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Satellite Power System: Concept Development and Evaluation Program

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INTRODUCTION AND SUMMARY

In October 1978, a reference Satellite Power System (SPS) was adopted (Reference 1) which provided technical and operational information required in support of environmental, socioeconomic, and comparative assessment studies. The reference SPS system included a reference space transportation system which was selected from alternative concepts which had been studied at various depth and to differing requirements at that time. Additional analyses and investigations have been conducted since that time to further define transportation system concepts that will be needed to support the developmental as well as the operational phases of the SPS program. To accomplish these objectives, transportations systems such as Shuttle and its derivatives have been identified; new heavy-lift launch vehicle (HLLV) concepts, cargo and personnel orbital transfer vehicles (COTV and POTV), and intra-orbit transfer vehicle (IOTV) concepts have been evaluated; and, to a limited degree, the program implications of their operations and costs were assessed. The results of these analyses have been integrated into other elements of the overall SPS concept definition studies.

This document presents a summary of the reference SPS space transportation system and its operations, a more detailed and updated description of applicable key elements, a description of the more promising alternative concepts, and recommendations for possible changes in the reference transportation concept. In addition, key issues such as propellant production, environmental impact, and technology advancement requirements are addressed.

The key requirements driver in SPS transportation systems synthesis is that of mass to orbit and the corresponding necessity to minimize those transportation costs. This singular requirement has led to the consideration of transportation elements with payload-carrying capabilities and launch/flight rates significantly greater than that perceived for any other contemporary program. Although many SPS options with different configurations and weights evolved during the course of transportation system synthesis, the impact on transportation options and their concepts requirements is considered negligible (i.e., the mass-to-orbit requirement is dominant). However, since the transportation systems costs have been developed for differing SPS concepts and traffic models, they can only be evaluated on a comparative basis with overall SPS systems approach. Specific transportation systems costs are, therefore, not included in this volume.

REFERENCE CONCEPT

The vehicles are distinguished by their primary payload, either cargo or personnel, and their area of operations between earth and low earth orbit (LEO) or between LEO and geosynchronous earth orbit (GEO). Cargo is transported from the earth's surface to LEO by the HLLV and personnel (and priority cargo) are transported from earth to LEO and back by the PLV. Transportation between LEO and GEO is provided by the COTV and the POTV.

The general ground rules followed in the development and evaluation of the transportation system are listed below.

- The SPS transportation system elements, with the possible exception of Shuttle-derived PLV's, are dedicated and optimized for the installation, operation, and maintenance of the SPS.
- The SPS transportation system will be designed for minimum total program cost consistent with technology advancement expectations of the early 1990's.
- Energy requirements will be minimized consistent with minimum cost.
- Environmental impact will be minimized and, so far as possible, protective measures needed will be factored into cost analyses.
- The use of critical materials will be minimized consistent with cost, energy, and environmental impact requirements.

Heavy-Lift Launch Vehicle (HLLV)

The reference HLLV is a two-stage, vertical takeoff, horizontal landing (VTOHL), fully reusable winged launch vehicle. The launch configuration and overall geometry are detailed in Figure 1. The vehicle uses sixteen CH_4/O_2 engines on the booster (first stage) and 14 standard SSME's on the orbiter (second stage). The booster engines employ a gas generator cycle and provide

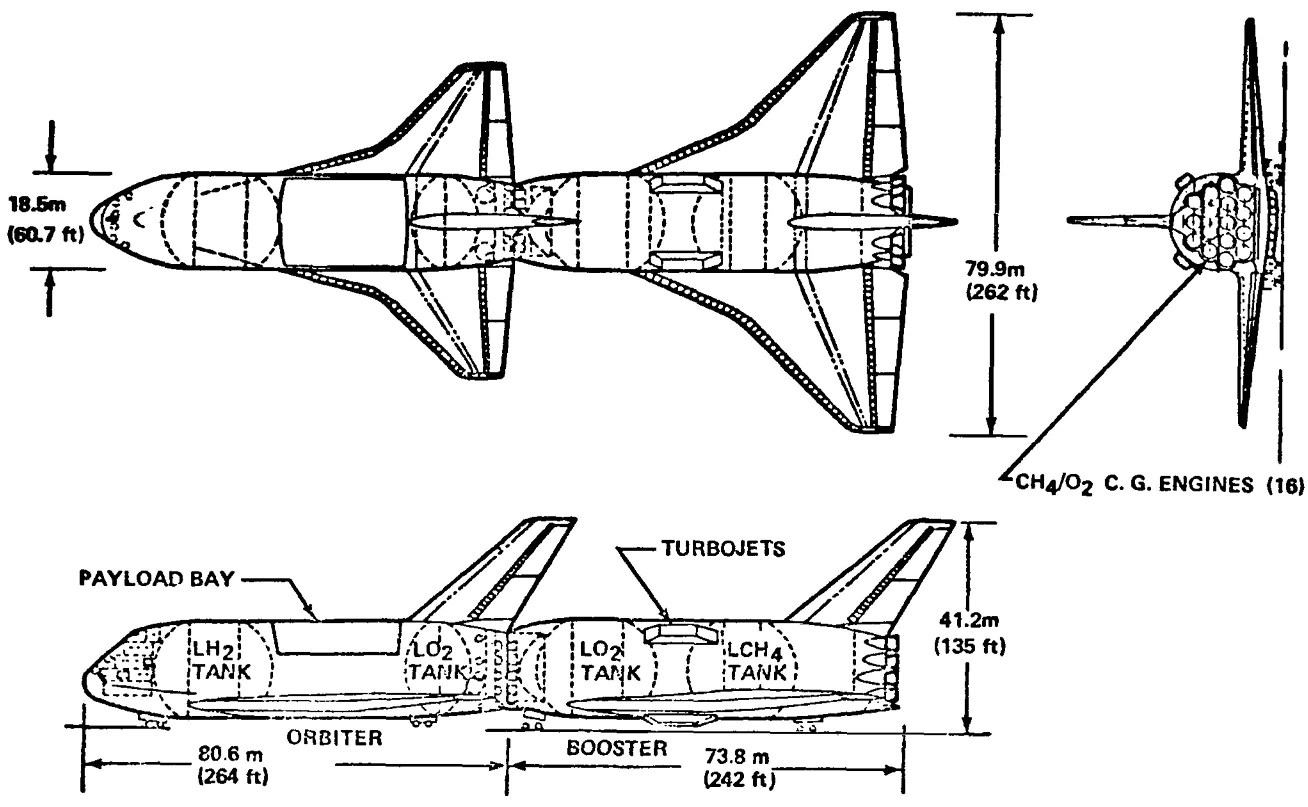


Figure 1. Reference Two-Stage Winged SPS Launch Vehicle

a vacuum thrust of 9.79×10^6 newtons each. The orbiter SSME's provide a vacuum thrust of 2.09×10^6 newtons each at 100% power level. The gross liftoff weight of the HLLV is 11,040 metric tons with a payload to LEO of 424 metric tons.

An airbreather propulsion system (aircraft jet engine) is provided on the booster to provide flyback capability and simplify the booster operations. Its landing weight is 934 metric tons. The orbiter deorbits and performs a glide-back landing maneuver. Its landing weight is 453 metric tons which includes an assumed returned payload of 63.5 metric tons, or 15% of the payload delivered to LEO.

Personnel Launch Vehicle (PLV)

The PLV provides for the transportation of personnel and priority cargo between earth and low earth orbit. The reference vehicle is derived from the current Space Shuttle system. It incorporates a winged liquid propellant fly-back booster instead of the solid rocket boosters and has a personnel compartment in the orbiter payload bay capable of transporting 75 passengers. The overall configuration and vehicle characteristics are shown in Figure 2. The passenger module is also illustrated in the figure.

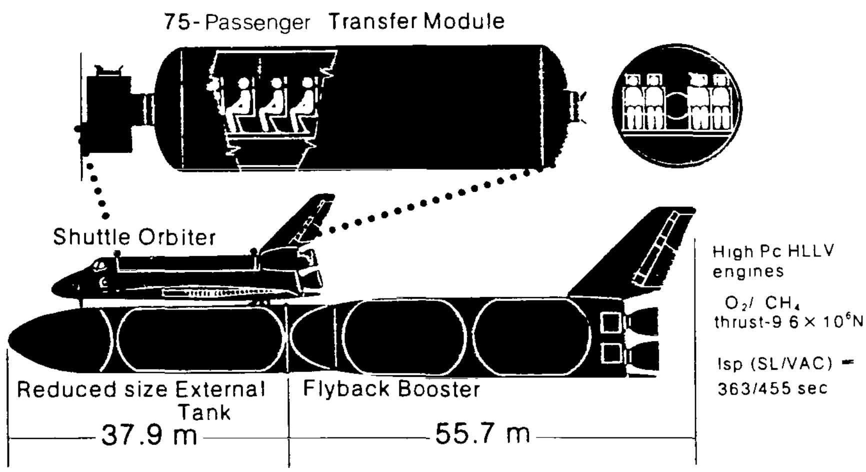


Figure 2. Reference Personnel Launch Vehicle

The booster employs four O_2/CH_4 engines similar to those on the HLLV booster. A series burn ascent mode is utilized and the external tank (ET) is a resized, smaller version of the Space Shuttle tank, carrying 546 metric tons of propellant versus 715 metric tons for the current space transportation system (STS).

Personnel Orbital Transfer Vehicle (POTV)

The functions of the POTV are to deliver personnel and priority cargo from LEO to GEO and to return personnel from GEO to LEO. The reference vehicle is a two-stage (common stage) LO_2/LH_2 configuration as illustrated in Figure 3.

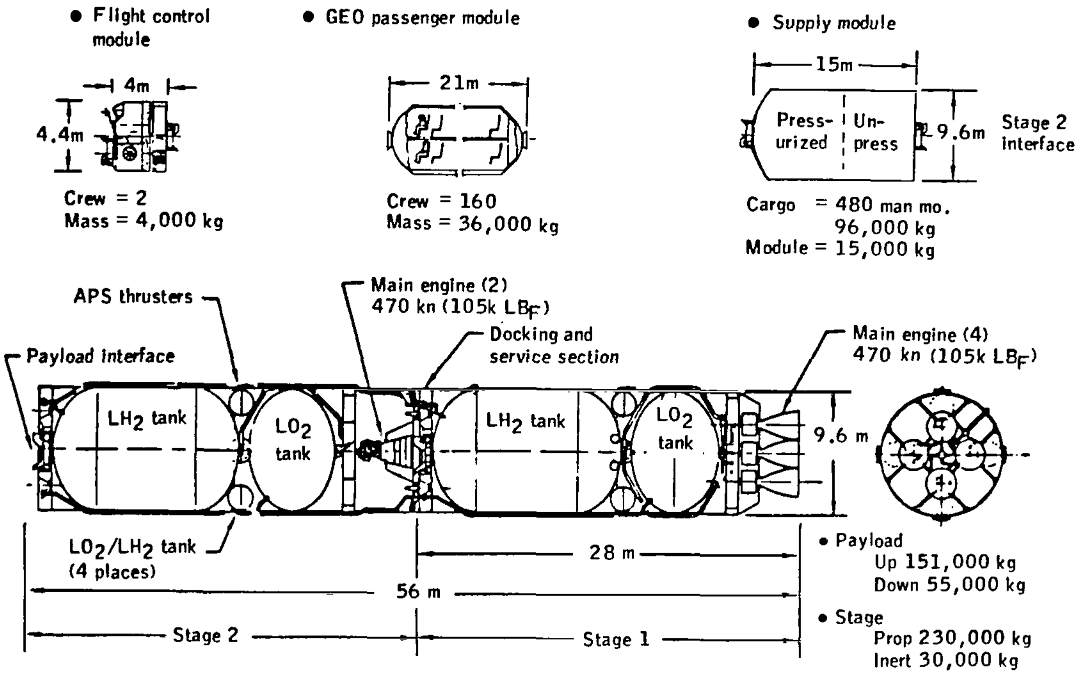


Figure 3. Reference LO_2/LH_2 Common Stage POTV

The start burn weight is 890 tons with an up-payload of 151 tons and a down-payload of 55 tons. The up-payload consists of 160 personnel in a passenger module, 480 man-months of consumables in a resupply module, and a flight control module piloted by a crew of two. The down-payload is identical except the resupply module returns empty to LEO.

Cargo Orbital Transfer Vehicle (COTV)

The function of the COTV is to deliver SPS cargo to GEO from the LEO staging area. The basic concept involves the construction of a fleet of reusable electric powered roundtrip vehicles and their dedicated solar array in LEO. The vehicle uses ion bombardment thrusters with cryogenic argon as the propellant. The ion thruster propellant was selected on the basis of availability, storability, absence of serious environmental impacts, cost, demonstrated performance, and technical suitability. Power conversion options are GaAlAs and Si photovoltaic array systems illustrated in Figure 4.

The first option utilizes a self-annealing GaAlAs array with a concentration ratio of 2, and provides a LEO-GEO trip time of 133 days and a total round trip time of less than 180 days. Ion bombardment thrusters of 100 cm diameter are used with an I_{sp} of 13,000 seconds and argon as the working fluid. The primary thruster array of 259 thrusters is suspended by cables and located at the vehicle center of gravity. Additional attitude thruster control packages are located at the structural extremities. The vehicle has a total mass of 4400 metric tons and a payload delivery capability of 3500 metric tons.

The second option utilizes a silicon photovoltaic solar array in a planar configuration with no concentration reflectors. Roundtrip time from LEO-GEO-LEO is approximately 160 days, which also allows two trips per year for each

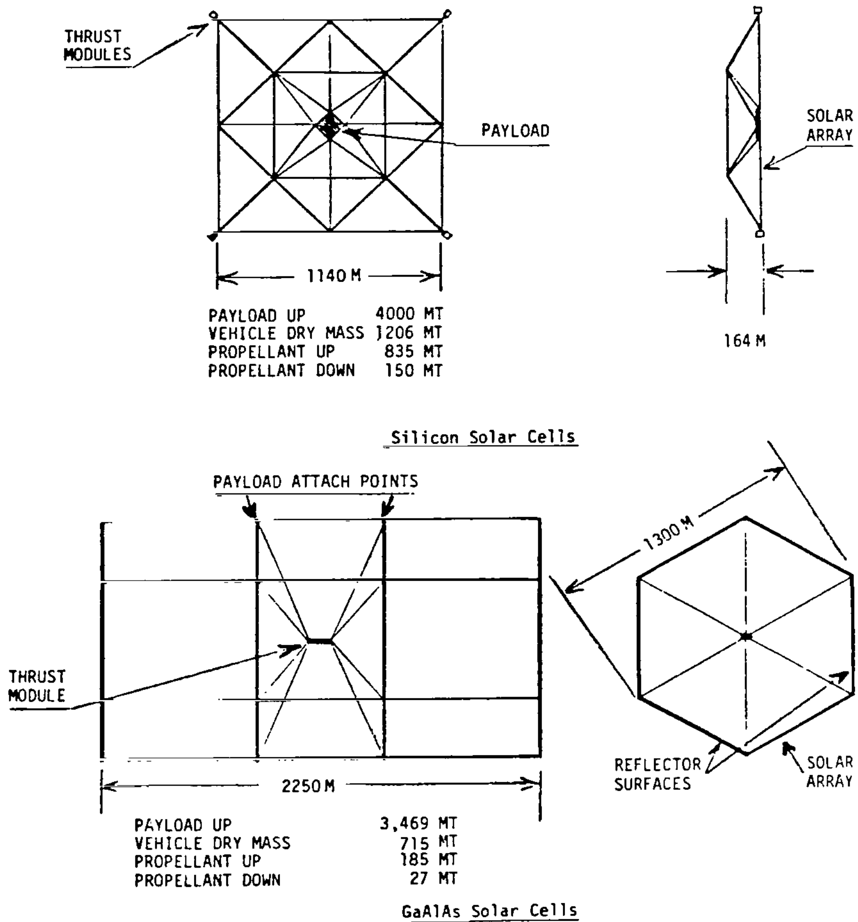


Figure 4. Reference Cargo Orbit Transfer Vehicle Options

COTV. Ion bombardment thrusters of 120 cm diameter are used with an I_{sp} of 7000 seconds and argon as the working fluid. Thruster modules of 296 electric thrusters each and an appropriate number of chemical thrusters are located at the four corners of the COTV. The vehicle has a total mass of 6200 metric tons and a payload delivery capability of 4000 metric tons.

Operations

SPS operations include those activities required to build SPS's and then to operate and maintain them. This requires a wide variety of activities as illustrated in Figure 5.

A significant mass production capability must be developed to produce the large quantity of diverse components required for satellite construction. Similarly, requirements for large quantities of propellants (oxygen, hydrogen, hydrocarbon, and argon) will demand greatly expanded processing capabilities. Also, the transportation of raw materials, fabricated components and assemblies, and propellants to the launch site will require extensive and efficient cargo handling and planning methods.

At the launch site, principal activities involve receiving, storing, and processing of material and propellants; launching vehicles; and refurbishing and checking out returning vehicles. Incoming material (via rail, air, etc.)

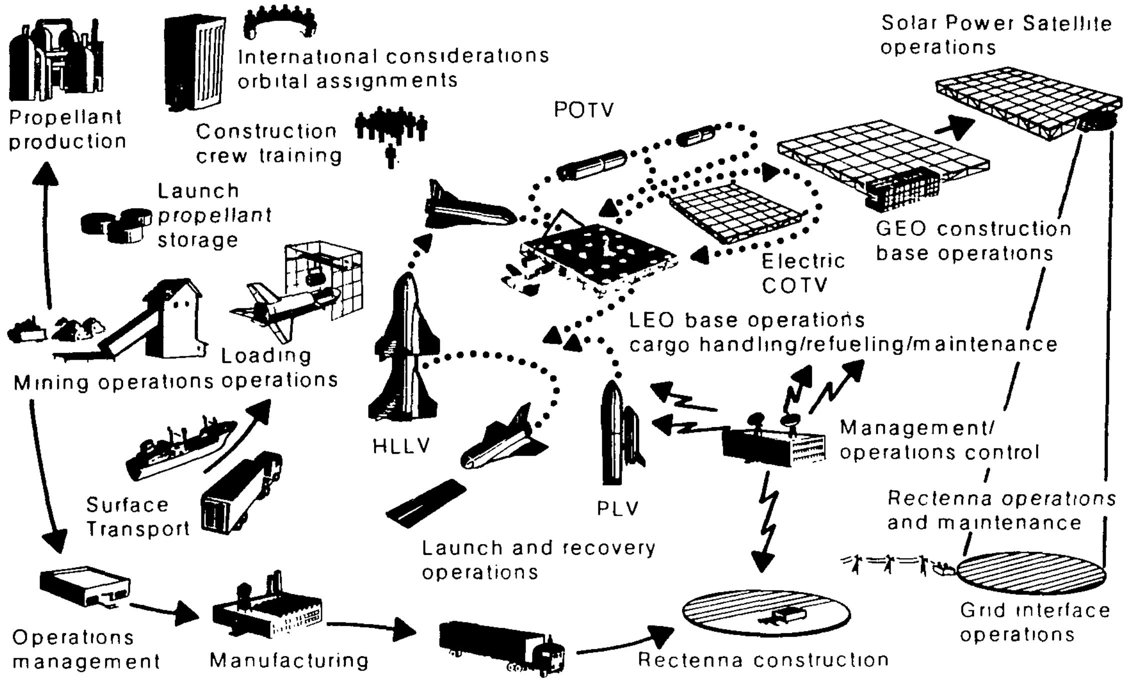


Figure 5. SPS Construction and Commercial Operations

is off-loaded, inspected, inventoried, and stored in warehouses. Component packaging (for construction material, consumables, spares) is very significant for construction as well as space transportation. Packages must meet dimensional and weight constraints of the launch vehicle and have appropriate mass density for cost-effective space transportation. Densities vary from a low of 12 kg/m^3 for antenna subarray elements to about 2500 kg/m^3 for power conductors. To obtain an efficient payload density, components must be packaged in appropriate mixes in order to minimize the number of launches, thereby reducing transportation costs. The silicon SPS option requires 375 HLLV flights and the GaAlAs option requires 225 HLLV flights to transport construction material for 10-GW (two 5-GW units) capability. Construction personnel are launched in an updated Shuttle PLV.

Operations in LEO include COTV construction and maintenance, payload transfers between HLLV's and COTV's, POTV stage mating, crew transfers, vehicle and base maintenance, and propellant storage and transfer.

After payload transfers, COTV's travel to GEO over a period of several months. At GEO, a small intraorbital transfer vehicle moves the cargo to the construction base. After off-loading, the COTV returns to LEO with packing materials, damaged or defective equipment, and parts and consumables containers. At LEO, argon tanks and thruster grids are replaced, the vehicles refurbished and readied for the next transit.

Construction personnel arriving at LEO in the updated Shuttle PLV from earth, transfer to POTV's for the trip to GEO, which takes a few hours. Personnel returning from GEO transfer to personnel launch vehicles for the trip back to earth.

Figure 6 presents a typical timeline for the silicon option for constructing the initial LEO and GEO bases and the COTV's required to then construct SPS's. Once the first COTV is completed, it begins to transport materials to GEO needed for the GEO construction base. Nine months are required to construct the GEO base. After two years, all of the major elements are available to begin production of the first SPS.

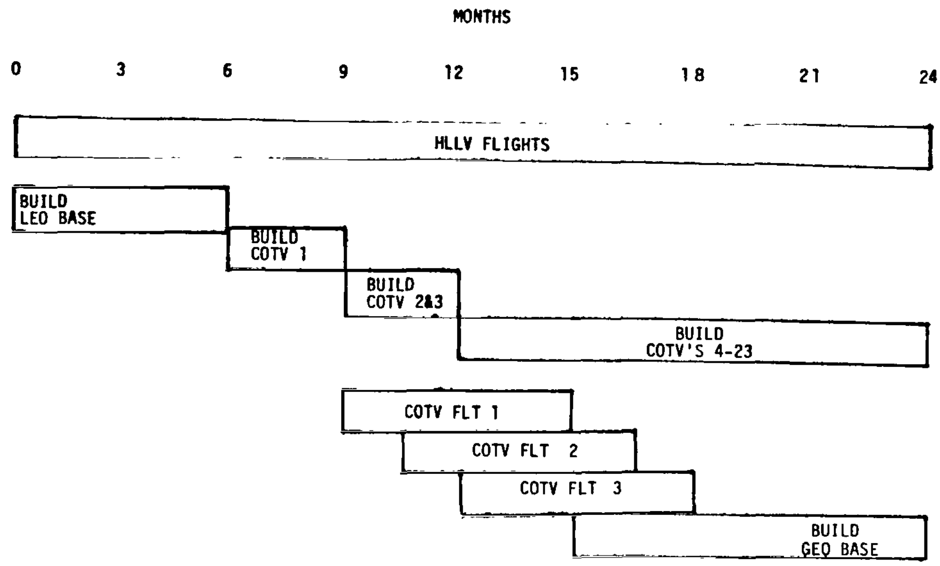


Figure 6. Construction Base Buildup for Silicon System

For the gallium option, the GEO base would be built first in LEO where it would construct the COTV's. Then, two COTV's would transfer the base to GEO and leave only staging facilities in LEO.

Figure 7 shows estimates of the number of flights required, payload characteristics, launch vehicle packaging factors assumed, and numbers of people associated with the initial two-year buildup period. Data are presented for both silicon and GaAlAs options.

Figure 8 presents estimates of the number of flights required, payload characteristics, packing factors assumed, and numbers of people associated with the construction of two SPS's per one-year period. Data are presented for both silicon and gallium options.

Table 1 shows the fleet sizes of HLLV's, PLV's, COTV's, and POTV's required for the buildup period prior to SPS construction and the construction of two SPS's per year. Data are presented for both silicon and gallium options. Fewer COTV's are needed for the gallium option due primarily to the different COTV design and flight times, and different satellite weights.

	SILICON	GALLIUM
LEO STAGING BASE		
MASS	2,405 MT	110 MT
PEOPLE	225	35
GEO CONSTR BASE		
MASS	8,353 MT	6,000 MT
PEOPLE	480	680
PAYLOADS		
HLLV	424 MT	424 MT
PLV	75 PEOPLE	75 PEOPLE
POTV	400 MT 160 PEOPLE	400 MT 160 PEOPLE
COTV	4,000 MT	3,500 MT
PACKING FACTORS		
HARDWARE	85%	90%
PROPELLANTS	95%	90%

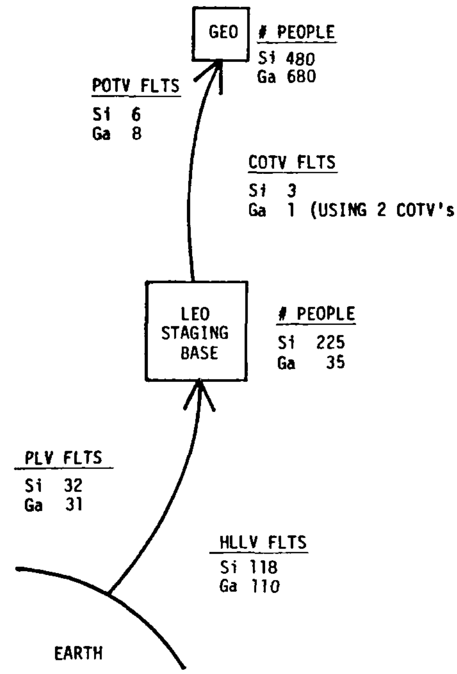


Figure 7. Scenario for Buildup of Construction Bases

	SILICON	GALLIUM
SPS MASS	50,984 MT	34,159 MT
PAYLOADS		
HLLV	424 MT	424 MT
PLV	75 PEOPLE	75 PEOPLE
POTV	160 PEOPLE 400 MT	160 PEOPLE 400 MT
COTV	4,000 MT	3,500 MT
PACKING FACTORS		
HARDWARE	85%	90%
PROPELLANTS	95%	90%

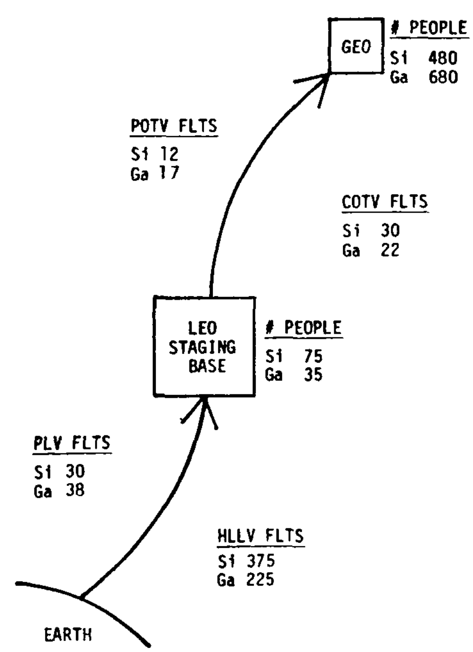


Figure 8. Scenario for Construction of Two 5-GW Satellites/Year

Table 1. SPS Fleet Sizes

	HLLV BOOSTERS	HLLV ORBITERS	PLV	COTV	POTV
STARTUP	2 (1)	3 (1)	2	3 (2)	2 (1)
ADDITIONAL REQUIRED FOR CONSTRUCTION OF TWO 5 GW SATELLITE/YEAR	2 (2)	2 (2)	-	20 (6)	-
TOTAL REQUIRED FOR CONSTRUCTION OF TWO 5 GW SATELLITE/YEAR	4 (3)	5 (3)	2	23 (8)	2 (1)

NOTE: PARENTHESIS () IDENTIFIES FLEET REQUIREMENTS FOR GALLIUM SATELLITE

ALTERNATIVE CONCEPTS

During and prior to the SPS Concept Definition Studies, the number and variety of transportation systems concepts and options evaluated have been quite extensive. The earliest configurations were synthesized during the MSFC HLLV study contract, NAS9-14710 (Reference 2) and the Future Space Transportation Systems Analysis Study, JSC Contract NAS9-14323 (Reference 3). These studies were followed by the SPS Feasibility and Concept Definition study phases (References 4 through 11) during which the space transportation systems were "tailored" to SPS requirements. The alternative concepts presented herein are necessarily limited and are believed to represent the most promising of those concepts evaluated.

Heavy-Lift Launch Vehicle (HLLV)

Of the many HLLV options investigated (i.e., one- and two-stage ballistic or winged, parallel or series burn, etc.), three of the more promising cargo delivery options are presented herein; the two-stage series burn vertical takeoff horizontal landing (SB/VTO/HL) HLLV (reference concept), Figure 8; a two-stage prallel burn vertical takeoff horizontal landing (PB/VTO/HL) HLLV, Figure 9; and an advanced technology option horizontal takeoff/landing single-stage-to-orbit (HTO/SSTO) HLLV, Figure 10. In addition, alternate payload (smaller) options have been evaluated for the first two configurations, Figures 11 and 12. Each configuration option offers its own unique advantages along with distinct technology advancement requirements. However, on the basis of technology advancement requirements, the smaller HLLV option (series or parallel burn) appears to best satisfy the needs of any SPS program while maintaining a utility for other potential space endeavors.

Personnel Launch Vehicle (PLV)

In the alternate concept, crew transfer from earth to LEO would be accomplished with the SPS HLLV, thus eliminating the requirement for maintaining a separate PLV fleet throughout the SPS construction and operational program.

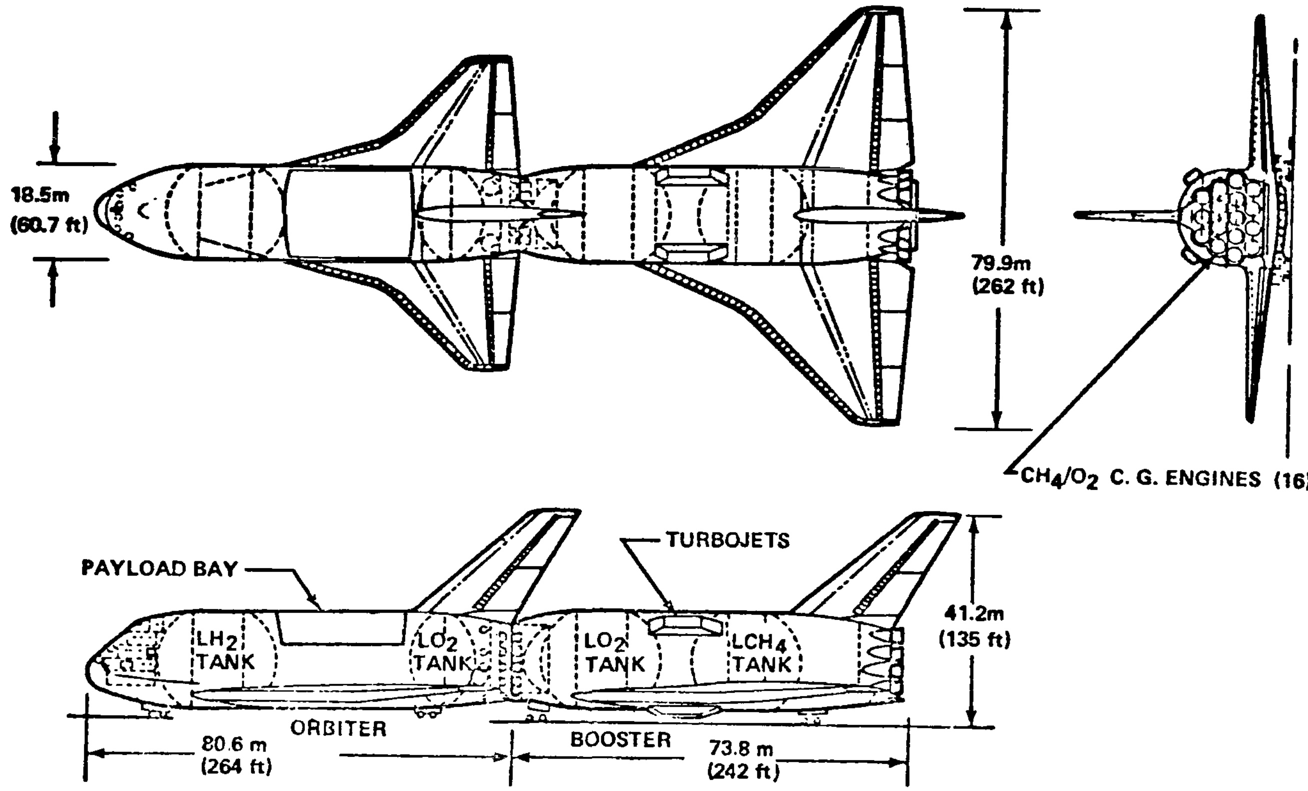


Figure 8. VTO/HL HLLV Concept—Series Burn (Reference Concept)

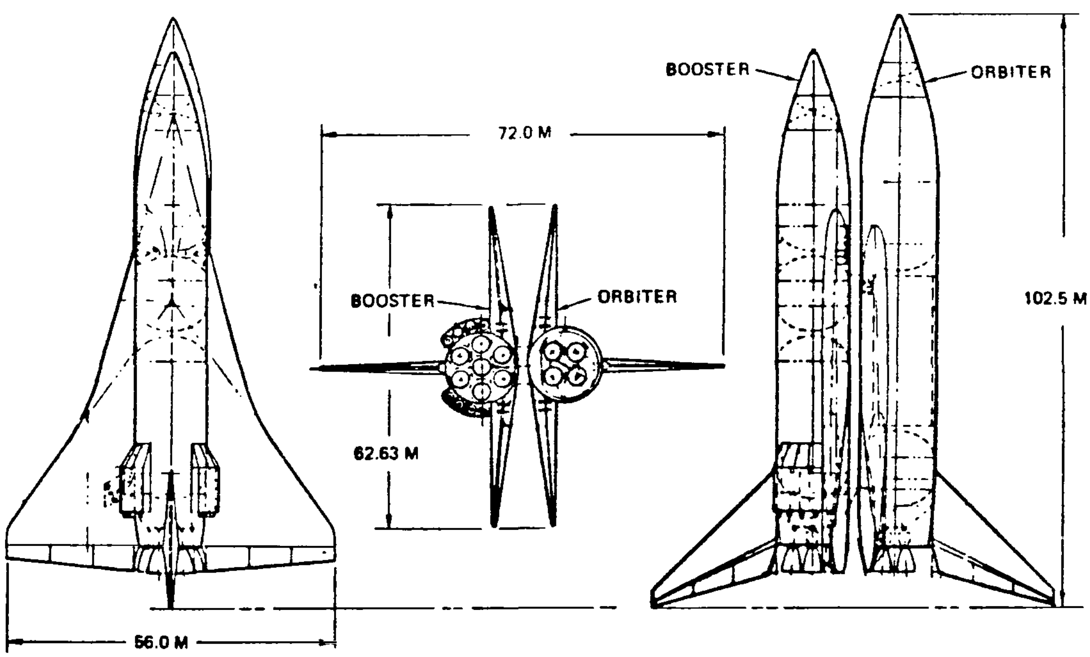


Figure 9. VTO/HL HLLV Concept—Parallel Burn

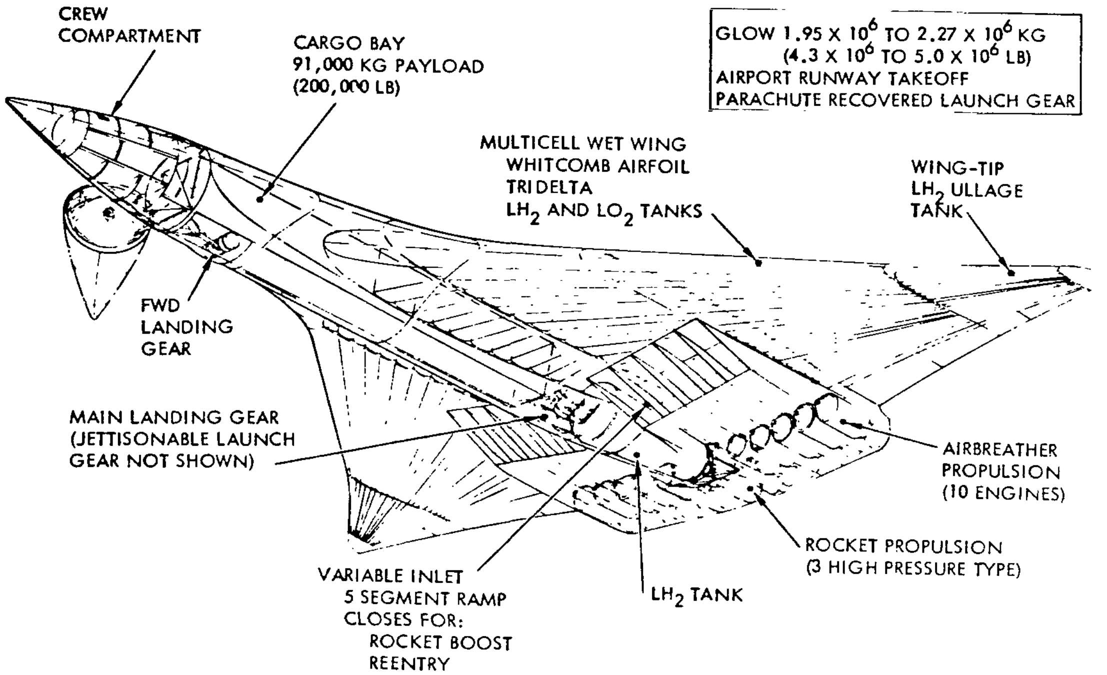
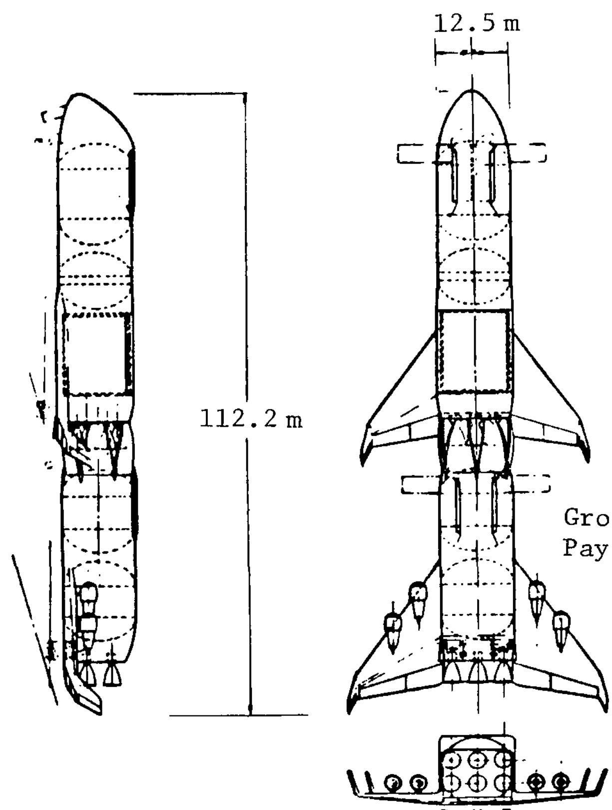
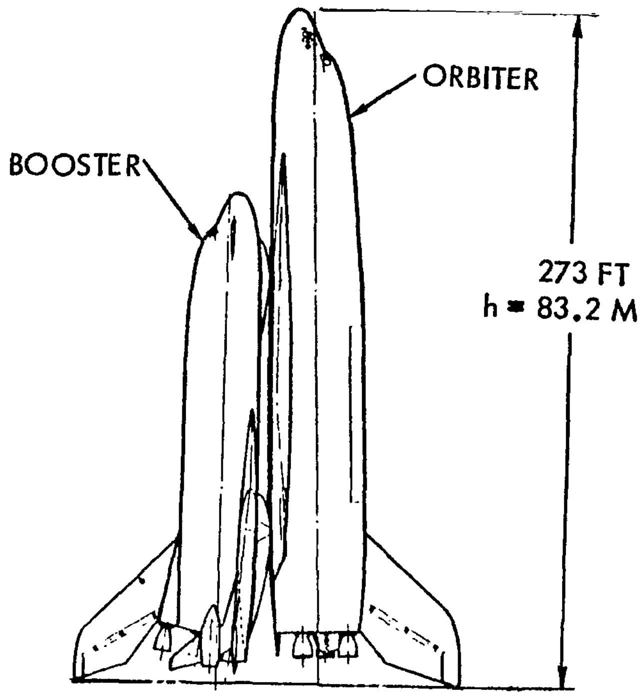


Figure 10. HTO/SSTO HLLV Concept



Gross weight: 4,034,000 kg
Payload: 126,000 kg

Figure 11. SB/VTO/HL HLLV (Small Payload Option)



PAYLOAD - 114,000 KG
GROSS WT - 3,560,000 KG

Figure 12. PB/VTO/HL HLLV (Small Payload Option)

Personnel Orbital Transfer Vehicle (POTV)

As previously stated, the reference POTV concept utilized a two (common) stage propulsive element to transport crew and crew supplies and priority cargo to GEO. The stages are fueled in LEO and are capable of a roundtrip mission. In the alternate concept(s), a single stage propulsive element is employed, Figure 13, to accomplish the transfer from LEO to GEO where the stage is refueled to accomplish the return trip to LEO. This approach is more cost-effective because of the reduced operational complexity and the lower cost of transporting return propellants to GEO by the COTV. The Figure 13 concept was optimized (crew module size) for the silicon SPS concept and is designed to transport both crew and crew supplies. Another concept, Figure 14, has been

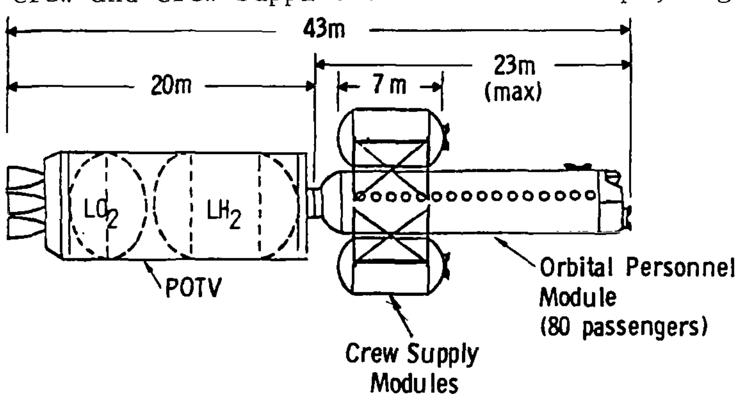
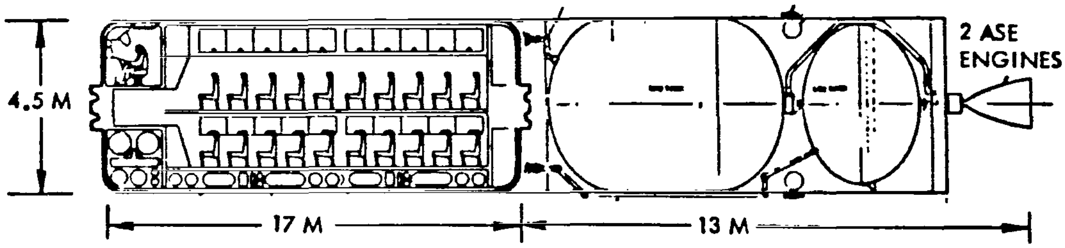


Figure 13. Orbital Crew Rotation/Resupply POTV Configuration



- 60 MAN CREW MODULE 18,000 KG
- SINGLE STAGE OTV (GEO REFUELING) 36,000 KG
- BOTH ELEMENTS CAPABLE OF GROWTH STS LAUNCH

Figure 14. Orbital Crew Rotation POTV Configuration

sized to satisfy the GaAs SPS concept and offers the additional advantage of being capable of transport to LEO in the growth STS (i.e., both elements can fit within the Shuttle cargo bay). This characteristic is of importance in the precursor or pilot plant phase of the SPS program when the HLLV is not available.

Cargo Orbital Transfer Vehicle (COTV)

The reference concept in itself offers the option of a silicon or gallium arsenide powered electrical orbital transfer vehicle (EOTV). However, the power source is only one of the several differences in technical approach between the two configurations. The GaAs concept utilizes high current density thrusters with direct power drive from the main solar array(s) as opposed to the use of low current density thrusters utilizing power processors for thruster primary voltage employed in the silicon concept. The higher current density thruster will result in shorter grid life which will possibly necessitate more frequent grid changes during EOTV life; however, the increased performance and reduced number of thrusters required are believed to offset that disadvantage. In addition, the silicon powered EOTV utilizes a chemical propulsion system for thrusting and attitude control in the shadow periods, whereas the GaAs concept employs an energy storage system (batteries) to provide the required electrical power for attitude hold only during periods of shadow. Again, the energy storage system weight is considerably less than the weight of the chemical propellant system and its fuels.

The updated COTV configurations and pertinent characteristics are presented in Figures 15 and 16.

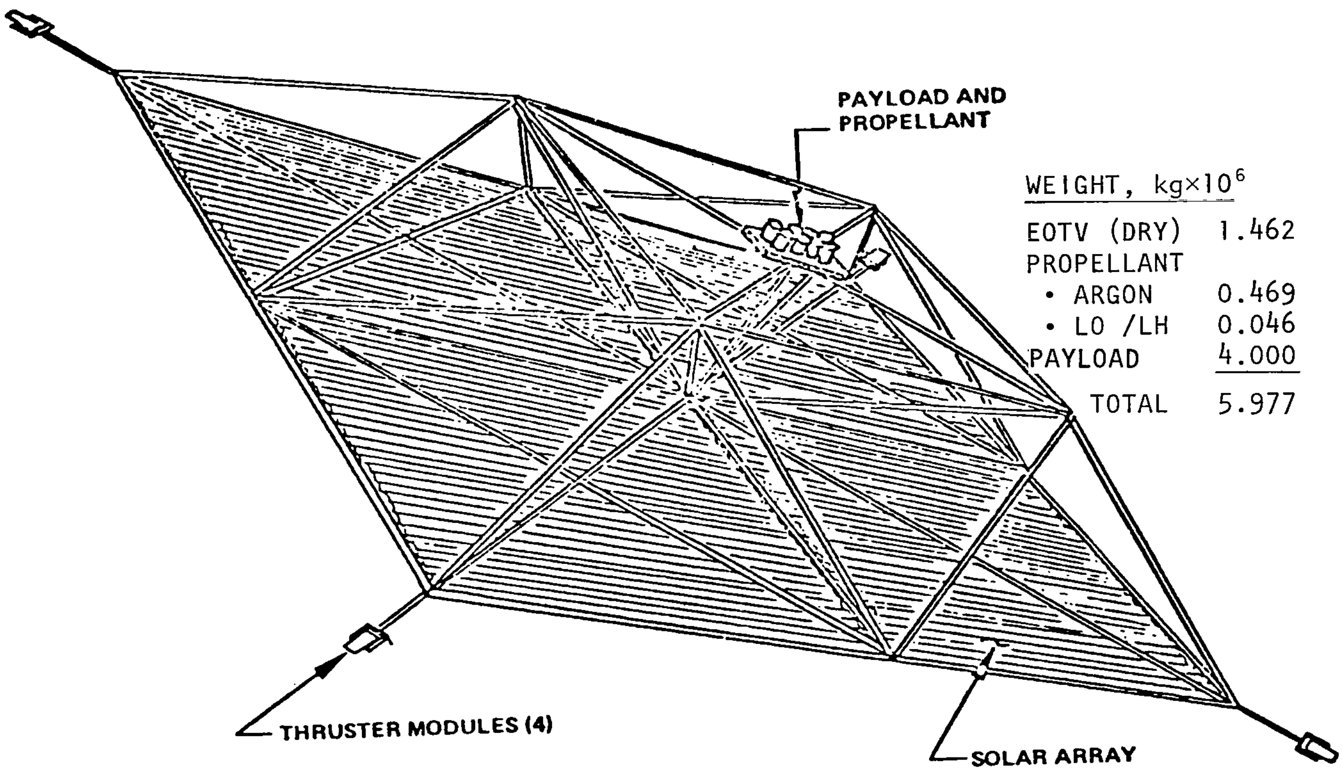


Figure 15. Silicon Cell EOTV Configuration

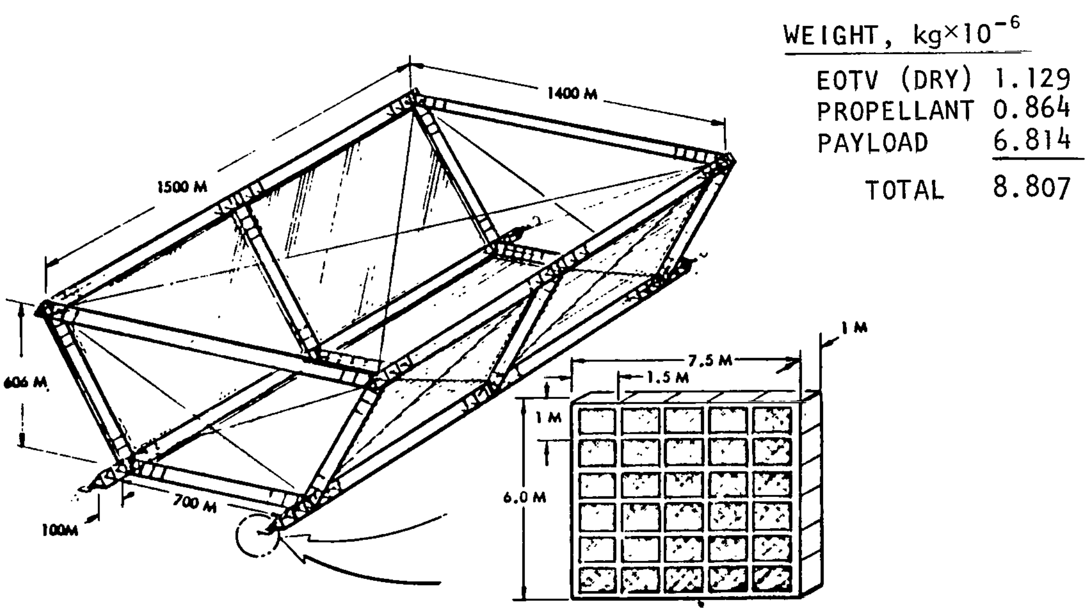
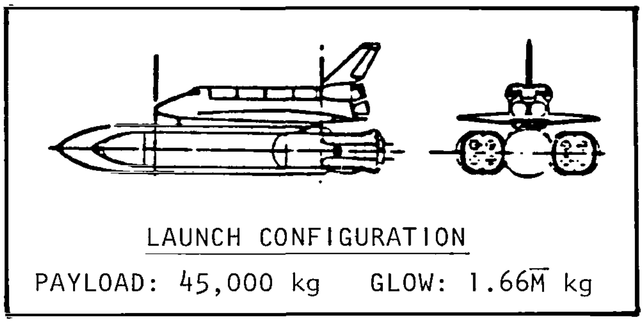


Figure 16. Gallium Arsenide Cell EOTV Configuration

Growth STS and STS-Derived HLLV

Numerous growth options for the Shuttle Transportation System have been proposed (References 12 and 13). An option selected for SPS pilot plant operations is essentially that shown in Figure 17 and described in Reference 14. This growth version is of the minimum change type (i.e., the STS solid rocket boosters are replaced with liquid rocket boosters). The proposed change will result in a Shuttle-delivered payload capability of approximately 45,000 kg, and when the Shuttle orbiter is replaced with an interim HLLV payload module, the vehicle will have a payload delivery capability of approximately 100,000 kg.



BOOSTER (EACH) x 10^3
GROSS WT = 395 kg
PROP. WT = 324 kg
INERT WT = 71 kg

SSME-35
F = 2043 KN (S.L.) (EACH)
I_{SP} = 406 SEC (S.L.)
ε = 35:1
MR = 6:1

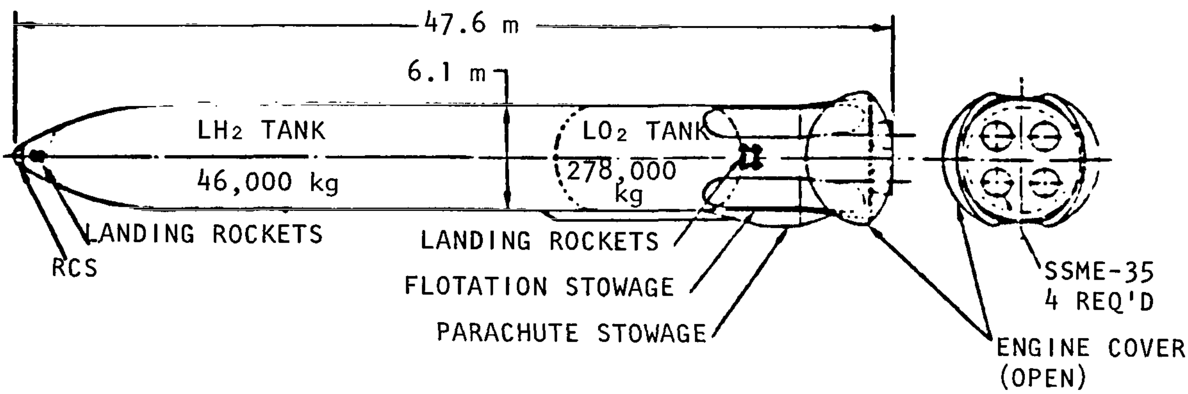


Figure 17. LO_2/LH_2 SSME Integral Twin Ballistic Booster

RELATED TRANSPORTATION SYSTEM ISSUES

At its peak, the SPS program will require the production of approximately 500 metric tons of liquid hydrogen and 5000 metric tons of liquid oxygen per day. Alternate production and storage concepts have been evaluated to satisfy these large quantities of propellant. In the short term, hydrogen production by coal gasification is the least expensive as well as the least flexible approach. On the other hand, SPS-powered electrolysis is the cleanest, least logistically complex, and most flexible technique potentially available.

The key transportation system related environmental concerns addressed during SPS feasibility and concept definition study phases have included:

- HLLV effluents in the lower and upper atmosphere
- HLLV acoustic emissions during launch and reentry
- Explosive hazards of HLLV propellants
- EOTV argon ion concentration in the magnetosphere

Although the sheer quantity of effluents introduced into the upper and lower atmosphere can result in some temporary changes in the atmospheric composition and properties, potential persistent and/or detrimental environmental effects have not been identified.

The development of a transportation system for the SPS poses a wide variety of technical and design challenges for the system designer. The vehicles are inherently very large—larger than any conventional aircraft envisioned today. This alone requires the development of advanced design and manufacturing techniques. Similarly, a postulated "return" to the airline operation concepts of earlier spacecraft development studies requires design for the near-elimination of post-flight refurbishment other than that required for refueling, payload installation, and mating.

A pacing technology at this point is the need to develop advanced high-temperature materials for reusable thermal protection systems for the HLLV that are an order of magnitude better than that employed on the STS. This calls for application of the advanced metallurgical technology available today. Coupled with the parallel development of thermostructural concepts to fully utilize advanced materials capability, a potential exists for significant accomplishments in vehicle thermostructural design.

The materials specified for the outer layers of the orbiter TPS must withstand an extreme thermal and stress environment. Those materials available today which can meet some of these requirements do not meet all of the desired criteria: coatings are subject to foreign object damage; embrittlement occurs after repeated exposure to high temperature environments reducing the physical strength of the material; the materials are heavy, costly, or in very short supply, etc.

All aspects of cryogenic tank design must be evaluated and resolved. These include the analysis of integral and non-integral tanks, insulation techniques, and operational utility.

So-called exotic or highly innovative new concepts in propulsion systems (i.e., multicycle air-breather engines or dual fuel liquid rocket engines) may also prove to be a pacing technology in advanced vehicle development—particularly in the area of reusability and expected life.

Transportation of the orbiter from the point of manufacture or alternate landing sites also requires early attention. Air-breathing engines are not incorporated in the vehicle in order to save weight, so the orbiter cannot operate in a ferry mode. Some form of an auxiliary propulsion system is necessary for the ferry mode since the development cost of a suitable carrier aircraft (i.e., SST concept) would very likely be prohibitively expensive. The design and operation of very large aircraft systems incur a new level of design analyses; such challenges have been met in the past as necessary in the cases of the B-29, the 747, and the C-5 aircraft and requires recognition of the large masses, inertias, and dimensions involved.

Since the EOTV solar array utilizes the same configuration, materials, and manufacturing processes as the satellite, common technology requirements are evident. The unique technology requirement is in the primary area of ion engine development. The key requirement is in large size (1.0×1.5 m), high current density (1000 A/m^2) thruster demonstration. Further analyses and demonstration testing of the "direct drive" concept, to minimize power processor weight and cost, are also required. The use of argon or another suitable propellant must be further evaluated and, a key issue is the feasibility of annealing the radiation damage incurred by the silicon solar array in transitioning of the Van Allen belt and/or further confirmation of the self-annealing properties of the GaAs solar array.

As a part of the SPS system definition effort, a workshop on SPS Space Transportation was held at Huntsville, Alabama on January 29-31, 1980. The Appendix to this report summarizes the results of the SPS Transportation Workshop.

1.0 EARTH-TO-ORBIT SYSTEMS

Evolving Satellite Power System (SPS) program concepts envision the assembly and operation of 60 solar-powered satellites in synchronous equatorial orbit over a period of 30 years. With each satellite weighing from 35 to 50 million kg, economic feasibility of the SPS is strongly dependent upon low-cost transportation of SPS elements. The minimum rate of delivery of SPS elements alone to LEO for this projected program is 70 million kg per year. This translates into as many as 350 flights per year, or approximately one flight per day, using a fleet of vehicles, each delivering a cargo of 200,000 kg.

The magnitude and sustained nature of this advanced space transportation program concept requires long-term routine operations somewhat analogous to commercial airline/airfreight operations. Ballistic vertical-takeoff, heavy-lift launch vehicles (e.g., 400,000-kg payload) can reduce the launch rate to less than 200 flights per year. However, requirements such as water recovery of stages with subsequent refurbishment, stacking, launch pad usage, and short turnaround schedules introduce severe problems for routine operations. The focus of attention has, therefore, been influenced in the direction of winged recoverable vehicle concepts. Three of the more promising configuration options evaluated, with varying payload capability, are summarized herein.

1.1 HLLV TWO-STAGE SERIES BURN (REFERENCE CONFIGURATION)

The launch configuration of the SPS series burn HLLV configuration is shown in Figure 1.1-1. This series burn concept uses 16 LCH_4/LO_2 engines on the booster and 14 standard SSME's on the orbiter. The LCH_4/LO_2 booster engines employ a gas generator cycle and provide a vacuum thrust of 9.79×10^6 newtons each. The SSME's on the orbiter provide a vacuum thrust of 2.09×10^6 newtons (100% power level). The nominal 100% power level for the SSME's was selected based on engine life considerations which indicated about a factor of 3 reduction in life if the 109% power level is used.

An airbreather propulsion system is provided on the booster for flyback capability. The reference wing area for both stages is:

$$S_W \text{ (Orbiter)} = 1446 \text{ m}^2 \text{ (15,560 ft}^2\text{)}$$

$$S_W \text{ (Orbiter)} = 2330 \text{ m}^2 \text{ (25,080 ft}^2\text{)}$$

Heat sink thermal protection system is provided on the booster and the Shuttle's Reusable Surface Insulation (RSI) is used on the orbiter.

The vehicle design weight characteristics are noted in Table 1.1-1. The net delivered payload is 424,000 kg. (An alternate configuration of 126,000 kg payload capacity was also evaluated, Section 1.4.) A return payload of 15% (63,500 kg) of the delivered payload was assumed for the orbiter entry and landing conditions. The resulting mass fraction is 0.875 for the booster and 0.841 for the orbiter.

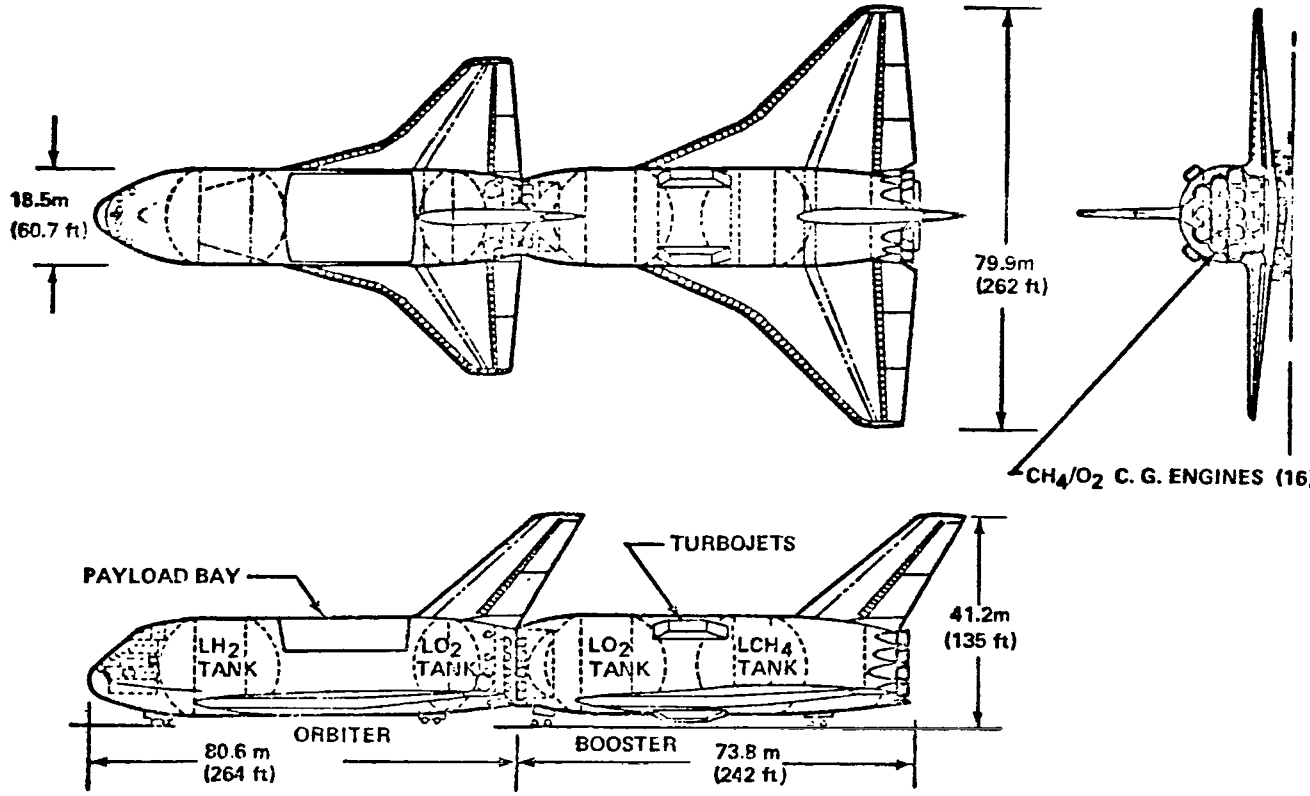


Figure 1.1-1. Two-Stage Winged SPS Launch Vehicle
(Fully Reusable Cargo Carrier)

Table 1.1-1. Two-Stage Winged Vehicle
Design Characteristics

	ORBITER		BOOSTER
GLOW		10,978,400	
BLOW	—		7,813.700
BOOSTER FUEL (LCH_4)	—		1,708,900
BOOSTER OXIDIZER (LO_2)	—		5,126,700
BOOSTER INERTS	—		978,100
OLOW-LESS PAYLOAD	2,740,700		—
ORBITER FUEL (LH_2)	329,400*		—
ORBITER OXIDIZER (LO_2)	1,976,200*		—
ORBITER INERTS	435,100		—
ASCENT PAYLOAD	424,000		—
RETURN PAYLOAD ~15%	63,500		—
MASS FRACTION	0.841		0.875
ENTRY WEIGHT			
- NO PAYLOAD	395,200		936,600
- WITH RETURN P/L	456,000		—
START CRUISE WEIGHT			
- NO PAYLOAD	—		932,900
- WITH RETURN P/L	—		—
LANDING WEIGHT			
- NO PAYLOAD	391,800		846,700
- WITH RETURN P/L	452,600		—

(ALL MASS DATA IN kg)

*MAINSTAGE + FLIGHT PERFORMANCE RESERVE

The vehicle ascent performance characteristics are noted in Table 1.1-2. A 3-g maximum acceleration thrust profile was used due to the manned capability and also to minimize the load conditions on the orbiter. The booster staging velocity of 2170 m/sec is well within the "heat sink" capability of the aluminum/titanium airframe.

Table 1.1-2. Ascent Performance Characteristics

First Stage			
T/W AT IGNITION	=	1.30	
MAXIMUM DYNAMIC PRESSURE	=	35.91 kPa	(750 psf)
MAXIMUM ACCELERATION	=	3.0 g	
STAGE BURN TIME	=	155.24 sec	
RELATIVE STAGING VELOCITY	=	2170 m/sec	(7,120 fps)
DYNAMIC PRESSURE AT STAGING	=	1.16 kPa	(24 psf)
Second Stage			
INITIAL T/W	=	0.94	
MAXIMUM ACCELERATION	=	3.0 g	
STAGE BURN TIME	=	350.24 sec	

The reentry characteristics for the booster and orbiter are noted in Table 1.1-3. The maximum deceleration for the booster is 4.27 g and the subsonic transition altitude is 17.86 km. The orbiter reentry has been limited to a normal load factor of 1.41 g until the subsonic transition which occurs at an altitude of 13.62 km.

Table 1.1-3. SPS Winged Vehicle Reentry Characteristics

BOOSTER	ORBITER
APOGEE CONDITIONS	
$h = 80.82$ km	
$V_{rel} = 1955$ m/sec	MAXIMUM DYNAMIC PRESSURE CONDITION
	$q = 13.17$ kpa
MAXIMUM DECELERATION CONDITION	$h = 15.55$ km
$q = 10.77$ kPa	$V_{rel} = 361$ m/sec
$h = 32.61$ km	NORMAL LOAD FACTOR = 1.41
$V_{rel} = 1327$ m/sec	
NORMAL LOAD FACTOR = 4.27 g's	SUBSONIC TRANSITION CONDITION
	$h = 13.62$ km
MAXIMUM DYNAMIC PRESSURE CONDITION	$\alpha = 6.4$ deg
$q = 13.29$ kPa	
$h = 22.96$ km	
$V_{rel} = 686$ m/sec	
NORMAL LOAD FACTOR = 1.49 g's	
SUBSONIC TRANSITION CONDITION	
$h = 17.86$ km	
$\alpha = 15$ deg	

The boost stage consists of the following subsystems:

- Structures
- Induced Environmental Protection
- Landing and Auxiliary Systems
- Ascent Propulsion

- Flyback Propulsion
- RCS Propulsion
- Prime Power
- Electrical Conversion and Distribution
- Hydraulic Conversion and Distribution
- Surface Controls
- Avionics
- Environmental Control

The booster stage structures subsystem consists of the wing, vertical tail, and body group. The body group consists of the nose section, oxidizer (LO_2) tank, intertank, fuel (LCH_4) tank, base skirt, thrust structure, aft body flap, and fairing structures. A preliminary sizing analysis was conducted to determine the individual structural element masses exclusive of heat sink requirements. The additional materials required to satisfy heat sink requirements are incorporated into the induced environmental protection subsystem. The wing box is constructed of 7075-T73 aluminum and the leading edge, trailing edge, and elevons are constructed of 6AL-4V titanium. A 4-g entry condition and a 2.5-g subsonic maneuver condition were considered in sizing the wing structure. A constant $t/c = 10\%$ was used. The wing mass is 129,700 kg. The vertical tail was sized for a boost max $q\beta$ condition of 177 kpa. The box structure is 7075-T73 aluminum and the remaining tail structure is 6AL-4V titanium. The mass of the vertical tail is 14,000 kg. The nose section consists of a fixed shell structure plus a deployable nose cap. The shell structure experiences maximum compressive loading of 35,200 N/cm forward and 24,000 N/cm aft during the boost 3-g condition. The smeared thickness of the 7075 aluminum skin-stringer panels is 0.82 cm forward and 0.68 cm aft. The smeared thickness of the 7075 aluminum nose cap is 0.38 cm. The nose section mass is 26,800 kg.

The oxidizer tank is an all welded 2219-T87 aluminum pressure vessel with integral sidewall stiffening in the cylindrical section. The smeared thickness of the sidewall panels varied from 0.79 cm forward to 0.93 cm aft. The dome membrane thickness varies between 0.28 cm and 0.40 cm for the upper dome and between 0.47 cm and 0.81 cm for the lower dome. The tank mass including slosh baffles is 36,100 kg. The intertank is approximately 18.5 m long and is constructed of 7075 aluminum. The intertank experiences a maximum compressive loading of 30,160 N/cm at the boost 3-g onset condition. The smeared thickness of the skin-stringer panels is 0.76 cm. The mass of the intertank, which incorporates the airbreather engine support structures, is 38,000 kg. The fuel tank is an all-welded 2219-T87 aluminum pressure vessel with integral sidewall stiffening in the cylindrical section. The smeared thickness of the sidewall panels is 0.89 cm. The dome membrane thickness varies between 0.28 cm and 0.40 cm for the upper dome and between 0.28 and 0.46 cm for the lower dome. The tank mass including slosh baffles is 32,600 kg. The base skirt is approximately 19.7 m long and is constructed of 7075 aluminum. The upper 14.4 m experiences maximum compressive loadings of 40,000 N/cm forward and 44,500 N/cm aft at the boost 3-g onset condition. The smeared thickness of the skin-stringer panels is 0.88 cm forward and 0.94 cm aft. The lower 5.3 m experiences a maximum combined compressive loading of 31,100 N/cm and shear flow of 18,900 N/cm during the tanked pre-ignition condition. The smeared thickness of the skin-stringer panels is 1.50 cm in the shear-out region and 0.64 cm outside the shear-out region. The base skirt mass is 47,200 kg.

The thrust structure consists of four major beam assemblies plus interbeam stabilizing members. Sixteen thrust posts are incorporated into the beam assemblies; 7075 aluminum is used throughout. The structural elements are sized for the ignition condition using a dynamic magnification factor of 1.25. Shear flows in the individual plates vary from 15,300 N/cm to 61,300 N/cm and the web plate thicknesses vary from 0.46 cm to 1.85 cm. The average cross area of a thrust post is 186 cm^2. The thrust structure mass is 23,900 kg.

The constant chord body flap provides the booster stage with pitch trim control and thermally shields the main engines during entry. The flap is constructed of 6AL-4V titanium and has a mass of 2100 kg. Fairing structures consist of the wing-to-body fairings located both forward and aft of the box carry-through section, the tail-to-body fairing, and the engine shroud/base region fairings. The fairings are constructed of 6AL-4V titanium and have an estimated mass of 8500 kg.

The induced environmental protection subsystem consists of the heat sink additions required to maintain the airframe outer skin within acceptable temperature limits, plus the base heat shield. Reusable Surface Insulation is used for thermal protection on the base heat shield. The heat sink additions weigh 38,300 kg and the base heat shield 8100 kg for a total system mass of 46,400 kg.

In addition to landing gear, a landing drag device and auxiliary systems for upper stage separation and nose cap deployment/latching are included. The landing gear weight is estimated at 3.2% of design landing weight. Total subsystem mass is 34,500 kg.

The ascent propulsion subsystem consists of the main engines, engines, accessories, gimbal provisions, and the fuel and oxidizer systems. Main propulsion is provided by 16 high pressure LO_2/LCH_4 gas generator cycle engines and the associated tank pressurization and propellant delivery system. The following engine characteristics were used in the analysis:

• Propellant	LO_2/LCH_4
• Chamber Pressure	34,500 kpa
• Area Ratio	60:1
• Mixture Ratio	3:1
• Thrust (S.L./Vac.)	8.76×10^6N/9.68×10^6N
• Specific Impulse (S.L./Vac.)	318.5 sec/352 sec

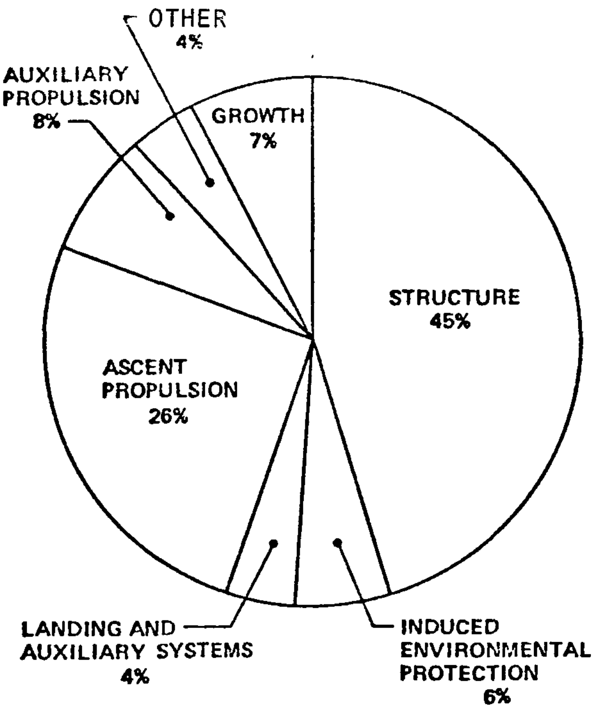
The mass of the 16 engines and associated accessories plus gimbal provisions (for 11 engines) is 162,400 kg. Pressurization gases are heated GO_2 for the LO_2 tank and heated GCH_4 for the LCH_4 tank. The total mass of the tank pressurization and propellant delivery systems is 42,200 kg.

