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Volume IX

Study Extension Results

Orbital Transfer Vehicle Concept Definition And System Analysis Study

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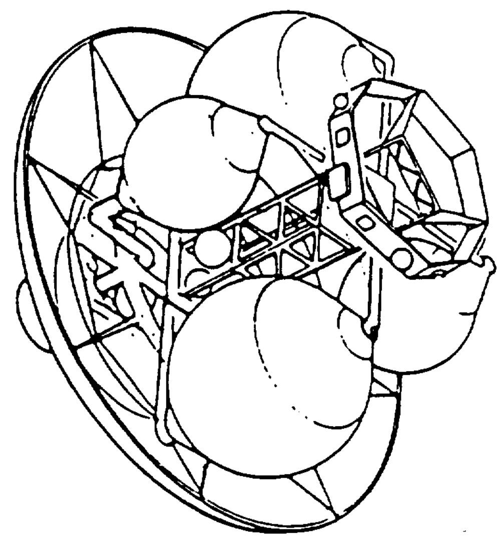

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ORBITAL TRANSFER VEHICLE
CONCEPT DEFINITION AND SYSTEM ANALYSIS STUDY

VOLUME IX
STUDY EXTENSION RESULTS

July 1987
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This report supplements the OTV Phase A study program results which are presented in eleven separate documents.

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Book 2	OTV Concept Definition
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ACRONYMS

ACC	Aft Cargo Carrier
AFE	Aeroassist Flight Experiment
ASE	Airborne Support Equipment
ATP	Authority to Proceed
BTU	British Thermal Unit
CDR	Critical Design Review
CPF	Cost per Flight
DD-250	Final End Item Acceptance by the Government
DDT&E	Design, Development, Test, & Engineering
DOD	Department of Defense
EKS	Expendable Kick Stage
EVA	Extra Vehicular Activity
GB	Ground-Based
GBOTV	Ground-Based OTV
GEO	Geostationary Orbit
GLOL	Gross Lift-off Length
GLOW	Gross Lift-off Weight
GN&C	Guidance, Navigation and Control
GPS	Global Positioning System
HEO	High Earth Orbit
IOC	Initial Operational Capability
IR&D	Independent Research and Development
Isp	Specific Impulse
IVA	Intra Vehicular Activity
L/D	Lift to Drag Ratio
LEO	Low Earth Orbit
LCC	Life Cycle Costs
LCV	Large Cargo Vehicle
LH_2	Liquid Hydrogen
LO_2	Liquid Oxygen
JSC	Johnson Spaceflight Center
MLI	Multi-Layer Insulation
MRMS	Mobile Remote Manipulator Systems
MSFC	Marshall Space Flight Center
NASA	National Aeronautics and Space Administration
OMV	Orbital Maneuvering Vehicle
OTV	Orbital Transfer Vehicle
PDR	Preliminary Design Review
PIDA	Payload Installation and Deployment Aid
PIM	Payload Interface Module
PLD	Payload
R&T	Research and Technology
RMS	Remote Manipulator System
RTV	Room Temperature Vulcanizing Sealant
SB	Space-Based
SBOTV	Space-Based OTV
SDI	Space Defense Initiative
SOFI	Spray On Foam Insulator
SRB	Solid Rocket Booster
SS	Space Station
STAS	Space Transportation Architecture Study
STS	Space Transportation System
TCS	Thermal Control System
TPS	Thermal Protection System
UPRCV	Unmanned Partially Reuseable Cargo Vehicle
WBS	Work Breakdown Structure
W/C_DA	Ballistic Coefficient, pounds/square foot

1.0 INTRODUCTION - OTV OVERVIEW

The NASA sponsored advanced upper stage studies conducted during the past decade provide major solutions to help determine the future program for advanced technology orbital transfer vehicles operating both from the ground and from a space base. The space-based systems will provide a new era of payload delivery capabilities with basing advantages and reduced costs to the users. This study describes our recommended cryogenic OTV that begins operations from the ground to meet mid-1990's user needs. The ground-based OTV evolves to a space-based system operating from the NASA Space Station now being defined. The proposed OTV plan incorporates the best features of a new OTV, the IOC and growth Space Station, the Orbital Maneuvering Vehicle (OMV) for support operations, and an unmanned large cargo vehicle (LCV).

The OTV design concepts resulting from our study of the mission requirements of the Rev. 9 (Preliminary) OTV Mission Model utilize cryogenic propellants and aerobraking which allow the OTV to be a low cost, fully reusable upper stage capable of transporting payloads from earth surface or the Space Station to GEO at costs less than $3300/lb.

The initial OTV is ground-based and launched in a new generation large cargo vehicle with a 25 foot diameter payload bay. When fully loaded with 52,000 lbs of propellant this vehicle can delivery a 15,000 lb payload to GEO and return empty to LEO for reuse. As mission requirements expand, the OTV propellant capacity is increased to 74,000 lbs allowing it to deliver 25,000 lbs or perform a manned mission consisting of 12,000 lb delivery and a 10,000 lb return. The growth vehicle can either be ground-based or space-based.

2.0 SUMMARY RESULTS

The purpose of this extension to the OTV Concept Definition and System Analysis Study was to improve the definition of the OTV program that will be most beneficial to the nation in the 1995 - 2010 timeframe. This activity built on the effort completed in prior study effort. It investigated the implications of the missions defined for, and the launch vehicle defined by the Space Transportation Architecture Study (STAS).

The key new mission requirements identified for STAS have been established and they reflect a need for greater early capability and more ambitious capability growth. The key technical objectives and related issues addressed are summarized. We have updated the OTV program approach previously selected in the area of vehicle design. New mission requirements, evolving Space Station definition, and proposed new launch vehicles were evaluated. We enhanced our analyses of selected areas including aerobrake design, proximity operations and the balance of EVA and IVA operations used in support of the OTV at the space-base.

These activities led to an improved definition of an OTV program that should receive favorable consideration for an early new start. An important aspect of this effort was developing a thorough understanding of the sensitivity of the OTV program to changes in use, economic environment and technology development. We conducted sensitivity studies to establish how the OTV program should be tailored to meet changing circumstances.

We conducted this study in two primary parts. The activities conducted in the first part were those that could be accomplished without a definition of the large cargo vehicle. When this definition became available from the STAS studies, the activities dependant on this information were conducted. These primarily delved into the effect of the availability of the large cargo vehicle on the preferred OTV program. Requirements assessments were ongoing throughout the whole study, as the definition of mission requirements is in a continuous state of change. Operations and accommodations assessments were also continuous, and supported all study activities as required. Study output includes definition of a baseline cargo vehicle supported OTV program and an assessment of the sensitivity of this baseline program selection to mission model options, to launch vehicle availability, and to variations in the Space Station development scenario.

The study data contained herein justify the design and development of a reusable, cryogenic, aerobraked OTV. Other major results of this study are:

- We recommend developing a space-based OTV capability
  - Enhances operation of advanced missions
  - Key to manned high altitude operations
  - Reduced booster launches
  - Economic viability depends on propellant 'hitchhiking' and efficient accommodations
- We recommend an OTV supported by large cargo vehicle
  - Standard 3-engine concept
  - Two vehicle sizes
  - Ground/Space operations compatible (large vehicle)
- High traffic options justify a specialized, smaller OTV
- Space-basing makes OTV operations cost less sensitive to launch operations cost

2.1 MAJOR PROGRAM SENSITIVITIES

2.1.1 Requirements Summary

Major program milestone schedules are shown in Figure 2.1.1-1. The various launch vehicle availabilities were a program ground rule, as was Space Station IOC in 1995. The full capability Space Station availability date was left open in the program ground rules; the contractors could specify their preferred dates any time after 1995.

Our analysis of the Rev. 9 mission model requirements show that a small OTV capable of transporting 15,000 lbs from LEO to GEO is required in 1995. The large OTV capable of delivering 25,000 lbs to GEO and also capable of delivering 12,000 lbs and returning 10,000 lbs is required in 1999. The large OTV must be man rated in 2002, but no increase in propellant capacity is required for the manned missions.

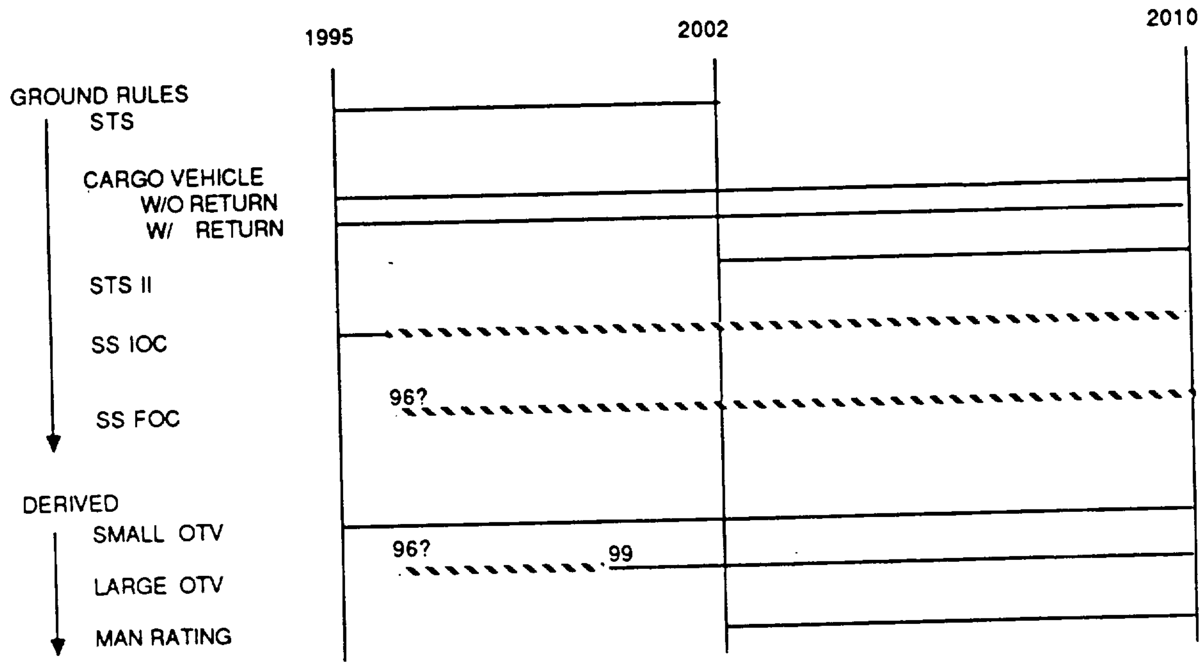


Figure 2.1.1-1 OTV Program Milestone Schedules

The Rev. 9 mission model defines five operational scenarios ranging from very constrained to highly ambitious; anywhere from 292 to 872 OTV missions over the 1995 - 2010 time frame, with the baseline, Scenario 2, containing 422 missions.

The results of our analyses of the various payload requirements show that OTV performance requirements are independent of Scenario. The top level derived requirements are summarized in Table 2.1.1-1. The only variations in these requirements is that most ambitious scenario will require the large OTV to be man-rated in 1999 rather than 2002, while the most constrained scenario does not require man-rating until after the year 2010.

Table 2.1.1-1 OTV Derived Requirements Summary

	SMALL OTV	LARGE OTV
• OPERATIONAL DATES		
DELIVERY/RETURN	1995+	1999 +
MANNED	N / A	2002
• PERFORMANCE		
GEO DELIVERY CAPABILITY, KLBS (SINGLE MISSION)	15	25
GEO DELIVERY CAPABILITY, KLBS (MULTIPLE MISSION)	33	33
ROUND TRIP CAPABILITY,KLBS	12/2	12/10
MAX DELIVERED P/L LENGTH, FT	30	50
MAX RETURNED P/L LENGTH, FT	10	30
LOW THRUST ACCELERATION	0.1G	0.1G
• METEOROID / DEBRIS SHIELDING		
PERMISSIBLE DAMAGE EVENTS PER HOUR		
UNMANNED MISSIONS	14E-6	14E-6
MANNED MISSIONS	N/A	3.5E-6

Table 2.1.1-1 shows a requirement to return a 30 foot payload. Discussions with the payload technical monitor revealed that this payload has deployed solar panels which limit acceleration levels to 0.1 G. Since aerobraking results in deceleration levels greater than 3. G's, this payload must be returned all-propulsively (See paragraph 5.2 for additional details and the rationale for selecting the size of the aerobrake)

2.1.2 Launch Vehicle Charging Impacts

Earlier OTV studies utilized only the STS as a launch vehicle with a baselined cost of $73M per flight and a LEO lift capability of 72,000 lbs. This extension study concentrated on utilizing a new launch vehicle with a 90 foot long, 25 foot diameter payload envelope. This vehicle had the capability to boost 150,000 lbs to LEO at a cost of $70M per flight. Sharing of launch costs with other payloads on the basis of the percentage of utilized launch vehicle capability has a major impact on reducing payload launch costs. The impact of using an STS type charging algorithm is shown in Figure 2.1.2-1.

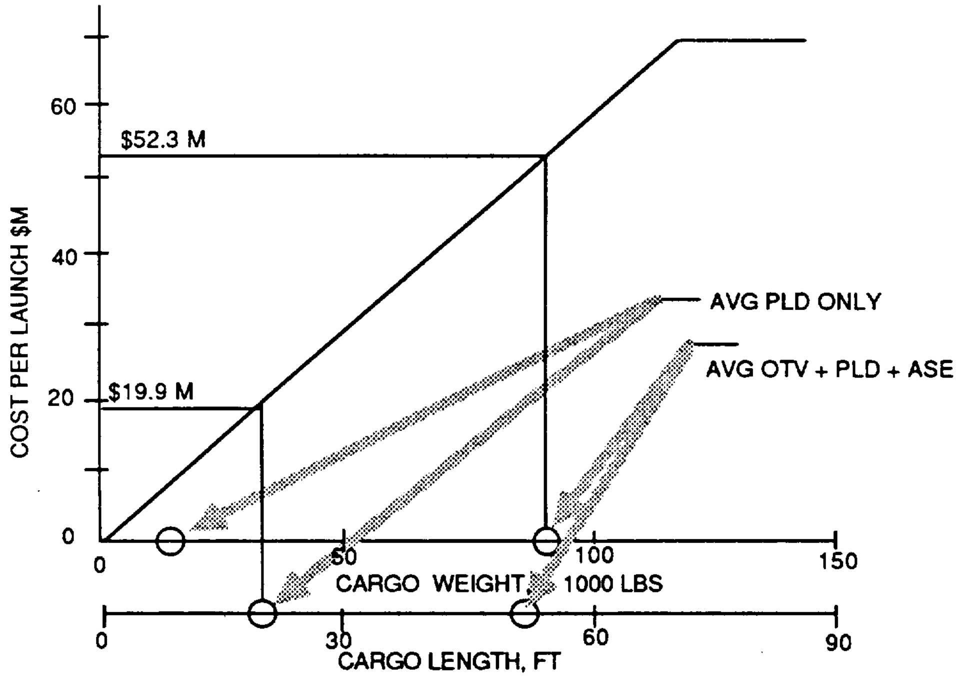


Figure 2.1.2-1 Typical Launch Costs for Large Cargo Vehicle

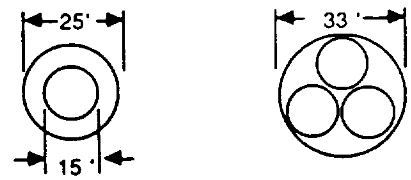
Each user is assessed a launch cost on the basis of either length or weight (only the largest is used). If a user requires 75 percent of the capability he is assessed the full launch cost. The length and weight data for the average of the 160 Rev. 9 payloads going to GEO are represented by the two left most circles on the ordinate of Figure 2.1.2-1. These data points consider the payload by itself, as would be the case if the OTV were space-based. As indicated by the circles, most of the payloads specified in the mission model will be charged on a length basis; weight is relatively unimportant.

The two circles on the right side of the ordinate show what happens when the payload and OTV are considered as a combined user, i.e., the lengths and weights are added together and launch costs calculated on this basis as is the case for a ground-based OTV. In this case, length and weight are shown to be of equivalent importance. Detailed analyses show a sensitivity of $200,000 per flight for a change of either 100 lbs dry weight or 1 foot of length.

The space-basing versus ground-basing trade described in paragraph 4.9 and the analyses of Figure 2.1.2-1 utilizes the payload data specified by the mission model. Essentially all of the payloads are specified to have a 15 foot diameter; a few are smaller, but none are larger. If the payload bay of the LCV were 33 foot diameter (instead of the specified 25 foot diameter) three payloads could fit alongside each other. The net result obtained by use of the shared launch cost algorithm when apparent payload length is reduced to 1/3 of the specified value is shown in Figure 2.1.2-2.

OBSERVATION

A 33 FT DIAMETER PAYLOAD BAY CAN ACCOMODATE 15 FT DIAM STS ERA PAYLOADS MORE EFFICIENTLY



ANALYSIS

AVERAGE LCV LAUNCH COSTS	GROUND BASED	SPACE BASED
BASELINE	\$52.3 M / FLIGHT (OTV + PLD + ASE)	\$19.9 M / FLIGHT
33 FT DIAM PAYLOAD BAY	\$49.3 M / FLIGHT (OTV + PLD + ASE)	\$ 8.8 M / FLIGHT
SAVINGS POTENTIAL OF LARGE DIA PLB	\$3.0 M / FLIGHT \$0.48 B FOR 160 GEO MISSIONS	\$11.1 M / FLIGHT \$1.78 B FOR 160 GEO MISSIONS

IMPACT

POTENTIAL OF ADDITIONAL \$1.3 BILLION ADVANTAGE FOR SPACE BASING IN GROUND / SPACE TRADE

Figure 2.1.2-2 Effect of a 33 Foot Diameter Payload Bay

The first conclusion to be reached is that the defined payloads are not optimized to utilize the large diameter payload bay. This is understandable because the mission model is based on known and planned payloads which were all designed for launch in the 15 foot diameter STS. The second conclusion is that the space-based - ground-based economic trade would shift towards space basing by an additional \$1.3B if the payloads were optimized to the launch vehicle. (Section 4.9 shows that space basing is approximately a \$1.0B life cycle cost winner over ground basing without this optimization of payloads.)

2.1.3 Propellant Transportation Costs

2.1.3.1 Hitchhiked Propellant

One of the study ground rules was that propellant for a space-based OTV could be loaded on the ground to fully utilize available lift capacity of the launch vehicle and not incur any transportation costs to LEO. Tankage, on-orbit operations and OMV charges are assessed however.

Figure 2.1.3-1(a) is a schematic representation of payload bay loading for a ground-based OTV. Payload bay loading for a space-based OTV is indicated in Figure 2.1.3-1(b). At first appearance, one might think the number of launches to capture a fixed number of payloads could be greatly reduced by space-basing since available payload bay capacity is increased. However, when the decreased launch vehicle performance of going to Space Station altitude and the propellant tanker flights necessary to supply the space-based OTV propellant and spares are accounted for, the number of LCV launches are roughly the same for the ground-based and space-based concepts shown in Figure 2.1.3-1(a) and (b).

The propellant "hitchhiking" concept is represented in Figure 2.1.3-1(c). Our analyses shows the hitchhiking concept to be both feasible and desirable. It eliminates 51 OTV propellant tanker flights and supplies a minimum of 63 percent of the propellant required for a space-based OTV. The cost per pound of hitchhiked propellant delivered into the SS tank farm is approximately $200/lb. The comparative cost of tanker supplied propellant is approximately $750/lb of which $650/lb is transportation costs to the SS.

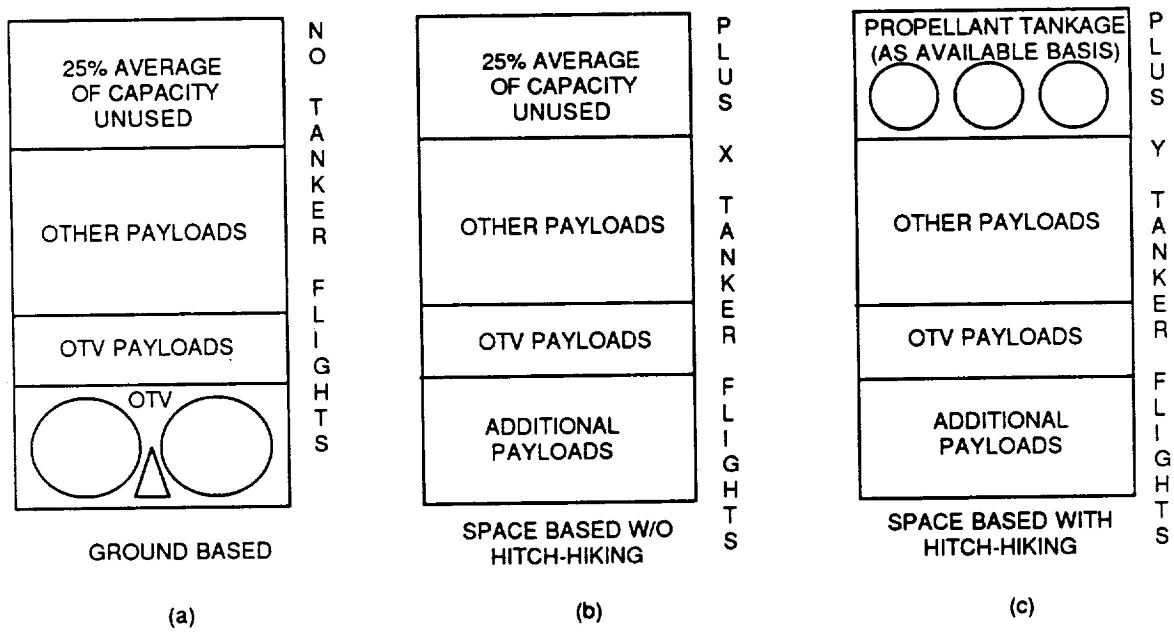


Figure 2.1.3-1 - Propellant Hitchhiking Concept

2.1.3.2 Scavenged Propellant

Propellant scavenging involves utilizing the unburned residual propellant from the launch vehicle. Previous Phase A studies published in Volume III, System and Program Trades showed that the Rev. 8 low traffic mission model (184 OTV flights) could provide 4.6M lbs propellant at an average cost of $272/lb. Scavenging from the LCV should be operationally less complex since it enters orbit as opposed to the ET which remained suborbital. Since more than 80% of the previous costs were associated with operations, LCV scavenging has the potential for supplying propellant at a cost equivalent to the hitchhiked propellant.

2.1.3.3 Propellant Transportation Costs Summary

Hitchhiking combined with STS, STS II, and LCV propellant scavenging can probably supply 100% of the space-based OTV propellant requirements. However, since LCV and STS II scavenging can not be analyzed in detail due to the lack of design details, scavenging with those vehicles was not considered in the life cycle cost analysis of this study.

2.2 MAJOR TRADES SUMMARY

2.2.1 System Level Trades

The system trades of the initial OTV Phase A study were updated, refined and modified to reflect the revised requirements and impacts of the Rev. 9 mission model.

The system level trades, with options and sub-options are summarized in Table 2.2.1-1. Although the table makes it appear that the trades are not interrelated, that certainly is not the case, as indicated in the following discussions.

Table 2.2.1-1 System Level Trade Study Summary

TRADE LEVEL		OPTIONS	SUB-OPTIONS	RECOMMENDATIONS
SYSTEM TRADES				
REUSABILITY	A	CURRENT EXPENDABLES		
	B	HI-TECH/LOW COST EXP		
	C	RE-USEABLE		RE-USEABLE
BASING MODES	A	ALL GROUND BASED		
	B	GROUND AND SPACE BASED		GROUND AND SPACE BASED,
			PURE SB VEHICLE	GB VEHICLE EVOLVED TO HYBRID
			HYBRID SB VEHICLE	
STAGE SIZES	A	LARGE ONLY	EXTRA SMALL	LARGE + SMALL
	B	LARGE + MEDIUM + SMALL	EXTRA LARGE	(PLUS EXTRA SMALL FOR SCENARIO 4)
	C	LARGE + SMALL		
LAUNCH VEHICLE	A	STS/STS II	CARGO BAY	TORUS BEST FOR CARGO BAY
			PARALLEL TANK	
			TANDEM TANK	
			TORUS TANK	
			ACC	ACC PREFERRED OVER CARGO BAY
	B	LARGE CARGO VEHICLE	LCV RETURN	LCV W/ STS RETURN PREFERRED OVER ACC
			STS/STS II RETURN	LCV WITH LCV RETURN BEST OF ALL OPTIONS
EXTENT OF AUTOMATED SERVICING AT SS	A	FULL RANGE FROM AUTONOMOUS TO MANUAL		REMOTE WITH IVA CONTROL
PROPELLANT TANK DE-ORBIT AND REENTRY	A	OMV DE-ORBIT		
	B	STS DE-ORBIT		
	C	AUXIL OTV PROP		
	D	NORMAL ORBIT DECAY		NORMAL ORBIT DECAY

2.2.1.1 OTV Reusability

The issue examined here was the merit of developing a reusable OTV as measured by the non-recurring and recurring life cycle costs of flying the missions in the various scenarios of the Rev. 9 mission model. Obviously, the use of current expendables will have the lowest non-recurring costs while a re-useable OTV will have the highest non-recurring costs, which are then off-set by lower costs per flight. The analyses described in Paragraph 4.1 shows that a re-useable OTV achieves payback after only three years of operation based on the Scenario 2 civilian traffic levels of the Rev. 9 mission model. In discounted costs, payback is achieved within six years.

