 m^2/year. Because of the large, long life thruster fewer were required on the advanced technology LCOTV, the production rates were not large enough to justify mature industry costing. Traditional aerospace costing was used to estimate an average cost of $8175/kg for the total production sum of 470 thrusters (50 kg each).

Operations Phase

The advanced technology LCOTV operations phase cost of $728.3 million is reduced by $229.7 million from the baseline LCOTV. Most of this reduction is attributable to the long life thrusters. The refurbishment rate on these is 10% per flight as is the remainder of the vehicle, except for the solar array. There are no array spares cost because it is annealed after each mission. The $285 million in the launch support subphase is the rotation and re-supply cost of the 12-man crew at LEO that refurbishes the vehicles between missions.

4.0 FINDINGS AND CONCLUSIONS

4.1 ACCELERATED TECHNOLOGY VALUE

Section 3 described the individual advanced technology assessments and their combined effect on each of the vehicles. Table 4.1-1 summarizes the LCC benefits of the individual assessments. These assessments were made separately on the reference vehicles. The benefits of combinations of the advanced technologies would be less than the sum of the individual assessments. The cost benefits shown are in constant 1977 dollars.

In order to further evaluate the effect of the advanced technologies the benefits were estimated using a 10% discount rate. The breakeven R&D costs were calculated assuming a 5-year R&D program preceding the beginning of a 5-year DDT&E phase. Table 4.1-2 lists LCC benefits in constant dollars, discounted dollars and the equivalent R&D funding in discounted dollars. Those technologies that had negligible or negative LCC benefits in constant year dollars were not evaluated in discounted dollars.

Table 4.1-1. Accelerated Technology Life Cycle Costs Benefits Summary

LIFE CYCLE COST DELTA $ x 10^6

TECHNOLOGY	SSTO	HLLV	POTV	LCOTV	TOTAL
Composite Structures	-3112	-986	—	—	-4098
Dual Expander Engine	-2118	-531	—	—	-2649
Eliminate Vertical Tail	-1652	-412	—	—	-2064
Extended Life SSME	-1261	-474	—	—	-1735
Integrated Subsystems	-434	-30	-102	—	-566
Slush LH_2	-300	—	—	—	-300
Improved Avionics	-219	-9	-18	—	-246
Metallic TPS	-76	-36	—	—	-112
Plus Cluster Engine	—	—	(+2)	—	(+2)
Gallium Arsenide Array	—	—	—	(+58)	(+58)
100 cm Thruster	—	—	—	(+216)	(+216)
Long Life Thruster	—	—	—	(+72)	(+72)
Direct Power Processing	—	—	—	-386	-386

•These data reflect the benefit when each item is evaluated "by itself" on the reference vehicles.

TABLE 4.1-2.—ADVANCED TECHNOLOGY IMPACT BENEFITS AND EQUIVALENT R&D FUNDING

TECHNOLOGY	$MILLION		
	BENEFITS		DISCOUNTED R&D FUNDING
	CONSTANT $	DISCOUNTED $	
Composite Structure	4098	2073	1544
Dual Expander Eng.	2649	878	653
CCV (No vertical tail)	2064	1007	750
Extended Life SSME	1735	331	247
Integrated Subsystems	566	257	191
Slush LH_2	300	179	133
Improved Avionics	246	119	89
Direct Power Processing	386	205	152

The R&D costs shown in table 4.1-2 represent the breakeven point between R&D expenditures and discounted LCC savings. These are R&D funds over and above DDT&E costs.

There are also substantial savings when the advanced technologies are applied to the whole tranportation system. Table 4.1-3 compares the total system LCC's for normal growth and advanced technology. Implementing the advanced technologies produces a savings of $8.310 billion in constant 1977 dollars. The savings are $3.821 billion, assuming a 10% discount rate. The breakeven level in R&D funding is $2.845 billion.

4.2 CONCLUSIONS

This study mets its basic objectives and in addition provided valuable insights into several other areas. Its perspective from a total and integrated transportation system was important in assessing the technology areas as noted below in a summary of the study findings:

- Accelerating technology pays off. A twenty percent reduction in life cycle costs was realized. Technology advancement was more important to the launch vehicles particularly the SSTO.

Table 4.1-3. Total System Cost Summary ($ in Millions)

	NORMAL TECH.	ADV. TECH.
TOTAL PROGRAM	41,636.73	33,327.27
DDT&E	9,783.53	8,638.80
PROGRAM MANAGEMENT	277.82	249.73
ENGINEERING	3,708.97	3,771.14
MANUFACTURING	4,385.84	3,355.19
TEST	1,407.47	1,262.74
PRODUCTION	8,542.05	6,183.23
PROGRAM MANAGEMENT	696.69	444.40
SUSTAINING ENGINEERING	193.35	159.10
MANUFACTURING	7,701.72	5,579.73
PRDD. TOOLING & S.T.E.	2,200.79	1,297.24
FLT. HARDWARE & SPARES	5,500.93	4,282.49
OPERATIONS	23,311.15	18,505.24
OPERATIONS SUPPORT	10,531.56	7,193.06
PROGRAM SUPPORT	2,185.65	2,156.53
SPARES PROCUREMENT	8,345.91	5,036.53
LAUNCH SUPPORT	12,779.59	11,312.18
OPERATIONS	5,054.71	4,778.65
PROPELLANT	7,724.88	6,533.53

- **ADVANCED TECHNOLOGY SAVINGS: $8,309.64 M**

o Normal growth represented a substantial improvement from todays state of the art. This is important in two respects; it emphasizes the improvement seen from accelerating technology and; it represents a substantial challenge in itself. A new high pressure, hydrocarbon booster engine and wide spread application of composites are two key items in the baseline.

o Composite structures with their promise of reduced weight and lower production costs is the most important technology area. Its high value was common accross the entire transportation system.

o A dual fuel/dual expander engine is critical technology for a single stage to orbit launch vehicle.

o Control configured vehicle technology, particularly as applied to the removal of the vertical tail, shows excellent potential for launch vehicles. It not only can lead to vehicle improvements but will also ease the problems of operations and facilities.

o Extended life engines have great value. The true benefits of a reusable STS via a substantial mission model will only be realized when engine technology provides improved life, low maintenance characteristics.

- o A. single-stage to orbit vehicle not only becomes viable but can be considered attractive when technology is accelerated for its purposes.
- o The HLLV improvements with technology, although less dramatic then the SSTO, were substantial. Life cycle costs were reduced 18 percent and cost per flight dropped ten percent.
- o The impact on the OTV's was not as significant as for the launch vehicles. The electric LCOTV suffered because of its mismatch with the mission model. This vehicle requires a mission requirement which is not only large but carried over a significant time period allowing amortization of its high production costs.

- o The POTV, configured as an all propulsive space based concept, had little room for improvement from its normal growth baseline. The two areas where a substantial change in vehicle characteristics could be effected, inert weight and propulsion, were already at high performance levels.

In addition to these basic study findings, the following conclusions can also be drawn:

- o Space basing of OTV's can offer significant advantages particularly in decoupling these vehicles from launch vehicle constraints. In this study, the operational requirements peculiar to space basing were allocated entirely to other parts of the system, particularly to the space base. Additional study in this area is required.

- o Significant improvements in the space based chemical OTV can only be made by changes in its operating modes such as use of aero-assist or GEO refueling. The impact of these operational changes may result in identification of additional high yield technology areas.

- Technology findings are sensitive to the vehicle concepts chosen. Some examples of the technology areas possibly affected are tabulated below.

o	Horizontal T.O. SSTO	- TPS/Structures
o	Less Than Two-stage HLLV or Reduced Size HLLV	- Increased Sensitvity To Technology
o	Ballistic HLLV	- TPS/Structures
o	Shuttle Tended-space Based OTV	- Thermal Control/ Maintenance Free Design
o	Aero-assisted OTV	- TPS/Guidance
o	Ground Based OTV	- Structures/Thermal Control
o	"Low G" Chemical LCOTV	- Propulsion

Figures 4.2-1, 4.2-2, 4.2-3 and 4.2-4 protray the vehicle designs as evolved under accelerated technology

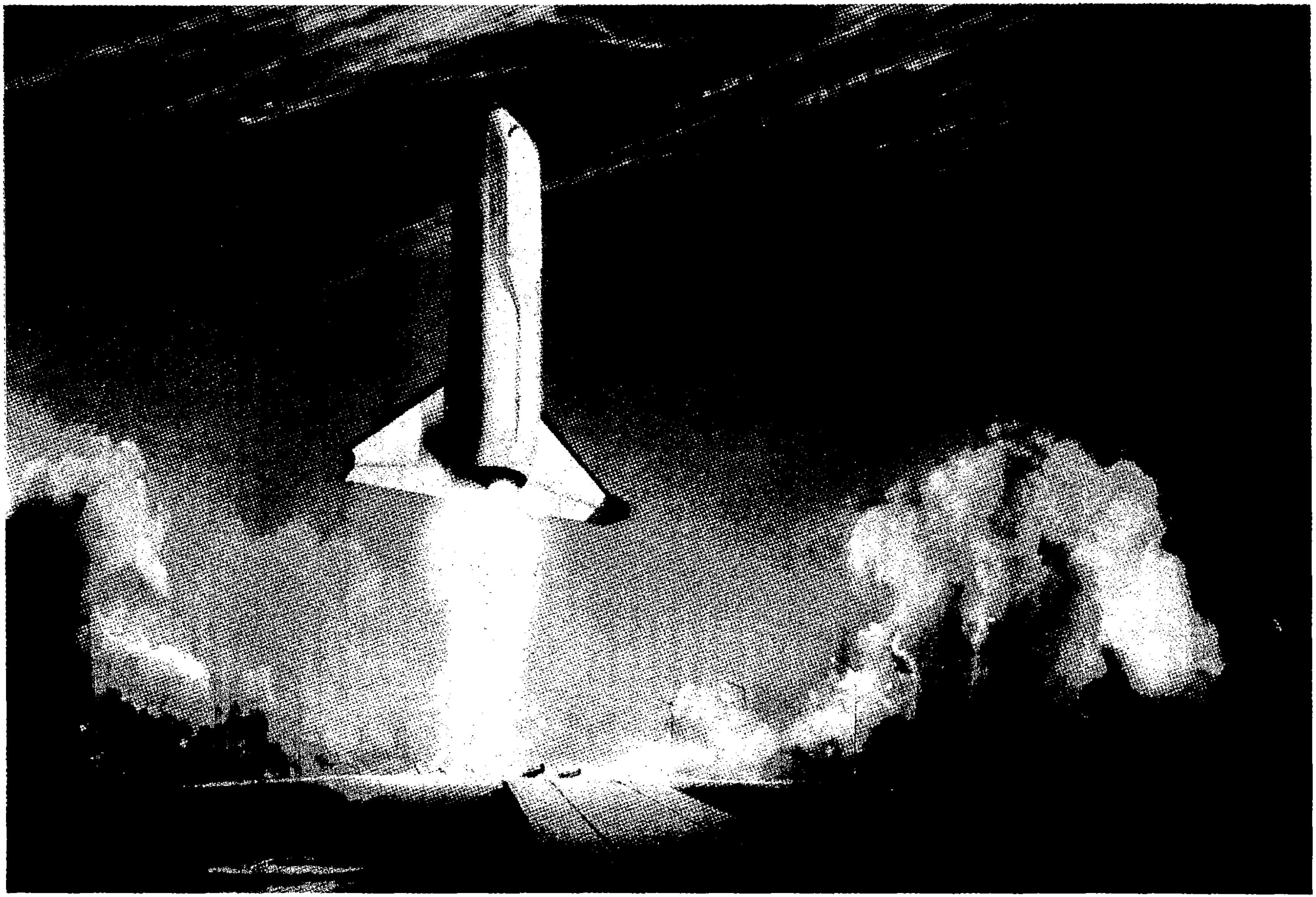


Figure 4.2-l. SSTO Vehicle Concept - Accelerated Technology

Figure 4.2-2. HLLV Concept - Accelerated Technology

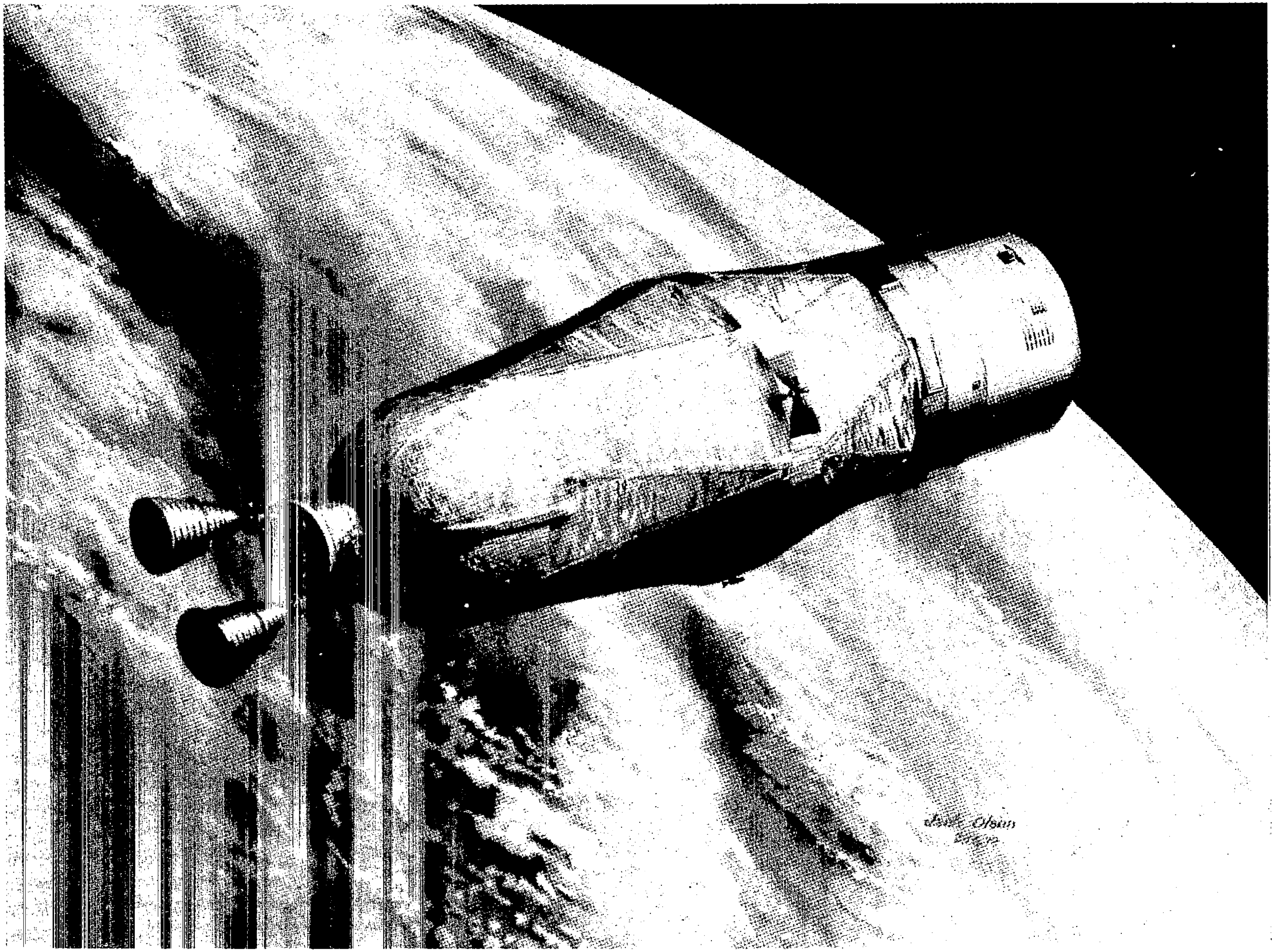


Figure 4.2-3 POTV Concept - Accelerated Technology

FIGURE 4.2-4- LCOTV CONCEPT-ACCELERATED TECHNOLOGY

5.0 GROUND VS SPACE BASED POTV ANALYSIS

5.1 INTRODUCTION

This "Ground vs Space Based POTV Preliminary Analysis" was undertaken as the major effort under a short extension of the basic contract.

Its objectives are summarized below.

- o Design a ground-based orbit transfer vehicle
  - o Single launch delivery by a heavy lift launch vehicle
  - o Maximum payload
- o Establish ground-based and space-based orbit transfer vehicle operation requirements
  - o Timelines
  - o Resources
  - o Facilities
- o Compare operation requirements
  - o Separate design issues from technology issues
  - o Identify enhancing and enabling technologies

The two subtasks undertaken to meet these objectives were: a) an operations analysis and, b) a design analysis. Figure 5.1-1 illustrates the operations analysis task approach. Due to the resource limitations of the extension and the wide scope of these objectives, the analysis reported herein must be called preliminary. Its limitations include: a) only five major functions were reviewed to a level which allowed definition of requirements; b) the operations analysis was not iterated; c) many important trades were not made. Subjective judgement was substituted in order to follow through with the identified analysis technique.

