NASA Contractor Report 3536

Future Orbital Transfer Vehicle Technology Study

Volume II - Technical Report

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Prepared for
Langley Research Center
under Contract NAS1-16088

NASA
National Aeronautics
and Space Administration

**Scientific and Technical
Information Branch**

1982

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ABBREVIATIONS AND ACRONYMS

AB OTV	aerobraked OTV
ACS	attitude control system
ASE	airborne support equipment
BAC	Boeing Aerospace Company
BITE	built-in test equipment
BOL	beginning of life
C/O	checkout
COTV	chemical OTV
CR	concentration ratio
DDT&E	design, development, test, and evaluation
DOD	Department of Defense
DOE	Department of Energy
EIS	Executive Information System
EOL	end of life
EOTV	electrical orbital transfer vehicle
EPS	electrical power system
EVA	extravehicular activity
FOTV	future orbital transfer vehicle
GaAs	gallium-arsenide (solar cell)
GB OTV	ground-based OTV
GEO	geosynchronous Earth orbit
GPS	global positioning system
IOC	initial operating capability
I_{sp}	specific impulse
IUS	inertial upper stage
IVA	intravehicular activity
LaRC	Langley Research Center
LCC	life cycle cost
LEO	low Earth orbit
LeRC	Lewis Research Center
LRB	liquid rocket booster
MER	man-hour estimating relationships
MLI	multilayer insulation
MMTR	mean missions to repair

MPS	main propulsion system
MSFC	Marshall Space Flight Center
MT	metric ton
NASA	National Aeronautics and Space Administration
OTV	orbital transfer vehicle
PCM	Parametric Cost Model
P/L	payload
PPU	power processing unit
PSMC	Payload and Sequential Mass Calculation (program)
RCS	reaction control system
R&D	research and development
RPS	reusable payload system
R&R	remove and replace
SB OTV	space-based OTV
SDNW	space disposal of nuclear waste
SDV	shuttle-derived vehicle
SEPS	solar electric propulsion system
SOC	Space Operations Center
SPS	solar power satellite
SRB	solid rocket booster
SRU	space removable unit
SSME	Space Shuttle main engine
SSUS	spin-stabilized upper stages
STS	Space Transportation System
t	tonne
$\bar{t}$	shield thickness
TFU	theoretical first unit
TVC	thrust vector control

FOREWORD

The Future Orbital Transfer Vehicle Technology Study, NASA Contract NAS1-16088, was managed by the NASA Langley Research Center (LaRC) and was performed by the Upper Stages and Launch Vehicles Preliminary Design organization of the Boeing Aerospace Company in Seattle, Washington. The NASA Contracting Officer's Representative (COR) was John J. Rehder.

This final report is organized into the following documents:

Volume 1: Executive Summary
Volume 2: Technical Report

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1.0 INTRODUCTION

This section discusses the background leading to the Future Orbital Transfer Vehicle (FOTV) Technology Study, the overall objective, key issues, guidelines, and the approach used in conducting the study.

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

1.1 BACKGROUND

During the last decade, numerous studies have been conducted concerning orbital transfer vehicles (OTV). These studies have considered a wide range of technologies and mission destinations, including various Earth orbits and lunar, planetary, and solar system exploration. Three OTV's or upper stages are currently included in the Space Transportation System (STS). These include the inertial upper stage (IUS) and two spin-stabilized upper stages (SSUS). When combined with the Space Shuttle, these systems provide GEO delivery capability for payloads weighing up to 2300 kg. Such capability is expected to satisfy the majority of the mission requirements through the late 1980's.

Missions beginning in the late 1980's are anticipated to be more ambitious; to satisfy these requirements, NASA has recently focused on two additional types of OTV's. These include a reusable cryogenic stage (defined in ref. 1) and a solar electric propulsion system (SEPS) (described in "Alternate System Design Concepts Study for the Solar Electric Propulsion System," NASA contract NAS8-33753). Both vehicles can be defined as "first-generation" systems for their respective technologies. The reusable cryogenic stage is envisioned primarily for transportation between LEO and GEO and planetary missions. The system is generally launched with its payload using a standard Space Shuttle and returned to Earth in the orbiter. Advanced versions of the system would employ an aerobraking device to significantly improve performance. With a growth version of the shuttle, the advanced OTV can provide delivery capability to GEO of 12 500 kg and 6000 kg for round trips. The SEPS is being initially designed for solar system exploration. Key features include solar array for power generation and ion thrusters for propulsion. It too is launched with its payload by a standard space shuttle. Further boost assistance is provided by the IUS to move the SEPS rapidly through the Van Allen radiation belts.

Although the first-generation cryogenic OTV and SEPS are expected to satisfy a large portion of the missions in the 1990's, several factors suggest further improvements in space transportation may be possible as well as necessary. These factors include: (1) by the mid to late 1990's, individual mission needs are expected to increase in addition to increases in annual mass being transported to GEO, (2) orbital support platforms are receiving serious consideration for this time frame with one role being the support of OTV's, and (3) OTV designs and operations for the first-generation OTV's were constrained by the launch system and technology available as of the early 1980's.

1.2 OBJECTIVES AND ISSUES

Recognition of the potential needs beginning in the mid 1990's led to the initiation of the Future Orbital Transfer Vehicle Technology Study. This study had the overall objective of building on the data base associated with the first-generation OTV's to determine the technologies required for the OTV fleet in the post-1995 time frame. Within this framework, the specific objectives were to:

1. Examine the roles of future orbital transfer vehicles within an integrated space transportation system in order to determine vehicle, operations, and technology requirements.
2. Develop baseline vehicle set(s) that satisfies the requirements.
3. Determine the benefit of accelerated technology improvements.
4. Define path of technology evolution from the near term to the far term (e.g., from first-generation OTV to future OTV baseline set).

The first objective dealt with defining the potential missions to be performed by OTV's and the types of vehicles and technologies which would satisfy mission/payload requirements. Since mission models generally contain a wide range of requirements, a vehicle set consisting of more than one type or size of OTV may be necessary to provide the least cost space transportation system. Use of accelerated technology generally tends to provide better performance but also entails additional research and development costs. Finally, to reduce cost and improve confidence, future OTV's should benefit from the evolution of near- to far-term technology.

The specific issues associated with the study objectives are:

1. Would space basing of future OTV's provide an improvement in terms of the total space transportation system and its operations?
2. Is there a role for an electric OTV in transporting cargo between LEO and GEO when near-term mission models are employed?

3. Would the use of accelerated technology rather than normal growth alter the results of either of the above issues?
4. What technological advances are necessary and which have the most payoff for future OTV's?

For the most part, these issues relate to establishing the characteristics for second-generation OTV's. One potential improvement to the assumed first-generation reusable cryogenic OTV with aeroassist capability was that of space basing the OTV rather than ground basing. Space-based OTV's have been analyzed in other studies and at times compared with ground-based OTV's. The studies, however, were limited by the amount of data available on the related support systems, confined to only a comparison of flight performance, or were not analyzed considering all the related aspects of space transportation and operations. The FOTV study, however, had the benefit of the recently completed Phase A OTV study (ref. 1) and the in-progress Phase A study, "Space Operations Center System Analysis," NASA contract NAS9-16151.

Electric orbit transfer vehicles, because of their high performance, also merit consideration as a means for transporting large quantities of cargo to GEO. Previous studies such as the Solar Power Satellite (SPS) (ref. 2) indicated solar electric OTV's to be more cost effective than LO_2/LH_2 OTV's. Contributing to their success was the use of a very large mission model, very low cost solar arrays, and a chemical stage that did not use aerobraking. When viewed in terms of more near-term mission models and technologies, such as those to be investigated in the FOTV study, a different outcome is quite possible.

The third issue involves the degree of readiness associated with technology. Normal growth is defined as that which should be available at a given point in time with current funding projections. Accelerated technology generally is viewed as providing higher performance and is technically feasible but little or no money is being funded for its development. Any resulting life cycle cost (LCC) improvement occurring as a result of the accelerated technology should also include the basic research and development (R&D) cost. The final issue is to identify those technologies that are necessary to achieve the optimum OTV's for the post-1995 time frame.

1.3 STUDY GUIDELINES

The key guidelines used in performing the study are listed below. Those followed by an asterisk (*) are from the statement of work; those followed by two asterisks (**) have been mutually agreed upon by NASA and Boeing.

1. Technology to be available in 1990**
2. Vehicle to have IOC of 1995*
3. Technology to be considered only in terms of OTV application*
4. 1995-2010 time frame to be considered for potential missions with major emphasis on Earth orbital missions*
5. Two levels of traffic models to be considered*
6. Most cost-effective launch system to be selected**
7. Figure of merit to be LCC of total space transportation system (1980 dollars)*

Technology was to be available 5 years before the IOC of the OTV to ensure a smooth development program in terms of final design and test. The emphasis on Earth orbital missions (rather than planetary) was specified to ensure that the OTV would be sized by these requirements, which are expected to dominate the bulk of the transportation needs in the next 20 years. Low and high traffic models were considered to test the sensitivity of the candidate OTV's. The opportunity to select the most cost-effective launch system was considered significant since in several past studies the system was edicted and, in most cases, had a significant bearing on the most effective OTV basing mode or technology employed. Finally, the total transportation system cost was to include all elements directly involved in transporting or providing services, including design, development, test, and evaluation (DDT&E), production, and operations costs.

A final note relative to guidelines used to conduct the study dealt with the type of OTV technology that would not be considered. Included within this area were nuclear electric rockets, fusion rockets, laser rockets, and nuclear fission rockets. These options were not considered for the following reasons: (1) it was judged there would be a very low probability of availability for the indicated IOC and (2) they were being examined in the Advanced Propulsion Systems Concepts for Orbital Transfer Study (NASA contract NAS8-33935), in progress at the same time.

1.4 CONTENT FORMAT

Section 2.0 summarizes key findings and conclusions of the study. The remainder of the document is formatted to emphasize the detailed analysis concerning the two system issues: (1) space- versus ground-based OTV's (sec. 3.0) and (2) electric versus chemical OTV's (sec. 4.0). Within each issue, mission considerations and implications concerning normal growth and accelerated technologies are included.

2.0 SUMMARY OF KEY FINDINGS AND CONCLUSIONS

2.1 KEY FINDINGS

Principal findings of the study are reported here as responses to questions that address the study issues.

Would Space Basing of Future OTV's Result in an Improved Space Transportation System?

In terms of total transportation costs, there was no clear-cut answer. Cost differences between the basing modes range from an 11% advantage for the SB OTV to a 7% advantage for the GB mode, depending on the mode used to recover (return to Earth) the key OTV elements. In the case of the ground-based (GB) OTV mode, the OTV's were to be recovered and reused (expendable OTV's were not cost effective). In the space-based (SB) OTV mode, propellant tankers were the key element requiring recovery consideration. The significance of the recovery operations was that they had an influence on which launch vehicle would be used which, in turn, was the largest contributor to the mission model total transportation cost. Differences in flight performance, refueling, and orbital support provisions were of secondary importance to the cost comparison.

This issue was analyzed using an advanced space scenario involving a mission model beginning in 1995, 11 years in duration, averaging 115t of GEO-equivalent payloads per year, and requiring 182 OTV flights. The basing issue was analyzed from a total transportation standpoint which involved all systems and operations necessary for launch and recovery, orbital support, and performance of the OTV mission itself. A permanently manned base was used to the best advantage of both basing modes. OTV's investigated were considered as second-generation reusable systems using LO_2/LH_2 propulsion and normal growth technology available as of 1990.

The most cost-effective launch system for the advanced space scenario involved use of both the Space Shuttle and a solid-rocket shuttle-derivative vehicle (SDV). The shuttle was used to launch personnel, supplies, and a portion of the OTV payloads. The SDV launched the majority of the payloads, OTV's, and/or propellant tankers. Cargo return (to Earth) capability was not provided by the initial SDV investigated. Design provisions were considered for the SDV that would allow cargo return, although this approach was judged to have relatively high technical risk concerning reentry control and payload survival with water landings.

The SB OTV mode was found to provide an 11% cost advantage for the case where return cargo capability was not provided by the SDV. This advantage was the result of the SB mode being able to resort to an expendable tanker but still use the SDV. The GB OTV mode, however, could not tolerate an expendable OTV (due to cost) nor were there sufficient numbers of shuttle flights to return the OTV's. This situation required the switch to a launch vehicle with return capability, such as the liquid-booster growth shuttle. Launch cost (per unit mass) was higher with this vehicle than with the combination of shuttle plus SDV, and this was the major contributor to the cost penalty of the GB OTV.

Should the higher risk SDV cargo return mode be considered, both basing modes would benefit in relation to the results of no SDV cargo return capability. In this case, the GB OTV mode showed the greatest improvement, resulting in a 7% cost advantage. Contributing to the result is the fact that both OTV modes used the same launch vehicles; however, the GB OTV does not require a tanker and has less space base support cost.

In addition to cost, other factors were assessed to determine if differences existed between the basing modes. The SB OTV was found to provide advantages in terms of flight performance, launch manifesting, and more rapid access to GEO. The performance advantage of 6% in payload for a fixed propellant loading occurs even after provisions were incorporated for on-orbit maintenance and space-debris protection. More effective launch manifesting occurs because with on-orbit propellant storage capability, launches involving GEO-type payloads can also include a tanker loaded with enough propellant to ensure a mass limited launch condition. A more rapid access to GEO also results from there being an OTV and propellant storage availability at a LEO space base. Missions that may require this feature include rescue of a manned system, servicing of a critical space system (assuming spares are available at the base), or special reconnaissance. The SB OTV could initiate the mission in less than 1 day because it is kept in a state of readiness except for refueling.

In summary, the cost difference between the basing modes was not overwhelming; however, the SB OTV mode can provide operational advantages and has a greater cost improvement potential with use of accelerated technologies.

Is There a Role for an Electric OTV in Transporting Cargo to GEO?

This issue must be viewed in the context of total OTV transportation requirements. An electric OTV (EOTV) with long delivery times (cost optimum of 180 days) and much exposure to Van Allen radiation does not satisfy the delivery needs of most payloads or high priority missions such as manned and DOD payloads requiring rapid delivery. These requirements, however, can be satisfied by a chemical OTV. Consequently, the issue becomes that of comparing two different fleets: the first is a mixed fleet of high-performance electric OTV's for trip-time-insensitive cargo plus chemical OTV's for high priority missions; the second is a fleet of chemical OTV's for all missions.

When viewed from this standpoint, the all-chemical SB OTV fleet provided a 23% advantage in transportation life cycle cost over the mixed fleet when normal growth technology was used. High production costs for the EOTV, in addition to the need for a chemical OTV, were the major contributors to the higher cost of the mixed fleet.

These results were based on a mission model that began in 1995, had a 16-year duration, and averaged 300 t/yr of GEO-equivalent payloads of which 110 t/yr were judged to be EOTV compatible. The launch vehicle fleet again consisted of a basic STS and SDV with reusable payload system (RPS). The EOTV used technologies that were considerably improved over those provided by SEPS, which was the assumed first generation electric OTV. Principal features of the power generation system were silicon cells that were 3% more efficient, six times larger, 25% as thick, and 50% as costly. Electric propulsion employed argon ion thrusters with twice the specific impulse and power processors with specific masses only 25% as large. The most dominating factor regarding sizing and ultimately the cost of the EOTV was the solar array degradation caused by Van Allen radiation. One LEO to GEO round trip with a lightweight array resulted in a 60% degradation of its initial power. Options investigated to minimize degradation and/or amount of power required were (1) a heavily shielded array, (2) faster transit through the radiation belts using chemical assistance, (3) concentrated arrays, and (4) thrusters using less power (arc jets). The heavy shielding concept using 300-μm cover, 50-μm cell, and 250-μm substrate had the best all-around characteristics when using normal growth technology that did not include annealing or GaAs cells.

Would Accelerated Rather Than Normal Growth Technology Alter the Results of Either of the Above Issues?

Use of accelerated technology provided improvements to all vehicles investigated—however, not to the extent of changing the major conclusion associated with either the basing or fleet makeup issues.

In the case of the OTV basing issue, use of accelerated technology such as LF_2/LH_2 provided substantial reductions in stage length (25%) and propellant loading (15%). Life cycle costs, however, were not appreciably different from the normal growth technology vehicle because of higher DDT&E and production costs. The SB OTV tended to benefit more from this technology because the reduction in propellant could be reflected in fewer SDV tanker launches.

Accelerated technology had a significant payoff for EOTV's. The most significant improvement was that of removing radiation damage by annealing. Little cost difference was found between silicon and gallium arsenide (GaAs) solar arrays when both incorporated annealing features. The lower performance and slightly higher radiation sensitivity of the silicon cells were offset by their better effectiveness in terms of annealing and lower unit cost. The most advanced accelerated technology EOTV investigated reduced the average unit cost by 50% relative to the normal growth EOTV. However, when viewed in the context of total OTV transportation requirements, the all-chemical OTV fleet employing normal growth technology still provided a 5% cost advantage, as well as operational advantages over a mixed fleet comprised of chemical OTV's and accelerated technology EOTV's.

What Technological Advances are Necessary and Which Have the Most Payoff for Future OTV's?

Based on the results of the two vehicle-level issues, the OTV having the greatest promise for the 1995-2010 time frame is an advanced, reusable LO_2/LH_2 system. The technologies suggested must be related to a point of departure—in this case a first-generation, ground-based, reusable LO_2/LH_2 OTV with RL-10 IIB main engine and an insulated ballute for aeroassist capability. The most significant critical/enabling technology associated with the second-generation OTV (GB or SB) is that of space-debris protection for large thin-walled cryogenic tanks designed to fracture mechanics criteria. Of particular interest are the shielding benefits provided by composite materials. On-orbit refueling and maintenance are necessary for the SB OTV. In the case of refueling, zero g propellant transfer provisions must be provided in addition to systems that minimize propellant storage and transfer losses. Maintenance considerations will dictate very high quality components, modularization, and computer-aided self-diagnosis. Normal growth in LO_2/LH_2 engine technology is expected to provide higher performance and longer life. Improvements in ballutes for aeroassist capability should also be pursued in the areas of advanced materials and techniques that would allow use of transpiration cooling, resulting in significant performance gains.

2.2 STUDY CONCLUSIONS

The following conclusions are presented with the assumptions that (1) the basic STS is an operational system, (2) a reusable ground-based LO_2/LH_2 OTV with aeroassist capability and a space base such as the SOC are firmly in the planning cycle, and (3) GEO-equivalent payloads can be as high as 300 t/yr.

1. Reusable LO_2/LH_2 OTV's can serve all general-purpose cargo roles between LEO and GEO for the foreseeable future.

2. Electric propulsion used with photovoltaics may be worthwhile for specialty missions (e.g., high energy, heavy payload, on-orbit stationkeeping) but not for LEO to GEO cargo delivery in the foreseeable future.
3. Space basing of OTV's can provide cost and operational benefits relative to ground-based OTV's.
4. Normal growth LO_2/LH_2 technology efforts (aeroassist and new engine) should continue because they pay for themselves and offer performance margins.
5. Accelerated technology for chemical OTV's does not appear justified if the most cost-effective launch system (SDV) is employed.
6. Key critical/enabling technologies that should be initiated for future OTV's include space-debris protection and propellant storage/transfer.
7. A possible OTV evolutionary path may include the following steps:
   a. Initiate operation with a shuttle-optimized, ground-based, reusable OTV.
   b. Once a space base (e.g., SOC) is available, use capability to integrate ground-based OTV/payload and OTV/Earth-return system.
   c. Switch to full space basing of OTV after key servicing features required by the OTV have been demonstrated at a space base. Key OTV support provisions to be provided by the space base include hangars and propellant storage facilities. A space-based OTV and hangar are shown in figure 2.2-1. The hangar has the dual role of providing OTV protection against space debris and serving as a facility in which to perform maintenance.
8. The most significant reduction in advanced space scenario transportation cost can be achieved through development of a shuttle-derivative cargo launch vehicle.

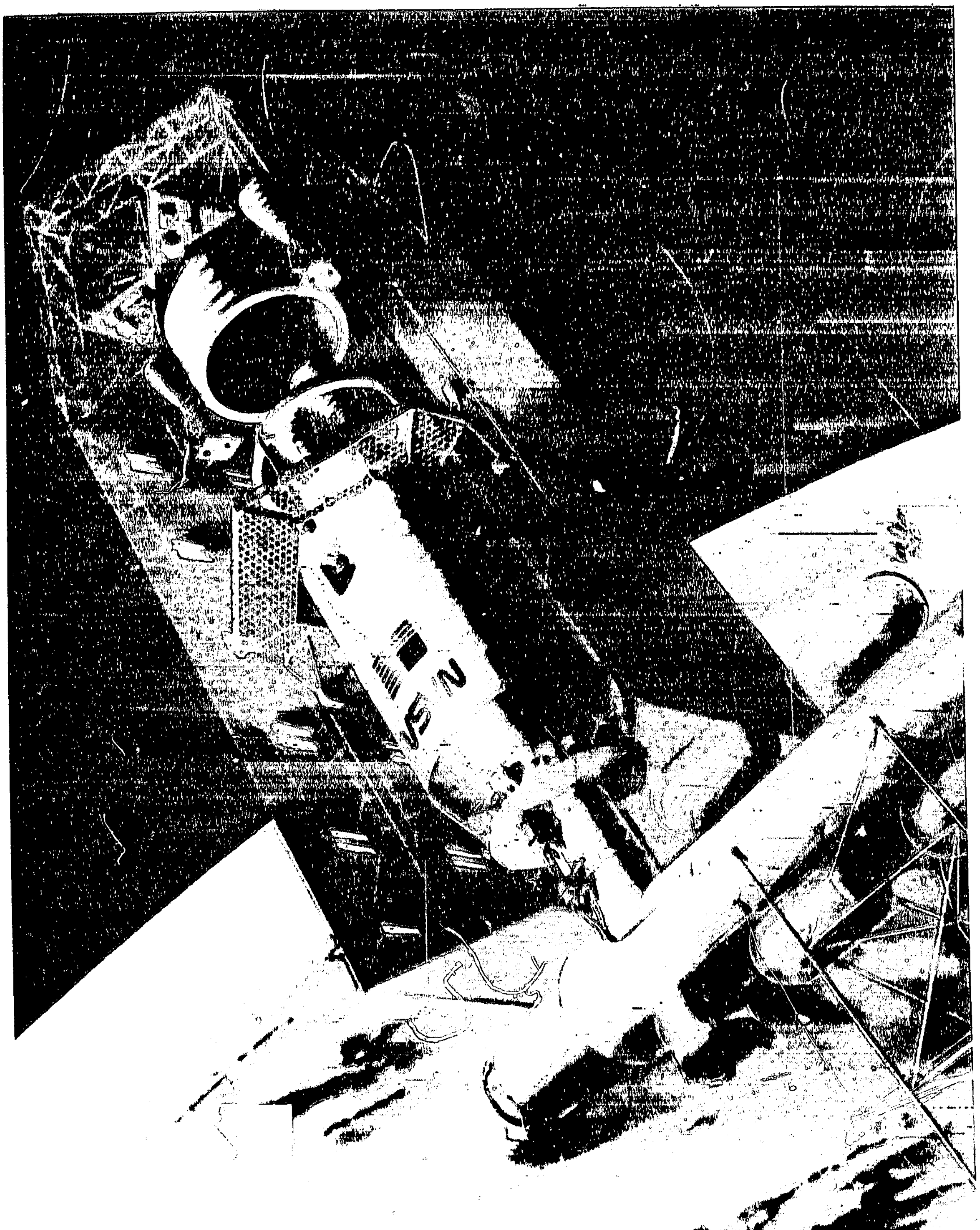

Figure 2.2-1 Space-Based OTV and Hangar

3.0 SPACE- VERSUS GROUND-BASED OTV's

This section presents the complete analysis associated with the comparison of space- versus ground-based OTV's. The principal subsections include mission analysis, the definition and comparison of OTV"s using normal growth and accelerated technology, and the overall findings and recommendations.

3.1 INTRODUCTION

The scope of the analysis associated with the OTV basing mode comparison is shown in figure 3.1-1. The SB OTV, once launched, essentially remains on orbit throughout its

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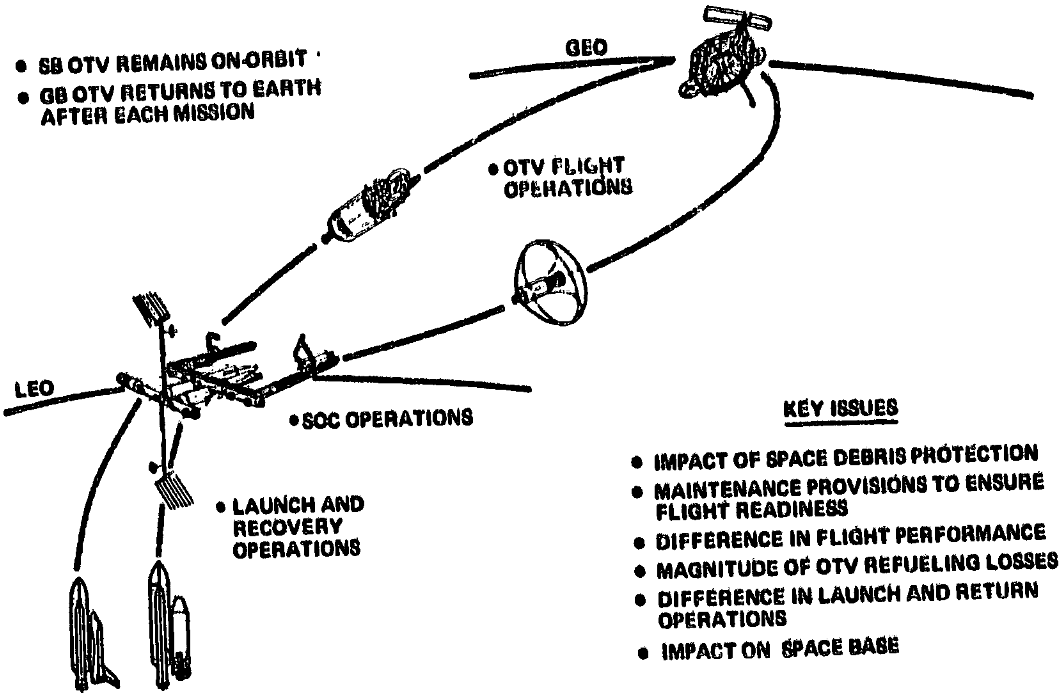

Figure 3.1-1 OTV Basing Concepts—Integrated Transportation Operations

design life. The GB CTV is returned to Earth after each flight to allow servicing. Included within the scope of an integrated transportation assessment of the basing modes are: (1) all launch and recovery operations, (2) all operations necessary at an orbital base, and (3) all operations associated with the actual OTV flight. Areas which were expected to show a difference between the basing modes are indicated as key issues and are discussed in detail in subsequent sections of this document. A brief discussion of each issue is included in the following paragraph.

The impact of space debris (meteoroids and manmade) protection in terms of mass penalty was expected to be greater for the SB OTV because its structural provisions normally would be designed to sustain only flight loads, whereas the GB OTV structure must sustain launch loads for a fully fueled condition. Continued on-orbit storage also presents a unique space debris protection requirement for the SB OTV. Vehicle design features to allow on-orbit maintenance in a reasonable amount of time and with a minimum of personnel are also expected to affect the SB more than the GB OTV because SB OTV maintenance manpower is much more constrained. Flight performance comparisons in the past have usually shown up to a 10% advantage for SB OTV. With dry mass increases in the areas of debris protection and maintenance, the advantage may be appreciably decreased. Refueling losses cover all aspects of the storage and transfer of propellant to an SB OTV. The impact of the losses is that of additional launch vehicle flights. Differences in launch operations will be expressed in numbers of launches as brought about by cargo manifesting. Recovery operations deal with how OTV's or propellant tankers are returned to Earth for servicing and reuse. The impact on the space base involves the number of personnel required and equipment directly associated with support of an OTV.

A final comment of the overall operational concept deals with a further explanation of the indicated orbital base. The base is indicative of NASA's Space Operations Center which is to be a permanently manned facility. Although SOC is still somewhat controversial in terms of required time frame, an availability by 1995 (5 years after JSC/NASA goal) appears within reason and would coincide with the operation of a second-generation OTV. It should be mentioned, however, that an OTV basing mode comparison with an unmanned platform available in LEO for support operations is also a possibility but is not analyzed in this study.

3.2 MISSION ANALYSIS

This section describes the mission model, the assumed time phasing of the missions, and the requirements imposed on the space transportation system.

3.2.1 Background

The mission model was developed using guidelines presented in the statement of work and data resulting from the Phase A OTV studies. The key guidelines from the SOW included the following:

1. Missions that could occur in the 1995-2010 time frame.
2. Two scenarios or mission models to be used, with one model being greater in magnitude in order to assess the merits of more advanced OTV concepts.
3. Mission categories to include: (1) large automated satellites (LAS) that require assembly or construction and are used at GEO, (2) free-flying automated satellites (FFAS) used in GEO, (3) cargo delivered to a GEO base, (4) manned round trips between LEO and GEO and, (5) other, relating to those missions which may fall outside the previous four categories.

Considerations for using data from the Phase A OTV study were based on several factors, including: (1) considerable effort had been expended in developing the model and involved participation by NASA, the Air Force, and study contractors, and (2) there was a 5-year overlap in model time frames (1987-2000 for Phase A and 1995-2010 for FOTV). A final decision relating to the mission models was that a space operations center would be available in LEO in the early 1990's; therefore, launch requirements should consider crew rotation and resupply of this base.

3.2.2 Model Description

An overview of the mission models considered is presented in table 3.2-1. Included are the mission names, brief descriptions, and assignment to either low or high models. The low model missions are those that were to be used in the comparison of SB and GB OTV's. It will be noted, the low model includes several types of missions which were considered outside the time frame of the Phase A OTV nominal model (revision 2). The high model was to be used in the comparison of OTV fleets consisting of electric and chemical OTV's versus all-chemical OTV's. Further discussion of the high model occurs in section 4.0.

The time phasing of the missions for the 16-year model is shown in table 3.2-2. Again, the first 5 years of the model are essentially the same as used in the Phase A nominal model (revision 2). The last 11 years involve more of the same types of missions and introductions of either the new mission, previously identified in table 3.2-1, or an advanced design such as for mission No. 2. Although the durations of the Phase A and FOTV models are nearly the same (14 versus 16 years), two of the mission types in the FOTV model involve a significantly greater number of payloads. First, there are a greater number of GEO base support missions (crew rotation/resupply), since the base is present for 12 rather than 2 years. Second, since there are more satellites in orbit with the later time frame, more servicing (manned and unmanned) missions are required.

Table 3.2-1 Mission Model Composition

CATEGORY	MISSIONS (TO GEO UNLESS INDICATED)	PHASE A OTV NOMINAL	FOTV LOW	FOTV HIGH	SUMMARY DESCRIPTION
LARGE ASSEMBLED SATELLITES (LAS)	1. COMMUNICATIONS PLATFORM (BLKI)	X	X	X	TRUNKLINE COMMUN. SYS; REPLACES INTELSAT, DOMSAT ETC.
	2. ADVANCED COMMUN. PLAT (BLK II)	-	X	X	IMPROVED TRUNKLINE SYS (1500 X POWDERS/PLAT. vs 230)
	3. PERSONAL COMMUN SAT.	X	X	X	PROVIDES WRISTSET TELEPHONE SERVICE
	4. SPACE BASED RADAR	X	X	X	AIRCRAFT, SHIP, GROUND VEHICLE SKIN TRACKING
	5. DOD CLASS III (HEAVY)	X	X	X	GENERIC PAYLOAD
	6. DEEP SPACE RELAY SAT	-	X	X	PROVIDES HIGH PERF LINK BETWEEN DEEP SPACE PROBES & EARTH
	7. SOLAR TERRESTIAL OBSERVATORY	-	X	X	MONITOR SUN AND EARTH
	8. SPS ENGR VERIF. TEST ARTICLE	-	-	X	DEMONSTRATE ENGINEERING PRINCIPALS OF KEY SATELLITE ELEMENTS
	9. SPS DEMO SATELLITE	-	-	X	PROVIDES COMMERCIAL DEMO OF COMPLETE SYSTEM; 100 MW GROUND OUTPUT
FREE FLYING AUTO SATELLITES (FFAS)	10. DOD CLASS IA	X	X	X	GENERIC PAYLOAD
	10a. DOD CLASS IB	X	X	X	GENERIC PAYLOAD
	11. DOD CLASS II (MEDIUM)	X	X	X	GENERIC PAYLOAD
	12. COMMERCIAL & NASA (LIGHT)	X	X	X	GENERIC PAYLOAD
CARGO TO MANNED GEO BASE	13. GEN PURPOSE BASE	X	X	X	STAGING BASE FOR SERVICING MISSIONS & EARTH OBSER PLAT.
	14. GEN PURPOSE MISSION EQUIP	X	X	X	APPLICATIONS MODULE FOR BASE AND GEO SERVICE SHUTTLE
	15. SATELLITE MAINT PROVISIONS	-	X	X	SPARES AND EXPENDABLES FOR LAS & FFAS
	16. CONST BASE ELEMENTS	-	-	X	LARGE BASED DEDICATED TO CONST OF DEMO SPS SATELLITE
	17. DEMO SATELLITE MAINT. PROVISIONS	-	-	X	SPARES AND EXPENDABLES
MANNED ROUND TRIPS	18. MAINTENANCE SORTIE (FROM LEO BASE)	X	X	X	UNSCHEDULED SERVICING OF GEO SATELLITES
	19. G. P. BASE SUPPORT	X	X	X	ROTATE CREW EVERY 90 DAYS AND PROVIDE SUPPLIES
	20. CONST BASE SUPPORT	-	-	X	ROTATE CREW EVERY 90 DAYS AND PROVIDE SUPPLIES
	21. SCIENCE SORTIES	-	X	X	INVESTIGATE EARTH BOW SHOCK; LUNAR SURFACE SURVEY ETC.
OTHER	22. UNMANNED SERVICING	X	X	X	SCHEDULED MAINT OF GEO SATELLITES
	23. PLANETARY	X	X	X	SOLAR SYSTEM RECONNAISANCE AND EXPLORATION
	24. NUCLEAR WASTE DISPOSAL	-	-	X	PROCESSED NUCLEAR WASTE TO 0.85 A.U.

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Table 3.2-2 Low Mission Model

FOTVTS-113

CATEGORY	MISSIONS	TOTAL	95	96	97	98	99	00	01	02	03	04	05	06	07	08	09	10	COMMENTS
			MISSIONS PER YEAR																
LARGE ASSEMBLED SATELLITES (LAS)	1. COMMUN PLATFORM	7	1	1	2	2	1												2700 XPONDERS (40 MHz) BY YEAR 2000
	2. ADV. COMMUN PLATFORM	4						1						1		1		1	ADDITIONAL 6000 XPONDERS BY 2010
	3. PERS COMMUN SAT	8	1		1		1		1 ◁10		1		1		1		1		80 MILLION SUBSCRIBERS; ◁10 CHANGE TO BLOCK II MODEL
	4. SPACE BASED RADAR	2									1	1							
	5. DOD CLASS III	9	1	1	1	1	1	1								1	1	1	
	6. DEEP SPACE RELAY SAT	1						1											
	7. SOLAR TERRES. OBSER	1								1									
FFAS	10. DOD CLASS IA 10a. DOD CLASS IB	32 10	2	2	2 1	2 1	2 1	2 1	2	2	2 1	2 1	2 1	2 1	2 1	2 1	2	2	
	11. DOD CLASS II	5						1	1	1	1							1	
	12. COMMER & NASA	16	2		2		2		2		2		2		2		2		
CARGO TO MAN MANNED GEO BASE	13. GEN. PURP. BASE	2					1 ◁1										1 ◁2		◁1 4 MAN CAPAB. ◁2 8 MAN CAPAB.
	14. G.P. MISSION EQUIP	5					1 ◁3		1 ◁4			1 ◁3		1 ◁4			1 ◁3		◁3 APPLIC MODULE; ◁4 SAT. SERVICE SHUTTLE
	15. SAT. MAINT. PROV.	16							1	2	1	2	1	2	1	2	1	3	ALSO REFLECTS MANNED MAINT. SORTIES PER YR
MANNED ROUND TRIPS	18. GEO MAINT. SORTIE	11	2	1	3	2	1	2	◁6								◁5		◁6 AFTER YR 2000 MAINT. SORTIE STAGED FROM GEO BASE
	19. G.P. BASE SUPPORT	54					2	4	4	4	4	4	4	4	4	4	8	8	◁5 REFLECTS 8 MAN BASE; 4 MEN & SUPPLIES/FLT
	21. SCIENCE SORTIES	3					1						1					1	
OTHER	22. UNMANNED SERVICING	103	2	4	4	5	5	6	7	8	7	8	7	8	7	8	7	10	
	23. PLANETARY	8		1		1		1		1		1		1		1			
	TOTAL MISSIONS	297	11	10	16	14	19	20	19	19	20	20	19	20	18	20	24	27	TOTALS FLIGHTS / PAYLOAD (MT) 11 YR MODEL 189 / 1260 16 YR MODEL 297 / 2045

3.2.3 Transportation Requirements

The major transportation requirements imposed by the missions/payloads are shown in table 3.2-3. A key feature of these data is that the number of missions reflects an 11-

Table 3.2-3 FOTV Low Model (11-Year) Mission-Imposed Transportation Requirements

FOTVT8-410

MISSION NO.	NAME	QTY (11YRS)	MASS (MT) EACH	LENGTH (M) EACH	G CONSTRAINT
1.	COMM PLAT	7	6.8	7.6	0.2
✓ 2	ADV COMM PLAT ▷1	1	31.8	27.4	0.2
3	PERS COMM-SAT ▷1	6	24.5	18.3	0.1
4.	SPACE BASED RADAR	2	11.4	18.3	0.1
5.	DOD CLASS 3	6	11.4	7.6	0.1
6	DEEP SPACE RELAY SAT.	1	6.8	7.6	0.1
7	SOLAR TERR. OBSERV.	1	11.0	18.3	1.0
10.	DOD CLASS IA	22	2.7	7.6	3.0
10a.	DOD CLASS IB	7	4.0	9-12	3.0
11.	DOD CLASS 2	4	5.5	6.1	1.0
12.	COMMER & NASA	12	4.5	7.6	3.0
13.	GEO BASE MODULES	2	15 & 20	12	3.0
14.	GEO BASE EQUIP.	3	9	6.1	3.0
15.	SAT. MAINT. PROV.	7 ▷2	2 ▷2	1.8	3.0
18.	GEO MAINT. SORTIE	11	5.9/5.9	4.0	3.0
✓ 19	BASE SUPP (CR/RS)	26	7.6/5.0	4.0	3.0
21.	SCIENCE SORTIES	2	8.1/8.1	4.0	3.0
22.	UNMANNED SERVICING	63	4.8/0.9	7.6	3.0
23.	PLANETARY	6	5	7.6	3.0
		182	Σ = 1280		

▷1 REQUIRES CONSTRUCTION ON ORBIT ▷2 INCLUDED WITHIN MISSION 19 UP

✓ DRIVER MISSIONS ○ CIRCLED MISSIONS ARE DIFFERENT FROM PHASE A

rather than 16-year model. This reduction was thought necessary because of the concern that with a large model, a potential advantage would be available to a more advanced system which, in this comparison, would be the SB OTV. Even with this limiting factor, however, approximately 40% more payloads are involved and the GEO delivery equivalent (accounts for round trip payloads) mass of almost 1300t is nearly twice that of the Phase A model. Should the 16-year model be used, a GEO delivery equivalent of over 2000t would occur.

Acceleration constraints are assumed for those payloads requiring final assembly or on-orbit construction. The largest delivery mission is nearly 32t compared with 11.5t for the Phase A model. The largest round trip mission was 7.6t up and 5.0t down, as compared with 5.9t up and down in the Phase A. Delta-V's and timelines associated with the missions are presented in section 3.3.5.

3.3 NORMAL GROWTH TECHNOLOGY VEHICLES

This section provides a complete description of both the SB and GB OTV which use normal growth technology. All aspects of vehicle design and operation are discussed, as well as a comparison of the life cycle cost. It should be noted that a number of topics in this section reflect the use of the selected launch vehicle family which consists of the basic STS and an SDV. The STS is used to launch crews and some payloads, while the SDV delivers OTV's, tankers, and most of the payloads. The analysis associated with the launch system selection is presented in section 3.3.11.

3.3.1 Assumptions and Guidelines

Key assumptions and guidelines used in defining the OTV's follow:

1. Point of departure—BAC Phase A OTV.
2. 1990 technology availability.
3. Stage design life—45 missions.
4. Propellant tankers and storage tanks design life—50 cycles.
5. OTV mission success goal = 0.97.
6. Space debris probability of not impacting OTV propellant tanks = 0.995.
7. Manned OTV to incorporate two engines.
8. Space transportation elements limited in size to allow return to Earth by STS orbiter.
9. OTV design reference missions: ground trip (LEO-GEO)—GEO base support (crew rotation/resupply) 7.6t delivery, 5.0t return; delivery only (LEO-GEO)—advanced communication platform, 31.8t, 0.2g maximum.
10. SOC altitude—370 km (by FOTV analysis).

A brief explanation concerning several of the assumptions follows. First, the Boeing Phase A OTV (also referred to as first generation) was used as a point of departure. This particular analysis had characterized the first-generation reusable LO_2/LH_2 OTV with aeroassist capability in considerable detail and provided a strong data base for developing a second-generation ground-based or space-based OTV. A stage design life of 45 missions seemed to be a reasonable point since the Phase A study assumed 20 (due in part because a number of stages had to be expendable and designing more capability into the system was not beneficial) and the shuttle orbiter assumes a 100-flight design life.

The mission success goal was the same as used in the Phase A study but, in the FOTV study, was to include the contribution of subsystems and space debris. Use of two main engines resulted partly from the situation where a new main engine rather than RL-10 derivative was to be used and its predicted reliability was not as high. It should be noted, however, that the number of engines is not a deciding factor between the basing modes as long as both modes use the same number. Transportation elements such as OTV's and tankers were to be sized (i.e., envelope) so they could be launched by the STS and returned by the orbiter. Launch system analysis would determine if the STS was the correct system to launch and return the elements. The two indicated design reference missions resulted after reviewing the mission model and conducting a preliminary performance analysis on several delivery and round trip missions. The indicated SOC altitude was the result of an analysis performed in the FOTV to determine the most cost-effective altitude considering launch and orbit transfer requirements.

3.3.2 Normal Growth Technology Projections

3.3.2.1 Background

The normal growth technology projection provides the fundamental basis for the design of the OTV's. With an IOC of 1995, an approximate readiness date of 1990 was assumed for the various technologies. Normal growth in the context of this forecast means that funds are either being expended or are planned to bring the technical risk down to a reasonable level for initiation of DDT&E by this readiness date. All R&D sources are considered, including NASA, the military services, Department of Energy (DOE), and other branches of Government, the academic community, and industry.

Input data were obtained from the technology forecast indicated in reference 3 and through discussions with individuals and organizations throughout Boeing. Various NASA and other sources were in turn consulted by Boeing specialists or study participants.

3.3.2.2 Projections

A summary of the normal growth projections for LO_2/LH_2 OTV's is presented in table 3.3.2-1. These data are presented to indicate the projections for the FOTV relative to the first-generation reusable cryogenic OTV assumed to be similar to that defined by the Phase A OTV study.

Structure - Continued use of 2219 T87 aluminum is envisioned for the main propellant tanks because there appears to be no identifiable substitute material which will yield a lighter and more reliable tank design. Composite overwrap designs do not appear to offer any significant weight or cost savings for the relatively low pressure main propellant

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Table 3.3.2-1 Chemical OTV Normal Growth Technology Projections

FOTVTD-328

SUBSYSTEM	BASELINE OTV (BAC PHASE A)	FOTV	BENEFIT
• STRUCTURE			
• TANKS	ALUM	NO CHANGE	–
• BODY SHELL	G/E SANDWICH	BETTER PROPERTIES	10% IN WT.
• AVIONICS RING	ALUM	G/E	40% IN WT.
• BALLUTE	INSULATED	TRANSPIRATION COOLED	50% IN WT.
• THERMAL CONTROL			
• RADIATOR	NO HEAT PIPES	WITH HEAT PIPES	10% LESS WT & AREA
• AVIONICS	PASSIVE	ACTIVE	20% NET WT REDUCTION
• AVIONICS	• REDUNDANT IMU • SIGNAL CONDITIONERS	• LASER GYRO • DATA BUS	35% LESS POWER 30% LESS WT IMPROVED RELIABILITY
• ELECTRICAL POWER			
• FUEL CELLS	• MODIF. SHUTTLE	• ADVANCED	• 38% IN POWER/WT
• BATTERIES	• Ni H_2	• ADVANCED	• 30% IN WHR/LB
• MAIN ENGINE	• RL-10 IIB	• NEW LO_2/LH_2 ENGINE	• ISP = +23 SEC (485 vs 462) • 100% IN LIFE (10 vs. 5 hrs) • WT + 16 KG
• ATTITUDE CONTROL	• N_2H_4 • DECAYING THRUST	• NO CHANGE • FIXED THRUST	• CONTROL AUTHORITY DURING DOCKING

tanks but are useful for high pressure tanks. Minimum aluminum thickness considerations will remain at approximately 0.6 mm (25 mils) for the main propellant tanks. Body shells will continue to use composites with a 10% improvement in strength and stiffness properties suggested. The resulting weight is 60% as heavy as compared with aluminum. Facing sheets for the sandwich design are expected to be 0.25 mm (10 mil) in thickness. The avionics ring assembly material has been allowed to change from aluminum to composite because an active rather than passive cooling system is used.

Ballute - The ballute is an inflatable drag device used to reduce the velocity of the OTV (rather than via propulsion) prior to insertion into LEO. The ballute used in the present Phase A aeroassisted vehicle uses Kevlar[1] cloth overlaid with insulation. Technology available by 1990 should allow transpiration cooling of the ballute. This is accomplished by redesigning the ballute structure to reduce or eliminate the insulation and increase porosity to provide natural transpiration cooling (ref. 4). The benefit of this approach is a 50% weight reduction (coolant gas plus bag) for the ballute unit and a 60% reduction in packaging volume.

[1]Kevlar and Kapton: registered trademarks of E. I. duPont de Nemours and Company, Inc.

Thermal Protection and Control - Improvements were identified for multilayer insulation (MLI) coatings, heat pipes, and radiators. The MLI mass per unit area should be reduced by 10%-20% due to thinner Kapton[1] film. Improved coatings are expected to enhance the absorptivity in the visible to emissitivity in the infrared ($\alpha_S/\varepsilon_{IR}$) characteristics from 0.277 for the first-generation OTV to 0.16 for the FOTV. Use of an active rather than passive thermal control system for avionics is suggested and will provide a weight reduction as well as an improvement in reliability due to operating at a lower temperature. Heat pipe performance will improve 10% by using electrostatic or mechanical pump assist to move condensed fluid back to the hot end. A radiator performance improvement of 10% is expected in terms of weight and area as a result of improved coatings and addition of heat pipes.

Avionics - Two significant changes in the avionics as a result of 1990 technology will be use of laser gyros in the guidance and navigation (G&N) system and a data bus for data management. These systems provide an improvement in reliability as well as decreases in power and weight. Significant improvements are also expected in computers because new processor designs make use of advanced microcircuits or nanocircuits. The other avionics components on the Phase A OTV design were also reviewed for 1990 characteristics with the net result of the total avionics system showing a 35% reduction in power and 30% in weight. An additional set of equipment required for the second-generation OTV will be rendezvous and docking avionics since all OTV basing options will interface with the Space Operations Center.

Electrical Power - Fuel cells currently being suggested for the Phase A OTV are modified STS orbiter cells. The 1990 technology OTV will use the lightweight fuel cells currently under study by NASA LeRC. The result is an approximate 38% reduction in weight for the same power output.

Main Engine - Phase A studies concerning a new LO_2/LH_2 engine for OTV application were completed in 1979 for NASA MSFC by Pratt and Whitney Aircraft Group (report FR12253), Rocketdyne Division (report ASR79-126), and Aerojet Liquid Rocket Co. (report 32999-F). These studies investigated both staged combustion and expander cycles for thrust levels of 66 000N. Chamber pressures ranged between 11 730 kPa (1700 psia) and 15 180 kPa (2200 psia) depending on engine cycle. Nozzle area ratios between 500 and 750 were investigated. The key benefits of these new engines over the RL-10 IIB of the

Phase A OTV were an Isp of 485 sec versus 462 sec and 10 hr (330 start/stops) versus 5 hr between overalls.

Engines using LO_2/MMH were not considered because (1) analysis performed in the Future Space Transportation System Study (ref. 5) indicated a considerable performance penalty relative to LO_2/LH_2 and (2) only a small length improvement was obtained. With strong consideration given to use of shuttle-derivative launch vehicles, the length constraint tends to be less severe; in addition, SB OTV's always have their payloads launched separately.

Engines using propellant such as LF_2/LH_2 were judged to be in the category of accelerated technology and are discussed in section 3.4.

Engine Inlet Pressure Trade - Peculiar to the SB OTV main engine technology projection was the issue of engine inlet pressure. The aforementioned engine studies assumed an inlet pressure of 110 kPa (16 psia) primarily because of application with a GB OTV. The analysis performed in the reference 5 study was primarily oriented to an SB OTV which allowed consideration of a lower inlet pressure, such as 69 kPa (10 psia), resulting in lower tank pressures which reduced the dry weight and, finally, less propellant for a given mission. The penalty to achieve the lower inlet pressure, however, was not assessed. A comparison between OTV's using these two inlet pressures was performed in the FOTV study, using the OTV definition available at the end of the first quarter, and in conjunction with the refueling analysis (sec. 3.3.9) since the OTV, storage tanks, and tanker are all involved.

The comparison of the two engine inlet pressures involves not only the flight elements but also ground testing operations. Testing of the low inlet pressure engine would most likely require rather extensive modifications to existing test facilities. The propellant tanks would have to be built to withstand the internal pressures of less than 101 kPa (14.7 psia). Also, continuous vacuum pumping facilities would be required to maintain 59 kPa ullage pressures during test runs because of heat leak and stratification of the cryogenic propellants. Actual propellant tank conditions would have to be maintained at less than 69 kPa ullage pressures due to the gravity head on the engine inlet because of the propellant height, thus increasing the costs of maintaining conditions stated above. In addition, increased refrigeration equipment would be required to provide propellants cooled to 69 kPa saturation conditions for each test run.

Difference between the flight elements originates with the engine inlet pressure (69 versus 110 kPa), translates to differences in OTV maximum vent pressures (110 versus 166 kPa), which in turn affects SOC propellant–storage tank pressures (124 versus 179 kPa), and finally propellant tanker pressures (159 versus 214 kPa).

The performance and cost comparison of the two inlet pressure options is summarized in table 3.3.2-2. Stage dry weight for the 69 kPa system is less due to lower tank

Table 3.3.2-2 Engine Inlet Pressure Comparison

OTV CHARACTERISTICS (kg)	110 kPa	69 kPa
STAGE DRY WT.	2263	2187
STAGE BURNOUT	2773	2605
PROPELLANT PER FLT.	28817	28243
SOC/OTV TRANSFER LOSS	620	822
NET PROP. Δ PER FLT		- 372
NET PROP. Δ 11 YRS.		- 66950
TANKER CAPACITY		
PROP PER FLT.		+ 318
SOC STORAGE TANK	NO SIGNIFICANT DIFFERENCE	
COST		
• PROP LAUNCH COST		-$32M
• OTV UNIT COST	NO SIGNIFICANT DIFFERENCE	
• GROUND TEST FACILITIES		+TBD

weight resulting from lower pressures and, to some degree, to smaller tank volumes due to less propellant and higher density propellant at the lower pressures as indicated below:

	69 kPa	110 kPa
LO_2 density (kg/m^3)	1158	1136
LH_2 density (kg/m^3)	72	70
LO_2 temp (^oR)	34	37
LH_2 temp (^oR)	180	186

Burnout weight reflects the dry weight as well as residual gases. Propellant per flight is based on the indicated burnout weights and a crew rotation/resupply mission with 7.6t delivered and 5.0t returned. Transfer losses relate to refueling concept B1, described in section 3.3.9, and are higher for the lower pressure system due to its thermodynamics (densities and enthalpy). SOC storage tank mass will be slightly less but, since it is only a one-time penalty, this is not significant. Tanker capacity for the 69 kPa system would probably result in an increase in usable propellant of approximately 320 kg out of 61 360 kg.

The comparison indicates the 69 kPa system to be approximately $30M less on a recurring cost basis. It is judged, however, that the ground facilities required for such a system will offset a large portion of this difference.

The recommendation at this time is that the engine inlet pressure remain at 110 kPa for an SB OTV being used to satisfy missions similar to those identified in the FOTV study. With larger OTV's and/or larger mission models, the 69 kPa system may have recurring cost benefits which would more significantly offset the ground facilities cost and make the lower inlet pressure worthwhile.

Attitude Control Propulsion - This system is used for small delta-V maneuvers as well as attitude control. The total estimated requirement is approximately 40 m/sec. An N_2H_4 system was used in the Phase A OTV. An LO_2/LH_2 system is judged to be feasible by 1990. A comparison of an N_2H_4 (Isp = 220 sec) and an LO_2LH_2 (Isp = 400 sec) attitude control system (ACS) was performed using the SB OTV definition resulting from the first quarter definition. The result of this trade is shown below:

	N_2H_4 ACS	LO_2/LH_2 ACS
Vehicle burnout (kg)	2 810	2 970
Total refueling req't (kg)	29 730	29 860
Main propellant (kg)	29 410	29 680
ACS propellant (kg)	320	180

The vehicle burnout weight using an LO_2/LH_2 ACS is more, primarily because of the LO_2/LH_2 accumulators (sized to supply up to 50% of propellant requirement) and more residuals. On a per-flight basis, the total refueling requirement (main propellant plus ACS) is 130 kg less for the N_2H_2 ACS vehicle as a result of its lower burnout weight.

Therefore, due to less refueling mass and a significantly simpler system, the N_2/H_4 ACS is recommended for this size of OTV. A change relative to the Phase A OTV is that a constant rather than decaying thrust level is recommended to allow full control authority because the OTV will be the active unit in its docking with SOC.

Refueling Systems - Technology associated with these systems is discussed in section 3.3.9.

3.3.3 Space-Based OTV Description

3.3.3.1 Operational Description

The SB OTV is initially launched without propellant and payload. The vehicle is based at an orbital space platform in LEO. In this case, the platform is the Space Operations Center which through analysis has been shown to operate most cost-effectively at 370 km. Payloads, fluids, and spares for the OTV are delivered to the base by the Earth launch systems in a sequence which allows at least two OTV flights prior to another launch system delivery. Prior to each of its flights, the OTV is serviced in terms of scheduled and unscheduled maintenance, payload mating, and loading of consumables and flight programs. Should a maintenance action not be possible on orbit, the OTV will be returned to Earth.

Typical flight operations for a delivery mission are as follows. Following separation from SOC, the OTV/payload combination phases in LEO until the correct nodal crossing occurs. Two perigee burns are used to inject the system into a LEO-GEO transfer orbit. A midcourse correction is performed during the coastout to GEO. Following circularization at GEO, a trim burn achieves the desired destination orbit. After separation from the payload at GEO, a transfer orbit injection burn places the OTV in a GEO-LEO transfer orbit, which will result in a perigee within the atmosphere so that aerobraking can be used to reduce the velocity to near-LEO circular velocity. The key characteristics of the aerobraking concept are illustrated in figure 3.3.3-1. A midcourse correction burn during the GEO-LEO transfer increases the perigee altitude accuracy to the necessary level for aerobraking. Upon completion of the aerobraking maneuver, the ballute is jettisoned and the OTV coasts to an apogee. A burn is performed to raise the perigee out of the atmosphere and up to the desired LEO altitude. A last burn at perigee circularizes the OTV at LEO. The final maneuver involves the docking at the space base. Further discussion of the aerobraking maneuver is provided in section 3.3.5.1.

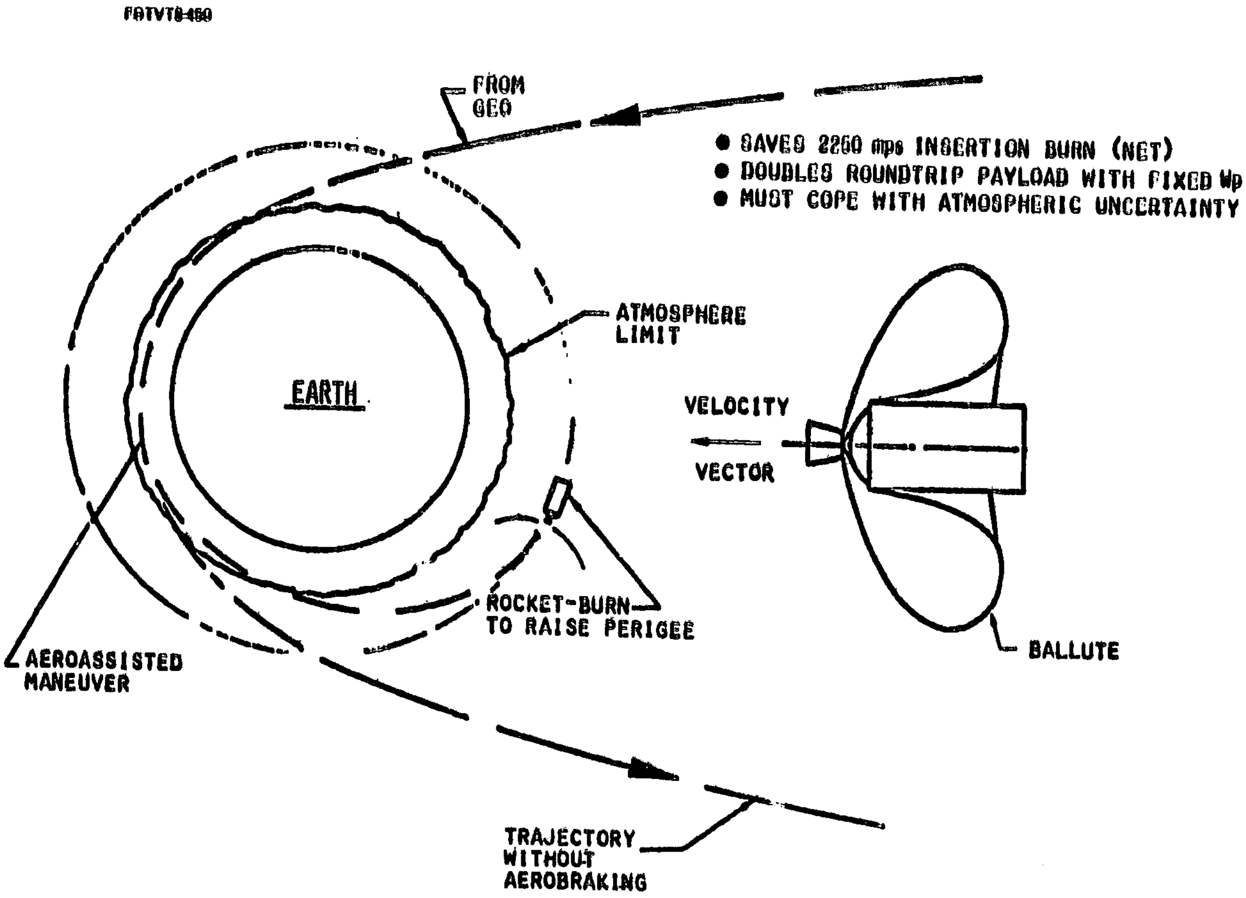

Figure 3.3.3-1 Aeroassisted Vehicle Maneuver

Once back at the base, the OTV is housed in a hangar for protection against space debris. The hangar also allows maintenance to be accomplished more easily. Housekeeping needs (power, thermal, and data links) for the OTV are provided by SOC systems.

3.3.3.2 Configuration Description

The configuration of the SB OTV is presented in figure 3.3.3-2, with overall geometry and physical charcteristics noted. Main propulsion thrust is provided by two advanced space engines, each having a vacuum thrust of 66 700N. The envelope for the engine is indicative of an expander cycle engine which would result in the greatest challenge (due to size) in physical integration. These engines provide thrust for all orbit transfer maneuvers including low thrust application during the GEO-to-LEO aero-maneuver. A hydrazine attitude control system provides thrust for vehicle orientation and rendezvous and docking maneuvers. The spacecraft structure consists primarily of a

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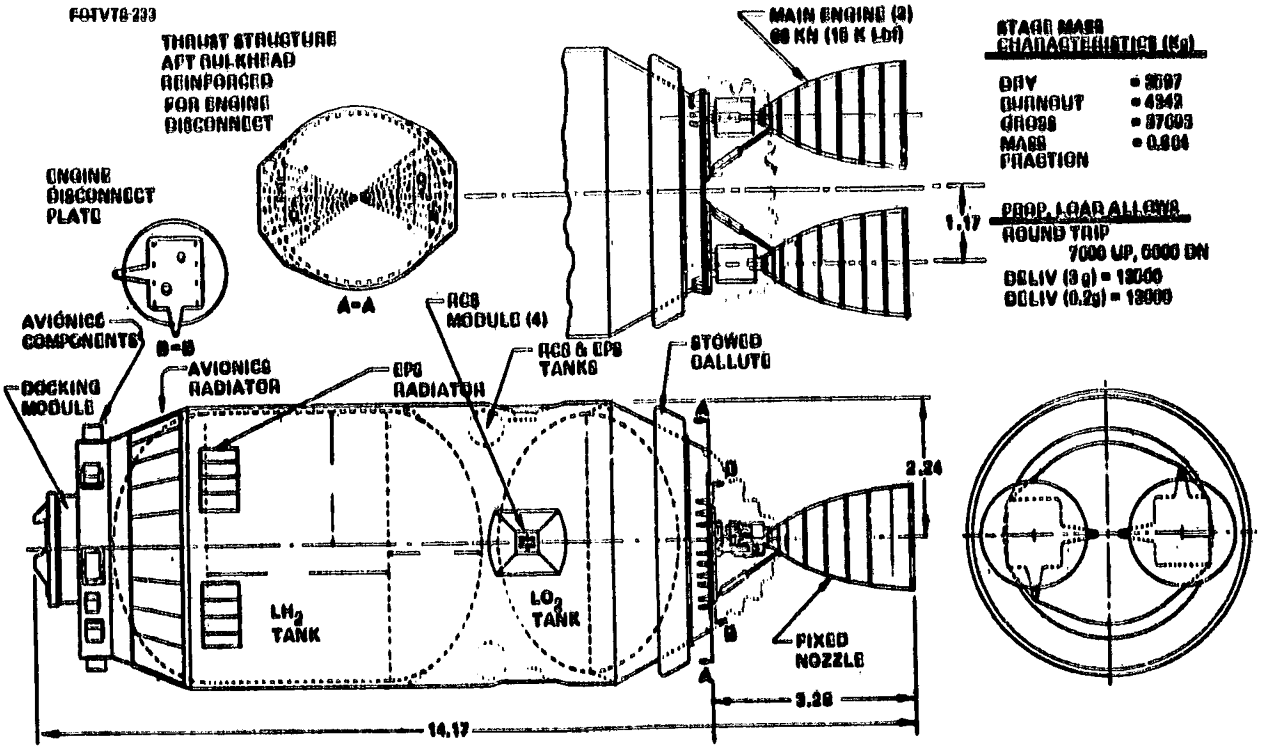

NOTE: ALL DIMENSIONS IN METERS

Figure 3.3.3-2 Space-Based OTV Configuration

body shell enclosing strut-supported main propellant tanks. The body shell is of composite design. Main propellant tanks are of 2219-T87 aluminum. Meteoroid/debris protection is provided by the combination of sandwich skin panels and a single-wall aluminum shield which is supported on standoffs located on the back side of the skin panels. A multilayer insulation (MLI) blanket around each main tank provides for propellant boiloff control. Electrical power is provided by O_2/H_2 fuel cells. A transpiration-cooled ballute is used to effect the GEO-to-LEO aeromaneuver. Space maintenance provisions allow for monitoring/replacement of selected critical components, including the main engine.

A summary mass statement is presented in table 3.3.3-1. The mass fraction of 0.8638 reflects the gross weight of 37 693 kg and the total main impulse propellant load of 32 560 kg (32 289 kg nominal, 271 kg reserve).

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

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Table 3.3.3-1 SB OTV Design Reference Mission Summary Mass Statement

ITEM	MASS (kg)
STRUCTURE	1447
THERMAL CONTROL -	124
AVIONICS	292
ELECTRICAL POWER SYSTEM (EPS)	234
MAIN PROPULSION SYSTEM (MPS)	681
ATTITUDE CONTROL SYSTEM (ACS)	128
SPACE MAINTENANCE PROVISIONS	214
WEIGHT GROWTH MARGIN	488
(DRY WEIGHT - LESS BALLUTE)	(3597)
RESIDUALS	413
RESERVES	332
(BURNOUT WEIGHT)	(4342)
BALLUTE	308
INFLIGHT LOSSES	382
FUEL CELL REACTANT	48
ATTITUDE CONTROL PROPELLANT	326
MAIN IMPULSE PROPELLANT	32,289
(OTV GROSS WEIGHT)	(37,693)
PAYLOAD	7687
(OTV + P/L WEIGHT	(45,380)
OTV MASS FRACTION	0.8638

1 GEO BASE SUPPORT
(CREW ROTATION/RESUPPLY)
7600 KG UP, 5000 KG DOWN

<u>Structure</u>—This group consists of the following: LH_2 and LO_2 tanks, body shell, docking/service/equipment/avionics assembly, thrust structure, meteoroid/debris shielding, and ballute installation fixed items. Total mass is 1447 kg.

<u>LH_2 and LO_2 Tanks</u>—The tanks containing the liquid hydrogen and liquid oxygen are all-welded 2219-T87 aluminum pressure vessels. The tank pressure shells are designed by room temperature pneumostatic proof test conditions corresponding to 0.4g initial acceleration with maximum ullage pressure of 152 kPa (22 psia). To satisfy a 45-mission service life requirement with minimal probability of leakage subsequent to a successful proof test and leak check, the tanks are designed using conservative fracture mechanics design data (i.e., "lower boundary" data in lieu of "best fit" data). Consistent with a lightweight design approach, the oxygen tank is not pressure cycled (purged) between missions. Refueling operation procedures, however, indicate a preference for a hydrogen tank which has been purged between missions. A minimum pressure shell thickness of 0.064 cm was incorporated. Fiberglass support struts are used to suspend the tanks from

the body shell. The LH_2 tank mass is 354 kg, 7.6% of the liquid hydrogen capacity. The mass of the LO_2 tank (no slosh baffling) is 163 kg, 0.58% of the liquid oxygen capacity.

Body Shell—The body shell is a composite structure consisting of honeycomb sandwich skin panels, tank-support rings, and miscellaneous equipment mounting and support structures. The sandwich skin panels incorporate 0.025-cm graphite/epoxy face sheets on a 0.2-cm-thick nomex core of 14.6 kg/m^3 density. This skin panel definition is considered minimum mass with respect to manufacturing considerations for a structure of this size. Total body shell mass is 394 kg.

Docking/Service/Equipment/Avionics Assembly—This structural/mechanical assembly incorporates the following: a 0.5m-high by 3.5m-diameter composite design ring assembly which provides for external mounting of equipment and avionics, a universal docking system, a peripheral latch/release system for payload accommodation, and service connector panels for fluids, gases, and electric power. Total assembly mass is 250 kg.

Thrust Structure—The thrust structure transmits loads from the two advanced space engines to the body shell. The structural assembly consists of a cone-frustum-shaped composite structure consisting of honeycomb sandwich skin panels (same definition as used for the body shell), a thrust distribution ring, and a thrust beam. The assembly attaches to the body shell at the LO_2 tank support ring. Total estimated mass is 95 kg.

Meteoroid/Debris Shielding—This group provides for the meteoroid/debris protection required in excess of that provided by the honeycomb skin panels of the body shell, equipment/avionics support section, and thrust structure. It consists of additional honeycomb skin panels to close off the forward and aft ends of the vehicle, plus a single-wall aluminum shield standoff installation located on the back side of all skin panels. In the LH_2 tank sidewell region, a 0.025-cm shield is located 6.4 cm from the sandwich skin panels. In all other regions, a 0.020-cm-thick shield is used, with a standoff distance of 7.6 cm. The masses of the added sandwich skin panels and of the internal shield installation are 37 kg and 109 kg, respectively.

Ballute Installation Fixed Items—An allowance of 45 kg has been incorporated for the ballute installation nonjettisonable items.

Thermal Control—Both active and passive techniques are used to provide thermal control. Thermal control of the fuel cells and of the avionics is provided by separate active thermal conditioning systems, each consisting of a fluid loop with a radiator (located on the body shell exterior) and associated pumps, valves, and control elements. Electric heaters are provided for ACS components and avionics equipment as required. MLI blankets enclose the LH_2 and LO_2 tanks. These blankets consist of layers (23 for LH_2 tanks, 15 for LO_2 tanks) of 3.75-μm double aluminized Kapton (DAK) radiation shields with Dacron[2] net spacers, enclosed within a protective bag of 12-μm DAK with rip-stop Dacron scrim attached. MLI blankets also enclose the fuel cell reactant tanks. Localized heat protection in the ACS thruster module region is provided by means of a highly polished thin gage shield of superalloy material. Mass estimates are: radiator systems, 54 kg; MLI blankets, 61 kg; miscellaneous, 9 kg.

Avionics—The avionics group includes elements for guidance and navigation, communications data mangement, rendezvous and docking, and data measurement. The guidance and navigation components (59 kg) include a laser gyro inertial measurement unit, a star scanner, and a global positioning system (GPS) receiver/processor, all of which are internally redundant. Included in the communications subsystem (35 kg) are redundant radiofrequency (RF) links which are NASA STDN/TDRS compatible. Deployable pairs of antenna pods are diametrically mounted on the avionics/equipment ring assembly. Each RF link contains a 20W S-band power amplifier and a STDN/TDRS transponder. The data management subsystem (89 kg) consists of two computers, a data bus system, and a signal interface unit. Rendezvous and docking components (46 kg) consist of a laser radar, a TV camera, and a high gain antenna. The instrumentation subsystem (63 kg) provides for monitoring of main propellant loading and usage and status monitoring of OTV subsystems. Total avionics mass is 292 kg.

Electrical Power System (EPS)—The primary power source is a set of lightweight design O_2/H_2 fuel cells, each rated at 2-kW nominal, 3.5-kW peak power. The fuel cells are actively redundant (i.e., both operating in the normal mode) with each fuel cell being capable of providing normal mission power. The O_2/H_2 reactant for the fuel cells is stored in the supercritical condition. A 25 A/hr nickel-hydrogen battery is provided for backup power and for smoothing of line transients. Distribution and control are provided by redundant power distribution units and a power transfer switch. System dry mass is estimated at 234 kg.

Main Propulsion System (MPS)—The two main engines are advanced space engines, each rated at a maximum vacuum thrust of 66 700N and providing a specific impulse of 485

[2]Dacron: registered trademark of E. I. duPont de Nemours and Company, Inc.

sec. The engines have fixed nozzles with 700:1 expansion ratio. The engines can also be operated in the tank head idle and pumped idle modes, with thrust levels of 300N and 6670N, respectively, and Isp of approximately 457 and 460 sec, respectively. The performance characteristics assumed are those which reflect a composite of the values provided by staged and expander cycle engines. Each engine has an estimated dry mass of 193 kg. Thrust vector control is provided by two electromechanical ball screw linear actuators per engine. Each actuator is equipped with redundant electric motor drive and has a mass of 8.5 kg. The propellant system consists of the following subsystems: propellant feed/fill/drain; autogeneous pressurant supply; main tank vent/relief; and helium-actuated pneumatics. Propellant system dry mass is 271 kg.

Attitude Control System (ACS)—The attitude control system uses hydrazine monopropellant pressurized by nitrogen gas supplied from a separate gas bottle. The ACS design uses 24 thrusters located in four modules positioned around the periphery of the OTV. The thrusters use a catalytic decomposition gas generator and produce 111N thrust each at 2208 kPa (320 psia) supply pressure. The propellant storage consists of five 0.53m-diameter titanium tanks, each having a storage capacity of 72.5 kg of hydrazine. Dry mass estimates are: thruster modules, 30 kg; hydrazine tanks and feed/fill/drain provisions, 83 kg; and nitrogen bottle and feed/relief/vent provisions, 12 kg.

Space Maintenance Provisions—Special interface provisions (structural/mechanical, fluid, electric, pneumatic, etc.) are provided for the removal of critical avionics assemblies, fuel cells, main engines, and thruster modules. To provide for expanded status monitoring, additional instrumentation and built-in test equipment are provided. Total estimated mass is 216 kg.

Weight Growth Margin—A margin allowance of 15% of subsystem dry weight has been incorporated. Total mass is 468 kg.

Residuals—This group consists of the fluids and gases onboard at end of mission under nominal conditions for EPS, MPS, and ACS. Total mass is 413 kg.

EPS Residuals—This subgroup consists of reactant trapped in the storage tanks, feed lines, and fuel cells, plus an allowance for trapped product water. Total mass is 7 kg.

MPS Residuals—This subgroup consists of the following: propellant trapped in main engines, propellant lines, and tank bottom sumps/propellant acquisition devices; bias fuel; gases in empty main tanks; and helium for pneumatic systems. Trapped propellant mass is

100 kg. A single-engine reference bias fuel allowance of 0.90% was used. For multiple engines, this allowance was factored by the inverse of the square root of the number of engines. The resulting LH_2 bias allowance for the two-engine installation is 0.64%. Bias fuel mass is 30 kg. Autogeneous pressurant conditions in the empty main tanks are: GO_2 at 148 kPa (21.5 psia) and 103°K (186°R); GH_2 at 140 kPa (20.3 psia) and 24.4°K (44°R). Gas masses are 141 kg of GO_2 and 99 kg of GH_2. The helium required for pneumatic control valve actuation reflects RL-10 technology and considers number of engines, number of engine firings, engine operating time, and engine shutdown time. The total helium mass is estimated at 18 kg.

ACS Residuals—This subgroup consists of the hydrazine trapped in the storage tank and feed lines, plus the nitrogen pressuran. ~quired to maintain the hydrazine at a supply pressure of 2208 kPa (320 psia). Total masses are 7 kg of hydrazine and 11 kg of nitrogen.

Reserves—Reserve allowances reflect the following: EPS, 100% loading of reactant tanks sized by the more stringent unmanned servicing mission; MPS, 2% of mission nominal idle delta-V requirement; ACS, 10% of nominal ACS propellant requirement. Reserve masses for EPS, MPS, and ACS are 28 kg, 271 kg, and 33 kg, respectively.

Ballute (Jettisonable)—An 18m-diameter transpiration-cooled ballute is used to effect the GEO-to-LEO aeromaneuver. The ballute is constructed of kevlar cloth. To provide a degree of porosity less than that of the basic Kevlar cloth, the cloth is coated with silicon rubber. The mass of the jettisonable ballute, including a 15% weight growth margin, is 308 kg.

Inflight Losses—This group consists of the following main propellant losses: boiloff losses for the 6-day mission, start-stop losses associated with nine firings of each of the two main rocket engines, losses for inflation and cooling of the ballute, and losses for main engine low thrust application during the aeromaneuver. Boiloff and start-stop losses total 244 kg, and aeromaneuver losses total 138 kg.

Nominal EPS Reactant—The nominal power requirement for the 6-day mission is 123 kW-hr. Based on a reactant power density of 2.7 kW-hr/kg, the nominal EPS reactant mass is 46 kg.

Nominal ACS Propellant—The ACS nominal delta-V budget for vehicle orientation and rendezvous/docking maneuvers is 43 m/s. Based on an average specific impulse of 220 sec for the hydrazine thruster and consideration of vehicle sequential mass, the nominal ACS propellant mass is 326 kg.

Nominal MPS Propellant—The MPS nominal delta-V budget for injection, coast, circularization, and trim maneuvers is 6316 m/s. This budget reflects a reduction due to ballute usage of 2213 m/s. Based on a specific impulse of 485 sec for the main engines and consideration of vehicle sequential mass, the nominal MPS propellant mass is 32 289 kg.

3.3.3.3 Airborne Support Equipment

The airborne support equipment (ASE) provides for the interfacing of the SB OTV to the launch vehicle which, in this case, is the SDV. The ASE consists of a structural assembly and interfacing pneumatic lines and cabling between OTV and SDV-provided interfaces. Total estimated mass is 430 kg.

The SB OTV is launched empty, without payload, in an inverted position (i.e., engines forward), cantilevered from an aft-located ASE structural assembly. The inverted position, relative to an upright position, allows a larger main engine dynamic envelope by being unconstrained by the ASE structural assembly. In addition, the inverted position eliminates the need for a latching interface at the aft end of the OTV because it utilizes the existing payload latching interface (including mechanisms and gas supply) located in the front end of the OTV. Cantilevering the OTV from the ASE structural assembly is practical because the OTV is launched empty and the resulting loads are well within its structural capability. Cantilevering is desirable because it eliminates the need for vehicle-mounted support trunnions and backup structures. (Small non-load-carrying trunnions are required to allow for deployment/retrieval of the OTV along a non-load-carrying rail system.)

Except for operation of its onboard pneumatics system to effect release from the ASE structural assembly, the OTV is in an inert condition until docked to SOC. During the total time period that the OTV is enclosed within the SDV shroud, the taking of vehicle measurements is limited to status monitoring of environmental data and OTV/ASE separation data. Processing of the data is accomplished by the SDV avionics.

3.3.4 Ground-Based OTV Description –

This section describes a large GB OTV sized for the same design reference missions as the SB OTV and a smaller GB OTV sized for less demanding missions. Two sizes of GB OTV's, compared to only a large GB OTV, provided a substantial reduction in the number of SDV launches required.

3.3.4.1 Operational Description

GB OTV's are normally launched fully fueled and w. .i their payloads. An alternative launch mode, however, is to launch the payload separately with integration of the OTV and payload occurring at a space base. The impact of launching with and without payload is discussed further in section 3.3.11.

The flight operations of the GB OTV are the same as for the SB OTV defined in section 3.3.3.1. Upon returning to LEO, the OTV would dock at the space base followed by placement within the launch vehicle recovery system for its return to Earth. Should a space base not be present, the OTV would rendezvous with and be returned to Earth by its launch vehicle which had been waiting in LEO.

Once back on Earth, all necessary maintenance is performed on the OTV and its ASE. Following checkout, the OTV, its airborne support equipment, and its payload (if appropriate) are mated and undergo integrated tests. The integrated assembly is then transferred to the launch pad and installed in the cargo bay of the launch system. Propellant loading of the launch vehicle and the OTV are accomplished at this time, followed by launch to LEO.

3.3.4.2 Configuration Description of Large Ground-Based OTV

The configuration of the large GB OTV, sized for the same missions as the SB OTV, is presented in figure 3.3.4-1 with overall geometry and physical characteristics noted. The large GB OTV is similiar in appearance to the SB OTV. Major differences are slightly larger main propellant tanks, a full diameter avionics/equipment ring assembly, and retractable nozzles on the main engines. The slightly larger tanks are necesary to accommodate an increase in main propellant mass of 861 kg (nominal plus reserve). The full diameter avionics/equipment ring assembly is a preferred configuration for payload accommodation during launch and ascent to LEO and for internal packaging of avionics/equipment. The retractable nozzles on the main engines are necessary to maximize payload length capability. The large GB OTV is nearly identical to the SB OTV with respect to all other aspects of overall configuration definition (number, type, and thrust of main engines; ACS propellant type; structural materials, methods of construction, and

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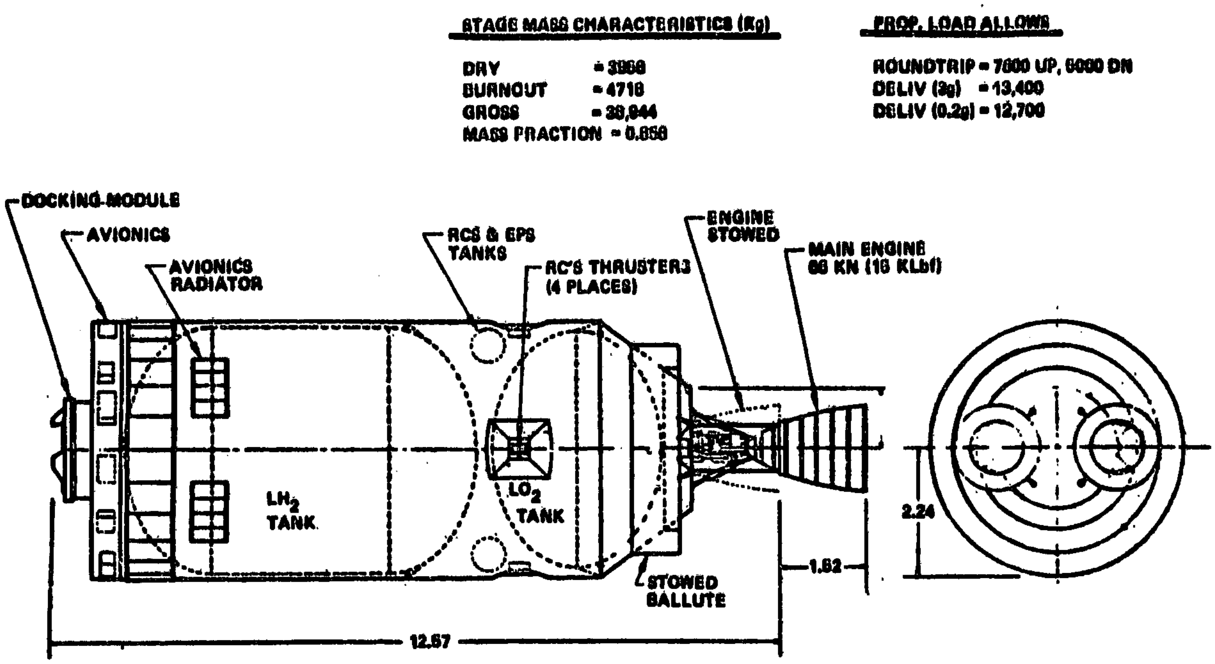

Figure 3.3.4-1 Large Ground-Based OTV Configuration

meteoroid/debris protection scheme; thermal control elements; electrical power source; ballute construction and materials; basic avionics; etc.). The major exception is provision for space maintenance of selected critical components, for which the large GB OTV has none.

A summary mass statement is presented in table 3.3.4-1. The mass fraction of 0.8582 reflects the gross weight of 38 944 kg and the total main impulse propellant load of 33 420 kg (33 142 kg nominal, 278 kg reserve).

Each of the items in the summary mass statement, exclusive of payload and airborne support equipment, is discussed in the following paragraphs, including definition of rationale for mass estimates. Pertinent comments concerning design differences of the large GB OTV relative to the SB OTV are incorporated. In those areas where the large GB OTV and SB OTV design definition and/or mass estimating criteria are identical, the reader is referred back to the SB OTV configuration description.

<u>Structure</u> - This group consists of the following: LH_2 and LO_2 tanks, body shell, docking/equipment/avionics assembly, thrust structure, meteoroid/debris shielding, and ballute installation fixed items. Total mass is 1774 kg.

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Table 3.3.4-1 Large GB OTV Design Reference Mission Summary Mass Statement

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ITEM	MASS (kg)
STRUCTURE	1774
THERMAL CONTROL	178
AVIONICS	292
ELECTRICAL POWER SYSTEM (EPS)	234
MAIN PROPULSION SYSTEM (MPS)	844
ATTITUDE CONTROL SYSTEM (ACS)	120
SPACE MAINTENANCE PROVISIONS	0
WEIGHT GROWTH MARGIN	616
(DRY WEIGHT - LESS BALLUTE)	(3958)
RESIDUALS	420
RESERVES	340
(BURNOUT WEIGHT)	(4718)
BALLUTE	316
INFLIGHT LOSSES	388
FUEL CELL REACTANT	46
ATTITUDE CONTROL PROPELLANT	334
MAIN IMPULSE PROPELLANT	33,142
(OTV GROSS WEIGHT)	(38,944)
PAYLOAD	7597
(OTV + P/L WEIGHT	(46,541)
AIRBORNE SUPPORT EQUIPMENT	2267 2
(LAUNCH WEIGHT)	(48,808)
OTV MASS FRACTION	0.8582

1 GEO BASE SUPPORT
(CREW ROTATION/RESUPPLY)
7500 KG UP, 5000 KG DOWN

2 ASSUMES LAUNCH BY SDV

<u>LH_2 and LO_2 Tanks</u> - The tanks containing the liquid hydrogen and liquid oxygen are all-welded 2219-T87 aluminum pressure vessels. The tank pressure shells are designed by room temperature pneumostatic proof test conditions corresponding to the flight condition which results in the maximum local pressure in the tank. For the hydrogen tank, the proof test condition corresponds to 0.4g initial acceleration with maximum ullage pressure of 152 kPa (22 psia)—the same condition used on the SB OTV hydrogen and oxygen tanks. For the oxygen tank, the proof test condition corresponds to 3g boost (Earth to LEO) with maximum ullage pressure of 124 kPa (18 psia). To satisfy a 45-mission service life requirement with low probability of leakage subsequent to a successful proof test and leakage check, the tanks are designed using median fracture mechanics design date (i.e., "best fit" data in lieu of "lower boundary" data). The SB OTV tanks were designed for

minimal probability of leakage and, hence, used conservative fracture mechanics design data (i.e., lower boundary data). Consistent with a lightweight design approach, the oxygen tank is not purged and repressurized prior to return to Earth. Safety procedures, however, dictate that the hydrogen tank be purged and repressurized prior to return to Earth. Subsequent to landing, both tanks are purged—the oxygen tank of its operational pressurant (gaseous oxygen) and the hydrogen tank of its reentry pressurant (helium). A minimum pressure shell thickness of 0.064 cm was incorporated. With respect to tank support, the hydrogen tank is suspended from the body shell by fiberglass struts—the same material used in the SB OTV hydrogen and oxygen tank struts. For the oxygen tank, a compromise between strut mass (as dictated by launch loads) and thermal heat leak indicates a preference for graphite/epoxy struts. the LH_2 mass is 345 kg, 7.2% of the liquid hydrogen capacity. The mass of the LO_2 tank (no slosh baffling) is 245 kg, 0.85% of the liquid oxygen capacity.