2.2.1.2 Basing Modes

The basing mode trade study compares a totally ground-based OTV system with a system that utilized a mixture of ground and space-basing. Pure space-basing was not considered as a candidate because the Rev. 9 missions start in 1995, while the earliest possible IOC for the Space Station was 1996. In addition, there may be a reluctance to operate DOD missions out of the Space Station because of security concerns resulting from the international aspect of the Space Station. Consequently, the basing mode trade considered only the 160 civilian missions of the Rev. 9, Scenario 2 model.

This trade study, described in detail in paragraph 4.9, concludes that space-basing does provide a payback of the non-recurring costs within the framework of the civilian GEO missions. The LCC savings of $1.0B is due primarily to the low cost of space-based propellant brought about by the hitchhiking concept which was described in paragraph 2.1.3. An additional cost benefit of $1.3B could be ascribed to space-basing if payloads were designed to better utilize the volume of the LCV as described in paragraph 2.1.2. Since space-basing does provide a cost savings, it should be started as soon as possible, within funding limit constraints. It must be noted that the economic advantage of space-basing is highly sensitive to such parameters as the cost of space-base accommodations and the concept of hitchhiking propellants with no transportation charge. Changes in ground rules can negate the apparent economic advantage of space-basing OTV.

The basing mode study also investigated the evolutionary growth path of modifying a ground-based vehicle to make it suitable for space-basing as opposed to designing a fully optimized vehicle for space-basing. This sub-trade, "hybrid" versus "clean-sheet", shows a minimal difference in LCC (paragraph 4.9.5). Thus, the hybrid approach is preferred since the common elements allow ground and space-based vehicles to serve as ready backups to each other in the event of unforseen changes in mission model, operational scenarios, or Space Station scenarios.

There are reasons for space-basing an OTV other than economics. Some of the more obvious reasons are:

1) Any launch vehicle, including foreign launch vehicles can boost the payloads to LEO. As long as the payload has an OMV/OTV compatible interface, a space-based OTV can be the upper stage. This would strengthen the international usage of Space Station.

2) Because the number of launches is reduced by space-basing with propellant hitchhiking, there are fewer opportunities for any type of threats including accidents or sabotage. Fewer launches also reduces environmental impacts such as noise and pollutants.

3) Final payload placement location will have no impact on the time of launch since each mission goes to the general vicinity of the Space Station. Since the launch window is greatly expanded, the concern about aligning weather and launch window is minimized.

4) A ground-based payload and OTV utilize approximately 1/2 the capacity of the LCV. A schedule slippage of just one payload either causes a major ripple in manifesting, or the other payloads will have to wait.

5) The need for rapid deployment of any non-scheduled OTV delivered payload would bump roughly 1/2 of the manifested payloads from the LCV. If the OTV were space-based, only the weight/volume of the payload would cause manifest rippling. Certain payloads could even be stored at Space Station for the ultimate in rapid deployment.

6) Payloads could go through a complete burn-in in LEO to eliminate the infant mortality. Also, the OTV failure rate will be reduced by not subjecting it to the launch environments for every mission.

7) User requirements will certainly increase in the future. A space-based OTV relaxes any upper limit on the weight and size of payloads.

2.2.1.3 OTV Propellant Quantity (Stage Size)

Analyses of mission model requirements and vehicle performance shows OTV with 52K propellant is needed in 1995 for delivery of a 15 Klb payload. A large stage is needed in 1999 for the 25 Klb delivery missions and in 2002 for the 12 Klb up/10 Klb down manned missions.

The trade study described in paragraph 4.7 investigates several other options for the ground-based program;

a) Is a mid-size stage worthwhile?

b) Is a smaller stage (10K delivery) worthwhile?

c) Is a super stage capable of performing lunar and planetary missions without multiple OTV stages and tanksets worthwhile?

With one exception, none of the three options make economic sense. The exception is that the high DOD traffic of Scenario 4 does justify a small ground-based stage optimized for heavy traffic in the mid-inclination regime.

The space-based OTV program will require a 74 Klb manrated OTV. The analyses of paragraph 4.9.5.1.3 shows that even though an additional smaller space-based OTV would save on propellant requirements, the savings do not justify the additional accommodations and spares costs.

2.2.1.4 Launch Vehicle Impacts on OTV

This trade study first concentrated on defining the best OTV for launch in the STS cargo bay. This design was found to be a single engine, flexible aerobrake, with a torus oxygen tank. This vehicle was then compared with

OTV's that could be launched in the dedicated aft cargo carrier (ACC). The ACC vehicles vehicles had much lower life cycle costs due to the fact that volume in the cargo bay was available for sharing launch costs with other payloads.

The optimum OTV for launch in the large cargo vehicle (LCV) was then defined. This design was a three-engine, flexible aerobrake, with four cylindrical propellant tanks. Even if the LCV does not have the capability to return the OTV to earth and the OTV pays the extra launch costs of STS ASE for return flights, and also disposes of hydrogen tanks onorbit, the LCV launch is lower cost than the STS/ACC launch.

The lowest cost option is an LCV with return to earth capability.

These trades are described in paragraphs 4.4, 4.5 and 4.9.

2.2.1.5 EVA vs IVA Servicing at Space Station

EVA servicing and maintenance of a space-based OTV has a small up-front cost which is rapidly offset by the high cost of crew labor. It also has an upper limit imposed by the number of crewmen available. At the other extreme, a completely autonomous robotic system that provides for inspection, diagnostics, task planning and execution of all actions carries a tremendous initial cost, but has the advantage of a very low recurring cost.

The operations trade study described in paragraph 7.1.3 concludes that, in general, human decision making and control of robotics devices that autonomously perform a limited set of tasks will be the lowest cost approach.

2.2.1.6 Deorbit of Expendable Propellant Tanks

If the LCV does not have a return to earth capability, the preferred OTV design concepts must expend propellant tanks since they cannot be fitted into a single STS flight for return along with the OTV core structure, avionics and propulsion systems. Three concepts were examined for ensuring the tanks reenter the atmosphere rather than contributing to LEO debris. These were: deorbit by OMV, deorbit by STS and OTV auxiliary propellant. In the latter case, a small set of propellant tanks would allow the OTV to drop the main tanks while on a re-entry orbit and then perform a burn to achieve a stable circular orbit.

This study, described in paragraph 7.2.4, concludes that an active deorbit system is not required due to the 30 to 40:1 ratio in ballistic coefficients. An orbit which allows the OTV to be parked for 30 days awaiting STS retrieval will cause the tanks to reenter by themselves in one day. Because the tanks are constructed of extremely thin skin aluminum, it is felt that uncontrolled orbital decay is an acceptable mode of disposal since it is probable that no elements can reach the ground intact. Certainly a more detailed anlaysis of the specific tankage configurations and re-entry dynamics will have to be conducted to validate this concept. However, this level of analyis is beyond the scope of the present study.

2.2.2 Subsystem Trades

The major subsystem trades and the options considered are summarized in Table 2.2.2-1.

Table 2.2.2-1 Subsystem Trade Study Summary

TRADE LEVEL		OPTIONS	SUB-OPTIONS	RECOMMENDATIONS
SUBSYSTEM TRADES				
AEROASSIST				
	A	BALLUTE	TURNDOWN RATIO	
			BACKWALL TEMPERATURE	
	B	RIGID	HIGH L/D	
			HIGH 1/3 OF CORRIDOR	
			MIDDLE CORRIDOR	MIDDLE OF CORRIDOR
			LOW 1/3 OF CORRIDOR	
	C	FLEX	LOW L/D	FLEX, LOW L/D
PROPULSION	1-A	ALL PROPULSIVE		
	1-B	AEROBRAKE		AEROBRAKE
	2-A	ENGINE QUANTITY	1	1 FOR STS CARGO BAY
			2	2 FOR PURE SPACE BASE OR STS/ACC
			3	3 FOR LCV
			4	

2.2.2.1 Aeroassist Configurations

The aeroassist configuration trade study described in paragraph 4.3 considered ballute, rigid structure and flexible-foldable brake configurations. The study shows that a rigid brake is not a viable candidate for ground-based missions and that the flexible fabric brake has the lowest life cycle costs of all the configurations, whether ground-based or space-based.

The aeroassist analyses in paragraph 5.2 examines the effects of varying L/D. Guidance and navigational error analyses show that a 5 nmi control corridor width is adequate to control the OTV. This can be achieved with a brake that has an L/D of 0.12. A brake with an L/D of 0.30 can be flown lift-up or lift-down in a 15 nmi aeropass envelope. Paragraph 5.2.2 shows that a brake design based on flying in the middle of the corridor is minimum weight for the high L/D design. However, the low L/D aerobrake is even lighter weight and is, therefore, less costly.

2.2.2.2 Propulsion System Trade Studies

2.2.2.2.1 All Propulsive vs Aerobrake

An OTV that utilizes propulsive burns for returning to a circular LEO has the advantage of reduced weight (no aerobrake) and less operational complexity. It has the disadvantage of utilizing considerably more propellant (up to 72% more for the 12 Klb delivery 10 Klb return mission). Paragraph 4.2 shows the LCC savings afforded by aerobraking amounts to approximately $13M per flight for space-based missions and approximately $9M per flight for ground-based missions.

2.2.2.2.2 Number of Engines

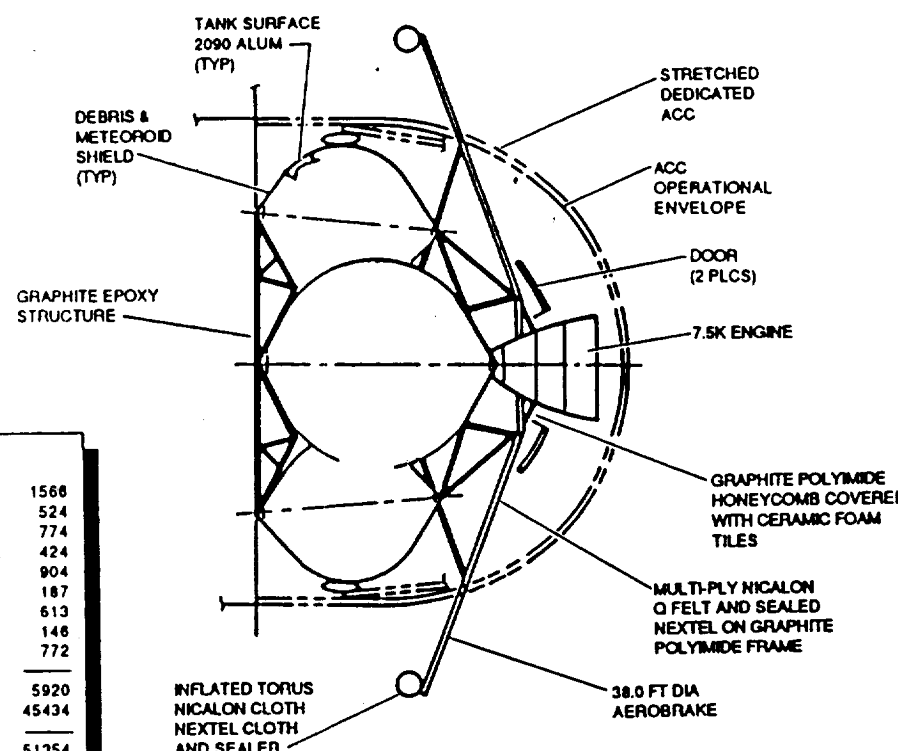
Paragraph 4.6 describes the trade study that shows a ground-based, LCV launched OTV should have three engines to achieve minimum life cycle costs. When launch costs based on vehicle length are relatively unimportant, as is the case for a space-based or an ACC launched ground-based OTV, two engines will have slightly lower LCC than three engines. For a non-manrated STS cargo bay launched OTV, a single engine nestled inside the torus oxygen tank provides the lowest LCC.

2.3 VEHICLE DESIGN SUMMARY

This section describes the selected OTV design concepts for the ground-based - STS launched OTV, the space-based OTV, two different size ground-based-LCV launched OTVs, and a hybrid OTV that can either be ground or space-based.

2.3.1 STS Ground-Based OTV

The general arrangement and weight breakdown of our selected ground-based STS delivered cryogenic OTV is shown in Figure 2.3.1-1. The four tank, single advanced technology engine configuration uses the volume and weight efficient principle suggested by Larry Edwards (NASA Headquarters) to fit easily into the Aft Cargo Carrier (ACC). The 38 foot diameter aerobrake folds for storage in the ACC. It is discarded after each flight. The aluminum/lithium propellant tanks are designed by engine inlet pressure requirements. Their thinnest gauges are 0.018 in. for the LO_2 tank and 0.014 in. for the LH_2 tank. The tanks are insulated with multi-layer insulation and spray-on foam insulation (SOFI). The hydrogen tanks are removed onorbit and, with the core system (LO_2 tanks, structure, avionics, and propulsion), are stowed in the orbiter bay for retrieval after OTV mission completion. The structure is of lightweight graphite epoxy. The propellant load was selected to enable full utilization of projected STS lift capability on GEO delivery missions.



WEIGHT	
AEROBRAKE	1566
TANKS	524
STRUCTURE	774
ENVIRONMENTAL CONTROL	424
MAIN PROPULSION	904
ORIENTATION CONTROL	187
ELECTRIC SYSTEMS	613
G,N & C	146
CONTINGENCY 15%	772
DRY WEIGHT	5920
PROPELLANTS, ETC.	45434
LOADED WEIGHT	51354

Figure 2.3.1-1 STS Ground-Based OTV

2.3.2 Space-Based OTV

The selected space-based OTV concept is shown in Figure 2.3.2-1. The brake/vehicle concept utilizes a wide "squatty" tankage package. This concept includes a central truss structure and subsequent side removable modular tankage. The vehicle utilizes a relatively low L/D (0.12) reusable 44 ft diameter aerobrake for control during the aerocapture maneuver which minimizes the thermal loads on the fabric brake and therefore its weight. This results in a low weight OTV with adequate control capability during the aerotrajectory. Two main engines are utilized to allow man-rating capability. The main engines have extendable/retractable nozzles which protrude through openings in the nose of the aerobrake. These openings are closed during the aerocapture maneuver with actuated doors.

This concept is intended to be launched only once and subsequently maintained in space. Therefore the design is relatively free of any launch vehicle constraints (such as diameter or length) except for the initial launch. The major components (tanks, structure, engines, etc.) are assembled into the flight configuration after their initial delivery to the Space Station.

WEIGHT	
AEROBRAKE	1800
TANKS	1025
STRUCTURE	1370
SUPPORT (ASE)	114
ENVIRONMENTAL CONTROL	730
MAIN PROPULSION	1288
ORIENTATION CONTROL	265
ELECTRIC SYSTEMS	533
G, N & C	180
CONTINGENCY 15%	1093
DRY WEIGHT	8378
PROPELLANTS, ETC.	74015
LOADED WEIGHT	82393

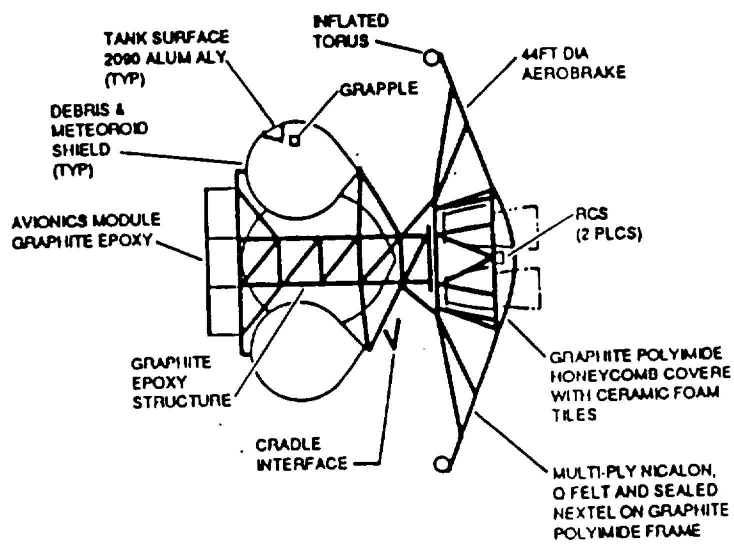
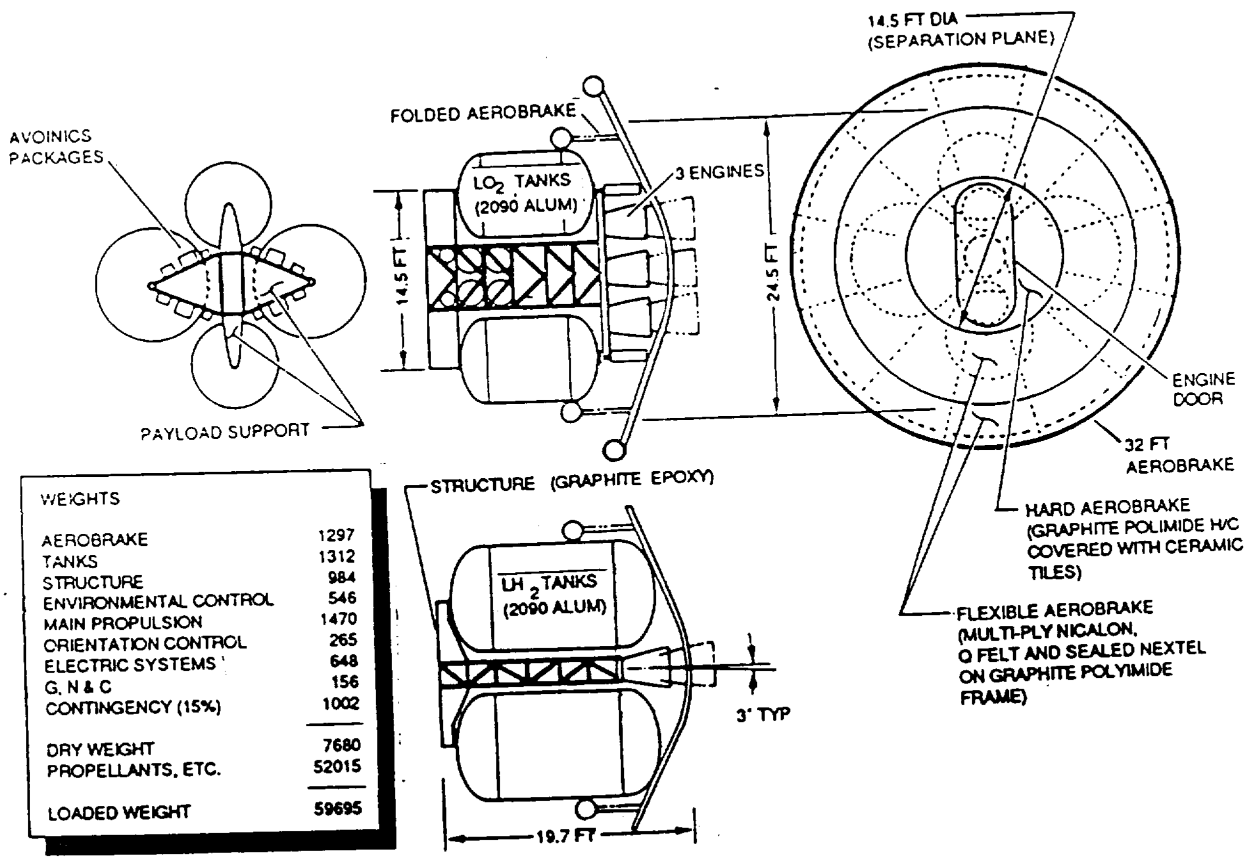


Figure 2.3.2-1 Space-Based Aeroassisted OTV

2.3.3 Large Cargo Vehicle OTV

2.3.3.1 52K OTV

The 3-engine OTV design concept shown in Figure 2.3.3-1 was developed for launch and return in a 25 ft diameter large cargo vehicle. The rationale for 3 engines is described in paragraph 4.6. The tankage diameters were chosen such that the combined length of the liquid oxygen tanks and the retracted engines would be the same length as the liquid hydrogen tanks. This results in the shortest vehicle length to minimize launch costs per the charging algorithm discussed earlier. The short length allows use of a 32 foot diameter aerobrake. The structure consists of a central core between the tanks that ties the tankage, aerobrake, and payload adapter together. This assembly remains as a unit after the mission when the aerobrake is jettisoned. If the LCV does not have the capability to return the OTV to earth after the mission, the OTV will be disassembled for return in the STS payload bay. The high volume, low cost cryogenic tanks are removed and the structural core is returned to earth with the high cost unit items such as main engines, power system, avionics, RCS, etc.

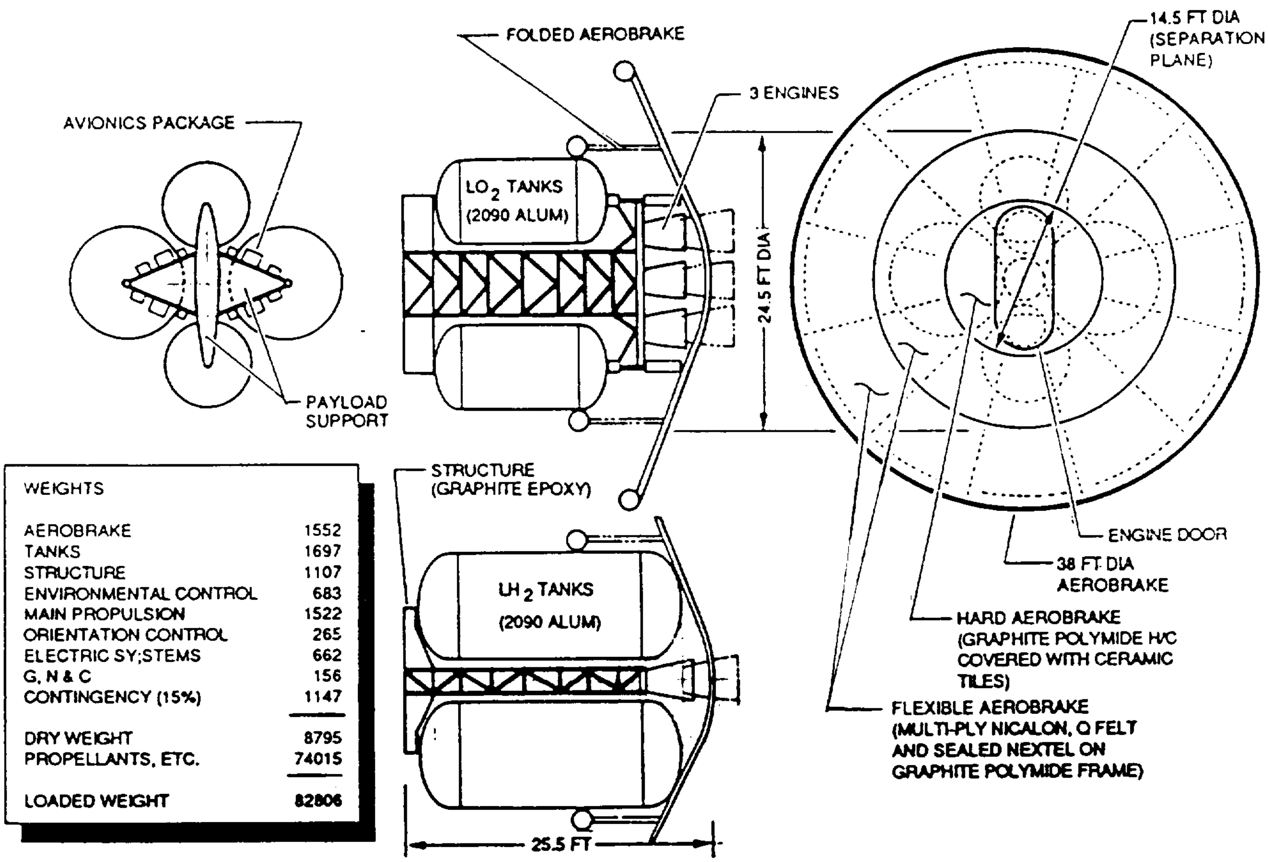


WEIGHTS	
AEROBRAKE	1297
TANKS	1312
STRUCTURE	984
ENVIRONMENTAL CONTROL	546
MAIN PROPULSION	1470
ORIENTATION CONTROL	265
ELECTRIC SYSTEMS	648
G, N & C	156
CONTINGENCY (15%)	1002
DRY WEIGHT	7680
PROPELLANTS, ETC.	52015
LOADED WEIGHT	59695

Figure 2.3.3-1 52K Ground-Based OTV

2.3.3.2 74K OTV

The vehicle concept depicted in Figure 2.3.3-2 is a "stretched" version of the 52 Klbm vehicle concept shown earlier. The major modifications are lengthened structure and added length in the propellant tank barrel sections. The aerobrake must grow in diameter from 32 feet to 38 feet to protect the longer stage and payloads. The core arrangement of the vehicle remains essentially the same with regard to vehicle diameter, engine configuration, avionics location, aerobrake hard shell design, etc. This vehicle is required to be capable of being man-rated.