The flyback propulsion subsystem consists of the airbreather engines, accessories, fuel system, tankage, and engine installation nacelles, ducts, and doors. Flyback thrust is provided by 12 turbojet engines, each having a S.L. static thrust of 356,000 N. The flyback fuel is RP-1. The dry mass of the subsystem is 57,400 kg.

The remaining subsystem masses have been estimated using historical or Shuttle predicted weights. These subsystems include RCS propulsion, prime power, electrical conversion and distribution, hydraulic conversion and distribution, aerosurface controls, avionics, and environmental control.

The reaction control system is required for stage orientation prior to entry and for control during entry. The subsystem dry mass is 5100 kg. Major power sources consist of batteries and airbreather engine driven generators for electrical power, and a hydrazine powered APU for hydraulic power. The subsystem mass is 4300 kg. The power conversion, conditioning, and cabling elements mass is 4200 kg. The stage functions requiring hydraulic power are serviced by the hydraulic conversion and distribution subsystem. The hydraulic power for rocket engine thrust vector control and valve actuation is included in the subsystem mass of 10,900 kg. The actuation system for the aerodynamic control surfaces is 10,300 kg. The avionics subsystem includes elements for guidance, navigation and control, tracking, instrumentation, and data processing and software. The subsystem mass is 1500 kg. The environmental control subsystem maintains a conditioned thermal environment for the avionics. The subsystem mass is 200 kg.

The flyback booster mass characteristics are shown in Figure 1.1-2. The structure, induced environment protection, ascent and auxiliary propulsion, and landing subsystems account for 89% of the dry mass. The induced environment protection subsystem mass includes the additional structural thickness required for the "heat sink capability" and the base heat shield.



DRY MASS BREAKDOWN

	MASS (kg)
STRUCTURE	360 800
INDUCED ENVIRONMENTAL PROTECTION	46 400
LANDING AND AUXILIARY SYSTEMS	34 500
ASCENT PROPULSION	204 600
AUXILIARY PROPULSION	60 600
PRIME POWER	4 300
ELECTRICAL CONVERSION AND DISTRIBUTION	4 200
HYDRAULIC CONVERSION AND DISTRIBUTION	10 900
SURFACE CONTROLS	10 300
AVIONICS	1 500
ENVIRONMENTAL CONTROL	200
GROWTH	58 600
DRY MASS =	796 900
RESIDUALS AND RESERVES	49 800
LANDING MASS =	846 700
LOSSES DURING FLYBACK	86 200
START FLYBACK MASS =	932 900
ENTRY IN-FLIGHT LOSSES	3 700
START ENTRY MASS =	936 600
IN-FLIGHT LOSSES PRIOR TO ENTRY	27 000
STAGING MASS =	963 600
THRUST DECAY PROPELLANT	14 500
INERT MASS =	978 100

Figure 1.1-2. Booster Mass Statement

The orbiter consists of the following subsystems:

- Structures
- Induced Environmental Protection
- Landing and Auxiliary Systems
- Ascent Propulsion
- OMS Propulsion
- RCS Propulsion
- Prime Power
- Electrical Conversion and Distribution
- Hydraulic Conversion and Distribution
- Surface Controls
- Avionics
- Environmental Control
- Personnel Provisions
- Personnel
- Payload Accommodations

The orbiter structures subsystem consists of the wing, vertical tail, and body group. The body group consists of the nose section, crew module, fuel (LH_2) tank, intertank, payload bay doors, oxidizer (LO_2) tank, aft skirt, thrust structure, aft body flap, and fairing structures. A preliminary sizing analysis was conducted to determine the individual structural element masses. The wing is constructed from 6AL-4V titanium. A 2.5-g entry condition and a 2.5-g subsonic maneuver condition were considered in sizing the wing structure. A constant t/c = 10% was used. The wing mass is 51,800 kg. The vertical tail was sized for a boost max $q\beta$ condition of 177 kpa. It is constructed of 6AL-4V titanium. The mass of the vertical tail is 12,300 kg. The nose section is constructed of 6AL-4V stiffened sandwich construction. Included in the nose section are the exterior windshields and the nose landing gear support bulkhead, wheel well and doors. The titanium sandwich is 3 cm thick and has a smeared thickness of 0.13 cm. The total mass of the nose section is 9200 kg. The crew module is an all-welded 2219-T87 aluminum pressure-tight vessel with integral stiffening. Included in the crew module are the interior (redundant) windshields, hatches for ingress and egress, and support provisions for other subsystem elements located within the module. The module accommodates a four-man flight crew plus a six-man passenger group. The crew module is 2800 kg.

The fuel tank is an all welded 6AL-4V titanium sandwich pressure vessel. The core thickness is 3 cm. The smeared thickness of the sidewall sandwich is 0.41 cm. The dome sandwich smeared thickness varies between 0.21 cm and 0.26 cm for the upper dome and between 0.22 cm and 0.28 cm for the lower dome. The tank mass is 21,200 kg. The intertank is constructed primarily of 6AL-4V titanium sandwich. It provides support for second stage payloads and the payload bay doors. The smeared thickness of the sidewall sandwich varies from 0.13 cm to 0.25 cm. The intertank mass is 25,900 kg.

The payload bay door is 24 meters long and has a surface area of 553 m^2. It consists of two panels that open at the upper centerline. Each panel consists of four equal length segments. The forward 6-m segment incorporates deployable radiators. The door primary structure is of honeycomb and frame construction employing composite materials and has a mass of 5100 kg.

The oxidizer tank is an all welded 2219-T87 aluminum pressure vessel consisting of two elliptical domes. The dome membrane thickness varies between 0.53 cm and 0.63 cm for the upper dome and between 0.62 cm and 1.00 cm for the lower dome. The tank mass including slosh baffles is 20,300 kg.

The aft skirt is approximately 12.2 m long and is constructed of 7075 aluminum. The skirt experiences maximum compressive loading of 26,200 N/cm forward and 33,800 N/cm aft during the booster 3-g condition. The smeared thickness of the skin-stringer panels is 0.71 cm forward and 0.82 cm aft. The aft skirt mass is 19,600 kg.

The thrust structure consists of an internal cone frustum with a cruciform beam system at its lower end. Ten thrust posts are incorporated into the lower section of the cone frustum and four thrust posts are incorporated into the cruciform beam system. A combination 7075 aluminum/6AL-4V titanium structure is used. The structural elements are sized for the ignition condition using a dynamic magnification factor of 1.25. The average compressive loading in the upper section of the cone frustum is 12,900 N/cm and the average smeared thickness of the aluminum skin panel is 0.49 cm. The average cross section area of a titanium thrust post is 23 cm^2. The thrust structure mass is 10,100 kg.

The constant chord body flap provides the orbiter with pitch trim control and thermally shields the main engines during entry. The flap is an aluminum structure with honeycomb skin panels. The flap mass is 640 kg.

Fairing structures consist of a forward wing-to-body fairing located in the transition region between the circular fuel tank and the "boxy" intertank, a wing-to-body fairing located under the lower half of the circular aft skirt, and a tail-to-body fairing. The fairings are aluminum structures with honeycomb skin panels. The total mass of the fairings is 3960 kg.

The induced environmental protection subsystem consists of (1) Reusable Surface Insulation (RSI) on the exterior surfaces of the wing, tail, and body, (2) a base heat shield incorporating RSI, (3) internal insulation for thermal control of pertinent components, and (4) purge, vent, and drain provisions. The masses of the foregoing are 44,800 kg, 1400 kg, 1100 kg, and 100 kg, respectively, yielding a total subsystem mass of 48,300 kg.

The landing and auxiliary subsystems includes the landing gear and payload handling manipulator arms. The landing gear weight is estimated at 3.2% of design landing weight. Total subsystem mass is 15,800 kg.

The ascent propulsion subsystem consists of the main engines, accessories, gimbal provisions, and the fuel and oxidizer systems. Main propulsion is provided by 14 standard SSME's and the associated tank pressurization and propellant delivery systems. The following engine characteristics were used in the analysis:

• Propellant	LO_2/LH_2
• Chamber pressure	20,700 kpa
• Area ratio	77.5:1
• Mixture ratio	6:1
• Specific impulse (vac)	473 sec

The mass of the 14 engines and associated accessories plus gimbal provisions (for 10 engines) is 43,540 kg.

Pressurization gases are heated GO_2 for the LO_2 tank and heated GH_2 for the LH_2 tank. The dry mass of the tank pressurization and propellant delivery system is 17,260 kg.

The orbital maneuver system consists of four ASE engines and accessories, and associated tank pressurization and propellant delivery and storage elements. The following engine characteristics were used in the analysis:

• Propellant	LO_2/LH_2
• Chamber pressure	13,800 kpa
• Area ratio	100:1/400:1
• Mixture ratio	6:1
• Thrust (vac)	89,000 N
• Specific impulse (vac)	473 sec

The mass of the four engines and accessories is 770 kg.

Tank pressurization is provided by a high-pressure low-temperature helium gas system. The dry mass of the tank pressurization and propellant delivery and storage elements is 483 kg.

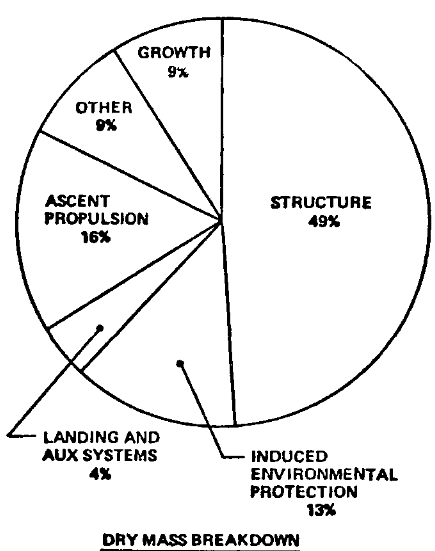
The remaining subsystem masses have been estimated using historical or Shuttle predicted weights. These subsystems include RCS propulsion, prime power, electrical conversion and distribution, hydraulic conversion and distribution, aerosurfaces controls, avionics, environmental control, personnel provisions, personnel and payload accommodations. The reaction control system provides for stage orientation on-orbit and prior to entry, and for control during entry. The subsystem dry mass is 3900 kg. Major power sources consist of an O_2/H_2 powered fuel cell subsystem to provide electrical power, and a hydrazine powered APU subsystem to provide hydraulic power. The dry mass of the prime power subsystem is 2500 kg. The power conversion, conditioning and cabling elements mass is 4800 kg. All stage functions requiring hydraulic power are serviced by the hydraulic conversion and distribution subsystem. The hydraulic power for rocket engine thrust vector control and valve actuation is included. The subsystem mass is 3600 kg. The actuation systems for the aerodynamic control surfaces and cockpit controls subsystem mass is 6800 kg.

The avionics subsystem includes elements for guidance, navigation and control, communications and tracking, displays and controls, instrumentation, and data processing and software with a subsystem mass of 2400 kg.

The environmental control subsystem maintains a habitable environment for the crew and passengers, and a conditioned thermal environment for the avionics. It provides the basic life support functions for the crew and passengers, and thermal control for several subsystems. It also provides for airlock pressurization. The subsystem mass including closed loop fluids is 2400 kg.

The fixed life support system and personnel accommodations for the 4-man flight crew is estimated to be 500 kg. The 4-man flight crew, their gear and accessories are 1200 kg. Removable payload support equipment mass allowance is 2900 kg.

The orbiter mass characteristics are shown in Figure 1.1-3. Structure accounts for approximately 50% of the study dry mass. The ascent propulsion and thermal protection subsystems are an additional 29% of the dry mass. The dry mass is 86% of the inert mass with the remainder including residuals and reserves, personnel and payload accommodations, and inflight losses.



	MASS (kg)
STRUCTURE	182,900
INDUCED ENVIRONMENTAL PROTECTION	48,300
LANDING AND AUX SYSTEMS	15,800
ASCENT PROPULSION	60,800
AUXILIARY PROPULSION	9,500
PRIME POWER	2,500
ELECTRICAL CONVERSION AND DISTRIBUTION	4,800
HYDRAULIC CONVERSION AND DISTRIBUTION	3,600
SURFACE CONTROLS	6,800
AVIONICS	2,400
ECLSS AND PERSONNEL PROV	2,900
GROWTH	32,900
DRY MASS =	373,200
PERSONNEL AND PAYLOAD ACCOMMODATIONS	4,100
RESIDUAL AND RESERVES	14,500
LANDING MASS =	391,800
ENTRY IN-FLIGHT LOSSES	3,400
START ENTRY MASS =	395,200
IN-FLIGHT LOSSES PRIOR TO ENTRY	39,900
INERT MASS =	435,100

Figure 1.1-3. Orbiter Mass Statement

1.2 HLLV TWO-STAGE PARALLEL BURN (ALTERNATE CONCEPT)

A parallel burn vertical-takeoff/horizontal-landing, heavy-lift launch vehicle (VTO/HL HLLV) concept has been evaluated as a candidate for SPS cargo and personnel transport to low earth orbit (LEO). Two vehicle payload capability options were synthesized—one with a payload capability of approximately 227,000 kg (500,000 lb), and the other 113,500 kg (250,000 lb), discussed in Section 1.4. Basic ground rules and assumptions employed in vehicle sizing are summarized in Table 1.2-1. Both stages have flyback capability to the launch site; the second stage is recovered in the same manner as the Shuttle Transportation System (STS) orbiter.

Table 1.2-1. HLLV Sizing—Ground Rules/Assumptions

- Two-stage vertical takeoff/horizontal landing (VTO/HL)
- Flyback capability both stages—ABES first stage only
- Parallel burn with propellant crossfeed
- LOX/RP first stage; LOX/LH_2 second stage
- High P_c gas generator cycle engine—first stage [I_s(vac) = 352 sec]
- High P_c staged combustion engine—second stage [I_s(vac) = 466 sec]
- Staging velocity—heat sink booster compatible
- Circa 1990 technology base—BAC/MMC weight reduction data
- Orbital parameters—487 km @ 31.6°
- Thrust/weight—1.30 liftoff/3.0 max
- 15% weight growth allowance/0.75% ΔV margin

The vehicle utilizes a parallel burn mode with propellant cross-feed from the first-stage tanks to the second-stage engines. The first stage employs high chamber pressure gas generator cycle LOX/RP fueled engines with LH_2 cooling, and the second stage employs a staged combustion engine similar to the Space Shuttle main engine (SSME) which is LOX/LH_2 fueled.

Although trade studies were conducted, a vehicle staging velocity compatible with a heat sink booster concept is considered desirable from an operations standpoint. Technology growth consistent with the 1990 time period was used to estimate weights and performance. The expected technology improvements are summarized in Table 1.2-2. Orbital parameters are consistent with SPS LEO base requirements, and the thrust-to-weight limitations are selected to minimize engine size and for crew/passenger comfort. Growth margins of 15% in inert weight and 0.75% in propellant reserves were established.

HLLV performance was determined by the use of a modified STS scaling and trajectory program. The engine performance parameters used in the analysis are given in Table 1.2-3.

In addition to pertinent trade studies (i.e., propellant type and loading, engine throttling, staging velocity, etc.) several technical issues were addressed; these included vehicle flight characteristics, ascent control analyses, thrust load distribution and structural requirements, and a preliminary thermal/structural assessment. The latter studies were performed with the lighter payload HLLV option.

Table 1.2-2. Technology Advancement

Body structure	17%
Wing structure	15%
Vertical tail	18%
Canard	12%
Thermal protection system	20%
Avionics	15%
Environmental control	15%
Reaction control system	15%
Rocket engines	
First stage thrust/weight = 120	
Second stage thrust/weight = 80	

Table 1.2-3. Engine Performance Parameters

Engine	Specific Impulse (sec) Sea Level	Specific Impulse (sec) Vacuum	Mixture Ratio	Thrust/Weight
LOX/RP GG Cycle	329.7	352.3	2.8:1	120
LOX/CH_4 GG Cycle	336.9	361.3	3.5:1	120
LOX/LH_2 Staged Comb.	337.0	466.7	6.0:1	80

A PB/VTO/HL HLLV configuration is shown in Figure 1.2-1 in the launch configuration. As shown, both stages have common body diameter, wing and vertical stabilizer; however, the overall length of the second stage (orbiter) is approximately 5 m greater than the first stage (booster). The vehicle gross

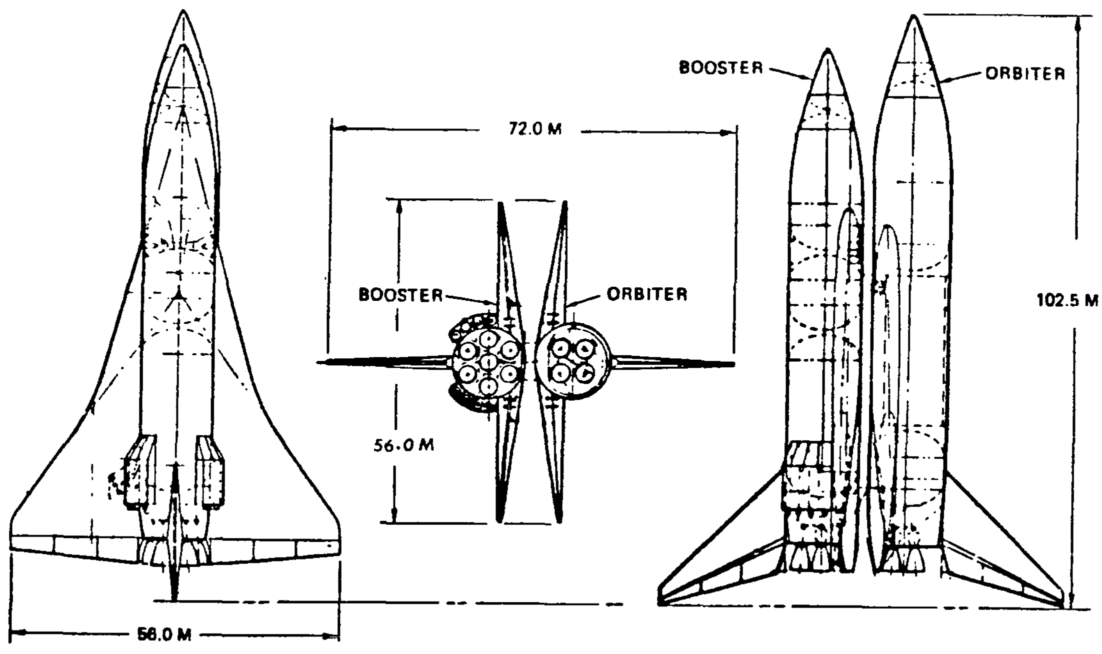


Figure 1.2-1. PB/VTO/HL HLLV Launch Configuration

liftoff weight (GLOW) is 7.14 million kg with a payload capability of 230,000 kg to the reference earth orbit. A summary weight statement is given in Table 1.2-4. The propellant weights indicated are total loaded propellant (i.e., not usable). The second-stage weight (ULOW) includes the payload weight. During the booster ascent phase, the second-stage LOX/LH_2 propellants are cross-fed from the booster to achieve the parallel burn mode. Approximately 730,000 kg of propellant are cross-fed from the booster to the orbiter during ascent.

Table 1.2-4. HLLV Mass Properties ($\times10^{-6}$)

	KG	LB
GLOW	7.14	15.73
BLOW	4.92	10.84
W_{P1}	4.49	9.89
ULOW	2.22	4.89
W_{P2}	1.66	3.65
PAYLOAD	0.23	0.51

The HLLV booster, shown in the landing configuration in Figure 1.2-2, is approximately 92 m in length with a wing span of 56 m and a maximum clearance height of 35 m; the nominal body diameter is 18 m. The vehicle has a dry weight of 450,000 kg. Seven high P_c gas generator driven LOX/RP engines are mounted in the aft fuselage with a nominal sea-level thrust of 10.2 million newtons each. Eight turbojet engines are mounted on the upper portion of the

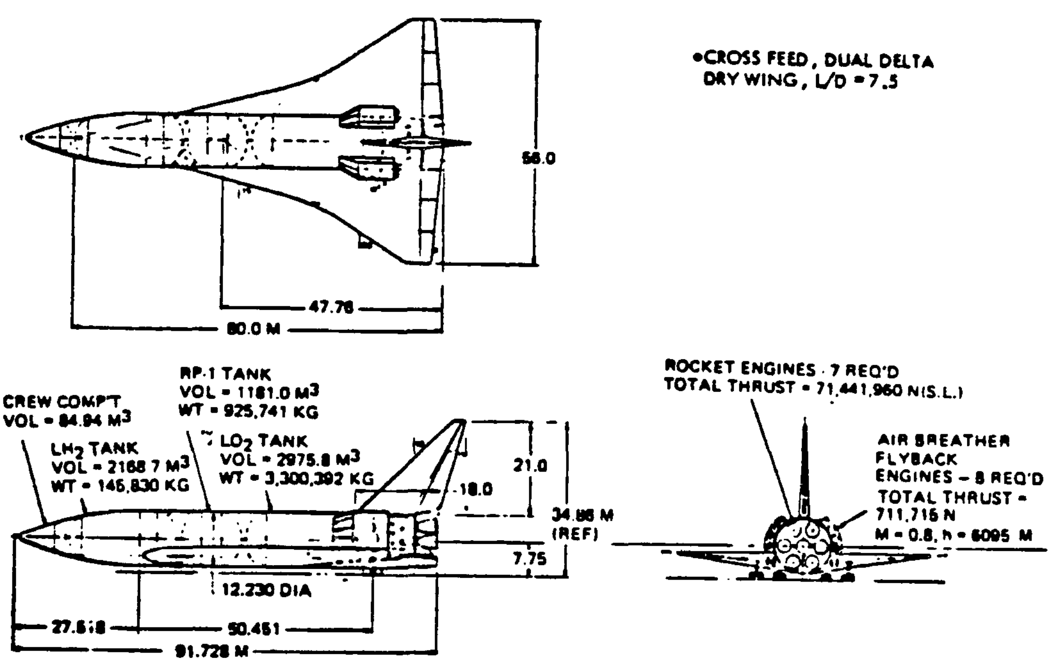


Figure 1.2-2. HLLV First Stage (Booster)—Landing Configuration

aft fuselage with a nominal thrust of 89,000 newtons each. A detailed weight statement is given in Table 1.2-5. The vehicle propellant weight summary is projected in Table 1.2-6.

Table 1.2-5. HLLV Weight Statement ($\times 10^{-3}$)

	Second Stage		First Stage	
	lb	kg	lb	kg
Fuselage	227.98	103.41	288.22	130.73
Wing	86.41	39.20	172.34	78.17
Vertical tail	12.57	5.70	15.89	7.21
Canard	3.07	1.39	4.87	2.21
TPS	115.94	52.59	-	-
Crew compartment	28.00	12.70	**	**
Avionics	8.50	3.86	7.50	3.40
Personnel	3.00	1.36	**	**
Environmental	5.70	2.59	**	**
Prime power	12.00	5.44	**	**
Hydraulic system	8.50	3.86	**	**
Ascent engines	59.38	26.93	148.70	67.45
RCS system	21.15	9.59	**	**
Landing gears	40.51	18.38	**	**
Propulsion systems	*	*	99.18	44.99
Attach and separation	-	-	10.12	4.59
APU	-	-	2.00	0.91
Flyback engines	-	-	62.95	28.55
Flyback propulsion system	-	-	40.54	18.39
Subsystems	-	-	56.80	25.76
Dry weight	632.71	286.99	909.12	-
Growth margin (15%)	94.91	43.05	136.37	-
Total inert weight	727.62	330.04	1045.49	-

*Included in fuselage weight
**Items included in subsystem

Table 1.2-6. HLLV Propellant Weight Summary ($\times 10^{-6}$)

	First Stage		Second Stage	
	kg	lb	kg	lb
Usable	4.358	9.607	1.579	3.481
Crossfeed	0.732	1.612	(0.731)	(1.612)
Total burned	3.626	7.995	2.310	5.093
Residuals	0.018	0.040	0.009	0.020
Reserves	0.020	0.045	0.011	0.024
RCS	0.005	0.010	0.008	0.018
On orbit	-	-	0.043	0.095
Boiloff	-	-	0.005	0.010
Flyback	0.085	0.187	-	-
Total loaded	4.486	9.889	1.655	3.648

The HLLV orbiter is depicted in Figure 1.2-3. The vehicle is approximately 97 m in length with the same wing span, vertical height, and nominal body diameter as the booster. The orbiter employs four high P_c staged combination LOX/LH_2 rocket engines with a nominal sea-level thrust of 5.3 million newtons each.

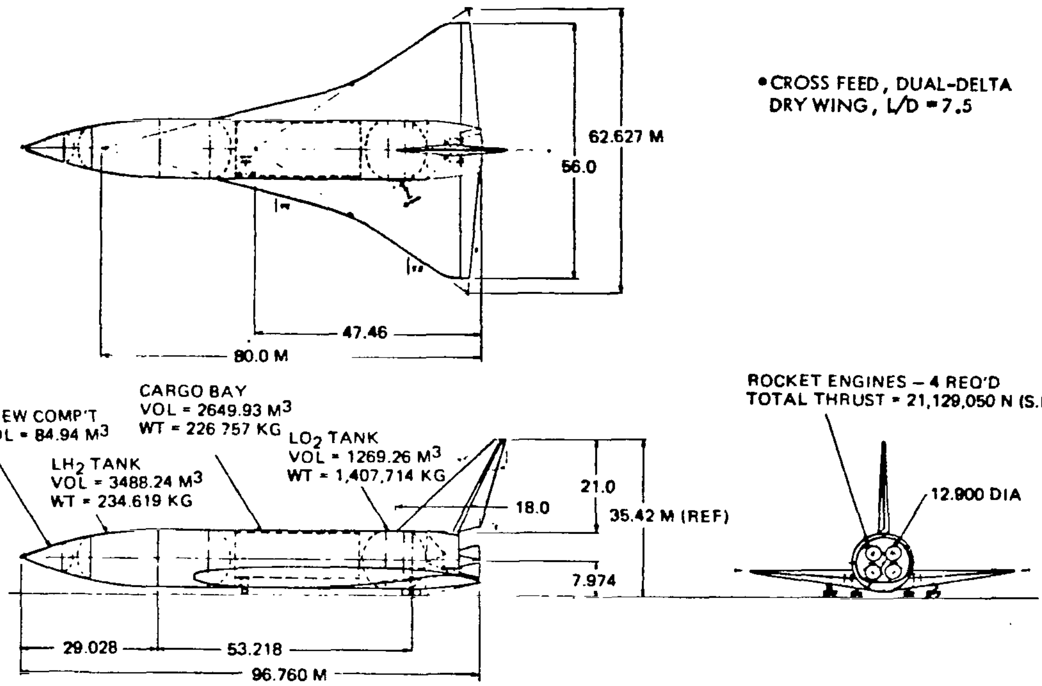


Figure 1.2-3. HLLV Second Stage (Orbiter) —Landing Configuration

The cargo bay is located in the mid-fuselage in a manner similar to the STS orbiter and has a length of approximately 27.5 m. The detailed weight statement and a propellant summary for the orbiter are included in Tables 1.2-5 and 1.2-6, respectively.

The HLLV performance has been determined by using a modified STS scaling and trajectory program. The vehicle can deliver a payload of approximtely 231,000 kg to an orbital altitude of 487 km at an inclination of 31.6°.

The vehicle relative staging velocity is 2127 m/sec (6987 ft/sec) at an altitude of 55.15 km (181,000 ft) and a first-stage burnout range of 88.7 km (48.5 nmi). The first-stage flyback range is 387 km (211.8 nmi). For this HLLV configuration, all engine throttling to limit maximum dynamic pressure during the parallel burn mode is accomplished with the first or booster stage engines only (i.e., second-stage engines operate at 100% rated thrust during boost).

1.3 HLLV SINGLE STAGE TO ORBIT (HTO/SSTO)—HIGH TECHNOLOGY ALTERNATE

The HTO-SSTO is a most advanced concept and, consequently, a higher technology risk option. This concept adapts existing and advanced commercial and/or military air transport system concepts, operations methods, maintenance procedures, and cargo handling equipment. The principal operational objective is to provide economic, reliable transportation of large quantities of material between earth and LEO at high flight frequencies with routine logistics operations and minimal environmental impact. An associated operational objective is to reduce the number of operations required to transport material and equipment from their place of manufacture on earth to low earth orbit. (Since this study was conducted under company discretionary funds and existing computer programs, some of the units in tables and figures have not been converted to the metric system.)

Some of the key operational features are:

- Single orbit up/down from/to the same launch site (at any launch azimuth subject to payload/launch azimuth match)
- Capable of obtaining equatorial orbit
- Takeoff and land on standard commercial or military runways
- Simultaneous multiple launch capability
- Total system recovery
- Self-ferry capability from manufacturing site to launch site
- Amenable to alternate launch/landing sites
- Incorporates Air Force (C-5A Galaxy) and commercial (747 cargo) payload handling, including rail, truck, and cargo-ship containerization concepts, modified to meet space environment requirements
- Swing-nose loading/unloading, permitting standard aircraft loading concepts
- Systems servicing with existing support equipment on runway aprons or service hangars

The HTO-SSTO utilizes a tri-delta flying wing concept, consisting of a multi-cell pressure vessel. The Whitcomb airfoil section offers an efficient aerodynamic shape for obtaining a high propellant volumetric efficiency. LH_2 and LO_2 tanks are located in each wing near the vehicle c.g., and extend from the root rib to the wing tip, Figure 1.3-1. In the aft end of the vehicle, three LOX/LH_2 high P_c rocket engines are attached with a double-cone thrust structure to a two-cell LH_2 tank.

Most of the cargo bay side walls are provided by the root-rib bulkhead of the LH_2 wing tank. The cargo bay floor is designed similar to the C-5A military transport aircraft. The top of the cargo bay is a mold-line extension of the wing upper contours, wherein the frame inner caps are arched to resist pressure. The forward end of the cargo bay provides a circular seal/locking mechanism to the forebody. Cargo is deployed in orbit by swinging the forebody to 90 or more degrees about a vertical axis and transferring cargo from the bay on telescoping rails.

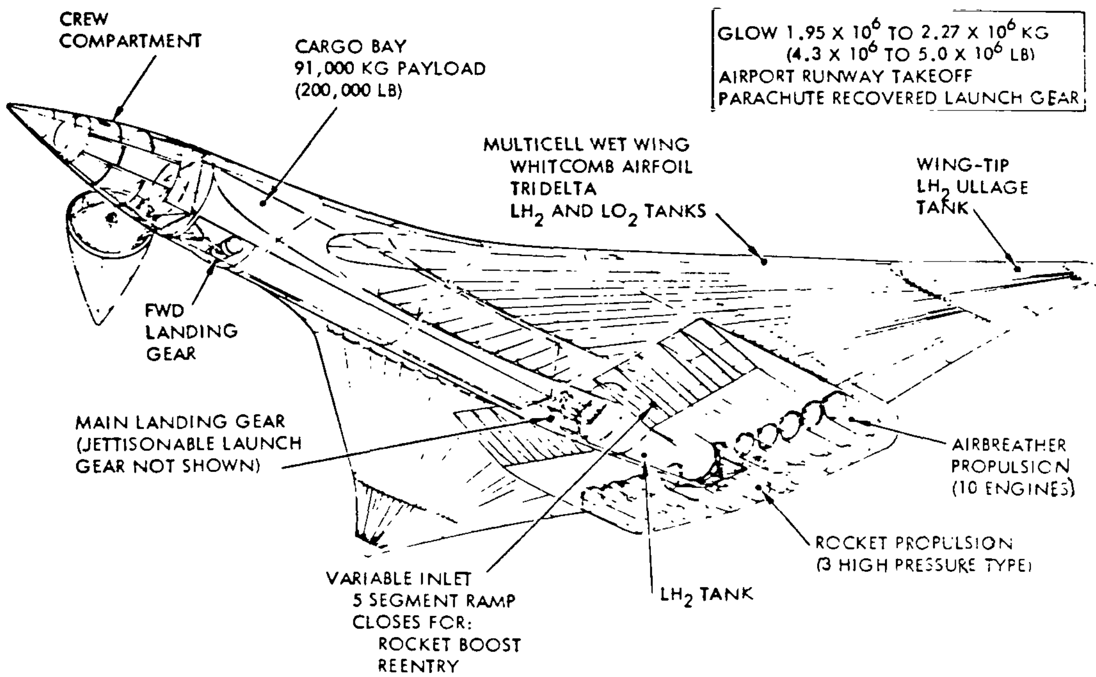


Figure 1.3-1. HTO-SSTO Design Features

The forebody is an ogive of revolution with an aft dome closure. The ogive is divided horizontally into two levels. The upper level provides seating for crew and passengers, as well as the flight deck. The lower compartment contains electronic, life support, power, and other subsystems including spare life support and emergency recovery equipment.

Ten high-bypass, supersonic-turbofan/airturbo-exchanger/ramjet engines with a combined static thrust of 6.68 MN are mounted under the wing. The inlets are variable area retractable ramps that also close and fair the bottom into a smooth surface during rocket-powered flight and for high angle-of-attack ballistic reentry. Figure 1.3-2 is an inboard profile of the vehicle, illustrating some of the details of vehicle construction.

Figure 1.3-3 presents details of the multi-cell structure of the wing. The upper figure illustrates the application of Shuttle-type RSI tile thermal protection system (TPS). The lower figure shows a potential utilization of a "metallic" TPS.

The wing is an integrated structural system consisting of an inner multi-cell pressure vessel, a foam-filled structural core, an inner facing sheet, a perforated structural honeycomb core, and an outer facing sheet. The inner

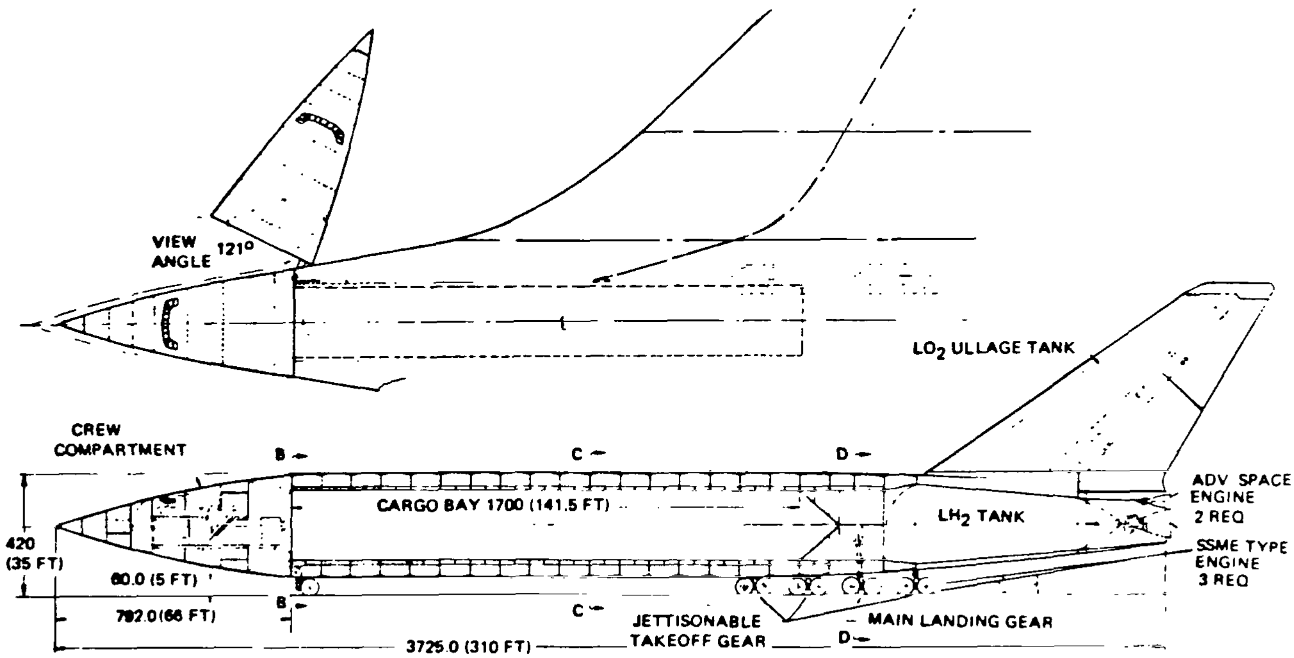


Figure 1.3-2. HTO-SSTO Inboard Profile

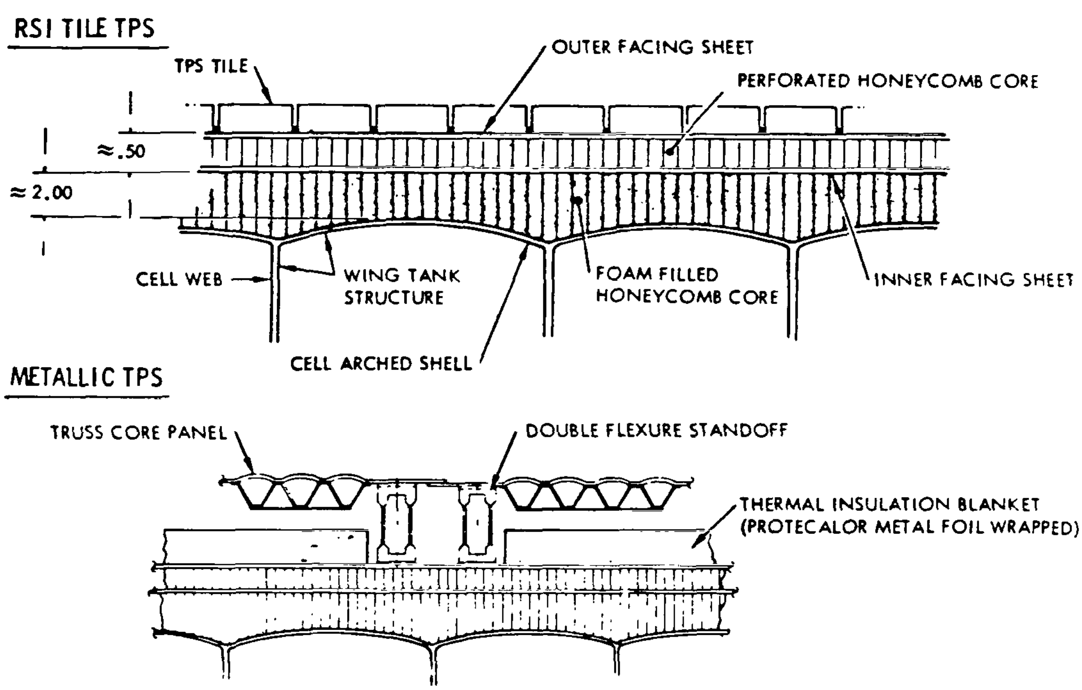


Figure 1.3-3. Wing Construction Detail with Candidate TPS Configurations

multi-cell pressure vessel arched shell and webs are configured to resist pressure. The pressure vessel and the two facing sheets, which are structurally interconnected with phenolic-impregnated glass fiber, honeycomb core, resist wing spanwise and chordwise bending moments. Cell webs react winglift shear forces. Torsion is reacted by the pressure vessel and the two facing sheets as a multi-box wing structure.

The outer honeycomb core is perforated and partitioned to provide a controlled passage, purge, and gas-leak detection system in addition to the function of structural interconnect of the inner and outer facing sheets.

The proposed multi-cycle airbreathing engine system, Figure 1.3-4, is derived from the General Electric CJ805 aircraft engine, the Pratt and Whitney SWAT-201 supersonic wraparound turbofan/ramjet engine, the Aerojet Air Turborocket, Marquardt variable plug-nozzle, ramjet engine technology, and Rocketdyne tubular-cooled, high-P_c rocket engine technology. The development of a multi-cycle engine of this type would require a most ambitious technology advancement program.

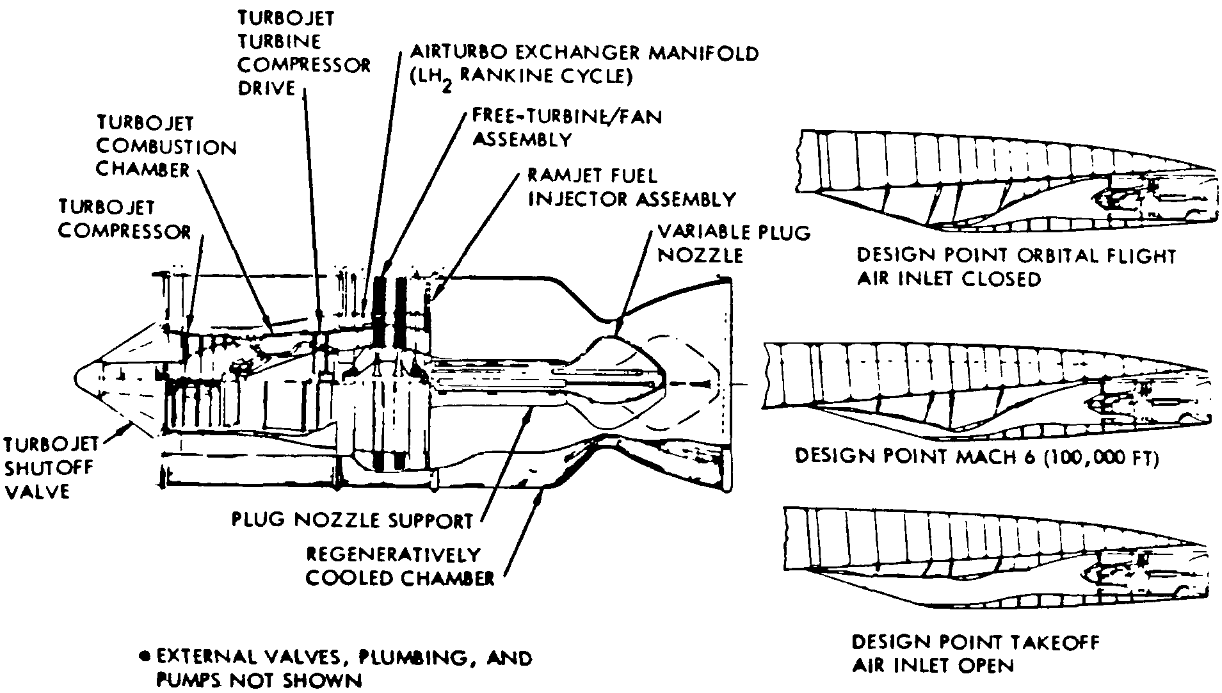


Figure 1.3-4. Multi-Cycle Airbreathing Engine and Inlet, Turbofan/Air-Turboexchanger/Ramjet

The multi-mode power cycles include: an aft-fan turbofan cycle, an LH_2 regenerative Rankine air-turboexchanger cycle; and a ramjet cycle that can also be used as a full-flow (turbojet core and fan bypass flow) thrust-augmented turbofan cycle. These four thermal cycles may receive fuel in any combination permitting high engine performance over a flight profile from sea-level takeoff to Mach 6 at 30-km altitude.

The engine air inlet and duct system is based on a five-ramp variable inlet system with actuators to provide ramp movement from fully closed (upper RH figure) for rocket-powered and reentry flight, to fully open (lower RH figure) for takeoff and low altitude/Mach number operation.

The inlet area was determined by the engine airflow required at the Mach 6 design point. The configuration required 6.68 MN thrust at the Mach 6 condition, and at least 5.8 MN for takeoff. This resulted in an inlet area of approximtely 10.5 m^2 for a 10-engine configuration. In order to provide pressure recovery with minimum spillage drag over the wide range of Mach numbers, the variable multi-ramp inlet is required. Estimated engine thrust (total of 10 engines) vs. velocity is given in Figure 1.3-5 in pounds.

*1 lb = 4.45 N

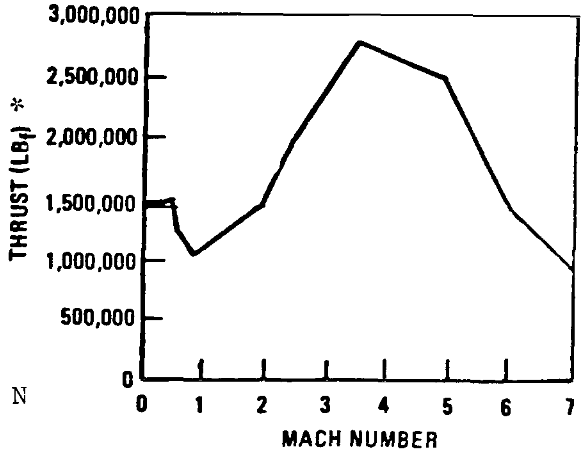


Figure 1.3-5. Airbreather Thrust Vs. Mach Number

Estimated aerodynamic coefficients and maximum lift/drag, lift coefficients, and angle-of-attack data are presented in Figures 1.3-6 and 1.3-7.

The SSTO uses aircraft-type flight from airport takeoff to approximately Mach 6, with a parallel burn transition of airbreather and rocket engines from Mach 6 to 7.2, and rocket-only burn from Mach 7.2 to orbit. Figure 1.3-8 illustrates a typical trajectory from KSC to an equatorial earth orbit. The prime elements of the trajectory are described below:

- Runway takeoff under high-bypass turbofan/airturbo exchanger (ATE)/ramjet power
- Jettison and parachute recovery of launch gear
- Climb to cruise altitude with turbofan power
- Cruise at optimum altitude, Mach number, and direction vector to earth's equatorial plane, using turbofan power
- Execute a large-radius turn into the equatorial plane still under turbofan power

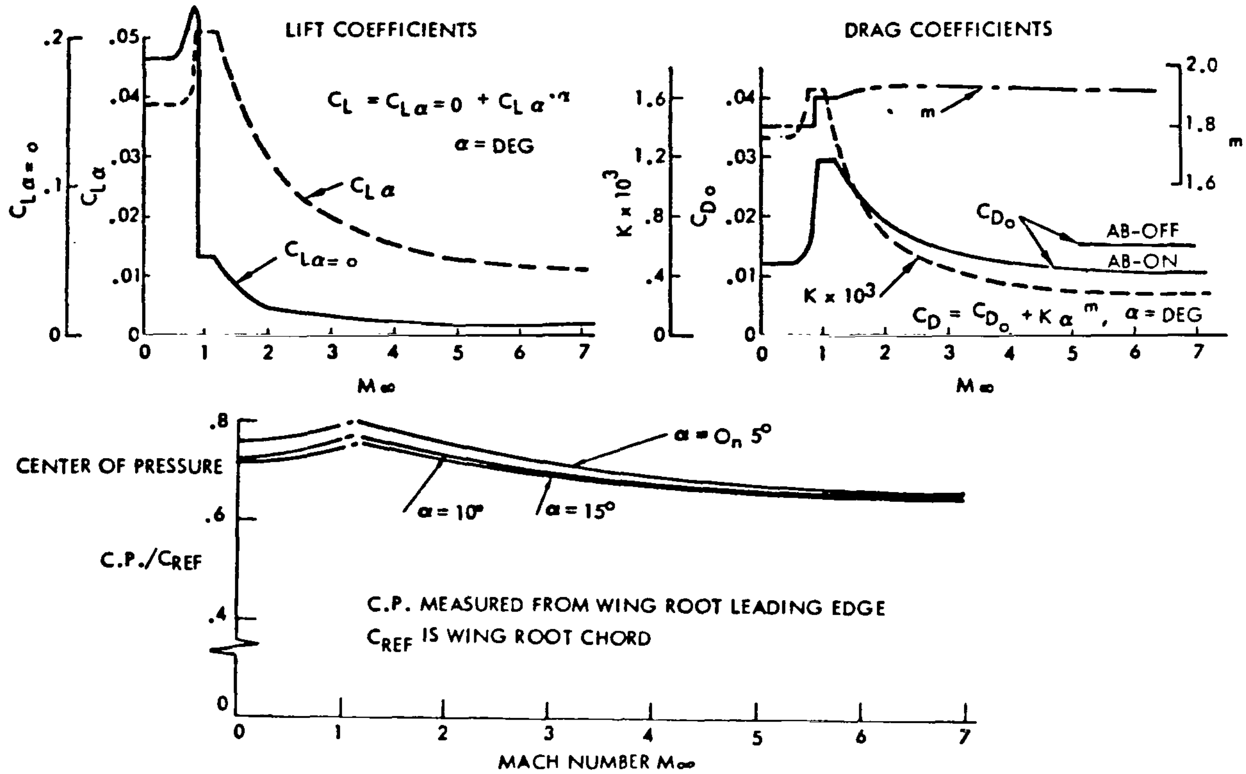


Figure 1.3-6. Aerodynamic Coefficients

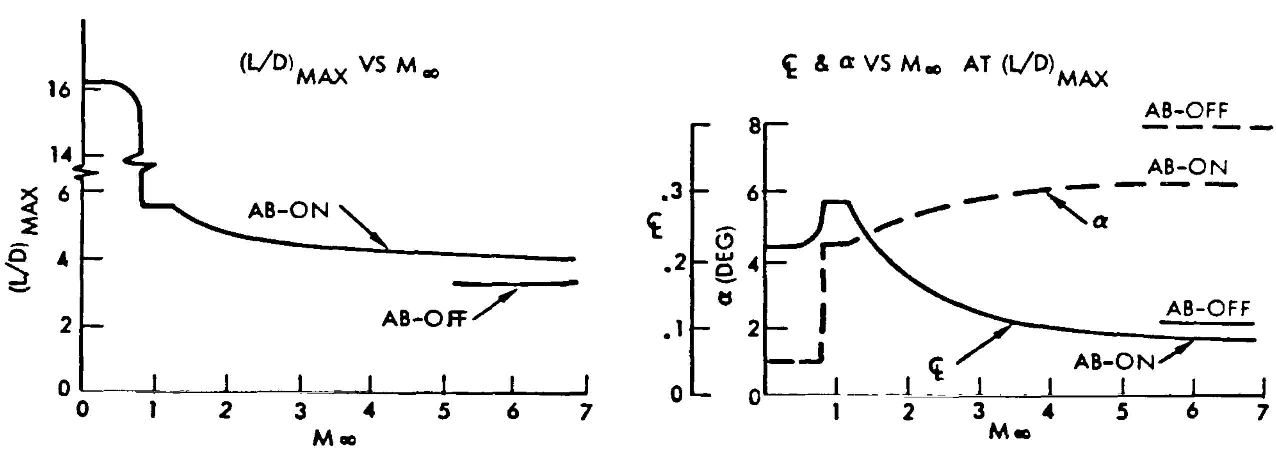


Figure 1.3-7. Maximum Lift/Drag

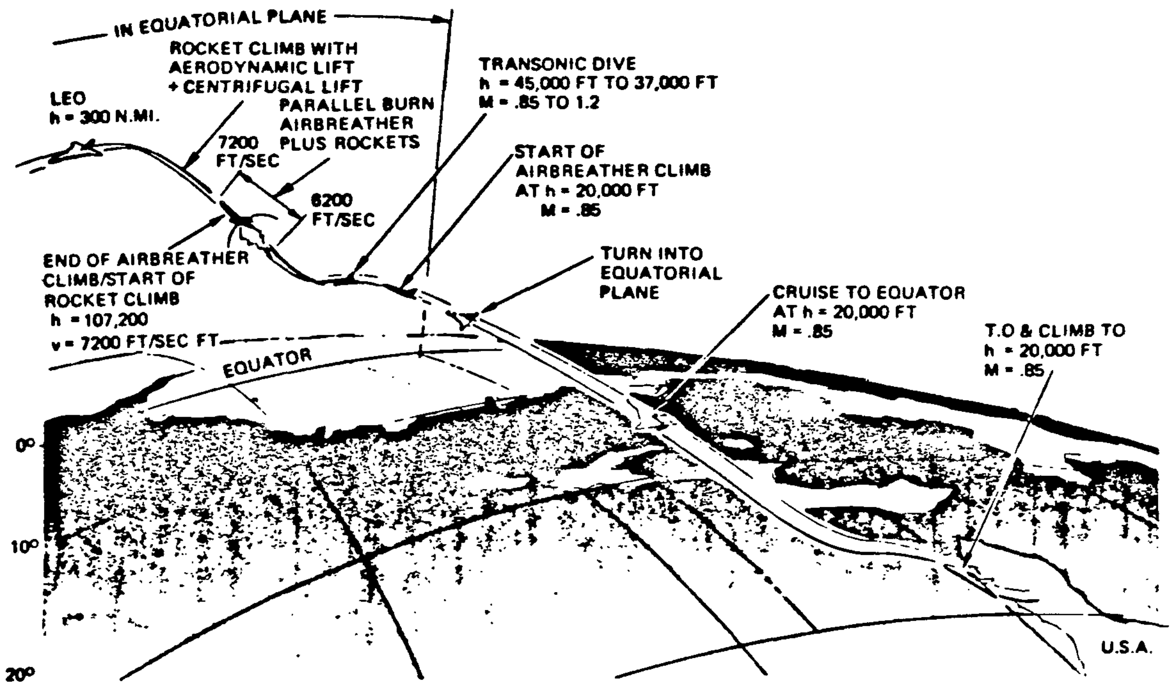


Figure 1.3-8. SSTO Trajectory

- Climb subsonically at optimum climb angle and velocity to an optimum altitude, using high bypass turbofan/ATE/ramjet power
- Perform pitchover into a nearly constant-energy (shallow γ-angle) dive and accelerate through the transonic region to approximately Mach 1.2, using torbofan/ramjet power
- Execute a long-radius pitch-up to an optimum supersonic climb flight path, using turbofan/ATE/ramjet power
- Climb to approximately 29 km (95 Kft) altitude and 1900 m/s (6200 fps) velocity, at optimum flight path angle and velocity, using proportional fuel-flow throttling from turbofan/ATE/ramjet, or full ramjet, as required to maximize total energy acquired per unit mass of fuel consumed as function of velocity and altitude
- Ignite rocket engines to full required thrust level at 1900 mps and parallel burn to 2200 mps
- Shut down airbreather engines while closing airbreather inlet ramps
- Continue rocket power at full thrust
- Insert into an equatorial elliptical orbit 91×556 km (50×300 nmi)
- Shut down rocket engines and execute a Hohmann transfer to 556 km (300 nmi)
- Circularize Hohmann transfer

The reentry trajectory is characterized by low γ (flight path angle), high α (angle of attack) similar to Shuttle. The main reentry elements are:

- Perform delta velocity maneuver and insert into an equatorial elliptical orbit
- Perform a low-γ, high-α deceleration to approximately Mach 6.0
- Reduce α to maximum lift/drag for high-velocity glide and cross-range maneuvers to subsonic velocity (approximately Mach 0.85)
- Open inlets and start airbreather engines
- Perform powered flight to landing field, land, and taxi to dock

Ascent and descent trajectories of the SSTO and Space Shuttle missions are compared in Figure 1.3-9. Because the performance of airbreathing engines and the aerodynamic lift of the winged vehicle depend on a high dynamic pressure, the SSTO flies at much lower altitude during the powered climb than the vertical ascent trajectory of the Space Shuttle for a given flight velocity. Light wing loading of the SSTO contributes to the rapid deceleration during deorbit.

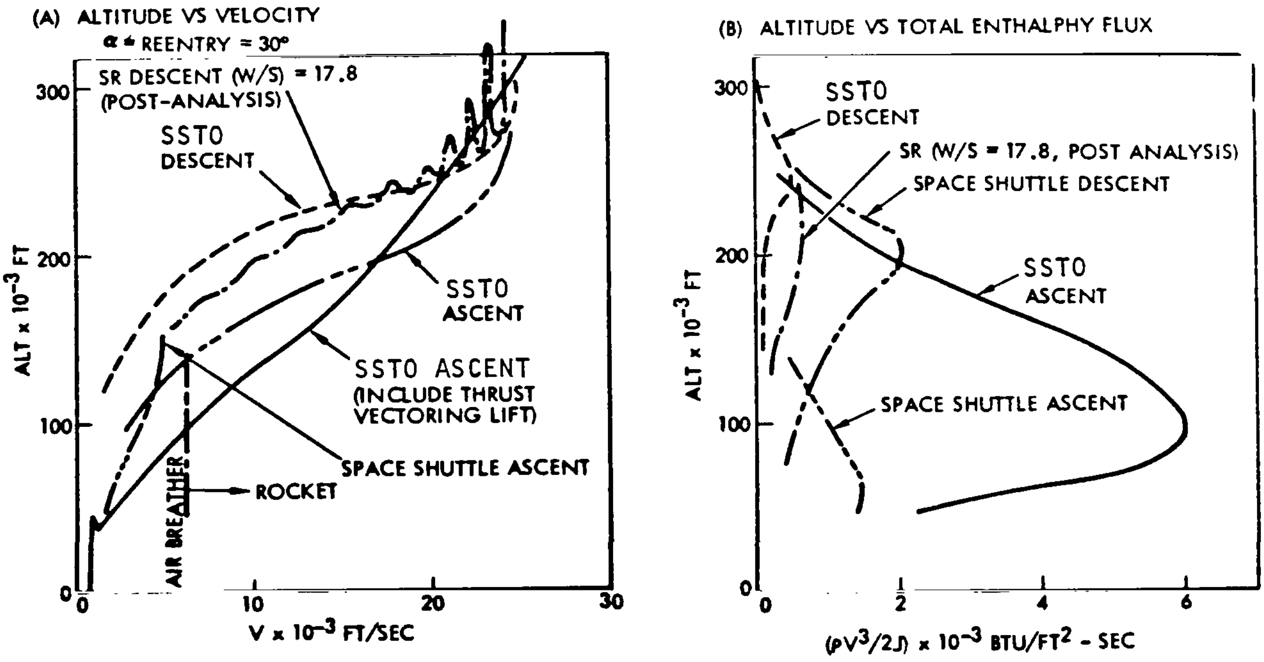


Figure 1.3-9. Ascent and Descent Trajectory Comparisons

The total enthalpy flux histories which indicate the severity of expected aerodynamic heating are also shown in Figure 1.3-9. As expected, the aerodynamic heating of ascent trajectory may design the SSTO TPS requirement. The maximum total enthalpy flux is estimated near the end of airbreather power climb trajectory. Except in the vicinity of vehicle nose, wing leading edge, or structural protuberances, where interference heating may exist, most of the ascent heating is from the frictional flow heating on the relatively smooth flat surface.

The descent heating is mainly produced by the compressive flow on the vehicle windward surface during the high angle-of-attack reentry, and is expected to be lower than the Space Shuttle reentry heating.

For the wing lower surfaces, heating rates were computed including the chordwise variation of local flow properties. Effects of leading edge shock and angle of attack were included in the local flow property evaluation. Leading edge stagnation heating rates were based on the flow conditions normal to the leading edge, neglecting cross-flow effects. All computations were performed using ideal gas thermodynamic properties.

Wing upper-surface heating rates were computed using free-stream flow properties, i.e., neglecting chordwise variations of flow properties. Heating rates were computed for several prescribed wall temperatures as well as the reradiation equilibrium wall temperature condition. Transition from laminar to turbulent flow was taken into account in the computations. Wing/body and inlet interference heating effects were not included in this preliminary analysis. The analysis was limited to the ascent trajectory, since the descent trajectory is thermodynamically less severe.

Isotherms of the peak surface temperatures for upper and lower surfaces (excluding engine inlet interference effects) for the SSTO and the STS orbiter are shown in Figure 1.3-10. Leading edge and upper-wing surface temperatures have similar profiles. The SSTO lower-surface temperatures are from 400°F to 600°F lower than the orbiter due to lower reentry wing loading (23 vs. 67 psf).

Preliminary data indicate that the titanium aluminide system (Figure 1.3-3) may be lighter than the RSI tile for the SSTO TPS system due to the lower average temperature (1000°F to 1600°F) profiles occurring over 80% of the vehicle exterior surface. The metallic truss core sandwich structure is similar to that developed for the B-1 bomber. The radiative surface panel consists of a truss core sandwich structure fabricated by superplastic/diffusion bonding. For temperatures up to 1500/1600°F, the concept utilizes an alloy based on the titanium-aluminum systems which show promise for high-temperature applications currently under development. For temperatures higher than 1500/1600°F, it is anticipated that the dispersion-strengthened superalloys currently being developed for use in gas turbine engines may be applicable. Flexible supports are designed to accommodate longitudinal thermal expansion while retaining sufficient stiffness to transmit surface pressure loads to the primary structure. Also prominent are expansion joints which must absorb longitudinal thermal growth of the radiative surface, and simultaneously prevent the ingress of hot boundary layer gases to the panel interior. The insulation consists of flexible thermal blankets, often encapsulated in foil material to prevent moisture absorption. The insulation protects the primary load-carrying structure from the high external temperature.

Unit masses of the SSTO TPS concept are compared with the unit mass of the STS orbiter RSI in Figure 1.3-11. The unit mass of the RSI includes the tiles,

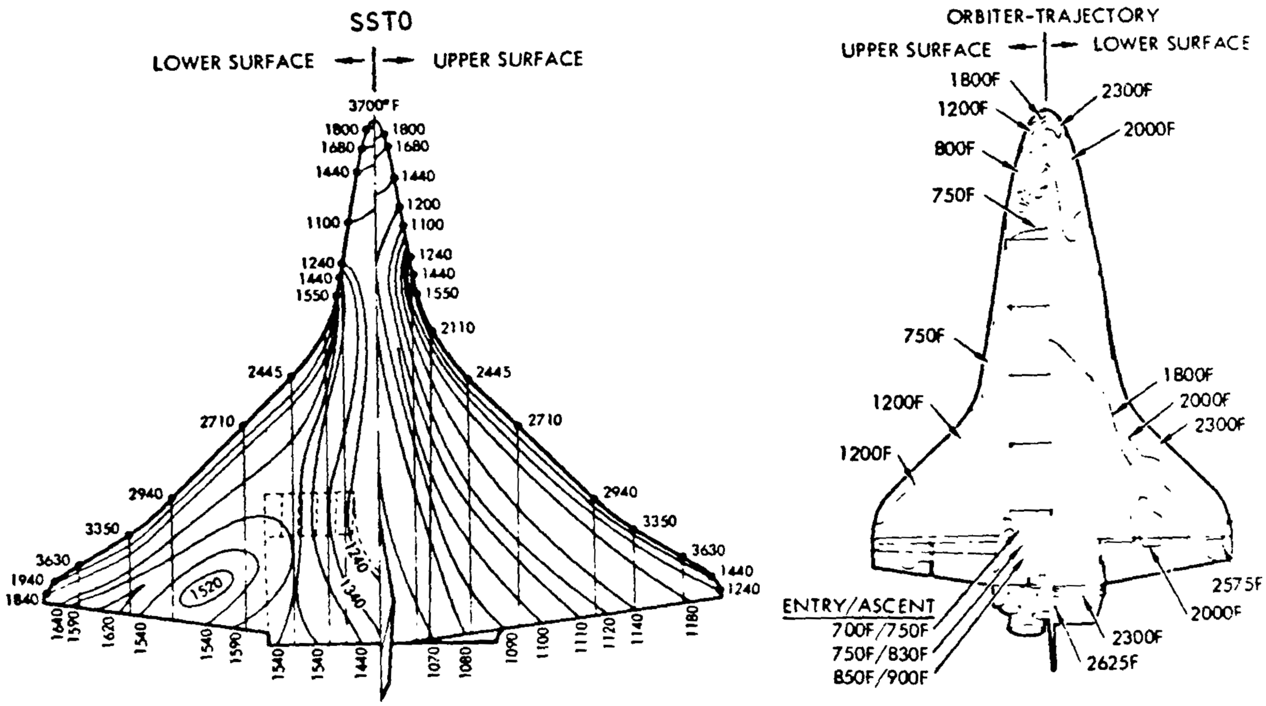


Figure 1.3-10. Isotherms of Peak Surface Temperature During Ascent

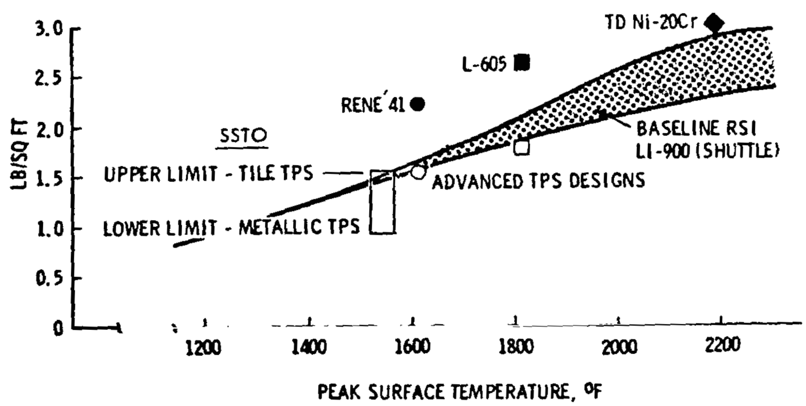


Figure 1.3-11. Unit Mass of TPS Designs

the strain isolator pad, and bonding material. The hatched region shown for the RSI mass is indicative of insulation thickness variations necessary to maintain mold line over the bottom surface of the STS orbiter. The RSI is required to prevent the primary structure temperature from exceeding 350°F. The unit masses of the metallic TPS are plotted at their corresponding maximum use temperatures. The advanced designs are seen to be competitive with the directly bonded RSI.

SSTO mass properties are dominated by the tri-delta wing structure, the thermal protection system, and the airbreather and rocket propulsion system. Estimated vehicle weights data are presented in Table 1.3-1.

Table 1.3-1. SSTO Weight Summary

Item Description	Weight (10^3 kg)
Airframe, aerosurfaces, tanks and TPS	167.8
Landing gear	12.3
Rocket propulsion	32.5
Airbreather propulsion	63.5
RCS propulsion	4.5
OMS propulsion	2.3
Other systems	17.2
Subtotal	300.4
Growth (10%)	30.0
Total inert weight (dry)	330.4
Useful load (fluid, reserves, etc.)	21.5
Inert weight and useful load	351.9
Payload weight	89.2
Orbital insertion weight	441.1
Propellant ascent	1826.9
GLOW (post-jett. launch gear)	2268.0

Again, it is emphasized that the SSTO concept represents a most advanced technology option and considerable further analyses are required to demonstrate viability of concept and definition of a much advanced technology program.

1.4 SMALL VTO/HL HLLV CONCEPTS (PREFERRED ALTERNATE CONCEPTS)

The primary driver in establishing HLLV requirements is the timely delivery of construction material to LEO; thus the payload magnitude becomes a major design parameter. The present-day use of the term "heavy lift" connotates a launch system with a payload capability substantially greater than the 30 metric tons of the Space Shuttle. A "small" heavy-lift system is a large vehicle; the term "small" is comparative to the very large SPS reference system. While reduced HLLV size would permit use of the already developed SSME with appropriate modifications to provide longer life, this in turn incurs an increased number of flights to deliver an equivalent mass to orbit. In addition, VTO/HL vehicle size may be severely limited by erection, mating, and launch wind conditions. A final resolution of the most practical payload from overall considerations will have to await the results of separate future studies. The basic ground rules and assumptions employed are the same as used for the larger payload versions.

1.4.1 HLLV Parallel Burn VTO/HL

An alternate (smaller payload) configuration of more conservative design (i.e., more closely resembling the STS configuration) is depicted in the launch configuration, Figure 1.4-1. This configuration was adopted to permit the use of documented STS aerodynamic and performance data in order to address certain specific technical issues relative to VTO/HL vehicle concepts.

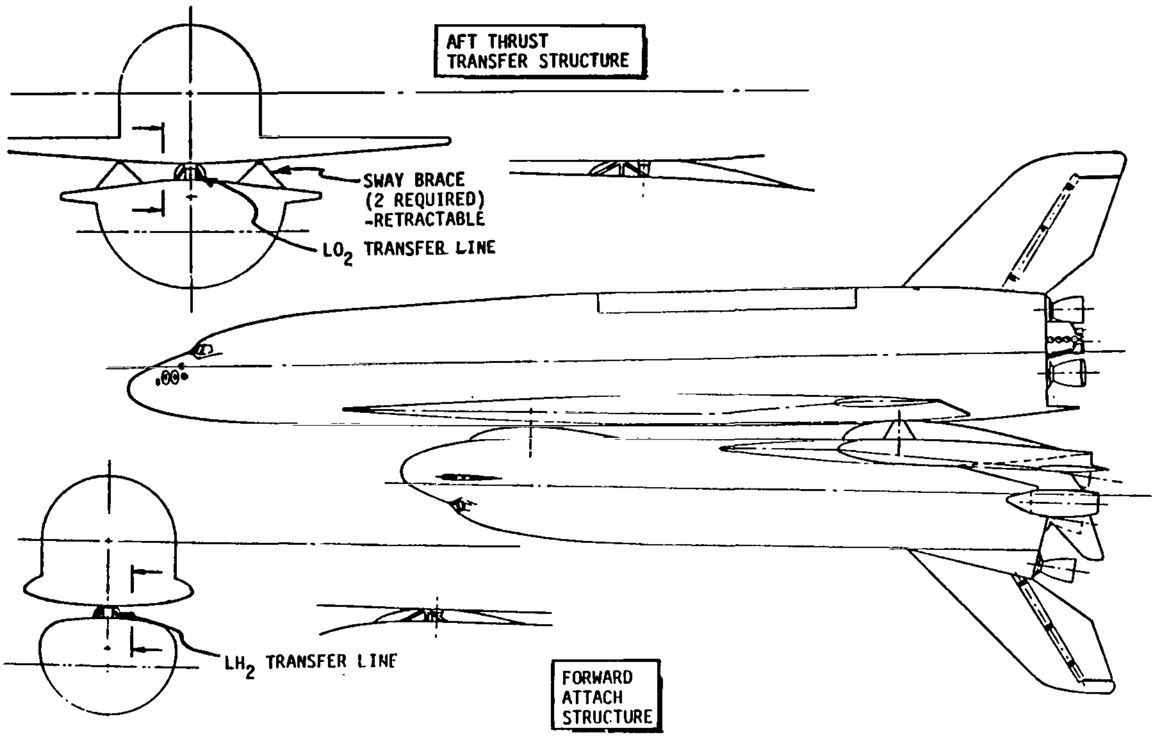


Figure 1.4-1. PB/VTO/HL HLLV Mated System and Attach Structure

Each of the two stages have return-to-base capability with vertical take-off and horizontal landing characteristics; the orbiter is unpowered at landing while the boosters fly back to the launch site with an airbreathing engine propulsion system. The launch vehicle utilizes a parallel burn propulsion mode with first-stage LO_2 and LH_2 being crossfed from the booster to the orbiter such that the orbiter stages with full propellant tanks. The booster utilizes high chamber pressure gas generator cycle LO_2/RP-1 fueled engines and the orbiter utilizes staged combustion LO_2/LH_2 engines developed from the Space Shuttle Main Engine (SSME) operating at zero NPSH.

The staging velocity was selected from earlier trade studies to be compatible with a heat sink structural concept for the booster. Material selection and development consistent with the 1990 time frame will ultimately play a significant role in the final selection of staging velocity. Thrust-to-weight requirements are selected to minimize engine size and crew/passenger discomfort. Orbital parameters are consistent with SPS LEO base requirements.

The mated system employs a fore and aft primary structural attach and sway brace attachment for differential roll stabilization. All attach points are released at staging through the application of explosive bolts.

The booster stage is approximately 61 m long and the orbiter, or second stage, is approximately 91 m long. Although the internal volume requirements are nearly the same, the boost vehicle employs eight LO_2/RP engines and, therefore, requires a wider base area. This wider base permits the application of the "double-bubble" type propellant tanks to accommodate hypersonic aerodynamic stability requirements and, hence, a foreshortening of the entire vehicle.

All ascent fuel to staging is contained in the boost vehicle. This necessitates a propellant transfer system. The LO_2 transfer system is supported by the aft structural attach system and is housed within the streamline fairing associated with the aft attach location. LH_2 is transferred at the forward attach point, and like the LO_2 system, is supported by the forward attach structure. It is housed within the forward streamlined fairing. The streamline fairings are applied at drag and interference heating points.