Body Shell - The body shell is a composite structure consisting of honeycomb sandwich skin panels, tank support rings, and miscellaneous equipment mounting and support structures (items common to the SB OTV), plus forward main support trunnions and ring, aft ring with interface latching provisions (for attachment to the ASE structural assembly), and aft-mounted service connector panels for fluids, gases, and electrical power (via the ASE). The sandwich skin panels incorporate 0.025-cm graphite/epoxy face sheets on a 0.2-cm-thick Nomex core of 14.6 kg/m^3 density. This skin panel definition, though considered minimum mass with respect to manufacturing considerations for a structure of this size, has sufficient strength (the positive margins are small) with respect to launch loads considerations. This is the same skin panel definition as used on the SB OTV. Total body shell mass is 619 kg.

Docking/Equipment/Avionics Assembly - This structural/mechanical assembly incorporates the following: a 0.5m-high by a 4.48m-diameter composite design ring assembly which provides for internal mounting of equipment and avionics, a universal docking system, and a peripheral latch/release system for payload accommodation. Total assembly mass is 259 kg.

Thrust Structure - This composite structural assembly, compared to the SB OTV, is similar in design detail, slightly different in geometry (smaller base region diameter associated with smaller expansion ratio engines), but equal in mass at 95 kg.

Meteoroid/Debris Shielding - Design definition is identical to that of the SB OTV. The masses of the added sandwich skin panels and of the internal shield installation are 52 kg and 114 kg, respectively.

Ballute Installation Fixed Items - An allowance of 45 kg (same as for SB OTV) has been incorporated for the ballute installation nonjettisonable items.

Thermal Control - Design definition is identical to that of the SB OTV with the exception that provisions have been incorporated to allow for MLI purge (liftoff/ascent) and repressurization (descent). Mass estimates are: radiator systems, 54 kg; MLI blankets, 61 kg; MLI purge/repress provisions, 54 kg; miscellaneous, 9 kg.

Avionics - Design definition and mass (292 kg) are identical to that of the SB OTV.

EPS - Design definition and dry mass (234 kg) are identical to that of the SB OTV.

MPS - Design definition is identical to that of the SB OTV except for incorporation of the following: retractable (in lieu of fixed) nozzles to maximize payload length; smaller nozzle expansion ratio (626 versus 700) to allow for nozzle clearance during separation from the ASE structural assembly; linkage-type thrust vector control (TVC) actuator support struts associated with use of retractable nozzles; larger diameter drain lines to accommodate an ascent phase emergency dump of main propellant (derived from consideration of the growth shuttle as a launch vehicle); and main tank vent/relief provisions for ground/ascent phases. Each engine has an estimated dry mass of 197 kg. Each TVC actuator installation has a mass of 11.5 kg. Propellant system dry mass is 404 kg.

ACS - Design definition is identical to that of the SB OTV except that the large GB OTV does not require special provisions for tank fill and drain in space. Dry mass of the hydrazine ACS is 120 kg.

Space Maintenance Provisions - The large GB OTV has none.

Weight Growth Allowance - A margin allowance of 15% of subsystem dry mass (same as used on SB OTV) has been incorporated. Total mass is 516 kg.

Residuals - Mass estimating criteria for EPS, MPS, and ACS residuals are identical to those of the SB OTV. Residual masses are: EPS, 28 kg (no change); MPS, 395 kg (7 kg higher); ACS, 18 kg (no change).

Reserves - Mass estimating criteria for EPS, MPS, and ACS reserves are identical to those of the SB OTV. Reserve masses are: EPS, 28 (no change); MPS, 279 kg (8 kg higher); ACS, 33 kg (no change).

Ballute (Jettisonable) - An 18.3m-diameter ballute is used to effect the GEO-to-LEO aeromaneuver. The ballute design definition is identical to that of the SB OTV. Ballute mass is 316 kg (8 kg higher due to use of slightly larger ballute).

Inflight Losses - Mass estimating criteria are identical to those of the SB OTV. The inflight losses mass is 388 kg (2 kg higher boiloff losses, 4 kg higher aeromaneuvering losses).

Nominal EPS, ACS, MPS Propellant - Mass estimating criteria are identical to those of the SB OTV except for incorporating a 1-sec reduced main engine specific impulse (484 versus 485) due to use of a lower nozzle expansion ratio (626 versus 700). Nominal propellant masses are: EPS, 46 kg (no change); ACS 326 (8 kg higher); MPS, 33 142 kg (853 kg higher).

3.3.4.3 Airborne Support Equipment for the Large Ground-Based OTV

The airborne support equipment provides for the interfacing of the large GB OTV to the launch vehicle (SDV). The ASE consists of a structural assembly, fluids systems, electronics/avionics, batteries, and cabling. Total estimated mass is 2267 kg.

The large GB OTV is launched with propellant, in an upright position, with payload attached. An aft-located ASE structural assembly transmits axial loads and aft lateral loads to the SDV. Forward lateral loads are transmitted to the SDV shroud via support trunnions located just forward of the LH_2 tank. The trunnions interface with latch/release fittings located on a major support ring incorporated into the shroud design. (The SDV shroud design also incorporates a non-load-carrying rail system to allow for deployment/retrieval of the OTV via the shroud nose section). Release of the OTV from the fixed ASE structural assembly is accomplished pneumatically.

The ASE fluids system consists of SDV to OTV fill, drain, vent pneumatic lines, umbilicals, and helium gas storage. The helium gas is used for ASE pneumatic functions (valve control) and for repressurization of the OTV LH_2 tank prior to return to Earth.

The electrical/avionics system provides for backup electrical power and control, ASE status monitoring command and control, cabling, and interfacing cabling between

OTV, payload systems, and SDV-provided data interfaces. A power control unit is provided to select ground power (SDV-supplied) or SDV power. The ASE batteries float online to prevent power dropout during periods when the ASE is powered up.

3.3.4.4 Configuration Description of Small Ground-Based OTV

The configuration of the small GB OTV is presented in figure 3.3.4-2 with overall geometry and physical characteristics noted. Basically, the small GB OTV is a shortened,

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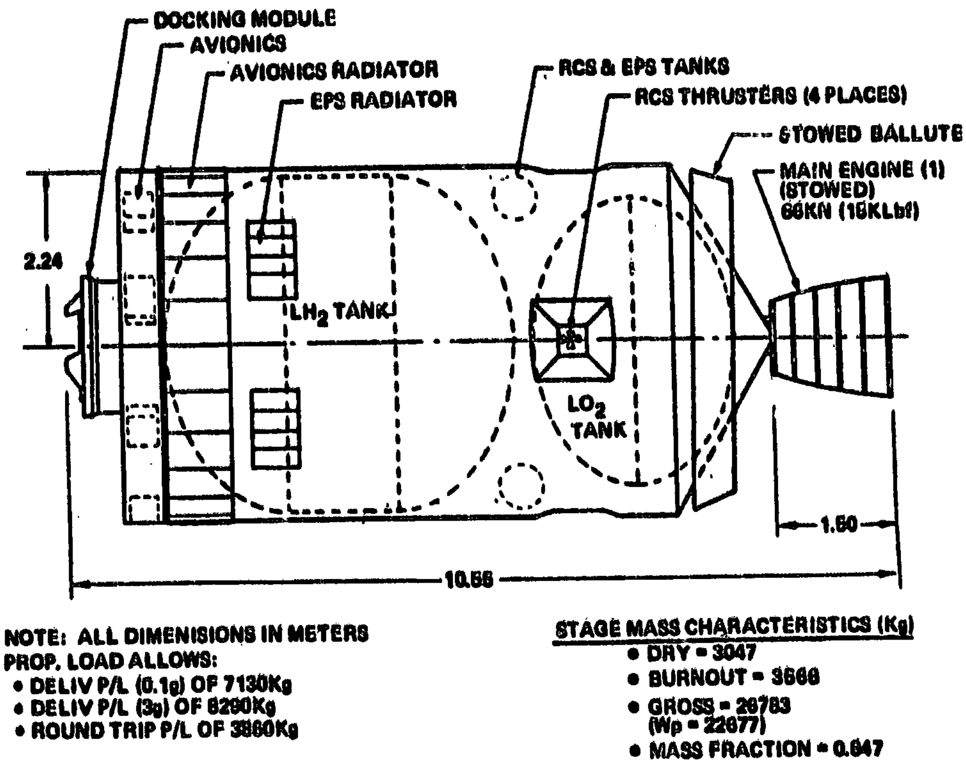

Figure 3.3.4-2 Small Ground-Based OTV Configuration

single-engine version of the large GB OTV. The main engine has a nozzle expansion ratio of 700 versus 626 for the large GB OTV. This expansion ratio (same as SB OTV) is permissible because adequate nozzle clearance exists during separation from the ASE structural assembly.

A summary mass statement for the small GB OTV is presented in table 3.3.4-2. The mass fraction of 0.8467 reflects the gross weight of 26 783 kg and the total main impulse

Table 3.3.4-2 Small GB OTV Summary Mass Statement—Unmanned Servicing Mission

FOTVT8-308

ITEM	MASS (KG)		
	BASIC VEHICLE	TWIN-LAUNCH CONFIGURATION	
		VEH.1(FWD.)	VEH.2(AFT)
STRUCTURE	1407		
THERMAL CONTROL	160		
AVIONICS	271		
ELECTRICAL POWER SYSTEM (EPS)	234		
MAIN PROPULSION SYSTEM (MPS)	402		
ATTITUDE CONTROL SYSTEM (ACS)	96		
WEIGHT GROWTH MARGIN	398		
(OTV DRY WEIGHT-LESS BALLUTE)	(3047)		
RESIDUALS	293		
RESERVES	226		
(OTV BURNOUT WEIGHT)	(3566)		
BALLUTE	201		
INFLIGHT LOSSES	244		
FUEL CELL REACTANT	46		
ATTITUDE CONTROL PROPELLANT	226		
MAIN IMPULSE PROPELLANT)	22,501		
(OTV GROSS WEIGHT)	(26,783)	(26,783)	(26,783)
PAYLOAD	6070	- -	- -
(OTV + P/L WEIGHT)	(31,853)	(N/A)	(N/A)
AIRBORNE SUPPORT EQUIPMENT		2041	1587
BALLAST			
(LAUNCH WEIGHT)		$\underbrace{(28,824) \quad (28,370)}_{(57,194)}$	
OTV MASS FRACTION	0.8467		

N/A - NOT APPLACABLE

propellant load of 22 677 kg (22 501 kg nominal, 176 kg reserve). The mass statement also reflects the fact that two small GB OTV's are launched together. Their payloads are launched separately, with mating occurring at SOC. This approach was found to reduce the number of launches significantly.

3.3.4.5 Airborne Support Equipment for the Small Ground-Based OTV

Airborne support equipment elements for the small GB OTV are similar to those of the large GB OTV. However, because of the twin-launch feature, the forward ASE structural assembly is a more complex design than the aft assembly. The increased complexity arises because the forward ASE must be deployed along with the forward OTV, separated from the forward OTV, stood off by means of a manipulator arm, and retrieved subsequent to deployment of the aft OTV. The masses of the forward and aft ASE, respectively, are estimated at 90% (2041 kg) and 70% (1587 kg) of the ASE mass for the large GB OTV.

3.3.5 Performance

3.3.5.1 Requirements

For the purpose of the performance analyses of this study, two destinations were considered: geosynchronous orbit at 0-deg inclination and planetary escape at a C_3 of 55 km^2/sec^2. Other Earth orbit missions were treated in terms of their GEO equivalents. Low Earth orbit departure altitude was 370 km (200 n. mi.) and the inclination was 28.5 deg, the orbit of the SOC. All missions were assumed to be staged from the SOC and the OTV's returned to the SOC, although some ground-based missions were capable of direct departure from the launch vehicle.

The types of missions analyzed consisted of (1) single-stage payload delivery to GEO, (2) two-stage payload delivery to GEO, (3) manned resupply to GEO, and (4) multiple satellite servicing at GEO.

LEO-GEO mission delta-V requirements were based on Hohmann transfer trajectories modified to include multiple perigee burns, finite burn delta-V losses, and aerobraking on the return to LEO. The ideal velocity requirements for the LEO-GEO transfer, including a 2.5-deg plane change during the perigee burn and a 26-deg plane change during the apogee burn, are: perigee delta-V = 2437.0 m/sec; apogee delta-V = 1770.9 m/sec.

Multiple perigee burns were used for phasing and to reduce losses for missions with maximum acceleration limits. The delta-V losses from the ideal were derived from previous study trajectory analysis.

The OTV Concept Definition Study (ref. 1) demonstrated the attractiveness of aerobraking to improve performance at low cost. This concept is based on use of the Earth's atmosphere to reduce vehicle velocity, largely eliminating the rocket burn required to enter low Earth orbit when returning from GEO or other high orbits. The net savings, amounting to 2250 m/sec, is accomplished by grazing the upper atmosphere and converting the vehicle's kinetic energy to heat through friction (i.e., aerodynamic drag). This must be done in a precise manner to avoid losing too much velocity and reentering or losing too little velocity and coasting back out to high orbit. This maneuver is complicated by navigation inaccuracies for density variations as high as 50%. An

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inflatable ballute device is used to provide aerobraking capability. Key operations associated with the aerobraking maneuver are shown in figure 3.3.5-1.

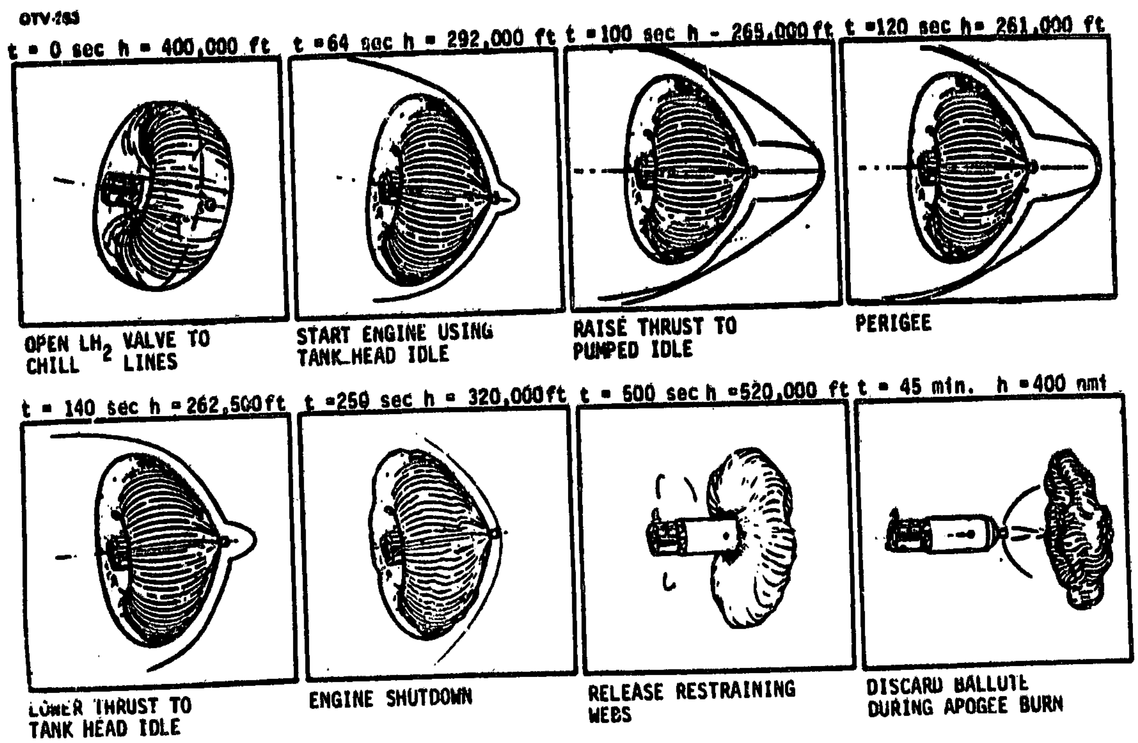


Figure 3.3.5-1 AB OTV Operating Scenario

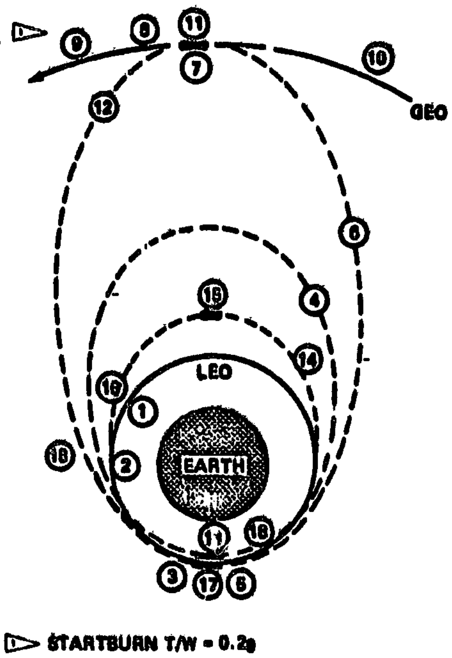
The basic design approach for the aerobraked OTV (AB OTV) concept makes minimum changes from the all-propulsive vehicle. In fact, the AB OTV is essentially an all-propulsive OTV with an aerobraking kit added. This kit consists of GPS navigation to improve perigee altitude accuracy and a large expendable ballute with its attachment and deployment hardware. In operation, the ballute is deployed prior to reentry maneuver and surrounds the vehicle and its payloads, with the exception of the main engine. The vehicle is oriented so that the engine is facing forward and is aerodynamically stable in this position. The unique capabilities of the concept are achieved by operating the engine during the maneuver at low thrust levels. Under thse low thrust conditions, the rocket exhaust gases act as an aerodynamic spike extending forward of the drag body that significantly affects its drag. Varying the thrust level varies the effecive spike length and affords a means for varying vehicle drag to control flightpath. The rocket exhaust gases shield the ballute surface from the shock-heated oncoming air and the exhaust gas momentum also moves the bow shock and stagnation point well off the vehicle. The engine, therefore, maintains a compatible environment for itself and for the ballute. The feasibility of the concept has been verified in wind tunnel testing.

The mission profile for the manned GEO base resupply mission is shown in table 3.3.5-1 along with the timeline and the delta-V requirements. Following separation from

Table 3.3.5-1 Aeroassist, Single-Stage Typical Mission Profile

FOTVTS-340

	EVENT	ELAPSED TIME (HRS)	ΔV(MPS)
1	SEPARATE	4.0	3.0
2	PHASE IN LEO	7.0	0
3	PHASING ORBIT INJECTION BURN	7.2	1370
4	COAST	10.2	0
5	TRANSFER ORBIT INJECTION BURN	10.3	1068
6	COAST & MIDCOURSE CORRECTION BURN	16.3	18.0
7	GEO CIRCULARIZATION BURN	16.4	1771.0
8	TRIM BURN	17.4	8.1
9	RENDZVOUS & DOCK	116.6	21.3
10	PHASE IN GEO	127.3	0
11	TRANSFER ORBIT INJECTION BURN	127.4	1846
12	COAST & MIDCOURSE CORRECTION BURN	132.4	19.8
13	AEROBRAKING MANEUVER & JETTISON BALLUTE	132.6	0
14	COAST	133.3	0
15	RAISE PERIGEE BURN	133.3	67
16	COAST	134.1	0
17	LEO CIRCULARIZATION BURN	134.1	122
18	RENDZVOUS & DOCK	140.6	18.3
19	RESERVES	140.6	137
20	UNLOAD P/L	140.6	-



the SOC, the OTV phases in LEO until the nodal crossing. The first of the two perigee burns is then performed. The magnitude of this delta-V determines the period of this intermediate orbit and allows further phasing. After one revolution in the phasing orbit, the second perigee burn provides the remainder of the delta-V necessary to inject into the LEO-GEO transfer orbit. A midcourse correction is performed during the coastout to GEO. Following circularization at GEO, a trim burn achieves the desired destination orbit parameters. The midcourse correction and GEO trim burns are performed by the main engine operating in idle mode. Delta-V to rendezvous and dock with the GEO station is provided by the ACS. After separation from the station and phasing, the transfer orbit injection burn places the OTV in a GEO-LEO transfer orbit with the perigee within the atmosphere so that aerobraking can be used to reduce the velocity to near LEO circular velocity.

A midcourse correction burn increases the perigee altitude accuracy to the necessary level for aerobraking. Upon completion of the aerobraking maneuver, the ballute is jettisoned and the OTV coasts to an apogee altitude slightly greater than LEO for phasing. At apogee, a burn is performed to raise the perigee out of the atmosphere and up to the LEO altitude. A last burn at perigee circularizes the OTV at LEO.

Tables 3.3.5-2 and -3 present the timelines for the single-stage and two-stage GEO payload delivery missions. A third perigee burn is added for the two-stage mission; the

Table 3.3.5-2 GEO Delivery Timeline and Delta-V Single-Stage SB OTV

POTV78-433

EVENT	DURATION (HR)	ELAPSED TIME (HR)	ΔV(MPS)	
SEPARATE	4.0	4.0	3	(RCS)
PHASE	3.0	7.0	0	
PHASE INJECTION	.2	7.2	1370	
COAST	3.0	10.2	0	
TRANSFER INJECTION	.1	10.3	1098	
COAST	5.0	15.3	15	(PHI)
GEO CIRC.	.1	15.4	1721	
TRIM	12.0	27.4	9	(RCS)
UNLOAD P/L	1.0	28.4	0	
PHASE	10.4	38.8	0	
TRANSFER INJECTION	.1	38.9	1845	
COAST	5.0	43.9	20	(PHI)
AEROMANEUVER	.1	44.0	0	
COAST	.8	44.8	0	
LEO INJECT	-	44.8	67	
COAST	.8	45.6	0	
LEO CIRC.	-	45.6	122	
RENDEZVOUS & DOCK	6.5	52.1	18	(RCS)
RESERVES		52.1	137	

first is provided by the booster stage and the second and third provide the remainder of the LEO-GEO transfer orbit delta-V. The split between the second and third burns is varied in the same manner as for the single-stage missions to allow appropriate phasing.

Ground- and space-based mission profiles differ only in the additional time prior to separation and during phasing required for deployment of the ground-based OTV's from the launch vehicle if SOC is not used.

Planetary missions were performed as shown in figure 3.3.5-2. The OTV injects the payload into the hyperbolic Earth escape orbit then separates from the payload and brakes into an elliptical geocentric orbit. At the apogee of the elliptical orbit, a small burn lowers the perigee into the atmosphere so that aerobraking can be used to circularize at

Table 3.3.5-3 Two-Stage SB OTV Timeline and Delta-V Budget

FOTVTG-355

(GEO DELIVERY MISSION)

	ELAPSED TIME (HRS.)	ΔV(M/S)
1. SEPARATE	4.0	3
2. PHASE	7.0	0
3. BOOST	7.2	915
4. COAST	10.2	
5. PHASE INJECT	10.4	V*
6. STAGE BOOSTER	14.4	0
7. TRANS. INJECT	14.6	353-V*
8. COAST	19.6	15
9. GEO. CIRC	19.8	1771
10. TRIM	33.8	7
(11. UNLOAD P/L)	34.8	0
12. PHASE	43.2	0
13. TRANS. INJECT	43.3	1844
14. COAST	48.3	20
15. AEROMANEUVER	48.4	0
16. COAST	49.2	0
17. INJECT	49.2	67
18. COAST	50.0	0
19. LEO CIRC.	50.0	122
20. REND. & DOCK	56.5	18
21. RESERVES	56.5	137
22. UNLOAD P/L	56.5	-
BOOSTER		
1. SEPARATE	4.0	3
2. PHASE	7.0	0
3. BOOST	7.2	914
4. COAST	10.2	0
5. PHASE INJECT	10.4	V*
6. STAGE	14.4	20
7. AEROMANEUVER	14.5	0
8. COAST	15.3	0
9. PHASE INJECT	15.3	67
10. COAST	16.1	0
11. LEO CIRC.	16.1	122
12. REND. & DOCK	22.6	18
13. RESERVES	22.6	61

*THIS ΔV VARIES WITH THE STARTMISSION MASS AND REPRESENTS THE REMAINDER OF THE BOOSTER ΔV CAPABILITY.

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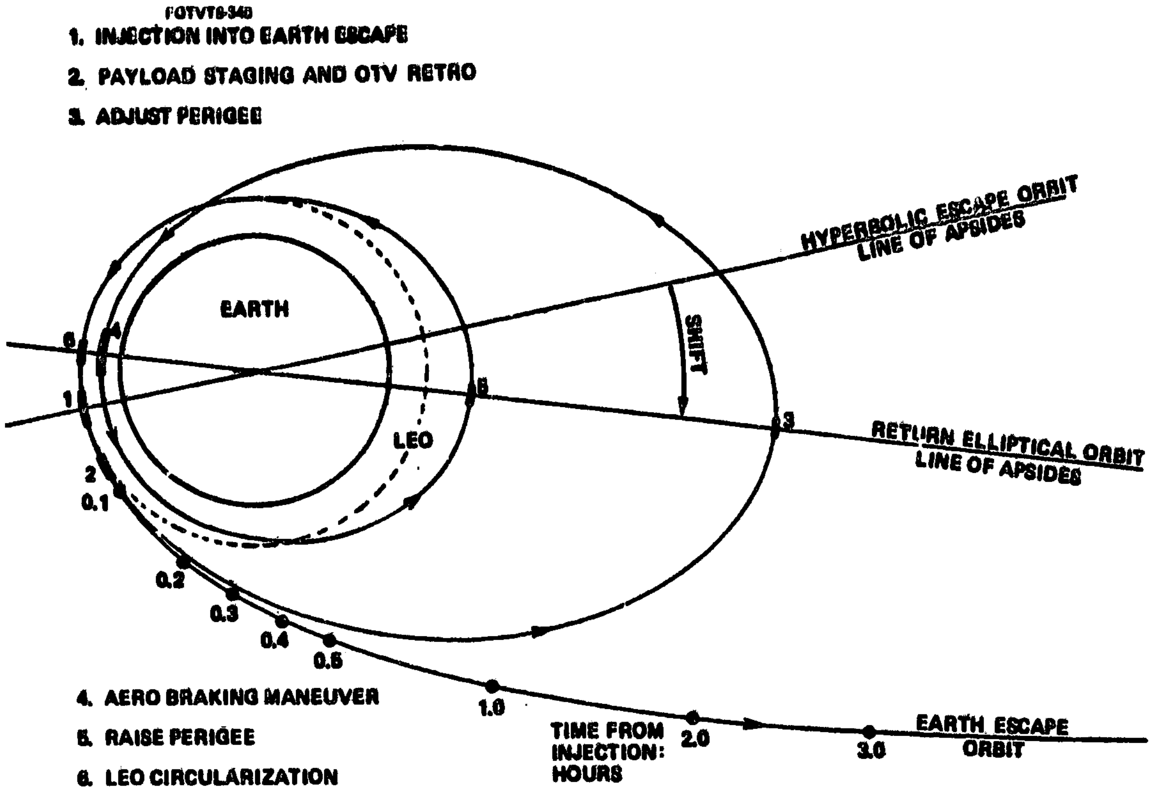

Figure 3.3.5-2 Planetary Mission Profile

LEO. The remainder of the mission is similar to the LEO-GEO missions. The shift in line of apsides is due to the retro delta-V occurring after perigee of the original orbit because of the time required for staging and reorientation. Table 3.3.5-4 presents a timeline and delta-V budget for the planetary boost mission.

3.3.5.2 Analysis

Stage sizing was accomplished in a two-step process. Preliminary mass trending relationships were developed by updating existing OTV data for FOTV characteristics. These relationships and the mission sequences described in the requirements section were input into our performance estimation program. The results were a series of parametric relationships between stage payload capability and propellant capacity for the different mission types. Comparing these to the required payloads for each mission type identified the sizing missions and determined the required stage propellant capacity. Point designs were then developed for the selected sizes to update the mass characteristics. Finally,

Table 3.3.5-4 Planetary Mission Timeline and Delta-V

EVENT	ELAPSED TIME (HR)	ΔV(MPS)
SEPARATE	4.0	3
PHASE	7.0	0
ESCAPE INJECTION	7.3	5482
STAGING	7.5	3
RETRO INJECTION	7.6	2800
COAST	53.1	0
TRANSFER INJECT	53.1	22
COAST	98.6	20
AEROMANEUVER	98.7	0
COAST	99.5	0
PHASE INJECTION	99.5	67
CCAST	100.3	0
LEO CIRCULARIZATION	100.3	122
RENDEZVOUS & DOCK	106.8	18
RESERVES		76

performance analyses for the selected stage were performed to verify payload capabilities.

Mass trending relationships used as inputs to develop performance parametrics are shown in figure 3.3.5-3. These relationships were developed by evaluating preliminary point designs at 31 750 kg and 58 970 kg propellant capacities. The design features and characteristics for the space-based stages are described in section 3.3.3.2 and for the ground-based, in sections 3.3.4.2 and 3.3.4.4.

The Payload and Sequential Mass Calculation (PSMC) program was used to determine parametric payload capabilities of the ground- and space-based OTV's. Given a stage burnout mass and propellant capacity, PSMC calculates propellant consumption, losses, and stage mass for each event in the mission profile. Payload and start mission mass are iterated until calculated propellant consumption and burnout mass match the specified values. The program incorporates a complete mission profile of time and delta-V for each event. The type of burn, either reaction control system (RCS) or main engine, and corresponding start-stop losses can be specified. Boiloff and EPS losses are calculated from the timeline and specified loss rates. The loss rate is specified as a function of propellant capacity to handle different stage sizes. A detailed mission

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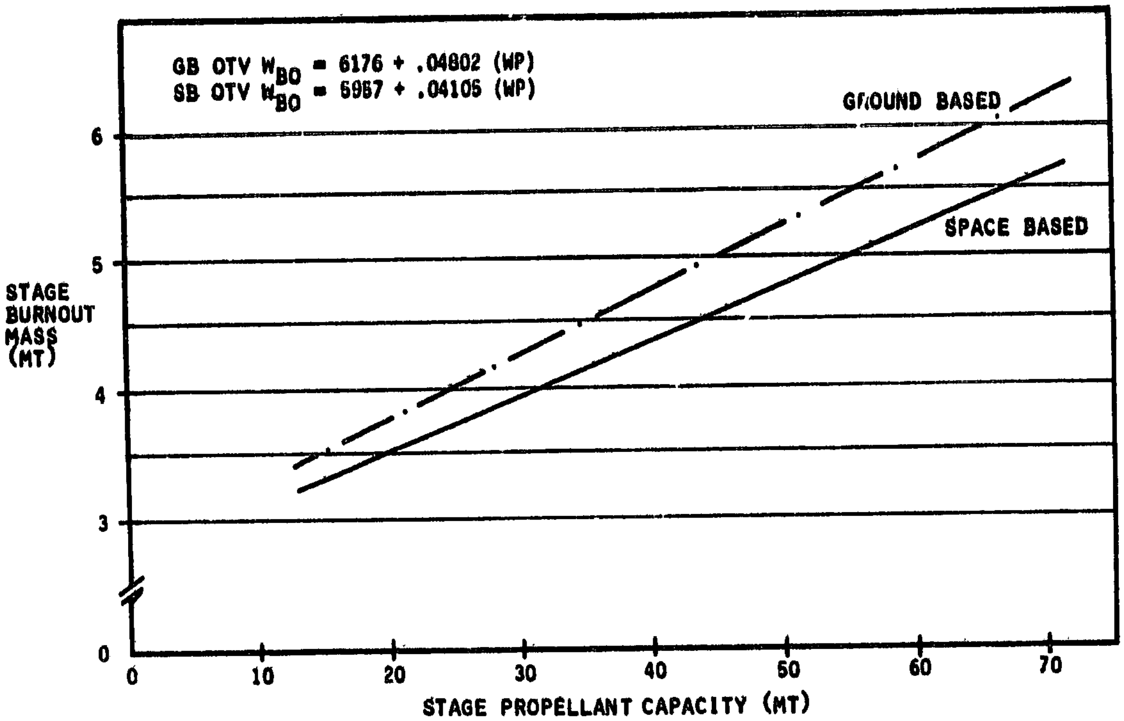

Figure 3.3.5-3 FOTV Mass Trending

sequential mass statement listing event, delta-V, propellant usage, losses, and mass is printed along with a summary mass statement.

Preliminary performance analysis had indicated the driving missions for stage sizing were the crew rotation/resupply of a GEO base and the two-stage GEO payload delivery. The required payloads were 31.75t for the two-stage delivery and 7.6t up with 5t down for the manned resupply. Figures 3.3.5-4 and 3.3.5-5 show parameteric performance for these missions for ground- and space-based OTV's. The space-based OTV is sized by the two-stage delivery at 32.57t propellant capacity and the ground-based, by the manned mission at 33.43t.

SB OTV sequential mass statements for the different missions are presented in tables 3.3.5-5(a) through (e). Mass statements for the GB OTV are essentially the same although heavier by approximately 1000 kg at startburn and 350 kg at burnout.

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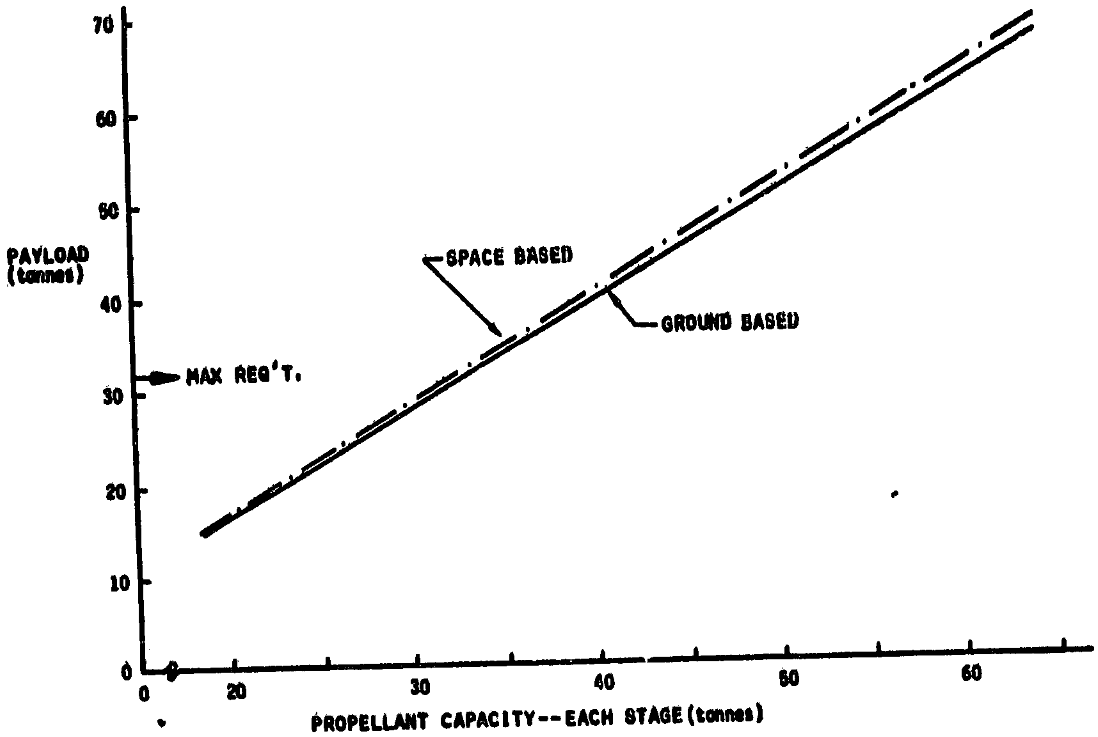

Figure 3.3.5-4 Two-Stage Parametric Performance—GEO Delivery

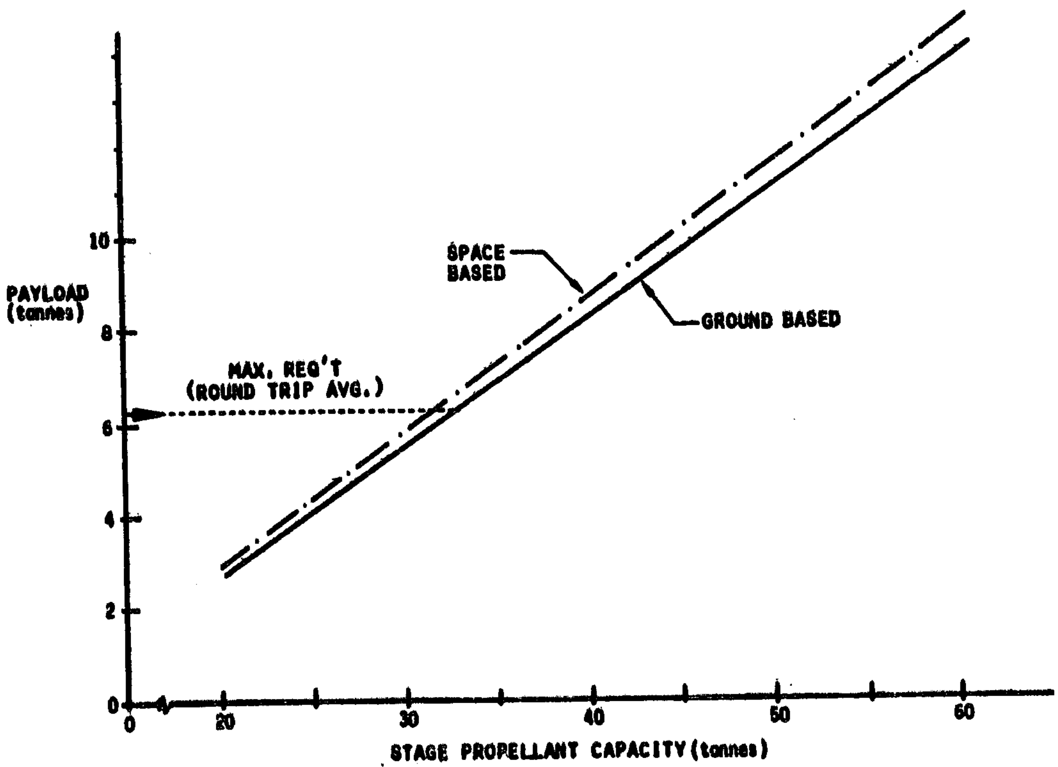

Figure 3.3.5-5 Single-Stage Parametric Performance—GEO Round Trip

Table 3.3.5-5(a) SB OTV Manned GEO Resupply Mission

	kg
Usable Main Prop Mass	32 568
Nominal Burnout Mass	4 125
Start Mission Mass	45 464
Payload Mass	5 089

Main Engine Isp = 485
Aux Prop Isp = 220

Event	Delta-V (m/s)	Prop Usage (kg)	Losses (kg)	Mass (kg)
Start Burn	---	---	---	45 476
Separate	3	64	3	45 409
Phase	0	0	2	45 407
Phase Inject	1370	11 360	42	34 003
Coast	0	0	2	34 001
Trans Inject	1098	7 007	17	26 968
Coast	15	98	20	26 850
GEO Circ	1771	8 347	17	18 486
Trim	9	40	18	18 428
Rend & Dock	21	181	71	15 577
Phase	0	0	7	15 569
Trans Inject	1844	5 006	17	10 548
Coast	20	50	20	10 478
Aeromaneuver	0	0	138	10 340
Coast	0	0	309	10 031
Inject	67	140	17	9 874
Coast	0	0	1	9 873
LEO Circ	122	250	17	9 606
Rend & Dock	18	81	5	9 521
Reserves	137	271	0	9 217
Unload Payload	---	---	5091	4 128

		kg
Nominal Main Propellant	=	32 295
Reserve Main Propellant	=	271
Nominal Aux Propellant	=	327
Reserve Aux Propellant	=	33
Total Losses	=	414

Table 3.3.5-5(b) SB OTV Two-Stage GEO Delivery

	kg
Usable Main Prop Mass	32 568
Booster Main Prop Mass	32 568
Nominal Burnout Mass	4 125
Booster Burnout Mass	4 125
Start Mission Mass	106 346
Payload Mass	31 761

Main Eng Isp = 485
Aux Prop Isp = 220

Event	Delta-V (m/s)	Prop Usage (kg)	Losses (kg)	Mass (kg)
Start Mission	---	---	---	106 346
Separate	3	150	6	106 191
Phase	0	0	4	106 186
Boost	1006	20 241	57	85 889
Coast	0	0	4	85 885
Phase Inject	772	12 093	23	73 769
Coast	0	0	3	69 063
Trans Inject	923	12 181	42	56 840
Coast	15	210	20	56 610
GEO Circ	1771	17 598	17	38 994
Trim	9	165	25	38 804
Unload Payload	0	0	1	7 043
Phase	0	0	7	7 035
Trans Inject	1844	2 261	17	4 757
Coast	20	23	20	4 714
Aeromaneuver	0	0	62	4 652
Coast	0	0	139	4 512
Phase Inject	67	63	17	4 433
Coast	0	0	1	4 432
LEO Circ	122	122	17	4 303
Rend & Dock	18	36	5	4 202
Reserves	137	121	0	4 121

	kg
Nominal Main Propellant	= 32 449
Reserve Main Propellant	= 121
Nominal Aux Propellant	= 201
Reserve Aux Propellant	= 20
Total Losses	= 262

Table 3.3.F-5(e) Two-Stage Delivery Booster Return

Event	Delta-V (m/s)	Prop Usage (kg)	Losses (kg)	Mass (kg)
Stage	20	23	26	4655
Aeromaneuver	0	0	62	4594
Coast	0	0	137	4456
Phase Inject	67	62	23	4371
Coast	0	0	1	4371
Leo Circ	122	111	23	4237
Rend & Dock	18	36	5	4197
Reserves	61	53	0	4125

	kg
Booster Nominal Aux Propellant =	196
Booster Reserve Aux Propellant =	19

Table 3.3.5-5(d) SB OTV GEO Delivery Mission

	kg
Usable Main Prop Mass	32 568
Nominal Burnout Mass	4 125
Start Mission Mass	50 830
Payload Mass	13 526

Main Eng Isp = 483
Aux Prop Isp = 220

Event	Delta-V (m/s)	Prop Usage (kg)	Losses (kg)	Mass (kg)
Start Burn	---	---	---	50 830
Separate	3	72	3	50 755
Phase	0	0	2	50 753
Phase Inject	1370	12 700	42	38 011
Coast	0	0	2	38 009
Trans Inject	1098	7 834	17	30 157
Coast	15	109	20	30 027
GEO Circ	1771	9 335	17	20 676
Trim	9	87	9	20 580
Unload Payload	0	0	1	7 053
Phase	0	0	7	7 046
Trans Inject	1844	2 265	17	4 764
Coast	20	22	20	4 721
Aeronameuver	0	0	62	4 659
Coast	0	0	139	4 520
Inject	67	63	17	4 439
Coast	0	0	1	4 439
LEO Circ	122	112	17	4 310
Rend & Dock	18	36	5	4 269
Reserves	137	121	0	4 128

		kg
Nominal Main Propellant	=	32 442
Reserve Main Propellant	=	121
Nominal Aux Propellant	=	196
Reserve Aux Propellant	=	20
Total Losses	=	259

Table 3.3.5-5(e) SB OTV for Planetary Mission

	kg
Usable Main Prop Mass	32 568
Nominal Burnout Mass	4 125
Start Mission Mass	41 799
Injected Mass	4 690

Main Eng Isp = 485
Aux Prop Isp = 220

Event	Delta-V (m/s)	Prop Usage (kg)	Losses (kg)	Mass (kg)
Start Mission	---	---	---	41 799
Separate	3	59	5	41 735
Phase	0	0	8	41 727
Escape Inject	5481	28 545	22	13 160
Staging	3	19	0	8 451
Retro Inject	2800	3 760	9	4 682
Coast	0	0	14	4 668
Trans Inject	22	21	9	4 638
Coast	20	22	23	4 593
Aeromaneuver	0	0	61	4 533
Coast	0	0	136	4 397
Phase Inject	67	62	9	4 327
Coast	0	0	2	4 325
LEO Circ	122	109	9	4 207
Trim	3	6	0	4 201
Dock	0	0	3	4 198
Reserves	76	67	10	4 121

3.3.5.3 Capabilities and Sensitivities

Maximum payload capabilities for the ground- and space-based OTV's are shown in table 3.3.5-6. Offloaded payload capability for the ground-based OTV is shown in table 3.3.5-7 and figure 3.3.5-6. Figure 3.3.5-7 presents the offloaded payload capability of the space-based OTV. Sensitivity to changes in Isp and burnout mass for the space-based OTV are shown in figure 3.3.5-8. These sensitivities represent the change in propellant capacity required to maintain payload due to a change in Isp or burnout mass.

Table 3.3.5-6 FOTV Maximum-Payload Capabilities

Mission	SB OTV	GB OTV
Manned GEO resupply (single-stage)	7 690 (up), 5 090 (down)	7 600 (up), 5 000 (down)
GEO delivery (single-stage)	13 530	13 340
0.2g GEO delivery (single-stage)	12 980	12 780
Two-stage GEO delivery	31 760	31 850

Table 3.3.5-7 GB OTV Offloaded Performance

GB OTV two-stage–GEO delivery (0.2g max)

W_p = 17 183 + 1.556 * P/L kg (up to P/L = 31 920 kg)

Small GB OTV (max W_p = 22 730 kg)

GEO delivery (3g limit)

W_p = 10 566 + 1.461 * P/L kg (up to P/L = 8290 kg)

GEO delivery (0.2g limit)

W_p = 10 832 + 1.507 * P/L kg (up to P/L = 7860 kg)

GEO delivery (0.1g limit)

W_p = 11 333 + 1.592 * P/L kg (up to P/L = 7130 kg)

GEO manned round trip (3g limit)

W_p = 10 766 + 3.086 * P/L kg (up to P/L = 3860 kg)

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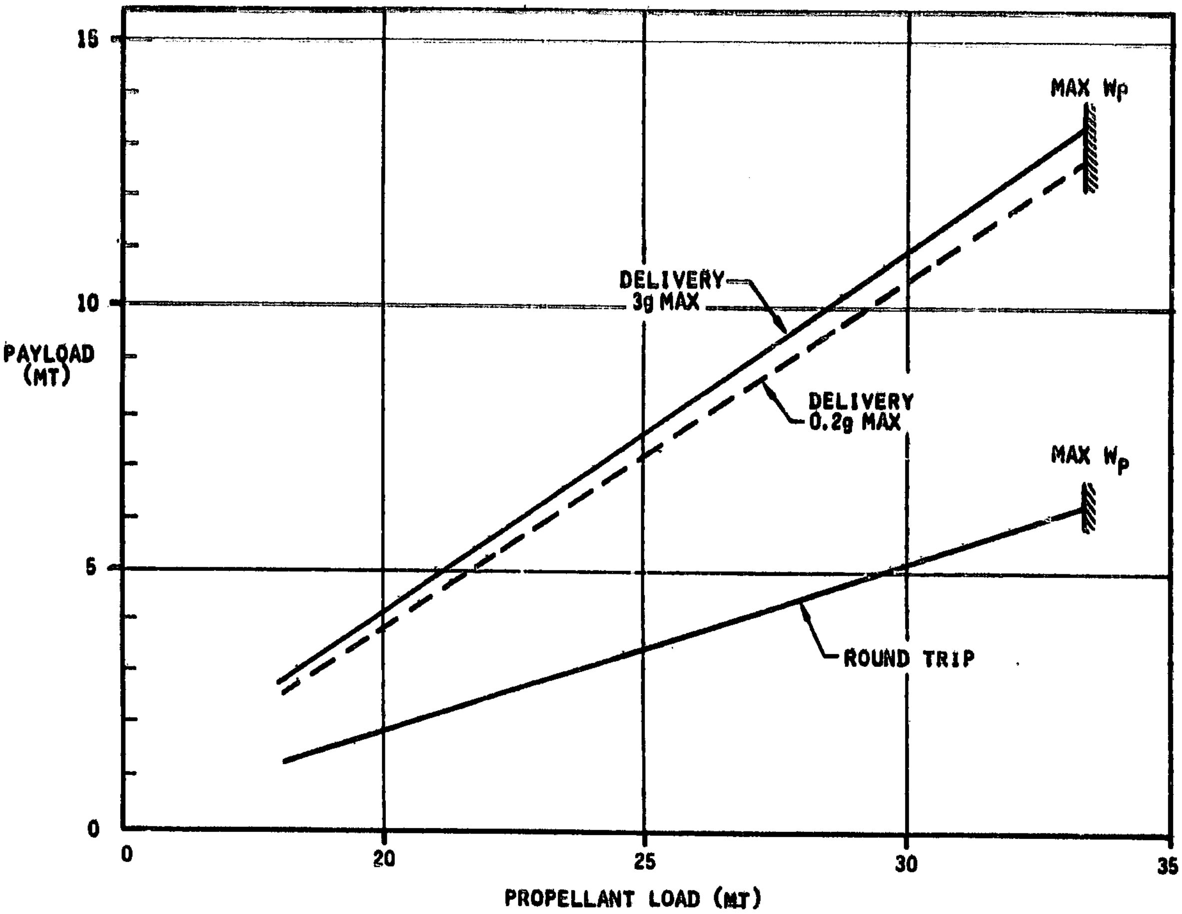

Figure 3.3.5-6 Offloaded Performance—Selected Single-Stage GB LO_2/LH_2 OTV

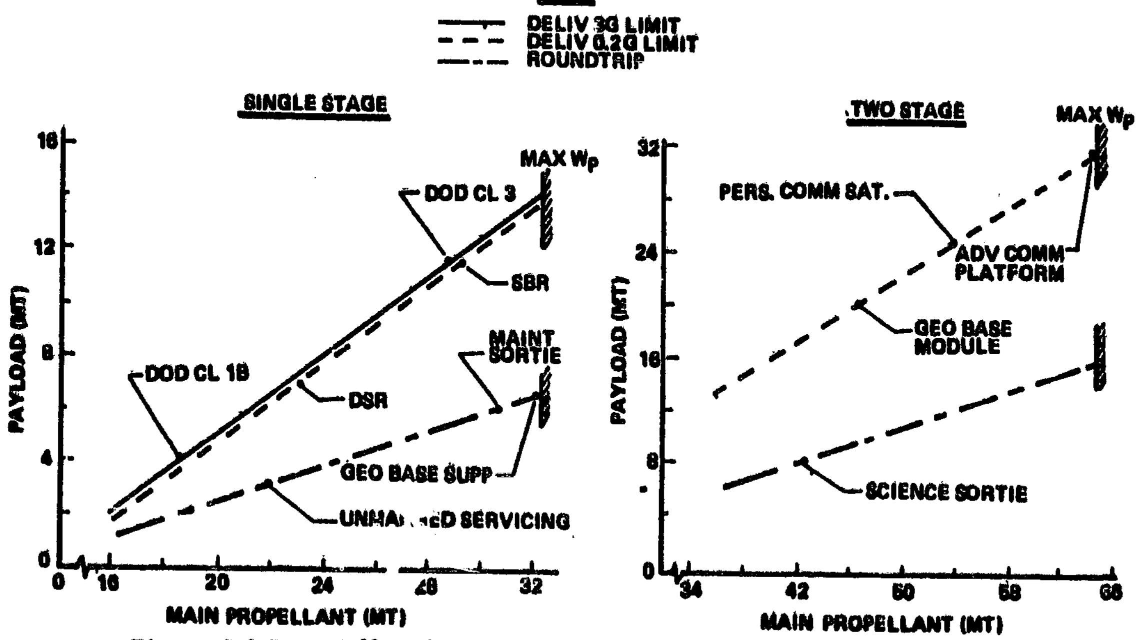

Figure 3.3.5-7 Offloaded Space-Based LO_2/LH_2 OTV Performance

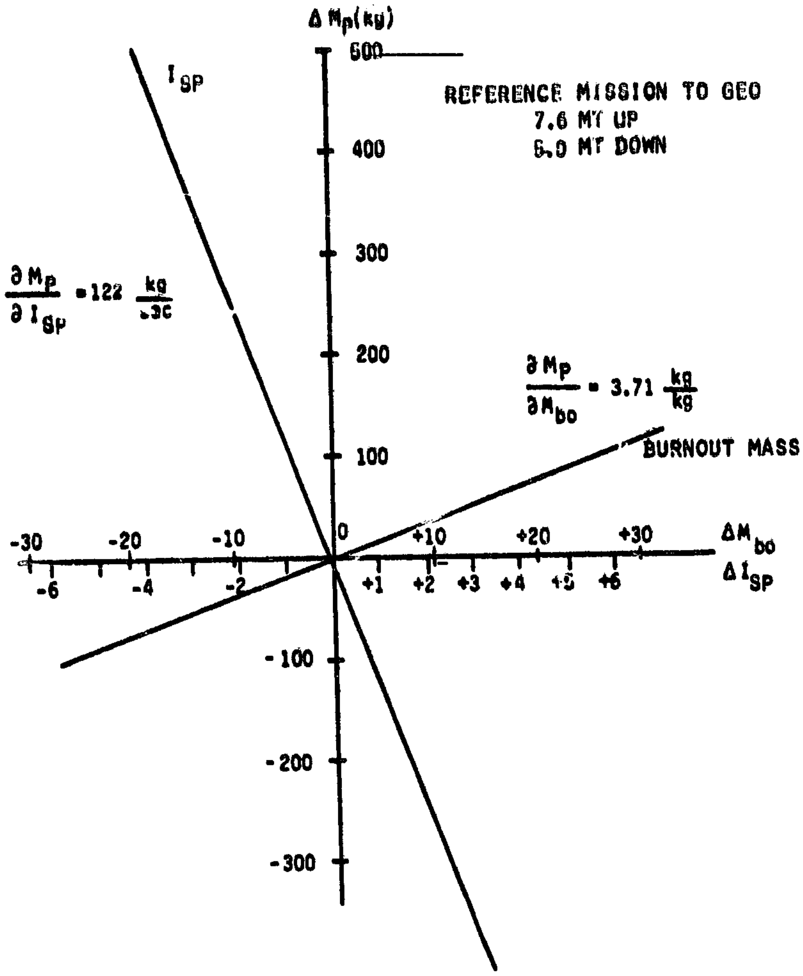

Figure 3.3.5-8 Space-Based OTV Sensitivities to Burnout Mass and Isp

3.3.6 Space Debris Protection

A major consideration in the development of a reusable system is to ensure its structural integrity including protection against space debris in the form of meteoroids and manmade objects. This section presents pertinent background information, the analysis leading to the required shield thickness, the design concept to be used, and the unresolved issues.

3.3.6.1 Background

Considerable effort was expended in this area in the 1960's and early 1970's. The major focus was on manned habitats and noncyclic pressure tanks. A review of these data for applicability to the operation of a reusable OTV for the post-1995 time frame indicated the need for further investigation. This conclusion was the result of a combination of the following factors relative to the 1973 MSFC tug studies.

1. Larger vehicle area
2. Longer exposure times
3. Permanent space basing not considered
4 Manmade debris not considered
5. Different viewpoint regarding sensitivity of propellant tanks to space debris damage

The larger vehicle area is the result of the FOTV systems containing approximately 32t of propellant as compared with 23t for the tug. The longest mission time was greater (8 versus 5 days), as was the average duration (5 versus 3 days). An additional factor for the SB OTV was that it was to remain on orbit (permanent space basing) and thus considerably increase its total exposure time. Prior studies concerning space debris protection only considered meteoroids. NORAD data now indicate a considerable number of manmade objects also exist in orbits that may impact an OTV. A different viewpoint is also suggested regarding the sensitivity of propellant tanks to meteoroid/debris damage. This viewpoint is summarized in table 3.3.6-1. In summary, the tank wall thickness should

Table 3.3.6-1 Propellant Tank Debris Protection Philosophy

- NASA CRITERIA FOR TANKS (NASA SP-8042, MAY1970):
  - ALLOWS PENETRATION UP TO 25% OF WALL THICKNESS (INTENDED FOR TANKS HAVING A NON-CYCLE SERVICE LIFE REQUIREMENT)
- CURRENT BOEING POSITION ON DAMAGE TO TANKS HAVING A CYCLIC SERVICE LIFE REQUIREMENT
  - CONSERVATIVE APPROACH OF ALLOWING NO DAMAGE (NO FLAWS OTHER THAN THOSE KNOWN AT TIME OF ACCEPTANCE TESTING (I.E. PROOF TEST AND LEAK TEST))
    - INSUFFICIENT DATA BASE FOR CORRELATING NON-PENETRATING DEBRIS DAMAGE TO REMAINING SERVICE LIFE VIA FRACTURED MECHANICS APPROACH
  - IF A TANK DESIGNED FOR NO DAMAGE IS DAMAGED, IT MUST BE SUBJECTED TO NEW ACCEPTANCE TESTING DESIGNED TO GUARANTEE (AS A MINIMUM) ITS REQUIRED REMAINING SERVICE LIFE.

not contribute to the required shield thickness. Furthermore, if a tank is damaged, it is strongly suggested that new acceptance testing be conducted to guarantee its required remaining service life.

3.3.6.2 Shielding Analysis

Guidelines and Assumptions—The guidelines and assumptions used to conduct the space debris analyses are presented in table 3.3.6-2. The tank area used was that established by

Table 3.3.6-2 Space Debris Analysis Guidelines and Assumptions

FOTVT8-178

- EXPOSURE AREA: SIDE PROJECTION OF TANKS
  GB AND SB OTV = 35 SQ M [1]
- EXPOSURE TIME:
  - MISSION ONLY (BOTH GB AND SB OTV)
    - TOTAL FLIGHT AVG = 5.3 DAYS (DESIGN POINT)
    - MANNED FLIGHT AND UNMANNED SERVICING AVG = 7.2 DAYS
  - BASING AT LEO BETWEEN MISSIONS
    - GB OTV --- 1 DAY
    - SB OTV --- 21 DAYS
- ALLOWABLE TANK PENETRATION: ZERO
- PROBABILITY OF NO TANK IMPACT $P_{(O)}$: 0.995
  (REQUIRED TO SATISFY VEHICLE MISSION SUCCESS CRITERIA)
- KEY EQUATIONS:
  - METEOROIDS: $\bar{t} = 0.162 \left(\frac{(AT)}{1-P_{(O)}}\right)^{.271}$ IN MILLIMETERS ALUMINUM
  - MAN MADE: $\bar{t} = 0.089 \left(\frac{(AT)}{1-P_{(O)}}\right)^{.431}$ IN MILLIMETERS ALUMINUM
- DEBRIS MODEL: KESSLER AND COUR-PALAIS, JGR VOL 83 METEOROIDS DOMINATE TRANSFER TRACTORY; MAN MADE DEBRIS DOMINATE LEO

[1] DESIGN PT. BUT FINAL DESIGN INDICATED SB = 38 SQ M
GB = 38.9 SQ M

side projection rather than wetted. As indicated earlier, the analysis was conducted without the tank wall contributing to the required shield thickness. The criteria are expressed as probability of no tank impact rather than penetration.

The indicated value of $P_{(o)} = 0.995$ is that which, when considered with the predicted subsystem reliability, gives the required mission success goal of 0.97. The shield thickness ($\bar{t}$) equations have different constants and exponents because meteoroids and manmade debris have different velocities, density, size, and flux. Both equations reflect use of a double-wall design with its general characteristics being one-third of mass in the bumper and the remainder in the back wall. Spacing between walls would be approximately 30 particle diameters. It should be noted that a single-wall shield would have a mass approximately four times greater than that of a double wall.

The debris model used in the analysis is shown in figure 3.3.6-1. The indicated meteoroid flux is essentially the same as that used in the Apollo, Skylab, and Space Shuttle programs. The manmade debris flux is established by NORAD. As of 1980 there were approximately 5000 objects of 10-cm diameter or greater. Approximately 60% of the objects are fragments resulting from explosions, 20% are mission related such as shrouds, rocket stages, etc., and 20% are payloads (either operating or nonoperating). The value used in the analysis was the predicted flux for 1990 which is approximately 8000 objects.

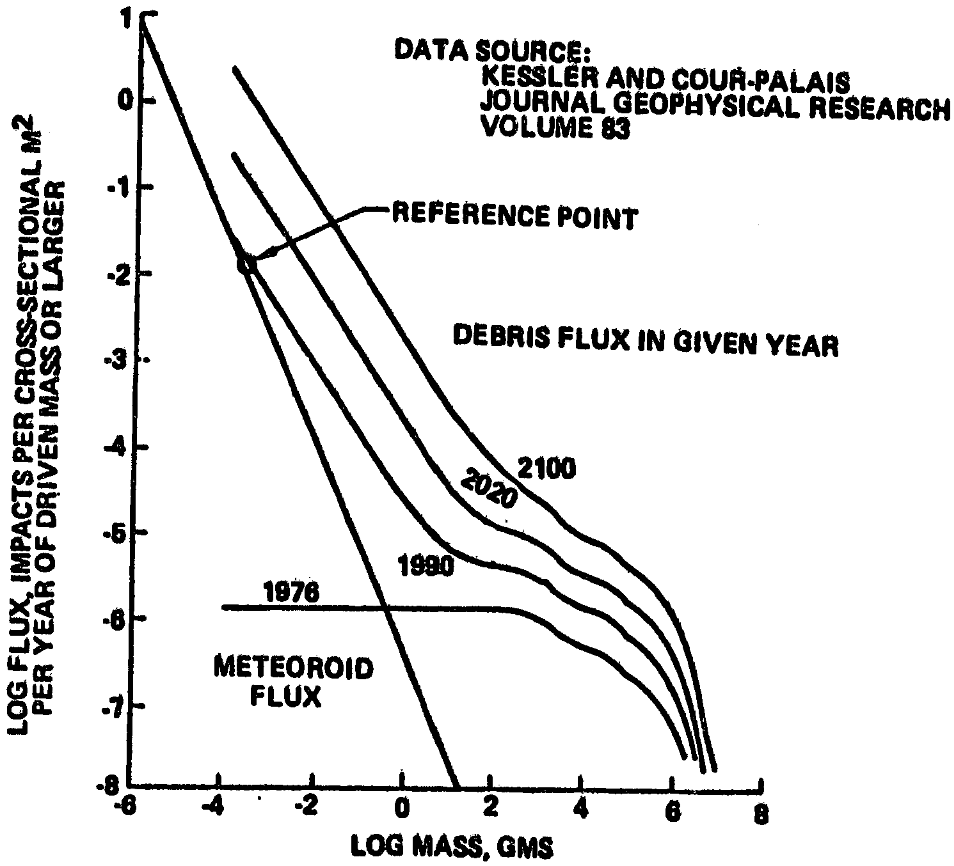

Figure 3.3.6-1 Debris Model

The number of impacts per year expected on an OTV is 10^{-2} with the indicated area and $P_{(o)}$. This also corresponds to protection against objects with a mass of 10^{-3}g.

Shield Thickness Requirement—The amount of protection or shielding required for various combinations of $P_{(o)}$ and vehicle AT (m^2-years) is shown in figure 3.3.6-2. The shield thickness is defined as $\bar{t}$ expressed in millimeters equivalent of aluminum. For the case of

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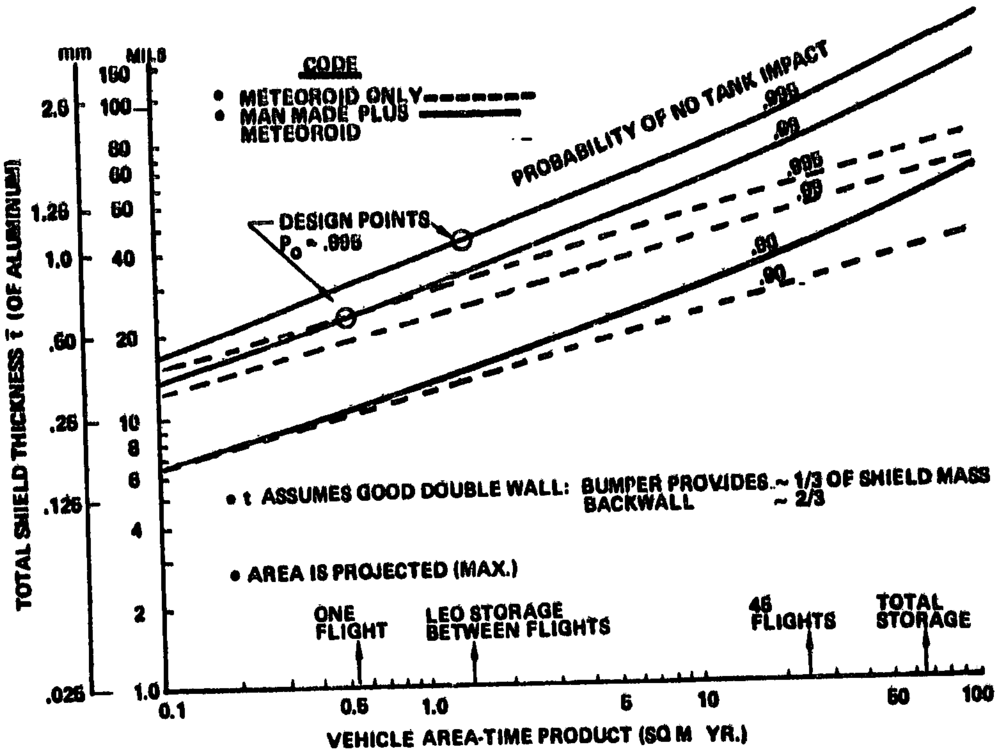


Figure 3.3.6-2 Debris Shield Requirement

a flight between LEO and GEO, meteoroids are the dominating environment. For a single flight and a $P_{(o)} = 0.995$, the resulting $\bar{t}$ is 0.62 mm. For the on-orbit storage period, the dominating environment becomes a combination of manmade and meteoroid debris. Assuming an average of 3 weeks between flights, the required $\bar{t}$ is 1.05 mm.

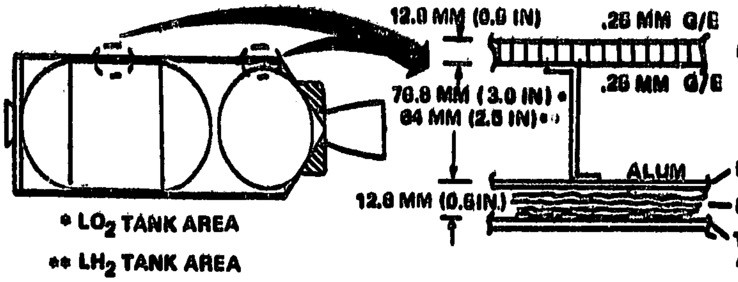
3.3.6.3 Design Approach

The selected design approach for space debris protection is shown in figure 3.3.6-3. The protection incorporated into the vehicle is only that necessary for a single flight since each flight is considered as an independent event. To provide the required protection during on-orbit storage (time between flights), the SB OTV will be placed in a hangar. The $\bar{t}$ provided by the hangar is the difference between the total requirement of 1.05 mm and that provided by the OTV (0.62 mm), or 0.45 mm. Using this approach, the OTV does not have to incur the structural penalty associated with on-orbit storage.

The $\bar{t}$ for the vehicle is provided through a combination of the shell, backwall, and MLI. The backwall was placed between the shell and tank, rather than outside the shell, because of vehicle-diameter constraints. The aluminum equivalent of the shell and MLI

FOTV78-179

1. MISSION REQUIREMENT
   - $\bar{t}$ = (.62 MM) 24 MILS FOR $P_{(O)}$ = .995 AND ONE LEO-GEO
   - APPLICATION: GROUND AND SPACE BASED OTV
   - DESIGN APPROACH



ALUM EQUIV. (MM)

	LO_2 TK	LH_2 TK
SHELL - LOADS & DEBRIS BUMPER	0.3	0.3
BACKWALL - DEBRIS PROTECT.	0.2	0.25
MLI - THERMAL & DEBRIS PROTECT.	0.1	0.1
TANK WALL ALUM	0	0
	0.6	0.65
	(MILS = 24	26)

2. MISSION TO MISSION (LEO STORAGE) REQUIREMENT
   - $\bar{t}$ = (1.06 MM) 42 MILS FOR $P_{(O)}$ = .995 AND 15 DAYS AT LEO
   - APPLICATION: SB OTV ONLY UNLESS GB OTV HAS WAIT TIME GREATER THAN 3 DAYS

3. DESIGN APPROACH
   - PROVIDE A HANGAR FOR OTV WHEN IT IS AT SOC.
   - HANGAR PROVIDES DIFFERENCE BETWEEN REQUIREMENT AND OTV PROVISION

   ∴ $\bar{t}$ OF HANGAR = 0.45 MM ALUM EQUIV.

Figure 3.3.6-3 Selected Design Approach

relates to their relative densities. A thicker backwall for the LH_2 tank was necessary since the spacing between walls in that area was not as great $\left[\bar{t} \sim (\text{spacing})^{-1/2}\right]$. Although the resulting design does not have the ideal double wall mass split (1/3-2/3), it is judged to be a reasonable compromise when launch or flight loads as well as debris protection must be considered.

In addition to the selected debris protection design, alternative concepts were also considered such as those indicated in figure 3.3.6-4. These designs relate to concepts which make greater use of MLI rather than "hard" structure for the protection system. For the GB OTV using a structural bumper which also carries the launch loads, approximately 140 layers of MLI would be required to provide the $\bar{t}$ of 0.62 mm. In the case of an SB OTV which only has a minimum of flight loads, consideration can be given to providing all of the protection with MLI, such as indicated in reference 3. To satisfy the FOTV $\bar{t}$ requirement, as many as 300 layers would be a "ballpark" estimate. In summary, use of a very large number of layers of MLI does not appear to provide as straightforward a solution as the hard structure approach indicated with the selected design. The reasons

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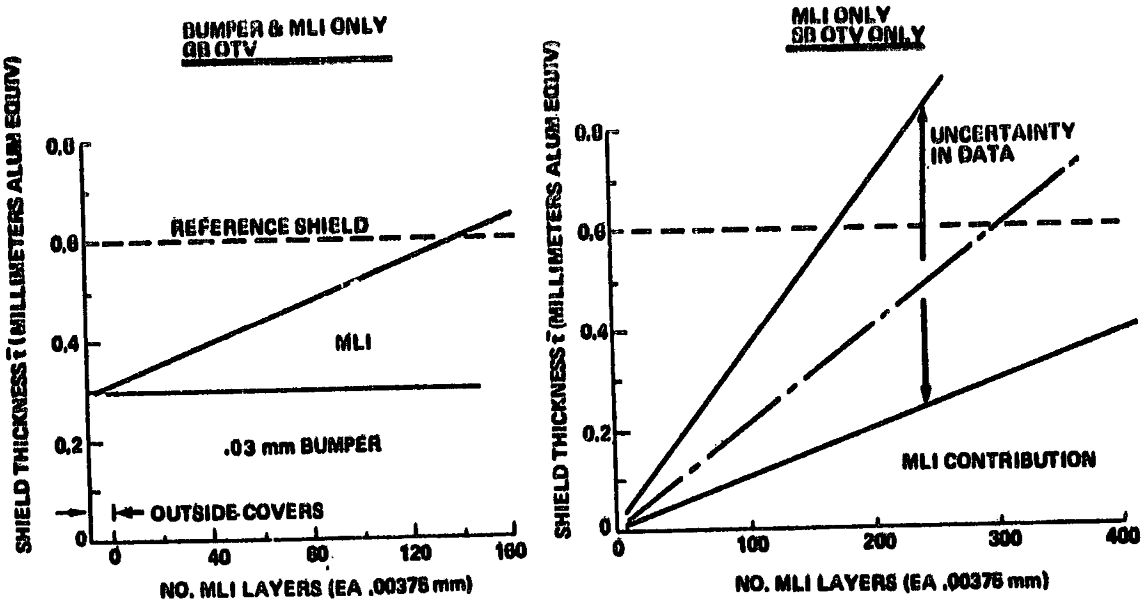

Figure 3.3.6-4 Alternative Debris Shield Designs

are as follows: (1) installation of such a large quantity offers considerable challenges, and (2) in making the literature survey associated with this analysis, very little information was found concerning the real shielding value of a large amount of MLI.

The structural mass impact of providing the indicated debris protection relative to idealistic SB and GB OTV's is shown in figure 3.3.6-5. In the case of the SB OTV, a vehicle that includes provisions for meteoroid protection has nearly a 500 kg penalty over one designed only for flight loads. The majority of the penalty is associated with the provision of the double wall shield. A GB OTV, although having a shell to sustain launch loads and fully loaded tanks, still must be provided with a backwall and results in a 200-kg penalty compared to a concept which does not consider space debris. To put these data into perspective, it must be noted that both of these vehicles are larger (larger propellant loads) and had longer on-orbit times than those which were investigated in the Phase A OTV studies. A ground-based and STS-compatible OTV did not require additional structure beyond that required to carry boost loads.

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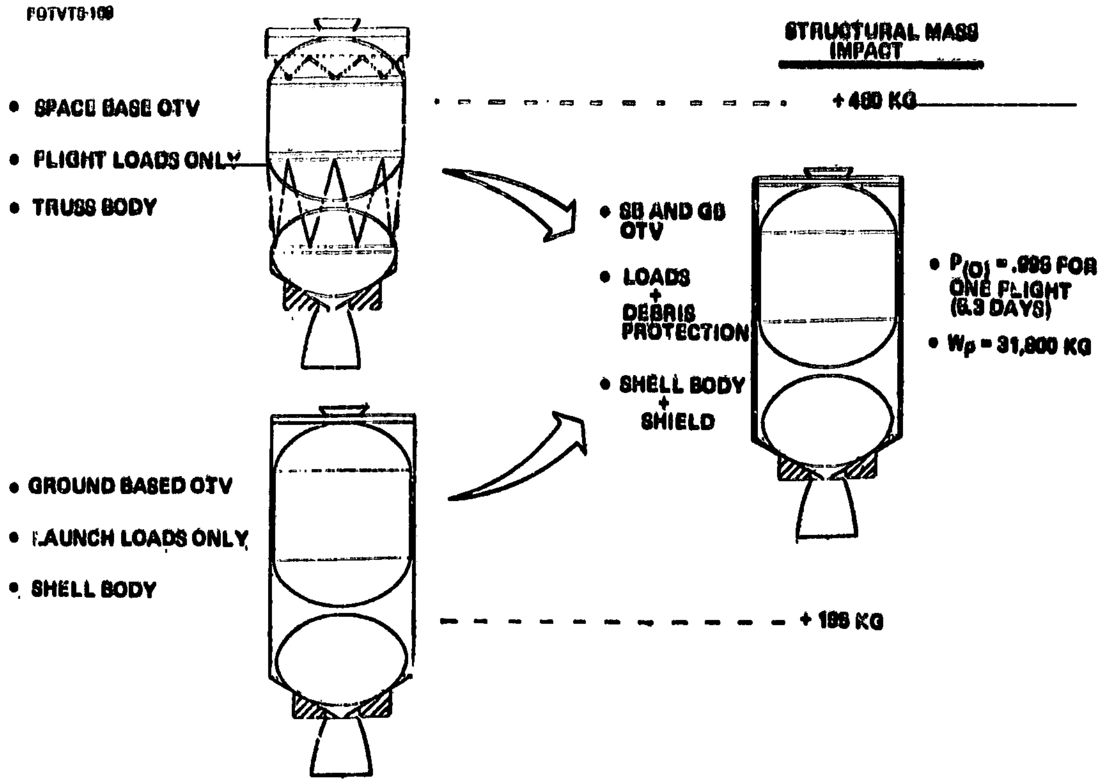


Figure 3.3.6-5 Debris Protection (Meteoroid) Mass Impact

3.3.6.4 Unresolved Issues

Although a workable design approach has been defined for space debris protection of advanced OTV's, it is recognized that there is considerable work remaining before an optimum design is achieved. Unresolved issues identified at this time are listed below:

1. Value of sandwich or honeycomb shell as bumper
2. Protection characteristics of graphite-epoxy structure
3. Value of three-layer shield
4. True benefit of MLI only for SB OTV
5. Proper viewpoint regarding $P_{(o)}$ and exposure time for propellant storage tanks at SOC

Of foremost importance is the need to establish the shielding characteristics of graphite-epoxy or composite structure as single sheets or in honeycomb or sandwich designs. The importance is indicated by the fact that this material is used extensively throughout the vehicle because it reduces weight; but at this time, there is very little, if any, test data regarding its debris protection qualities. In the case of MLI only for protection, there is the uncertainty as to whether a multilayer design would actually provide more protection than its mass equivalence. Three-layer (walls) shields have been indicated to improve the protection. Uncertainty also exists in the design criteria for the propellant storage tanks at SOC in terms of debris protection. These tanks have fewer pressure cycles than an OTV; however, their exposure time is longer. Use of a $P_{(o)}$ such as 0.995 would result in a $\bar{t}$ as large as 2.5 mm; however, since each tank is only launched once, the long-term impact may not be too significant.

3.3.7 Reliability Assessment

This section presents the analyses related to the mission success prediction of both the space- and ground-based OTV's.

3.3.7.1 Overview

Mission success is defined as the probability that the OTV will perform its intended function for the duration of its mission. The mission, in turn, is defined as the time period from the initiation of one mission to the initiation of another. Subsystem reliability, as well as the reliability of the space debris protection system, must be considered. This section, however, only discusses the subsystem reliablity prediction.

The purpose of the reliability analysis was threefold: first, to determine if there is a difference between SB and GB OTV's (a potential difference was how the launch portion of the GB OTV compared with the space storage portion of the SB OTV); second, to determine if the mission success goal could be met with the assumed subsystem redundancy; and, third, to establish the framework and data base to conduct the maintenance analysis discussed in section 3.3.8.

3.3.7.2 Analysis

The assumptions and guidelines used to perform the reliability assessment are shown in table 3.3.7-1. First and foremost in the reliabilty prediction is the assumption that all subsystems have been restored to their original status prior to each flight. The subsystem definitions (schematics) were essentially the same as defined for the OTV in Phase A (ref. 1) with the indicated exceptions being unique to the second-generation reusable OTV. Launch was not included for the SB OTV since this event theoretically occurs only once and at that time all subsystems are dormant. The slightly longer LEO-GEO time for the GB OTV was the result of the vehicle having to separate from its launch vehicle and wait

Table 3.3.7-1 Reliability Analysis Assumptions and Guidelines

FOTVTS-203

- 160 OTV MISSIONS
- PHASE A OTV SUBSYSTEMS WITH REVISIONS:
  - AVIONICS - ADD GPS AND RENDEZVOUS/DOCKING EQUIPMENT
  - ACTIVE AVIONICS COOLING
  - ADVANCED MAIN ENGINE (2)
  - MORE ACS THRUSTERS
- SOC AVAILABLE FOR SPACE BASING
- MISSION PROFILE

	SHUTTLE LAUNCH	LEO - GEO	STORAGE
• GROUND BASED OTV	0.2 HR	160 HR * (1)	—
• SPACE BASED OTV		140 HR (1)	480 HR

(1) INCLUDES 0.42 HR OTV ENGINE BURN (8 BURNS)

- EXPOENTIAL RELIABILITY MODEL AT COMPONENT LEVEL
- MISSION SUCCESS GOAL = 0.97
- SAFETY GOAL = 0.9996

on orbit (in an active state) until the proper position is reached to initiate the first burn. (Note: This operating mode was used prior to the selection of two sizes of GB OTV's.) The SB OTV can remain at SOC until a short time period before the initial burn. The storage time reflects an average of 20 days between OTV flights. The SB OTV will be in a dormant/semiactive condition, while the GB OTV is completely deactivated when it is on the ground. The mission success and safety goals were the same as used in Phase A. An assessment in terms of meeting the safety goal was not performed.

The subsystem mission profiles assumed are presented in table 3.3.7-2. Again, almost all of the GB OTV subsystems are active during launch and while waiting in orbit. The SB OTV has some subsystems dormant (not active) and others semiactive during its storage time. A semiactive condition means valves or tanks are under pressure. Failure rates for dormancy are assumed to be 10% of the value when active.

The data base employed in the analysis is as follows:

1. IUS subsystem and component failure rates
   a. MIL HDBK 217C, Failure Rates for Electronic Parts
   b. SAMSO-LV6S-77-005, Space Effectiveness Requirements for STS/IUS
   c. Boeing document 180-15480, Program Failure Rates Standards Manual
2. Engine and inlet valve data from Pratt and Whitney

Table 3.3.7-2 Subsystem Mission Profiles

	GROUND BASED OTV		SPACE BASED	
	LAUNCH (.2HR)	MISSION (100HR)	MISSION (140HR)	STORAGE (400HR)
AVIONICS	ACTIVE	ACTIVE	ACTIVE	DORMANT
ACS	SEMI-ACTIVE	ACTIVE	ACTIVE	SEMI-ACTIVE
AVIONICS THERMAL CONTROL	ACTIVE	ACTIVE	ACTIVE	DORMANT
MLI PURGE	ACTIVE	DORMANT	NA	NA
BASIC STRUCTURE	ACTIVE	ACTIVE	ACTIVE	DORMANT
ELECTRIC POWER				
GENERATION	ACTIVE	ACTIVE	ACTIVE	SEMI-ACTIVE
COOLING	ACTIVE	ACTIVE	ACTIVE	DORMANT
PROPULSION	SEMI-ACTIVE	ACTIVE	ACTIVE	SEMI-ACTIVE

3. Boeing experience analysis center
4. PRC Doc. R-1863, On-Orbit Spacecraft Reliability

Component failure rates (and failure modes) used in the IUS were applicable to both avionics and RCS systems of the OTV. Data for the main propulsion (particularly the advanced engine) were provided by Pratt and Whitney.

Subsystem reliability block diagrams for the main propulsion system are presented in figures 3.3.7-1, -2, -3; for attitude control in figures 3.3.7-4, -5, -6; for avionics in figures 3.3.7-7, -8, -9, -10; and for electrical power in figures 3.3.7-11, -12, -13. Thermal control and structure subsystems do not contribute significantly to the unreliability and for that reason their diagrams are not shown. The diagrams are based on subsystem schematics developed for the Phase A OTV but modified for FOTV application. Included within each diagram are the failure rates for launch (3g maximum), spaceflight (0g), and cyclic operations, all expressed as events per million hours or cycles. Also included is the extent of the redundancy within each subsystem.