WEIGHTS	
AEROBRAKE	1552
TANKS	1697
STRUCTURE	1107
ENVIRONMENTAL CONTROL	683
MAIN PROPULSION	1522
ORIENTATION CONTROL	265
ELECTRIC SYSTEMS	662
G, N & C	156
CONTINGENCY (15%)	1147
DRY WEIGHT	8795
PROPELLANTS, ETC.	74015
LOADED WEIGHT	82806

Figure 2.3.3-2 74K Ground-Based OTV

2.3.3.3 Hybrid OTV

An alternative exists to developing a space-based OTV in addition to a ground-based OTV. This alternative consists of utilizing kits to modify a ground-based vehicle to the extent that makes it suitable for space basing. The kits provide the required debris shielding, thermal protection, and modularity for onorbit servicing. Table 2.3.3-1 shows the weight impact to the ground-based 74 Klbm vehicle concept. These weight adjustments do not include a 15% contingency that would be reflected in the total vehicle dry weight.

ITEM	WT CHANGE (LBM)	REASON
DEBRIS SHIELD	+ 104	INCREASED METEOROID EXPOSURE TIME
ENGINE Q/D	+ 171	NOT ON GB
THERMAL - LH2	- 90	REPLACE 1/2 IN SOFI WITH MLI FOR 1 IN TOTAL
NET DIFFERENCE	+ 185	

Table 2.3.3-1 Modifications for Ground- to Space-Basing

Figure 2.3.3-3 shows the 74 Klbm propellant capacity OTV (ground-based) modified for use as a space-based vehicle. The weights reflect the modifications mentioned above. The vehicle is intended to be delivered to its

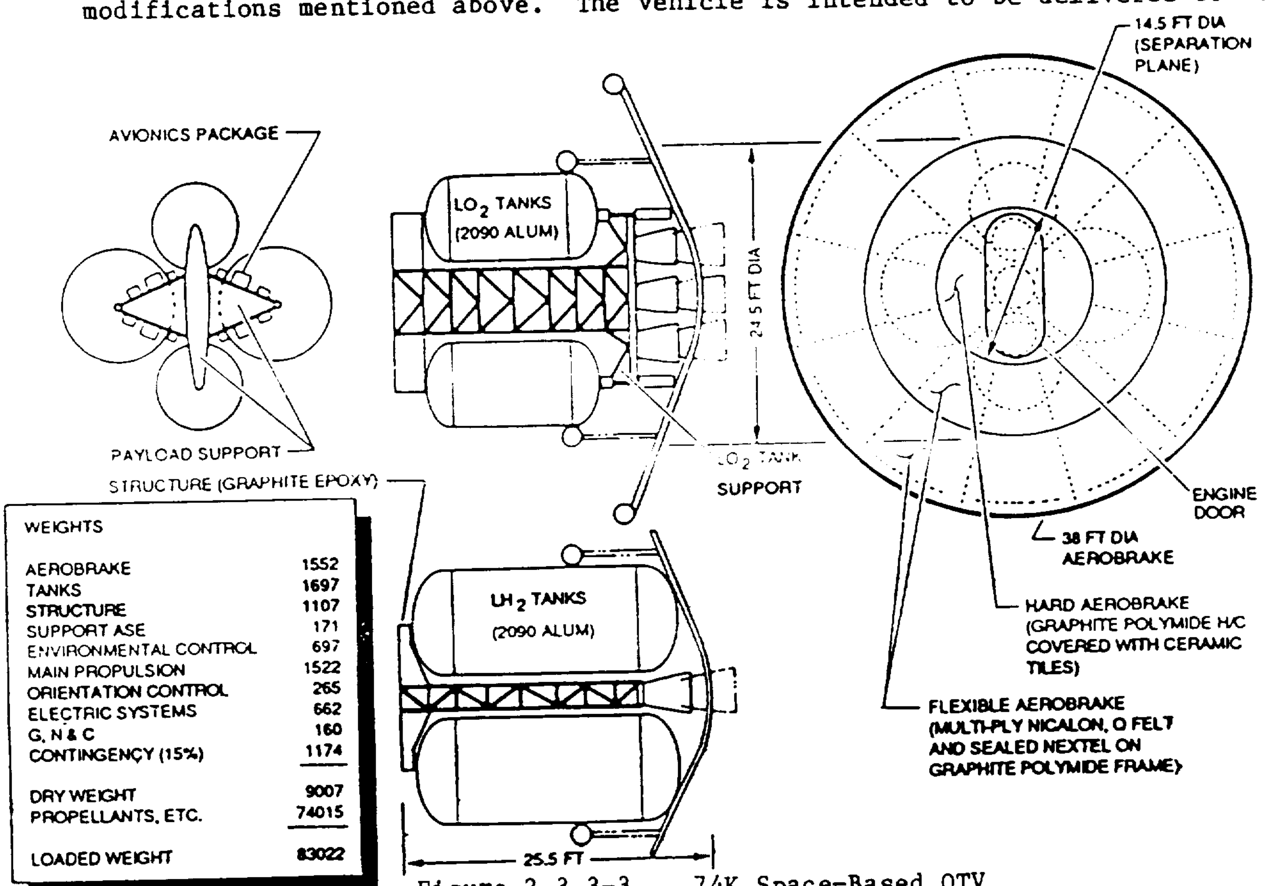


Figure 2.3.3-3 74K Space-Based OTV

space-base in one piece by the large cargo vehicle, and then accommodated and operated out of this space-base for its useful life. The reason for only one size of space-based OTV is that the cost of the propellant that could be saved by having a smaller OTV (in addition to the large one) is small compared with the development cost and Space Station accommodations costs for the extra stage.

Figure 2.3.3-4 summarizes the dry weight comparisons between the OTV concepts. The dashed line is typical of the weight-propellant capacity growth relationship.

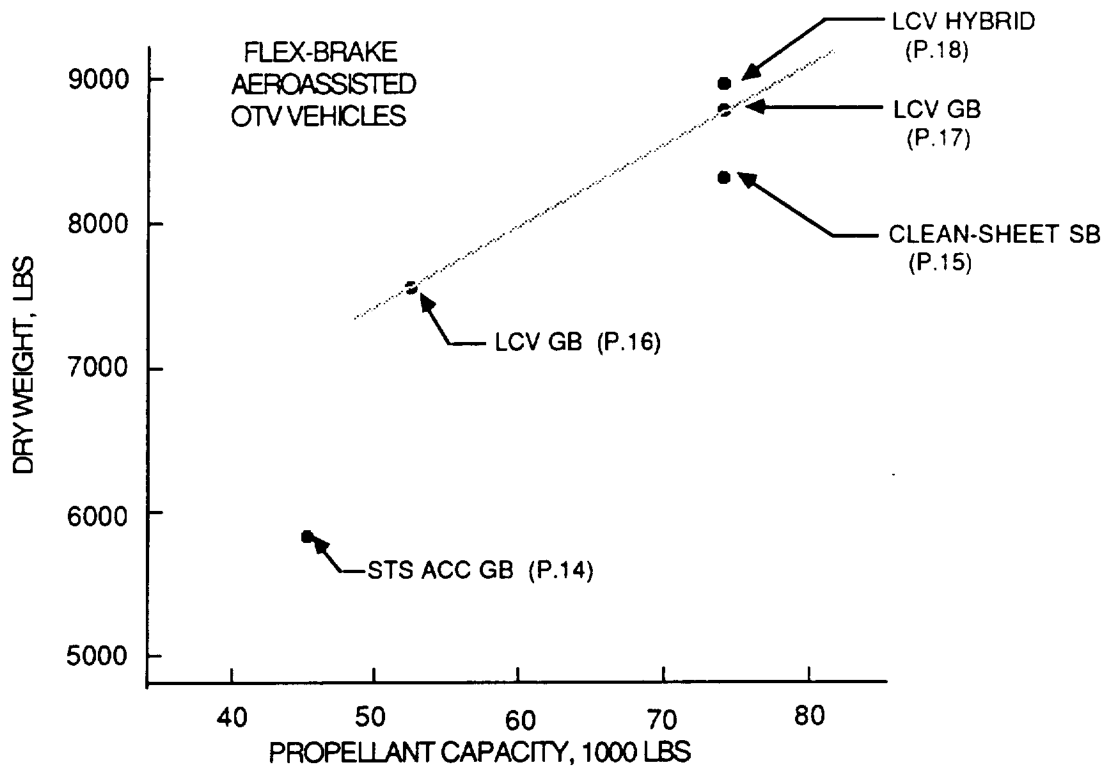


Figure 2.3.3-4 Space-Based OTV Weight Comparisons

2.4 OPERATIONS AND ACCOMMODATIONS SUMMARY

Operations and accommodations issues were reviewed to assess the impact of the Revision 9 mission model and delivery to LEO by a large cargo vehicle (LCV) which allows a wide body OTV.

Proximity operations near the Space Station were analyzed and three possible solutions investigated. It is recommended that a joint working group representing Space Station, OMV, and OTV review these proposals and designate the best solution.

Operational time lines were reviewed and event times substantiated for GEO, Lunar, and Planetary type missions. A review of the Ford Aerospace and LMSC documentation on geostationary platforms proposed for the 1995 - 2000 time period show that the OTV system can meet all performance and support requirements for delivery of these systems to orbit.

Flight Operations and Ground Operations were analyzed and requirements defined for ACC, Shuttle Payload Bay, and LCV delivery of an OTV system. Operational requirements in support of the various aerobrake configurations for both space-based and ground-based OTV were defined and methodology developed. Aerobrake TPS inspection techniques were evaluated and recommendations made for inspection aids.

A number of trade studies were also performed: an operational comparison of the flexible brake, ballute, and shaped brake; comparison of methods to deorbit expended propellant tanks; and change out methodology for the 3-engine wide body OTV. Turnaround times needed for space-based and ground-based OTVs were determined, minimum required fleet size was determined; and production rates were established for the OTV system and major replaceable components.

Space Station accommodations from the initial study phase were reviewed and changes recommended. Changes included a smaller hangar, a smaller propellant storage facility, and a revised estimate of robotic software and hardware requirements. Reduction in requirements lowered the estimated cost of IOC accommodation to 45% of that proposed in the initial study phase. A trade study analysis of EVA/IVA requirements was conducted with the resultant recommendation for SBOTV, that processing and servicing be performed by IVA supervisory control of a robotic manipulator arm. Space-based operations for servicing, checkout, maintenance, and propellant loading/unloading were reviewed, operations times and IVA involvement evaluated and accommodation requirements assessed.

2.4.1 Flight Operations

2.4.1.1 Proximity Operations

Further study is necessary to determine the best approach to performing the proximity operations involved with returning an OTV and an attached spacecraft to the Space Station. OMV, OTV and Space Station all are involved, and the best solution may involve compromises in all three programs. We started this activity by identifying and assessing the candidate approaches as described in Paragraph 7.2.1. Option 1 uses the OMV and also adds hot and cold gas RCS clusters to the OTV to provide a full capability for six-degree-of-freedom control of the integrated package through to final Space Station docking. Option 2 provides a complete capability within the OTV design so it can safely approach the Space Station with no support from OMV. The third option leaves the OTV with its current minimal RCS capability and relies on procedural changes to implement the solution. The OTV and payload are separated from one another and ferried to the Space Station by the OMV on two separate trips. This enables the OMV to dock at the payload interface and minimizes interference between the OMV RCS system and the OTV aerobrake.

2.4.1.2 Flight Operations Requirements, LCV Delivery of a Wide Body GBOTV

The OTV and payload will be delivered to LEO fully assembled and intact. The OTV/Payload will be released from the LCV and allowed to coast for up to 12 hours for prepositioning prior to launch. Ground control will conduct checkout of both the OTV and payload prior to initiating and engine burn. Launch-from-LEO operations are then conducted, the mission performed, and the returning OTV executes the aeropass maneuver. At the end of the aeropass maneuver, the OTV jettisons the flexible portion of the aerobrake. The OTV is then injected into a low circular orbit in the range of 100 - 150 nmi. As the OTV reaches its desired orbit, the accumulators are fully charged and the LH_2 tanks are jettisoned. In the case of the larger OTV (74K), one of the LO_2 tanks is also jettisoned. The OTV then performs an ignition burn utilizing the accumulator gases to gain a higher orbit. Once there, all systems are shut down and the inert OTV awaits STS rendezvous. The STS

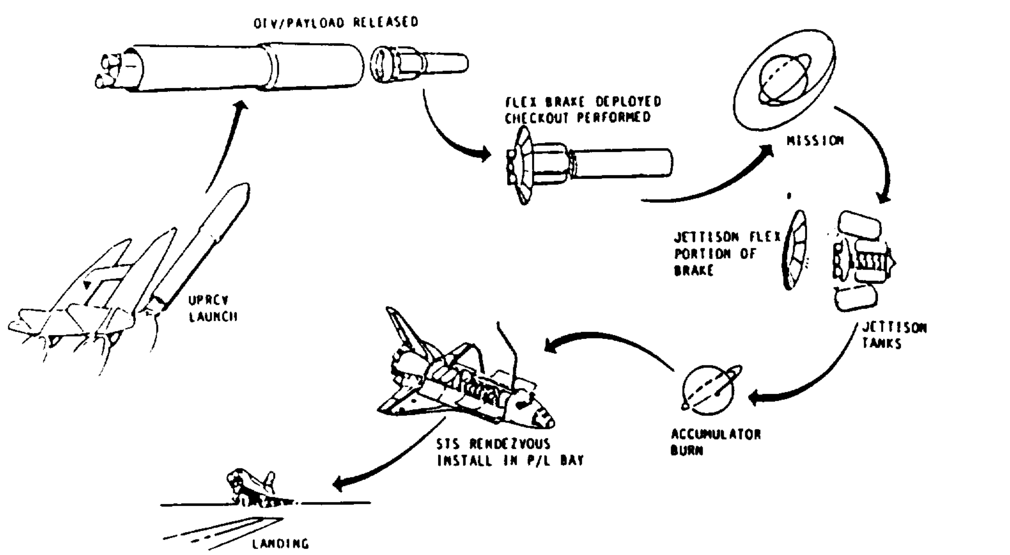


Figure 2.4.1-1 LCV Delivery, Unmanned GBOTV

performs rendezvous with the OTV, grapples it, and secures it to the Payload Installation and Deployment Aid (PIDA). Using the RMS, the LO_2 tank(s) are removed and installed in the payload bay. The remaining core structure with engines, avionics, and rigid core portion of the brake are then loaded into the bay.

2.4.1.3 Flight Operations Requirements, LCV Delivery of a Wide Body SBOTV

For the space-based Wide Body OTV, each new OTV delivery will be handled as a GBOTV launch. Subsequent delivery of payloads and OTV spare components by the LCV are to ZONE 4 behind the Space Station. The OMV rendezvous with the LCV and ferries the payload and/or component spares to Space Station. At Space Station, for each subsequent mission beyond the initial delivery of each OTV, payload mating, propellant loading, checkout, and deployment from the station are performed. Ground control conducts Launch-from LEO operations, the mission performed, and the returning OTV executes the aeropass maneuver. OTV will be injected into orbit behind Space Station at the designated pickup point to await rendezvous with the OMV for transport to Space Station. Once at Space Station, propellant detanking is performed and inspection of the returned OTV takes place. Diagnostic testing will be performed and any necessary maintenance action taken. The OTV is then placed in storage to await the next mission.

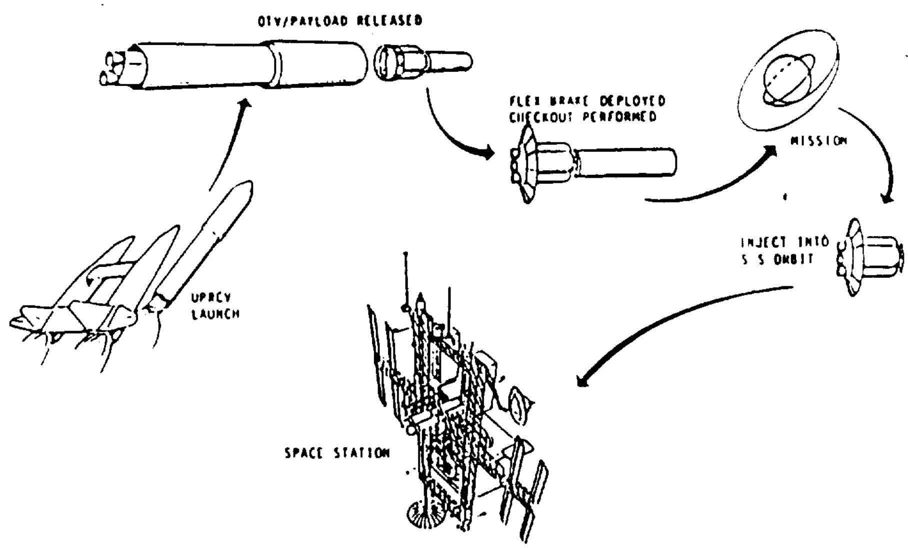


Figure 2.4.1-2 LCV Delivery, SBOTV

2.4.2 Space Station Accommodations

2.4.2.1 EVA/IVA Comparison

When considering whether to perform processing operations at Space Station by EVA or IVA, it is not just a decision between robotics and manual EVA. Automation is a continuum stretching from hands-on operations through to autonomous robotics. Level of complexity and development costs soar as operations are made completely automated. A degree of manual intervention tends to keep cost down by allowing human decision making to determine what to do next, and then have the robot do a limited set of tasks. This is referred to as supervisory control.

For OTV processing support from the Space Station, we must also consider the availability of personnel at the station for OTV related activities. By utilizing an IVA astronaut, supervisory control, and an RMS robotic arm we minimize both the demands made on the astronaut and the time necessary for turnaround of an OTV mission.

We conducted an in-depth trade study to assess the level of automation that should be incorporated in space-based OTV support operations. This assessment included evaluation of the parameters listed below. Consideration was given to performing specific operations with EVA, remote operations with an IVA crew member providing control, and fully automated robotic operation. We found that remote operations were preferable to fully automated operations in most cases, although the precise level of automation depends on the specific task. The ranking shown in the chart below is generically indicative of the preferred approach, however, we felt that operations should be biased toward automation due to the restriction of crew availability at the Space Station.

Table 2.4.2-1 EVA/IVA Trade Study Results

PARAMETER	EVA	RMS (TELEOP)	AUTO ROBOTICS
OPERATIONAL CREW REQUIREMENTS	1	5	10
MAINTENANCE CREW REQUIREMENTS	10	5	1
DEVELOPMENT COST	10	8	1
OTV DESIGN DRIVERS	10	9	8
TPS INSPECTION AND REPAIR	5	4	2
PROPELLANT LOADING	1	8	10
OPERATIONAL COST	1	7	10
PAYLOAD MATING	1	10	6
PRE-LAUNCH TESTING	1	10	9
SCHEDULED/UNSCHEDULED MAINTENANCE	1	9	10
TOTALS	41	75	67

2.4.2.2 Space Station Accommodations Cost Revision

A revised accommodations cost estimate was generated for the various cost trades being performed as part of the study effort. As can be seen in Table 2.4.2-2, the revised cost figures are significantly lower than those used during the initial study phase. It had been assumed that OTV would have to bear the entire development cost of robotic hardware. It is now felt that this cost should drop drastically due to two separate factors: firstly that Space Station and OMV have an equal need for the development of this hardware and should share the cost. Secondly, with the many advances currently occurring in this field, cost should be dropping. Imaging system requirements for OTV could well be adapted from that developed for OMV to meet the needs for on-orbit satellite servicing. Software requirements, hangar size and tank farm needs are discussed in Section 7. Transportation costs represent the difference between the Shuttle and the LCV launch costs and capabilities.

Table 2.4.2-2 IOC Accommodations Costs for OTV

ITEM	PHASE A COST $M	REVISED COST $M	COMMENTS
ROBOTIC HARDWARE	165	96	SHARED COST ITEM (OTV, OMV, & SS)
STEREO-VISON IMAGE SYSTEM	100	30	ADAPTATION OF OMV SYSTEM
SOFTWARE	285	57	RE-ASSESSMENT OF REQUIREMENTS REDUCES LOC FROM 2M TO 400K
HANGAR	76	65	43X42X90 FT 1 OTV + 55 FT PL SIZED FOR GEO MISSIONS
TANK FARM	170	120	100 LBS PROP CAPACITY
TRANSPORTATION	140	50	LCV LAUNCH COSTS
TOTAL	936	418	

2.5 MAJOR PROGRAM AND VEHICLE RECOMMENDATIONS

2.5.1 Baseline Program Description

We have concluded that the preferred Orbital Transfer Vehicle program in the era where a large cargo vehicle is available and Scenario 2 missions are to be performed will be as summarized in Figure 2.5.1-1. It will comprise two types of orbital transfer vehicles. A three in-line engine, four side-by-side tanks, unmanned, ground-based vehicle with a 52,000 pound propellant capacity will support initial missions. This vehicle will be used throughout the operational period. A generally similar manned, space-based vehicle with a 74,000 pound propellant capacity will be made operational as soon as it can be supported by the Space Station. All manned missions will be launched from a space-base, but the space-based vehicle can be launched from the ground as well. Its initial mission will be ground-based -- returning to residence at the Space Station upon return.

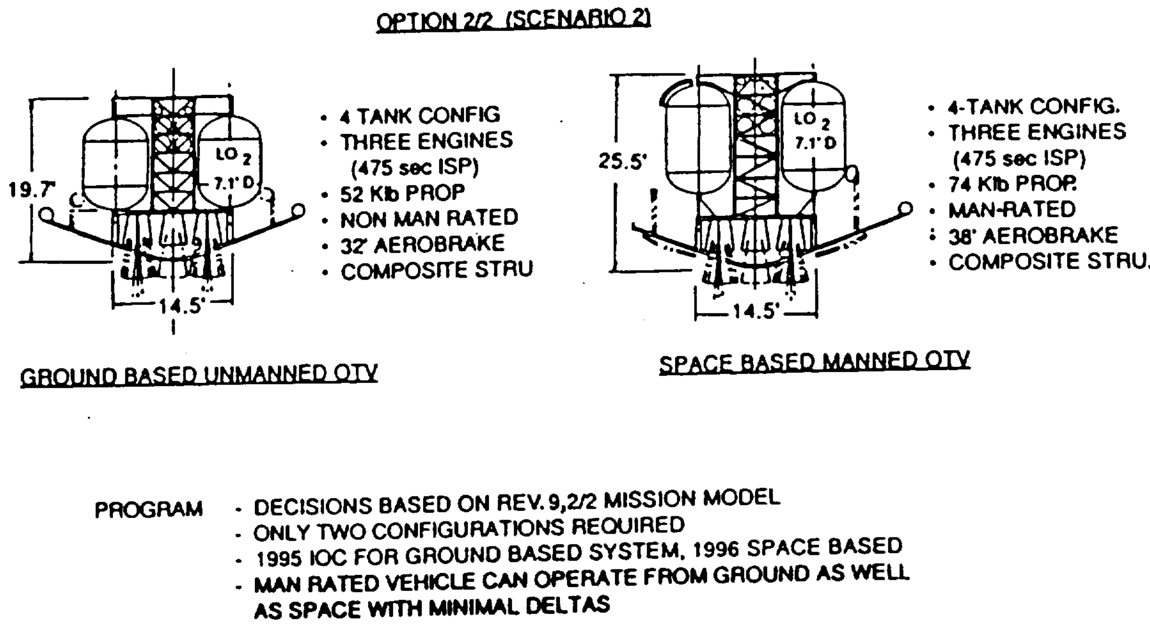


Figure 2.5.1-1 Nominal C/V OTV program

The major cost and schedules associated with the OTV program summarized in Figure 2.5.1-1 are summarized in Figures 2.3.1-2 through -4. Figure 2.5.1-2 shows a spread of the major cost elements involved in capturing the Scenario 2 DOD and Civil Mission Model. The total acquisition cost for R&T, DDT&E for both ground and space-based stages and space-base accommodations, and vehicle and accommodations production is \$2B. The total cost of operations through CY 2010 is \$22.1B. The bulk of the operations cost is associated with DOD missions.