This study was valuable, however, in a number of ways. The design analysis (section 5.2) clearly establishes the dry mass advantage of the space based design over the ground based for the size range addressed. This advantage is sufficient to override liberal application of system redundancy which may be required to reduce "in space" servicing. This is clearly

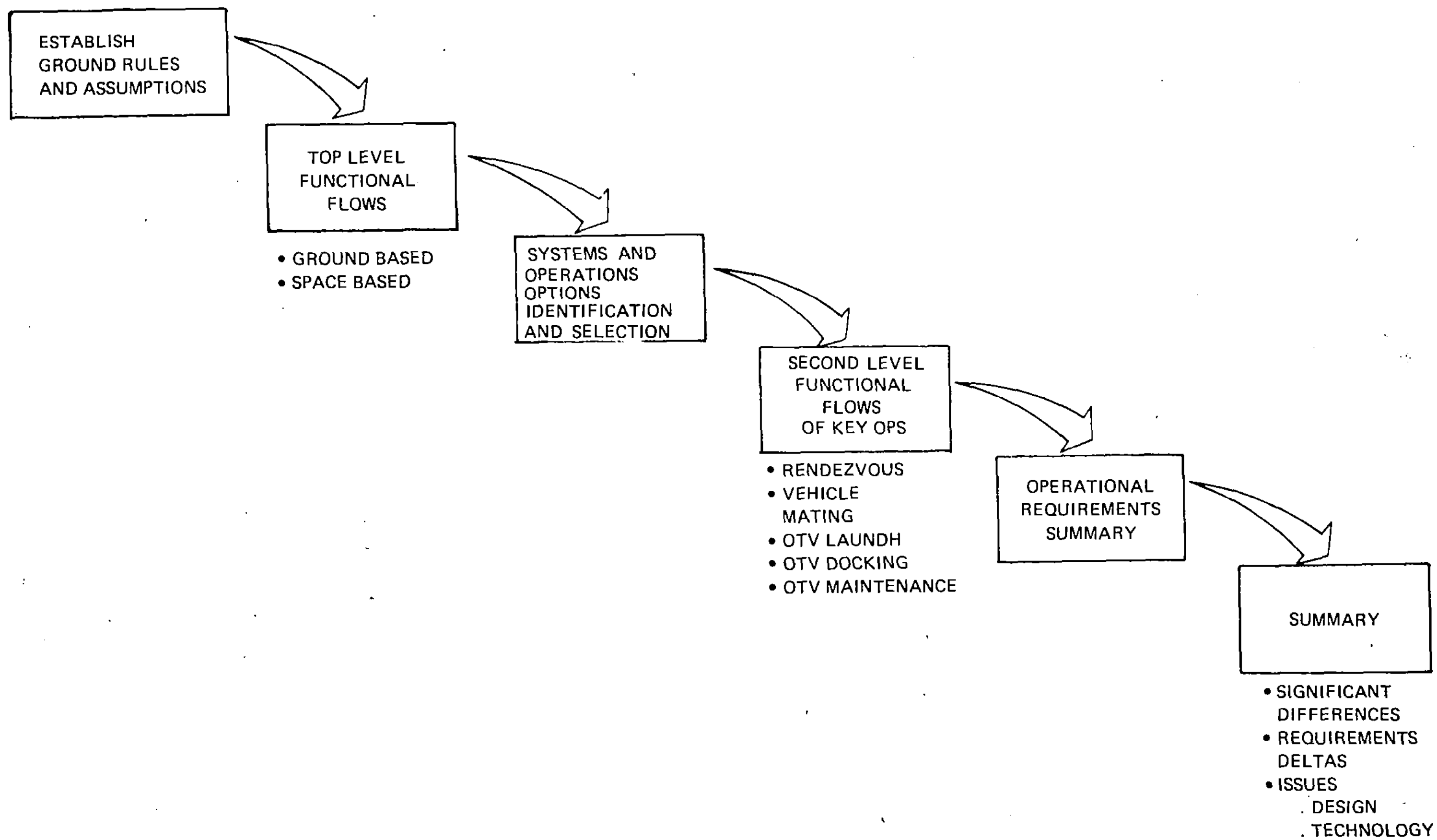


Figure 5.1-1.—Operations Analysis Task Logic

one of the major technology issues identified by this analysis. These issues are summarized in section 5.5. The analysis approach used in this task, and in a sense tested in its accomplishment, is a valuable tool for driving out the operational issues which must be addressed before the definitive case between ground and space basing can be made.

5.2 DESIGN EVALUATION - GROUND BASED VS SPACE BASED POTV

LO_2 Dump Considerations - The HLLV has an ascent payload capability of 222 259 Kg (490 000 lbs) and a landing payload capability of 22 680 Kg (50 000 lbs). The ground based POTV has a gross weight (no GEO payload) of 84 368 Kg (186 000 lbs) and a 100% LO_2 dump weight of 16 329 Kg (36,000 lbs). ASE will yield a net 100% LO_2 dump weight of approximately 20 412-22 680 Kg (45 000-50 000 lbs). Thus, with respect to HLLV orbiter landing weight, a POTV 100% LO_2 dump is mandatory. The HLLV orbiter landing cg is located at 71.9% body length with 22 680 Kg (50 000 lbs) of payload located at mid-bay. This landing cg is acceptable. With the POTV positioned in the orbiter payload bay with the LH_2 tank forward, with 100% LO_2 dump, and with consideration of ASE, the cg of the POTV/ASE is only slightly forward of mid-bay and the orbiter landing cg is at 71.6% body line. Thus, with respect to HLLV orbiter landing cg, a POTV 100% LO_2 dump is feasible. In summary, a POTV 100% dump is both mandatory (landing weight) and feasible (landing cg) and will be reflected in the structural design of the ground based POTV.

Structural Configuration Impact - The structural configuration impact of converting the POTV from a space based vehicle to a ground based vehicle is summarized below.

The space based vehicle is mounted inverted in the HLLV orbiter payload bay such that its GEO payload attach ring functions as the OTV support ring that transmits x-axis loads to the ASE adapter. The y-axis and z-axis loads are transmitted to ASE lateral bracing via fittings attached to the external tabs on the main tankage belly-band section.

For the ground based vehicle, the 16-strut intertank truss assembly has been replaced with a 16-strut upper assembly and a 32-strut lower assembly. These truss assemblies interface with each other at an OTV support ring located at the longitudinal cg of the fully tanked OTV. The OTV support ring transmits x-axis, y-axis, and z-axis loads to the ASE adapter which is laterally braced. In addition, during an abort landing, the large $\pm$ y-axis and -z-axis loads of the fully loaded LH_2 tank are transmitted to ASE lateral bracing via pads attached to the external tabs on the tank belly-band section. The POTV structural configuration and support scheme allows for excellent thermal isolation of the POTV propellant tanks during prelaunch operations. The space based and ground based POTV configurations are shown in figure 5.2-1.

Weight Impact - The performance change due to this weight change is a reduction in payload from 12 129 Kg (26 740 lbs) from the space based vehicle to 9 126 Kg (20 120 lbs) for the ground based vehicle (reference figure 5.4-1). The payload weight reduction is 3003 Kg (6 620 lbs) (25%). A weight comparison at the subsystem level is presented in figure 5.2-2. Note that the structure weight has more than doubled and that the stage mass fraction has decreased from 0.936 to 0.910. A weight change analysis is presented in figure 5.2-3.

5.3 TOP-LEVEL OPERATIONAL REQUIREMENTS ANALYSIS

The ground rules and assumptions that were given as the starting point for operations analysis are shown in table 5.3-1. A manned GEO sortie was used as the mission model for both the ground-based and the space-based OTV operations analysis. This mission was selected since it includes the widest scope of operations.

The top-level functional flow sequence for the ground-based OTV operations is shown in figure 5.3-1. The corresponding top-level functional flow for the space-based OTV operations is shown in figure 5.3-2. These top-level functional flows would be completed by including emergency modes and non-manned missions.

The same block numbers are given for both the ground-based and space-based operations where the operations are identical. Where there are significant differences in corresponding blocks, the space-based functional blocks are given an "A" notation. Where space-based OTV operational blocks have no equivalent in the ground-based functional flow, the unique blocks are given unique numbers. Table 5.3-2 summarizes these differences.

The functional blocks that are circled in table 5.3-2 were selected for a second level functional analysis. Time did not allow all of the functional delta's to be detailed. The ones selected were those that might display the key differences between the ground-based and space-based scenarios.

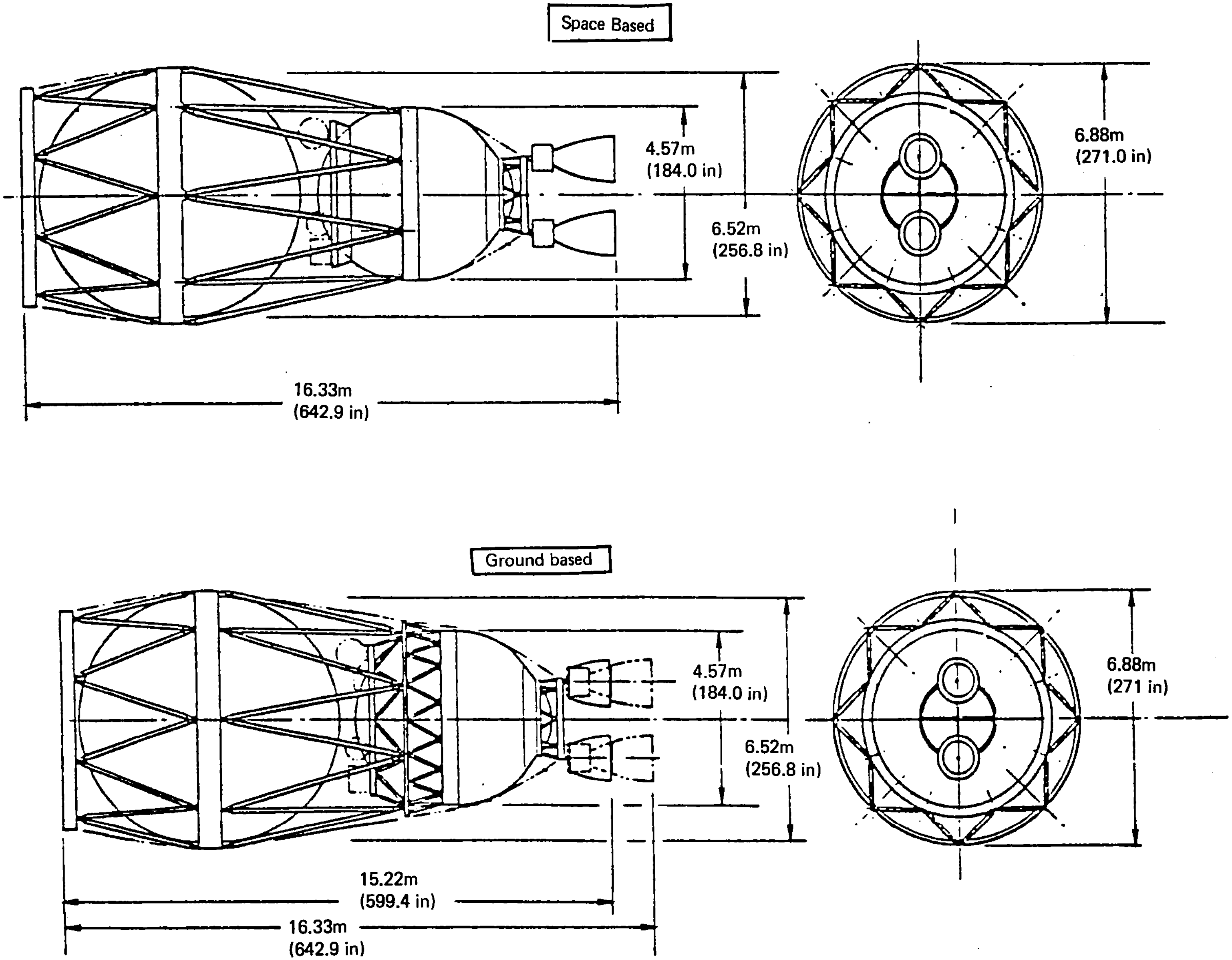


Figure 5.2-1. Point design chemical fueled OTV concept.

	Space Based		Ground Based		Weight Change	
	Kg	Lb	Kg	Lb	Kg	Lb
Structures and Mechanisms	1,306.8	2,881	2,843.6	6,269	+1,537	+3,388
Thermal Control	487.6	1,075	623.7	1,375	+136	+300
Main Propulsion System (MPS)	752.9	1,660	780.2	1,720	+29.9	+60
* Auxiliary Propulsion System/Reaction Control System (APS/RCS)	428.6	945	428.6	945	—	—
Electrical Power System (EPS)	79.4	175	80.3	177	+0.9	+2
Avionics	213.2	470	213.2	470	—	—
Margin	327.0	721	497.1	1,096	+120.1	+375
(Dry Weight)	(3,595.6)	(7,927)	(5,466.7)	(12,052)	(+1,871)	(+4,125)
MPS Residual Fluids and Gases	807.4	1,780	1,061.4	2,340	+254	+560
APS/RCS Residual Fluids and Gases	70.8	156	70.8	156	—	—
EPS Residual Fluids and Gases	2.3	5	2.3	5	—	—
MPS Inflight Losses	236.8	522	332	732	+95.3	+210
MPS Flight Performance Reserves	503.5	1,110	503.5	1,110	—	—
APS/RCS Reserves	58.9	130	58.9	130	—	—
EPS Reserves	6.4	14	6.4	14	—	—
MPS Nominal Propellant	78,544.1	173,160	76,743.3	169,190	-1,800.8	-3,970
APS/RCS Nominal Propellant	595.1	1,312	595.1	1,312	—	—
EPS Nominal Reactant	19.9	44	22.2	49	+2.3	+5
(Gross Weight)	(84,440)	(186,160)	(84,862)	(187,090)	(+421.8)	(+930)

$$' = \frac{\text{MPS Nominal Propellant + FPR}}{\text{Gross Weight}}$$

*Dedicated Subsystems

Figure 5.2-2. POTV Weight Comparison–Ground Based vs. Space Based

	Kg	Lb
Main Fuel Tank	441.3	+973
Main Oxidizer Tank	342.5	+755
Redesign tanks for increased cyclic life, higher vent pressures (due to higher propellant vapor pressures), larger head pressures, and 9g crash loads distribution.		
Intertank Truss Assembly	653.2	+1440
Replace existing 16-strut assembly with a 16-strut upper assembly and a 32-strut lower assembly, the truss assemblies interfacing with each other at an OTV support ring. Design upper assembly for 9g cash load (tension) and 3g boost load (compression). Design lower assembly for 3g boost load (tension). Design support ring for 3g boost load/ASE interface.		
Support Pads	68.0	+150
New item. During abort landing, pads transmit large lateral loads of fully loaded LH2 tank to ASE lateral bracing.		
Secondary Structures	31.8	+70
Increased allowance.		
Insulation Blanket Purge System	136.0	+300
New item. Consists of plumbing distribution and control components to provide for prelaunch conditioning to remove contaminants and for pressurization during entry to prevent the absorption of atmospheric contamination.		
Main Oxidized Dump Provisions	27.2	+60
New item. Consists of plumbing distribution and control components to provide for abort dump of main oxidizer.		
Fuel Cell Reactant Storage Bottles	.91	+2
Larger storage bottles to accommodate reactant usage during prelaunch/ascent.		
Margin	170.0	+375
Adjustment to maintain 10%.		
GH2 In Empty Fuel Tank	113.4	+230
GO2 In Empty Oxidizer Tank	149.7	+330
Adjustments to reflect higher vent pressures (and temperatures) associated with higher vapor pressures.		

Figure 5.2-3. POTV Weight Change Analysis Summary Ground Based vs. Space Based

	Kg	Lb
H2 Boiloff	68.0	+150
O2 Boiloff	27.2	+60
Adjustment to reflect losses during prelaunch/ ascent.		
Fuel Cell Reactant	2.3	+5
Adjustment to reflect usage during prelaunch/ ascent.		
Nominal MPS LH2	-172.4	-380
Nominal MPS LO2	-176.9	-390
Adjustments associated with increased mass of vapor in empty main tanks.		
Nominal MPS LH2	-244.9	-540
Nominal MPS LO2	-1206.5	-2660
Adjustments associated with decreased densities due to higher vapor pressures (load at 110kPa (16 psia) in lieu of 689kPa (10 psia).		

Figure 5.2-3. POTV Weight Change Analysis Summary Ground Based vs. Space Based (continued).