The booster, Figure 1.4-2 employs hot structure with metallic heat sink as required for the entry flight regime of the booster. Initial investigations indicate that utilization of advanced metal matrix technology wherever feasible will result in a substantial weight savings.

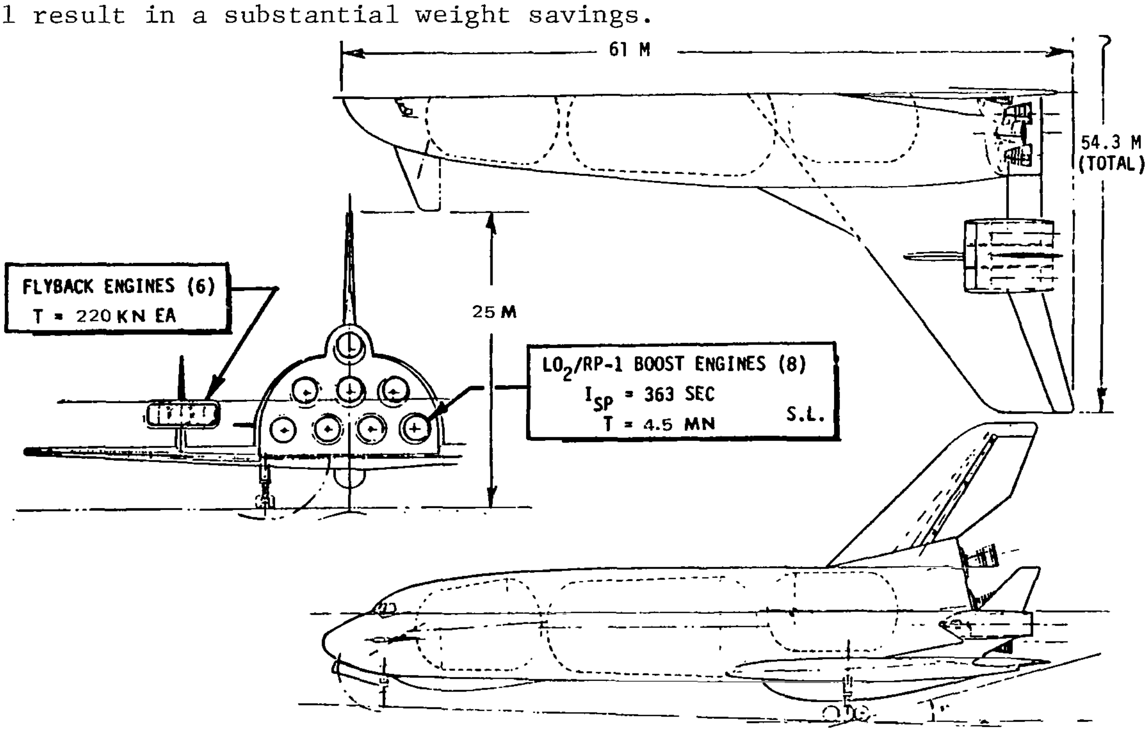


Figure 1.4-2. PB/VTO/HL HLLV—Booster

The wing is sized to produce a nominal 333 km/hr landing speed and is optimized to minimize flyback propulsion requirements. Six turbojet engines are provided to accommodate the return-to-base mode after a launch. This flyback propulsion system weighs approximately 45,000 kg (with 9000 kg of JP-5 fuel). Ascent propulsion is provided by eight advanced development engines of 4.5 MN thrust each.

The system employs a belly-to-belly mating system for structural and propellant transfer continuity. Drag loads are reacted through a centerline attach truss located within the aft mounted fairing which also houses the LO_2 transfer line. The forward attach reacts yaw and pitch inputs and supports the LH_2 transfer line within the forward fairing. Retractable outboard sway braces (two) are employed to stabilize the system in differential roll.

The orbiter configuration, Figure 1.4-3, has been established to accommodate a payload of 113,500 kg in a volume of 1382 m^3 with a payload bay length of 21.3 m. The payload density is 82 kg/m^3.

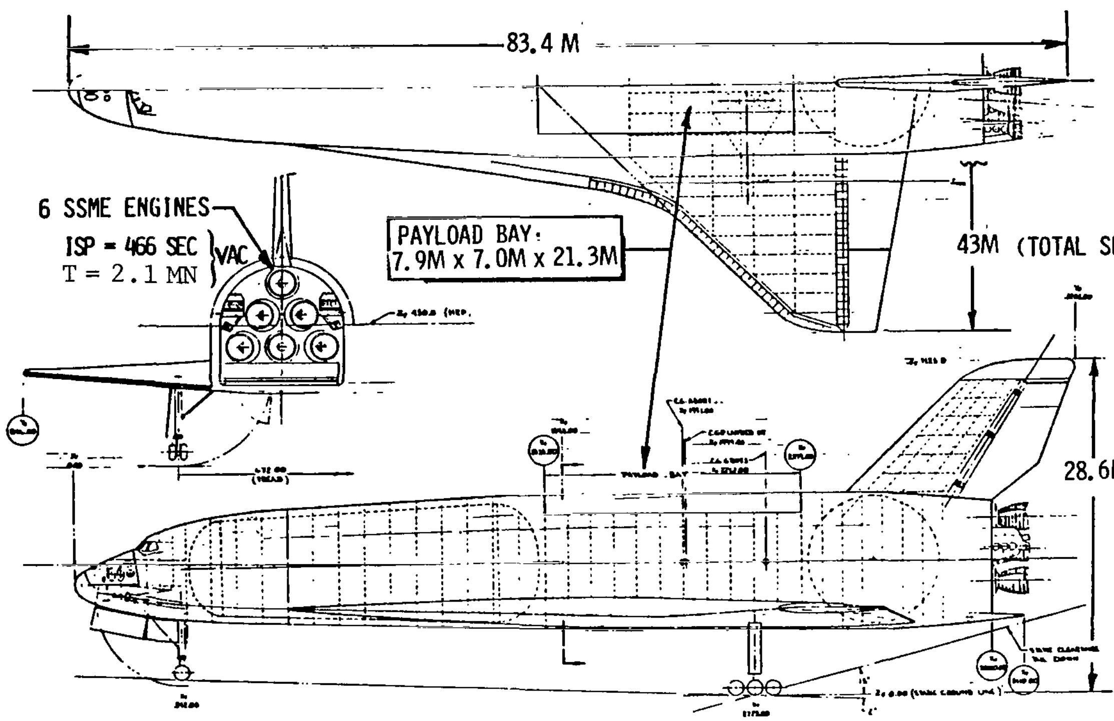


Figure 1.4-3. PB/VTO/HL HLLV—Orbiter,
Payload = 113,500 kg

The orbiter wing has been scaled from the Shuttle orbiter which permits the application of documented Shuttle orbiter aerodynamic data for performance estimation. The wing has been sized for the abort-once-around flight condition (payload onboard) to provide a nominal landing speed of 333 km/hr.

For the purposes of the present study, graphite-polyimide (GR/PI) has been selected as the primary structural material with RFCI tile for the TPS. Reentry thermal gradients are very similar to Shuttle orbiter because of the similar wing loading and planform. Thus, the FRCI can be tailored to accommodate the 600°F backface temperature allowable through the application of the GR/PI. It is assumed, for the time frame of the application, that a direct bond system will have been developed through the application of GR/PI. The structural weight fraction of the system is reduced by approximately 20% from conventional metallic structures.

The propulsion system employs six SSME engines which produce 2.1 MN thrust each (vacuum). The cryogenic tankage is non-integral to minimize the requirement for a high-risk developmental technology. However, additional weight savings could be realized through the application of integral cryogenic tankage, but would require an intense design and development program to achieve the reliability, inspectability, and maintainability required for a reusable system.

Additional weight savings have been realized by the judicious location of the avionics and ancillary systems. Communications between systems will be accomplished by the application of fiber-optics. Power supply systems will be located at the point of application (i.e., separate systems fore and aft), thus reducing the amount and run length of the power cables.

The substantial increase in orbiter size when designed for transporting much heavier payloads than the present Space Shuttle orbiter (29,500 kg) is readily apparent when the SPS HLLV orbiter is compared to the Shuttle orbiter at the same scale, Figure 1.4-4. Dimensionally, such a comparison is somewhat misleading since the larger orbiter is a "wet" design, containing its own fuel, while the smaller is "dry."

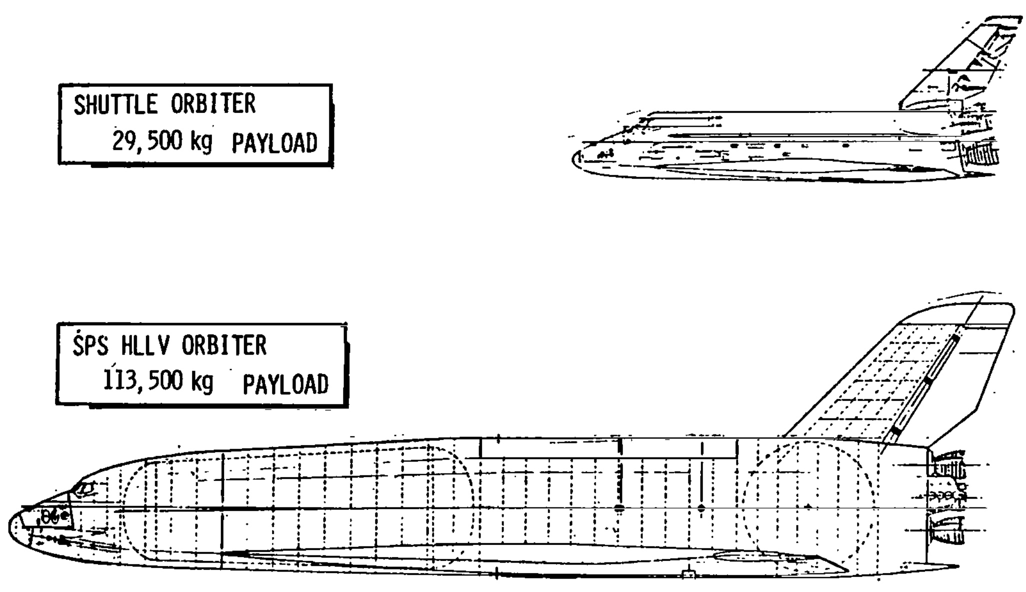


Figure 1.4-4. Size Comparison—Orbiters

The combined mass properties of the vehicle are presented in Table 1.4-1. At liftoff, the HLLV weighs 3.56M kg. At sea level, the thrust of the six orbiter engines is 10 MN and the thrust of the eight booster engines is 35.6 MN. The total thrust at liftoff is 45.6 MN for a thrust-to-weight of 1.306.

Table 1.4-1. Combined Mass Properties

Condition	WT (10^6 kg)	X_O
Booster @ liftoff	2.410	2175
Booster @ liftoff	1.150	2262
Liftoff	3.561	2203
Booster propellant	-1.702	2127
Crossfed orbiter propellant	-0.446	
Staging	1.413	2320
Booster @ staging	-0.262	2573
Solo orbiter	1.151	2262
Orbiter propellant	-0.830	2367
Orbiter @ burnout	0.321	1991
Inert orbiter	-0.208	2015
Delivered payload	0.114	1950

During the booster flight of almost 160 sec, 1.70M kg of LO_2/RP are burned by the booster engines and almost 445,000 kg of LO_2/LH_2 are transferred to the orbiter for SSME engine use. After separation from the booster at a relative velocity of about 1980 mps the orbiter continues to orbit with a payload of 114,000 kg.

The booster mass properties are given in Table 1.4-2. The structure represents about 37% of the dry weight. Of this total, 58% is fuselage, 32% is wing, 6% is tail, and 1.5% is canard. Use of advanced hot structure results in unit weights of 4.8 psf for the body surface area, 11.7, 8.5 and 8.0 psf for the planform area of the wing, tail, and canard, respectively. Allowances for a pressurized crew module for a crew of two have been provided. The landing gear weight was at 3.4% of the landing weight of 4.0% of the dry weight.

The propulsion system is almost 34% of the dry weight. Of this total, 51% is for engines, 18% for the RP tank, the orbiter crossfeed LH_2 and the combination LO_2 tank, 20% for the delivery systems, including the LO_2/RP feed and LO_2/LH_2 crossfeed systems, and 11% for the primary thrust structure.

A small auxiliary propulsion system for attitude control is provided. The flyback system represents 15% of the dry weight and includes feed and wet wing tankage for the propellant.

The total inert weight of the booster is also the staging weight and represents about 11% of the gross weight for a stage mass fraction of 0.89.

The booster lands with a c.g. of about 73.3% of the reference body length (L_B). At liftoff, the booster has a weight of slightly over 2.4M kg at a c.g. of 56.4% L_B.

Table 1.4-2. PB/VTO/HL HLLV Booster Mass Properties

ITEM	WT ($kg \times 10^3$)	X_B
STRUCTURE	85.98	
TCS & PV&D	1.77	
LANDING GEAR	9.21	
PRIMARY PROPULSION	78.79	
AUXILIARY PROPULSION	1.13	
FLYBACK PROPULSION	34.47	
HYDRAULICS AND ACTUATION	8.05	
ELECTRICAL POWER	1.95	
AVAIONICS & EPD&C	7.17	
ECLSS	1.77	
PERSONAL PROVISIONS	0.81	
ORBITER/BOOSTER ATTACH STRUCT	1.00	
DRY WEIGHT	232.10	
RESIDUALS	3.66	
RESERVES	0.09	
LANDED WEIGHT	235.85	1671
USED IN FLIGHT	15.81	-
AUXILIARY PROPELLANT	0.91	-
FLYBACK PROPELLANT	9.07	-
STAGING WEIGHT	261.63	1683
BOOSTER—LO_2/RP	702.28	-
ORBITER—LO_2/LH	446.34	-
GROSS LIFTOFF WEIGHT	2410.26	1285

	X_B	L_B
LANDED	1671	73.3%
SEPARATION	1683	73.8%
GLOW	1285	56.4%

The orbiter mass properties are presented in Table 1.4-3. The structure when combined with the thermal protection system (TPS) represents almost 60% of the dry weight. Of this total, 66% is fuselage, 29% is wing, and 5% is tail. Use of advanced composite structure and reusable surface insulation results in unit weights of 5.9 psf for the body surface area, 12.65 psf and 9.2 psf for the planform area of the wing and tail, respectively. Allowances for a pressurized crew module, for internal thermal control (TCS) and purge, vent and drain (PV&D) have been provided. Landing gear weight was estimated at 3.4% of the abort weight, or 5.5% of the dry weights.

The propulsion system is almost 24% of the dry weight. Of this total, 52% is for six modified SSME engines, 24% is for non-integral LO_2 and LH_2 tanks, 18% for delivery systems, including tank, crossfeed, fill, vent and drain lines and valves. The basic thrust structure is 6.4% of the propulsion system weight.

Table 1.4-3. PB/VTO/HL HLLV Orbiter Mass Properties

ITEM	WT (kg×10³)	X_0
STRUCTURE	78.81	
TPS, TCS & PV&D	40.00	
LANDING GEAR	10.87	
PRIMARY PROPULSION	46.67	
AUXILIARY PROPULSION	2.06	
HYDRAULICS & ACTUATION	4.01	
ELECTRICAL POWER	1.95	
AVIONICS & EPD&C	7.68	
ECLSS	1.77	
PERSONAL PROVISIONS	0.81	
PAYLOAD PROVISIONS	1.13	
ORBITER/BOOSTER ATTACH STRUCT	1.00	
DRY WEIGHT	196.81	
RESIDUALS	0.95	
RESERVES	0.03	
LANDED WEIGHT	198.03	1999
USED IN FLIGHT	6.36	-
AUXILIARY PROPUL. PROP.	3.14	-
TOTAL INERT WEIGHT	207.53	2015
PAYLOAD	113.5	1950
ABORT WEIGHT	320.93	1991
ASC PROPELLANT	830.14	-
GROSS LIFTOFF WEIGHT	1151.08	2262

	X_0	L_B	MAC
ABORT	1991	64.2%	13.9%
LANDED	1999	64.5%	14.8%
INERT	2015	65.0%	16.7%
GLOW	2276	73.4%	47.1%

The remaining systems weigh about 20,400 kg, or 10.5% of the dry weight. All weights are based on similar elements of the STS orbiter. The auxiliary propulsion system (APS) is basically that of the STS orbiter, while the hydraulic system is double that of the STS orbiter. Two redundant/separate fuel cell/cryo tank sets are employed—one for the forward equipment, and the other for the aft equipment. Two redundant and separate environmental control systems are also provided. The forward system also includes the life support system. The avionics are located functionally and are connected only by fiber optical wiring.

Personnel provisions are for a crew of two for two days. Allowances are provided for payload installation and mechanical/electrical/fluid connections to the booster. Residuals account for trapped line and tank fluids and gases. The reserves are for the APS. Almost 9500 kg of fluids are used during ascent, flight and descent, including 3130 kg of APS propellants.

The total inert weight represents about 18% of the gross weight, and the payload 10%, for an overall stage mass fraction of 0.72 (including payload).

The orbiter normally lands with a center of gravity (c.g.) at 64.5% of the reference body length (L_B = 79 m), or 14.8% of the mean aerodynamic chord (MAC). The abort c.g. is only slightly aft of the normal landing c.g. From ground liftoff to booster separation, the orbiter weight is slightly greater than $1.13\bar{M}$ kg with a c.g. at 73.4% L_B, or 47.1% MAC.

1.4.2 HLLV Series Burn VTO/HL (Smaller Reference Concept)

Certain hardware items in the reference SPS system were sized to take advantage of the large (17-m diameter by 23-m length) payload bay of the reference launch vehicle. Principal items are the electrical rotary joint (slip ring) and the crew habitats of the orbital bases. Clearly, a smaller payload bay volume will impose penalties on these elments of the system or require added construction labor in space. The realizable reduction of size of the launch vehicle without reduction of the large payload bay envelope would be extremely limited. Accordingly, it was necessary to make a reasonable judgment as to how much envelope reduction could be accommodated by SPS systems without excessive penalties. A smaller crew habitat will house fewer crew per unit, but there is nothing special about the 100-man reference capacity. Smaller habitats will incur operational inconveniences, but will provide nonrecurring cost reductions and may avoid the necessity (presently shown in the reference SPS development scenario) to develop an intermediate-sized habitat (larger than SOC, but smaller than the ultimate article) for a demonstration project.

Based on these and similar considerations, it was concluded that the limiting article is the power transmitter subarray. The subarrays are 10.4 m^2 by about 30 cm thick. Accordingly, it was decided to employ a square cross-section payload bay 11 m^2, with some convenient length.

Vehicle scaling was based upon prior parametric scaling studies and included consideration of the variation in structural efficiency with stage size and propellant load.

Based on these prior studies, a liftoff mass of 4000 tons was selected for a point design study. The payload capability anticipated from these parametric analyses is 120 metric tons. SPS packaging studies have indicated that the payload bay density (lift capability/volume) should be in the range of 75 kg/m^3 to 100 kg/m^3. The forcing function is the relatively low density of transmitter subarrays; they average much less than 75 kg/m^3 but by mixing subarrays with high-density items, an average in the range stated is obtained. At 120 metric tons lift capability, an 11-m^2 payload bay cross-section requires a length of 13.2 m to reach 75 kg/m^3. Anticipating the 120-metric-ton estimate to be slightly conservative, a length of 14 m was selected. Note that this payload bay, although it has 5.6 times the volume of the Shuttle payload bay, is actually about 4 m shorter. The analysis did not include booster flyback range as a parameter. For typical boosters, flyback propellant is 10% to 20% of inert mass; the variation of flyback propellant with staging conditions is a significant overall optimization parameter.

The selected small HLLV series burn configuration is shown in Figure 1.4-5. The orbiter includes a swept-back delta wing with a small subsonic foldout canard. The payload bay is aft of the propellant tanks and is 11 m^2 by 14 m long. The orbiter uses six Space Shuttle main engines with extended exit bells. Four of the six engines are gimbaled; the center two are fixed. The upper stage also uses a small yaw ventral for head-end steering to improve controlability in yaw.

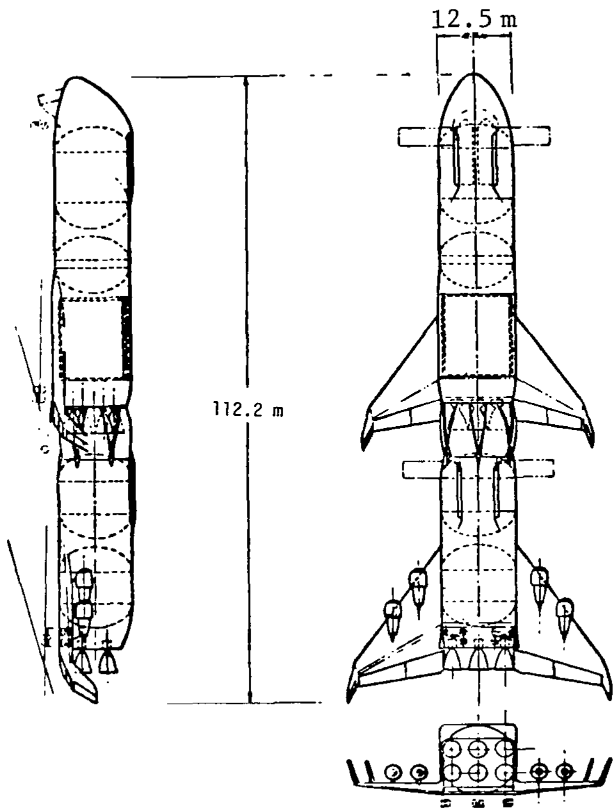


Figure 1.4-5. SB/VTO/HL HLLV Configuration

The vehicles are control configured in yaw, thus eliminating the large vertical tail. Elimination of the vertical tail assists in balancing the vehicle and makes practical an aft payload bay on the orbiter. The booster employs a "flower-petal" opening nose with a truss structure as an interstage structure. This approach avoids expendible interstage hardware and allows the second-stage engine start sequence to be initiated during the first-stage tail-off as the open nose allows room for gas venting during the start sequence. After stage separation, a hinged actuator mechanism closes the nose to a streamlined, aerodynamic configuration.

The booster employs six oxygen-methane engines of approximately 1835 KN thrust. Four high-thrust airbreather engines are mounted on top of the wings for flyback. The airbreather engine inlets are closed by a blow-off cover until subsonic transition, at which time the engines undergo start sequence. Engine location was selected to avoid flow attachment to either the wing or the body as a flow attachment will result in higher drag during the flyback.

Table 1.4-4 presents the mass statement for the small HLLV. The estimated payload, based on the detailed mass statement, is 126 metric tons as compared to a parametric figure of 120 metric tons.

The vehicle launch trajectory employs zero-lift "gravity-turn" boost trajectory followed by a roughly optimized second-stage trajectory. Injection conditions are 90 km altitude, due east, with injection velocity appropriate to coast to 447 km altitude.

Shortly after liftoff, the mated vehicle (under booster thrust) executes a slight "tilt" away from vertical flight, in the downrange direction. This initiates the "gravity turn." The amount of tilt sets the staging conditions. With a fixed amount of boost propellant, more tilt (1) reduces staging altitude, (2) reduces staging path angle, and (3) increases relative velocity at staging. Figures 1.4-6 and 1.4-7 show the characteristics of a preliminary reference trajectory with near-optimal characteristics.

Final selection of a reference trajectory requires evaluation of flyback range effects. For any flyback range, there will be an optimal booster wing area. Increasing wing area increases with flyback cruise L/D, decreasing both installed thrust and flyback fuel. Since increasing wing area reaches a point of diminishing returns, i.e., further increases in area add little to L/D, whereas wing mass increases nearly linearly with area, it is apparent that an optimal area must exist (for any given flyback range). Since booster inerts affect payload (1 kg of booster inerts is worth roughly 1/6 kg payload) there is a joint optimum among staging conditions and booster wing area. These optimizations are nearly decoupled, however, because of the sharpness of the optimum of tilt (= staging conditions). The flyback range at optimal staging conditions will be between 250 and 300 km. Over this range, the optimal wing area will change little. Consequently, our analysis assumed that optima to be entirely decoupled.

A further parametric study was conducted to select the reference wing area. Wing area was dictated by landing speed with a desire to maintain landing speed at no more than 300 km/hr. The result was a selection of a reference wing area of 760 m^2 with a canard for subsonic trim. A hypersonic trim investigation showed that the vehicle could be trimmed between 30 and 40 degrees angle of attack with reasonable aileron deflections.

The orbiter wing area was also selected for landing speed of 300 km/hr. Again, a canard was used for subsonic trim to avoid large wing areas.

Comparative costs between the small HLLV and the large reference system were evaluated. Satellite design changes resulted in increased costs for the space construction systems. The necessity to use smaller crew modules results

Table 1.4-4. Small HLLV Mass Properties

BOOSTER	KG	LBM
STRUCTURE-AEROSURFACES	28,235	62,245
WING	25,509	56,236
CANARD	1,452	3,200
TIPLETS	1,020	2,249
YAW VENTRAL	254	560
STRUCTURE - BODY & TANKS	69,107	152,357
NOSE	9,761	21,519
NOSE GEAR SUPPORT	693	1,528
METHANE TANK	9,684	21,349
OXYGEN TANK	13,610	30,006
INTERTANK	10,592	23,353
AFT BAY & FAIRINGS	10,513	23,178
THRUST STRUCTURE	8,130	17,924
BODY FLAP	1,860	4,100
FAIRINGS	4,264	9,400
TPS	0	0
MECHANISMS	9,043	19,936
LANDING GEAR	8,090	17,836
DRAG DEVICE	953	2,100
MAIN PROPULSION	68,750	151,596
ROCKET ENGINES	50,000	110,229
ENGINE ACCESSORIES	6,250	13,779
PROPELLANT SYSTEMS	12,500	27,588
AUXILIARY PROPULSION	30,615	67,495
FLYBACK ENGINES	25,000	55,115
FUEL SYSTEM	3,039	6,700
RCS	2,576	5,680
SUBSYSTEMS	7,804	17,205
AUXILIARY POWER	703	1,550
ELEC. CONV & DISTR.	2,667	5,880
FLT CONTROL ACTUATION	2,073	4,570
FLIGHT CONTROL SYSTEM	1,111	2,450
AVIONICS	1,000	2,205
EC/LSS	250	550
GROWTH	21,355	47,083
TOTAL DRY	234,909	517,917
FLUIDS	61,466	135,506
BIAS PROPELLANT	11,300	24,911
PRESSURANT	11,300	24,911
RESIDUALS & TRAPPED	8,475	18,684
FLYBACK FUEL	30,391	67,000
NET INERTS	296,375	653,423
IMPULSE PROPELLANT	2,260,000	4,982,396
BOOSTER LIFTOFF MASS	2,556,375	5,635,819

ORBITER	KG	LBM
STRUCTURE-AEROSURFACES	22,552	49,720
WING	20,135	44,390
CANARD	1,560	3,440
TIPLETS	635	1,400
YAW VENTRAL	222	490

ORBITER (CON'T)	KG	LBM
STRUCTURE-BODY & TANKS	66.328	146,211
NOSE	2,440	5,380
NOSE GEAR SUPPORT	529	1,166
LH_2 TANK	10,928	24,093
LO_2 TANK	11,719	25,835
INTERTANK	6,231	13,737
PAYLOAD BAY BODY SECTION	10,282	22,668
PAYLOAD BAY DOORS	2,255	4,971
AFT BODY	10,979	24,204
THRUST STRUCTURE	3,390	7,473
BODY FLAP	2,270	5,000
FAIRINGS	2,137	4,700
CREW CAB STRUCTURE	3,168	6,984
INDUCED THERMAL PROTECTION	19,923	43,922
WING RSI	4,799	10,580
BODY RSI	10,136	22,345
TANK SIDEWALL PANELS	1,571	3,465
WING TIPLETS RSI	386	850
LH_2 INTERNAL INSULATION	2,169	4,782
PROPELLANT PURGE, VENT, & DRAIN	862	1,900
MECHANISMS	7,198	15,869
LANDING GEAR	6,439	14,196
DRAG DEVICE	759	1,673
MAIN PROPULSION	31,694	69,873
SSME's	19,336	42,630
TOTAL INERTS	221,650	488,641
ASCENT PAYLOAD	126,260	278,359
TOTAL ORBITER INJECTED	347,910	767,000

INTEGRATED VEHICLE	KG	LBM
IMPULSE PROPELLANT	1,130,000	2,491,198
ORBITER AT LIFTOFF	1,477,910	3,258,198
BOOSTER AT LIFTOFF	2,556,375	5,635,819
VEHICLE AT LIFTOFF	4,034,285	8,894,017

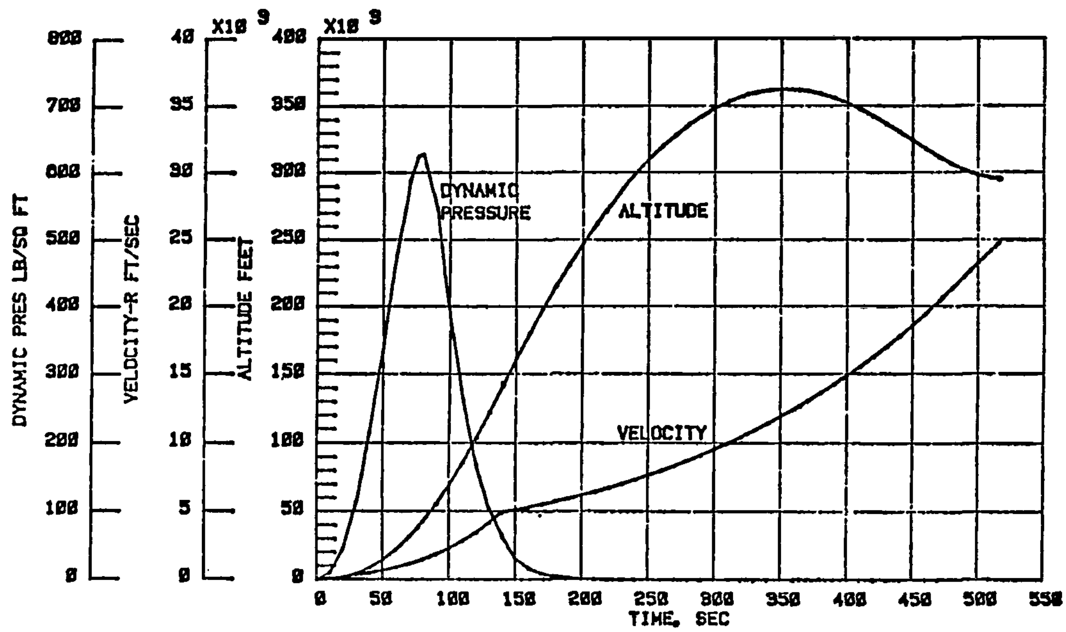


Figure 1.4-6. Small HLLV Reference Trajectory

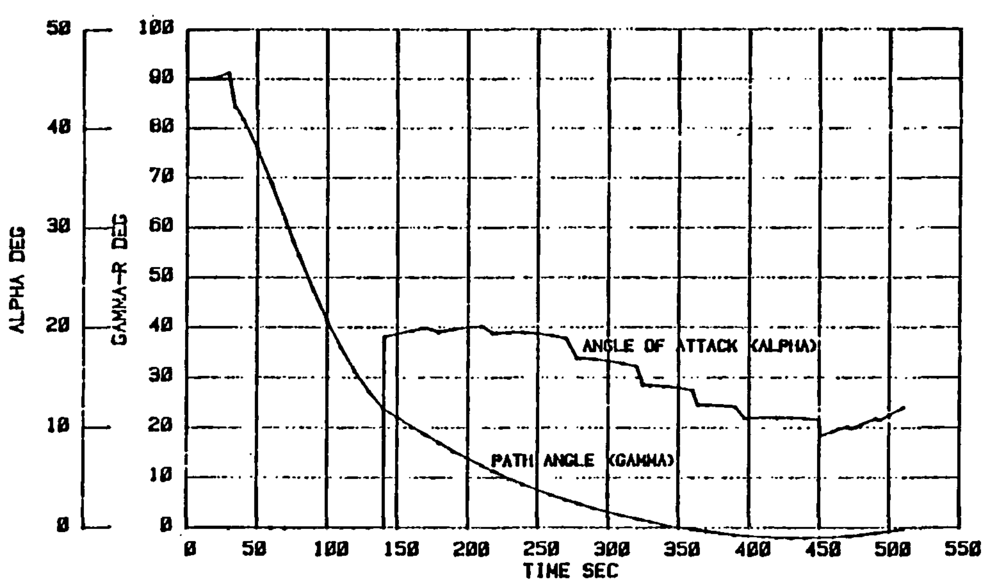


Figure 1.4-7. Small HLLV Reference Trajectory

in a DDT&E savings, but an investment increase from the need to buy more of the smaller modules. Transportation cost effects include direct DDT&E savings on the smaller launch vehicle, savings resulting from less complex facilities and increase in the fleet investment and in the HLLV factory, and savings resulting from less development activity on Shuttle derivatives as a result of having the small heavy lift launch vehicle.

In summary, the small HLLV has positive features and some negative features. Table 1.4-5 summarizes these positive and negative features. In general, the positive features outweigh the negative ones and the small HLLV appears to be a better option for SPS.

Table 1.4-5. Small HLLV Net Effects

POSITIVE

- LESS NONRECURRING COST: MORE COMMONALITY WITH SHUTTLE
- REDUCED NOISE AND SONIC OVERPRESSURE
- LESS FACILITIES COST: OFFSHORE PADS NOT NEEDED
- SIZE APPROPRIATE FOR ALTERNATIVE MISSIONS
- CREW AS WELL AS CARGO DELIVERY

NEGATIVE

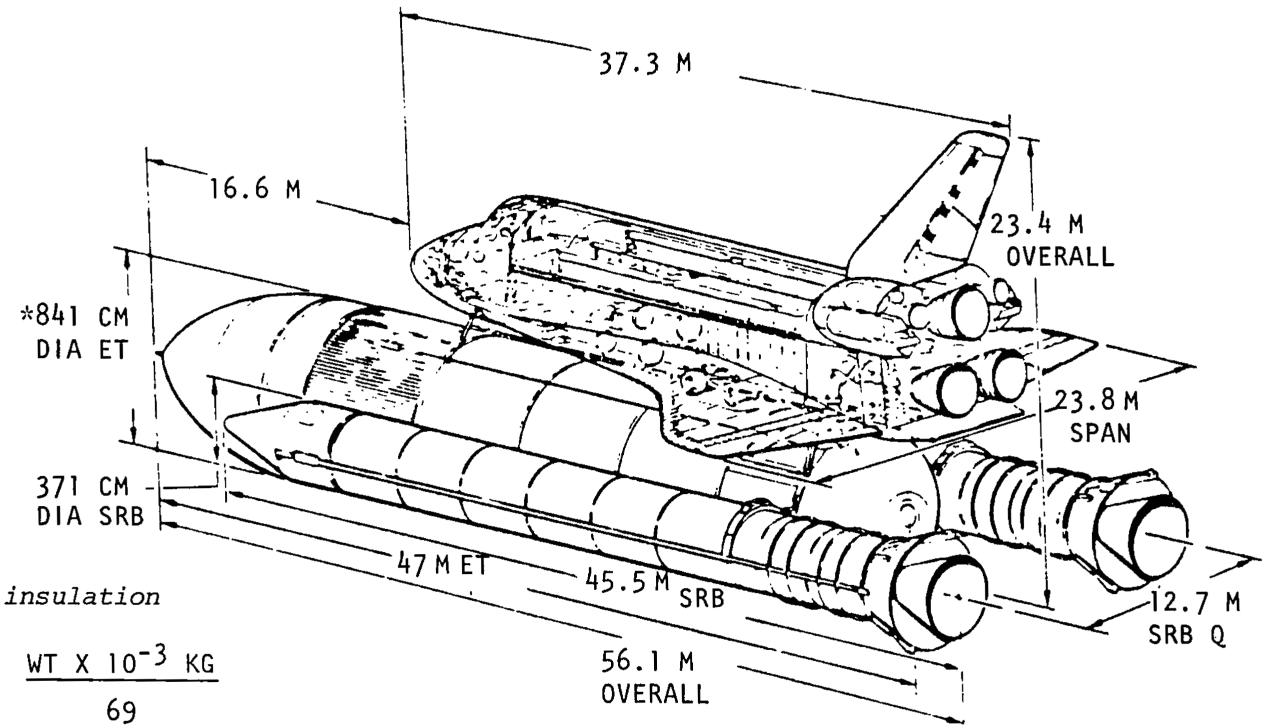
- SLIGHTLY HIGHER RECURRING COST
    - GREATER NUMBER OF CONSTRUCTION CREW
    - MORE PROPELLANT CONSUMED
- MORE FREQUENT FLIGHTS
- MORE EFFLUENT DEPOSITED IN UPPER ATMOSPHERE

1.5 PERSONNEL TRANSFER SYSTEM

The personnel transfer system consists of three basic elements: (1) a personnel launch vehicle (PLV) to transfer construction personnel from earth to LEO; (2) a personnel orbital transfer vehicle (POTV), a chemical propulsive stage to transfer the PM from LEO to GEO; and (3) the PM, a self-contained crew/personnel module containing all the necessary guidance, navigation, communication, and life support systems for construction crew transfer from LEO to GEO. Only the PLV is discussed in this section.

The PLV is a derivative or growth version of the currently defined Space Shuttle Transportation System (STS). The configurations selected for SPS studies are representative of various growth options evaluated in company-funded studies and NASA contracts.

The current STS configuration is depicted in Figure 1.5-1.



Less external insulation

	WT X 10^{-3} KG
ORBITER (INERT)	69
ORBITER (LIFTOFF)	115
ET (LIFTOFF)	738
SRB (LIFTOFF)	1167
PAYLOAD	29
GROSS (LIFTOFF)	2020

TO 300 KM CIRCULAR ORBIT @ 28.5° INCLINATION

Figure 1.5-1. Baseline Space Shuttle Vehicle

1.5.1 Personnel Launch Vehicle (Reference)

The personnel launch vehicle (PLV) is used to transport cargo and personnel to low earth orbit during the demonstration phase of the program and only personnel during the commercial phase.

The PLV is derived from the current Space Shuttle system. The vehicle consists of a winged liquid propellant flyback booster that employs four O_2/CH_4 engines similar to the HLLV booster, a resized smaller version of the Space Shuttle external tank, and the Space Shuttle orbiter. The payload capability to the LEO base in a 447 km/31-degree orbit is approximately 89 MT. The configuration, vehicle characteristics, and engine characteristics are shown in Figure 1.5-2.

VEHICLE CHARACTERISTICS (KG)

GLOW	2,714,750
BLOW	1,959,140
W_{P1}	1,688,820
OLOW (ET)	666,880
W_{P2}	551,720
PAYLOAD	88,730

ENGINE CHARACTERISTICS

STAGE	E	NO.	TYPE	I_{SP} (SL/VAC)	THRUST (VAC)
1	60	4	HIGH P_C LO_2/LCH_4	318.5/352	2.15×10^6 LBF 9.564×10^6 N
2	77.5	3	SSME	363.2/455.2	$.470 \times 10^6$ LBF 2.091×10^6 N

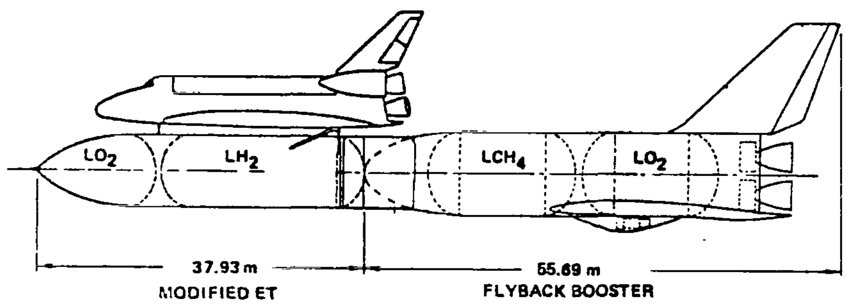


Figure 1.5-2. Shuttle-Derived PLV

Studies conducted early in the SPS contract analyzed a PLV employing a ballistic recoverable booster, modified ET and orbiter. However, once a decision was made to use a two-stage winged HLLV, it appeared reasonable to develop a PLV that also employed a winged booster in order to provide an evolutionary path. The reference configuration is the result.

The mass statement for the flyback booster is shown in Table 1.5-1. The mass statement for the ET which has a propellant load of 547 MT (rather than 703 MT when used with the Space Shuttle) is shown in Table 1.5-2.

1.5.2 Personnel Launch Vehicle (Alternate Concept)

As stated previously, should the alternate (smaller payload) HLLV configuration be adopted, personnel would be transferred from earth to LEO in that vehicle during the operational phase of the SPS; and even if a larger payload HLLV is adopted, it appears that using the HLLV for personnel transport (along

Table 1.5-1 Flyback Booster Mass Summary

Element		kg
Wing		31,940
Tail		4,930
Body		68,490
Induced environ. protection		9,050
Landing and auxiliary systems		9,710
Propulsion—ascent		51,320
Propulsion—RCS		960
Propulsion—flyback		13,800
Prime power		1,190
Elec. conv. and distribution		960
Hyd. conv. and distribution		4,230
Surface controls		2,020
Avionics		1,450
Environmental control		210
Growth allowance		16,200
Dry mass		(216,460)
Residuals and reserves		12,700
Landing mass		(229,160)
Flyback fuel		26,260
Inflight losses	3,900	
Inert mass		(259,320)

Table 1.5-2. ET Mass Summary

Element		kg
Structures		21,146
LO_2 tank	4,446	
Intertank	3,276	
LH_2 tank	13,424	
Thermal protection		1,631
Propulsion and mech. sys.		1,710
Electrical sys.		66
ORB attachments		1,492
Change uncertainty		686
ET inert mass		26,731
Unusables		1,530
ET meco mass		28,261

with cargo) might still be the preferred option. However, it has been determined that an interim vehicle for both cargo and personnel would be desirable during the SPS development and pilot plant stages of the program. A "minimum change" growth version (PLV) is shown in Figure 1.5-3. As indicated in the figure, the growth version or PLV is achieved by replacing the existing solid rocket boosters (SRB) with a pair of liquid rocket boosters (LRB). The existing orbiter and external tank are used in their current configuration. The added performance afforded by the LRB increases the orbiter payload capability to the reference STS orbit by approximtely 54%, or a total payload capability of 45,350 kg (100,000 lb).

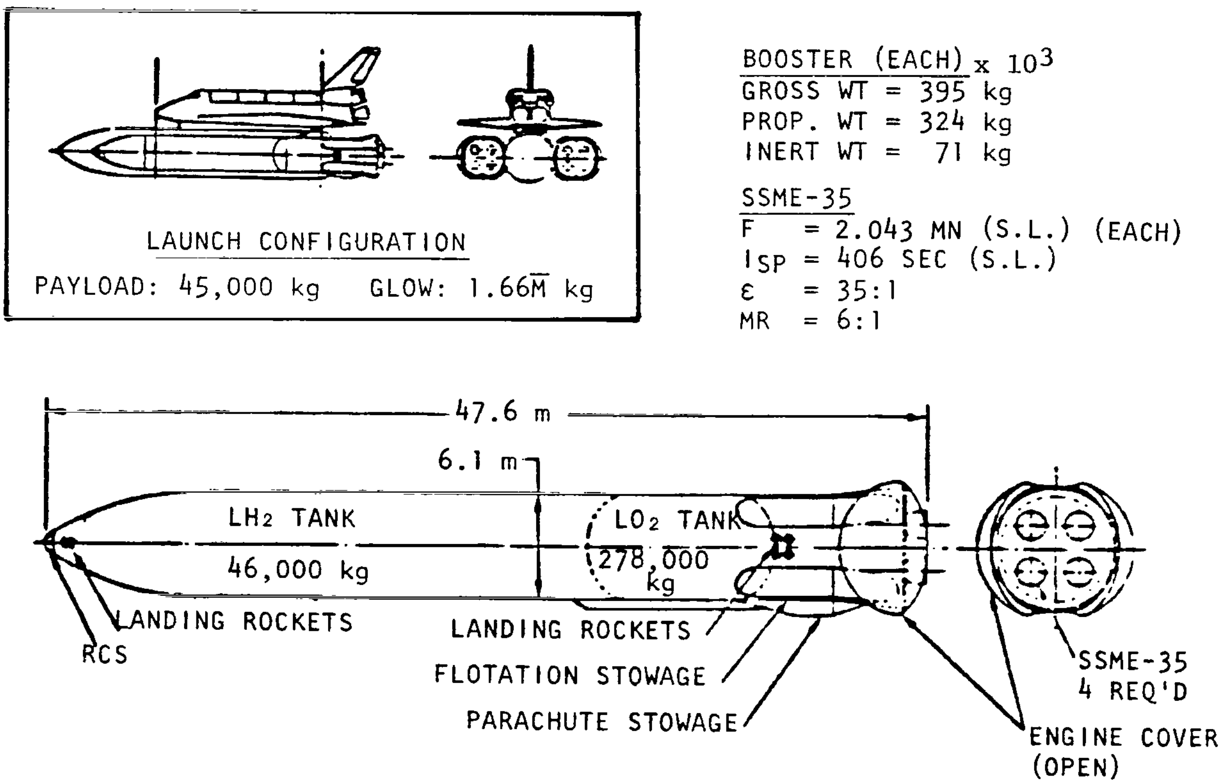


Figure 1.5-3. LO_2/LH_2 SSME Integral Twin Ballistic Booster

The LRB illustrated in Figure 1.5-4 has a gross weight of 395,000 kg made up of 324,000 kg of propellant (278,000 kg of LO_2 and 46,000 kg of LH_2), and 71,000 kg of inert weight. The overall length of the LRB is 47.55 m with a nominal diameter of 6.1 m. Four Space Shuttle main engine (SSME) derivatives are employed with a gross thrust of 8.17 MN (sea level), providing a liftoff thrust-to-weight ratio of approximtely 1.3.

The STS-derived heavy-lift launch vehicle (STS-HLLV), employed in the precursor phase of SPS is derived by replacing the STS orbiter on the PLV with a payload module and a reusable propulsion and avionics module (PAM) to provide the required orbiter functions. The PAM may be recovered ballistically or, preferably, as a down payload for the PLV. These modifications yield an STS-HLLV with a payload capability of approximately 100,000 kg (Figure 1.5-4).

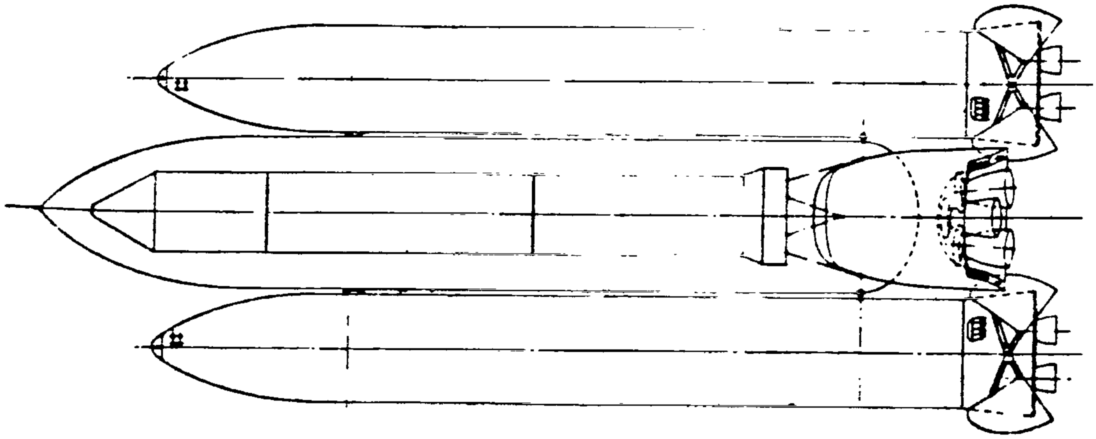


Figure 1.5-4. STS-HLLV Configuration

Unique design features of the LRB, as compared to an expendable liquid booster system, are presented in Table 1.5-3. The necessity to preclude ice damage to the orbiter requires the LH_2 tank to be located forward since the insulation system, which must be internal to avoid water impact damage, is not compatible with LO_2. In addition, the thickness of insulation required on the LH_2 tank is about twice that required to maintain propellant quality.

Table 1.5-3. Shuttle LRB Unique Design Features

ORBITER ICE DAMAGE AVOIDANCE	• LH_2 TANK FWD, INSULATED TO PRECLUDE ICE
ENTRY PROVISIONS	• RCS TO ORIENT BOOSTER • CLAMSHELL COVERS FOR ENGINE PROTECTION • HEAT SINK STRUCTURE
WATER LANDING PROVISIONS	• PARACHUTES & RETRO-SUSTAINER ROCKETS • INTERNAL LH_2 TANK INSULATION • RCS FOR WAVE ALIGNMENT • REINFORCED STRUCTURE • AVIONICS TO CONTROL LANDING
WATER PROTECTION PROVISIONS	• CLAMSHELL COVER FOR ENGINE PROTECTION • SEALED STRUCTURE • FLOTATION BAGS FOR ORIENTATION
RECOVERY PROVISIONS	• RADIO BEACON AND LIGHTS • HANDLING HARDPOINTS

Other unique features are the provisions required for entry, water landing, water protection, and recovery. In addition to these supplementary provisions, the structure (unlike that of an expendable system) must act as a heat sink for reentry heat loads, be reinforced to absorb landing loads, and be sealed to prevent sea water contamination.

The basic structure consists of the propellant tank assembly and an engine compartment. The tank assembly is made up of the LH_2 tank and the LO_2 tank, with a common bulkhead similar to the Saturn S-II separating the propellants. The engine compartment comprises a skirt section, thrust structure, launch support structure, heat shield, and movable covers that protect the engines during atmospheric reentry and water recovery. The locations of the landing rockets, the APU, avionics packages, parachutes, the flotation bag, and RCS system are indicated in Figure 1.5-3.

The structural design of a recoverable LRB is governed by five basic load conditions—water impact, high-Q boost, internal tank pressures, prelaunch loads, and maximum thrust.

The nose cap primary structure and tank frames are designed to withstand loads due to initial water impact and subsequent water penetration with resultant slap-down loads being reacted by the tank ring frames. Launch maximum aerodynamic pressure (high-Q) loads influence the structural design of the main frames, forward portions of the LH_2 tank, and engine thrust structure. The LH_2 and LO_2 tank walls and domes are structurally sized for maximum internal tank pressures. Equivalent tank wall thickness due to internal pressure exceeds those required by other load conditions. The maximum body bending moment occurs at the aft end of the booster. The design of the aft skirt and frames is governed by prelaunch loads when the boosters are loaded and free-standing on the launch pad. The ET attachment thrust structures are designed by maximum thrust loads at launch.

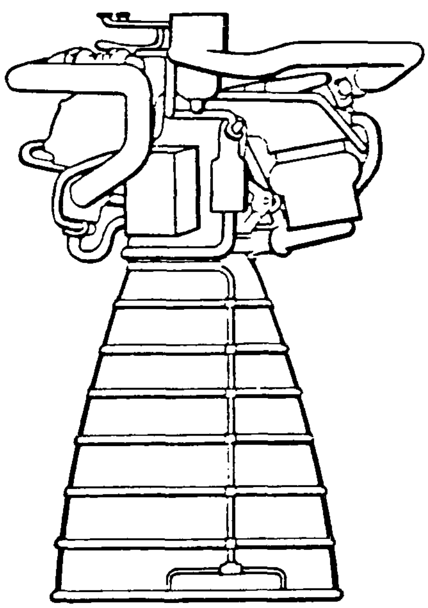
There are four structural attachments between the ET and each booster. The three aft attachments take lateral shears and bending moments, and the forward attachment takes lateral shears and thrust loads. This four-point interface is statically determinate, so that structural loads are not induced by deformations in the adjacent body. This interface arrangement is the same as that for the baseline Shuttle.

The electrical interface between the booster and ET is accomplished by external cables mounted on one of the aft struts. They are separated at pull-away connectors when the strut is cut. The increased number of wires required for the LRB may increase the number of cables and connectors.

The LRB utilizes a derivative of the Space Shuttle main engine (SSME). The only difference between the LRB engines and the SSME is in nozzle expansion ratio—35 in lieu of 77.5 to 1. The SSME-35 and its characteristics are depicted in Figure 1.5-5.

After the boosters separate from the orbiter ET, the engine covers close and the reaction control system (RCS) fires to pitch the boosters over and align them for reentry (Figure 1.5-6). The drogue and then the main chutes deploy to slow ascent. Retro motors are fired to minimize landing velocity. Upon splashdown, the chutes release and flotation bags inflate at the aft end to hold the engine area out of the water.

The booster will be commanded by the recovery vessel to start depressurization (one propellant at a time) upon landing. The recovery vessel will pick



THRUST, 10^6 N	2.043 (SL) 2.239 (VAC)
EXPANSION AREA RATIO	35:1
CHAMBER PRESSURE, PSIA	3230
MIXTURE RATIO	6.0:1
SPECIFIC IMPULSE, SEC	406 (SL) 445 (VAC)
ENGINE WEIGHT, KG	2876
SERVICE LIFE	
HOURS	7.5
START	55
ENVELOPE	
LENGTH, CM	371
DIAMETER, CM	
POWER HEAD	267
NOZZLE EXIT	160

Figure 1.5-5. Liquid Rocket Booster Main Engine (SSME-35)

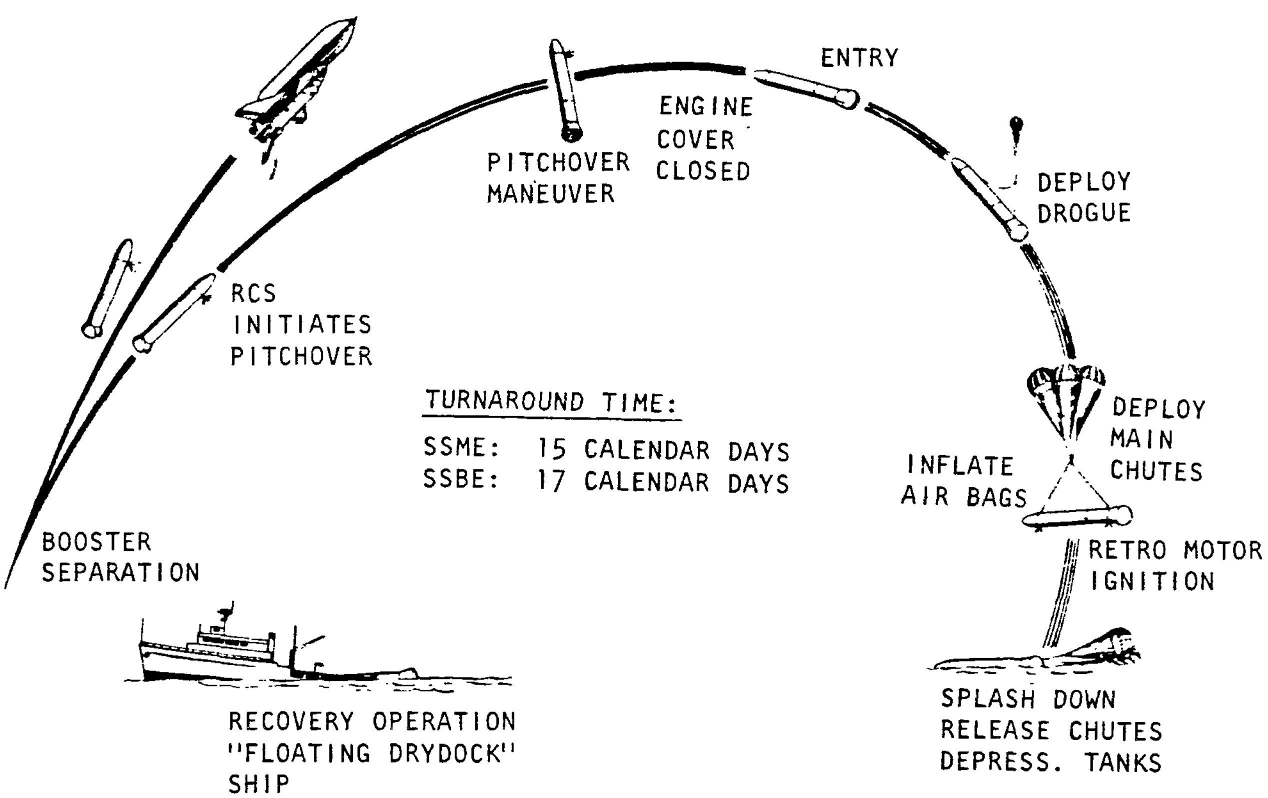
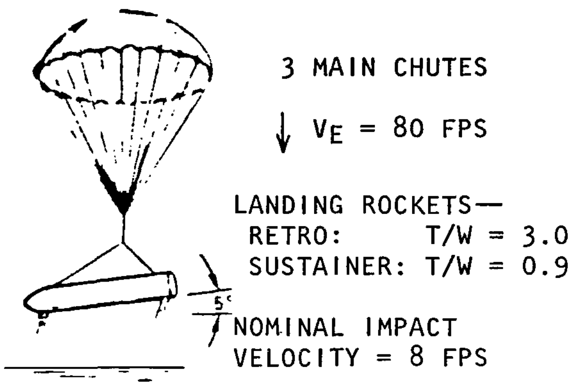


Figure 1.5-6. Integral Booster Recovery Concept

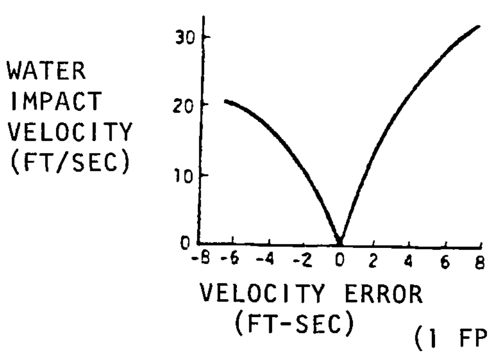
up chutes during booster depressurization. After the booster is depressurized, the aft end of the ship is aligned to the booster, the aft gate is lowered, and the compartment is flooded (<30 minutes). A craft is then launched to attach tow lines to the booster, which is then pulled into the ship. The booster is positioned over the contour supports or lifted in a crane cradle, rear gate is closed, and the compartment is pumped dry. The booster undergoes washdown and inspection as the ship returns to port. Utilizing this system, a booster can be retrieved and returned to port in 20 to 24 hours maximum (a function of distance and sea state). Booster recovery will be accomplished in waves up to 2.5 meters. The booster recovery system is shown in Figure 1.5-7.



SYSTEM ERRORS

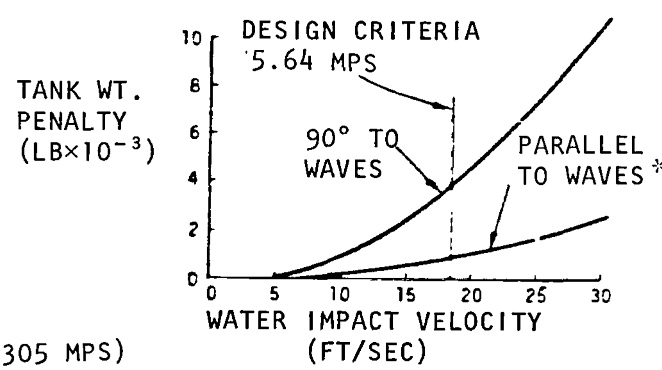
ERROR SOURCE	VALUE
CHUTE VARIATIONS	±1.43 M/SEC
AIR DENSITY	±1.06 M/SEC -0.72 M/SEC
THRUST	±1%
WEIGHT	±1186 KG
ALTIMETER	±0.61 M
SIGNAL TIME	±1.22 M

EFFECT OF VELOCITY ERRORS ON IMPACT VELOCITY



(1 FPS = 0.305 MPS)

STRUCTURAL WEIGHT PENALTY



*AZIMUTH CONTROL

Figure 1.5-7. Booster Recovery System

Should the final SPS HLLV lead to a vehicle concept capable of personnel as well as cargo transport, this PLV would be phased out of the SPS program prior to first satellite construction.

2.0 ORBIT-TO-ORBIT SYSTEMS

Independent of SPS assembly location, there is a significant demand for OTV transportation from LEO to GEO due to the magnitude of the SPS program. For the LEO-assembled SPS, hardware flights dominate the early years but in later years, logistics flights become a significant factor. Since propellant to support these OTV flights represents a significant portion of the total HLLV payloads, alternate advanced OTV concepts having high specific impulse appeared to be worthwhile candidates to satisfy cargo mass transfer requirements.

Because of the need for rapid transfer of personnel to and from GEO, a conventional chemical propulsion element is deemed to best satisfy that requirement.

2.1 SILICON ELECTRIC OTV (REFERENCE UPDATE)

The cargo Orbit Transfer Vehicle is used to transport satellite components from the LEO staging depot to the GEO construction base. This vehicle uses electric propulsion and is referred to as the electric orbit transfer vehicle (EOTV).

The selected EOTV configuration for the Silicon Cell Reference SPS is shown in Figures 2.1-1 and 2.1-2 and consists of four solar array bays, with each bay formed by a pentahedron. The apexes of the pentahedrons are tied together to serve as a mounting location for the payload and propellant tanks. This location provides a good moment of inertia balance to minimize gravity

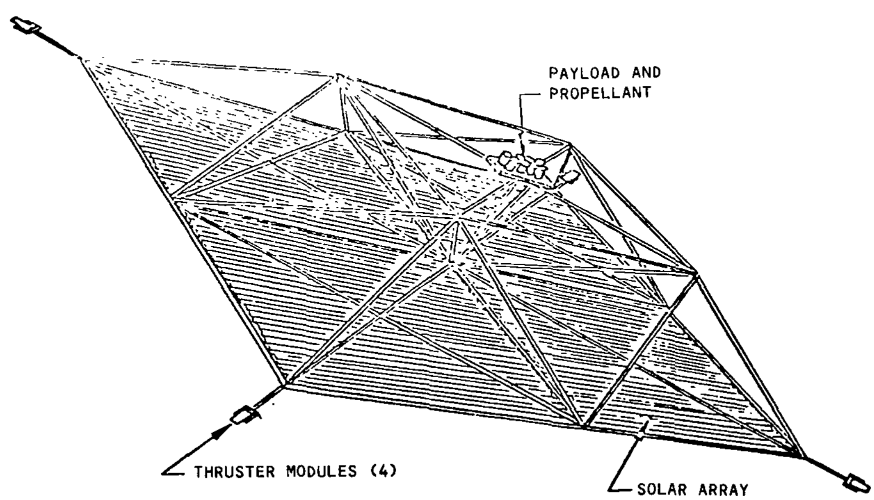


Figure 2.1-1. Si EOTV Configuration Concept

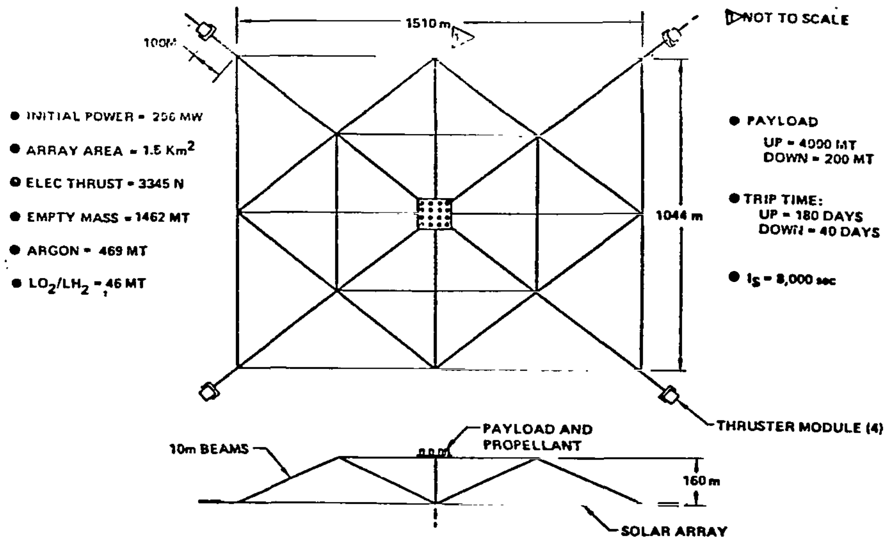


Figure 2.1-2. Si Electric OTV Configuration

gradient torque control requirements and simplifies the docking of the payloads as well as propellant tankers. Thruster modules are attached to beams protruding from the four corners of the configuration. Power for the thrusters is drawn from solar arrays in the bay adjacent to the thruster module. The vehicle is sized to deliver 4000 metric tons and return 200 metric tons with an uptrip time of 180 days and down time of 40 days, with a specific impulse of 8000 sec. The total dry mass of the vehicle is 1462 metric tons while the total propellant loading is approximately 500 metric tons. The 1510 m dimension of the configuration is a function of cell size and voltage requirements.

In terms of power generation and distribution systems, the EOTV is divided into four separate bays with each bay providing power to a thruster module as shown in Figure 2.1-3. Each bay is divided into fifty-four 14.5 m segments and produces approximately 74 MW. The optimum voltage was found to be 2685 V as shown in Figure 2.1-4. Each segment consists of 20 strings, with each string in turn consisting of 498 panels. Each of the panels include (140) 5x10 cm cells. The cell shape change is the result of compromise between a desired square satellite shape and the power and voltage requirements dictated by the propulsion system.

Power buses are located on three sides of each bay of the EOTV as illustrated in Figure 2.1-5. Each bay is divided into 7 sectors in order to minimize the impact on the switch gear complexity should a fault occur. Five sectors each collect power from 8 segments while two sectors collect power from 7 segments. A bus from each sector runs to the associated thruster module where the power is processed. Each of the buses is 1mm thick by 80 cm deep. The optimum bus temperature was found to be 50 C as shown in Figure 2.1-6.