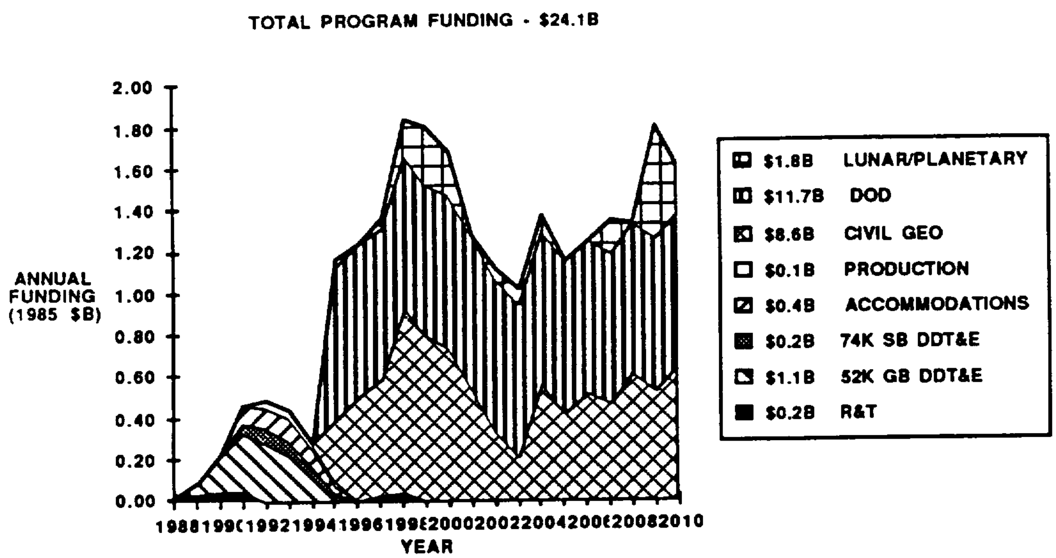


Figure 2.5.1-2 Nominal C/V OTV Program Funding

The development schedule for the ground-based OTV is summarized in Figure 2.5.1-3. An ATP on January 1, 1989 supports an Initial Operational Capability in January 1995. A space-based OTV program ATP in January 1990 (Figure 2.5.1-4) supports an Initial Operational Capability in January, 1996. It is currently anticipated that this is the earliest space-based operational capability that can be supported, and that an initial capability near the turn of the century would be more likely to occur.

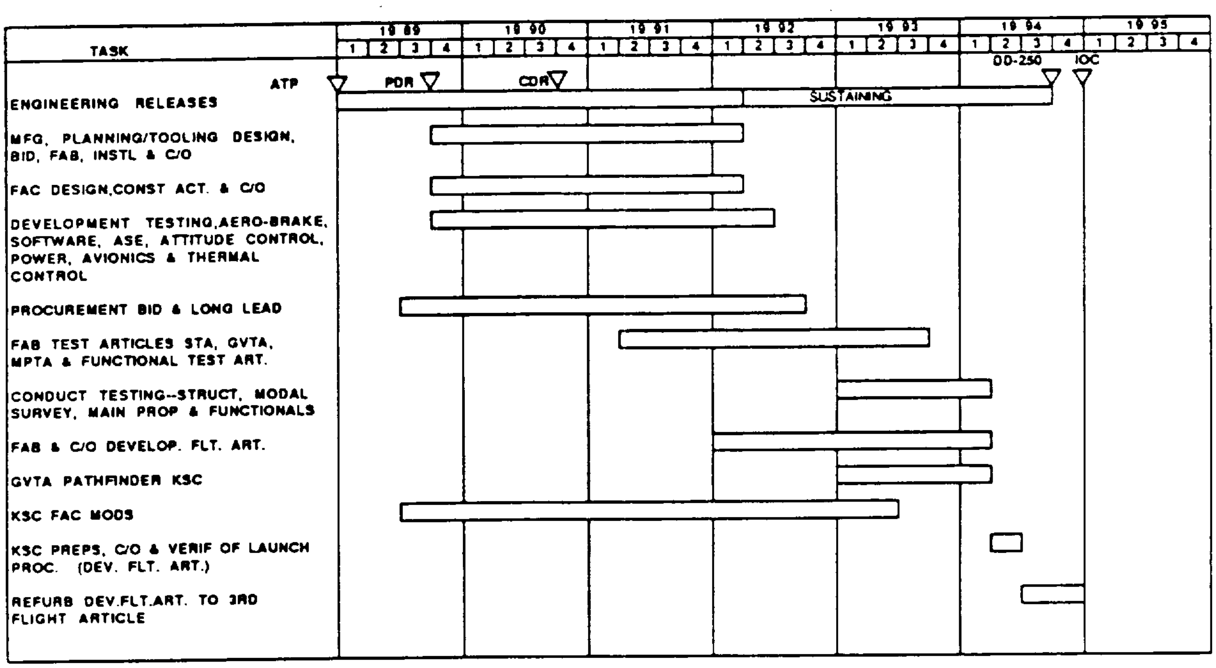


Figure 2.5.1-3 Ground-Based OTV Schedule

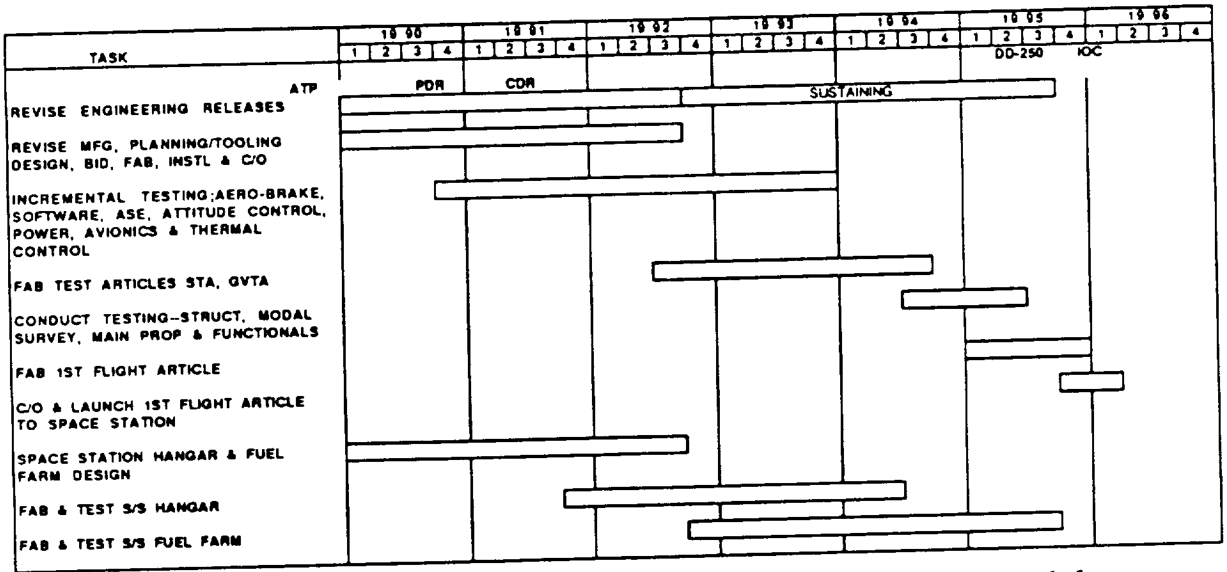


Figure 2.5.1-4 Earliest Capability Space-Based OTV Schedule

2.5.2 Program Sensitivities

The major characteristics of the five mission scenarios investigated are shown in Figure 2.5.2-1. Scenario 1 does not present a driver for space-basing, particularly because it contains no manned GEO missions. Scenario 2 justifies the nominal OTV program just discussed. Scenario 3 requires nothing different from the OTV program, assuming that the limited SDI mission activity is not multiple-launched on OTVs. Scenario 4 justifies a specialized OTV directed at the low mid-inclination and other DOD traffic. Scenario 5 justifies a specialized nuclear waste OTV which has a strong possibility of being able to perform selected DOD missions more effectively as well. This scenario also requires build-up of multi-stage OTVs at the space-base and requires that more OTVs be resident in space.

	MAJOR CHARACTERISTIC	IMPACT ON OTV PROGRAM
SCENARIO 1	NO MANNED GEO PRIOR TO 2010	NO SPACE BASED DRIVERS
SCENARIO 2	A BALANCED, BUT ACTIVE PROGRAM	NOMINAL
SCENARIO 3	MINIMAL CHANGE FROM SCENARIO 2	NO CHANGE
SCENARIO 4	HEAVY DOD TRAFFIC TO MID-INCLINATION	SPECIALIZED 40K, DOD OTV
SCENARIO 5	AGGRESSIVE PROGRESS TOWARDS 50-YEAR INITIATIVES	•MULTI-STAGE BUILDUP AT SPACE BASE • SPECIALIZED NUCLEAR WASTE DISPOSAL OTV

Figure 2.5.2-1 Mission Model Impact on OTV Program

The major characteristics of the five launch scenarios investigated are shown in Figure 2.5.2-2. The problem with STS growth is that there may not be much of it. If the OTV program is limited to the use of a shuttle with a 65,000 pound payload capability, many of the early missions in all the models will involve multiple launches with attendant operational problems. In this scenario, space-basing has even more virtue than in the cargo vehicle supported era we concentrated on in this extension study. The large cargo vehicle without retrieval capability results in the recommended OTV program previously discussed. The preferred OTV configuration for this case has been shown to be the wide body configuration. This approach leads to the operational complexities cited in the ground-based case. We would, of course, prefer the wide retrieval capability if only its operational cost is involved. The justification of the development cost of this capability is beyond the scope of this OTV study. Propellant hitchhiking and scavenging are the economic savior of the space-based OTV concept. This justification is real, but will likely prove upsetting to the users that are paying the launch bill. They would likely prefer to share in the cost benefit. The impact of STS II on OTV program selection appears to be minimal.

	MOST SIGNIFICANT FEATURE	IMPACT ON OTV PROGRAM
STS GROWTH	• HEAVIER LEO CARGO • ACC 'UP' VOLUME	MULTIPLE LAUNCH MISSIONS: SPACE BASE BENEFIT
LARGE CARGO VEHICLE (NO RETRIEVAL)	LOW COST TRANS-PORTATION TO LEO	GND BASED OPS COMPLEX - VEHICLE DISASSEMBLY - EXPENDABLE TANKS
LARGE CARGO VEHICLE (WITH RETRIEVAL)	LARGE OTV RETRIEVAL CAPABILITY	ENHANCES GROUND BASED OTV PROGRAM
PROPELLANT HITCHHIKING & SCAVENGING	NO PROPELLANT TRANSPORT CHARGE	PROVIDES ECONOMIC JUSTIFICATION FOR SPACE BASING
STS II	LOW COST MANNED LAUNCH	MINIMAL

Figure 2.5.2-2 Launch Vehicle Impact on OTV Program

Four possible space-basing scenarios are identified in Figure 2.5.2-3. With no space-based support, missions that cannot be launched from the ground on a single flight require complex orbiter support operations. For example, launching a manned GEO mission would require two current capability orbiter launches on one week centers with orbiter supported onorbit mission assembly. With a 65,000 pound capability STS, the occurrence of this problem is frequent. With a large cargo vehicle, the problem will eventually occur. Space tending with the Space Station would ease this problem, but the timing would still be constrained unless the ability to top propellants were provided as a part of the space tending package. This approach does not enable acquiring the potential benefit of the hitchhiked propellant concept. The nominal space-based approach achieves all the operational benefits previously discussed, and mitigates the cost of this capability with the benefit of hitchhiked propellants. If OTV Space Station activities were delayed until the manned missions are scheduled, the impact would be: The large early missions would require either complex ground-based operations or more payload segmentation; and the operational base that is required to pay off Space Station accommodation developmental costs would be beyond the horizon of this study.

	MOST SIGNIFICANT FEATURE	IMPACT ON OTV PROGRAM
NO SPACE BASE SUPPORT	—	REQUIRES COORDINATED RAPID L/V TURNAROUND AND COMPLEX ORBITER SUPPORTED LEO OPERATIONS
SPACE TENDING	SUPPORTS LEO MISSION ASSEMBLY	DECOUPLES L/V AND OTV OPERATIONS AND PROVIDES LEO OPNS SUPPORT
NOMINAL SPACE BASE	AVAILABLE FOR LARGE UNMANNED GEO	ENABLES: - SUPPORT OF ALL LARGE MISSIONS - PERMANENT OTV SPACE RESIDENCE - 'HITCHHIKE' BENEFITS (FEWER L/V LAUNCHES)
DELAYED SPACE BASE	AVAILABLE FOR MANNED GEO	EARLY LARGE GEO MISSIONS REQUIRE COMPLEX LEO OPNS

Figure 2.5.2-3 Space-Basing Impact on OTV Program

Development of the reusable OTV is economically justified, even in the most modest projected mission scenarios. We believe that, even though it is difficult to justify on a discounted life cycle cost basis, the lower operational costs justify investment in space-basing. Further Phase A effort should be directed at identifying an initial OTV that will be useful whether or not a large cargo vehicle program is initiated in the near future, and one that has a good growth path to space-based capability. We believe the key to meeting this objective is to develop a concept that can fly in an Aft Cargo Carrier or a large cargo vehicle with minimal design penalty. After this concept is delineated, an extended Phase B study contract should optimize the concept; and a full scale development directed at achieving a mid 90's initial operational capability should be undertaken.

3.0 MISSION AND SYSTEM REQUIREMENTS ANALYSES

3.1 MISSION MODEL AND GROUND RULES

The analyses described in this document differs from the analyses published in Volumes I through VIII in that it is based on a new mission model (Rev. 9) and different launch vehicles. The previous studies were constrained to the low and nominal versions of the Rev. 8 mission model; this study examines the five Scenarios of Rev. 9. The previous studies used only STS (with or without an ACC) for launch; this study considers STS, STS II, and a new large cargo vehicle with and without return-to-earth capabilities.

3.1.1 Mission Model Analyses

The Rev. 9 mission model is derived from the Space Transportation Architecture Study (STAS) mission models. The STAS model defines four traffic options for both the civilian and the DOD programs. The OTV study was ground ruled to consider five of the 16 possible combinations, as shown in Figure 3.1.1-1. The circled numbers are used to designate the scenarios. Scenario 2, which represents the baseline civil and normal growth DOD requirements, was designated by MSFC to be the basis for all design decisions and recommendations. The other scenarios were to be examined for sensitivities.

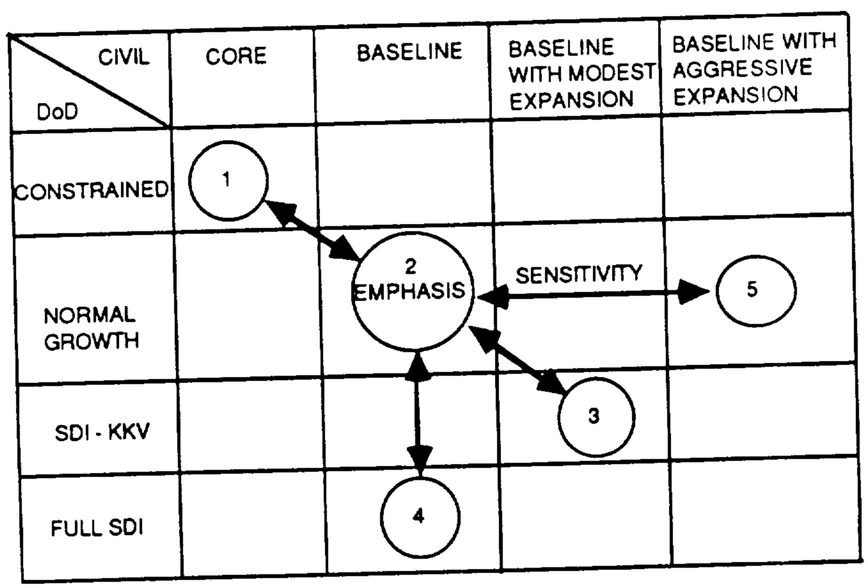


Figure 3.1.1-1 STAS vs Rev. 9 Mission Model Scenario Designations

Significant features of each of the scenarios are:

Scenario 1 has no manned missions and no lunar missions, but overall traffic is equivalent to the Rev. 8 nominal model.

Scenario 2 contains manned GEO missions, an early requirement for a 15 Klb GEO delivery with subsequent 25 Klb GEO delivery missions, and considerable traffic in multiple payload deliveries.

Scenario 3 in the STAS model shows a considerable increase in LEO traffic, but this does not reflect in additional OTV missions. The main difference from Scenario 2 is three additional high energy planetary missions.

Scenario 4 includes very heavy traffic of large payloads to mid-inclination orbits of relatively low altitude.

Scenario 5 includes 100K payloads to GEO (segmented into 25K deliveries), a manned lunar program, a large lunar station with many lunar logistics missions and missions designated as nuclear waste disposal.

Table 3.1.1-1 shows the total traffic from 1995 - 2010 for the Rev. 9 scenarios and the Rev. 8 options. Since OTV design decisions prior to this study extension were based on the low Rev. 8 model, the mission composition of Scenario 2 can be expected to cause changes in previous conclusions.

Table 3.1.1-1 Mission Model Comparisons

TYPE OF MISSION	REVISION 8		REVISION 9 SCENARIOS				
	LOW	NOMINAL	1	2	3	4	5
GEO TOTAL CIVIL (MANNED) ((EARLIEST))	68 3 2008	144 17 2002	102 0 N/A	160 16 2002	160 16 2002	165 16 2002	202 22 1999
LUNAR (MANNED) ((EARLIEST))	2 0 N/A	14 3 2006	0 N/A N/A	8 0 N/A	8 0 N/A	8 0 N/A	14 8 2006
PLANETARY	6	14	14	14	17	14	25
CIVILIAN SUB-TOTAL	76	172	116	182	185	187	241
DOD	68	85	176	240	240	480	240
NUCLEAR WASTE	0	0	0	0	0	0	391
TOTAL	144	257	292	422	425	667	872

The civilian GEO missions are categorized by Scenario in Table 3.1.1-2. As shown, the quantity of multiple payload delivery missions combined with the 12 Klb delivery/10 Klb return missions dominate the civilian GEO missions. The data used to define the multiple payload delivery mission is shown in Table 3.1.1-3. In all cases the multiple payload adapter is assumed to weigh 2000 lbs and have a return length of 10 feet.

Table 3.1.1-2 Civilian GEO Missions

	SCENARIO				
	1	2	3	4	5
NUMBER OF 12K UP/ 2 K DOWN (MULTIPLE PAYLOAD)	84	84	84	84	84
NUMBER OF 12 K UP/ 10 K DOWN (NUMBER MANNED)	0	53 (16)	53 (16)	53 (16)	77 (22)
NO. OF OTHER DELIVERY (AVG. DELIVERY WGT.)	16 (16.0)	21 (17.0)	21 (17.0)	26 (17.6)	39 (17.8)
NUMBER OF 10 K RETURN	2	2	2	2	2
TOTAL NUMBER	102	160	160	165	202

Table 3.1.1-3 Multiple Payload Delivery Mission

REV 9 DESIGNATION	WGT	QTY	PERCENT ASSUMED AVAILABLE FOR OTV LAUNCH	QTY MANIFESTED PER OTV MISSION	AVG. LENGTH FT	MAX. LENGTH FT
CLASS "A"	0-2030LBS	245	50	4	9.7	16.4
CLASS "B"	2031-2500LBS	38	50	3	10.7	15.1
CLASS "C"	2501-5005LBS	117	100	2	23.5	35.1

THE RESULTANT REV 9 MISSION MODEL MULTIPLE GEO PAYLOAD DELIVERY MISSION IS DEFINED AS:

PLD NO	WGT UP	LENGTH UP	WGT DN	LENGTH DN	TOTAL QTY
18912 (a)	12,000 LBS	35 Ft	2,000 LBS	10 FT	28
(b)		20 Ft			28
(c)		12 Ft			28

Lunar mission requirements (by year) are elaborated in Table 3.1.1-4. As shown, Scenario 5 includes manned missions which dictate returning a 20 Klb payload that is 22 feet long.

Table 3.1.1-4 OTV Lunar Missions

	SCENARIO				
	1	2	3	4	5
TOTAL NUMBER	0	8 (0 MANNED)	SAME AS 2	SAME AS 2	14 (8 MANNED)
AVG WGT	-	42.3			52.1
LARGEST WGT,KLB (YR)	-	5K (99) 33K(03) 73K(09)			5K(96) 33K(00) 93(08)
LARGEST UP/DOWN	-	0			73K/20K (06)
AVG LENGTH,FT	-	24.5			45.7

Table 3.1.1-5 shows that the DoD missions are essentially identical for all cases, except that Scenario 4 adds a large number of heavy, low altitude, mid-inclination missions.

Table 3.1.1-5 Generic DOD OTV Missions

MISSION TYPE	PLD WGT	CIRC ORBIT ALT K nm	CIRC ORBIT INC DEG	QUANTITY BY SCENARIO 1	2	3	4	5
POLAR	5 K	4.0	90	16	16	16	16	16
GEO	10 K	19.3	0	32	96	96	96	96
MID-INC.	10 K	19.3	63	128	128	128	128	128
LOW-MID INC	110 K	1.0	63	0	0	0	240	0
TOTAL				176	240	240	480	240

Table 3.1.1-6 summarizes the Rev. 9 missions which drive the design of the OTV. The missions are the same for all five scenarios. In Scenario 5, payload 15009 (manned portion of the GeoShack payload) flies in 1999 rather than 2004. However, since the propellant required for the 12 K up/10 K down missions (which occur in 1999 in Scenario 2) is essentially the same, the schedule change does not impact design; but it does require man rating in 1999 rather than 2002.

Table 3.1.1-6 Rev. 9 Design Driver Missions

REV 9 MISSION NUMBER	NAME	WEIGHT (Lb)	L x Diam (ft)	Flight (yr)	g Limit
18072	MOBILE SAT - B	14,550	19.7 x 13.1	1995	0.1
18308 / 18309	H-F DIRECT BROADCAST SATELLITE (VOA)	33,070(1)	30 x 14.9	1996	0.1
18751	COMM.SAT. CLASS IV	10,030 (D & R)	30 x 14.8	1998 2001	0.1
18074 / 18075	SETI GEO ANTENNA	33,070(2)	30 x 14.9	1999	0.1
15011	GEOSHACK LOGISTICS	12,000 D 10,000 R	15 x 15	1999 annual	
15009	GEO SHACK (MANNED PORTION)	25,080	19.8 x 14.9	2004	

(1) CAN BE SEGMENTED INTO 2 OR 3 PIECES (WITH 10% WGT PENALTY) TO KEEP DELIVERY WEIGHT BELOW 15,000 LBS.

(2) CAN BE SEGMENTED INTO 2 PIECES (WITH 10% WGT PENALTY) TO KEEP DELIVERY WEIGHT BELOW 22,000 LBS.

3.1.2 Study Ground Rules

Major ground rules that formed an integral part of this study, and which affect study results are summarized below.

- Space Station IOC is 1996; FOC can be as desired, but no earlier than 1996.
- GEO payloads in excess of 25,080 lbs can be segmented and flown on multiple missions.
- OTV's can be staged and may utilize tank kits to perform high energy lunar and planetary missions.