Table 5.3-1. Operations Analysis Ground Rules and Assumptions

COMMON TO BOTH

- o Use a manned GEO sortie as the mission model
- o HLLV unmanned
- o SSTO manned
- o OTV mission crew transported to LEO in the crew module carried within the SSTO
- o OTV mission components attached to crew module on earth

GROUND BASED OTV OPERATIONS

- o HLLV launched before the SSTO
- o HLLV transports the OTV

SPACE BASED OTV OPERATIONS

- o HLLV transports propellants in an HLLV tanker orbiter vehicle
- o Vehicle Operations Center located on a multi-purpose space base in LEO
- o OTV propellant transferred from tanker (tanks)
- o Limited maintenance on orbit

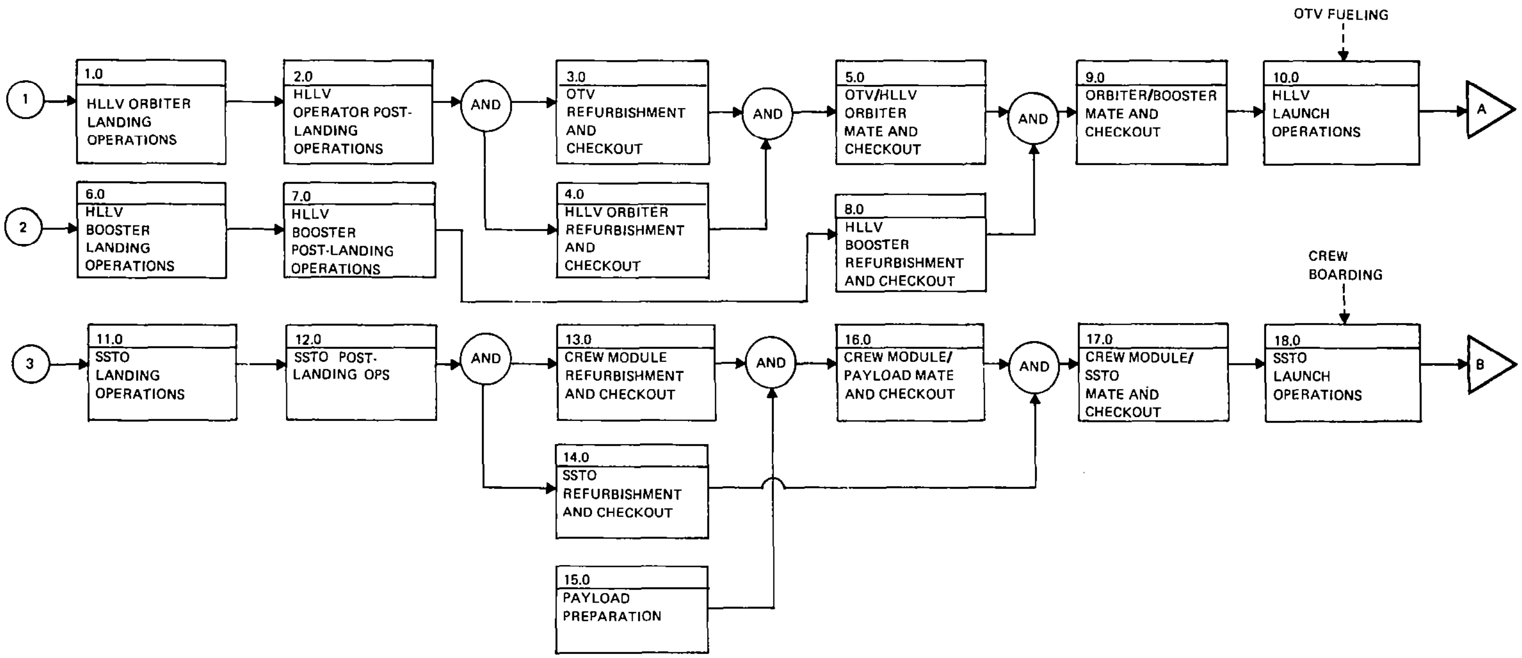


Figure 5.3-1.—Ground Based POTV Operations Top-Level Functional Flow

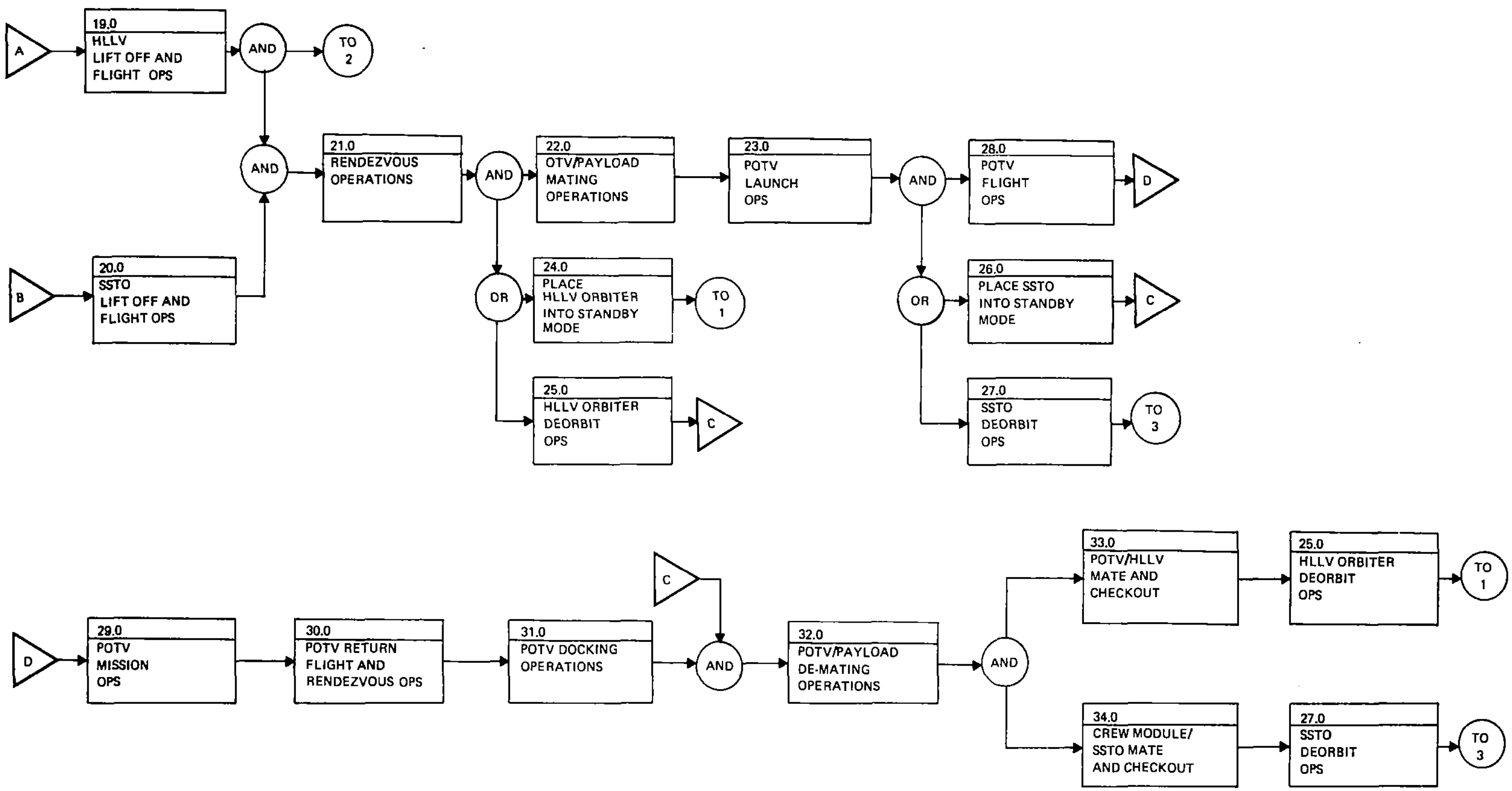


Figure 5.3-1.—Ground Based POTV Operations Top-Level Functional Flow (Continued)

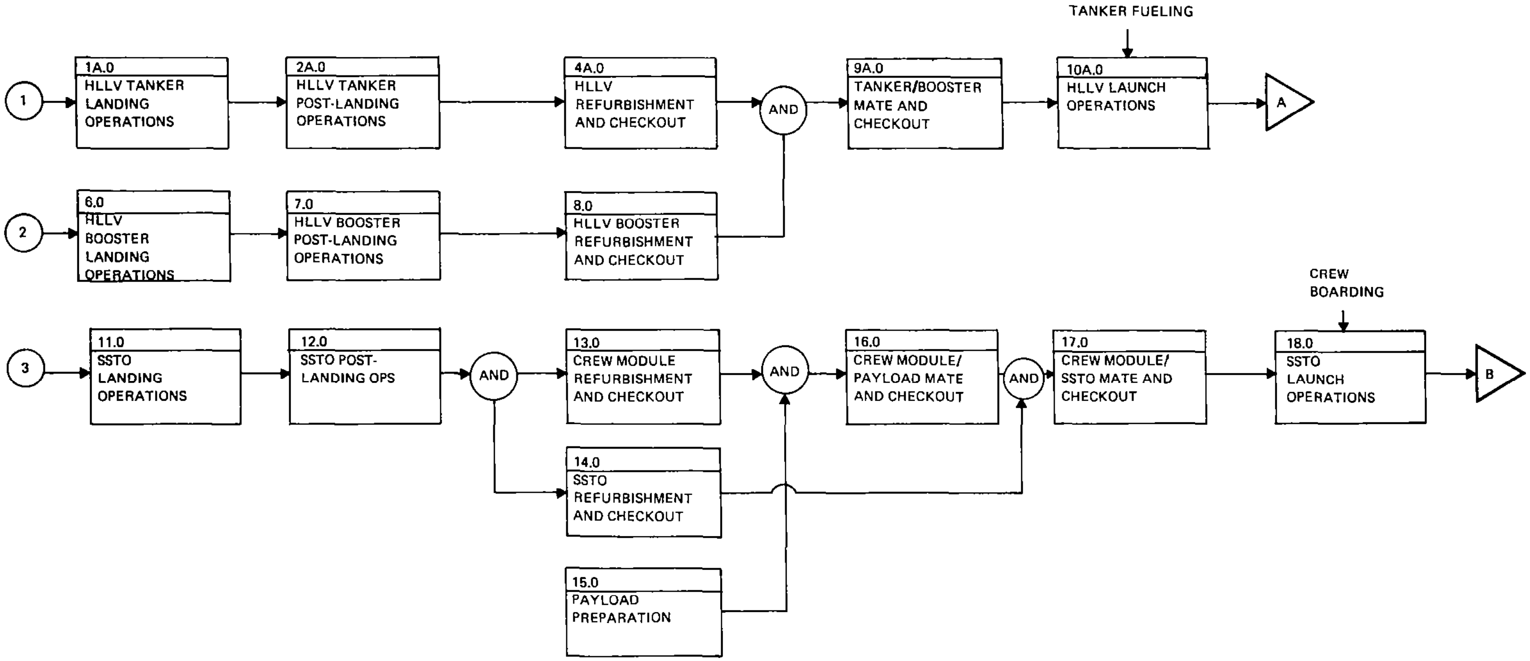


Figure 5.3-2.—Space Based POTV Operations Top-Level Functional Flow

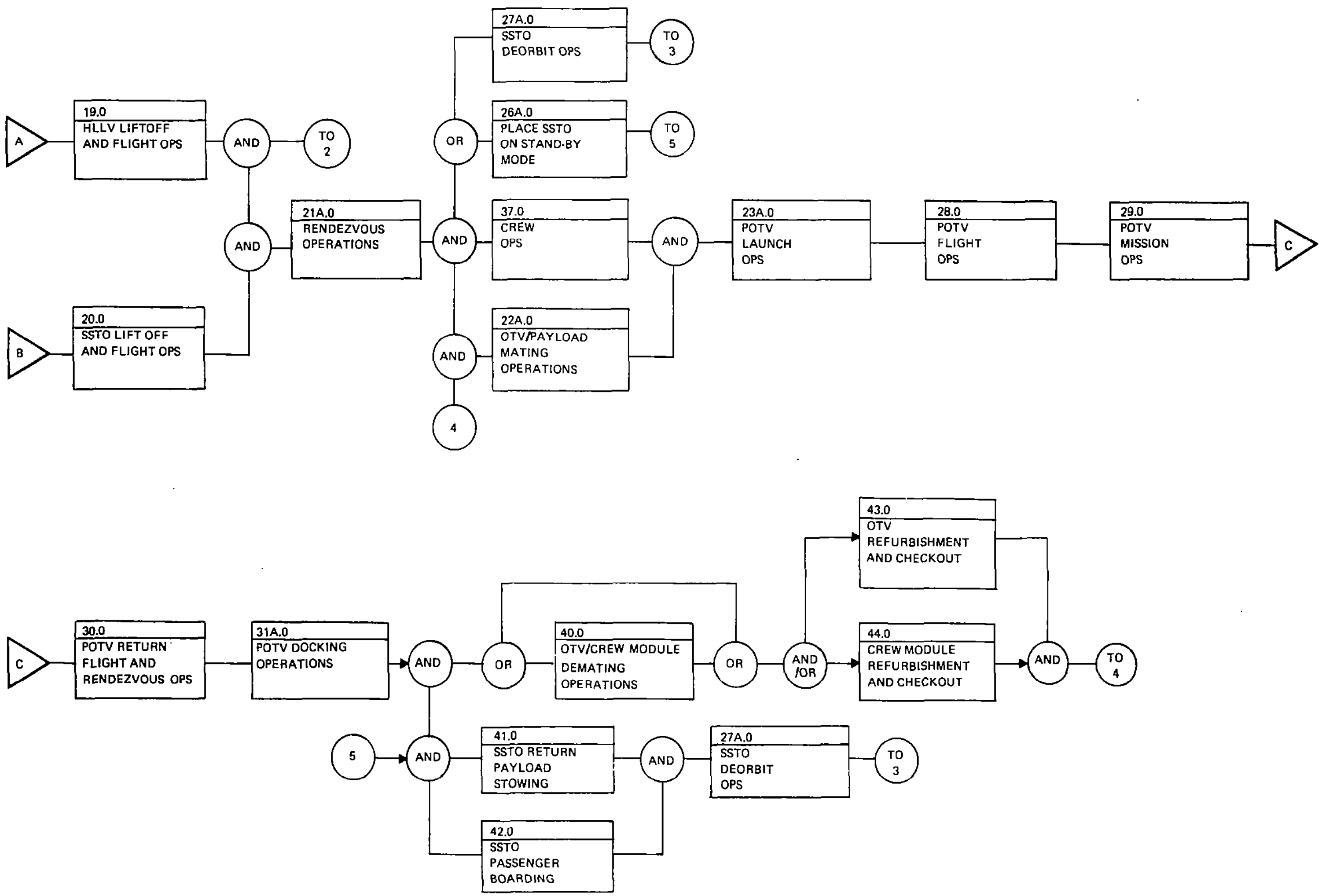


Figure 5.3-2.—Space Based POTV Operations Top-Level Functional Flow (Continued)

Table 5.3-2.—Ground-Based vs Space-Based OTV Operations—Top Level Functions Deltas

GROUND BASED		SPACE BASED	
1.0	HLLV Orbiter Landing Ops	1A.0	HLLV Tanker Landing Ops
2.0	HLLV Orbiter Post-Landing Ops	2A.0	HLLV Tanker Post-Landing Ops (see 43.0)
3.0	OTV Refurbishment and Checkouts		
4.0	HLLV Orbiter Refurbishment and Checkout	4A.0	HLLV Tanker Refurbishment and Checkout
9.0	Orbiter/Booster Mate and Checkout	9A.0	Tanker/Booster Mate and Checkout
21.0	Rendezvuos Operations	21A.0	Rendezvous and Docking Operations
22.0	OTV/Payload Mating Operations	22A.0	OTV/Payload Mating Operations
23.0	POTV Launch Operations	23A.0	POTV Launch Operations
25.0	HLLV Orbiter De-orbit Operation	25A.0	Tanker De-orbit Operations
26.0	Place SSTO on Stand-by Mode	26A.0	Place SSTO on Stand-by Mode
27.0	SSTO De-orbit Operations	27A.0	SSTO De-orbit Operations
31.0	POTV Docking Operations	31A.0	POTV Docking Operations
		37.0	Propellant Transfer/Storage Operations
		38.0	POTV Fueling Operations
		39.0	Crew Operations
		40.0	OTV/Crew Module Demating Operations
		41.0	SSTO Return Payload Stowing
		42.0	SSTO Passenber Boarding
		43.0	OTV Refurbishments and Checkout
		44.0	Crew module Refurbishment and Checkout

NOTE: Items Circled were selected for second-level functional analysis.

5.4 SECOND-LEVEL OPERATIONAL REQUIREMENTS ANALYSIS

The selected top-level functional blocks were developed to the second level in order to derive operational requirements. To create the second-level functional flows, it was necessary to select some specific operational and/or hardware options as models for the analysis. The report on each of the selected operations begins with a "trade tree" that shows the various alternative approaches that were considered. The subjective rationale for the selection of the options is described.

After the second-level functional flows are shown, a list of the operational requirements for the function is given.

5.4.1 Rendezvous Operations (Blocks 21.0 and 21A.0)

5.4.1.1 Systems and Operations Options

The systems and operations for this function are shown in figure 5.4-1.

5.4.1.1.1 Ground-Based Scenario

The two alternative operational approaches for the rendezvous were 1) to eject the OTV from the HLLV Orbiter and then have the SSTO rendezvous with the OTV, or 2) to have the OTV remain attached to the HLVV Orbiter and then have the SSTO rendezvous with the orbiter. The first option was selected as it was judged to be preferable to dock to the small OTV rather than the large HLLV.

5.4.1.1.2 Space-Based Scenario

For the HLLV Tanker, the rendezvous options were 1) to formation fly with the space base (no hard-docking) or 2) to rendezvous and dock to the space base. The second option was selected as vehicle - space base docking will be required for other vehicles. However, the first option (formation flying) is worthy of serious consideration (this operation would be similar to aerial refueling or at-sea refueling of ships).

Another set of options for the HLLV Tanker were whether 1) propellant pallets, or 2) propellant pumping should be used. The second option was selected as the HLLV Tanker propellant capacity will probably not be an even increment of the OTV propellant capacity.