Electric propulsion modules are located at four corners of the EOTV. The key characteristics of each module are shown in Figure 2.1-7. Each module

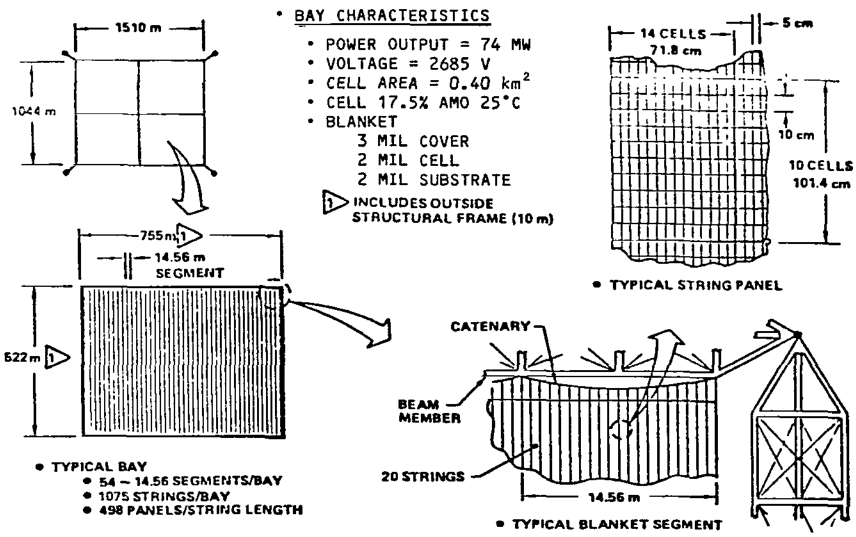


Figure 2.1-3. Si EOTV Power Generation System

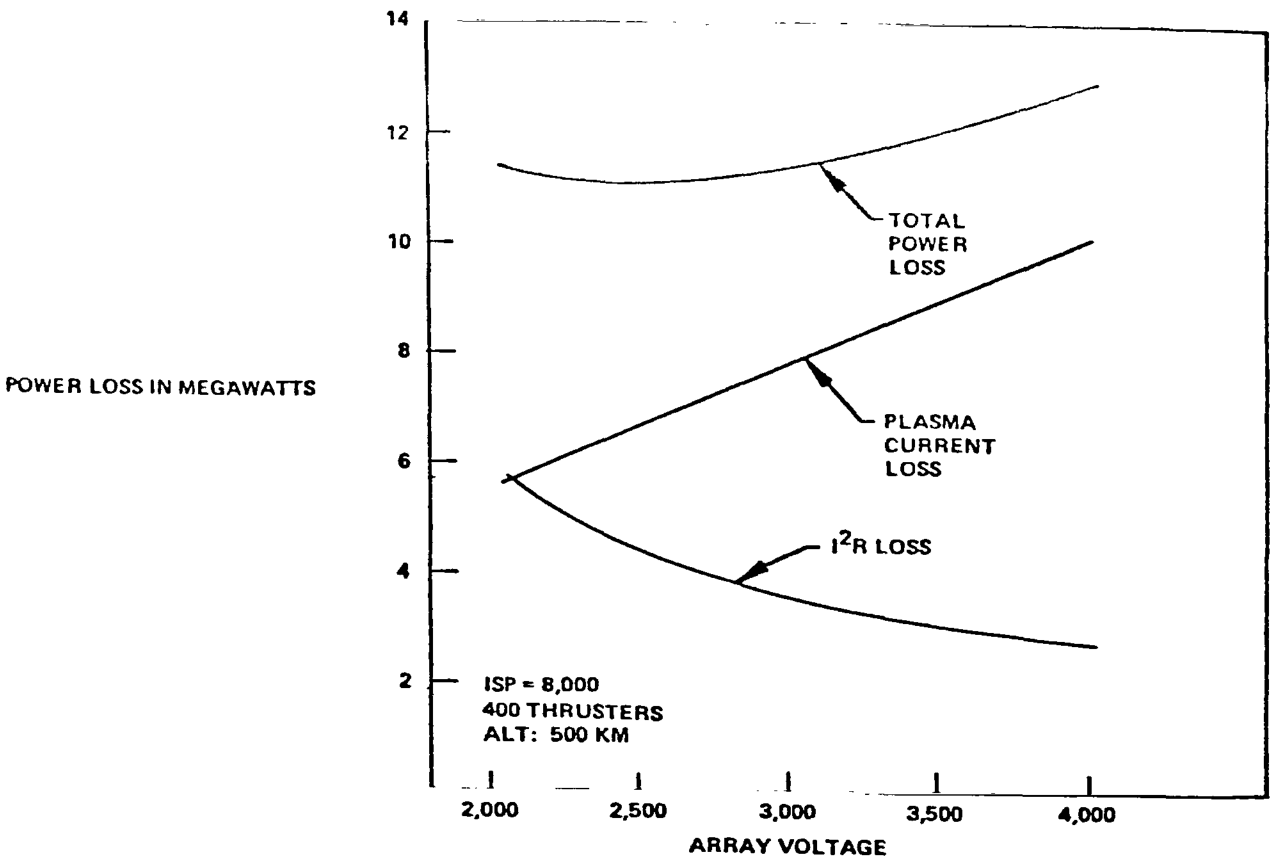


Figure 2.1-4. Optimum Array Voltage

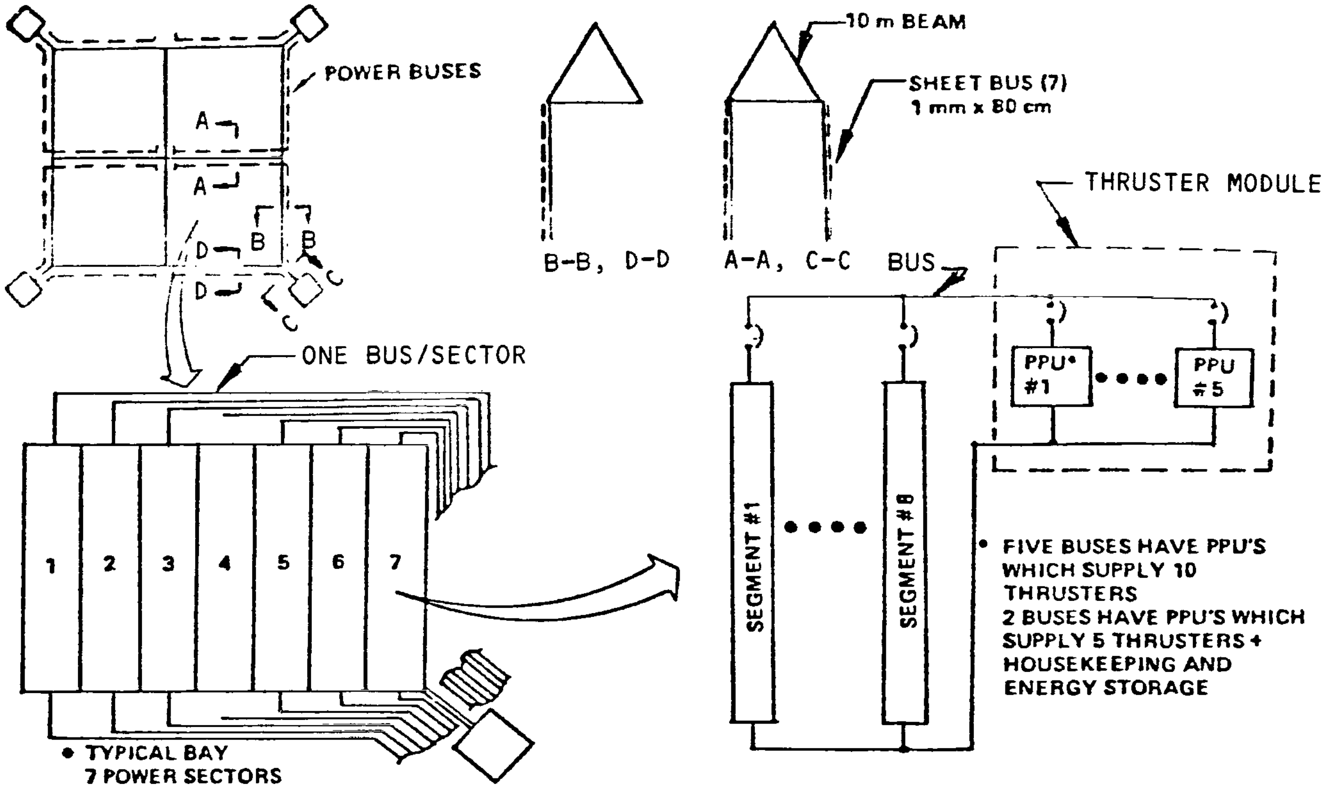


Figure 2.1-5. Power Collection and Distribution

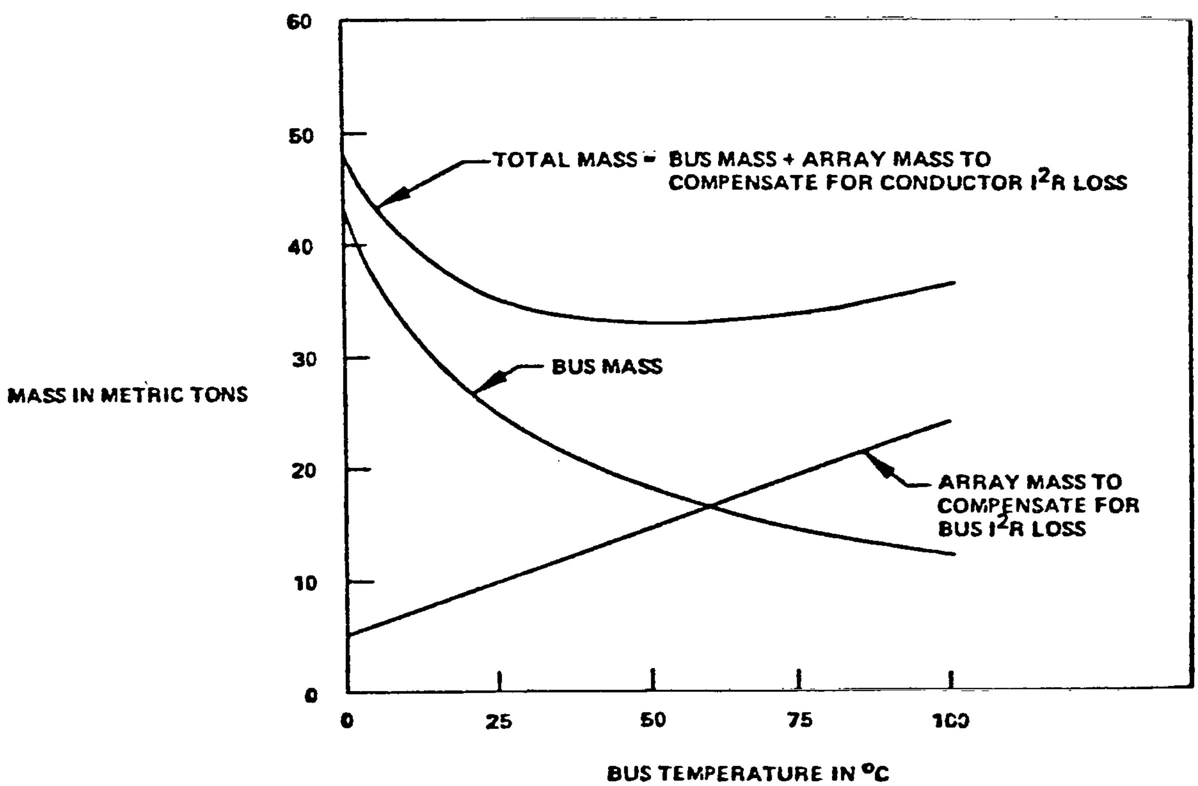
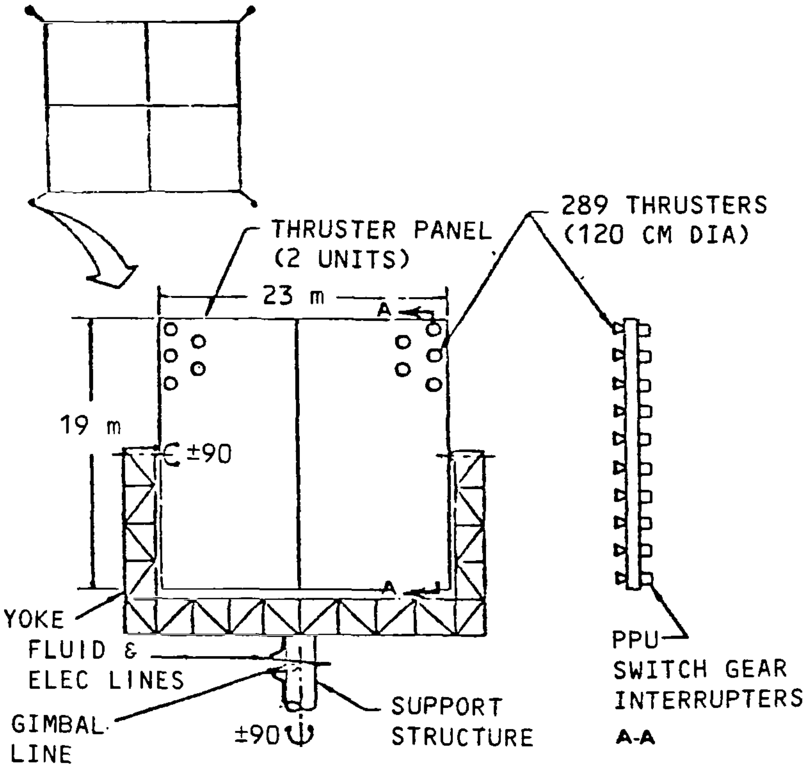


Figure 2.1-6. Optimum Power Bus Temperature

consists of a gimbal, yoke, thruster panel containing thrusters and power processing units and a thermal control system. For the reference design, 289 thrusters are used at each of the four corners. The principal components of the 1.2 m diameter ion thruster and performance characteristics associated with a specific impulse of 8000 sec are shown respectively in Figure 2.1-8 and Table 2.1-1.



- THRUSTER POWER SUPPLY
  - DIRECTLY FROM ARRAY
    - NO PROCESSING
    - NO REGULATION
  - NO PROCESSING
    - ARRAY REGULATION
  - ✓ PROCESS ALL POWER
- TYPE OF PROCESSING
  - MOTOR/GENERATOR
  - ✓ SOLID STATE
- PROCESSING THERMAL CONTROL
  - ACTIVE RADIATOR
  - 915 M^2
  - LIMIT ELECTRONICS TO TO 200°C

Figure 2.1-7. Si Electric Propulsion System

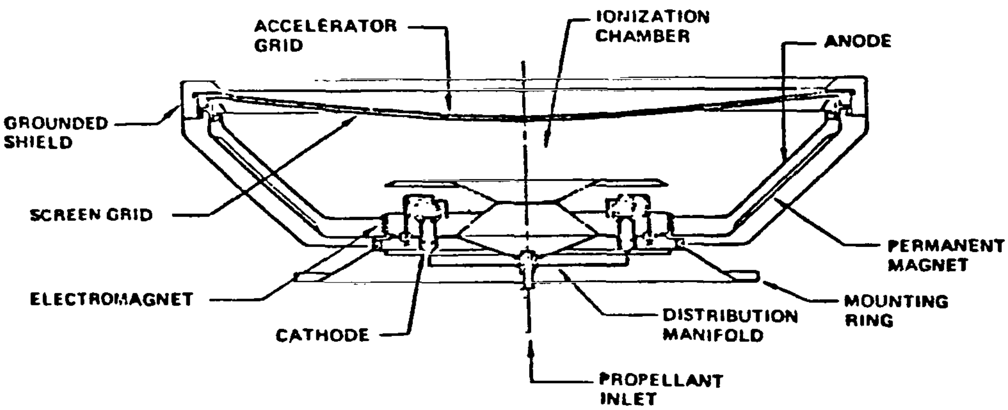


Figure 2.1-8. 120 CM Argon Ion Thruster

Table 2.1-1. Selected 1.2 m Argon Ion Thruster Characteristics

FIXED CHARACTERISTICS	
BEAM CURRENT:	80.0 AMPS
ACCEL VOLTAGE:	500.0 V
DISCHARGE VOLTAGE:	30.0 V (FLOATING)
COUPLING VOLTAGE:	11.0 V
DBL ION RATES:	0.16 (J2/J1)
NEUTRAL EFFLUX:	4.8384 AMP EQUIV
DIVERGENCE:	0.98
DISCHARGE LOSS:	187.3 EV/ION
OTHER LOSS:	1758.0 W
UTILIZATION	0.892 W
LIFE:	8000 HR
WEIGHT:*	50. KG
SELECTED CHARACTERISTICS	
SCREEN (BEAM) VOLTAGE:	1700 V
INPUT POWER	130 KW
THRUST:	2.9 N
EFFICIENCY:	78

*WEIGHT PREDICTION COURTESY OF T. MASEK OF HRL.

Several methods were considered for supplying power to the thrusters. One of these options involves obtaining power directly from the arrays with no processing or no regulation. The chief disadvantage in this option is that the voltage is decreasing at the same time the power is degrading as a result of radiation damage. As the flight proceeds, the lower voltage will result in a loss of approximately 1000 sec of specific impulse. A second option regulates and sectionalizes the array so that as additional power is required, additional sectors can be switched into operation. The main disadvantage of this concept is the extremely complicated switch gear system. The power supply method employed for the Si concept involves processing all the power. The array voltage generated in this concept is the optimum voltage from the standpoint of I^2R and plasma losses. The resulting voltage is 2685 V as compared to 1700 V required by the thrusters. A complete comparison of these concepts was not accomplished, however, the all-processing method appeared to be the most straightforward and since some of the power requires processing, this method was selected for the Si reference. The type of processing equipment selected was solid state due to its longer MTBF. Thermal control of the processing equipment is required and is accomplished using an active radiator.

The mass characteristics of the EOTV are summarized in Figure 2.1-9. The empty mass for the configuration is shown for both initial (mid-term) and final values. The most significant change was that associated with the solar array mass, which increased as a result of using a more accurate model reflecting the power requirements for I^2R losses, storage provisions, changing power conditioning efficiencies as a result of using solid state equipment rather than motor generator equipment and also a revision in the radiation degradation analysis. These changes to the solar array, in turn, have reflected or resulted

ITEM	• EMPTY MASS (MT) MIDTERM	FINAL
POWER GEN & DISTRIB	(736)	(951)
SOLAR ARRAY	608	780 [1]
STRUCTURE	95	122
DISTRIBUTION	33	42
ENERGY STORAGE	–	7
ELECTRIC PROPULSION	(447)	(496)
THRUSTERS	71	79
POWER CONDITIONING	195	219
THERMAL CONT	55	88
STRUCT/MECH	80	61
PROPELLANT FEED SYS	46	49
AUXILIARY SYSTEMS	(12)	(15)
TOTAL	1195	1462

• STARTBURN MASS (MT)

PAYLOAD		4000
EMPTY		1462
PROPELLANT		
	ARGON	469
	LO_2LH_2	46
		5977

[1] MORE ACCURATE MODEL
- POWER REQ'T ADDITIONS
  - I^2R & STORAGE
  - PPU EFF
  - REVISED RADIATION DATA
- ARRAY AREA
  - BASED ON DESIGN POWER NOT ELEC
- OTHER CHANGES ARE RESULT OF [1]

Figure 2.1-9. Si EOTV Mass Summary

in changes in all other elements of the vehicle resulting in approximately a 300 metric ton increase over the initial values. Accordingly, the startburn mass also reflects a 300 metric ton increase over the initial value.

2.2 GALLIUM ARSENIDE ELECTRIC OTV (REFERENCE UPDATE)

The rationale for electric OTV selection over the conventional chemical systems is clearly illustrated in Figure 2.2-1. Because of the limited

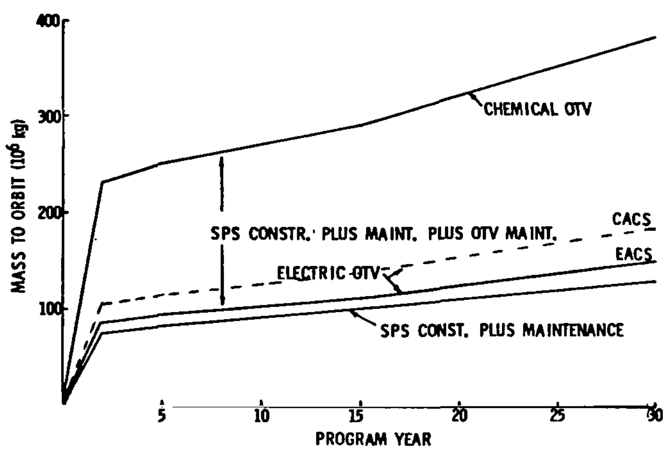


Figure 2.2-1. Mass-to-Orbit Requirements

specific impulse of chemical rocket systems (i.e., <500 sec), the mass to low earth orbit requirement is increased approximately three-fold due to chemical propellant requirements. Also indicated, is a comparison of mass to orbit requirements for a chemical attitude control system (CACS) versus an electric thruster attitude control system (EACS). Again, a decreased mass to orbit (i.e., ~25%) requirement is indicated for an EACS. Since transportation costs from earth-to-LEO is the prime contributor to overall SPS transportation cost, the electric system offers a considerable cost advantage over chemical systems.

The major technology options for the electric OTV propulsion subsystem concern the thruster type, size, and design operating point; the power interfaces between the thrusters and the solar array or other primary source; and the propellant type, storage, and distribution.

Thruster types considered for this application were ion bombardment, magnetoplasmadynamic (MPD), and resistojet. Other types, such as RF excitation, were rejected a priori because of development risk and lack of evidence of performance superior to the types first mentioned.

Resistojet thrusters were discarded because their low I_{sp} (<1200 s) offers insufficient propellant mass savings compared to chemical propulsion. MPD thrusters were initially considered on the basis of reported I_{sp} values up to 10,000 s. An independent investigation established that high I_{sp} values were measured in small vacuum chambers which allowed exhaust propellant to be recirculated through the thruster; this appeared to reduce the propellant flow rate and proportionately increase I_{sp}. The state-of-the-art I_{sp} is actually believed to be in the range of 2000 to 2500 s, with 4000 s the realistic growth potential. For this reason, and because MPD thruster development has been largely abandoned except for long range research at Princeton University, this type was dropped from immediate consideration.

The surviving candidate, for which a current development program has established reliable performance data, is the ion bombardment thruster.

Conventional power conditioners for ion bombardment thrusters regulate all supplies, serving as an interface between the power source (solar array) and the thrusters. Various so-called direct-drive concepts have been proposed in which some of the thruster supplies are obtained directly from the solar array. This approach reduces power conditioner mass, power loss, and cost, and improves propulsion system reliability.

The power conditioners proposed for the SPS propulsion system process only the low-voltage fixed power. The other supplies are taken directly from solar arrays. The beam power is obtained from the main SPS or OTV solar array. To avoid significant power loss from plasma discharge, the array voltage is maintained at 2000 V. Solar eclipse produces solar cell temperature, efficiency, and output voltage variations which cause acceptable transients in the beam voltage during the first few minutes after each eclipse.

The ion thruster propellant selection criteria are availability, storability, absence of serious environmental impacts, cost, demonstrated performance, and technical suitability. Technical factors are as follows:

- *High specific impulse* - At a given beam voltage, $I_{sp} \sim 1/\sqrt{m_i}$, where m_i is the ion mass.
- *High thrust* - At a given beam voltage and current, $T \sim m_i$.
- *Low vaporization temperature* - Allows instantaneous thruster restart after solar eclipses without power storage for preheating.
- *Low first-ionization potential* - Limits thruster discharge loss and minimizes the efficiency loss due to neutral atoms.
- *High second-ionization potential* - Minimizes the efficiency loss due to multiple ions.

Obviously, the first two factors are mutually contradictory and are best compromised by an ion of medium mass.

The propellants for which ion bombardment thruster experimental data exist are evaluated against the above criteria in Table 2.2-1. The selection of argon is self-evident.

Table 2.2-1. Ion Propellant Selection Criteria

PROPELLANT	AVAILABILITY	STORABILITY	ENVIRONMENTAL FACTORS	COST ($/KG)	THRUSTER TECHNOLOGY STATUS	ATOMIC WEIGHT	VAPORIZATION TEMP. (K)	IONIZATION POTENTIALS (V) 1	IONIZATION POTENTIALS (V) 2
ARGON	HIGH (0.9% OF AIR)	CRYOGENIC	INERT	0.50	GROUND TESTS	39.9	97	15.76	27.62
CESIUM	PROBABLY INADEQUATE	SOLID	EXTREMELY REACTIVE	300	SPACE FLT	132.9	951	3.89	25.1
XENON	VERY SCARCE	CRYOGENIC	INERT	1000	LABORATORY DEVELOP.	131.3	167	12.13	21.2
MERCURY	MARGINAL	LIQUID	TOXIC	55	SPACE FLT	200.6	530	10.43	19.13

The argon thruster design and performance characteristics used were based on work conducted at NASA Lewis Research Center.

A thruster aperture in the 100 cm range has been selected. Experience with the development of 8- and 30-cm thrusters, now at an advanced stage, suggests that the performance of 100-cm thrusters can be analytically predicted with only minor deviations. The cathodes and ion extraction systems require major modifications. Multiple cathodes are employed to improve lifetime, reliability, and performance. It is assumed that more resistant cathodes can be constructed with lifetimes comparable to the OTV (10 years). However, the grid sets will have to be refurbished periodically because of positive ion bombardment.

The concept of a dished grid, which proved successful for the 30-cm mercury thruster, appears feasible for the larger argon thrusters. Dished grids, which enable closer-spaced accelerator grids, effectively result in greater thrust density but impose a limit on the specific impulse.

The GaAs electric orbital transfer vehicle concept, Figure 2.2-2, is based on the same construction principles of the GaAs reference satellite configuration. The commonality of the structural configuration and construction processes

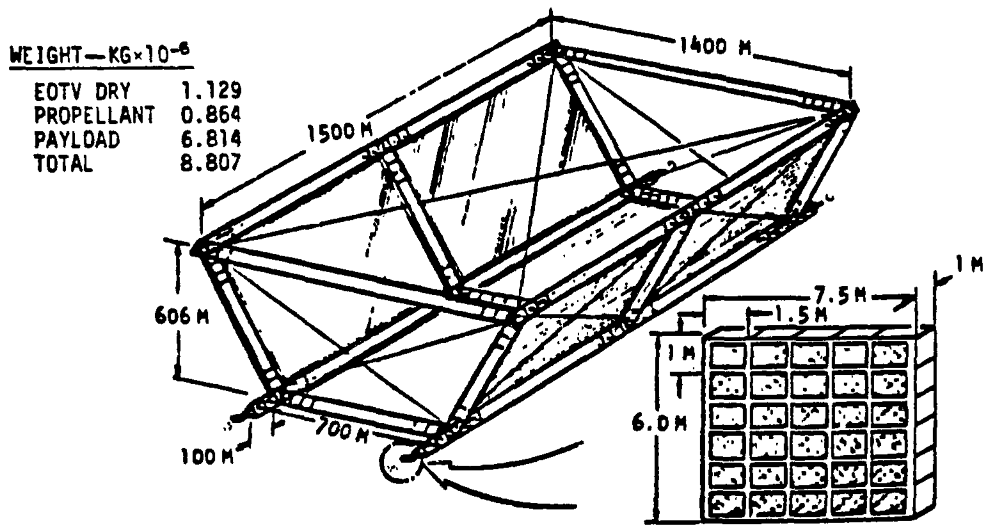


Figure 2.2-2. GaAs EOTV Configuration

with the satellite design is evident. The structural bay width of 700 m (solar array width of 650 m) is the same as that of the satellite. The structural bay length is reduced from 800 m to 750 m for compatibility with the lower voltage requirement of the EOTV. The concept utilizes electric argon ion thruster arrays.

The primary assumptions used in EOTV sizing are summarized in Table 2.2-2. The orbital parameters are consistent with SPS requirements and the delta "V" requirement was taken from previous SEP and EOTV trajectory calculations. A 0.75 delta "V" margin is included in the figure given.

During occultation periods, attitude hold only is required (i.e., thrusting for orbital change is not required).

Since thruster grid changes are assumed after each mission, a minimum number of thrusters are desired to minimize operational requirements.

An excess of thrusters are included in each array to provide for potential failures and primarily to permit higher thrust from active arrays when thrusting is limited or precluded from a specific array due to potential thruster exhaust impingement on the solar array or to provide thrust differential as required for thrust vector/attitude control. A 5% specific impulse penalty

Table 2.2-2. EOTV Sizing Assumptions

• LEO ALTITUDE - 487 KM @ 31.6° INCLINATION
• SOLAR INERTIAL ORIENTATION
• LAUNCH ANY TIME OF YEAR
• 5700 M/SEC ΔV REQUIREMENT
• SOLAR INERTIAL ATTITUDE HOLD ONLY DURING OCCULTATION PERIODS
• 50° PLUME CLEARANCE
• NUMBER OF THRUSTERS - MINIMIZE
• 20% SPARE THRUSTERS - FAILURES/THRUST DIFFERENTIAL
• PERFORMANCE LOSSES DURING THRUSTING - 5%
• ACS POWER REQUIREMENT - MAXIMUM OCCULTATION PERIOD
• ACS PROPELLANT REQUIREMENTS - 100% DUTY CYCLE
• 25% WEIGHT GROWTH ALLOWANCE

was also applied to compensate for thrust cosine losses due to thrust vector/ attitude control.

An all-electric thruster system was selected for attitude control during occultation periods. The power storage system was sized to accommodate maximum gravity gradient torques and occultation periods. A very conservative duty cycle of 100% was assumed for establishing ACS propellant requirements. A 25% weight growth margin was applied as in the case of the SPS.

The solar array size is dictated primarily by the requirement to maintain the same construction approach as the satellite, consistent with specific EOTV voltage requirements. The solar array voltage must be as high as possible to reduce wiring weight penalties and to provide high thruster performance, yet, power loss by current leakage through the surrounding plasma must be minimized. At the proposed LEO staging base, with very large solar arrays and high efficiency cells, an upper voltage limit of 2000 volts is postulated.

Since GaAs solar cells are employed in this concept with a concentration ratio of 2 on the solar cell blanket, the resulting cell operating temperature of 125°C allows continuous self-annealing of radiation damage during transit through the Van Allen radiation belt.

The solar blanket width of the satellite (650 m) is retained for the EOTV. A blanket length (per bay) of 1400 m is determined by the solar cell string length required to achieve the desired operational conditions of 2000 V (string length of approximately 63.5 m). Eleven such strings result in a solar blanket length of approximately 700 m. Twenty-five meters of additional structural length at each end of the solar blanket are required to provide for catenary support. These considerations lead to the selection of a two-bay configuration with structural dimensions of 700x1500 m (solar blanket size 650x1400 m) with a total power output of 309 MW (includes 6% line losses).

The solar array weights were scaled from satellite weights and are summarized in Table 2.2-3.

Table 2.2-3. EOTV Solar Array Weight Summary, 10^{-6} kg

STRUCTURE		0.095
PRIMARY	0.041	
SECONDARY	0.054	
MECHANISMS		0.004
CONCENTRATORS		0.033
SOLAR PANELS		0.229
POWER DIST. & CONTROLS		0.262
MAINTENANCE PROVISIONS		0.003
INFORMATION MANAGEMENT		0.002
TOTAL		0.628

Having established the solar array operating voltage, the maximum screen grid voltage is established, which in turn fixes propellant ion specific impulse. In order to assure adequate grid life, to assure a minimum round trip capability of approximately 4000 hours, a maximum beam current of 1000 amp/m^2 was selected. Based on the available power and a desire to maintain reasonable thruster size, the remaining thruster parameters are established. A rectangular thruster configuration (1x1.5 m) is assumed. Primary thruster characteristics are summarized in Table 2.2-4.

Table 2.2-4. Argon Ion Thruster Characteristics

MAXIMUM TOTAL VOLTAGE, VOLT	4405
MAXIMUM OPERATING TEMP., °K	1330
SCREEN GRID VOLTAGE, VOLT	1880
ACCELERATOR GRID VOLTAGE, VOLT	-2525
BEAM CURRENT, AMP	1500
BEAM POWER, WATT	2.82×10^6
SPECIFIC IMPULSE, SEC	7963
THRUST, NEWTON	56.26

Based on the individual thruster power requirements and the available array power, 100 thrusters may be operated simultaneously. An additional 20 thrusters are added to provide a thrust margin when thruster array orientation might preclude firing due to potential ion impingement on the solar array. The thrusters are arranged in 4 arrays of 30 thrusters each. The thruster array mass summary is presented in Table 2.2-5. (The mass indicated is for all four arrays.)

Table 2.2-5. Thruster Array Mass Summary, kg

THRUSTERS & STRUCTURE	24,000
CONDUCTORS	6,000
BEAMS & GIMBALS	2,200
POWER PROCESSING	2,000
ATTITUDE REFERENCE SYSTEM	1,000
BATTERIES & CHARGER	154,000
TOTAL	189,200

The EOTV performance is based on a 120 day trip time from LEO-GEO (obtained from trade studies). Knowing the propellant consumption rate of the thrusters and the thrusting time, the maximum propellant which can be consumed is determined; which in turn defines the payload capability. The vehicle is also sized to provide for the return to LEO of 10% of the LEO-to-GEO payload. The EOTV weight summary is presented in Table 2.2-6.

Table 2.2-6. EOTV Mass Summary, 10^{-6} kg

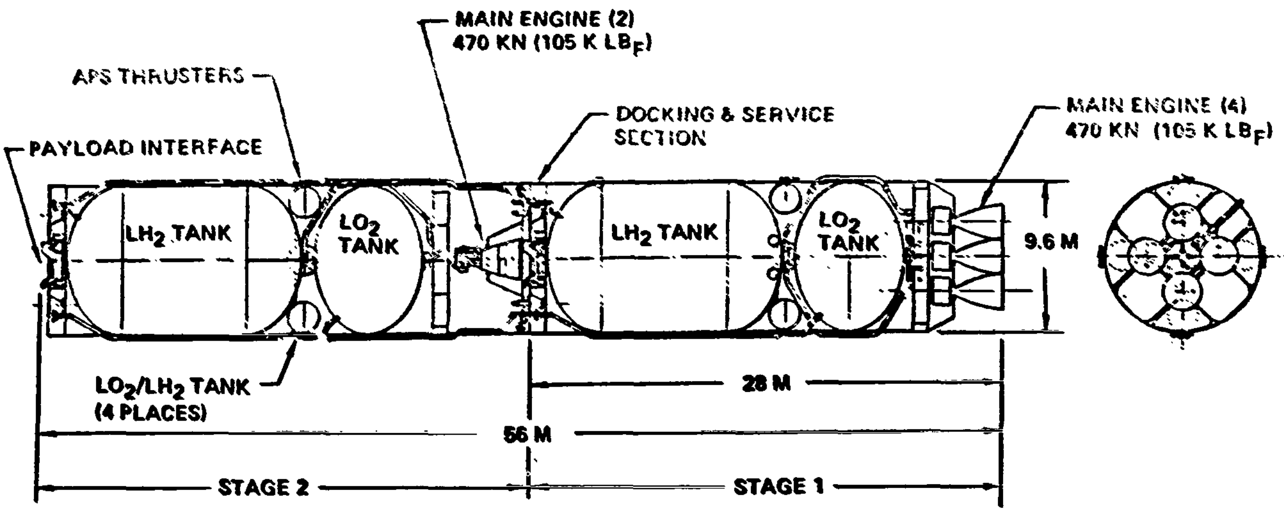
SOLAR ARRAY		0.628
THRUSTER ARRAY (4)		0.189
PROPELLANT TANKS & DIST.		0.086
EOTV (DRY)		0.903
GROWTH (25%)		0.226
EOTV, TOTAL		1.129
PROPELLANT		0.864
MAIN LEO-GSE	0.655	
MAIN GEO-LE)	0.143	
ATTITUDE CONTROL	0.066	
EOTV (WET), TOTAL		1.993
PAYLOAD		6,814
LEO DEPARTURE		8.807
GEO ARRIVAL		8.116
GEO DEPARTURE		1.971
LEO ARRIVAL		1.822

2.3 PERSONNEL TRANSFER SYSTEM

The personnel transfer system is used to transfer personnel between LEO and GEO bases. Alternate POTV concepts are presented; a larger two-stage vehicle capable of round trip flight from LEO to GEO and return, and a smaller single stage vehicle which is refueled in GEO for the return trip to LEO. The POTV traffic model is of course dependent upon a selected SPS scenario.

2.3.1 Two-Stage Personnel Orbit Transfer Vehicle (Reference Concept)

The POTV configuration is a spaced-based common stage OTV, a two-stage system with both stages having identical propellant capacity as shown in Figure 2.3-1. The first stage provides approximately 2/3 of the delta V requirement for boost out of low Earth orbit at which point it is jettisoned for return to the low Earth orbit staging depot.



- PAYLOAD CAPABILITY = 150 000 Kg UP
  90 000 Kg DOWN
- OTV STARTBURN MASS = 890 000 Kg
- ONE FLIGHT PER MONTH PER CONSTRUCTION BASE

Figure 2.3-1. POTV for GEO Construction

The second stage completes the boost from low earth orbit as well as the remainder of the other delta V requirements to place the payload at GEO and also provides the required delta V to return the stage to the LEO staging depot. Subsystems for each stage are identical in design approach. The primary difference is the use of four engines in the first stage due to thrust-to-weight requirements. Also, the second stage requires additional auxiliary propulsion due to its maneuvering requirements including docking of the payload to the construction base at GEO. The vehicle has been sized to deliver a payload of

150,000 kg and return 90,000 kg. As a result, the stage start-burn mass without payload is approximately 890,000 kg with the vehicle having an overall length of 56 m.

Main propellant containers are welded aluminum with integral stiffening as required to carry flight loads. Intertank, forward and aft skirts, and thrust structures employ graphite/epoxy composites. An Apollo/Soyuz type docking system is provided at the front end of each stage for docking with payloads, refueling tankers and orbital bases. The stage-to-stage docking system provides for docking the stages together with flight loads carried through full-diameter structures. Propellant transfer connections allow either stage to be fueled independently with the stages either separated or docked together. Structure of the two stages is identical to the extent practicable.

Main engines are based on shuttle engine technology, operating with a staged-combustion cycle at 20 MN/m^2 (3000 psia) chamber pressure, a LO_2/LH_2 mixture ratio of 5.5 to 1.0 and a retractable nozzle with extension expansion area ratio of 400 providing a specific impulse of 470 sec. Advanced low NPSH pumps are used to minimize feed pressures. A 6° square gimbal pattern is employed. The engines are capable of operating in a tank-head idle (THI) mode (pumps not turning: mixed-phase propellants) for chill-down and self-ullaging at a specific impulse of 350 sec; 60 sec (time) in self-ullaging mode is assumed needed prior to bootstrapping to full thrust. Throttling between tank-head idle and full thrust is not required. Main propellant pressurization is derived from engine top off after an onboard helium prepressurization.

Auxiliary propulsion is used for attitude control and low delta V maneuvers during coast periods and for terminal docking maneuvers. An independent LO_2/LH_2 system is used and provides an Isp of 375 sec averaged over pulsing and steady state operating modes. Thrusters are mounted in quad packages analogous to the Apollo Service Module installation. Each quad has its own propellant supply to facilitate change out. Auxiliary propulsion for the two stages uses common technology but capacities and thrust levels are tailored.

Primary electric power is provided by fuel cells based on shuttle technology, tailored to the OTV requirement. Reactants are stored in vacuum-jacketed pressure vessels. Product water is assumed retained onboard to minimize payload contamination potential. Ni-Cad batteries are employed for peaking and smoothing; 28 V dc power is rough-regulated and filtered with fine regulation provided by power using subsystems as needed. A potential inert mass saving (not assumed) would use low pressure reactants provided from main propellant tanks. Electric power systems for the two stages are identical except for reactant capacity and harnesses.

Avionics functions include onboard autonomous guidance and navigation, data management, and S-band telemetry and command communications. Navigation employs earth horizon, star and sun sensors with an advanced high performance inertial measurement system. Cross-strapped LSI computers provide required computational capability including data management, control and configuration control. The command and telemetry system employs remote-addressable data busing and its own multiplexing. Although the avionics systems in the two stages are identical, software for each stage is tailored to the stage functions.

Main propellant tanks are insulated by aluminized mylar multilayer insulations contained within a purge bag. The insulation system is helium purged on the ground and during Earth launch. Environmental control of the avionics systems is accomplished using semi-active louvered radiators and cold plates. Active fluid loops and radiators are required for the fuel cell systems. Superalloy metal base heat shields are employed to protect the base areas from recirculating engine plume gas.

Performance characteristics associated with the common stage LO_2/LH_2 OTV are shown in Figure 2.3-2. Propellant requirements are shown as a function of the payload return and delivery capability. Performance ground rules used in these parametrics are as follows (values are main propellant qualtities):

- THI mode — Stg 1—100 kg per start; Stg 2—50 kg per start
- Stop loss — Stg 1—20 kg; Stg 2—10 kg
- Boiloff rate — 6 kg/hr each stage
- Burnout mass scaling equations:

  Stage 1 3430 kg = $0.05567\ WP_1 + 0.1725\ WP_2$
  Stage 2 3800 kg = $0.05317\ WP_1 + 0.1725\ WP_2$

  Where WP_1 and WP_2 are main and auxiliary propellant capacities respectively
- Stage μ of 0.93
- Staging base at 477 km, 31°

Summary level mass estimates are presented in Table 2.3-1 for the selected satellite OTV. A weight growth factor of 10% was used rather than 15% as in FSTS based on the judgment that the SPS LO_2/LH_2 OTV would be a second generation vehicle. Mass estimates for the systems reflect the design approach previously described.

2.3.2 Single Stage Personnel Orbit Transfer Vehicle (Alternate Concepts)

As stated previously, the POTV is the propulsive element used to transfer the personnel module (PM) from LEO to GEO and return. In previous scenarios, the POTV reference concept used two common stage LO_2/LH_2 propulsive elements. The first stage provided an initial delta-V and returned to LEO. The second stage provided the remaining delta-V required for PM ascent to GEO and the requisite delta-V for return of the PM to LEO.

The alternate concepts described herein use a single stage to transport the PM and its crew and passengers to GEO. Two concepts are presented herein; the first is of a larger size which has been optimized for the silicon SPS concept and the other, a smaller version optimized for the GaAs satellite concept (i.e., different crew rotation requirements). Other than size another major difference in the operations mode exists. In the case of the larger vehicle the crew is transported from earth-to-LEO in the PLV where they are

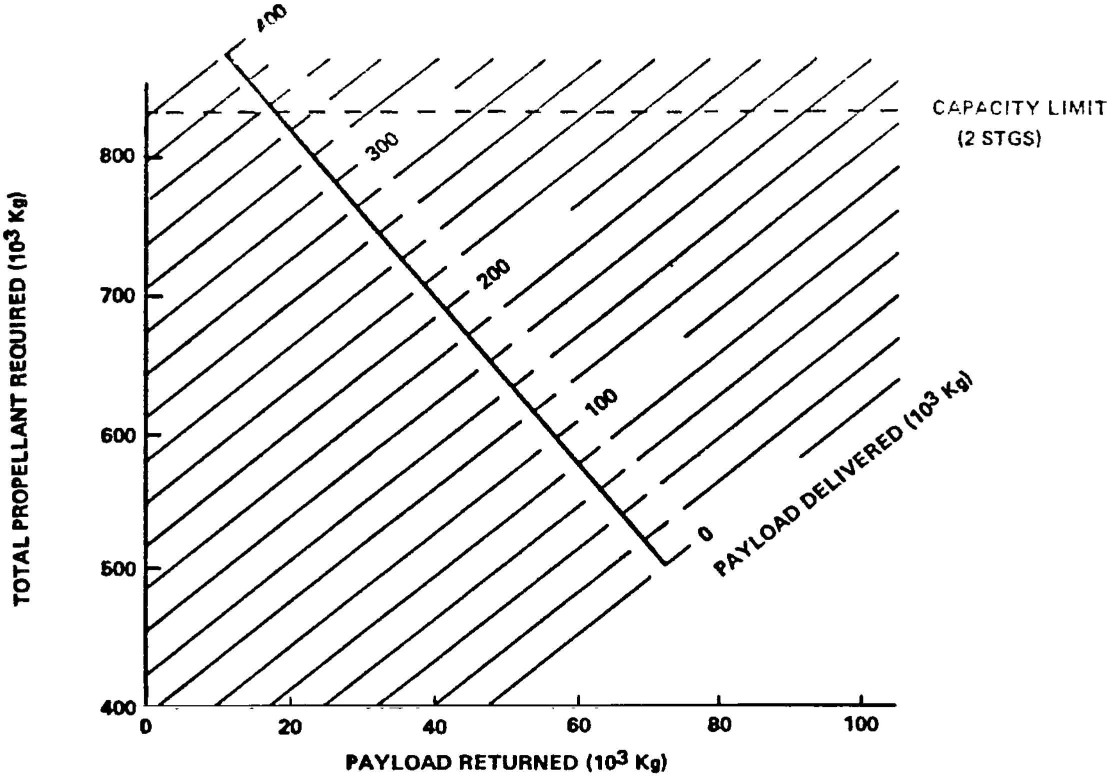


Figure 2.3-2. Two-Stage LO_2/LH_2 OTV Performance

Table 2.3-1. Chemical OTV Mass Summary

	Stage 1 (KG)	Stage 2 (KG)
Structure and Mechanisms	13,500	14,780
Main Propulsion	7,000	4,020
Auxiliary Propulsion	820	1,120
Avionics	300	310
Electrical Power	850	820
Thermal Control	1,850	2,310
Weight Growth (10%)	2,420	2,340
Dry	26,630	25,790
Fuel Bias	640	640
Unusable LO_2/LH_2	1,810	1,810
Unusable and Reserve APS	290	660
Burnout	29,370	28,990
Main Impulse Prop	415,000	407,000
APS	2,700	6,100
Startburn	447,070	442,090

then transferred to the orbit personnel module (OPM) and POTV for transfer to GEO. In the smaller GaAs concept, the crew is transported from earth-to-LEO in the OPM which is part of a PLV payload. In LEO the OPM is mated to the POTV for transfer to GEO. The return from GEO to earth is accomplished in reverse for both systems.

2.3.2.1 Large Single Stage POTV

The POTV is a single stage LO_2/LH_2 vehicle which in the normal mode has the capability of transporting 90 MT of payload between LEO and GEO. Return of the vehicle and payload requires refueling at GEO. The POTV can be flown in a roundtrip mode without refueling if necessary but with a reduction in payload capability. An inboard profile of the POTV is shown in Figure 2.3-3.

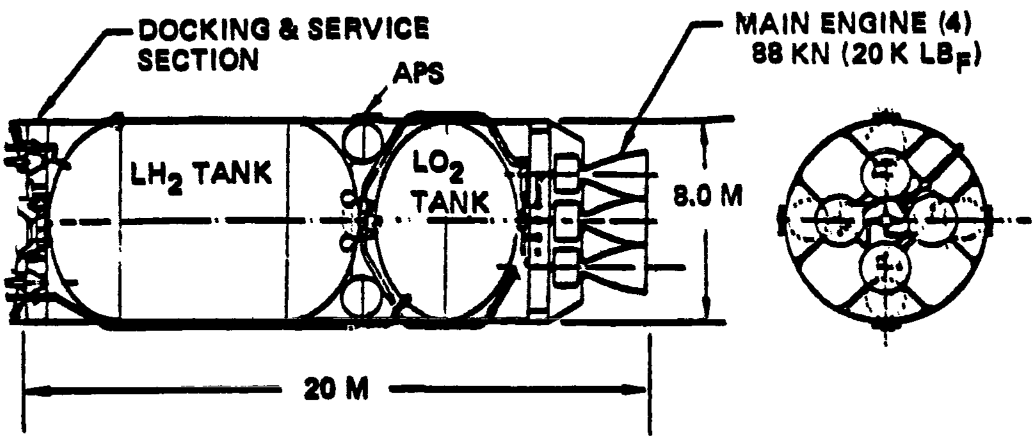


Figure 2.3-3. Inboard Profile of the POTV

Main propellant containers are welded aluminum with integral stiffening as required to carry flight loads. Intertank, forward and aft skirts, and thrust structures employ graphite/epoxy composites. An Apollo/Soyuz type docking system is provided at the front end of the stage for docking with payloads and orbital bases.

Five ASE type engines (staged combustion) are used for main propulsion. A thrust level per engine of 88 KN (20,000 lb_f) is assumed along with an extension expansion area ratio of 400 and a specific impulse of 470 sec.

Auxiliary propulsion is used for attitude control and low delta V maneuvers during coast periods and for terminal docking maneuvers. An independent LO_2/LH_2 system is used and provides an Isp of 375 sec averaged over pulsing and steady state operating modes.

Primary electric power is provided by fuel cells based on shuttle technology, tailored to the OTV requirement. Reactants are stored in vacuum-jacketed pressure vessels. Ni-Cad batteries are employed for peaking and smoothing the 28 V dc power.

Avionics functions include onboard autonomous guidance and navigation, data management, and S-band telemetry and command communications. Navigation employs Earth horizon, star and sun sensors with an advanced high performance inertial measurement system. Cross-strapped LSI computers provide required computational capability including data management, control and configuration control. The command and telemetry system employs remote-addressable data busing and its own multiplexing.

Main propellant tanks are insulated by aluminized mylar multilayer insulations contained within a purge bag. Environmental control of the avionics systems is accomplished using semi-active louvered radiators and cold plates. Active fluid loops and radiators are required for the fuel cell systems. Superalloy metal base heat shields are employed to protect the base areas from recirculating engine plume gas.

The nominal transfer time beginning with separation from the LEO base and docking at the GEO base is approximately 11 hours. Orbit phasing requirements could add an additional 12 hours.

The present single stage vehicle requiring refueling at GEO was selected to reduce the total POTV propellant per flight since the highly efficient EOTV could be used to deliver the POTV return propellant to GEO. The POTV savings per flight is approximately 265 MT with the net savings including EOTV penalty of 175 MT per POTV flight.

The 90 MT payload capability was the result of sizing the vehicle to be delivered by the shuttle derivative HLLV (without propellant). This capability is sufficient to deliver up to 80 GEO workman and food and crew accommodations for 6600 man days.

The total dry mass is 13,420 kg while the main impulse propellant is 200,000 kg. The breakdown is shown in Table 2.3-2.

Table 2.3-2. POTV Mass (kg) Summary

Structure & Mechanisms	6,900	Unusable LO_2/LH_2	1,130
Main Propulsion	2,500	Unusable & Reserve APS Prop	500
APS	500		
Avionics	300	Fuel Cell Reactant	150
		Boil-off	100
Electric Power	450	Burnout	
Thermal Control	1,030	Main Impulse Prop	200,000
Contingency (15%)	1,750	APS	1,200
Dry	13,430	Stage Start-burn	216,510

The orbit personnel module (OPM) is used to accommodate crews during the transfer between the LEO and GEO bases. The launch personnel module (LPM) accommodates crews during transit between Earth and the LEO base. The OPM differs from the LPM in terms of the mission duration and associated needs as well as the environmental protection requirements.

The mated configuration of OPM and POTV with crew supply modules is shown in Figure 2.3-4.

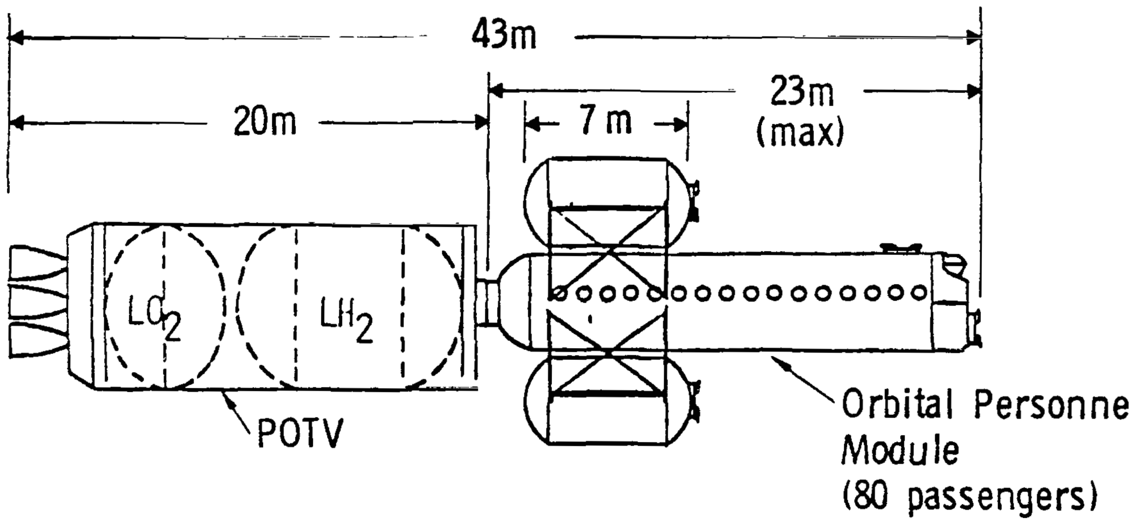


Figure 2.3-4. Orbital Crew Rotation/Resupply Configuration

The configuration for the OPM is shown in Figure 2.3-5. This design has the flight control deck integrated with the passenger crew cabin. The OPM has been sized to transport 78 orbital workman in a single deck, 6-abreast arrangement in addition to a flight crew (pilots, flight engineer, flight attendants).

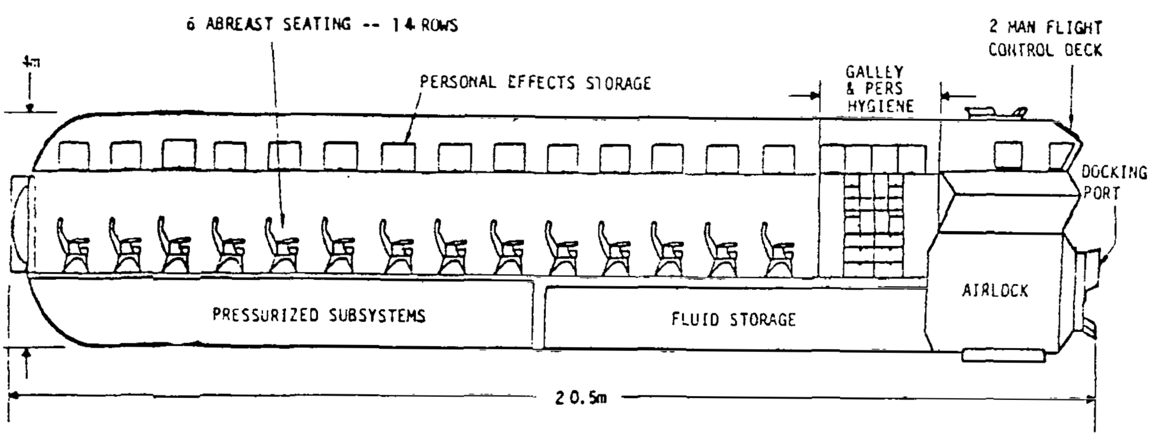


Figure 2.3-5. OPM Configuration

The structural shell is aluminum with an inside layer of tantalum to improve radiation protection characteristics. An average electrical power requirement of 15 kW has been estimated for the OPM. Advanced LO_2/LH_2 fuel cells are the primary power source with batteries for peaking and emergency. The environmental control/life support system is a scaled down version of the system used in the crew modules at the LEO and GEO bases. This closed loop system was selected due to the large crew size and frequent use. The system employs a sabatier reactor for CO_2 reduction, water recovery and electrolysis for oxygen production. Thermal control consists of water loops inside the cabin and freon loops for the space radiator.

Crew accommodations include pressure suit garmets for all on-board personnel, several EVA suits for emergencies, food storage and preparation, furnishings such as seats and mobility aids and limited recreation provisions. The information system includes data processing, displays and controls to operate the POTV as well as OPM, supplemental G&N equipment to that incorporated on the POTV and communications equipment. Crew consumables are based on a nominal transfer of 1-day for both delivery and return flights and an additional day provided for emergency.

The major issue considered thus far in the design of the OPM is that of defining the crew capacity. The principle considerations in selecting the crew capacity of the OPM are the payload capability of the POTV and the payload envelope of the launch vehicle which will initially deliver the OPM to orbit. As previously described the POTV was sized for refueling at GEO and delivery to LEO by the shuttle derivative HLLV. These conditions resulted in a payload capability that allowed delivery of approximately 80 people. Using crew rotation cycles of 90 days results in POTV flights approximately every 15 days. Previous analysis had a flight every 30 days but the more frequent flight arrangement is judged to provide more flexibility for the case of delivering a small amount of priority cargo.

The hardware mass of the OPM including growth is 43,685 kg with the total flight weight including crew is 53,285 kg.

A mass summary and detail hardware breakdown is presented in Table 2.3-3.

2.3.2.2 Small Single Stage POTV

The POTV operations scenario for the smaller single stage POTV is presented in Figure 2.3-6. After initial delivery of the POTV to LEO by the STS or SPS-HLLV, the propulsive stage is subsequently refueled in LEO (at the LEO station) with sufficient propellants to execute the transfer of the PM to GEO. At GEO, the stage is refueled for a return trip of crew and passengers to LEO. The HLLV delivers crew consumables and POTV propellants to LEO and the EOTV delivers the same items required in GEO. The PM with crew/personnel is delivered to LEO by the PLV.

Although significant propellant savings occur with this approach, as compared to the reference concept, the percentage of total mass is small when compared with satellite construction mass. However, the major impact is realized in the smaller propulsive stage size and the overall reduction in orbital operations requirements.

Table 2.3-3. OPM Mass Summary

Hardware		(349.50)
Structure	12,560	
Electrical Power	710	
Envir. Cent/Life Support	18,300	
Crew Accommodations	2,980	
Avionics System	400	
Consumables		(1,450)
EPS - LO_2/LH_2	400	
Afm Supply - LO_2/LH_2	150	
Water - Crew	585	
Food & Pkg	315	
Fluids		(1,060)
Thermal Cont. (H_2O & Freon)		
Payload		(7,090)
Crew & Pers Effects		
	Subtotal PM	44,500 kg
Growth (Contingency) - 25% on Hardware		8,735
	PM Grand Total	53,285 kg

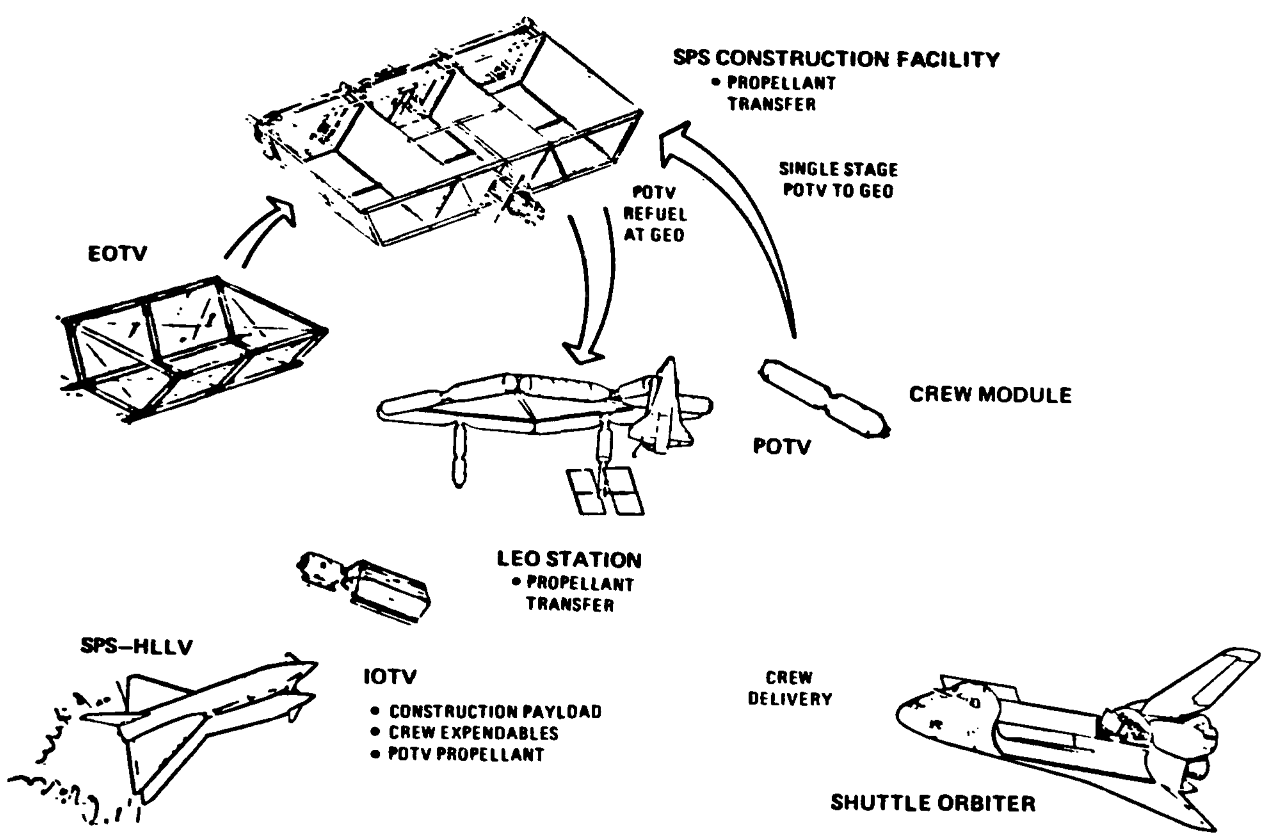
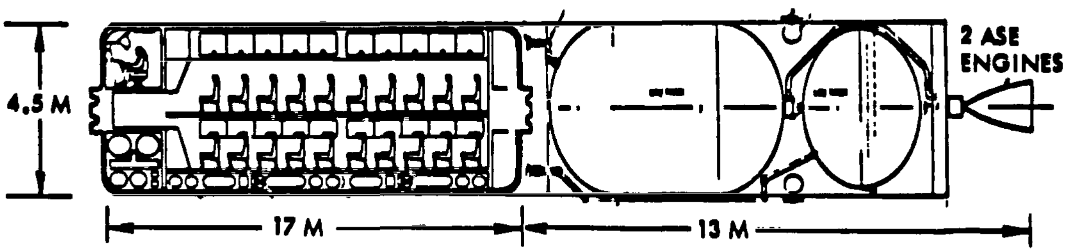


Figure 2.3-6. POTV Operations Scenario

The single stage POTV configuration is shown in Figure 2.3-7 in the mated configuration with the PM. Either element is capable of delivery from earth to LEO in the STS; however, subsequent propellant requirements for the POTV will be delivered to LEO by the HLLV because of the lesser $/kg payload cost.

Individual propellant tanks are indicated for the LO_2 and LH_2 in this configuration because of uncertainties at this time in specific attitude control requirements. With further study, it may be advantageous to provide a common bulkhead tank as in the case of the Saturn-II, and locate the ACS at the mating station of the POTV and PM, or in the aft engine compartments—space permitting.



• 60 MAN CREW MODULE	18,000 KG
• SINGLE STAGE OTV (GEO REFUELING)	36,000 KG
• BOTH ELEMENTS CAPABLE OF GROWTH STS LAUNCH	

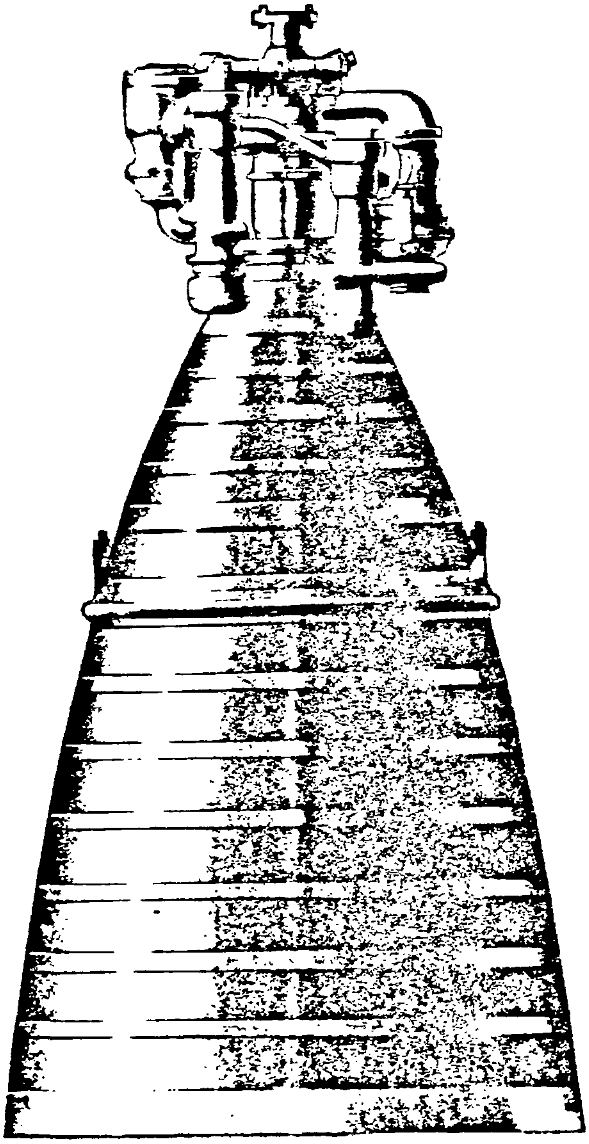
Figure 2.3-7. Single Stage POTV Configuration

The POTV utilizes two advanced space engines (ASE), which are similar in operation to the Space Shuttle main engine (SSME). The engine is of high performance with a staged combustion cycle capable of idle-mode operation. The engine employs autogenous pressurization and low inlet NPSH operation. A two-position nozzle is used to minimize packaging length requirements. The ASE and pertinent parameters are shown in Figure 2.3-8. A current engine weight statement is given in Table 2.3-4.

Since the POTV concept utilizes an on-orbit maintenance/refueling approach, an on-board system capable of identifying/correcting potential subsystem problems in order to minimize/eliminate on-orbit checkout operations is postulated.

The recommended POTV configuration has a loaded weight of 36,000 kg and an inert weight of 3750 kg. A weight summary is presented in Table 2.3-5.

Although the current POTV configuration provides a suitable concept for identifying and developing other SPS programmatic issues, further trade studies are indicated such as tank configuration and ACS location(s). Also, future studies might be directed toward the evolution of a configuration that would be compatible with potential near-term STS OTV development requirements.



THRUST (N)	89,000
CHAMBER PRESSURE (MN/M^2)	14
EXPANSION RATIO	400
MIXTURE RATIO	6.0
SPECIFIC IMPULSE (SEC)	473.0
DIAMETER (cm)	123.2
LENGTH (cm)	
NOZZLE RETRACTED	128.3
NOZZLE EXTENDED	238.8

Figure 2.3-8. Advanced Space Engine

Table 2.3-4. Current ASE Engine Weight

Fuel boost and main pumps	33.8
Oxidizer boost and main pumps	40.7
Preburner	5.6
Ducting	11.3
Combustion chamber assembly	28.5
Regen. cooled nozzle (ε = 175:1)	26.5
Extendable nozzle and actuators (ε = 400:1)	55.3
Ignition system	2.8
Controls, valves, and actuators	33.6
Heat exchanger	6.4
Total (kg)*	244.5

*Based on major component current measured weights.

Table 2.3-5. POTV Weight Summary

Subsystem	Weight (kg)
Tank (5)	1,620
Structures and lines	702
Docking ring	100
Engine (2)	490
Attitude control	235
Other	262
Subtotal	3,409
Growth (10%)	341
Total inert	3,750
Propellant	32,750
Total loaded	36,000

A construction sequence has been developed which requires a crew rotation every 90 days for crew complements in multiples of 60. The PM was synthesized on this basis. A limitation on PM size was established to assure compatibility with the STS cargo bay dimensions and payload weight capacity (i.e., 4.5x17 m and 45,000 kg).

The PM shown in Figure 2.3-9 is based on parametric scaling data developed in previous studies. It is assumed that a command station is required to monitor and control POTV/PM functions during the flight. This function is provided in the forward section of the PM as shown. Spacing and layout of the PM is comparable to current commercial airline practice. Seating is provided on the basis of one meter, front to rear, and a width of 0.72 m. PM mass was established on the basis of 110 kg/man (including personal effects) and approximately 190 kg/man for module mass. The PM design has provisions for 60 passengers and two flight crew members.

Several other POTV/PM options were evaluated (Figure 2.3-9 and Table 2.3-6). All options utilize a single-stage propulsive element which is fueled in LEO and refueled in GEO for the return trip. The various options considered transfer of both crew and consumables as well as crew only. Transfer of consumables by EOTV was determined to be more cost effective. Another potential option, which is yet to be evaluated, is a 30-man crew module and integral single-stage capable of storage within the STS cargo bay.

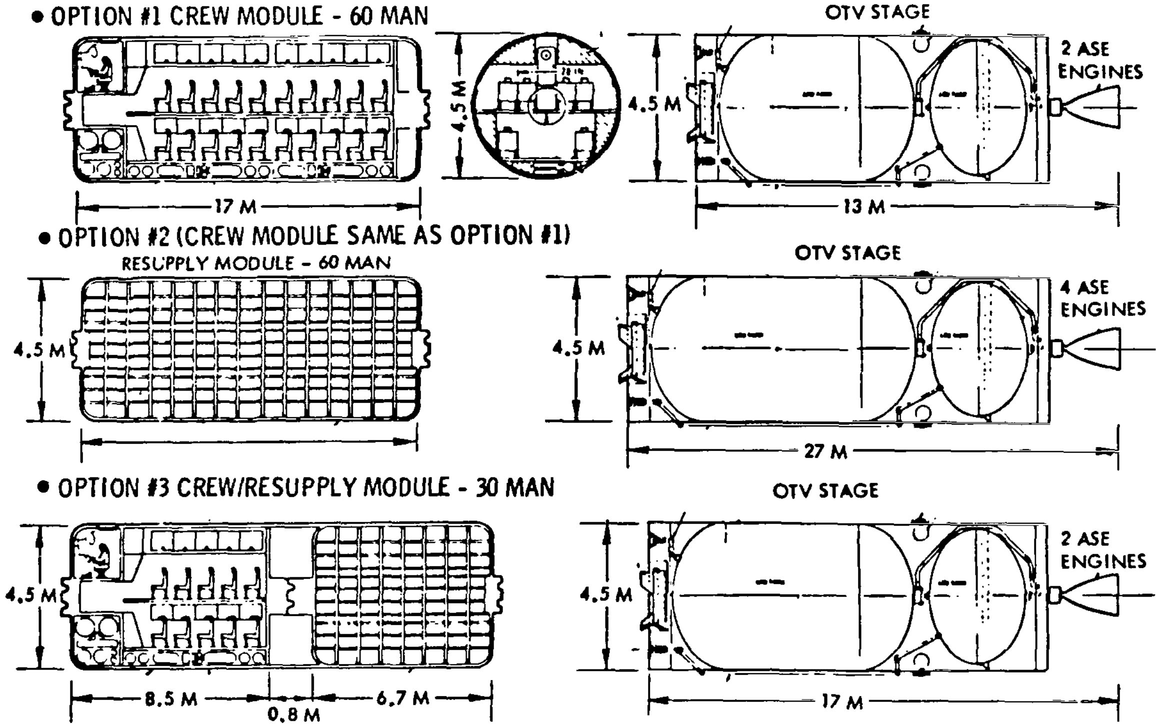


Figure 2.3-9. POTV/PM Configuration Options

Table 2.3-6. POTV/PM Options—Element Mass

	kg
60-man crew module	18,000
60-man resupply module	26,000
Integrated 30-man crew/resupply module	22,000
Option 1 OTV	36,000
Option 2 OTV	87,000
Option 3 OTV	44,000

3.0 OTHER SYSTEM ELEMENTS

A ground and flight operations summary, intra-orbit vehicle description, propellant production and storage analyses, and environmental considerations are presented. Since many vehicle options have been studied to varying degrees, the data presented have by necessity been structured to the NASA/DOE reference configuration only.

3.1 GROUND AND FLIGHT OPERATIONS

The major element of ground operations is related to launch vehicle turnaround requirements. The high launch frequency demands an airline operations concept which, in turn, dictates vehicle design requirements which will result in the near-elimination of post-flight refurbishment and checkout other than that required for payload installation, mating, and fueling.

A great dependence must be placed upon on-board monitoring and fault detection/isolation systems in order to preclude the requirement for ground interfacing and checkout requirements. All previous ground and flight performance data will be computer analyzed to determine performance trend data indicative of potential impending failures. The line replaceable unit (LRU) concept must be employed with a primary design consideration of accessibility and internal isolation features to permit rapid replacement of worn or failed components. Launch site operations will be restricted to LRU maintenance and replacement (i.e., overhaul and repair will be performed at a suitable depot).

All cargo must arrive at the launch site in a pre-palletized configuration in order to minimize handling. Cargo manifests will be computer controlled with automated cargo handling and transfer.

Communications between the launch vehicle and ground stations will be restricted such that the launch vehicle is essentially capable of autonomous operation other than launch and landing clearances.

As previously stated, a LEO staging base will be required for crew/cargo transfer and orbital vehicle maintenance. The HLLV will rendezvous only with the LEO base (i.e., docking not required). Cargo will then be transferred from the HLLV to the EOTV by LEO based intra-orbit transfer vehicles. Down-payload, as required, will be transferred to the HLLV. A maximum stay time in orbit for the HLLV should not exceed 12 hours.

The payload may rendezvous or dock with the LEO base in order to effect crew transfer. The crew module will be removed from the PLV cargo bay and mated to a POTV element for immediate transfer to GEO. Crews returning to earth will have already boarded a crew module, which will then be loaded into the PLV cargo bay. The maximum stay time for the PLV in LEO will be 12 hours.

LEO base maintenance or orbital vehicles will be primarily restricted to component (LRU) replacements on the EOTV, POTV, and IOTV; and the propellant servicing requirements of the POTV and IOTV. (EOTV propellant tanks will be transferred directly from the HLLV to the EOTV.)

The EOTV, POTV, and IOTV GEO operations will be essentially the same as those conducted in LEO. Transportation system maintenance provisions in GEO will also be the same as those in LEO.

The two-stage series burn HLLV operations plan includes prelaunch, launch, and recovery activities associated with the SPS launch vehicle. The launch site operations plan includes:

- Both vehicles landing at the launch site
- Stage maintenance and checkout in dedicated facilities for both the booster and orbiter
- Mating, vehicle integration, and fueling at the launch pad

A horizontal mating operation is planned on the launcher where the two stages will be joined and then rotated to the vertical. This concept is depicted in Figure 3.1-1. The upper portion of the launcher/erector is rotated away from the vehicle after the vehicle is in the vertical position to provide clearance for launch.

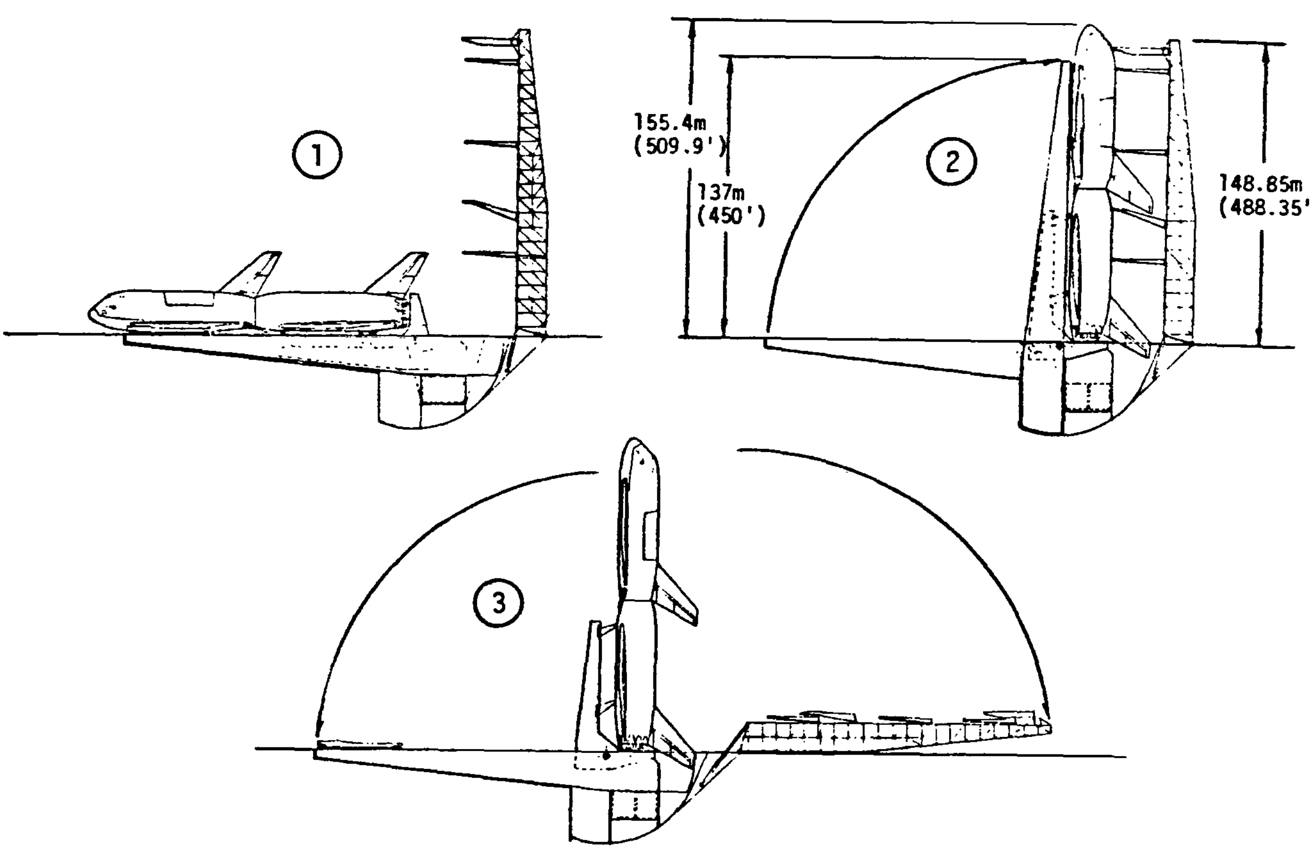


Figure 3.1-1. Launcher/Erector Concept

The booster timeline from launch to its move in the integration positon is shown in Figure 3.1-2. The timelines reflect the average timelines for the operational vehicle system. A total of 62 hours is estimated for this portion

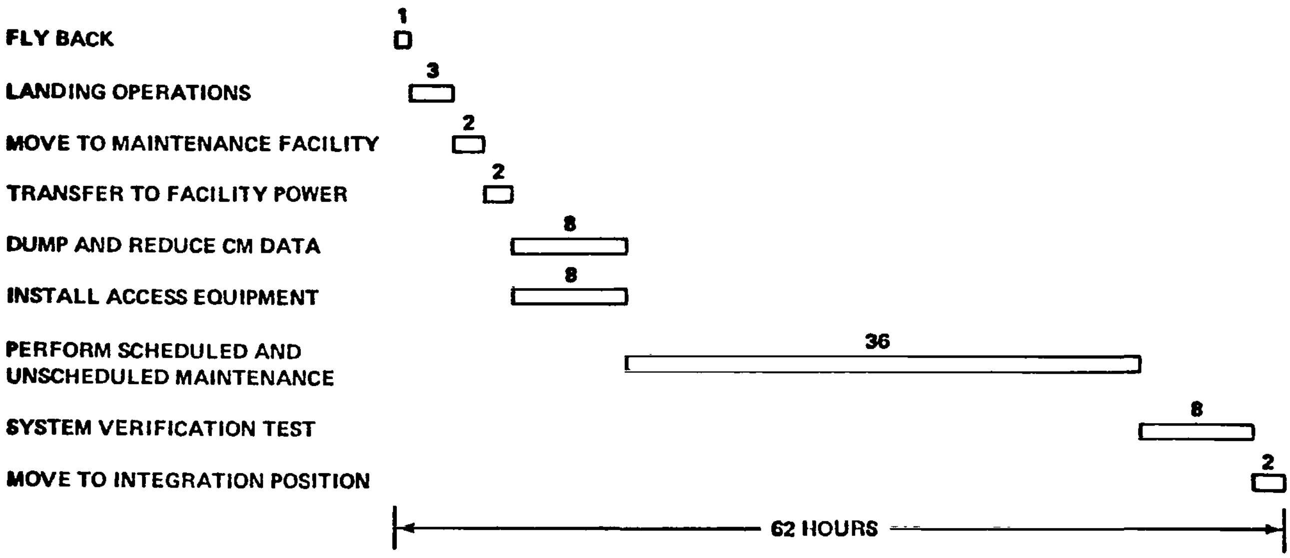


Figure 3.1-2. Booster Processing Timelines

of the turnaround with the scheduled and unscheduled maintenance activity requiring 36 hours. On-board condition monitoring equipment will enhance the operations by:

- Providing performance monitoring of the stage subsystems
- Aiding in fault isolation and detection

Rocket engine maintenance is anticipated to be the major portion of the booster operations.