- DOD payloads are not to be used as design drivers, but the mission traffic can be utilized to amortize development costs.
- Each mission shall have a probability of 0.999 or greater that there will be no debris or meteoroid impact on propellant tank walls.
- Launch vehicle performance, schedules and costs are as described in paragraph 3.2.
- OMV and Space Station operations costs are as described in paragraph 8.0.
- Mission analyses and duration ground rules are as described in paragraph 3.3.
- Low cost transportation for propellant for a space-based OTV is as described in paragraph 3.2.
- OTV hardware life requirements are as described in paragraph 7.0.

3.2 LAUNCH VEHICLE CHARACTERISTICS

3.2.1 Ground Rule Capabilities

Table 3.2.1-1 lists the IOC, weight and volume capacities, launch costs and the parametric sensitivities of candidate launch vehicles specified in the study ground rules. The cost data is presumed to be operational costs only, not including amortized DDT&E and production costs.

The STS was specified to have 30,000 lb normal and 61,000 lb abort landing limits. The charges associated with returning an OTV were baselined as consisting of the STS launch costs for the return ASE and the extra on-orbit operations time involved with rendezvous, recovery and stowage.

Table 3.2.1-1 Ground Rules for Launch Vehicles

VEHICLE	IOC	CAPACITY	COSTS	SENSITIVITIES
STS	NOW	60 FT x 15 FT 72 Klbs TO LEO	$73M / FLT ($1123 / LB, $1.2M / FT)	65 - 81 KLBS TO LEO
DEDICATED STRETCHED ACC	1995	21.2 FT x 27 FT	$2.4M / FLT + $171M DDT&E	
STS II	2002	60 FT x 15 FT 65 Klbs TO LEO	$20M / FLT ($307 / LB, $0.33M / FT)	20 - 30 $M / FLT 250 - 500 $ / LB 40 - 80 KLBS TO LEO 15 - 23 FT DIAM 30 - 70 FT LENGTH
LCV w/o RETURN	1995	90 FT x 25 FT 150 Klbs TO LEO	$70M / FLT ($467 / LB, $0.78M / FT)	50 - 85 $M / FLT 250 - 600 $ / LB 100 - 200 KLBS TO LEO 22 - 33 FT DIAM 90 - 100 FT LONG
LCV w/ RETURN	1995	40 KLB RETURN 90 FT x 25 FT 150 Klbs TO LEO	$85M / FLT ($567 / LB, $0.94M / FT)	20 - 85 $M / FLT 350 - 1100 $ / LB 40 - 150 KLBS RETURN 15 - 25 FT DIAM 40 - 90 FT LONG

3.2.2 Launch Cost Charges

The study ground rule was, "launch charges for cargo vehicles and Shuttle II will assume the same user charge policy as the STS." The STS charging algorithm defined in JSC-11802, "STS Reimbursement Guide", is graphically depicted in Figure 3.2.2-1 for a large cargo vehicle with 150K capacity, 90 foot long payload bay and $70M launch costs. Payloads can share launch costs provided they do not require more than 75% of the launch vehicle capacity. (The weight fraction and length fraction of available capacity are calculated separately; only the largest value is used). When a payload requires 75% or more of capacity, the payload is assessed the full launch cost. As shown by the local slope on Figure 3.2.2-1, shared payloads have more sensitivity to length and weight variations than indicated by using average slope data.

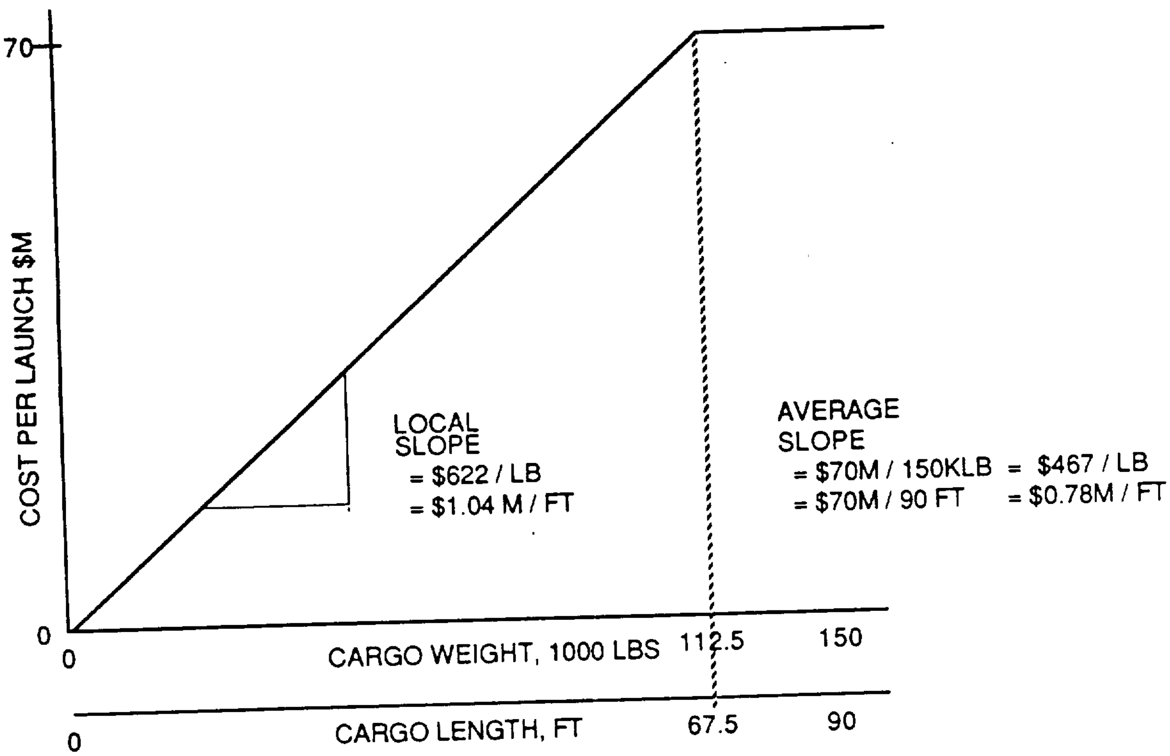


Figure 3.2.2-1 Shared Launch Cost Charging

3.2.3 Propellant Cost Charges for Space-Based OTV

"Propellant loaded on the ground to fully utilize the available lift capability of the launch vehicle will not be charged for transportation to LEO, but will incur any OMV charges for transfer to the propellant storage facility." This ground rule is similar to the Reduced Airfare Rate Program authorized by Federal Resolution allowing airlines to transport their employees on a no-charge, as-space-available basis.

Flight manifesting data published in the 1983 Green Book showed that of 25 STS missions (starting with STS-9), 15 had weight and length available that could have been used for hauling "hitchhiked" propellant. Assuming the STS had a 65K capacity with a 56 foot long payload bay (4 ft reserved for EVA access), and also assuming a 6 foot long propellant tank set weighing 2000 lbs (including ASE) which can contain up to 7640 lbs propellant at a constant mixture ratio of 6:1, the 15 flights could have transported 286,000 lbs propellant to LEO. This is equivalent to 17.6% of the total STS lift capability (286K/25/65K).

Considerably more propellant could be transported if the mixture ratio were varied from mission to mission. Heavily loaded missions with volume left over could haul all liquid hydrogen; lightly loaded mission with only small available volume could have all liquid oxygen. Our analyses conservatively neglected this effect.

Unpublished data from the STAS program shows that 119 LCV launches and 374 STS/STS II launches will be utilized to support the payloads in the civil Option II mission model. If we assume a 25% reduction in the LEO lift capacity to get to the 270 nmi Space Station altitude, the propellant available from hitchhiking is conservatively (because of the constant mixture ratio) estimated as

$$0.176 \times 0.75 \times (119 \times 150,000 + 374 \times 65,000) = 5.5 \text{ million lbs.}$$

The space-based/ground-based trade in paragraph 4.9 uses this as a baseline for low cost propellant. Sensitivities ranging from 0 to 9 million lbs are also shown in the cost data.

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3.3 DESIGN DRIVER MISSION ANALYSES

Mission analysis conducted on the mission model define time of flight and velocity requirements for use in performance calculations. All parameters are computed using Keplerian analysis (spherical gravity fields) and impulsive burns. Mission timeline ground rules are shown in Figure 3.3-1.

- 12 HOUR PHASING COAST IN LEO TO ACHIEVE ANY GEO LONGITUDE
- 3 HOUR INTERMEDIATE ORBIT FOR PRECISE PAYLOAD POSITIONING
- 1 DAY BETWEEN LAUNCHES FOR MULTIPLE LAUNCH EVENTS
- MISSION STAY TIMES:

-UNMANNED GEO DELIVERY	1 DAY AT GEO
-MANNED GEO SORTIE DEMO	6 DAY AT GEO
-MANNED GEO SORTIE TO SHACK	12 DAY AT GEO
-UNMANNED LUNAR DELIVERY	7 DAY IN LUNAR ORBIT
-MANNED LUNAR SORTIE	16 DAY IN LUNAR ORBIT
-DOD	1 DAY AT DESTINATION ORBIT

Figure 3.3-1 Mission Timeline Ground Rules

3.3.1 Geosynchronous Missions

3.3.1.1 Unmanned Missions

The bulk of missions performed were in this class. Two varieties, ground-based from a 140 nmi/28.5° and space-based from a 270 nmi/28.5° inclination part orbit, were considered. These two missions are shown in Figure 3.3.1-1 and 3.3.1-2.

	A	B	C	D	E	F	G	H	I	J	K
1	MISSION	TOTAL OTV	MPS PROP	SEGMENT	GEO	DELTA-V	PAYLOAD	CRYO	FUEL CELL	ACS PROP	MPS PROP
2	SEGMENT	WEIGHT	REMAINING	DURATION	DELTA-V	GRAVLOSS	WEIGHT	BOILOFF	USAGE	USAGE	USAGE
3	NUMBER	(LBS)	(LBS)	(HRS)	(FPS)	(FPS)	(LBS)	(LBS)	(LBS)	(LBS)	(LBS)
4											
5	0	57154	48794	0	0	0	14550	0	0	0	
6	1	25698	19336	12.00	8073.99	192	14550	59	6	26	3036
7	2	18456	8095	5.30	5855.68	-68	14550	23	4	12	1920
8	3	8868	1508	24.00	6051.61	0	0	106	16	53	441
9	4	8825	1465	4.20	20.00	0	0	18	3	9	1
10	5	8600	1240	3.00	350.00	0	0	13	2	7	20
11	6	8560	1200	5.50	0.00	0	0	24	4	12	
12											

Figure 3.3.1-1 Geosynchronous Mission Summary (Ground-Based)

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	A	B	C	D	E	F	G	H	I	J	K
1	MISSION	TOTAL OTV	MPS PROP	SEGMENT	GEOSB	DELTA-V	PAYLOAD	CRYO	FUEL CELL	ACS PROP	MPS PROP
2	SEGMENT	WEIGHT	REMAINING	DURATION	DELTA-V	GRAVLOSS	WEIGHT	BOILOFF	USAGE	USAGE	USAGE
3	NUMBER	(LBS)	(LBS)	(HRS)	(FPS)	(FPS)	(LBS)	(LBS)	(LBS)	(LBS)	(LBS)
4											
5	0	62633	57273	0	0	0	25023	0	0	0	0
6	1	26510	21150	2.08	7856.39	100	25023	8	1	9	36108
7	2	9714	4354	5.30	5798.29	100	25023	23	4	18	16757
8	3	6369	1009	24.00	6051.61	0	0	106	16	63	3170
9	4	6330	970	4.20	20.00	0	0	18	3	8	8
10	5	6122	762	3.00	450.00	0	0	13	2	7	187
11	6	6058	698	4.10	84.48	0	0	16	3	9	34
12	7	6035	675	3.10	0.00	0	0	14	2	7	0

Figure 3.3.1-2 Geosynchronous Mission Summary (Space-Based)

The missions are broken up into mission segments, each segment consists of a coast period followed by a burn. For the GEO missions the burns associated with these segments are as follows:

#1 - Perigee burn into GEO-transfer orbit
#2 - Apogee burn into GEO
#3 - Deorbit burn from GEO
#4 - Midcourse correction during GEO downleg
#5 - Post-aero maneuvers
#6 - Hohmann transfer to Space Station (Space-Based only)

An optimal inclination split is used to compute the first two burns. For the ground-based mission this split is 2.2° inclination change in the first burn and 26.3° in the second. For the space-based mission this split is 2.3° and 26.2°.

An additional factor added to the first two burns is a gravity loss factor to account for finite burn losses. A series of integrated trajectories was used to derive this gravity loss term. For the perigee burn it results in an increase to the impulsive Delta-V required. For the apogee burn it results in a decrease to the impulsive Delta-V because of the raising of perigee in the first finite burn. The loss factors are represented as polynomials which are a function of burn time:

$$T_{burn} = \frac{\text{Propellant Burned}}{\text{Thrust Level}} \times Isp$$

$$\text{Perigee Loss} = 0.050625\ T_b + 1.792969 \times 10^{-4}\ T_b^{2} - 2.490234 \times 10^{-8}\ T_b^{3}$$

$$\text{Apogee Loss} = 0.0473248\ T_b + 8.5038 \times 10^{-5}\ T_b^{2} - DV_{\text{Loss Per.}}$$

The GEO-deorbit burn was computed to put the OTV downleg perigee at 40 nmi (in the atmosphere) with an inclination of 28.5°. The midcourse and post-aero maneuvers are derived from aeroassist GN&C work. For the ground-based mission, the post-aero maneuvers are 350 fps which puts the vehicle into a 140 nmi circular orbit with allowances for aeroassist dispersions in apogee and inclination. For the space-based mission the corresponding velocity (sized for a nominal 245 nmi post-aero orbit) is 450 fps.

The segment duration times generally correspond with pure orbital mechanics requirements with the following exceptions. A coast period of 12 hours prior to the first burn is required in the ground-based mission to achieve any possible earth-relative longitude at GEO-inject. This coast period is not required for space-based missions because the station deploy time can be adjusted to achieve the same thing. GEO-deorbit opportunities occur every 12 hours when the pickup vehicle's orbital node intersection is reached. The ground-based mission requires that this duration at GEO be 24 hours to be consistent with Shuttle crew cycle constraints. The same duration is also used on the space-based missions, but more to keep commonality with the ground-based profile than for any hard constraint. Finally, 5.5 hours is allocated at the end of all missions to allow for rendezvous maneuvers.

3.3.1.2 Manned GEO Servicing Missions

The manned GEO servicing mission (#15010)) is rather loosely defined. In order to derive vehicle requirements a mission analysis effort was conducted to define mission duration and velocity requirements.

Figure 3.3.1-3 shows basic orbital data used to design the GEO servicing missions. The curves show Delta-V required to establish drift rates for moving from point to point in the GEO lane. This is displayed as drift angles and the time required to transit them (in days). The velocities required include the start Delta-V and the stop Delta-V.

- CURVES SHOW DRIFT TIMES V.S. TOTAL DELTA-V (ΔV1 + ΔV2)
- 4 DRIFT ANGLES SHOWN: 20°, 45°, 90°, 180°

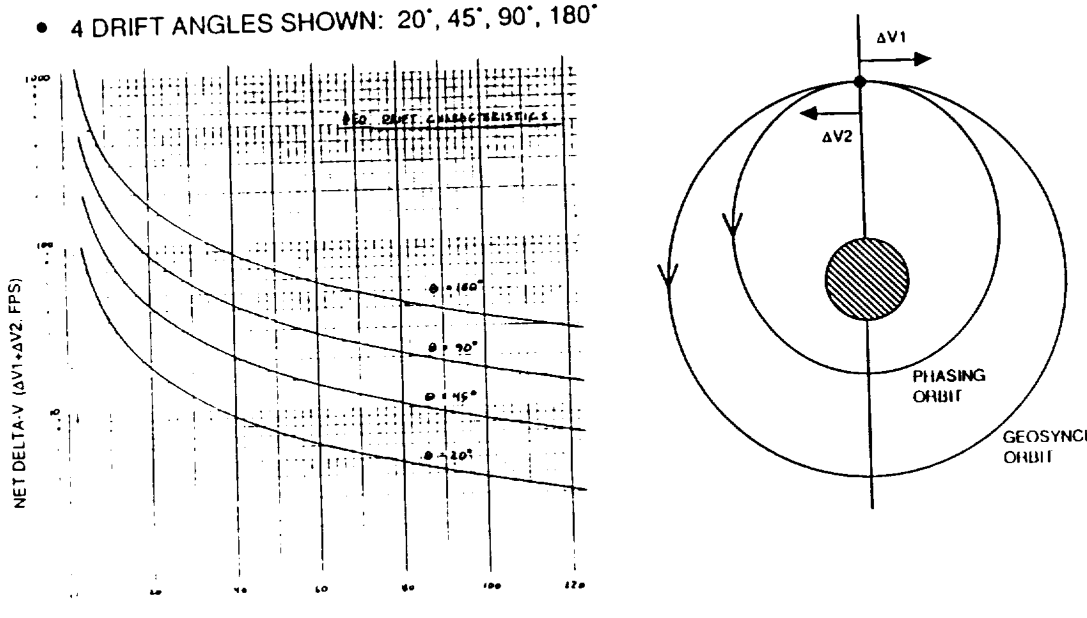
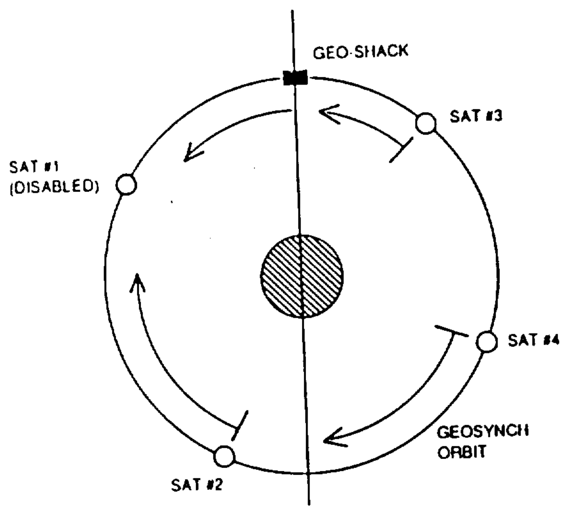


Figure 3.3.1-3 GEO Servicing Drift Data

Because of the time critical nature of manned missions, it is more efficient to perform multiple servicing with satellites that have been gathered into a tight service cluster. This minimizes the amount of time and propellant required for inter-satellite transit.

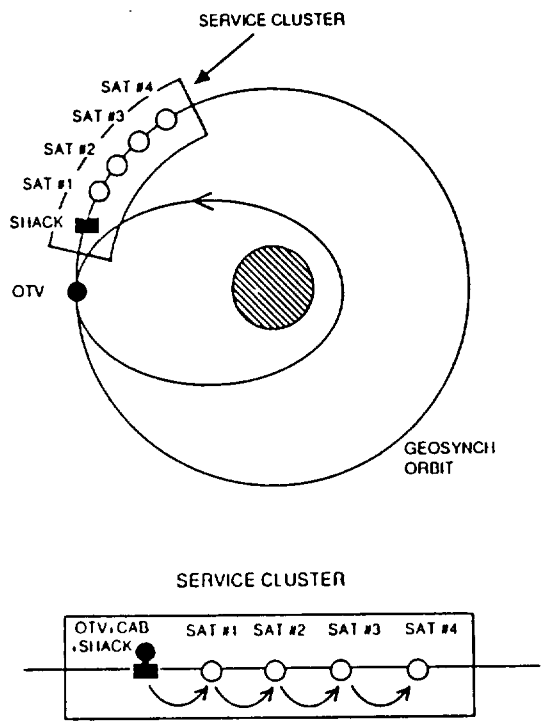
In order to establish capabilities a worst case servicing scenario (Figure 3.3.1-4) was used which assumes that 4 satellites are to be visited with one of them having no propulsive capability due to engine failure or propellant shortage. A 45 day roundup period is used which requires a net Delta-V of 50 fps per satellite (on average). The active satellites along with the GEO shack are gathered to the disabled satellite forming a service cluster. Once this is accomplished, the OTV and attached manned servicer are deployed from low earth orbit.



- MANNED MISSION IS TIME AND PERFORMANCE CRITICAL
- OPTIMIZE BY MANEUVERING SATELLITES AND GEO-SHACK TO SERVICE CLUSTER
- ASSUME WORST CASE SCENARIO:
  4 SATELLITES TO BE SERVICED
  1 SATELLITE DISABLED (NO PROP.)
- 45 DAY ROUNDUP PHASE REQUIRES 50 FPS PER SATELLITE (AVERAGE)
- START ΔV (25 FPS) SUPPLIED BY SATELLITES
- END ΔV (25 FPS) SUPPLIED BY SATS OR OMV
- GEO SHACK ΔV SUPPLIED BY OMV

Figure 3.3.1-4 Manned GEO Servicing - Roundup Phase

With the service cluster established at a satellite spacing of 1/2 deg., the OTV delivers the manned cab to the GEO shack which is stationkeeping with one of the satellites (Figure 3.3.1-5). The shack's OMV retrieves the OTV plus cab to the GEO shack, the shack is manned and checked out (1 day), and servicing operations commence on the nearest satellite. Three days have been allocated to perform this operation. Once a satellite has been serviced the OTV is used to move the GEO shack to the next one in a 1/2 day transfer which requires 88 fps total. This sequence of operations is repeated for each satellite, requiring a total of 21 days to service all four. This time also includes 3 days at the end of the servicing mission to initiate redeploy of the satellites and to prepared the shack for unmanned operation. Lesser numbers of serviced vehicles and their time requirements are also shown.

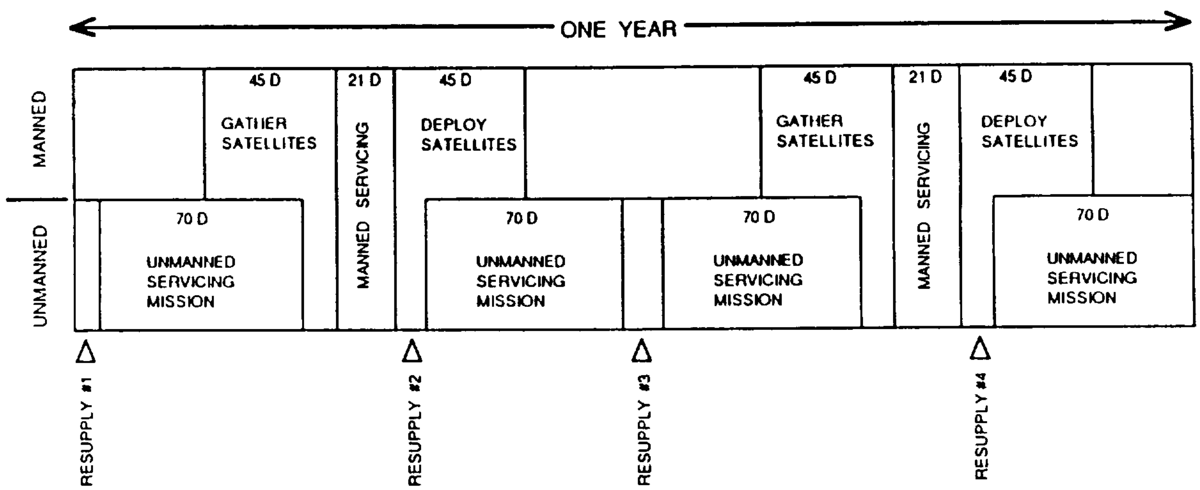


- ESTABLISH SERVICE CLUSTER OF SATELLITES (1/2° = 200 NM SPACING)
- GEO-SHACK STATIONKEEPS 10-20 NM FROM FIRST SATELLITE
- OTV + MANNED CAB INJECTS 10 NM FROM SHACK, OMV RETRIEVES
- OTV PROVIDES MAJOR ΔV'S TO MANEUVER FROM SAT. TO SAT. (88 FPS PER SAT.)
- SERVICING PERFORMED BY CAB+OMV+SHACK
- 45 DAY REDEPLOY OF SATS INITIATED BY SAT. ITSELF OR OMV+OTV
- SERVICING MISSION DURATIONS:

1 SAT.	8 DAYS
2 SATS	12 DAYS
3 SATS	17 DAYS
4 SATS	21 DAYS

Figure 3.3.1-5 Manned GEO Servicing - Service Phase

A years worth of servicing missions is shown in Figure 3.3.1-6 in an integrated timeline. Two 21-day manned servicing missions are illustrated along with their associated 45 day satellite gathering and re-deploy phases. In the time remaining the GEO shack's OMV can be used for unmanned servicing. Because this vehicle is unmanned it does not have the time constraints of the manned sortie and thus can use a longer mission duration to save on maneuvering propellant. It also does not require continuous OTV presence and so is not a mission driver for the OTV.