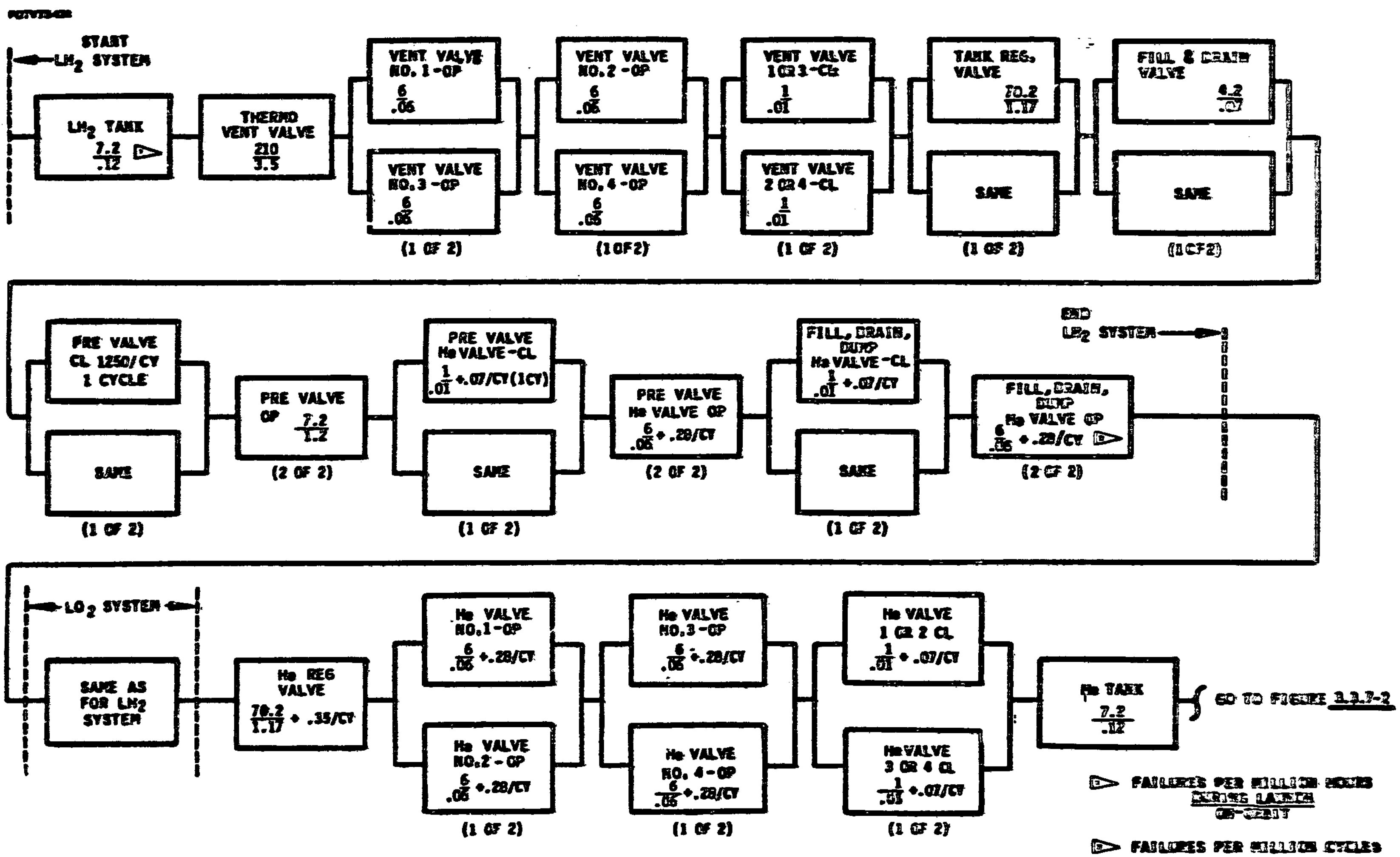


Figure 3.3.7-1 Main Propulsion Reliability Block Diagram

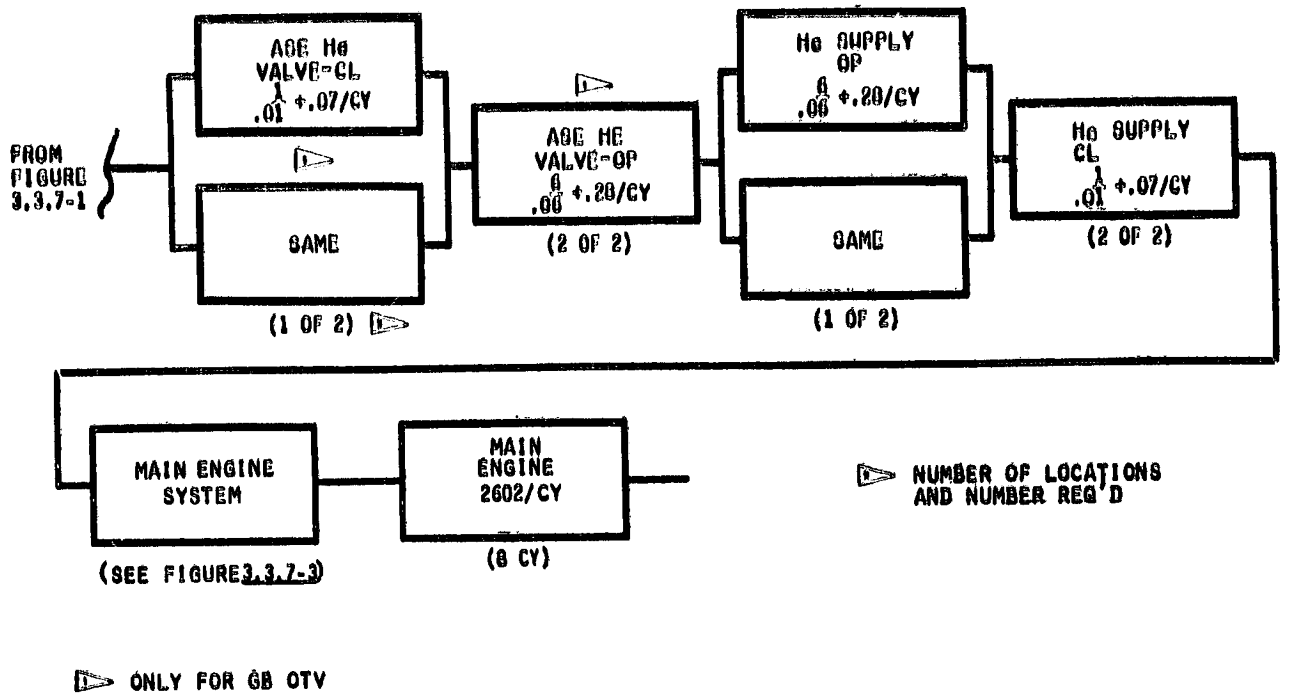


Figure 3.3.7-2 Main Propulsion Reliability Block Diagram

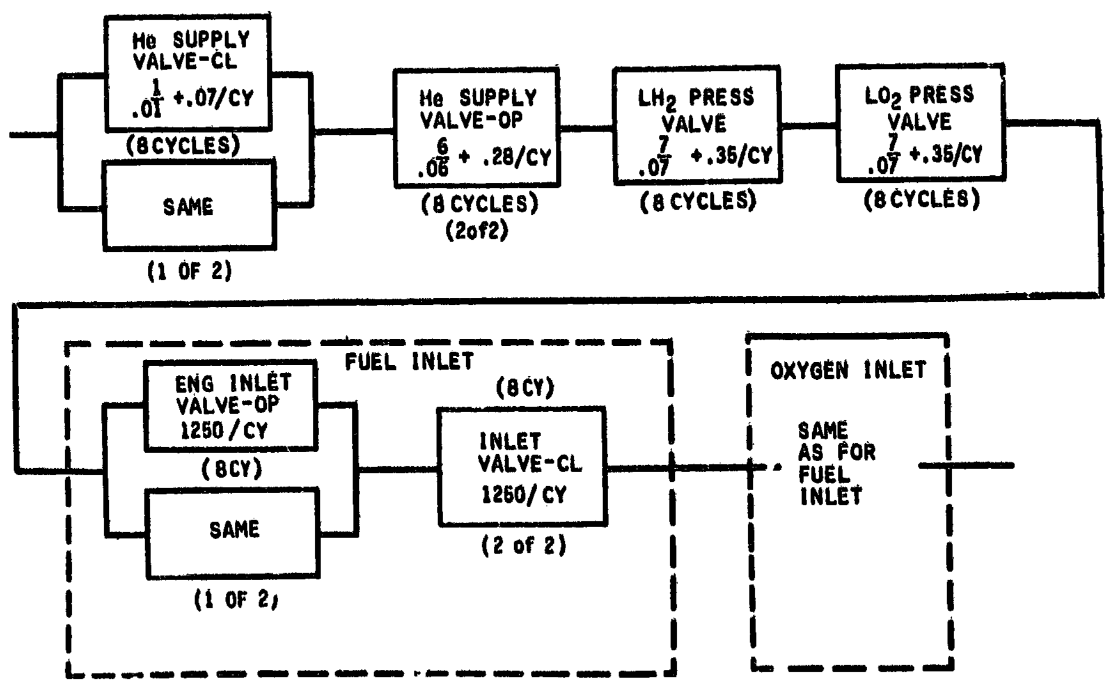


Figure 3.3.7-3 Main Engine System Reliability Block Diagram

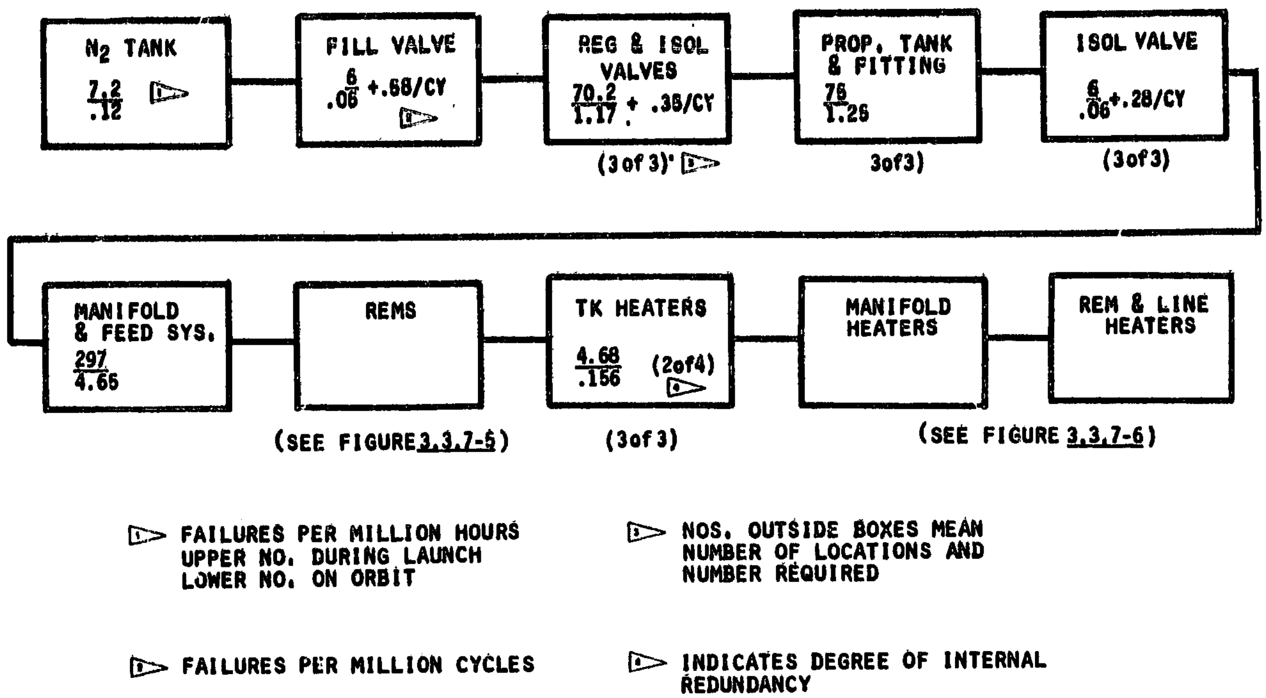


Figure 3.3.7-4 ACS Reliability Block Diagram

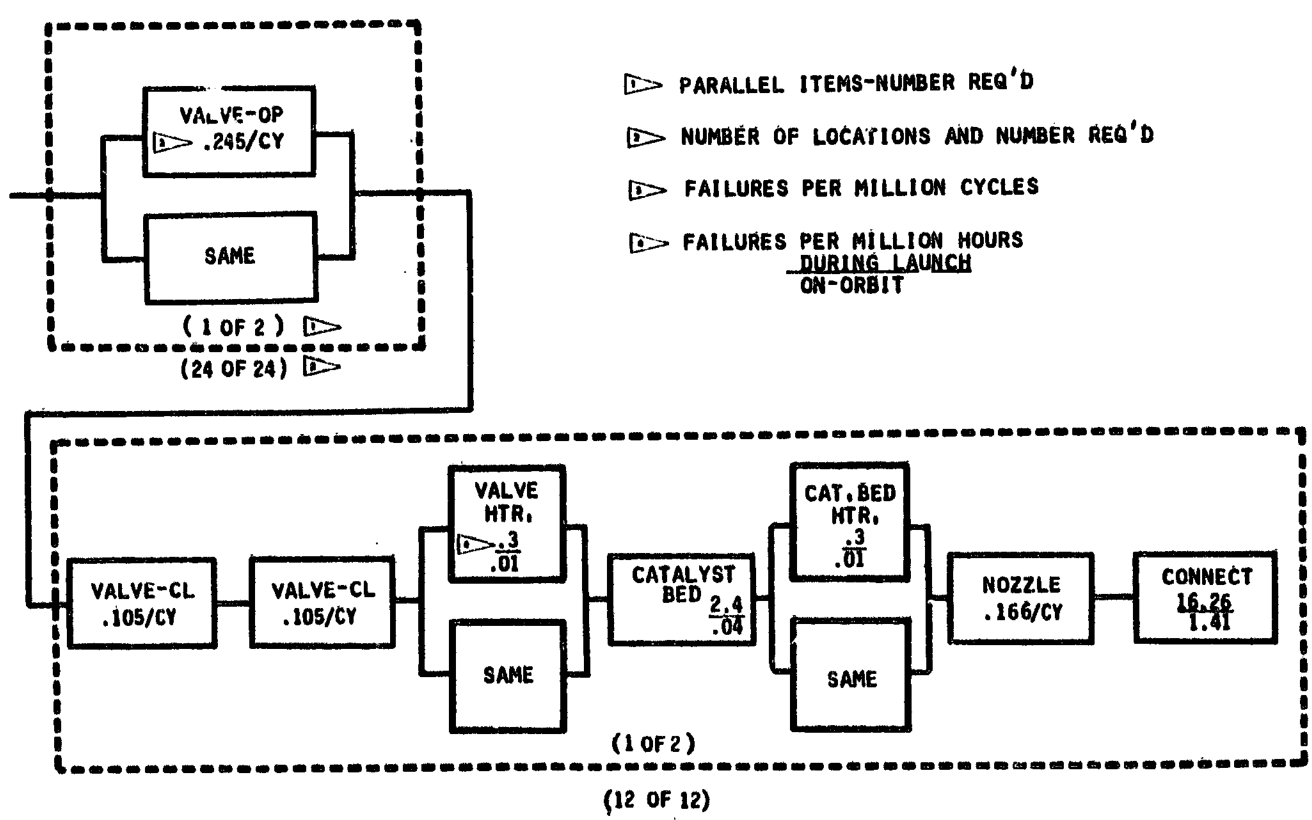


Figure 3.3.7-5 REMS Reliability Block Diagram

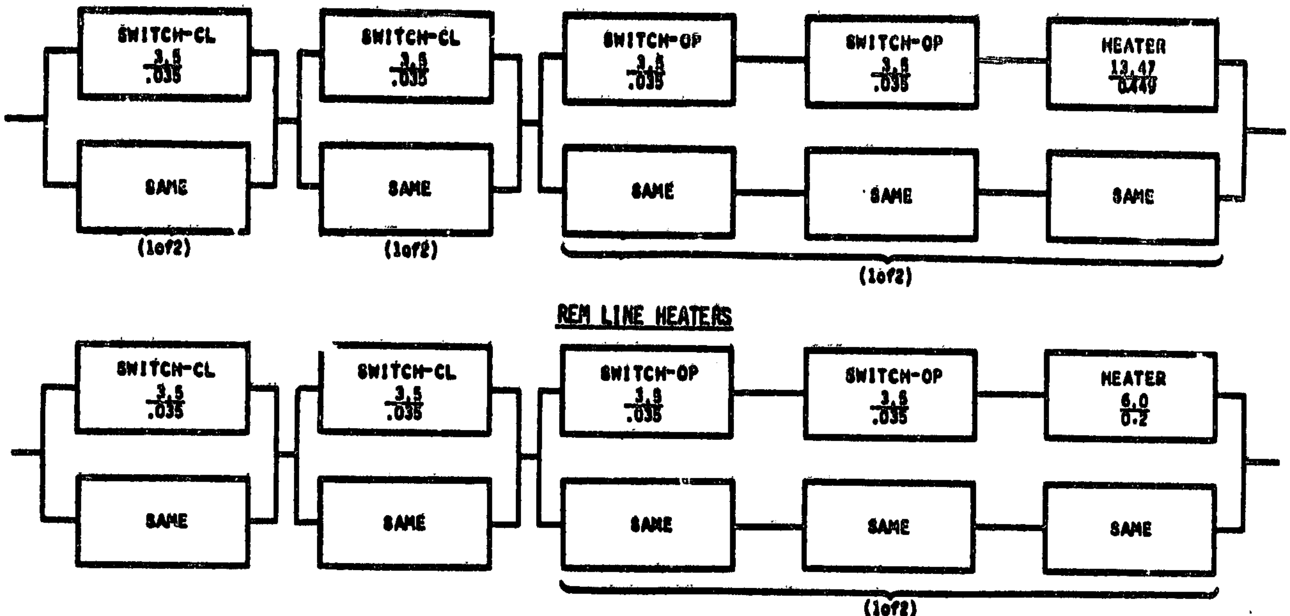
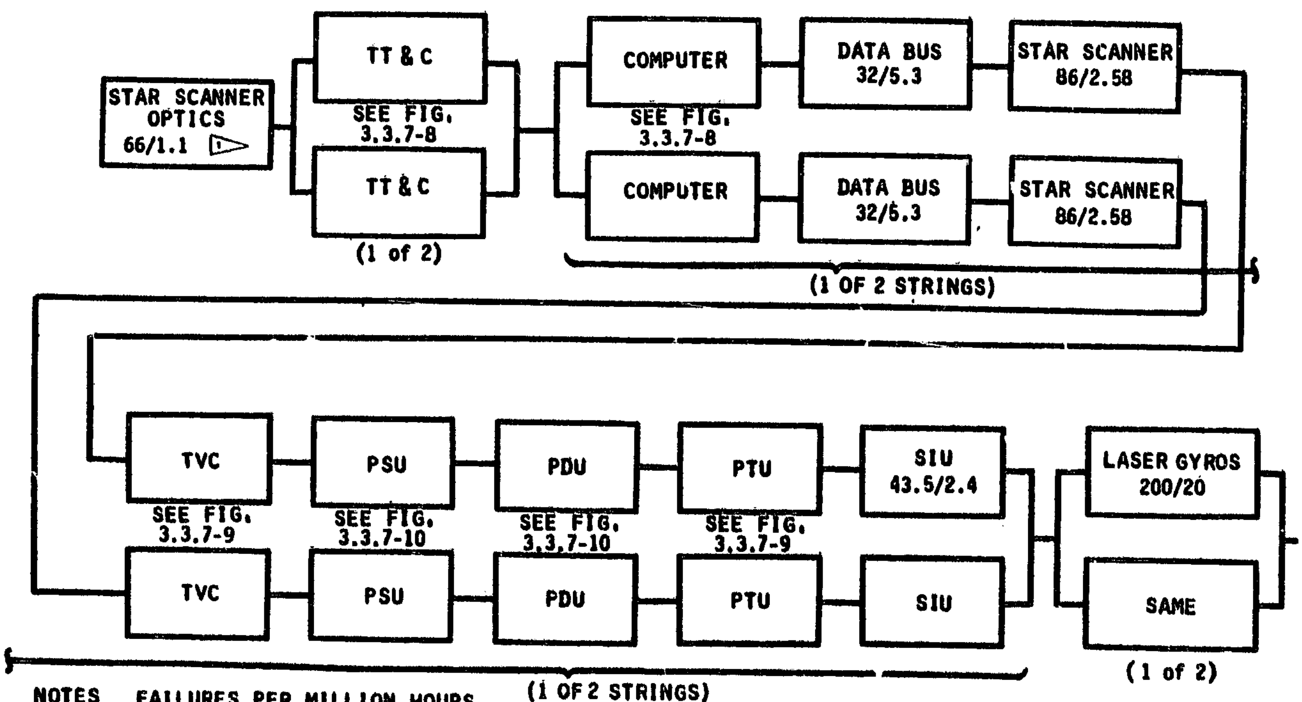


Figure 3.3.7-6 ACS Reliability Block Diagram



NOTES FAILURES PER MILLION HOURS
▷ DURING LAUNCH/ON-ORBIT

TT & C = TELEMETRY, TRACKING & COMMAND
TVC = THRUST VECTOR CONTROL
PSU = POWER SWITCHING UNIT
PDU = POWER DISTRIBUTION UNIT
PTU = POWER TRANSMISSION UNIT
SIU = SIGNAL INTERFACE UNIT

Figure 3.3.7-7 Avionics Reliability Block Diagram

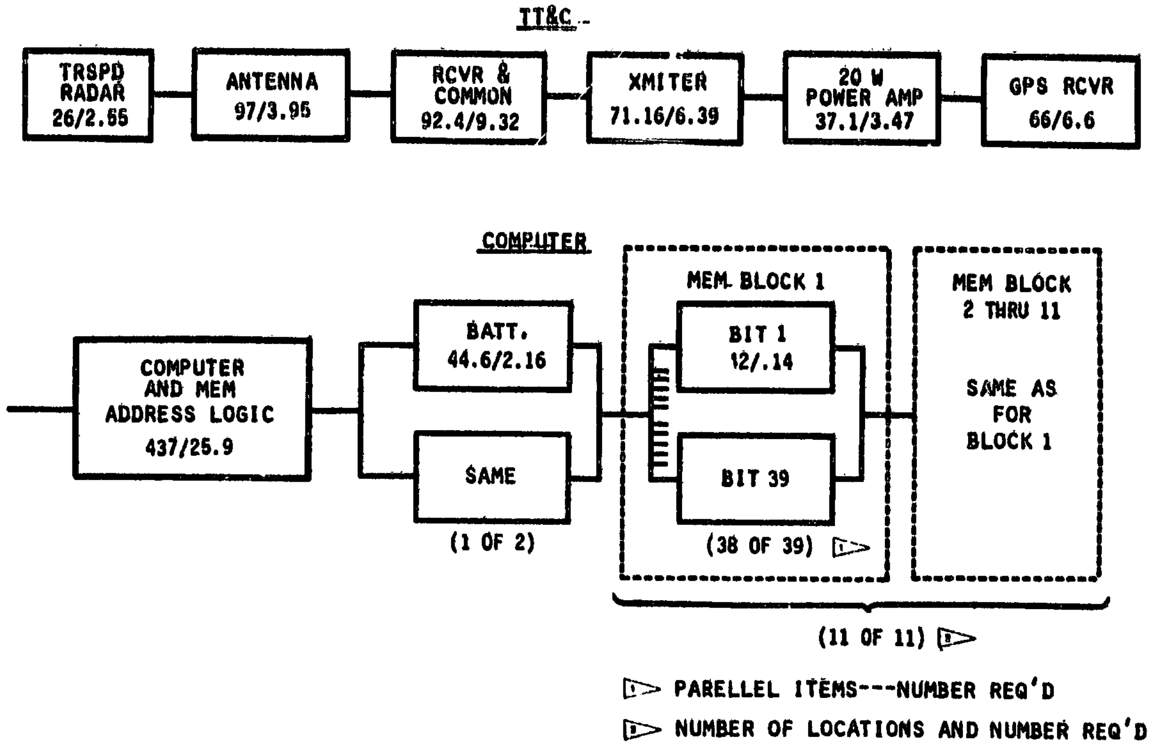


Figure 3.3.7-8 Avionics Reliability Block Diagram

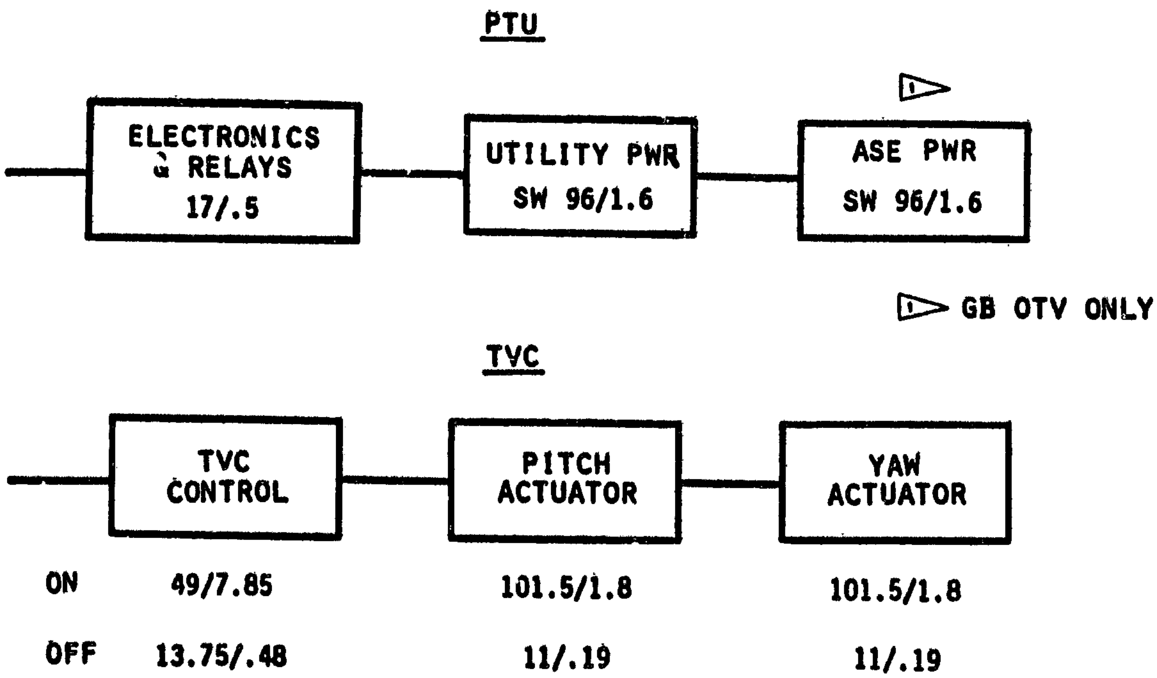


Figure 3.3.7-9 Avionics Reliability Block Diagram

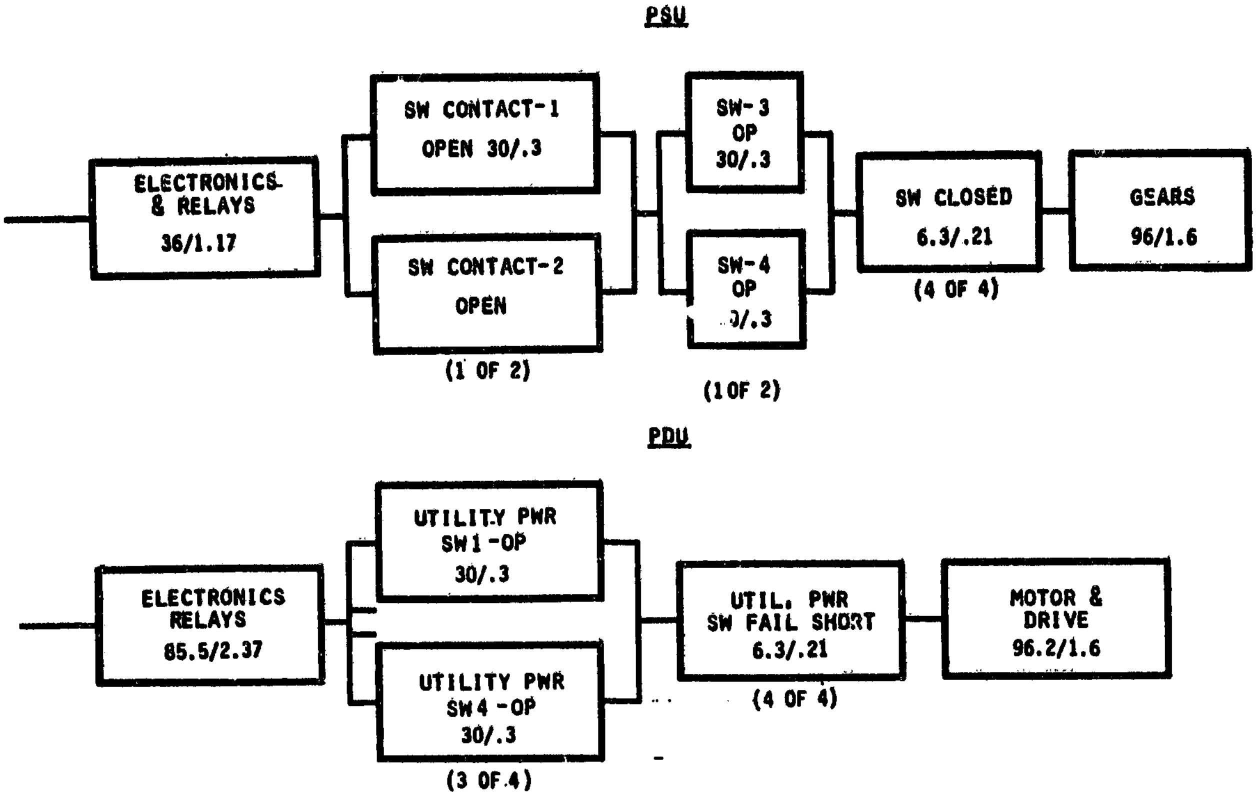


Figure 3.3.7-10 Avionics Reliability Block Diagram

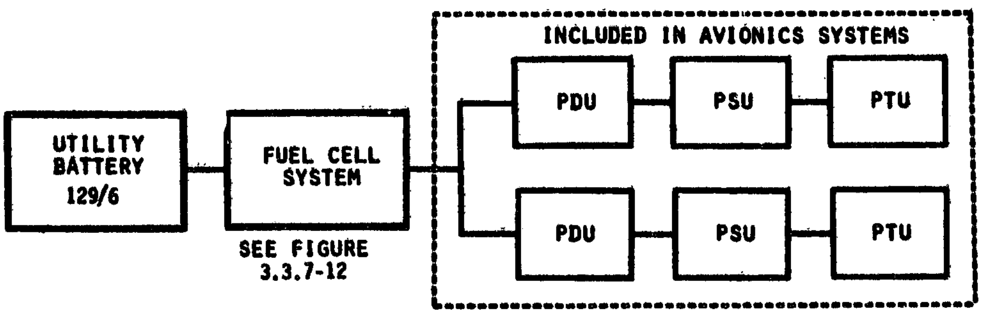


Figure 3.3.7-11 Electrical Power System Block Diagram

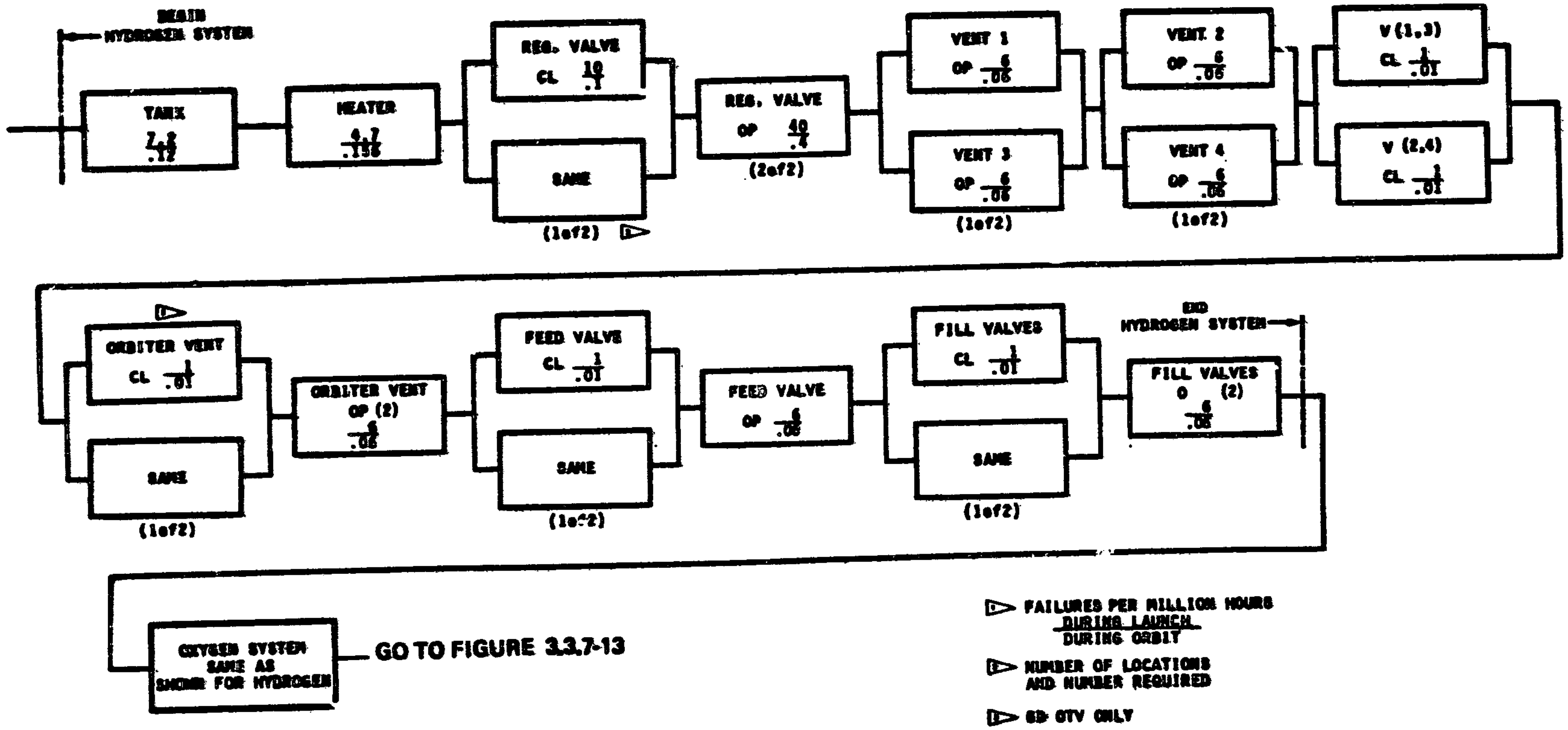


Figure 3.3.7-12 Fuel Cell Reliability Block Diagram

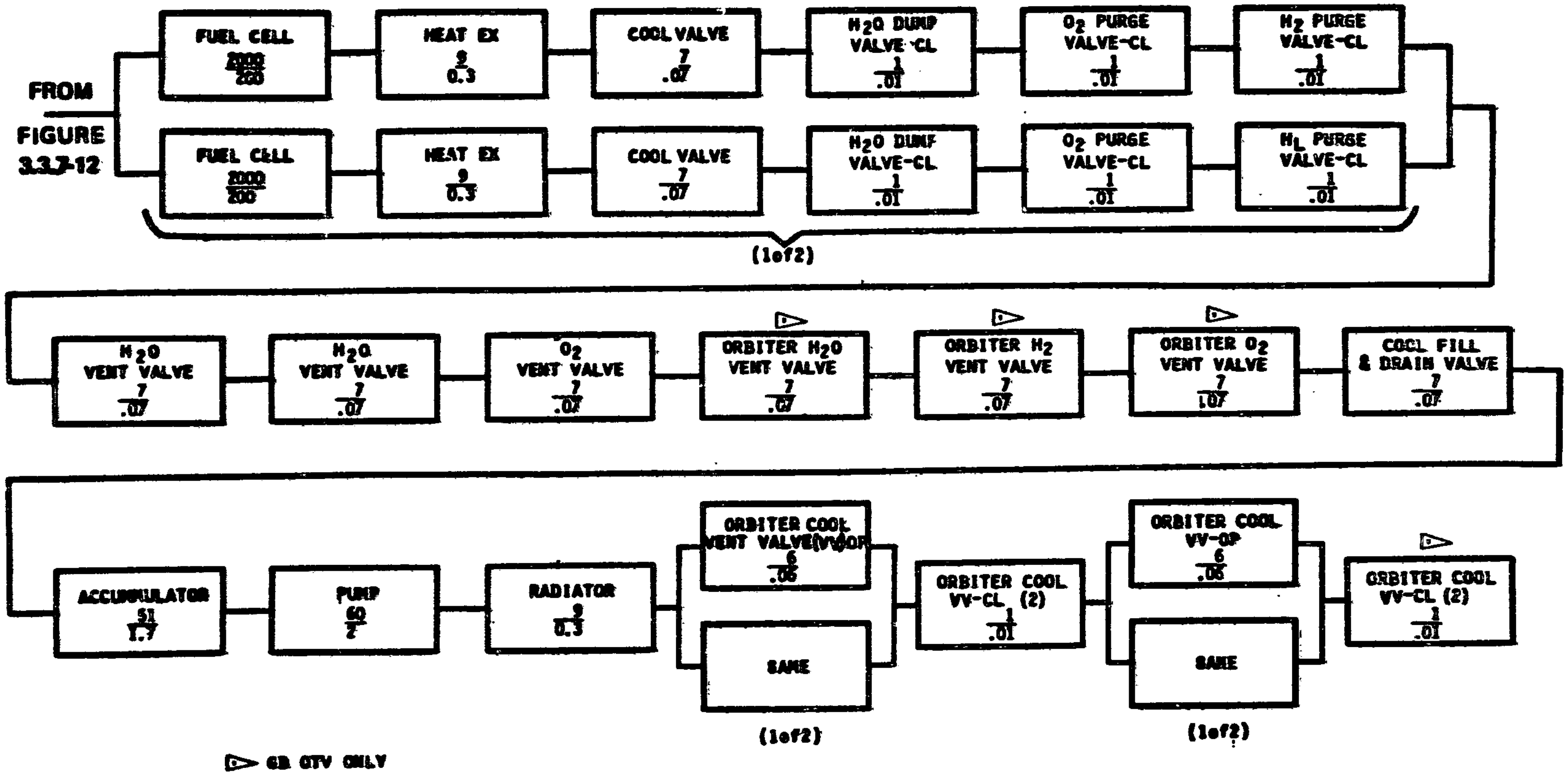


Figure 3.3.7-13 Fuel Cell Reliability Block Diagram

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3.3.7.3 Results

Mission success predictions were determined by applying the subsystem operating profiles to the reliability block diagrams. A summary reliability prediction of the SB and GB OTV's is presented in table 3.3.7-3. From a mission (flight)-to-mission standpoint, the

Table 3.3.7-3 Mission Reliability Prediction Summary

EVENT	GROUND BASED OTV	SPACE BASED OTV
LAUNCH	.998566	N/A
LEO - GEO	.987521	.988660
SINGLE MISSION (CUM)	.986106	.988660
STORAGE	N/A	.989382
TOTAL MISSION TO MISSION	.986106	.978163

MISSION SUCCESS GOAL = 0.97

difference (0.986 versus 0.978) is not significant. It should be noted, however, that this portion of the prediction only includes the contribution of the subsystems. Consequently, in order to satisfy the mission success goal of 0.97, the reliability of the space debris protection system must be 0.995. The impact of satisfying the space debris requirement was presented in section 3.3.6. A final observation concerning the subsystem reliability prediction is that the launch reliability of the GB OTV is greater than the storage period reliability for the SB OTV. This occurs even though the GB OTV is active and operating under much higher stress levels; however, these factors are offset by the very short period of time (0.2 hr).

Subsystem reliabilty predictions are shown in table 3.3.7-4. The subsystems with the lowest reliability are the ACS, MPS, and EPS. Critical components (those with the highest total failure rates) within each subsystem are identified and discussed as part of the maintenance analysis in section 3.3.8.

A reliability comparison was also made for OTV's utilizing either one or two main engines. Utilization of two main engines was the assumed reference case primarily due to the large number of manned flights. The mass impact of the second engine, however, was

Table 3.3.7-4 FOTV Subsystem Reliability Predictions

FOTVT5-207

SUBSYSTEM	GROUND BASED LAUNCH	GROUND BASED LEO - GEO	GROUND BASED TOTAL MISSION	SPACE BASED LEO - GEO (MISSION)	SPACE BASED STORAGE	SPACE BASED TOTAL
ACS	.999849	.993718	.993568	.994434	.996104	.990560
EPS	.999890	.997220	.997110	.997429	.997736	.995170
PROPULSION	.999873	.997915	.997791	.998028	.995932	.993968
AVIONICS	.999987	.999748	.999735	.999773	.999891	.999664
AVIONICS T/C	.999971	.999319	.999290	.999372	.999754	.999126
MLI PURGE	.999992	1.0	.999992	N/A	N/A	N/A
STRUCTURE	.999	.99955	.998550	.99958	.999928	.999508
---------	------	------	-------	-------	-------	-------
SYSTEM	.998566	.987521	.986105	.988660	.989382	.978163

substantial (up to 400 kg considering engine and all scar penalties) and, therefore, it was of interest to determine the reliability impact of using only one engine. Again, the engine involved was an advanced expander type with a reliability prediction of 0.979 for eight burns versus 0.991 for RL-10 IIB. The results of this analysis are shown in table 3.3.7-5.

Table 3.3.7-5 Reliability Sensitivity to Engine Quantity

	LAUNCH	LEO - GEO	SINGLE MISSION	STORAGE	TOTAL MISSION TO MISSION
TWO ENGINE OTV					
GROUND BASED	.998566	.987521	.986105	N/A	.986105
SPACE BASED	N/A	.988660	.988660	.989382	.978163
SINGLE ENGINE OTV					
GROUND BASED	.998559	.963842	.962463	N/A	.962463
SPACE BASED	N/A	.965079	.965079	.989344	.954795

MISSION SUCCESS GOAL = 0.97

In both the GB and SB cases, the mission-to-mission predictions are approximately 0.02 below the two-engine case which can be translated into 2 lost missions per 100 flown. The two-engine prediction assumes both are operating; however, one engine is capable of doing the mission if necessary so, in effect, complete redundancy is provided. Although these data are not conclusive, they do indicate that with an advanced engine, a two-engine system may, in fact, be necessary if the desired mission success goal is to be satisfied.

3.3.8 Maintenance

This section discusses the OTV system requiring unscheduled and scheduled maintenance, the means to perform checkout before and after maintenance, and the impact on the vehicle and space base. This analysis has been confined to the SB OTV for the following reasons:

1. Maintenance needs on a reusable space-based OTV had not been well defined in past studies.
2. The impact of accomplishing the maintenance, particularly in terms of crew requirements at a space base, has a much greater significance than for a comparable number of personnel being used on the ground.
3. A high-level analysis of a ground-based OTV had already been performed in the Phase A study.

The key issues involved in SB OTV maintenance were: (1) Is maintenance necessary? (2) What systems (components) are involved? (3) What is required to ensure the SB OTV has the same degree of readiness as a GB OTV? (4) Must the SB OTV ever be returned to Earth?

3.3.8.1 Unscheduled Maintenance

Definition—Unscheduled maintenance is defined as the repair of components that fail on a random basis or due to an unscheduled event (e.g., accident). This function is important because the mission success prediction is based on full restoration of the complete system prior to each flight.

Need for Unscheduled Maintenance—The issue of the necessity for unscheduled maintenance was addressed by performing an OTV reliability analysis to determine if any components fail during flight and, if so, what are the consequences if the failures are not corrected.

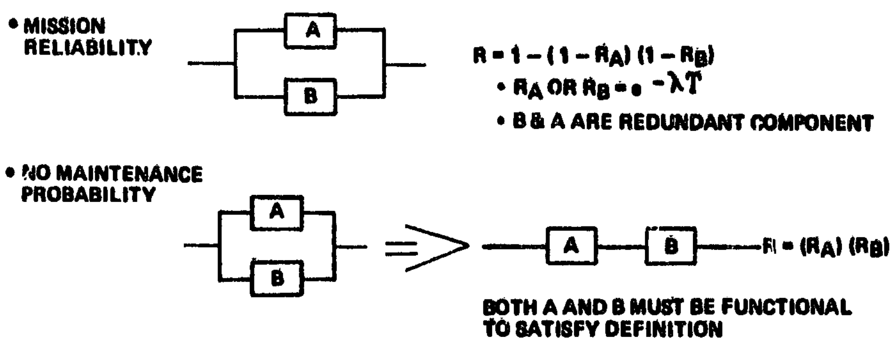
Determination of the probability that no component fails (which also means no maintenance is necessary) was accomplished by placing all parallel redundant components in series and applying the appropriate failure rates and time factors. This approach is necessary since each component must be functional to satisfy the mission success prediction of 0.978 as indicated in section 3.3.7. An illustration comparing the reliability block diagrams for mission-success probability and no-maintenance probability is shown in table 3.3.8-1. The reliability block diagrams used for the analysis are those previously

Table 3.3.8-1 No-Maintenance Probability Overview

- THE PROBABILITY THAT NO COMPONENT WITHIN THE OTV WILL REQUIRE CORRECTIVE MAINTENANCE PRIOR TO THE INITIATION OF ANOTHER FLIGHT

- STORAGE TIME INCLUDED

- NO MAINTENANCE PROBABILITY VS MISSION RELIABILITY MODELING



shown in section 3.3.7.2 but adjusted for the series format. The results of this analysis are shown in table 3.3.8-2. The probability of no maintenance being necessary (i.e., no component has failed) is only 0.388, which gives a strong indication that a failure is likely. These data were also used to establish the mean missions to repair (MMTR), which indicate a value of 1.06, meaning a failure can be expected on nearly every OTV flight.

The consequence of not correcting (or restoring) the failed components is shown using several methods. The first is for the reliability prediction of an individual mission as shown in figure 3.3.8-1. The data indicate, for example, that if the OTV has been fortunate enough to make it through nine flights without any repair, the predicted reliability for the 10th flight would be 0.90. With restoration (repair), however, a

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Table 3.3.8-2 SB OTV Subsystem Reliability Assessment

SUBSYSTEM	TOTAL MISSION RELIABILITY	NO MAINTENANCE PROBABILITY	MMTR*
ACS	.9910	.487	1.39
EPS	.9982	.9236	12.58
PROPULSION	.9940	.8425	7.37
AVIONICS	.99966	.938	15.7
AVIONICS T/C	.99913	.999126	1144
STRUCTURE	.99961	N/A	N/A
-------	-----	------	------
SYSTEM	.97826	.388337	1.06

* MEAN MISSIONS TO REPAIR = $\frac{1}{N\lambda T}$; N = NO. OF COMPONENTS; λ = FAILURE RATE; T = 1 MISSION

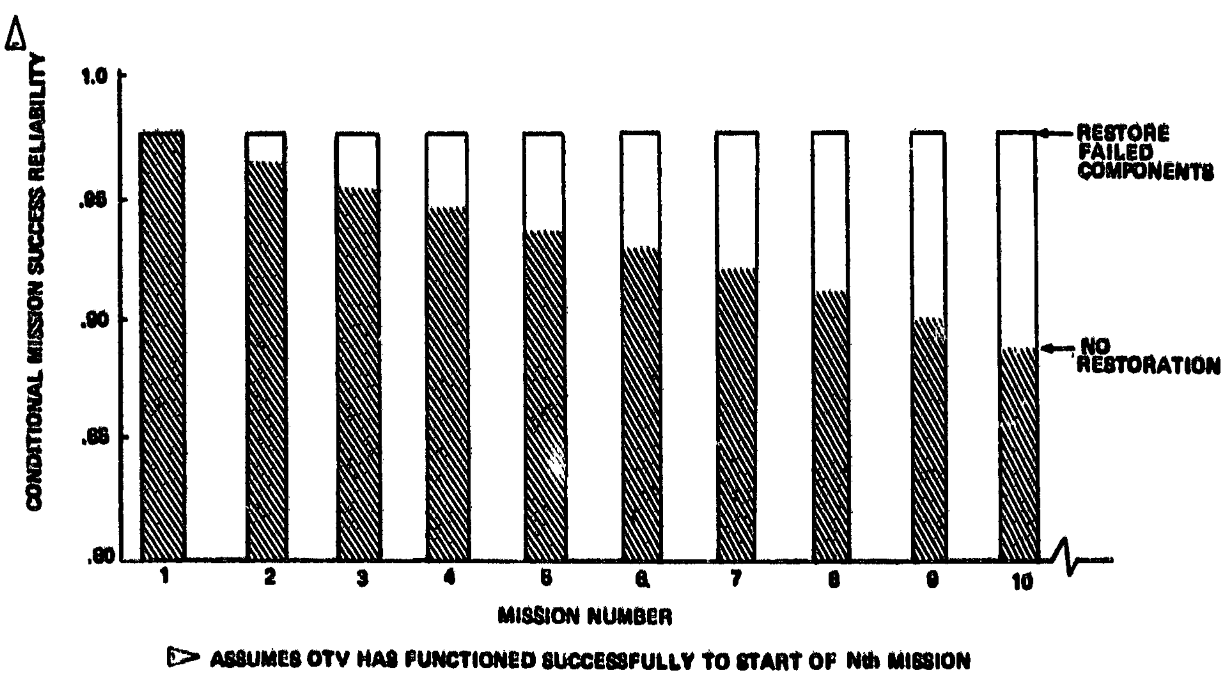

Figure 3.3.8-1 Predicted Reliability for Individual Mission—SB OTV

prediction of 0.975 would always occur. Figure 3.3.8-2 illustrates the difference in cumulative predicted reliability for a given flight. For the case of predicting the reliability of performing 10 flights, a value of 0.80 occurs with restoration; while the no-restoration approach results in only 0.50. In summary, with the high priority characteristics of many of the DOD missions and the high cost of the payloads, operating an OTV without full restoration seems unacceptable. Consequently, the need to provide the capability to perform unscheduled maintenance appears substantiated.

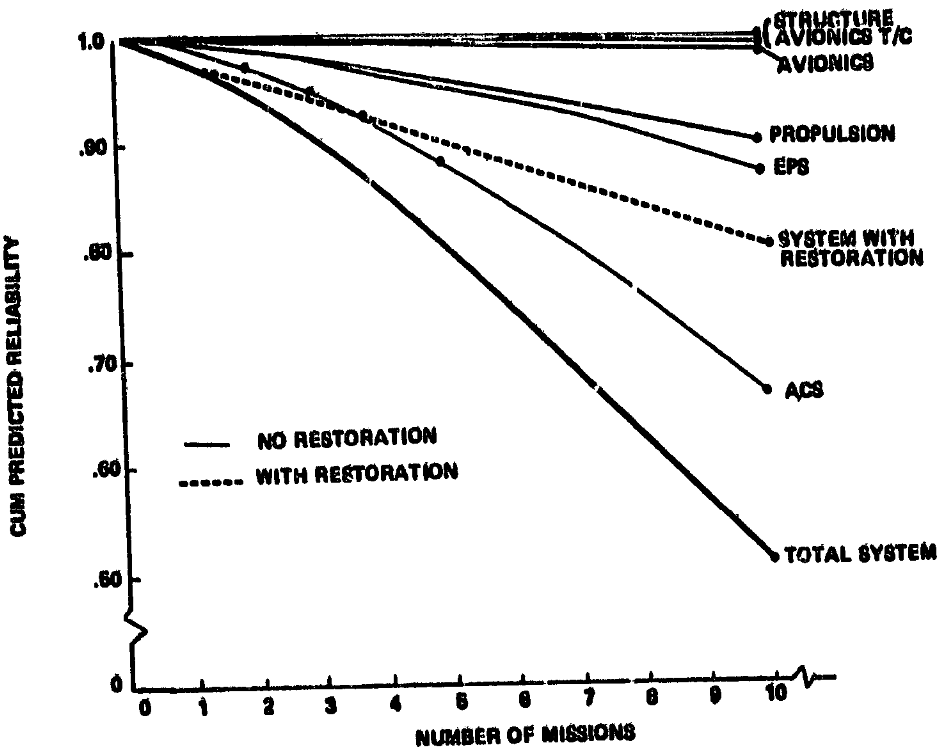

Figure 3.3.8-2 Multimission Predicted Reliability—SB OTV

Maintenance Concept—The overall maintenance concept employed was generally that of removing failed components and replacing them with ready-to-go components rather than on-site repair. To accomplish this operation, the component to be removed and replaced (R&R) was to be of sufficient size that the task could be accomplished in a zero gravity environment and by crewmen in pressure suits.

A listing of the components in terms of their total expected failures and MMTR for a single mission is presented in table 3.3.8-3. These values reflect the total number of components of a given type as well as their failure rates. It should also be noted that

Table 3.3.8-3 Component Unscheduled Maintenance Requirements

SUBSYSTEM	COMPONENT	QUANTITY	TOTAL FAILURES (MISSION & STORAGE)	MMTR ▷
ACS	REM VALVES	40	.600	1.67
ACS	REM NOZZLE	24	.139	7.2
EPS	FUEL CELL	2.	.076	13.2
PROPULSION	MAIN ENGINES	2	.0416	24.
AVIONICS	COMPUTER MEMORY	850	.0228	44.
AVIONICS	COMPUTER CPU	2	.00974	103
PROPULSION	FUEL INLET VALVES	4	.008	125
AVIONICS	LASER GYROS	2	.00752	133
ACS	REM CONNECTORS	24	.005982	167
PROPULSION	THERMO VENT VALVES	2	.004340	230
AVIONICS	TT&C RECEIVER	2	.003504	285
PROPULSION	TANK REG VALVES	4	.002902	345
	13 COMPONENTS			400 - 1000
	68 COMPONENTS			> 1000
				Σ =1.08

▷ MEAN MISSIONS TO REPAIR

many of these components are small and/or are integrally a part of another unit or assembly, thus making R&R difficult.

The approach selected to satisfy the R&R criteria was consequently that of grouping the components into more easily handled units called space removable units (SRU). The resulting SRU's and the components they contain are shown in table 3.3.8-4. Four basic SRU categories are indicated, with the avionics modules actually involving six different types of units. It should also be noted that some items have been assigned as a ground maintenance task, meaning the SB OTV must be returned to Earth. This assignment is the result of the components being in locations or having design features which make them essentially impossible to R&R in space. Other components had MMTR's so large that the design provision to allow R&R would not be justified.

Benefit of On-Orbit Maintenance—The principal benefit in providing on-orbit maintenance provisions is that it increases the number of missions that can be flown before the OTV must be returned to Earth for unscheduled maintenance. This point is shown in table 3.3.8-5 through the use of the four different types of SRU's. Again it should be

Table 3.3.8-4 Component Maintenance Assignment

SUBSYSTEM	COMPONENT	SRU ASSIGNMENT				GROUND MAINTENANCE REQUIRED
		ACS THRUSTER MODULE	FUEL CELL MODULE	MAIN ENGINE MODULE	AVIONICS MODULES	
ACS	REM VALVES	✓				
ACS	REM NOZZLE	✓				
EPS	FUEL CELL		✓			
PROPUL	ENGINE			✓		
AVIONICS	COMPUTER MEM BIT				✓	
AVIONICS	COMPUTER CPU				✓	
PROPUL	FUEL INLET VALVE			✓		
AVIONICS	LASER GYRO IMU				✓	
ACS	REM CONNECTOR	✓				
PROPUL	THERMO VENTS					✓
AVIONICS	TT&C RECEIVER				✓	
PROPUL	TANK REG VALVES					✓
	OTHERS	• HEATERS • CAT BEDS	• VALVES - O_2, H_2, H_2O	• PRESS VALVES • PUMP SEAL PURGE VALVES	• TTC TRANS • TTC PWR AMP	• COMPONENTS WITH MMBR > 400

Table 3.3.8-5 SB OTV Mean Missions to Earth Return (MMTER)

LEVEL OF SPACE MAINTENANCE	OTV MMTER ▷
• NONE	1.06
• THRUSTER MODULE	4.76
• PLUS FUEL CELLS	7.38
• PLUS MAIN ENGINE	11.7
• PLUS AVIONICS MODULES	29.04

▷ RETURNED TO EARTH FOR NON SPACE MAINTAINABLE COMPONENTS

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noted that if no unscheduled maintenance provisions are provided, the SB OTV would have to return to Earth essentially every flight to repair the failure. By providing R&R provisions for the ACS thruster module, the number of missions between Earth return is increased to one in about every five flights. By providing all the indicated R&R capability, the OTV only has to be returned to Earth every 29 flights. Over the entire mission model of 180 OTV flights, this results in approximately six relaunches of OTV's, assuming only the failed components are restored when on the ground. These relaunches are in addition to the four OTV launches necessary to satisfy design life considerations (45 flights per stage).

The sensitivity of the total number of OTV launches (excluding those for backup vehicles) to the frequency of Earth return is shown in figure 3.3.8-3. These data indicate the number of OTV launches is quite sensitive if the OTV must be reutrned to Earth more frequently than the reference of every 29 flights. Incorporating additional R&R provisions into the OTV to the point of matching the frequency of Earth return with the design life (45 flights) is also a possibility. This approach, however, only decreases the number of OTV launches by two. Out of a total of approximately 120 SDV launches, this is not judged too significant, in addition to resulting in more burnout weight and, consequently, a performance penalty.

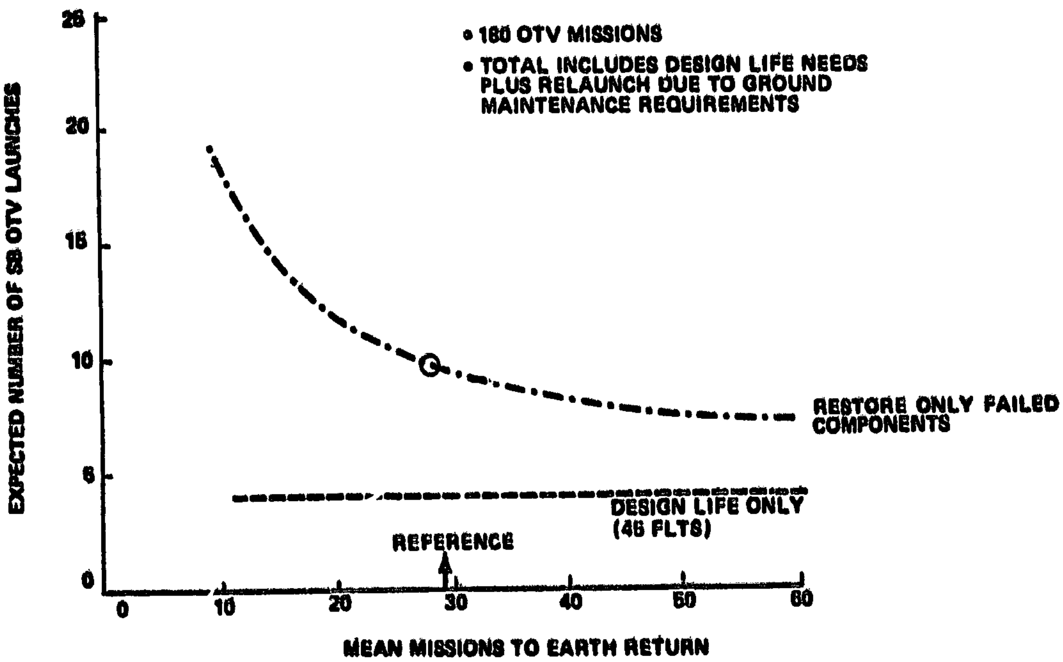

Figure 3.3.8-3 Sensitivity of SB OTV Launches to OTV Mean Missions to Earth Return

C-2

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Penalties—Providing the necessary maintenance features to enable the SB OTV to remain on orbit as indicated does result in some penalties to the vehicle and SOC. In the case of the vehicle impact, each module must have design modifications, such as simplified mounting provisions and quick-disconnect electrical and fluid connections. Examples of the mounting provisions for the main engine and ACS thruster modules are illustrated in figure 3.3.8-4. The quick disconnects (QD) and mass characteristics associated with the

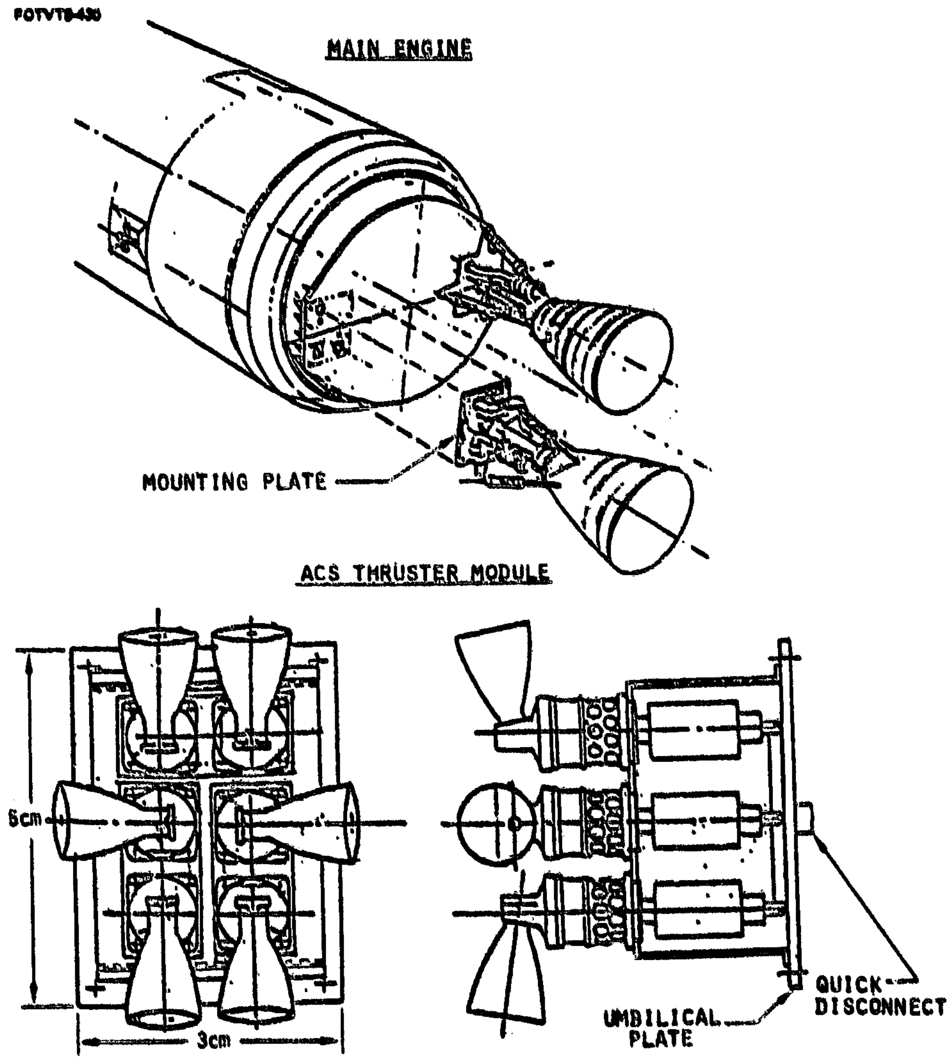


Figure 3.3.8-4 SRU Installations

SRU's are presented in table 3.3.8-6. The mass impact for each SRU reflects the use of QD's and special mounting plates and indicates a range of 10% to 25% of the mass of the basic unit. The total mass penalt was 234 kg, including the required built-in test equipment and instrumentation.

Table 3.3.8-6 SB OTV Maintenance Provision Penalty

SRU TYPE	QTY	CHARACTERISTICS PER UNIT					
		FLUID QD	ELEC QD	FAST-NERS	SIZE (CM) LxWxH	MASS IN KG BASIC	MASS IN KG Δ FOR SRU [1]
AVIONICS							
• LASER GYRO IMU	2	--	2	4		30	7
• GPS RCVR/PROC.	2	--	2	4		18	5
• TRANSPONDER	2	--	2	4	20 X 14 X 42	14	2
• RF AMPLIFIER	2	--	2	4	60 X 35 X 23	9	2
• COMPUTER	2	--	3	4	39 X 35 X 23	27	5
FUEL CELL	2	7	2	4	46 X 30 X 20	23	6
THRUSTER MODULE	4	1	1	4	30 X 20 X 30	30	3
MAIN ENGINE	2	5	1	10	356 X 187 X 187	209	62

- TOTAL SRU MASS CHANGE = 170
- OTHER CHANGES = 64
  - BUILT-IN TEST EQUIP (23)
  - INSTRUMENT & CABLING (41)

TOTAL MAINT PENALTY 234 KG

[1] INCLUDES MOUNTING PLATES AND ALL QUICK DISCONNECTS

Crew size and time requirements to accomplish the unscheduled maintenance are indicated in table 3.3.8-7 and have been based on the worst (extreme) case. As indicated

Table 3.3.8-7 Unscheduled Maintenance Activity—Extreme Case

SRU ACTIVITY	MAINTENANCE TIME (IN MINUTES) AVIONICS MODULE	THRUSTER MODULE	FUEL CELL MODULE	ENGINE MODULE
• MAINTENANCE PREP (GATHER TOOLS, SRU AND REACH POSITION)	20	20	20	40
• REMOVE AND REPLACE (INCLUDES 50% FACTOR)	28	28	28	105
• INSPECT AND C/O	20	60	180	240
• WRAP UP (RETURN TOOLS, ETC)	20	20	20	30
SUBTOTAL	88	128	248	415

(COMBINED TOTAL = 879 MINUTES (14.6 HRS)

WORK SHIFTS FOR SRU'S
- 300 MIN AVAIL/SHIFT (1)
- 2.3 SHIFTS

OTHER RELATED EVENTS
- PRE-MAINT = 60
- POST-MAINT = 56
- VISUAL INSPECTION = 120-180

CREW REQUIREMENTS
- ONE SHIFT PER DAY
- 2 EVA (ELECTRICAL/MECH)
- 1 IVA (C/O SPECIALIST)
- GROUND SUPPORT REQUIRED

- TOTAL TIME 3 SHIFTS

(1) TOTAL WORK SHIFT = 480 MIN.

earlier, there is a 99.8% chance of there being no more than four repair jobs per flight. Four R&R jobs were, therefore, assumed and consisted of one of each type of SRU (i.e., a main engine, an ACS thruster module, an avionics module, and one fuel cell module). Each maintenance action involved four types of activity. The times indicated reflect timeline work associated with tests being performed in the NASA MSFC Water Immersion Facility using components similar in mass and size. It should also be noted that achievement of the indicated times to perform the maintenance actions strongly reflects "clean sheet" installation interfaces rather than use of existing types of interfaces. In addition to the actual maintenance activity, there are related events involving premaintenance which include time for the extravehicular activity (EVA) crew to reach the hangar and hangar activation. Post-maintenance activity is essentially the reverse. Visual inspection will also be required.

Two EVA crewmen are necessary to perform the actual hands-on work while a third member remains within SOC and conducts the checkout operations. Based on a 390-min useful work period in a work shift, three shifts are required. To minimize the crew size at SOC, only one shift per day was assumed.

3.3.8.2 Scheduled Maintenance

This area generally deals with: (1) those items that have wearout characteristics less than the total OTV design life; (2) expendable hardware elements and; (3) those components which require regular servicing. A listing of scheduled maintenance activities for the SB OTV is shown in table 3.3.8-8. The most significant items are those of replacing the main engines and ballute. In the case of the main engine, the mean mission to repair prediction is every 24 flights, so the scheduled replacement every 20 flights (10 hr) may preclude some of the unscheduled maintenance activity for this unit. Ballute installation after each flight appears to offer a significant challenge in that there are numerous attachments occurring over most of the vehicle external surface.

Table 3.3.8-8 Scheduled Maintenance Activities

ACTIVITY	DESCRIPTION	RESOURCES PER ACTIVITY
• REPLACE MAIN ENGINES	• EVERY 10 HOURS OF OPERATION (APPROXIMATELY 20 FLIGHTS)	• SAME AS FOR UNSCHEDULED MAINTENANCE
• REPLACE BALLUTE SYSTEM • INSTALL BALLUTE UNIT - AFT END • STRING AND ATTACH 80 RESTRAINING STRAPS AT FWD END	• EVERY FLIGHT • 310 KG • TORROIDAL SHAPE 3.56 M MAJOR DIA 0.31 M MINOR DIA	• 2 PEOPLE EVA • 1 PERSON AT CONT. CENTER • 1 SHIFT
• BATTERY CHARGE	• EVERY FLIGHT • AUTO HOOK-UP	• 1 PERSON AT CONT. CENTER -
• FUEL CELL PURGE	• EVERY FLIGHT • AUTOMATIC	• 1 PERSON AT CONT. CENTER
• MAIN LH_2 TANK VENT	PRIOR TO LOADING AUTOMATIC	• 1 PERSON AT CONT. CENTER
• PROVIDE THERMAL CONTROL FOR AVIONICS AND ACS	• WHILE AT BASE IN - STORAGE MODE	• HANGAR WALL COATINGS CAN PROVIDE SUFFICIENT CONTROL • BASE STANDBY PWR AVAIL.

3.3.8.3 Checkout Concept

An integral part of the maintenance operation is the function of checkout. This function is defined to include the ability to assess the condition of the system as well as the detection of faults and isolation of faults to the appropriate SRU. A major source of condition assessment is the evaluation of data after a flight has been completed. Real-time assessment is also required, such as after a new SRU has been installed. Fault detection and isolation to an SRU will also require extensive built-in instrumentation.

Concepts—Three checkout concepts were considered with the key difference being the location for initiating the checkout and analysis of data. One concept used ground-based control, another used SOC-based equipment and personnel, and the third had all the capability built into the OTV. A brief description of each follows.

1. Concept 1: ground-based control—Instrument the vehicle to the level required for condition assessment and provide vehicle-mounted built-in-test equipment (BITE) to stimulate or simulate equipment for the purpose of obtaining condition assessment and fault detection and isolation data. The data from the vehicle would be

transmitted to the ground where the data analysis would be accomplished resulting in the identification of the faulty SRU and indication of system status. The required maintenance actions would then be transmitted back to the maintenance crew located at the SOC.

2. Concept 2: SOC-based control—This concept is the same as Concept 1 except all equipment and personnel associated with data analysis are located at SOC.
3. Concept 3: OTV control—Instrument the vehicle to the level required for condition assessment and provide vehicle-mounted BITE to stimulate and simulate equipment and also equipment to detect and isolate faults to the SRU level without any assistance from external sources.

Comparison—The comparison of the three concepts is summarized in table 3.3.8-9. For Concept 1 (ground-based control), the most significant impact to the LEO base is the data and command link to the Earth. The particular areas of concern are the bandwidth required and the methods required to ensure the integrity of the data path. The estimated time for automatic checkout using either Concept 1 or 2 would be 2 to 3 hr.

Table 3.3.8-9 Automatic Checkout Techniques—SB OTV

AUTOMATIC CHECKOUT TECHNIQUE	VEHICLE MASS IMPACT (KG) ①	DATA LINK ②	SOC PERSON.	DATA STORAGE	DISPLAYS & CONT. & S/W
GROUND BASE CONTROL	64	MAXIMUM	MIN.	MOD.	MIN.
LEO BASE CONTROL	64	MODERATE	MAX.	MAX.	MAX.
VEHICLE CONTROL (GRD FLT DATA ANALYSIS)	114	MINIMUM	MOD.	MIN.	MOD.

- AUTOMATIC CHECKOUT UTILIZES BUILT-IN-TEST EQUIP.
  - NOT USED DURING MISSION
  - PROVIDES STIMULATION AND SIMULATION INPUTS AND TEST SEQUENCES
  - REQUIRES APPROX THREE HOURS

① - COVERS INSTRUMENTATION AND BUILT IN TEST EQUIPMENT
② - USES TDRS AND STDN

For the case of Concept 2 (LEO-base control), the principal impacts are the increased number of personnel required for checkout (versus vehicle maintenance personnel only for ground-based control) and the increased hardware and software required for the test and checkout task.

In Concept 3, where the vehicle itself contains sufficient built-in-test equipment and software for fault isolation to the SRU level, the vehicle avionics are considerably more complex. Since the avionics are more complex, the equipment failure rates will most likely increase. While the failure rates of that portion of the equipment used for the mission may be negligibly affected by the addition of BITE, the total failure rate (derived from the sum of mission plus BITE) will increase and the equipment can be expected to require more maintenance. Additional instrumentation is also required to monitor the performance of the vehicle-mounted BITE. The additional mass penalty for this concept is due to the equipment which must be added to each SRU to indicate its status (acceptable or nonacceptable) in addition to extra computer memory. The estimated time for accomplishing automatic checkout using this concept is 0.5 to 1 hr.

Selection—Based on this brief study of checkout concepts for a space-based OTV, it is recommended that ground-based control of the automatic checkout be used as the baseline. Except for the integrity of the base-to-Earth data and command communications link, this concept minimizes the impact to both the vehicle and the LEO base (SOC). Further discussion regarding checkout in terms of the total turnaround operations associated with an OTV is presented in section 3.3.10.

3.3.8.4 Maintenance Facility

As previously discussed in section 3.3.6, OTV protection against space debris while at SOC can be most effectively accomplished by providing a hangar, which could also have all the necessary provisions for OTV maintenance. Such a hangar, with an OTV, was shown in figure 2.2-1. Such a facility is judged to be beneficial in that it could provide the necessary lighting, containment of personnel and spares, work platforms, and a storage area for spares and maintenance equipment. A shirtsleeve environment is not viewed as necessary since the SRU's are envisioned to be replaced by crewmen in pressure suits. In addition, the penalty for a shirtsleeve environment would involve the loss of the atmosphere each time an OTV is removed or, if the atmosphere is to be recovered, a large amount of electrical power would be required. Although each OTV (stage) based at the SOC will need a hangar for space debris protection, only one hangar needs to be provided with maintenance provisions.

3.3.9 Refueling

This section discusses the comparison of several refueling options for the main propellant of the OTV, a system description of transferring other fluids required by the OTV, and a definition of the associated propellant tanker.

Refueling is defined as that function dealing with replenishment of all consumables (fluids). Included within this category are primary propellants, secondary propellants, reactants for electrical power, and pressurization fluids. This analysis deals only with the SB OTV since fueling of systems at a launch pad (such as for a GB OTV) is a fairly well understood operation.

The overall goal of the SB OTV refueling analysis was to select a concept which provided the best combination of acceptable cost, complexity, and risk. Within this goal was the desire to determine if active propellant conditioning systems (e.g., refrigerators, liquifiers) are beneficial with the OTV traffic rates, as indicated by the FOTV mission model and when used in conjunction with SOC.

The major factors considered in the refueling analysis are shown in figure 3.3.9-1.

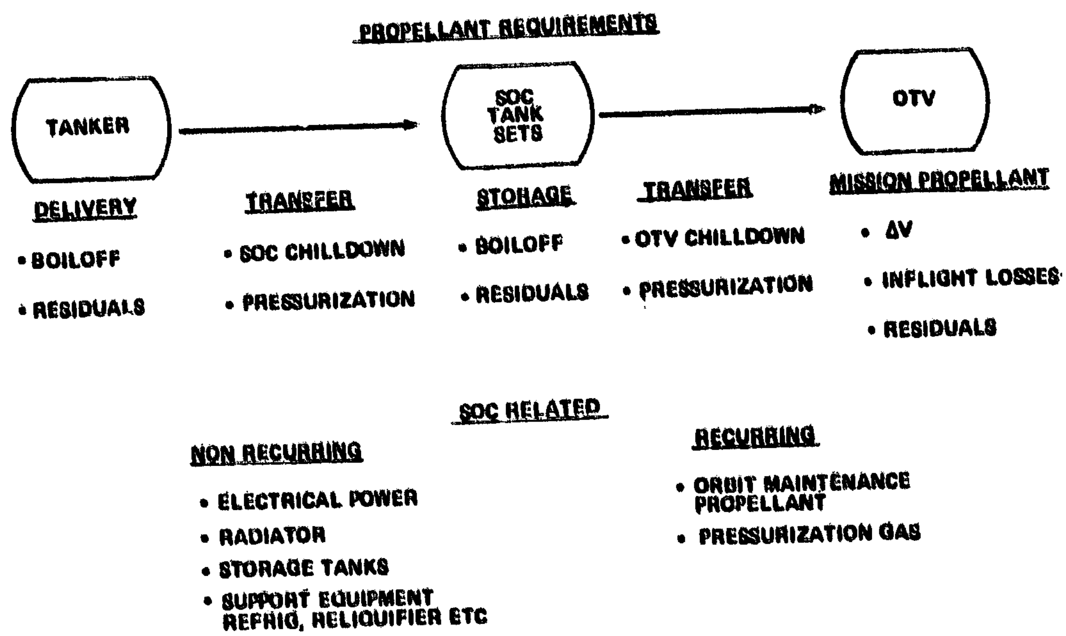


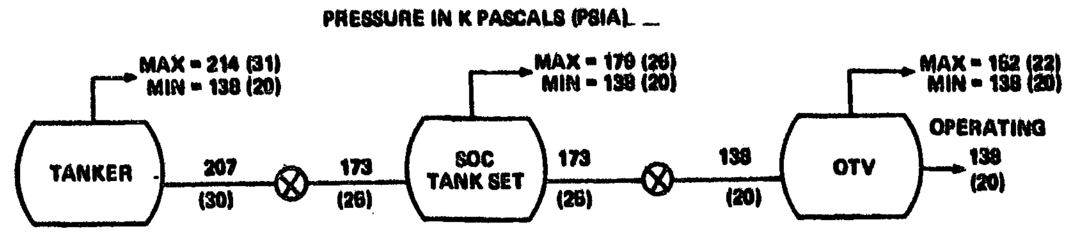
Figure 3.3.9-1 OTV Refueling Factors

The major system elements in the refueling operations are (1) the tanker which delivers propellant from Earth to the SOC, (2) the SOC storage tanks (referred to as tank sets as each contains an LO_2 and LH_2 tank and tanks for other refueling fluids), and (3) the OTV. Propellant requirements are to account for mission propellant as well as all that which is lost or unavailable either during the delivery to orbit, transfer between tanks, or on-orbit storage. Refueling may also require various types of equipment and provisions located at the space base, which is assumed to be the SOC. A review of the applicable data base, most notably reference 6, indicated a reexamination of the refueling operation was necessary due to the differences in ground rules and the desire to investigate several different concepts.

The major issue associated with the refueling of an SB OTV is the amount of losses associated with the delivery, storage, and transfer of main propellant. This issue is presented in section 3.3.9.1. Other fluids such as hydrazine, nitrogen, and helium also require resupply; however, their losses constitute only a small fraction of the main propellant. Consequently, it was decided the transfer of these fluids would only receive a minimum of analysis. These data are presented in section 3.3.9.2. Design characteristics of the tanker used in the refueling operation are presented in section 3.3.9.3.

3.3.9.1 Main Propellant Refueling

Systems Characteristics - The characteristics assumed for the major refueling system elements and their utilization are shown in figure 3.3.9-2. These characteristics were



- REUSABLE
- 20 LAYERS MLI
- ~61.4 MT CAPACITY
- FULL SCREEN PROP ACQUISITION SYS
- HELIUM PRESS SYS
- TRANSFER TIME: <20 HRS

- PROP STORAGE REQ'T ~88.4 MT
- STORAGE SYSTEM: (2) 59.4MT CAPACITY TANK SETS
- FULL SCREEN ACQUISITION SYSTEM
- 50 LAYER MLI
- TANK WALLS 500° R WHEN LOADED
- TRANSFER TIME: <20 HRS
- UTILIZATION PROFILE

OTV FLT	TIME (DAYS)	TS NO 1	TS NO 2
	0	FULL	FULL
1	20	½	FULL
2	40	0	FULL
3	60	FULL	½
4	80	FULL	0
5	100	½	FULL
6	120	0	FULL
7	140	FULL	½
	ETC		

- ENGINE INLET: 110 K Pa (16 PSIA)
- PROP CAPACITY:
  SINGLE STG ~ 29.8 MT
  TWO STG ~ 59.4 MT
- TOTAL FLIGHTS
  11 YR: 180
  16 YR: 280
- FLIGHT INTERVAL
  18 - 20 DAYS
- TANK WALLS 500° R WHEN LOADED

Figure 3.3.9-2 OTV Refueling Assumptions established to provide the framework to define and evaluate candidate refueling concepts. The pressures indicated are based on the reference OTV engine inlet pressure of 110 kPa (16 psia) and estimated line losses.

The tanker was sized for delivery by a shuttle-derivative launch vehicle. An MLI thermal control approach was used rather than a dewar for this application because of being generally lighter, less costly, and having less risk. The propellant acquisition system is the same as proposed in reference 6. This system consists of screens located near the tank walls that provide a capillary action to acquire the propellant so no g-field is necessary.

Propellant storage requirement at SOC (based on the initial SB OTV point design) was established as 86.4t. This resulted from a situation involving an OTV mission requiring 59.4t of propellant (for the largest payload in the model) in addition to a need to perform a rescue mission (27t of propellant) to a manned OTV or a GEO base prior to a tanker delivery. The storage tanks were sized for the amount of propellant available after transfer from the tanker. Full screen propellant acquisition systems were used and additional layers of MLI applied to reduce boiloff; however, more than 50 layers does not show much benefit. Data presented in figure 3.3.9-3 indicate the resulting boiloff would

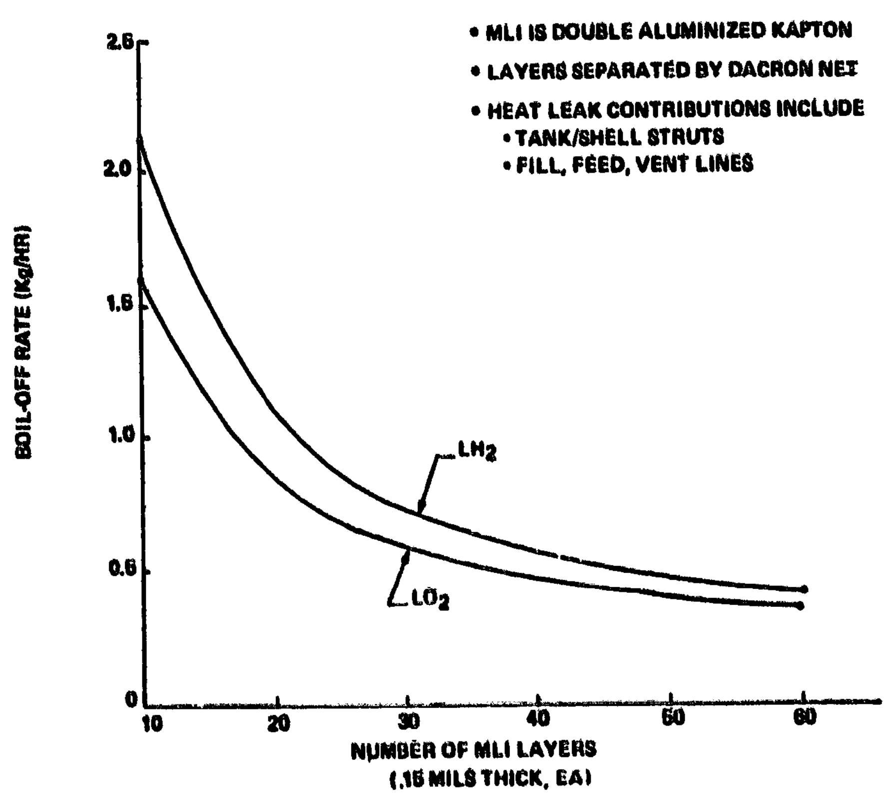


Figure 3.3.9-3 Propellant Storage Tank Boiloff

be 0.45 kg/hr for LH_2 and 0.36 kg/hr for LO_2. Two tank sets (each containing an LO_2 and and LH_2 tank) are suggested to satisfy the storage requirement, with each capable of handling a full tanker load after transfer losses and residuals are considered. The utilization plan (approximation) assumed for each tank set is indicated and results in 80-day cycles between loadings. For this analysis, one-half of the propellant of each tank set is assumed used for each OTV flight.

The propellant capacities indicated for the OTV reflect values that were associated with the initial SB OTV point design. This vehicle did not incorporate space debris protection, maintenance provisions, and other features that were incorporated into the final OTV design described in section 3.3.3. The final SB OTV single-stage design required 32.5t of propellant. The 10% difference in mission propellant requirement, however, is judged insufficient to change the results of the refueling analysis since the associated losses would increase only by approximately 1%.

Refueling Concepts - The refueling concepts analyzed generally relate to the pressurization method used to transfer propellant from the SOC tank set to the OTV. All concepts use helium pressurization to transfer the propellant from a tanker to SOC storage tanks. The names given to the concepts are listed below:

Concept A: Independent pressurization
- A1: Helium pressurization
- A2: Thermal pressurization
- A3: Thermal pressurization with boiloff recovery

Concept B: Recovered vapor pressurization
- B1: Recovered vapor pressurization
- B2: Recovered vapor pressurization with boiloff recovery
- B3: Recovered vapor pressurization with boiloff recovery and transfer loss recovery

Concept C: Subcooled propellant with thermal pressurization

Concept D: Tank exchange with recovered vapor pressurization

The independent pressurization concepts are shown in figure 3.3.9-4. Concept A1 uses helium to pressurize the SOC tanks to enable the transfer but this needs replenishment with each refill of a SOC tank. Concept A2 provides the required pressure via thermal means using a heater to vaporize a portion of the propellant. Concept A3 also

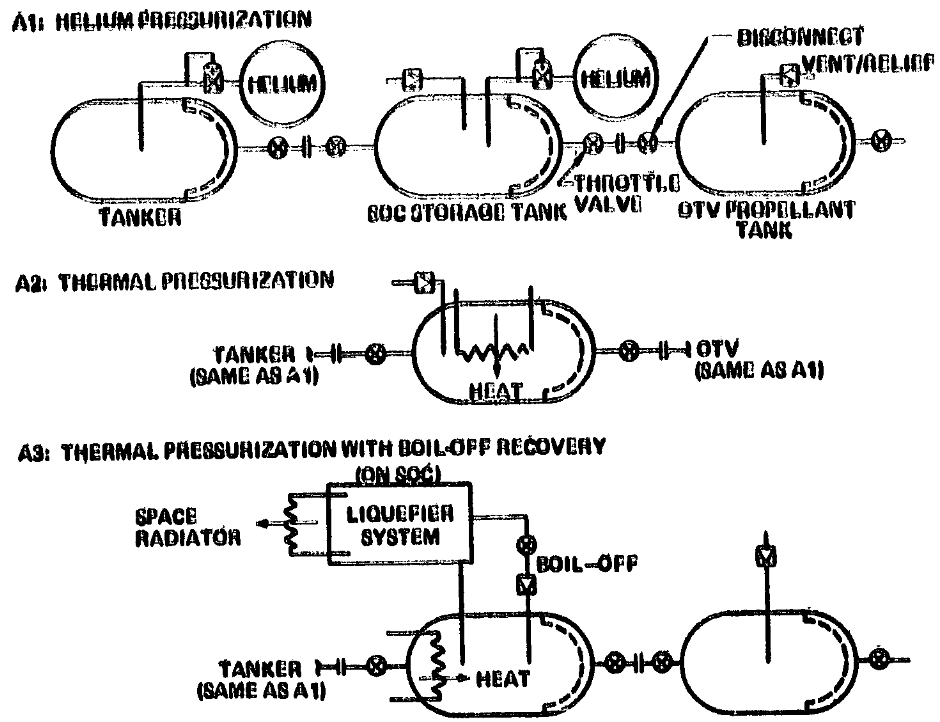


Figure 3.3.9-4 OTV Refueling Concept A—Independent Pressurization

uses thermal pressurization and incorporates a liquifier system to collect and condense the boiloff occurring from the SOC storage tanks. A reverse Brayton cycle system is assumed for the liquifier system.