The orbiter timeline from launch to its move to the integration position is shown in Figure 3.1-3. A total time of 97 hours for orbiter processing

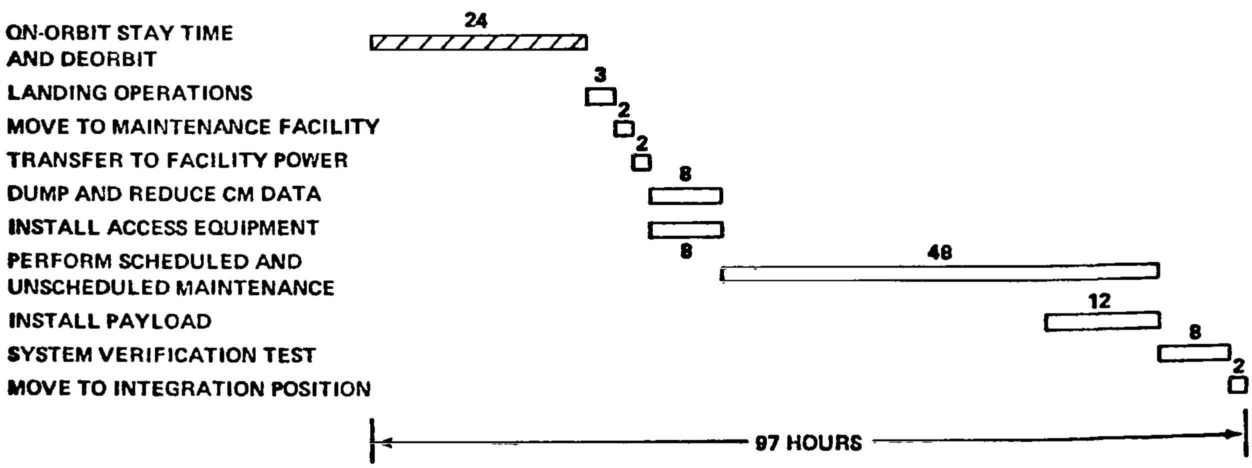


Figure 3.1-3. Orbiter Processing Timelines

including 24-hour on-orbit staytime is estimated. The maintenance portion of the activity is estimated to require 48 hours due to the thermal protection system and the additional systems/equipment required for the manned stage. A total of 12 hours has been allocated for payload installation in a parallel operation with the orbiter maintenance.

The vehicle integrated operations timeline is shown in Figure 3.1-4. These activities are at the launch site and reflect all the operations from vehicle mating through launch. This portion of the launch operations requires

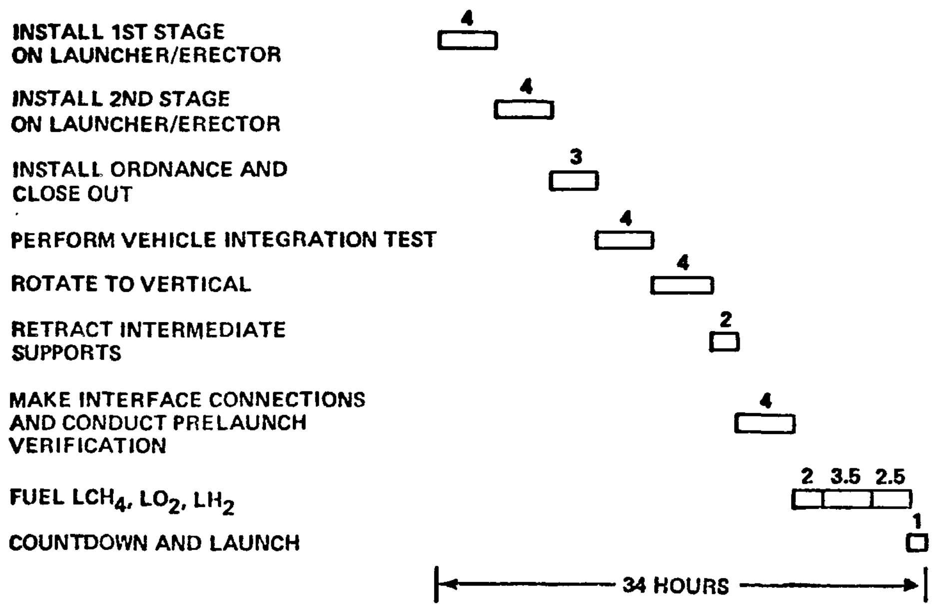


Figure 3.1-4. Integrated Vehicle Operations Timelines

34 hours for the booster and 30 hours for the orbiter. The total turnaround times for the booster and orbiter are summarized in Table 3.1-1. Also shown on the table for reference is the anticipated turnaround times for the two-stage ballistic recoverable concept studied earlier. The two-stage winged vehicle results in turnaround times which are less than those for the ballistic vehicle.

Table 3.1-1. Vehicle Turnaround Analysis Summary

VEHICLE CONCEPT	STAGE OPS ONLY	INTEGRATION AND LAUNCH OPERATIONS	TOTAL TURNAROUND
WING/WING			
BOOSTER	63 HOURS	34 HOURS	97 HOURS
ORBITER	97 HOURS	30 HOURS	127 HOURS
BALLISTIC/BALLISTIC			
BOOSTER	93 HOURS	34 HOURS	127 HOURS
UPPER STAGE	102 HOURS	30 HOURS	132 HOURS

Typical orbit transfer operations from LEO to GEO for the common-stage OTV are illustrated in Figure 3.1-5. The majority of the delta-V for boosting from LEO is provided by Stage 1. Stage 1 then separates and returns to the staging depot following an elliptical return phasing orbit. Stage 2 completes the boost and puts the payload into a GEO transfer and phasing orbit, as well as injecting the payload into GEO and performing the terminal rendezvous maneuver with the GEO construction base. Following removal of the payload, Stage 2 uses two primary burns in returning to the LEO staging depot. A detailed mission profile indicating events, time, and delta-V is presented in Table 3.1-2.

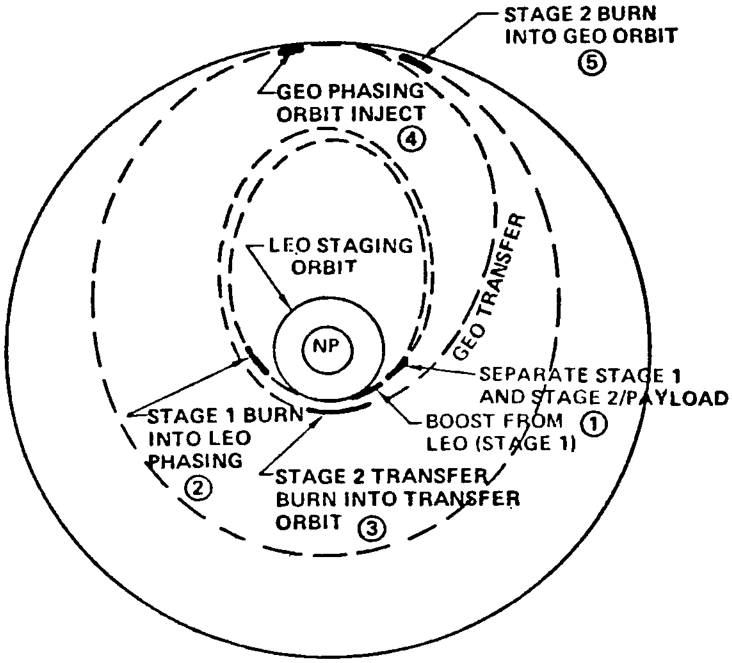


Figure 3.1-5. Chemical OTV Transfer Operations

A total mission timeline for each stage is presented in Figure 3.1-6. Allowing approximately eight hours for refueling and refurbishment results in 40 hours elapsed time before a given Stage 1 can be reused. A typical Stage 2 however, has an elapsed time of 85 hours before reuse, including time for assembly between stages and between OTV and payload.

Mission events that occur while using an EOTV for GEO construction are indicated in Table 3.1-3. A total of 16 days of on-orbit time has been indicated for the turnaround the the vehicle, in addition to the 219 days of time required for the up and down transfers.

Once the vehicle reaches GEO, it will be placed in a standby condition approximately 1 km from the base. At that time, small LO_2/LH_2 tug(s) will be used to move the cargo from the EOTV to the GEO construction base. Annealing of the solar arrays will occur at GEO. Once the vehicle has returned to low earth orbit, it will again be placed in a stationkeeping standby condition approximately 1 km from the LEO base. Again, small tugs will fly out from the

Table 3.1-2. Mission Profile

MISSION EVENT NO. AND NAME	REQ'D TIME (HR)	ΔV M/SEC	PROPULSION (MAIN OR AUX.)	REMARK
			MISSION	
1. STANDOFF	0	3	A	PROVIDE SAFE SEPARATION DISTANCE BETWEEN FACILITY & VEHICLE
2. PHASE	12	3	A	ΔV IS ATTITUDE CONTROL
3. COAST	0.5	1715	M	OTV 1st STAGE SEPARATES AFTER THIS ΔV
4. COAST	4.2	3	A	ELLIPTIC REV
5 INJECT	0.1	750	M	INCL. 60 M/SEC ACCUM. FINITE BURN LOSS
6. COAST	5.4	3	A	TRANSFER TO GEO
7. PHASE INJ.	0.1	1780	M	REPRESENTATIVE FOR 15° PHASING
8. PHASE	23	3	A	
9. TPI*	0.1	56	M	INCL. 15 M/SEC OVER IDEAL TO ALLOW FOR CORRECTIONS
10. RENDEZ.	2	10	A	TPI ASSUMED TO OCCUR WITHIN 50 KM OF TARGET
11. DOCK	1	10	A	
12. WAIT	8	0	-	ASSUMED DOCKED
13. STANDOFF	0.1	3	A	
14. DEORBIT	0.1	1820	M	
15. COAST	5.4	10	A	TRANSFER TO LEO
16. PHASE INJ.	0.1	2356	M	
17. PHASE	12	3	A	ORBIT PERIGEE AT STAGING BASE ALT.
18. TPI	0.1	50	M	
19. RENDEZ.	2	20	A	
20. DOCK	1	10	A	
21. RESERVE	-	130	M	2% OF STAGE MAIN PROPUL. V BUDGET
			FIRST-STAGE RECOVERY	
1. COAST	4.2	30	A	ΔV TO CORRECT DIFFERENTIAL NODAL REGRESSION BETWEEN COAST ORBIT & STAGING BASE
2. PHASE INJ.	0.1	1645	M	ELLIP. ORB PERIGEE AT STG BASE ALT.
3. PHASE	0.1	50	M	
4. TPI	12	3	A	ALTITUDE CONTROL
5. RENDEZ.	2	20	A	
6. DOCK	1	10	A	
7. RESERVE	-	85	M	2% OF STG MAIN PROP. V BUDGET

*TERMINAL PHASE INITIATION

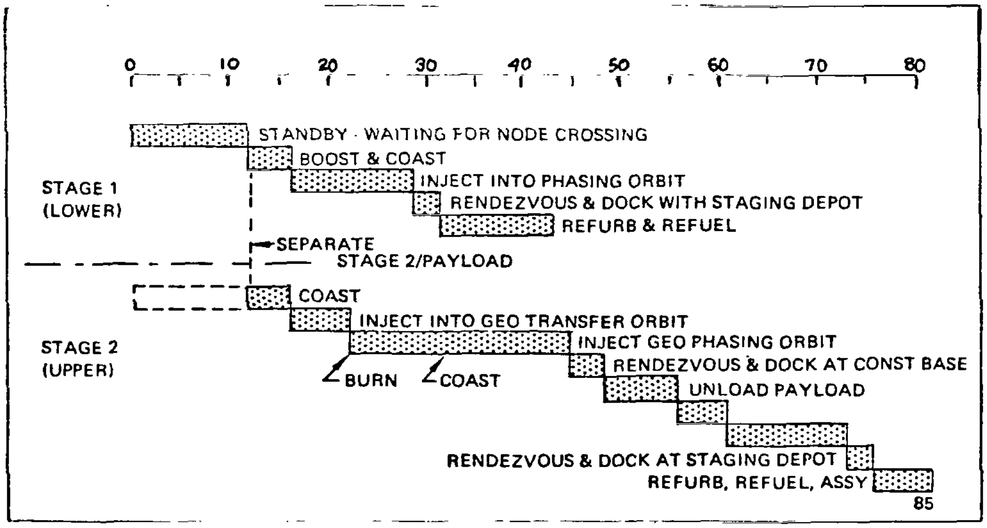


Figure 3.1-6. Chemical OTV Flight Operation Timeline

Table 3.1-3. Mission Events

EVENT	DESCRIPTION	Δ TIME (DAYS) ON-ORBIT	Δ TIME (DAYS) TRANSFER
• TRANSFER TO GEO	COST OPTIMIZED FIRST FLIGHT		180
• TERMINAL MANEUVERS	RENDEZVOUS AND PLACE ON STANDBY CONDITION	1	
• UNLOAD CARGO	(10) 400 MT UNITS	1	
• ANNEAL SOLAR ARRAY	1.2 MILLION SQ METERS	4	
• PREPARE FOR RETURN	ACTIVATE, CHECKOUT AND LOAD CARGO	1	
• TRANSFER TO LEO	DICTATED BY POWER AVAILABLE		39
• TERMINAL MANEUVERS	RENDEZVOUS AND PLACE ON STANDBY CONDITION	1	
• REFURB ELEC THRUSTERS	1600 UNITS	4	
• CARGO HANDLING	UNLOAD CARGO AND LOAD (10) 400 MT UNITS	1	
• UNSCHEDULED MAINT	---	1	
• PROPELLANT RESUPPLY	ARGON, LO_2, LH_2	1	
• PREPARE FOR TRANSFER	ACTIVATION AND CHECKOUT	1	
	TOTAL	16	219

LEO base to the EOTV to perform refurbishment operations on the thrusters, unload and load cargo propellant, and deliver propellant. The propellant resupply will be done by tankers rather than removal of the propellant tanks.

3.2 INTRA-ORBIT TRANSFER VEHICLE

Intra-orbit transfer systems have been synthesized in terms of application and concept only. On-orbit elements considered here are powered by a chemical (LOX/LH_2) propulsion system. At least three distinct applications have been identified: (1) the need to transfer cargo from the HLLV to the EOTV in LEO and from the EOTV to the SPS construction base in GEO; (2) the need to move materials about the SPS construction base; and (3) the probable need to move men or materials between operational SPS's. Clearly the POTV, used for transfer of personnel from LEO to GEO and return, is too large to satisfy the on-orbit mobility systems requirements. A "free-flyer" teleoperator concept would appear to be a logical solution to the problem. A propulsive element was synthesized to satisfy the cargo transfer application from HLLV-EOTV-SPS base in order to quantify potential on-orbit propellant requirements. This transportation element has been designated intra-orbit transfer vehicle (IOTV).

Sizing of the IOTV was based on a minimum safe separation distance between EOTV and the SPS base of 10 km. It was also assumed that a reasonable transfer time would be in the order to two hours (roundtrip), which equates to a ΔV requirement on the order of 3 to 5 m/sec. A single advanced space engine (ASE) is employed with a specific impulse of 473 seconds. Typical IOTV parameters are summarized in Table 3.2-1.

Table 3.2-1. IOTV Weight Summary

SUBSYSTEM	WEIGHT (kg)
ENGINE (1 ASE)	245
PROPELLANT TANKS	15
STRUCTURE AND LINES	15
DOCKING RING	100
ATTITUDE CONTROL	50
OTHER	100
SUBTOTAL	525
GROWTH (10%)	53
TOTAL INERT	578
PROPELLANT	300
TOTAL LOADED	878

3.3 PROPELLANT PRODUCTION AND STORAGE

At its peak, the SPS program will require the construction of two SPS's per year. This production rate will necessitate the placement of significant materials and personnel into low-earth orbit. In order to meet this high mass-to-orbit rate, the earth-launch-vehicles (HLLVs) will consume even greater quantities of propellant, primarily LH_2 and LO_2. In fact, peak daily propellant consumption is anticipated to be in the order of 1000 metric tons of liquid hydrogen and 10,000 metric tons of liquid oxygen. These large amounts of hydrogen and oxygen are approximately 6% and 30% of the present daily U.S. production rate, respectively, and it is important to assess the nation's ability to meet these demands.

A study was performed to analyze various techniques available for liquid hydrogen and liquid oxygen production. Figure 3.3-1 summarizes the scope of the analysis.

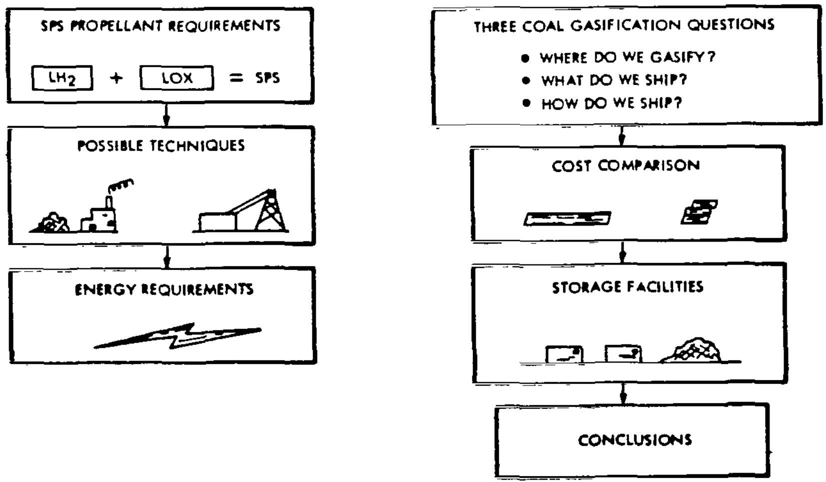


Figure 3.3-1. Scope of Analysis

The ground rules for the study were:

- Launch from Cape Kennedy
- Solar electric OTV with argon as propellant
- Mass-to-orbit to support two SPS/year
- Packing factor of 15%

The total mass-to-orbit as a function of SPS production rate can be translated into ELV propellant required as a function of year; and these data are presented in Figure 3.3-2.

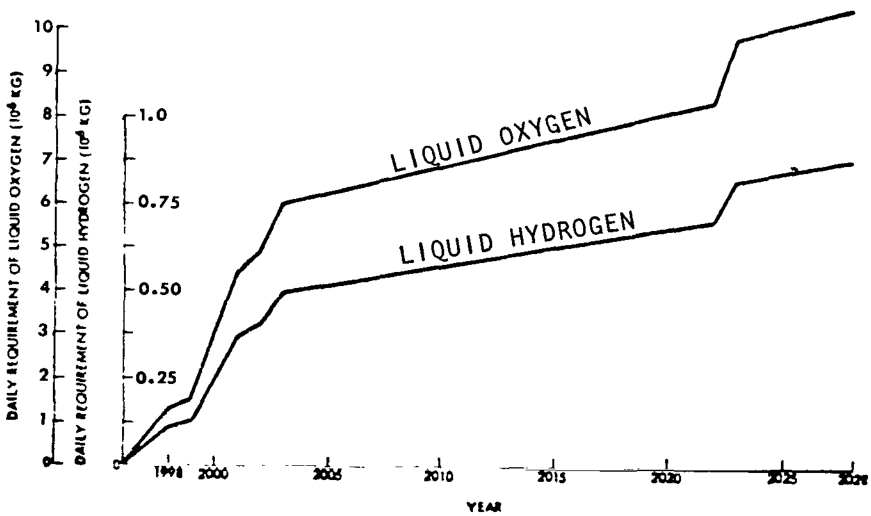


Figure 3.3-2. HLLV Propellant Requirements

There are a number of techniques which can be utilized in order to produce these large amounts of propellant. Currently, liquid oxygen is produced by liquification of air as well as by electrolysis of water. Both these techniques are viable sources of LO_2 for the SPS program.

There are several techniques which may be used in the production of liquid hydrogen. The most feasible alternatives are the production of hydrogen from natural gas, from coal gasification, by the electrolysis of water, by thermochemical processes, and from photosynthetic processes.

Natural gas as a source of hydrogen was considered not to be a viable source. Natural gas is expensive and will be more expensive in the future. It is unreasonable to allow SPS hydrogen production to be dependent upon a natural resource that will be very scarce at the time when the SPS program will require peak hydrogen production.

Thermochemical and photosynthetic processes are awaiting development and there are no assurances that either of these techniques will be able to provide the necessary hydrogen.

The only two techniques which appear capable of providing the required SPS program hydrogen are coal gasification and electrolysis of water.

Figure 3.3-3 presents a block diagram of a typical electrolysis process. The electrical energy for electrolysis can be supplied by a variety of sources; here, it is provided by a nuclear power plant. Desalinized ocean water is split into oxygen and hydrogen, then liquified and stored. This process has the advantage that for every pound of hydrogen produced, eight pounds of oxygen are produced simultaneously; more than enough to serve as oxidizer for the HLLV (mixture ratio 6:1). Thus, both propellants are produced in a single operation and at the same production facility.

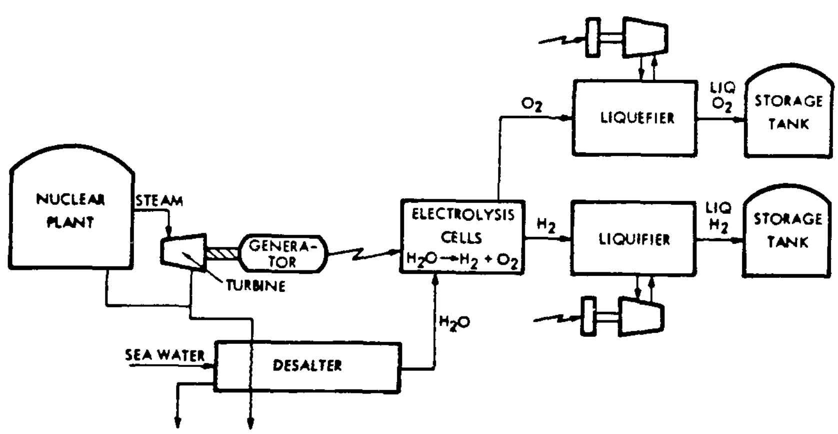


Figure 3.3-3. Typical Electrolysis Process

The power required to produce LH_2 and LO_2 by electrolysis of water is indicated in Figure 3.3-4 (data are for the General Electric solid polymer electrolytic cell). Most of the power is consumed in the splitting of water. At a hydrogen production rate of 1000 metric tons per day, nearly 3 GW of power are required. This means that, near the end of the SPS production phase, the energy equivalent of nearly one half of an SPS will be required.

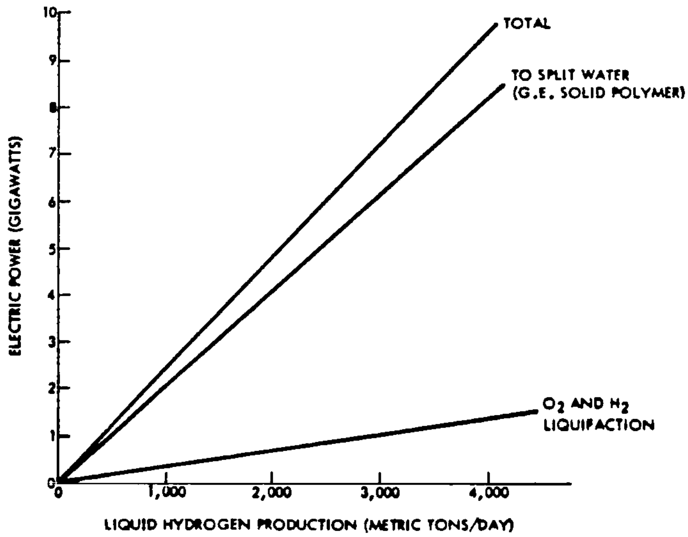


Figure 3.3-4. Electrical Power Required by Electrolysis

Although the power requirements of electrolysis may seem high, the ease of operating such a plant makes it an attractive alternative. The plant can be located along the east coast of Florida, thus eliminating logistical problems, and desalinization of ocean water can be accomplished for only a fraction of a percent of the total energy required for electrolysis.

Coal gasification, on the other hand, is a much more complicated operation. The schematic presented in Figure 3.3-5 depicts a typical coal gasification process. Pulverized coal is vaporized in the presence of steam and oxygen to release hydrogen. After purification, the gaseous hydrogen is liquefied and stored.

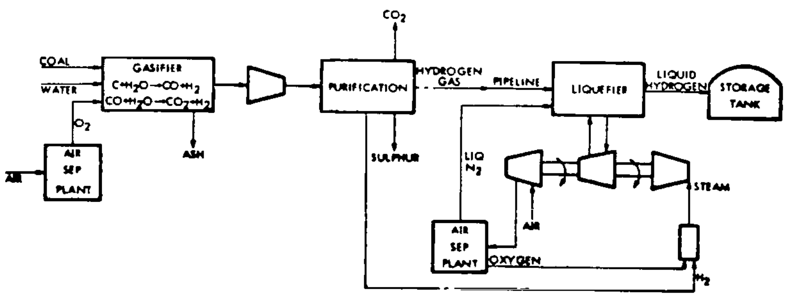


Figure 3.3-5. Typical Coal Gasification Process

This technique, unlike the electrolysis of water, produces only hydrogen, and the required oxygen must be produced by the liquefaction of air.

Coal gasification also produces significant percentages of carbon, carbon-dioxide, and other pollutants. Every kilogram of hydrogen produced requires 6.4 kg of coal, 5.3 kg of water, and 6.9 kg of oxygen, and liberates 0.6 kg of carbon and ash. The quantity of oxygen necessary to liberate hydrogen from coal is nearly equivalent to that needed as oxidizer for the HLLV flights, and the total coal consumed throughout the SPS program will be approximately 15% of all the coal mined in the U.S. in 1970.

Figure 3.3-6 presents the power necessary for coal gasification along with that needed to produce liquid oxygen. At the peak SPS production rate, the electric power required to produce propellant is approximately 0.5 GW.

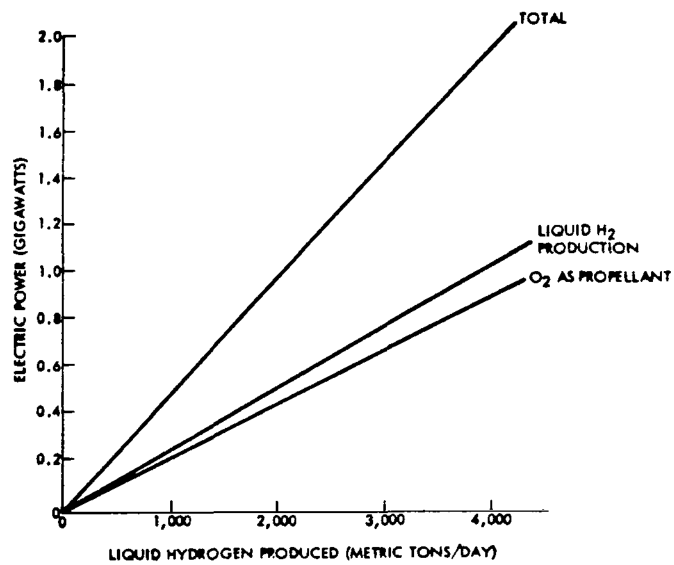


Figure 3.3-6. Power Required for Coal Gasification

A comparison of the energy requirements of electrolysis and coal gasification indicates the power needed by coal gasification is nearly one-sixth that necessary for electrolysis.

Logistically, however, coal gasification is more complex than electrolysis, since it would require transporting large amounts of coal or hydrogen over long distances—from the coal mine to Cape Kennedy. It is, therefore, important to delve more deeply into the specific logistical alternatives of coal gasification.

The major U.S. coal reserves are located in three geographic areas: the Appalachian region, the Mid-Western region, and the Western region (Figure 3.3-7). The Appalachian coal reserves are essentially committed to eastern energy requirements. This coal is located underground and must be mined using costly underground mining techniques. The mid-western coal has a high sulfur content and presently cannot meet the pollution standards of most cities—making it nonusable. The western coal is low in sulfur and is essentially undeveloped. It is surface coal and, therefore, relatively inexpensive to mine. Abundance, low sulfur content, and undeveloped nature make the western coal reserves the prime source of coal for SPS hydrogen production.

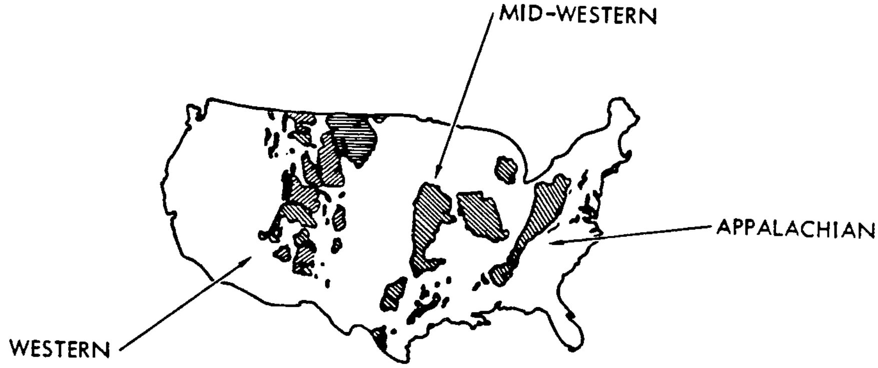


Figure 3.3-7. Geographical Location of U.S. Coal Reserves

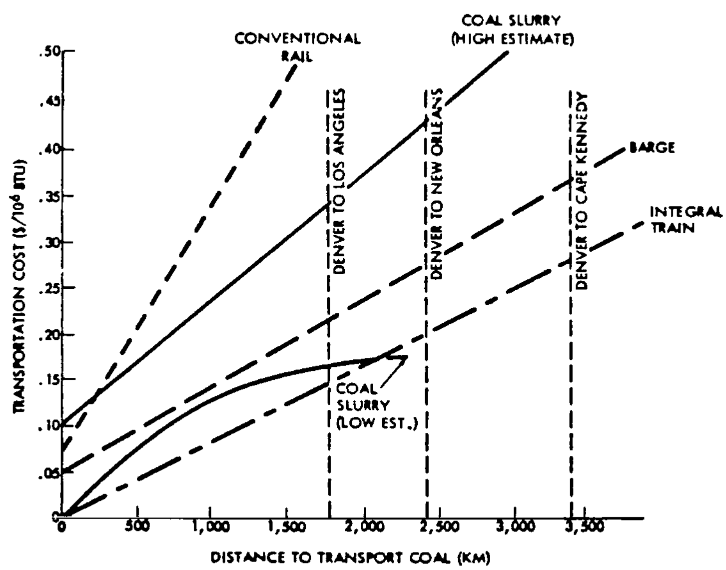
However, the location of the western coal reserves necessitates a complicated logistics scenario. There are numerous questions which must be answered in order to develop the most efficient and cost-effective means of handling coal gasification from the mine to the launch site.

The main question is whether coal should be shipped from the mine to the launch site (where it would be gasified), or whether coal should be gasified at the mine and then the hydrogen shipped to the launch facility.

Since coal gasification requires large amounts of water, it may be advantageous to ship the coal to a location with an abundant water supply. It is, therefore, important to analyze the various alternatives available for transporting coal.

Figure 3.3-8 presents the relative cost of transporting coal by various techniques. Coal slurry is 50% water and 50% coal by weight. The difference between the two coal slurry curves indicates differences in estimated terrain effects and construction costs.

Figure 3.3-8. Cost of Transporting Coal



An integral train is a unique concept which does not exist at the present time. It consists of a system of cars which are much larger than conventional train cars, having the capability of quick side-dumping; motors at both ends alleviating the need for turning the cars around for the return trip; and semi-permanently attached cars.

Barging coal is not a feasible alternative since the coal must be barged through the Panama Canal and the long distance involved makes barging too costly.

Figure 3.3-8 indicates that, for the distances considered here, the integral train concept may be the least expensive means by which to transport coal from the mine site to a coal gasification plant (at Cape Kennedy), although coal slurry may also be competitive once further information has been compiled.

Another alternative is to gasify coal at the mine site and ship gaseous hydrogen to the launch site. A comparison of the costs of shipping coal, with those of shipping gaseous hydrogen, is presented in Figure 3.3-9. The two cost curves for shipping gaseous hydrogen result from considering the construction of new pipelines as opposed to using portions of existing natural gas lines.

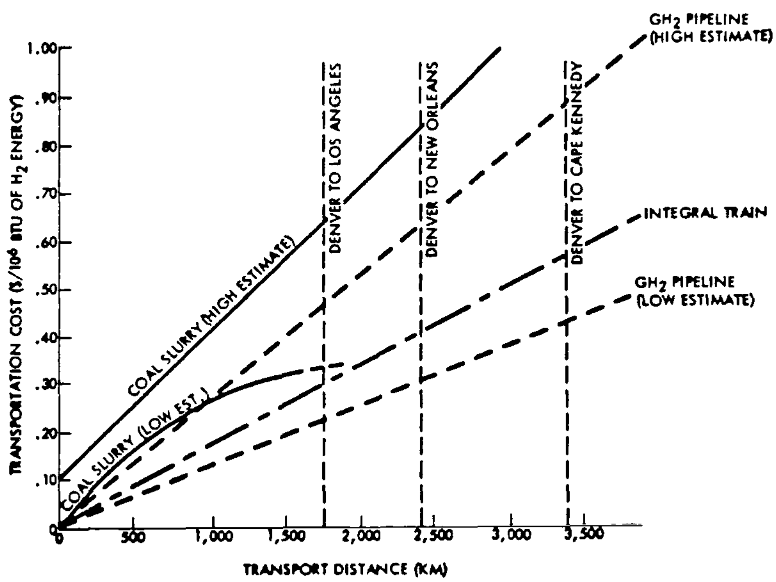


Figure 3.3-9. Relative Transportation Costs

As is evident in the curves, no firm conclusion can be drawn at the present time concerning the most cost-effective technique. Until a more definitive scenario is developed, it is not clear whether coal should be shipped from the mine to the launch site and gasified, or whether hydrogen should be produced at the mine and shipped to the launch facility.

A factor which may influence this choice is the amount of water required by the coal gasification process. The SPS program will require approximately 10,000 acre-feet of water per year for nominal coal gasification production of hydrogen. This is a very small percentage of the total watershed available in the area; although this resource is highly dispersed and not concentrated in rivers and lakes. The watershed is sufficiently large, however, so that by judicious planning, the necessary water can be accumulated for coal gasification.

An alternative solution to the water requirement would be to ship water from the Pacific Ocean. Figure 3.3-10 presents the power required to transport water to the western coal region from the west coast. The data indicate that the energy needed is on the order of 0.01 GW, which is a very small percent of the power necessary for coal gasification.

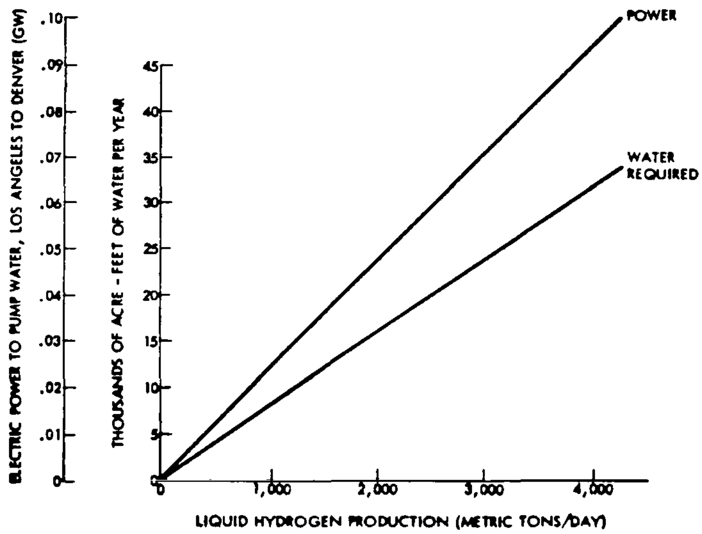


Figure 3.3-10. Water/Power Required for Coal Gasification at Mine

The conclusion, then, is that even if there is insufficient water within the western region environment, the power necessary to transport it from the west coast is not significant when compared to the total coal gasification power requirement.

Figure 3.3-11 presents a summary of the costs for various alternatives in the production of liquid hydrogen and liquid oxygen. The costs are indicated as the cost of producing one pound of liquid hydrogen and six pounds of liquid oxygen per pound of liquid hydrogen.

Although the integral train seems to be the least expensive alternative for coal gasification at the present time, the uncertainty in the production scenarios precludes making this final decision.

It is also important to note that, although electrolysis requires five times the power necessary for coal gasification, electrolytic production of propellant is only twice as expensive as coal gasification—after considering the logistical costs of transporting coal or hydrogen from the western coal reserves to Cape Kennedy.

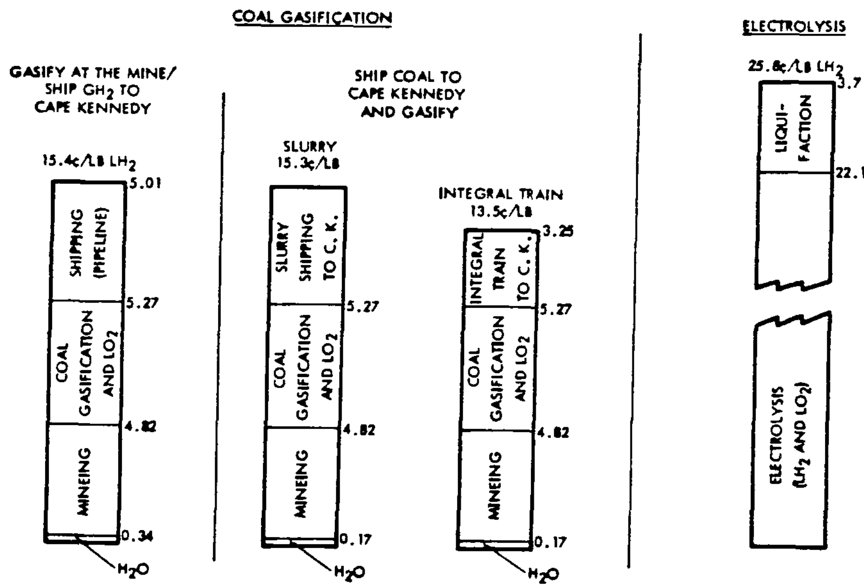


Figure 3.3-11. Cost Summary

This analysis has not considered environmental factors, operation, maintenance, and other problems unique to a system which transports material 3000 km. It is clear that operational considerations could easily make electrolysis (at the launch site) the most attractive technique.

Regardless of the technique which is selected to manufacture hydrogen and oxygen, a storage facility will be required to absorb the effects of unforeseen circumstances and ensure a smooth HLLV launch schedule. The size of the storage facility will depend on the reliability of the propellant production scenario. Figure 3.3-12 presents liquid hydrogen storage area as a function of storage capacity. These data take into account peripheral dikes and advanced techniques in the construction of liquid hydrogen storage facilities.

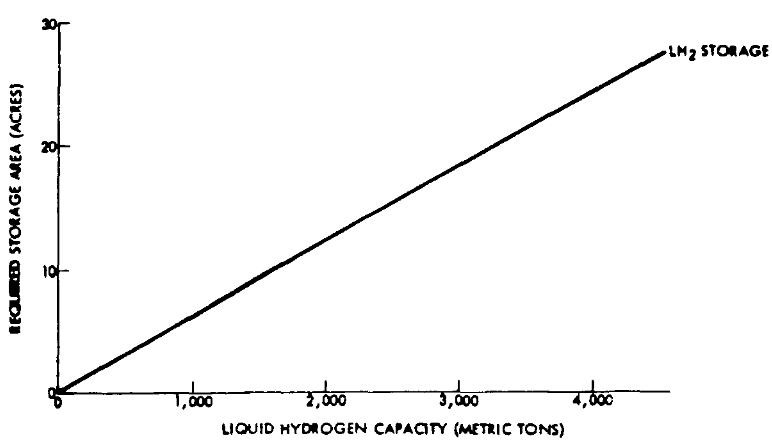


Figure 3.3-12. Liquid Hydrogen Storage Requirements

3.4 ENVIRONMENTAL CONSIDERATIONS

Since the entire atmosphere from the ground to GEO will be subject to rocket exhaust products, it is correspondingly expected that all regions of the atmosphere will be perturbed by these effluents at least to some extent. The main reason for concern arises form both the sizes of the vehicles, i.e., their effluent emission rate, and their launch frequency. In the troposphere, the ground clouds formed during launch of the HLLV and, to a lesser extent, the PLV, could give rise to some local inadvertent weather modifications and air-quality effects. The weather modifications can result from two sources. First, the injection of thermal energy and moisture can cause a cloud scale dynamic response of the local atmosphere which may lead to changes in local circulation and cloud population. Second, the injection of cloud condensation and ice nuclei can, at a micro scale, effect cloud physics processes that would ultimately influence cloud formation, precipitation, and possibly haze or fog formation. The air-quality effects arise from the entrainment of surface debris and dust, after-burning of exhaust product in the ambient air, and injection of rocket fuel impurities. Use of fuels such as RP may lead to concentration of SO_2 and other pollutants that would lead to or exacerbate local air pollution problems. After-burning of even clean fuels may result in levels of oxides of nitrogen that could lead to air pollution problems, especially if the U.S. EPA sets a fairly low ambient NO_x air-quality standard. Emissions of sulfur and nitrogen compounds could also contribute to acid rain problems, but the levels are not expected to be very significant at this time.

Moving higher into the atmosphere, we do not anticipate any significant stratospheric impacts from the use of CH_4, H_2 type fuels since the exhaust products are indistinguishable from ambient constituents present in substantially higher concentrations. However, as we move higher up in the atmosphere, it becomes increasingly more rarified and, consequently, more susceptible to large-scale perturbations. By the same token, our understanding of such perturbations, not to mention the natural state of the upper atmosphere, declines with increasing altitude. We are currently at the stage of having identified what effects could occur, but are severely limited in our ability to predict what will occur when the SPS is implemented. Effects that could arise in the mesosphere include chemical composition and dynamic changes brought about by the addition of water vapor, especially above 70 or 80 km. This water vapor could also contribute to the formation of high-altitude ice crystal clouds. The rate and location of water vapor injections will also influence ionization levels in all regions of the ionosphere from the D-region (50-90 km) up through the F-region which is around 350 km. Injections of rocket exhaust directly in the F-region will produce dramatic reductions in local plasma density and, therefore, influence radio wave propagation and perhaps other high-altitude physical phenomena. Avoiding injections directly into the F-region will mitigate processes (not fully understood at present) and will move at least some of the exhaust products injected both above and below into the F-region. What is of greatest concern in this regard are the long-term chronic effects in the ionosphere of once or twice daily injections of water and hydrogen molecules over a 30 or more year period.

Above the F-region, the principal exhaust products will be AR^+ ions from EOTV flights and H_2O and H_2 from POTV flights. Effects may arise both from the next accumulations of H-atoms and the energy associated with these injections combined with that of HLLV and PLV circularization and deorbit burns. This addition of thermal energy and mass may lead to changes in temperature and density that would influence satellite drag and stability of the Van Allen radiation belts. Interactions of these exhaust products with ambient neutrals and plasma will give rise to enhanced background levels of airglow which may interfere with remote sensing. Also, the thermal or radiation transfer properties of the thermosphere may be altered by the addition of large amounts of water vapor.

Finally, the injection of energetic AR^+ ion beams containing both mass and energy large in magnitude, compared with that naturally present in the plasma and magnetosphere, may significantly alter both the composition and structure of this most rarified region of the satellite environment. In addition to possible alterations of the radiation doses received by vehicle passing through or residing in the radiation belts, such injections may given rise to alterations in the intensity and frequency of high-energy particle precipitation events at mid to high latitudes. Electromagnetic wave propagation could be influenced by plasma instabilities triggered by the AR^+ ion injections. Finally, some speculation has been given to the influence that SPS injections in the magnetosphere may have on the so-called solar weather effect. A related effect would be changes that may result from AR^+ injections on the manner in which the magnetosphere responds to changes in the solar wind and magnetic storms. Large ionospheric auroral currents associated with such storms have been observed to cause current surges and circuit breaker trips in long-line telephone systems and electric power transmission lines in northern latitudes. Alteration of the latitude at which these events occur could make their impacts on populated areas more significant.

While present knowledge does not permit a definitive statement regarding mitigating strategies, some suggestions deserve future attention. These include the use of alternative ions such as H+ or the use of neutrals instead of ions. Trajectory shaping, thrust scheduling, and selection of propellant type on the basis of altitude range should also be considered.

Data are required on the concentrations and fluctuations of upper atmospheric ambient constituents and on perturbations caused by rocket effluents. Hard data are especially needed on effects of AR+ and chemical injections above 200 km. The SPS ground-based exploratory development (GBED) program should include ample opportunity to design experiments that could combine technology testing with the atmospheric effects studies. Without these experimental data, it will be difficult to substantially reduce uncertainties—especially regarding effects above 500 km. Small-scale space experiments should be conducted during the GBED program to at least stimulate the refinement of theoretical modeling techniques and planning of larger-scale more sophisticated experiments. In addition, GBED time frame experiments will provide a basis for development and refinement of both ground-based and airborne diagnostic instrumentation.

4.0 CONCLUDING REMARKS

The reference SPS transportation system has satisfied its intended objective in meeting the needs of other SPS related studies in support of technical and operational information required to conduct environmental, socioeconomic, and comparative assessments. However, as in most studies, potential improvements are recognized and/or developed which lead to better systems definition and improved concepts or approaches. Some of the major changes recommended in the SPS transportation system reference concept are briefly summarized in the following paragraphs.

HEAVY-LIFT LAUNCH VEHICLE (HLLV)

The consensus at this time indicates that the selection of a smaller payload HLLV configuration (i.e., between 100,000 and 150,000 kg) would be more desirable from the standpoint of lower nonrecurring cost and commonality with the STS, reduced noise and sonic over-pressure, thus eliminating the need for off-shore launch pads, and the potential for alternate programs application. The selection of series vs. parallel burn requires further study. The ability to utilize smaller/fewer engines in the parallel burn concept must be traded against aerodynamic interference effects and mating/separation issues along with engine propellant feed transfer from the booster to the orbiter prior to separation. Further analyses of ground-level wind effects on the larger (taller) series burn concept along with vehicle erection techniques must be pursued.

PERSONNEL LAUNCH VEHICLE (PLV)

The utilization of the HLLV for personnel transport during the SPS operational phase proves to be a more cost-effective approach by elimination of the cost and complexity of an additional transportation element in the SPS inventory.

CARGO ORBITAL TRANSFER VEHICLE (COTV)

The significant advantages to be gained by the use of an electric propulsion system, primarily in the area of a reduced number of HLLV flights required to transport orbital transfer propellants, essentially drives that concept selection. In addition, the potential for self-annealing of radiation damage experienced by the solar cells in transitioning the Van Allen radiation belt would certainly favor the use of a GaAs power source, regardless of SPS concept selection. The further advantages offered by technology advancement features in ion engines such as large diameter (one meter or more), high-current density operation, and direct power drive deserve continued study and technology feasibility verification.

The specific size, payload and trip time variables are very flexible and must be firmly established later on the basis of a specific SPS concept selection. The final selection of argon as a propellant awaits the results of further environmental assessment studies. Other propellants, such as hydrogen, might prove to be more viable environmentally.

PERSONNEL ORBITAL TRANSFER VEHICLE (POTV)

Again, the present consensus is that the POTV should be a single-stage chemical element capable of transporting the required personnel and priority cargo from LEO and GEO and refueling in GEO for the return trip to LEO. In addition to reducing operational complexity of multiple-stage operation, the transport of return propellants to GEO by the COTV is most cost effective. The specific size and payload capability is dependent upon SPS scenario selection and personnel rotation requirements. The need for compatibility with the STS cargo bay is not necessarily a valid requirement.

GROWTH SHUTTLE TRANSPORTATION SYSTEM (STS)

The reference growth STS concept was selected when it was assumed that it would serve as a PLV throughout the construction and operational phase of the SPS. With the alternate HLLV concept (both smaller and serving as the personnel carrier), this extensive modification of the STS may not be required. The minimum-growth alternative of replacing the SRB with LRB could satisfy the early developmental and pilot plant SPS requirements. Final-growth STS selection should consider the needs of other potential contemporary space endeavors.

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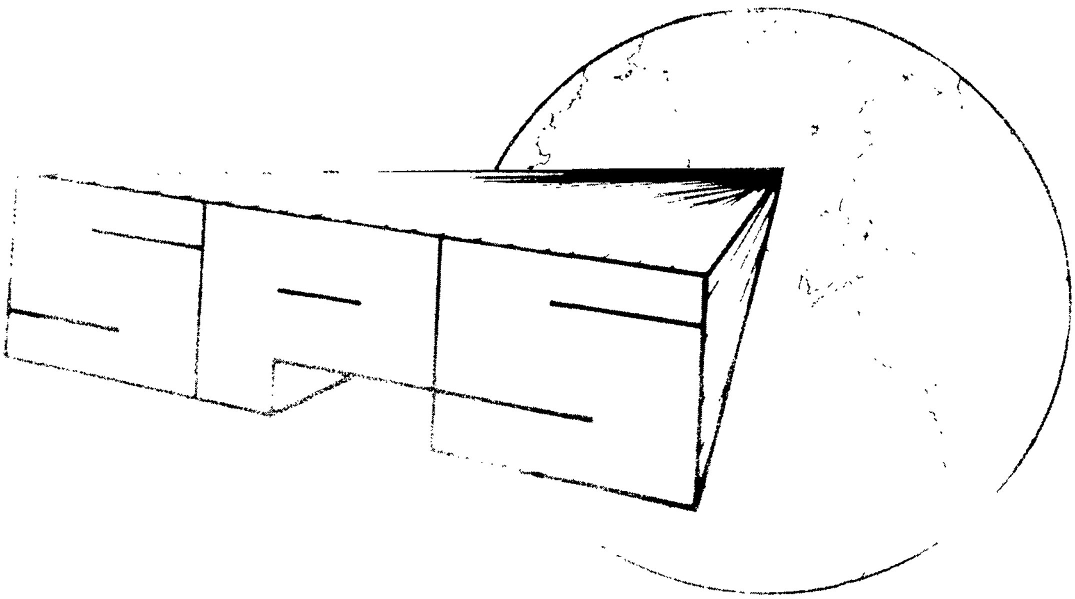
APPENDIX

SPS SPACE TRANSPORTATION WORKSHOP

January 29-31, 1980
Huntsville, Alabama

THE FINAL REPORT OF THE SPS SPACE TRANSPORTATION WORKSHOP

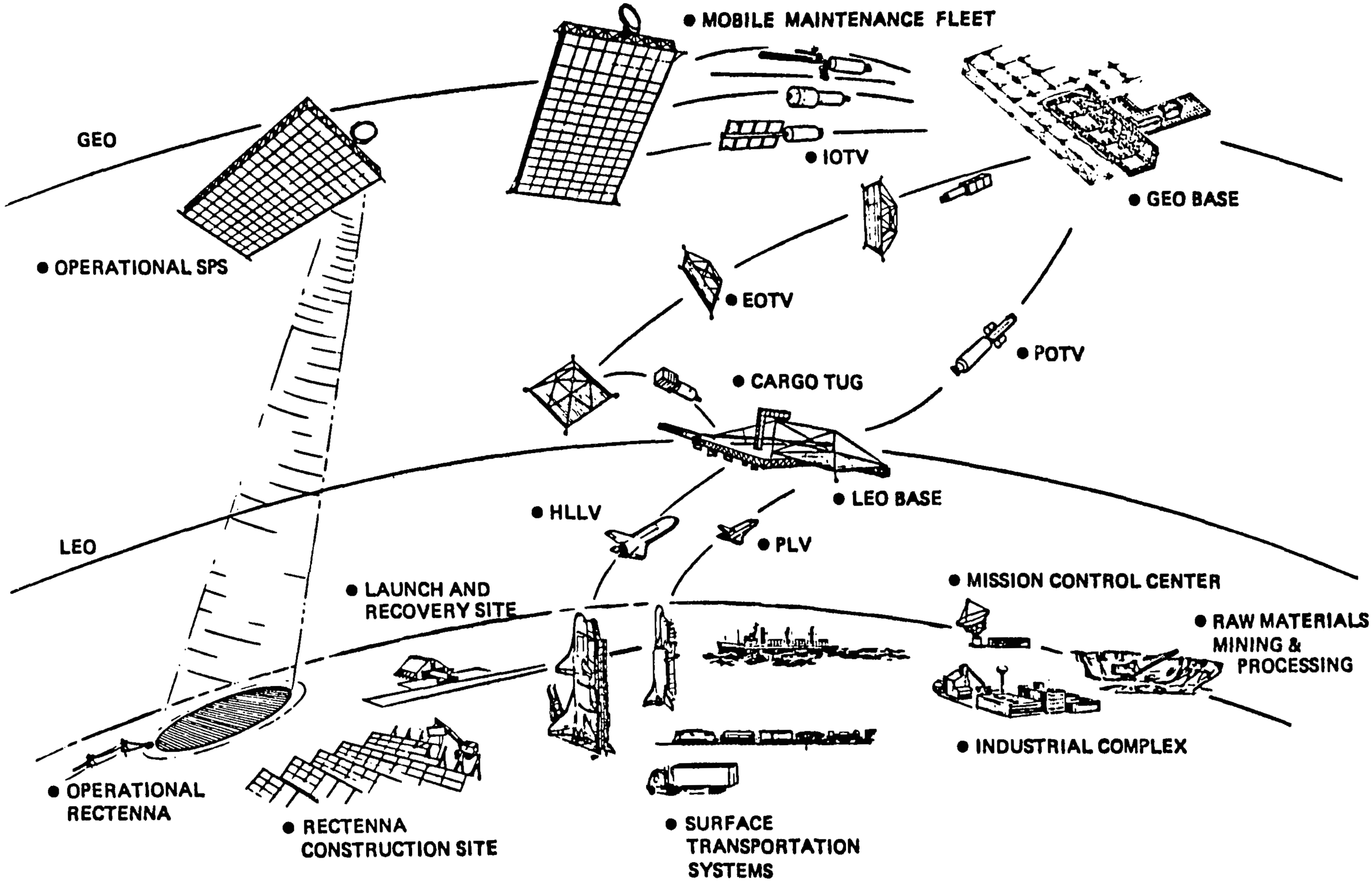
January 29-31, 1980
Sheraton Inn — Huntsville
Huntsville, Alabama



October 1980

Prepared for the
Advanced Systems Office
Program Development Directorate
Marshall Space Flight Center
Huntsville, Alabama

Prepared by the
Johnson Environmental and Energy Center
The University of Alabama in Huntsville



Satellite Power Systems (SPS)—Space Transportation Vehicles and Operations

THE FINAL REPORT OF THE SPS SPACE TRANSPORTATION WORKSHOP

January 29-31, 1980
Sheraton Inn — Huntsville
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Prepared for the
Advanced Systems Office
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The University of Alabama in Huntsville

Reviewed and Approved by:

David L. Christensen
Senior Research Associate

NOTICE

FOREWORD

Although this workshop was not intended to reach major decisions on satellite power system (SPS) transportation technology, it was expected to assist in mapping the next phase of work. In the opening words of Carl Schwenk, its purpose was to search the contemporary reference system for "show-stoppers" and to ask such questions as the following:

- Does space transportation pose insurmountable difficulties in realizing an economical SPS?

- Do space transportation operations create unavoidable environmental disasters?

- Can the aerospace community state with confidence that space transportation-systems technology will evolve to provide low-cost delivery of massive payloads to orbit?

- Will technology permit low-cost operations and maintenance of space-based transportation systems?

In addition, the workshop was asked to identify the dominant issues that call for the earliest, more detailed studies, and to assess the credibility of the prevailing plans for further efforts.

In all frankness, none of these tasks could be fully dispatched, initially because of the brevity of the meeting compared to the volume of relevant material to be digested, but fundamentally because the problems are not so simply defined.

Statements of technical viability and economic competitiveness are meaningful only when normalized in terms of all tangible and intangible benefits which derive from a successfully completed program, and in terms of full costs of alternative energy strategies. Neither parameter has been, nor likely can be, determined with any confidence over the projected development or operating span of the SPS at the present time.

What did clearly emerge from the vigorous discussions in the working groups, however, and persists through the resulting sections of this report, was that SPS is an attractive, challenging, worthy project, which the aerospace community is well prepared and able to address. The mature confidence and authority with which the assembly of contractors, agency delegates, and consultants dealt with the long succession of technical, social, economic and political issues left the clear impression that if some persuasive constellation of purposes--public or private, peaceful or military, national or internationl--should assign this particular energy strategy a high priority, it could be accomplished.

Robert G. Jahn

Chairman

ACKNOWLEDGMENTS

The SPS Space Transportation Workshop, held under the auspices of The University of Alabama in Huntsville January 29 through 31, 1980, at the Sheraton Motor Inn, addressed in two and a half days questions that will require the efforts of many workers for the next 5 to 10 years before a rational decision can be made concerning the variety of vehicles and transportation systems needed for the erection and operation of these potentially vital energy systems.

Approximately 60 participants, listed in the following pages, provided expert and devoted efforts that are presented in the body of this report. Their wholehearted participation represents an essential contribution to the ongoing development of an understanding of the promise of satellite power systems and, in particular, their space transportation aspects.

The administrative support of David L. Christensen, Kenneth Rossman, and David B. Cagle of the Johnson Environmental and Energy Center at The University of Alabama in Huntsville was essential to the success of the workshop. The secretarial assistance of H. Barbara Guillet, Marionette Bishop and Patricia Hein is gratefully acknowledged.

It is hoped that the workshop will assist in identifying further work necessary to the realization of SPS as a significant element in meeting future energy requirements of the Earth.

Pres Layton

J. Preston Layton

Co-chairman

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SUMMARY

In the course of studies of SPS over the past 10 years, it has become apparent that the space transportation requirements are major elements in the technical and economic realization of the concept.

The space transportation system generally consists of a trajectory from Earth's surface to a low-Earth orbit (ESLEO) and a transfer from low-Earth orbit (LEO) to a geosynchronous altitude (GEO) or an orbit-to-orbit (OTO) transfer, which includes both a transfer through the Van Allen Belts and intraorbital operations.

A number of concepts have been studied for enhancing the capabilities of the current Shuttle Transportation System (STS) so its role can be extended to early SPS demonstrations. Beyond the growth and derivative versions of the present Shuttle concept lie the possibilities for relatively low-cost transportation for ESLEO, which is a major factor in the economic feasibility of SPS.

The initial steps in enhancing the operational capabilities of the Shuttle will probably include using the liquid-propellant boost module, derived from the Titan ICBM, and liquid-propellant, strap-on boosters to replace the current solid-propellant, strap-on boosters. Following this modification, there may come advanced versions employing boosters with aerodynamic surfaces. Such developments will be consequences of the direction that the national space program takes in the next two decades.

Entirely new heavy-lift launch vehicles (HLLV) will need to be identified before the economic and environmental problems of the prototype, or even demonstration, SPS can be resolved. The need for single-stage vehicles capable of

achieving low-Earth orbits, using either vertical or horizontal take-off and landing, remains to be determined by future analyses or the course development of events in booster technology. In any event, considerable analysis, research, and technology will be required before the choice can be made. Social impacts in environmental areas will need to be considered.

The ESLEO operational requirements and costs dominate the SPS space transportation scene. Launch-vehicle technology must be driven to a rather sophisticated extent to meet the needs as currently perceived and this perception is immature at the present time. The workshop decided that, although rather advanced technology and well-developed operational management would be required, it was proper to target the average cost of gross cargo payloads into LEO at $30 (1979)/kg for construction of the initial SPS. The further cost goal for repetitive construction of 30 to 60 SPS would need to be reduced to $15 (1979)/kg for all operational payloads for ESLEO and would require the use of advanced, long-lived vehicles with a sophisticated operational organization, probably utilizing offshore equatorial launch sites.

The wide variety of OTO missions in support of the SPS demonstration, construction and operation needs to be better defined before the vehicle concepts can be identified. Chemical orbital transfer vehicles (OTV) require further analysis, technology refinement and a reasonably early start on development to provide a capability that is needed in even the present STS. OTO, including intra-orbit, requirements of the 1980s need to be coordinated with SPS needs for chemical rocket OTVs in the 1990s and beyond. In-orbit propellant processing should be fully assessed for early employment.

Much work is needed on the conceptualizing and research on electric rocket propulsion systems for SPS applications. Mission analyses including optimized high- and low-thrust acceleration trajectories are needed that serve the SPS requirements. High-power ion thrusters and magnetoplasmadynamic (MPD) thrusters urgently need development to ascertain their characteristics. Much better coordination between research in the electric-rocket propulsion system technology planning and support, and the overall future requirement for this kind of propulsion, including the SPS, is needed.

More advanced propulsion systems such as dual-mode solid-core nuclear fission systems, gas-core nuclear rocket stages and mass driver reaction engines (MDRE) need sustained attention. OTO propulsion using high-power lasers should also be given attention.

The present ground-based exploratory development program in space transportation for SPS is inadequate and such content as it has needs to be restructured. Its primary efforts should be directed toward strengthening the present concepts but, at the same time and just as importantly, we should be carefule not to close off any promising concepts or technologies. Operations and social impacts are also important considerations. If the program is intended to be the next phase for SPS, it needs to be reconceived from the ground up with an increase of an order of magnitude in funding.

A greatly increased program of SPS space transportation analysis, research and technology is clearly needed. Efforts must be devoted to areas of system analysis and technology readiness (including ground and space testing) that will reduce space-transportation cost uncertainties in the next five to ten years.

Although the consensus of the workshop supported the future prospects of the SPS, it was generally believed that much work is needed before space transportation choices can be made.

I. INTRODUCTION TO SPS SPACE TRANSPORTATION

A. Historical Background

The Sun provides the basis of all life on Earth and is the primary energy source. Man has been tapping the Sun's energy in various forms for many centuries. Dependence on different energy forms has varied as the demands of man's societies have changed and increased, especially in the past several hundred years. The rate of energy usage has increased exponentially under the global pressures of the industrial revolution and the pervasiveness growth of technology throughout the world.

It has been evident for some years that petroleum fuels, on which industrial activity and the standard of living of most countries depend, would reach the peak of their economic production within a few decades and be exhausted in a foreseeable time thereafter. Coal is a major fossil fuel with extended reserves, but also with economic and societal difficulties. At present, nuclear-fission energy is seen to have only a limited and special usefulness, while controlled-fusion concepts must still be found to be feasible and practicable.

The use of direct solar energy for base electrical utility power is being studied as a renewable source of almost limitless power and is believed to hold great promise; however, the state-of-the-art of the various system concepts has not yielded a clear direction for solar power systems development. A large number of technologies and systems are being studied and developed under the energy programs of the United States and elsewhere. Thermal and photovoltaic ground-based central power systems are both under development. The possiblity of space-based, solar-utility power was first suggested in the late 1960s by Dr. Peter Glaser of Arthur D. Little, Inc. Early SPS design concepts are shown in Figure 1. These concepts were based on the use of solar photovoltaic (silicon) cells and microwave transmission to Earth

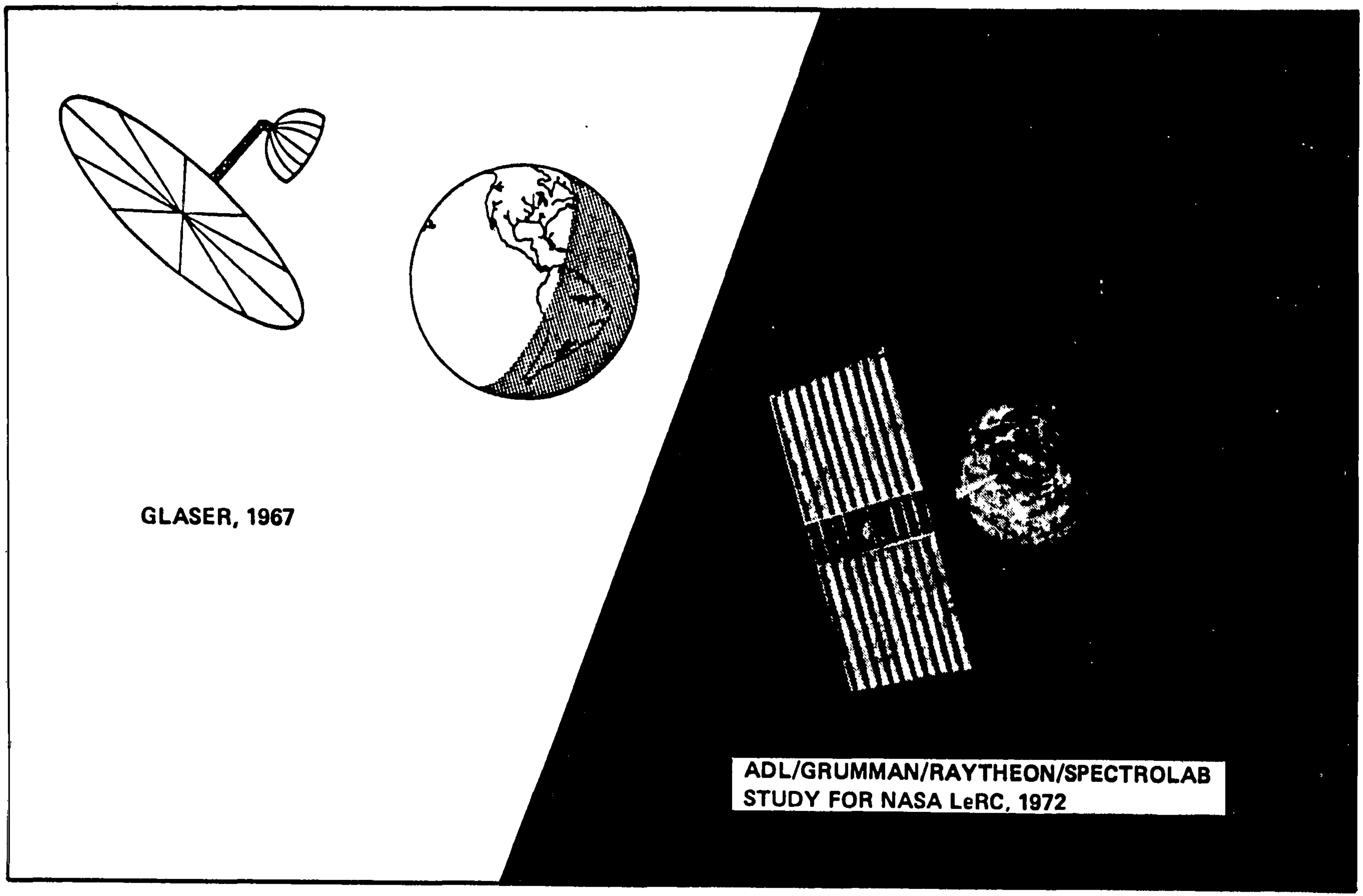


Figure 1 Early Satellite Power System (SPS) Design Concepts

at the 10-GWe power level.

B. Description of SPS Concepts

A considerable number of SPS concepts have been studied in more or less detail (as shown in Figure 2) by Boeing Aerospace Corp. The photovoltaic designs are primarily planar with silicon solar cells in rectangular areas of 50 to 100 km^2 and a mass in geostationary orbit of 50 to 100 Gg. Other designs with thermal solar collectors and Brayton- or Rankine-cycle power conversion have similar areas and masses. Similar concepts have been studied by Rockwell International and others with essentially the same results. Rockwell has shown a preference for gallium arsenide photovoltaic cells.

Figure 3 shows SPS space construction detail that gives an appreciation of the scale of the undertaking. In this illustration a construction base in geostationary orbit is shown with surrounding SPS structure and heavy-lift and personnel vehicles.

C. Current Status of SPS Program

The SPS studies and analyses have been carried out on a very broad base under the direction of the Department of Energy (DOE) in a joint effort with the National Aeronautics and Space Administration (NASA). The work has been distinguished by the breadth of a long-term conceptual development and consideration of broad societal and environmental issues. Economic factors relative to competing energy systems have also been considered in the year 2000 and beyond.