- 2 MANNED SERVICING MISSIONS (21 DAY DURATION)
- 4 GEO-SHACK RESUPPLY MISSIONS
- 4 UNMANNED SERVICING MISSIONS (70 DAY DURATION)

Figure 3.3.1-6 Servicing Timeline (Manned and Unmanned)

Based on a mission model average, two satellites are serviced in each manned sortie, and an on-station duration of 12 days is required of the OTV for each mission. Additionally, 176 fps must be supplied by the OTV for moving the 53.8 Klb GEO shack plus cab.

This mission profile is summarized in Figure 3.3.1-7.

	A	B	C	D	E	F	G	H	I	J	K
1	MISSION	TOTAL OTV	MPS PROP	SEGMENT	GEOMANSE	DELTA-V	PAYLOAD	CRYO	FUEL CELL	ACS PROP	MPS PROP
2	SEGMENT	WEIGHT	REMAINING	DURATION	DELTA-V	GRAVLOSS	WEIGHT	BOILOFF	USAGE	USAGE	USAGE
3	NUMBER	(LBS)	(LBS)	(HRS)	(FPS)	(FPS)	(LBS)	(LBS)	(LBS)	(LBS)	(LBS)
4											
5	0	75748	67420	0	0	0	12000	0	0	0	0
6	1	39032	30704	2.08	7856.39	262	12000	9	1	5	36700
7	2	22643	14515	5.30	5798.29	-92	12000	29	4	12	16151
8	3	21418	13090	144.00	178.00	0	10000	634	99	317	875
9	4	10278	1950	144.00	6051.61	0	10000	634	99	317	10091
10	5	10220	1892	4.20	20.00	0	10000	18	3	9	27
11	6	9601	1272	3.00	450.00	0	10000	13	2	7	598
12	7	9481	1133	4.10	84.48	0	10000	18	3	9	110
13	8	9458	1110	3.10	0.00	0	10000	14	2	7	0
14											

Figure 3.3.1-7 Manned GEO Servicing Mission Summary

3.3.2 DOD Missions

The revised mission model contains 4 generic DOD missions (unclassified): Geosynchronous delivery (identical to civil), mid-inclination delivery, generic polar, and generic low inclination. The ground-based missions are boosted with the large cargo vehicle directly into a park orbit with the proper mission plane (except for GEO delivery). Upon completing its mission the OTV returns to 28.5° inclination where it waits for Shuttle retrieval.

The geosynchronous delivery mission (#19035) is identical to the mission profile derived for the civil mission model, see Section 3.3.1.1.

The mid-inclination mission (#19036) delivers a 10000 lb spacecraft to a circular geosynchronous orbit inclined 63° to the equator. In general, this mission is almost identical to the GEO delivery mission except for the plane change required (34.5° vs 28.5° for standard GEO delivery). The optimum plane change splits for the first two burns of the space-based mission are 2.5° and 32.0°. The ground-based and space-based mission data are shown in Figure 3.3.2-1 and 3.3.2-2.

	A	B	C	D	E	F	G	H	I	J	K
1	MISSION	TOTAL OTV	MPS PROP	SEGMENT	DODHI63	DELTA-V	PAYLOAD	CRYO	FUEL CELL	ACS PROP	MPS PROP
2	SEGMENT	WEIGHT	REMAINING	DURATION	DELTA-V	GRAVLOSS	WEIGHT	BOILOFF	USAGE	USAGE	USAGE
3	NUMBER	(LBS)	(LBS)	(HRS)	(FPS)	(FPS)	(LBS)	(LBS)	(LBS)	(LBS)	(LBS)
4											
5	0	46458	38641	0	0	0	10000	0	0	0	0
6	1	22777	15160	12.00	7996.80	128	10000	59	6	26	23594
7	2	13793	6176	5.27	4826.30	-45	10000	23	4	12	8645
8	3	8820	1203	24.00	6508.20	0	0	106	16	53	4790
9	4	8778	1161	4.20	20.00	0	0	18	3	9	12
10	5	8497	880	3.00	450.00	0	0	13	2	7	258
11	6	8420	803	4.10	84.48	0	0	18	3	9	48
12	7	8397	780	3.10	0.00	0	0	14	2	7	0
13											

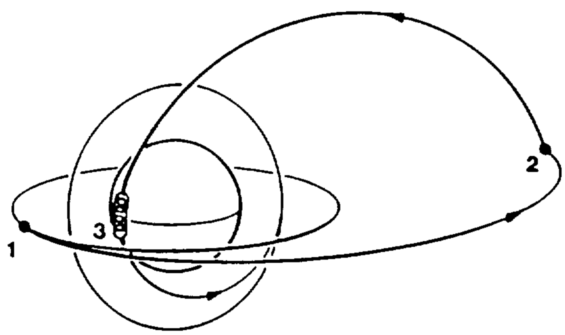
Figure 3.3.2-1 DOD Mid-inclination Mission Summary (Ground-Based)

	A	B	C	D	E	F	G	H	I	J	K
1	MISSION	TOTAL OTV	MPS PROP	SEGMENT	DODHI63S	DELTA-V	PAYLOAD	CRYO	FUEL CELL	ACS PROP	MPS PROP
2	SEGMENT	WEIGHT	REMAINING	DURATION	DELTA-V	GRAVLOSS	WEIGHT	BOILOFF	USAGE	USAGE	USAGE
3	NUMBER	(LBS)	(LBS)	(HRS)	(FPS)	(FPS)	(LBS)	(LBS)	(LBS)	(LBS)	(LBS)
4											
5	0	51463	43866	0	0	0	10000	0	0	0	0
6	1	25087	18370	2.08	7876.50	144	10000	8	1	5	25480
7	2	13793	6176	5.31	6234.20	-51	10000	23	4	12	12155
8	3	8820	1203	24.00	6508.20	0	0	106	16	53	4790
9	4	8778	1161	4.20	20.00	0	0	18	3	9	12
10	5	8497	880	3.00	450.00	0	0	13	2	7	258
11	6	8420	803	4.10	84.48	0	0	18	3	9	48
12	7	8397	780	3.10	0.00	0	0	14	2	7	0
13											

Figure 3.3.2-2 DOD Mid-inclination Mission Summary (Space-Based)

The generic polar mission (#19517) delivers a 5000 lb spacecraft into a 4000 nmi orbit inclined 90° to the equator. The primary driver for this mission is the 61.5° plane change required.

Figure 3.3.2-3 illustrates an efficient method of performing large plane changes through the use of aeroassist. In this technique, an apogee raising maneuver is performed which allows the plane change burn to be executed at apogee where orbital velocities are low. Once the plane change has been performed an aerobraking maneuver is executed at perigee to reduce apogee down to the final desired altitude. This technique is contrasted against the all propulsive method which substitutes a third rocket burn for the aeroassist, but still raises apogee to perform the plane change. The velocities required to perform the return transfer of the DOD polar mission are 10950 fps for the aeroassisted technique and 18050 fps for all propulsive which gives an idea of the savings via aerobraking.



- USE OF AEROASSIST IN PLANE CHANGES
  - (1) BOOST APOGEE VIA ROCKET BURN
  - (2) PERFORM INC.CHANGE AT APOGEE WHERE VELOCITY IS LOW
  - (3) UTILIZE AEROASSIST AT PERIGEE TO REDUCE APOGEE
- DELTA-V SAVINGS UP TO 50% OVER-ALL PROPULSIVE

Figure 3.3.2-3 Large Inclination Changes Via Aeroassist

This technique is optimum for recovering an empty OTV where greater than a 30° plane change in low orbit is required. If a large plane change is required with payload attached, however, a problem is encountered with the use of aeroassist. Although many payloads will be able to protect themselves from the entry environment, it seems unlikely that this will be true in general. Therefore, a ground rule was made that the OTV can perform aeroassisted plane changes only if a payload is not attached.

With this in mind, the ground-based mission data is shown in Figure 3.3.2-4. The first two burns boost the OTV and payload to the 4000 nmi mission orbit via a coplanar Hohmann transfer. In segment 3 the apogee is boosted to 19000 nmi for the 61.5° plane change burn which is performed in segment 4 and results in an orbital inclination of 28.5°. This plane change altitude was selected to lie within the normal capabilities of the aerobrake. A standard aeroentry then results in a Space Station compatible orbit (270 nmi, 28.5° inc) for Shuttle pickup.

	A	B	C	D	E	F	G	H	I	J	K
1	MISSION	TOTAL OTV	MPS PROP	SEGMENT	DODPOLAR	DELTA-V	PAYLOAD	CRYO	FUEL CELL	ACS PROP	MPS PROP
2	SEGMENT	WEIGHT	REMAINING	DURATION	DELTA-V	GRAVLOSS	WEIGHT	BOILOFF	USAGE	USAGE	USAGE
3	NUMBER	(LBS)	(LBS)	(HRS)	(FPS)	(FPS)	(LBS)	(LBS)	(LBS)	(LBS)	(LBS)
4											
5	0	32944	25327	0	0	0	5000	0	0	0	0
6	1	28706	16089	12.00	4116.40	29	5000	53	8	26	9150
7	2	17854	10237	1.43	3418.70	-8	5000	6	1	3	5842
8	3	13649	6028	2.24	4016.60	0	0	10	2	5	4195
9	4	8890	1208	6.47	6483.40	0	0	26	4	14	4776
10	5	8778	1161	4.20	20.00	0	0	15	3	9	12
11	6	8497	880	3.00	450.00	0	0	13	2	7	259
12	7	8420	803	4.10	84.48	0	0	18	3	9	48
13	8	8397	780	3.10	0.00	0	0	14	2	7	0
14											

Figure 3.3.2-4 DOD Polar Mission Summary (Ground-Based)

The space-based polar mission is summarized in Figure 3.3.2-5. This mission requires two large plane changes. The first is accomplished all propulsively (since the payload is attached) and the second via aeroassist. In segment #1 the apogee is boosted to 30000 nmi, segment #2 performs the plane change at apogee and then segment #3 burn circularizes at a 4000 nmi polar orbit. The return leg (segments 4 through 9) is identical to that used for the ground-based mission.

	A	B	C	D	E	F	G	H	I	J	K
1	MISSION	TOTAL OTV	MPSPROP	SEGMENT	GEOSB	DELTA-V	PAYLOAD	CRYO	FUELCELL	ACSPROP	MPSPROP
2	SEGMENT	WEIGHT	REMAINING	DURATION	DELTA-V	GRAVLOSS	WEIGHT	BOILOFF	USAGE	USAGE	USAGE
3	NUMBER	(LBS)	(LBS)	(HRS)	(FPS)	(FPS)	(LBS)	(LBS)	(LBS)	(LBS)	(LBS)
4											
5	0	82809	74071	0	0	0	5000	0	0	0	0
6	1	43717	34585	2.08	8533.60	290	5000	9	1	5	39073
7	2	31014	22282	8.82	4608.10	-101	5000	39	6	18	12639
8	3	20678	12144	10.18	4922.30	0	5000	45	7	22	10064
9	4	15854	7222	2.24	4016.60	0	0	10	2	5	4905
10	5	10320	1588	6.47	6483.40	0	0	28	4	14	5588
11	6	10275	1543	4.20	20.00	0	0	18	3	9	14
12	7	9950	1218	3.00	450.00	0	0	13	2	7	303
13	8	9865	1133	4.10	84.48	0	0	18	3	9	56
14	9	9842	1110	3.10	0.00	0	0	14	2	7	0

Figure 3.3.2-5 DOD Polar Mission Summary (Space-Based)

The DOD generic low-inclination mission (#19036) is summarized in Figures 3.3.2-6 and 3.3.2-7 for ground and space-based missions. This mission delivers an 110000 lb payload to a 1000 nmi orbit inclined at 63° from the equator. The technique utilized in the ground-based mission is identical to that used in the ground-based DOD polar. The mission orbit is reached via a coplanar Hohmann transfer from park orbit and return to the pickup orbit is achieved with an aeroassisted large plane change as described above. The space-based mission is identical to the space-based DOD polar with the exception of the transfer from station orbit to mission orbit. Rather than using a 3-burn transfer with a high apogee as was done in the DOD polar, a simpler two-burn Hohmann transfer (with most of the plane change occuring at apogee) is used because of the smaller amount of plane change required. This transfer is accomplished in segment #1 and #2. Beginning at segment #3 the return mission is identical to that in the ground-based low-inclination profile.

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	A	B	C	D	E	F	G	H	I	J	K
1	MISSION	TOTAL OTV	MPS PROP	SEGMENT	DODLo 63	DELTA-V	PAYLOAD	CRYO	FUEL CELL	ACS PROP	MPS PROP
2	SEGMENT	WEIGHT	REMAINING	DURATION	DELTA-V	GRAVLOSS	WEIGHT	BOILOFF	USAGE	USAGE	USAGE
3	NUMBER	(LBS)	(LBS)	(HRS)	(FPS)	(FPS)	(LBS)	(LBS)	(LBS)	(LBS)	(LBS)
4											
5	0	41500	33883	0	0	0	110000	0	0	0	0
6	1	28104	20487	12.00	1327.20	51	110000	53	8	26	13309
7	2	17117	9500	0.89	1257.60	-16	110000	4	1	2	10980
8	3	10893	3218	2.06	6840.80	0	0	9	1	5	6268
9	4	8820	1203	5.53	3024.30	0	0	24	4	12	1973
10	5	8778	1161	4.20	20.00	0	0	18	3	9	12
11	6	8497	880	3.00	450.00	0	0	13	2	7	259
12	7	8420	803	4.10	84.48	0	0	18	3	9	48
13	8	8397	780	3.10	0.00	0	0	14	2	7	0
14											

Figure 3.3.2-6 DOD Low-inclination Mission Summary (Ground-Based)

	A	B	C	D	E	F	G	H	I	J	K
1	MISSION	TOTAL OTV	MPS PROP	SEGMENT	GEOSB	DELTA-V	PAYLOAD	CRYO	FUEL CELL	ACS PROP	MPS PROP
2	SEGMENT	WEIGHT	REMAINING	DURATION	DELTA-V	GRAVLOSS	WEIGHT	BOILOFF	USAGE	USAGE	USAGE
3	NUMBER	(LBS)	(LBS)	(HRS)	(FPS)	(FPS)	(LBS)	(LBS)	(LBS)	(LBS)	(LBS)
4											
5	0	65876	57144	0	0	0	10000	0	0	0	0
6	1	58175	47443	2.08	2014.90	32	10000	9	1	5	9686
7	2	20014	11282	0.91	11854.00	-9	10000	4	1	2	36155
8	3	12683	3936	2.06	6840.80	0	0	9	1	5	7331
9	4	10320	1588	5.53	3024.30	0	0	24	4	12	2306
10	5	10275	1543	4.20	20.00	0	0	18	3	9	14
11	6	9950	1218	3.00	450.00	0	0	13	2	7	303
12	7	9885	1153	4.10	84.48	0	0	18	3	9	58
13	8	9842	1110	3.10	0.00	0	0	14	2	7	0
14											

Figure 3.3.2-7 DOD Low-inclination Mission Summary (Space-Based)

3.3.3 Lunar Missions

Two distinct classes of lunar missions exist, flights to low lunar orbit (60 nmi altitude) and flight to the L1 libration point. In order to perform mission analyses a three body integrated simulation was utilized which propagates motion of the earth, moon and spacecraft within their mutual gravity fields. Flight to low lunar orbit make up the bulk of the mission model (#17201, 17202, 17203, 17206, 17207). Because of the difficulty in simulation targeting, no distinction is made between the polar and equatorial orbiters. This mission is summarized in Figure 3.3.3-1. Major burns are trans-lunar injection (segment #1), lunar orbit insertion (segment #3), and trans-earth injection (segment #4). The mission completes with an aeropass, post-aero circularization (segment #8), and Hohmann transfer to the Space Station. Midcourse corrections are indicated at segments #2,5,6, and 7. Gravity loss is accounted for only in the translunar injection where its effect is largest. As a function of burn time (T_{burn}, see Geosynchronous mission summary), the following factor is added to the impulsive velocity:

$$DV_{loss} = 1.32\ 0.050625\ T_b + 1.792969 \times 10^{-4} T_b^{\ 2} - 2.490234 \times 10^{-8}\ T_b^{\ 3}\ )$$

	A	B	C	D	E	F	G	H	I	J	K
1	MISSION	TOTAL OTV	MPS PROP	SEGMENT	LUNAR	DELTA-V	PAYLOAD	OTV	FUEL CELL	ACS PROP	MPS PROP
2	SEGMENT	WEIGHT	REMAINING	DURATION	DELTA-V	GRAVLOSS	WEIGHT	BOILOFF	USAGE	USAGE	USAGE
3	NUMBER	(LBS)	(LBS)	(HRS)	(FPS)	(FPS)	(LBS)	(LBS)	(LBS)	(LBS)	(LBS)
4											
5	0	39402	31785	0	0	0	5072	0	0	0	0
6	1	17310	9693	2.08	10132.00	151	5072	9	1	5	22077
7	2	16911	9294	34.50	100.00	0	5072	152	24	76	147
8	3	12694	5077	34.50	3019.00	0	5072	152	24	76	3966
9	4	9393	1776	165.30	3019.00	0	0	727	119	364	2097
10	5	9081	1464	34.50	100.00	0	0	152	24	76	61
11	6	8812	1195	34.50	30.00	0	0	152	24	76	18
12	7	8776	1161	3.00	20.00	0	0	13	2	7	12
13	8	8497	880	3.00	450.00	0	0	13	2	7	250
14	9	8420	803	4.10	84.48	0	0	18	3	9	48
15	10	8397	780	3.1	0	0	0	14	2	7	0

Figure 3.3.3-1 Lunar Orbiter Mission Summary

Timing is important to the lunar missions as the moon moves rapidly out of the Space Station orbit plane. To avoid broken-plane type trajectory analysis (beyond the scope of this study) it was ground ruled that the moon must lie in the projected plane of the Space Station for coplanar transfer. This establishes time of flight restrictions to keep earth and lunar departures within the station's plane. This is expressed as a flight duration as follows:

$$T_{flight} = \frac{n\,180^{o} - 2\,T_{transit}\,R_{station}}{R_{station} + R_{moon}}$$

Where $T_{transit}$ is the transit time to and from the moon (days), $R_{station}$ is the Space Station nodal regression rate (positive, deg/day), and R_{moon} is the lunar inertial orbital rate (deg/day).

This translates to mission durations of 12.8, 21.8, 30.7, etc. days assuming a 2.9 day trans-lunar transfer and a space station at 270 nmi. Based on this, a nominal flight duration of 12.8 days was used for unmanned lunar missions and 21.8 days for manned flights.

The structure of the L1 libration point mission (#17200, Figure 3.3.3-2) is identical to that for the low lunar orbit mission. Because the libration point is far from the moon (and on the opposite side to the earth) a fairly long transfer time (5.8 days) is required along with lower inject velocities at the libration point.

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	A	B	C	D	E	F	G	H	I	J	K
1	MISSION	TOTAL OTV	MPS PROP	SEGMENT	LUNAR17	DELTA-V	PAYLOAD	CRYO	FUEL CELL	ACS PROP	MPS PROP
2	SEGMENT	WEIGHT	REMAINING	DURATION	DELTA-V	GRAVLOSS	WEIGHT	BOILOFF	USAGE	USAGE	USAGE
3	NUMBER	(LBS)	(LBS)	(HRS)	(FPS)	(FPS)	(LBS)	(LBS)	(LBS)	(LBS)	(LBS)
4											
5	0	44140	36523	0	0	0	5072	0	0	0	0
6	1	18722	12105	2.08	10089.00	178	5072	0	1	5	24403
7	2	19050	11483	70.00	100.00	0	5072	306	48	154	162
8	3	13494	5877	70.10	3602.00	0	5072	306	48	154	5046
9	4	9942	2325	116.60	3602.00	0	0	513	80	257	2703
10	5	9370	1753	70.00	100.00	0	0	306	48	154	63
11	6	8812	1195	74.10	31.00	0	0	326	51	163	18
12	7	8778	1161	3.00	20.00	0	0	13	2	7	12
13	8	8497	880	3.00	450.00	0	0	13	4	7	259
14	9	8420	803	4.10	84.48	0	0	18	3	9	48
15	10	8397	780	3.1	0	0	0	18	2	7	0

Figure 3.3.3-2 Lunar Libration Mission Summary

3.3.4 Planetary Missions

Boosting of planetary missions by a recoverable upper stage is difficult because of the energies involved. A strategy for performing this type of mission is shown in Figure 3.3.4-1. After injecting the payload into its desired trajectory (sometimes through the use of an expendable kick stage), the OTV separates to a safe distance and then deorbits into a large looping earth orbit (typically about 4 days in duration). Near the apogee of this orbit a two-burn dog leg maneuver is performed which corrects for nodal regression of the pickup vehicle. An aeroassist is then performed which reduces the orbit size to that compatible with Shuttle/Space Station retrieval. No attempt was made to compute out-of-plane impacts resulting from launching from the Space Station as this level of analysis is beyond the scope of the OTV study. The effect of this Space Station nodal drift has very significant impacts on mission velocity and departure windows, requiring further analysis at a future date.

Gravity loss is computed from the following:

$$DV_{loss} = -25.232769 + 0.2549762\ T_b + 1.72078047 \times 10^{-4}\ T_b^{\ 2} - 2.1662239 \times 10^{-8}\ T_b^{\ 3} + 7.7525435 \times 10^{-13}\ T_b^{\ 4}$$

The basic planetary mission strategy was coded into an optimization program utilizing gradient search techniques to minimize the OTV/spacecraft stack mass through the use of offloading and expendable kick stages, if necessary. The results of this program are shown in Figure 5.6.4-1 for the 24 planetary missions.

For a more extensive description of this program and planetary mission analysis see MMC OTV TM I.1.2.0.0-1.

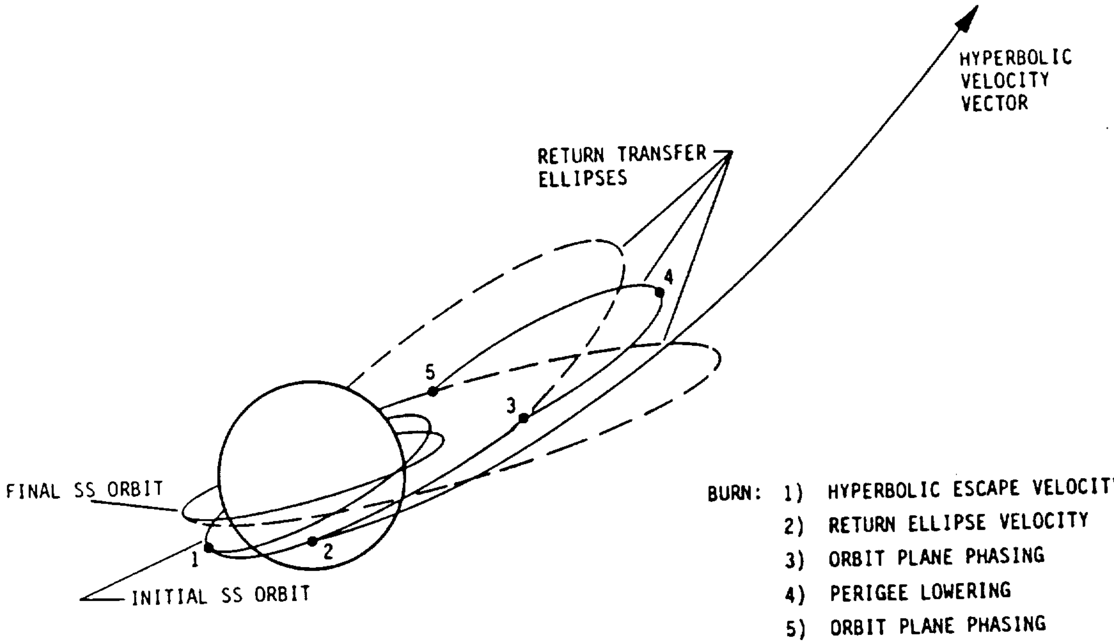


Figure 3.3.4-1 Planetary Mission Overview

4.0 SYSTEM SELECTION TRADE APPROACH

MMC concentrated on the refinement and optimization of STS launched OTV's during the first three months of the study extension. Updated Rev. 9 mission model requirements, STS and ACC launch vehicle characteristics, Space Station requirements and design and program requirements were categorized and their impacts on OTV system and subsystem trade studies were evaluated.