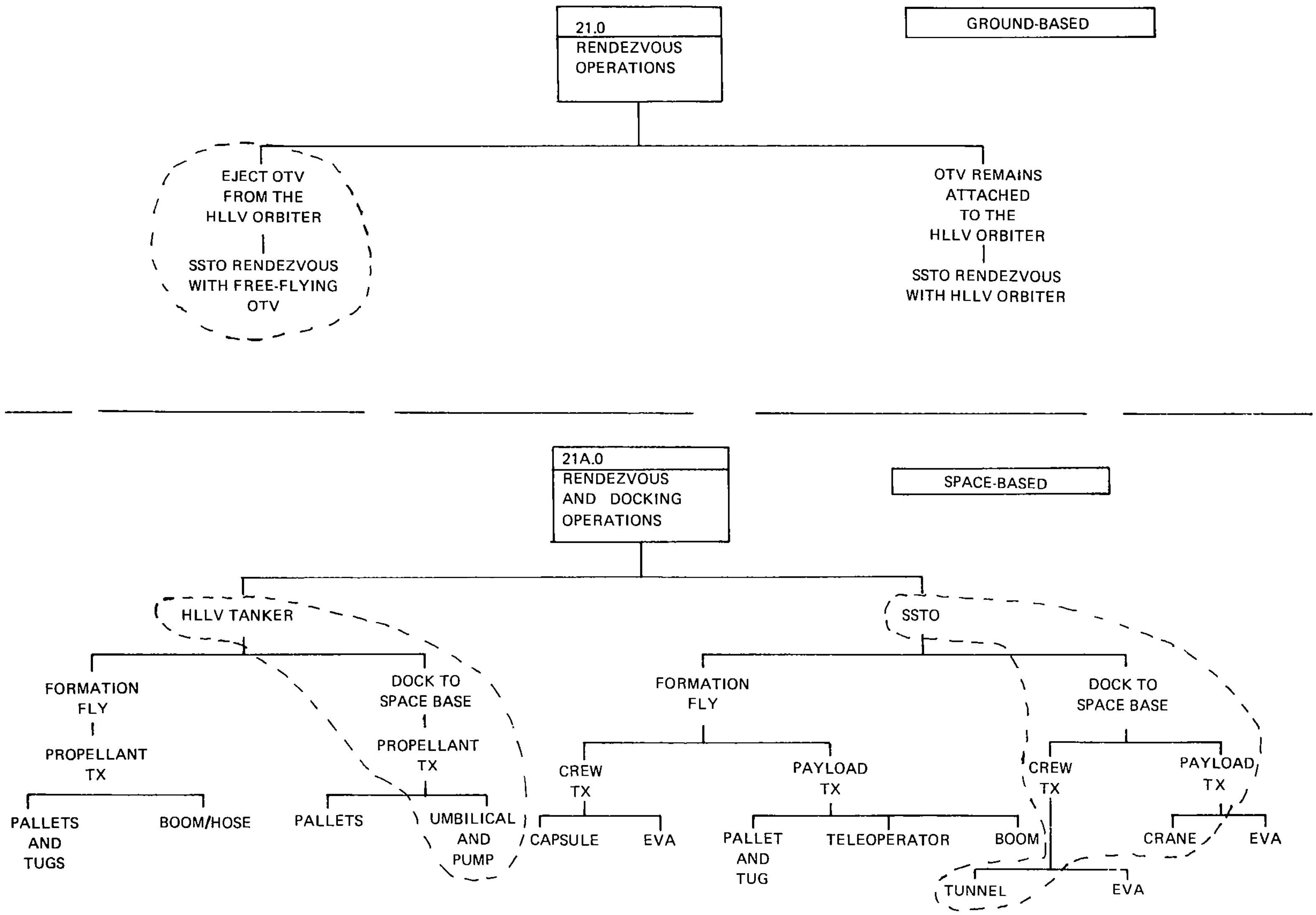


Figure 5.4-1.—Rendezvous Operations "Trade Tree"

For the SSTO, the rendezvous and docking options were the same as for the tanker. In this case, hard docking to the space base was selected as the mechanics of transferring the SSTO payload to the base would be more straight forward.

It was elected to use crew transfer tunnels rather than extra vehicular activity (EVA) to move the SSTO passengers to the space base crew habitat as the tunnel system would allow the crew to be transferred immediately after vehicle docking, EVA would require several hours of prebreathing.

The SSTO payload transfer will be addressed in section 5.4.2.1.2.

5.4.1.2 Second-Level Functional Flow

The second-level functional flow diagrams for the rendezvous operations are shown in figure 5.4-2.

5.4.1.3 Operational Requirements

Table 5.4-1 lists the operational requirements placed on the various vehicles and the Space Base that result from the second-level functional flow analysis.

5.4.2 OTV/Payload Mating Operations (Block 22.0 and 22A.0)

5.4.2.1 Systems and Operations Options

The systems and operations options for this function are shown in figure 5.4-3.

5.4.2.1.1 Ground Based

The two vehicle mating options were 1) to capture the OTV and then dock it directly to the payload, or 2) to capture the OTV, dock it to the SSTO and then mate the OTV to the payload. The first option was selected because of its relative simplicity.

5.4.2.1.2 Space Based

The vehicle mating options were 1) horizonbtal mating or 2) vertical mating. The horizontal mating option was selected because this would be more compatible with the crew module handling equipment that would be used for removing the crew module from the SSTO cargo bay. The vertical mating option should not be discarded without a more detailed trade study.

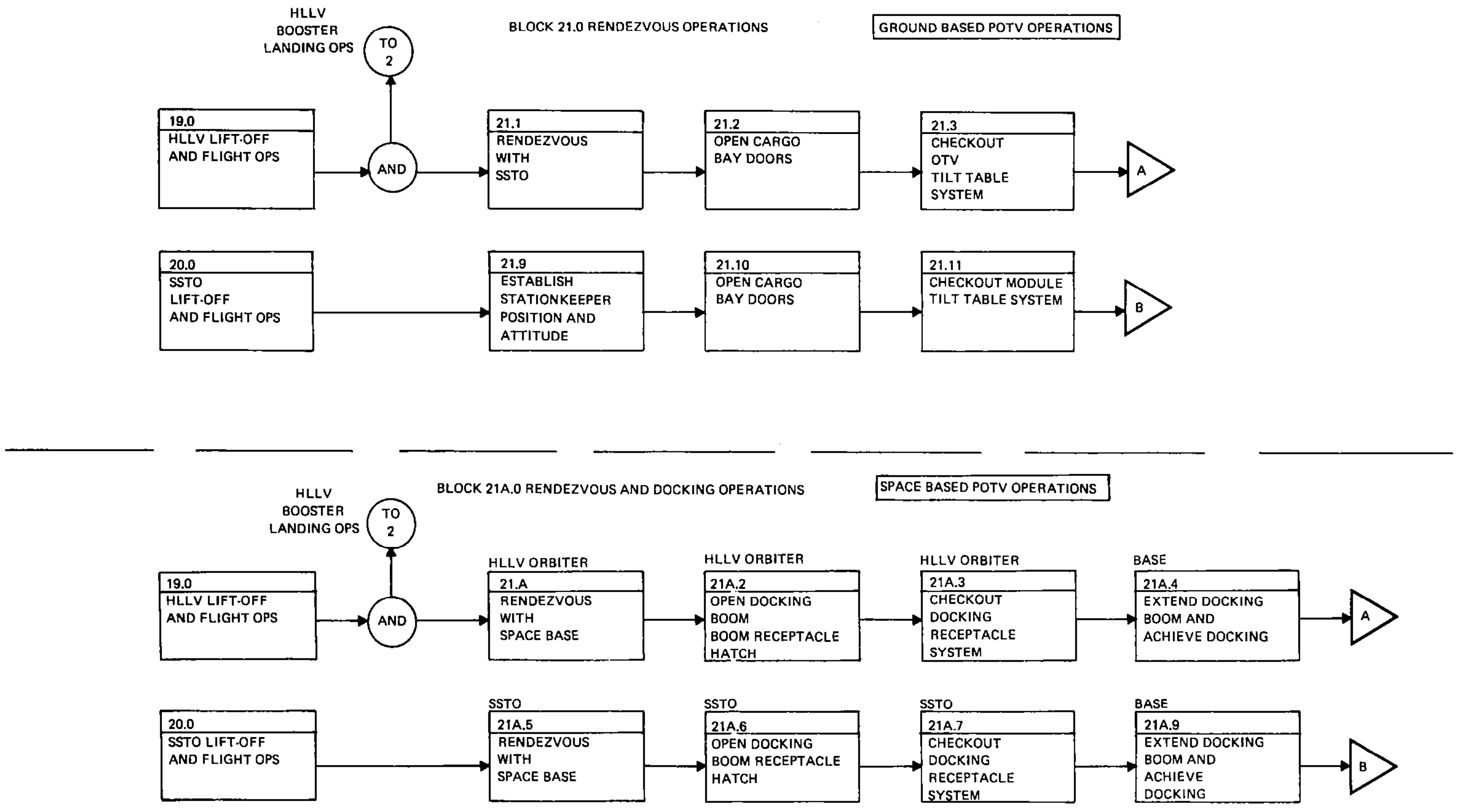


Figure 5.4-2.—Rendezvous Operations Second-Level Functional Flow

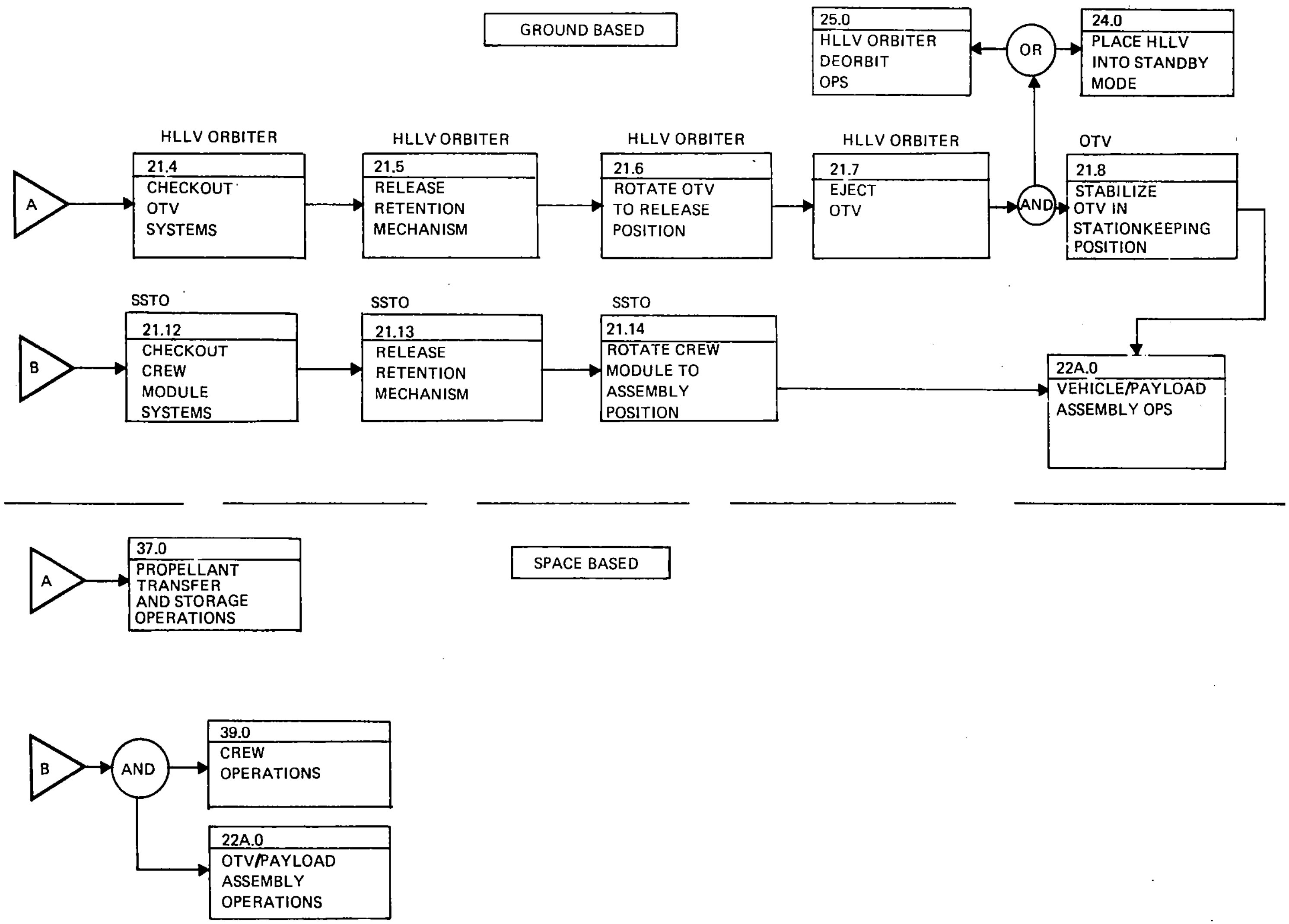


Figure 5.4-2.—Rendezvous Operations Second-Level Functional Flow (Continued)

Table 5.4-1.—Operational Requirements/Summary: Rendezvous and Docking Operations

VEHICLE \ OPERATIONAL BLOCK		21.0 GROUND BASED	21A.0 SPACE BASED

HLLV	ORBITER	. Data link to SSTO and Earth
		. Stationkeeping capability
		. Remote control by data link of
		  . POTV tilt table/ejection mechanism
		  . Payload retention system
		  . Cargo bay doors
		  . HLLV attitude
			N/A

	TANKER	N/A	. Rendezvous and stationkeeping with space base. . Space base compatible docking provisions . Data link to space base/Earth . Remote control of attitude during docking. . Propellant transfer umbilical receptacle

SSTO		. Rendezvous & stationkeeping with HLLV orbiter
		. Data link to HLLV and POTV
		. Command by data link
		  - HLLV & POTV systems
		  - POTV & orbiter tilt table ejection mechanism checkout
		  - POTV ejection
		  - HLLV preparation for de-orbit (with ground)
		  - POTV stationkeeping
		. Crew displays for monitoring above operations
			. Rendezvous and stationkeeping with space base
			. Space base compatible docking provisions
			. Data/voice link to space base
			. Remote control by space base of attitude during docking.
			. Crew displays for rendezvous and docking monitoring.
			. Hatch compatible with mating to space base crew transfer tunnel for crew transfer to space base.

(Continued)

Table 5.4-1.—Operational Requirements/Summary: Rendezvous and Docking Operations (Continued)

VEHICLE \ OPERATIONAL BLOCK	21.0 GROUND BASED	21A.0 SPACE BASED
POTV	. Rendezvous and dock with SSTO for crew module exchange - During payload mating - End of mission . Rendezvous and dock with HLLV for return to Earth . Remote control from SSTO of systems, HLLV rendezvous and docking activities, attitude, etc. . Data link to SSTO/Earth . Automatic interface with HLLV. Tilt table and payload retention system (active) . Autonomous attitude hold and stationkeeping capability.	. Rendezvous and dock with space base - end of mission. . Passive space base docking interface . Remote control by space base of terminal dock . Data/voice link to space base/Earth. . Autonomous attitude hold & stationkeeping capability. . Passive berthing interface (independent of forward payload interface) for parking at space base.
SPACE BASE	N/A	. Data and voice link to SSTO, POTV. . Data link to HLLV tanker. . Command HLLV tanker and OTV - Systems - Rendezvous & stationkeeping - Attitude during docking (also SSTO) . Mount provisions for terminal docking of POTV, SSTO, HLLV (docking boom system) . Monitor above functions - Radar - Crew Displays - Computation

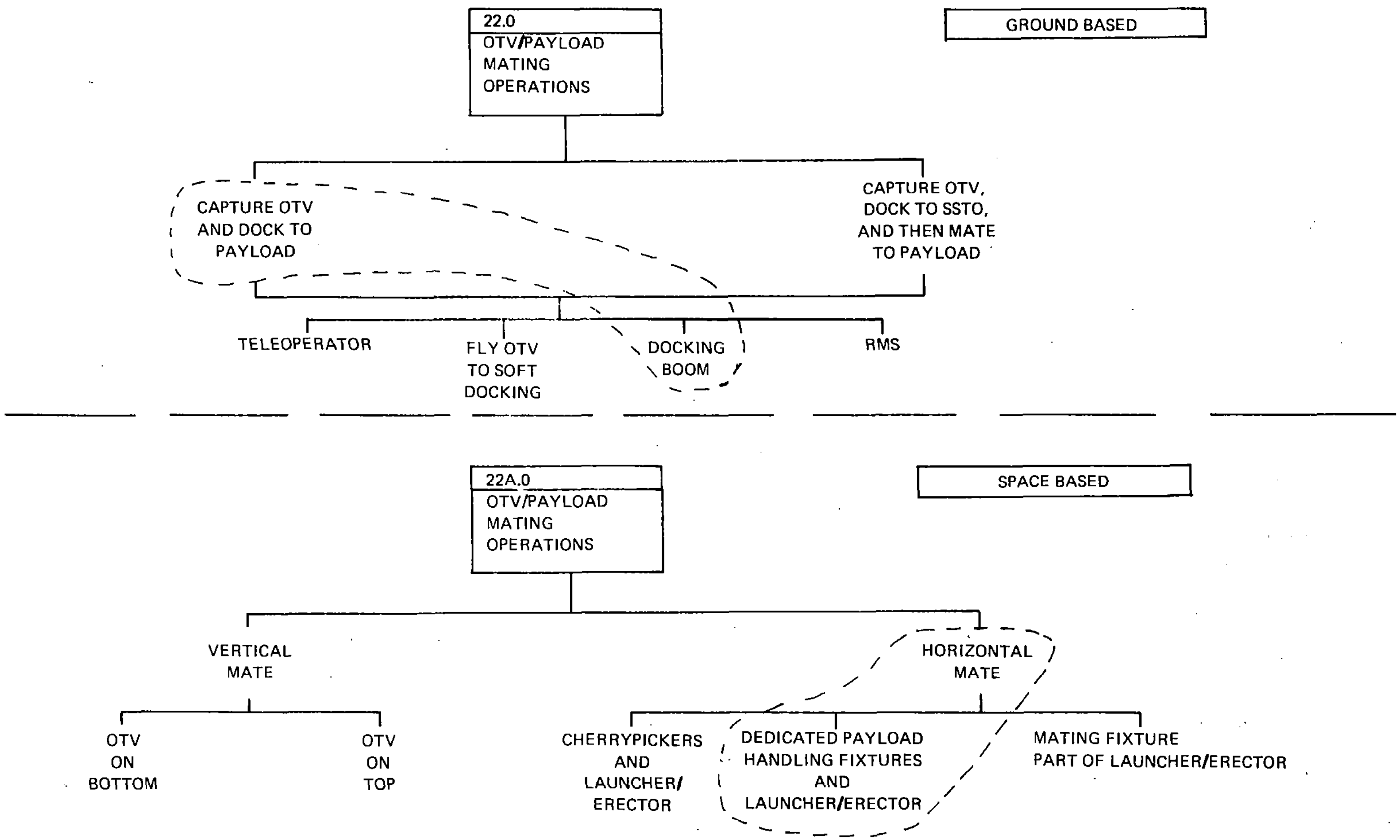


Figure 5.4-3.—OTV/Payload Mating Operations "Trade Tree"

Having selected a horizontal mating mode, the sub-options were 1) to use cherrypickers to handle the payload during the mating operations, 2) to use a special purpose handling equipment to remove those payload from the SSTO and to move it to the OTV where the mating operation is conducted, or 3) to have the payload handling apparatus be a portion of the launcher/ejector system. Option 2 was selected. Option 1 was deleted because of the risk involved in trying to extricate a payload from the SSTO cargo bay and to achieve the vehicle mating using one or more cherrypickers. Option 3 was deleted due to its apparent complexities.