1. Reference Systems

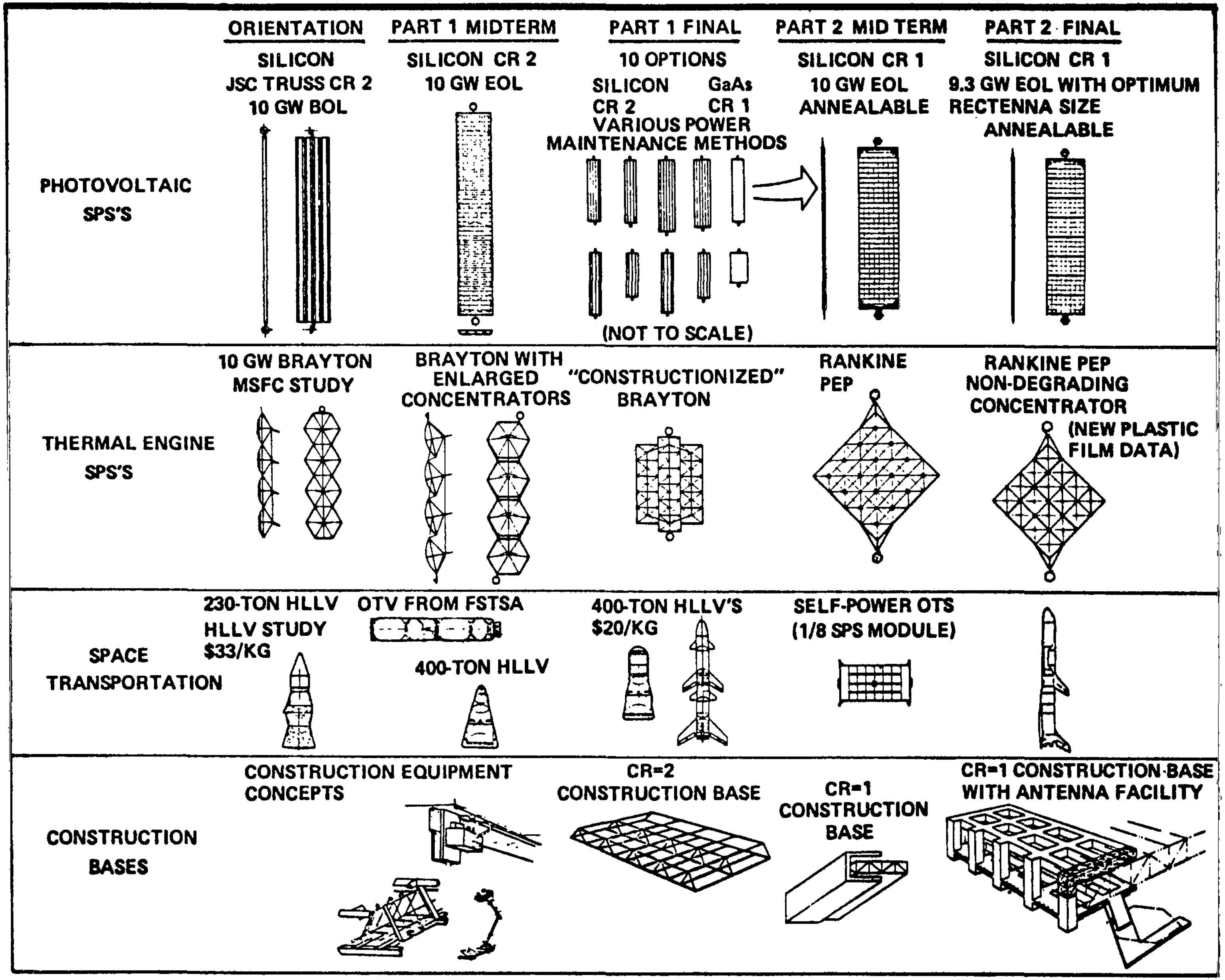


Figure 2 Various Satellite Power System Design Concepts - Boeing 1970s

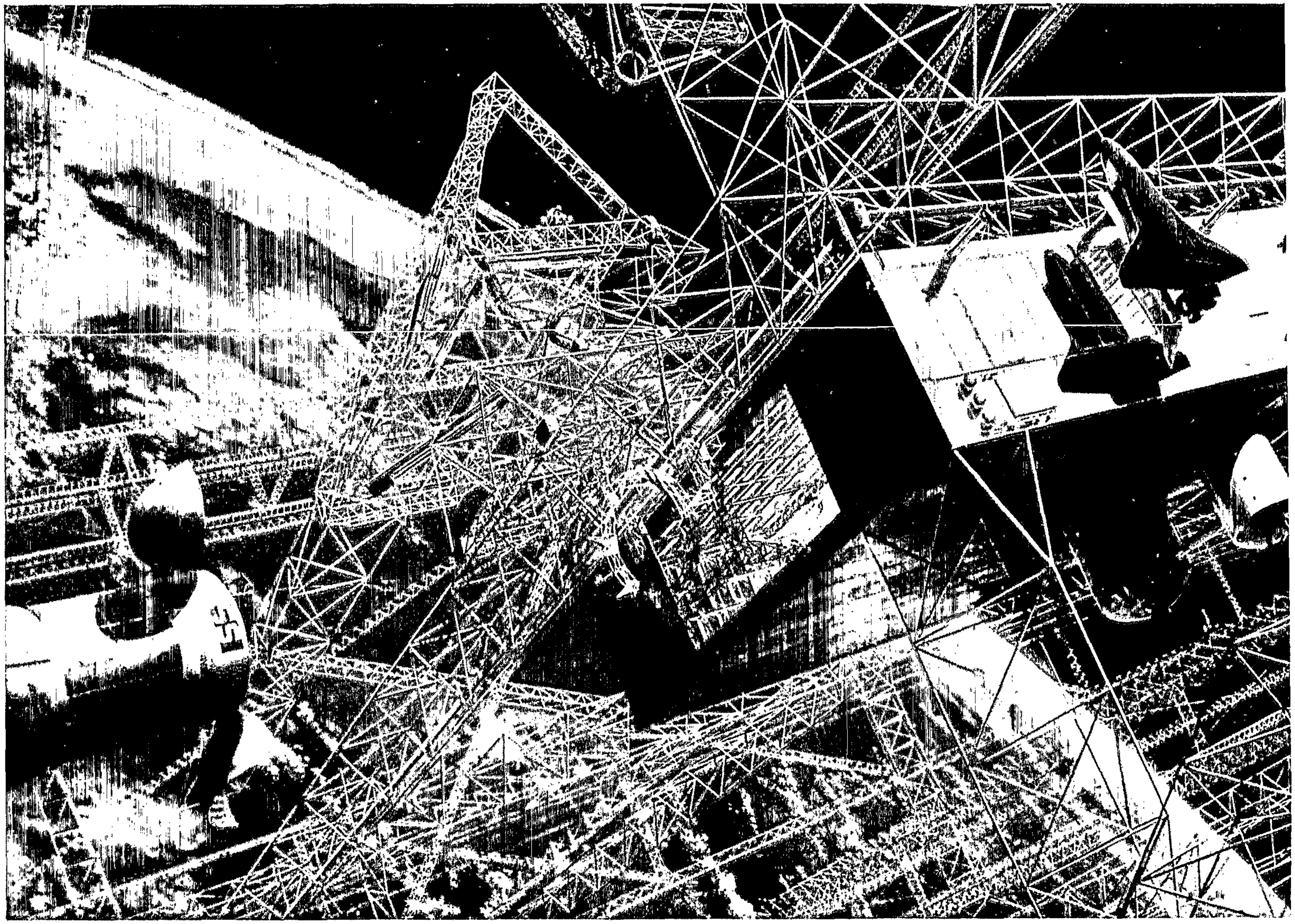


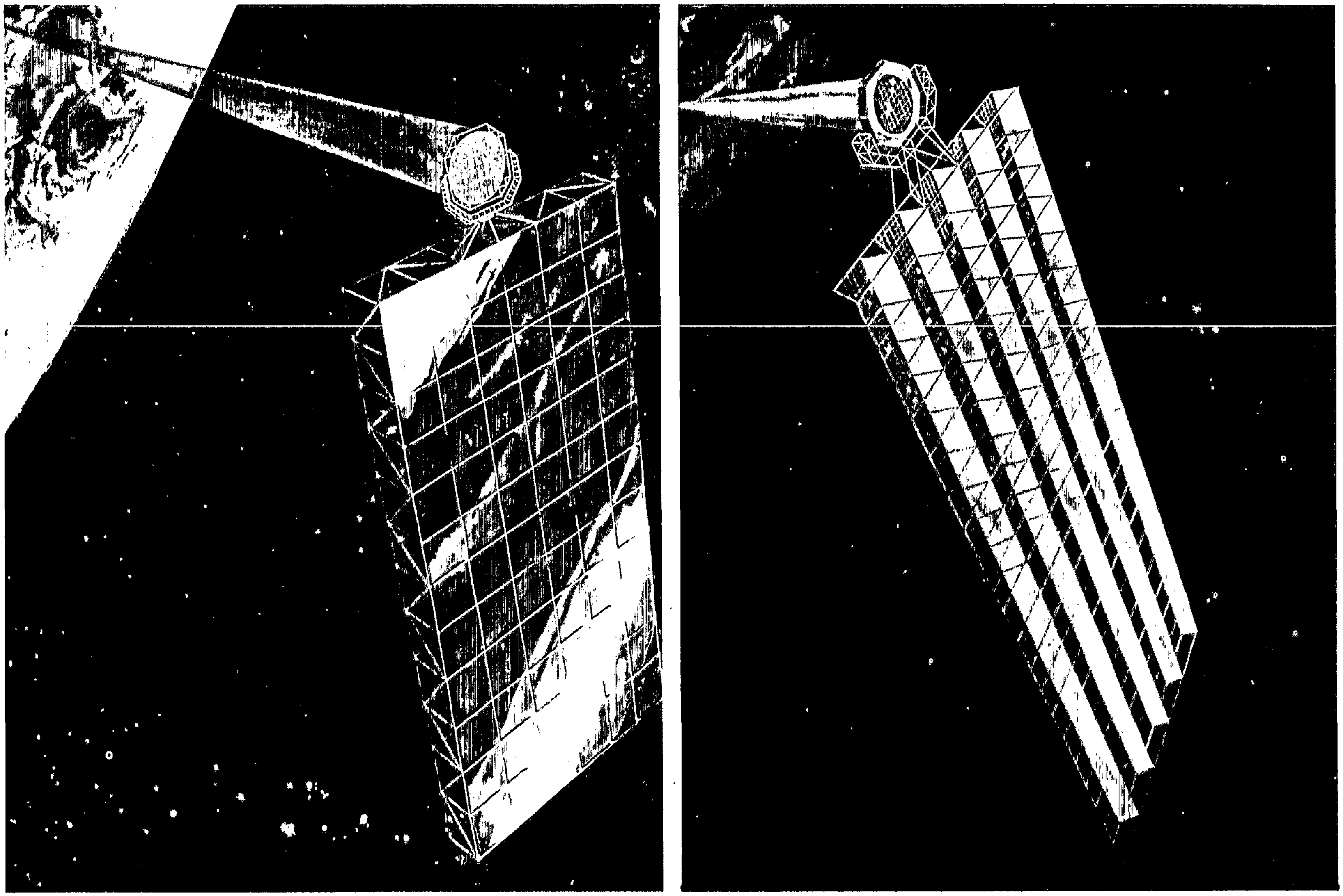
Figure 3 Satellite Power System - Space Construction Facility Detail

In recent months two photovoltaic reference systems have been identified, as shown in Figure 4, to serve as mileposts in further consideration of SPS from the standpoint of basic feasibility and in competition with other energy systems in the early years of the 21st century.

Alternative concepts still need to be considered carefully in some detail before development is undertaken, and much research and technology effort, including ground and space tests, is required before a definitive conclusion can be reached or a system configuration selected. Two recent concepts are shown in Figure 5, and many others will need to be considered.

2. Space Transportation Requirements

All studies of the SPS have identified the space-transportation element as a major, and even critical factor in the overall prospects of the system. The frontispiece shows the variety of space vehicles and operations currently identified in the construction and maintenance of the SPS. The ESLEO-transportation requirement represents the most substantial challenge in advanced large chemical rocket vehicle technology and costs. The OTO requirement, especially from LEO to GEO, and intra-orbit operations are also very demanding and will necessarily involve new vehicle technology and operations. Space basing will certainly be required. Electric rockets and other advanced propulsion capabilities may be needed. The current status and future prospects for satisfying the SPS space-transportation requirements as viewed by the workshop participants are presented in the sections that follow.



Silicon Solar Cells

Gallium Aluminum Arsenide Solar Cells

Figure 4 SPS Reference Systems - Late 1979

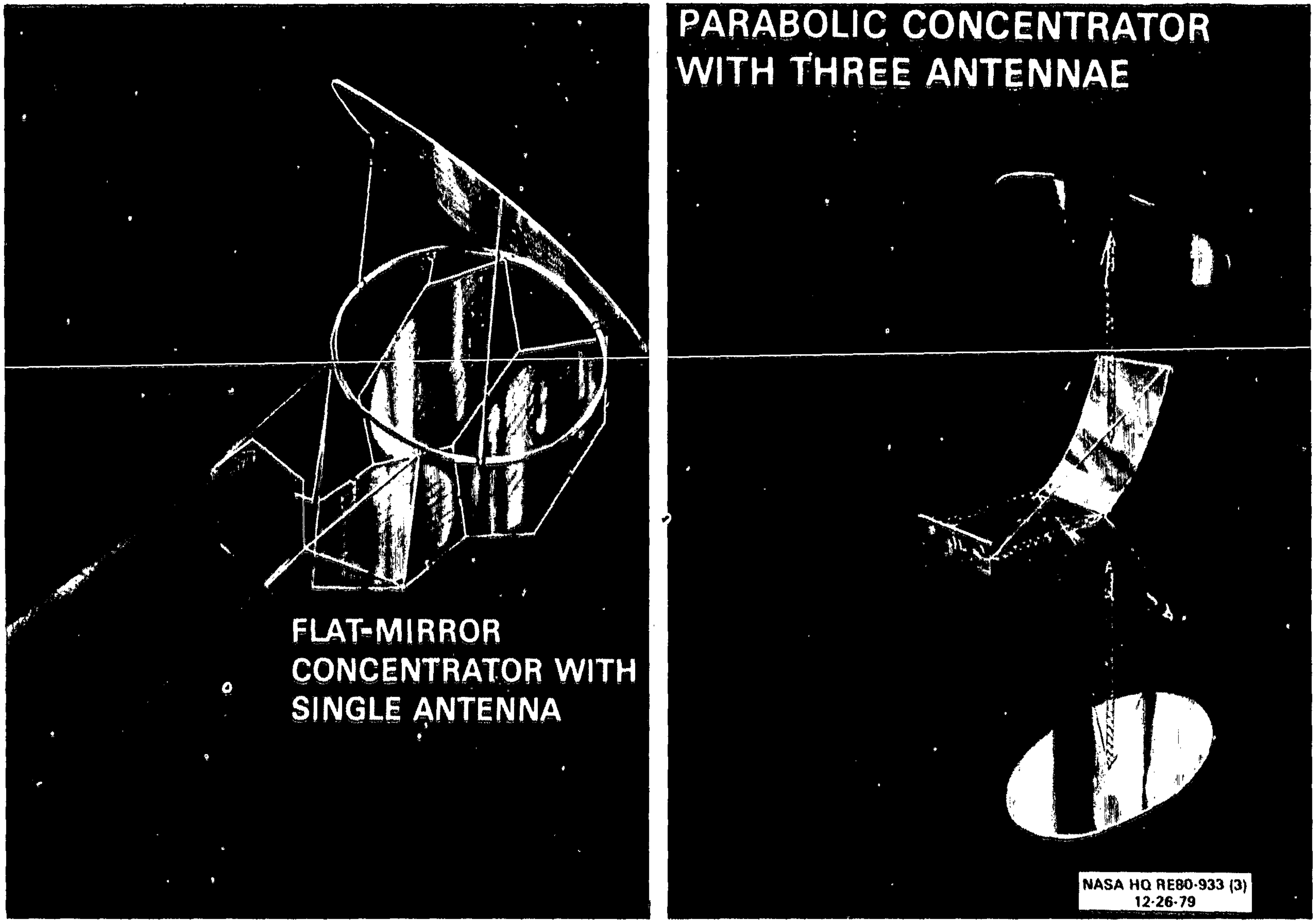


Figure 5 Alternate SPS Concepts - Early 1980

II. EARTH SURFACE TO LOW EARTH ORBIT (ESLEO) TRANSPORT

A. Vehicle Systems Concepts

1. Shuttle Transportation Systems (STS)

a. Current baseline

It was agreed that the baseline current (1980) Shuttle transportation system, as shown in Figure 6, will be capable of supporting space-data-acquisition projects necessary for SPS feasibility evaluation during the middle years of the 1980s. These early experiments would undertake to verify analyses and ground-based experiments essential to early demonstration of SPS feasibility. NASA has already established the Orbiter Experiments (OEX) program to perform this function. If it proves desirable to conduct a subscale SPS demonstration program during the early 1990s, substantial uprating of the Space Shuttle delivery capability is feasible. The approach taken in uprating will be impacted by early operational experience and actual recurring costs per flight.

b. Growth using liquid propellant boosters

It is understood that near-term Shuttle performance growth capability will be provided by the Titan LBM. The LBM was originally conceived for use at the Western Test Range (WTR) to give the Shuttle a performance increase from a predicted 1984 capability of 10,885 kg (24,000 lbm) to over 16,325 kg (36,000 lbm) into a near-polar orbit (98-deg inclination). The LBM, to be available in mid-1985, can also be used at the Eastern Test Range (ETR) to raise the Shuttle payload from a predicted 1984 capability of 29,480 kg (65,000 lbm) to a 36,280 kg (80,000 lbm) equivalent payload on due-east launch. This increased payload capability will undoubtedly have utility in any SPS on-orbit system demonstration program, and its availability should be recognized and incorporated into SPS planning.

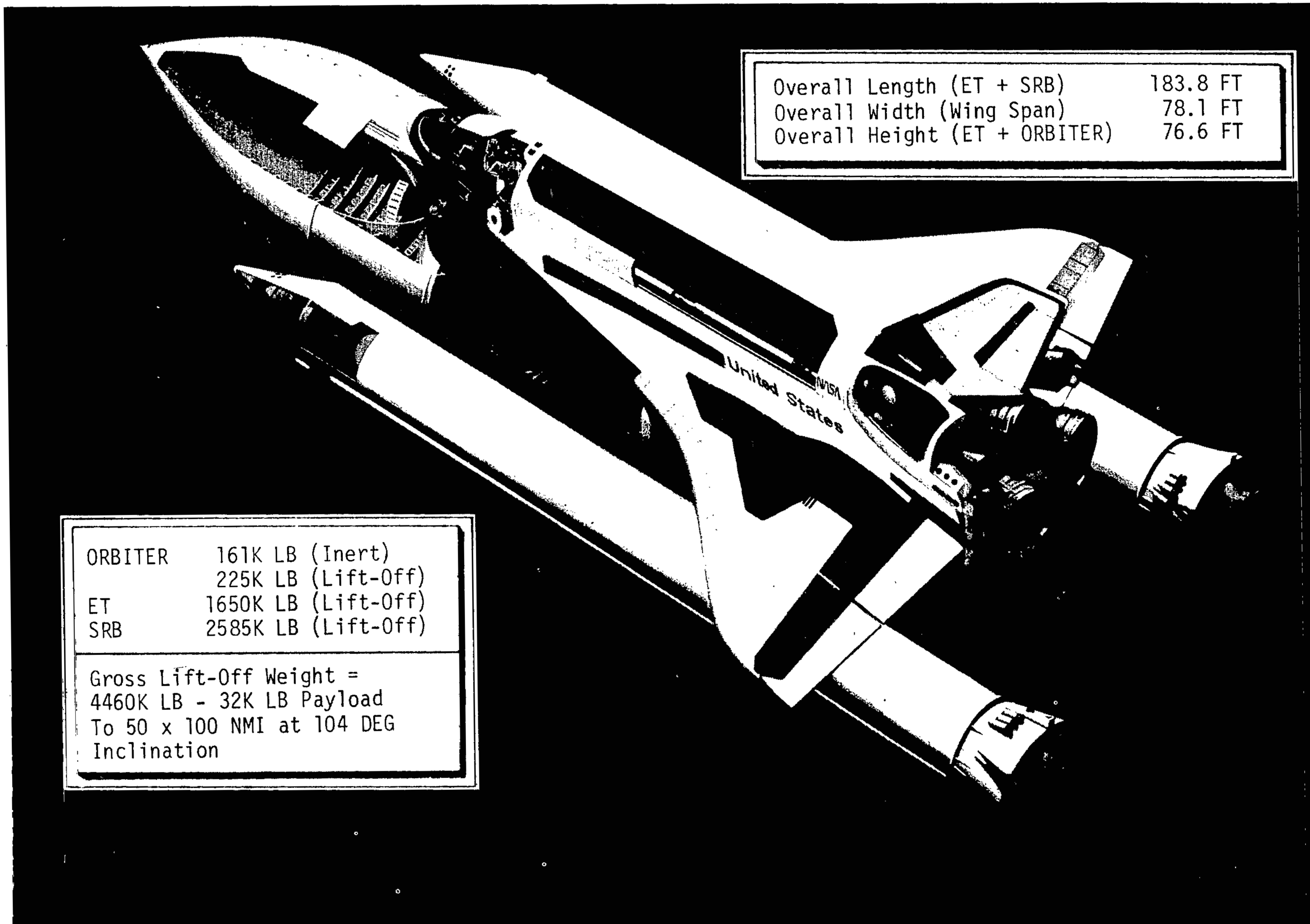


Figure 6 Shuttle Transportation System (STS) - Current (1980) Configuration

The LBM airborne configuration consists of the Titan 3 first-stage engine, a new thrust structure and modified fuel and oxidizer tanks. The LBM is a self-contained propulsion system which mounts on the aft of the external tank. It has a 200-sec burn time, starting 5 sec after Shuttle liftoff.

The LBM is currently in the program-definition phase with full-scale development anticipated to start in October, 1982, to support a June, 1985, first flight at the WTR. The development program contains testing of the structural and propulsion systems, as well as an LBM flight-duration demonstration. Further growth configurations of the LBM with additional engines and tankage are also being evaluated.

According to Rockwell studies, the basic Orbiter vehicle can be adapted to transport about 75 personnel to low-Earth orbit within the cargo bay. This capabiltiy should be adequate to support probable requirements of the SPS program well into the 1990s. This concept is illustrated in Figure 7.

Studies have shown the feasibility of increasing the Orbiter payload for SPS-scale demonstrations to nearly 54,420 kg (120,000 lbm) by replacing the present solid rocket boosters (SRB) with a pair of reusable liquid propellant rocket boosters (LRB) that would be recovered from the water and refurbished, in an operation similar to that planned for the SRBs. The largest uncertainties in this conceptual approach involve the operations for undamaged water landing, retrieval and turnaround, and the costs associated with achieving the required confidence level for these operations. The proposed LRB configuration is shown in Figure 8.

2. Heavy Lift Launch Vehicles (HLLV)

a. Shuttle derivatives

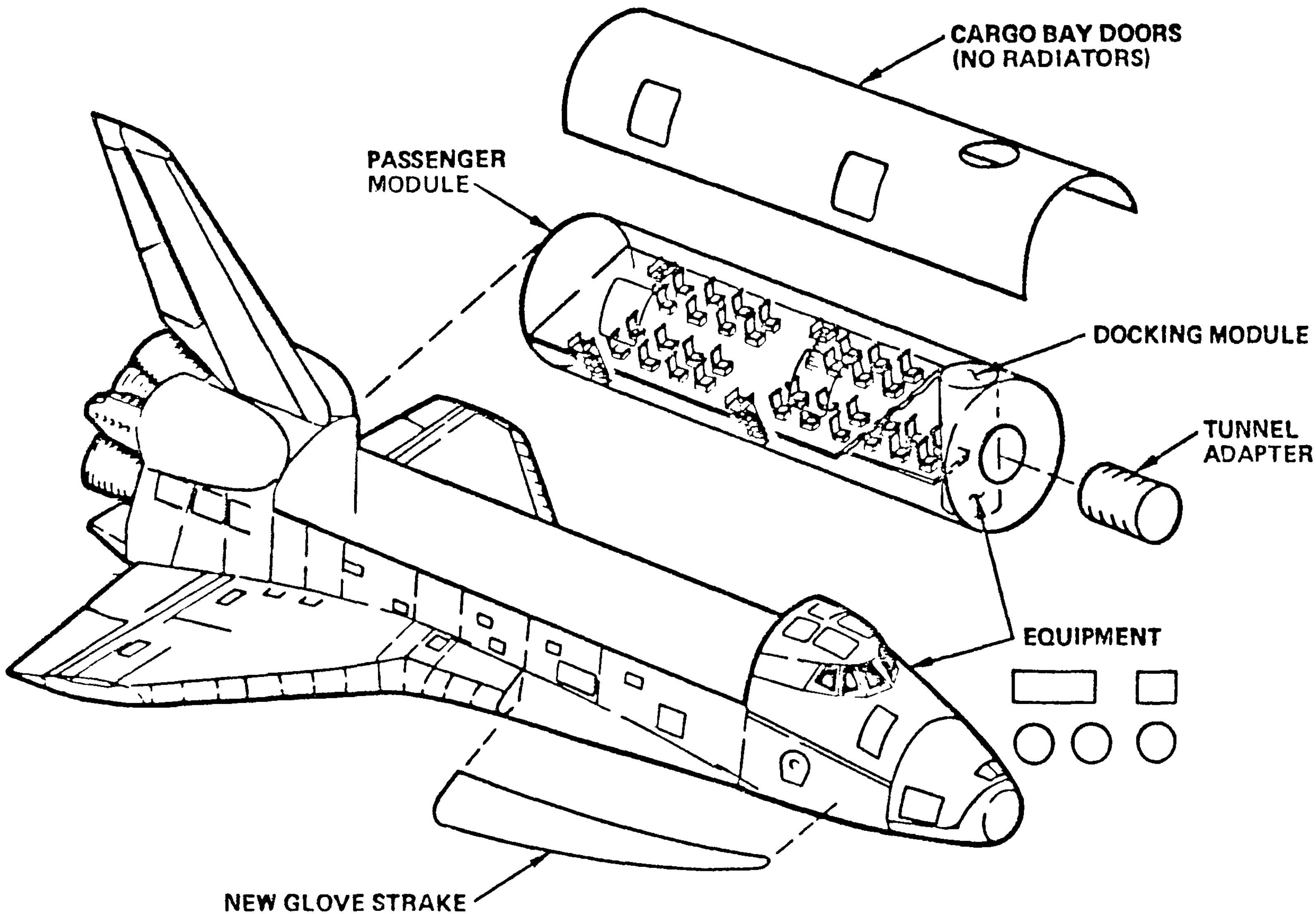


Figure 7 Shuttle Orbiter Personnel Payload Configuration

The present STS hardware can be adapted to deliver heavy-lift class payloads. Several studies have indicated the feasibility of using the LRB, the external tank and a new recoverable propulsion module containing the Space Shuttle main engines (SSME) and appropriate elements of the STS guidance, navigation, flight control, data systems auxiliary power and reaction control systems. The configuration, illustrated in Figure 9, could deliver more than 68,000 kg (180,000 lbm) of payload to low-Earth orbit. This configuration provides an effective contender for intermediate SPS demonstration program support by utilizing an expendable shroud that would permit payload dimensions to exceed those now imposed by the Shuttle cargo bay constraints.

The Shuttle derivative concepts assume present specifications plus modest technology growth, such as the following:

- Space Shuttle main engine being fully in accord with current specifications
- A new liquid-propellant booster engine using current technology
- Shuttle-type thermal protection system (TPS)
- Automated diagnostics to facilitate maintenance operations
- Aluminum and titanium airframes with modest use of composites
- Cryogenic orbital maneuvering system (OMS)
- Off-line processing of palletized payloads to minimize loading time

These design assumptions lead to an expected vehicle life of 300 flights (500-flight design life with 0.1 per cent attrition per flight). Engine-life limitations would probably result in a substantial maintenance load and

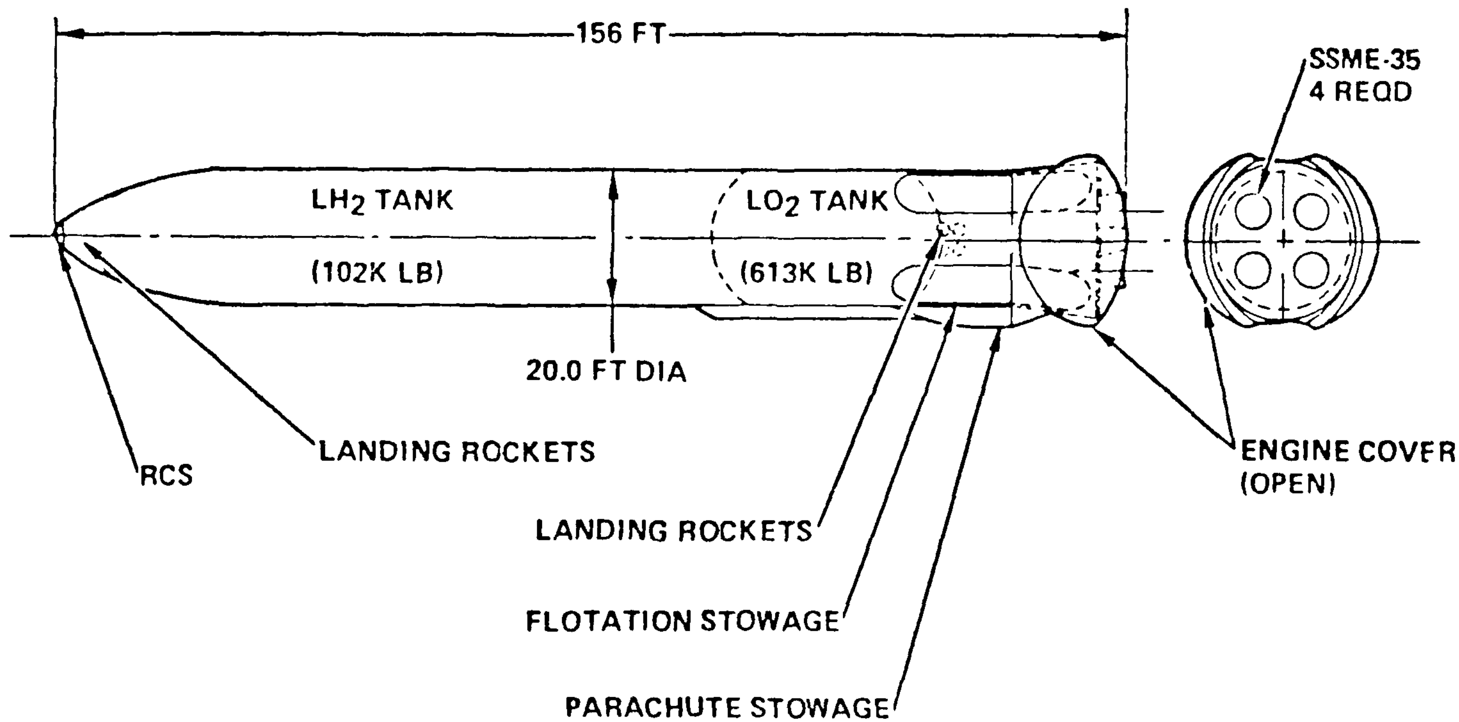


Figure 8 Liquid Propellant Rocket Recoverable Booster (LRB)

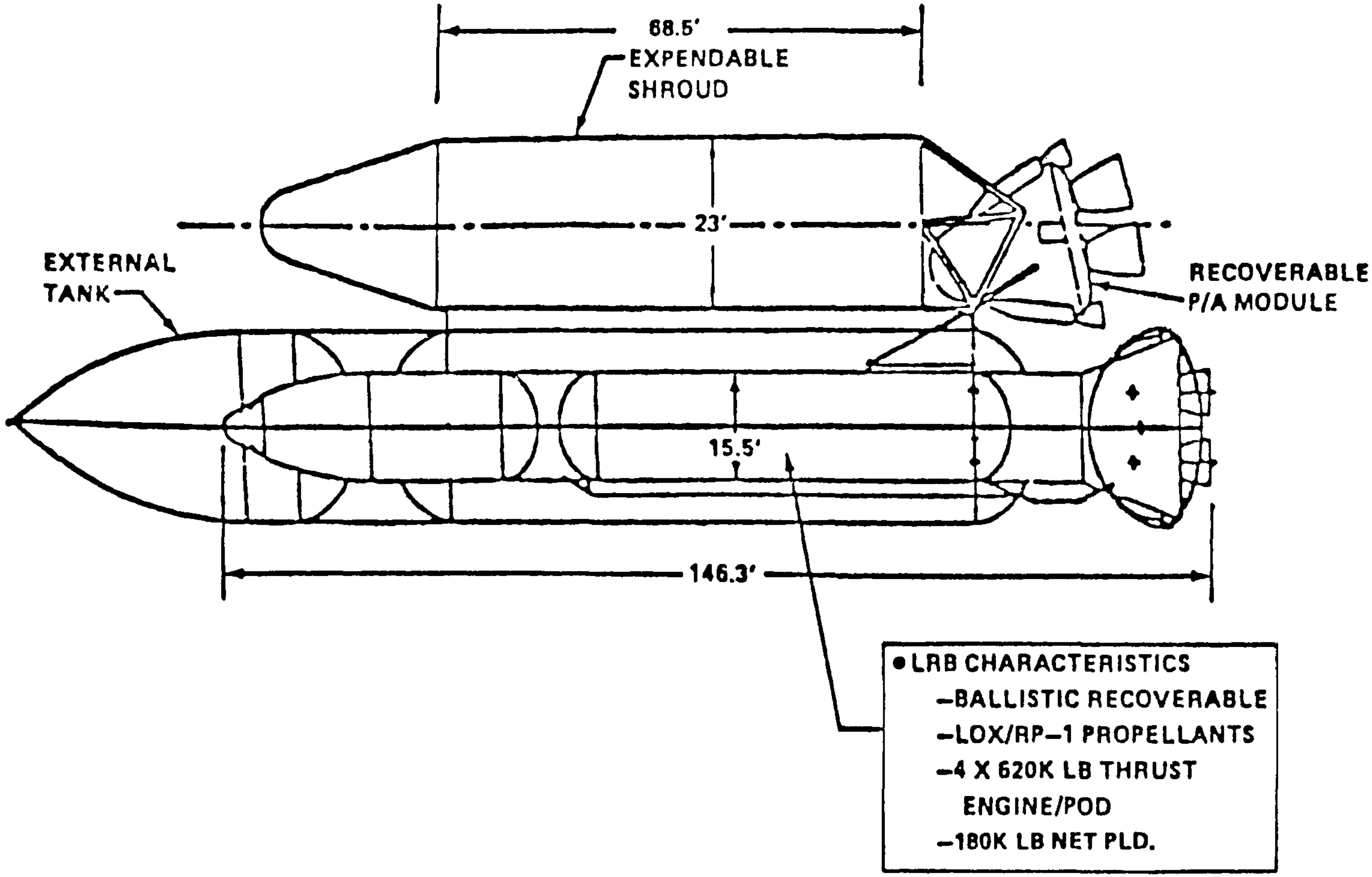


Figure 9 Heavy Lift Launch Vehicle - Shuttle Derivative - MSFC

TPS refurbishment is an unknown quantity. Airframe spares of 0.18 per cent flight have been estimated with somewhat higher engine spares in accordance with the current SSME specification.

b. New vehicles

It was the consensus of the workshop that more ambitious goals in performance, reusability, and operations technology must be advanced, utilizing new vehicles to develop a potential for substantial reductions in projected transportation cost. This is a critical area in terms of overall SPS economics. To achieve significant reductions in costs, a representative set of goals must include the following items which require, in effect, new vehicles:

- Vehicle design life exceeding 1,000 flights with reduced attrition
- Improvements in engine life and maintainability beyond the SSME specification by major factors
- A TPS technology that would require only routine visual inspection and infrequent maintenance, and would offer very high confidence that catastrophic failure would not occur
- Vehicle and airframe subsystems requiring infrequent maintenance
- A means of leak detection (for propellants and hazardous fluids) that would obviate extensive pressure checking, purging, etc.
- More aggressive use of composites and other mass-reduction means
- Vehicle sizing and capabilities appropriate to alternative uses, so that the SPS program will not have to bear the entire development cost

- Advanced operational capabilities similar to airline freight operations

Given the goal of an HLLV system capable of placing about 100,000 kg (220,000 lbm) into LEO, there is little reason to question our present ability with current technology, although new large vehicles, such as the flyback booster shown in Figure 10, would be required. With more massive payloads and a greatly reduced cost of payload to LEO, it will be necessary to utilize advanced techology and very large, completely reusable HLLVs, such as those shown in Figures 11 and 12. Although the conceptual designs need further study, it is essential that they have minimum costs for production, operation and maintenance.

Assuming that the cost to operate, primarily fuel cost, is about 15 per cent of the total over the vehicle lifetime, the costs of hardware (manufacturing and spares) and labor (maintenance and operating personnel) can be taken to be divided at 40-45 per cent each.

The key drivers of the technology, then, may be identified initially as those which reduce labor and hardware costs. Eventually, as these costs are minimized, the cost of fuel will become more significant, so attention must also be given to those technologies which will reduce it (i.e., improve performance).

The SPS studies performed by governmental and industrial teams have repeated to a considerable degree the findings of earlier pre-Shuttle studies performed between 1962 and 1969. The common denominator is to achieve "airline operation," high reliability, long time between failures, little delay between flights (i.e., maintenance relegated to scheduled periods, turnaround limited to refueling and mating with rapid payload installation, and launch).

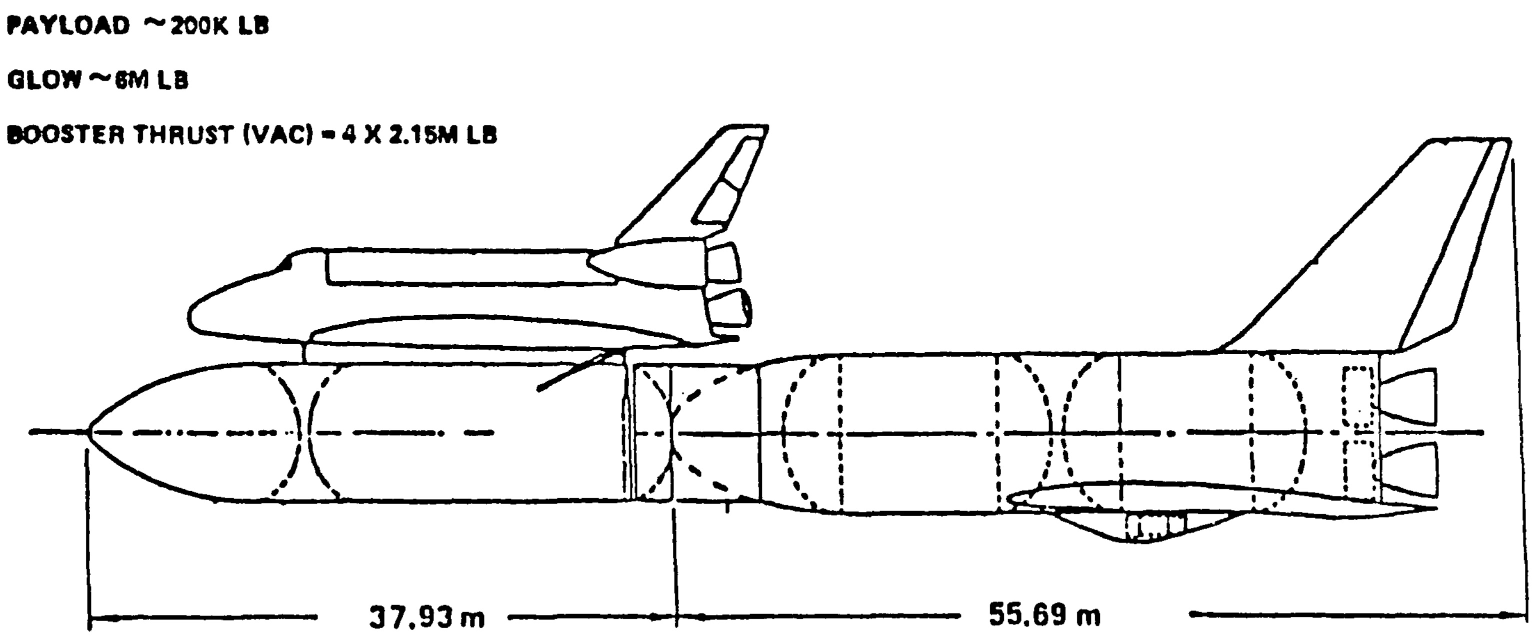


Figure 10 Personnel/High Priority Cargo Vehicle with Flyback Booster

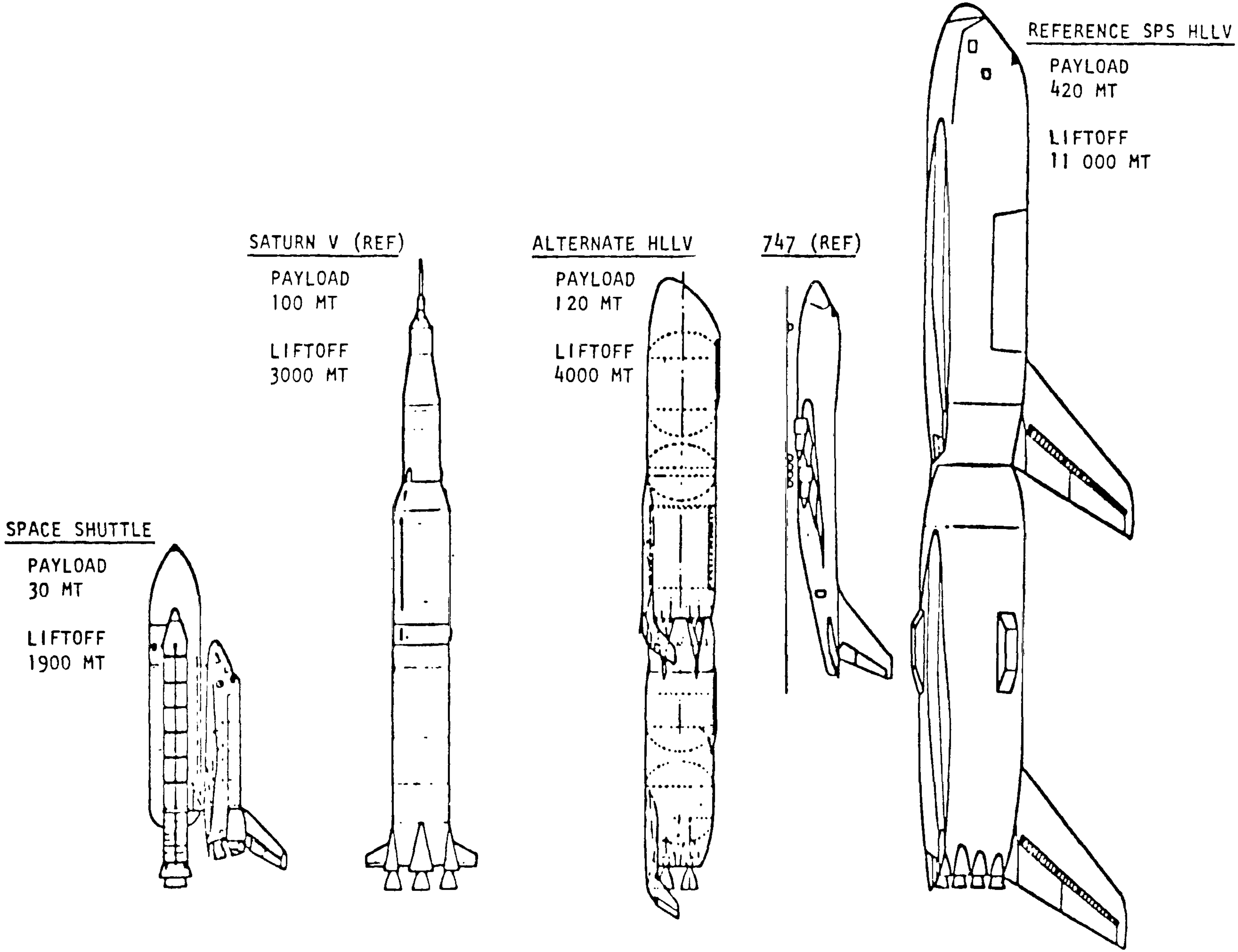
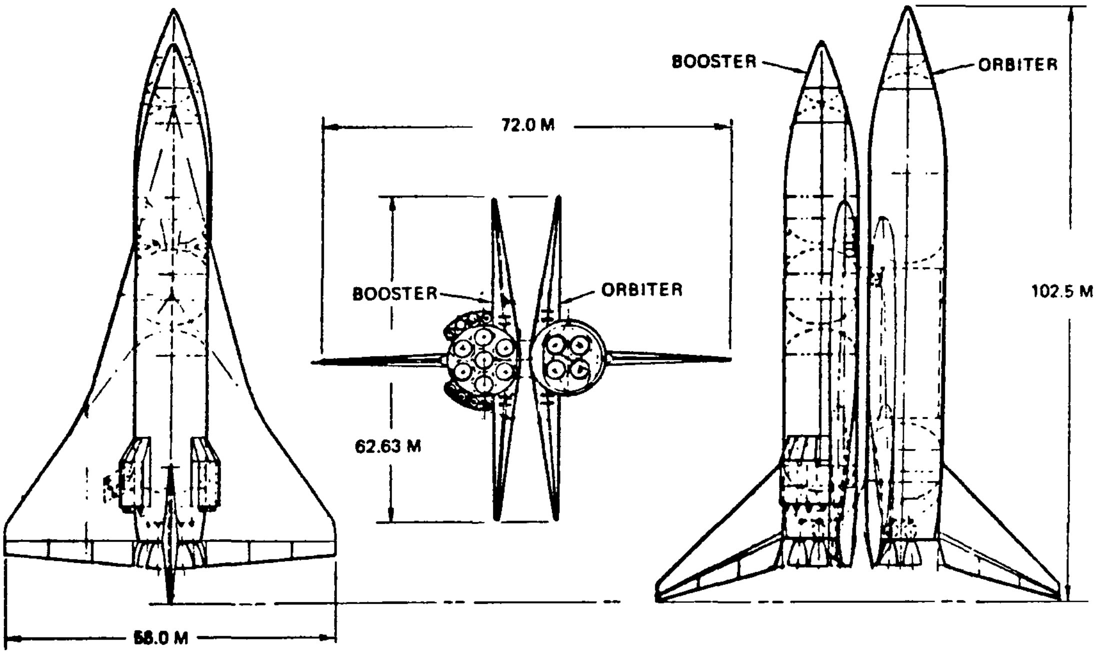


Figure 11 Heavy Lift Launch Vehicle (HLLV) Concepts - Boeing



HLLV Mass Properties $(x10^{-6})$

	kg	lb
GLOW	7.14	15.73
BLOW	4.92	10.84
W_{P1}	4.49	9.89
ULOW	2.22	4.89
W_{P2}	1.66	3.65
PAYLOAD	0.23	0.51

Figure 12 Reference Heavy Lift Launch Vehicle (HLLV) Configuration - Rockwell

c. Critical vehicle technologies

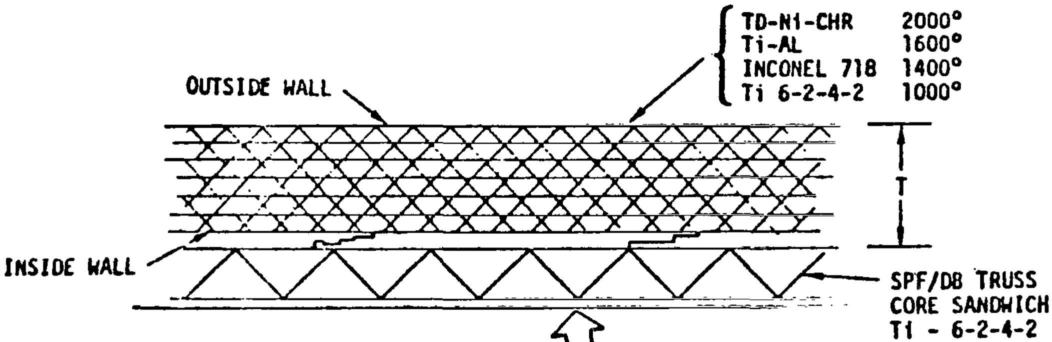
The critical vehicle technologies, among others, must be emphasized early and aggressively if SPS goals, identified above, are to be met.

(1.) Reusable thermostructure

In the broad sense, thermostructure refers to both the TPS and the primary structure. The TPS, in particular, must require no inspection or refurbishment between flights; to do so would induce prohibitive labor costs considering the extended surface involved with these very large systems. This strongly suggests the use of metallic material for both the TPS and primary structure as shown in Figure 13. The TPS thickness and mass are dependent upon the allowable backface temperatures of the primary structure. High thermal-gradient joints are characteristics of the interfaces between hot external surfaces and cooler internal structures.

These requirements vary, with boosters or orbiters, since their thermal environments are different. Boosters stage at lower velocities and therefore have less energy to dissipate. The maximum temperatures are typically not greater than 1,090°K (1,500°F), as shown in Figure 14. The local temperatures are generally well within the realm of conventional heat-sink structure with perhaps some localized TPS. The design emphasis is on minimizing structural mass while not increasing manufacturing or maintenance costs.

Orbiters encounter much higher thermal environments with maximum temperatures of approximately 1,750°K (2,700°F), as shown in Figure 15. These temperatures exceed the capability of currently available materials which do not require special surface coatings (to retard oxidation) and which can experience repeated thermal cycles without degradation. Much work is needed to bring the candidate materials listed in Table 1 to full technology readiness.



T FOR MULTIWALL TPS

OUTSIDE TEMP °F \ INSIDE TEMP °F	1000	600	350
2000	1.20	1.60	2.20
1600	1.00	1.20	1.70
1400	.50	1.00	1.30
1000	-	.50	1.00

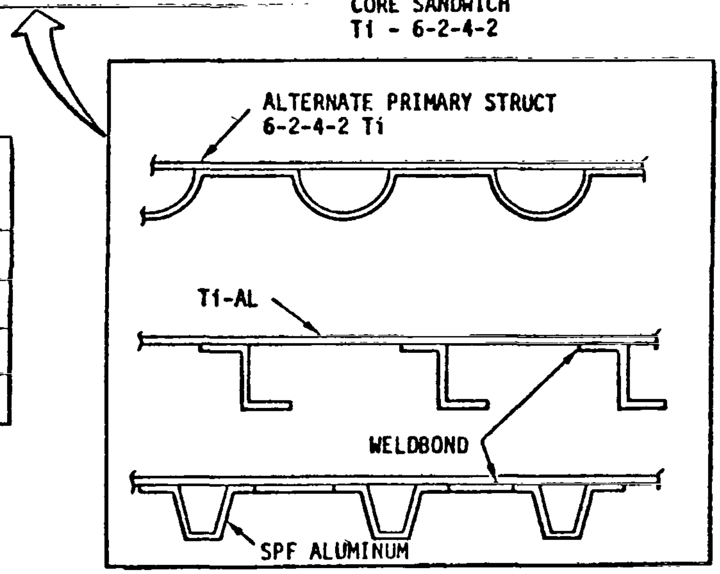


Figure 13 Multiwall Thermal Protection System (TPS) Configuration

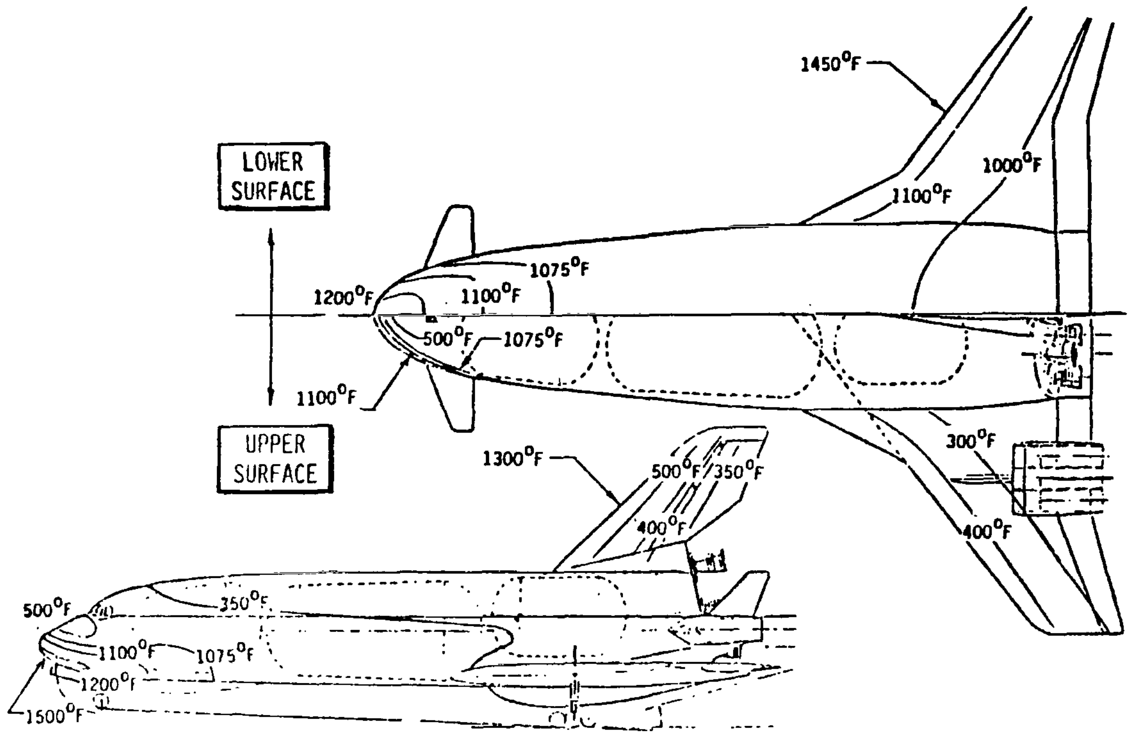


Figure 14 SPS Booster Maximum Radiation Equilibrium Isotherms

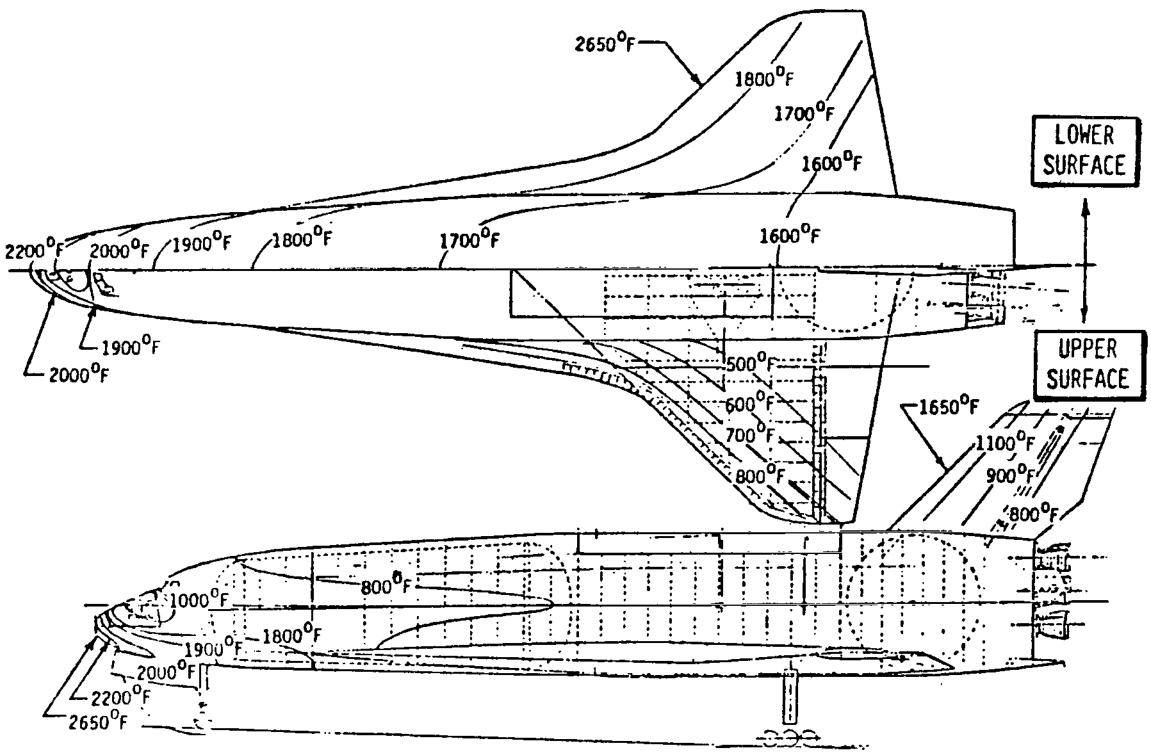


Figure 15 SPS Orbiter Maximum Radiation Equilibrium Isotherms

REF: AFFDL-TR-69-94

Table 1
Merit Indexes for Candidate Materials

TEMP RANGE	VEHICLE STRUCTURAL APPLICATION	CANDIDATE MATERIAL (1)	PHYSICAL PROPERTIES (3) P S G E	TENSILE PROPERTIES (3) P S G E	CREEP RESISTANCE (2)(3) P S G E	FORMABILITY (3) P S G E	WELDABILITY (3) P S G E	OXIDATION RESISTANCE (3) P S G E	LEADING CANDIDATE MATERIALS	REMARKS
-421° - 1000°F	WING, TANKS, BODY STRUCTURE	2219 T81							*2219 T81	GOOD FABRICABILITY AND STRENGTH
		6061 T6								GOOD WELDABILITY, MODERATE STRENGTH
		7075 T6							*7075 T6	HIGH STRENGTH, WELDING NOT PRACTICAL
		BE 38A1					NA			NOT WELDABLE
		AMS 7902					NA			
		5A1 1MO 1V							8A1-1MO-1V	GOOD STRENGTH AND FABRICABILITY
		6A1-4V								GOOD STRENGTH AND FABRICABILITY
		5A1 2.5Sn								LOW STRENGTH, GOOD FABRICABILITY
1000° - 2000°F	UPPER SURFACE PRIMARY AND SECONDARY STRUCTURE HEAT SHIELD	COBALT BASE ALLOY								
		HAYNES 25							*HAYNES 25	ANNEALED MATERIAL WITH MODERATE TENSILE PROPERTIES, GOOD OXIDATION RESISTANCE TO 1600°F
		NICKEL BASE ALLOY								
		INCO 625 (1400°F)							*INCO 625	MATRIX STRENGTHENED ALLOY WITH MODERATE TENSILE PROPERTIES, METALLURGICALLY UNSTABLE ABOVE 1400°F
		INCO 718 (1400°F)								AGE HARDENABLE ALLOY WITH HIGH TENSILE PROPERTIES, MODERATE CREEP RESISTANCE
		HASTALLOY X								SUPERIOR OXIDATION RESISTANCE TO 2000°F
		RENE 41 (ANN) (1600°F							*RENE 41	ANG WELDABILITY, SUBJECT TO EMBRITTLEMENT AND ALLOY DEPLETION ABOVE 1600°F
		TD NICKEL								NOT COMPETITIVE WITH MECHANICAL PROPERTIES OF OTHER SUPERALLOYS
		TD NiCr							*TD NiCr	CANDIDATE UNCOATED MATERIAL FOR APPLICATION TO 2200°F
2000° - 2500°F	LOWER SURFACE LEADING EDGE AND HEAT SHIELD	CHROME 30					NOT PRACTICAL			EXTREMELY BRITTLE MATERIAL AT ROOM TEMPERATURE
		TD NICKEL								NOT TO BE USED AS PRIMARY STRUCTURAL APPLICATION, NOT COMPETITIVE WITH Cb ALLOYS
		TD NiC (2200°F)							*TD NiC	CANDIDATE UNCOATED MATERIAL FOR APPLICATION TO 2200°F
		COLUMBIUM (4) ALLOY								
		D 43								POOR WELDABILITY
		B 66								SUPERIOR DENSITY COMPENSATED STRENGTH VALUES, POOR FORMABILITY AND WELDABILITY PROPERTIES
		FS 85								HIGH DENSITY
		C 129Y								EXCEPT FOR LOWER CREEP RESISTANCE, SIMILAR TO Cb 752
		Cb 752							*Cb 752	MODERATELY HIGH MECHANICAL PROPERTIES
2500° - 3500°F	LEADING EDGE	TANTALUM (4) ALLOY								
		90Ta 10W							*90Ta 10W	MODERATE MECHANICAL PROPERTIES, VERY GOOD WITH RESPECT TO FABRICABILITY
		T 222								HIGH MECHANICAL PROPERTIES, MILL ROLLING PROBLEMS
		TZM					NA			
3500° - 4000°F	NOSE CAP	TUNGSTEN THORIA			NA		NA		*TUNGSTEN-THORIA	TESTED IN REENTRY PROFILE
		TUNGSTEN					(4)			LIMITED BY OXIDATION PROTECTIVE SYSTEM
		ZIRCONIA ROD								TESTED IN REENTRY PROFILE

NOTES (1) MAXIMUM STRUCTURAL TEMPERATURE LIMIT
(2) BASED ON 0.5 PERCENT CREEP AT SPECIFIED TEMPERATURE
(3) RATING LEGEND IS AS FOLLOWS.
E – EXCELLENT
G – GOOD
S – SATISFACTORY
P – POOR
(4) WITH OXIDATION PROTECTIVE COATING
NA – NOT APPLICABLE

In addition, many of these materials have high densities, are very expensive, and are available only from foreign sources. Little or no effort has been expended in metallurgical development since the late 1960s. Therefore, a major development program is required to provide advanced thermostructures which meet the needs of the SPS and other advanced space transportation systems. Primary emphasis should be placed as follows:

- Materials - metallurgical development of new materials which are readily manufacturable, maintainable, reusable, highly damage resistant, and made from domestically available raw materials
- TPS - extensive development and evaluation of metallic thermal protection systems with or without nonmetallic insulative material. Active cooling or heat-pump systems are back-up candidates for local high-heating areas
- Primary structure - principal structural components which may be metallic, composite or metal matrix, and which may also be hot or cold. High-strength structural gradient joints must also be developed

(2.) Cryogenic tank insulation

The cryogenic tanks of both the boosters and orbiters must be designed so that they require little or no inspection other than normal maintenance cycles. Similar requirements are placed on the tank insulation. Whether the tanks are integral or nonintegral does not relieve this requirement significantly. Insulation systems must be developed which satisfy these requirements and prohibit cyropumping and eliminate external ice buildup. The latter is especially important for horizontal takeoff vehicles.

(3.) Other critical technologies

Efforts need to be made to identify all critical areas of vehicle technology and to be certain that they receive adequate attention to remove substantial problem areas. Propulsion, in all ESLEO applications, is discussed below.

3. Other Vehicle Concepts Including the Advanced Single-Stage-to-Orbit (SSTO) Vehicle

a. Baseline personnel launch vehicle

The requirement for the personnel launch vehicle (PLV) is to transport SPS construction and maintenance personnel. Roughly 600 people are required for the steady-state construction period while approximately 30 people per satellite are needed for maintenance. Assuming a three-month duty tour in space, annual man-trips start at 2,400 and approach 10,000 when 60 satellites are operational.

The payload and launch-rate requirements in the early program phases are compatible with a Space Shuttle system which incorporates modest payload uprating -- possibly the augmented STS or an uprated liquid rocket booster.

The total cost of personnel transportation within the overall SPS scenario is "relatively insignificant"--representing approximately 10 per cent of the total SPS transportation cost, or about 2.5 per cent of the total SPS cost.

The Shuttle-derivative approach provides a required capability at low investment cost and risk. The high operational cost associated with high HLLV traffic flow raises the possibility of substantial cost savings through personnel transportation on the HLLV. This approach, suggested by both study contractors, eliminates the requirement for all but occasional use of this vehicle but puts an additional man-rating requirement on the HLLV.

The relative total cost of the PLV compared to the HLLV is small, and thus the criticality of this system from a total cost standpoint is low. A modest uprating of the Shuttle can meet the initial requirements at low investment cost and risk. However, the PLV operational trips required and the tradeoffs need to be evaluated against the development of a new vehicle with lower operational costs. The requirments and justification for such a vehicle would come not only from SPS but also from the broad range of other space activities--both civilian and military. Within that broad range of transportation requirements, it is quite likely that the development of a new PLV will be attractive.

b. Advanced PLV and HLLV concepts

The PLV and HLLV baseline concepts presented by the study contractors have emphasized low risks and low technology. Relatively little treatment has been accorded to options associated with alternate system concepts and/or the possible benefits to be derived from the incorporation of technology improvements. In trying to prove feasibility, the obvious motivation is to show a capability while using low-risk technology. However, the best system options will strike a balance between low risk and benefits/improvements to be derived from alternate vehicle concepts and/or technology advancements.

A new PLV/priority cargo vehicle must, first of all, be fully reusable and meet a payload requirement in the range of from 20,000 to 50,000 kg (40,000 to 100,000 lbm). Beyond that there are concepts with a broad matrix of operational modes, staging options and propulsion system with potential application for a PLV. Key issues appear to be vertical vs horizontal takeoff, one vs two stages, and rocket vs air-breathing propulsion. Air-breathing propulsion is generally associated with horizontal takeoff

Six PLV concepts are discussed below:

- Concept 1 - Two stages, vertical takeoff, and horizontal landing (VTOHL). All rocket propulsion is the most conventional approach offering potentially low risk

- Concept 2 - Single stage, VTOHL, all-rocket propulsion shares basic technology elements with Concept 1; however, it needs a high level of performance in order to become attractive. Potential benefits accrue in development, vehicle purchase, and operations by having a single vehicle
- Concept 3 - Air-breathing, first-stage accelerator offers versatility of horizontal takeoff (HTO) operations. Large vehicle size, and propulsion system mass and cost are key issues
- Concept 4 - A sled-assisted, rocket-powered HTO concept which shares many technology issues with Concept 2
- Concept 5 - An air-launch assist by in-flight fueling which has many similarities to Concept 4
- Concept 6 - A single-stage vehicle utilizing multicycle, air-breathing propulsion system offers great versatility; however, it also presents a very substantial challenge to the mass and performance of the propulsion system. A Rockwell concept of such a vehicle, called the "Star Raker" is presented in Figure 16. Although this vehicle employs very advanced technology, it represents the direct thrust of future aerospace development and may incorporate a substantial capability for a variety of missions after the turn of the century. However, it is too soon to determine how such a vehicle would fit into the SPS or other uses. Never the less, it is necessary that the essential technologies be pursued actively

It is essential that a systematic evaluation of these various advanced concepts be included in order to identify the most desirable concepts and their associated technology requirements. A balanced series of system studies and technology is required to guide the development of the concept.

The proposed ground-based exploratory development (GBED) program contains a long list of detailed technology programs which support a rather specific set of reference vehicles. There does not appear to be enough depth in the systems-level studies to justify selection of these reference vehicles to the extent that critical technology requirements should be predicted for them. The GBED program should initiate adequately funded, feasibility studies of competitive systems: and parallel supporting-technology programs should be tailored appropriately. The system studies should initially consider multiple concepts and only later narrow to preferred concepts.

There are many areas of common technology requirements between the advanced PLV concepts and the baseline, two-stage, VTOHL rocket-powered concepts. Concepts 1 and 2 above do not create any basically new technology issues. However, the hybrid and single-stage concepts tend to require a higher level of performance than the VTOHL options. Although single-stage-to-orbit (SSTO) concepts are not baselined, the GBED program does include specific SSTO propulsion items.

The horizontal takeoff concepts as a group generate a number of technological implications not common to the baseline HLLV. These are most critical in the area of air-breathing propulsion and range from adaptations of existing turbojets to advanced-technology, multicycle engines operable to hypersonic speeds. Air-breathing propulsion applied to accelerator vehicles

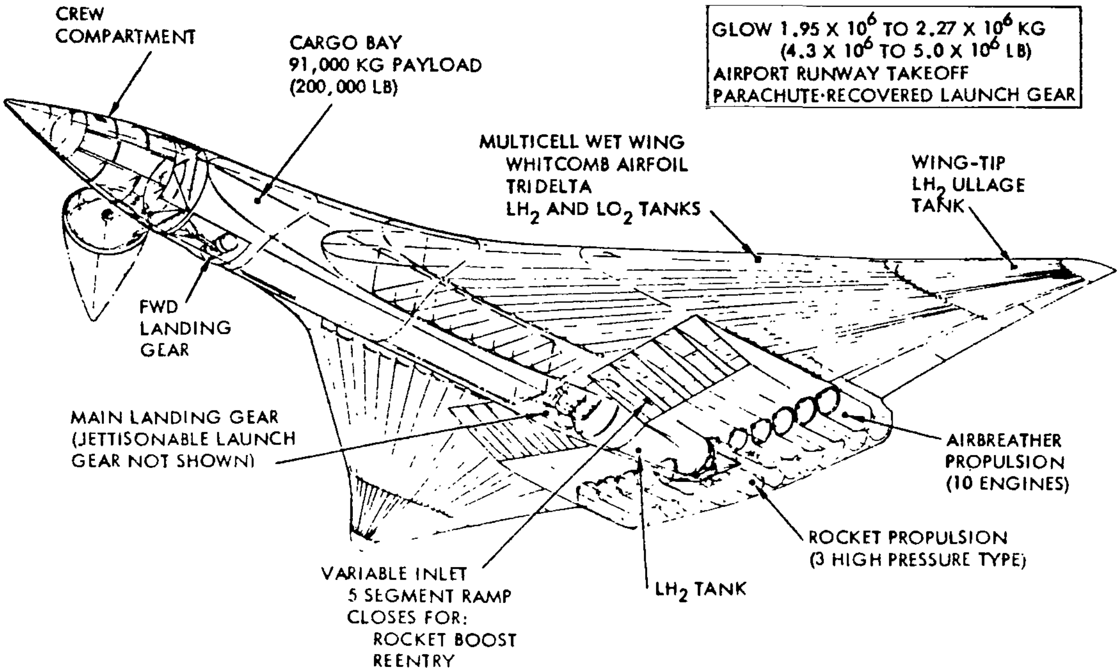


Figure 16 Heavy Lift SSTO Launch Vehicle with Winged Recovery - Rockwell "Star Raker"

offers the benefits of high specific impulse; however, the penalties of propulsion-system weight create a special technology effort on reducing engine weight. The horizontal takeoff mode presents additional challenges in aerodynamic configuration and structural loading not required in the vertical-takeoff mode. The SSTO vehicles incorporate an aircraft-development approach which includes taxi, takeoff and landing, subsonic flight, supersonic flight, low- and high-altitude tests, etc.

The technology program of the baseline HLLV will create benefits to potential PLV system concepts. Additional activity related to PLV should focus on broad system/technology option assessment prior to committing substantial resources to specific developments.

B. Propulsion Technology Options

The propulsion systems used in the ESLEO SPS transportation are discussed in this section. The reference vehicles are the PLV and the HLLV. These vehicles use liquid oxygen/liquid hydrogen propellants for high-altitude operation and either oxygen/RP-1 or oxygen/H_2 propellants during the low-altitude operation. Engine-thrust levels in these two vehicles are not identical; and therefore there are potentially four different rocket engines while only one engine, the SSME, is currently under development.

One of the advancements in technology that should be pursued for the three new engines is to improve engine service life and reduce turn-around maintenance. It is also important to understand the sensitivity of engine performance and life and their impacts on transportation cost. Both of these affect the operational cost of SPS in terms of labor to perform maintenance and spares to overhaul or replace engines. Since labor and hardware are large percentage shares of the total SPS cost, research and technology funding in propulsion should be concentrated on them.

The next phase of the SPS program should address the features of the rocket engines of the reference vehicle that impact the operational costs. A generic approach to increasing life would apply across the board to all three new engines. However, there are specific areas that must be considered for the liquid oxygen/RP-1 engine that are not appropriate to liquid oxygen/liquid hydrogen. Carbon formation within the turbomachinery and in cooling circuits could be significant factors degrading performance and life of engines using RP-1 fuel. Techniques to clean the engine between flights without significant penalties to cost and time are necessary. Past programs with RP-1-fueled engines have relied upon purging and flushing the engines on the launch pad prior to launch. Technological advances in this area are expected to have great influences in reducing operational costs and should be included in the following program phase.

Research and technology associated with materials development and advancing fabrication techniques to increase engine life, reduce maintenance, lower weight, and reduce cost are not addressed in the present propulsion program. Initiation of new development programs needs an advanced technological base in these areas. There are numerous potential advances that could be applied in a development program if their feasibility is demonstrated. The reference SPS system does not depend upon advances of this type, but there should be significant returns if the subsequent program includes activity to permit assessment of these advances. There have been essentially no funds spent by NASA for rocket-engine research in this area for nearly a decade.

Ballistic recovery of the PLV liquid-propellant boosters assumes complete protection of the propulsion system from the sea. There is no research and technology (R&T) in the next program to assess the capability of the engine to survive a sea-water environment without increased maintenance. It could be a key factor in the decision between ballistic and fly-back boosters for the reference PLV. Therefore, it is recommended that the next program phase include this issue.

Alternate propulsion systems have emerged in SPS studies. Dual-fuel engines for SSTO vehicles, as in Figure 17, multiple-cycle, air-breathing engines for SSTO and HLLV, as in Figure 18, and LOX/CH_4, high-thrust engines are alternatives that are not yet developed. These propulsion systems may not be required for the reference-system performance, but it is strongly recommended that sufficient funds be invested in R&T of these systems becaose of their potential for ultimately reducing costs. By omitting alternate propulsion concepts, options are closed for future decisions on the best propulsion improvements on the reference SPS system. It is recommended that the next program phase be structured to give equal priority to all promising propulsion systems.

The major technology issue for the liquid-propellant rocket engines that may be utilized for the SPS transportation system is the means of achieving low-cost operation of a highly reusable, complex system. The implications of this issue demand long life for the engine and its components, ease of inspection and maintenance, basic reliability of components, and high confidence in the ability to avoid random catastrophic failures. An appraisal of existing, successful, and reusable propulsion systems provides a good model to adopt for minimizing the operating costs of the SPS transportation life cycle.