The B-group of concepts uses recovered vapor for pressurization and is presented in figure 3.3.9-5. The basic characteristics of these concepts are shown by Concept B1. In this concept, vapor is produced when saturated liquid in the SOC tank is throttled to the lower OTV tank pressure. A portion of the vented vapor is passed through a compressor and returned to the SOC tank to maintain SOC tank pressure. Due to the differences in densities in the two tanks, some flashed vapor is dumped through the compressor relief valve to maintain SOC tank pressure. Concept B2 uses the same pressurization approach as Concept B1 and also incorporates a liquifier system to eliminate SOC tank boiloff. Concept B3 goes even further in reducing propellant losses by collecting and liquefying the flashed vapor occurring during transfer as well as boiloff. The much higher rates associated with the propellant transfer require considerably more power for liquefying than just for boiloff (46 kW versus 10 kW).

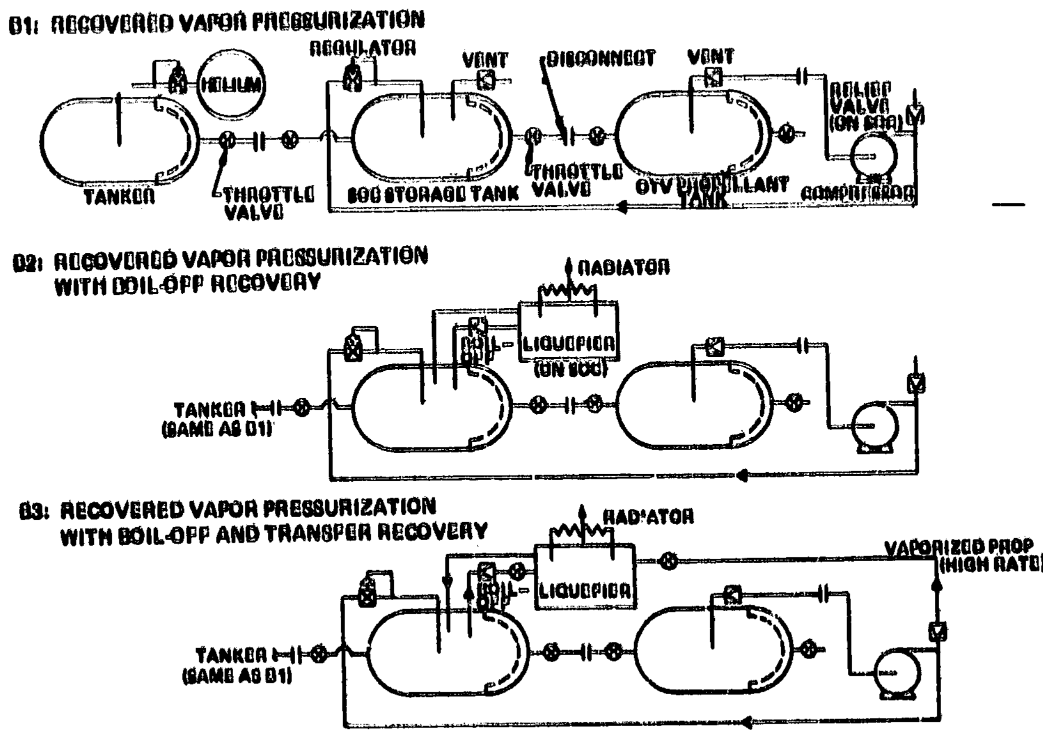


Figure 3.3.9-5 OTV Refueling Concept B—Recovered Vapor Pressurization

Concept C, shown in figure 3.3.9-6, achieves subcooled propellant prior to the transfer to the OTV. In this concept, a closed loop refrigerator system (reverse Brayton cycle) is used to cool the propellant over a 7-day period after its delivery to a SOC tank set. This approach avoids the flashing of 2% to 3% of the propellant as it is throttled to the OTV tank pressure. An internal heater is used to thermally pressurize the SOC storage tanks during propellant transfer.

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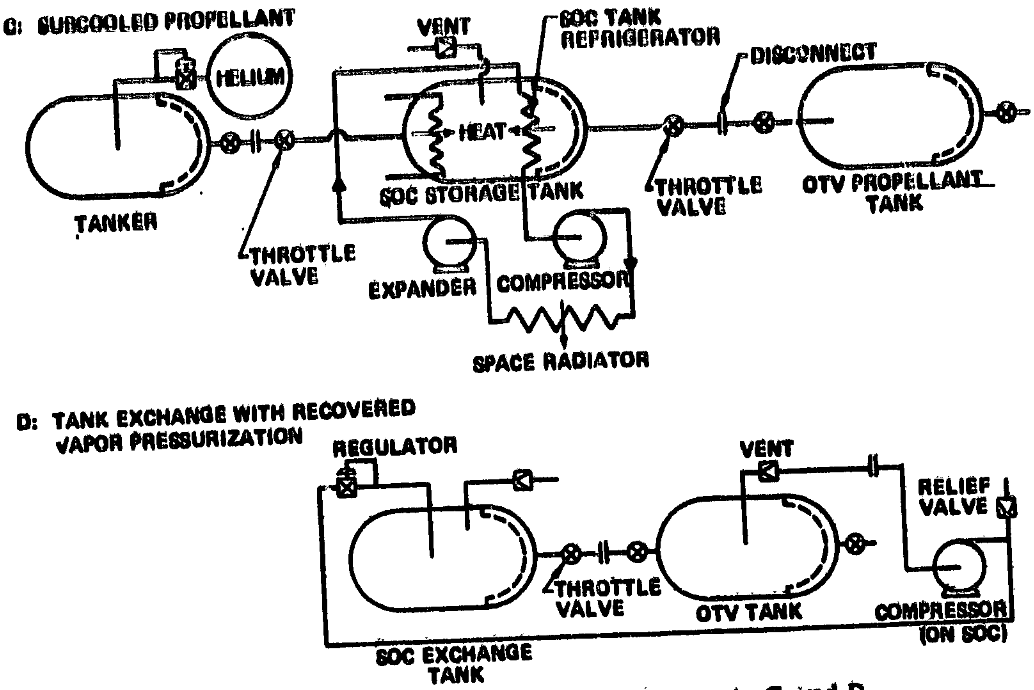

Figure 3.3.9-6 OTV Refueling Concepts C and D

Concept D, also shown in figure 3.3.9-6, is called "tank exchange." This concept uses the same pressurization approach as B1 but instead of transferring propellant from a tanker to the SOC storage tanks, an empty SOC tank set is replaced with a full tank set, thus eliminating the transfer losses.

The concept of pump transfer between the tanker and SOC tanks and between the SOC tanks and OTV was not analyzed for the following reasons. Acquisition of the propellant to the pump inlet requires screened surface channels limited in cross-sectional area as dictated by surface tension forces and screen sizes. The allowable flow rates (for which no data base exists) in the acquisition channels to prevent breakdown of the surface-tension-supported surfaces are severely limited and thus will restrict the allowable propellant transfer rates. Removal of propellant from the supply tank without repressurization will reduce the tank pressure and cause vaporization of saturated propellants. The most likely place for vaporization is within the acquisition channels and/or the screened surfaces thus tending to block liquid flow or dry off the screened surface. In either case, entry of vapor into the acquisition channels is likely to stop liquid flow into the tank outlet resulting in breakdown of propellant transfer and an unknown but potentially large amount of residuals. Design solutions to these problems can only be verified by full-scale orbital experiments. Therefore, the small potential weight advantage of pumped transfer over pressurized transfer did not warrant further analysis of this concept.

Support Requirements - The support requirements associated with each refueling concept are presented in table 3.3.9-1. The reference propellant transfer time between a SOC

Table 3.3.9-1 Refueling Support Requirements

	CONCEPTS							
CHARACTERISTICS	A1	A2	A3	B1	B2	B3	C	D
PERMANENT STORAGE TANK ON SOC	YES	YES	YES	YES	YES	YES	YES	NO, USE TANK
TANKER	YES	YES	YES	YES	YES	YES	YES	EXCHANGE
TRANSFER TIME (HRS)	4	20 ▷	4	4	4	20	4 ▷	4
POWER (KW)	0	1.3	10	NIL	10	46	10.4	NIL
SOLAR ARRAY (SQ M)	0	29	220	NIL	220	1060	230	NIL
ARRAY MASS (MT)	0	13	1.1	NIL	1.1	5.2	1.14	NIL
RADIATOR (SQ. M)	0	0	18	0	18	86	18	0
LIQUIFIER MASS (MT)	0	0	1.8	0	1.8	4.1	1.6	0
SOC W/C_D A(KG/M^2)	160	156	138	160	138	94	137	160.

▷ COULD BE 4 HRS WITH 6.6 KW ▷ BUT TAKES 7 DAYS TO SUB COOL PROP AFTER DELIVERY

tank set and an OTV was 4 hr. As indicated, several concepts show 20 hr, primarily because the power requirement becomes prohibitive for the shorter time; however, 20 hr is probably acceptable since it is still faster than a ground-based OTV could be launched. Power requirements are relatively large for those concepts using liquifier systems and extremely large for Concept B3 since the vent losses which occur at high rates during transfer are collected. The power for liquifying was taken from reference 6 which indicated 9.9 kWh/kg for LH_2 and 0.66 kWh/kg for LO_2. The SOC solar array satisfies the power requirement during the sunlight portion of the orbit, as well as recharging a secondary power system that is used during the dark portion of the orbit. Mass indicated for the power system covers both the solar array and secondary power supply. Radiator provisions are assumed to be separate from those required for the basic SOC. The average radiator was assumed to be always perpendicular to Earth and had an average temperature of 37.8°C. The indicated area also assumes the use of a two-sided design, with each square meter of radiator dissipating 0.5 kW of waste heat. The W/C_DA parameter is the key factor in establishing the amount of orbit maintenance propellant required by the SOC. Those concepts which result in lower SOC W/C_DA will have higher orbit maintenance requirements.

Propellant Requirements - The propellant requirement associated with each aspect of the refueling operation is shown in table 3.3.9-2 for each refueling concept. The values reflect the requirement associated with a single repetition. The number of repetitions

Table 3.3.9-2 Propellant Requirements for SB OTV

ITEM	REPETITIONS	CONCEPTS AND MASS PER REPETITION (MT)							
		A1	A2	A3	B1	B2	B3	C	D
1. OTV MISSION PROP	180 ▷1	28.9	28.9	28.9	28.9	28.9	28.9	28.9	28.9
2. SOC/OTV TRANSFER									
CHILLDOWN	180	0.9	0.9	0.9	0.6	0.6	0	0.3	0.6
PRESSURIZATION	90 ▷2	0.8	0.3	0.3	0	0	0	0.3	0
3. STORAGE									
BOILOFF	90	1.6	1.6	0	1.6	0	0	0	1.6
RESIDUALS	90	1.8	1.8	1.8	1.8	1.8	1.8	0	1.8
4. TANKER/SOC TRANSFER									
CHILLDOWN	90	0.6	0.6	0.6	0.6	0.6	0.6	0	0
5. TANKER									
BOILOFF	90	0.1	0.1	0.1	0.1	0.1	0.1	0.1	0.1
RESIDUALS	90	1.3	1.3	1.3	1.3	1.3	1.3	1.3	0
6. Δ SOC ORBIT MAINT PROPELLANT	180	0.6	0.63	0.68	0.6	0.68	0.96	0.7	0.6

▷1 180 REFLECTS NUMBER OF OTV FLIGHTS ▷2 90 REFLECTS SOC TANK PROVIDING 2 OTV FLTS

indicated relates to the 11-year mission model. The number of repetitions for the OTV mission propellant relates to the number of flights, while those items showing 90 repetitions reflect the number of tanker launches or SOC storage tank cycles. The orbit maintenance values reflect the requirement for the 20 to 22 days between each of the 180 OTV flights.

The OTV mission propellant requirement reflects the definition of the space-based OTV at the time of the first quarter review and, as indicated previously, is approximately 10% less than required in the final design. Childown losses occurring during the transfer from SOC to OTV reflect 28.9t being transferred to the OTV tanks, which have wall temperatures of approximately 400°C. This loss is greatest for the A concepts since no propellant vapor is recovered as in B and D concepts or subcooled as in the C concept. Pressurization loss for the transfer between SOC tanks and OTV is the greatest for Concept A1 since it uses an expendable helium system. All of the B and the D concepts show no requirement since part of the OTV tank vent loss is recovered, compressed, and sent back to the storage tanks. The boiloff value of 1.6t is associated with each tank-set (two are present) during each 60-day period in which they contain propellant. This value considers wall, support struts, and penetration contributions to heat leak. With 50 layers of double aluminized Kapton, the LH_2 boiloff rate is assessed to be 0.45 kg/hr while the LO_2 rate is 0.36 kg/hr. Use of liquifier systems in Concepts A3, B2, and B3 reduces the losses to zero. In Concept D, continuous refrigeration eliminates boiloff from occurring. Chilldown losses indicated for the tanker to SOC transfer also result from the 400°C tank walls in the receiver and the transfer of 60t of propellant. Concept C, which has continuous refrigeration, and Concept D, which exchanges SOC tanks each time, can avoid this loss. The difference in orbit maintenance propellant reflects the different W/C_DA. The mass number indicated reflects the propellant (N_2H_4) required for solar normal conditions and the SOC located at 370 km.

The total mission model refueling associated with the concepts is shown in figure 3.3.9-7. This total is found by adding the actual OTV mission propellant plus the losses peculiar to each refueling concept and variation in SOC orbit maintenance propellant. An average annual refueling requirement of over 500t occurs for the 11-year reference mission model. The losses plus orbit maintenance propellant for the refueling concepts represent a range of 6% to 14% of the actual OTV mission propellant.

Cost Comparison - The total cost associated with the refueling operation is shown in figure 3.3.9-8. A spread of approximately 5% exists between the lowest and highest cost concepts. Propellant launch cost is based on use of a standard shuttle-derivative vehicle (see sec. 3.3.11 for description) with 61.2t of propellant delivery capability. The small delta launch cost associated with Concept D reflects the fact that the exchange tank is heavier than a refueling tanker resulting in it less propellant available per launch. Liquifier/refrigerator values were established by scaling relative to the amount of power required. The 10-kW units had a DDT&E estimated to be \$50M and a unit cost of \$10M. A 46-kW unit was estimated at \$100M for DDT&E and \$25M for unit cost. Solar array costs reflect unit costs of \$57,000/m^2, with no DDT&E included since the relatively small change in area would not significantly affect the design.

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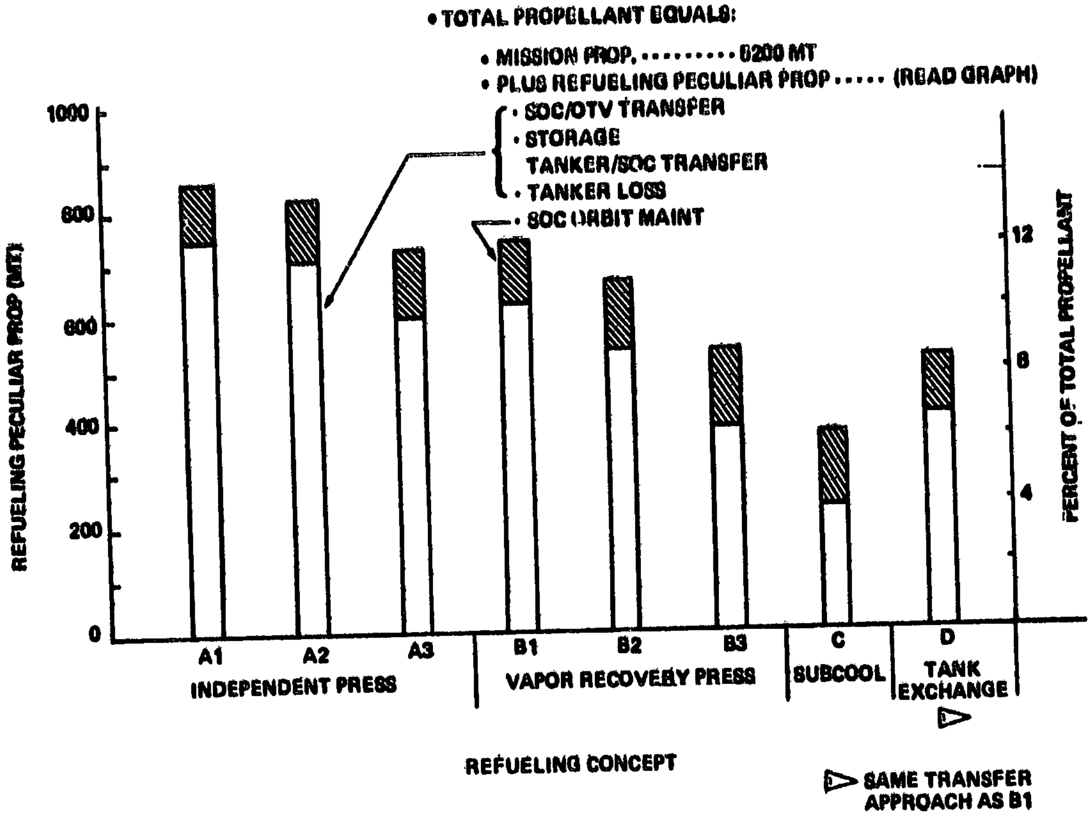


Figure 3.3.9-7 Total Refueling Requirement

Concept Comparison and Selection - Concept B2, although not the least-cost system, is judged to provide the best overall characteristics which include acceptable risk and operational complexity. Concepts A1, A2, A3, B2, and B3 are all more expensive; and A3, B2, and B3 also are more complex due to use of liquifiers. Concept C, which provided the least cost, has higher risk and uncertainty in performance due to the lack of data base concerning space-qualified refrigerators. In terms of the FOTV study, space-type refrigerators and liquifiers may be considered as accelerated rather than normal growth technology. Concept D is judged to be a major contender, although the complexity of moving large propellant tanks around the base presents some concern.

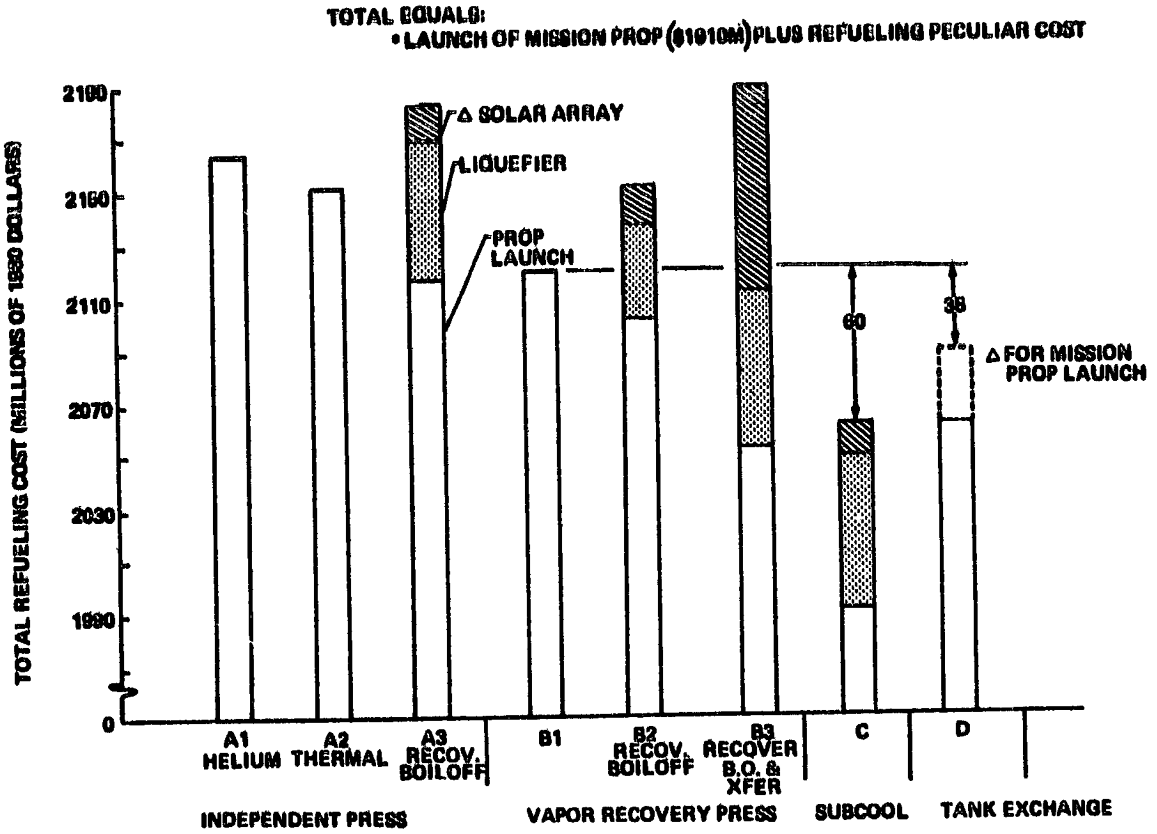


Figure 3.3.9-8 Total Refueling Cost

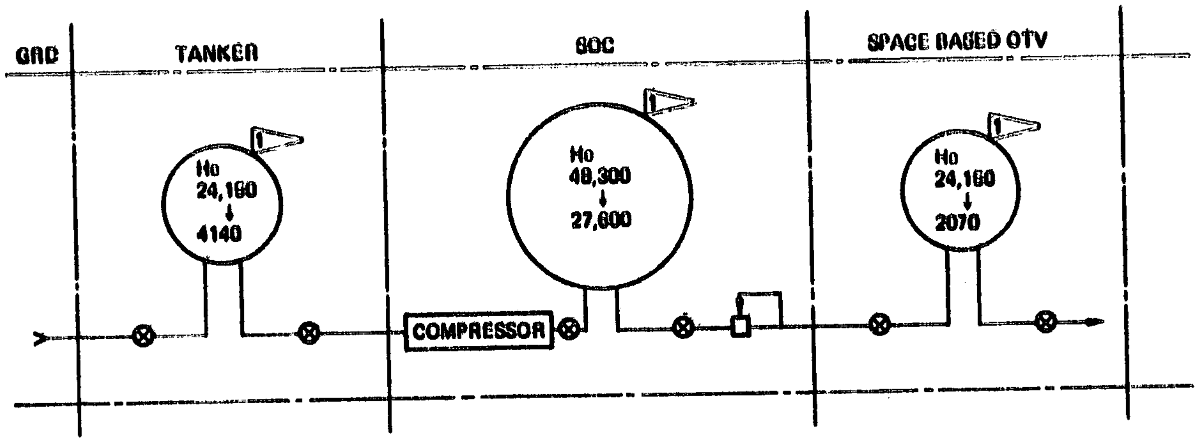
3.3.9.2 Other Fluids Transfer

Other fluids required by the SB OTV consist of reactants (O_2/H_2) for the fuel cells, pressurant (GHe) for the pneumatics system, and propellant (N_2/H_2) and pressurant (GN_2) for the ACS system.

With respect to fuel cell reactants, it is assumed that the advanced design fuel cells of the SB OTV are capable of operating on propulsion-grade O_2/H_2 (with minimal purging). Consequently they are supplied from the same storage supply as the main propulsion system.

Helium for the SB OTV pneumatics system is stored in the LH_2 tank (to minimize bottle size and mass) and as it is used, expands isothermally from a pressure of 24 150 kPa (3500 psia) to a pressure of 2070 kPa (300 psia). A fluid transfer schematic, between the tanker, SOC, and SB OTV is presented in figure 3.3.9-9. Two other notable features are that the largest and heaviest helium bottle and the compressor are both incorporated into the SOC because it is the vehicle least affected by the impact of the added mass.

ALL PRESSURES IN kPa (1 kPa = 0.145 psia)



1 BOTTLE STORED IN LH_2 TANK.
ISOTHERMAL EXPANSION @ 21°K (38°R)

Figure 3.3.9-9 Pneumatic System Helium Transfer Schematic—Tanker to SOC to OTV

The SB OTV ACS propellant is expelled from its storage tank at a constant supply pressure of 2208 kPa (320 psia) by means of GN_2 pressurant. The GN_2 is stored in a separate bottle. Both the propellant and the pressurant are maintained at (or near) room temperature. As it is used, the GN_2 expands isothermally from a pressure of 24 150 kPa (3500 psia) to a pressure of 3450 kPa (500 psia). A fluid transfer schematic, between tanker, SOC, and SB OTV is presented in figure 3.3.9-10. Two notable features are that the propellant and pressurant are always maintained at (or near) room temperature and that the largest and heaviest pressurant bottle and the compressor are incorporated into the SOC.

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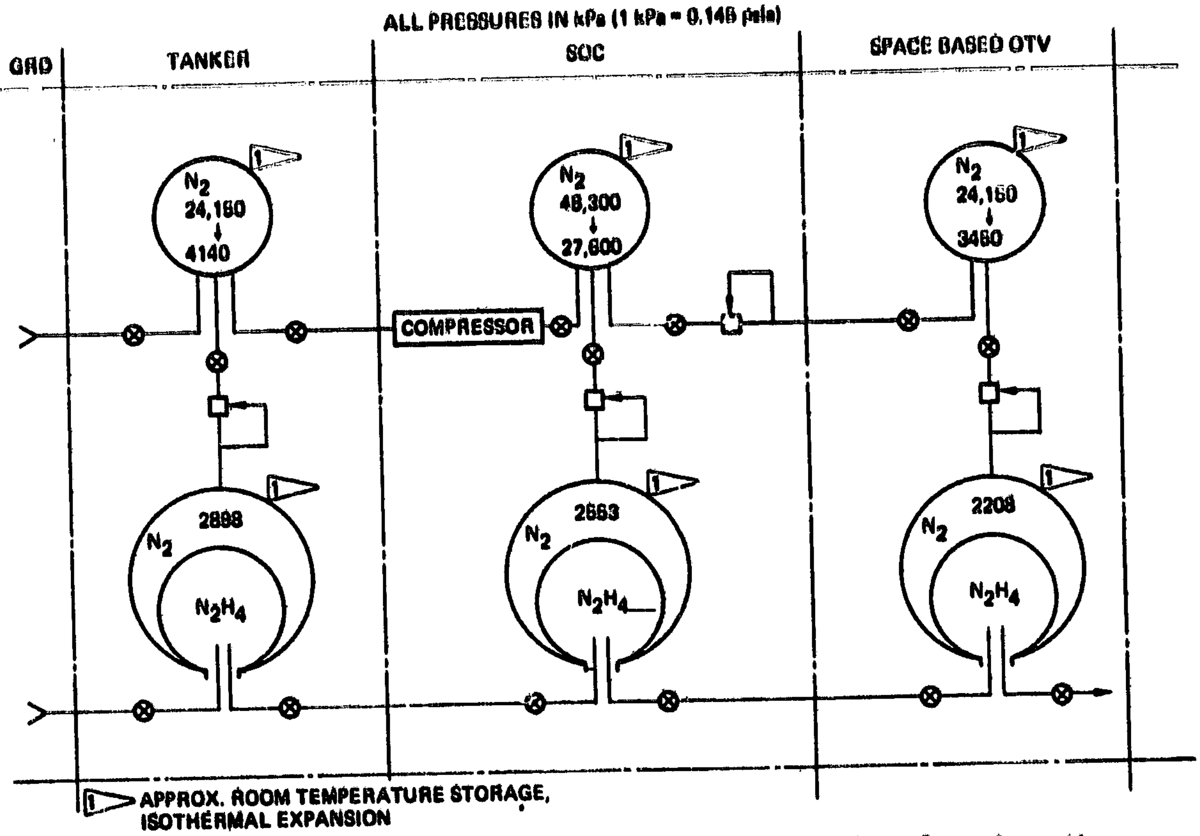
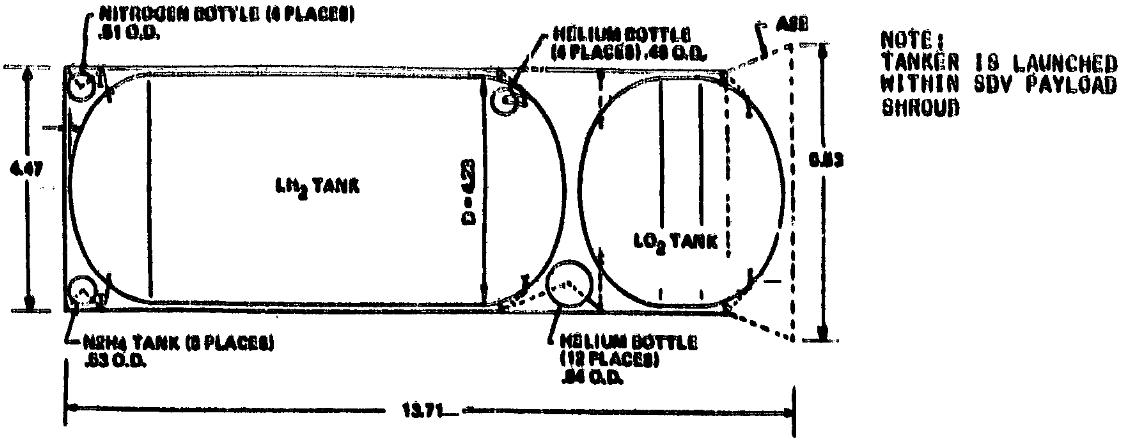


Figure 3.3.9-10 Attitude Control Propellant and Pressurant Transfer Schematic—Tanker to SOC to OTV

3.3.9.3 Propellant Tanker Configuration Description

The configuration of the propellant tanker is presented in figure 3.3.9-11 with overall geometry and physical characteristics noted. The tanker is sized for containment within the reusable payload system of the SDV solid rocket booster (SRB). The structural arrangement, details of structural design (main tankage, body shell), and thermal control design (main tankage MLI) are similar to those of the large GB OTV. Avionics consist primarily of propellant loading/transfer instrumentation and associated data management electronics. A battery provides for onboard electrical power. The main propulsion transfer system utilizes helium gas pressurant (stored in the intertank region) and propellant acquisition screens (lining the total inner surface area of the tanks) to effect transfer of the LO_2 and LH_2. Transfer helium for pneumatics is stored in the LH_2 tank. Transfer hydrazine and nitrogen pressurant are stored forward of the LH_2 tank. ASE for the tanker consists of an aft-located structural adapter. All other ASE-chargeable items are assumed incorporated in the design of the dedicated reusable payload system.

- SIZED FOR SDV/SRB WITH REUSABLE PAYLOAD SYSTEM
- MASS IN Kg • DIMENSIONS IN METERS



TANKER MASS CHARACTERISTICS (kg)

• DRY	4168
• RESIDUALS	1343
• LOSSES	612
• TRANSFER FLUIDS	53046
GROSS WT.	59169

Figure 3.3.9-11 Propellant Tanker Configuration

A summary mass statement for the propellant tanker is presented in table 3.3.9-3. The mass fraction of 0.8862 reflects the gross weight of 59 169 kg, including a transferred LO_2/LH_2 weight of 52 435 kg.

Table 3.3.9-3 Propellant Tanker Summary Mass Statement

STRUCTURE	2018
THERMAL CONTROL	146
AVIONICS	77
ELECTRICAL POWER	23
MAIN PROPULSION XFER SYSTEM	1184
ATTITUDE CONTROL XFER SYSTEM	177
WEIGHT GROWTH MARGIN	544
(TANKER MODULE DRY WEIGHT)	(4168)
RESIDUALS	1343
(TANKER END OF TRANSFER WT)	(5511)
LOSSES	612
TRANSFER FLUIDS*	(53046)
LO_2/LH_2 @ MR = 5.600:1	52435
GHe	21
N_2H_4	560
GN_2	30
(TANKER GROSS WT)	(59169)
ASE	1134
(LAUNCH WT)	(60303)
TANKER MASS FRACTION	0.8862

*TRANSFERRED TO SOC

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3.3.9.4 Fluids Inventories for Propellant Tanker, SOC Storage Tank, and SB OTV

Fluids inventories for the propellant tanker, a SOC tank module, and an SB OTV are presented in tables 3.3.9-4, -5, and -6, respectively. These inventories reflect the

Table 3.3.9-4 Propellant Tanker Flu l Inventory

	H_2	O_2	He	N_2H_4	N_2
MAIN PROPELLANT & EPS REACTANTS					
XFER TO SOC (52,435 @ MR = 5.646)	7891	44,544			
XFER LOSSES (1.0% H_2, 1.0% O_2)	77	444			
TRAPPED IN ACQ SCREENS (2.0% H_2, 2.0% O_2)	159	898			
TRAPPED IN SUMP/LINES	9	91			
BOILOFF LOSSES – ASCENT	77	14			
PRESSURANT IN MAIN LH_2 TANK			87		
PRESSURANT IN MAIN LO_2 TANK			26		
PRESSURANT RESIDUAL IN MAIN TANK BOTTLE			12		
PRESSURANT XFER TO SOC – PNEUMATICS			21		
PRESSURANT RESIDUAL IN PNEUMATICS BOTTLE			15		
ACS FLUIDS					
XFER TO SOC				560	30
XFER LOSSES				—	—
TRAPPED IN N_2H_4 TANKS				11	23
PRESSURANT RESIDUAL IN BOTTLE				—	12
	8213	45,991	161	571	65
	54,204 @ 5.600:1				

following guidelines: a propellant tanker sized for 100% recharging of a SOC storage tank, two SOC storage tanks sized for 100% recharging of three single-stage SB OTV's, 100% purging/venting of all onboard cryogenics and ACS propellant prior to a recharging operation, and no purging/venting of onboard helium and nitrogen pressurants from depleted gas bottles prior to a recharging operation.

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Table 3.3.9-5 SOC Storage Tank Fluids Inventory

	H_2	O_2	He	N_2H_4	N_2
MAIN PROPELLANT & EPS REACTANTS					
XFER TO OTV (49,887 @ MR = 5.716)	7429	42,458			
XFER LOSSES (1.4% H_2, 1.9% O_2)	104	807			
TRAPPED IN ACQ. SCREENS (2% H_2, 2% O_2)	150	848			
TRAPPED IN SUMP/LINES	11	181			
PRESSURANT IN MAIN TANKS	197	250			
PRESSURANT XFER TO OTV – PNEUMATICS			21		
PRESSURANT RESIDUAL IN BOTTLE			121		
ACS FLUIDS					
XFER TO OTV				549	14
XFER LOSSES				—	—
TRAPPED IN N_2H_4 TANKS				11	16
PRESSURANT RESIDUAL IN BOTTLE				—	64
	7891	44,544	142	560	94
	$\underbrace{7891 \quad 44,544}_{62,435 \text{ @ } 5.645:1}$				

Table 3.3.9-6 SB OTV Fluids Inventory

	H_2	O_2	He	N_2H_4	N_2
MAIN PROPELLANT & PRESSURANT					
USABLE (32,560 @ 6.000:1)	(4651)	(27,909)			
NOMINAL	4613)	27,676			
RESERVE	38	233			
RESIDUAL/LOSSES	(312)	(440)			
TRAPPED	9	91			
BIAS	30	—			
PRESSURANT – MAINTENANCE	99	141			
PRESSURANT – PNEUMATICS	—	—	18*		
CHILLDOWN/START/STOP	65	107			
BOILOFF/VENT	37	35			
BALLUTE INFLATION	61	—			
THRU ENGINES DURING A/B	11	66			
EPS REACTANTS					
NOMINAL	5	40			
RESERVE	3	20			
RESIDUAL	[illegible]	[illegible]			
ACS FLUIDS					
NOMINAL				326	
RESERVE				33	
TRAPPED				7	
PRESSURANT				—	11**
TOTAL	4973	28,420	18	366	11
	$\underbrace{4973 \quad 28,420}_{33,393 \text{ @ } 5.710:1}$				

* 14 USABLE, 4 RESIDUAL

** 9.2 USABLE, 1.8 RESIDUAL

3.3.10 Turnaround

A key factor associated with any reusable transportation system is the amount of time and number of personnel required to prepare the vehicle for another flight. The term used to describe this effort is called "turnaround." The time element is significant since if turnaround takes more time than that available between flights, then another vehicle must be available or additional crew must be added to turn the vehicle around in the required time. The turnaround operations are particularly significant for the SB OTV since the impact of equipment and personnel in orbit is generally more costly than if provided on the ground. For this reason and because turnaround at a space base had not been previously defined, the turnaround analysis was only performed on the SB OTV. (GB OTV turnaround was analyzed in the Phase A studies.)

Turnaround Flow - The turnaround flow for an SB OTV is shown in figure 3.3.10-1. Six

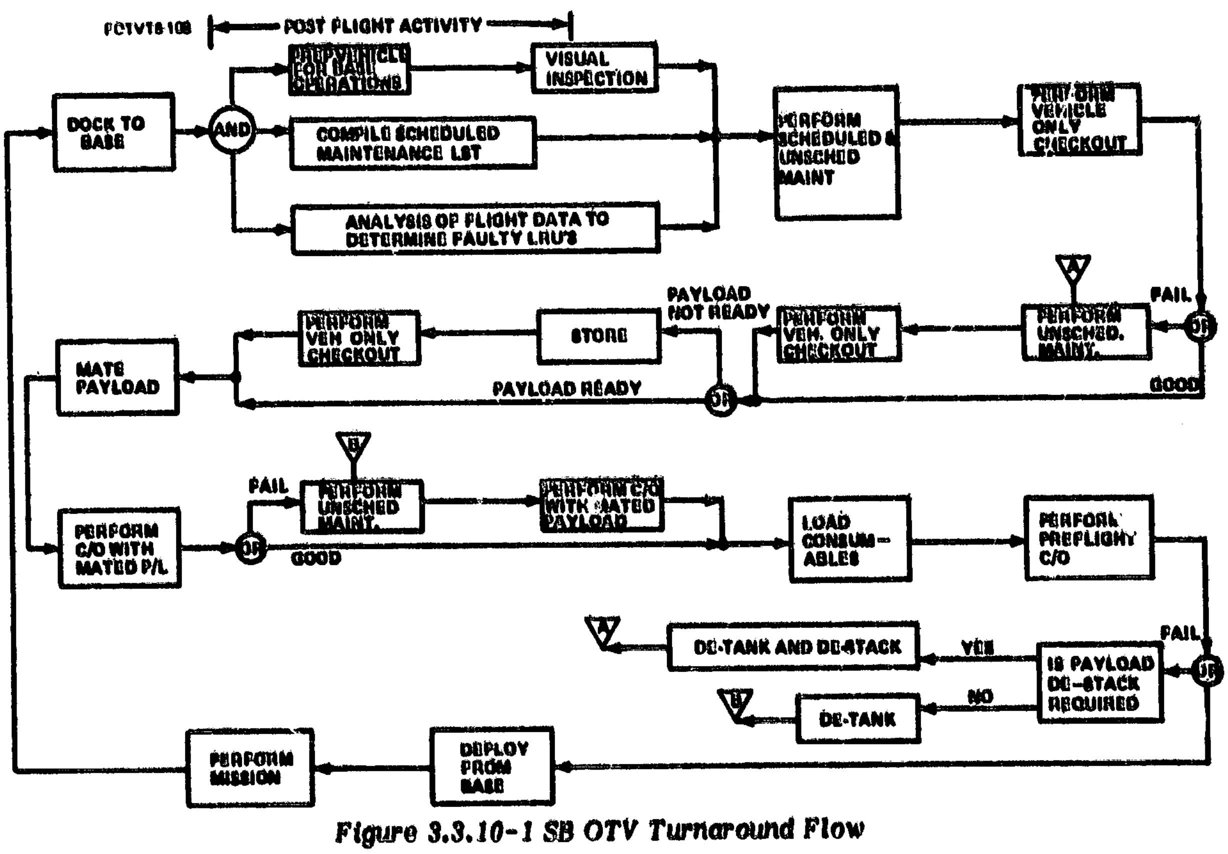


Figure 3.3.10-1 SB OTV Turnaround Flow

major types of operations or events occur. These include: (1) post-flight activity, (2) scheduled and unscheduled maintenance (referred to as refurbishment in this study), (3) checkout, (4) storage or standby, (5) payload handling and mating, and (6) loading of consumables (refueling). Each of these operations, except payload mating, has been discussed in preceding sections. The payload mating operation involves the physical attachment of the payload and OTV in terms of any required structural, electrical, fluid, or avionics interfaces. It should also be noted that three major checkout periods occur in the normal flow and relate to each major configuration change. One checkout occurs prior to the payload mating to allow better access should some failure be detected. Another checkout period occurs prior to propellant loading to avoid detanking of the vehicle in the event a failure occurs during the payload mating operation. The final checkout occurs just prior to deployment from the space base to ensure that all systems are good prior to the start of the mission.

Crew and Timeline.--The time and personnel requirements associated with the turnaround are shown in table 3.3.10-1. As noted, these data present an extreme case in that the

Table 3.3.10-1 Turnaround Time—Extreme Case

● ON-ORBIT ACTIVITY

EVENT	DURATION (HRS)	PERSONNEL/SHIFT
● PREP OTV FOR BASE OPS	4	2
● UNSCHED. MAINTENANCE	16 [1]	3
● SCHEDULED MAINTENANCE [2]	6	3
● CHECKOUT OTV ONLY	2	2
● PAYLOAD MATING OPS	4	3
● OTV/PAYLOAD INTERFACE C/O	1	2
● LOAD CONSUMABLES	13 [1]	2
● PRE-FLIGHT CHECKOUT	4	2
SUBTOTAL	50 HRS 8 DAYS (1 SHIFT)	
● REQUIRED GROUND TIME (MAINTENANCE PLANNING)	3 DAYS	
● TOTAL REQUIRED TIME	11 DAYS	
● TIME BETWEEN FLIGHTS (WORST CASE)	15 DAYS (AVG 15-17 DAYS)	
● TIME REMAINING (MARGIN) • EXTRA PLANNING TIME • REPAIR AFTER A C/O	4 DAYS	

[1] 4 SRU'S PLUS BALLUTE INSTALLATION

[2] BALLUTE ONLY, ENGINE IN UNSCHEDULED CLASS

unscheduled maintenance time reflects the removal and replacement of four SRU's (see sec. 3.3.8) rather than the mean of one SRU per flight. The time requirements also reflect one work shift per day, each containing 6.5 hr of actual worktime. The total on-orbit turnaround time is approximately 8 days, while another 3 days are allocated for analysis and planning at the ground mission control center. Even with the time between flights being as short as 15 days, a margin of 4 days exists. Crew requirements include two to perform the actual removal and replacement operations within the hangar and a third person to perform checkout operations in the control center of the SOC.

3.3.11 Launch and Recovery

As indicated in the study guidelines, a key task was to determine the most cost-effective launch system for the indicated mission model. Once selected, the launch system was then assessed to determine differences between the OTV basing modes in terms of recovery (Earth return of key elements) and detailed launch manifesting.

3.3.11.1 Launch System Screening and Selection

The significance of the launch system is indicated by the fact that (1) it contributes the majority of the total transportation cost, (2) its cost per pound of payload has a major influence on the technology selected for the OTV, and (3) it establishes constraints on the OTV and propellant tanker size in terms of mass, envelope, and operating modes. The selection of the launch system was initially based on preliminary OTV performance estimates and the assumption that all launches would be mass limited. This approach was used in order to reduce the number of launch system candidates as quickly as possible so a more detailed assessment of the selected system could be made.

Launch Requirements - The launch requirements for the assumed 1995-2005 mission model are shown in table 3.3.11-1. A total of 72 crew launches are required in addition to 6525t and 7345t, respectively, for the SB and GB OTV concepts.

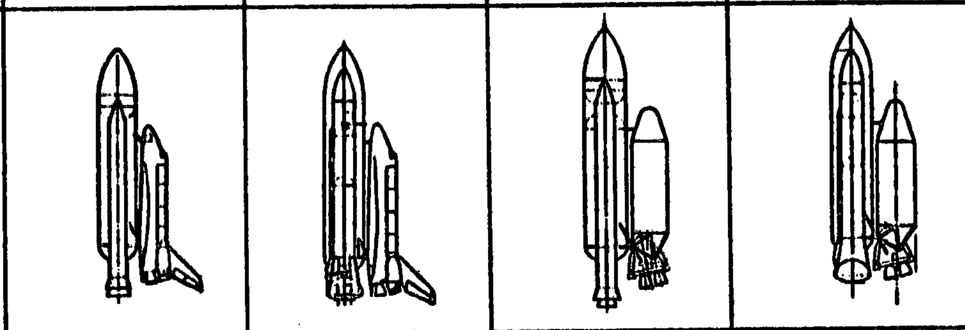
Table 3.3.11-1 Launch Requirements

Crew launches	(72)	
LEO base	44	
GEO base	28	
	SB OTV	GB OTV
Cargo (t)	(6525)	(7345)
GEO payloads	860	860
LEO payloads	825	825
OTV propellant	4000	4400
Propellant handling and transfer losses	400	0
Tanker or stage dry weight and ASE	440	1260

Crew launches reflect crew on-orbit staytimes of 90 days. Separate launches are assumed for each base. The GEO payloads are those discussed in section 3.2. LEO payloads include those associated with SOC as well as LEO-type spacecraft.

The SOC payloads consist of crew and base supplies amounting to 65 t/yr (including dry weight of logistics module) for an eight-man crew and the base located at 370 km. The supplies are delivered every 90 days. A total of 10 t/yr was allocated for LEO spacecraft and/or their supplies. The OTV propellant mass reflects preliminary rather than final OTV design features. As such, the SB OTV showed a 10% performance advantage versus 2.5% for the final design. Propellant losses associated with SB OTV refueling were assumed to be 10% of the flight propellant (prior to refueling analysis), thus making the total propellant to be launched the same for both concepts.

Launch System Candidates - Four types of launch systems were considered for FOTV application. The overall configurations and key characteristics of the systems are shown in figure 3.3.11-1. In all cases, characteristics were obtained from prior studies (refs. 1

KEY CHARACTERISTICS / LAUNCH VEHICLE	STANDARD SHUTTLE	SHUTTLE GROWTH	SHUTTLE DERIVATIVE (SOLID ROCKET BOOSTER)	SHUTTLE DERIVATIVE (LIQUID ROCKET BOOSTER)
CONFIGURATION				
MAJOR ELEMENTS	• ORBITER • EXTERNAL TANK • SOLID ROCKET BOOSTER (2)	• ORBITER • EXTERNAL TANK • LIQUID ROCKET BOOSTER (2)	• EXPENDABLE CARGO SHROUD • RECOVERABLE PROPULSION AND AVIONICS MODULE • SOLID ROCKET BOOSTER (2)	• EXPENDABLE CARGO SHROUD • REC VERABLE PROPULSION AND AVIONICS MODULE • LIQUID ROCKET BOOSTER (2)
APPLICATION	CREW AND CARGO	CREW AND CARGO	CARGO	CARGO
CARGO BAY (m) DIA x LEN NO.	4.57 x 18.3	4.57 x 18.3	7 x 24	7 x 24
PAYLOAD TO LEO (KG) @ 370 Km	29,500	47,000	67,700	84,000
DDT&E	-0-	2.08 B	1.2 B	1.0 B (4)
PROD COST (1) (2)	0	460 M	335 M	750 M
COST/FLT (3)	28.6 M (5)	27.0 M (5)	22.0 M (6)	18.7 M (6)

ALL COST IN 1980 DOLLARS (1) ORBITERS NOT INCLUDED (2) FOR FLIGHT RATE 23/YR (3) WHERE APPROP INCLUDES ET AND P/A MOD (4) Δ TO SHUTTLE GROWTH (5) FROM PHASE A OTV (6) FROM NAS8-32396

Figure 3.3.11-1 Candidate Launch Vehicle System Key Characteristics

and 7) with the only adjustments in this study being to reflect 1980 dollars and payload delivery capability to 370 km.

The key distinguishing features of the systems relative to the standard shuttle are as follows: the shuttle growth employs liquid rocket boosters (LRB) using LO_2/hydrocarbon propellant rather than SRB's. The shuttle derivative with SRB's has the orbiter replaced with a payload container (shroud) that is expendable but has a reusable propulsion/avionics module contining three Space Shuttle main engines (SSME) and the key avionics systems used during the flight. The shuttle derivative with LRB's also uses an expendable payload shroud and reusable propulsion/avionics module.

Four different combinations of these vehicles were considered to satisfy the integrated transportation requirement of launching both crews and cargo.

1. Basic STS only—Launches crew and all cargo.
2. Basic STS plus shuttle derivative with SRB—STS launches crew and a portion of the cargo while the SDV SRB launches the bulk of the cargo.
3. Shuttle growth only—Launches crew and all cargo.
4. Shuttle growth plus shuttle derivative with LRB—Shuttle growth launches crew and a portion of the cargo while the SDV LRB launches the remainder of the cargo.

Comparison and Selection - The life cycle cost comparison of the launch system combinations is presented in figure 3.3.11-2. The ordinate value in the crossover plot

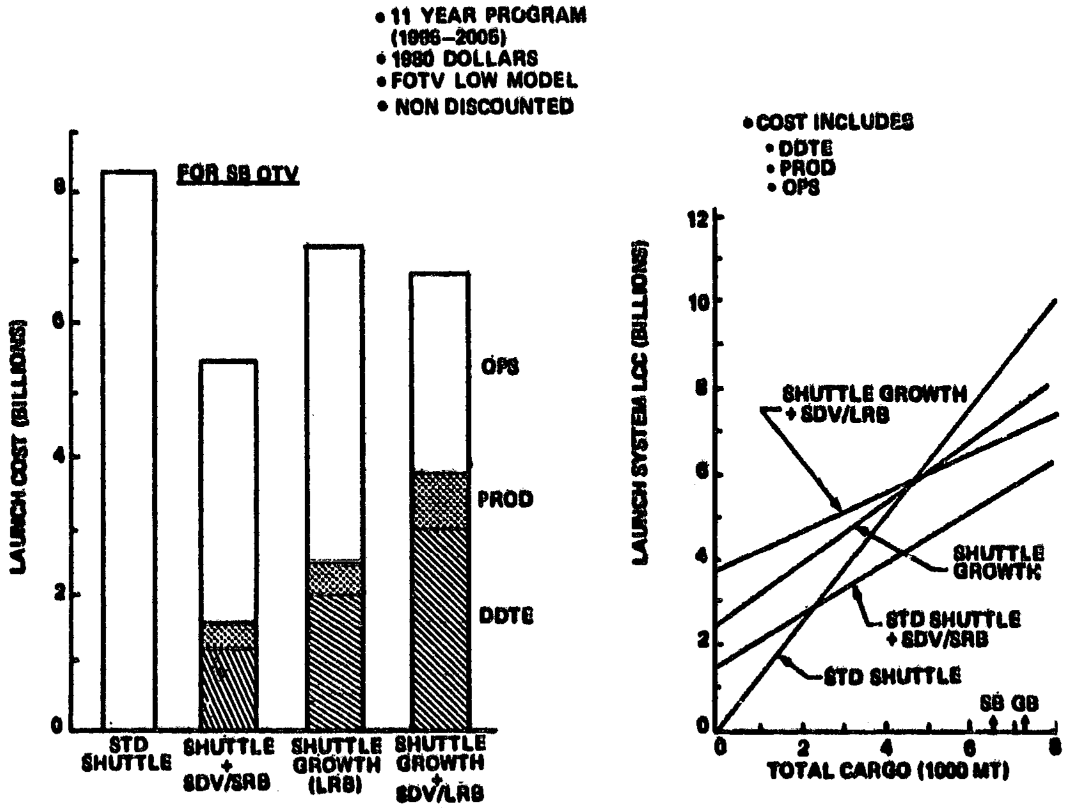


Figure 3.3.11-2 Launch System Comparison Initial Selection

reflects the DDT&E and production cost while the slope relates to the operations launch cost, including the 72 crew launches.

These data indicate that the standard STS plus SDV SRB combination provides a considerable cost margin at the indicated cargo requirement. In addition, this combination has the characteristics that provide the least cost over a cargo range from 3000t to 15 000t. A system with this degree of cost effectiveness tends to offset reservations about extreme accuracy on the mission model and the fact that the final SB OTV propellant launch requirements were about 1000t higher than the preliminary estimates.

3.3.11.2 Recovery of OTV-Related Elements

The least-cost launch fleet (STS plus SDV SRB) does present a key problem, however, in terms of providing capability to return OTV-related elements to Earth. The extent of this problem and alternatives are shown in figures 3.3.11-3 and 3.3.11-4 for the

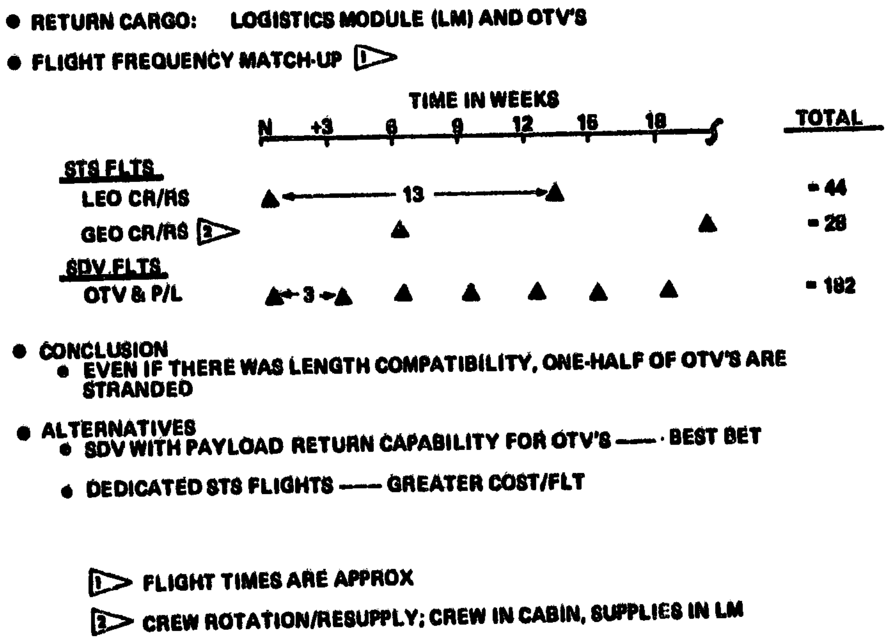
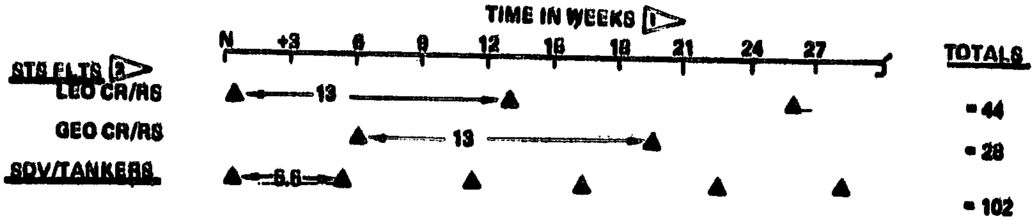


Figure 3.3.11-3 GB OTV Return Cargo Requirements

GB and SB OTV, respectively. The problem is basically that with an SDV using an expendable payload shroud, the only system available to return OTV elements is the STS orbiter. For the reference mission model, the total number of SB OTV propellant tankers (102) or the number of GB OTV's (182) both exceed the number of orbiters available (72). In addition, since the orbiter also contains a SOC logistics module, the length remaining in the cargo bay is not sufficient for either an OTV or tanker.

- RETURN CARGO: LOGISTICS MODULE (LM) AND PROP. TANKERS
- FLIGHT FREQUENCY MATCH-UP



PRELIMINARY CONCLUSION

- WITH SOME DIFFICULTY TANKERS MAY BE MATCHED TO RETURN ON STS FLTS — IF LENGTH COMPATIBLE.

- LENGTH MATCH-UP
  LOGISTICS MODULE + DOCKING MODULE = 8.2 M
  TANKER LENGTH = 13.7 M
  TOTAL LENGTH = 22 M > 18.3 M OF CARGO BAY
- FINAL CONCLUSION
  TANKERS CANNOT BE RETURNED ON STS FLIGHTS
- ALTERNATIVES
  SDV WITH PAYLOAD RETURN CAPABILITY FOR TANKERS — BEST BET
  EXPENDABLE TANKERS — GREATER COST
  LESS FLEXIBILITY

1 TIMES ARE APPROX

2 CREW ROTATION/RESUPPLY; CREW IN CABIN, SUPPLIES IN LM IN CARGO BAY

Figure 3.3.11-4 SB OTV Return Cargo Requirements

Several options are available to each basing mode to overcome this problem. The one selected, since it was common to both modes, was the use of a reusable payload system. This concept combines the previously expendable payload shroud and reusable propulsion/avionics module into one integral unit so the whole system is reusable. In this manner, either OTV's or tankers can be returned. The configuration and system characteristics of the recoverable and expendable shrouds are shown in figure 3.3.11-5. The key disadvantages of the recoverable system include a decrease of 10t in payload and additional DDT&E cost. It should also be mentioned that reentry and recovery of such a system presents some challenging technical problems and, therefore, must be viewed as having relatively high risk.

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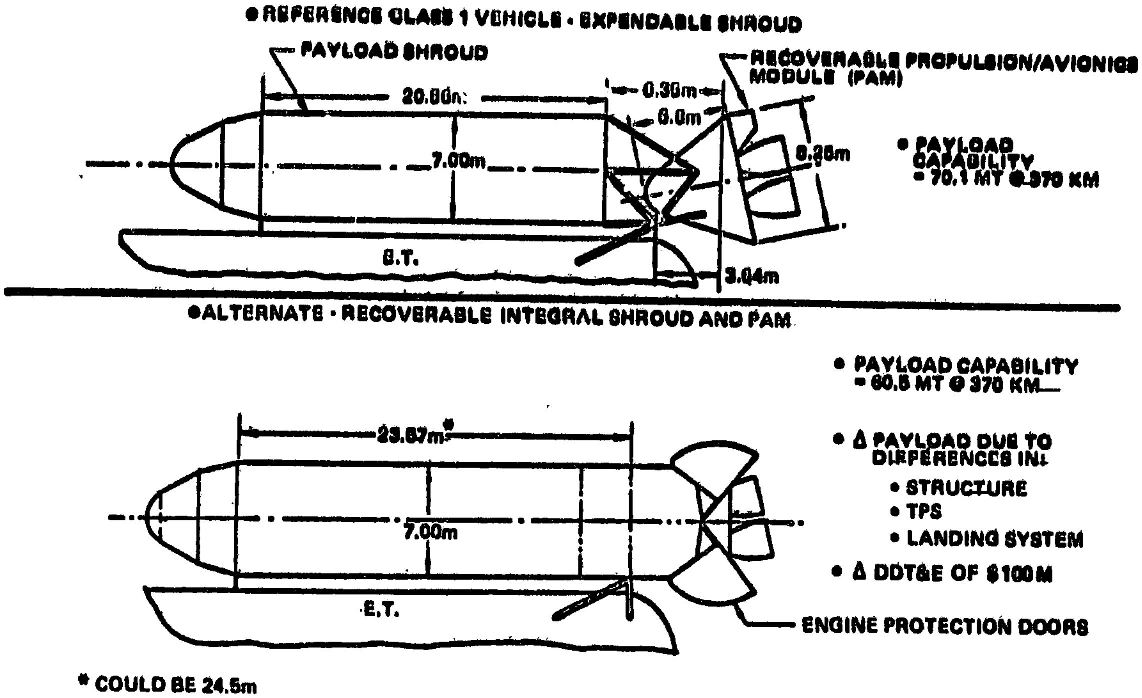

Figure 3.3.11-5 SDV Shroud Options

3.3.11.3 Launch Manifesting Results

The final number of SDV launches required for the payloads and OTV-related elements was influenced by several factors. These included (1) the utilization of the reusable payload system, which provided less payload, and (2) a more detailed launch manifesting analysis that considered payload lengths and allowable mixes rather than payload mass only, thereby potentially resulting in some volume-limited launches.

A summary of the guidelines and assumptions used in conducting the manifesting analysis is presented in table 3.3.11-2. The goal in the manifesting analysis was to try to achieve mass-limited launches. The number of STS launches for crew rotation/resupply was the same for all the OTV options; however, the payloads included in these launches could be different. This analysis also reflects the propellant requirements associated with the final OTV design features and propellant refueling losses.

Manifesting for a single-size GB OTV and the mission model payloads is shown in table 3.3.11-3. This OTV is always launched with its payload. A total of 196 SDV launches were required, with the majority being for the combined launch of an OTV and its payload. It should be noted that the majority of these launches have a mass load factor of approximately 65%. Launching payloads with the OTV also results in 21 STS launches being considerably underused.

Table 3.3.11-2 Launch and Return Manifesting Guidelines and Assumptions

- NO MORE THAN TWO PAYLOADS PER LAUNCH (IF POSSIBLE)
- DO NOT MIX DOD PAYLOADS WITH NASA OR CIVIL (IF POSSIBLE)
- CREWS OCCUPY STS ORBITER CABIN
- BASE LOGISTICS MODULES DELIVERED WITH STS (INCLUDES DOCKING MODULE)
- OTV IMPLEMENTATION OPTIONS
  - GROUND BASE - ONE SIZE (LAUNCH WITH PAYLOAD IF POSSIBLE)
  - GROUND BASE - BIG AND LITTLE (TWO LITTLE OTV'S AT ONCE IF POSSIBLE)
  - SPACE BASE - ONE SIZE
    - BIG AND LITTLE TANKERS
- KEY PAYLOAD CHARACTERISTICS

TYPE	MAX. MASS (MT)	LEN (M)
LEO LM + DM + PM	20	11.3
LEO LM + DM	8	8.2
GEO LM + DM	4	6.4
SB OTV	37.7	14.2
TANKER - BIG	60.4	14
TANKER - LITTLE	47.2	10.4
GB OTV BIG	38.9	12.6
GB OTV LITTLE	26.8	10.5
GEO PAYLOADS	SEE MISSION MODEL SECTION	

CODE

LM = LOGISTICS MODULE
DM = DOCKING MODULE
PM = PROPELLANT MODULE (SOC ORBIT KEEPING)

Table 3.3.11-3 One-Size GB OTV Launch Vehicle Manifesting

STS LAUNCHES (72)

TYPE	REMAINING CAPAB [1] MASS	LEN
LEO CR/RS FLTS (44)		
LM + DM + PM (36)	6	7
LM + DM + PM (2) + P/L 2	0	0
LM + DM + PM (6) + P/L 3 [2]	0	0
GEO CR/RS FLTS (28)		
LM + DM (21)	24	11.6
LM + DM + P/L 2 (1)	0	0
LM + DM + P/L 3 (6)	0	0

REUSABLE SDV LAUNCHES (195)

TYPE	REMAINING CAPAB [1] MASS	LEN
OTV + PAYLOADS (182)		
AVERAGE FLT	21	6
MAX. FLT	7.2	0
OTV ONLY (11)		
AS SECOND STAGE	18	11
GEO PAYLOADS ONLY (3)		
[2] 4. SBR	49	6.1
[2] 7. STO	49	6.1

[1] MASS IN MT, LEN. IN METERS
[2] MISSION NUMBER

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A more effective manifesting approach for the ground-based OTV mode was to utilize two sizes of OTV's and, on occasion, have the payloads launched separately. The results of this approach are shown in table 3.3.11-4 and indicate that the number of SDV launches was reduced to 138. The smaller OTV was used for delivery of GEO payloads with masses up to 8300 kg or for round trips of approximately 3900 kg. Payloads larger than these values would use the larger size OTV. With the assumed mission model, the small OTV was used in 116 out of 182 OTV missions. The launch mode employed when using the small OTV's consisted of having two of the OTV's launched together, preceded by their payloads, with mating of OTV's and payloads occurring at SOC. Launch of the large size OTV also included its payload with the noted exceptions.

Table 3.3.11-4 Two-Size GB OTV Launch Vehicle Manifesting

STS LAUNCHES -(72)

TYPE		REMAINING CAPAB. [2] MASS	REMAINING CAPAB. [2] LEN
• LEO CR/RS FLTS (44)			
• LOGISTICS MODULE (44) PLUS ONE OF FOLLOWING ON EACH FLIGHT:			
[1] 2. ADV COMM PLAT	(1)	7	2.4
3. PERS COMM SAT	(6)	9	2.4
10. DOD CL IA	(21)	12	2.1
11. DOD CL 2	(4)	9	3.6
12. & 15. NASA P/L AND SAT MAINT PROV	(12)	7	1.8
• GEO CR/RS FLTS (28)			
• LOGISTICS MODULE (28) PLUS ONE OF FOLLOWING ON EACH FLIGHT:			
[1] 1. COMM PLAT	(7)	8	2.1
2. ADV COMM PLAT	(2)	9	2.4
3. PERS COMM SAT	(6)	7	2.4
[3] 5. DOD CL 3 / 22. UNMANNED SERV PROV	(6)	4	1.0
10a. DOD CL IB	(7)	16	0

REUSABLE SDV LAUNCHES (138)

TYPE		REMAINING CAPAB. [2] MASS	REMAINING CAPAB. [2] LEN
• BIG OTV & PAYLOAD FLTS (50)			
• BIG OTV & ONE OF THE FOLLOWING ON EACH FLT			
[1] 13. GEO BASE MODULES	(2)	3	-1.2
14. GEO BASE EQUIP	(3)	17	6.0
22&18. MAINT SORTIE	(11)	9	1.0
22&18. GEO BASE SUPPORT (CR/RS)	(26)	9	1.0
21. SCIENCE SORTIE	(2)	10	6.0
23. PLANETARY	(3)	13	4.2
• BIG OTV ONLY FLTS (27)			
• DUE TO P/L LAUNCHED SEPARATELY (MISSION NUMBER 2, 3, 4, 5, 7, 22 + SOC PM	(16)	3	2.0
• AS SECOND STAGE MISSIONS 2, 3, 13, 21, 22	(5)	16	6.0
• AS SECOND STG + SOC PM + 22	(6)	3	2.0
• GEO PAYLOAD ONLY FLTS (3)			
[1] 4. SBR	(2)	49	6.1
7&6 STO + DSR	(1)	42	0
• LITTLE OTV ONLY FLTS (66) TWO PER LAUNCH		4	1.5

[1] MISSION MODEL NUMBER, NAME & NO. OF FLTS
[2] MASS IN MT, LENGTH IN METERS
[3] COMBINE ON EACH FLT.

Manifesting of the SB OTV elements and mission model payloads is presented in table 3.3.11-5. This concept required only 121 SDV launches. The reduced number of launches occurs primarily because propellant makes up the bulk of the cargo (80%) and is relatively easy to achieve mass-limited conditions, particularly when using two sizes of tanke.s and offloading as required when GEO payloads are included in the launches. It should also be noted that this analysis reflects the final propellant launch requirement, which was 5400t rather than the 4400t used in selection of the launch system. The

Table 3.3.11-5 SB OTV Launch Vehicle Manifesting

STS LAUNCHES (72)

TYPE		REMAINING CAPAB. [2] MASS	REMAINING CAPAB. [2] LEN
• LEO CB/BS FLTS (44)			
• LOGISTICS MODULE (44) PLUS ONE OF FOLLOWING ON EACH FLIGHT:			
[1] 2. ADV COMM PLAT	(1)	7	2.4
3. PERS COMM SAT	(6)	8	2.4
10. DOD CL IA	(21)	12	2.1
11. DOD CL 2	(4)	9	3.6
12&16. NASA P/L AND SAT MAINT PROV	(12)	7	1.8
• GEO CB/BS FLTS (28)			
• LOGISTICS MODULE (28) PLUS ONE OF FOLLOWING ON EACH FLIGHT:			
[1] 1. COMM PLAT	(7)	8	2.1
2. ADV COMM PLAT	(2)	9	2.4
3. PERS COMM SAT	(6)	7	2.4
6. DOD CL 3	(6)	8	4.0
10a. DOD CL IB	(7)	16	0

[1] MISSION MODEL NUMBER, NAME; (NO. OF FLTS)

[2] MASS IN MT, LENGTH IN METERS

REUSABLE SDV LAUNCHES (121)

TYPE		REMAINING CAPAB [2] MASS / LEN
• TANKER ONLY (14) W_p = 61.4 MT		MASS LIMITED
• TANKER & GEO P/L (91)		W_p AVAIL PER FLT
• OTV PROP PLUS ONE OF THE FOLLOWING EACH FLT		
• SOC ORBIT MAINT PROP	(18)	31 (ST) [3]
[1] 8. DSR	(1)	44 (LT) [4]
10. DOD CLASS 1A	(1)	48 (LT)
13. GEO BASE MODULES	(2)	41 (ST)
22. UNMANNED SERVICING PROVISIONS	(63)	47 (LT)
23. PLANETARY	(6)	47 (LT)
• GEO PAYLOAD ONLY (3)		VOLUME LIMITED
[1] 4 & 14 SBR & GEO BASE EQUIPMENT	(2)	
7 & 14 STO & GEO BASE EQUIPMENT	(1)	
• OTV & SHORT TANKER W_p IN TANKER = 41 (ST)	(13)	

[3] ST = SHORT TANKER, MAX W_p = 41 MT

[4] LT = LONG TANKER, MAX W_p = 61.4 MT

number of launches identified for OTV's satisfies the wearout, relaunch due to unscheduled ground maintenance, and standby units to serve as second stages or backup.

In summary, a more detailed manifesting analysis indicated that the ground-based OTV concept could not achieve the degree of mass-limited launches possible with SB OTV systems. The ground-based mode could achieve a fairly high degree of manifesting, however, by using two sizes of OTV's. This approach saved 58 SDV launches, compared with a single-size GB OTV, and consequently was the mode used in the cost comparison with SB OTV's.

3.3.12 Impact on Space Base

A summary of the OTV basing mode impact on the LEO space base (assumed to be SOC) is shown in table 3.3.12-1. The data for the GB OTV mode are indicative of the mode using two sizes of OTV's. The most apparent impact of the SB OTV is its need for propellant storage tanks and hangar, as illustrated in figure 3.3.12-1. Again, it should be emphasized that the hangar serves a dual role in terms of providing debris protection while at LEO and being a maintenance facility.

Table 3.3.12-1 OTV Basing Mode Impact on SOC

IMPACT	GROUND BASED OTV (2 SIZES)	SPACE BASED OTV
• HANGAR • DEBRIS PROTECTION • MAINTENANCE CAPAB.	• NONE, UNLESS OTV STAYS AT BASE MORE THAN 3 DAYS (DEBRIS PROTECTION)	• 4 (ONE FOR EACH OTV) • ONLY ONE WITH MAINTENANCE CAPABILITY
• MAINTENANCE CAPAB.	• NONE	• SCHEDULED & UNSCHED.
• CHECKOUT CAPAB.	• OTV/PAYLOAD	• OTV • OTV/PAYLOAD
• REFUELING	• NONE	• (2) 62 MT TANK SETS AND ALL ASSOCIATED PLUMBING & CONTROL SYSTEMS
• DOCKING PORTS	• OTV (3) • PAYLOADS (3)	• OTV (4) • TANKER (1) • PAYLOADS (3)
• HANDLING (MATING) PROVISIONS FOR:	• OTV/OTV (11) • OTV/PAYLOAD (135) • OTV/RECOVERY VEHICLE (193)	• OTV/OTV (11) • OTV/PAYLOAD (182) • OTV/RECOV. VEH (6)
• PERSONNEL	• 1-2, 10% DUTY CYCLE	• 3, 40% DUTY CYCLE

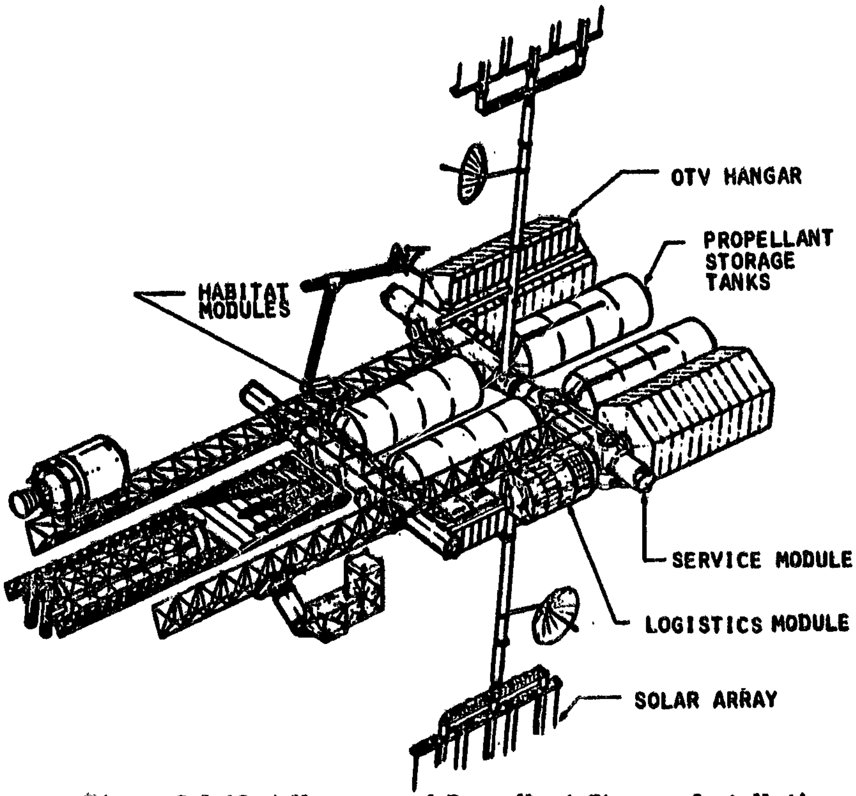

Figure 3.3.12-1 Hangar and Propellant Storage Installation

Handling and mating operations associated with the OTV and payloads are nearly as high with the ground-based OTV mode primarily because of the 116 small OTV's being launched separately from their payloads. The GB OTV approach also requires considerably more mating operations between OTV and recovery vehicle because all OTV's return to Earth. In the case of the SB OTV, the only OTV and recovery vehicle operations are those which return an OTV for unscheduled ground maintenance. The SB OTV refueling tanker remains within the SDV payload shroud and transfers propellant via lines; therefore, no handling is necessary.

The crew size and duty cycle are greater with an SB OTV; however, the magnitude appears acceptable when considering that the nominal crew is eight and OTV support is one of the three primary roles specified for SOC. The impact of the crew size will be expressed as a SOC users charge.

3.3.13 Cost Analyses

This section presents an overview of the scope, methodology, and guidelines used in the cost analyses, cost breakdown for the key elements, and total transportation life cycle cost summaries.

3.3.13.1 Overview

Scope - As specified by the study guidelines, the figure of merit for comparing space- and ground-based OTV's was to be life cycle cost of the total transportation system in performing the indicated mission model. This has been defined to include the DDT&E, production, and operations cost associated with the OTV's, all directly related orbital support systems, and launch systems. This task was accomplished by using a combination of both study-developed costs and utilization of costs from other studies when appropriate. The study-developed costs included those associated with all OTV's, propellant tankers, and space propellant storage tanks. Costs associated with launch systems were for the most part taken directly from prior studies but updated to 1980 dollars. Payload costs were not included for two reasons: (1) this was a transportation analysis and (2) as long as payloads are separate from transportation elements, their cost will be a constant factor.

Methodology - The primary tool used for estimating DDT&E and production costs is the Boeing-developed Parametric Cost Model (PCM). PCM develops costs from physical hardware descriptions and program schedules and allows the integration of any known costs (or outside-generated costs such as subcontractor or vendor estimates) into the total estimate. In this way, Boeing can assemble a program cost from the best available source data.

An overview of the PCM estimating method is illustrated in figure 3.3.13-1. As depicted in the illustration, the scope of the program relative to quantities, program time period, work breakdown structure, and associated ground rules and assumptions is established by the customer. Contractor program planners amplify the customer-furnished directives into a design, development, fabrication, test, and spares philosophy required to support the implementation of the program. These data, along with financial information relative to labor, support, and overhead rates, are assembled on a PCM "global" level input sheet, which defines the program-level constraints that the cost model will work within. To develop individual component hardware estimates, engineering and manufacturing functionals describe the components that make up the subsystem. This description requires a weight, hardware-type, redundancy, hardening, and circuitry-type

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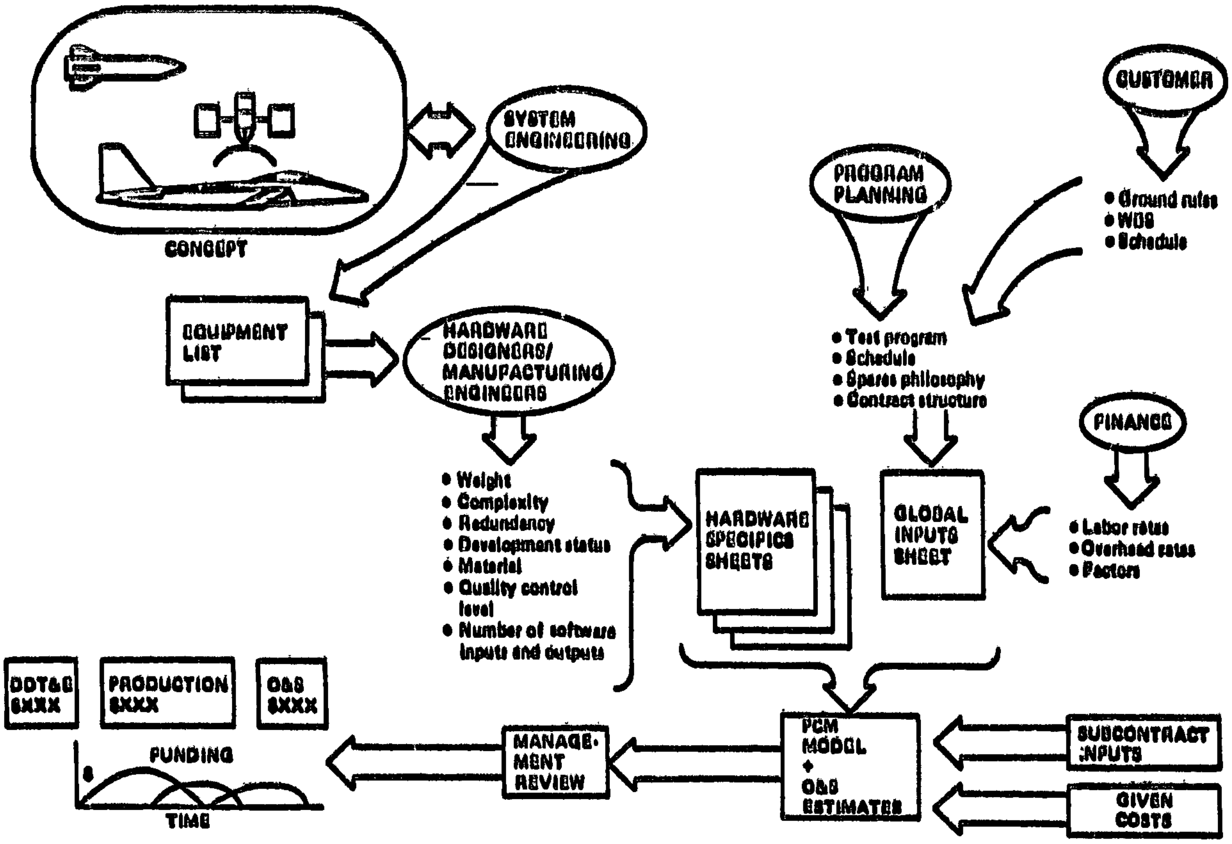

Figure 3.3.13-1 PCM Estimating Model

definition and an assessment of complexity, development status, manufacturing process, and required quality control level. These hardware data, in conjunction with programmatic-level global inputs, are processed in the PCM cost model to generate cost estimates.

The PCM is a collection of relationships and factors that have been developed from Boeing's historical data base, consisting of man-hour and dollar data contained in the Executive Information System (EIS). EIS is a company-wide data bank providing raw information from which (in the case of PCM) functional man-hour estimating relationships (MER) have been derived. These MER's relate program inputs to the model's internal working logic. Each major functional area (project engineering, developmental shop, etc.) making up Boeing's organization is represented and interrelated in the model. These functional areas are ultimately expressed in terms of man-hours required to fulfill the objectives of the program, which are converted to dollars using dollar-per-hour rates and estimating factors that are appropriate for the time period of the estimate.

Development of operations costs for OTV's and propellant tankers was achieved by adjusting data inputs to the analysis performance for the Phase A OTV study. Primarily this involved the items sensitive to annual flight rate and the total number of flights in the mission model.

Guidelines and Assumptions - The guidelines and assumptions used to determine DDT&E and production costs are as follows:

1. ROM ASE costs derived from Phase A study costs
2. 2.75 equivalent units of test hardware (flight vehicles and ASE)
3. Two sets of GSE included in DDT&E
4. One test flight included in DDT&E
5. Flight test units refurbished for operational fleet
6. 10% initial spares
7. 90% production learning curve on stages
8. 95% production learning curve on engines
9. Engine costs from MSFC
10. All costs expressed in 1980 dollars

Hardware quantities used in developing production cost are presented in table 3.3.13-1. The stage quantities reflect the fleet size which is necessary on a continuous

Table 3.3.13-1 Production Quantities

	SBOTV (10)	GB-1 SIZE (10)	GB-2 SIZES (12)
● STAGES			
● FLEET SIZE	(4)	(4)	(6)
● MANNED OTV	1	1	1
● BACKUP MOTV	1	1	1
● CARGO OTV	1	1	-
● BACKUP CARGO OTV OR 2ND STAGE	1	1	1
● CARGO LITTLE (2)			2
● CARGO LITTLE BACKUP			1
● WEAR OUT	(5)	(5)	(5)
● 209 STG FLTS			
● 45 FLT DESIGN LIFE			
● ATTRITION	(1)	(1)	(1)
● ENGINES	(30)	(30)	(26)
● ASE	(3)	(3)	(3)
● GSE	(3)	(3)	(3)
● TANKERS	(4)	-	-
● SOC STORAGE TANKS	(4)	-	-

basis, wearout based on 45 flight design life, and attrition. Engine quantities are based on 10 hr or 20 flights design life.

3.3.13.2 System Cost Summary

Space-Based OTV - The DDT&E and theoretical first unit (TFU) costs for the SB OTV are shown in table 3.3.13-2. The total DDT&E cost is $695M. The flight hardware design

Table 3.3.13-2 SB OTV DDT&E and TFU Cost Estimate

DDT&E		(695)
• FLIGHT HARDWARE DESIGN		369.5
STRUCTURE	15.3	
THERMAL CONTROL	10.2	
AVIONICS	32.2	
POWER	10.6	
PROPULSION	275.0	
ATTITUDE CONTROL	1.2	
BALLUTE	25.0	
• SYSTEMS ENGRG. & INTEGRATION		15.1
• INITIAL TOOLING		10.3
• SYSTEMS TEST		202.9
TEST HARDWARE	128.5	
TEST OPERATIONS	74.4	
• ASE		11.4
• GSE		18.9
• SOFTWARE		19.3
• LAISON/DATA		8.2
• PROGRAM MANAGEMENT		39.3

TFU		(29.6)
STRUCTURE	4.5	
THERM. CON	0.6	
AVIONICS	11.1	
POWER	3.8	
PROPULSION	5.5	
ATT. CON.	0.9	
ASSY. & C/O	3.2	

MILLIONS OF 1980 DOLLARS

portion is estimated at $370M, including $270M for the advanced engine. Avionics costs reflect use of a radiation-hardened system. Test operations include ground test, flight test operations, and users charge for a launch vehicle. The TFU for an assembled stage is estimated to be nearly $31M. The main engine TFU is $1.86M.

The total production cost for the SB OTV is estimated at $365M; a breakdown is presented in table 3.3.13-3. The largest flight hardware contributions are avionics and main propulsion. The ballute is also a major contributor because of being expended each flight.