5.4.2.2 Second-Level Functional Flows

The second-level functional flows for the OTV/Payload Mating Operations is shown in figure 5.4-4.

5.4.2.3 Operational Requirements

Table 5.4-2 lists the operational requirements levied on the various vehicles and the space base that results from the second-level functional flow analysis.

5.4.3 POTV Launch Operations (Blocks 3.0 and 23A.0)

5.4.3.1 Systems and Operations Options

The systems and operations options for this function are shown in figure 5.4-5.

5.4.3.1.1 Ground Based Scenario

The launch mode options are 1) a fly-away mode wherein the OTV engines are ignited and the vehicle is self-propelled away from the SSTO, 2) a launching boom is used to transport the OTV away from the SSTO before the OTV engines are started and 3) a cast-away launch wherein the OTV is propelled away from the SSTO via an ejection system and the OTV engines are started after the vehicle is well-clear of the SSTO. Obviously, the first option is not feasible due to the hazard presented to the SSTO. The second option is eliminated due to the additional single purpose equipment that would have to be taken up by the SSTO. The third option was selected as the launcher and is also used as the tilt-table and OTV transport interface. This technique is also that selected for the IUS system.

5.4.3.1.2 Space Based Scenario

The vehicle launch orientation options are 1) horizontal or 2) vertical. The vertical launch was selected. However, the horizontal launch should not be disregarded. The cast-away

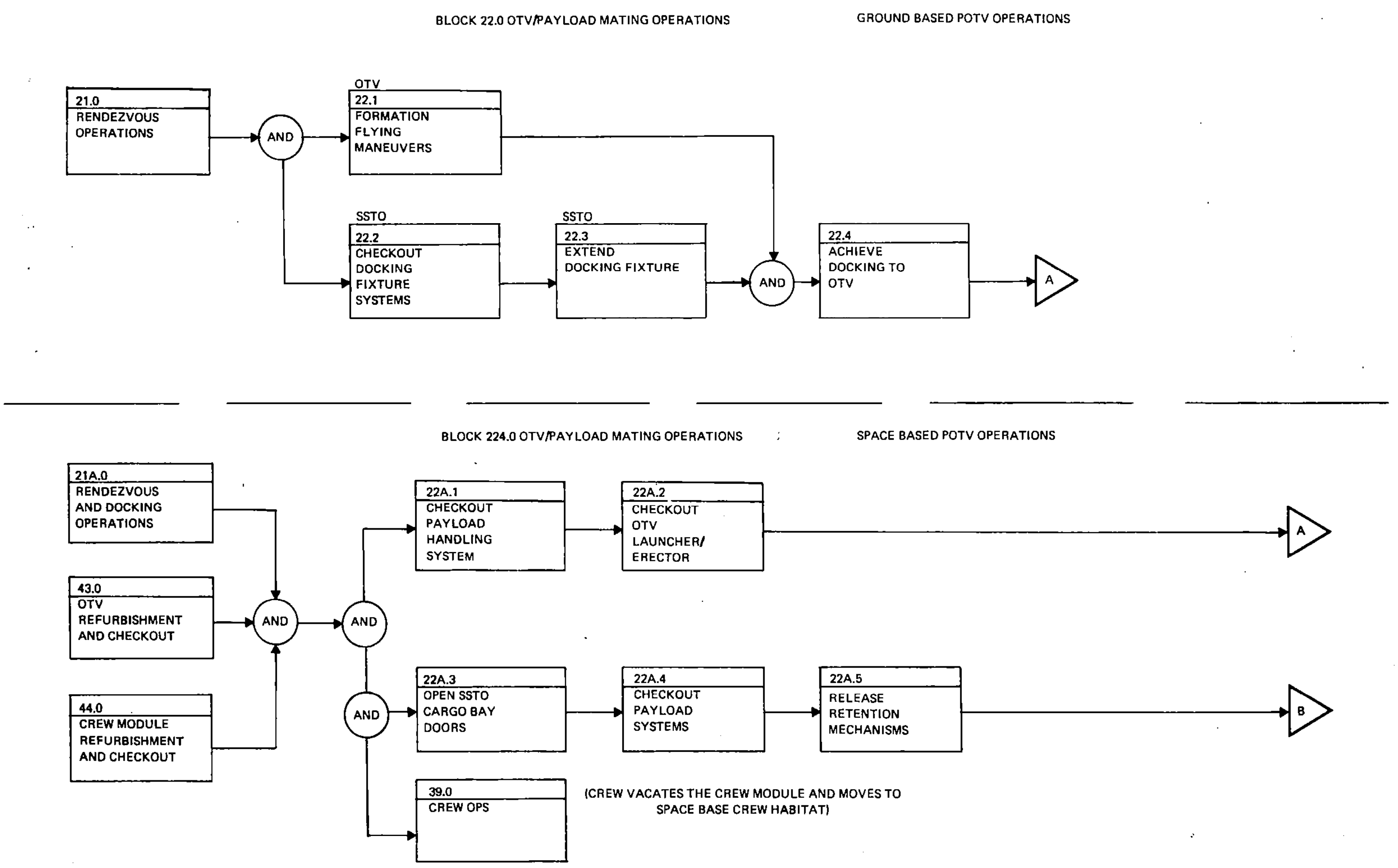


Figure 5.4-4.—OTV/Payload Mating Second-Level Functional Flow

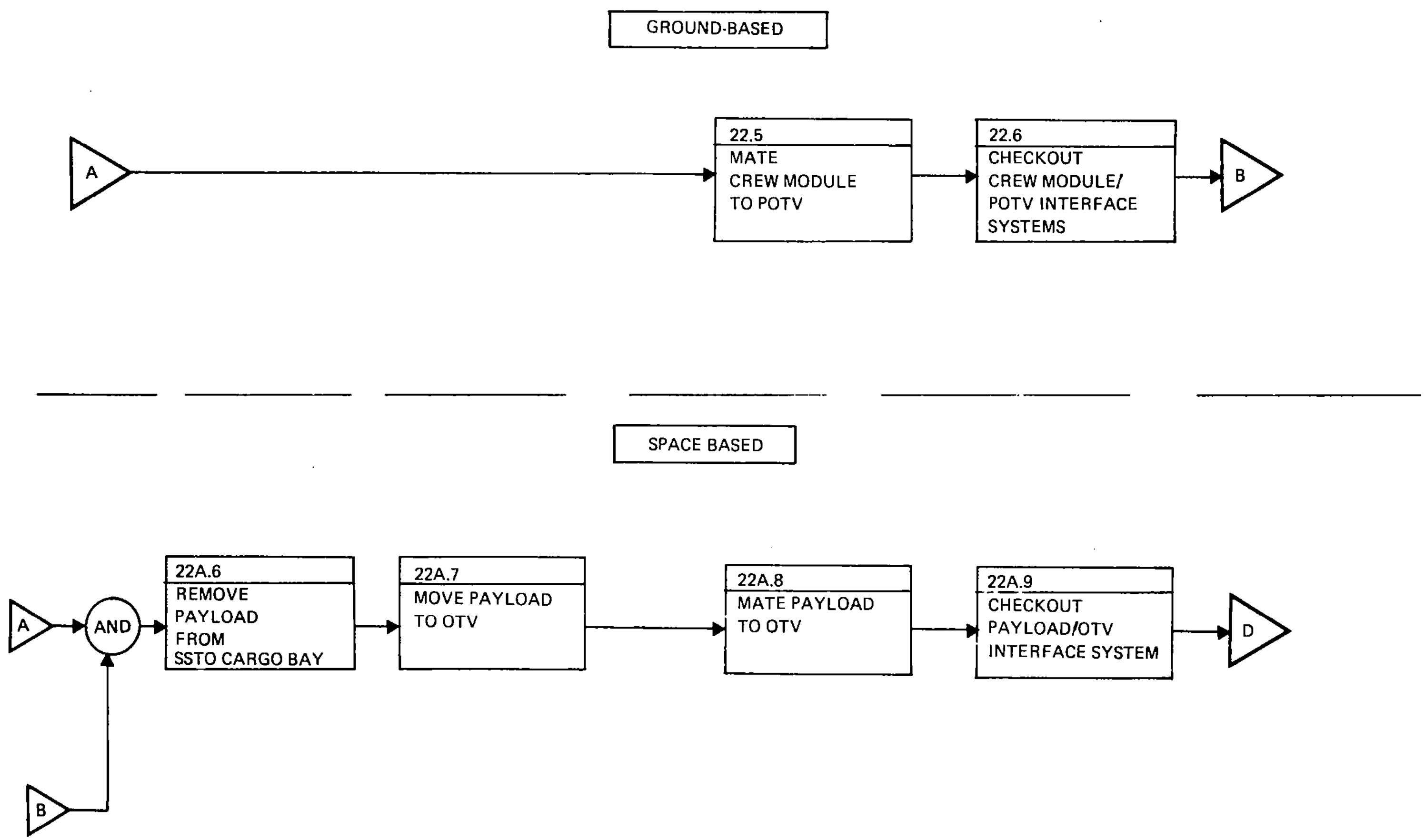


Figure 5.4-4.—OTV/Payload Mating Second-Level Functional Flow (Continued)

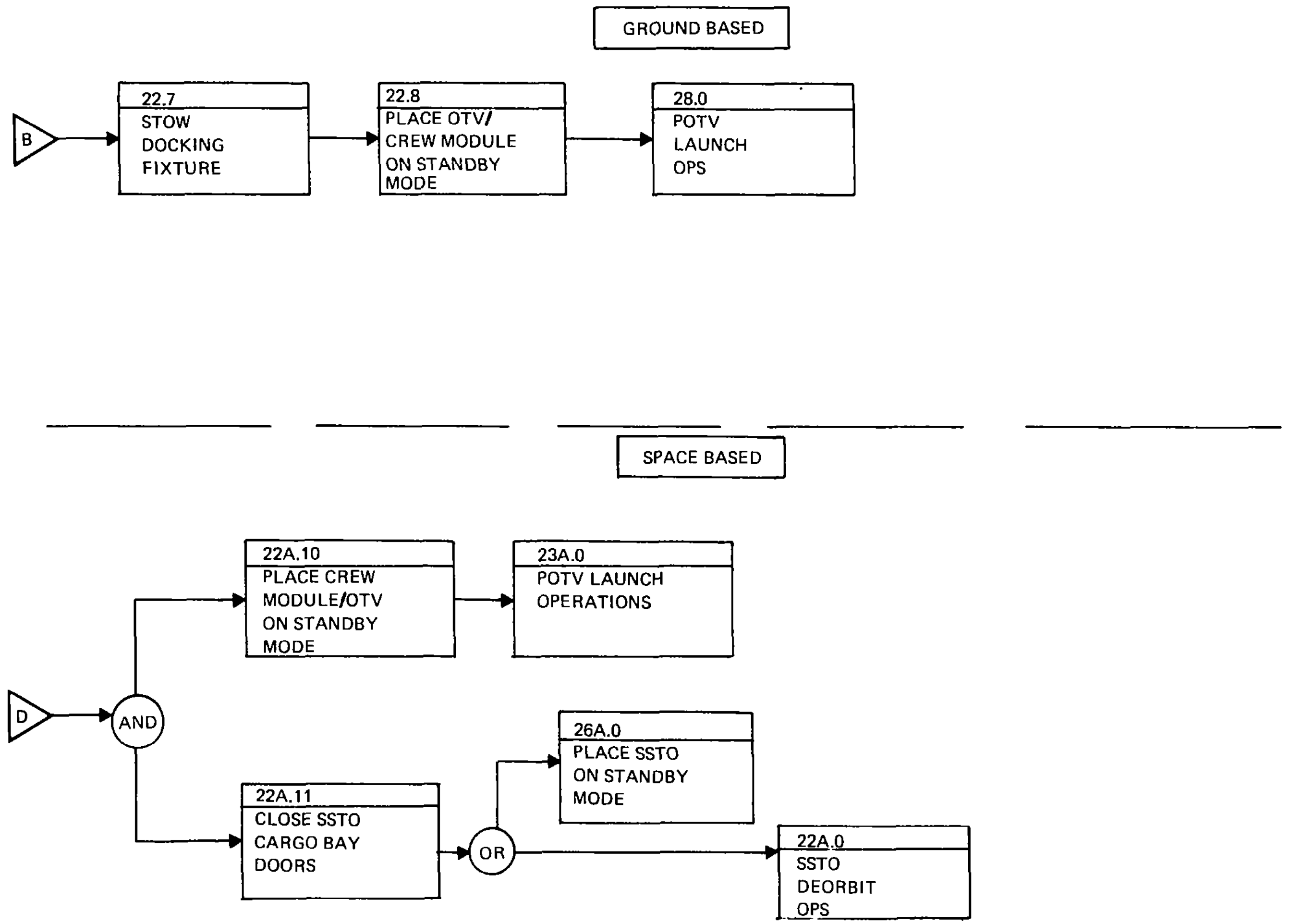


Figure 5.4-4.—OTV/Payload Mating Second-Level Functional Flow (Continued)

Table 5.4-2.—Operational Requirement Summary: OTV/Payload Mating Operations

VEHICLE \ OPERATIONAL BLOCK	22.0 GROUND BASED	22A.0 SPACE BASED
SSTO	o Active Payload (Manned Module) Retention System. For remote unlatching of manned module after OTV mating. o Crew Module/OTV Interface Monitoring Capability o Payload Retention System Monitoring Capability	o Active Payload Retention System o Payload Retention System Monitoring Capability
CREW MODULE	o Passive docking interface (aft end) for OTV docking o Passive berthing interface (aft end) for hard attachment of OTV. o Active docking interface and personnel port (forward end) for docking to SSTO at end of mission. o Fittings for SSTO payload retention system pickup. o SSTO and POTV electrical umbilical connectors (remote mating).	o Passive berthing interface (aft end) for attachment to OTV. o Active docking interface and personnel port (forward end) for contingency docking (end of mission) o Fittings for SSTO payload retention system pickup o Payload handling system interface. o SSTO and POTV electrical umbilical connectors (remote mating).
OTV	o Automatic active forward berthing interface for payload hard attachment (hard dock). o Payload electrical umbilical (remote mating)	o Remote controlled active forward berthing interface for payload hard attachment. o Payload electrical umbilical (remote mating)
SPACE BASE	N/A	o Payload Handling System -Removes payload from SSTO cargo bay -Transfers and aligns payload to OTV forward berthing interface -Activates remote OTV berthing latches and umbilical -Reverses functions at end of mission o Payload handling systems monitor capability -CCTV/Windows -Displays -Controls