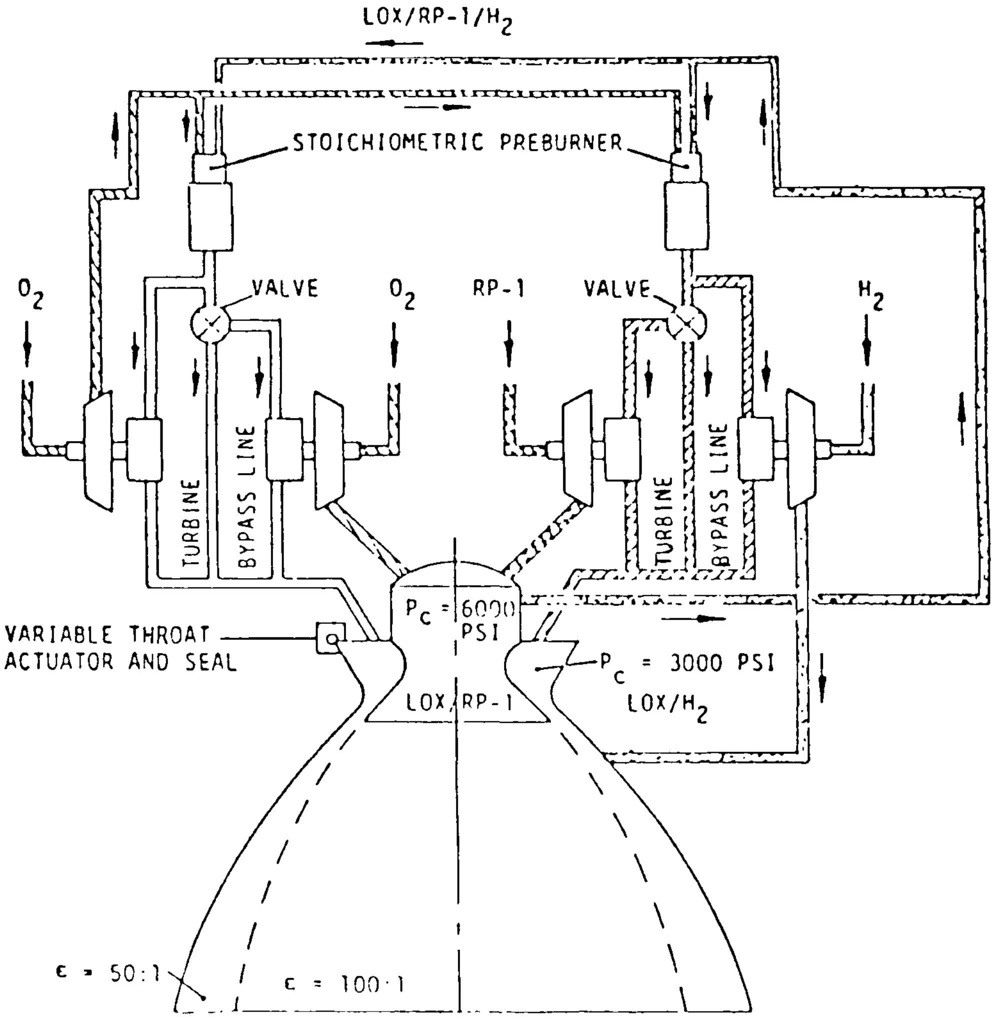


Figure 17 Dual Expander Rocket Engine Concept - LOX/RP-1/LH_2

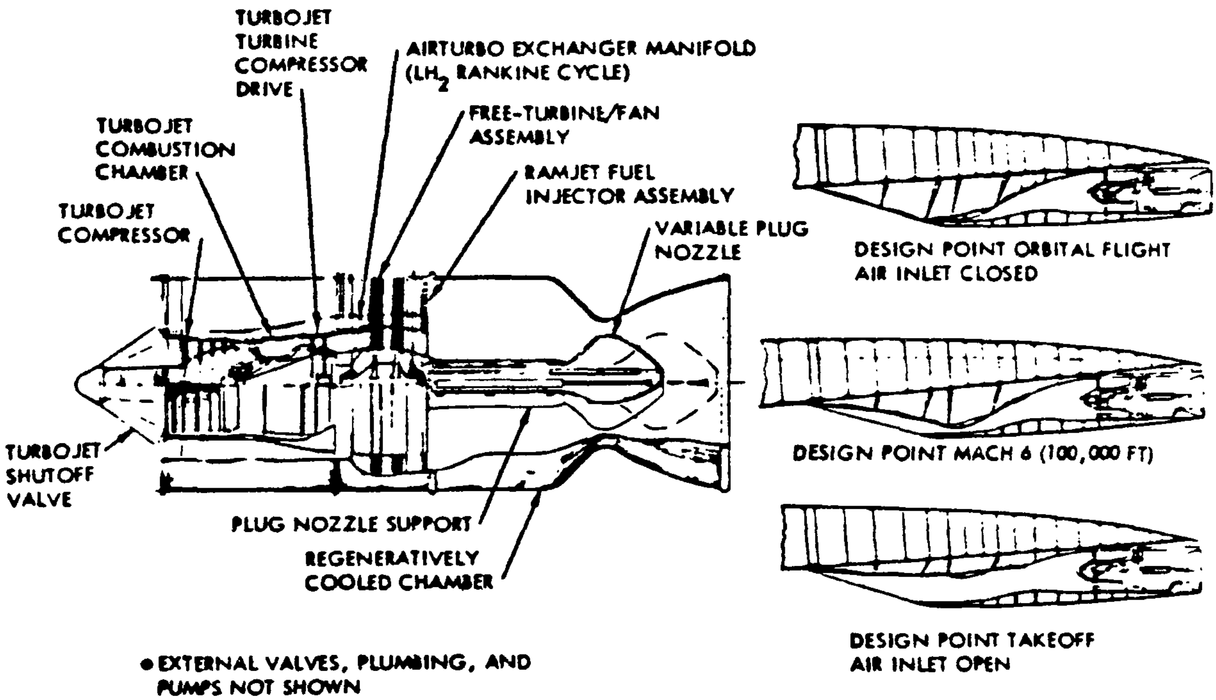


Figure 18 Multicycle Airbreathing Engine and Inlet Configuration - Rockwell

These successful reusable systems are as follows:

- The automotive engine (a simple, low-technology system)
- The aircraft turbojet engine (a complex, high-technology system)

The engines initially introduced should perform to the conditions and limits identified up to the point of qualification. At the same time, additional operational experience will be accumulated on a test stand through the "fleet leader" concept. This approach accumulates additional experience far in excess of the operating fleet. This additional experience is the only way to identify certain types of random failures and weak points in the engine design. As the combined experience, inspection, and overhaul observations of the operational engine and the fleet leader are accumulated, the ultimate operational and maintenance procedures are developed, and the operational limitations can be expanded. As a result, the ultimate maximum life and minimum maintenance operations are developed to the desired level of confidence.

Since this approach is novel in the field of rocket propulsion, considerable new experience will evolve from the SSME, which is the nation's first reuseable, high-performance engine. This experience with the SSME and serious attention to reuseability in the beginning of the SPS transportation system should develop the necessary operational results approaching the success of propulsion systems for aircraft and automobiles.

C. Operational Considerations

In the construction and maintenance phases of the SPS, one to two launches of an HLLV (of the 400 mg or 180,000 lbm payload variety) are required each day. A fleet consisting of five to six boosters and six to seven orbiters is needed to place either the silicon (weighing $51x10^6$ kg or $112.2x10^6$ lbm) or gallium (weighing $34x10^6$kg or $74.8x10^6$ lbm) satellites in LEO. It is estimated that the turn-around time for each of the HLLVs is approximately four to five days. The number of reuses is based on current Shuttle criteria.

Operating cost is driven significantly by the degree of reuseability and the amount of refurbishment required on launch vehicles. It is expected that over the next three to six years, the present STS will mature operationally through flight experience in much the same manner as does a new commercial airplane. Improvements in subsystem performance, reduced turn-around times and reduced refurbishment needs will all contribute to providing information for SPS. However, additional advances in vehicle design and life (engine, insulation, structure, etc.) could significantly reduce operational cost.

Results from examining nonspace systems (airline and water transportation), that have undergone significant changes in the past 25 years, lead to the conclusion that an HLLV system would benefit from automation and reduced manpower support by incorporation of on-board, self-test, and performance-monitoring equipment. Possible design features were also identified which could minimize operational flow and the manpower associated with launch operations.

In summary, key factors for low-cost operations include the following:

- Design for long life and maintainability throughout the life cycle
- Automation of preflight check-out and servicing
- On-board, self-test, and performance monitoring

- Continual subsystem or component tests to gain experience and confidence for extending inspection intervals
- Reduction of skill level for maintenance through simplified design
- Streamlined management for maximizing productivity

D. System Support Requirements

This section discusses three key areas of SPS transportation system support. The first consideration is the capability of the industrial base to support the STS transportation system by providing as an example the liquid-propellant rocket industry's current and projected status. The logistics considerations provide an indication of the magnitude of the area of logistics support needed. Logistics alternatives must be addressed early as they are major contributors to life-cycle cost. Launch-facility definition and location, the last area, not only can have an impact on program planning and funding if located outside of the U.S.A., but also will have an impact on personnel, propellants, spares, and payloads. All three areas have received limited study by SPS transportation-system contractors and a minimum of discussion during this workshop.

1. Industrial Base

Industrial base concerns arise for the SPS transportation system due to the current low level of funding in view of projected requirements for the 1990s and beyond. In assessing the industrial base, the following questions must be answered:

- What industrial base is required for SPS transportation?
- Will it be in place when required?

- What are areas of concern?
- What is needed to maintain or develop these areas?

To illustrate the potential overall problem, the following discussion of the liquid-propellant rocket industry is provided. It is recommended that this area and others which are identified are properly addressed in any near-term planning for an SPS transportation system. The American Institute of Aeronautics and Astronautics (AIAA) has recognized the problem of this industry and is preparing a position paper based on the use of cross-cut techniques. The discussion below reflects the tenor of the study.

Space Shuttle is a step toward establishment of routine, low-cost space operations. However, it is not an end point, and continued progress in lowering the cost of space operations depends on continuing development of propulsion technology. Unfortunately, at present, propulsion technology and system development are at a low ebb. The extensive funding commitment required to bring the STS to fruition and funding constraints imposed by current national priorities have severely restricted propulsion R&D. This tight budget situation, placing a strain on the propulsion industry, is resulting in the loss of some previously developed capabilities.

SPS and other future missions need new propulsion capabilities not included in the present STS. R&D lead time for a propulsion system is 5 to 10 years; therefore, delays in needed R&D can have significant downstream effects. Mission-performance capabilities become frozen; and the impact of a lack of propulsion system progress will be felt on the SPS, on the space program, and on industry by limited payload or mission opportunities and flexibility.

Because of the vital role of propulsion in the evolutionary growth of SPS and other space mission capabilities and because of the adverse effects that inadequate R&D support is having on the liquid-propellant propulsion industry, there is a need to renew the commitment to liquid-propellant propulsion R&D and to support restoration of an adequately funded effort. That effort must focus on promising options in propulsion systems and must be keyed to future requirements such as SPS transportation. These requirements are identified below as typical R&D options that should be pursued.

Space Shuttle will provide low-cost transportation to LEO for manned and unmanned missions. Economic analyses have identified Shuttle modifications which could improve its cost effectiveness.

SPS transportation studies have identified technological options which should be pursued for HLLV or SSTO; advanced liquid-propellant rocket propulsion is a key requirement. Advanced, high-density, high-pressure, liquid-propellant rocket engines are required by HLLV to maximize specific impulse while minimizing engine system volume and weight. High-density fuel is required to minimize vehicle size. High levels of specific impulse, and either an advanced version of the SSME or an entirely new dual-fuel engine are needed by SSTO vehicles. Lead times, up to 10 years, are required for some areas of this technology.

The liquid-propellant rocket propulsion industry is currently in a state of decline when it is needed to advance technologies which support development of necessary propulsion systems to maximize STS utilization, STS payload systems, and the SPS transportation system. The low level of the R&D budget has forced universities to turn to other areas of research, government laboratories to reassign their propulsion staffs, and industrial organizations to diversify and enter other markets or to leave the marketplace altogether. This situation has resulted in a rapidly declining liquid-propellant engine R&D capability, a national asset which took more than 30 years and billions of dollars to develop. This capability, if lost, will not be easy or cost effective to reestablish. It represents knowledge and experience not found in textbooks. If it is not supported by meaningful technology and development efforts at a significant funding level, it will be lost to SPS and other future space programs. The present austere planning of NASA and DOD, unless supplemented by a focus such as SPS, will not protect this technological base.

2. Logistics

In order to define the logistics requirements-- both on Earth and in space and to establish the feasibility of meeting these requirements--a comprehensive, end-to-end analysis was conducted of space and ground operations for construction, operation, and maintenance of the Nth satellite and rectenna. From these analyses, the time-phase, personnel and material-flow requirements on Earth and in space were derived.

Within the context of the systems and mission timelines defined, no operational or technological barriers to performing the logistics functions were uncovered. There were, however, cost-sensitive issues highlighted which bear on the problem of space-transportation economics, e.g., costs of hydrogen at the launch facility. The two more promising near-term processes identified for liquid hydrogen production are coal gasification and water electrolysis. Coal gasification involves manageable but expensive logistics problems. Water electrolysis requires a lot of energy and costs more. It is recommended, as a part of the GBED program, that technological studies of the more advanced liquid hydrogen production processes, such as thermochemical and photosynthetic processes, be undertaken.

3. Launch Facilities

The SPS reference system assumes use of Kennedy Space Center (KSC) at Cape Canaveral, Florida, as a launch site. Three potential limitations at KSC are space for the launch pads, noise and sonic booms, and other concurrent activities. These limitations together with the potential of performance improvements from equatorial launch sites led to an examination of alternate sites, primarily near the equator. The following discussion summarizes this examination.

Cape Canaveral can probably support an SPS emplacement up to approximately 10 GWe of power per year. A suggested site plan is shown in Figure 19. This figure has a high uncertainty, being dependent on achieving recycle rates for the pad. To the first order, it is not heavily driven by vehicle size. Vehicles smaller than the reference HLLV will alleviate concerns for noise and sonic booms.

Performance gains due to low-altitude launch are negligible with an electric-propulsion OTV. Reduction in ΔV is countered by increased shadowing by Earth for the EOTV. Appreciable gains are available (roughly 15 per cent) in chemical-OTV performance. The gains did not appear to offset the likely higher costs of remote site operations.

Low-inclination (23 deg) launch to an equatorial LEO provides frequent (about 15 times per day) launch windows and a lesser radiation environment for the crews.

No desirable equatorial land sites were found, given political,

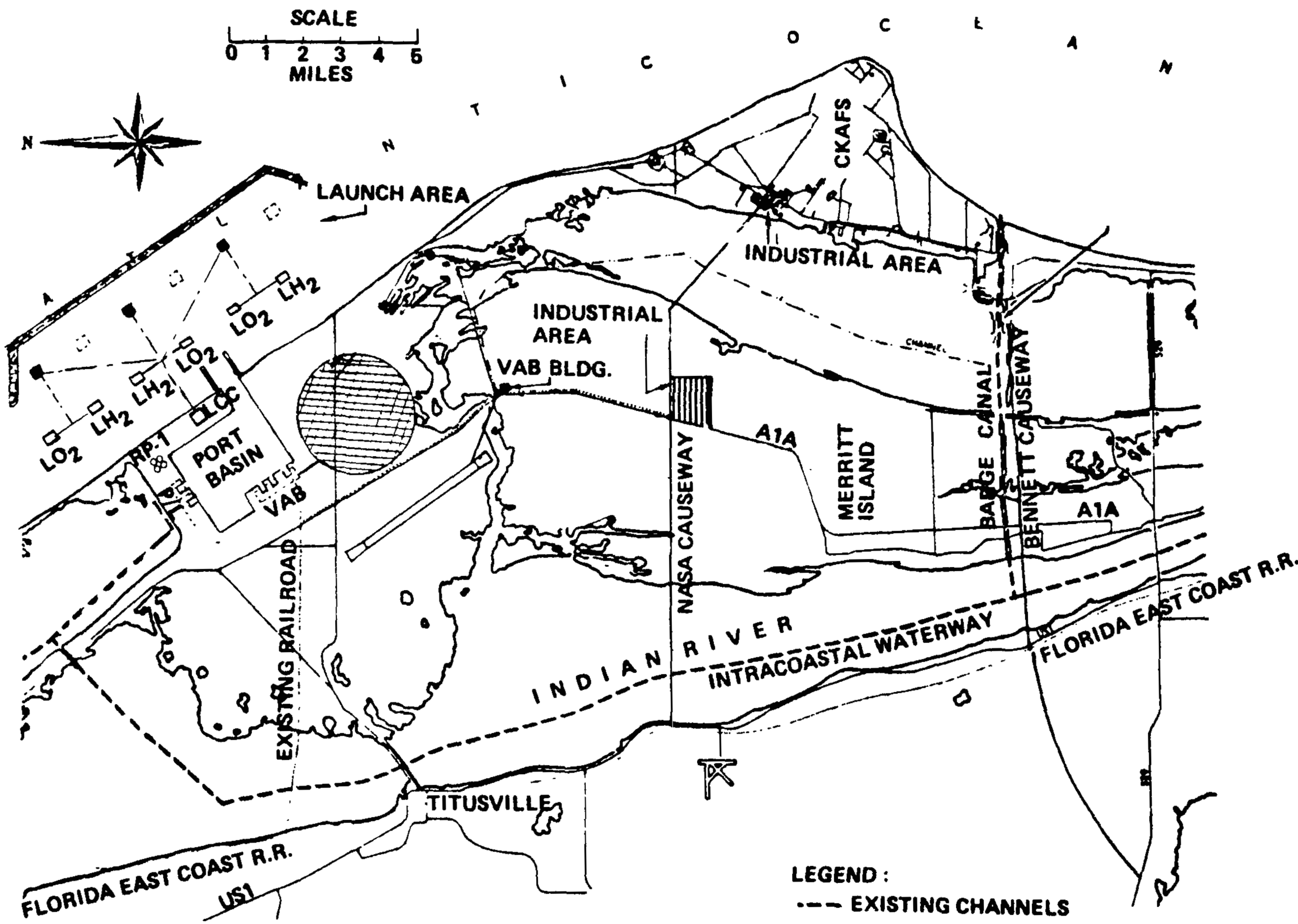


Figure 19 HLLV Site Plan - Cape Canaveral, Florida

environmental, and safety considerations. However, a potentially attractive off-shore installation concept was developed with the characteristics shown in Figure 20. Other features studied include the following:

- Location off the west coast of South America in international waters at a latitude of some 3 deg
- Mild climatology, weather, sea states and low currents
- Water depth on the order to 600 ft (180 m), well within offshore technology
- Brown and Root, Inc. examined moored, semisubmersible, and jacketed structures and projected an installation cost of $3 to $4 billion. Facilities and equipment costs are additive to this base structure cost. The structures provide areas for landing runways, processing, cargo hauling, propellant storage, and launch operations. Facilities and equipment would be installed on the structures in a continental shipyard before towing to the emplacement site
- Estimated cost of this approach is less than a remote, land-based facility

Further study is required in this area to refine system size limitations for KSC use and to develop credible cost data to support a launch site location trade-off study. An input to this study should be the results of a complete logistic study to define launch rates, material, propellant supply, and personnel supply rates.

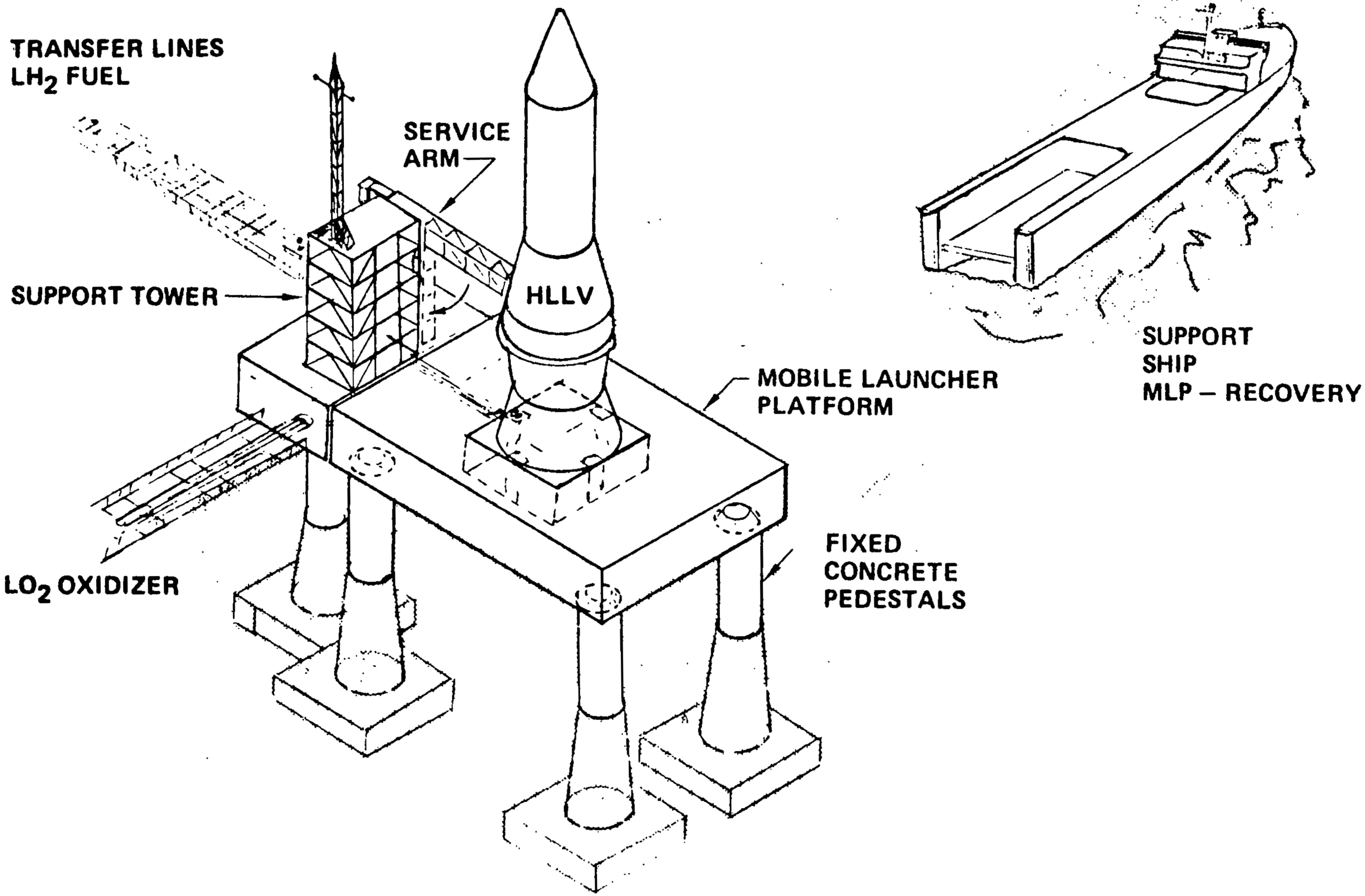


Figure 20 Off-Shore HLLV Launch Site Installation

III. ORBIT-TO-ORBIT (INCLUDING INTRA-ORBIT) TRANSPORT (OTO)

This section provides an overview of OTO and intra-orbit transport and traffic requirements associated with the reference SPS concept. In addition, it directs attention to some important areas of uncertainty and issues bearing on OTO transportation requirements that require more thorough investigation which may lead to substantial changes and improvements in the definition of SPS and its operations. Based on these observations, it identifies some key items which should be treated in the next phase of the SPS program.

A. Orbital Transfer Vehicle (OTV) Missions

The SPS system is to be developed in three major overlapping phases according to the current reference systems:

- Orbital base construction (LEO and GEO) and on-orbit construction of electric orbital transfer vehicles (EOTV)
- Construction of the SPS satellites
- Operation and maintenance of the SPS satellites

The orbital transfer modes required by each of these three phases are as follows:

- Intra-orbit transfers (transfers typically less than a few kilometers, except during maintenance)*
- Personnel and cargo transfers between LEO and GEO
- Emergency personnel and high-priority cargo transfers directly to and from GEO orbit (The last transfer mode has not previously been included and probably should be considered as a side issue)

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*The maintenance phase requires intra-orbit transfers between all deployed satellites (spaced 2 deg apart in GEO) at the rate of twice per year.

Table 2 summarizes the number of OTO transfer flights required for a two-satellite SPS system. A more detailed analysis of these parameters should be performed; ultimately these data could be used to determine the propulsion system characteristics.

Using Table 2, an overall timeline and sequence of activity can be developed. First, intra-orbit transfers at LEO must be performed for each HLLV and PLV. The second GEO intra-orbit transfer represents that required to perform SPS maintenance and corresponds to servicing 20 satellites in a period of 90 days. This 90-day servicing is performed twice a year as indicated by the two LEO-GEO-LEO transfers required. Personnel and cargo (4,000 klystron tubes, for example) are transferred to GEO by a single vehicle.

1. Cargo Transport From LEO to GEO

The LEO to GEO cargo transfers required for construction of the SPS satellites and vehicle returns in the reference system scenario are not performed in series, but overlap in their timelines. Even with this overlap, given a number of EOTVs in simultaneous operation, the 120-day transfer required seriously restricts the time to load and unload cargo and refurbish the EOTV vehicle and propulsion system. The requirement for priority cargo OTV with chemical rocket propulsion systems needs to be assessed, especially during the demonstration and construction periods.

In summary, the assumed SPS construction rate of two satellites per year is an overriding system driver and the resulting nominal timelines are probably unrealistic. It is suggested that OTO transfer traffic models should be developed as a function of transfer time (i.e., thrust acceleration levels), SPS deployment rate, and SPS mass required in GEO. With this, OTO transfer vehicles can be sized and optimized; and the mass rate required in LEO by HLLVs can be

Table 2

Orbit-to-Orbit Transfer Mission Requirements

Mission	No. of Flights to Build Construction Bases and EOTVs		No. of Flights for SPS Construction (2-5 GWe per Year)		No. of Flights for SPS Maintenance (one Year Period)	
	Personnel	Cargo	Personnel	Cargo	Personnel	Cargo
a. LEO Intra-Orbit	32	118	30	375	6	N.A.
b. LEO-GEO-LEO	6	3	12	30	2	N.A.
c. GEO Intra-Orbit #1	5	2	11	29	-	-
d. GEO Intra-Orbit #2	-	-	-	-	25	N.A.
e. LEO Intra-Orbit	6	3	12	30	2	N.A.
f.	1	N.A.	1	N.A.	1	N.A.
g.	N.A.	1	N.A.	-	N.A.	4

accommodated. Accordingly, system timelines can then be developed including appropriate cargo transfer vehicle construction, cargo loading and unloading, and vehicle refurbishment.

2. SPS Module Transfer From LEO to GEO

While recognizing the importance of the reference SPS system concepts as a stepping-off point for technical and economic assessments, it is observed that areas of uncertainty exist, which should remain open as subjects for investigation and which could lead to substantial changes and improvements in the character of SPS and their operations.

The option of constructing SPS modules in LEO for transfer to, and final assembly in, GEO is a potentially competitive approach which could be technically and economically superior if:

- EOTV reusability cannot meet or exceed ten round-trip flights
- Solar-cell annealing capability cannot be reliably held above 50 per cent
- Operational factors are significantly different than currently foreseen, including the docking problem

3. Personnel Transport

The importance of transporting large numbers of personnel from LEO to GEO for construction of the SPS must receive full consideration from the initial to the final system and their subsequent operation. The vehicles configured for this use have chemical rocket propulsion to minimize transfer time, especially through the Van Allen belts, and are presented in the following section.

4. Emergency Personnel and High-Priority Cargo

The reference SPS concept does not include provision for emergency transfer to Earth or for quick-reaction delivery of high-priority cargo ("Federal Express").

The need for these mission capabilities should be assessed. The vehicles for this use have not yet been configured. Such vehicles should incorporate the capability for direct flights to Earth from LEO or GEO with airstrip landing.

B. Chemical Rocket Orbital Transfer Vehicles

A reusable cryogenic Shuttle upper stage has been considered to be part of the STS program for over 10 years. This program is more than twice as far away as it was seven years ago, as is shown in the following table.

Concept	IOC Date	Δ Years
Space Tug (1973)	1979 (Initial)	6
	1982 (Final)	9
Interim Upper Stage (1975)	1980	5
To be followed by Orbital Transfer Vehicles	1983	8
Orbital Transfer Vehicle (1980)	~1992	12

The reasons for this increased delay shown above are a combination of lack of near-term funding (which will still be unavailable for a number of years because of the need to bring the Space Shuttle to operational status) and the decision to use the available time to go to the direct development of a "clean sheet" advanced system in 1992.

If STS upper stage and early OTV capability is to be obtained within a desirable future (say, within this decade), a feasible approach is to pursue an evolutionary program.

Such an evolved program would initially make maximum use of existing subsystems, which would be improved as technology became available and introduced as the capability was required.

Initially, the cryogenic stage would be used in an expendable mode. During this time, experience in operating a O_2/H_2 stage from the Shuttle would be obtained. This stage would increase STS payload capability to GEO by a factor of approximately 2.5 over that of the Shuttle/IUS. With modification and operating at low thrust, initial experience can be gained in the erection and deployment of large structures in LEO and GEO.

This stage would then be modified to allow it to be returned in the Shuttle Orbiter payload bay and brought back to Earth for re-use. This re-use capability would provide operational experience, rather than economic pay back, and would include an improved cryogenic space insulation, in-orbit servicing and eventually manual operation.

The feasibility of the chemical OTV does not have to be established. Rather, the uncertainties facing the chemical OTV are in the realm of life and cost, not performance, and these are the issues that need to be better defined. The eventual approach to the design, development and operation of the chemical OTV engine will be nearer to commercial aero engine practice and possibly even the industrial gas turbine, rather than that used for the present generation of liquid-propellant rocket engines. The combination of low cost and long life engines therefore are expected to require the following actions:

- Reduce dependence on strategic materials
- Enhance reliability and life
- Extend in-service periods
- Employ fail-safe design
- Accelerate minimum cycle development

Requirements for space-based operation include use of condition monitors and engine diagnostic systems (EDS) techniques.

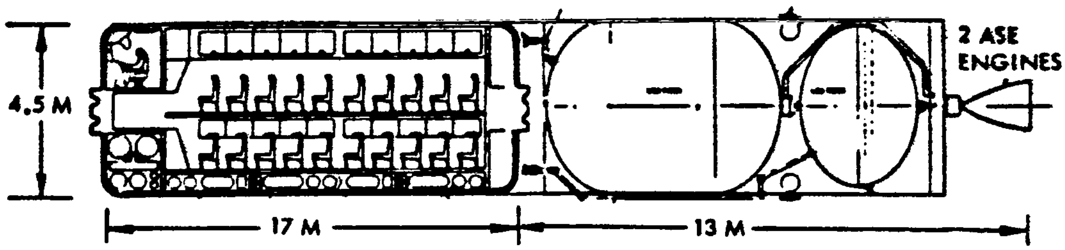
Three reference OTV missions are envisioned in the SPS program as follows:

- Cargo OTV, for transfer of intermediate cargoes from LEO to GEO and return
- Large cargo OTV from LEO to GEO and vehicle return (EOTV baseline)
- Personnel/priority cargo OTV for short turnaround between LEO and GEO. An emergency ballistic re-entry vehicle may also be required

Chemical rocket OTV options currently identified that could meet the personnel/priority cargo transport requirement are shown in the next three figures. Figure 21 shows a single-stage OTV and a crew module, which could also carry cargo, that are compatible with the payload bay of the baseline Shuttle. This vehicle would find use during the space test and demonstration phases of SPS. A growth version OTV is shown in Figure 22 that would find use during the establishment of the GEO construction base and the construction of the initial SPS. A derivative cargo STS is needed for transport of this space-based OTV which would be refueled for the return to LEO in GEO. The two-stage personnel/high priority cargo OTV, shown in Figure 23, is a fully developed concept that would find continued use between LEO and GEO throughout the construction phase and during the operation phase of the SPS. Such a vehicle would make effective use of the in-orbit propellant-processing facility concept presented in Figure 24.

A range of chemical rocket OTV engines will be required from low thrust (~4,500 Newton [1,000 lbf] for low acceleration and reaction control) to much higher thrust (≈470,000 Newton [100,000 lbf] for primary propulsion of the above personnel/high priority cargo and intermediate cargo OTVs).

For some years NASA has had an advanced space engine (ASE) under development with the configuration and characteristics shown in Figure 25. The further development of such an engine should be continued but its cycle, thrust level and



- 60-MAN CREW MODULE 18,000 KG
- SINGLE-STAGE OTV (GEO REFUELING) 36,000 KG
- BOTH ELEMENTS CAPABLE OF GROWTH STS LAUNCH

Figure 21 Personnel Orbital Transfer Vehicle (POTV) Configuration - Rockwell

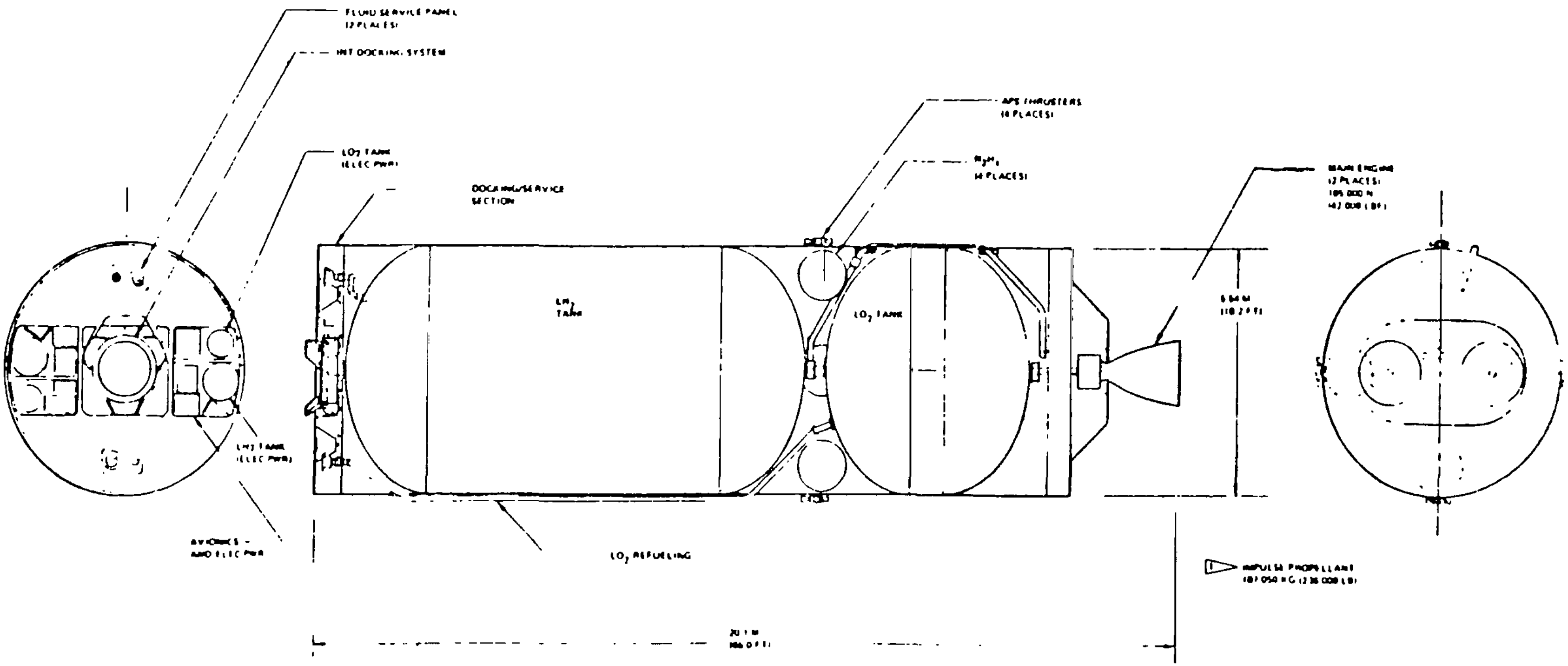
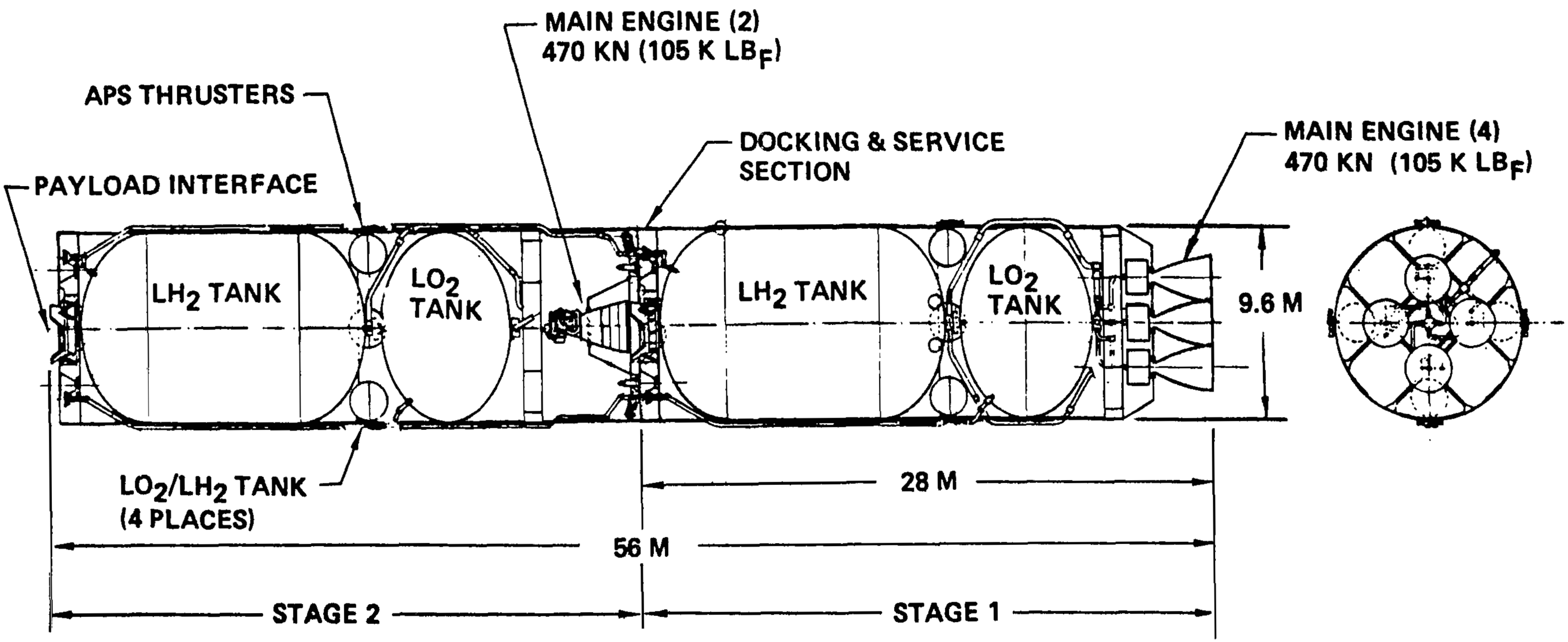


Figure 22 Single-Stage Advanced Personnel/High Priority Cargo OTV - Boeing



- PAYLOAD CAPABILITY = 400,000 KG
- OTV STARTBURN MASS = 890,000 KG
- STAGE CHARACTERISTICS (EACH
  - PROPELLANT = 415,000 KG
  - INERTS = 29,000 KG
    (INCLUDING NONIMPULSE PROPELLANT)
- 280 OTV FLIGHTS PER SATELLITE

Figure 23 Two-Stage Personnel/High-Priority Cargo OTV - Boeing

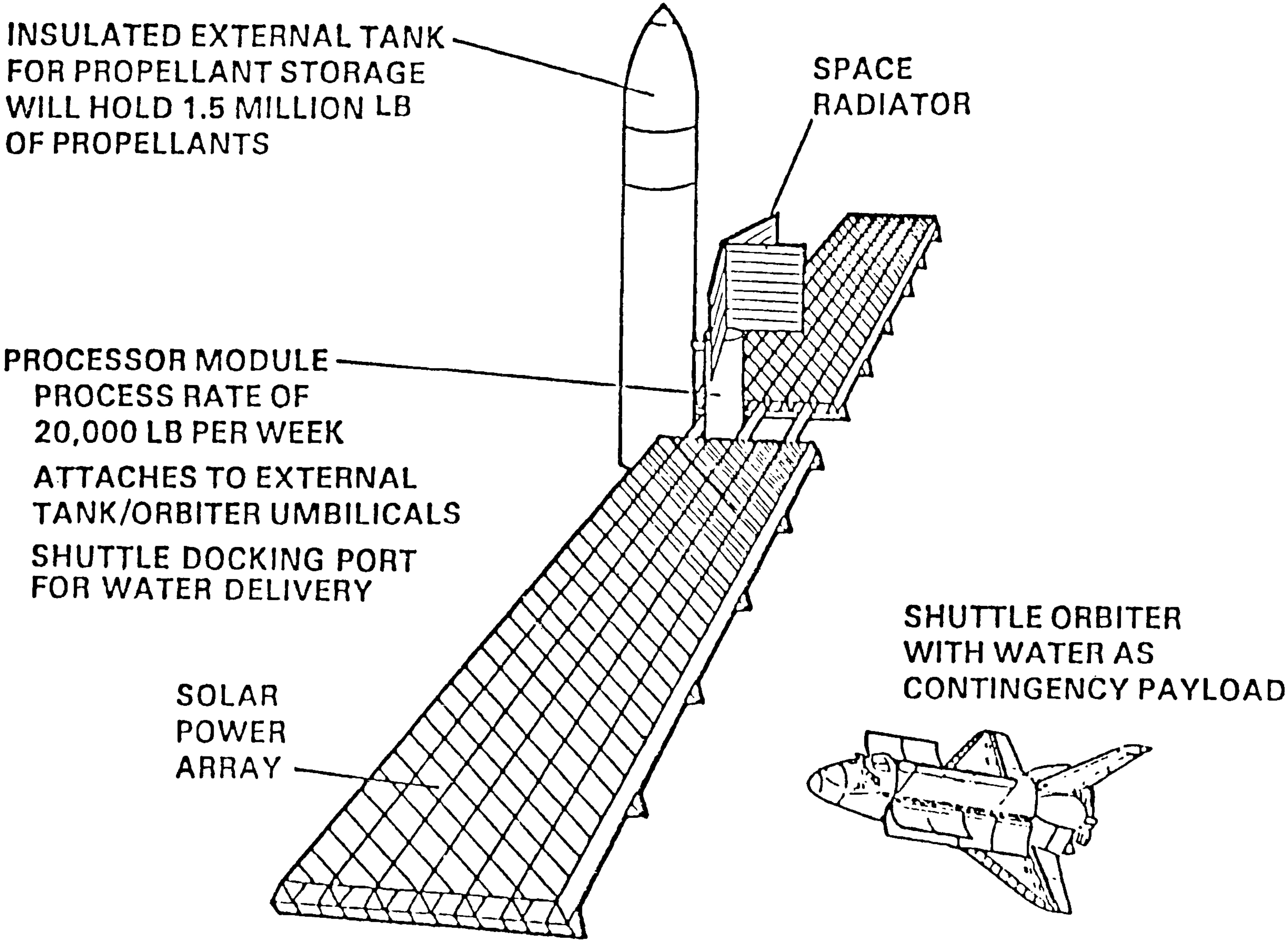
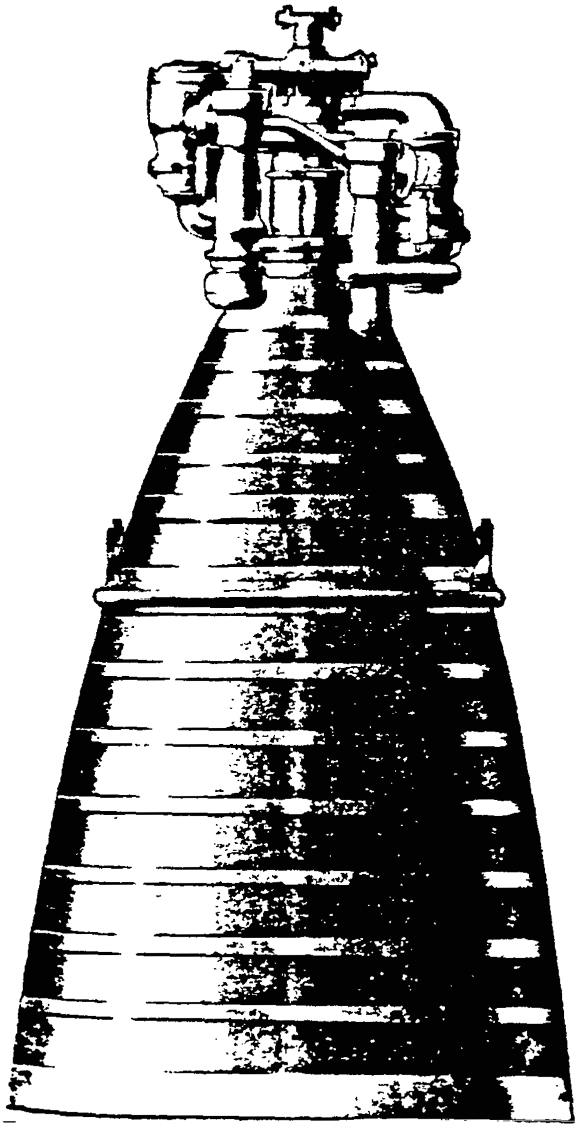


Figure 24 In-Orbit Propellant Processing Facility Concept



THRUST (LB)	20,000
CHAMBER PRESSURE (PSIA)	2000
EXPANSION RATIO	400
MIXTURE RATIO	6.0
SPECIFIC IMPULSE (SEC)	473.0
DIAMETER (IN.)	48.5
LENGTH (IN.)	
NOZZLE RETRACTED	50.5
NOZZLE EXTENDED	94.0

Figure 25 Advanced Space Engine (ASE) Characteristics

other characteristics must be reviewed so they are compatible with the perceived needs of the orbital transfer vehicles in the future U.S. space program including the SPS.

Certain low-thrust chemical rocket propulsion technological efforts have already been initiated to meet NASA and DOD requirements for transfer of acceleration-limited structures from LEO to GEO. These programs should be examined for their applicability to SPS and augmented where appropriate to meet those operating requirements that are peculiar to SPS. Systems analysis should be undertaken to evaluate promising concepts from the standpoint of life-cycle cost, mass, performance and environmental considerations.

Other programs in component technology should be undertaken in the areas of propellant-feed systems designed for maintainability and long operating times or intermittent operation, long-life reusable thrust chambers, control systems and utilization of low-cost materials. At the end of the next phase of the SPS, several low-thrust chemical rocket concepts will be defined to a sufficient degree to permit their evaluation for use in various SPS vehicles. Breadboard system demonstrations of the most attractive concepts could then be initiated to verify the technical merit.

A recommended program of activities is presented which will undertake to show the merits, potential and costs of chemical propulsion systems tailored to meet mission needs. The goal of this activity is to reduce uncertainties in the following:

- Performance, mass, lifetime, maintenance and on-orbit operation
- Cost comparisons and cost-estimating relationships
- Range of applicability of chemical rocket systems

With this goal accomplished, a comparison of chemical rocket and other candidate approaches (i.e., electrical and more advanced) can be conducted by the systems contractors with the knowledge that the chemical-rocket data base will be at a high confidence level. It is recognized that the technology of other candidate systems is not as mature. Therefore, this base will serve as a measurement standard against which the performance characteristics of other candidates can be judged. Following these judgments, suitable trade-off studies can then be conducted and the lowest cost systems (including unreliability impacts) can be selected.

C. Electric Orbital Transfer Vehicles (EOTV)

It is the consensus of the working group that ion propulsion for transfer from LEO to GEO is feasible and may offer major cost savings relative to chemical propulsion. The cost savings result primarily from the reduced mass delivered to LEO. The feasibility of ion propulsion has been demonstrated in the development of a substantial body of technology, including space tests, during the past years. Since ion thrusters are more developed, they were selected for the initial systems analyses; however, other options that should be considered are described in Section F below.

Although a considerable amount of technical work must be performed before a suitable electric propulsion system is available for OTV application, the cost of this work will be small compared to the cost savings that can result. To be more specific, ion propulsion permits a reduction by a factor of 2 or 3 for the mass required at LEO to place a given payload at GEO. This major mass reduction has an associated reduction in overall cost.

EOTVs currently defined in the reference SPS by the major contractors are shown in Figures 26 and 27. A typical electric rocket propulsion system with 120-cm (46.8-in) diameter ion thrusters, using argon as the propellant, is shown in Figure 28.

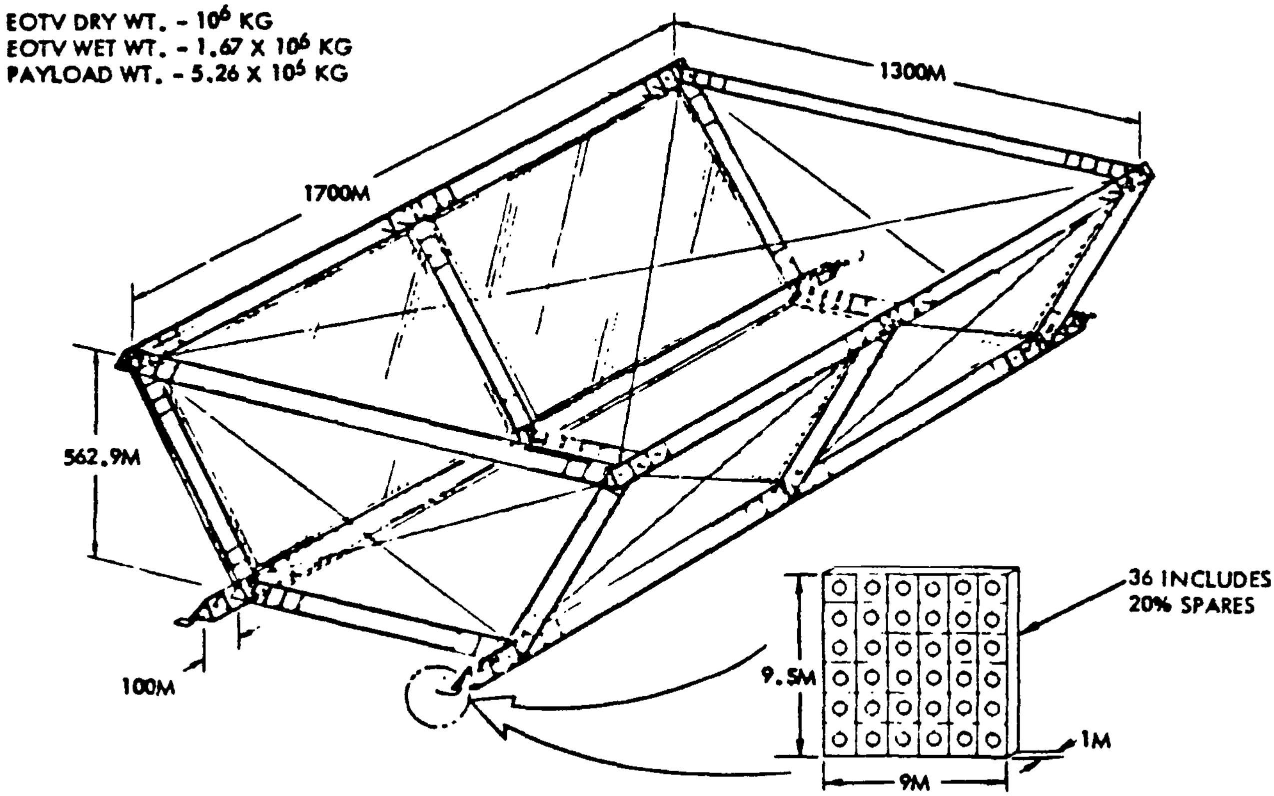


Figure 26 Electric Orbital Transfer Vehicle (EOTV) Configuration - Rockwell

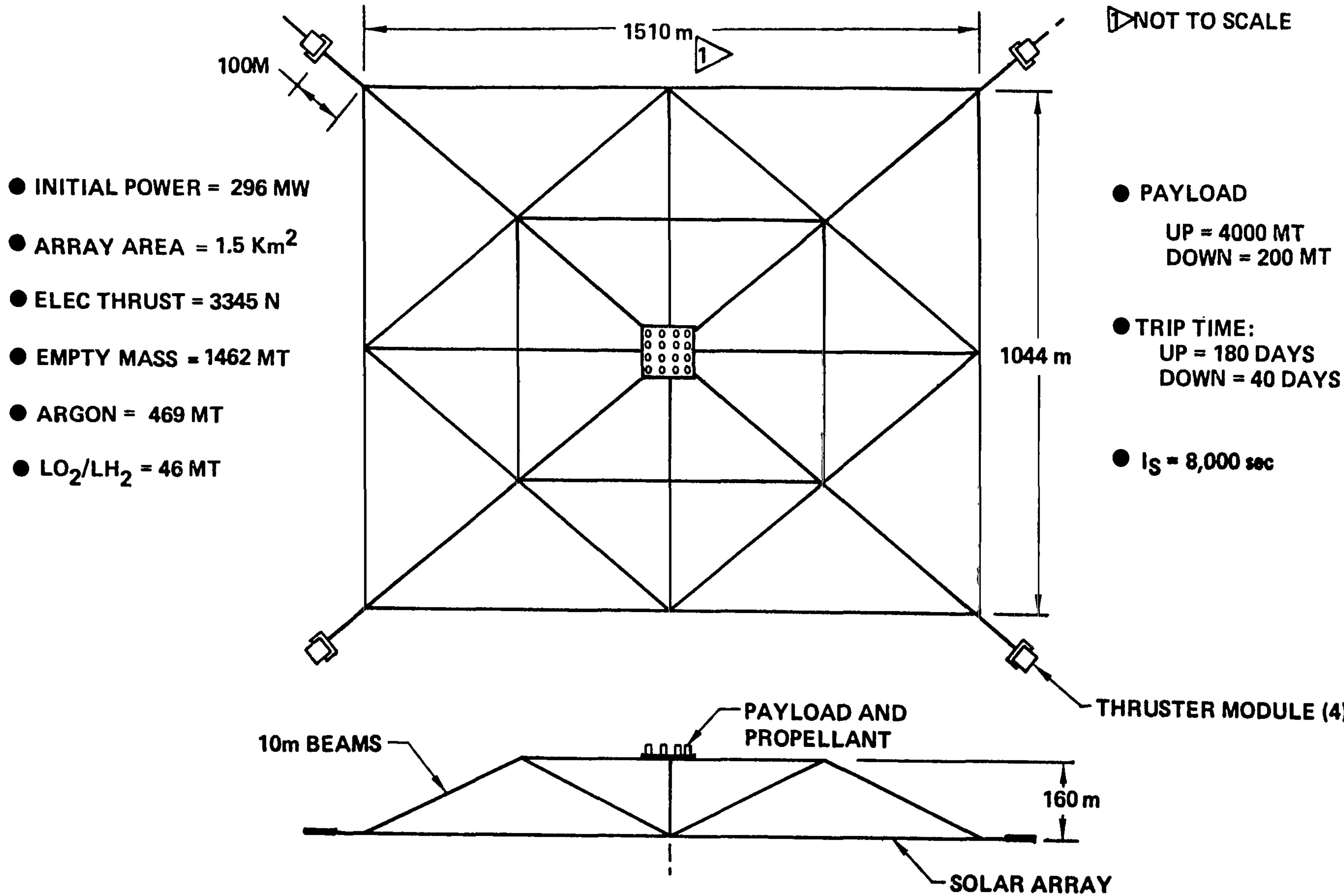
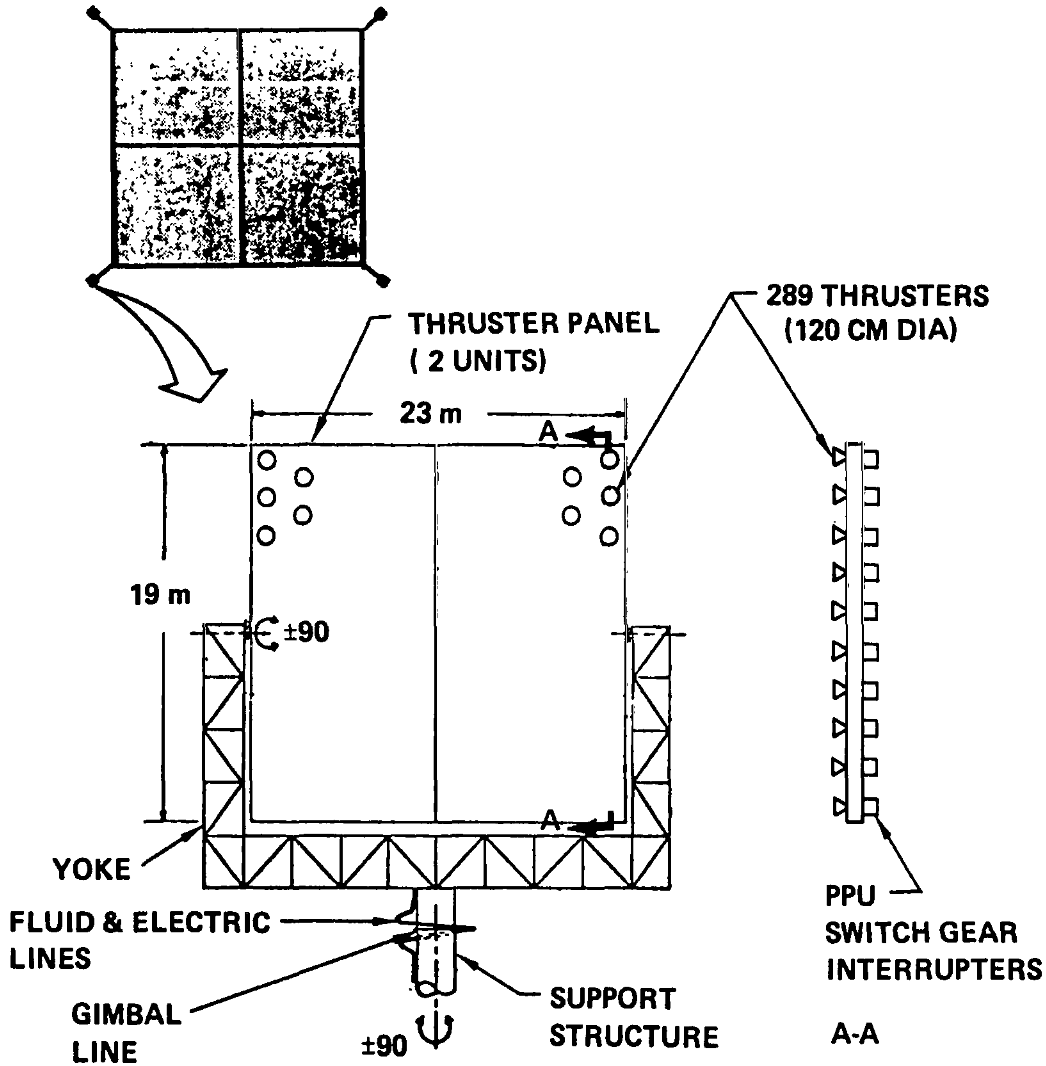


Figure 27 Electric Orbital Transfer Vehicle (EOTV) - Boeing



- THRUSTER POWER SUPPLY
  - DIRECTLY FROM ARRAY
    - NO PROCESSING
    - NO REGULATION
  - NO PROCESSING
    - ARRAY REGULATION
  - PROCESS ALL POWER
- TYPE OF PROCESSING
  - MOTOR/GENERATOR
  - SOLID STATE
- PROCESSING THERMAL CONTROL
  - ACTIVE RADIATOR
  - 915 M^2
  - LIMIT ELECTRONICS TO 200°C

Figure 28 Electric Rocket Propulsion System - Boeing

Because of the major advantages of ion propulsion for the OTV, it is clear that the following tasks should be adequately addressed at an early time:

- Demonstrate that system performance is verified and that operating constraints are quantified
- Define the operating interfaces of the system
- Establish the ecological acceptability of the system

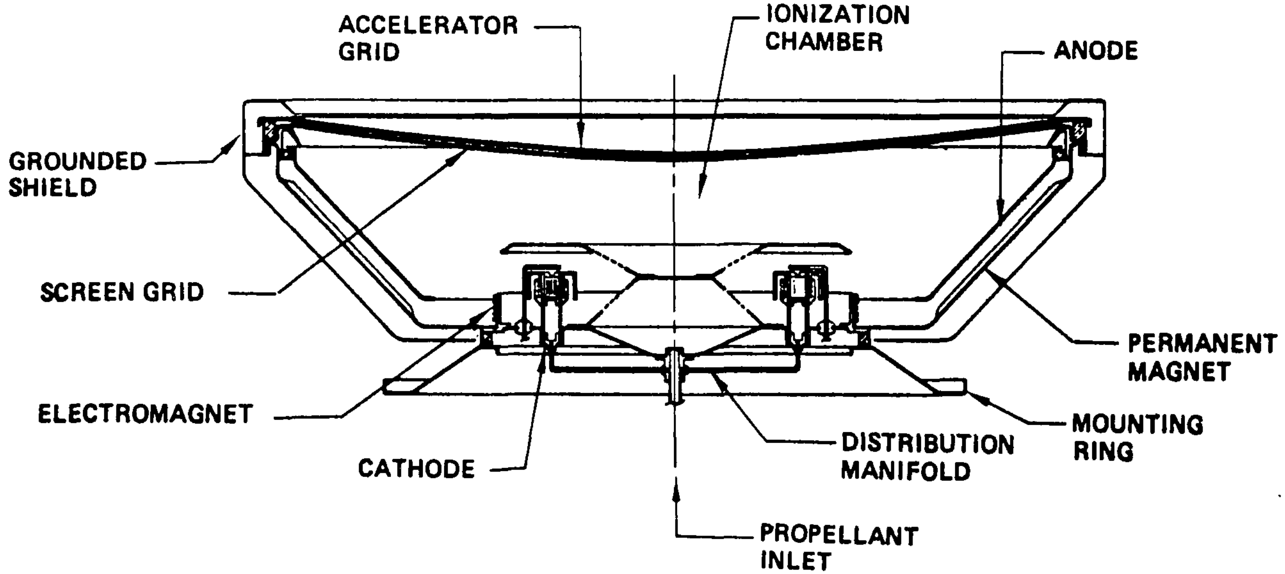
1. Electric Thrusters

As indicated above, an argon ion thruster, of approximately 1-m diameter, with conventional power conditioning similar to solar electric propulsion system (SEPS), is the reference system for ion rocket OTV. Such a thruster extrapolated from current practice is presented in Figure 29. The performance of this thruster (thruster efficiency of over 60 per cent at specific impulses above 6,000 sec) is a major driver for system cost. Performance estimates that have been made in SPS studies to date have ranged from either conservative to overly optimistic. Adequate performance appears likely, but the extrapolation from present work is quite large.

The importance of ion thruster performance results in a requirement for ground tests of the ion thruster of the size planned. In the absence of adequate facilities, a space test would be required for verification. The facility requirements for an approximately 1-m thruster emphasize the need for preliminary tests with a smaller thruster at the earliest possible time. This smaller thruster should be significantly larger than existing 30-cm (9-in) thrusters and can be assumed to be roughly 50 cm (19.5 in) in diameter. The development of this intermediate-size thruster should permit extension and verification of scaling relationships.

Thruster lifetime is also a cost driver. The major thruster components involved in the lifetime are cathodes (both main and neutralizer) and ion optics (accelerator system or grids). Because of the larger size and mass, the ion optics

• 120 CM THRUSTER



PARAMETER	30 CM	120 CM
SPECIFIC IMPULSE – SEC	3000	7500
THRUST – N	0.13	2.8
POWER – KW	2.6	140
BEAM VOLTAGE – V	1100.	1500
BEAM CURRENT – A	2.0	80
EFFICIENCY – %	72.0	75
LIFE – HR	15000.	5000.
MASS – KG	7.8	50

Figure 29 Ion-Thruster Technology Extrapolation - Boeing

are felt to be most important for cost. The problems involved in ion optics are replacement (refurbishment) and assembling and aligning ion-optic grids in GEO, a rather delicate operation, or replacement of ion optics complete with structural support sufficient to maintain alignment during transport from ground to GEO.

Lifetime tests should be conducted after adequate performance data have been obtained. These lifetime tests should be conducted both on the ground and in space. It is also felt that sufficiently sensitive diagnostic tests exist to permit adequate test duration to be of the order of 100 hr, if the thruster is recovered.

2. Power Conditioning

The power conditioning, like the thruster, represents a major extrapolation from present technology. To keep the cost low and reliability high, the module size of this power conditioning should be large, much larger than any existing in space or considered for any other space application. Heat rejection would, in the absence of other developments, result in modules having larger than present kg/kw ratios. A major need in power conditioning then, is to develop large, efficient and lightweight modules. A possible example of the type of development required is integration of heat pipes with the transformers.

The sequence of test proposed is, first, to develop thruster power-conditioning modules with adequate overall performance parameters. Then, at a lower priority, the interactions with an active load (ion thruster) should be evaluated and resolved.

A major reduction in power-processing mass (and perhaps also losses) could result from direct drive of ion thrusters from solar arrays. The largest power block is for the screen (ion beam) supply. The next largest block is for the discharge supply. The effective use of direct drive would be for the screen supply or for the screen and discharge supplies, with all other functions associated with conventional power conditioning.

The use of direct drive results in several interactions that should be evaluated. A major interaction is the dynamic one between the thruster and solar array. At the very least, switching of incremental array areas in and out of the circuit should be required for control. The voltages required for screen and discharge functions determine the sign and magnitude of associated array areas. The nature of interactions of these array areas with ambient and charge-exchange plasmas is thus partially determined by the choice of direct drive, if used. These plasma interactions are discussed below.

3. Solar Array

The basic, solar-array technology required for the SPS is assumed as an available base. The requirements discussed below are in addition to this base. The low end of the orbit-raising mission involves a high plasma density of $\geq 10^5$ cm^{-3}. Plasma interactions with high voltage (2kV) array surfaces will therefore be more intense than at GEO. Near the thrusters there will be additional contributions to this plasma density due to charge exchange of escaping propellant atoms with beam ions. The propagation of this charge-exchange plasma is not well understood, nor are the effects of the space plasma on a high-voltage array. Other thruster/array interactions should also be included.

The plasma environment under some conditions will be sufficently dense to assure near spacecraft-ground potential will exist outside all insulator surfaces surrounding solar arrays. Under such conditions, the insulators must continuously withstand the full local array voltage relative to the spacecraft ground. The large areas, the possibilities of manufacturing defects, defects due to poor handling during assembly, or micrometeoroid holes require that electrical breakdown failures be self-limiting. The physical processes involved in these breakdowns and the means of making them self-limiting are important areas for further experimental work.

The radiation degradation of the solar array in transfer from LEO to GEO is an important factor in solar array selection. This is in addition to the special plasma interactions faced by the OTV solar array. These special considerations for the OTV solar array indicate that serious consideration be given to a modified solar array design from that used in the SPS. For example, the inability to anneal radiation damage in silicon solar cells as indicated by the Boeing reference system, might make gallium arsenide a viable alternative for the ion rocket OTV, even if silicon cells are used on the SPS.

Environmental Interactions - Large quantities of ionized and atomic argon are expelled from the thrusters during orbit-raising operations. These large quantities raise the possibility of interactions with portions of the upper atmosphere. Because such interactions could be critical in the decision to use or not use an ion thruster, further study of these interactions is important. (See section on atmospheric effects of the SPS transportation system.)

4. Alternative Electric Thruster Systems

Other electric-thruster systems should be studied as possible alternatives. Emphasis here should be on propellants having minimal interactions with the upper atmosphere. Hydrogen appears to be a possible propellant from this viewpoint. Thruster concepts to be considered should include magnetoplasmadynamic (MPD) thrusters as discussed in the following section.

5. SPS-Focused Technology Program

The propulsion requirements for SPS require major extensions from the ion-propulsion system technology under development for planetary and geocentric applications. A focused program which would build upon the established technology is thus required to establish confidence in and define the performance envelopes of ion-thruster systems appropriate for SPS.

The three generic areas, as follows, require focused technology efforts:

- Hardware
- Field and particle interfaces
- Ecological and societal impacts

Brief discussions of each area, including summaries of the proposed technology efforts, are presented in the following paragraphs. Ground-based analyses and experiments comprise the bulk of the activity, but a Shuttle-based space test may be required to refine and corroborate the data obtained in ground tests.

Hardware Technology - Table 3 shows some of the technical areas deserving evaluation along with a summary of specific areas and rationales. It is presently estimated that a 4-yr program would be required to perform the key ground evaluation with a thruster intermediate in size between the present 30-cm (11.7-in) size and the sizes of interest of ~1 m for SPS. A flight test of a full-size thruster may be required to confirm lifetime and performance due to the expected limitations in vacuum-facility pumping capabilities in the 1986 time-frame. The power-processor technology program (primarily evaluation of high-power components) could be performed completely in ground tests.

6. Field and Particle Interfaces

The bulk of the field and particle interfaces will be adequately addressed in on-going programs. The characteristics and impacts of the low-energy plasma from SPS-size thrusters would, however, require focused evaluation. At present, adequate scaling laws applicable to the relevant thrusters' dimensions and operating conditions are not available nor are plans in existence to obtain them. As a special consideration, due to anticipated vacuum-facility limitations, the Shuttle flight test mentioned earlier would be required to refine and verify the models and experimental data obtained during the ground-based program.

Table 3

MAJOR EOTV HARDWARE TECHNOLOGY AREAS

GENERIC	SPECIFIC	COMMENTS
• Thruster Lifetime	• Ion optics and cathode lifetime versus thrust level	• Thruster refurbishment presently assumed in system studies. Large EOTV-mission life-cycle-cost reductions possible if refurbishment requirements are eliminated/alleviated through increased life
• Scaling Phenomena	• Increased thrust per module - Thruster shape - Increases thrust/area - Advanced plasma containment - Multiple cathodes	• EOTV costs directly related to number of thrusters/PPU's. Strong cost benefits accrue for large increases in thrust/module
	• Power processor high power component technology	• Component powers much higher than currently demonstrated in space. Heat-removal technology required (such as heat pipes) to maintain or reduce power-processed specific mass
• Reduced Cost/Mass Power Management and Control	• Simplified power processor concepts • Direct drive	• Power processor and associated thermal control systems are cost and mass drivers in proposed EOTV designs. Simplification will affect system reliability
• Thruster Extended Performance and Operating Envelope	• Increased and variable thrust/power • Variable specific impulse operating range	• Increased T/P will reduce trip times and reduce EOTV fleet-size requirements • Variable specific impulse will allow - Use of primary propulsion systems for on-orbit propulsion - Minimize power (energy storage) requirements during occultation phases of orbit raisings
• Solar Array	• Radiation-resistant solar arrays • Radiation-recovery technology	• EOTV power system environments much different than on-orbit • Power degradation during orbit transfer strongly affects EOTV scenario

7. Ecological and Societal Impacts

The impact of the argon-ion beams on the upper geosphere is presently under study. The present situation is that large-scale uncertainties exist as to the exact interaction phenomena to be expected. Ground and space tests will probably both be required to fully understand and accommodate as necessary the operation of ion beams on the scale of SPS.

D. SPS Station-Keeping and Attitude Control

Station-keeping and attitude-control operations are performed at LEO during transfer from LEO to GEO and at GEO. These operations are required for the LEO base, for the EOTV and POTV during transfer from LEO to GEO and return, and for both the GEO base and the satellites maintained at GEO.

1. Baseline Definition

Based on the several workshop presentations and discussions with Boeing and Rockwell study personnel, information on the baseline systems for station-keeping and attitude control was obtained. Both the Boeing and Rockwell baseline systems are noted in Table 4 according to function. Differences and open issues are readily identified by this comparison.

2. Baseline Difference/Open Issues

The two contractors have decided upon varying attitude-control and station-keeping scenarios based on assumptions that greatly differ.

For several of the attitude-control system (ACS) functions and locations, Boeing has decided to use chemical (O_2/H_2) rather than electric propulsion. Their differences in Isp greatly affect the amount of propellant which must be transported, stored, etc. The rationale for Boeing's baseline is that they believe the high-velocity ions coming out of the electric thrusters may be detrimental to the personnel and materials located at the LEO and/or GEO bases. Rockwell, on the other hand, has decided to use high-performance electric thrusters using SPS satellite technology. Personnel and equipment protection would be achieved by using

Table 4

Baseline Systems - Station-Keeping and Attitude Control

FUNCTION	BOEING	ROCKWELL
LEO Base	Chemical (LO_2/LH_2) Isp 400 sec	Electric (Ion) Isp 13,000 sec
EOTV	Electric (Ion) Isp 7,500 sec and chemical	Electric Isp 8,300 sec and batteries
POTV	Chemical	Chemical
GEO Base	Chemical	Electric Isp 13,000 sec
Satellites	Electric Isp 20,000 sec and chemical	Electric Isp 13,000 sec and batteries

appropriate shielding and configurations to preclude ion impact. This "health" issue must be addressed in greater depth to decide whether this is a go or no-go decision.