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Table 3.3.13-3 SB OTV Production Costs

	COST IN MILLIONS	
FLIGHT HARDWARE		259.6
STRUCTURE	32.7	
THERMAL CONTROL	4.6	
AVIONICS	76.0	
EPS	26.8	
MAIN PROPULSION	61.3	
ACS	6.7	
BALLUTE	48.0	
AIRBORNE SUPPORT EQUIPMENT	4.7	
ASSEMBLY AND CHECKOUT		31.6
TOOLING		20.7
SPARES		21.0
SUSTAINING ENGINEERING		9.7
PROGRAM MANAGEMENT		22.0
TOTAL		364.6

The operations cost per flight is shown in table 3.3.13-4. The cost of $3.5M per flight is based on 16-18 flights per year. Additional operations cost, such as the launching of propellant, SOC users charge related to the turnaround of the OTV, and the tanker reuse cost are charged separately against each of these items.

Table 3.3.13-4 Operations Cost Per Flight

GROUND OPS	0.45
FLIGHT OPS	2.55
SUSTAINING ENGR.	0.28
OPERATIONAL SPARES	0.43
PROP. & GASES	0.02
	$3.73M

[1] BASED ON 18-20 FLIGHTS/YR

Ground-Based OTV - Costs presented for the GB OTV reflect a program utilizing two sizes of vehicles: a large OTV (approximately the same size as the SB OTV) and a small OTV with approximately 70% as much propellant. As indicated in section 3.3.11, this approach considerably reduced the number of launches as compared with a single-size GB OTV.

The DDT&E and TFU for the combined program are presented in table 3.3.13-5. The DDT&E cost of $815M reflects a high degree of commonality between the vehicles in terms of subsystems, ASE, and GSE. Key differences include tank size and propellant system as influenced by two versus one main engine. If costed alone, the large GB OTV

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Table 3.3.13-5 GB OTV DDT&E and TFU—Two Sizes

DDT&E		(773.2)	TFU	(30.9)
• FLIGHT HARDWARE DESIGN		379.2	STRUCTURE	6.4
STRUCTURE	25.0		THERM. CONTROL	0.8
THERMAL CONTROL	14.4		AVIONICS	9.2
AVIONICS	27.4		POWER	3.7
POWER	9.9		PROPULSION	5.4
PROPULSION (1)	276.3		ATT. CON.	0.9
ATTITUDE CONTROL	1.2		ASSY. & C/O	4.5
BALLUTE	25.0			
• SYSTEMS ENGRG. & INTEGRATION		21.6		
• INITIAL TOOLING		18.5	(MILLIONS OF 1980 DOLLARS)	
• SYSTEMS TEST		261.2		
TEST HARDWARE	169.9			
TEST OPERATIONS	91.3			
• ASE		22.7		
• GSE		19.1		
• SOFTWARE		5.8		
• LIAISON/DATA		11.0		
• PROGRAM MANAGEMENT		34.1		

(1) INCLUDES ENGINE AT $ 271M

would have a DDT&E of approximately $700M. The small OTV, therefore, requires approximately another $115M. TFU's of $30.4M and $24.4M are estimated for the large and small GB OTV's, respectively.

The total production cost for the combined GB OTV program is $392M; breakdown of the cost is presented in table 3.3.13-6. Although two types of vehicles are involved, the high degree of commonality makes the total number of components produced not too different from that required for a single-size vehicle.

Table 3.3.13-6 GB OTV Production Cost—Two Sizes

FLIGHT HARDWARE		$ 266.5M
STRUCTURE	40.3	
THERMAL CONTROL	5.6	
AVIONICS	71.2	
EPS	28.4	
MAIN PROPULSION	56.2	
ACS	6.3	
BALLUTE	49.0	
AIRBORNE SUPPORT EQUIP	9.5	
ASSEMBLY AND CHECKOUT		32.6
TOOLING		37.2
SPARES		21.8
SUSTAINING ENGR.		10.0
PROG. MGT.		23.9
	TOTAL	$ 392M

The operations cost per flight for the GB OTV is the same as for the SB OTV with the exception that all maintenance costs are reflected, as opposed to a major portion being chargeable to the SOC. Accordingly, the cost per flight excluding launch is \$5.5M.

Propellant Tanker for SB OTV - Two sizes of propellant tankers were used for the SB OTV concept so that the number of SDV launches would be minimized. One tanker is sized for a usable propellant load of 52t of LO_2/LH_2 and the other for 42t.

DDT&E and TFU cost estimates were developed for the large tanker only and are shown in table 3.3.13-7. The DDT&E cost was \$315M and the TFU \$10.9M. An additional DDT&E cost of \$125M was assumed for the smaller tanker based on cost scaling relationships between the large and small GB OTV's.

Table 3.3.13-7 Large Propellant Tanker—DDT&E and TFU

DDT&E		(314.4)
• FLIGHT HARDWARE DESIGN		41.5
STRUCTURE	19.7	
THERMAL CONTROL	11.0	
AVIONICS	5.7	
POWER	0.2	
PROPULSION	4.1	
ATTITUDE CONTROL	0.8	
BALLUTE		
• SYSTEMS ENGRG. & INTEGRATION		11.9
• INITIAL TOOLING		13.1
• SYSTEMS TEST		161.9
TEST HARDWARE	78.4	
TEST OPERATIONS	83.5	
• ASE		22.7
• GSE		13.9
• SOFTWARE		16.0
• LIAISON/DATA		6.4
• PROGRAM MANAGEMENT		29.0

TFU	(10.9)
STRUCTURE	4.1
THERM. CONTROL	0.3
AVIONICS	0.9
POWER	0.2
PROPULSION	3.0
ATT. CON.	0.5
ASSY. & C/O	1.9

(MILLIONS OF 1980 DOLLARS)

The operations cost per flight for the tankers was estimated as \$1.5M. The ground operations associated with the tanker are less than for a GB OTV because not as many subsystems are involved; software development is considerably less since the system is not flown as an independent vehicle. The production cost for both sizes of tankers was \$150M.

Propellant Storage Tanks (at SOC) - These two tanks have essentially the same propellant capacity as the large propellant tankers. The DDT&E cost of the storage tank was estimated to require an additional \$125M. This cost was to cover differences in thermal control provisions, space debris protection, and propellant transfer equipment including plumbing, compressors, etc. Production cost per unit was assumed to be 20% greater than the tanker, resulting in the total cost of four units being \$80M. An operations cost for the storage tank was not defined.

3.3.13.3 Total Transportation Cost Comparison

The final comparison of GB and SB OTV's for total transportation cost includes two key factors. First, only the GB OTV concept employing two sizes is included since it required 58 fewer SDV launches than the single-size GB OTV. Second, it was decided that since the RPS associated with the shuttle-derivative vehicle had a relatively high risk, the comparison should be done with and without the use of this system.

The comparison of basing modes when using an RPS is shown in table 3.3.13-8 by program phase and in table 3.3.13-9 by hardware element. In t. ' case, the GB OTV mode

Table 3.3.13-8 - Life Cycle Cost Summary With Reusable Payload System

• COST IN MILLIONS • 1980 DOLLARS	GROUND BASED OTV-2 SIZES	SPACE BASED OTV-ONE SIZE
DDTE	(1915)	(2360)
OTV	815	695
TANKER	—	440
SOC SYSTEMS	TBD	125 (ROM) + TBD
SDV/RPS	1100	1100
PRODUCTION	(785)	(1045)
OTV	380	365
TANKER	—	150
SOC SYSTEMS [1]	TBD	80 (ROM) + TBD
SDV/RPS	450	450
OPERATIONS	(5865)	(5590)
OTV	770	640
TANKER	—	140 (ROM)
SDV/RPS	3035	2750
STS	2060	2060
SOC SYSTEMS [1]	TBD	TBD
SOC USER CHARGE	TBD	TBD
TOTAL COST TO DATE	8620	8995
RANGE OF REMAINING COST (TBD'S)	< 100	< 300
POTENTIAL TOTAL COST	8720	9295

[1] INCLUDES PROP. STORAGE TANKS
NOT INCLUDED: HANGAR, DOCKING SYSTEMS, REFUELING PLUMBING AND CONTROL

provides a total transportation cost savings of approximately \$600M or 7% compared with the SB OTV. Moreover, GB OTV provides approximately \$700M savings in front-end costs (DDT&E and production). The cost increment for the GB mode is greater than for the SB mode primarily because two vehicle sizes rather than one were involved. The operations cost directly associated with the OTV in the GB mode is larger because the majority of its cost is shown here as compared with the SB OTV which has a portion of its operations cost included as part of the tanker and SOC systems. Tanker costs for the SB OTV mode

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Table 3.3.13-9 *Life Cycle Cost Summary With Reusable Payload System*

- COST IN MILLIONS
- 1980 DOLLARS

	GROUND BASED OTV - 2 SIZES	SPACE BASED OTV - 1 SIZE
OTV	(1975)	(1700)
DDTE	815	665
PRODUCTION	390	395
OPERATIONS	770	640
TANKER	(0)	(730)
DDTE		440
PRODUCTION		150
OPERATIONS		140 (ROM)
SOC SYSTEMS	(TBD)	(205)
DDTE		125 (ROM) + TBD
PRODUCTION		80 (ROM) + TBD
OPERATIONS		TBD
SDV/RPS	(4580)	(4300)
DDTE	1100	1100
PRODUCTION	450	450
OPERATIONS	3035	2750
STS	(2060)	(2060)
DDTE	---	
PRODUCTION	---	
OPERATIONS	2060	2060
TOTAL COST TO DATE	8620	8995
TBD ESTIMATE	100	300
POTENTIAL TOTAL	8720	9295

reflect two sizes and include a total of four units. The SOC system cost identified is that for the propellant storage tanks. Both the GB and SB OTV's have a hangar and users charge cost that is to be determined. A rough order of magnitude (ROM) cost for the GB OTV mode is $100M, while the SB OTV concept is $300M because of more hangars and additional personnel. Operations costs associated with the SDV RPS are due to the GB OTV requiring 138 launches versus 121 for the SB OTV.

The second cost comparison considers the SDV without RPS, which means the use of an expendable payload shroud and no Earth return capability with the SDV. These data are presented in table 3.3.13-10 and indicate the SB OTV mode provides a benefit of approximately $1.1B or 11% as compared with the GB OTV mode. The lower cost provided by the SB OTV is primarily due to its being able to use a more cost-effective launch system for cargo. The approach used by the SB OTV is to continue using the SDV but to switch to an expendable tanker. In the case of the GB OTV, however, the only option available in this study was to utilize a launch system that could return the OTV to Earth for servicing and reuse. The least-cost launch system which satisfies this requirement is the shuttle-growth vehicle (see fig. 3.3.11-2 for cost comparison with the basic STS).

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Table 3.3.13-10 Life Cycle Cost Summary Without SDV RPS

- SPACE BASED OTV
  SWITCH TO EXPENDABLE TANKER AND SHROUD BUT RETAIN STANDARD SDV AND STS
- GROUND BASED OTV
  SWITCH TO THE NEXT LEAST COST LAUNCH SYSTEM WHICH CAN RETURN REQUIRED PAYLOADS ---- SHUTTLE GROWTH
- COST COMPARISON:

	GROUND BASED OTV - 2 SIZES	SPACE BASED OTV - ONE SIZE	COMMENT
OTV	(1976)	(1700)	SAME
TANKER	N/A	(1126)	EXPENDABLE
DDTE		340	
PRODUCTION		765	
OPERATIONS		20	
SOC SYSTEMS	(TBD)	(205) + TBD	SAME
SDV	(N/A)	(4160)	EXPENDABLE SHROUD
DDTE		1000	
PRODUCTION		400	SHROUD INCL. IN OPS
OPERATIONS		2760	
SHUTTLE GROWTH (SG)	8600	(N/A)	
DDTE	2000		
PRODUCTION	400		
OPERATIONS	6200		
STS	(N/A)	(2060)	SAME
• TOTAL COST TO DATE:	10576	9240	
• TBD ESTIMATE	100	300	
• POTENTIAL TOTAL	10676	9540	

A final comment regarding the GB OTV mode without use of an RPS concerns the concept of an expendable OTV to enable use of the SDV. First, all OTV's cannot be expendable. Approximately 40 flights are to be manned and will return to LEO. There are also 60 unmanned satellite-servicing flights that could be considered expendable rather than reusable; however, they involve expensive servicing equipment. Although the manned OTV's could be expendable after reaching LEO, an interest in their reuse would require additional STS flights to return them to Earth for servicing. The other 140 OTV's could be expendable. The difference in their production cost versus reusable OTV's is estimated to be approximately \$2.5 billion, even after the number of units and learning curve have been applied (average cost of \$16 million versus \$30 million for expendable and reusable). There would be some savings in launch costs in the expendable concept because the SDV rather than shuttle-growth vehicle would be used. This may be offset, however, by the need to use more STS launches as indicated earlier. In summary, the cost increase over a reusable GB OTV with SDV and RPS with a combination expendable and reusable GB OTV using STS and SDV (without RPS) is estimated to be about the same (approximately \$2 billion) as the cost increase with the reusable GB OTV using only a shuttle-growth vehicle. The reusable GB OTV mode is therefore judged to be the most desirable because it has the potential to capitalize more from technology improvements and does not have the operational nuisance of OTV disposal.

3.4 ACCELERATED TECHNOLOGY VEHICLES

This section identifies the characteristics assumed for accelerated technology OTV's, describes the vehicles and their performance, identifies the impact on launch recovery, estimates their costs, and compares the life cycle cost with normal growth technology OTV's.

The major emphasis of this analysis was to evaluate the benefits of a liquid fluorine/hydrogen (LF_2/LH_2) main engine and an advanced LO_2/LH_2 main engine. Both engines have better performance than that provided by normal growth and, in the case of the LF_2/LH_2 system, a higher propellant bulk density which results in smaller vehicles as compared with LO_2/LH_2 systems.

For the most part, these data are presented in a format of direct comparison with the normal growth OTV's.

3.4.1 Accelerated Technology Projections

Accelerated technology in the context of this forecast is defined as that which is judged to be technically feasible by the 1990 readiness date but, at this time, is receiving little or no funding to bring about its development.

Improvements in subsystems other than LF_2/LH_2 and advanced LO_2/LH_2 main engines were judged not as significant as those provided by the main engines; therefore, only normal growth technology projections for these subsystems have been incorporated. A summary of the technology projections is presented in table 3.4.1-1.

In the case of the LF_2/LH_2 main engine, the existing data base was that provided by the Pratt & Whitney report PDS-2687, "O_2H_2 and F_2H_2 Rocket Engine Parametric Data," dated 1968. The indicated specific impulse of 511 sec was projected by doing the following: (1) the engine length was allowed to be the same as for the normal growth LO_2/LH_2 engine (1.524m stowed, 3.048m overall) and (2) the ratio of performance improvement in LO_2/LH_2 engines between 1968 and that projected for 1990 (as suggested in sec. 3.3.2) was applied to the 1968 LF_2/LH_2 data to obtain its 1990 performance projection. The associated mixture ratio, chamber pressure, and weight are also indicated and an area ratio of 600 was assumed. Engine life and DDT&E costs reflect a compromise between data available from reference 8 and from discussions with Pratt and Whitney (P&W). The LF_2/LH_2 engine DDT&E cost is estimated to be 75% greater than for a comparable LO_2/LH_2 engine, while the design life for the engine and the stage is assumed to be 75% as long as for an LO_2/LH_2 system. The significantly greater bulk density of the LF_2/LH_2, as compared with LO_2/LH_2, relates to its atomic structure resulting in a stoichmetric combination with hydrogen at a mixture ratio of 19 as compared with 8 for LO_2/LH_2.

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Table 3.4.1-1 Chemical OTV Accelerated Technology

- SUBSYSTEMS
  - STRUCTURE ····· COMPOSITES ALREADY USED EXTENSIVELY
  - AVIONICS ····· NOMINAL REDUCTION IN WEIGHT & POWER
  - MAIN PROPUL. ····· FLUORINE/HYDROGEN, ADV LO_2/LH_2
  - ACS ····· N_2H_4 STILL MOST EFFECTIVE
  - THERMAL ····· MODEST IMPROVEMENTS
- MAIN ENGINE CHARACTERISTICS (66 KN) – – ····· TYP. OF SB OTV (EXPANDER TYPE)

KEY PARAMETER	NORMAL LO_2/LH_2	LF_2/LH_2	ADV. LO_2/LH_2
SPECIFIC IMPULSE (SEC)	485	511	499
MIXTURE RATIO (O/F)	6/1	13/1	6/1
EXPANSION RATIO	700	600	700
CHAMBER PRESS (K Pa)	11032	11032	11730
WEIGHT (KG)	193	193	174
LIFE (HOURS)	10	7.5	10
DDTE COST ($M)	270	470	335
• PROP. BULK DENSITY (KG/M^3)	360	612	360

Improvements also have been projected in LO_2/LH_2 engines beyond that characterized for normal growth engines. An Isp of 499 sec can be envisioned through improvements in combustion chamber thermal performance and/or turbomachinery efficiencies. A 10% weight reduction is also suggested through development of lighter weight turbomachinery. DDT&E costs reflect a 25% increase over that of a normal growth new LO_2/LH_2 engine.

3.4.2 LF_2/LH_2 OTV's

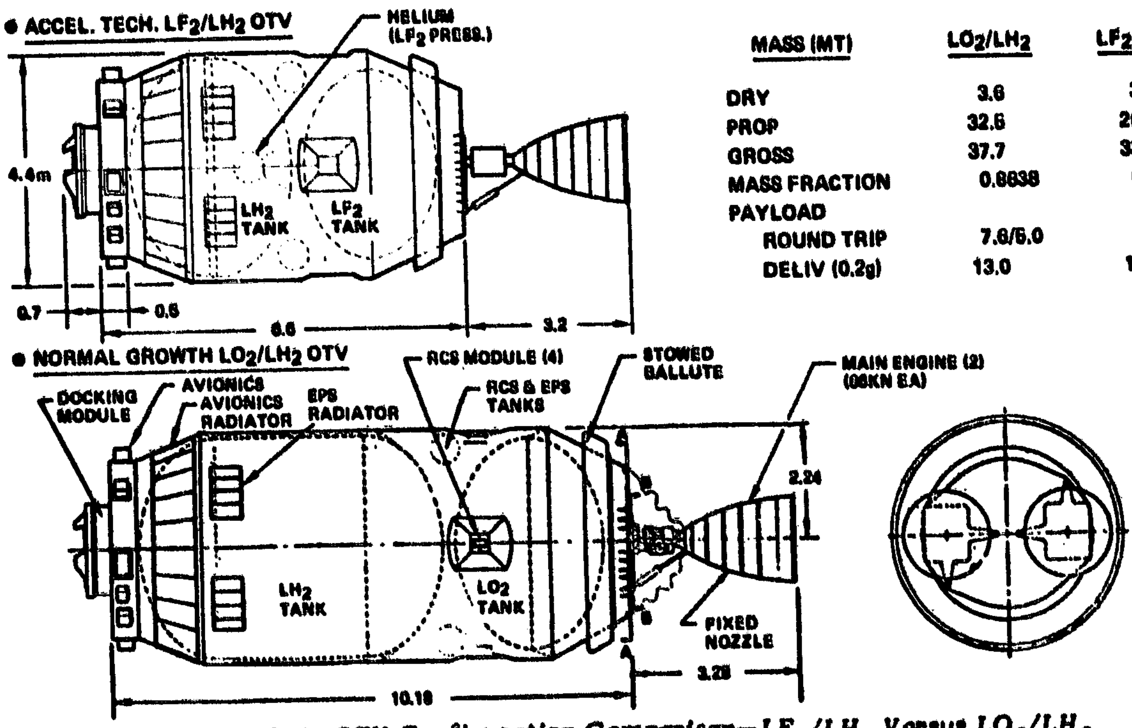
3.4.2.1 Design Considerations

Rocket propulsion systems using LF_2/LH_2 propellants have been successfully designed and tested including modification of a P&W RL-10 engine that used LF_2 as an oxidizer. Successful operation of the RL-10 was accomplished for a total of 1000 sec of engine operation with only minor damage. The primary design problems are those associated with materials compatibility with fluorine. Metals that have been suitable for use with fluorine, as recommended in reference 9, include: 2219-T87 and 6061-T6 aluminum, 304L and 17-4 CRES, A286 steel, 6AL-4V titanium, nickel, and nickel-copper alloys such as Monel, Inconel 600V, and Hastalloy B. These metals form hard, tough, fluoride surface films when passivated by exposure to gaseous fluorine. Most organic compounds react readily with fluorine. Teflon (carefully prepared) is resistant to fluorine in a static application, but its tendency to cold flow can lead to edge conditions or voids which are conducive to attack in a dynamic fluorine system so that it becomes unsatisfactory for valve seats or dynamic seals. Metal valve seats and seals are preferred. Cleanliness of fluid systems exposed to liquid or gaseous fluorine is vitally important. Contaminants such as grease, oil, inclusions or flux residues in welds, or moisture cause reactions which may become self-sustaining. Moisture reacts with fluorine to form HF (hydrogen fluoride or hydrofluoric acid) which, in turn, reacts with the fluoride surface layers on some metals causing brittleness or complete breakdown. System designs should avoid sharp corners or dead-ended pockets where turbulent flow conditions may produce accelerated corrosion or provide traps for contaminants which can cause self-sustaining reactions.

Both fluorine and HF are highly toxic and are safety hazards to personnel.

3.4.2.2 Space-Based OTV

The configuration for the SB LF_2/LH_2 OTV is presented in figure 3.4.2-1 along with the normal growth LO_2/LH_2 OTV. Sizing of the `age was established by the 0.2g delivery mission of 32t to GEO. Such a mission requires two stages of the size indicated. A



MASS (MT)	LO_2/LH_2	LF_2/LH_2
DRY	3.6	3.1
PROP	32.6	28.1
GROSS	37.7	33.6
MASS FRACTION	0.8638	0.8648
PAYLOAD		
ROUND TRIP	7.6/5.0	7.6/5.0
DELIV (0.2g)	13.0	13.0

Figure 3.4.2-1 OTV Configuration Comparison—LF_2/LH_2 Versus LO_2/LH_2

single-stage LF_2/LH_2 OTV provides a length reduction of 3.7m (25%) when compared with a normal growth LO_2/LH_2. The major reasons for this reduction are less propellant due to higher specific impulse and higher propellant bulk density. The majority of the subsystem design approaches for this OTV are the same as for the SB LO_2/LH_2 OTV defined in section 3.3.3.2 (although in some cases the sizing may be a little different). Exceptions are the main engine, LF_2 tank pressurization system, and pressure levels. Characteristics of the main engine have already been discussed in the preceding section. The pressurization system for the LF_2 tank has been switched to a regulated helium system because insufficient data were available to characterize an autogenous GF_2 system for an advanced expander cycle LF_2/LH_2 engine. The high-pressure helium supply is stored in the LH_2 tank to minimize storage bottle weight. The pressurant flow is heated in a heat exchanger, using the LH_2 tank pressurization gas as a heat source, to minimize the helium usage. A schematic of this system is shown in figure 3.4.2-2. The LF_2 tank operating

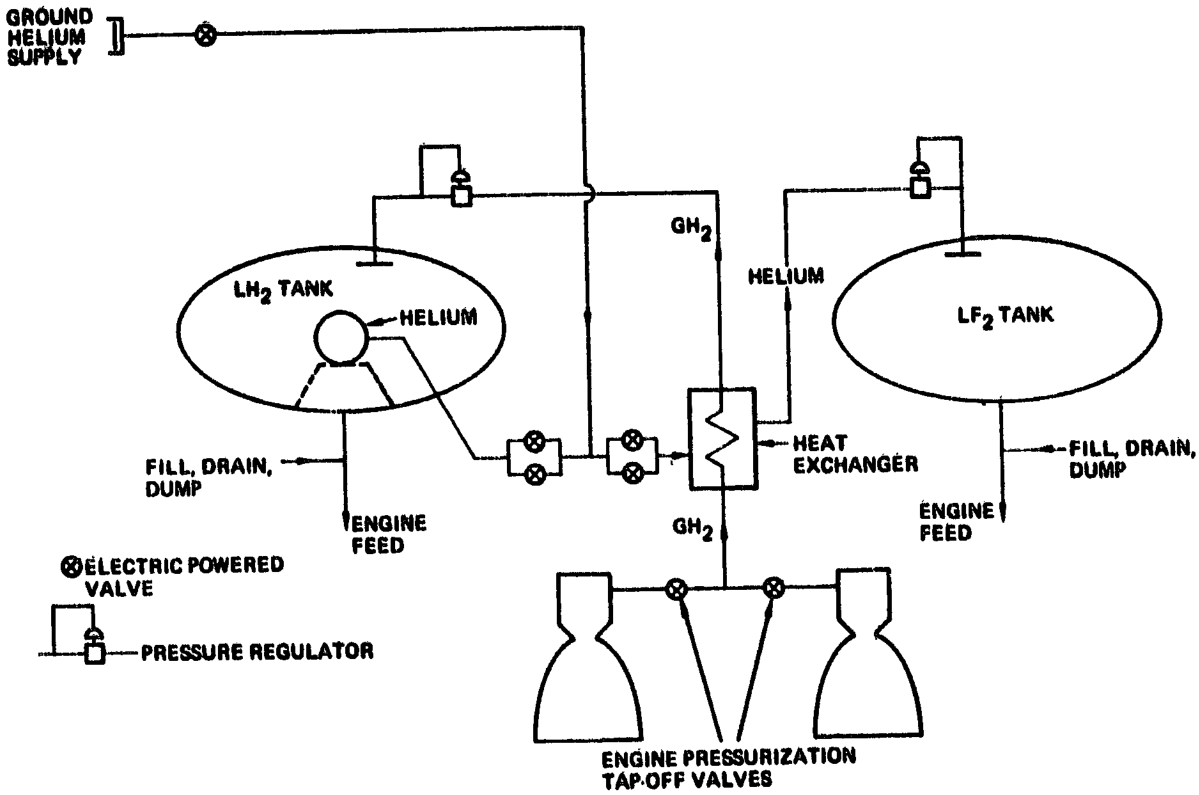


Figure 3.4.2-2 LF_2/LH_2 Propulsion System Tank Pressurization

pressure and maximum vent pressure are respectively 165.6 kPa (24 psia) and 186.3 kPa (27 psia). Comparable pressures for the LO_2 tank in the LO_2/LH_2 OTV were 138 kPa (20 psia) and 151.8 kPa (22 psia). The primary reason for the differences is associated with net positive suction head (NPSH) requirements. NPSH requirements for the LF_2 engine inlet have not been well defined. In RL-10 engine tests with LF_2 propellant, a minimum NPSH of 55 kPa (8 psi) was required which was approximately the same as required for operating the engine on LO_2. Advanced expander cycle LH_2/LO_2 engine studies have projected required oxidizer inlet NPSH = 6.9 kPa (1.0 psi). However, due to the lack of data on LF_2 pump development, it was estimated that an FOTV era engine could achieve an LF_2 inlet NPSH = 27.6 kPa (4.0 psi). This NPSH requirement, plus estimated feed system pressure drop and LF_2 tank vapor pressures, resulted in the indicated pressures.

A summary mass statement for the SB LF_2/LH_2 OTV is presented in table 3.4.2-1. The mass fraction of 0.8648 reflects the gross weight of 33 563 kg and the total main impulse load of 29 087 (28 777 kg nominal, 310 kg reserve).

Table 3.4.2-1 Single-Stage SB OTV Summary Weight Statement

	ASCENT TO LEO	GEO MISSION
STRUCTURE	1063	1063
THERMAL CONTROL	103	103
AVIONICS	292	292
ELECTRICAL POWER SYSTEM (EPS)	234	234
MAIN PROPULSION SYSTEM (MPS)	667	667
ATTITUDE CONTROL SYSTEM (ACS)	122	122
SPACE MAINTENENCE PROVISIONS	216	216
WEIGHT GROWTH MARGIN	405	405
(OTV DRY WEIGHT – LESS BALLUTE)	(3102)	(3102)
RESIDUALS	10	362
RESERVES	—	310
(OTV BURNOUT WEIGHT)	(3112)	(3774)
BALLUTE *	299	299
INFLIGHT LOSSES		355
FUEL CELL REACTANT		46
ATTITUDE CONTROL PROPELLANT		312
MAIN IMPULSE PROPELLANT		28,777
(OTV GROSS WEIGHT)	(3411)	(33,563)
PAYLOAD	—	8041
(OTV + P/L WEIGHT)	(N/A)	(41,604)
CONTRACTOR FURNISHED ASE	658	
GOVERNMENT FURNISHED ASE		
BALLAST	—	
(LAUNCH WEIGHT)	(4069)	
OTV MASS FRACTION		0.8648

* INCLUDES MARGIN

3.4.2.3 Ground-Based OTV

The configuration of the large GB LF_2/LH_2 is presented in figure 3.4.2-3 with overall geometry and physical characteristics noted. This OTV is similar in appearance to the SB LF_2/LH_2 OTV shown in figure 3.4.2-1 and was also sized by the 32t delivery mission. Major differences are slightly larger main propellant tanks, a full diameter

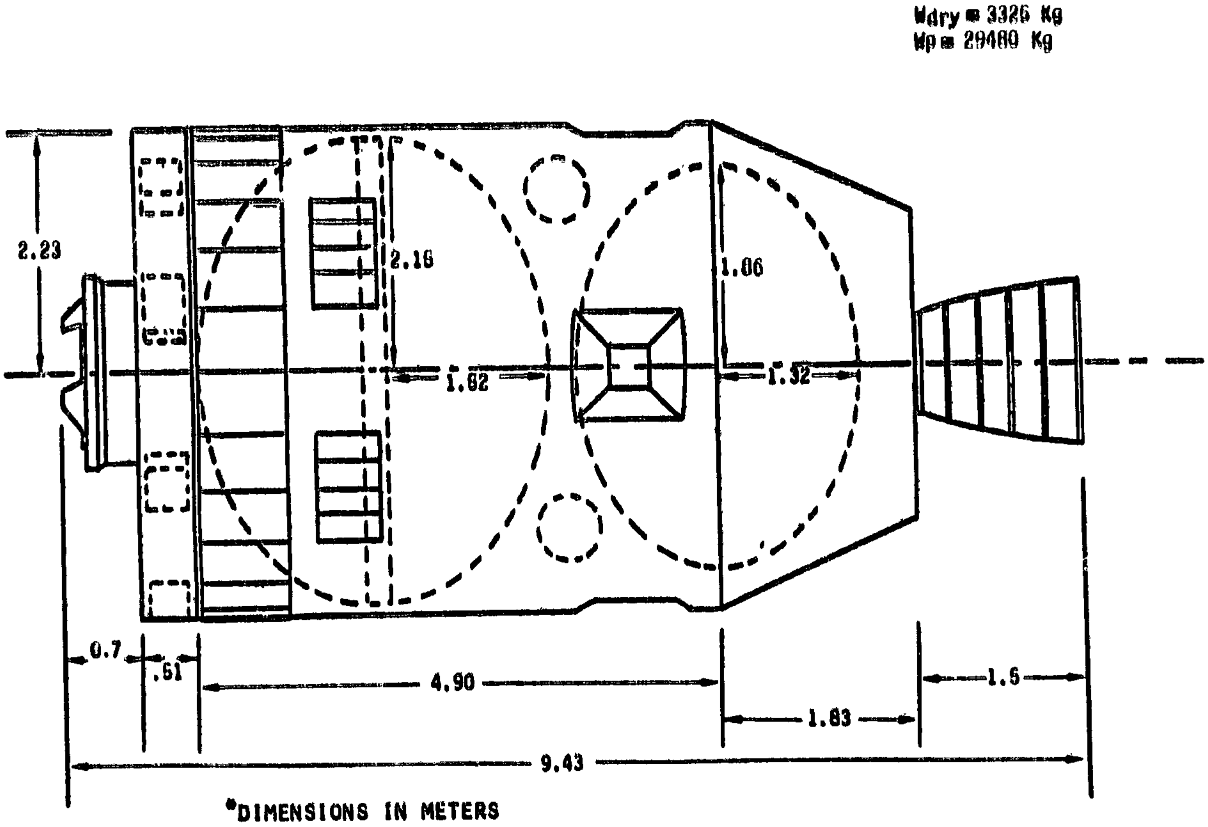


Figure 3.4.2-3 Ground-Based LF_2/LH_2 OTV

avionics/equipment ring assembly, and stowed nozzles on the main engines. The slightly larger tanks are necessary to accommodate an increase in main propellant mass of 454 kg (nominal plus reserve). The full diameter avionics/equipment ring assembly is a preferred configuration for payload accommodation during launch and ascent to LEO and for avionics/equipment packaging. The retractable nozzles on the main engines are necessary to maximize payload length capability. The large GB OTV is nearly identical to the SB OTV with respect to all other aspects of overall configuration definition (number, types, and thrust of main engines; ACS propellant type; structural materials, methods of construction, and meteoroid/debris protection scheme; thermal control elements; electrical power source; ballute type; basic avionics; etc.). The major exception is provisions for space maintenance of selected critical components, of which the large GB OTV has none.

A summary mass statement for the large GB LF_2/LH_2 OTV is presented in table 3.4.2-2. The mass fraction of 0.8606 reflects the gross weight of 34 258 kg and a total main impulse propellant load of 29 543 kg (29 229 kg nominal, 314 kg reserve). ASE mass reflects use of a shuttle derivative.

Table 3.4.2-2 Single-Stage GB OTV Summary Mass Statement [1]

	MASS (kg)
STRUCTURE	1302
THERMAL CONTROL	164
AVIONICS	292
ELECTRICAL POWER SYSTEM (EPS)	234
MAIN PROPULSION SYSTEM (MPS)	788
ATTITUDE CONTROL SYSTEM (ACS)	118
WEIGHT GROWTH MARGIN	436
(OTV DRY WEIGHT—LESS BALLUTE)	(3324)
RESIDUALS	366
RESERVES	314
(OTV BURNOUT WEIGHT)	(4004)
BALLUTE*	304
INFLIGHT LOSSES	366
FUEL CELL REACTANT	46
ATTITUDE CONTROL PROPELLANT	318
MAIN IMPULSE PROPELLANT	29,220
(OTV + P/L WEIGHT)	(34,258)
PAYLOAD	8005
(OTV + P/L WEIGHT)	(42,263)
CONTRACTOR FURNISHED ASE } [2]	} 2268
GOVERNMENT FURNISHED ASE }	
BALLAST	
(LAUNCH WEIGHT)	(44,531)
OTV MASS FRACTION	0.8608

*INCLUDES MARGIN

[1] MANNED RESUPPLY MISSION
7600 UP/5000 DN

[2] LAUNCH BY SDV

A small GB LF_2/LH_2 OTV was also considered to enable two of these vehicles to be launched at once, thus reducing the number of launches as was done for the small LO_2/LH_2 OTV. Although a configuration was not developed, the key features would be the same as for the large GB LF_2/LH_2 OTV with the exception of (1) smaller propellant tanks, because the vehicle was sized for a delivery of approximately 8t, and (2) use of a single engine, because the vehicle was not used for manned missions.

3.4.2.4 Performance

The parametric relationship between burnout mass and propellant capacity, which was used for performance calculations, is shown in figure 3.4.2-4. Round trip parametric performance for the SB and GB LF_2/LH_2 OTV's is presented in figure 3.4.2-5. GEO delivery parametric performance for the SB OTV is presented in figure 3.4.2-6. Offloaded

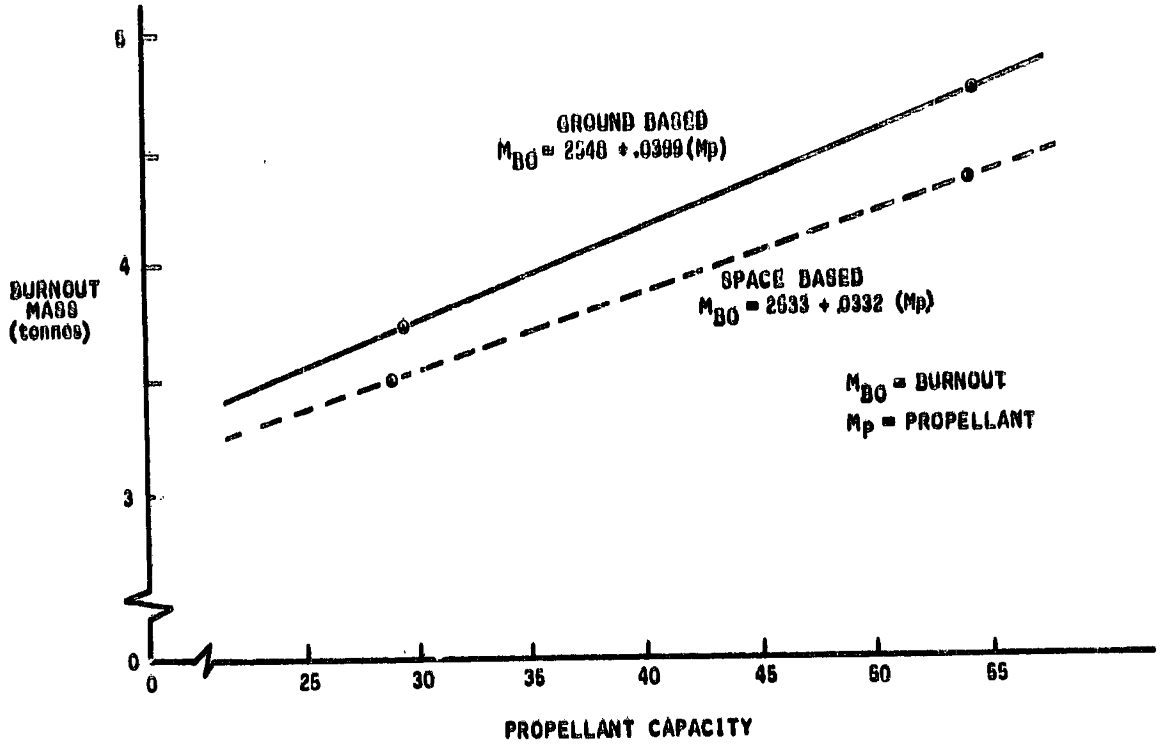


Figure 3.4.2-4 LF_2/LH_2 FOTV Parametric Mass Relationships

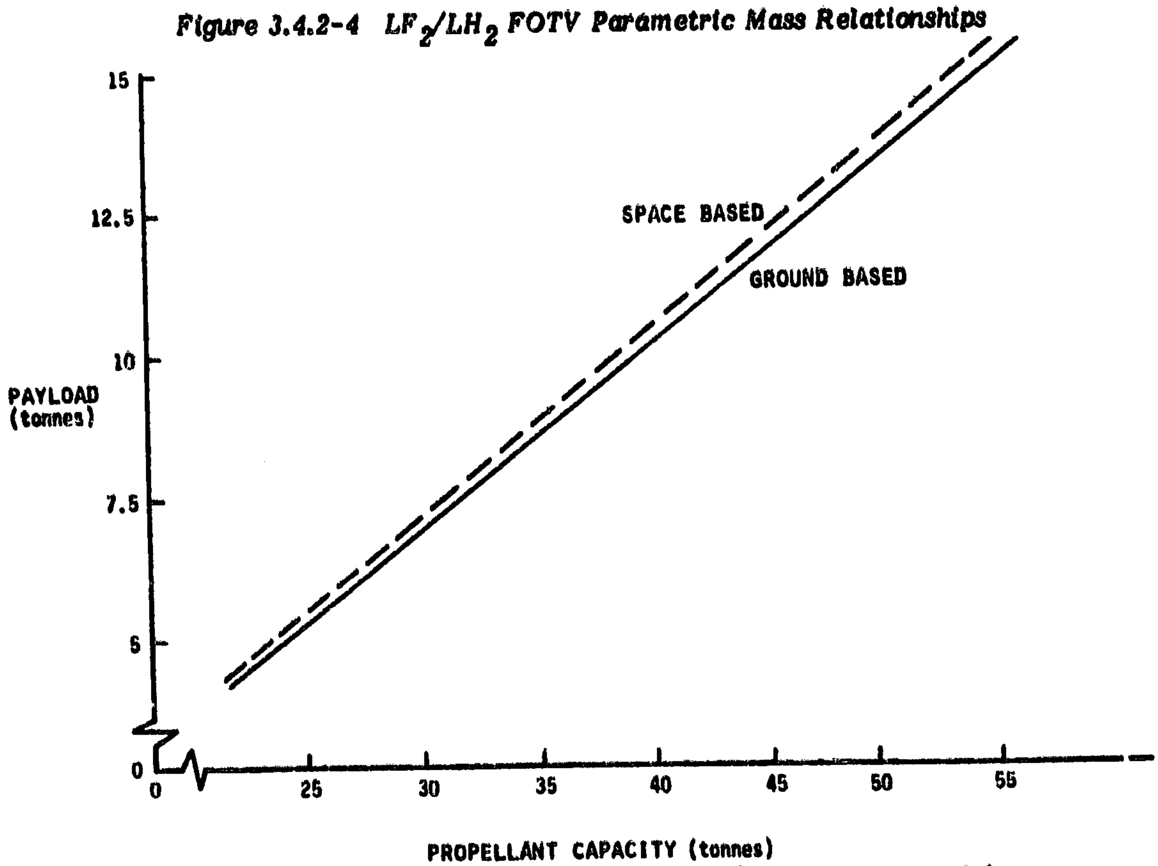


Figure 3.4.2-5 LF_2/LH_2 OTV Round Trip Performance Parametrics

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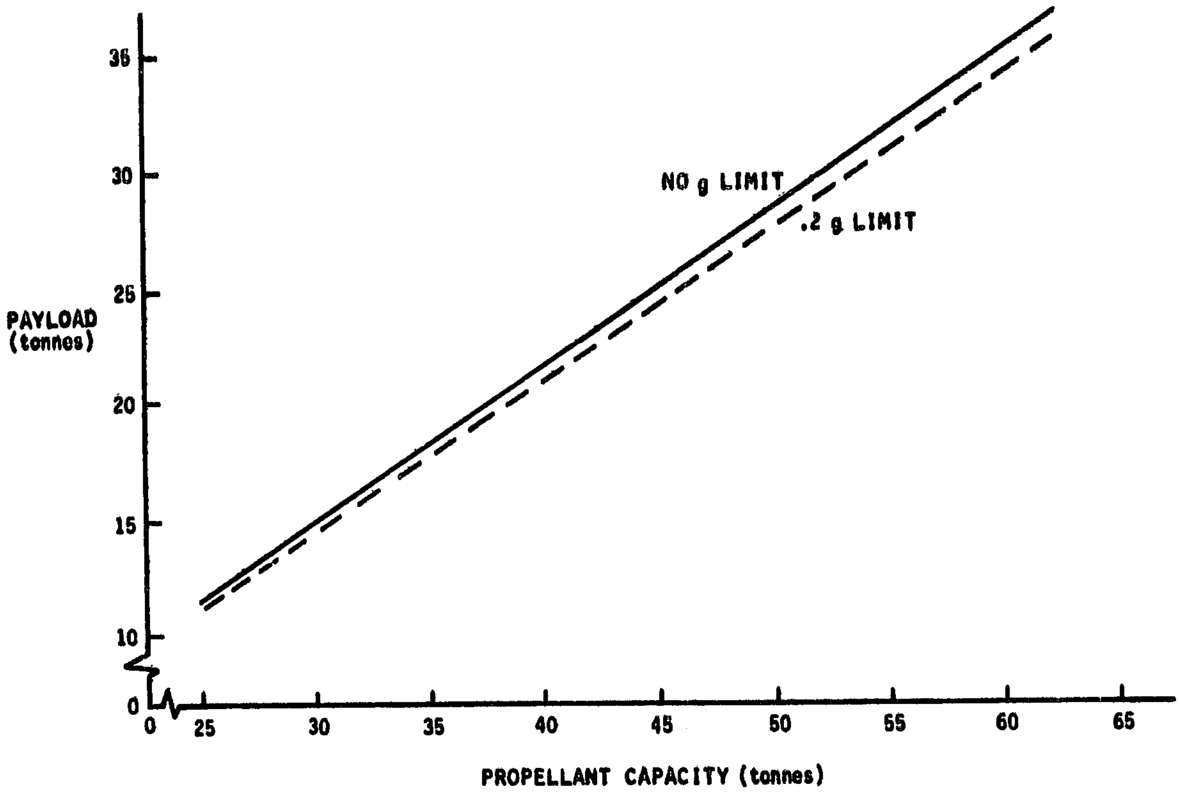

Figure 3.4.2-6 Space-Based LF_2/LH_2 OTV Delivery Parametric Performance

performance capability for two-stage SB and GB OTV's is shown in figure 3.4.2-7. Single-stage offloaded performance capability of the SB OTV is shown in figure 3.4.2-8.

The small GB LF_2/LH_2 was sized for the delivery of up to 7955 kg with no g constraint and 6820 kg for a 0.1g constrained delivery. The propellant loading for these payload requirements is 20 450 kg.

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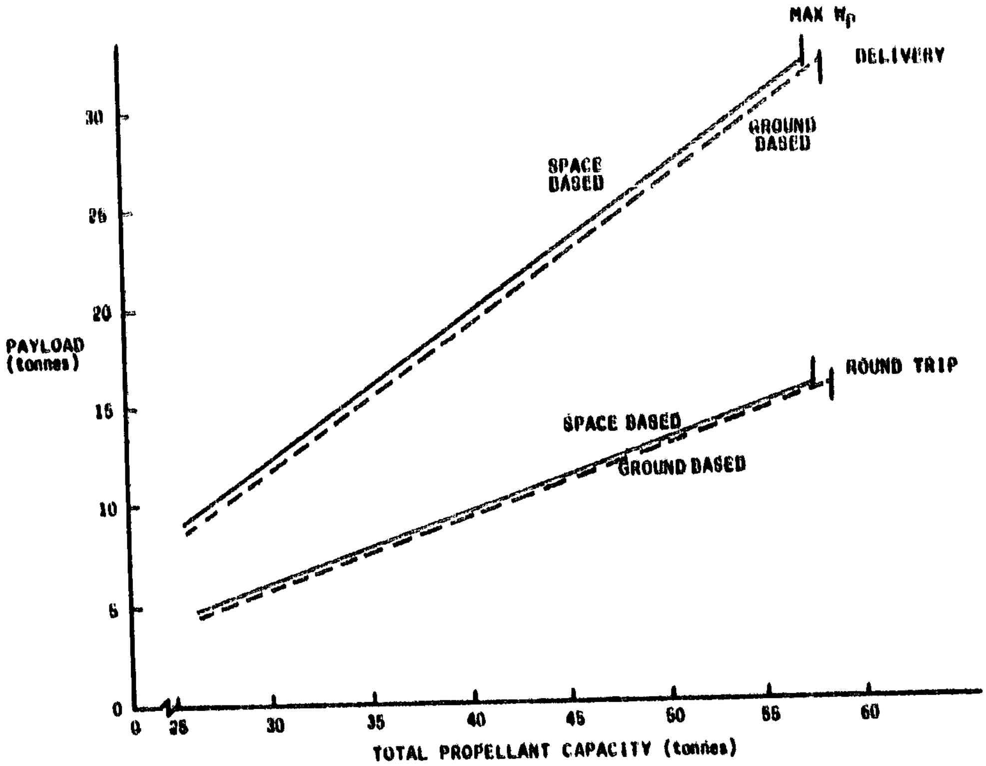

Figure 3.4.2-7 Two-Stage LF_2/LH_2 OTV Offloaded Performance

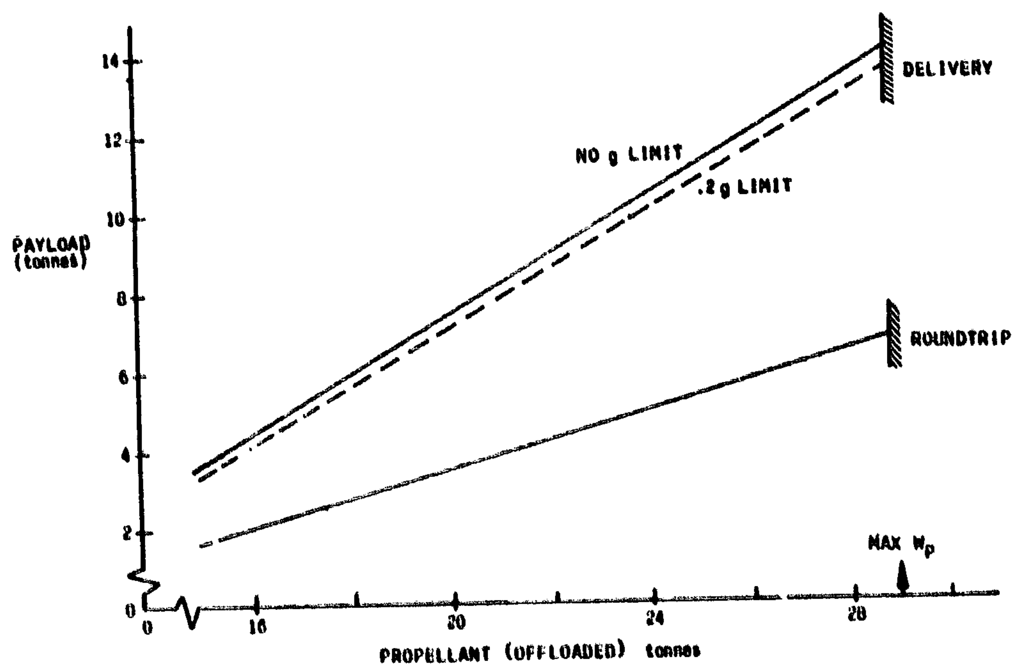

Figure 3.4.2-8 Space-Based LF_2/LH_2 OTV Offloaded Performance

3.4.2.5 Launch and Recovery Operations

The impact of LF_2/LH_2 OTV's on launch operations appears to be more significant for the SB OTV because it enables the number of launches to be reduced relative to the normal growth SB LO_2/LH_2 OTV. This occurs primarily because of less total propellant being required and the ability to mass limit propellant launches. As a result, when compared on a relative basis of dedicated propellant tanker launches, the LF_2/LH_2 SB OTV reduces the number of SDV launches from 100 to 84 as compared with the normal growth LO_2/LH_2 SB OTV. No benefit in recovery operations appears possible since the tankers are still too large and too numerous for return by the shuttle orbiter when used for SOC resupply.

In the case of the GB LF_2/LH_2 OTV concept using two sizes, benefits occur in terms of increasing the length and mass margins on each launch but not in terms of reducing the number of launches. Stage length reductions result in increasing the length margins to 5m to 7m out of 24m available on each SDV launch of a large GB LF_2/LH_2 OTV. Mass reductions of approximately 3t per large OTV result in launch margins of 10t to 12t out of 60t. Whether the length and mass margins can be used together to reduce launches is the subject of a more detailed launch manifesting analysis. In terms of recovery operations, the large GB OTV is still too large to return with a shuttle orbiter which also includes a SOC logistics module. The small OTV would be compatible for return; however, with 116 small OTV's and only 72 orbiter return flights, a mismatch still occurs.

3.4.3 Advanced LO_2/LH_2 OTV's

As an alternative to LF_2/LH_2 for accelerated technology, consideration was given to an advanced LO_2/LH_2 system providing a higher Isp than that of the normal growth engine. Configurations were not developed for the advanced LO_2/LH_2 OTV because of its similarity to normal growth vehicles, with the exception of slightly smaller propellant tanks resulting from the higher Isp.

The advanced SB OTV used an Isp of 499 sec and an area ratio of 700, while the advanced GB OTV had an Isp of 498 sec because of a lower area ratio (626) due to stowing limitations. Offloaded performance parametrics for the SB and GB OTV's are shown in tables 3.4.3-1 and 3.4.3-2, respectively.

A reduction in number of dedicated propellant tanker launches from 100 to 94 would occur for the advanced SB LO_2/LH_2 OTV relative to the normal growth OTV because of less propellant. Only a small benefit in length and mass margins would occur for the advanced GB LO_2/LH_2 OTV.

Table 3.4.3-1 SB Advanced LO_2/LH_2 OTV Performance Equations

Stage size = 31 160 kg

1. GEO round trip: Maximum payload = 6500 kg
   W_p = 2.923 * payload + 12 178 kg

2. GEO delivery: Maximum payload = 13 730 kg
   W_p = 1.399 * payload + 11 964 kg

3. Two-stage GEO delivery: Maximum payload = 31 960 kg
   W_p = 1.485 * payload + 14 864 kg

Table 3.4.3-2 GB Advanced LO_2/LH_2 OTV Performance Equations

Stage size W_p = 31 750 kg; W_{bo} = 4358 kg

1. GEO round trip: Maximum payload = 6340 kg
   W_p = 2.941 * payload + 13 101 kg

2. GEO delivery: Maximum payload = 13 440 kg
   W_p = 1.404 * payload + 18 641 kg

3. Two-stage GEO delivery: Maximum payload = 31 880 kg
   W_p = 1.490 * payload + 16 012 kg

3.4.4 Cost Comparison with Normal Growth

The life cycle cost comparison of accelerated technology OTV's relative to normal growth technology OTV's is presented in table 3.4.4-1. It should be noted that these data are presented for the case of using an SDV with RPS. Should the RPS not be available, the

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Table 3.4.4-1 Accelerated Versus Normal Technology Chemical OTV Life Cycle Cost

- MAIN ENGINE IMPACT
- 1980 DOLLARS (IN MILLIONS)

HARDWARE ELEMENT	HYBRID GB OTV		SB OTV		
	NORMAL LO_2/LH_2	ACCEL. LF_2/LH_2	NORMAL LO_2/LH_2	ACCEL. LF_2/LH_2	ACCEL. LO_2/LH_2
OTV	(1976)	(2276)	(1700)	(1880)	(1770)
• DDTE	816	1046	695	905	765
• PRODUCTION	390	460	365	435	365
• OPERATIONS	770	770	640	540	640
SOC SYSTEMS	(TBD)	(TBD)	(206 + TBD)	(206 + TBD)	(206 + TBD)
TANKER	N/A	N/A	(730)	(730)	(730)
SDV/RPS	(4640)	(4540)	(4300)	(3960)	(4170)
STS	(2060)	(2080)	(2060)	(2060)	(2060)
COST TO DATE	8620	8920	8995	8925	8935
	REFERENCE	+300	REFERENCE	−70	−60

general conclusions regarding the value of accelerated technology versus normal growth are expected to remain the same.

In the case of the GB LF_2/LH_2 system, a \$300M LCC penalty exists over a normal growth LO_2/LH_2 concept as a result of the higher OTV DDT&E associated with the engine and the additional production cost because of shorter life (assumed to be 25% less than for an LO_2/LH_2 OTV). No difference occurred in the launch cost for the GB OTV because there was no reduction in the number of launches as indicated earlier and no credit was given to the mass margins available. An advanced GB LO_2/LH_2 OTV cost is not shown but it would have had a new LCC penalty of approximately \$80M due to its engine development.

The SB LF_2/HF_2 OTV concepts show a \$70M advantage over normal growth LO_2/LH_2, primarily as a result of the savings in propellant launches offsetting the development cost of the engine. The advanced SB LO_2/LH_2 OTV system also shows a small cost advantage for the same reasons.

A comparison of SB and GB OTV's, both using LF_2/LH_2, indicates essentially no difference in cost as compared with a \$375M advantage for GB OTV when both OTV's used normal growth LO_2/LH_2 engines. Again, this is primarily because the SB OTV concept can take advantage of the higher performance in the form of fewer propellant tanker launches.

In summary, the accelerated technology OTV's do not provide an LCC which justifies the additional developmental risk. Consequently, no engine advances beyond normal growth LO_2/LH_2 appear warranted. Finally, the value of accelerated technology appears to be more beneficial to SB OTV's than to GB OTV's.

3.5 VALUE OF NORMAL GROWTH TECHNOLOGY

The previous section indicated the use of accelerated technology chemical OTV's did not significantly improve the total transportation cost relative to normal growth technology. As a result, there was an interest in defining the value of the assumed normal growth technology for the second-generation OTV relative to technology assumed available for the first-generation LO_2/LH_2 OTV defined in the Phase A studies.

The results of this assessment are presented in table 3.5-1. Several combinations of

Table 3.5-1 LO_2/LH_2 OTV Value of Normal Growth Technology

- FOTV LOW MISSION MODEL (1280 MT OF GEO PAYLOADS)
- COST REFLECTS NET DIFFERENCE IN DDTE, PRODUCTION AND LAUNCH

TECHNOLOGY FEATURES	LAUNCH SYSTEM: STS + SDV	LAUNCH SYSTEM: STS ONLY
• NORMAL GROWTH [3] NEW ENGINE [1] NEW BALLUTE [2]	REFERENCE TOTAL COST OF $9 BILLION	REFERENCE TOTAL COST OF ≈ $ 11.5 BILLION (+ 27%)
• WITHOUT NEW BALLUTE (BUY NEW ENGINE)	+$65M (0.7%)	+$160M (1.4%)
OR		
• WITHOUT NEW ENGINE (BUY NEW BALLUTE	+$30M (0.3 + %)	+$320M (2.8%)
OR		
• WITHOUT NEW ENGINE OR NEW BALLUTE (USE RL-10IIB & STD BALLUTE)	+$116M (1.3%)	+$575M (5.0%)
OR		
• WITHOUT ANY BALLUTE (NEW ENGINE/ALL PROPULSIVE)	+$250M (2.7%)	+$820M (7.1%)

[1] LO_2/LH_2 AT 485 SEC, 10 HR LIFE [2] TRANSPIRATION COOLED [3] INCLUDES ALL N.G. SUBSYSTEMS

technology features are examined in conjunction with two different launch system options. The technology features are those judged to have the most leverage, such as the engine and ballute as applied to the SB LO_2/LH_2 OTV. The results indicate that the penalty for not using the normal growth technology is not too significant if the STS plus

SDV launch systems are available but becomes more significant if only the STS is used. In summary, launch systems significantly influence the value of OTV technology.

Specific results of this assessment are as follows for the STS plus SDV launch systems.

1. The cost penalty is less if the new ballute (+\$30M) is developed rather than a new engine (+\$65M).

2. Without either a new engine or new ballute, the penalty is only 1.3% (\$115M).
3. Using an all-propulsive mode (no aerobraking ballute) but with a new engine, the cost penalty is less than 3% (\$250M). This relatively small penalty for an all-propulsive mode appears to be in conflict with the Boeing Phase A study results which indicated a 15% advantage for aerobraking and their mission. There are several factors, however, which explain this result. First, the FOTV data assume the use of a more cost-effective launch fleet in the form of STS plus SDV rather than STS plus STS growth used in the Phase A study. Secondly, the Phase A study mission model required a significant amount of expendable hardware for the all-propulsive mode due to launch vehicle constraints, whereas the FOTV OTV's were sized so all units were reusable.

A more significant cost impact occurs when the launch system is confined to the basic STS only. The key points are as follows:

1. As a point of reference, the use of normal growth technology in the OTV would result in a 27% (\$2.5B) increase in LCC.
2. Without a new engine or new ballute, the increase is 5% (\$575M).
3. With an all-propulsive OTV, the penalty exceeds 7% (\$820M).

As a final note, it should be stated that had the launch system been confined to the STS when evaluating the accelerated technologies, a greater benefit would have been shown for the more advanced systems. The magnitude of the total transportation cost, however, would have been greater than that provided by the normal growth OTV's when using STS plus SDV. The conclusion, therefore, is that procurement of SDV in conjunction with normal growth OTV's is more beneficial than accelerated technology OTV's used with the basic STS.

3.6 FINDINGS

A summary of the principal findings resulting from this comparison of SB and GB OTV's is presented below. These findings are highly related to the assumptions used, particularly to that of a first-generation reusable LO_2/LH_2 OTV with aeroassist capbaility being the point of departure.

1. There was no clear-cut winner. The cost comparison is very dependent on recovery and reuse considerations, available launch systems, and orbital support facility.
2. Configuration, design features, and performance are very similar. This was the result of subjecting the SB OTV to a thorough total transportation and operations analysis. The most significant impact on the SB OTV is protection against space debris and on-orbit maintenance provisions.
3. Accelerated technology, such as LF_2/LH_2 engines, does not provide a cost benefit. The engine does reduce stage length and improve performance, benefitting an SB OTV More than a GB OTV because the reduced propellant allows fewer tanker launches as long as on-orbit propellant storage capability is available.
4. Accelerated technology propellant storage/transfer has a payoff. Concepts have the potential to reduce the refueling losses from 12% to 5%. Such systems include space-qualified refrigerators and liquifiers.
5. SB OTV's can provide a total transportation cost savings. For an advanced space scenario employing a low-risk shuttle-derivative launch vehicle, without reusable payload system, and a manned orbit facility, a savings of 11% was provided.
6. OTV stage and propellant tanker return needs are key considerations in launch system selection. This situation is caused by both length availability in the shuttle orbiter when supporting SOC and the number of orbiter flights compared with OTV flights or tanker launches.
7. The launch system employed is the single most dominating factor. Use of a basic shuttle plus its solid rocket cargo derivative results in a 15% savings over the next most effective system employing a shuttle using liquid rocket boosters and liquid rocket cargo derivative vehicle.
8. Mission model size and makeup have the most direct influence on launch vehicle selection. The launch vehicle selection, in turn, will infuence the selected OTV basing mode.
9. Space-based OTV impact on SOC appears acceptable. A crew size of three is required at 40% duty cycle. Hangars are necessary for maintenance and debris

protection. Propellant storage tanks should provide sufficient capacity for an emergency OTV flight at any time.

10. A space base would have a valuable role with either a GB or SB OTV. In the case of the GB OTV, it could be used for mating payloads and OTV's to enable more effective launch manifesting. This same function is provided for the SB OTV in addition to supporting the maintenance and refueling operations.

11. Significant technology efforts are necessary for future OTV's. The most significant new technology associated with the second-generation OTV (GB or SB) is that of space-debris protection. Refueling technology needs to be addressed for the SB OTV in addition to demonstrations of on-orbit maintenance. Normal growth in technologies, such as new LO_2/LH_2 engines and transpiration-cooled ballute, offers performance, operation, and cost benefits that justify their development.

In summary, SB OTV's appear to offer the lowest total transportation costs for the least-risk approach regarding recovery and also provide flexibility in launch and flight operations for the case of normal growth technology. In addition, greater potential exists for reducing cost when accelerated technology is employed. Finally, development of a shuttle-derivative cargo launch vehicle provides the most significant means in reducing transportation cost in the 1995-2010 time frame.

3.7 RECOMMENDATIONS

The recommendations below are based on the assumption that a reusable LO_2/LH_2 OTV with aeroassist capability is in the procurement cycle. In summary, continued emphasis is recommended on all system elements including launch vehicle, OTV, and orbital support platform. The specific recommendations are as follows.

1. Continue investigations concerning the most effective shuttle-derivative launch vehicle. This is judged to be extremely important since operation of the SDV proved to be the most dominating cost factor. The work should reflect related performance and cost data from actual space shuttle flights rather than the preliminary design data used in the 1977 SDV studies. Consideration should also be given to the fact that advanced space scenarios may require a fleet of both STS and SDV systems and may thus impact the cost-per-flight characteristics. Cargo return needs also must be included. Accordingly, special emphasis should be given to investigating the feasibility of the reusable payload system and its related performance and cost features.

2. Consider the system implications of the following:

a. An unmanned platform instead of SOC for orbital support. Although support for SOC is increasing, the required time frame is still somewhat controversial. Accordingly, an unmanned platform–that can provide a "parking" location and housekeeping functions for the SB OTV is a possible precursor to SOC. Crew support for maintenance and checkout operations would be provided via shuttle launches. Associated crew launch cost and/or revisions to the maintenance provisions onboard the OTV are the key features to be defined.

b. Launch system confined to basic STS. Although the cost analysis indicated a substantial benefit when using the SDV, this does not ensure its development. Consequently, the effects of the mass and envelope constraints associated with the STS need assessing in terms of the impact on launch manifesting and number of required launches. Theoretically, the SB OTV concept should be less affected than the GB OTV since propellant via tankers can be more effectively manifested than hard cargo such as payloads and/or OTV stages.

3. Initiate future OTV technology efforts.

a. Space debris protection studies and demonstrations. The primary emphasis should be to establish the protection characteristics of materials associated with reusable cryogenic OTV's rather than extrapolated from data developed for habitats or expendable OTV's. Of major interest would be composite sheet and sandwich as well as MLI.

b. Propellant storage and transfer demonstrations. Cost effectiveness of the SB OTV is influenced by the additional amount of propellant which must be launched to cover all refueling losses. Further studies regarding the most effective means of accomplishing this function need to be performed as do relatively large-scale demonstrations of the top contender prior to committing to an SB OTV.

c. Maintenance needs for SB OTV. Consider on-orbit maintenance features during preliminary design phase of those systems requiring maintenance. Particular attention should be directed to the main engine. Demonstration of maintenance crew and time requirements also appears warranted before committing to an SB OTV due to its impact on SOC crew size and related users charge.

d. Development of key normal growth technologies. Most significant of these is a new LO_2/LH_2 engine and transpiration-cooled ballute. Although the cost benefit of these systems over the first-generation systems was not significant when used in conjunction with an SDV, they did pay for themselves and provided increased performance when necessary. Moreover, should only the basic STS be available, a savings of over 5% in total transportation costs would occur.

4. Maintain surveillance of all aerospace products for development of OTV-type subsystems. The most likely areas will include avionics (laser gyros and data bus), structures (composites), and electrical power generation systems.

4.0 ELECTRIC VERSUS CHEMICAL OTV's

This section presents the complete analysis associated with the comparison of electric versus chemical OTV's. The principal subsections include mission analysis, the definition and comparison of OTV's using normal growth and accelerated technology, and the overall findings and recommendations.

4.1 INTRODUCTION

Consideration of an electric orbit transfer vehicle for LEO to GEO cargo delivery is based primarily on its high specific impulse (up to 10,000 sec) as compared with 485 sec for an LO_2/LH_2 OTV. There are several key disadvantages, however, with the most notable being (1) relatively long trip times (typically 180 days between LEO and GEO), (2) solar array damage when passing through the Van Allen radiation belts (typically 40% power loss), and (3) relatively high costs associated with solar arrays and electric propulsion elements. A favorable comparison of the EOTV with an LO_2/LH_2 OTV, therefore, depends on how well the disadvantages can be minimized and whether the savings in recurring costs can offset the expected high production costs.

4.1.1 Scope

The comparison of electric versus chemical OTV's must take into consideration the total transportation requirements associated with a given mission model. In most cases this means high-priority cargo (rapid delivery), manned missions, and general cargo. Consequently, the comparison actually involves an assessment of the following OTV fleets:

1. EOTV's for trip-time insensitive payloads plus chemical OTV's for manned and high-priority cargo
2. Chemical OTV's for all payloads

4.1.2 Guidelines

The principal guidelines affecting the overall comparison are shown below:

1. Technology to be available in 1990
2. Vehicle to have IOC of 1995
3. Mission model to include projected activity between 1995-2010
4. EOTV to be confined to photovoltaic power supply
5. Chemical OTV to be a space-based LO_2/LH_2 system

6. Launch system to include the basic STS plus shuttle derivative with reusable payload system
7. Figure of merit to be total transportation system life cycle cost to accomplish the mission model

The technology availability and vehicle IOC dates, as well as mission model duration, were discussed in the study guidelines of section 1.3. The mission model for this comparison, however, was to be more ambitious than the SB versus GB OTV model in order to provide a greater opportunity to utilize the high performance of the EOTV. The EOTV power supply was confined to photovoltaics because of (1) the desire to build on the knowledge base established by the solar electric propulsion system, which is assumed to be the first-generation EOTV, (2) another in-progress study (Advanced Propulsion System Concepts for Orbital Transfer, NASA contract NAS8-33935) was investigating solar, thermal, and nuclear electric concepts, and (3) solar photovoltaics may be the only electric system available at the time of IOC. An SB LO_2/LH_2 OTV is assumed, based on the results of the space versus ground comparison and the feeling that an even more ambitious mission model would further justify space basing. The launch systems to be used are also based on the results of the SB versus GB OTV trade and the projected effectiveness of these systems for LEO delivery requirements exceeding those of the anticipated electric versus chemical OTV mission model. The transportation cost was to include that associated with launch vehicles, OTV's, and orbital support.

4.1.3 Emphasis and EOTV Issues

The major emphasis associated with this trade was the definition of EOTV because the chemical OTV had already been defined in the SB versus GB OTV trade. This definition was to include both design and operational features. The key issues associated with the utility of an EOTV were judged to be the following:

1. Payload compatibility—How many payloads could accept the long trip times? Should large payloads be transported as finished systems (meaning LEO construction) or as components (meaning GEO construction)?
2. Van Allen radiation impact—This involves the extent of the over-sizing of the EOTV due to solar array degradation, the design life limits imposed on other EOTV elements, and penalties imposed on payloads being transported.
3. Cost sensitivity to trip time and Isp—Short trip times are desirable from the standpoints of fleet sizing and minimizing radiation degradation; however, large amounts of propulsion power are required. High Isp reduces propellant requirements

but again requires more propulsion power. The goal then is to find the combination of Isp and trip time which gives the least system cost.

4.2 MISSION ANALYSIS

This section describes the background associated with the mission model used in comparing the electric and chemical OTV's and the imposed transportation requirements.

4.2.1 Background

The initial mission model developed for the electric versus chemical OTV trade consisted of the FOTV low model plus a demonstration phase of the solar power satellite and the space disposal of nuclear waste (SDNW). The resulting payload mass to GEO (delivery equivalent) during the 16 years was nearly 20 000t as compared with 2500t for the low model used in the space- versus ground-based OTV comparison. The SPS and SDNW payloads contributed nearly equally to the difference between the two models.

The actual comparison of the electric and chemical OTV's, however, took place nearly 6 months later. By this time, several events had occurred which made a reassessment of the model appropriate. First, both the SPS and SDNW were rather controversial and completely dominated the model. Selection of an OTV based on such conditions, therefore, seemed rather risky. Second, in the case of the SPS, in November 1980 the DOE elected not to provide any additional funding. A final curtailment of the SPS occurred in June 1981 when the National Academy of Sciences reported the program was not justified. In the case of SDNW, a dedicated study by MSFC/Boeing (Analysis of Space Systems Study for the Space Disposal of Nuclear Waste, NASA contract NAS8-33847) was in progress with one of the OTV options being an EOTV.

Consequently, with the above factors, deletion of the SPS and SDNW appeared justified. Development of a mission model sufficiently large to assess the benefit of more advanced OTV's concentrated, therefore, on expanding the number of missions in the areas of communication platforms, DOD payloads, science and observation platforms, and manned activity.

4.2.2 Mission Model

The resulting mission model is shown in table 4.2.2-1. In general, the model consists of additional payloads in the mission categories described in the SB versus GB OTV model, as well as new missions. New missions include a very large DOD payload (No. 25), lunar exploration payloads (Nos. 26, 27), larger science satellites (Nos. 28, 29), and a relatively small engineering verification test article (EVTA) associated with the SPS. The model also contains an additional manned GEO base (eight-man) to be used for construction of

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Table 4.2.2-1 FOTV High Model (16-Year) Mission-Imposed Transportation Requirements

NO.	NAME	QTY	MASS (t) EA.	LARGE DIMEN. PAYLOADS (m)	FOTV COMPAT.
1	COMMUN PLAT	13	6.8	130 x 60 x 16	✓
2	ADV COMMUN PLAT	6	31.8	130 x 60 x 15	✓
3	PERS COMM SAT	10	24.6	67 x 130	✓
4	SPACE BASED RADAR	3	11.4	190 x 300	✓
5	DOD CLASS	14	11.4	SOME	60%
6	DEEP SPACE RELAY	2	6.8	37 x 67	✓
7	SOLAR TERREST. OBSERV	1	11.0		✓
10	DOD CLASS 1A	32	2.7		NO
10A	DOD CLASS 1B	12	4.0		✓
11	DOD CLASS	10	5.5		NO
12	COMMER. & NASA	16	4.5		✓
13	GEO BASE MODULES	2	15 & 20		NO
14	GEO BASE EQUIP	7	9.0		NO
15	SAT. MAINT PROV.	23	2.0		✓
18	GEO MAINT. SORTIE	16	5.9/5.9 [2]		NO
19	BASE SUPP (CR/RS)	40	5.6/5.0 [2]		NO
		46	12.4/10.5 [2]		NO
21	SCIENCE SORTIES	3	8.1/8.1 [2]		NO
22	UNMANNED SERV.	183	4.8/0.9 [2]		✓
23	PLANETARY	12	5.0		NO
[1] 25	DOD CLASS	12	27.3	SOME	16%
[1] 26	UNMANNED LUNAR	2	2.0		NO
[1] 27	MANNED LUNAR SORTIE	4	12/7.5 [2]		NO
[1] 28	LG. SCIENCE SAT	2	25.0	100 x 50	✓
[1] 29	MED. SCIENCE SAT	3	15.0	50 x 25	✓
[1] 30	SPS EVTA	1	42.0	480x 80 x 20	✓
		Σ = 477	Σ = 4600 (GEO DELIVERY EQUIVALENT)		

[1] NEW MISSIONS RELATIVE TO FOTV LOW MODEL

[2] UP/DOWN

large payloads. Further discussion of the rationale for this base is presented in the next section. Related to the GEO construction base is the crew rotation/resupply mission involving eight men and constituting the largest round-trip mission.

The model includes 477 payloads resulting in a total GEO delivery equivalent mass of approximately 4600t (300 t/yr) which is nearly twice the size of the low model. Approximately 45% of this mass is related to round trip payloads. A comparison of the low and high models is presented in figure 4.2.2-1.

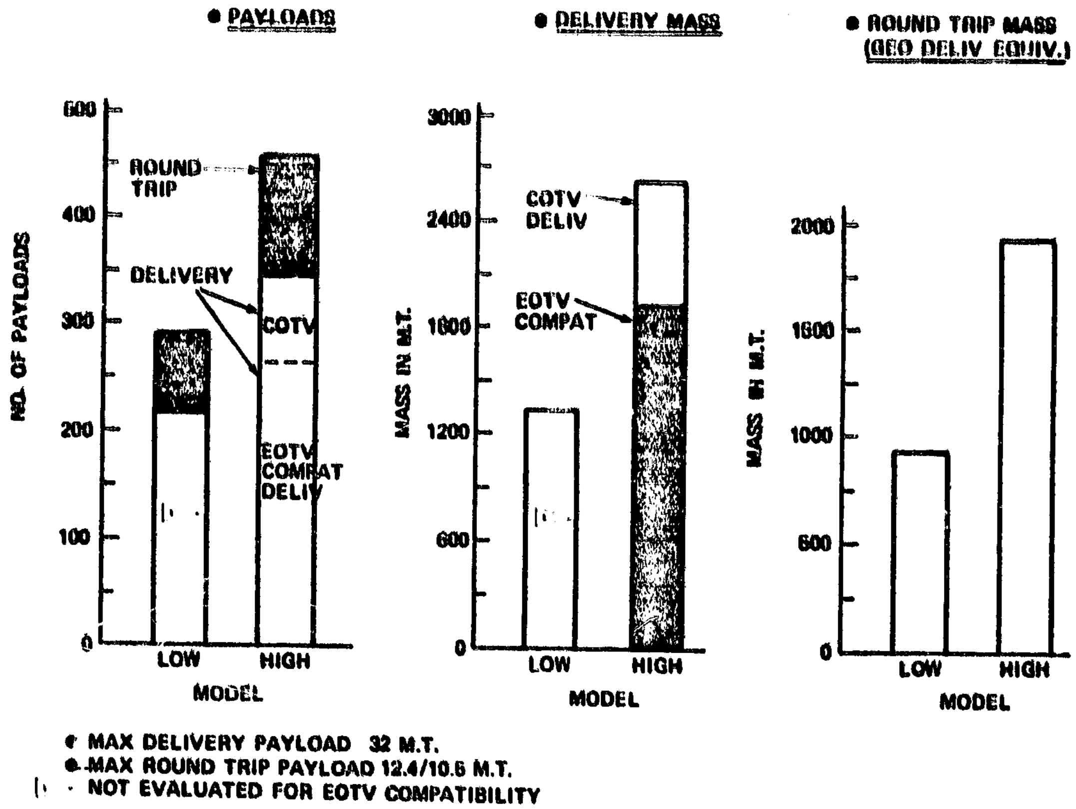


Figure 4.2.2-1 FOTV 16-Year Mission Model Summary

4.2.3 Payload Compatibility With EOTV's

A key issue associated with EOTV's is their compatibility with payloads (or vice versa). Those payloads judged to be compatible in terms of relatively long trip times were indicated in table 4.2.2-1. A total of 284 payloads were identified. In general, those judged incompatible due to trip times are the manned missions and some DOD missions. The compatible payloads result in a delivery mass of approximately 1900t or 40% of the total model mass, which indicates considerable need for a chemical OTV in order to satisfy the total mission model transportation requirement.

Another compatibility issue dealt with whether payloads requiring construction should be transported as finished systems from LEO or as components with construction occurring at GEO. A summary of this issue is presented in figure 4.2.2-2. In the model, 30 to 40 payloads are classified as very large in dimension (100m to 200m in length or diameter) when fully deployed with the majority of these requiring on-orbit construction. Several problems occur when construction of these payloads is done at a LEO SOC and then transported to GEO using the EOTV. First, the transfer of these payloads from SOC

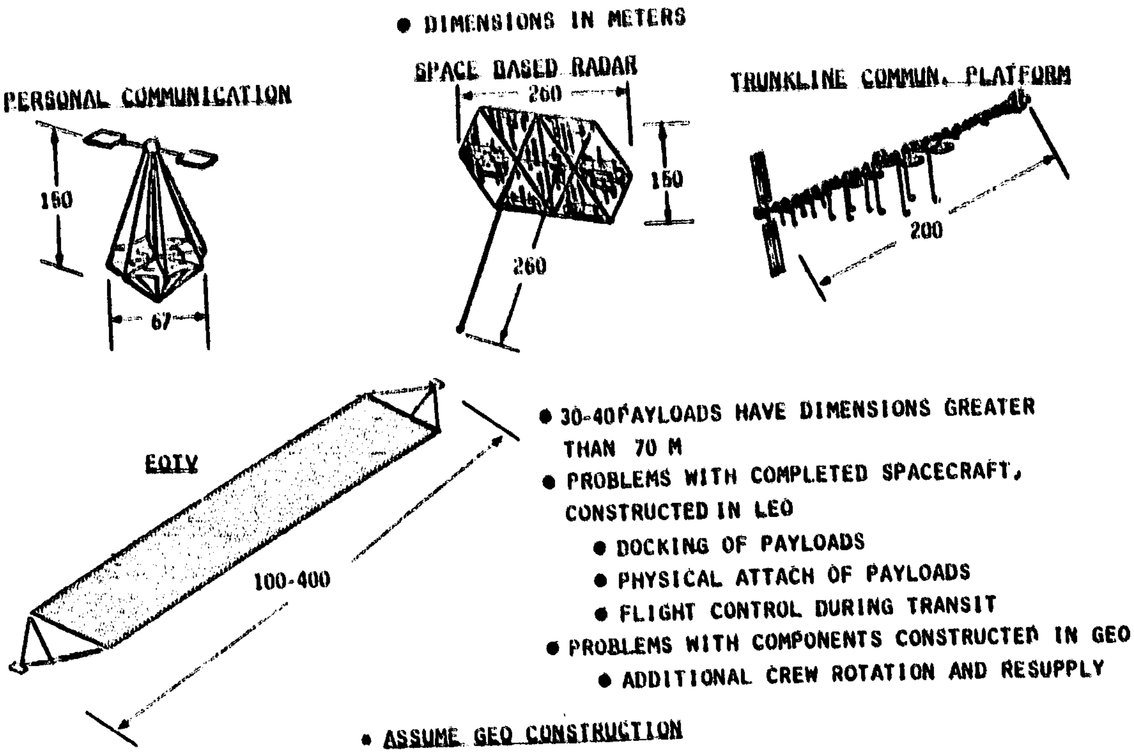


Figure 4.2.2-2 EOTV Transportation of Large Payloads as Completed Spacecraft or Components

to an EOTV, in terms of docking and attachment, present challenging problems due to the aerodynamic and gravity gradient disturbances (note: the EOTV stationkeeps near the SOC). Secondly, the wide range in payload configuration as well as dimensions would present an orbit transfer configuration that would be difficult in terms of flight control problems, particularly during the early part of the transfer when gravity gradient disturbances are the greatest. The recommended approach to overcoming these problems is to have a construction/final assembly base located at GEO. Although this approach solves the indicated problems, it does require more crew rotation/resupply transportation. A transportation mode to reduce this penalty involves GEO refueling of the chemical OTV with propellant delivered by the EOTV. A brief assessment of this mode indicated it was not cost effective for this particular mission model. The GEO refueling mode is discussed further in section 4.4.5.

4.3 NORMAL GROWTH TECHNOLOGY VEHICLES

This section provides the definition of the best possible EOTV using the assumed normal growth technology, summarizes the space-based LO_2/LH_2 OTV, and compares the vehicles in terms of total OTV fleet considerations.

4.3.1 Electric OTV Definition

4.3.1.1 EOTV Concept

To establish the framework for the definition of an EOTV, a brief description is provided in terms of its key subsystems and its unique flight operations.

The major system elements associated with an EOTV and their relationships are shown in figure 4.3.1.1-1. The power generation system for this vehicle becomes a major

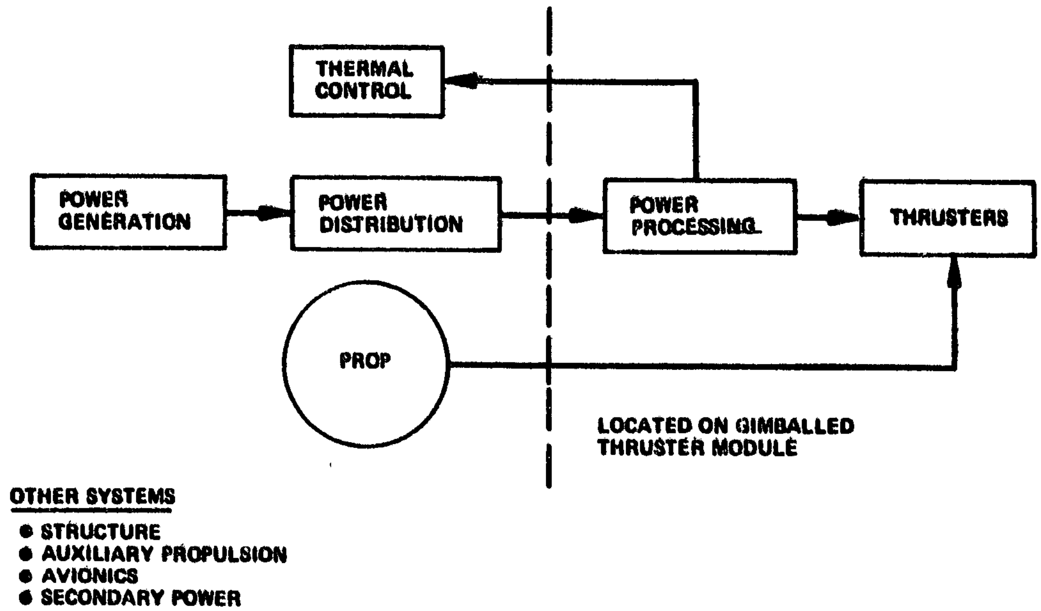


Figure 4.3.1.1-1 EOTV System Elements

driver both in terms of performance and cost. As discussed in the guidelines, the analysis was restricted to photovoltaics. Power collection and distribution have a unique challenge in selecting the optimum voltage. From an I^2R standpoint, a high voltage is advantageous; however, this presents problems in terms of plasma losses while in low Earth orbits (below 1000 km). Power processing becomes a major factor due to the differences in the optimum collected voltage and the voltages required by the thrusters. A thermal control system is required to handle the waste heat associated with power processing. A monopropellant is used in the auxiliary propulsion system. Other subsystems such as structure, auxiliary propulsion, avionics, and secondary power are also required but generally do not play a major role in optimizing the EOTV.