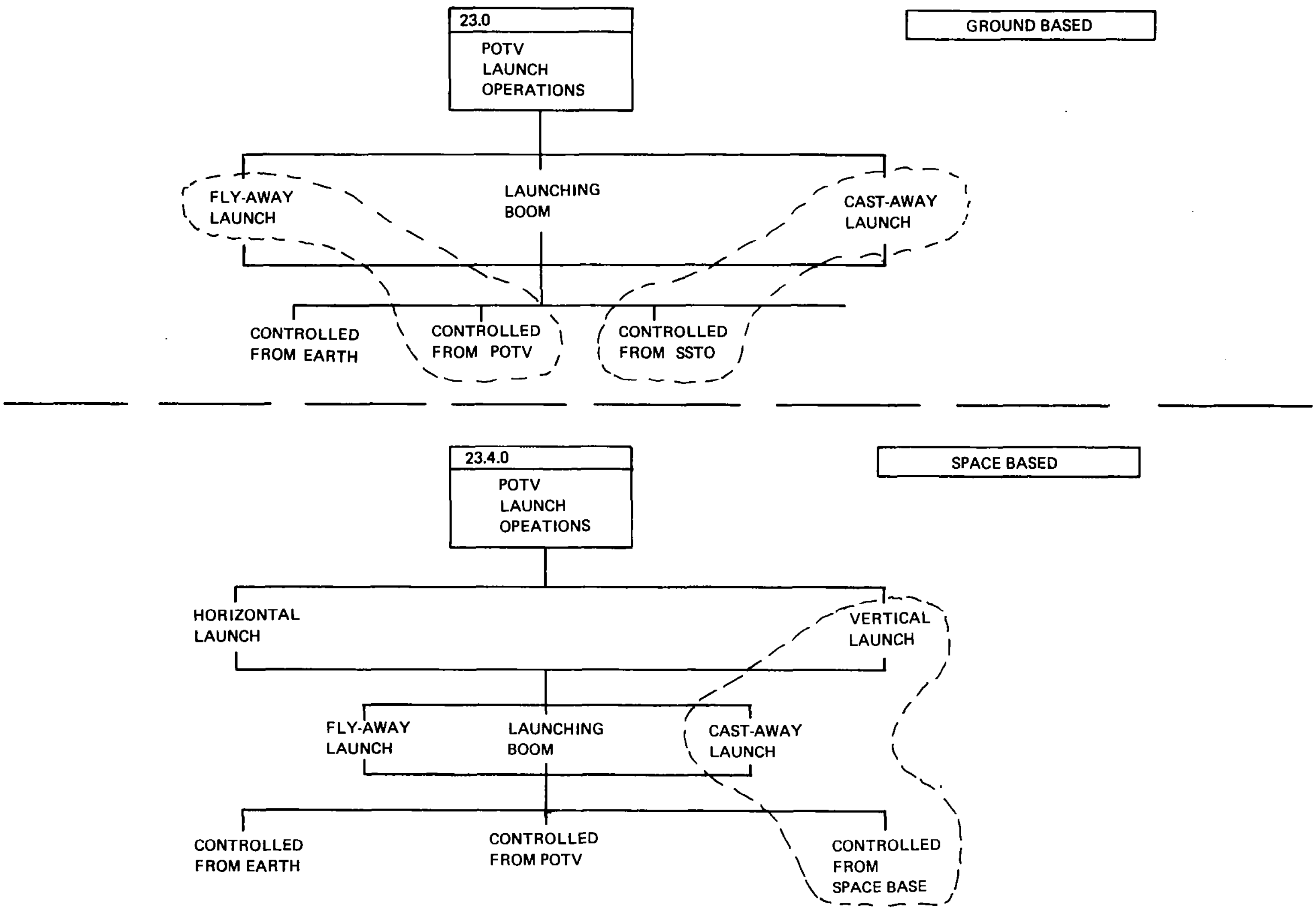


Figure 5.4-5.—POTV Launch Operations "Trade Tree"

launch was selected for the same reasons that this mode was chosen for the ground-based scenario.

5.4.3.2 Second-Level Functional Flow

The second-level functional flows for the POTV Launch Operations are shown in figure 5.4-6.

5.4.3.3 Operational Requirements

Table 5.4-3 lists the operational requirements levied on the various vehicles and the Space Base that results from the second-level functional flow analysis.

5.4.4 POTV Docking Operations (Block 31.0 and 31A.0)

5.4.4.1 Systems and Operations Options

The systems and operations options for this function are shown in figure 5.4-7.

5.4.4.1.1 Ground-Based Scenario

The docking boom system was selected as the preferred concept by a matter of elimination. The teleoperator system creates another free-flying vehicle and fly-to-docking operation that is riskier and more complicated. The "fly OTV to soft docking" mode is too risky. The RMS would not be husky enough to maneuver the vehicle. It can be designed to be stiffer and act as a docking boom.

5.4.4.1.2 Space-Based Scenario

The same options are available here and the docking boom was again selected for the same reasons.

5.4.4.2 Second-Level Functional Flows

The second-level functional flows for the POTV Docking Operations are shown in figure 5.4-8.

5.4.4.3 Operational Requirement

The operational requirements are not significantly different than those described in table 5.4-1 for the Rendezvous Docking Function.

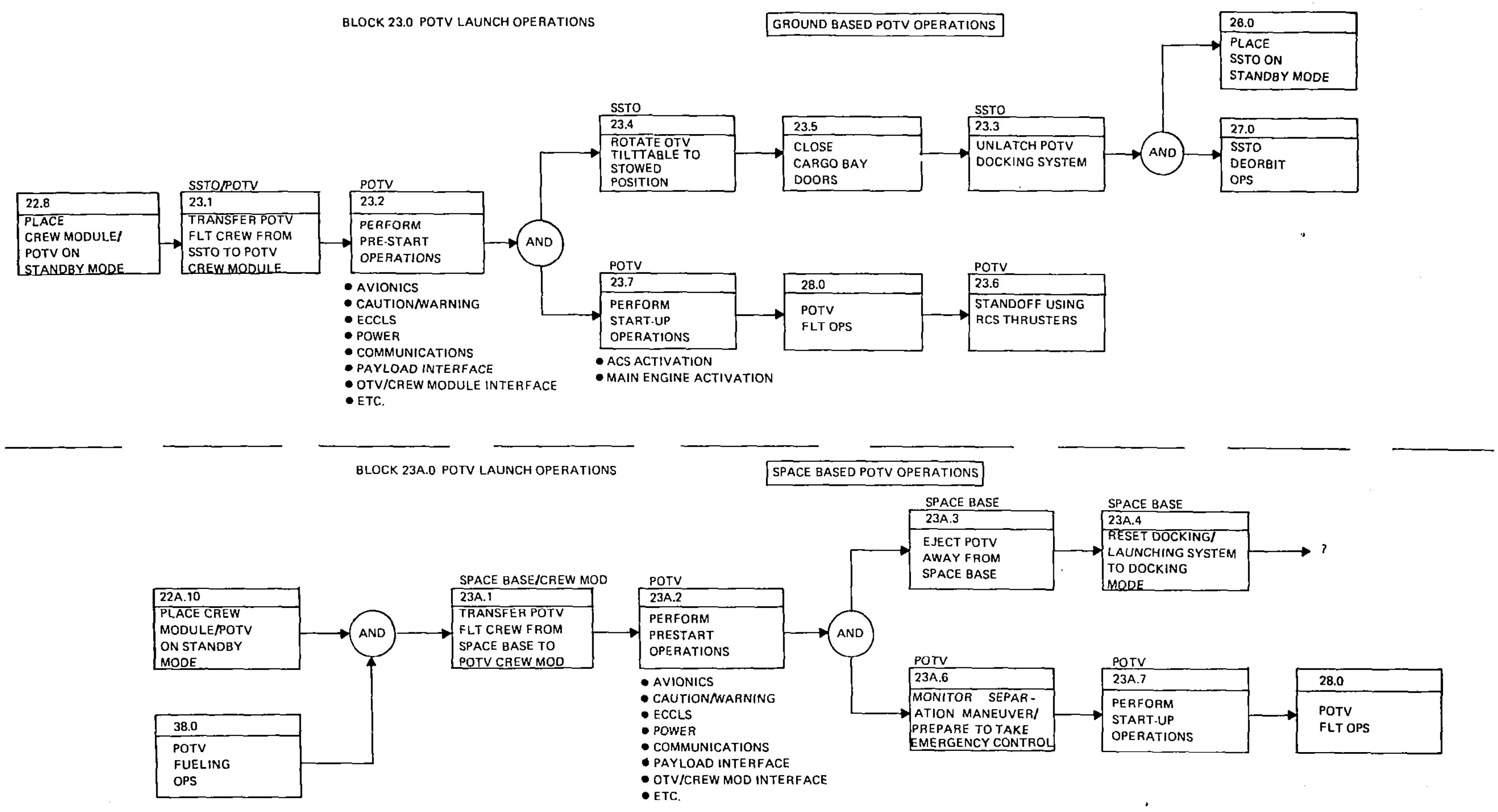


Figure 5.4-6.—POTV Launch Operations Second Level Functional Flow

Table 5.4-3.—Operational Requirement Summary: OTV Launch Operations

OPERATIONAL BLOCK / VEHICLE	23.0 GROUND BASED	23A.0 SPACE BASED
HLLV	o No specific HLLV launch provisions required - OTV stands off using RCS thrusters	N/A
SSTO	o No specific SSTO launch provisions required after undocking of crew module with OTV attached, OTV stands off from SSTO using RCS thrusters. (1m/sec ΔV) o Crew tunnel provides for crew transfer from SSTO cab into manned module	N/A
POTV	o No specific launch provisions req'd. except 1 m/sec ΔV. per launch for RCS system o Crew controls + monitors standoff using existing POTV controls and display o Crew module hatch compatible with remote mating to SSTO crew transfer tunnel	o OTV is passive with standoff function accomplished by docking/launching boom. o Crew monitors standoff only. o Crew module hatch compatible with remote mating to space base crew transfer tunnel
SPACE BASE	N/A	o Crew transfer tunnel capable of reaching and mating to crew module for crew transfer. o Remote disconnect of space base OTV/ umbilicals. o Docking/launching boom capable of standing off fully loaded OTV. o CCTV or windows for space base visual monitoring of launch. o Controls displays & computation as required for control and monitoring of launch and standoff operations

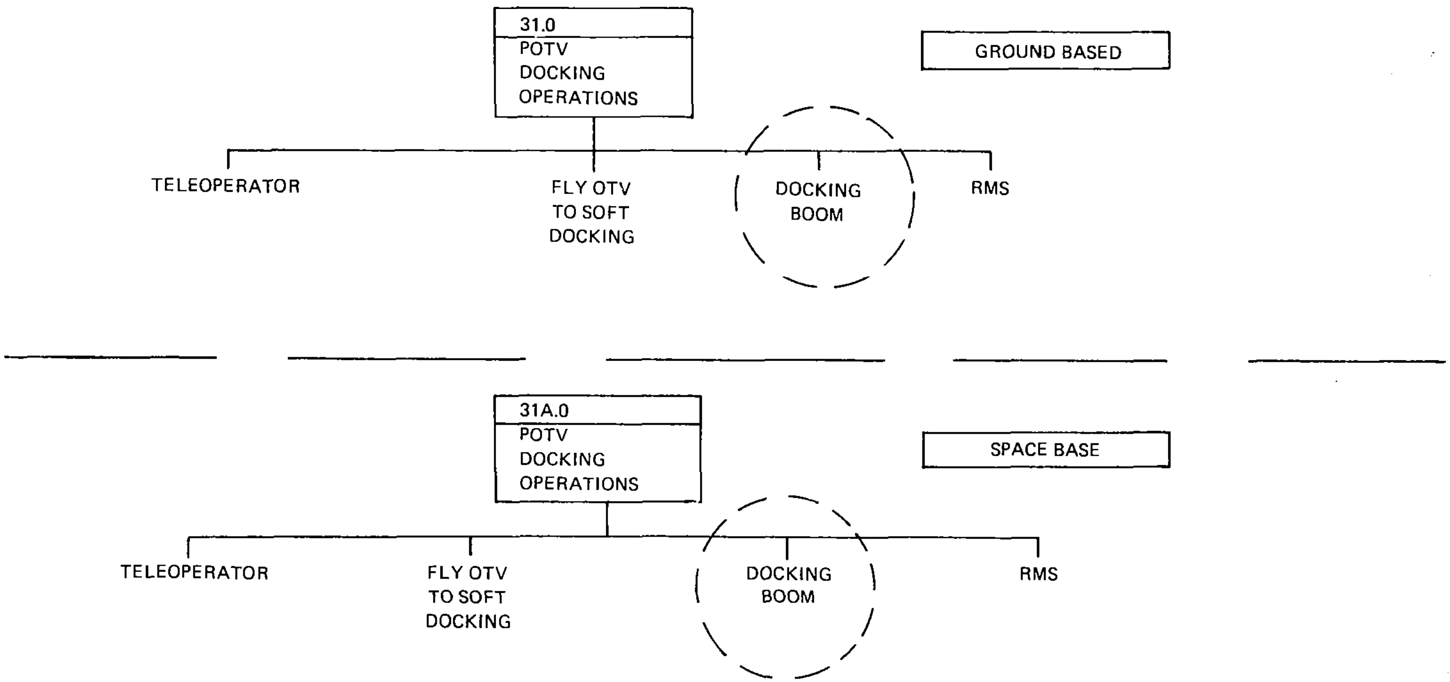


Figure 5.4-7.—POTV Docking Operations "Trade Tree"