The OTV trade studies evaluated the following:

- Reusable versus expendable
- All propulsive versus aerobrake
- Aeroassist configuration
- ACC versus cargo bay
- Diameter of large cargo vehicle GB OTV
- Main propulsion system
- GB OTV vehicle/fleet sizing
- Alternative OTV options
- Ground-based versus space-based trade

These trades resulted in the definition of three different cryogenic, reusable, aerobraked OTV designs as indicated in Figure 4.0-1.

The best cargo bay vehicle was a single engine vehicle with a 40 foot diameter flexible aerobrake and utilized a toroidal oxygen tank. This vehicle weighed 5360 lbs and contained 45,000 lbs propellant.

The best ACC launched OTV was also single engine with a 38 ft diameter aerobrake. It weighed 5920 lbs and contained 45,500 lbs propellant. Both of the ground-based vehicles were capable of delivering 15,000 lbs to GEO.

The space-based OTV utilized 2 engines and a 44 ft diameter aerobrake. It weighed 8378 lbs, contained 74K propellant and was capable of delivering 28,000 lbs to GEO.

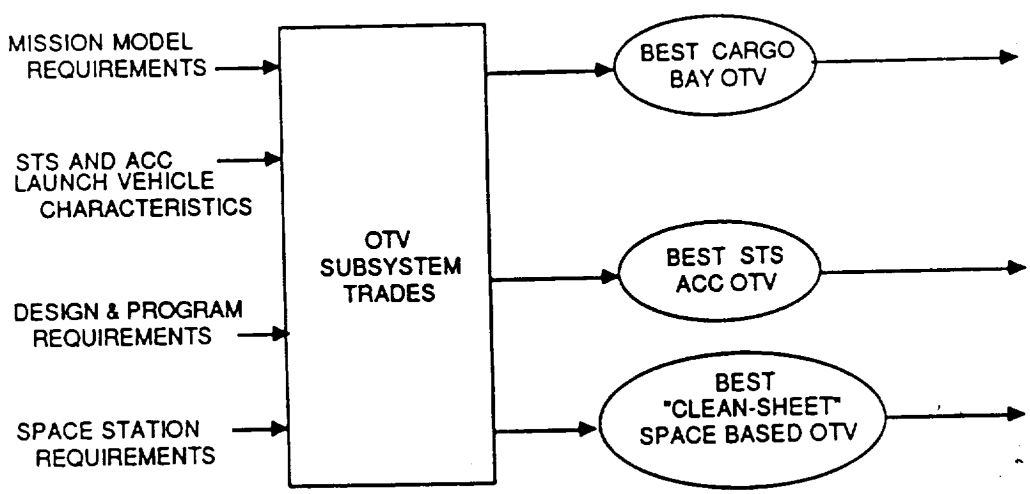


Figure 4.0-1 Extension Study Plan, STS Constrained OTV's

During the second half of the study extension, we initially concentrated on re-doing all the system and subsystem trades to determine the best ground-based OTV to be launched in the low cost ($70M/Flt), large capacity (150 Klbs to LEO) cargo vehicle. The study flow is indicated in Figure 4.0-2. After determining the best ground-based LCV launched configuration, we then determined the extent of modifications that would be required to allow this OTV to be man rated and space-based. This configuration is referred to as the hybrid.

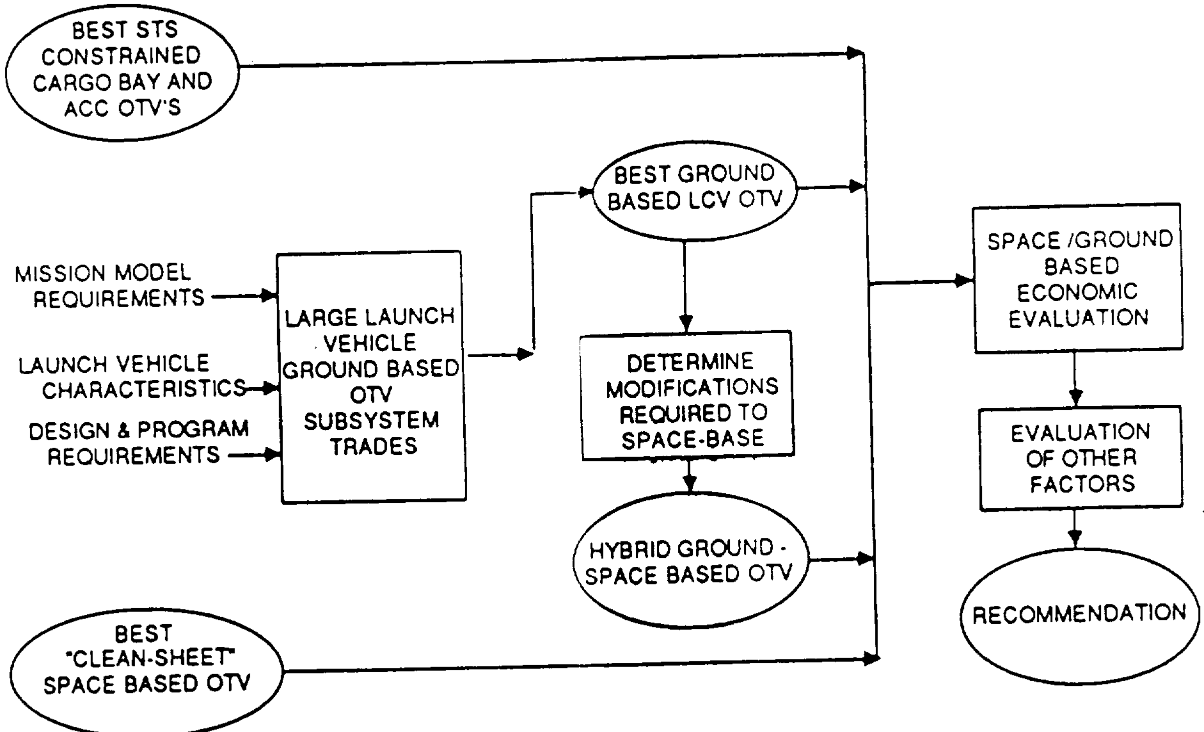


Figure 4.0-2 - Conclusion of Extension Study Plan

The following list of candidate vehicles were then evaluated in terms of Life Cycle Cost:

Designation	Capacity to GEO
STS Cargo bay GBOTV	15 K
STS ACC GBOTV	15 K
LCV GBOTV	15 K
LCV GBOTV (man-rated)	25 K
LCV SBOTV	15 K
LCV SBOTV (man-rated)	25 K
LCV Hybrid OTV (man-rated)	25 K

The economic evaluation was based only on the 160 civil payloads going to GEO. The DOD missions were intentionally omitted from the ground-based - space-based trade because of programmatic uncertainties regarding the military usage of Space Station. As shown in paragraph 4.9, space basing recovers the investment costs within the 160 civilian GEO missions. Any DOD missions that might be space-based would decrease the time for payback to occur.

4.1. REUSABLE VS EXPENDABLE TRADE STUDY

The objective of the reusable/expendable upper stage trade study was to assess the relative technical/economic merits of the alternative expendable concepts for STAS era launch vehicles against those of a reusable OTV program.

4.1.1 Criteria

The evaluation criteria for this trade focused on the economic performance of the alternative candidates, primarily development and launch costs, unit costs and onorbit operations.

4.1.2 Concepts

The trade was conducted within the 160 civil GEO missions (53 delivery/ return; 107 delivery only). The trade actually incorporated two different expendable OTV concepts. The first concept consisted of employing existing expendable upper stage concepts to perform the GEO civil mission model (Table 4.1.2-1). The only deviation from this was to develop an upgraded "stretched" Centaur G' concept to perform the more demanding return missions and to accommodate the 16 manned missions. The second concept involved the development of a "new technology" expendable upper stage. The approach here was to provide the new stage with the performance/dry weight advantages of new technology engines and structures while focusing on "must cost" unit estimates to provide a breakeven point with the reusable concept. This part of the trade includes 107 delivery only missions from the civil GEO mission model. The top level vehicle attributes are shown in Figure 4.1.2-1.

Table 4.1.2-1 Existing Upper Stage Vehicle Characteristics

Stage Name	IOC Year	Capacity GEO klb	Thrust klb	Engine Type	Gross Wt.klb	Propel. Wt.klb	Dry Wt. klb	ASE Wt. klb	L Ft	D Ft
PAM D	1982	1.4	14.9	Solid	4.82	4.4	0.4	2.5	7.8	4.4
PAM A	1982	2.2	35.2	Solid	8.26	7.6	0.7	4.6	7.5	4.4
IUS	1982	5.1	45/18	Solid	32.5	27.4	5.1	7.4	16.5	10.0
CENTAUR G	1986	10.0	2 x 15	LH2/LO2	37.2	29.9	7.3	9.2	19.5	14.2
CENTAUR G'	1986	19.5	2x 16.5	"	42.3	34.7	7.6	9.5	29.1	14.2
CENTAUR G"	1996	25.0	TBD	"	81.9	64.0	8.4	9.5	35.0	14.2
CENTAUR G" (2 Stages)	1999	12/10	TBD	"	165.5	140.0	8.0/8.0	9.5	70.0	14.2

CHARACTERISTICS

CRYOGENIC PROPELLANTS	
DRY WEIGHT	5500 Lbs
LENGTH	17 Ft
AVG. PROP. LOAD	27.3Klbs
MAX. PROP. LOAD	49.2Klbs

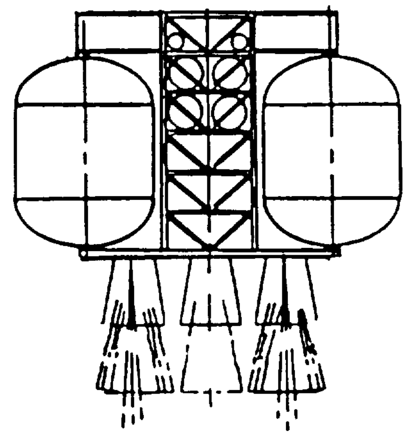


Figure 4.1.2-1 New Technology GBOTV Characteristics

The two expendable programs were traded against a ground-based reusable OTV program consisting of 52 klb and 74 klb stages. The two stages were utilized according to the requirements of the GEO civil missions. (See Section 4.9.2 for a comprehensive description of the reusable ground-based OTV program).

4.1.3 Ground Rules and Assumptions

The ground rules and assumptions used for this trade are consistent with the overall study ground rules enumerated in Section 8.0. Clarifications/ exceptions to the ground rules include the following:

A) Existing expendable OTV program - Stage hardware/ground processing costs were developed from government supplied cost data (STAS ground rules);

B) New technology expendable upper stage - Parametric expendable DDT&E cost estimates were made to determine concept breakeven points with reusable OTV program reference.

4.1.4 Assessments

4.1.4.1 Existing Expendable Upper Stage

The existing expendable upper stage manifesting of the 160 GEO civil missions was performed as shown in Table 4.1.4-1. The 84 multiple payload missions were divided into the smaller individual payloads they were originally developed from. The payloads were then manifested on a combination of PAM A, PAM D and IUS upper stages. This translated the 84 missions to 216 individual payloads. The other 76 missions were manifested to either Centaur G, G' or a G' derivative that involved both a stretch and manrating upgrade. Return missions were accomplished all propulsively by a two stage stretched Centaur configuration.

Table 4.1.4-1 Existing Upper Stage Vehicle Manifesting

Stage Name	IOC Year	Missions	P/l Wt. Class
PAM D	1982	9	2,500
PAM A	1982	92	2,000
IUS	1982	116	5,000
CENTAUR G	1986	1	10,000
CENTAUR G'	1986	11	14,000
CENTAUR G"	1996	11	25,000
CENTAUR G" (2 Stages)	1999	53	12/10

A ROM DDT&E estimate of $0.3B was made for the stretch/manrating of the Centaur G'. This also includes integration of the Centaur class of vehicles to the UPRCV. Operations costs included hardware production, ground processing and launch costs for the expendable stages.

Table 4.1.4-2 includes the composite CPF and total operations cost estimate for each class of existing upper stage. The data highlight the high launch cost of the Centaur class of vehicles, especially the two-stage concept required to service the 12 klb up, 10 klb down, GEO servicing and manned missions. All-propulsive return propellant requirements of 140 klb for these missions force the use of a second UPRCV to launch the missions. The other Centaur missions are more competitive in terms of launch costs, with the reusable GOBTV reference ($52.3/mission [Section 4.9.4]), but incur a large penalty for expendable hardware. The IUS and PAM missions display poor manifesting attributes within the 25 foot diameter UPRCV payload envelope, resulting in a relatively high payload delivery cost per pound measurements to GEO of approximately $16K/lb.

Table 4.1.4-2 Existing Upper Stage Vehicle Operations Costs (1985 $M)

Stage Name	Launch Cost ($M)	H/W Ground Processing CPF	Total CPF ($M)	Operations ($M)
PAM D	25.0	16.0	41.0	369
PAM A	21.3	10.1	31.4	2,889
IUS	45.3	33.6	78.9	9,152
CENTAUR G	39.4	50.3	89.7	90
CENTAUR G'	52.4	35.0	87.4	961
CENTAUR G"	59.5	47.4	107.2	1,179
CENTAUR G" (2 Stages)	110.5	86.9	197.4	10,462
				25,102

Figure 4.1.4-1 highlights the cumulative LCC of the reusable GBOTV (Section 4.9) vs existing upper stages. The cumulative cost curve displays the high operating costs of expendable systems vs the reusable GBOTV. The nonrecurring investment of the reusable system achieves a payback in 1998. The two programs diverge from that point on. The total LCC estimate for existing stages exceeds that of the reusable program by over 100% within the GEO civil mission model.

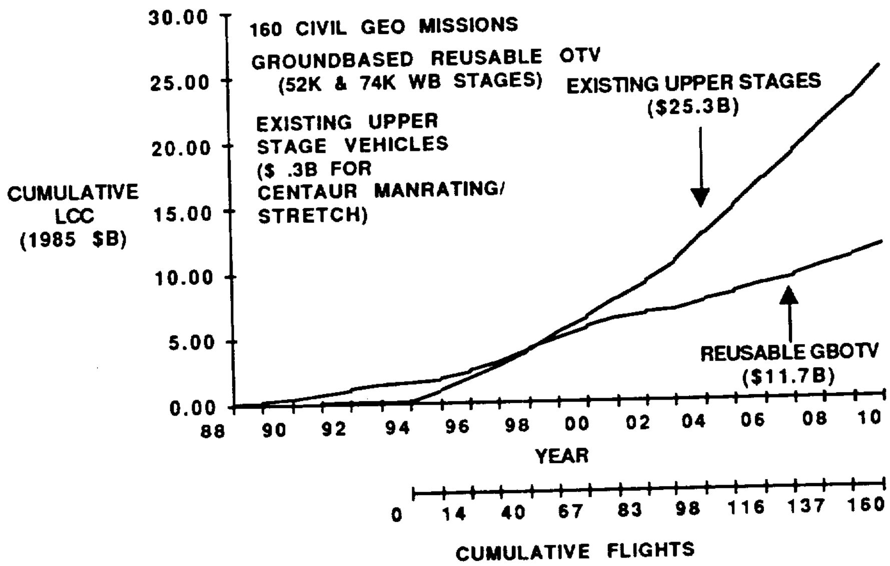


Figure 4.1.4-1 Existing Upper Stage Cumulative LCC (1985 $B)

Figures 4.1.4-2 and -3 highlight the annual cumulative delta LCC in constant and discounted dollars between the existing expendable program and the ground-based reference. The charts are generated by plotting the cumulative cost difference between the two program funding profiles on an annual basis. Both cases clearly show that this expendable is very uncompetitive with the reusable program. Two major cost areas contribute to this. First, the expendables as defined do not manifest well within the 25 ft UPRCV bay diameter. The second major factor is obviously the cost impact of expendable hardware as compared to reusable hardware turnaround costs.

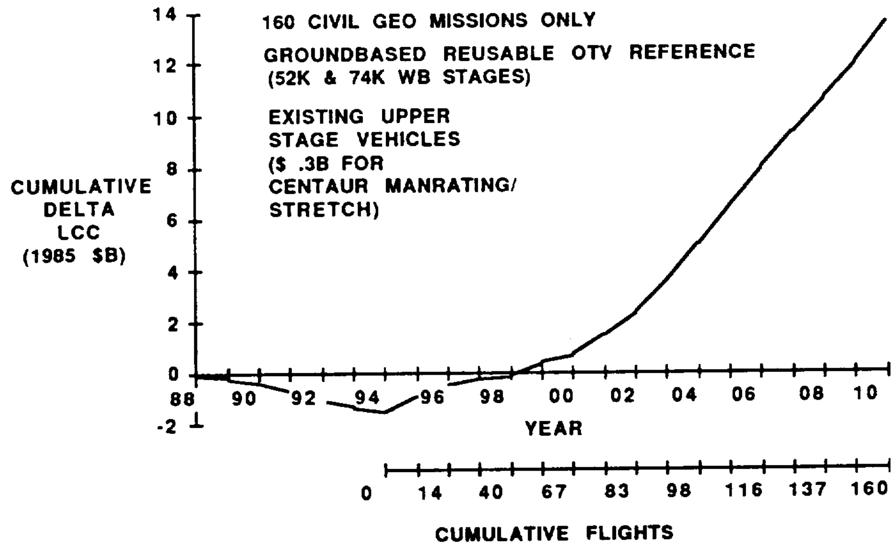


Figure 4.1.4-2 Existing Upper Stage Cumulative Delta LCC (1985 $B)

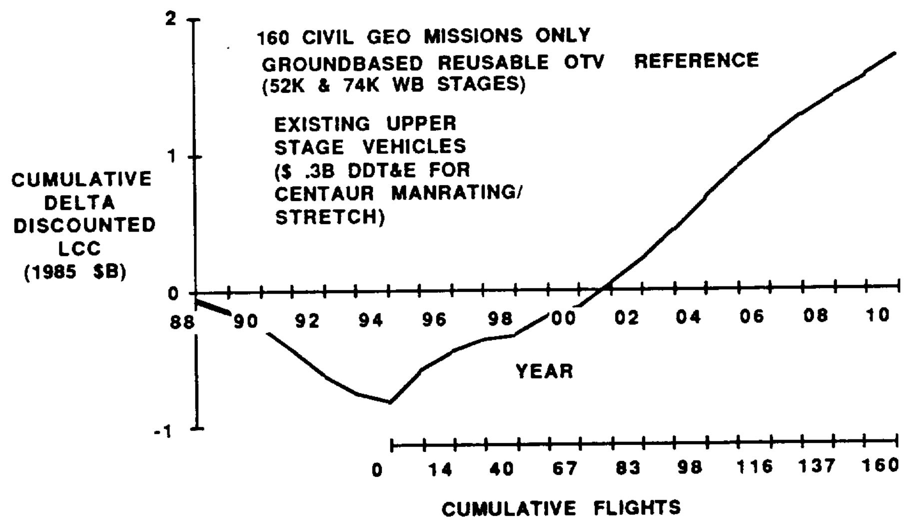


Figure 4.1.4-3 Existing Upper Stage Cumulative Delta Discounted LCC

4.1.4.2 New Technology Expendable

A second ground-based expendable upper stage concept was developed in order to provide a more competitive performance/economic expendable stage candidate to trade against the reusable OTV program. In order to simplify the trade and view the new technology expendable OTV under optimum conditions, only the 107 delivery missions of the civil model were considered. This allowed the expendable OTV to be unaffected by the severe penalty of an all propulsive return of the 53 geoshack logistics and manned missions.

The expendable OTV acquisition costs were estimated at three discrete points in order to allow a view under a wide range of investment cost conditions. The lower estimate of $0.3B for DDT&E would, at best, include development of a high performance engine concept. It is fairly unrealistic in that no allowance for other subsystem development has been included. The mid-range estimate of $0.6B begins to approach a program cost that would perhaps include expenditures for new structures, propulsion and tankage subsystems but with little left over for high technology avionics and power subsystems. The high estimate reflects a fairly complete high technology expendable stage DDT&E estimate.

The operations costs of the new tech expendable OTV were arrived at in two ways. Launch costs and multiple payload carrier cost (when applicable) were discretely estimated for each of the 107 missions under consideration. Given the length and dry weight (thus propellant) advantages of the expendable over that of the reusable OTV, a launch cost savings of $7.8M per flight over the 107 missions were realized ($42.7 vs $50.5M). An additional $1.0M per flight penalty was assessed to the expendable for 84 missions due to expending the multiple payload carrier.

In order for the expendable to break even within 107 flights, the remainder of the operations costs for the expendable vehicle were calculated on a discounted "must" cost basis (Figure 4.1.4-4). A constant year dollar unit/ground processing cost was then determined for the three different investment amounts.

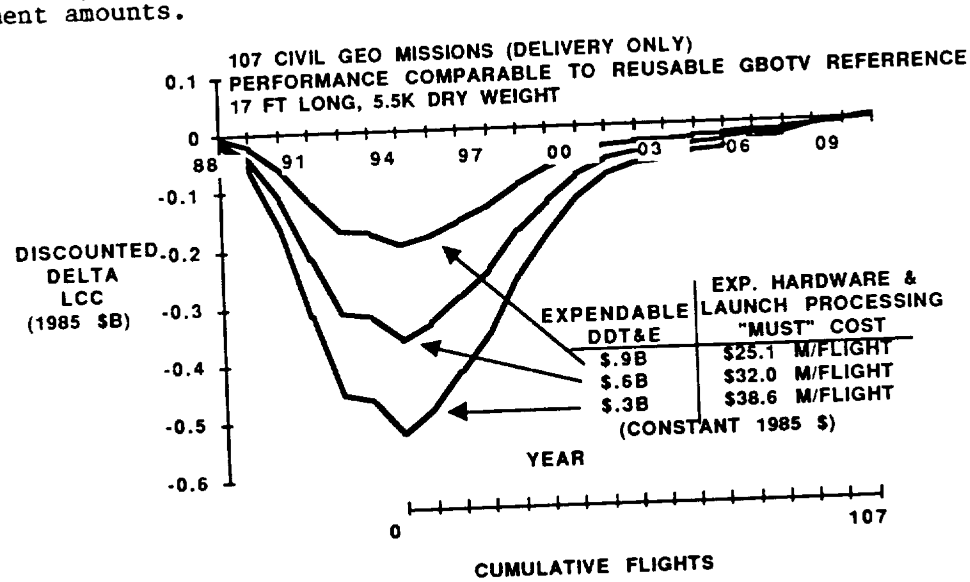


Figure 4.1.4-4 New Technology Expendable Cumulative Delta Discounted LCC

The higher the investment cost required the lower the unit/ground processing costs "must" be. The $25.1M, $32M and $38.6M CPFs reflect learning curves of approximately 88%, 91% and 94%, respectively.

4.1.5 Sensitivities

No overt sensitivities were performed within the expendable OTV trades other than the "must" CPF sensitivity of the new tech expendable to investment cost. This subject as been previously addressed in Section 4.1.4.2.

4.1.6 Recommendations

It is apparent that employment of existing upper stages cannot compete with a new technology reusable upper stage capability. Existing upper stage cost history reflects minimal learning impacts and poor launch vehicle manifesting attributes. Existing upper stages cannot take full advantage of the UPRCV payload volume. The combined impacts more than doubles the cost over that of the ground-based reusable OTV program.

A new tech expendable OTV that combines the performance/manifesting advantages of the reusable OTV while maintaining low investment costs and optimistic production/ground processing learning attributes fares considerably better. The most likely investment cost to support the newly defined technology expendable stage characteristics would approach $0.9B. This estimate would require an overall unit cost/ground processing improvement curve of at least 88%. An 88% unit cost improvement curve is fairly optimistic for a vehicle of this type since over 80% of the unit cost is due to the engine, avionics, and propulsion systems. The combination of these two factors would allow the expendable to break-even with the reusable GBOTV within the 107 civil GEO missions in discounted dollars. If the return missions are included the expendable vehicle growth would impact both launch and unit costs and would reduce the launch cost delta and force considerably better production improvement profiles. For these reasons, the reusable ground-based vehicle is preferred over either of the two expendable vehicle candidate programs.

4.2 ALL PROPULSIVE VS AEROBRAKE TRADE

The purpose of this trade was to determine the best mode of return for the OTV. Aeroassist offers potential benefits in propellant and launch cost savings, but at a cost of new technology and hardware development. The two basic approaches were compared and costed, based on their ability to fly the geosynchronous portion of the Rev. 9 mission model.