The overall operational features of an EOTV performing a LEO-GEO cargo delivery are shown in figure 4.3.1.1-2. Because of its anticipated size, the EOTV will be based in LEO near the SOC rather than attached to it. The transfer consists of a spiral trajectory, typically involving up to 1000 revolutions for transit times of 180 days. While in the

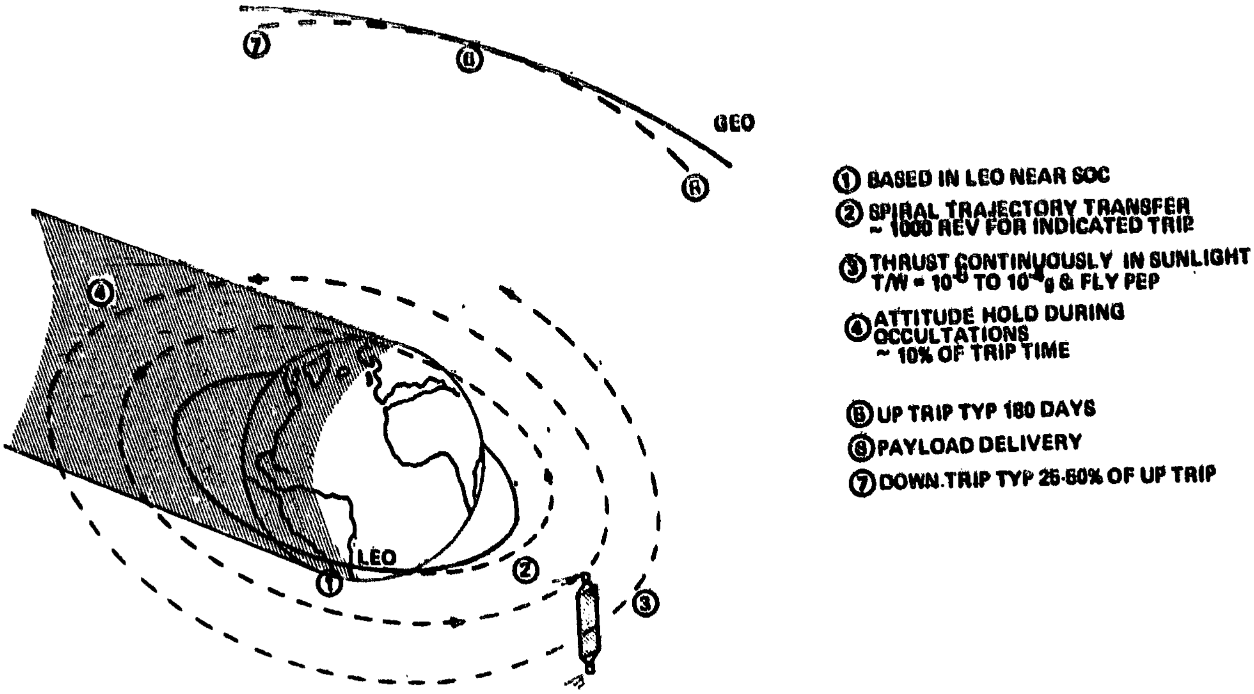


Figure 4.3.1.1-2 Electric Orbit Transfer Vehicle Operational Concept

sunlight, the array remains pointed toward the Sun so that thrust is continually provided. Because of the Earth's shadow, however, occultations occur during each orbit until a relatively high altitude is reached. During occultations, attitude is held but no orbit-raising propulsion is applied. Return flights to LEO include the same operations; however, the downtime usually is only 25% to 50% of the up time because the payload is no longer present but essentially the same amount of power and propulsion is available (except for that lost from radiation degradation).

The subsystem and operational features are discussed in greater detail in subsequent sections.

4.3.1.2 Technology Projections

As defined in section 3.3.2, normal growth technology is defined to mean funds are either being expended or are planned to bring the technical risk down to a reasonable level for initiation of design, development, test, and evaluation by the 1990 readiness date. The technology projections for the EOTV are presented in tables 4.3.1.2-1 and -2, along with the characteristics of the SEPS vehicle which is assumed to be the first-generation EOTV-type system. Further characterization of all EOTV subsystems is presented in section 4.3.1.4.

Table 4.3.1.2-1 EOTV Solar Array Normal Growth Technology Projection

AREA	1980 (SEPS TYPE)	1990 EOTV	BENEFIT
● SOLAR ARRAY			
● CELL -- TYPE	SILICON	SILICON	LESS AREA
-- EFF(%)	~ 13	16	
SIZE (CM)	2 x 4	5 x 5	LOWER ASSY COST
THICK (μm)	200	50	LESS DEGRADATION & WT.
● COVER MAT'L	FUSED SILICA	CERIUM DOPED MICROSHEET	CHEAPER
● SUBSTRATE MAT'L	KAPTON	MICROSHEET	CHEAPER
● BLANKET THICK (MIL) (COVER-CELL-SUBST)	200-150-50	75-50-50	
● BOL POWER (W/M^2)	175	216	
● SPEC. MASS (KG/KW)	8.7	2.4	
● STRUCTURAL DEPLOYMENT & SUPPORT			
● TYPE	● MOTOR DRIVEN MAST	● SPACE FAB. TRI-BEAMS ● TRAMPOLINE SUSPENSION	
● SPEC. MASS (KG/KW)	3.1	0.60	

- DOES NOT INCLUDE ANY LOSSES

Table 4.3.1.2-2 EOTV Electric Propulsion Normal Growth Technology Projection

AREA	1980 (SEPS)	1990 EOTV	1990 EOTV
● THRUSTER			
● TYPE	ION	ION	ARC JET
● DIA (CM)	30	50	TBD
● PROP	MERCURY	ARGON	HYDROGEN
● ISP (SEC)	3000	5-10,000	900
● EFF (%)	72	63-82	90
● SPEC. MASS (KG/KW)	3	1	0.5-1.0
● POWER PROCESSING			
● EFF. (%)	87-90	92	93
● POWER SUPPLIES PER THRUSTER	12	5	1
● PPU'S PER THRUSTER	1	1	1
● SPEC. MASS (KG/KW)	13	3.1	1.8
● PPU THERMAL CONTROL			
● RADIATOR TYPE	HEAT PIPE	ACTIVE	ACTIVE
● SPEC. MASS (KG/KW)	15	8	8

Solar Array—The solar array for the EOTV consists of the cell, cover, and substrate. The technology projections for the solar array are presented in table 4.3.1.2-1. Silicon cells are suggested as the only candidate for normal growth. Although GaAs cells are receiving considerable emphasis (including funding), a thin cell design desired by an EOTV still represents considerable challenge. Consequently, the GaAs cell has been placed in the category of accelerated technology which is analyzed in section 4.4. Insufficient data, particularly in terms of radiation sensitivity and cost, prevented consideration of other advanced cells.

The 16%-efficient silicon cell is that which is assumed to be the overall average of very large production quantities rather than that obtained under laboratory conditions (which could be 1% to 2% higher). Cell thickness of 50 μm should be commonplace and reduce mass as well as sensitivity to radiation. Improvements are also envisioned in cell size which benefits assembly cost and in thickness which reduces radiation degradation and weight. Use of cerium-doped microsheet coverglass eliminates the need for an ultraviolet filter and also is cheaper. Microsheet can also be used for the substrate which contributes to a less costly array. The specific mass of the 1990 array is only 35% that of the 1980 array due to differences in thickness of the cover-cell-substrate.

Structure (Solar Array Support)—Improvements in the EOTV structure are also presented in table 4.3.1.2-1. Assuming a SOC-type space base is available, consideration can be given to lightweight space-fabricated composite tribeams. When assembled, the beams form the framework to support the array through the use of a trampoline suspension system. The specific mass of the 1990 structural system based on SPS-type design criteria is only 20% of the 1980 system.

Thrusters—Electric propulsion thrusters in this study have been confined to ion and arc-jet systems. The projections for these thrusters are presented in table 4.3.1.2-2. Through mutual agreement between the study manager and the NASA COR, magnetoplasmadynamic (MPD) thrusters were not considered since they were receiving special emphasis in the advanced propulsion concepts study (ref. 14).

The projected ion thruster characteristics are indicative of those resulting from studies being conducted by Hughes Research Labs (HRL) and XEOS for NASA LeRC. These studies have baselined a 50-cm-diameter thruster which uses argon propellant. The diameter may have been limited by available screen grid material width. Argon is selected to eliminate environmental objections associated with mercury when used for LEO-GEO application. The 50-cm thrusters also use multipole containment schemes rather than the divergent beam used for the SEPS thruster. This characteristic improves beam flatness which results in higher efficiency and high allowable beam current density at any selected lifetime. Performance and design life parameters for the thruster are presented as part of the EOTV point design characterization in section 4.3.1.4.

Although no current U.S. funding is being applied to develop thermal arc jets, a variety of concepts were designed and tested in the 1960's. Efficiencies as low as 0.3 were common and, consequently, research was halted. The work, however, did confirm analytical design procedures for arc heating and nonequilibrium expansion. More recently, Dr. Rolf Buhler of the University of Stuttgart has been investigating hydrogen arc jets and has developed an idea to improve the performance of this thruster. The concept consists of adding a mixing chamber downstream of the arc chamber to homogenize the propellant which is subsequently expanded in a conventional nozzle. In small sizes, the resulting total temperature will be limited by allowable structural temperatures and/or chemical reaction rates with the hydrogen. Based on chamber temperatures of 4500°R, an Isp of 900 sec is postulated.

Power Processing Unit (PPU)—Improvements in the area of PPU's are also shown in table 4.3.1.2-2. The ion jet PPU improvement in specific mass (3.1 versus 13) is primarily the result of reducing the number of power supplies by combining functions. Efficiencies as high as 92% can be expected.

Arc-jet PPU's are expected to have slightly higher efficiency and lower specific mass than the ion thruster PPU because the arc jet requires only a single voltage, typically as low as 100V.

PPU Thermal Control—Use of an active (pumped-fluid) radiator rather than heat pipe radiator is expected to reduce the specific mass of this system to 8 kg/kW from 15 kg/kW.

4.3.1.3 Sizing Considerations

The sizing of an EOTV which uses photovoltaics and low-g transfer must take into account several considerations which are unique as compared with a typical chemical OTV. These include variable trip time and Isp, flight (trajectory) parameters, array radiation degradation, array sizing philosophy in terms of beginning versus end of life, payload size, and design life. A discussion of each of these factors follows.

Trip Time and Specific Impulse—As discussed earlier, both trip time and specific impulse directly affect the amount of power required which strongly influences the system cost. The combination of trip time and Isp which gives the least cost is discussed in detail in section 4.3.1.5.

Flight Parameters—Flight parameters which influence EOTV sizing include the transfer delta-V, flight profile, and flight control.

Delta-V: The most significant factor affecting the orbit transfer delta-V requirements for an EOTV is the g losses when operating typically at 10^{-3} to 10^{-4} m/sec^2. This factor, in conjunction with the plane change from 28.5 to 0-deg inclination, results in an ideal delta-V of 5750 m/sec as compared with 4300 m/sec for a 0.2g chemical OTV. Losses associated with gravity gradient effects (2%) and flight performance reserves (2%) result in a one-way delta-V of approximately 6000 m/sec.

Flight Profile: Flight profile characteristics in terms of the relationships between orbit plane, altitude, and elapsed time for a typical orbit transfer are shown in figure 4.3.1.3-1. A significant point that can be seen from these data is that with a low-g

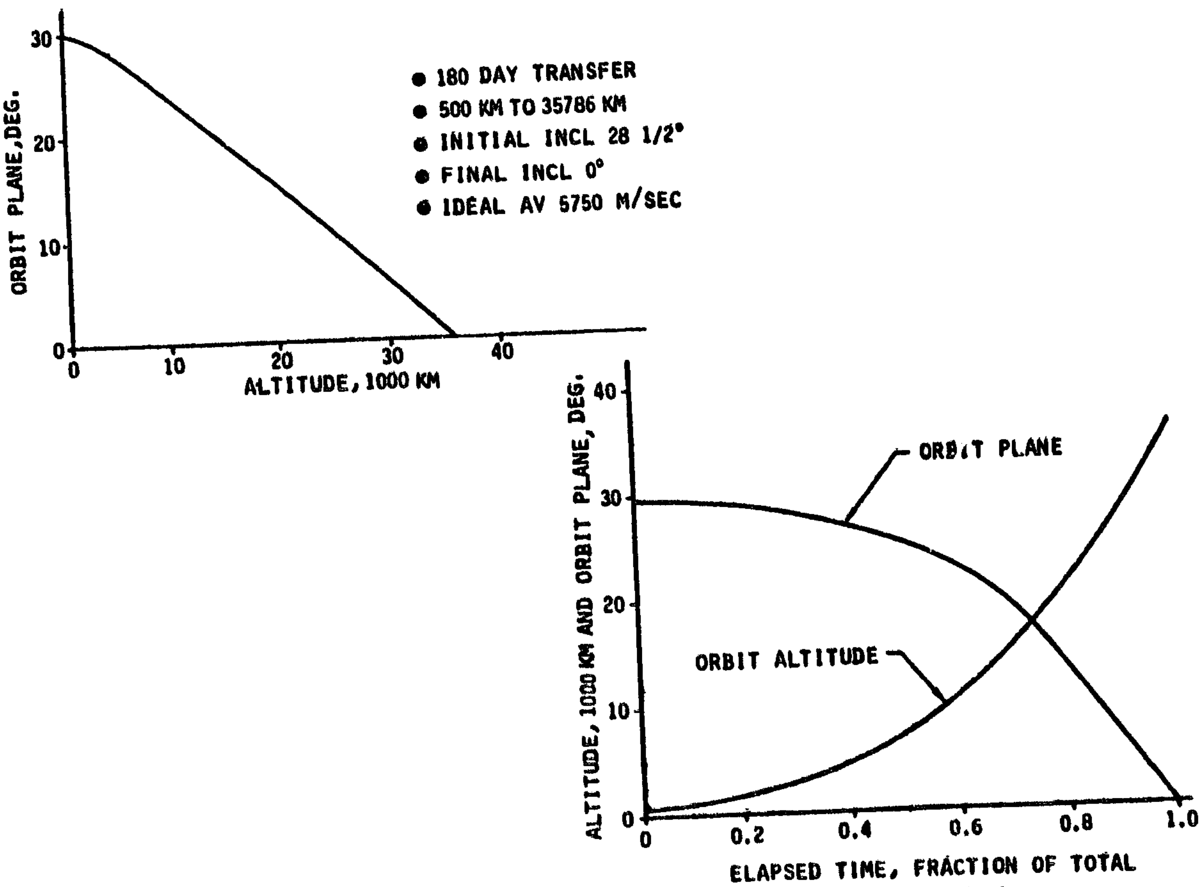


Figure 4.3.1.3-1 Low-Thrust Orbit Transfer Characteristics

transfer, 60% of the time is spent below 10 000 km, which includes the most damaging regions of the Van Allen radiation belts.

The relatively slow rate of increasing altitude also means a large number of revolutions are involved. Each of the revolutions contains an occultation or shadow period when the vehicle passes on the backside of the Earth and out of sunlight. The number of occultations that can be expected as a function of transfer time is presented in figure 4.3.1.3-2. The band indicated illustrates the range in number of occultations depending on

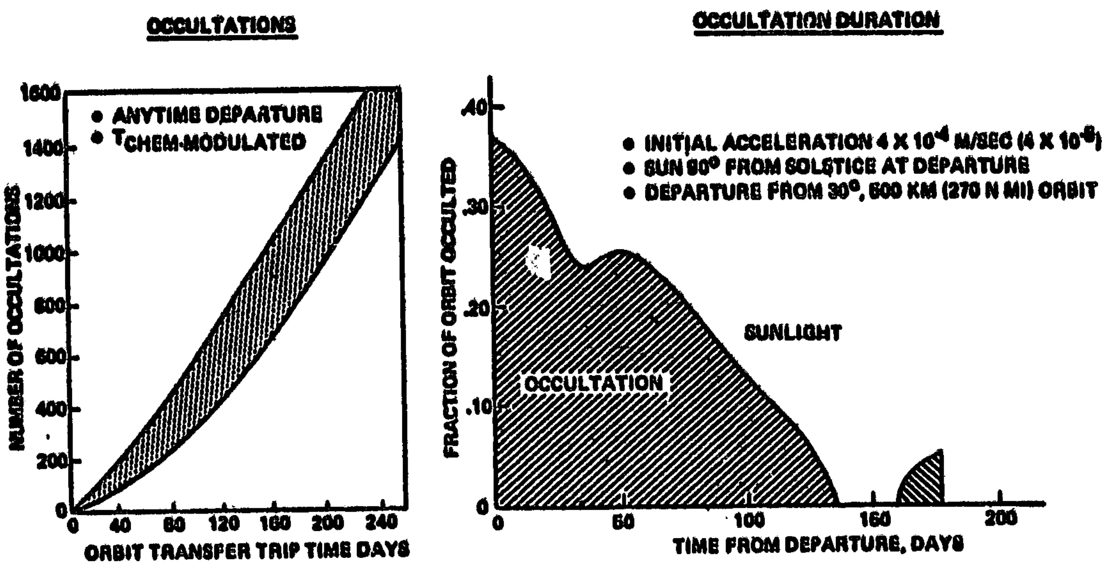


Figure 4.3.1.3-2 Orbit Transfer Occultations

whether the transfer is initiated at the best or worst time of the year relative to the orbit and Sun position. Therefore, for typical transfer times of 180 days, as many as 1000 occultations can be expected. The impact of this factor is that it identifies the number of starts and stops which must be experienced by the system as well as providing an input to the fraction of time a vehicle is occulted and cannot generate power for propulsion purposes. The fraction of time as a function of time from departure is also shown in figure 4.3.1.3-2. The decrease with time is the result of the orbit getting larger and the shadow zone staying constant. The average for the complete orbit transfer with departure at solstice is approximately 16%; with equinox departures, approximately 10%. The yearly average due to occultation is, therefore, judged to be approximately 13%. An additional nonthrusting time increment relates to each startup of the system. In this case, there is a brief period to stabilize the high voltage of the array immediately after breaking out into the sunlight as well as the small increment of time to start the thrusters. A nonthrusting time of 2% is assumed for startup. Therefore, the total nonthrusting time period for a transfer is 15%. The impact of this is that a higher acceleration level is required for a given transfer time.

Flight Control: The flight control task associated with the transfer of an EOTV from LEO to GEO involves directing the thrust vector in a manner to change the plane of the orbit and raise the altitude while maintaining the attitude of the satellite so that electric power can be generated for the thrusters. During the transfer, the solar arrays are always directed toward the Sun as shown in figure 4.3.1.3-3 to eliminate incident

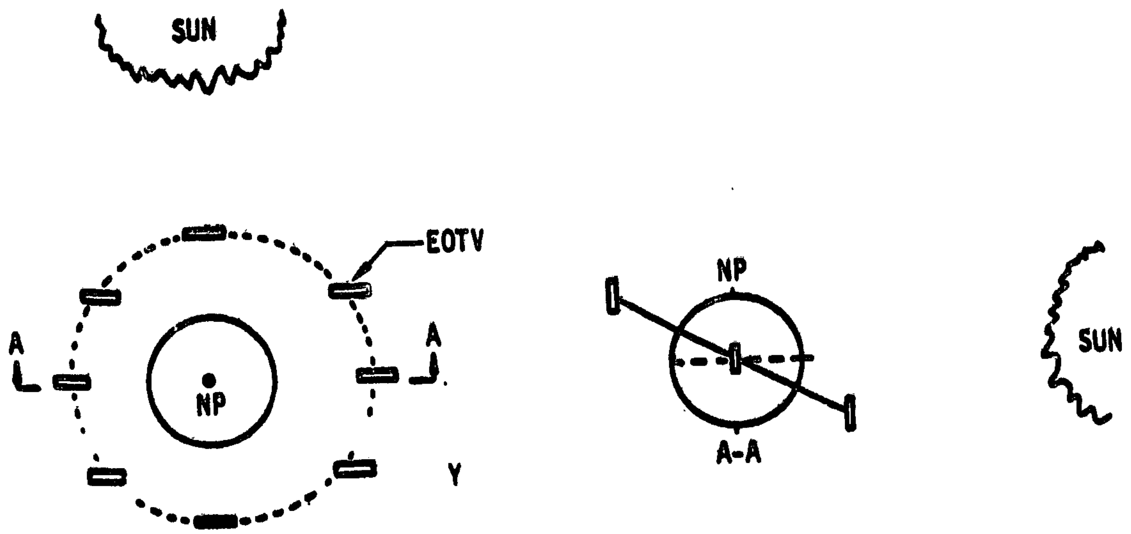


Figure 4.3.1.3-3 Flight Control Orientation

angle losses. This attitude, however, results in a disturbance in the form of gravity gradient torque. A simplified illustration of the torque characteristics is shown in figure 4.3.1.3-4. The largest disturbance will occur when the vehicle is nearest the Earth (diminishes with the cube of the radius from the Earth's center) and with its principal axis of inertia at 45 deg to nadir.

The impact of the flight control consideration is to configure the EOTV in a manner that maximizes the moment of inertia differences between axes and to install the thrusters to best counter the torque and provide climbout thrust.

Solar Array Radiation Degradation—Perhaps the most dominating factor in the sizing of a photovoltaic EOTV is that of the power degradation of the array as a result of radiation exposure. The following paragraphs discuss the environment and degradation prediction.

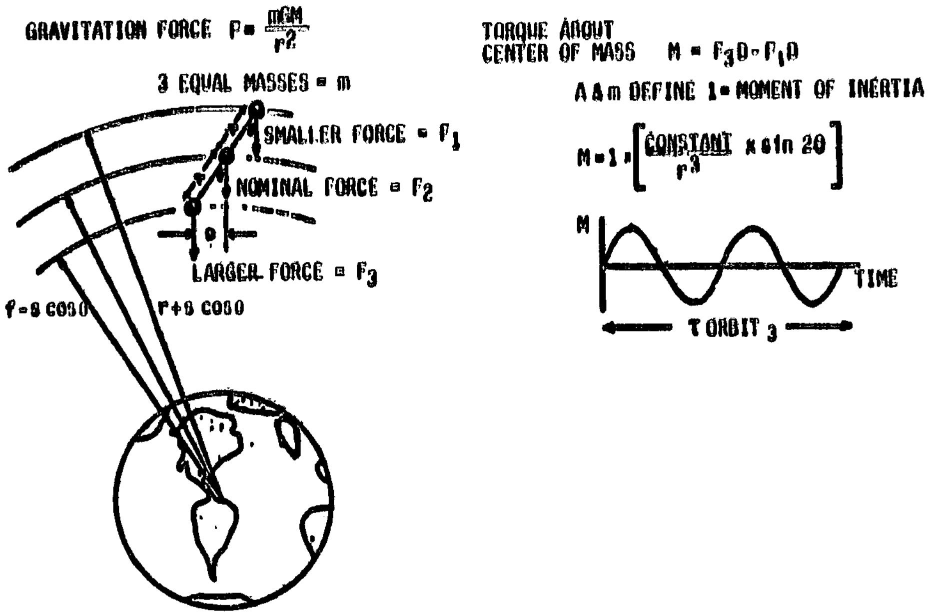


Figure 4.3.1.3-4 Gravity Gradient Torques

Environment: During transfer from LEO to GEO, the EOTV will be exposed to the most intense regions of the Earth's trapped radiation belts. The trapped protons are described by flux map AP-8 (ref. 10) while the trapped electrons are described by AE-6 and AE-4 (ref. 11 and 12). The actual environment experienced by the EOTV varies with each orbit. The total fluence is determined by integration of the flux maps and the vehicle flight profile. The flux of protons and electrons as a function of altitude and inclination is shown in figure 4.3.1.3-5 taken from reference 13. The maximum flux and peak degradation occurs at approximately 6000 km. Although there are a greater number of electrons, protons actually are the dominating factor in cell damage since they have a much higher displacement damage cross section in silicon. The energy spectra of the protons and electrons encountered in a typical transfer are shown in figure 4.3.1.3-6.

The actual environment experienced by the solar cell is reduced by the shielding provided by the coverglass and substrate. The effect of several shielding densities is shown in figure 4.3.1.3-7 also taken from reference 13. These data indicate for shielding as heavy as 0.07 gm/cm^2 (12 mils), the penetrating protons will have a peak flux at energies less than 2 MeV. Particles of these energies are very damaging to the cell because they stop shortly after entering and produce heavy localized damage near the end of their tracks.

Degradation Prediction: The prediction of the solar cell degradation involved two basic steps. First, cell performance as a function of 1-MeV electron fluence was

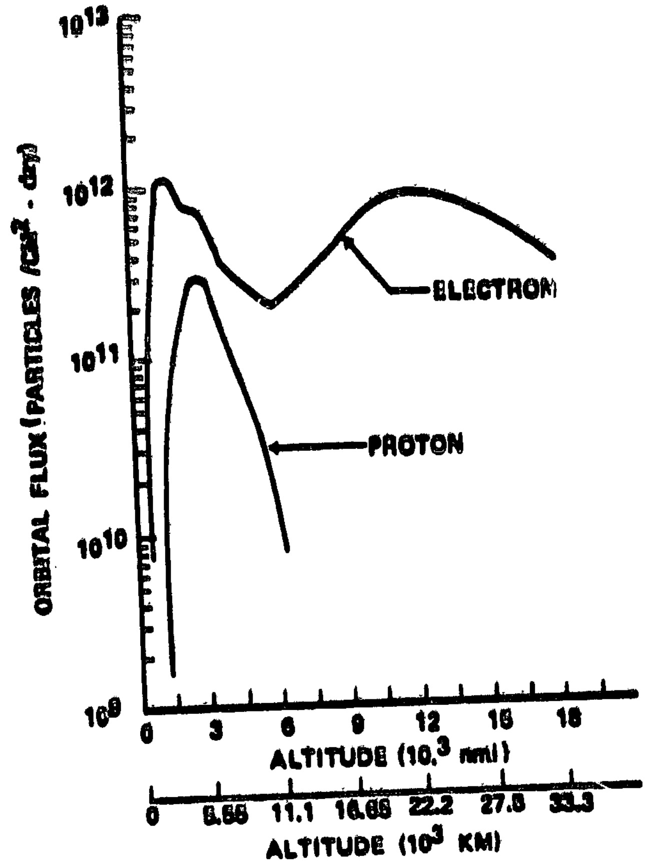
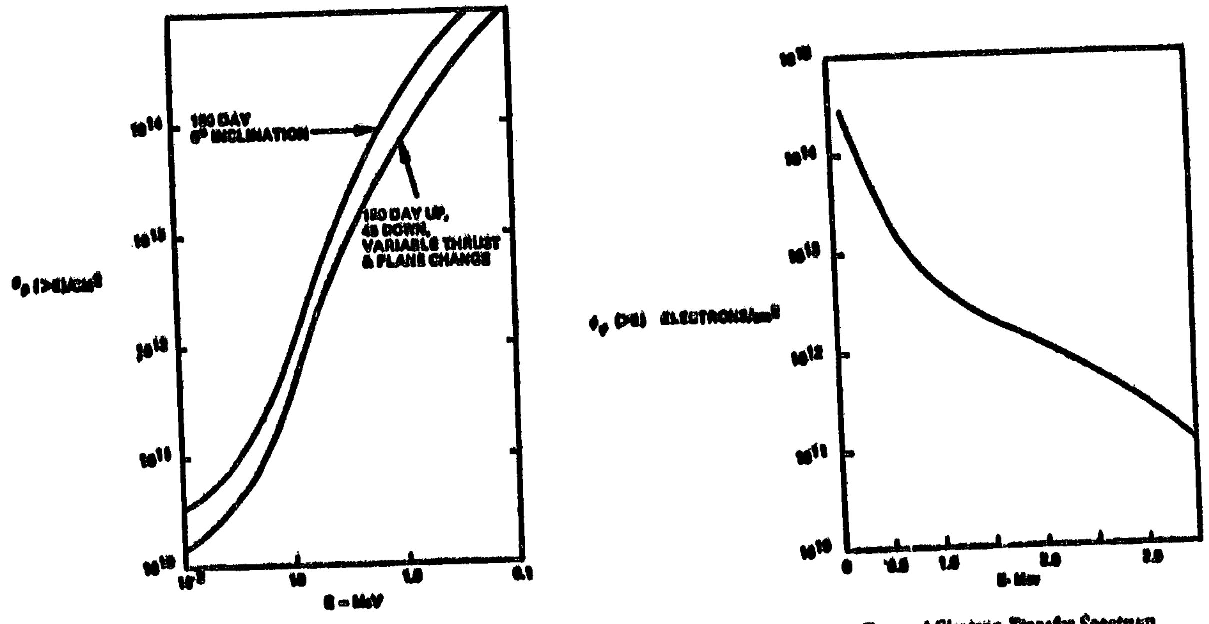


Figure 4.3.1.3-5 Radiation Flux Versus Altitude



Trapped Proton Transfer Spectrum

Trapped Electron Transfer Spectrum

Figure 4.3.1.3-6 Energy Spectra

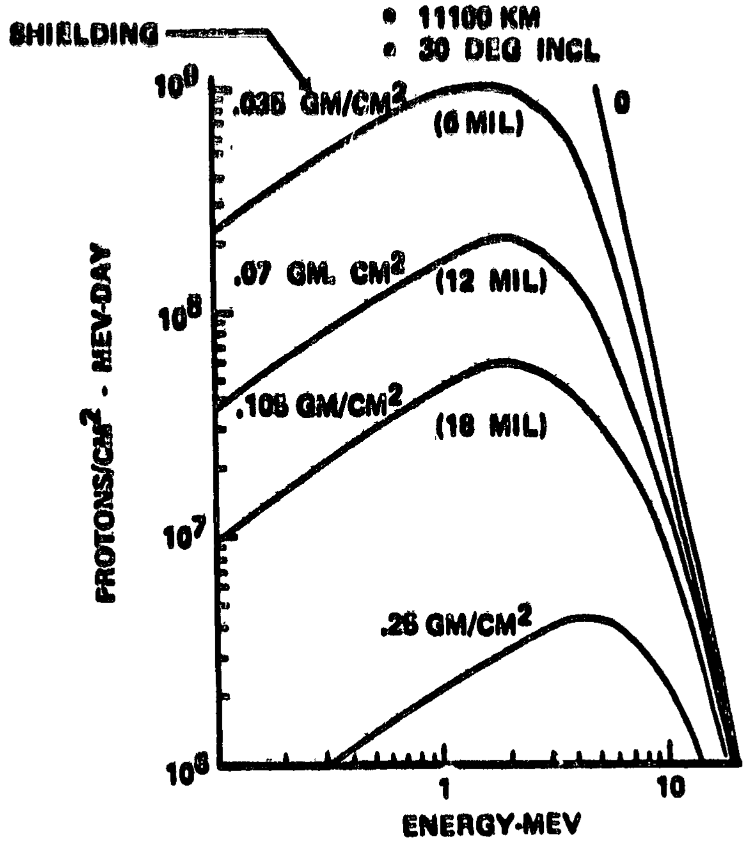


Figure 4.3.1.3-7 Modified Spectrum Inside Cell

established by surveying the literature and consulting cell manufacturers. The second step required converting the expected fluence of protons and electrons penetrating a given shield into 1-MeV electron equivalence.

Cell performance data were obtained from a study (ref. 14) being conducted by Boeing for the Air Force concerning radiation-hardened solar cells. The best performing planar silicon cell was one produced by Spectrolab, called HESP II, with the following characteristics: 50-μm (2 mil) thick, n/p, BSR, 16%. A 50-μm cell developed by Solarex offered slightly higher performance at the maximum test fluence of 10^{16} (1-MeV equivalence) but not as good at extrapolations beyond 5×10^{16}.

The conversion of the protons and electrons received by a cell to 1-MeV equivalence assumed no combined effects as discussed in reference 15. When using a 75-50-50 μm (3-2-2 mil) blanket, one round trip involving 180 days up and 45 days down results in a cell dose of 1.07×10^{17} (1-MeV equivalence).

The resulting performance of the 75-50-50 μm solar array when used for power generation between LEO and GEO is shown in figure 4.3.1.3-8. Several key observations should be noted. First, the fluence of one round trip is nearly 1000 times more severe

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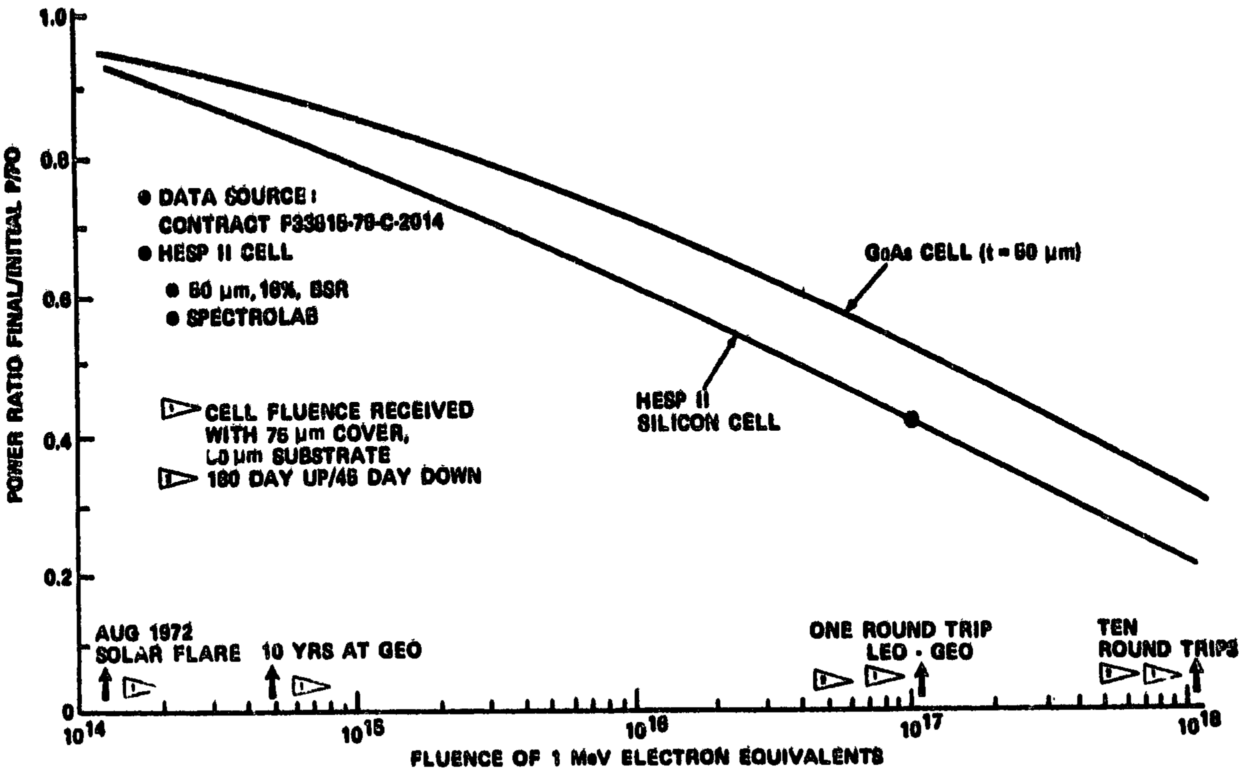

Figure 4.3.1.3-8 EOTV Design Driver—Van Allen Radiation Impact

than the largest recorded solar flare and 500 times larger than the design environment for satellites operating 10 years at GEO. In terms of performance, the round trip results in a final power output only 42% of the initial output, or a degradation of 58% for the first trip. Should 10 round trips be flown, the output is only 22% of the initial, indicating a 78% degradation. The impact of the degradation can be that of oversizing in order to have a fixed amount of power available for the last trip. This is discussed further in this section under the heading of "Array Sizing Philosophy." The radiation sensitivity of a GaAs cell is also presented in figure 4.3.1.3-8. Further discussion of this cell is found in section 4.4.1.1.

The impact of the radiation degradation for the reference cell with no annealing can be altered by different trip times and shielding thickness as indicated in the left-hand plot of figure 4.3.1.3-9. Reducing the trip time reduces the degradation but it also requires more propulsion power. Increasing the shielding significantly improves the power ratio; however, it also means more dry mass requiring more propulsion power. Less degradation would also occur if the EOTV flight was essentially begun at a higher altitude in a manner similar to the proposed SEPS mission profile. For example, as indicated in the right-hand plot of figure 4.3.1.3-9, if operation is initiated at 10,000 km, 90% of the fluence has been

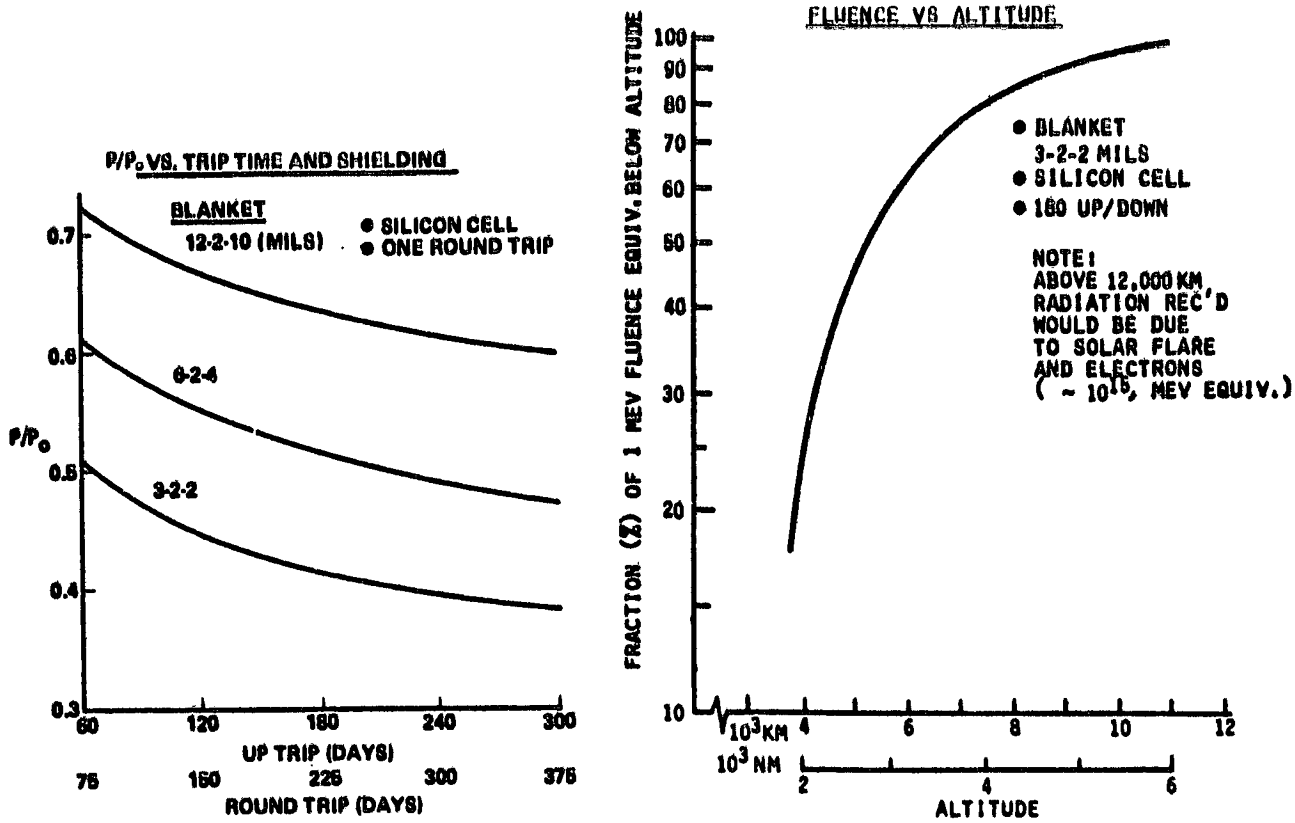

Figure 4.3.1.3-9 Radiation Degradation Sensitivity

bypassed, meaning a round trip would have a fluence of approximately 10^{16} (1 MeV) and a P/Po of 60%. The significance of the above variations is discussed in section 4.3.1.4.

Design Life—Design life is defined as the number of flights that can be made by a given EOTV. More flights, of course, means fewer units and this becomes a major factor when the units are expensive.

The principal concern regarding design life is the impact of radiation. Once again the component of major concern is that of the solar arrays. A comparison of power output versus number of round trips is shown in figure 4.3.1.3-10. These data indicate that by the time the 10th flight has occurred (for the 75-50-50 μm array), the rate of P/Po reduction is not too significant and additional flights could be performed without excessive penalty. A design life of 10 flights, however, although not long by reusable chemical OTV standards, does result in a significant amount of time in total operation. This occurs as a result of each trip (typically) requiring a total operating time of 225 days—thus, for 10 flights, a total of over 6 years. GEO communication satellites are currently being designed for 7-10 years of life; however, once again, the environment at

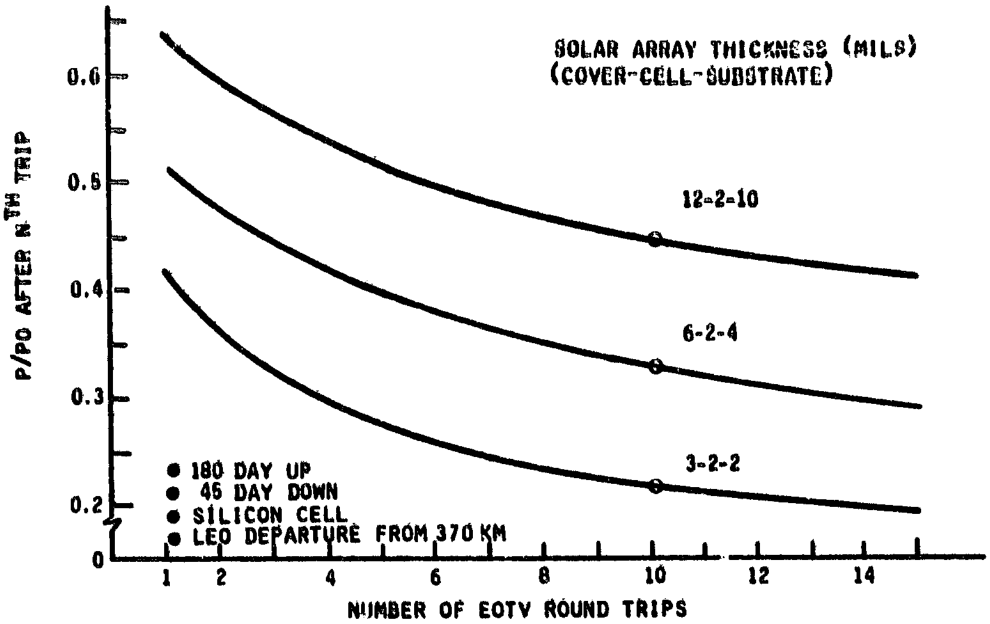

Figure 4.3.1.3-10 Power Output Sensitivity to Number of Trips

GEO is not nearly as severe as that associated with LEO-GEO transfer. Another factor to be considered is that at the 10-flight point, the power output of the 25-50-50 μm array is approximately 20% of the initial output which means an oversizing of the array of approximately five times. In addition to the power decay, there is also a decay in the voltage produced. Data developed during the SPS studies indicated that, at the indicated degradation levels and with continued operation at the maximum power point, the voltage would be reduced 25% of the power reduction. Therefore, if the power was reduced by 80%, the voltage would be reduced by 20%. The impact of this situation could be a more complex power processing system to handle the variations in voltage. Other concerns associated with multiuses of a solar array in this application are the cell-to-cell mismatch and the number of thermal cycles. Cell-to-cell mismatch can occur because each cell will be affected slightly differently in terms of radiation, thereby resulting in further degradation of the overall power output. Thermal cycles become a consideration as a result of the occultations which occur during the orbit transfer. In this case, one typical EOTV transfer has more occultations (thermal cycles) than a GEO satellite experiences in 15 years.

Other components potentially affected by radiation include structure and solid-state electronics. Composite-type structure is suggested for the EOTV. There is indication, however, that this material exhibits outgassing characteristics when exposed to radiation doses on the order of 10^9 rads. One round trip of the EOTV is expected to give a dose

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level of this magnitude at a depth of 0.0025 mm. The resulting outgassing could consist of contaminants which would affect the performance of the solar array. Solid-state electronics generally can accept radiation doses as high as 10^4 rads. Typical structural enclosures provide 0.25 to 0.375 cm of aluminum which would result in a received dose of 10^4 rads on the electronics. Ten flights would mean 10^5 rads and the need for radiation-hardened electronics or considerably more shielding (although, in some applications, this presents physical integration problems).

Array Sizing Philosophy: BOL Versus EOL—Two extremes can be considered in the sizing philosophy for the solar array: (1) design for beginning of life (BOL) or first flight and (2) design for end of life (EOL) or last trip (assumed to be 10 trips). The EOL approach has been selected since it appears to require about the same amount of solar array and provides better operational features, particularly trip time per flight. The overall features of the two approaches are shown in figure 4.3.1.3-11 and described in the following paragraphs.

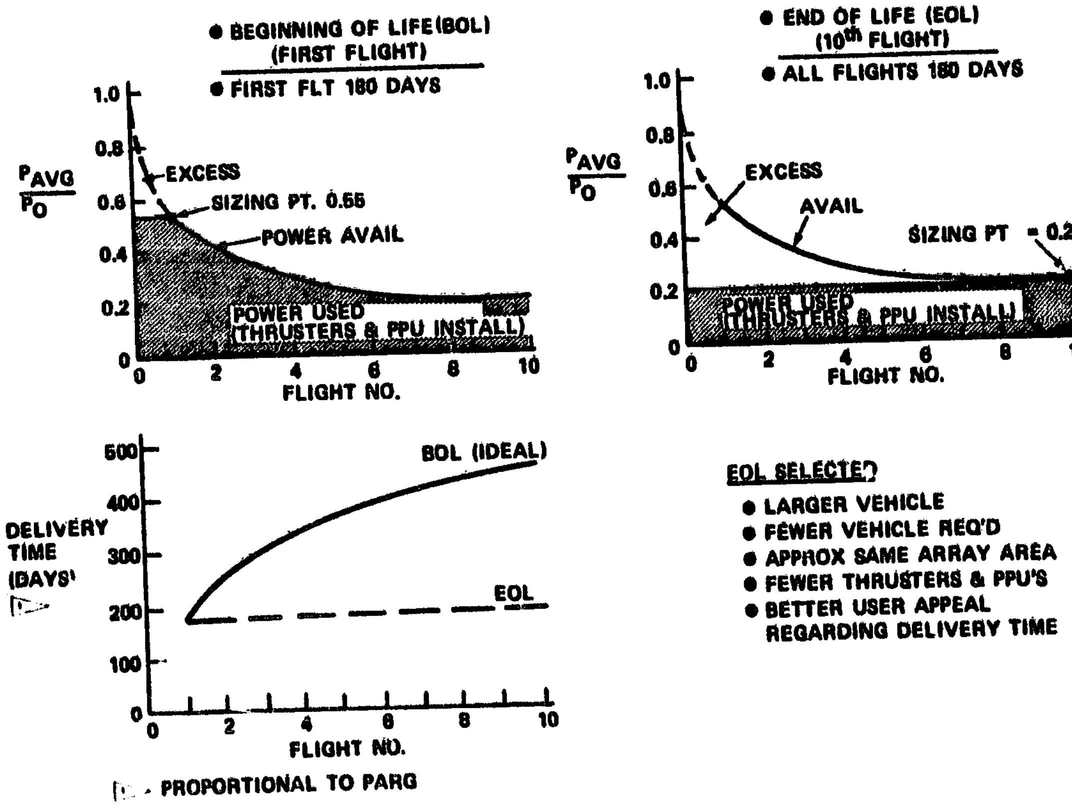


Figure 4.3.1.3-11 BOL Versus EOL Vehicle Solar Array Sizing

When designing for the first flight, the average power available is approximately 55% of the initial, meaning the array must be oversized by nearly a factor of 2. Designing for the 10th flight or EOL by definition incorporates the total expected degradation and consequently shows an average power of only 22% of the initial power. This means the array must be oversized by nearly a factor of 5. The BOL option, therefore, provides a smaller vehicle. Trip times for the BOL option get longer, however, due to the P/Po decreasing with each trip. For comparison purposes, the BOL option on the 10th flight requires approximately 760 days for a round trip (600 up/160 down) as compared with 225 days (180 up/45 down) for the EOL design option. The longer trip times mean more vehicles (12 versus 4) are required to fly the assumed six flights per year. Consequently, the total solar array requirements are about the same when considering the size and numbers of vehicles.

The long trip times of the BOL option would also be a major impact on the payload owner. At this point in time it is difficult to imagine many revenue-producing payload owners who would be content with 1- to 1-1/2-year delivery times. It should also be mentioned that the actual trip times would be longer because the radiation dose received was assumed to be that related to 180 days up and 45 days down. In reality, however, each trip gets longer because more radiation is received which means even more degradation per flight and, consequently, slower trips.

A final disadvantage of the BOL option is that more electric propulsion hardware is required because thrusters and PPU's are matched to the initial power available but, unfortunately, are of no significant benefit as the array degrades and less power is available for propulsion.

Payload Size—An EOTV sized for one payload versus another generally does not show any appreciable difference in effectiveness. If sized for one payload and then the same vehicle is used to transport a different size (mass) payload, a variation will occur in trip time since EOTV's are power-limited devices. Key considerations in selecting an EOTV relative to payload size are: (1) a size that minimizes the number of payloads taken up on a given flight (dedicated flights are optimum but not practical), (2) a size that minimizes the number of flights required to get the components of the largest constructible payloads to orbit, (3) a physical size that does not dictate the size or design of the construction base, and (4) a size that offers reasonable compatibility in launch vehicle manifesting so payloads do not have to wait at LEO for long time periods before being transferred to GEO. After reviewing the mission model and preliminary vehicle sizing data, a payload mass of 25t was judged to adequately satisfy the above considerations.

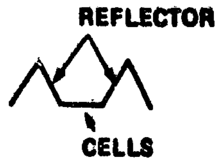
4.3.1.4 System Design Options and Characterization

The EOTV options considered for analysis are generally related to concepts which would reduce solar array degradation and/or the amount of power required. This motivation was prompted by the desire to minimize the array oversizing and the recognition that the solar array will most likely be the most costly component in the vehicle.

A listing of the options and their basic features is presented in table 4.3.1.4-1. A

Table 4.3.1.4-1 EOTV Normal Growth Design Options

- **POINT DESIGN — · · · KEY PARAMETERS**
  - **BLANKET THICKNESS (MILS) — · · · 3-2-2 (COVER · CELL · SUBSTRATE)**
  - **TRANSFER MODE · · · SELF POWER**
  - **CONCENTRATION RATIO (CR) · · · 1**
  - **THRUSTER · · · ION**
- **OPTIONS**
  1. **INCREASE CELL SHIELDING ▷ OTHER PARAMETERS SAME AS POINT DESIGN**
     - **UP TO 12 MILS (12-2-10)**
  2. **CHEMICAL ASSIST TRANSFER MODE ▷**
     - **DELIVERY AND RETURN**
     - **DELIVERY ONLY**
  3. **MORE POWER OUTPUT PER UNIT AREA ▷**
     - **CONCENTRATION RATIO = 2**



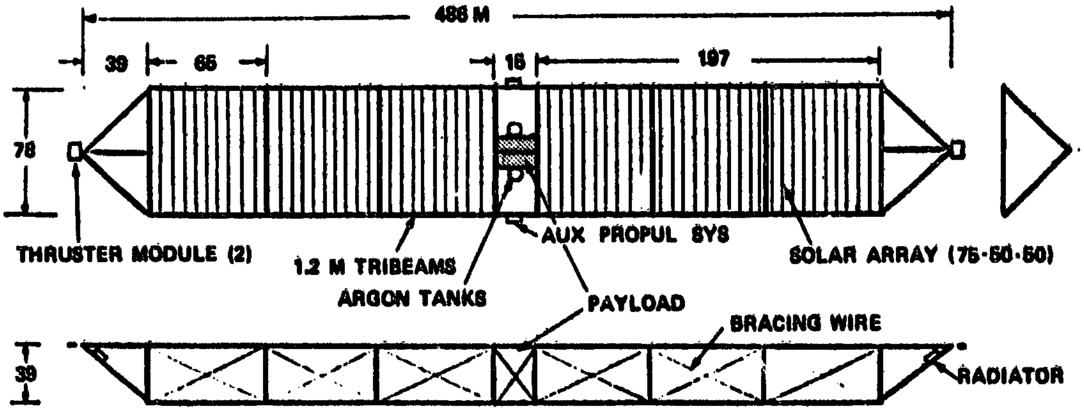
  4. **LOWER ISP (LESS POWER) ▷**
     - **ARCJETS**
- **KEY DESIGN REQUIREMENTS**
  - **PAYLOAD = 26 MT UP/0 DOWN**
  - **10 FLIGHT DESIGN LIFE**

point design was developed to serve as a point of comparison as well as to establish the basic characteristics of EOTV-type subsystems. Option 1 was to determine if the reduced degradation brought about by heavy shielding would offset the additional mass. The chemical assist transfer mode of Option 2 involves a chemical OTV transporting the EOTV rapidly to/from some altitude which would be above all or a major portion of the radiation belts. Option 3 involved a concentrated array which would reduce the amount of array required. Option 4 was an approach that employed an arc jet which operated at a lower Isp than ion thrusters and, thus, required less power and less array.

The remainder of this section consists of a more detailed description of each of these options. Again, the point design vehicle was used to establish the specific mass characteristics of all subsystems. The other system design options are discussed only in terms of the differences to the point design. Sizing and optimization of each concept are presented in section 4.3.1.5; they are compared in 4.3.1.6.

Point Design - As mentioned earlier, a point design EOTV was defined to serve as a means to establish overall subsystem characteristics. The key guidelines used to establish the point design were a specific impulse of 6000 sec, up-trip time of 180 days, and a payload capability of 25t up/0t down. Furthermore, the array was to be planar (CR = 1) with a 75-μm cover, 50-μm cell, and 50-μm substrate. The system was also to provide all its own propulsion (self-power) and utilize ion thrusters.

Configuration: The configuration and key characteristics of the point design EOTV are shown in figure 4.3.1.4-1. For the indicated design conditions, an array of over



- PAYLOAD UP = 25 MT
- PAYLOAD DOWN = 0
- SPECIFIC IMPULSE = 6000 SEC
- UP TRIP = 180
- DOWN TRIP = 96
- INITIAL POWER = 6620 KW
- MAX THRUST(EOL) = 29 N
- FIXED MASS = 33 MT
- ARGON MASS = 10 MT
- ARRAY AREA = 30000 M^2
- NO OF THRUSTERS (50 CM) = 84

Figure 4.3.1.4-1 EOTV Point Design Configuration

30 000 m^2 is required. The total vehicle dry weight is 33t, of which 10t is main propellant. The main propulsion modules are mounted on the centerline of the vehicle at each end. Through means of a yoke and gimbal system, the modules can be properly directed and operate whenever the vehicle is generating power. The modules contain thrusters as well as power processing units. Payload and propellant are located at the center of the vehicle to provide the most optimum moment-of-inertia characteristics. The solar array is

designed so one-half is dedicated to each main thruster module. Auxiliary propulsion modules are located at the vehicle center on the lateral axis to provide roll control. The structural framework is made up of space-fabricated tribeams. Fabrication and assembly of the beams, as well as installation of all subsystems, take place at a space base such as SOC. During the orbit transfer, the longitudinal axis of the vehicle is generally aligned with the north-south axis of the Earth.

Structure: Structural analysis of the EOTV relied heavily on similar work performed in the SPS studies. The key criterion in designing the beams within the truss was to sustain the bending moment induced by the uniaxial (lateral direction) edge loading of the array on the longitudinal beams and the column loading on the lateral beams. A minimum solar array edge loading of 2 N/m was assumed to provide the necessary array smoothness for maximum power output.

A number of truss cross sections are possible for the EOTV. Four candidates investigated are shown in figure 4.3.1.4-2. The V-bottom concept was selected primarily

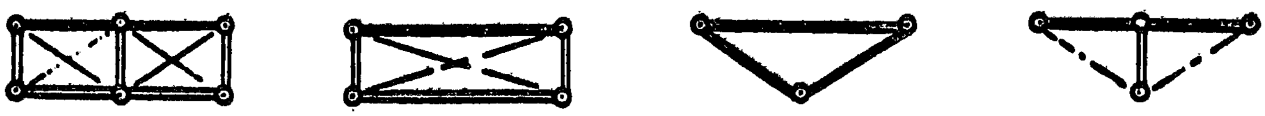


Figure 4.3.1.4-2 Truss Cross-Section Concepts

because of having less total beam length but also due to the considerations of construction ease, cross-section stability, and bending stiffness effectiveness. The braced-T truss, however, is also a viable option. A baseline length-to-depth ratio of 8% was selected for the V-bottom and, when analyzed for dynamics, was found to provide satisfactory separation between the natural frequency of the array and vehicle. In addition, the total truss weight was relatively insensitive to this ratio.

The basic member in the truss is a composite tribeam similar to that shown in figure 4.3.1.4-3. This type of beam has been analyzed in the General Dynamics/Convair Division Space Construction Automated Fabrication Experiment Definition Study (SCAFEDS) (NASA contract NAS9-15310). When used in an EOTV application, however, it was necessary to use rigid diagonal cord cross bracing due to the nature and magnitude of the imposed loads. The mass of the beam is approximately 1.0 kg/m. Fabrication of the beam is accomplished by an automated beam machine at the LEO construction base.

The resulting EOTV configuration had a natural frequency ratio (array-to-vehicle) of 20 to 25 at the 2 N/m edge loading.

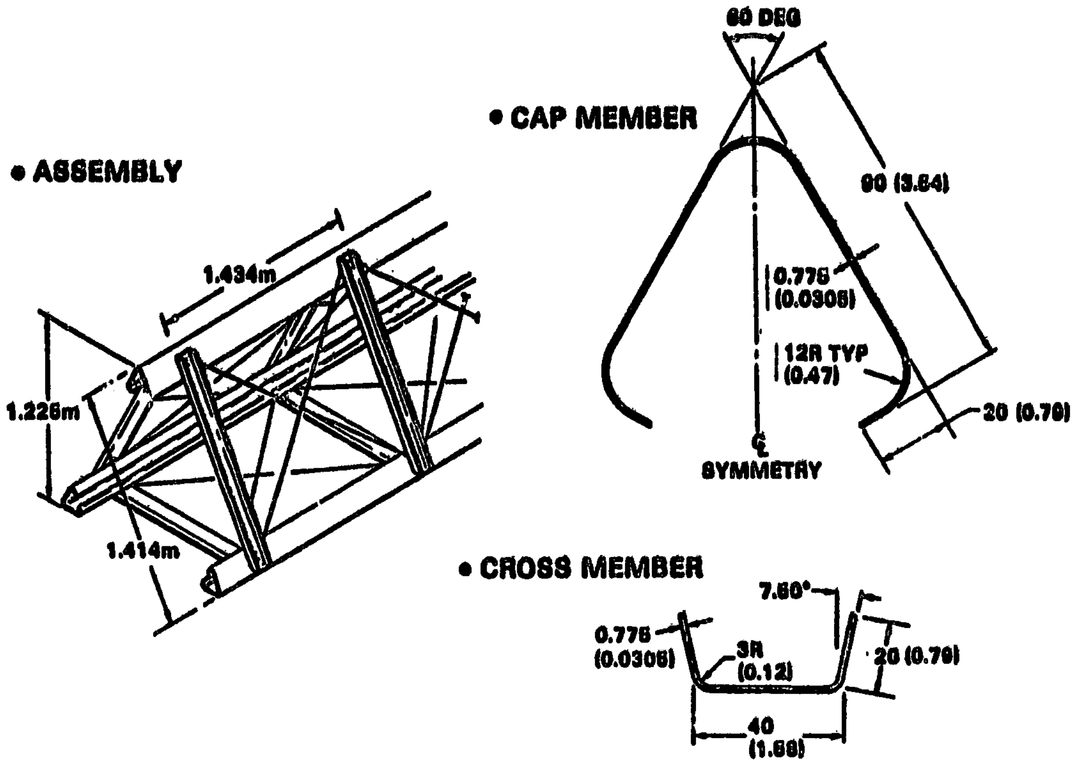


Figure 4.3.1.4-3 Beam Geometry

The trampoline method of supporting the solar blanket within the primary structural bay is shown in figure 4.3.1.4-4. It provides a uniform tension to the end of each solar

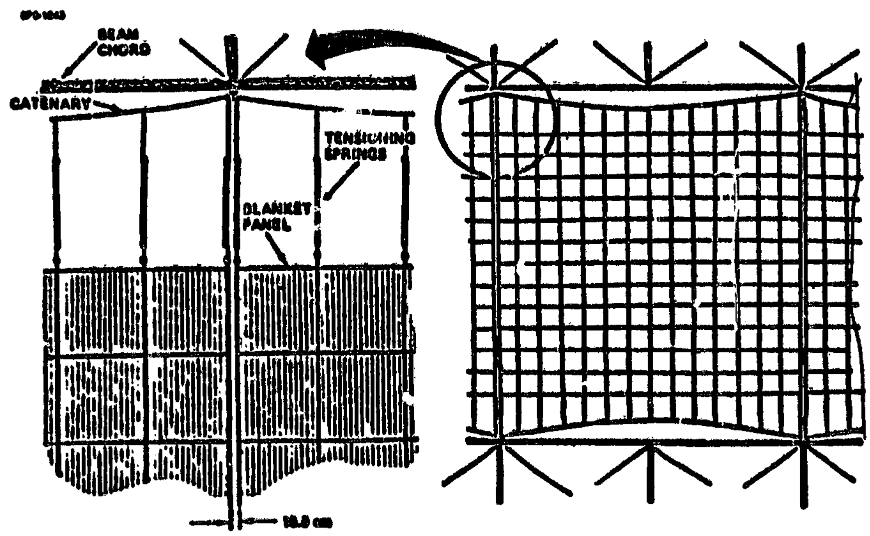
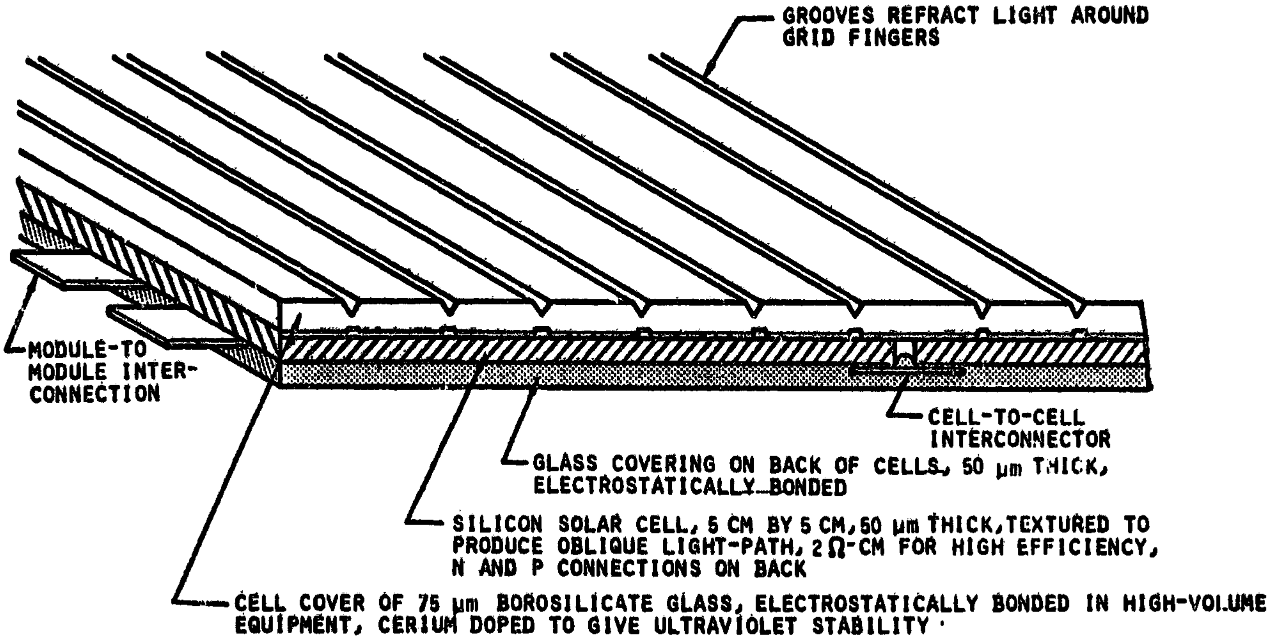


Figure 4.3.1.4-4 Array Support System

array segment by the use of constant-force blanket-tensioning springs at each blanket support tape. These springs are also attached to a catenary cable that is then attached to the primary structure. This method of support is lightweight as well as being most adaptable to the length changes in the structure due to temperature variations occuring as a result of occultations.

Solar Array: The solar array converts the incident sunlight into electrical power at the required voltage and supplies the power to the distribution system. The configuration of the solar array blanket is shown in figure 4.3.1.4-5. The definition of the array was performed in the SPS studies (ref. 2).



INTERCONNECTORS: 12.5 µm COPPER, WITH IN-PLANE STRESS RELIEF, WELDED TO CELL CONTACTS

Figure 4.3.1.4-5 Solar Array Blanket

The basic energy conversion device is a 50-µm-thick, 5- by 5-cm silicon cell, with a textured surface to reduce reflectance and a BOL efficiency of 16%. This results in a voltage of 0.6V and a current of a 0.9A at the maximum power point. The cover glass is 75-µm-thick cerium-doped borosilicate glass which is electrostatically bonded to the solar cell. The substrate is 50-µm-thick glass which is electrostatically bonded to the back of the cell. The cell is designed with both p and n junctions brought to the back of the cell. The interconnects are 12.5-µm-thick silver-plated copper. Complete panels are assembled by welding together the module-to-module interconnections. Detail mass estimates of this array indicated 0.426 kg/m^2. The power output of the array (without radiation degradation) was 179 W/m^2, as shown in table 4.3.1.4-2. As indicated pre-

Table 4.3.1.4-2 Silicon Array Output

SOLAR INPUT	= 1353 WATTS/M^2
CELL CONVERSION (0.16) AMO 25^o TEXTURED CELL	= 216.5
BLANKET FACTORS (.93) UV LOSSES CELL-CELL MISMATCH INTERCONNECT LOSS	= 201.3
APHELION INTENSITY (.9675)	= 194.8
THERMAL DEGRADATION (.918)	= 178.8
ORIENTATION LOSS (NONE, FLY PEP)	= 178.8
RADIATION DEGRADATION (INCLUDE AS PART OF ORBIT TRANSFER OPTIMIZATION)	= 178.8

viously, however, Van Allen radiation reduces the power output after one round trip to 42% of the initial power. The power decay curve as a function of altitude for the point design is shown in figure 4.3.1.4-6. The majority of all the degradation occurs by the time 10 000 km is reached; however, this also constitutes 60% of the total triptime for the up leg.

Selection of the optimum operating voltage for the solar array is complicated due to several competing factors as well as other considerations. The competing factors are that minimum I^{2}R losses occur with high voltage but plasma losses, particularly at low altitudes, are minimum with low voltage. Other considerations in the voltage selection issue are the decreasing voltage as a flight proceeds, due to array radiation degradation, and the optimum thruster Isp being 6000 sec, which requires a voltage of 850. The relationship of the competing factors as a function of voltage and array size, power specific mass, and percentage of plasma loss is shown in figure 4.3.1.4-7. At altitudes as

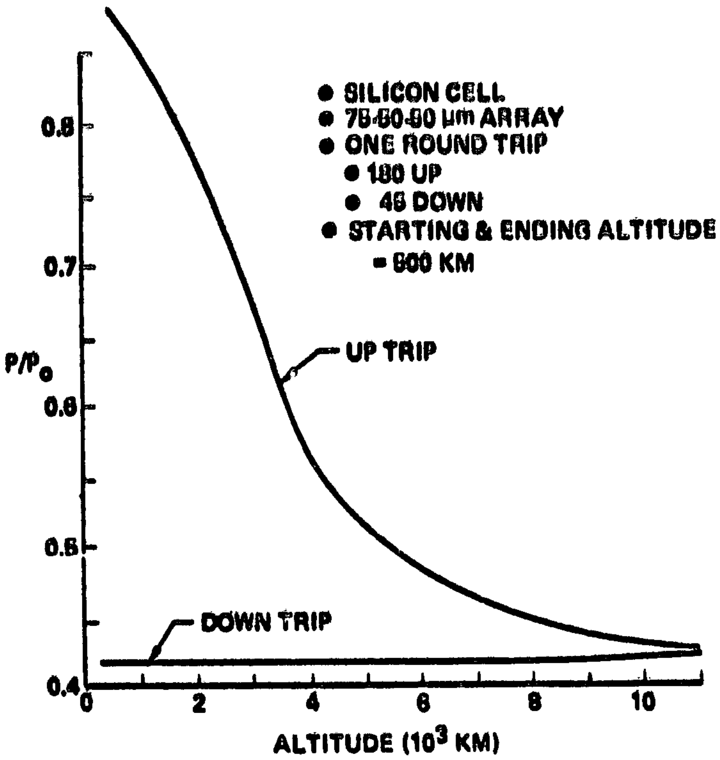


Figure 4.3.1.4-6 Power Output Versus Altitude

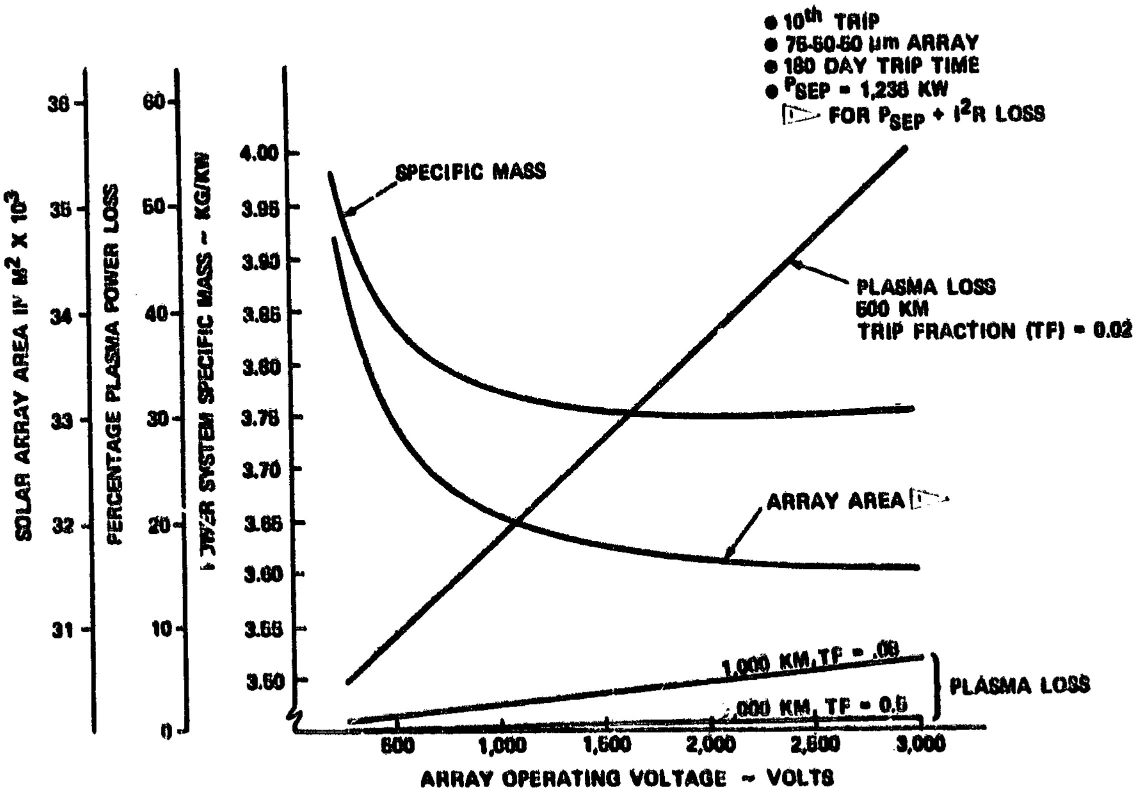


Figure 4.3.1.4-7 Power System Sensitivity to Operating Voltage

low as 500 km (starting altitude is 370 km), plasma losses amount to 30% at 1500V, which is the point where the array area and specific mass are minimized. Once the EOTV reaches altitudes beyond 1000 km (which takes approximately 8% of the up-trip time), the plasma losses are reduced considerably even at higher voltages. The array area is larger at low voltages because of the required higher current. The additional array area, as well as heavier conductors (buses) for the higher currents, result in specific masses at the low voltage. It should also be noted that the area and specific mass do not, however, reflect the plasma losses. The reason for this is because up until the 4th or 5th flight, excess power is available because the array has been sized for EOL rather than BOL (refer back to fig. 4.3.1.3-11). In addition, the amount of time in a given flight which is affected by large plasma losses is relatively small. The issue, therefore, becomes one of possible impact in terms of trip time with less available power due to plasma losses as a function of voltage. This relationship is shown in figure 4.3.1.4-8. A trip-time penalty exists at

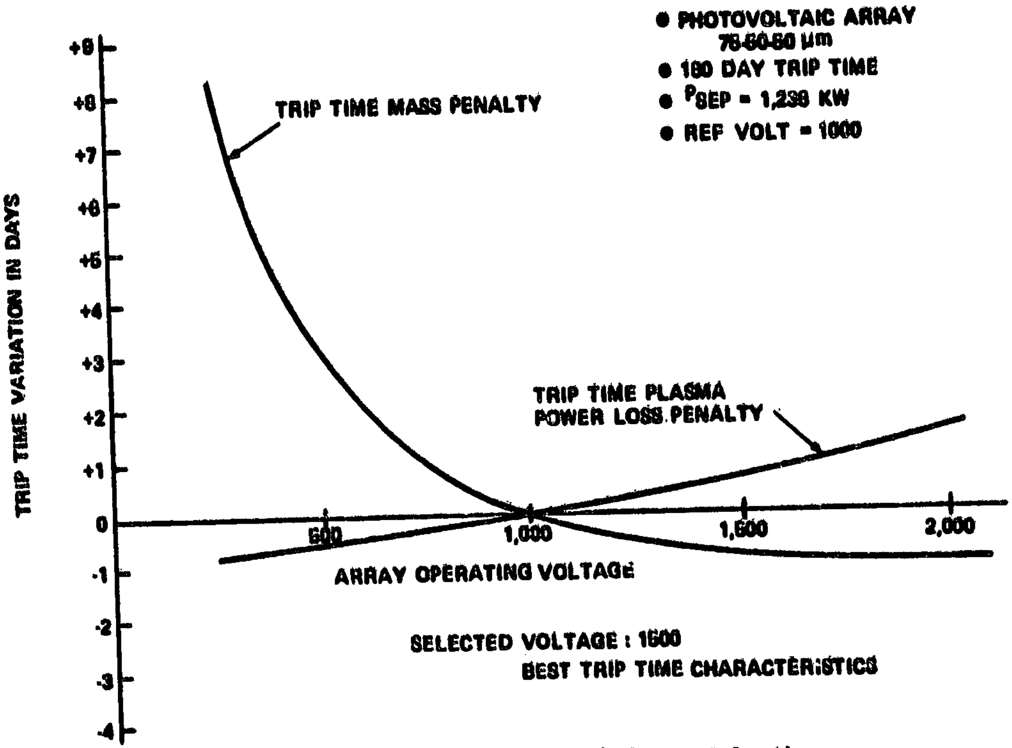


Figure 4.3.1.4-8 Operating Voltage Selection

low voltages because the vehicle is heavier and trip time is directly proportional to startburn mass for a fixed electric propulsion system. The optimum operating voltage is that which results in the minimum trip time. This point is found when the trip times from the mass penalty and plasma loss cancel each other. These data indicate an operational voltage of 1500V to be most desirable which was subsequently used in the design of the

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power distribution system and power processing system.

Power Distribution System: The selected power distribution system design for the point design EOTV is shown in figure 4.3.1.4-9. The power system design uses multiple

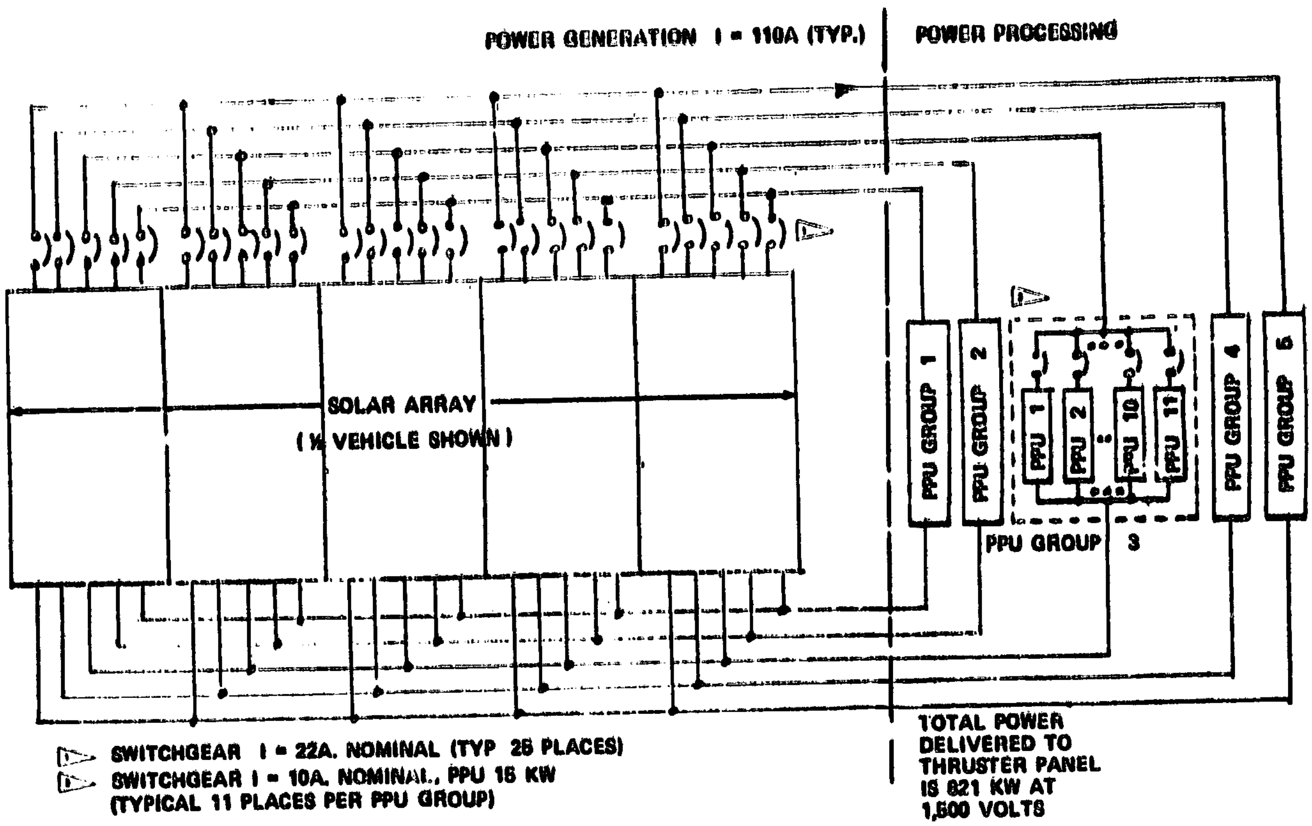


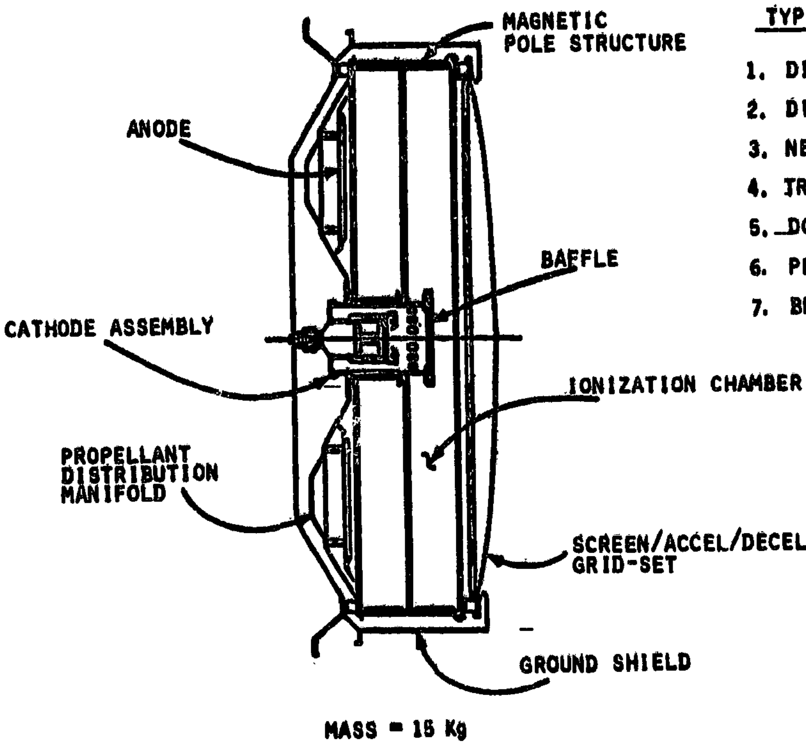
Figure 4.3.1.4-9 Power System for Electric OTV

main power conductors (five pairs) to limit potential main bus current during any system fault. As the distance from the thruster panel to the array increases, the conductor voltage drop increases. Thus, if a group of PPU/thrusters were powered solely from a section of array farthest from the group, the power delivered to that PPU/thruster group would be less than if it were powered from a section nearer the PPU/thruster panel. The multiple connections to the main power buses from the solar array enable each main bus to deliver approximately the same power to each PPU/thruster group.

The conductors selected for the main buses are thin aluminum sheets. Sheet conductors maximize the ratio of the surface area (for heat rejection) to conductor area (for current conduction). Aluminum was selected as the conductor material since the product of resistivity and density is less than that for other candidate materials (copper, silver, etc.). The conductor operating temperature was selected at 25°C.

Ion Thruster: Design studies for 50-cm-diameter argon ion thrusters are currently in progress under contracts being administered by NASA LeRC. Due to the competitive nature of the contracts, design and performance characteristics were somewhat guarded. The design and performance data which follow were developed by Boeing personnel who have been active in thruster characterization for over 5 years. A review of these data by LeRC personnel indicated the characteristics to be compatible with the findings of the LeRC contractual studies.

The assumed design concept of the thruster and typical characteristics are shown in figure 4.3.1.4-10. Thrust is produced by electrostatic acceleration of ions extracted from



TYPICAL CHARACTERISTICS ▷

1. DISCHARGE VOLTAGE: 37.0V
2. DISCHARGE CURRENT: 67.5A
3. NEUTRAL EFFLUX: 1.7135 A. eq.
4. TRANSMISSIVITY: 0.706
5. DOUBLE ION FRACTION: 0.0624 (J2/J1 avg.)
6. PEAKING FACTORS: 1/0.8
7. BEAM DIVERGENCE: 0.98

▷ FOR BEAM VOLT. OF 1000
BEAM CURRENT 12 AMPS
SEPS TYPE OPTICS

Figure 4.3.1.4-10 50-cm Argon Thruster

an electron bombardment ionization chamber. This design reflects a multipole containment field to improve beam uniformity (flatness) and maintain primary electron confinement in the plasma volume. The indicated characteristics are indicative of a thruster using a screen voltage of 1000V, beam current of 12A, and an optics similar to the NASA SEPS thruster. Variations in voltage and current would alter the indicated characteristics.

Performance characteristics for the thruster as a function of Isp are shown in figure 4.3.1.4-11. These data are indicative of that related to a 15A thruster and would

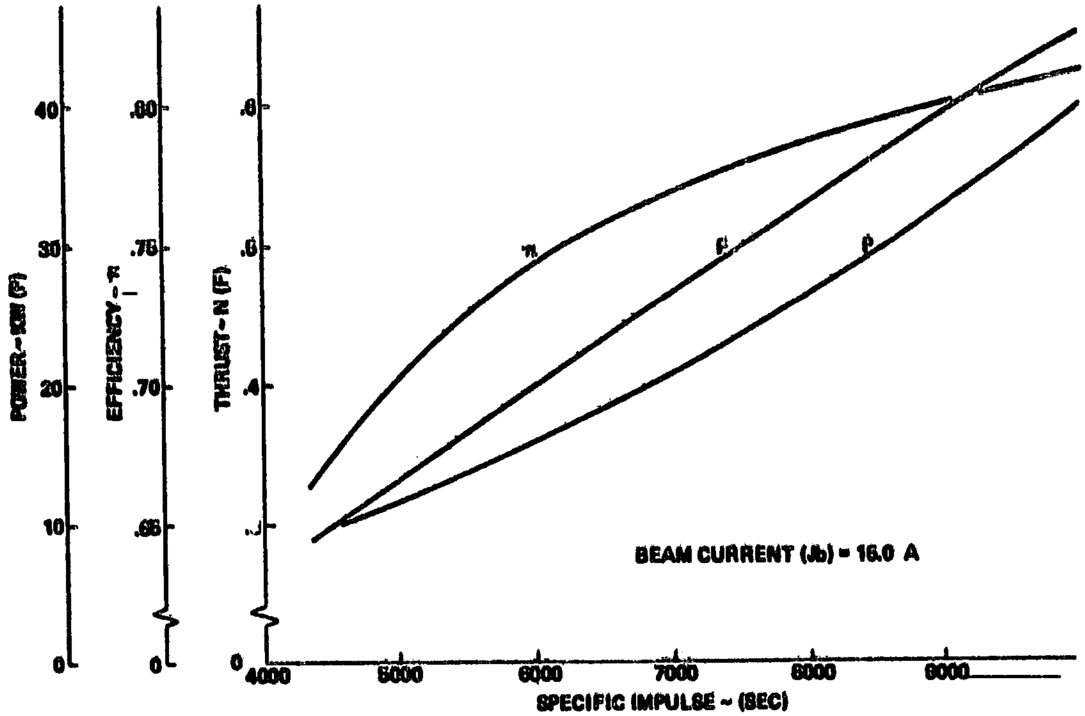


Figure 4.3.1.4-11 50-cm Argon Ion Thruster Characterization

vary for different currents. It should be noted that for Isp's below 5000 sec, the efficiency falls off dramatically, which will have a strong bearing on the required size of the vehicle as well as propellant. Screen grid life characteristics and voltage relationship with Isp are presented in figure 4.3.1.4-12. Life is shown to be highly sensitive to beam current as well as to screen voltage. The approach employed for selecting the beam current for a given Isp was to operate at the highest possible beam current that would allow the grids to have sufficient life to perform a given mission. At the completion of the trip, the grid sets would be replaced. Since a typical round trip involves over 5000 hr of propulsion, designing for no replacement of grids in 10 flights would necessitate a beam current of only 6A or 7A, which would considerably penalize the performance of the vehicle.