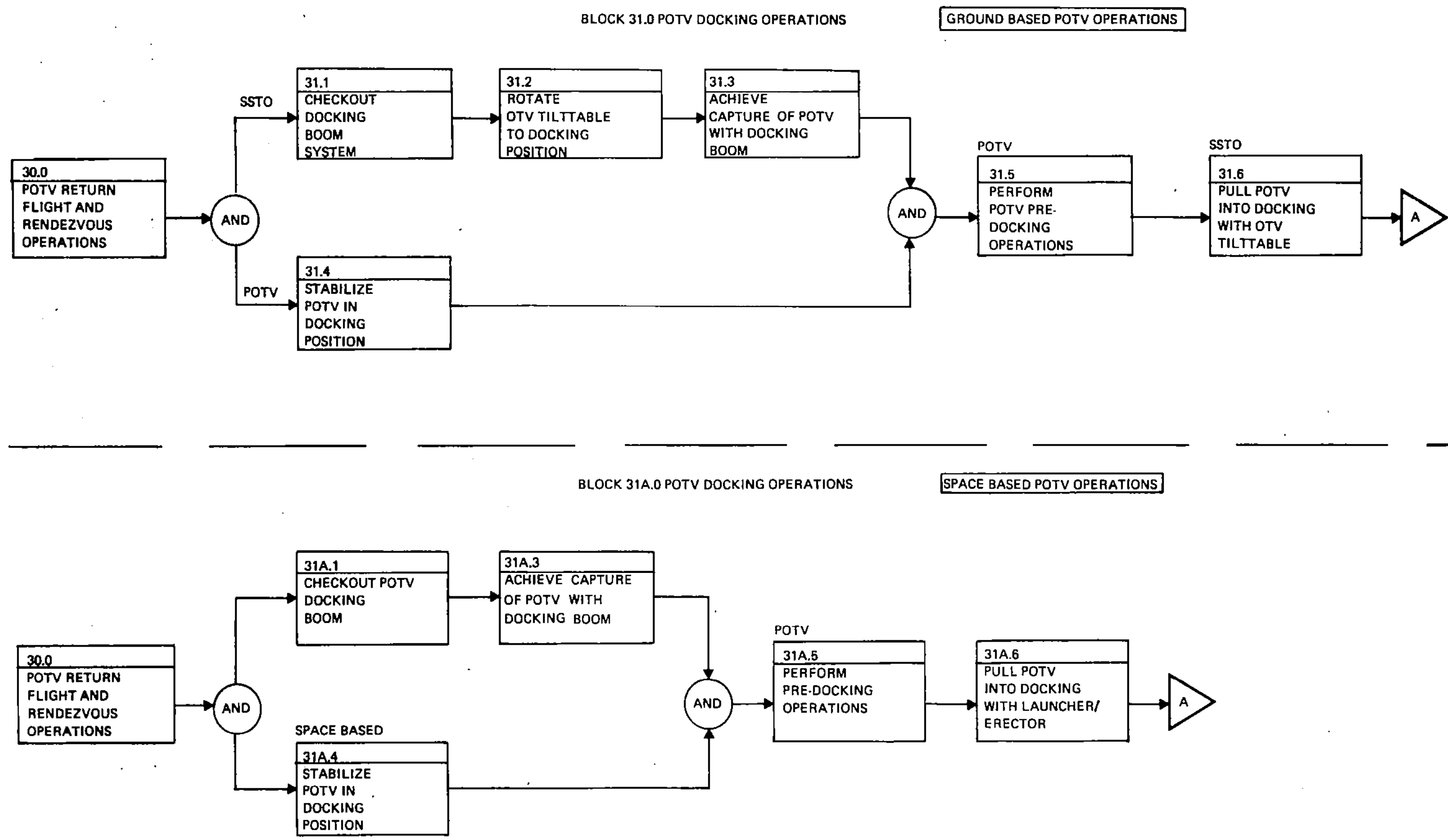


Figure 5.4-8. POTV Docking Operations Second Level Functional Flows

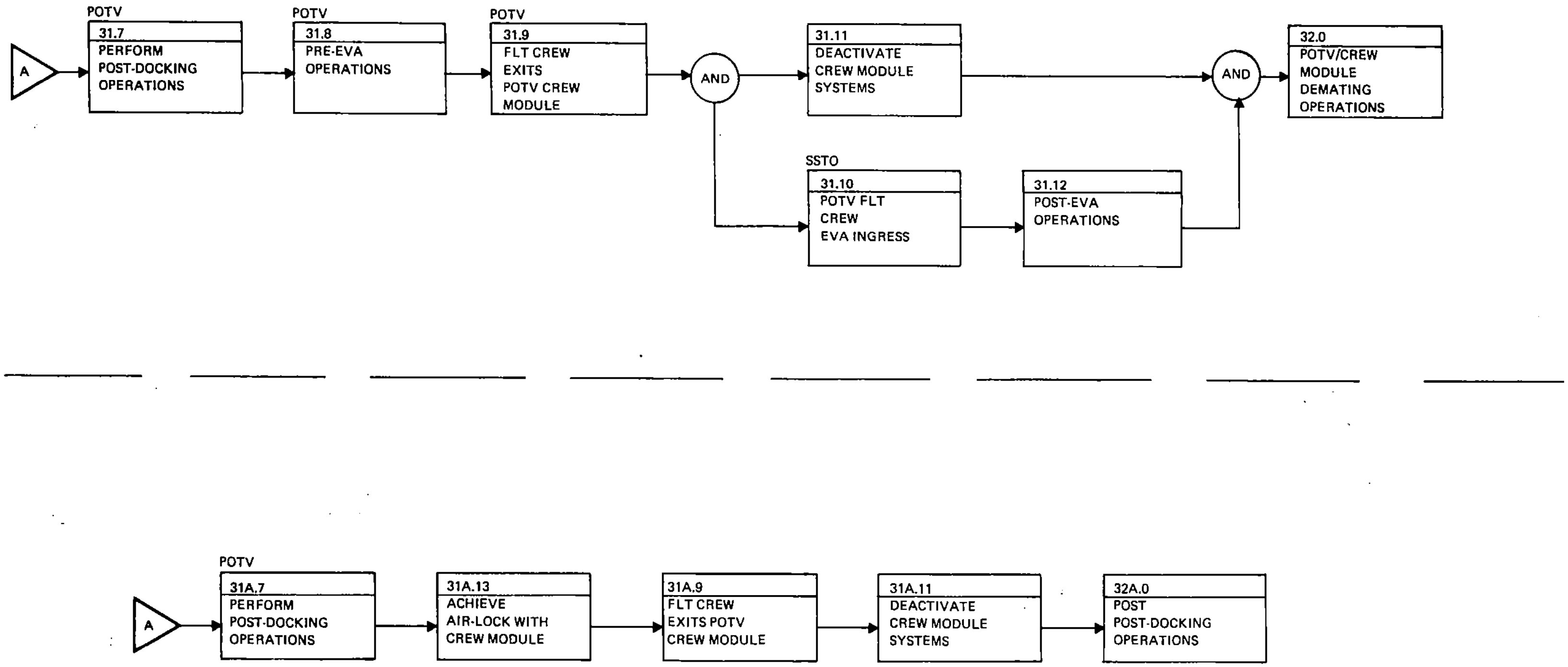


Figure 5.4-8.—POTV Docking Operations Second-Level Functional Flows (Continued)

5.4.5 OTV Refurbishment and Checkout (Blocks 3.0 and 43.0)

5.4.5.1 Systems and Operations Options

5.4.5.1.1 Ground Based Scenario

OTV refurbishment and checkout options were not considered. The space tug refurbishment and checkout systems and operations were used as a model.

5.4.5.1.2 Space Based Scenario

Figure 5.4-9 shows the various options that were considered. Returning the OTV to Earth would be a last-resort option. The rationale for the selection of the various suboptions shown in figure 5.4-9 will be addressed in section 5.4.5.3.

5.4.5.2 Second-Level Functional Flows

The second-level functional flows for the OTV Refurbishment and Checkout Operations are shown in figure 5.4-10, (these functional flows were adopted from the reference 25. Note that the ground-based and space-based functional flows are essentially identical. The notable differences are that in the space-based scenario 1) the flight data analysis and maintenance planning will be conducted on Earth and 2) at the end of the refurbishment operations there is no need to transport the OTV (Block 3.23 is deleted).

The real differences between ground-based maintenance and space-based maintenance show up at the third level where the means for implementing the various functional requirements will be substantially different. For instance, table 5.4-4 list the various maintenance equipment items that would be required for the ground and space-based OTV refurbishment operations. Note the many deltas.

5.4.5.3 Operational Requirements

No attempt was made to derive a complete list of operational requirements for this operational function. The maintenance operations are much more complex to attack at this time. However, some general observations of the operational requirements are given in table 5.4-5.

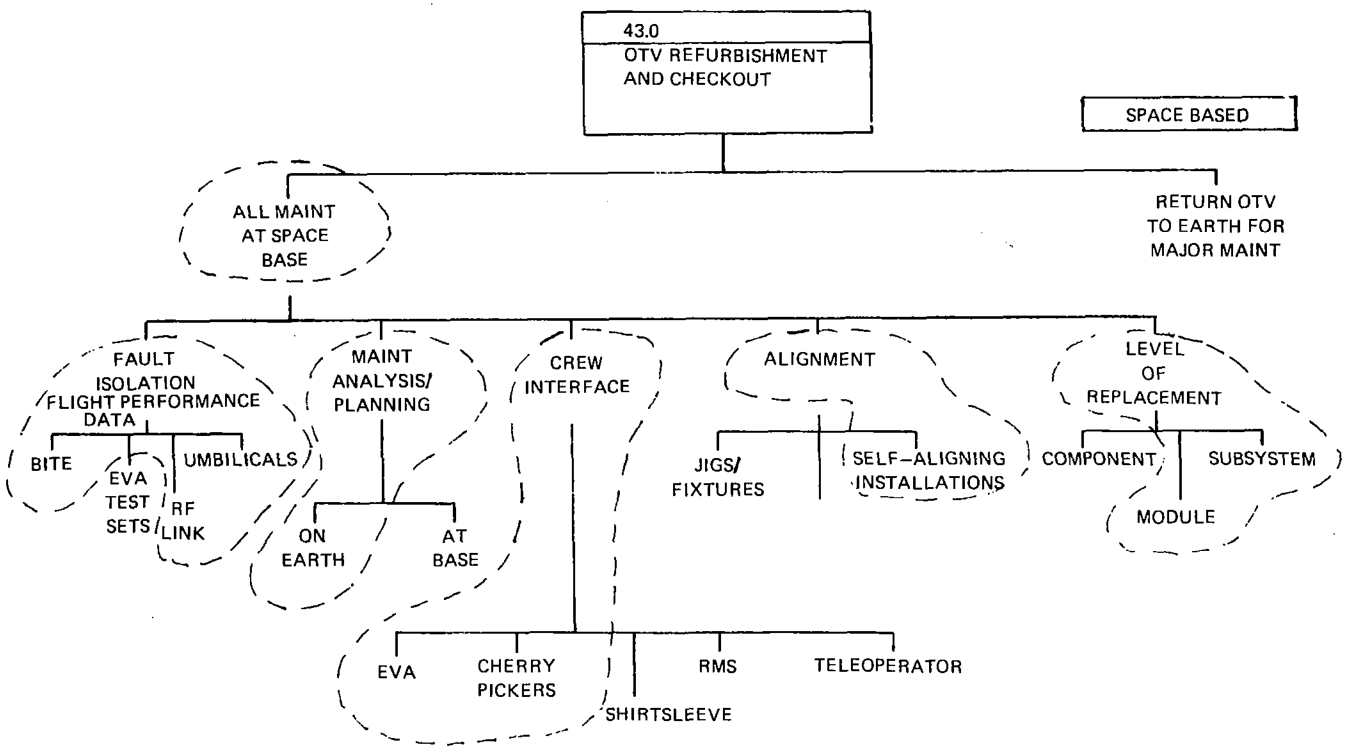


Figure 5.4-9.—OTV Refurbishment and Checkout "Trade Tree"

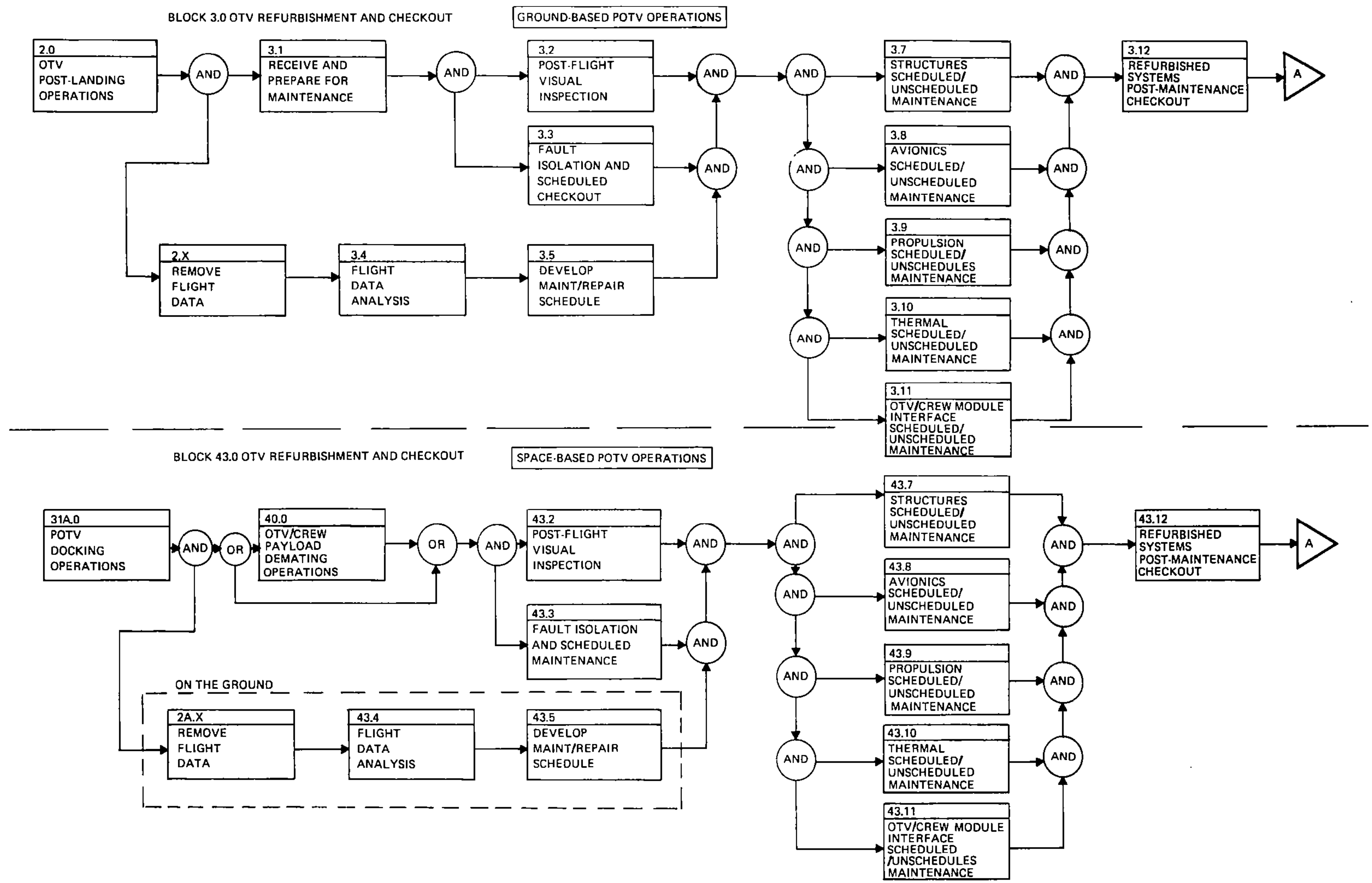


Figure 5.4-10.—OTV Refurbishment and Checkout Second-Level Functional Flow

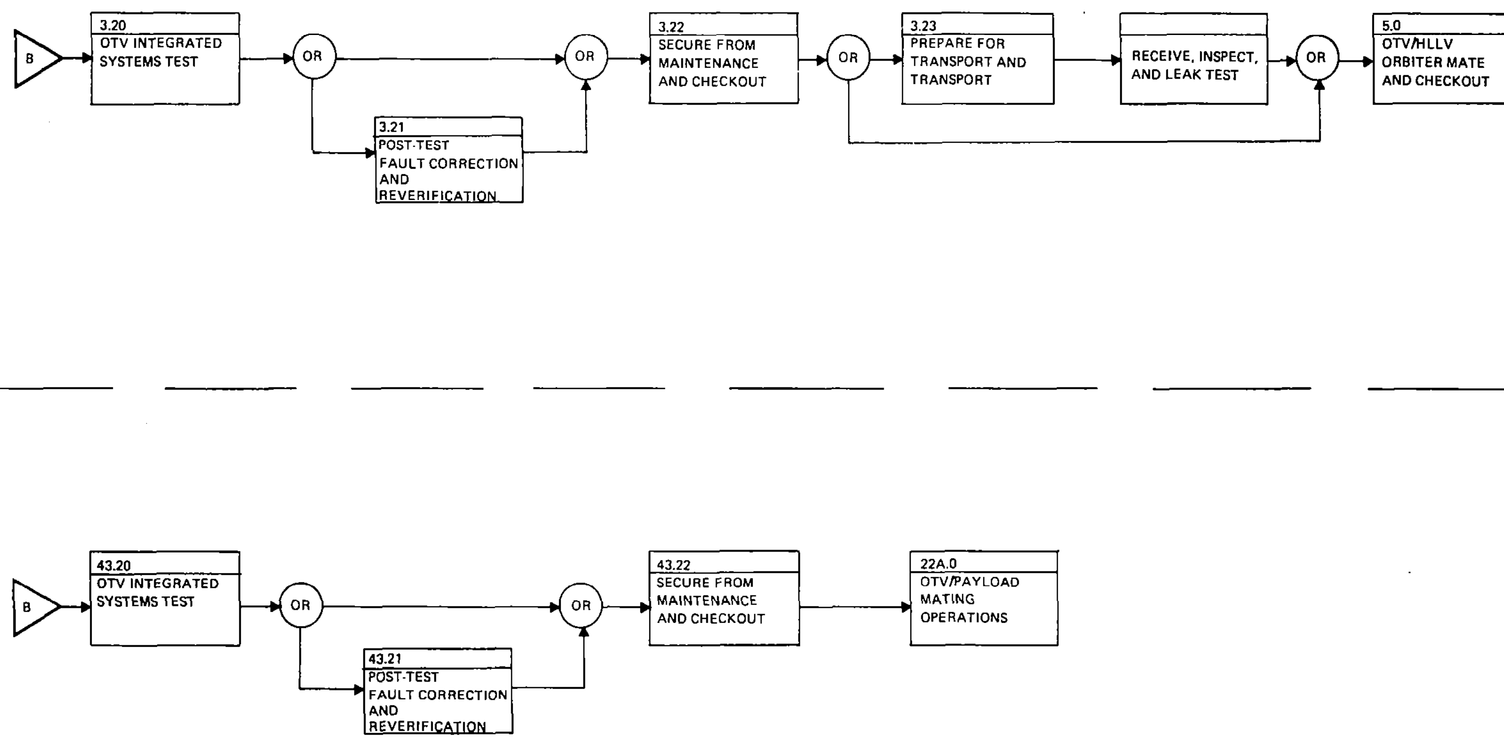


Figure 5.4-10.—OTV Refurbishment and Checkout Second-Level Functional Flow (Continued)

Table 5.4-4.—OTV Maintenance Support Equipment

GROUND BASED EQUIPMENT	SPACE BASED EQUIPMENT
OTV Workstand	OTV Docking Port
Engine Workstand	Engine System Pallet
Adapter Workstand	-
Mating Workstand	-
Cargo Bay Workstands	-
Inspection Workstand	-
Internal Platforms	-
OTV/Orbiter Adapter Workstand	-
Electrical Power Supply	Electrical Power Supply Umbilical
Electrical Power Test Set	Electrical Power Test Umbilical
Electrical Load Banks	Electrical Load Banks
Battery Test Set	BatteryTest Umbilical
Battery Charger	Battery Charger Umbilical
Communications Test & Checkout Equipment	Communications Test and Checkout Umbilical
Guidance & Navigation Test & Checkout Equipment	Guidance and Navigation Test & Checkout Umbilical
Control & Data Acquisition Console	Control & Data Acquisition Console
EMI Test Equipment	EMI Test Equipment
Memory Load & Verify Unit	Memory Load and Verify Unit

Table 5.4-4.—OTV Maintenance Support Equipment (Continued)

GROUND BASED EQUIPMENT	SPACE BASED EQUIPMENT
Electronics Calibration Equipment	Electronics Calibration Umbilical
Overhead Crane	-
Mobile Crane	Cherrypicker
Portable Clean Enclosure	-
Engine Handling Kit	Engine Handling Fixture
Engine Alignment Fixture	-
Engine Actuator Support Fixture	-
Engine Actuator Adjustment Kit	-
Stage Alignment Kit	-
Spacecraft Adapter Handling Kit	-
OTV Adapter Handling Kit	-
Insulation Transport Pallet	-
OTV Horizontal Support Fixture	-
Handling Lifting Sling Unit	-
Vertical Lifting Sling Unit	-
OTV Protective Cover	-
Outer Cover	-
Dessicant Kit	-
Battery Handling Kit	-
Pyrotechnics Container	-