Boeing has baselined the use of a backup chemical ACS system for both the EOTV and satellite. Rockwell relies on the use of an electric propulsion system for these functions. Rockwell utilizes energy-storage devices (batteries) to power the electric propulsion during these periods. They would possibly have to add more thrusters and batteries to cover the higher thrust periods. It seems clear that a much more detailed trade needs to be made relative to which of these baselines is more cost effective.

Both contractors have decided to resupply the satellite ACS propellants on a regular basis. However, Boeing's baseline is that this propellant will be stored at the GEO base and transferred to the satellite's tankage. Rockwell decided that it would be better to replace the empty tanks with new tanks that have been refilled after transport down to Earth. This differing philosophy probably has a great effect on the mass transport quantities and their costs. No clear definition of why these differing philosophies have been used is apparent. Therefore, a more detailed trade study should be undertaken which will highlight which of these approaches is more cost effective.

In addition to the above, several of the baseline decisions of both contractors seem to <u>not</u> have a good base in existing technology. These items are discussed below.

3. Technology Issues

Included here are items which must be evaluated and tested.

a. Electric ion thrusters

A new, large-diameter (~120 cm/46.8 in) thruster which must be developed exceeds the size of any fully qualified thruster to date. The largest previous device has a diameter of 30 cm (11.7 in). This advanced technological undertaking becomes an item of considerable concern. Further compounding the situation is the probable need for two different thrusters (low Isp, high thrust and high Isp, lower thrust). A demonstration of this capability for long lifetime is definitely required.

A detailed study is also needed to assess the effect of thruster exhaust particles on the vehicle and any adjacent personnel. This latter information is needed to determine if electric propulsion can be used for control operations of the LEO and GEO bases.

b. Chemical thruster

While of lesser concern than the electrical-thruster questions, information is also required on pulsing oxygen/hydrogen thrusters. Performance (400 sec Isp pulsing) and life-testing are required to show SPS applicability.

E. Intra-Orbit Transport

The need to provide an intra-orbit transport capability is implicit in the construction and maintenance approach for SPS. It should be recognized, however, that a versatile vehicle is required to meet the varying on-orbit operations requirements independent of the construction site, i.e., LEO or GEO. First, there is the requirement for delivering payload from the HLLV depot to a LEO construction base located several kilometers away or to an EOTV for eventual delivery to a GEO construction facility. Second, there is a similar requirement in GEO for an intra-orbit transportation vehicle (IOTV) to off-load payload from the EOTV and deliver it to the GEO construction base. The characteristics of such a vehicle depend on the mass of payload being delivered and the number of payload modules which must be transferred to the construction site.

A small teleoperator IOTV will be required for local utilization and a version was conceived by Rockwell. IOTV sizing assumed a minimum safe separation distance between the EOTV and SPS base of 10 km (6 mi) and a round-trip transfer time of 2 hr. This equates to a ΔV of 3 to 5 m/sec (9.8 to 16.4 ft/sec). A single advanced space engine is employed with an Isp of 473 sec.

In contrast, the Boeing design concept for an intra-orbit personnel/cargo tug shown in Figure 30 is a much bigger, manned vehicle which obviously has a much larger payload-carrying capability.

In either case the technology to build such a vehicle is well in hand at the present time. The only technological issues concern on-orbit refueling and engine life since it is expected that the IOTV will be reusable. On the first issue, General Dynamics has done considerable work in the area of on-orbit refueling; and it is suggested that such work continue. As for the second issue, engine-life requirements, although not defined at this time, are not thought to be critical. Definitive studies to determine whether these vehicles need to be manned or can be operated remotely (e.g., teleoperator operations) also remains open to further study. It is apparent that such vehicles, if manned, could profit if dexterous manipulator capability were added to the crew cabin. Payloads could thus be moved about with comparative ease from within the cabin, thus reducing the amount of extra-vehicular activity (EVA) required of the crew and increasing their productivity. The development of a flight station incorporating such dexterous manipulators is also strongly recommended. These same manipulators are needed for closed-cabin, cherry-picker operations on the construction base. Thus, such a program would serve a dual purpose.

Once SPSs are operational, a third requirement for satellite-maintenance sortie transportation also exists. The primary function of this class of IOTV is the resupply of SPS expendables, and any maintenance support equipment needed to

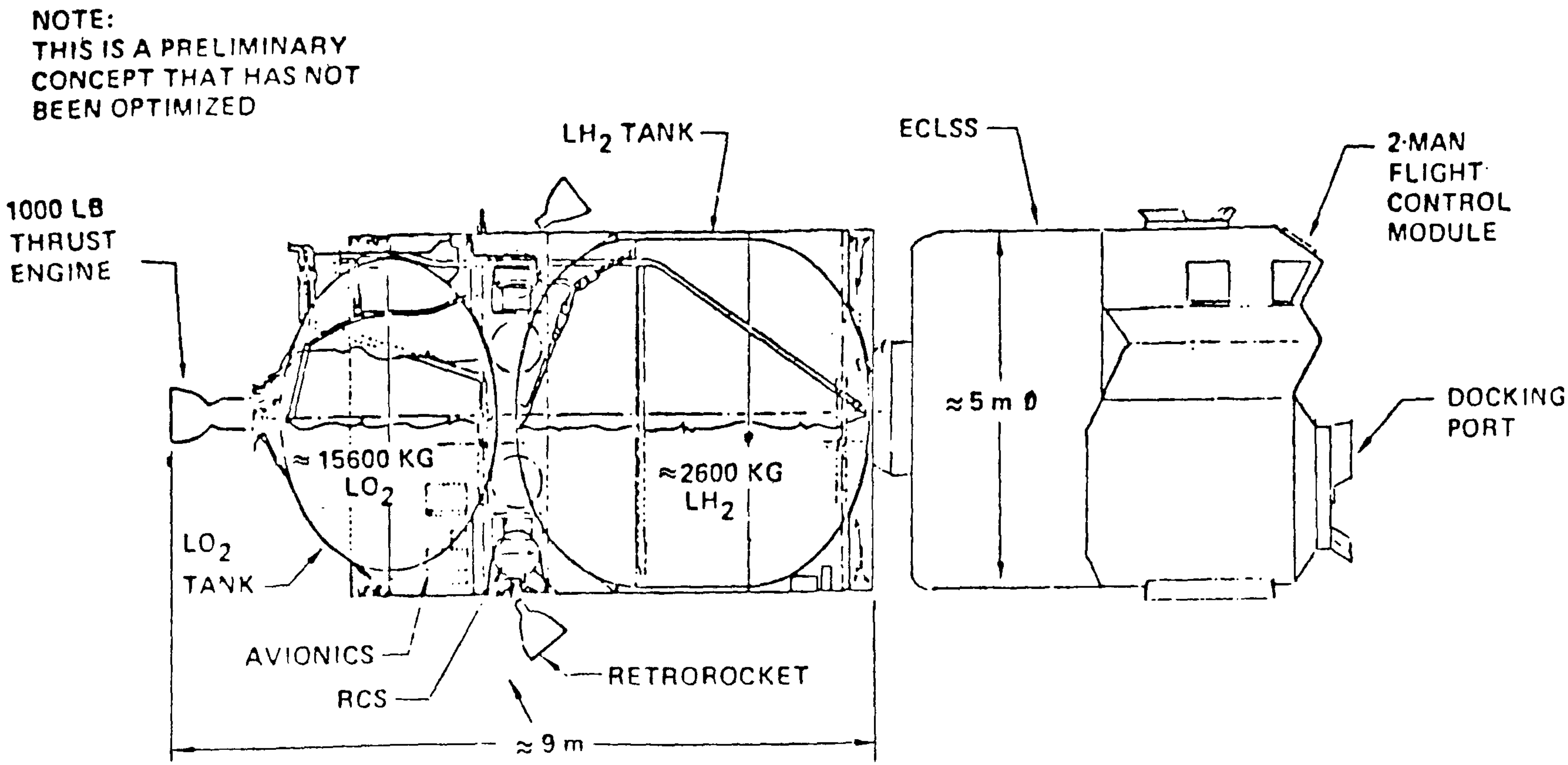


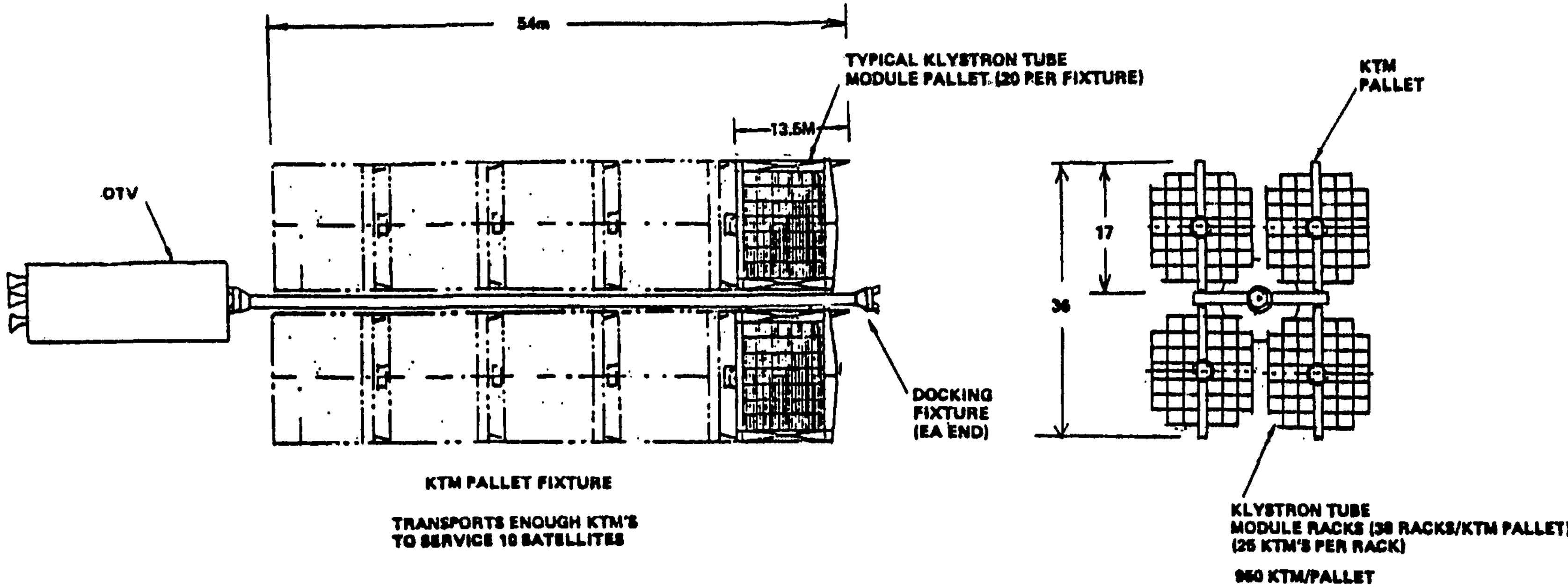
Figure 30 Intra-Orbit Personnel/Cargo Tug Concept - Boeing

keep the SPS operational. The characteristics of such a vehicle are shown in Figure 31 as seen by Boeing. This vehicle is designed to deliver supplies to 10 satellites in GEO per sortie. Its size is the same as the POTV. Beyond any technological issues mentioned previously for the POTV propulsion system, no further issues are foreseen.

Based on IOTV requirements as they are presently understood, there does not appear to be any impediments in the development of any of the different types of IOTVs needed to support SPS construction or maintenance. It is strongly recommended, however, that the following technologies be pursued over the next three to five years for the benefit of SPS:

- Development of dexterous manipulators for IOTV and cherry-picker operations to maximize man's productivity while working in space
- Continued funding of cryogenic engine development to assure the safety, reliability, and life requirements for man-rated OTVs
- Funding for the development of fluid transfer systems, and broadened scope of such studies to include all critical fluids needed to resupply operational SPSs
- Teleoperator simulations should be undertaken to determine whether construction and repair operations can be done remotely or whether man is required in close proximity to the work site

The above issues should be addressed immediately with ground-based simulations and later with STS flight simulations. The issue of man's productivity in space hinges on the results of such studies and provides credibility to the SPS concept.



SUPPLY OTV MISSION

- SUPPLIES TO 10 SATELLITES/SORTIE

Maneuver	Longitude Change (Deg)	Time (Days	Payload (MT)
Base to 1st	5	5	2050
Between ea.	2	1	1850 (avg)
Return to Base	25	5	1720

- OTV PROP. REQMT = 77 MT
  (CAPACITY = 200 MT)

CREW OTV MISSION

- TRANSPORT MOBILE CREW MODULE to 20 SATELLITES IN 90 DAYS
- MODULE MASS = 287 MT
- OTV PROP REQMT - 32 MT

Figure 31 SPS Maintenance Sortie Transportation Vehicle

F. OTO Advanced Propulsion and Vehicle Concepts

The large impact that the OTO transfer vehicle has on the overall SPS requires the best possible choice be made of the propulsion devices for this mission. While the ion thruster selected in both the Boeing and Rockwell studies is certainly a viable candidate for this task, it is by no means the only available option. Furthermore, the reference ion engine (~ 120 cm/46.8 in diameter, 8,000 sec Isp) is not within the state-of-the-art and will require substantial technological development. The decision for the choice of this engine over other candidates is not assured. In the selection of a propulsion system for the OTOTV, the evaluation of the candidate systems has been inconsistent, with a disproportionate effort being placed on the argon-ion thruster. It is recommended that the evaluation of alternate systems be given a more substantial treatment to account for both near-term applications and long-term potential. The systems to be studied, and compared, need to include MPD thrusters; solid-, gaseous-, and plasma-core nuclear reactors; and the electromagnetic mass driver as well as the argon-ion thruster. Other advanced concepts, such as laser or microwave power transmission for electric propulsion, should also be considered as should dual-mode nuclear/electric and very advanced chemical systems, which may be less conceptually developed but offer considerable potential.

The recommended study should concentrate on the optimum way to accomplish the task of transferring material from LEO to GEO. Operating costs should be included; but for the first round, it may be desirable to discount the engineering and developmental costs. Environmental considerations should be given a high priority in this study to preclude encountering severe problems later on. It is recommended that a comprehensive but relatively short-term (perhaps 1 yr) study be made of the competing advanced propulsion concepts to determine which ones best fulfill the needs of the SPS. At the conclusion of this study, technological assessment and component development should begin on those which prove most promising. This decision should determine which system is most practical for the GBED, which requires high reliability but not necessarily lowest cost, and the system which would best provide low operating cost but may not be available in the time frame seen for the initial power stations.

The following paragraphs briefly describe the candidate advanced-propulsion systems. Each is described with the advantages it offers, disadvantages it may have, the current status of technology, and the required technological program.

1. MPD Thrusters

a. Potential

The MPD thruster offers a highly attractive alternative to the low-thrust devices for OTO transportation. The advantages that the MPD thruster offers include the following:

- High-thrust density (10,000 N/m^2) that allows one MPD thruster system to replace a large number of ion thruster systems while providing an equivalent thrust level
- Potential of reducing LEO-to-GEO transfer times down to several weeks as compared to ion thruster transfer times which are on the order of several months
- Capability for steady state or pulsed operation that permits close impulse bit control for attitude control and station-keeping functions
- Simpler system that offers potentially lower costs
- Capability of operation over a wide range of propellants that permits selection of a working fluid that can provide low costs and minimal

interactions with the environment.

b. Status

The MPD thruster is in a development phase while concurrently being supported by a strong research base. The physics of this type of thruster have been researched extensively over a sizable period of time with a high level of confidence being generated in the results and with no technological barriers identified. The data base for this thruster is therefore extensive and continuously expanding. Under an on-going technological development program receiving support from both NASA and the Air Force, thruster research apparatus has provided inferred steady-state performance data in the neighborhood of 5 mW, which represents a power of interest for SPS applications and does not require an extrapolation to a desired operating power level. Performance goals of 50 per cent at 3,750 sec with argon has been established for the thruster. Recent results (40 per cent at 1,500 sec) from the research effort suggest these goals may be conservative and that performance somewhat in excess of these goals may be expected.

Major areas that are currently being addressed in the existing development program include direct measurements of thruster performance and erosion rates. The performance measurements will be undertaken in the near future. Specially designed fiberglass facilities, which provide minimum interaction between the exhaust plume and the vacuum tank walls, have been installed in a new electric propulsion laboratory at Princeton. A thruster and thrust stand have been designed, fabricated, and checked out. Installation and check out of the test set-up within the vacuum tank will occur within two months. After a shakedown phase, verification of the thruster performance data will commence. This thrust stand will also represent a powerful tool for the evaluation of changes in thruster geometry.

Tests are also underway to establish erosion-measuring techniques. Erosion rates of operating thrusters will begin to accumulate as pulsed thruster operation at a high repetition rate can be established. Although possible in space, steady state operation in a laboratory is precluded by the high propellant throughput and low environmental pressures required. Efforts are underway, both at Princeton and JPL, to provide high repetition-rate thruster operation. A test facility to provide a high repetition rate has been designed and is expected to be in place at JPL in about one year. Erosion-rate indications will begin to accumulate at that time.

The thruster system is presently in a study phase with some experimental experience with inductive and capacitive energy storage for pulsed operation. A completely steady-state thruster system required for the SPS application has not been studied.

c. Needs

The needs represented here below require an augmentation of the present baseline MPD development program:

- System studies for SPS applications
- Development of MPD thrust system components
- Flight experiment demonstrating steady-state 5-mW operation
- Thruster interactions study for multiple thruster operation
- Augmentation of the thruster development effort for thruster optimization and lifetime demonstration tests
- System demonstration tests

2. Nuclear Electric OTV

A solid-core, nuclear-electric OTV concept is shown in Figure 32. This advanced concept has shown economical transport performance in previous studies and should continue to be studied as the SPS concepts evolve.

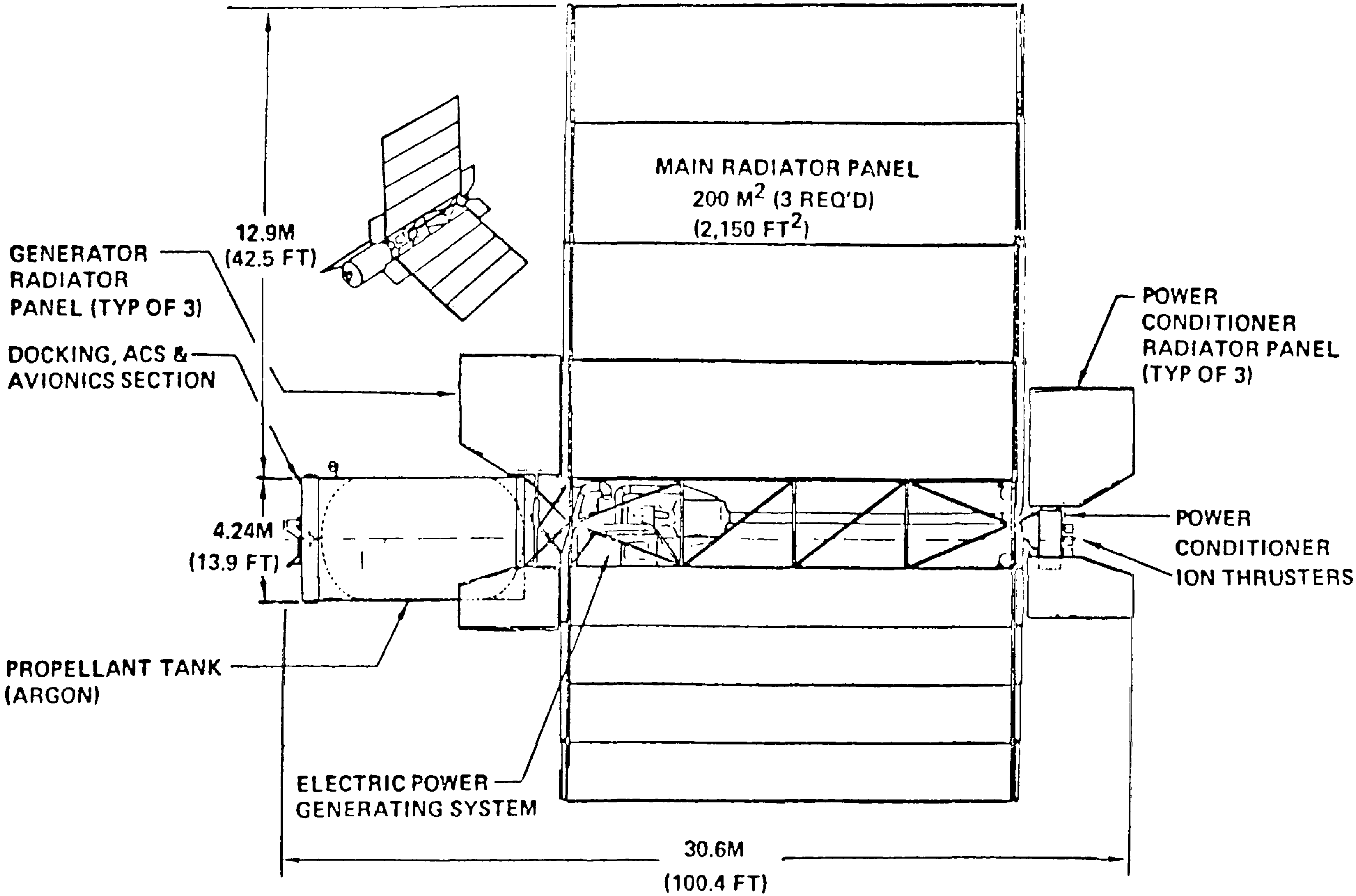


Figure 32 Nuclear Electric OTV - Boeing

3. Gas-Core Reactor OTV

A nuclear-reactor heat source was considered as an alternative to the solar array to power the OTVs. The gas-core reactor was studied as the concept most adaptable to this mission and is presented in Figure 33.

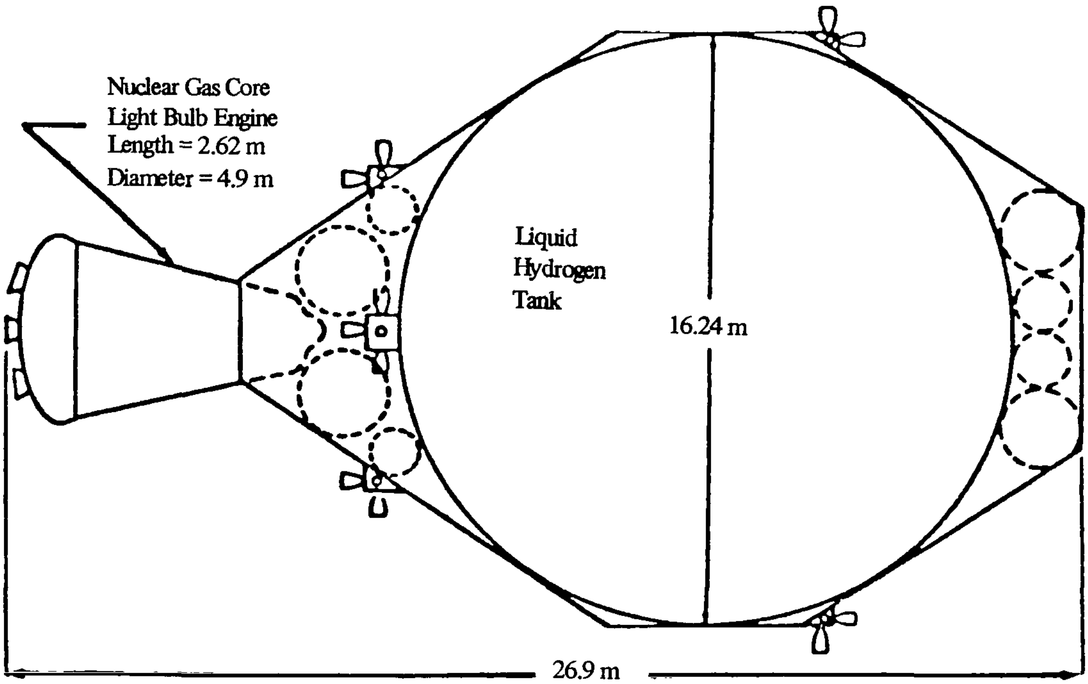
The specific impulse of a nuclear propulsion system is intermediate between that of chemical systems and electrical propulsion systems as indicated below:

- LO_2/LH_2 ~ 470 sec
- Nuclear rocket ~ 2,000 sec
- Electric rocket propulsion ~ 6 to 8,000 sec

Mass in orbit, hence cost, can be expected to be less with higher Isp.

Neutron and X-ray radiation shielding is required for reactor usage in proximity to personnel. This consideration would seriously limit the flexibility with which such a vehicle could be used. Shielding is heavy and shielding design is a difficult problem. After-heat disposal during reactor shut-down is also an important consideration. Unshielded reactors, on the other hand, would require remote handling so that malfunction repair and maintenance in space could be expected to be very difficult; however, it warrants further consideration.

The basic concept of the gas-core reactor relies on the use of thermal radiant energy transfer from a high temperature (~80,000°K) radiating fissioning uranium plasma to a submicron tungsten particle-seeded hydrogen propellant stream. The plasma is vortex-confined by a cool nonabsorbing buffer gas. In one of the several gas-core reactor concepts which have been conceived, the fuel and buffer gas flows are separated from the propellant stream in the core by a transparent wall which allows containment of the fuel within a closed-loop circuit.



Nuclear Gas Core Reactor OTV Mass Summary

Energy Source	Uranium - 233 dioxide
Propellants	LH_2
Specific Impulse	2080 to 2425 s
Thrust	445 to 1780 kN
Engine Mass	42,000 to 91,000 kg
Pressures	271 kN/m^2 (operating)
	384 kN/m^2 (maximum)

Stage Element	Mass, kg
Structures and Mechanisms	18,780
Main Propulsion System	56,800
Auxiliary Propulsion	600
Avionics	260
Electric Power	480
Thermal Control	1,220
Growth Allowance (15%)	11,730
Dry Mass	89,920
Auxiliary Propellants and Fluids	2,000
Total Inert Mass	91,920
Mainstage Propellants LM_2	124,290
OTV Total Mass	216,210

Figure 33 Nuclear Gas Core Reactor OTV – Rockwell

The fuel would be processed for subsequent reinjection into the core region. Propellant exit temperatures in the range of $4{,}000^{\circ}$ K to $6{,}700^{\circ}$ K are predicted for the previous range of fuel-radiating temperatures. Corresponding specific impulse in the range of 1,000 sec to 1,900 sec and thrust-to-weight ratios of 0.3 to 1.3 have been estimated for engine powers of 600 mW to 4,600 mW. (Engine mass without propellant is 39,000 kg or 85,800 lbm.)

The gas-core reactor engine offers the combination of high thrust and moderate specific impulse with the result that rapid LEO-to-GEO trips can be made. Thus, perhaps as few as one vehicle would be required, consequently reducing mass in LEO. However, it must be realized that crew shielding (shallow shielding) must be incorporated that, depending on the safety considerations, will add to the engine basic weight. An assessment must also be made of potential upper atmospheric pollution.

The technology development for the gas-core reactor would probably be longer than electric propulsion devices, but the high thrust, high specific impulse combination may make the gas-core reactor a promising candidate for use in applications beyond the initial deployments.

Although the gas-core reactor requires advanced development of several disciplines, numerous "proof-of-principles" experiments have been conducted over the past 15 years. For instance, a seeded flowing gas stream (simulating the propellant) has been heated by radiation from a dense plasma to temperatures exceeding $4{,}000^{\circ}$ K. A radiating plasma (equivalent black-body temperature of $6{,}000^{\circ}$ K) consisting of argon and UF_6 has been successfully contained within a container of cooled fused silica without causing coating of the walls and transmitting over 90 per cent of the source radiation through the walls. A system was developed that permitted separation of the uranium from the argon and demonstrated that recirculation of the UF_6 was indeed feasible.

While these experiments have been on a small scale relative to that required for the nuclear light bulb engine, they do demonstrate that much of the technological "know-how" necessary has been developed.

4. Mass Drivers

The mass driver reaction engine (MDRE) shown in Figure 34 and other electromagnetic accelerators such as the rail gun provide promising alternatives to electric propulsion for LEO to GEO cargo-transfer missions. The MDRE is capable of accelerating its reaction mass to 1,000 G to 10,000 G and has a high efficiency of 70 per cent to 96 per cent, which permits extremely high performance. Thrust is produced by using electromagnetic forces to accelerate a reaction mass to high exhaust velocities (10 km/sec to 30 km/sec [6.2 mi/sec to 18.6 mi/sec]). In the mass driver, reaction mass is carried in a superconducting bucket which is accelerated to the desired exhaust velocity. This reaction mass is then released and expelled from the mass driver while the empty bucket is decelerated and returned for refilling with a new reaction mass. The mass driver is a linear synchronous motor and is based on the well-proven technology of electric motors. The superconducting bucket is magnetically supported by the guide strips lining the mass driver coils and therefore has no physical contact (i.e., no friction or wear).

The MDRE has the unique feature of being able to use any material for a reaction mass and thus eliminates the need for specialized propellants. Because of this fact the reaction mass is not ionized and will be in a retrograde escape orbit, thereby eliminating the possibility of harmful effects to the ionosphere. For safety reasons, liquid oxygen or other similar material can be used as a reaction mass, rather than a solid pelletized reaction mass, to eliminate a

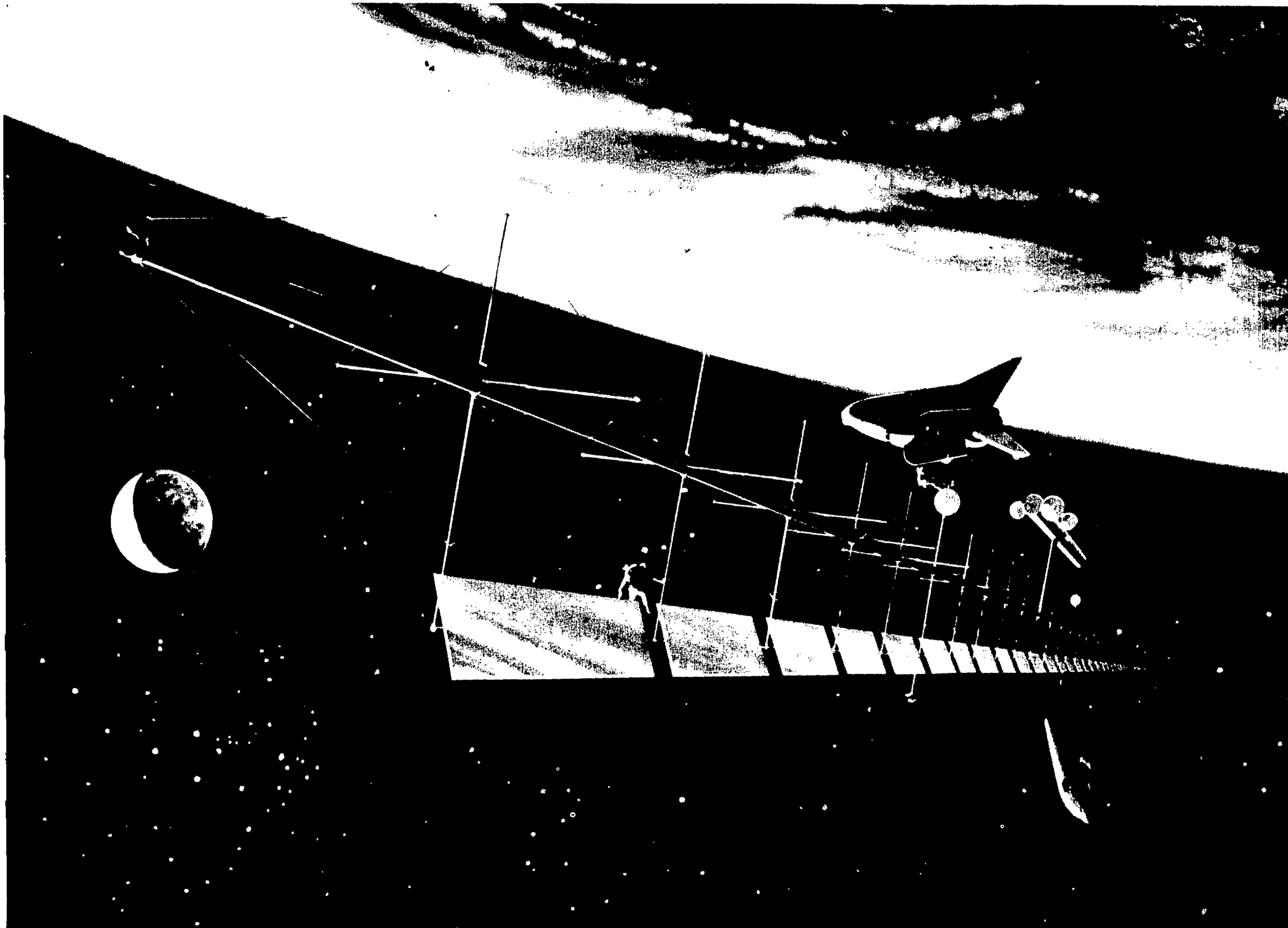


Figure 34 Mass Driver Reaction Engine Concept

potential hazard to other orbiting vehicles. The MDRE also has the feature of variable Isp which is easily chosen by the value of acceleration and length of the MDRE. Since the MDRE operates in the 1,000 sec to 3,000 sec Isp range with a higher thrust than that of the ion engine, shorter trip times (~90 days) are possible with little sacrifice in payload delivered per mission as compared to the 180- to 210-day missions for argon engines. This makes it possible to reduce the fleet size by a factor of 2 to 3 over the ion engine EOTV and reduces initial lift into LEO.

5. Other Concepts

In addition to the specific propulsion alternatives that have been discussed, there is one other propulsion system/vehicle concept that may merit serious attention. This transportation concept uses a chemical or electric propulsion system and a remote power supply with energy transmitted to the OTV by microwaves or lasers. In the SPS scenario, where the cargo OTV makes many trips between LEO and GEO, the removal of the power supply from the OTV and subsequent decrease in its mass may significantly decrease the transportation trip time. The transmission of the power by microwaves or lasers would surely be made feasible by the large development put into the SPS power transmission and conversion systems. A remote power supply for an electric propulsion OTV would eliminate the anticipated problem of degradation of the onboard exposed solar array during long transfer spirals through the Van Allen belts.

IV. OTHER MAJOR CONSIDERATIONS

A. Professional and Industrial Capabilities

The universities and technical institutes need to contribute to an SPS program in three important ways. First, they can identify areas of fundamental science and engineering which underlie the SPS in general and specific. Second, they can assist by performing research on topics of basic and applied science needed to undergird the technical and more applied tasks of industry and government. Third, they alone can provide the requisite flow of educated young people who will necessarily step into leadership roles of the program in the critical next decades.

For any of these to occur in a healthy and productive fashion requires deliberate attention from cognizant federal agencies and other interested parties. Without their close attention, financial support, technical liaison, and mutual concern, any academic effort will be sterile and the entire program will suffer. This history of past collaborations between the government agencies and the academic community bears out these generalized assertions. In those fields, especially in aerospace and associated engineering and scientific disciplines, where the pattern of sponsored research in universities has been established in the past and is generally representative of the industrial and governmental interest in an area, the reservoir of basic knowledge and the flow of creative personnel have been sustained, and the overall enterprise has been the more efficient. Where such academic support patterns have been inadequate or poorly composed, the field as a whole has tended to stumble, stagnate and overrun its supply of basic data and creative people. The specific mechanisms for stimulating the academic sector are well tried and would be equally effective in context of the SPS. They are as follows:

- A substantial support of basic research in specific areas appropriate to the SPS program to the university prerogatives and to the interests and capabilities of the faculty and students
- A careful selection of major grants to allow the most qualified institutions to establish centers of excellence in particular fields by acquiring suitable capital, research facilities, and then developing incisive academic programs in those fields
- A program of undergraduate scholarship, graduate fellowships and assistantships to encourage the best engineering and science students to undertake studies in these fields
- Involvement of productive and articulate faculty in program planning and assessment processes by membership on advisory groups and private consultation arrangements

With the above elements functional, an ambiance of relevance and excitement develops in the academic community which seems to invigorate the professional sector and enhances enthusiasm for the program.

The aerospace industrial complex today possesses the fundamental skills, knowledge, and many of the facilities needed to accomplish the SPS program. These capabilities include conceptual design; systems engineering; experimental, development and qualification testing and manufacturing; as well as ground and flight checkout and operations.

Only a fraction of this total capability is currently directed to advanced activities of the space program. While Shuttle and some spacecraft programs are related to the SPS program, broad research and technology, and direct SPS tasks are insufficiently funded to maintain a satisfactory industrial base during the coming decade. This base, then, will not be available when needed. This is especially true of SPS space transportation in the areas of advanced vehicle technology, propulsion, and operation in space. Unless specifically provided for, capabilities in these and other areas will be dissipated before SPS funding rises above the threshold level.

Industry needs full insight into the SPS program in order to relate its requirements to business projections. They must be able to identify the SPS-unique requirements for special skills and will need government support for technological work and special facilities as well as access to government facilities. Industry access to SPS studies, program asssessment activities and policy issues can go a long way toward preparing the aerospace sector for a program of the magnitude of SPS, so it can plan activities to match its expertise. Reviews of on-going programs that relate current capabilities to future SPS should also be made. Finally, an informed and involved industry can provide positive support to the SPS program through meeting with the decision makers and through support of congressional hearings.

Professional societies offer a capability that should be utilized in support of the SPS program. They should have access to study findings and recommendations, should be invited to participate in program assessments, and should be encouraged to promote symposia. Additionally, they should write position papers and inform members of Congress.

In conclusion, the capabilities of the university and industrial communities are needed to support and participate in the SPS program. They have a large stake in defining, justifying, supporting and performing their roles in the ultimate success of such a vast undertaking. Support of these vital capabilities is necessary so their participation will be available in support of a national commitment to the SPS program.

B. Cost and Decision-Making

The viability of the SPS concept must be assessed on the ability to deliver competitive electrical power at the utility bus bar compared to other options. An estimated cost of approximately $92 billion (1979) was derived from cost-estimating relationships and includes all research, technology and development and production of the first 5-GWe SPS. As shown in Figure 35a, transportation represents ~45 per cent of the R,T&D costs and ~ 25 per cent of the initial 5 GWe SPS, shown on Figure 35b. Of the transportation costs approximately half is for the HLLV. The recurring transportation, as shown in Figure 35c, is dominated by ESLEO transport which represents ~60 to 70 per cent of the costs and the recurring costs from LEO to GEO is ~ 20 per cent. The uncertainty in ESLEO transportation costs is significantly less than from LEO to GEO. Therefore, reduction in cost-risks and technology enhancement must be addressed to critical areas of the latter. In the former, low-cost operations are the key to providing competitive SPS energy for HLLV and later SSTO vehicles.

Because the final SPS must be cost-competitive with other energy systems, reduction of cost and cost uncertainty must be an objective of much of the R, T&D work. SPS space transportation has already been identified as a major cost element so that reduction of transport cost uncertainty deserves a substantial share of next-phase resources. Costs and decision-making conclusions include the following:

- ESLEO - Vehicle cost uncertainty can be reduced by specific R,T&D. Operational costs uncertainty can best be reduced by STS operations experience

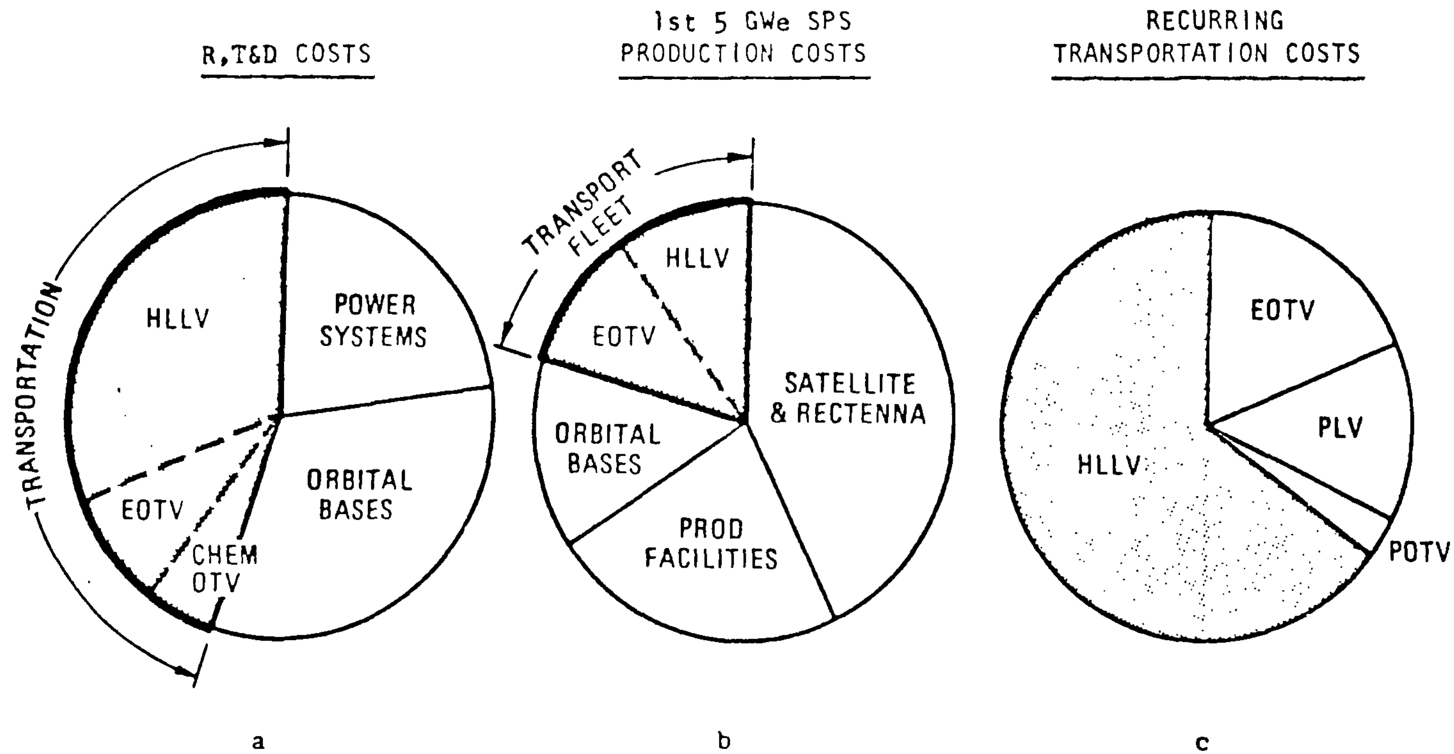


Figure 35 SPS Space Transportation Costs

- LEO Operations - Man-hours are a significant cost driver; R,T&D and STS operations are needed to improve estimates of man-hours for SPS transport
- LEO to GEO (Electric Transfer Vehicle Assumed) - Long life of photovoltaic arrays in trapped radiation environment is needed; ground-based R,T&D can significantly reduce the associated cost uncertainty. Ion-thruster development should include in-space testing; otherwise, major cost uncertainties will remain
- GEO Operations - Personnel transfer assumes using chemical rockets and GEO activity entails major cost uncertainties. Also, extra costs are needed for safety provisions; R,T&D is necessary but not sufficient to remove these uncertainties

At this early stage in a major program such as SPS, arguments based on comparing cost estimates for different technical approaches cannot yield valid decisions. The only way to get valid decisions based on cost-related choices is to perform continuous studies and analyses. Carefully selected research and technology work will assist in selection and in reducing cost uncertainties.

Cost and decision items that require R,T&D to reduce cost uncertainty are given below:

- ESLEO

  Composite structure

  Reusable cryogenic insulation

  Reusable thermal protection systems

  Long-life engines (many starts, few man-hours of maintenance between flights)

  Self-test technology to reduce checkout man-hours (on ground and in orbit)

Facility/vehicle design integration, including pre-launch cargo processing and design for maintenace

- LEO TO GEO

  Solar photovoltaic system degradation and annealing

  Ion thrust design and qualification

  Large, low-density structures

  Electric power processors

  Automated rendezvous and docking of large, flexible items

  On-orbit servicing

  Man-rating, risk assessment, safety hazard protection, rescue

  Guidance and attitude control (interaction with large flexible structures)

Reasons for priority selection of the above list from the longer lists presented to the workshop are as follows:

- Propulsion efficiency (hence, mass of inert components and of propellant brought up from Earth) of the LEO-to-GEO vehicles has impact on other transport elements. Thus, the size and the cost uncertainties of all vehicles are magnified, and substantial efforts to reduce these uncertainties are justified

- Heavy lift (and, to a lesser degree, personnel transport) from ESLEO for SPS will involve vehicles of unprecedented size and number of flights. It is much too early to select a vehicle from among the practical possiblities; a design reference is useful for study but both options must be vigorously pursued. Composite structures, instead of metal tanks, are but one example of new technologies whose cost impact could be very favorable but is today unknown

- Personnel and operational costs in all phases are a major cost element with uncertainties reducible to some extent by study, analysis and simulation, but substantial cost uncertainties will remain that can only be reduced by flight experience. STS operations might well be considered as a source of data for the next phase of SPS

- Safety criteria for personnel and redundancy of space vehicles for mass transport have not been explicitly addressed in technical planning. While percentage reserves appear to have been applied to individual designs for vehicles and in the number of vehicles hypothesized for total fleets, explicit treatment of accidents has not been undertaken. The next phase of SPS must assess the risks which can be accepted and determine the technical requirements and costs to provide STS redundancy (including design requirements for individual vehicles and extra vehicles) to reduce unacceptable risks

- The present plan addresses hardware technology at the component and subsystem level to reduce cost and uncertainty. Transport operations, their requirements and costs for ground, LEO, GEO and intra-orbital activities are less well known and do not appear to be addressed other than in terms of the most elementary construction and manipulation capabilities. This especially applies to emergencies and recovery therefrom. While the next phase program probably cannot undertake significant efforts in this area, it should conduct studies to provide detailed estimates of these requirements to define future plans and programs for a technology verification phase

There are other significant issues that will remain open despite analysis. One of these is the effect of all engine exhausts on the atmosphere, ionosphere, and plasma around the Earth. Because these environmental effects have a public interest element as well as a technical/cost element, extra and early attention to them is needed. The technological readiness date is an important factor in the selection of technologies to be pursued and the specific form of the R,T&D program. The rationale for the selection of specific dates appears not to be fully appreciated nor is the effect of varying the date understood.

The basic concept of technological readiness needs clarification. Can it be defined in terms of the range of uncertainty and the form of the uncertainty of both performance and cost? If this can be achieved, the user of the technology can than make the decision as to when the technology is ready. For SPS, in distinction to certain other space efforts, this question may prove crucial.

Since the operational system envisions construction of two 5-GWe satellites per year for a 30 yr-time span and eventual maintenance of 60 SPS satellites at GSO, ground and space transportation operations represent major cost and manpower uncertainties. HLLV must be turned around in 4 to 5 days. Short launch pad operations are necessary.

Airline-type operations using on-board failure prediction and autonomous operational sequencing will be required. In addition, airline cargo-type processing must be achieved, and computer-based cargo manifesting is essential to maximize payload mass per flight.

At LEO, base maintenance and repair will be required for the EOTV. Logistics and depot maintenance must be addressed to minimize manpower.

If transportation system operations and maintenance can be streamlined to levels similar to other mature transportation systems, significant cost reductions might be achieved. This would require a design that is structured for ease of maintenance and one not requiring refurbishment after each flight.

Finally, it is necessary to develop and utilize overall program evaluation and formulation tools that do not explicitly consider performance and cost uncertainties. It is also necessary to establish the value of the R&D projects and program in terms of the information to be obtained in the form of performance and cost uncertainty reductions in each program phase. In other words, the R,T&D program for SPS should be considered as one aimed at the sequential resolution of uncertainty through R,T&D. This is elaborated in Appendix A. It is strongly urged that these tools be developed and utilized in the continuing formulation of the R,T&D program of the SPS.

Cost is defined as the summation of price times quantity where the summation is across all components and encompasses labor, material, and capital. It should be noted that even if all these quantities were known precisely, the cost would still be uncertain by a possibly large factor because of price uncertainties (the cost-effectiveness ratios yield values of price but require significant assessments and do not reflect changes in the "world" relative to the "historic" world). Many of the prices may be correlated and thus averaging of higher and lower outcomes may not result. Because of performance uncertainties, the quantities (ranging from number of solar cells to number of flights) required will be uncertain. The net result is that the SPS cost (and its transportation component) will also be uncertain. In fact, the cost must be considered as being a random variable with a large standard deviation. At present, it is not realistic to state a cost for the SPS. The only thing that can be stated with a high level of confidence is that the SPS will not be built unless it is economically competitive. This implies that there must be a high probability that the present value of the SPS will be equal to or less than the present value of the cost of other alternatives.

Concluded are the following:

- Since cost is a random variable with a long standard deviation, a single specific value of cost should not be used for decision-making
- It is inappropriate at this time to consider a decision to build (or not to build) an SPS
- It is appropriate to consider the next phase in a multi-phase program. The decision should only be to commit (or not to commit) to the next phase
- An important element of each phase is its impact on the probability of commercialization
- The SPS program should be considered aimed at the sequential resolution of uncertainty through R&D
- It is too early to base major SPS transportation decisions on the comparative costs of identified technical options. Cost uncertainties must be reduced by transport system R,T&D
- In addition to present transportation approaches, some very advanced technological options should be given a reasonable amount of attention since, if successful, they could result in a major improvement of SPS cost-effectiveness. Examples are SSTO vehicles and MPD thrusters

- Personnel costs in all phases will be significant, and current operational cost uncertainty is not reducible by analysis; simulation and actual flight experience are needed to support SPS design decisions and cost estimates

C. SPS Transportation System Funding and Timing

A GBED program has been defined for the period of 1981-1986. It incorporates the major technological areas and focuses the efforts required to resolve key issues that would affect a decision to proceed with an SPS technology-verification phase; to support societal and other nontechnical assessments; to define preferred system concepts; and to define plans for a post-GBED phase. This program is to provide a logical stepping stone toward the initial visibility needed for an evolutionary phased program defintion in support of SPS program needs. A more detailed analysis of the GBED items for the transportation area shows, however, that assumptions have been made which tend to prematurely close out technical program options and constrain technological requirements. These assumptions are responsible for curtailing the level of funding effort by a factor of between 2 and 5 below that deemed productive for some areas. For example, the GBED reliance on other NASA programs to provide timely technological answers is probably misplaced; and current configuration assumptions force an underestimate of augmentation required. The GBED program for transportation alone needs a funding of $100 million for the 5-yr period of 1981-1986.

Figure 36 shows the present SPS scenario in terms of annual mass that must be lifted into LEO versus calendar years to 2040. Increases of five and ten per cent over the STS baseline capability are also indicated so that the scenario is shown to be rather unrealistically loaded in the 1990s unless a

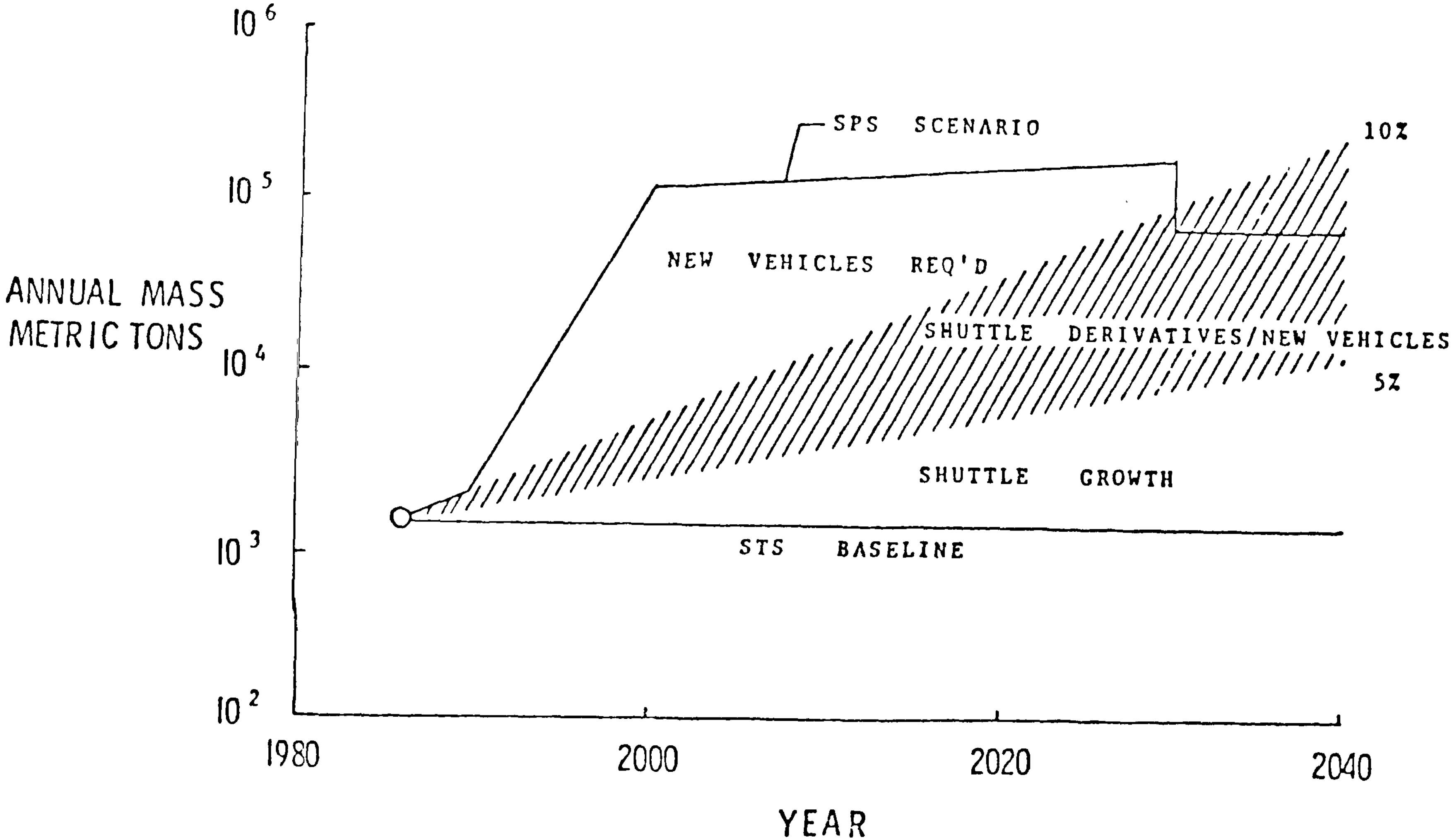


Figure 36 NASA Space Transporation Technology Planning Scenario
(Assume Shuttle STS Baseline: 30 Tons Per Flight, 50 Flights Per Year)

major national commitment were to be made by the mid-1980s. This appears to be highly unlikely and should not be the sole or primary basis for overall planning

Figure 37 shows the present NASA space transportation planning for SPS in terms of primary reference vehicles, and it is even more clearly shown that the technical readiness for SPS can surely not be realized while even the moderate (10 per cent) growth would be very ambitious. This would indicate that substantial revision in SPS timing should be contemplated.

D. Atmospheric Effects of the SPS

Since the atmosphere from the ground to GEO will be subject to rocket exhaust, it is expected that all regions will be perturbed by its effluents to some extent. The main reason for concern arises from both the size of the vehicles (their effluent-emission rate) and their launch frequency. In the troposhere, the ground clouds formed during launch of the HLLV, and to a lesser extent, the SSTO could give rise to some local weather modifications and effects on the quality of the air. Weather modifications can result from two sources. First, injection of the local atmosphere and possible changes in local circulation and numbers of clouds. Second, the injection of cloud condensation and ice nuclei can at a microscale affect the physical processes of clouds, a process that could ultimately influence cloud formation, precipitation, and possibly haze or fog formation. These effects arise from the entrainment of surface debris and dust, after-burning of exhaust products in the ambient air, and injection of fuel impurities. Use of fuels such as RP may lead to concentration of sulfer dioxide and other pollutants that would lead to local air-pollution problems. After-burning of even clean fuels

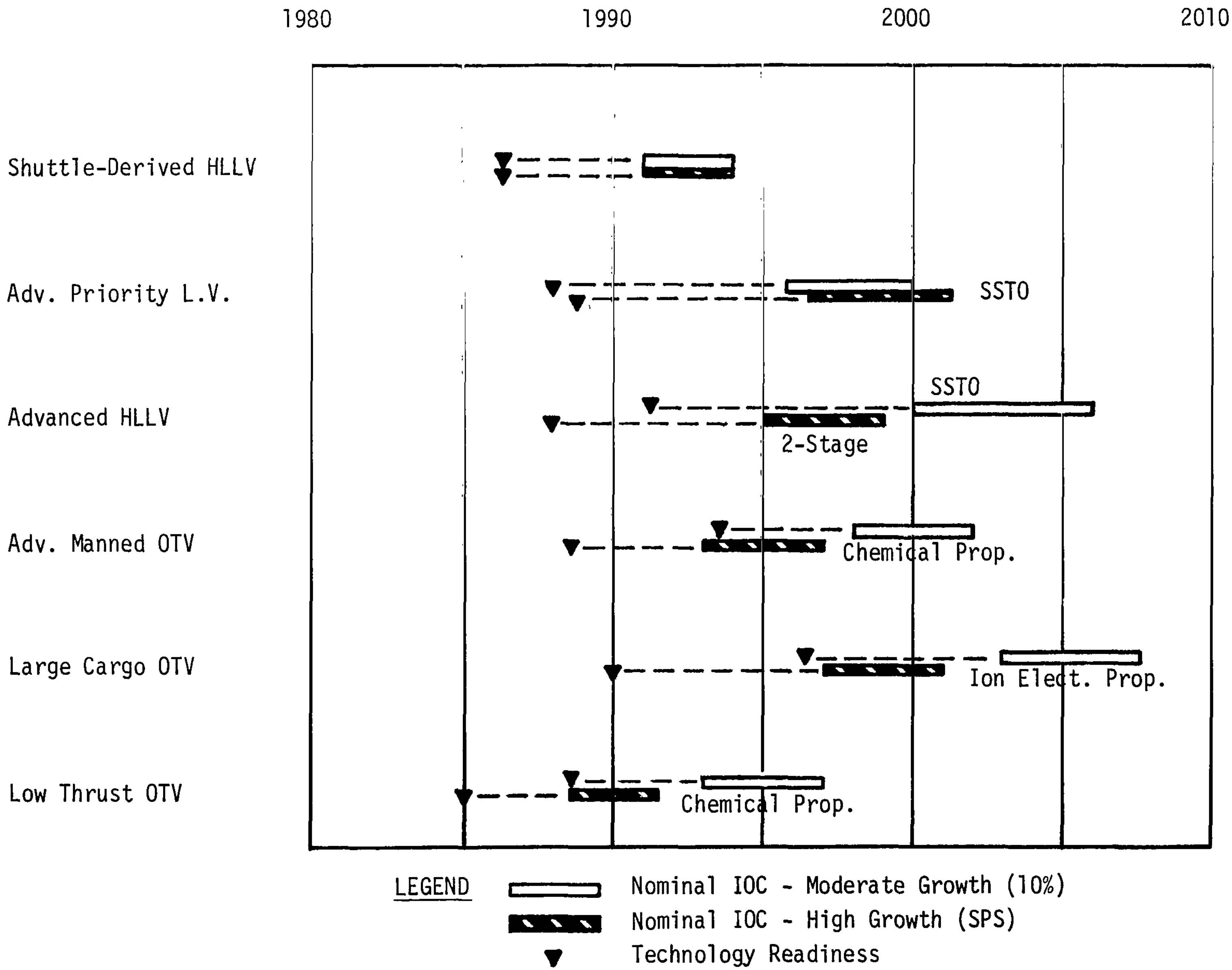


Figure 37 NASA Space Transportation Planning - Reference Vehicles

may result in levels of oxides of nitrogen that could lead to air-pollution problems, especially if the Enviromental Protection Agency sets a fairly low NO_x standard. Emissions of sulfur and nitrogen compounds could also contribute to acid rain, but the levels are not expected to be significant.

Higher in the atmosphere, no significant stratospheric impacts from the use of CH_4 or H_2 fuels are anticipated, since the exhaust products are indistinguishable from ambient constituents present in substantially higher concentrations. However, at greater heights, the atmosphere becomes increasingly rarified and consequently more susceptible to large-scale perturbations. By the same token, our understanding of such perturbations, as well as the state of the upper atmosphere, becomes less clear at very high altitudes. Scientists are currently identifying what effects could occur but are limited in their ability to predict what will occur when the SPS is a reality. Effects that could arise in the mesosphere include chemical composition and dynamic changes brought about by the addition of water vapor especially above 70 km to 80 km (42 mi to 48 mi). This water vapor could also contribute to the formation of ice-crystal clouds. The rate and location of water vapor injections will also influence ionization levels in all regions of the ionosphere from the D-region through the F-region. Injections of rocket exhaust directly in the F-region will produce dramatic reductions in local plasma density and therefore influence radio-wave propagation and, perhaps, other physical phenomena. Avoiding injections will mitigate processes (not fully understood at present) which will remove at least some of the exhaust products injected both above and below into the F-region. Of greatest concern are the long-term, chronic effects in the ionosphere of once or twice daily injections of water and hydrogen molecules over 30 or more years.

Above the F-region, the principal exhaust products will be argon (AR^+) ions from EOTV flights and H_2O and H_2 from POTV flights. Effects may arise both from the next accumulation of H atoms and the energy associated with these injections combined with that of HLLV and PLV circularization and de-orbit burns. This addition of thermal energy and mass may lead to changes in temperature and density that could influence satellite drag and the stability of the Van Allen radiation belts. Interactions of these exhaust products with ambient neutrals and plasma will give rise to background levels of airglow which may interfere with remote sensing. Also, the thermal or radiation transfer properties of the thermosphere may be altered by the addition of large amounts of water vapor.

Finally, the injection of AR^+ ion beams, containing both mass and energy large in magnitude compared with that naturally present in the plasma and the magnetosphere, may significantly alter both the composition and structure of this most rarified region of the satellite environment. In addition to possible alternations of the radiation doses received by vehicles in the radiation belts, such injections may alter the intensity and frequency of high-energy particle precipitation events at mid-to-high latitudes. Electromagnetic wave propagation could be influenced by plasma instabilities triggered by the AR^+ ion injections. Finally, some consideration has been given to the influence that SPS injections in the magnetosphere may have on the solar-weather effect. A related effect would be changes that may result from AR^+ injections on the manner in which the magnetosphere responds to changes in the solar wind and magnetic storms. Large ionospheric auroral currents associated with such storms have been observed to cause current surges and trips of circuit breakers in long-line telephone systems and power transmission lines in northern latitudes. Alteration of the latitude at which these events occur could make their impacts on populated areas more significant.

While present knowledge does not permit a definitive statement regarding mitigating strategies, some suggestions deserve future attention. These include the use of alternative ions such as hydrogen or the use of neutrals instead of ions. Trajectory shaping, thrust scheduling, and selection of type of propellant on the basis of altitude should also be considered.

Data are needed on the concentrations and fluctuations of upper-atmospheric constituents and on perturbations caused by rocket effluents. Definitive data are needed on effects of AR^+ and chemical injections above 200 km (120 mi). The GBED program should include opportunity to design experiments that could combine technology testing with atmospheric effects studies. Unless some experimental data are obtained in GBED, it will be difficult to substantially reduce uncertainties especially regarding effects above 500 km (300 mi). It is recommended that small-scale space experiments be conducted during the GBED program to stimulate the refinement of theoretical modeling technique and planning of larger-scale, more sophisticated experiments. In addition, GBED time-frame experiments will provide a basis for development and refinement of both ground-based and airborne diagnostic instrumentation.

V. CONCLUSIONS AND RECOMMENDATIONS

The primary conculsion is that the SPS space transportation studies so far conducted are well done and give confidence that with further systems analyses and substantial R,T&D existing technological concepts could provide a basis for the SPS, although timing and costs are at present highly uncertain. Advanced space transportation concepts that appear to offer greatly improved operations and reduced costs should receive emphasis in the next phase of SPS.

While it is too soon to commit to the development of specific vehicles (except a low-thrust OTV), additional analyses, and R&T (including ground and space testing) can reduce uncertainties within the decade of the 1980s. Whereas the timing of the present SPS program is clearly unrealistic with respect to space transportation, time is available to plan a proper program and establish a firm foundation. SPS should be considered basically as a global energy source of great potential that may contribute to meeting the Earth's future power needs.

The ESLEO transport requirement of SPS is a great challenge in scale and character of operations. However, an evolutionary series of heavy-lift and personnnel-launch vehicles with chemical rocket propulsion can be targeted realistically to move heavy masses into LEO for $30(1979)/kg by the year 2000. More advanced propulsion technology and vehicles may make $15(1979)kg a goal in the foreseeable future.

Although LEO to GEO (including intra-orbit transport) with electric orbital transport vehicles appears to be promising for massive cargoes, this requirement will probably need a variety of vehicles including chemical rocket stages and much further analysis and technology attention, especially the advanced concepts.

Based on its promise as a major global energy source, it is strongly recommended that SPS be carried into a next phase with approximately an order of magnitude increase in funding.

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APPENDIX A

SPS Research, Development and Demonstration (R,D&D)* Program Evaluation and Formulation

It is assumed that the objective of the present SPS program is the sequential resolution of uncertainty through research and development. The existence of performance and cost uncertainties leads to risk associated with continued SPS-related investments. The risk can be viewed in terms of the likelihood that energy from the SPS will cost more than the energy from other technologies. It is assumed that this objective will have to be accomplished within budgetary constraints and that all desired projects cannot be undertaken simultaneously. It is therefore necessary to be in a position continuously to evaluate projects and select the mix of R,D&D projects that maximizes benefits from limited resources.

It is important to observe that an R,D&D project yields a tangible product of economic value only upon complete development of a technology and only upon commercialization of it. In general, only the commercialization phase of every R,D&D results in direct benefits to society. There are, however, indirect benefits of energy R,D&D such as price shifts on nonrenewable resources brought about by expectations deriving from the R,D&D activities. The earlier phases of the R,D&D can be used in the decision-making process to continue the project, to change it, or to terminate it. The economic value of the earlier phases of the R,D&D process is thus the value of the information which

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*Research, development and demonstration (R,D&D) used in this appendix is the approximate equivalent of research, technology and development (R,T&D) used in the text.

they produce. It is this value which one should compare to the cost of performing an R,D&D subproject when making the decision to fund it, and not to the economic value which is obtained by commercialization of the technology that might ultimately be developed as a result of the R,D&D project.*

The information becomes valuable when it is used in a decision-making process by increasing the probability of choosing the best alternative. For example, consider the decision to wager on the outcome of flipping a coin. Most would agree that a bet of $1 to 10 cents that the coin will land heads is not a good wager to enter (an expected-value decision-maker clearly would not make this wager). But it would obviously be a good wager if it could somehow be known in advance that the coin would land on heads.** In this case, the value of the information that the coin would land heads is 10 cents, the amount to be gained from its use. On the other hand, the value of the information that the coin would land tails is zero because, since the bet is on heads, the decision to not enter the wager is unchanged by this information. Before knowledge of the outcome of the flip is obtained, one can only know that there is a 50-50 chance that the coin will land on heads. Thus, before obtaining this information one can only say that there

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*Although the value of information produced by an R,D&D subproject is a function of the economic value obtained by commercialization, they are significantly different quantities.

**This example is, of course, somewhat artificial since no one would wager against a sure thing; and since, if the outcome of the flip were really known in advance of the flip, the flip would be superfluous.

is a 50 per cent chance that the information will be worth 10 cents and a 50 per cent chance that it will be worth nothing; hence, the information has an expected value of 5 cents (0.5 x 10 cents + 0.5 x 0 cents). An expected-value decision-maker should be willing to pay up to 5 cents to obtain this information prior to entering the wager.

It is not easy to see how one could obtain knowledge of the outcome of a flip of a coin in advance. Nonetheless, it does seem intuitive that even imperfect information could have some value. For example, suppose the coin were selected at random from a bucket of coins, some of which were fair coins and some of which were weighted to land heads a high fraction of the time, perhaps 95 per cent. It would clearly be of value to know which type of coin was chosen and this could be determined easily by "test flipping" the coin.

Energy R,D&D is a similar process. Each R,D&D phase is a process of "buying" information on the ultimate outcome of the overall project or program. If this information makes clear the fact that the technology cannot be developed to a point of successful commercialization, the project can be terminated, thus preventing the expenditure of additional funds. If, on the other hand, the project is continued, it will be with the confidence gained from having eliminated some of the uncertainty that existed at the start.

A major difference between flipping the coin and energy R,D&D lies in the fact that the latter involves the purchase of information from a sequence of R,D&D projects that has, in the past, caused analytical complications which have prevented proper analysis of more practical problems. Recently developed techniques overcome these complications.*

The problem can be stated as the evaluation of the decision to initiate or continue an energy R,D&D project, or to commit to the next subproject, recognizing and accounting for the following:

- Nearly all R,D&D projects are multiphasic efforts. They consist of a sequential set of subprojects, each of which is funded independently based upon the results of previous subprojects (and, perhaps, upon a set of external variables, such as prices and availability of competing technologies)
- The outcome of an energy R,D&D project (or subproject) cannot be known before completing the project. If it could, there would obviously be no need to do the project.** All that can be known in advance is the range of possible outcomes and the relative likelihood that any particular outcome will occur, compared to any other outcome

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*See "A Energy RD&D Project/Program Evaluation Methodology," ECON, Inc. Report No. 79-221-1, April 15, 1979, prepared under DoE Contract No. ER-78-C 05-5863. This section has been abstracted from this report.

** It is sometimes thought that R,D&D is a process of buying technology improvements. It is not. The technology improvements are available options prior to any R,D&D effort. What the R,D&D effort does is to provide the information necessary better to discriminate between the available options. The technology improvement which appears to result from an R,D&D effort actually results from the decision process following the R,D&D effort, in which the better available options are chosen for further consideration. For example, consider a battery test which determines performance as a function of a number of design parameters. Prior to the test, all design options are available alternatives but, since performance cannot be predicted as a function of design option, the better alternatives cannot be discriminated from the worse alternatives. The test provides the information necessary to make the choice between design options, but it is the choice of design option (the decision) that results in a good battery design, not the information gained by the test. Recognition of the role of the decision process in the evolution of a technology through an R,D&D project is key to this methodology of evaluation.

- The economic output of each R,D&D phase is a sequential resolution of the uncertainty that exists at the start. Such an output is information upon which one may choose a future course of action from the set of alternative courses; for example, to continue, to terminate, or to continue in a modified form
- The result of an R,D&D project, if successful, is a commercial technology which, if implemented, yields economic benefit

This statement of the problem is focused on an evaluation of the next increment of an energy R,D&D project as it is only the increment for which a commitment will be made. Since, in general, the economic output of the next increment (or subproject) of the R,D&D project will be information, the problem may be equivalently stated: Evaluate the information to be obtained in the next subproject in an energy R,D&D project. It is implicit that the next subproject is deemed economically desirable if the value of the information which it provides exceeds its cost. Other methodologies which do not explicitly address the net value of the information produced by the next increment of the project will systematically underestimate the value of pursuing the technology. This is true because they do not account for all the alternate courses of action available to the project manager.

To accomplish the above requires that both cost and performance be considered as uncertainty variables described by ranges of uncertainty and the form of the uncertainty. These estimates must be made and without the specific projects. In order to utilize these uncertainty assessments it is necessary to develop the following:

- An engineering system model that interrelates that technical performance of the pertinent subsystems and results in the determination of quantities
- A cost model that forecasts prices based upon specified economic parameters and as per the engineering system model
- A benefit model that uses market parameters and the cost (from the cost model) to obtain benefits

It is only through the use of this technique that R,D&D programs can be formulated that quantitatively consider uncertainty and risk reduction and the value of information. It is strongly urged that these techniques be developed and utilized in the continuing formulation of the R,D&D program of the SPS.

APPENDIX B

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15. Supplementary Notes

16. Abstract

During the several phases of the Satellite Power System (SPS) Concept Definition Study, various transportation system elements were synthesized and evaluated on the basis of their potential to satisfy overall SPS transportation requirements and their sensitivities, interfaces, and impact on the SPS. Additional analyses and investigations were conducted to further define transportation system concepts that will be needed for the developmental and operational phases of an SPS program. To accomplish these objectives, transportation systems such as the Shuttle and its derivatives have been identified; new heavy-lift launch vehicle (HLLV) concepts, cargo and personnel orbital transfer vehicles (COTV and POTV), and intra-orbit transfer vehicle (IOTV) concepts have been evaluated; and, to a limited degree, the program implications of their operations and costs were assessed. The results of these analyses have been integrated into other elements of the overall SPS concept definition studies.

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