<u>Propellant Storage and Feed System</u>: Tankage for liquid argon is very similar to that associated with liquid oxygen since both have similar temperatures. Density of the argon is 1440 kg/m^3 versus 1140 kg/m^3 for LO_2. Propellant delivery from the tanks to the main propulsion modules is accomplished by heating the liquid argon and using the boiloff pressure to drive the gaseous argon. Flow rates resulting from this approach can be adjusted to match those required by the thrusters, thereby eliminating any boiloff

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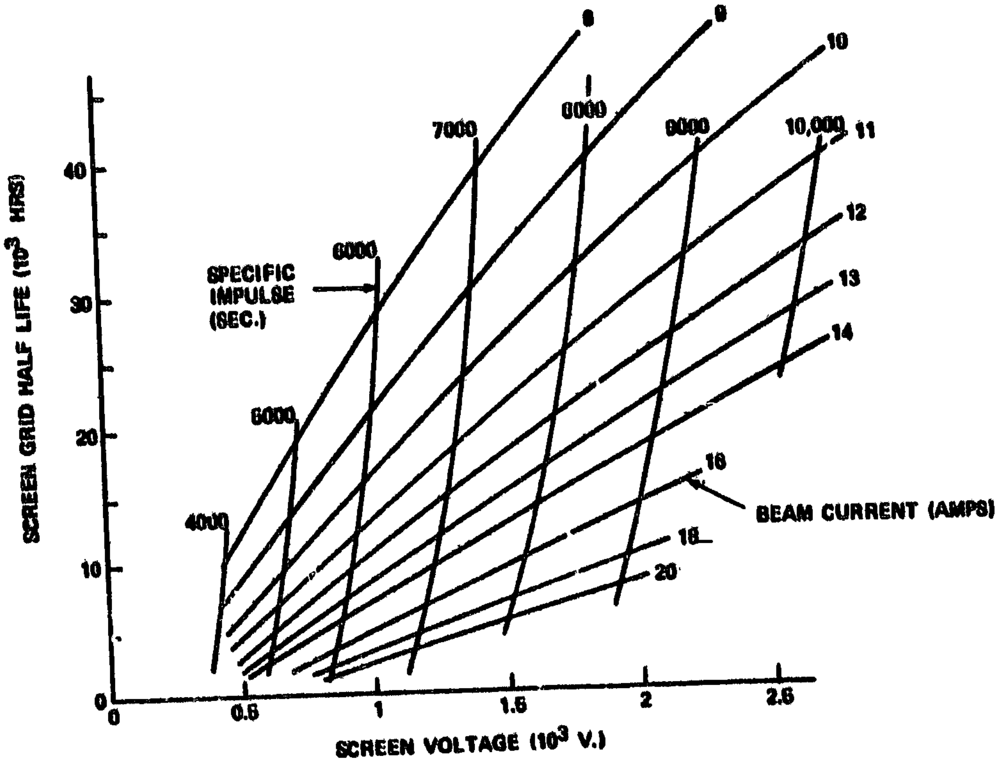

Figure 4.3.1.4-12 50-cm Argon Thruster Screen Grid Half-Life Trends

penalty associated with the long trip time. Based on LO_2 tank mass fractions, a value of 4% of the propellant mass was assumed for the mass of the tankage and feed lines.

Power Processing Units (PPU): The function of the PPU is to adjust the voltage provided by the array to that required by the electric thrusters. The extent of this adjustment is indicated by the selected array voltage beginning at 1500V and by the 10th flight decaying to approximately 1200V. The range of thruster Isp's to be investigated required voltages between 500V and 1500V. Another consideration was that for the sake of performance analysis simplicity, a constant Isp was desired, meaning a constant voltage. It should also be noted that direct drive of the thrusters (power taken from the array without processing) was judged to be beyond normal growth technology but was examined as part of the accelerated technology effort of section 4.4.

The power requirements for an ion thruster with an Isp of 6000 sec are shown in the following table:

Power Source	Voltage (V)	Current (A)	Power (W)
Screen	850	18	15 300
Discharge/cathode	37	96	3 552
Accelerator	566	0.004	2
Total power			18 854

Note: In addition to the operating-mode power requirement above, the discharge/cathode power support is switched to the cathode heater to bring the cathode up to operating temperature during thruster startup.

The discharge/cathode supply was specified to be well regulated with respect to the screen supply voltage. The discharge supply was to be switched to the thruster cathode to heat the cathode to operating temperature during thruster startup. The thruster keeper supply, if required for thruster operations, could be derived from the regulated discharge supply.

Based on these power requirements and the change in solar array characteristics with radiation degradation, the ion thruster power processing unit concept shown in figure 4.3.1.4-13 was developed. The input filter section is followed by a preregulated power to the remainde of the PPU. The liquid-cooled transformer used in the transformer coupled inverter section is based on the lightweight transformer development effort by the USAF Aeropropulsion Laboratory (ref. 16). Regulation of the screen and accelerator supplies is accomplished by the input regulator. However, the discharge supply requires further regulation and a regulator is provided for this output.

The mass and power loss of each major element of the ion thruster PPU are shown in table 4.3.1.4-3. The PPU has an efficiency of 92% and a specific mass of 2.87 kg/kW (without growth allowances).

Thermal Control: The dc-dc solid-state converter used in power processing requires an active thermal control system in order to control its operating temperature to a maximum of 70°C. A heat exchanger transfers heat from the gas circulating in the dc-dc converter to the Therminol-60 coolant loop. The coolant loop delivers the waste heat to a space radiator which emits heat from both sides. A specific mass of 8 kg/kW of radiated heat results with this system.

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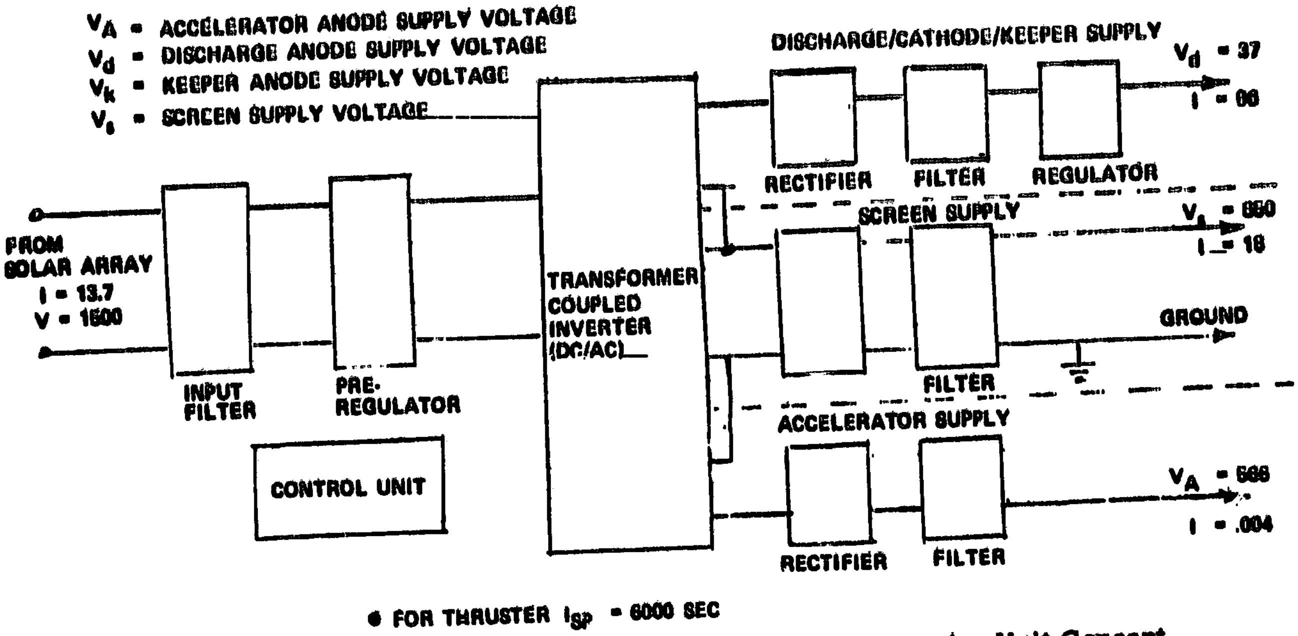

Figure 4.3.1.4-13 50-cm Argon Ion Thruster Power Processing Unit Concept

Table 4.3.1.4-3 Power Processing Unit Characteristics

● ION THRUSTERS ● TOTAL THRUSTER POWER = 18,854 WATTS

PPU SECTION	MASS IN KILOGRAMS	LOSSES IN WATTS
INPUT FILTER	(2.5)	(166)
PRE-REGULATOR	(3.5)	(39)
TRANSFORMER COUPLED INVERTER	(8.0)	(475)
SCREEN SUPPLY	(7.0)	(42.5)
RECTIFIER (I = 18A)	5.0	18
FILTER	2.0	407
ACCELERATOR SUPPLY	(2.5)	(120)
RECTIFIER (I = .004A)	1.0	1
FILTER	1.5	119
DISCHARGE/CATHODE/KEEPER SUPPLY	(9.0)	(454)
RECTIFIER (I = 96A)	4.0	67
FILTER	3.0	107
REGULATOR	2.0	280
CONTROL UNIT	(5.0)	(30)
INSULATORS	(3.0)	(15)
WIRING & CONNECTORS	(2.5)	
INTERNAL THERMAL CONTROL	(5.0)	
PACKAGING	(11.0)	
TOTALS	59.0	1,714

KG/KW = 2.87 EFFICIENCY = 92%

Avionics: Guidance and navigation equipment includes two Sun sensors, two star trackers, and redundant laser gyro assemblies for a total mass of 60 kg. Data handling makes use of two central computers, 20 remote units, and three interface units totaling 140 kg. S-band communications equipment totals 20 kg. Power control and wiring provisions contribute 760 kg, with 70% of this attributed to wire harnesses. The total avionics mass is 980 kg.

Secondary Power: Secondary power is required during the time the vehicle is occulted. Regenerative fuel cells provide the main power source with a small utility battery used for peaks. The total estimated mass is 250 kg including wiring.

Auxiliary Propulsion: Auxiliary propulsion is required to maintain attitude during the occultations, to provide initial propulsion when near the LEO space base, terminal propulsion near a GEO base, and attitude control/stationkeeping during the turnaround operations at LEO. To circumvent cryogenic storage problems and the low use rate, an N_2H_4 ACS system is employed.

Performance Parameters: The key performance parameters used in the optimization of each EOTV option as well as the values resulting with the point design are shown in table 4.3.1.4-4. The majority of the values were described in the preceding paragraphs; however, several factors represent a combination of several subsystems and merit explanation. The power generation system includes the mass of primary structure, solar array, and power distribution system, divided by the indicated blanket output. All masses reflect growth allowances. The EPS structure includes the yoke and gimbal system associated with the main propulsion modules. The other subsystems include the total of the avionics, secondary power, and auxiliary propulsion.

Option 1: Heavy Shielding - The motivation for a heavy shielded solar cell option was to see if the benefit of reduced radiation degradation would offset the additional mass per unit area and result in a lower cost EOTV. The performance parameters which differ from the point design are the specific mass of the power generation system and the P/Po for the 10th flight.

Based on array power output versus shield thickness data shown in figure 4.3.1.4-14, 300-μm (12-mil) shields appear to be an optimum design point. The reference heavy shield array is therefore 300-50-250 μm (12-2-10 mils) in terms of cover-cell-substrate, respectively. The additional substrate thickness is necessary since the radiation is omnidirectional.

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Table 4.3.1.4-4 Point Design Performance Parameters

VARIABLE PARAMETERS ▷	VALUE	FIXED ▷ PARAMETERS	VALUE
• ΔV (ONE WAY) M/S	6000	• RESERVES (%)	2
• SPECIFIC IMPULSE (SEC) ▷	6000	• RADIATOR (kg/kw)	8
• NON THRUST TIME %	15	• EPS STRUCT (% PM)	15
• P/PO (10 TH FLT @ 180 DAYS)	0.72	• OTHER SUBSYS (kg)	2200
• POWER GEN SYS (kg)	4		
• BLANKET OUTPUT (W/M^2)	179		
• PPU EFF. ▷	92		
• PPU (kg/kw)	3.3		
• PROP TANKS (%WP)	4.		
• THRUSTER ▷ (kg/kw)	1		

▷ VALUE IS TYPICAL
▷ TYPICAL WITH INDICATED ISP
▷ VALUES VARY WITH DESIGN OPTIONS
▷ VALUES SAME FOR ALL DESIGN OPTIONS

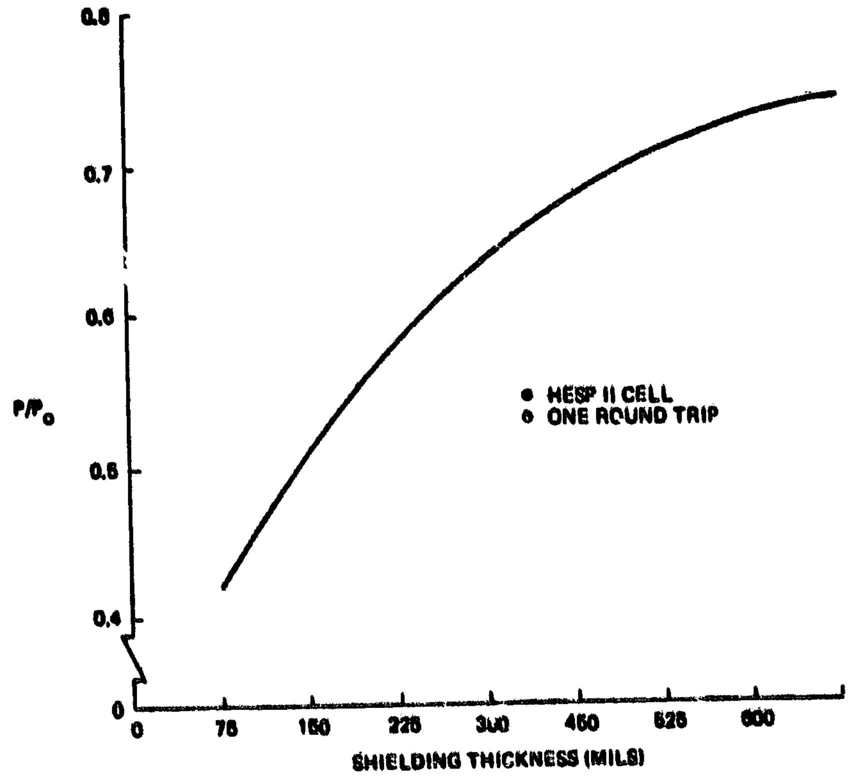

Figure 4.3.1.4-14 Shielding Effectiveness

The additional mass per unit area of the 300-50-250 array is based on 0.057 kg/m^2 for each 25 µm (1 mil) of cover or substrate. As a result, the 300-50-250 has an area mass of 1.4 kg/m^2 versus 0.427 kg/m^2 for the point design. Because of the heavier array, a 15% structural penalty was assumed. No additional penalty was assumed for the power distribution system.

The total power generation system involves the array, structure, distribution system, and growth allowances (21%). The power output of the array (without degradation) was 179 W/m^2. The resulting specific mass of the 300-50-250 was 10 kg/kW versus 4 kg/kW for the point design.

The benefit of the heavy shield option, however, is much lower degradation. As previously shown in figure 4.3.1.3-10, at the completion of the first trip, the P/Po is 0.64 versus 0.42 for the point design. At the completion of the 10th flight, the heavy shield option has a P/Po of 0.45 versus 0.22 for the point design.

The system level comparison of this option and others is presented in section 4.3.1.5.

Option 2: Chemical Assist Transfer Mode - This option is based on the concept of moving the EOTV rapidly through all or the majority of the Van Allen radiation belts using a chemical OTV. This concept is illustrated in figure 4.3.1.4-15. Following staging of the

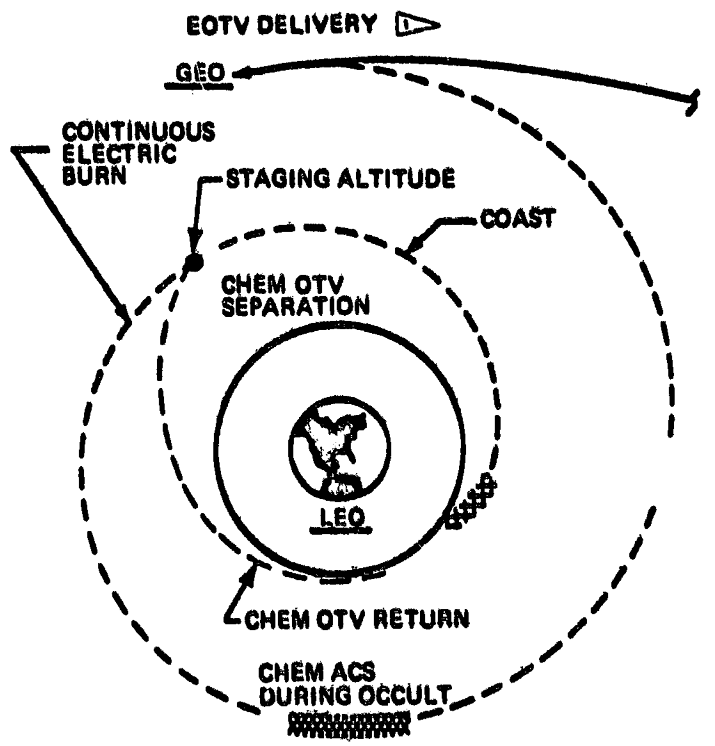

Figure 4.3.1.4-15 EOTV/Chemical OTV Mission Profile

chemical OTV, the EOTV completes its delivery mission to GEO. Return of the EOTV can be accomplished either by itself or by a chemical OTV rendezvousing at a specific altitude and again moving the EOTV rapidly through the belts back to LEO. The key mission variable in this option is the altitude selected for staging. The performance parameters, which will vary from the point design, include the EOTV delta-V, nonthrust time, P/Po, and the use of a chemical OTV for delta-V assistance.

EOTV delta-V requirements as a function of altitude and inclination are shown in figure 4.3.1.4-16. In this analysis, altitudes of 7400 km and 11 100 km were considered for

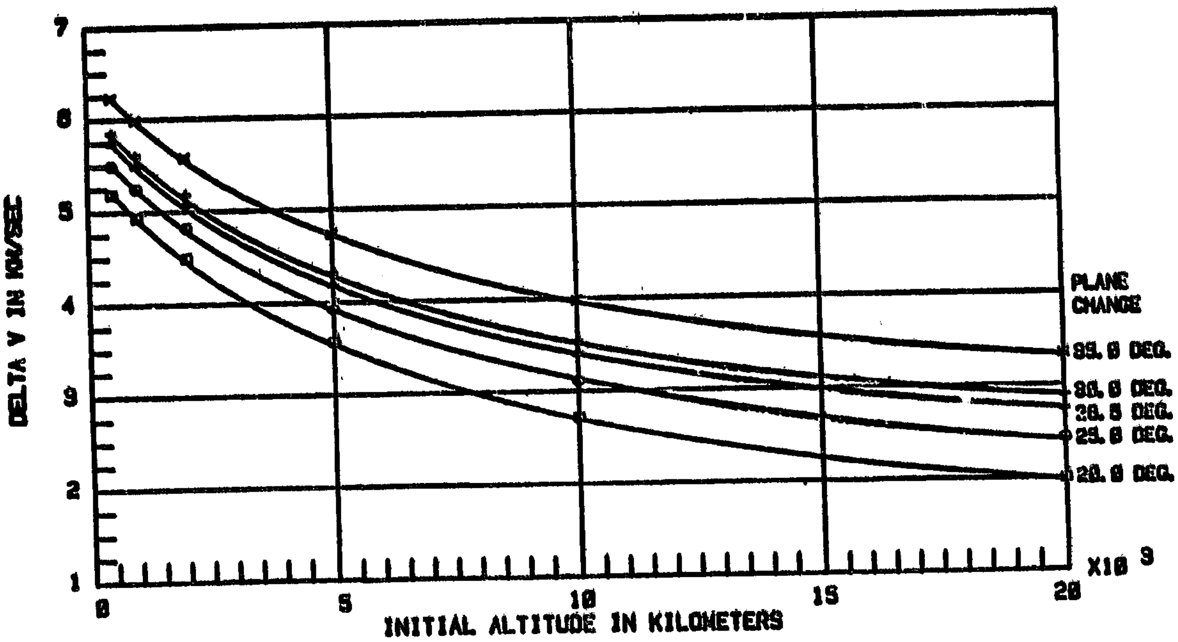

Figure 4.3.1.4-16 Low-Thrust LEO to GEO Delta-V

investigation. At 28.5 deg, the 7400-km altitude required a delta-V of 4050 m/s (one-way and including reserves) while 11 100 km requires 3550 m/s as compared with 6000 m/s for the self-power point design which begins its mission at 370 km.

The nonthrust time for this option is 2% rather than 15% for the point design. The reduction is the result of the EOTV being nearly in full sunlight once the indicated altitudes are reached. A small amount of occultation occurs as the vehicle approaches GEO (within 20 days) during the equinox periods due to array and thruster startup considerations.

The P/Po associated with this option is indicated in figure 4.3.1.4-17. The left-hand plot shows the fraction of fluence below a given altitude. For the altitude of 7400 km,

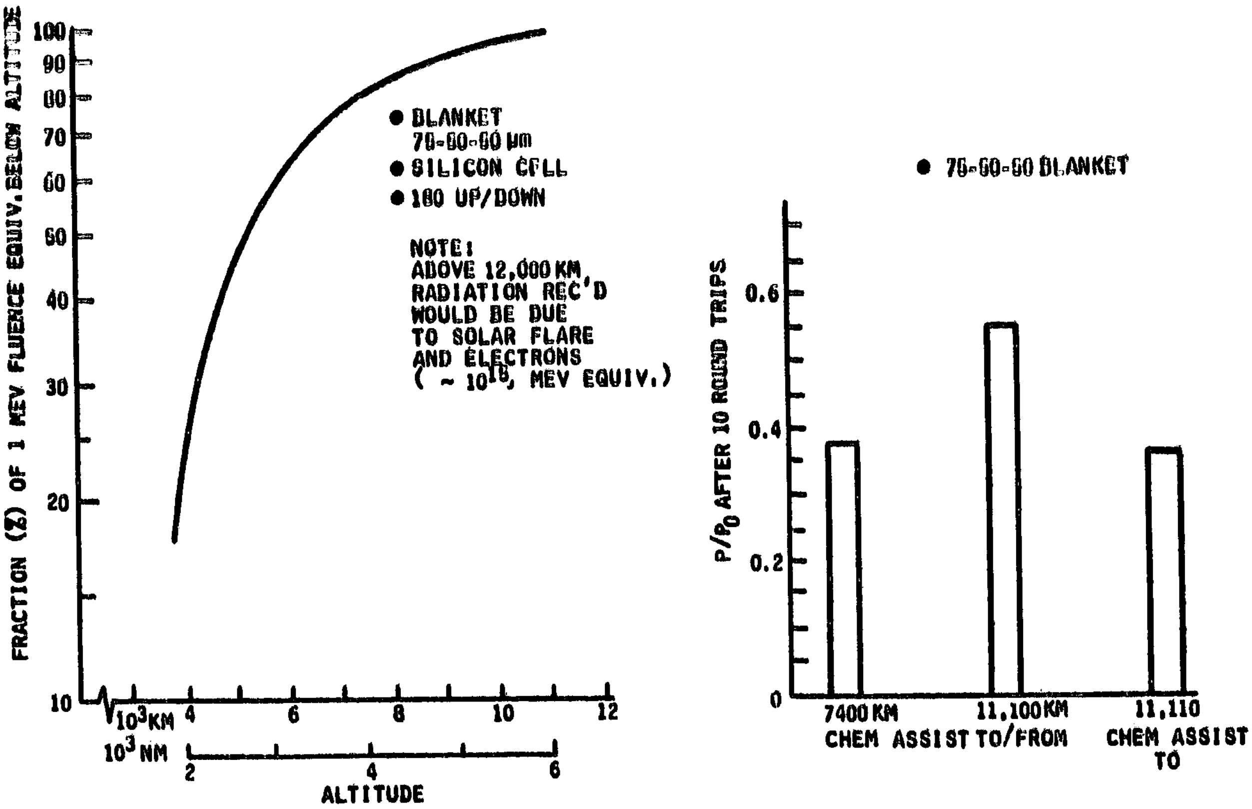

Figure 4.3.1.4-17 Power Outp.: Sensitivity to Altitude

nearly 80% of the radiation is bypassed and with staging at 11 100 km, approximately 98% is avoided. At the higher altitude, however, trapped electrons and solar flares become contributing factors. In terms of 1-MeV fluence per round trip, the 11 100-km altitude would have 3.5×10^{15} versus 1.07×10^{17} for the point design. The resulting 10th flight P/Po is 0.55 for the chemical assist to and from 11 100 km versus 0.22 for the point design. The other chemical assist options both have a P/Po of approximately 0.4.

The significant improvement in power with this op:ion unfortunately involves the use of a chemical OTV to achieve the desired altitude. The propellant required for the chemical OTV to deliver and return an EOTV is shown in figure 4.3.1.4-18. On a mission which delivers an EOTV to a given altitude, the chemical OTV returns to LEO using aeroassist where it would wait for the return of the EOTV to the staging altitude. When a chemical OTV returns an EOTV to LEO, the propellant includes that which is necessary to reach the EOTV at the staging altitude and the return using an all-propulsive return mode due to g constraints.

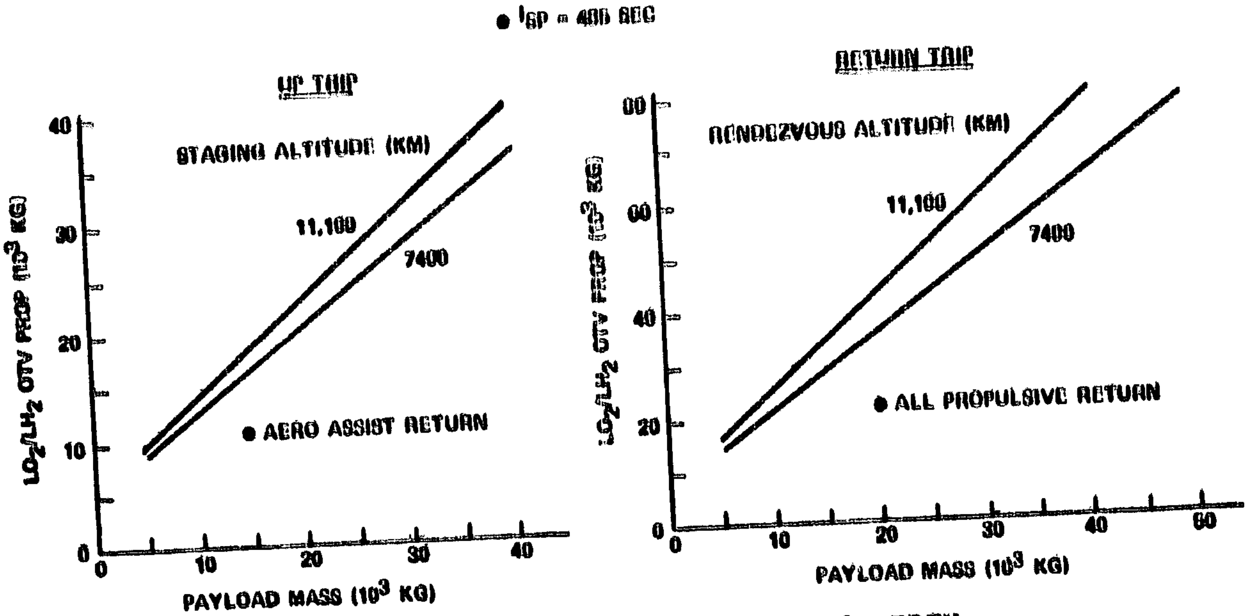

Figure 4.3.1.4-18 Chemical OTV Assist for EOTV

The optimization of this concept, in terms of which suboption is the most effective, is presented in section 4.3.1.5; the comparison with other system options is in section 4.3.1.6.

Option 3: Concentrated Solar Array - Another method which can be used to reduce solar array area (and resulting high cost) is to have a concentrated design which provides more power per unit area. A number of concentrator designs were considered in the SPS studies (ref. 2). The one judged to be the best when considering factors such as performance, constructibility, and required attitude control accuracy was CR = 2. An illustration of this concept is shown in figure 4.3.1.4-19. In this concept, sunlight is received by the array via direct impingement as well as from reflection from the concentrators. Lightweight space-fabricated tribeams for the structure form the V-ridges which, in turn, are covered by aluminized Kapton to provide the reflecting capability. The performance parameters which are different from the point design include the power output and the specific mass of the power generation system.

The power output for the CR = 2 design was estimated to be 260 W/m^2 (before degradation) versus 179 W/m^2 for the point design, which was planar. The higher output is the result of effectively twice as much sunlight hitting the array, but it is partially offset because the cells operate at a lower efficiency due to the higher temperature caused by the concentration (106°C versus 55°C for CR = 1). A 5% penalty was also included to compensate for lack of perfect reflector flatness. The effective CR was therefore 1.45.

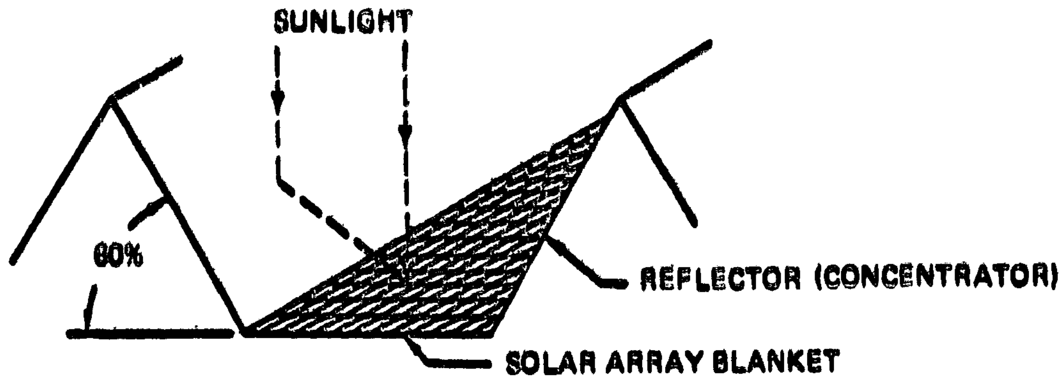

Figure 4.3.1.4-19 CR = 2 Concept

The power generation system for this option includes the array, structure, concentrators, and distribution system. A specific mass of 3.5 kg/kW was estimated versus 4.0 kg/kW for the point design. This value was based on CR = 2 to CR = 1 specific mass ratios established in the SPS study.

Option 4: Lower Power With Arc Jets - This option also has the potential to reduce the required solar array area by means of operating at a significantly lower Isp and, thus, less propulsion power. This is accomplished through use of hydrogen propellant arc jets rather than ion-thrusters.

As indicated in section 4.3.1.2, arc jets have been investigated in the past but not until recently has a concept been put forth which had the potential to improve their efficiency to an acceptable level. This concept was that of adding a mixing chamber downstream of the arc chamber to homogenize the propellant, which is subsequently expanded in a conventional nozzle. A concept for a 25-kW unit is shown in figure 4.3.1.4-20. Thruster size considered in the FOTV analysis, however, ranged between 100 to 200 kW. Most significant performance parameters which vary from the point design include thruster performance, PPU performance, and propellant tank mass fraction. Performance for the arc jet includes an Isp of 900 sec (versus 6000 sec for point design) and an efficiency of 0.9 (versus 0.75). Higher Isp is possible; however, chamber temperatures become a concern and the assumed efficiency would be lower.

The PPU design concept for the arc jet thruster was based on a unit sized for 200 kW (2000A at 100V). The concept efficiencies and specific mass for lower power levels would not be appreciably different. The arc jet PPU concept is shown in

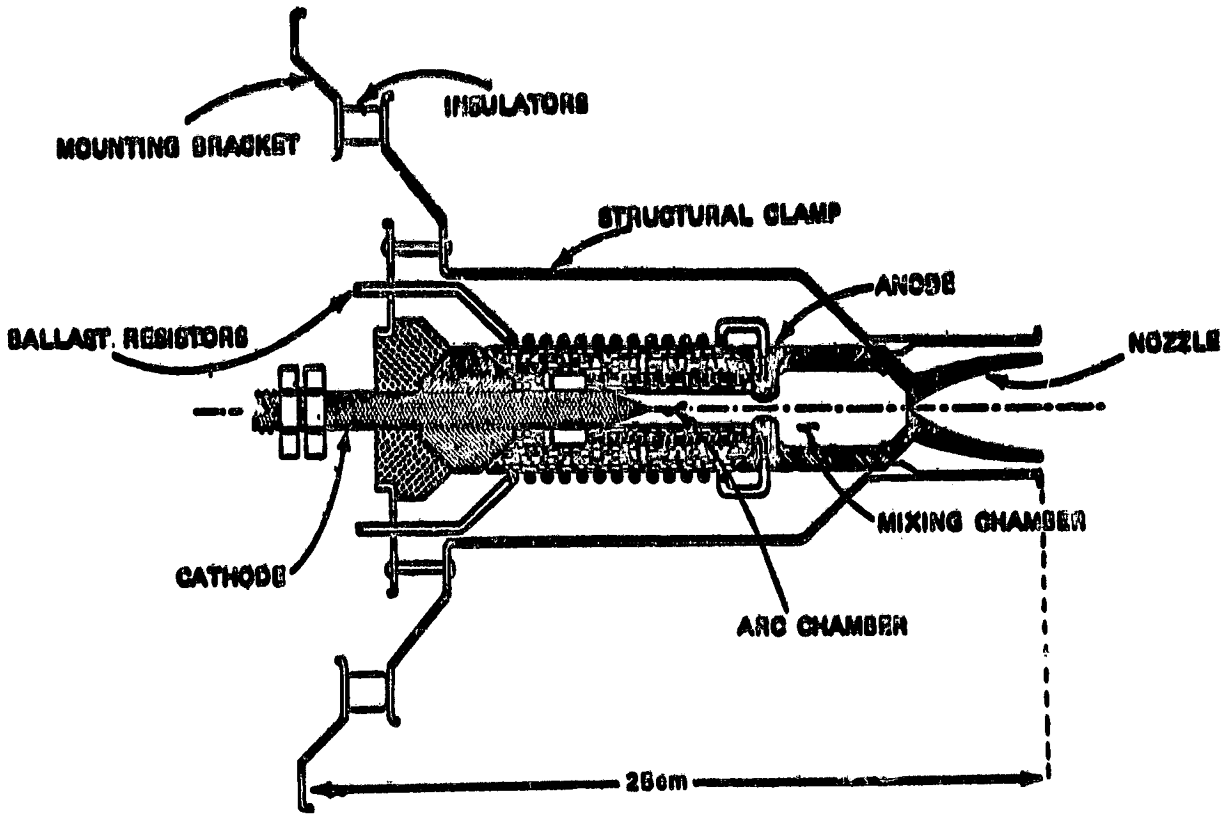

Figure 4.3.1.4-20 25-kW Thermal Arc-Jet Concept

figure 4.3.1.4-21. Related mass and power loss of each major PPU element are shown in table 4.3.1.4-5. The PPU for the arc jet has an efficiency of 93% and a specific mass of 1.83 kg/kW of processor input power (without growth allowances). The major causes for differences in arc jet PPU efficiency and specific mass from the ion thruster PPU are the number and types of PPU outputs required for the ion thruster. The increased number of supply voltages for the ion thruster primarily account for the specific mass difference, and the regulation requirements of the ion thruster discharge voltage account for the efficiency difference.

The propellant tank fraction to propellant mass was estimated to be 18% versus 4% for the point design. This significantly higher value is the result of the arc jet using hydrogen propellant with a density of 67 kg/m^3 rather than argon (1440 kg/m^3).

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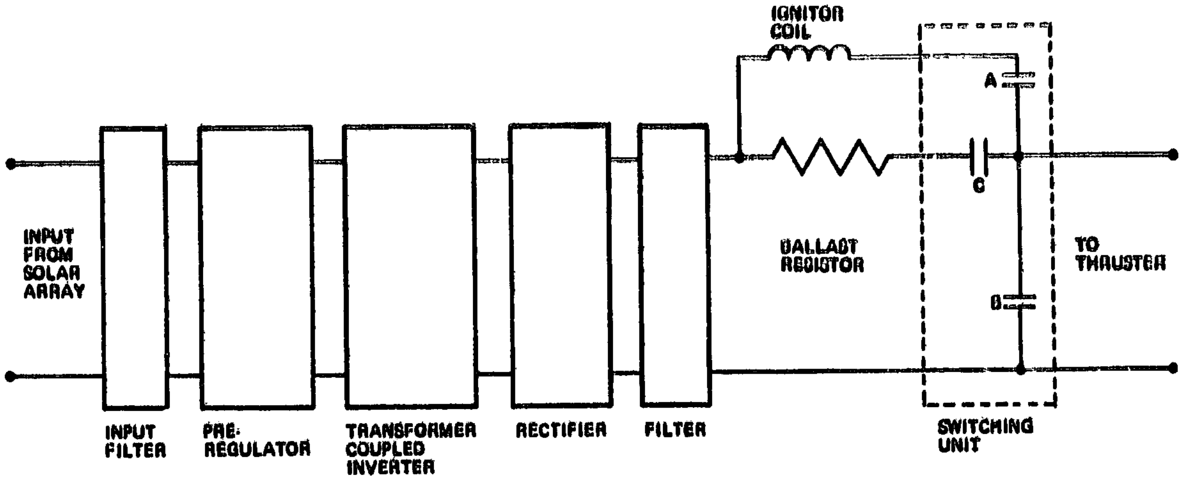

Figure 4.3.1.4-21 Arc Jet PPU Block Diagram

Table 4.3.1.4-5 Arc-Jet PPU Characteristics

V = 100V
I = 2,000A

POWER PROCESSING UNIT SECTION	MASS IN KILOGRAMS	LOSSES IN WATTS
INPUT FILTER	25.5	1,760
PRE-REGULATOR	36.0	415
INVERTER	82.5	5,515
RECTIFIER	80.0	1,400
FILTER	31.0	5,320
IGNITER	4.0	1
SWITCHING UNIT	12.0	200
CONTROL UNIT	5.0	50
WIRING AND CONNECTORS	8.0	150
INTERNAL THERMAL CONTROL	37.3	–
PACKAGING	71	
TOTAL	392.3	14,810

1 NOT USED IN NORMAL OPERATIONS

kg/kW = 1.83 (w/o GROWTH)

EFFICIENCY = 93%

4.3.1.5 System Optimization

Each of the design options characterized in the preceding section can be operated with a certain trip time and Isp which results in minimizing power, mass, and, most importantly, cost. This section presents the guidelines and assumptions used in the optimization and the results for each design option.

Guidelines and Assumptions - System level performance factors which contribute to the cost optimization included the size of the array, dry weight of the EOTV, and propellant requirement per flight. A summary of the top-level parameters used in establishing these performance parameters is shown below (each has been discussed previously in section 4.3.1.3).

1. Payload up: 25t
2. Payload down: 0t
3. Design life: 10 flights
4. Size for end of life: 10th flight
5. Up-trip time: variable (120 - 300 days)
6. Isp: variable (4000 - 8000 sec)

A summary of the key subsystem and flight characteristics contributing to the performance of each EOTV design option is presented in table 4.3.1.5-1. Each of these parameters has been described in section 4.3.1.4.

Table 4.3.1.5-1 EOTV Options Performance Parameter Summary

PARAMETER	POINT DESIGN	OPTION 1 SHIELDING	OPTION 2 CHEM ASSIST	OPTION 3 CR = 2	OPTION 4 ARCJET
ΔV (ONE WAY) M/S	6000	✓	3560 & 4000	✓	✓
SPECIFIC IMPULSE (SEC)	6000	✓	✓	✓	800
NON THRUST TIME %	15		2	✓	✓
P/PO (10TH FLT @ 100 DAYS)	.22	.48	.64	✓	✓
POWER GEN SYS (Kg/Kw)	4	10	✓	3.6	✓
BLANKET OUTPUT (W/M²)	170	✓	✓	260	✓
PPU EFF.	92	✓	✓	✓	93
PPU (Kg/Kw)	3.3	✓	✓	✓	2.1
PROP TANKS (% Wp)	4	✓	✓	✓	18
THRUSTER (Kg/Kw)	1	✓	✓	✓	0.8 - 1.0
RESERVES (%)	2	✓	✓	✓	✓
RADIATOR (Kg/Kw)	8	✓	✓	✓	✓
ESP STRUCT (% EPS)	15	✓	✓	✓	✓
OTHER SUBSYS (Kg)	2200	✓	✓	✓	✓

PARAMETER VARIES BUT INDICATED VALUE IS TYPICAL
✓ SAME AS POINT DESIGN
TYPICAL WITH INDICATED ISP

A high-level costing approach was used for the initial optimization and comparison due to the wide range of vehicle sizes possible with variable trip time and Isp. The key parameters were launch cost, vehicle cost, and trip time interest cost. It was also judged that a distinction between design options could be found by comparing the costs associated with each vehicle concept performing 10 flights (its design life). Once an EOTV concept was selected, a more detailed-cost assessment would be made including DDT&E, total production and operations costs associated with the EOTV fleet, and the delivery of compatible mission model payloads.

Launch costs were to include those associated with the LEO delivery of payloads, EOTV hardware, and propellant. A shuttle-derivative vehicle was to be used with a cost of $22M per flight.

EOTV costing was done using parametrics associated with the power generation system and the electric propulsion system. The power generation system cost parametric is shown in figure 4.3.1.5-1. The indicated cost reflects three key points: First, the

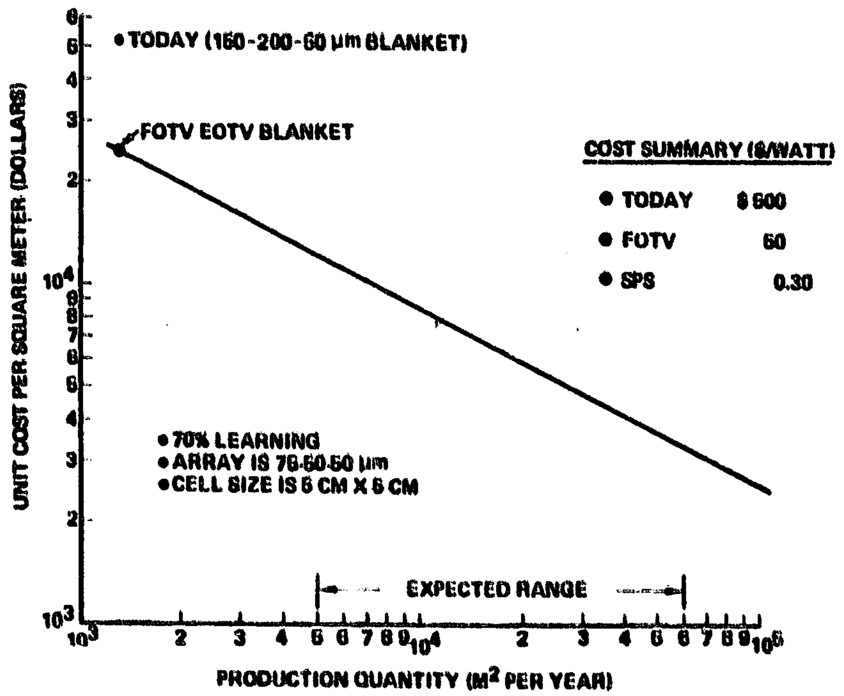

Figure 4.3.1.5-1 EOTV Design Driver—Solar Array Production Cost

initial starting point for the EOTV array is approximately one-half the cost associated with a typical 1980 technology array. The basis for this reduction is presented in table 4.3.1.5-2. The biggest reduction is the result of using 5- x 5-cm cells, although a

Table 4.3.1.5-2 Silicon Solar Blanket Cost Projection Production

	1980 (TYPICAL)	1990 TECH. THIN CELL	1990 TECH. THIN WITH LARGE SIZE
BLANKET MAKE-UP (COVER-CELL-SUBST t IN μm)	150-200-50	75-50-50	75-50-50
CELL SIZE (CM)	2 x 2	2 x 2	5 x 5
CELL EFF. (%)	12.5	16	16
MATERIAL COST	($10)	($11.0)	($66.0)
CELL	5 ▷1	7.50 ▷4	45.80 ▷4 ▷8
COVER	4 (FUSED SILICA)	3.0 ▷5	18.70 ▷5 ▷8
OTHER	1 ▷2	0.50 ▷6	1.50 ▷8
LABOR-BLKT. ASSY	($15)	($11) ▷7	($5.00) ▷9
BLANKET COST/CELL	($25)	($22)	($71)
CELLS/SQ. M	2260	2260	365
COST/SQ. M	$56,500	$49,720	$25,915

▷1 BASED ON 1700 M^2 (200KW) PER YEAR

▷2 CONVENT INTERCONNECT, ADHESIVE, INSUL.

▷3 REFLECTS CURRENT SPLINT IN SEPS BLANK

▷4 IF THERE WAS ▷1

▷5 CERIUM DOPED MICROSHEET

▷6 PRINTED INTERCONNECT – NO ADHES, INSUL

▷7 FEWER PIECES IN BLANKET

▷8 PROPORT. TO AREA & DIMEN

▷9 FEWER CELLS/BLKT.

small benefit also occurs with an advanced thin cell (50 μm). Second, the EOTV array production cost reflects a 70% cost reduction rate which may be expected with highly automated production associated with large numbers of units. Third, the production rate per year reflects a total of nine vehicles during the 16-year mission model. The total power generation system cost was found by adding 5% to the array cost to cover structure and distribution systems.

The electric propulsion system cost parametric is presented in figure 4.3.1.5-2. These data reflect average unit costs of the thrusters and PPU's based on a buy of 9 vehicles. Thruster and PPU costs constitute 75% of the cost of the EPS. For a given trip time, a higher Isp requires more thrust, thus more units and a lower average unit cost. For a given Isp, longer trip times require less thrust, thus fewer units and a higher average unit cost.

Trip time cost relates to the interest which must be paid on borrowed money during the time it takes to make the delivery. Typically this is a half year. The interest was

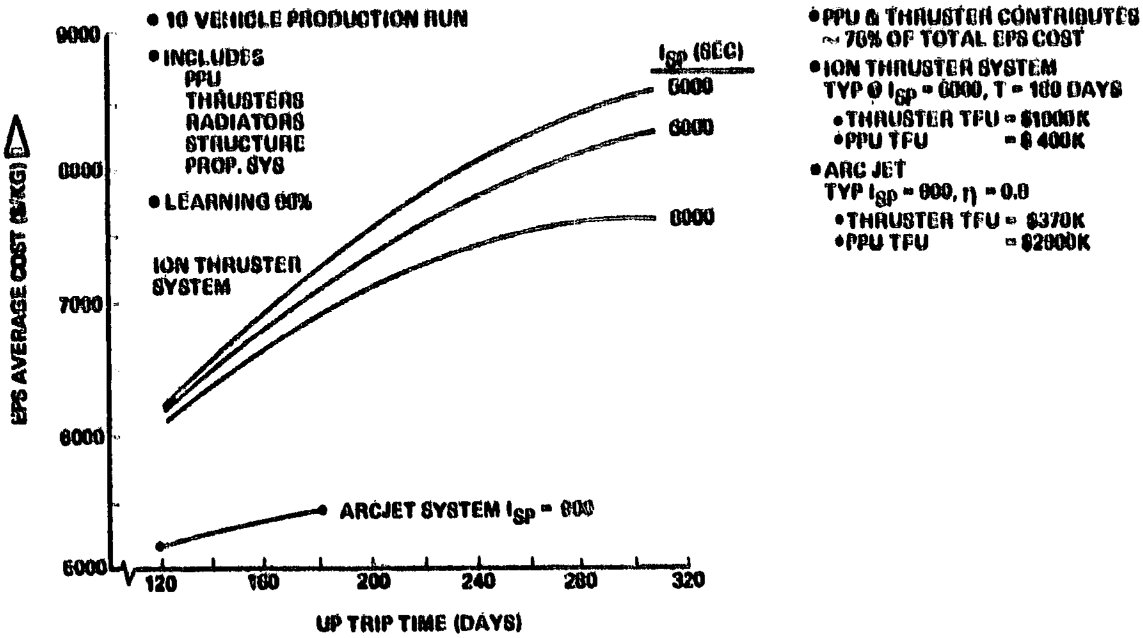

Figure 4.3.1.5-2 Electric Propulsion System Cost Parametrics

determined using a rate of 15%, paid against the captial invested which included launches and payloads. The cost for a 25t payload was assumed to be $100M.

Point Design Optimization - Optimization for the point design is presented in terms of power, mass, and cost. The power optimization is presented in figure 4.3.1.5-3. In the left-hand plot, which uses a fixed Isp, the emphasis is on sensitivity to trip times. Longer values require less acceleration and thus less propulsion power. The degradation percentage increases with longer trip time. The right side of the figure presents the same data but from the viewpoint of the sensitivity to Isp as well as trip time. Higher Isp generally requires more power although, at values as low as 4000 sec, the thruster efficiency falls off so rapidly that more power is required. In summary, an Isp of 5000 sec appears optimum over the range of trip times of most interest (180 to 240 days).

The mass optimization is shown in figure 4.3.1.5-4. Once again, these data reflect the mass which must be delivered to LEO in order to perform 10 EOTV flights. From a fixed-trip-time standpoint shown in the upper left, it indicates the hardware mass is minimum at 6000 sec and gets heavier with more Isp. The higher Isp's require more power which means more solar array and propulsion equipment if a constant thrust is maintained. Propellant mass, however, comes down significantly with higher Isp. The upper right plot shows both hardware and propellant decreasing with longer trip times. Again this is due

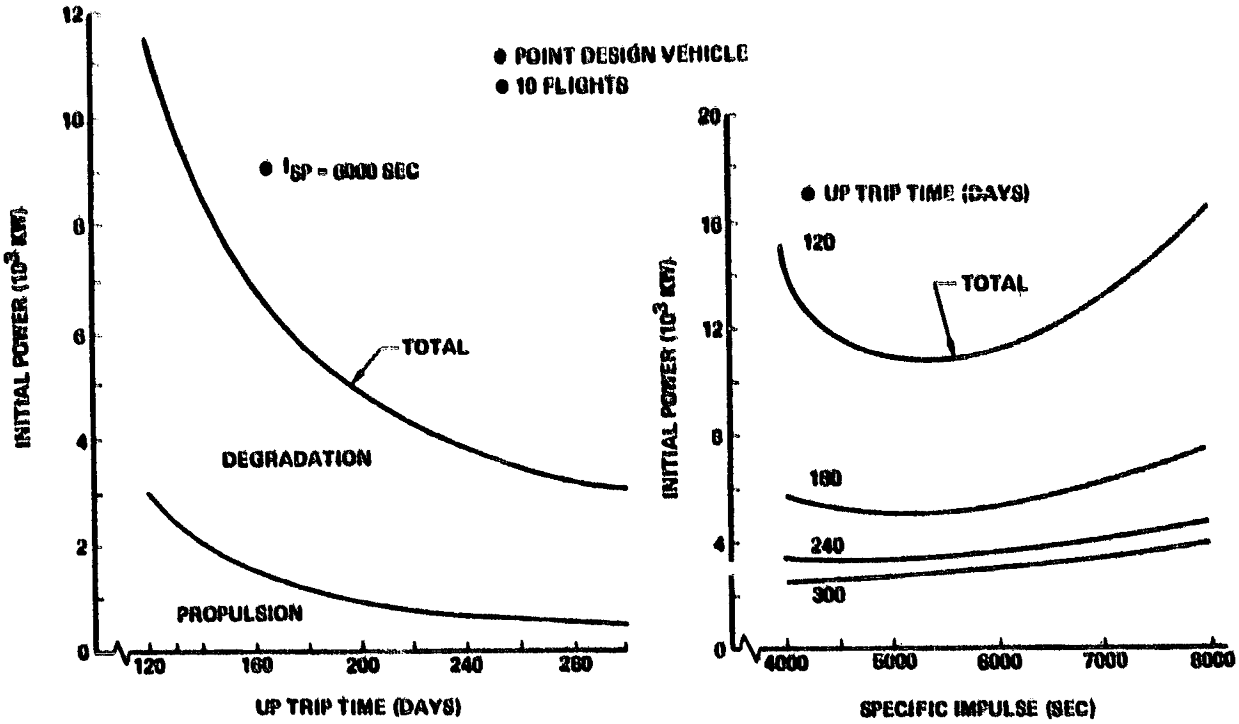

Figure 4.3.1.5-3 EOTV Power Optimization

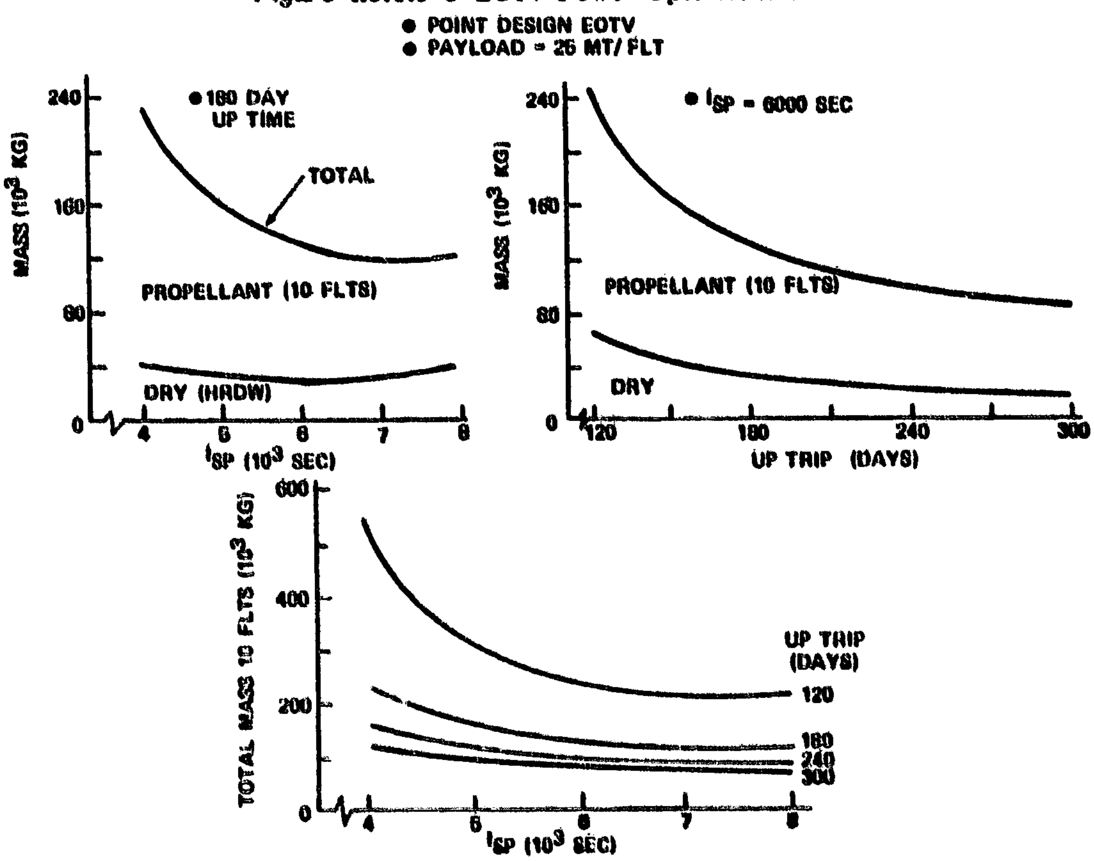

Figure 4.3.1.5-4 EOTV Mass Optimization

to less acceleration required. The lower plot shows the combined effect and indicates the least mass occurs with long trip times and high Isp. The cost optimization of the point design is presented in figure 4.3.1.5-5, again from several viewpoints. The left-hand plot

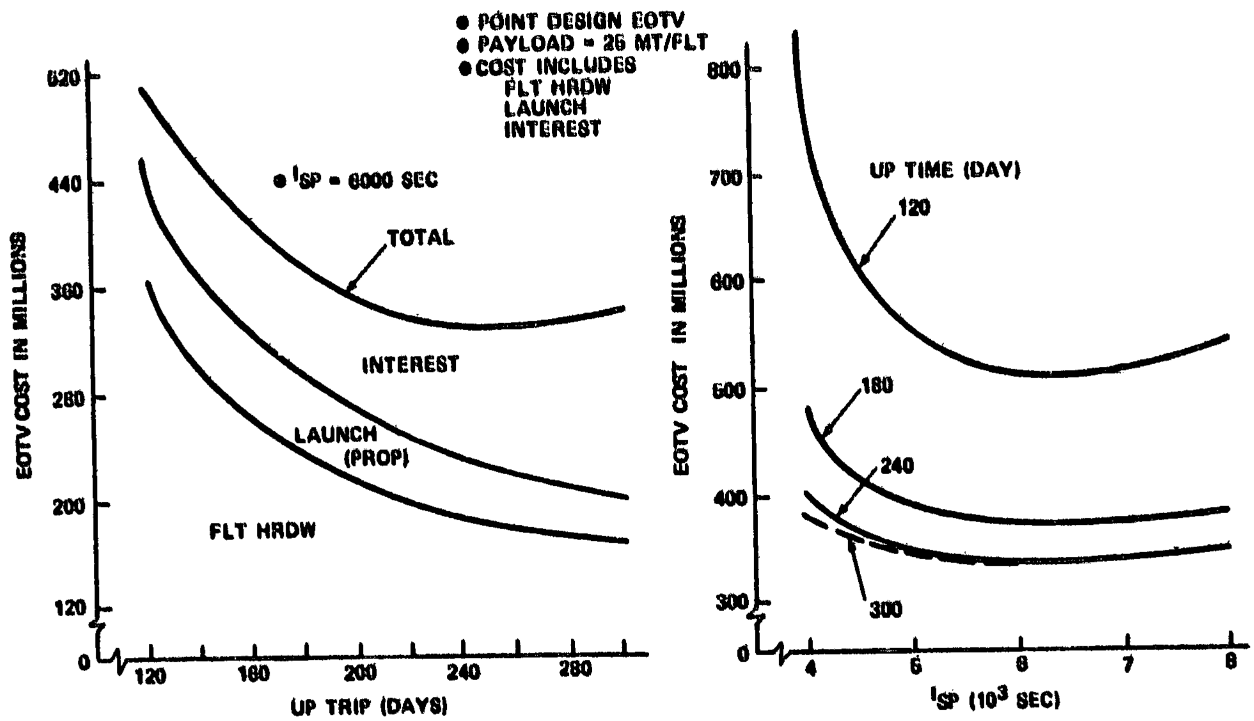

Figure 4.3.1.5-5 EOTV 10-Flight Design Life Recurring Cost Optimization

emphasizes the influence of trip time. If considering only the traditional elements of costing such as hardware and launch, longer trip times are better. As indicated in the guidelines, however, in the case of long trip times, the cost of borrowed money expressed as interest should also be included. Once this factor is applied, the optimum trip time moves back to 220-240 days. It should also be noted from this plot that the cost of hardware (one EOTV) is much greater than that of launching its propellant (even for 10 flights), which is dramatically different from chemical OTV costs. The right-hand plot emphasizes the effect of Isp for several trip times. In this case, the cost optimum for all trip times occurs with an Isp of 6000 sec; however, the cost does not vary significantly between 5000 and 8000 sec.

In summary, the point design, when using an up-trip time such as 180 days, requires minimum power with an Isp of 5000 sec, has a minimum mass with 8000 sec, and is cost optimum at 6000 sec. The cost optimum Isp, however, occurs with up-trip time of 240 days.

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Other Self-Power Options - Other options which use self-power include the heavy shielding and CR = 2. Each of these exhibit optimization trends similar to those shown for the point design. Their cost optimization occurred with Isp's between 6000 and 8000 sec and up times of 220 to 240 days. The arc jet option was analyzed only for an Isp of 900 sec and optimized also at 240 days.

Chemical Assist Option - Because this mode was considerably different from the self-power mode, inasmuch as a large portion of the Van Allen belt was to be avoided, it was thought the optimization would occur at a different Isp and trip time. In addition, there was the factor of considering several delivery modes and altitudes.

The cost optimization for the chemical assist option is shown in figure 4.3.1.5-6. In the case of a chemical OTV assisting in delivery and return from an altitude, the higher

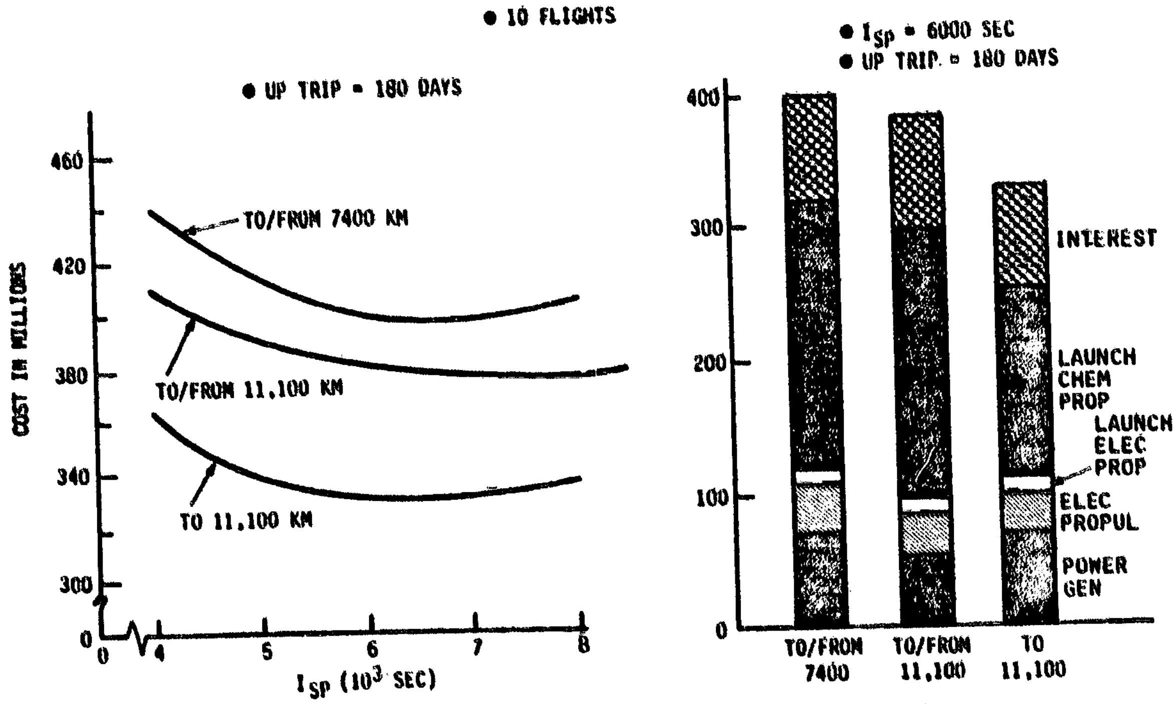

Figure 4.3.1.5-6 Chemical Assist EOTV Cost Optimization

altitude optimized at a higher Isp and provided a lower cost. This occurs because the array is smaller and its cost savings more than offset the launch cost of additional chemical OTV propellant, as shown in the right-hand plot.

Chemical OTV assist only for delivery to the high altitude provided the least cost. The array in this case was not penalized too excessively because the down leg flown by the EOTV without payload is relatively fast (approximately 1/4 the upleg to the same

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altitude). The savings in chemical propellant were significant, however, since it only had a payload for one leg of its trip.

In summary, chemical OTV assistance only for the delivery leg is preferred. A staging altitude of 11 100 km is probably near optimum since very little Van Allen proton radiation is present. At higher altitudes, electrons and solar flares become the elements of major concern.

4.3.1.6 Option Comparison and Selection

This section presents a summary comparison of the normal growth technology EOTV options. Data for all options are presented using 180 day up-trip time. Although this trip time is not cost optimum, the cost penalty is small and the duration is judged to be more acceptable from a users' standpoint. All options use an Isp of 6000 sec, except arc jet which used 900 sec.

Power Requirement - The power requirements for the options are presented in figure 4.3.1.6-1. The propulsion power is that related to the Isp and trip time while the

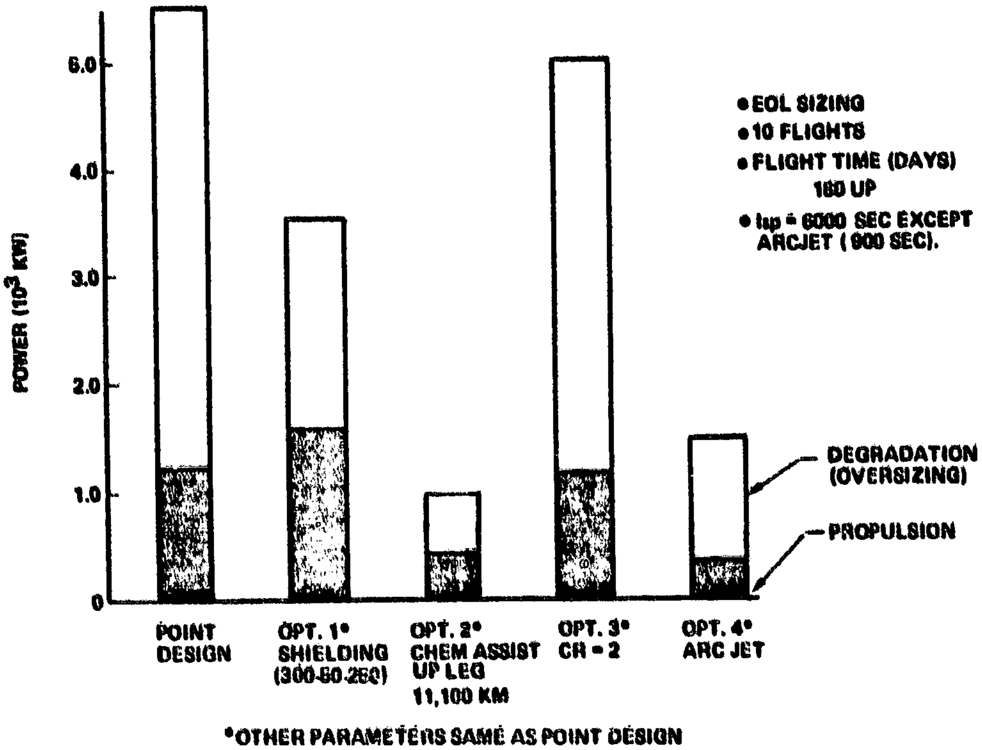

Figure 4.3.1.6-1 EOTV Power Requirements

degradation reflects the oversizing to compensate for radiation damage. The heavy shielding option (No. 1) required the largest amount of propulsion power because it is the heaviest vehicle. The arc jet option needed the least amount of propulsion power. From the standpoint of percentage of oversizing, the heavy sh ding and chemical assist options are almost equal. The heavy shielding option minimizes the amount of radiation reaching the cell while the chemical assist option avoids the major portion of the radiation fluence.

Vehicle Size - A comparison of the vehicle size which results from the power requirements is shown in figure 4.3.1.6-2. In addition to the significance of the array size, other

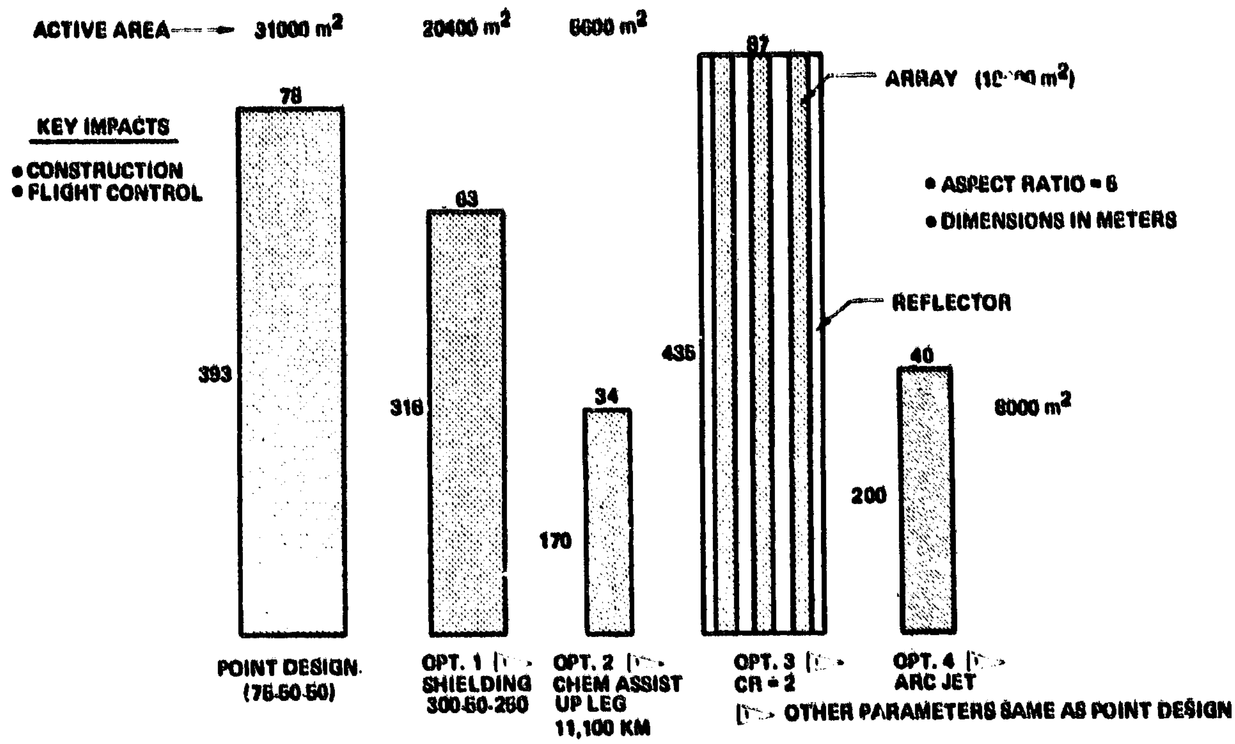

Figure 4.3.1.6-2 EOTV Options and Size Comparison

key factors related to vehicle size include construction and flight control difficulties. Due to their power needs, the chemical assist and arc jet options have the smallest size. It is also of interest that the CR = 2 option, although not requiring the largest amount of power, results in the largest configuration by nature of the reflectors requiring the same area as the array. Due to a higher power output per unit area, however, the CR = 2 array is about the same size as the heavy shielding option, even though its total power requirement is nearly 35% greater.

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System Mass - The system mass comparison is shown in figure 4.3.1.6-3 and reflects the dry weight of the vehicle and the propellant for the 10-flight vehicle design life. The arc-jet option, although being nearly the smallest EOTV, results in a huge 10-flight mass

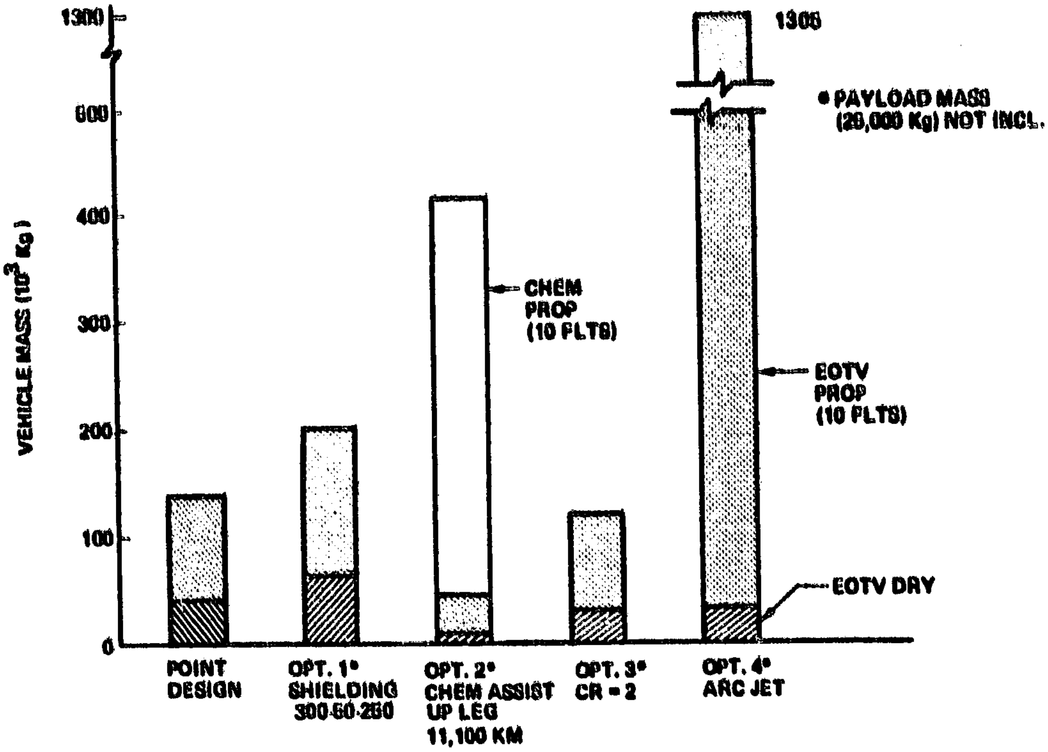

Figure 4.3.1.6-3 10-Flight System Mass Comparison

the main contributor being propellant as a result of the Isp of 900 sec and the large propellant tank mass fraction when using hydrogen. The chemical assist option also involves a considerable amount of mass. Although it is the smallest EOTV and uses the least amount of EPS propellant (due to its reduced delta-V when starting at a higher altitude), it requires 38t of chemical propellant each flight to deliver the EOTV to its starting altitude. The least massive system was the point design, even though the heavy shielding option was smaller. Again, this occurs because the vehicle is less massive per unit area which results in less propellant.

Cost - The final comparison of the options involves the cost to procure one EOTV and have it perform 10 flights. These data are presented in figure 4.3.1.6-4. Although large variations occurred with the other comparison parameters, relatively little difference exists in cost with the exception of the arc-jet option. The biggest contributor for the arc

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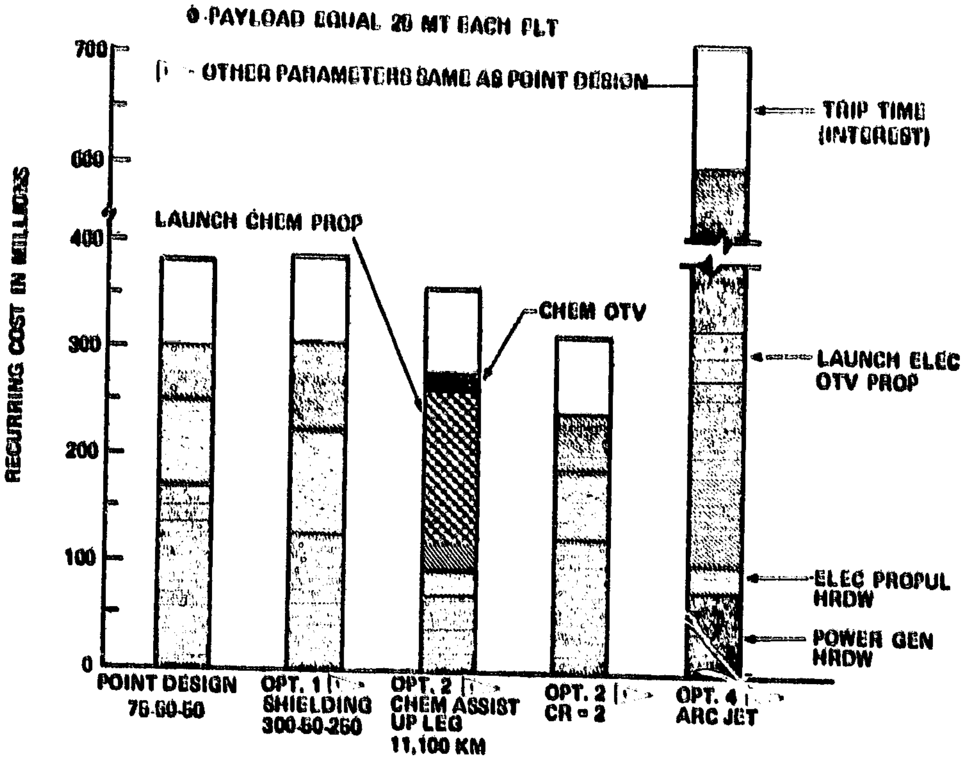

Figure 4.3.1.6-4 10-Flight System Cost Comparison

jet was the launch of the large amount of hydrogen proepllant. The chemical assist option provides the lowest EOTV hardware cost; however, the launching of the chemical propellant increases the cost to nearly the same as the point design and heavy shielding option. As compared with the point design, the heavy shielding option required less array and, thus, less power generation cost; but because of being heavier, it required more EPS hardware and propellant. The CR = 2 option results in the least cost primarily by virtue of its high power output per unit area and relatively low weight resulting in a relatively small amount of propellant.

Assessment and Selection - An overall assessment of the options based on the preceding comparison parameters and other factors indicates the following. Only the arc-jet option is ruled out due to cost. The CR = 2 option is not preferred because it is more difficult to construct, has performance uncertainties relative to the reflector, and most likely (although not determined) would have a higher DDT&E cost because of its design complexity (V-ridge reflectors must be integrated in with the array). The chemical assist option, having the lowest cost of the remaining options, presents an additional operational

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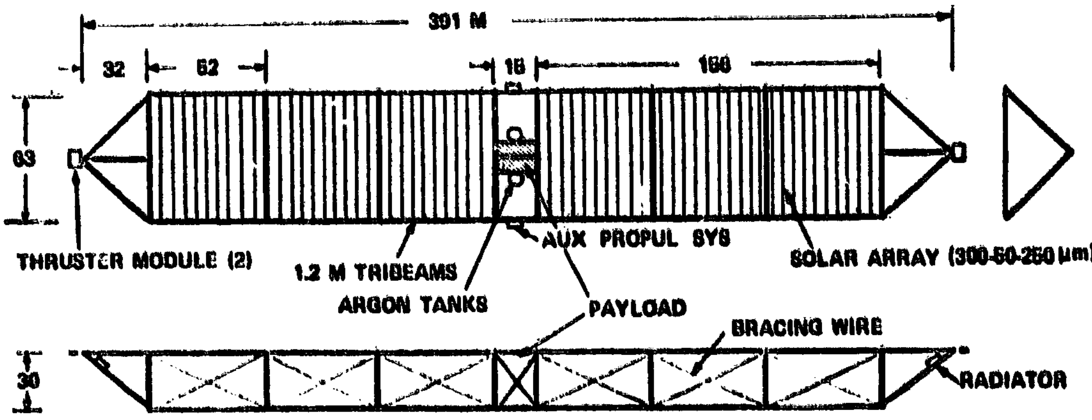
complexity as compared with the self-power modes. A key advantage of this option, however, is that it has less sensitivity to trip times as fast as 120 days. Both the point design and shielding options could be improved in terms of reduced size, with no impact on total fleet cost, by decreasing the mass of the payload to 12t to 15t (there would be more vehicles but they would be smaller in size). The shielding option, however, is judged to be more forgiving relative to uncertainty of the radiation impact and currently requires 30% less array.

The recommendation is, therefore, to use the heavy shielding EOTV option in the normal growth comparison of electric versus chemical OTV's. Furthermore, this concept and the point design with lightweight array will be assessed for improvements resulting from accelerated technology. It should be stated, however, that the chemical assist option is still a reasonable alternative; it is not being eliminated but rather "shelved" due to limitations on the number of options which can be further investigated at this time.

4.3.1.7 Selected EOTV Description

This section provides a summary of the design and operational features associated with the selected normal growth EOTV. This system will be used in the normal growth technology comparison of electric versus chemical OTV's.

Configuration Description - The configuration and key characteristics of the selected EOTV are shown in figure 4.3.1.7-1. In general, the design approach for the subsystems is



- PAYLOAD UP = 25 MT
- PAYLOAD DOWN = .0
- SPECIFIC IMPULSE = 6000 SEC
- UP TRIP = 180
- DOWN TRIP = 112

- INITIAL POWER = 3600 KW
- MAX THRUST(EOL) = 38N
- FIXED MASS = 61 MT
- ARGON MASS = 14.8 MT
- ARRAY AREA = 19,600 M^2
- NO OF THRUSTERS = 110 (60 CM)

Figure 4.3.1.7-1 EOTV Normal Growth Reference Configuration the same as described for the point design in section 4.3.1.4. The system is sized to deliver 25t in 180 days, using an Isp of 6000 sec. The BOL power is 3600 kW (1600 kW EOL), which requires 19 600 m^2 of 300-50-250 µm array. The main propulsion modules are mounted on the centerline of the vehicle at each end through means of a yoke and gimbal system which allows them to be properly directed and operate whenever the vehicle is generating power. The modules contain a total of 110 50-cm thrusters, producing 38N of thrust. The modules also contain 110 power processing units. The solar array is designed so one-half is dedicated to each main propulsion module. Auxiliary propulsion modules are located at the vehicle center on the lateral axis to provide roll control during flight, stability during occultation, and stationkeeping. The framework is made up of space-fabricated tribeams. Payload and propellant are located at the center of the vehicle to provide the most optimum moment-of-inertia characteristics. The total vehicle dry weight is 51t, of which 14.5t is propellant.

A mass breakdown of the configuration is shown in table 4.3.1.7-1. The solar array represents nearly 70% of the dry mass, partly due to its heavy shielding. All dry masses reflect a growth allowance of 15%.

Table 4.3.1.7-1 Normal Growth Reference System EOTV Mass Summary

ITEM	MASS (M.T.) ▷		DESCRIPTION
POWER GEN & DISTRIB.		(37.9)	
SOLAR ARRAY	33.4		300-50-250 µm BLANKET, 19600 SQ. M
PRIMARY STRUCTURE	3.0		1.2 M TRI BEAM, 2700 M
DISTRIB. & CONT.	1.5		BUSES & SWITCHGEAR, 1500 VOLTS
ELECTRIC PROPULSION		(10.8)	
POWER PROCESSING	6.0		(110) 15 KW UNITS
THRUSTERS	1.7		(110) 50 CM UNITS
THERMAL CONTROL	1.2		ACTIVE RADIATOR, 210 SQ. M
PROP. STOR. & FEED	0.6		
STRUCT & MECHANISMS	1.3		GIMBALS, YOKE, PANELS
AVIONICS		(1.5)	G & N, COMMUN, DATA MGT
SECONDARY POWER		(0.5)	4 REGENERATIVE FUEL CELLS
AUXILIARY PROPULSION SYS.		(0.2)	N_2H_4 SYSTEM
FIXED WT.		(50.9)	
ARGON PROPELLANT		(14.8)	
MAIN IMPULSE	14.5		
RESID & RESERVE	0.3		
N_2H_4		3.3	

▷ EACH EQUIP ITEM INCLUDES 15% MARGIN FOR GROWTH ALLOWANCE

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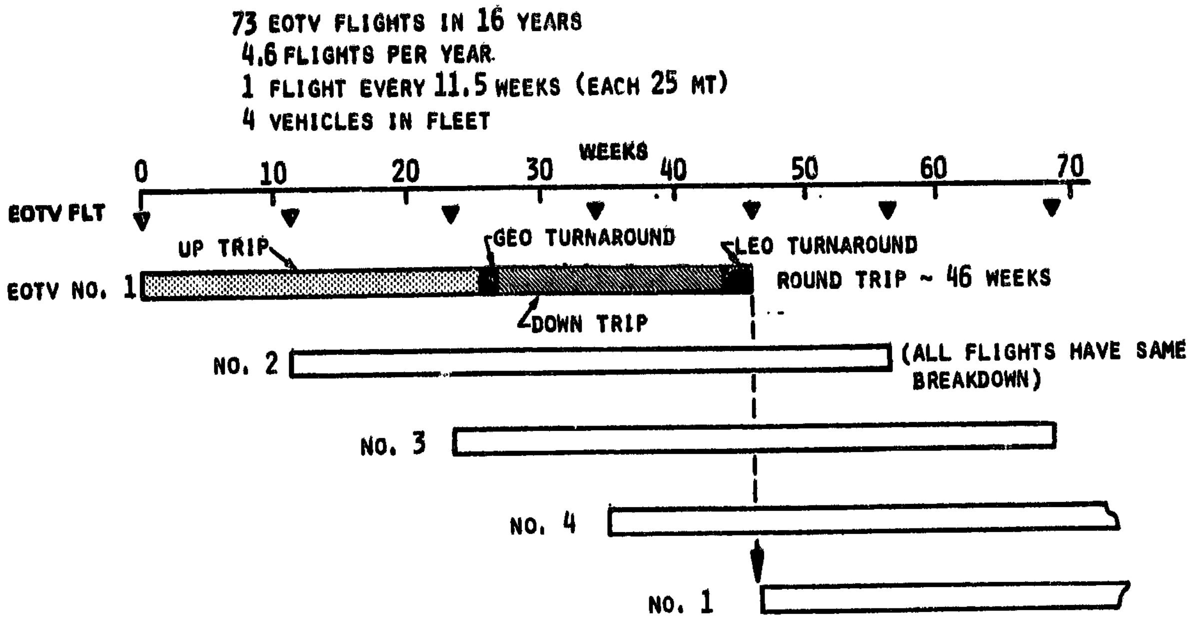
Operations - The flight scheduling (utilization) and fleet sizing of the EOTV's are indicated in figure 4.3.1.7-2. Based on a total of 73 flights over 16 years, a total of four vehicles are required in the fleet at a given point in time. Each flight consists of a 26-week up trip, 1 week allocated to offloading payloads at GEO and any unscheduled maintenance, a downtrip of 16 weeks (dictated by the amount of thrust provided for the



- 10 FLIGHTS PER VEHICLE
- 8 VEHICLES REQ'D FOR ALL MISSIONS

Figure 4.3.1.7-2 EOTV Flight Scheduling and Fleet Sizing

up trip), and 2 to 3 weeks for LEO operations, for a total of 46 weeks from the beginning of one flight to the initiation of another for a given EOTV.