Table 5.4-4.—OTV Maintenance Support Equipment (Continued)

GROUND BASED EQUIPMENT	SPACE BASED EQUIPMENT
Erector	-
OTV Lifting Ring	-
Weight & Balance Measurement Set	Zero G Weight and Measurement System
Propellant Control Set-Cryongenic	1
LH_2 Unit	
LOX Unit	
Propellant Supply & Transfer Set-Cryogenic	1
LH_2 Unit	
LOX Unit	
APS Propellant Control Unit	1
APS Propellant Supply & Transfer Unit	1
Ground Pressurization Kit	-
APS Pressure instrumentation Kit	1
Main Propulsion System Checkout Accessoried Kit	Main Propulsion Systems Checkout Accessories Kit
APS Checkout Accessoried Kit	APS Checkout Accessories Kit
Preservation Kit	-
APS Drain, Flush, & Purge Set	1

1 Accounted for in Propellant Handling & Storage

Table 5.4-4.—OTV Maintenance Support Equipment (Continued)

GROUND BASED EQUIPMENT	SPACE BASED EQUIPMENT
Fuel Unit	
Oxidizer Unit	
MLI Purge Metering Unit	-
APS Pneumatic Console	-
Ground Coolant Unit	Coolant Umbilical
Heat Exchanger Console	Heat Exchanger System
Vacuum Pump and Gauge Unit	-
Hydraulic Servicer	-
Cargo Lift Trailer	-
Tug Transfer Dolly	-
Air Arry Tiedown Kit	-
Transport Data Recorder	-
Transport Environmental Kit	-
Flight Component Dolly	-
Inspection Equipment	Inspection Equipment
Ultrasonic Scan Unit	Ultrasonic Scan Unit
Radiography Unit	Radiography Unit
Mass Spectrometer Leak Detection Unit	Mass Spectrometer Leak Detection Unit
Acoustic Leak Detection Unit	Acoustic Leak Detection Unit
Borescope and Fibre Optics	Borescope and Fibre Optics

Table 5.4-4.—OTV Maintenance Support Equipment (Continued)

GROUND BASED EQUIPMENT	SPACE BASED EQUIPMENT
Theodolite	
Orbiter Simulator	-
Payload Docking Simulator	-
Umbilical Simulator	-
Star Simulator	-
Ground Servicing Umbilical Set	2

2 Called out in preceeding items.

Table 5.4-5.—Operational Requirements Summary: OTV Inspection/Checkout/Maintenance Operations

OPERATION/REQUIREMENTS \ BASING MODE	GROUND BASED	SPACE BASED
MOUNTING	OTV horizontally mounted on a cradle in clean room area.	OTV berthed to space base
ACCESS	Many access platforms and workstands	Single cherrypicker, EVA mobility aids
TOOL HANDLING	Personnel carry handtools, test kits, etc. to OTV as required.	Cherrypicker tool kit prepacked with all tools required
MAINTENANCE LOCATION	OTV disassembled as required. Parts and sub-assemblies transported to dedicated areas for maintanence	Maintenance operations performed at OTV. No large scale disassembly
WORK MODE	Conventional "hands on" approach	Dextrous manipulators or EVA crewmen
COMPONENT TRANSFER	Overhead and portable cranes move subassemblies	Cherrypicker transfers modules to and from berthed OTV
ELECTRICAL HOOKUPS	Checkout and test equipment will be located in vicinity of OTV and connected by many umbilicals.	A centralized umbilical plate will be remotely mated to the berthed OTV simultaneously completing all umbilical connections. Checkout equipment will be located remotely in the space base.
MAINTENANCE CREW CHARACTERISTICS	Many skills. Numerous (20-50 men)	Small and cross-trained (2 or 3 at a time)

Table 5.4-5.—Operational Requirements Summary: OTV Inspection/Checkout/Maintenance Operations (Continued)

BASING MODE / OPERATION/ REQUIREMENTS	GROUND BASED	SPACE BASED
TASK TIMELINES	Many tasks performed simultaneously	Most tasks performed sequentially due to limited crew. EVA or dextrous manipulator tasks will take 4 to 6 times as long as equivalent task performed in shirtsleeve environment
AVIONICS SERVICE	Remove avionics at unit level for servicing at dedicated facility. Re-install after servicing.	Changeout avionics modules at unit level using standardized module form factors, remote mating connectors, single mechanical interface. (similar to space telescope or MMS)
PROPULSION SYSTEM	Changeout any component	Changeout main engine module. -Centralized interface -Seal integrity verification
STRUCTURE SUBSYSTEM	Changeout/repair any component	No structural ops on orbit. (50 flt scheduled or contingency return)
THERMAL CONTROL SUBSYSTEM	Changeout of any component	No thermal control system maintenance ops on orbit (50 flt scheduled or contingency return
ACCESS	Removal of structural panels required for access to many subsystems. Shirt-sleeve environment and "hands on" access allow servicing of components in restricted access areas.	Systems accessed by removal of thermal blankets only. (Velcro or button type closures). Access clearance must be very generous to allow dextrous manipulator/EVA access.

Table 5.4-5.—Operational Requirements Summary: OTV Inspection/Checkout/Maintenance Operations (Continued)

OPERATION/ REQUIREMENT \ BASING MODE	GROUND BASED	SPACE BASED
INSTALLATION AND ALIGNMENT OF SUBSYSTEM UNITS	Alignment accomplished using many different jigs and gauges with optical verification. Large numbers of mechanical fasteners for attachment using handtools Many electrical connectors per unit ; hand attached. Cable lacing operations Fluid couplings individually made & verified Many detail parts, handtools, cable clamps. Extensive Q.A. operations during all phases.	Self aligning installations or self verification Single actuator attachment for mechanical attachment using special tools (MMS). Electrical & fluid connections automatically mated during mechanical attachment. Fluid connections seal integrity is self checking (plenum with pressure sensor surrounds seal). Q.A. confined to visual inspection on completion and system checkout.
SYSTEM TEST AND FAULT ISOLATION	Extensive ground support equipment available Checks performed at subsystem or unit level. Fault isolation to board level using portable test equipment	Fault isolation to unit level using BITE. Backup capability in space base supported link to ground based diagnostic equipment.
SYSTEM RELIABILITY	Essentially one mission criteria for reliability	Comprehensive trade required of mission reliability criteria versus cost of space maintainable design.
INSPECTION	Extensive visual inspection using eyeball, borescopes, mirrors, etc. Use of gauges and portable tools magnaflux, zyglo, radiography, ultrasonic, acoustic leak detection. Mass spectrometer leak detection	Limited visual inspection (EVA, CCTV) otherwise restricted to built-in sensors with extensive computer processing of output.

5.5 SUMMARY & CONCLUSIONS

The task focused on creating a preliminary comparison of ground-based and space-based OTV operations. The OTV, SSTO, and HLLV defined in the earlier portion of this study were used as the basis for this operations analysis task. Some key assumptions and groundrules were established and were defined in table 5.3-1. A manned GEO-sortie was used as the mission model. For the space-based scenario, it was assumed that the Space Base was already on-station and it was equipped to handle the OTV mission operations.

Top-level functional flows for both the ground-based and space-based operations were developed. The similarities and deltas exposed by this first level functional flow comparison are listed in table 5.3-2.

Five of the key operations were analyzed to a second level of detail and are listed below:

- HLLV/SSTO Rendezvous Operations
- OTV/Payload Mating Operations
- POTV Launch Operations
- POTV Docking Operations
- OTV Maintenance Operations

It was necessary to select some system/operational options in order to develop the second level detail.

A design analysis was conducted to quantify the mass differences between ground and space based OTV versions. (Section 5.2).

OTV operational requirements (tables 5.4-1, 5.4-2, 5.4-3, and 5.4-5) in their selected areas are compared in tables 5.5-1 through 5.5-5. Primary differences between ground-based and space-based concepts are identified along with key design and technology issues. Major issues identified are summarized in table 5.5-5 along with a description of enhancing and enabling technology issues identified.

Table 5.5-1.—Operational Requirements Comparison: Rendezvous and Docking

OPS REQMTS. / OPERATIONAL BLOCK / VEHICLE	21.0 GROUND BASED	21A.0 SPACE BASED
RENDEZVOUS	EQUIVALENT FOR GROUND AND SPACE BASED MODES	
DOCKING	POTV unmanned active docking mode to contact and latching for docking to SSTO/manned module, and HLLV. Three docking operations - manned module at mission start, SSTO and HLLV at mission end.	Docking requires vehicle stationkeeping and attitude hold. Vehicle passive for terminal docking maneuvers. One docking operation at end of mission. (space base)
COMMUNICATIONS	DATA LINK REQUIREMENTS EQUIVALENT FOR GROUND AND SPACE BASED MODES	
DOCKING PROVISIONS	. HLLV payload retention system fittings. . Active forward docking interface (OTV/MM)	Space base docking system receptacle. Active forward berthing interface (OTV/MM)

o TECHNOLOGY ISSUES:

. Safety aspects of unmanned active vehicle docking with manned module/SSTO
  - Reliability
  - Fail operational
  - Remote monitoring and override by SSTO crew
  - NHB 1700.7 (OSTS Safety Policy and Reqmts for STS payloads) Page a-3 states that the minimum safe distance from the STS orbiter for firing or RCS thrusters is 200 ft

o DESIGN ISSUES:

. Two additional docking operations (1 fueled and 1 empty) on ground based OTV impose ΔV penalty (approx. 40 m/sec)
. Plume impingement from OTV thrusters on SSTO and HLLV during terminal docking.
. Space base docking/launch system
. Active forward docking interface required for unmanned OTV docking with manned module/HLLV tilt table
  - Mass
  - Reliability

o ENHANCING/ENABLING TECHNOLOGIES

. The technology of rendezvous and docking, both manned and unmanned, is well established for active vehicles.
Enhancement of reliability is required for active docking of unmanned vehicles to manned spacecraft.

Table 5.5-1.—Operational Requirements Comparison: Rendezvous and Docking (Continued)

- Technical Issue

  Establish design philosophy and assess mass and cost penalties to achieve reliability sufficient for a 50 mission life without maintenance.

  . Establish life and cycle # requirements for subsystems

  . Evaluate alternate approaches (minimize main engine cycles by extensive use of integrated LOX/LH_2 RCS for auxiliary propulsion) (extensive redundancy vs reliability)

  . Assess penalties for 50 mission design life

- Design Issue

  Space based maintenance; assessment of mass, cost and operational constraints imposed by maintenance compabible design.

  . Assess constraints imposed by

    - Limited crew size
    - Task time limits imposed by use of EVA/manipulator
    - Necessity of sequential phasing of operations.

  . Component Changeout

    - Which components? (avionics, engines, RCS modules ?)
    - How often ?
    - Scheduled or on condition ?
    - Component access
    - Component attachment

Table 5.5-2.—OTV Operational Requirements Comparison: OTV Payload Mating Operations

OPS REQMTS \ BASING MODE	GROUND BASED	SPACE BASED
BERTHING	Initial positioning of payload with respect to OTV established by OTV active docking mechanism. OTV active berthing system is activated automatically by completion of docking mechanism retration cycle, along with automatic OTV/payload umbilical mate.	Space base payload handling mechanism transports and aligns payload to OTV berthing interface; verifies alignment and activates (by remote control) OTV active berthing interface, along with automatic OTV/payload umbilical mate.

o Primary Difference(s)	Ground based OTV requires active terminal docking system to activate payload mating operations. See operational requirements comparison for rendezvous and docking.
o Technology Issues	
o Design Issues	Design of lightweight, reliable active berthing interface for OTV.

Table 5.5-3.—OTV Operational Requirements Comparison: Launch Operations

BASING MODE / OPS REQMTS.	GROUND BASED	SPACE BASED
STANDOFF	Active standoff under OTV control using RCS	OTV passive standoff using space base docking and launching system. Upper Under space base control
CREW TRANSFER	OTV crew transfer requirements equipment for ground and space based modes	

- o Primary. Differences: Active standoff for ground based vehicle versus passive for space based
- o Technology Issues
- o Design Issues:
  - o Space Base Docking/Launch System
    - sizing/requirements
    - mechanization
    - control
  - o Space Base Launch Control
    - visual monitoring
    - degree of automation/crew participation

Table 5.5-4. OTV Operational Requirements

COMPARISONS: MAINTENANCE, INSPECTION AND CHECKOUT

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o Table 5.4-5 presents a comparison of requirements in these areas. Inspection reveals that the ground based vehicle requirements are conventional. Primary technical and design issues in the space based configuration are:

o Inspection:
- Analysis of Inspection Requirements (TECHNOLOGY)
  - . What needs inspection
  - . What kinds of inspection are needed?
    - Visual ?
    - Leak ?
    - EMI ?
- Mechanization of Inspection (DESIGN)
  - . EVA
  - . Remote Optical (periscope)
  - . Remote CCTV: Pan, tilt, zoom, boom mount, color; stereo?
  - . Built-in sensors - computer processing for fault detection.
  - . Inspection access requirements

o Checkout:
- Electrical Systems Test
  - . BITE provisions
  - . Test interface lead outs (separate bus?)
  - . Type, number & location of sensors (temp? Current?)
  - . Sensor reliability and redundancy. data bussing
  - . Checkout computation
    - On OTV ?
    - Space Base ?
    - Data link to ground?
  - . Software: Enhancement of sensor redundancy
- Mechanical Systems Test
  - . What is tested? (i.e., main engine TVC actuators)
  - . How is testing accomplished (actuator, valve cycling ?)
  - . What parameters are monitored and how
  - . How are systems powered during test

o Cost Effective Achievement of 50 Mission Lifetime:
- Primary issue is reliability (technical) versus maintenance (design)

(Continued)

Table 5.5-4. OTV Operational Requirements (continued)

- Technical Issue

  Establish design philosophy and assess mass and cost penalties to achieve reliability sufficient for a 50 mission life without maintenance.

  . Establish life and cycle # requirements for subsystems

  . Evaluate alternate approaches (minimize main engine cycles by extensive use of integrated LOX/LH_2 RCS for auxiliary propulsion) (extensive redundancy vs reliability)

  . Assess penalties for 50 mission design life

- Design Issue

  Space based maintenance; assessment of mass, cost and operational constraints imposed by maintenance compabible design.

  . Assess constraints imposed by

    - Limited crew size
    - Task time limits imposed by use of EVA/manipulator
    - Necessity of sequential phasing of operations.

  . Component Changeout

    - Which components? (avionics, engines, RCS modules ?)
    - How often ?
    - Scheduled or on condition ?
    - Component access
    - Component attachment

Table 5.5-5.—OTV Operational Requirement Issue Summary

o MAJOR ISSUES IDENTIFIED	
- Rendezvous and Docking:	Active docking of unmanned vehicle with man rated spacecraft
- Payload Mating:	Design of OTV active berthing interface
- Launch Operations:	Space base docking and launch systems design
- Maintenance, Inspection and Checkout:	Trade to define proper balance between reliability and maintainability
o ENHANCING AND ENABLING TECHNOLOGIES:	Increased reliability in subsystems is a technology area which:
	- Enables unmanned docking operations with manned SSTO
	- Potentially reduces or eliminates requirement for space based maintenance
	- Enhances vehicle life and mission reliability

Major conclusions reached during the operations analysis study include:

- o Significant design advantages exist for the space-based configuration.
  - Stage mass fraction of .936 as opposed to .910 for the ground based.
  - Elimination of the requirement for active unmanned terminal docking operations
- o Reliability is the major technology issue uncovered by the study.
  - Space-based OTV reliability is a critical factor in determining the amount and frequency of space maintenance required. Limitations on the scope of space-based maintenance operations will impose stringent reliability requirements.
  - Basic ground-based OTV reliability requirements are less stringent due to ground-based refurbishment after each flight. Additional requirements are imposed by the unmanned active docking required with the manned SSTO transport.
- o Many critical options were not addressed in the study such as space basing of the crew module, which offers operational and design advantages for OTV and SSTO. Any reliability improvements for space basing can also enhance ground basing.

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15. Supplementary Notes

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16. Abstract

Studies were conducted under this NASA contract to define "Technology Requirements for Future Earth-to-Geosynchronous Orbit Transportation Systems," and are summarized in this final report. The objective of this work was to identify those technologies either critical to performance or which offer cost advantages compared to the investment required to bring them to usable confidence level. Several previous studies have addressed similiar objectives, however, none have had the unique perspective of a total transportation system as an evaluation yardstick. Vehicles included in the system are: (1) a single-stage-to-orbit (SSTO) launch vehicle used in a priority cargo role; (2) a matching orbit transfer vehicle; (3) a heavy-lift launch vehicle with a low-Earth orbit delivery capability of 226 757 kg (500 000 lb); and (4) a matching solar electric cargo orbit transfer vehicle (LCOTV). The system and its reference technology level are consistent with an initial operational capability in 1990. The 15-year mission scenario is based on early space industrialization leading to the deployment of large systems such as Solar Power Satellites. Life-cycle cost benefits in discounted and undiscounted dollars for each vehicle, technology advancement, and the integrated transportation system are calculated. A preliminary functional analysis was made of the operational support requirements for ground based and space based chemical propulsion orbit transfer vehicles.

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