A more detailed description of the operations which occur at LEO is indicated with figure 4.3.1.7-3. During this time period, the EOTV stationkeeps near SOC, rather than being docked, due to its physical size. The EOTV maintains a gravity gradient stabilized attitude and minimum drag profile to minimize propellant requirements. The major turnaround tasks involve vehicle refurbishment and loading of propellants and payload. The stationkeeping propellant expenditure of 29 kg/day is based on the data shown in figure 4.3.1.7-4, using a nominal atmosphere.

A concern associated with basing a large but low-thrust vehicle at LEO is whether or not sufficient thrust is available to overcome drag (worst case) as the vehicle begins its orbit transfer. This situation is further complicated by the fact that at the low altitudes,

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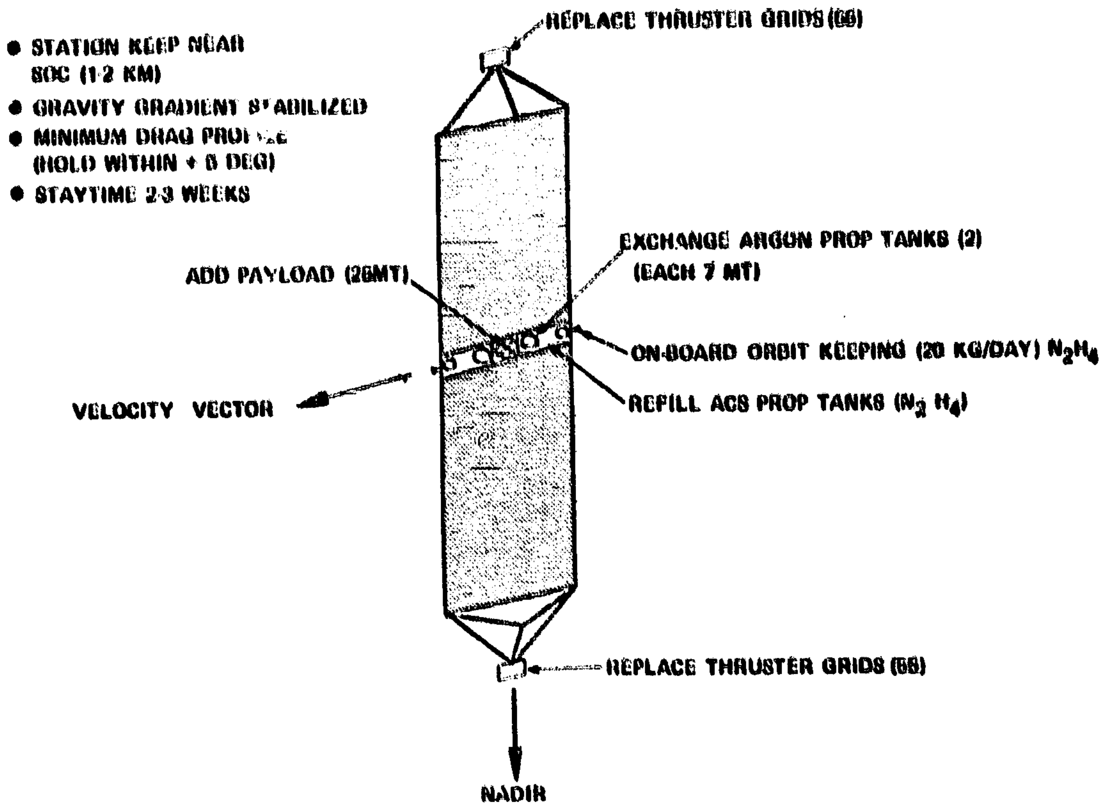

Figure 4.3.1.7-3 EOTV LEO Operations

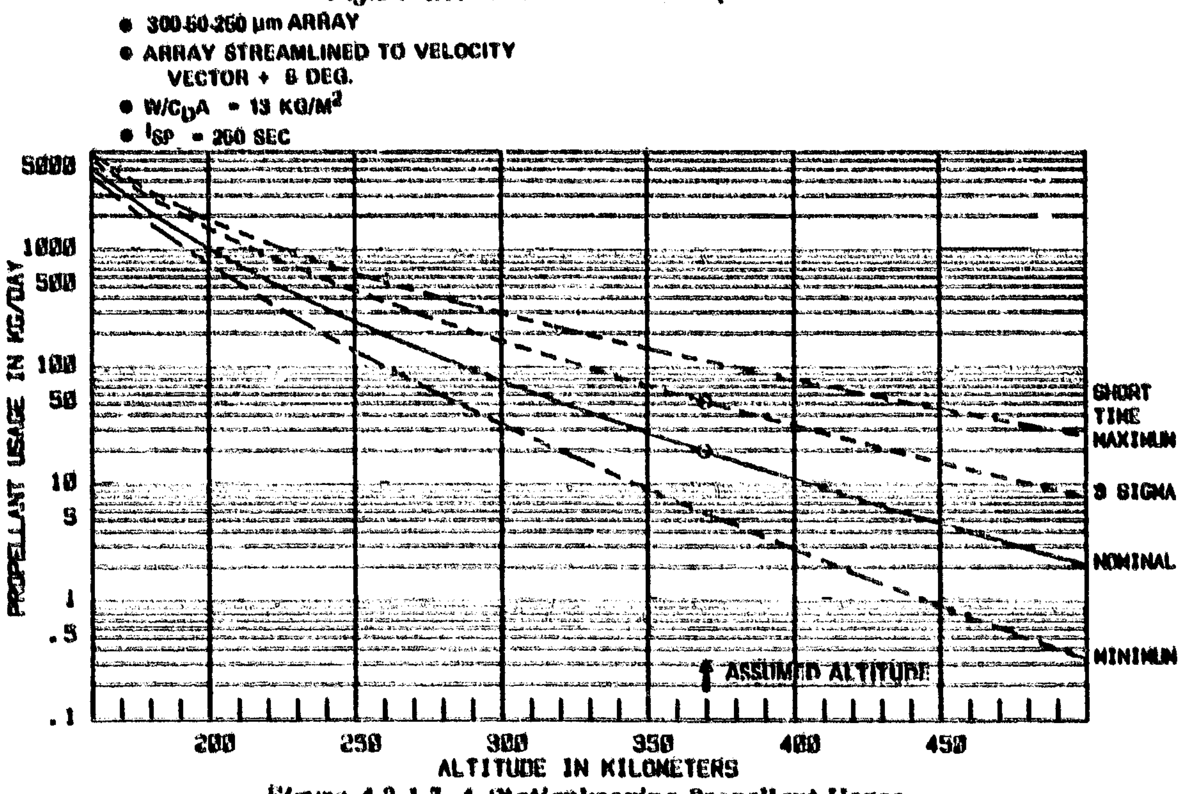

Figure 4.3.1.7-4 Stationkeeping Propellant Usage

plasma-losses also occur which decrease the amount of power available for thrusters. Data pertaining to this topic are presented in figure 4.3.1.7-5. When considering the worst-case, which is the beginning of the 10th flight, a nominal atmosphere would require 6N of thrust. Even with a 14% allowance for plasma loss, a total of 33N is available; therefore, no problem exists. This thrust level, in fact, is sufficient for even the worst atmosphere-density case.

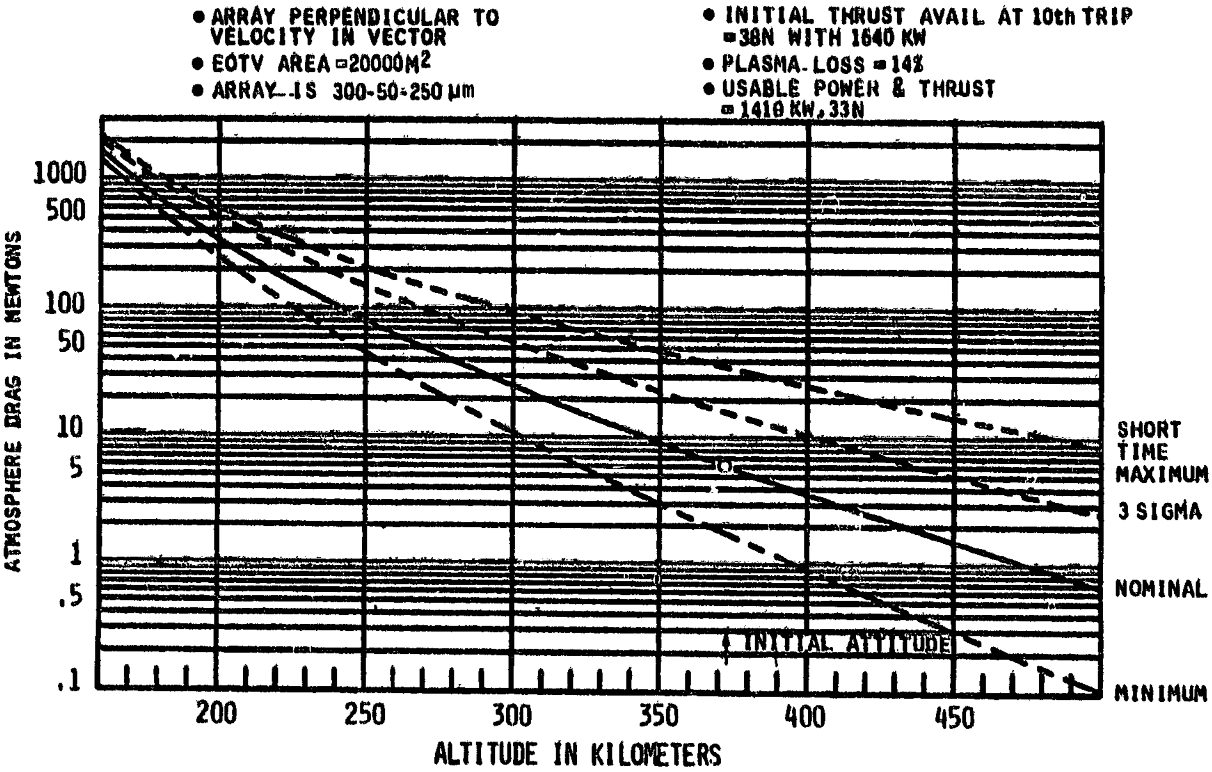

Figure 4.3.1.7-5 EOTV Climbout Drag

4.3.1.8 Cost

This section presents the DDT&E, average unit, and flight operations costs for the selected normal growth EOTV.

<u>DDT&E</u> - The DDT&E for the selected EOTV is estimated to be $900M. This value was not determined by use of the PCM cost model due to the lack of appropriate data base. Instead, use was made of scaling relationships with the SEPS vehicle. In this method, the vehicle is divided into solar-array and non-solar-array elements and the cost determined for each. The total DDT&E cost was the sum of these two elements.

The DDT&E cost of the solar array was found by scaling area rather than mass because the bulk of the 300-50-250 μm array consists of relatively low cost glass. The equation used was as follows:

$$\text{EOTV array cost} = \text{SEPS array cost}\left(\frac{\text{½ array area of EOTV}}{\text{total array of SEPS}}\right)^{0.5} \times 1.2$$

One-half of the EOTV array area was used since both sides are identical. A 1.2 factor is applied, however, to cover integration and total system aspects of the array. The 0.5 exponent is a rule of thumb used in DDT&E estimating.

The DDT&E cost of the non-solar-array (S/A) components was found by scaling mass in the following manner:

$$\text{EOTV non-S/A component cost}$$
$$= \text{SEPS non-S/A component cost}\left(\frac{\text{EOTV non-S/A mass}}{\text{SEPS non-S/A mass}}\right)^{0.5}$$

Although the indicated method of calculation is not traditional, when compared with more detailed DDT&E estimates–made for SEPS and the EOTV for SPS, the value appears to be reasonable. The key characteristics of these vehicles and their DDT&E cost estimates are shown below.

Vehicle	Power (kW)	Dry Mass (t)	DDT&E ($M)
SEPS	32	1.5	200
FOTV EOTV	3 600	50.0	900
SPS EOTV	300 000	1500.0	2400

Average Unit Cost - The key factors used in establishing the average unit cost are shown in table 4.3.1.8-1.

Table 4.3.1.8-1 EOTV Unit Cost Factors

Item	Value	Rationale
Production units	9	8 flight + 1 spare
Solar array	$6500/m^2	Production rate of 18 000 m^2/yr (see fig. 4.3.1.5-1)
Main structure	$4000/kg	Composite material
Distribution and control	$1000/kg	Mostly aluminum sheet
PPU	$1900 TFU	Scaling relative to SEPS and SPS
Thruster	$400 TFU	Scaling relative to SEPS and SPS
Thermal control (radiator)	$3900/kg	Scale to SPS
Tankage	$400/kg	Scale to chemical OTV
Avionics	$10M/set	Scale to chemical OTV
ACS	$1M/set	Scale to chemical OTV
Secondary power	$5M/set	Scale to chemical OTV

The resulting EOTV average unit cost breakdown is shown in table 4.3.1.8-2. The total cost is $361M and is comprised of $243M for the flight hardware and $118M for related support cost (each of these is a percentage of the flight hardware cost). In the case of the flight hardware, the solar array is the most dominating element, although the combination of PPU and thruster is also a major contributor.

Table 4.3.1.8-2 EOTV Average Unit Cost

- ASSUMES 9 UNITS IN PRODUCTION RUN
- COST IN MILLIONS
- 1980 DOLLARS

FLIGHT HARDWARE COST		
POWER GEN. & DISTRIB.		(146)
SOLAR ARRAY	132	
PRIMARY STRUCTURE	12	
DISTRIB. & CONTROL	2	
ELECTRIC PROPULSION		(81)
PPU	54	
THRUSTERS	19	
THERMAL CONTROL	4	
PROP. STORAGE & FEED	1	
STRUCT/MECHANISMS	3	
AVIONICS		(10)
AUX. PROPULSION		(1)
SECONDARY POWER		(5)
SUB TOTAL		243

SUPPORT COST	
ASSY & CHECKOUT	36
SUSTAIN. ENGR	10
TOOLING	24
SPARES	24
PROG. MGT	24
SUB TOTAL	118

TOTAL COST
$361M

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Flight Operations Cost - The EOTV flight operations cost was judged to be 30% higher than that associated with a chemical OTV. This resulted in a value of \$4M per flight. The increase was judged necessary to cover more command/control activity and software due to the long trip time and relatively complex transfer trajectory.

4.3.2 Chemical OTV Definition

The chemical OTV (COTV) to be used in the fleet comparison is the same as that defined throughout section 3.3 and, specifically, section 3.3.3. This is an LO_2/LH_2 space-based system sized for 32 500 kg of main impulse propellant. The configuration and key mass characteristics of the vehicle are shown in figure 4.3.2-1.

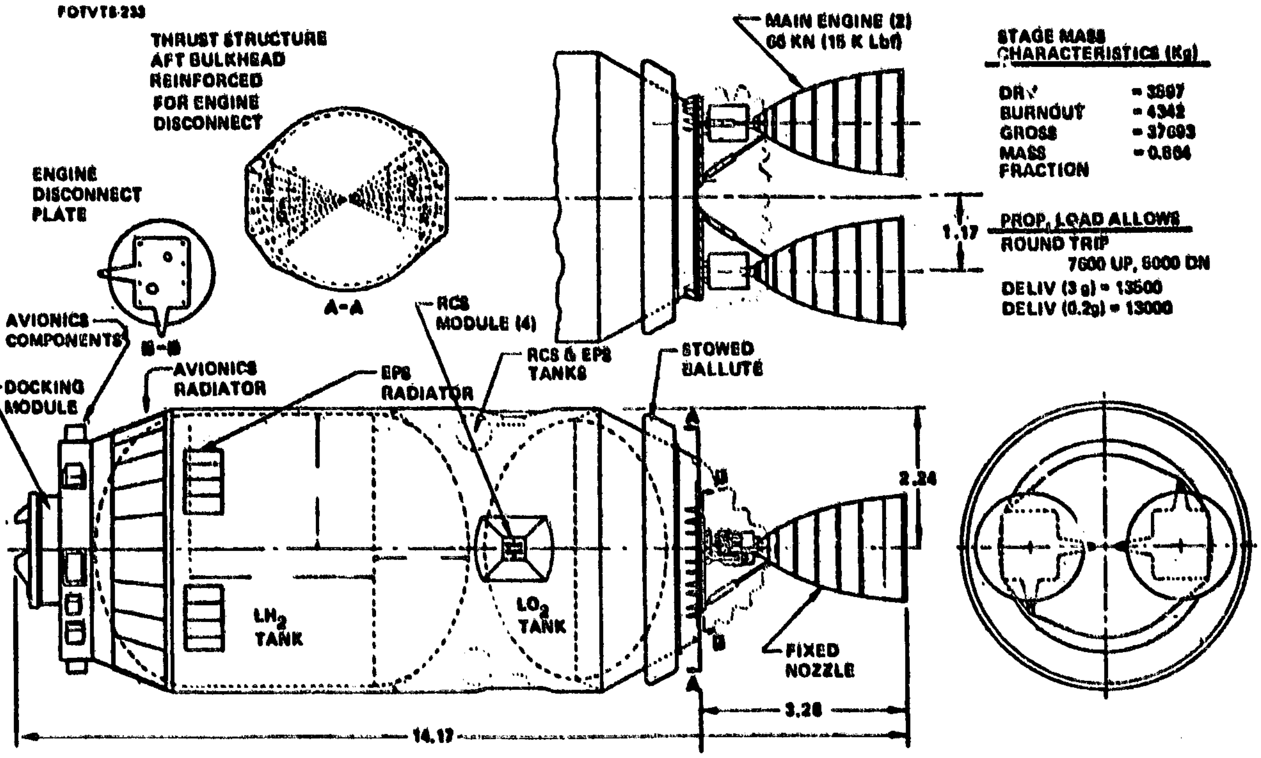

Figure 4.3.2-1 Space-Based OTV Configuration

This vehicle can be used both as a single- or two-stage OTV. The offloaded performance of the vehicle is shown in figure 4.3.2-2. When compared with the EOTV in terms of propellant required for delivery of 25t, the COTV requires approximately 54t, while the EOTV requires 15t.

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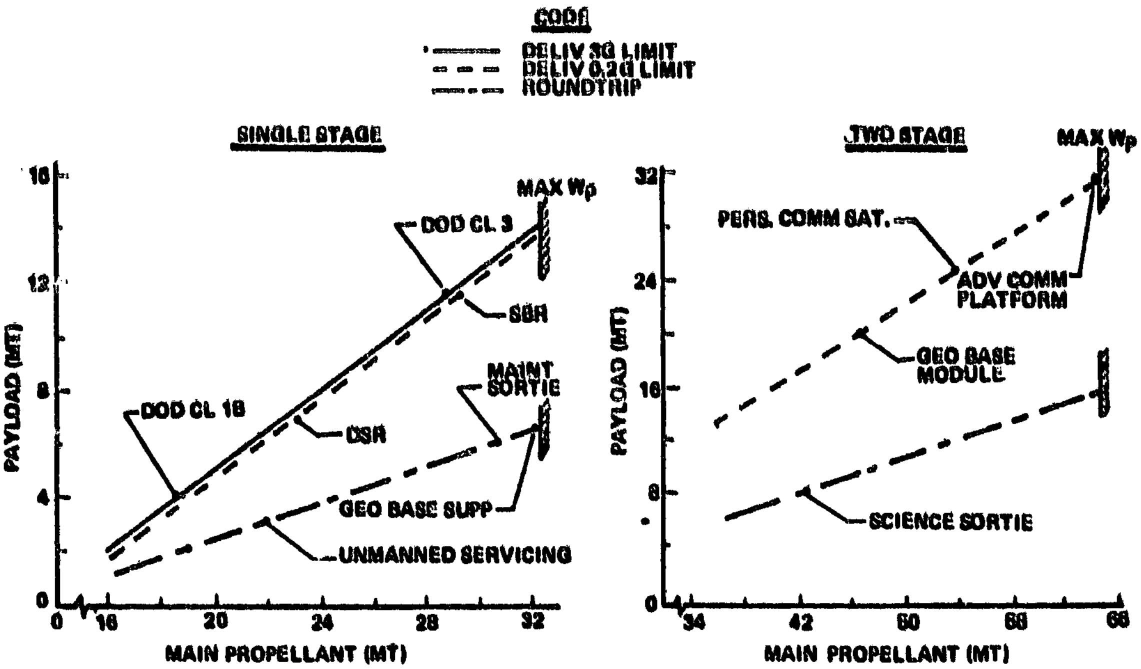

Figure 4.3.2-2 Offloaded SB LO_2/LH_2 OTV Performance

4.3.3 Launch Operations

The launch system employed in the OTV fleet comparison was the basic STS and its solid derivative cargo version with reusable payload system, as defined in section 3.3.11. The STS had a gross payload capability of 29t to 370 km and the SDV/SRB/RPS, a payload of 60t.

Launch requirements for the 16-year mission model are presented in table 4.3.3-1 and launch system assignments are in table 4.3.3-2.

Table 4.3.3-1 Launch Requirements

Crew Launches		(112)
LEO/GEO base	64	
GEO base	48	
Cargo (t)	EOTV + COTV	COTV Only
Manned base payloads	2 080	2 080
GEO payloads	2 600	2 600
EOTV propellant	1 120	-
COTV propellant*	6 800	10 900
Stages	410	40
Tankers and ASE (dry)	1 020	1 640
	14 030	17 260

*Assumes subcooled propellant storage concept (4% for propellant handling and transfer losses).

Table 4.3.3-2 Launch Assignments and Flights

STS (same as both OTV options)	
Crews (8 people per flight)	
Base and crew supplies	2080
GEO payloads	1220
	3300t (in 112 flights)
SDV (all remaining cargo)	
For EOTV + COTV fleet at 60t per flight	10 730t = 178 flights
All COTV fleet at 60t per flight	13 960t = 231 flights

4.3.4 OTV Fleet Comparison

4.3.4.1 Key Factors

The key factors associated with the comparison of an OTV fleet comprised of all-chemical OTV's versus electric plus chemical OTV's is presented in table 4.3.4-1. For

Table 4.3.4-1 Cost Comparison Key Factors

- OTV COMBINATIONS
  - EOTV FOR TRIP INSENSITIVE PAYLOADS PLUS COTV FOR ALL OTHER PAYLOADS
  - COTV FOR ALL PAYLOADS
- PAYLOAD MANIFESTING
  - TRIP INSENSITIVE PAYLOADS CAN BE MANIFESTED ON A COTV THE SAME AS EOTV
  - ALL OTHER PAYLOADS TRANSPORTED INDIVIDUALLY
- KEY CHARACTERISTICS

	EOTV + COTV	COTV	
OTV FLIGHTS			
EOTV	73	—	
COTV	133	200	
OTV PROD UNITS (1)			
EOTV	8	—	(~$ 360 M/UNIT)
COTV	6	10	(~$ 30 M/ UNIT)
LAUNCH FLIGHTS			
STS	112	112	(CREWS, LOGISTICS, 40% PAYLOADS)
SDV/HPS	178	231	(60% PAYLOADS, PROP, VEHICLES)

(1) WEAROUT ONLY

sake of simplicity, it was assumed that the 25t of payload delivered by an EOTV could also be delivered by a COTV. Consequently, both fleet options involved a total of 266 OTV flights. In the all-COTV option, a total of 136 flights require two stages. The COTV used in the mixed fleet involved 63 flights using two stages

The number of production units reflect wearout and backup needs. Wearout is based on the total number of stage flights and a design life of 45 flights for COTV and 10 flights for the EOTV. One backup stage is included for the COTV. Eight EOTV's are necessary from the delivery rate standpoint and provide the capability for 80 flights. Only 73 flights are required so, in effect, an extra or backup unit is provided in terms of flight capability.

The number of launch flights was determined in the preceding section.

4.3.4.2 Life Cycle Cost

A summary of the total transportation cost associated with the two-fleet options is presented in figure 4.3.4-1. When compared for the total mission model, the all-COTV option provided a savings of approximately $3 billion or 23%. This savings results from lower DDT&E, considerably less production cost, and no delta interest cost, which more than offsets higher launch operations cost. The right-hand plot presents the cost related

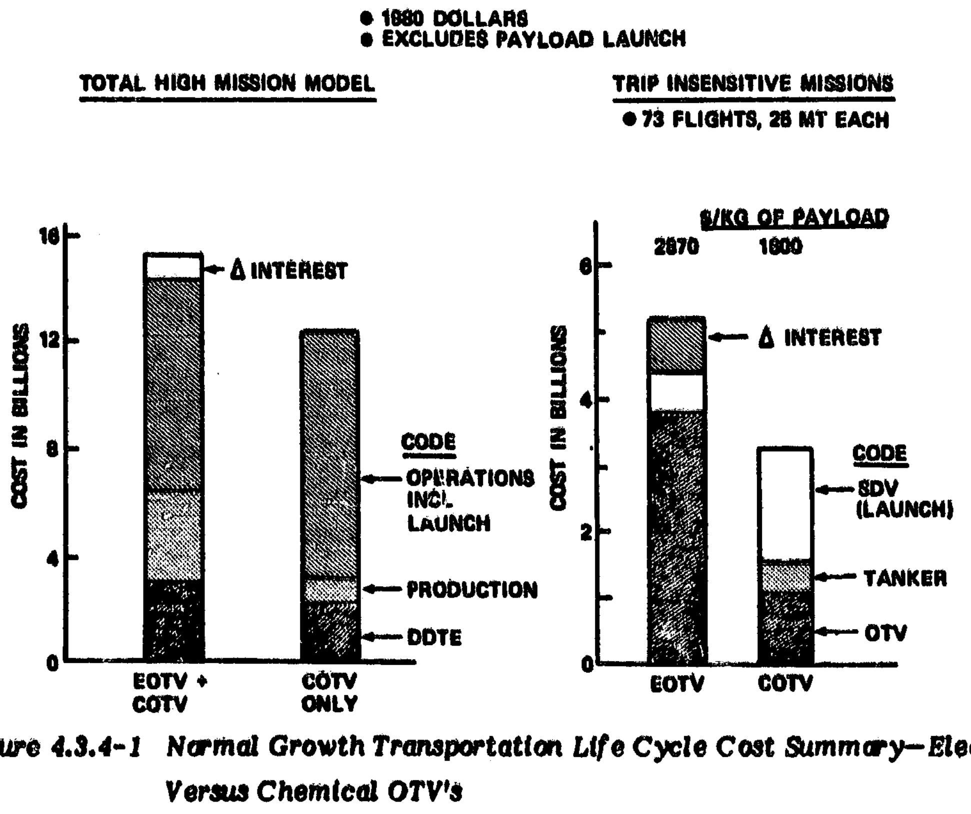

Figure 4.3.4-1 Normal Growth Transportation Life Cycle Cost Summary—Electric Versus Chemical OTV's

only to the trip insensitive payloads (73 flights) and emphasizes the differences in terms of hardware element. The high cost of the EOTV is clearly the key factor in the $3 billion deficit.

A breakdown of the costs associated with the total mission model is shown in table 4.3.4-2. In terms of DDT&E, the difference is a result of developing the EOTV. The

Table 4.3.4-2 Complete FOTV High Mission Model Transportation Cost Summary

• COST IN MILLIONS
• 1980 DOLLARS

	EOTV + COTV		ALL COTV	
DDTE		(3140)		(2240)
EOTV	900		–	
COTV	700		700	
TANKER	440		440	
SDV/RPS	1100		1100	
SOC	TBD		TBD	
PRODUCTION		(3495)		(935)
EOTV	2760		–	
COTV	210		360	
TANKER	75		125	
SDV/RPS	450		450	
SOC	TBD		TBD	
OPERATIONS		(8020)		(9205)
EOTV	290		–	
COTV	660		780	
TANKER	130		210	
SDV/RPS (LAUNCH)	3805		5080	
SOC	TBD		TBD	
STS (LAUNCH)	3135		3135	
SUBTOTAL		14655		12380
OTHER				
TRIP TIME (INTEREST)		(605)		(–)
TOTAL		15260		12380

production cost difference is dominated by the high cost associated with the EOTV. Launch costs are less with the mixed fleet, primarily because less total propellant is required and, thus, 50 fewer flights of the SDV. Again, a $3 billion net difference exists between the fleets. From a front-end-cost standpoint (DDT&E plus one-half of production), the all-COTV fleet would have a cost advantage of over $2 billion.

4.3.4.3 Sensitivity to EOTV-Compatible Payloads

As a sensitivity to the basic fleet comparison, consideration was given to arbitrarily doubling the number of trip insensitive payload flights. Since this generally meant more large platforms, more STS launches were required to support their construction. Additional EOTV flights also meant more units, which reduced the average unit cost.

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The results of this comparison are shown in figure 4.3.4-2. For this case, the all-COTV fleet increases its cost advantage to \$4.5 billion; however, the percentage difference is still 25%. The change in cost occurs as a result of the increased delta in launch cost for the all-COTV fleet, but it is more than offset by the cost delta associated with the production cost of the EOTV and its trip time interest cost.

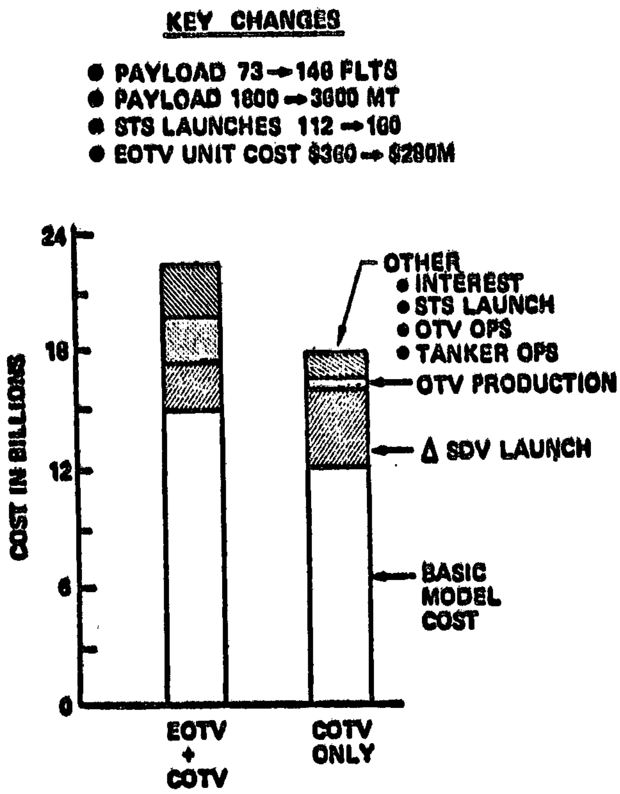

Figure 4.3.4-2 Sensitivity to EOTV-Compatible Payloads

4.3.4.4 Conclusions

Based on transportation life cycle cost considerations, when both options use normal growth technology, an all-chemical OTV fleet provides a significant advantage over a fleet consisting of electric and chemical OTV's for a mission module composed primarily of GEO payloads.

4.4 ACCELERATED TECHNOLOGY VEHICLES

This section identifies the improvements assumed for accelerated technology, characterizes and compares the system design options, and reassesses the OTV fleet comparison using the best possible EOTV.

The major emphasis of this analysis was the definition of an accelerated technology EOTV. Specifically, the goal was to try to reduce the unit cost of the system. Based on the results of the normal growth analysis, the key to achieving this goal was to considerably reduce the cost contribution of the solar array through the use of accelerated technology.

4.4.1 Accelerated Technology Projections

Accelerated technology projections pertaining to the solar array focused on two areas: (1) improved solar cells in terms of performance and/or cost and (2) solar array annealing which would effectively reduce the amount of oversizing required and thus the cost. The projections for these two areas are described in the following paragraphs.

4.4.1.1 Solar Cells

Survey of the literature reveals a wide variety of advanced cells are being investigated by the photovoltaic industry. Some of these include gallium arsenide (GaAs), vertical junction silicon, multiband gap (tandem) cells, and thin films (currently considered for terrestrial application). Consideration of an advanced cell for EOTV application, however, requires its characterization in terms of performance, radiation sensitivity, annealability, and cost. Based on this criteria and the existing data base, only the GaAs cell was judged to be adequately (marginally) characterized. Accordingly, the only accelerated technology cell to be used in analyzing accelerated technology EOTV's is GaAs.

The key characteristics projected for the GaAs cell (1990 readiness date) are as follows:

1. Efficiency: 18%

   This reflects the overall average associated with automated production of very large quantities.
2. Thickness: 50 μm (2 mil)

   Current cells are in the 200-250 μm (8-10 mil) range. SPS studies indicated thin film cells on the order of 5-10 μm in thickness; however these were judged to be not available for 1990.
3. Junction depth: 0.3 μm

   Ranges of 0.3 to 1.0 μm have been considered; however, the lower value considerably improves radiation resistivity.
4. Size: 5 x 5 cm

   Today, 2- x 2-cm cells are being produced. As in the case of silicon cells, a larger size is possible with improved manufacturing techniques.

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5. Mass: 4.8 gm/m^2/μm of thickness

GaAs is approximately twice the density of a silicon cell. Based on a 75-50-50 μm blanket, the GaAs array is 540 gm/m^2 versus 427 gm/m^2 for silicon.

6. Cost: see figure 4.4.1-1

Based on information in reference 17, today's GaAs cells cost \$3/cm^2 more than silicon cells. It was also stated that larger quantities (not specified) would reduce the cost difference to only \$1/cm^2. Other elements of an array, such as coverglass and substrate, could be the same as for the silicon cell array. Therefore, using the lowest indicated GaAs cell cost, the total array results in 1.5 times the cost of a silicon array for the same given annual production rate.

7. Radiation sensitivity: (see fig. 4.3.1.3-8)

These data indicate that for one round trip, the GaAs cell will have a P/Po of 0.52 versus 0.42 for the silicon array when both use a 75-50-50 μm blanket.

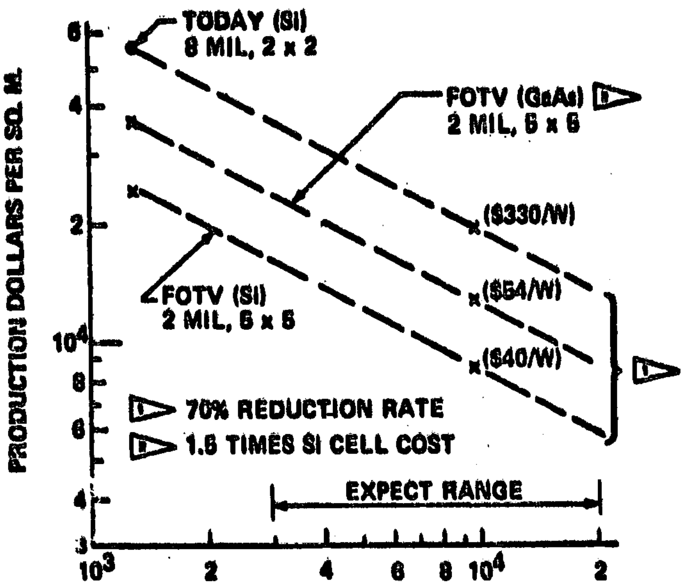

Figure 4.4.1-1 Solar Array Cost

4.4.1.2 Annealing

Background - An alternative to heavy shielding to minimize array degradation is that of thermal annealing. Stated simply, this means subjecting the irradiated solar cells to elevated temperatures for certain durations with the result being the removal of a portion of the damage and, thus, restoration of the power output. This concept is illustrated in figure 4.4.1-2 using an EOTV with a 75-50-50 μm silicon array. During one round trip, the array receives a fluence of 10^{17} of 1-MeV electron equivalence, resulting in a power

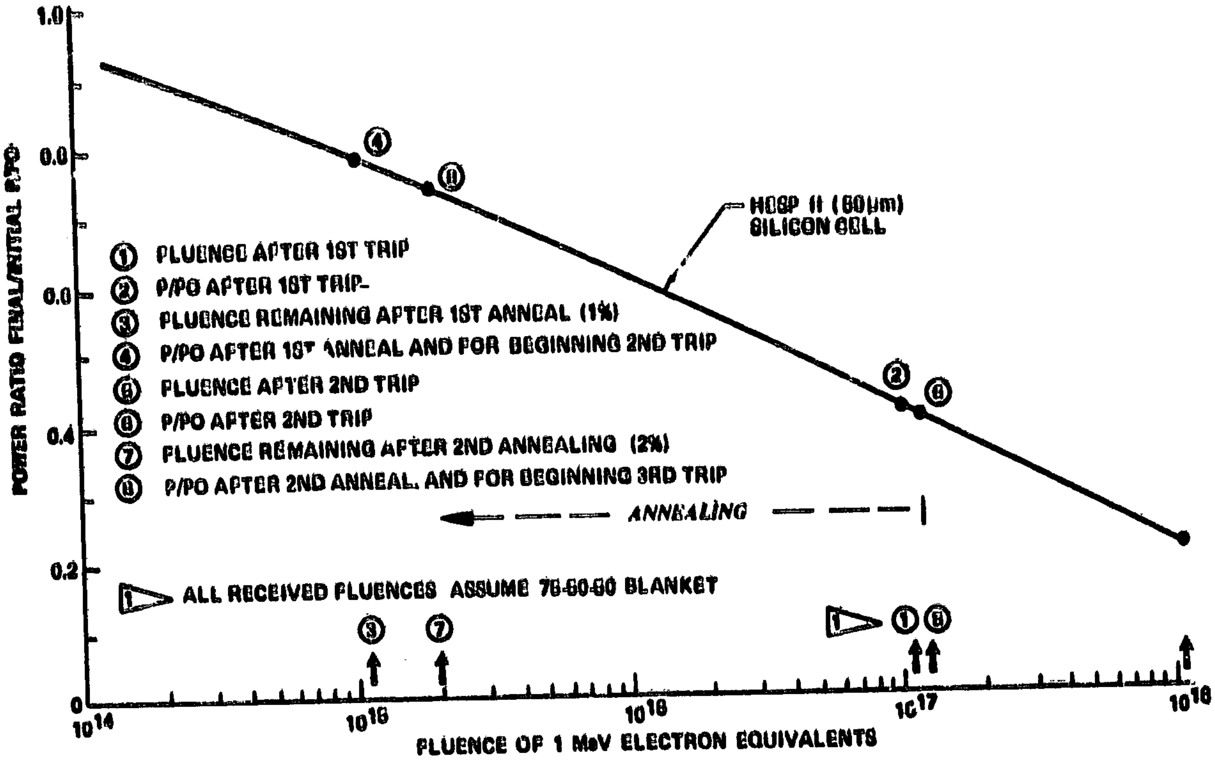

Figure 4.4.1-2 Radiation/Annealing/Power Relationships

decay to about 40% of the initial output. If the annealing operation removes all but 1% (assumed) of the displacement damage (used synonomously with fluence), a fluence of 10^{15} will remain in the array and the power output will be restored to 80% of the initial output. Power output after subsequent degradations and annealings can be found by repeating the indicated technique. A key factor, however, is that the amount of fluence (damage) which cannot be removed by a given annealing is additive to the amount received on the next trip. In other words, the damage gradually accumulates even after annealing.

The key issue involved in the annealing operation is not whether it works but the degree of its effectiveness in terms of how much damage (fluence) is removed. Projections regarding effectiveness are somewhat difficult to obtain since very limited data are available for proton damage in silicon cells—and even less in GaAs cells. This is particularly true in the case of low-energy proton damage in GaAs cells. This area is a major concern since the GaAs cells generally have shallow active regions, which is where the majority of the damage occurs with this type of proton. It probably is also worthwhile to reestablish the concept that solar cells will experience low-energy protons during an orbit transfer (originally discussed in section 4.3.1.3). This occurs because a given coverglass can stop incident protons up to a given energy level, but protons of higher energy levels can penetrate the glass and enter the cell with lower energy levels. Therefore, since the LEO-GEO proton environment contains a wide energy spectrum, low-energy particles are always assured of reaching the cell.

Effectiveness - The annealing effectiveness assumed for this study is shown in figure 4.4.1-3 for several cell types and operating conditions. In general, the values indicated are based on extrapolations made from data presented in references 18 through 21.

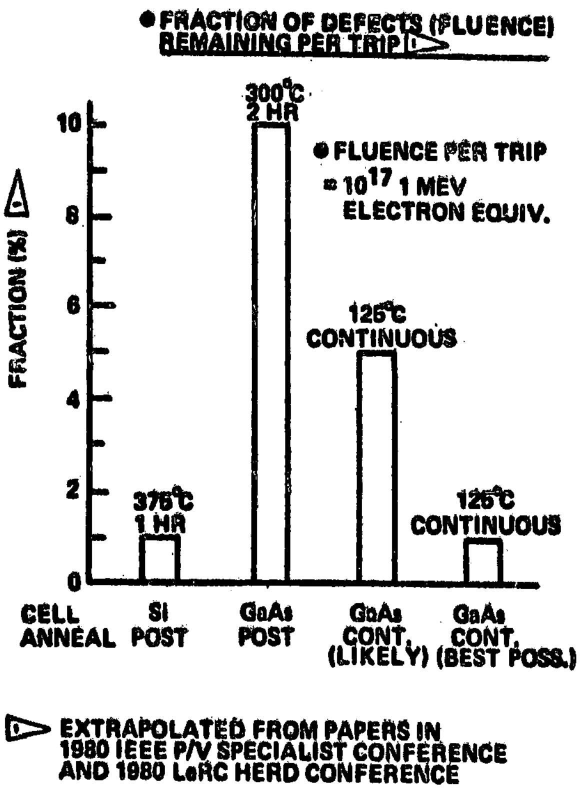

Figure 4.4.1-3 Assumed Shielding Effectiveness

In the case of a silicon cell, a postflight annealing approach is indicated, which means the annealing occurs after the total damage for one trip has been received. Investigations discussed in reference 18 found that cells exposed to 7×10^{11} p/cm^2 (250 keV) could be annealed so no more than 5% of the damage remained if a temperature of 375°C and 40 min were used. Annealing could occur even faster at 400°C; however, concern was expressed regarding thermal damage to the cell. The 1% damage remaining value is based on extending (extrapolating) the annealing time out to 50 to 60 min at 375°C.

Postflight annealing is also indicated for a GaAs cell with the result being that 10% of the defects remain after annealing for 2 hr at 300°C. This value was derived by extrapolating data presented in reference 19 to reflect a proton environment similar to that used in the silicon annealing test described previously. The apparent difference in annealing effectiveness between GaAs and silicon cells involves a variety of factors, with one of the most significant being that GaAs defects are more complex.

Several continuous annealing operations are also indicated for the GaAs cell. This method of annealing involves operating the cell at a hotter temperature than would be desired from an efficiency standpoint, but the benefit is that not so much damage occurs. Temperatures on the order of 125°C have been successful in preventing damage caused by electrons, according to reference 20. Removal or prevention of proton damage, however, may require higher temperatures. Such an approach is not used with silicon cells because their efficiencies suffer considerably at these temperatures. GaAs cells, however, are less sensitive and, therefore, are candidates for continuous operation at elevated temperatures. Use of a CR = 2 design can result in array temperatures on the order of 125°C.

The GaAs continuous case indicated as most likely shows 5% of the defects (fluence) remaining. This value was influenced to a large degree by data presented in reference 21. Extrapolation of these data for the type of proton environment discussed previously indicated an improvement over a no-annealing situation of approximately 10 to 12 points in P/Po. This, in turn, translated into receiving a fluence approximately 5% as large as that normally experienced.

A best possible GaAs value of 1% damage remaining is also suggested. This improvement was judged a possibility since the irradiation of the test cells discussed in reference 20 took place in 1 hr, which was much more rapid that that which would normally occur during a flight. Offsetting this factor, however, was the consideration that the annealing temperatures utilized in reference 20 were 150°C to 200°C. Temperatures available with the CR = 2 design would not be higher than 125°C; consequently, the annealing may not be as effective.

In summary and as indicated above, considerable extrapolation had to be done to make projections relative to annealing effectiveness, particularly in the case of GaAs cells. Any firm conclusions regarding this operation should certainly await test data which more closely simulate the environment in terms of the fluence and rate for applicable combinations of protons.

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Benefit - As indicated previously, the real benefit of annealing is that it reduces the oversizing of the array. To illustrate the benefit, an EOTV with a 75-50-50 μm silicon array is used for 10 flights, as shown in figure 4.4.1-4. Although the power ratio drops

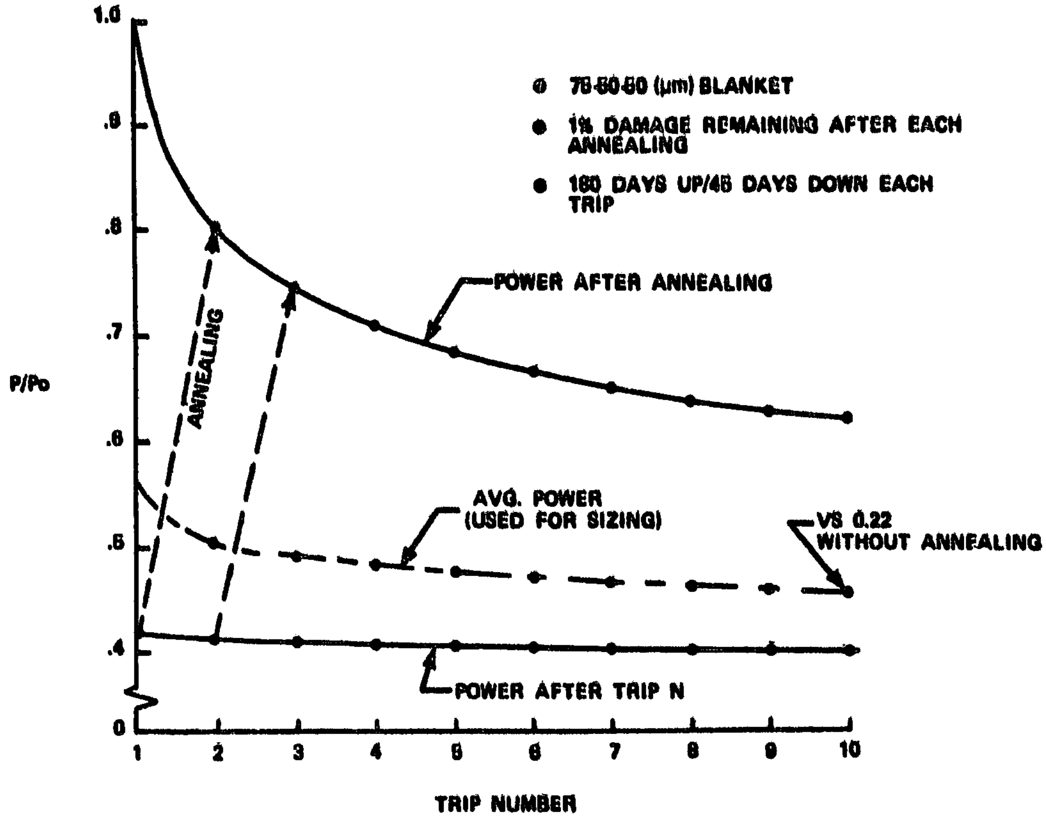

Figure 4.4.1-4 Benefit of Solar Array Annealing—Silicon Cells

down to nearly 0.4 each flight, annealing returns the ratio to a respectable level to begin the following flight. The key factor, however, is the average power (based on mission time) that is available for sizing the array remains relatively high. In this case, the sizing power ratio for completing 10 trips is 0.45 versus 0.22 if annealing is not performed, which means the size of the array is reduced by more than 50%. This same approach is used with the other annealing concepts discussed in the preceding paragraph. The power ratios of all concepts investigated are presented in section 4.4.2.

Implications - Although a benefit can result through annealing, it must also be pointed out that there are some implications. In the case of a planar array (CR = 1), several factors merit consideration. First there must be a means to elevate the array temperatures to those required. The SPS studies considered the use of laser devices which were attached to gantries that moved across the array. Other studies have considered a greenhouse approach; however, reaching the required temperatures may prove difficult. At this point it will be stated that the mass to cover the annealing devices and their cost have not been included in the analysis. As will be indicated in the final fleet comparison, however, the exclusion of these items would not alter the outcome. Another factor which applies to a silicon array is array characteristics themselves. Most important is the need to develop cells that can withstand 350°C to 400°C temperatures without being thermally damaged. Also related to the temperature is the need for electrostatic bonding of thin coverglasses to cells, rather than use of adhesives which could lead to outgassing. Progress has been made in this area during the last several years.

The GaAs array has the best potential and does not require an external annealing source if use is made of a concentrated design (CR = 2). This concept in itself represents design and construction challenges not present in a planar array. Coverglass attachment also must be addressed with this cell.

4.4.2 System Options

4.4.2.1 Characterization

The accelerated technology EOTV system options investigated were oriented to take advantage of the cell improvements and various annealing options. Three basic options were considered:

1. Option 1: silicon array with post-annealing—Both a 75-50-50 μm array (Option 1A) and a 300-50-250 μm array (Option 1B) were included to determine if annealing would make use of the lightweight array more beneficial.
2. Option 2: GaAs array with annealing—Two different annealing options were to be assessed when using the higher performance array: postflight annealing (Option 2A) and continuous annealing (Option 2B).
3. Option 3: most optimistic GaAs EOTV—This option would determine the effects of the most optimistic projections in technology and design features through use of higher performance cells, direct drive (minimum power processing), high beam current, and improved continuous annealing.

The principal performance and cost features of these options are shown in table 4.4.2-1. Differences relative to the normal growth technology EOTV are emphasized. All options continued to use 50-cm argon ion thrusters, similar configuration arrangements, and were to be optimized over a range of trip times and specific impulses. A brief discussion of the key features relative to the normal technology vehicle follows.

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Table 4.4.2-1 Accelerated Technology Electric OTV's Key Assumptions

PARAMETER	NORM. TECH.	ACCELERATED TECHNOLOGY				
		OPT. 1A	OPT. 1B	OPT. 2A	OPT. 2B	OPT. 3
• CELL TYPE	SILICON	✓	✓	GaAs	GaAs	GaAs
• BLANKET (μm) (COVER-CELL-SUBST)	300-50-250	75-50-50	✓	75-50-50	75-50-50	75-50-50
• CONCEN. RATIO	1	✓	✓	✓	2	2
• CELL EFF (AMO 25°C)	16	✓	✓	18	18	20
• POWER OUTPUT (W/M²) [1]	179	✓	✓	210	342	380
• PWR GEN (KG/KW) [3]	10.4	4.0	10.4	4.3	3.7	3.7
• ANNEALING	NO	POST	POST	POST	CONTINUOUS	CONTINUOUS
• DAMAGE REMAINING EACH TRIP [2] (% OF TRIP FLUENCE)	100	1	1	10	5	1
• P/Po AFTER 10th TRIP (%)	45	48	70	63	63	73
• PWR PROCESS (KG/KW)	3.1	✓	✓	✓	✓	1.6
• NORMALIZE COST PER CELL (SAME QTY)	1.0	✓	✓	1.5	1.5	1.5

[1] BOL REFLECTING ALL EXPECTED LOSSES EXCEPT RADIATION
[2] 180 DAYS UP/45 DAYS DOWN
✓ SAME AS NORMAL TECH.
[3] SOLAR ARRAY, STRUCTURE & DISTRIBUTION

Option 1A, using a lightweight silicon cell array, provides a lower specific mass for the power generation system. Annealing of the array improves the power ratio to a value greater than that provided by the heavy shielded normal technology vehicle. Option 1B also uses silicon cells, but heavy shielding, and with annealing improves the power ratio to 0.70.

Option 2A, using higher performing GaAs cells, has an improved power output. The specific mass of this planar array is higher than that of the silicon array because of the heavier cell. The annealing effectiveness of this option is not as good as that provided by the corresponding silicon option (1A), but this is offset by the fact that the GaAs cell did not degrade as much for a given amount of radiation. Cost for a given size array will be 50% greater than for silicon. Use of a CR = 2 design, as in Option 2B, considerably improves the power output. The specific mass of the power generation system (PGS) is lower than the planar array GaAs system (Option 2A) by the same ratio as was used in the normal growth vehicle analysis. The power ratio for Option 2B is higher because of the more effective annealing that is assumed.

Option 3 is the most optimistic design considered. This system includes a 20% efficient cell, which gives a higher power output; an annealing effectiveness, which results in the highest power ratio; direct drive, which means power is obtained directly from the array and supplied to the thruster screens to reduce the amount of power processing; and a thruster design approach, which allows a beam current of 20A to be used, resulting in more thrust for a given Isp while still satisfying burn-life constraints.

4.4.2.2 Optimization

The optimization technique employed was the same as that used in the normal growth vehicles. A key point of interest was whether the higher performance and annealable GaAs option would optimize differently than the normal growth vehicles.

The results of the optimization for Option 2A (GaAs with 75-50-50 array and postflight annealing) are shown in figure 4.4.2-1. The left-hand plot indicates an optimum

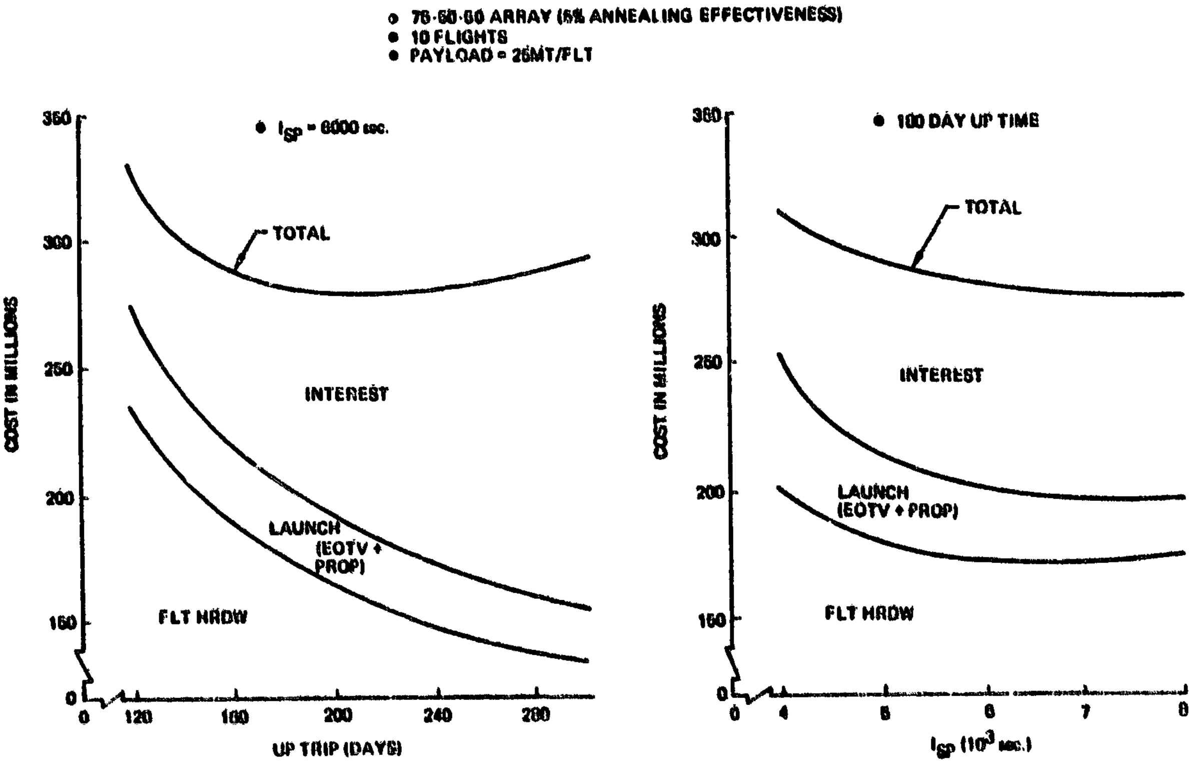


Figure 4.4.2-1 GaAs EOTV Cost Optimization

transfer time of 190 to 220 days as compared with 240 days for the normal nonannealable silicon growth vehicles. With a reference trip time of 180 days, as shown on the right-hand plot, the optimum Isp appears to be 8000 rather than 6000 sec, as occurred with the normal growth vehicles. Therefore, the higher power output vehicle with annealing optimizes at faster trip times and higher Isp than normal growth vehicles.

Optimization of the other GaAs options had similar characteristics. The annealable silicon options, however, still optimized at trip times near 240 days and 6000 sec.

4.4.3 Accelerated Versus Normal Growth EOTV

The comparison of EOTV's using accelerated versus normal growth technology was again performed for the case of one design life or 10 flights. Because cost optimizations have shown only small differences for trip times between 180-220 days and Isp between 6000-8000 sec, the lower values were used for the comparison points. The EOTV concept found to have the best characteristics would then be used in another assessment of the OTV fleet options.

The required array area for the accelerated technology EOTV options is presented in figure 4.4.3-1. All accelerated technology vehicle concepts show an improvement over

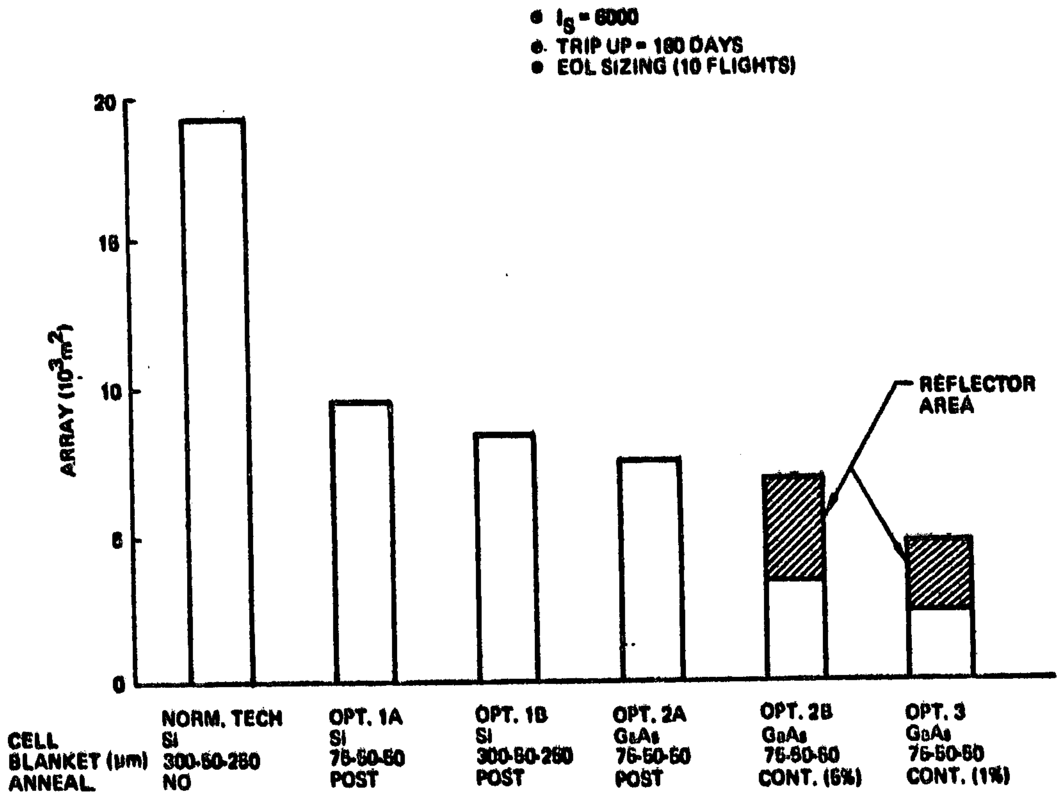


Figure 4.4.3-1 Area Comparison

the normal growth vehicle. Another point of interest is that the GaAs options (2A and 2B) require less array area than the heavy shielded silicon option (1B), even though the latter had a higher power ratio. This occurs primarily because the GaAs options have higher power output per unit area and less vehicle mass. A considerable improvement in active array also is seen when using the continuous annealing approach rather than post-annealing (Option 2B versus 2A).

The mass comparison of the options is shown in figure 4.4.3-2. Some of the key observations are as follows. All options considerably reduce the mass relative to the

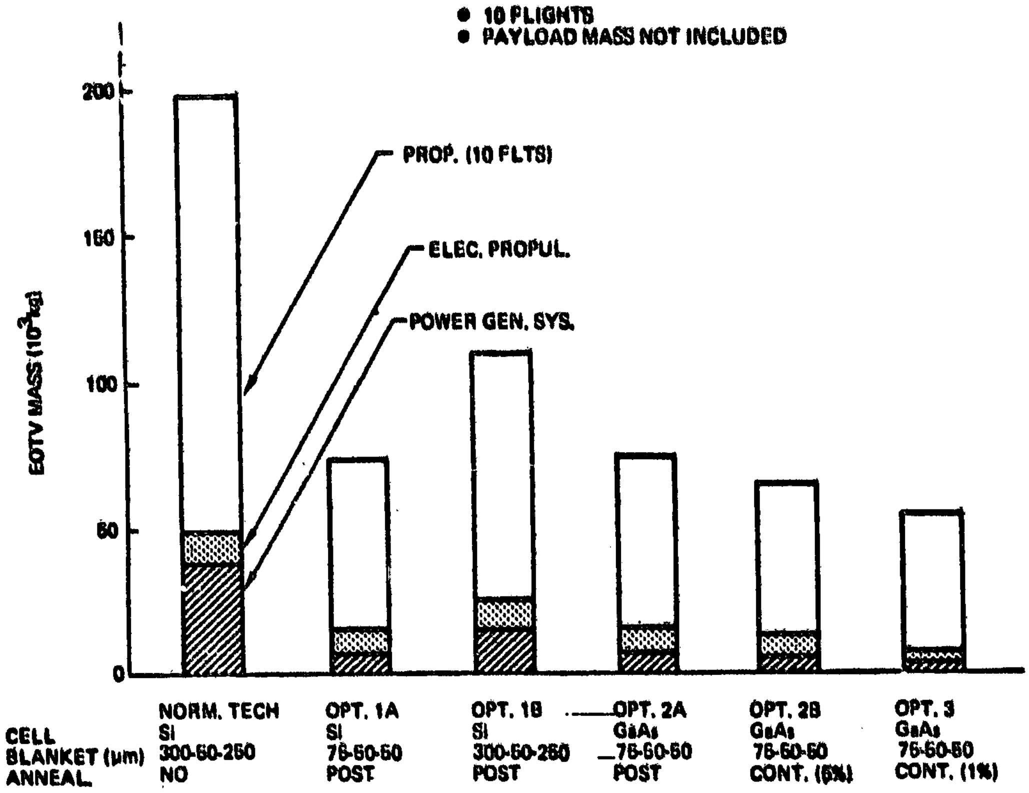


Figure 4.4.3-2 Mass Comparison

normal growth vehicle. This occurs primarily because the higher power ratios available with annealing result in smaller arrays which, in turn, reduce the propellant requirement. A final point of interest is that although the heavy shielded option (1B) requires less array area than the lightweight array option (1A), it requires considerably more total mass, primarily due to the propellant requirement.

The cost comparison of the accelerated technology options is presented in figure 4.4.3-3. Again, all options offer considerable improvements over the normal growth vehicle, primarily as a result of smaller solar arrays which, in turn, reduce the amount of electric propulsion.

In the case of the silicon options, when annealing is incorporated, the lightweight array provides an advantage over the heavy shielded option primarily due to less launch cost. The advantage of a 5% delta in annealing effectiveness of the continuous annealing GaAs EOTV (Option 2B) over the post-annealable GaAs EOTV (Option 2A) escalated to an approximate 14% advantage in terms of cost, which clearly indicates the leverage for continuous annealing if it is technically feasible. The GaAs Option 2B, however, only provides a small cost margin over the lightweight silicon option since its cost per unit area offsets a smaller required area.

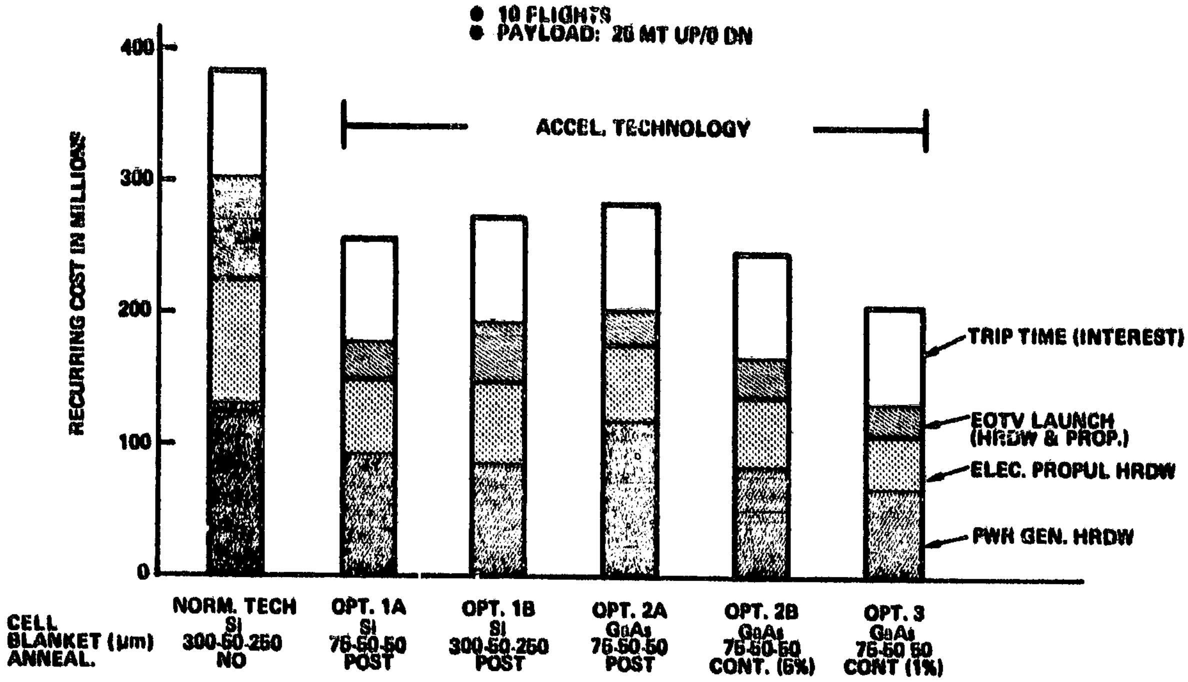
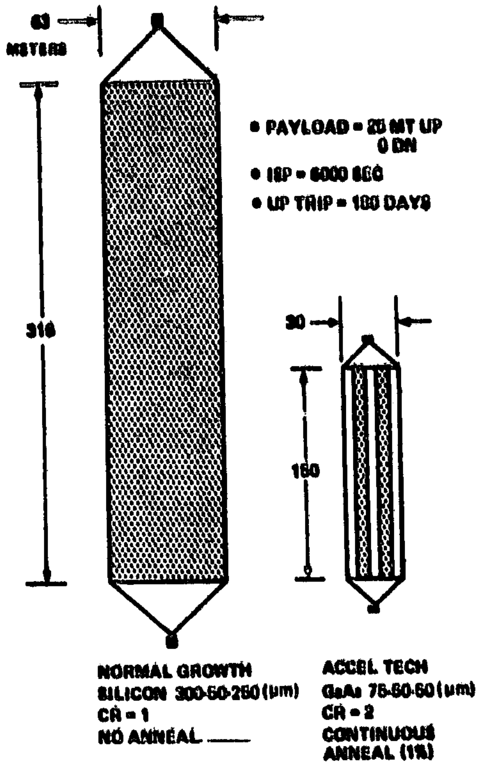


Figure 4.4.3-3 Accelerated Versus Normal Technology EOTV Cost Comparison

Option 3, which used the most optimistic performance assumptions, resulted in the least-cost system. Since this option used the same degree of annealing effectiveness and array thickness as the lightweight silicon option, the advantage of Option 3 can be related to its higher array performance offsetting its greater specific mass.

A comparison of only the least-cost accelerated technology vehicle and the normal growth EOTV is shown in figure 4.4.3-4. This comparison involves the most optimistic GaAs EOTV (Option 3) and the nonannealable heavy shielded silicon EOTV. The advantages associated with the accelerated system are significant because a high degree of annealing is available. The BOL power is only 25% that necessary for the normal growth vehicle. The array area requirement is only 16% as large due to the higher power resulting from more efficient cell and a concentrated design. Dry weight is considerably reduced because the annealing approach takes the place of the heavy shielding. The average unit cost is reduced by 50%. This reduction is not as large as that indicated for the other parameters because the GaAs array has a higher cost per square meter. In summary, the advantages of this accelerated technology option appear significant enough to offset, temporarily, the concerns associated with some of its optimistic design and performance features. Consequently Option 3 will be used to reassess a mixed OTV fleet (electric plus chemical OTV's) versus an all-chemical OTV fleet.



KEY CHARACTERISTICS

	NORMAL	ACCEL
• BOL POWER (KW)	3000	930
• AREA (10^3 SQ M)	10.0	2.4 (ACTIVE)
• DRY WT (MT)	61.0	7.8
• PROP/FLT (MT)	14.6	4.8
AVG UNIT COST ($ MILLIONS)		
POWER GEN	146	72
ELECT PROPULSION	81	30
AVIONICS	10	10
OTHER	6	6
FLT HRDW	243	122
ASSY & C/O	36	18
SUPPORT	82	39
TOTAL	361	179

Figure 4.4.3-4 Accelerated Versus Normal Technology Electric OTV

4.4.4 OTV Fleet Cost Comparison

Two OTV fleet options are compared. One option uses a mixed fleet consisting of the selected accelerated technology EOTV for trip time insensitive payloads and a normal growth technology LO_2/LH_2 OTV for high-priority cargo. The second option involves the use of the normal growth LO_2/LH_2 OTV for all payloads. The number of OTV flights for each vehicle is the same as defined in the normal growth comparison of section 4.3.4.

The cost comparison of the two OTV fleet options when performing the total mission model is summarized in figure 4.4.4-1.

The left-hand plot is for the case of using the reference launch cost for the SDV ($22M/flight). In this comparison, the mixed fleet using an accelerated technology EOTV shows a reduction of $2B or 14% as compared with the mixed fleet using the normal growth EOTV. This occurs as a result of there being nearly a 50% reduction in EOTV DDT&E and production costs and a 10% reduction in SDV launch costs. When compared with the all-chemical fleet, however, the cost of the best mixed fleet is still 5% higher. A further breakdown of the cost is provided in table 4.4.4-1.

The right-hand plot presents a cost sensitivity in terms of a 50% increase in cost per

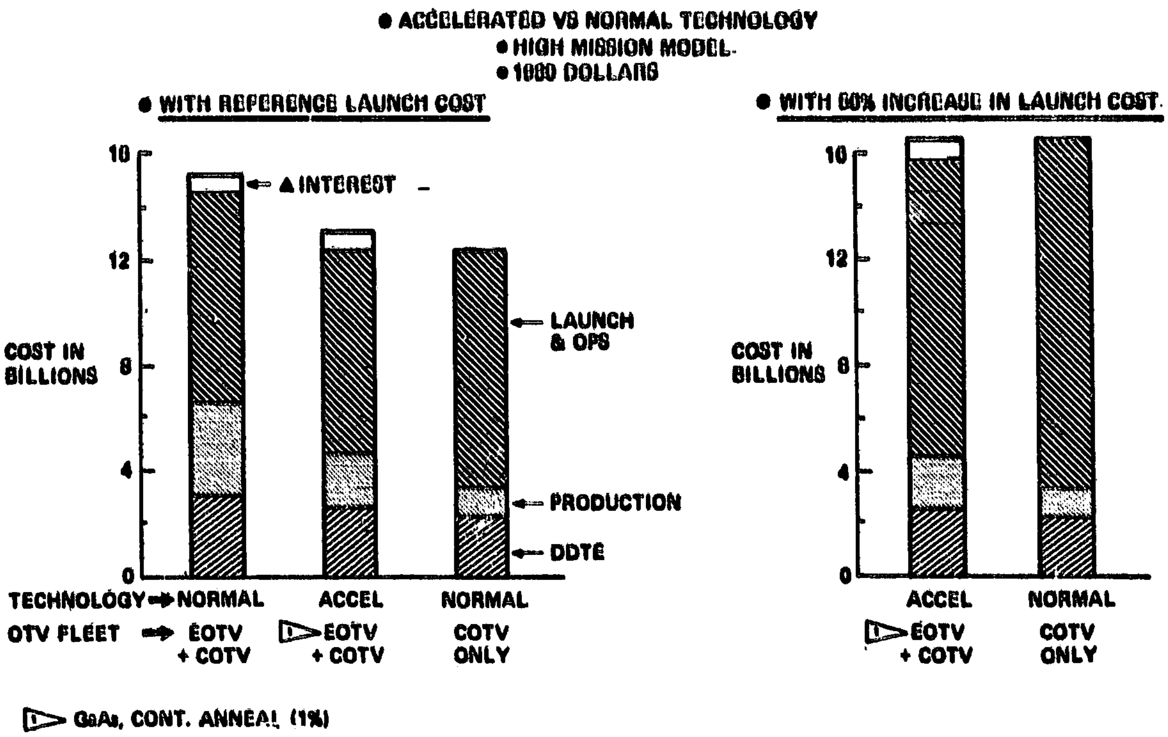


Figure 4.4.4-1 Electric Versus Chemical OTV Transportation Cost Summary

Table 4.4.4-1 Transportation Cost Summary—OTV Fleet Comparison

	NORMAL GROWTH EOTV + COTV		ACCEL. TECHNOLOGY EOTV + COTV		NORMAL GROWTH ALL COTV	
DDTE		(3140)		(2700)	–	(2240)
EOTV	900		460		700	
COTV	700		700		440	
TANKER	440		440		1100	
SDV/RPS	1100		1100		TBD	
SOC	TBD		TBD			
PRODUCTION		(3496)		(2110)	–	(936)
EOTV	2760		1376		380	
COTV	210		210		126	
TANKER	76		76		460	
SDV/RPS	460		460		TBD	
SOC	TBD		TBD			
OPERATIONS		(8020)		(7636)	–	(9206)
EOTV	290		290		780	
COTV	660		660		210	
TANKER	130		130		5080	
SDV/RPS (LAUNCH)	3906		3620		TBD	
SOC	TBD		TBD		3136	
STS (LAUNCH)	3136		3136			
SUBTOTAL		(14656)		(12445)		12380
OTHER						
TRIP TIME (INTEREST)		(606)		(605)		(–)
TOTAL		15260		13050		12380

flight for the launch systems. Although this value appears rather arbitrary, it actually reflects a more up-to-date (1981) cost projection for the solid propellant used by the SRB's. Consequently, the increase affects both the STS and SDV. With this sensitivity, the mixed fleet with the high-performance EOTV shows less impact than the all-chemical fleet to the ~~extent that the~~ total transportation ~~cost~~ becomes essentially equal.

Several other cost factors should be taken into consideration, however, regarding either method of comparison. In the case of the EOTV, costs have not been included for the following:

1. R&D to achieve the design and performance features identified for the most optimistic EOTV
2. Construction costs (SOC users charge)—a total of at least eight EOTV's
3. Cost impact on EOTV payloads that require radiation protection

Cost variations could also occur with the chemical OTV. In this case, the change could be a reduction if an accelerated technology vehicle using LF_2/LH_2 was employed. A cost reduction would occur in both fleets but would be most significant for the all-chemical fleet.

In summary, the potential additions and deletions that could occur in the cost comparison tend to substantiate the belief that an all-chemical OTV fleet contributes to the achievement of the least transportation cost within the constraints of the analysis.

4.4.5 EOTV Utilization for GEO Refueling

An EOTV in a mixed fleet could also be used to transport propellant for later use by the LO_2/LH_2 OTV. This mode is sometimes referred to as GEO refueling. The operations associated with this concept are as follows. The LO_2/LH_2 OTV's would be sized only for GEO delivery requirements. Upon reaching GEO, their propellant would be essentially depleted (excluding reserves). A propellant storage facility would be available at GEO, however, to provide propellant for the return to LEO. In this manner, the LO_2/LH_2 OTV can be physically smaller than if designed to carry propellant for the complete trip. The potential benefit in this approach is that less propellant needs to be launched. Delivery of the propellant to the GEO storage facility would be done with the high-performance EOTV. The key issue in this approach is whether the savings in launch cost would offset the cost of propellant delivery by the EOTV.

A key consideration in this concept is how many chemical OTV flights could really be benefited. Several factors are involved in establishing this number. First, the OTV flights which would benefit most are those involving round trip payloads because they require larger amounts of return propellant than delivery-only missions. Secondly, the OTV's GEO operations must include a stop at the refueling facility which most likely would be located at manned GEO bases. The missions which best fit these criteria are those of crew rotation and resupply for the two GEO bases in the mission model. Payload-delivery-only OTV flights to GEO would generally go to locations other than the bases. Even if these OTV's would maneuver to the propellant facility for refueling, there would be little savings since without a return payload, very little propellant is required when using aeroassist (typically only 3t). Therefore, the payload-delivery-only missions are judged to be noncandidates for GEO refueling. The OTV flights that appear to be the best candidates for GEO refueling are those with round trip payloads.

The comparison of GEO and LEO refueling was done using the selected accelerated technology EOTV defined in section 4.4.3. The propellant and launch requirements of the GEO and LEO refueling option are shown in table 4.4.5-1. Propellant requirements for the chemical OTV using GEO refueling reflect a stage sized essentially as an expendable vehicle because propellant for the return is provided after reaching GEO. The GEO propellant-delivery requirements for the EOTV are those of the return propellant for the chemical OTV, amounting to 56 t/yr and resulting in the need for two EOTV flights per year. The total mission model chemical propellant savings is 1984t, but this is partially offset by the need to launch EOTV-type propellant. The net result is an 1824t reduction in propellant to be launched, which translates into 35 fewer SDV launches.

The cost of GEO refueling relative to LEO refueling is shown in table 4.4.5-2. Three extra EOTV's are required during the mission model to satisfy the delivery requirements when using GEO refueling. The cost of these vehicles is $540 million. Launch cost savings are $748 million, resulting in a net savings of $208 million out of a total transportation cost of approximately $13 billion. It should be noted, however, that several costs have not been included in the GEO refueling concept. Most notable of these are for the propellant storage facilities at GEO and the means used to minimize boiloff during the 180-day transfer to GEO.

If the above savings were included in the fleet cost comparison of section 4.4.4, the cost difference between the fleets would be reduced to approximately $500 million out of a total of nearly $13 billion, with the all-chemical OTV still having the least cost.

In summary, it does not appear that GEO refueling provides sufficient cost benefits to offset the operational complexity. The major factor which contributes to the indicated results and conclusion is that the chemical OTV's use aeroassist for return to LEO and thus significantly reduce the amount of required return propellant. Another factor is that

Table 4.4.5-1 GEO and LEO Refueling Propellant and Launch Requirements

OTV Propellant Requirements

Mission	Payload (t) up/down	Chemical OTV propellant (t) GEO refuel up/down	Chemical OTV propellant (t) LEO refuel round trip	EOTV propellant (t) round trip
Construction base (CR/RS)	12.4/10.5	24.5/8.6	54	—
Maintenance base (CR/RS)	7.6/5.0	17.5/5.4	32	—
Propellant delivery (EOTV)	25/0	—	—	5

EOTV Propellant Delivery Flights

Payload (t) up/down	Chem prop down (t)	Flights per year	Annual propellant (t)	
12.4/10.5	8.6	4	34.4	56t at 25t per flight
7.6/5.0	5.4	4	21.6	= 2 EOTV flights per year

Propellant Launch Requirements

Payload (t)	LEO refuel prop/flt(t)	GEO refuel prop/flt(t)	Delta per flight	Delta per year	Total delta (16-yr)	
12.4/10.5	54	33	21	88	1408	1984
7.6/5.0	32	23	9	36	576	
25/0 (EOTV)	-	-	5	10	→	-160
					Net =	1824t

at 52t per launch
= 35 SDV launches

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Table 4.4.5-2 GEO Versus LEO Refueling Cost

<u>EOTV unit cost</u>			
	Flights per year:	2	
	Flights in mission model:	32	
	Flights per vehicle:	10	
	Vehicles needed:	3 at $180 million	
		= $540 million	
<u>SDV launch cost</u>			
	Propellant savings:		35 flights
	Extra EOTV's:		<1 flight
		Net savings = 34 flights	
		at $22 million per flight	
		= $748 million	
<u>Net cost</u>			
	Launch savings:		$748 million
	EOTV unit cost:		<u>$540 million</u>
		Savings	$208 million

there are not a large number of round trip payloads which benefit from this concept. These two factors are different from the Boeing SPS analysis, which assumed all-propulsive OTV's, involved large numbers of heavy round trip payloads (crew rotation and resupply), and did cost justify GEO refueling. It is judged, however, that the SPS scenario is considerably removed from space scenarios now projected for the next 20 to 25 years. The recommended approach for refueling chemical OTV's that are used in a mixed-fleet mode with EOTV's continues to be LEO refueling. As such, the overall transportation cost for the mixed-fleet would be essentially the same as that indicated in section 4.4.4.

4.5 FINDINGS

A summary of the principal findings resulting from the comparison of electric and chemical OTV's is presented below. These findings only apply when viewed in terms of the guidelines and assumptions which were used. Most significant of these were that applications were for cargo missions between LEO and GEO and that comparisons should be done in the context of total transportation system requirements.

1. An all-LO_2/LH_2 OTV fleet is a clear winner. Mixed fleets involving EOTV's do not provide cost or operational benefits. This was true with normal growth EOTV's (+24% LCC) and accelerated technology EOTV's (+5%) for GEO payload mission models up to 300 t/yr. Launch costs would have to increase by at least 50% before a mixed fleet (with accelerated technology OTV's) could provide approximately the same cost as the all-chemical fleet.
2. Use of EOTV's for GEO refueling of chemical OTV's does not provide a cost benefit relative to LEO refueling.
3. Accelerated technology has a payoff for EOTV's. The most significant improvement was that of annealing which reduced EOTV LCC by 50%. Annealing effectiveness is still an open issue.
4. Annealable silicon and GaAs arrays result in comparable EOTV costs. Lower performance and higher radiation sensitivity of the silicon array were offset by annealing and lower costs per unit area.
5. Solar cell cost prediction is speculative. This applies to large quantities (5000 to 10 000 m^2/yr). GaAs cell costs have greater uncertainty than silicon.
6. EOTV utilization has major uncertainties. Key concerns include design life as affected by radiation and payload exposure to radiation.

9.6 RECOMMENDATIONS

The recommendations resulting from the EOTV definition and OTV fleet comparison must also be viewed within the constraints of the study guidelines and assumptions.

1. Give no further consideration to photovoltaic (silicon or GaAs) EOTV's for GEO cargo delivery. An exception would be if there is some major deviation in the assumed performance or costing of these systems.
2. Focus on improving performance and operational capabilities of a space-based reusable LO_2/LH_2 OTV. Mission models of the size investigated could justify accelerated technology refueling concepts and, potentially, LF_2/LH_2 systems.
3. Focus any further EOTV technology on radiation and cost data.
   a. Conduct extensive radiation/annealing analyses including:
      (1) Development of radiation tests which use rates related to cost-effective trip times (180 days).
      (2) Multiple annealings of cells with radiation degradation comparable to that received in one round trip.
      (3) Development of a common presentation format.
   b. Obtain radiation and cost data on advanced cells identified by this study but not included in the analysis.
   c. Assess design-life limits due to multiple trips between LEO and GEO.
   d. Develop cost data associated with large quantities of solar cells (5000 to 10 000 m^2/yr).
   e. Improve cost prediction of thin (2-mil) GaAs cells.

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*U.S. GOVERNMENT PRINTING OFFICE 1982 539-007/8 1-3