4.2.1 Criteria

The primary factor is propellant savings of the aerobraked over the all propulsive vehicle. This translates directly into lower launch costs because of the reduced liftoff weight and volume of the OTV. This is contrasted against the development cost of aerobraking technology as well as the production and refurbishment costs of brake hardware.

Although these are the primary factors, other cost impacts include the price of purchasing additional propellant for the all propulsive program, the increased program support overhead required for an aerobraked vehicle (more subsystems to track and support), as well as mission loss differences.

The analysis was conducted for both a ground-based and space-based OTV program. The 160 geosynchronous missions of Scenario #2, Rev. 9 mission model were used as the traffic basis.

4.2.2 Concepts

The design concepts considered for the aeroassist trade study included both ground and space-based OTV configurations, all launched by the large cargo vehicle (LCV).

The aeroassisted vehicles utilized were the basic family described in Section 2.3. For the ground-based option this included a 52K propellant capacity vehicle with a dry weight of 7680 lb and a 74K vehicle with a dry weight of 8795 lb. The 52K vehicle was used for missions requiring less than 16500 lb equivalent GEO delivery. The space-based OTV is a 74K propellant capacity vehicle with a dry weight of 9007 lb which is used for all the missions after space-based IOC in 1996. The performance of these vehicles is summarized in Section 6.2.3.

To perform ground-based missions with the all-propulsive option also required two vehicles. The small vehicle was a 74K propellant capacity stage with a dry weight of 6947 lb which was derived from the 74K aerobraked stage. This vehicle was capable of delivering a 17600 lb payload in GEO. The large all-propulsive stage was a 122K capacity vehicle weighing 8760 lb dry which was sized to perform the 12K up/10K down mission (#15011). This vehicle was also the workhorse for all space-based missions. Upgrading to space-basing requires about 200 lb of additional dry weight which was neglected for this all-propulsive vehicle in the interests of time. Thus the space-based all propulsive propellant requirements are slightly optimistic, which does not affect the final answers. The all-propulsive vehicle performance is summarized in Figure 4.2.2-1.

THE 74K PROP. CAPACITY OTV CAN PERFORM THE FOLLOWING ALL-PROPULSIVE GEO MISSIONS. THE DRY WEIGHT OF THE STAGE IS 6947 LB. (NO AEROBRAKE) :

PAYLOAD NO	MISSION NAME	PAYLOAD (UP / DOWN)	PROP. USAGE (LB)
-	MAX CAPACITY DELIVERY	17,594 / 0	74,000
18912	MULT. P/L DELIVERY	12,000 / 2,000	72,962
18076	SOLAR TERR. GEO EXP	7,055 / 0	57,302
18075	SETI GEO ANTENNA-B	14,551 / 0	69,340
15008	UNMANNED GEO SHACK	16,720 / 0	72,873

A LARGER OTV IS REQUIRED TO PERFORM THE FOLLOWING ALL-PROPULSIVE GEO MISSION THE PROP. CAPACITY IS 122K & THE DRY WEIGHT IS 8760 LB. (NO AEROBRAKE) :

PAYLOAD NO	MISSION NAME	PAYLOAD (UP / DOWN)	PROP. USAGE (LB)
15011	GEO SHACK LOGISTICS	12,000 / 10,000	121,734
10100	REFLIGHTS	20,000 / 0	93,632
15009	MANNED GEO SHACK	25,080 / 0	102,355

Figure 4.2.2-1 All-Propulsive Vehicle GEO Performance

4.2.3 Assumptions

The cost comparisons were based on the 160 geosynchronous missions contained in Scenario #2 of the Rev. 9 mission model. The reasoning behind using this subset is explained in Section 4.9. Because the space-based IOC occurs in 1996 it was assumed that the 5 missions in 1995 must be flown ground-based. Thus the space-based option consists of 5 ground-based and 155 space-based flights. All flights were launched by the large cargo vehicle which has a lift capability to low park orbit of 150,000 lb.

For space-based missions a dedicated tanker was assumed to be able to deliver propellant to orbit at a cost of 550 dollars/lb. Hitchhiked propellant was costed at 200 dollars/lb.

Although the all-propulsive vehicle requires more burn time of its engines, it requires the same number of starts as the aeroassisted OTV. Because it is felt that OTV engine wear-out is primarily a function of the number of restarts the wear-out and failure rates are assumed equal between the two vehicles.

Because of technical and time constraints no assessment of space-based accommodation differences was attempted. Here the primary areas are brake refurb/replace accommodations hardware (a net cost for the aerobraking option) and tank farm capacity increase (a net cost for all-propulsive). There will probably be a small net benefit to the all-propulsive option if these two areas are considered but it will not be large enough to alter the net results of the trade.

Other costing rules and assumptions are contained in Section 8.0, "Cost Estimates".

4.2.4 Assessments

The propellant sensitivity to payload delivered is shown for the small and large vehicle options (Figure 4.2.4-1 and 4.2.4-2). The propellant differences between space-based and ground-based missions are not significant. When this data is applied to the 160 GEO missions it is found that the aerobraked option requires 9.0 million pounds of propellant, with 14.4 million pounds being required by the all-propulsive option.

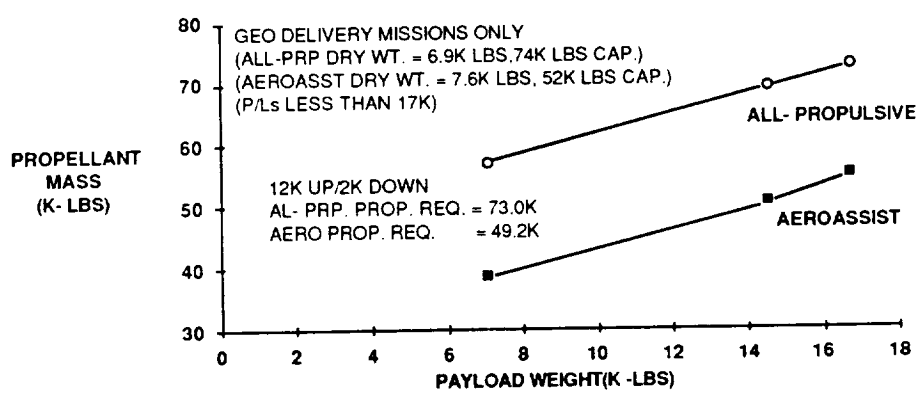


Figure 4.2.4-1 All-Propulsive vs Aerobrake Propellant Requirements (Small Vehicles)

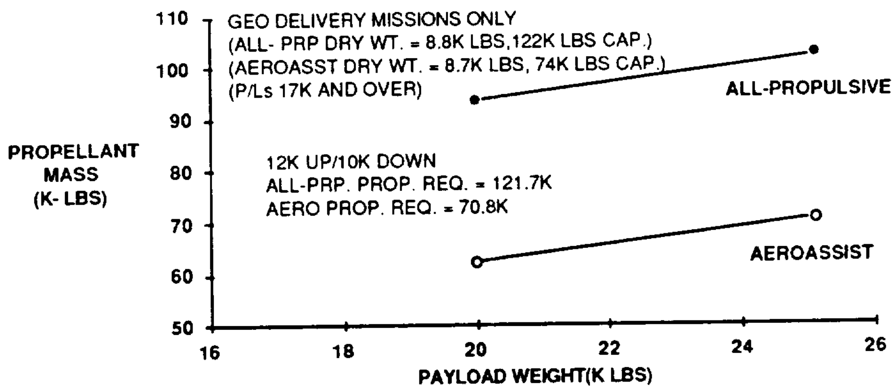


Figure 4.2.4-2 All-Propulsive vs Aerobrake Propellant Requirements (Large Vehicles)

4.2.4.1 Ground-Based Assessment

The delta life cycle cost curve for the ground-based OTV is shown in Figure 4.2.4-3 for 1985 dollars and in Figure 4.2.4-4 for discounted dollars. Over the program life aerobraking shows a net savings of $1.3B with a break-even point in 1997.

The primary factor in this difference is the higher launch costs for all-propulsive of $1.7B. For each option 95 payloads were delivered with the small OTV and 65 were delivered with the large one. For the small vehicle missions, the length load factor averaged 11% higher for all-propulsive (33 missions, on average, were length charged). The weight load factor averaged 33% higher for all-propulsive (62 missions, on average, being length charged). This translated to a net delta launch cost to the small all-propulsive vehicle of $ 870M. For the large vehicle missions, all were charged on a weight basis with the average weight load factor being 62% higher for the all-propulsive option. However, because many of the all-propulsive launch loads lie within the LCV's 75 - 100% charging algorithm plateau they are not penalized as heavily as might otherwise be expected. The net delta cost for the large vehicles winds up being $880M more for all-propulsive.

Other factors which influence the life cycle cost are aeroassist technology DDT&E ($200M penalty to aero), recurring brake hardware build and refurbishment ($265M penalty for aero), propellant cost ($11M cost to all-propulsive), and program support ($9M cost to aero).

The strongest single driver is the higher launch costs of the all-propulsive option which swings the trade in favor of aerobraking.

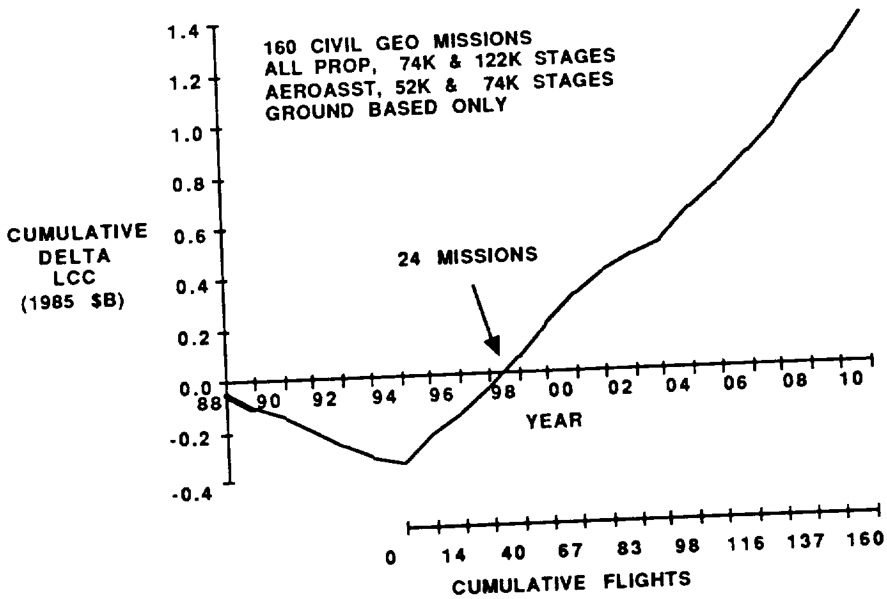


Figure 4.2.4-3 Ground-Based All-Propulsive vs Aero
Delta LCC (1985 $B)

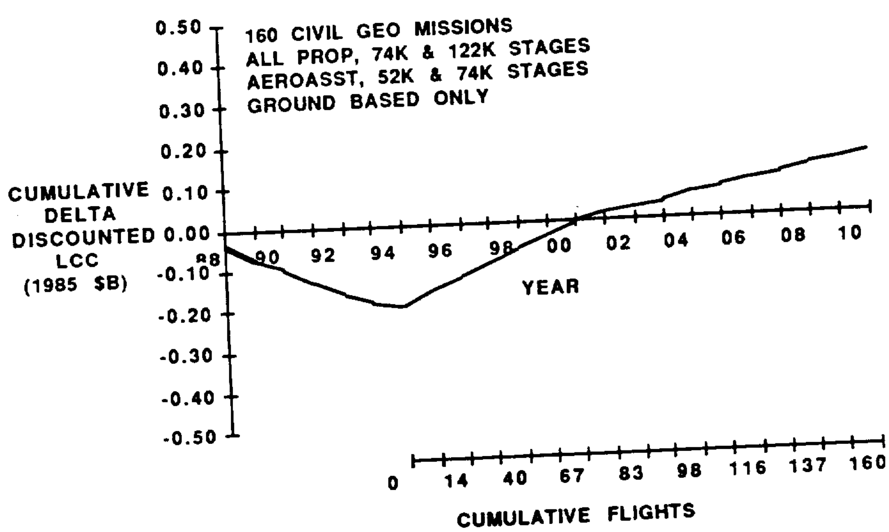


Figure 4.2.4-4 Ground-Based All-Propulsive vs Aero
Delta Discounted LCC (1985 $B)

4.2.4.2 Space-Based Assessment

The delta life cycle curve for the space-based OTV is shown in Figure 4.2.4-5 for 1985 dollars and in Figure 4.2.4-6 for discounted dollars. The curves are shown for varying amounts of hitchhiked propellant. Because the all-propulsive option requires much more propellant than the aerobraked one, a given hitchhiked propellant quantity for aero is less for all-propulsive on a percentage basis. The study final results indicate that 63% of aero and 38% of all-propulsive propellant requirements can be supplied from hitchhiking. Over the program life, then, aerobraking shows a total LCC benefit of $2.0B with a break-even point in 1996.

The primary factor in this difference, as with the ground-based option, is in the higher launch costs for all-propulsive. This cost is made up of two parts: First, the propellant delivery cost for all-propulsive is higher, as one might expect, by $2958M (for the 63% aero/38% all-propulsive hitchhiking mode); secondly, the aerobrake delivery costs of $653M (reflecting one new brake every 5 OTV flights) is charged to the aero option and partially offsets the propellant delivery cost advantage.

Other delta life cycle costs that were significant are the aeroassist DDT&E cost of $200M, stage hardware recurring costs of $17M to aero (which includes brake and tankage costs), and onorbit operations of $48M to aero for refurb and replace of brake hardware.

As mentioned earlier, delta life cycle costs due to differences in onorbit accommodations were not included but their impact cannot change the overall outcome.

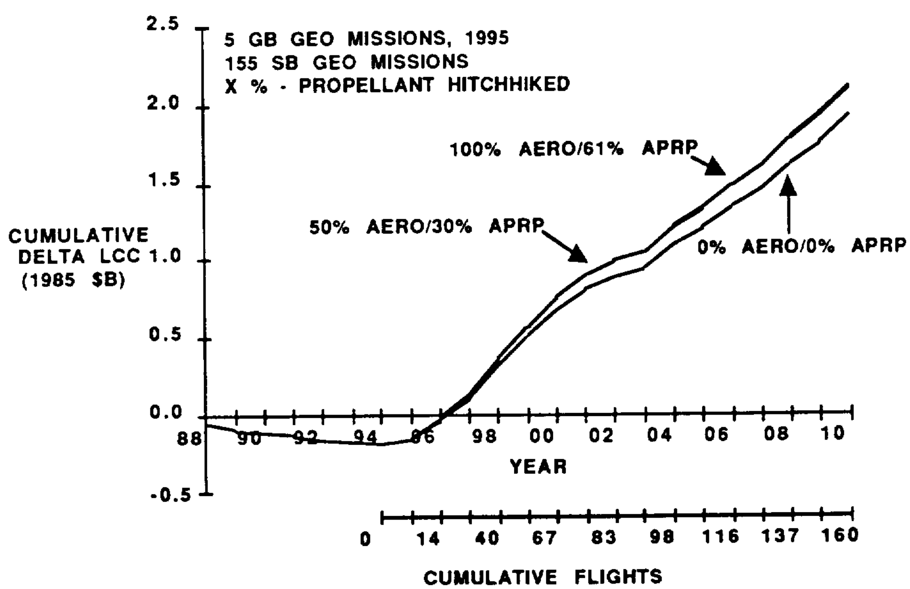


Figure 4.2.4-5 Space-Based All-Propulsive vs Aero Delta LCC (1985 $B)

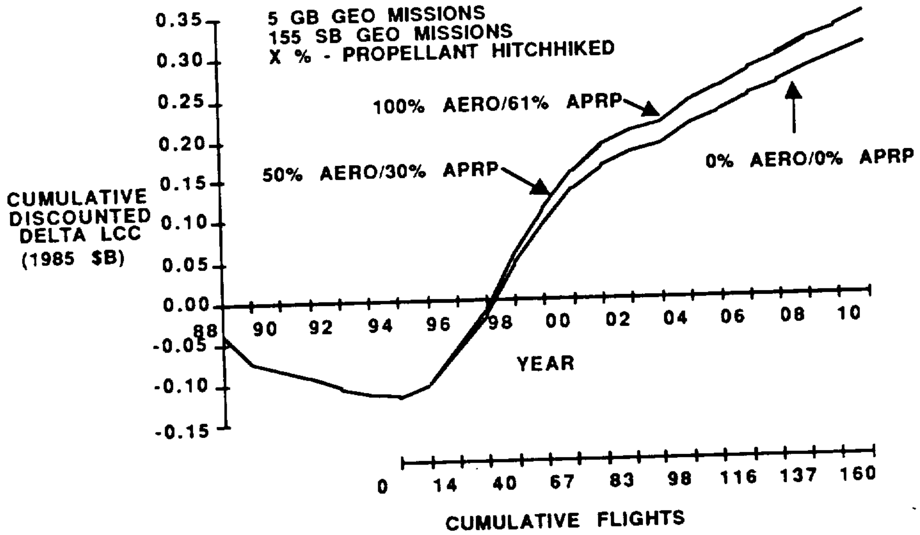


Figure 4.2.4-6 Space-Based All-Propulsive vs Aero Delta Discounted LCC (1985 ($B)

4.2.5 Sensitivity

This trade was conducted with a dedicated tanker cost of $550 per pound of propellant delivered to orbit. Subsequent analysis of actual LCV costs has upped that rate to $750 per pound. At the 63% aero/38% all propulsive hitchhiking rate this should increase the delta life cycle cost by about $1.0B in favor of aerobraking.

Rocket engines were assumed to wear out at equivalent rates for both options based on the near equivalent number of engine starts per mission. If wear out is based on engine burn time, however, the engine replacement costs will go up for the all-propulsive option, further favoring aerobraking.

If hitchhiking is disallowed a small ($100M) benefit to the all-propulsive option is realized because hitchhiking benefits aerobraking at a faster rate due to its lower propellant usage. This does not change the final answer.

Finally, as has been mentioned before, a more detailed look at onorbit accommodations will probably reveal some benefits for the all-propulsive option. The cost of aerobrake support hardware will probably be higher to the aero-option than the cost of a larger tank farm to the all-propulsive option. In any case, this savings for all-propulsive cannot be enough to change the outcome of the trade.

4.2.6 All-Propulsive vs Aerobraking Recommendations

Because of its large economic benefit, both in a ground-based mode and in a space-based mode, aeroassist is the clear choice for the OTV. This is true if accounting is done either with constant or discounted dollars. The impacts to the cost analysis mentioned in the sensitivity section above do not alter this conclusion.

4.3 AEROASSIST CONFIGURATION TRADE STUDY

A trade study was conducted to determine the optimum aeroassisted vehicle configuration and aerobrake design. To minimize impacts of configuration-peculiar delivery modes only space-based vehicles were considered in this trade.

4.3.1 Criteria

Propellant consumption is the largest cost driver in considering the various OTV candidate concepts. This is due to the high cost of delivering propellant to orbit. The maintenance and servicing operations costs are not significant comparison items between concepts because of the relatively low proportion of overall life cycle cost and also because of the similarity between concepts. Launch costs associated with replacement aerobrakes, however, are large enough to at least account for and include in any reasonable cost comparisons. The other items significant in total life cycle cost calculations are the development and production costs. Production costs include any spares or items that are replaced on a routine basis.

4.3.2 Concepts

The candidates selected for the trade study are vehicle concepts that package most optimally with the ballute, flexible fabric, and rigid brake concepts. For instance, the best tankage and structural concept for the ballute brake concept is the tandem ellipsoid/cylindrical shell configuration. Only space-based vehicles were considered because the rigid brake cannot be ground-based due to its size.

4.3.2.1 Flexible Brake OTV

The flexible fabric brake OTV concept is shown in Figure 4.3.2-1.

WEIGHT	
AEROBRAKE	1800
TANKS	1025
STRUCTURE	1370
SUPPORT (ASE)	114
ENVIRONMENTAL CONTROL	730
MAIN PROPULSION	1288
ORIENTATION CONTROL	265
ELECTRIC SYSTEMS	533
G,N & C	160
CONTINGENCY 15%	1093
DRY WEIGHT	8378
PROPELLANTS, ETC.	74015
LOADED WEIGHT	82393

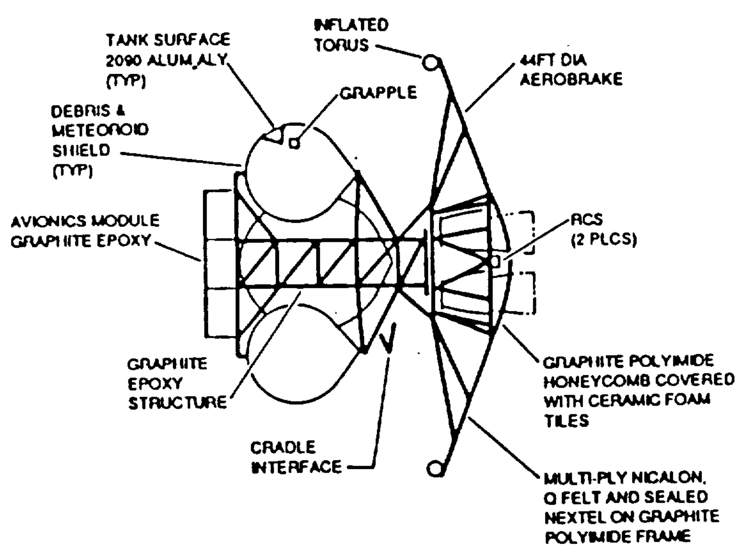


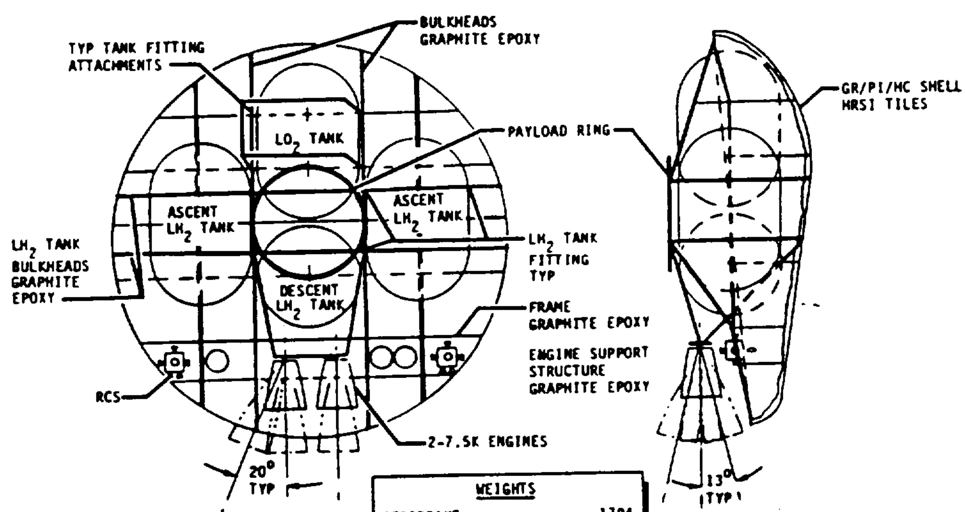
Figure 4.3.2-1 Flexible Fabric Aerobrake - Space-Based OTV

The flexible brake/vehicle concept optimizes with a wide "squatty" tankage package. This resulted in a central truss structure and subsequent side removable modular tankage. The two main engines have extendable/retractable nozzles which protrude through openings in the nose of the aerobrake. These openings are closed during the aerocapture maneuver with actuated doors. The vehicle and brake utilize a relatively low L/D (0.12) for control during the aerocapture maneuver and thus minimize the thermal loads on the fabric brake and therefore its weight. This results in a minimum weight OTV concept with adequate control capability during the aerotrajectory.

The aerobrake must incorporate a folding feature to allow delivery by either the STS or the LCV, since replacement is required after every five missions.

4.3.2.2 Rigid Brake OTV

The rigid brake vehicle concept shown in Figure 4.3.2-2 utilizes an all tile brake construction rather than an inflatable or flexible fabric surface. Since the rigid or "shaped" brake is also inherently capable of higher L/D, it can provide the vehicle with a greater degree of control capability, although it may not be required. The rigid brake concept represents the most near term technology due to the incorporation of tiles similar to those used on STS. This may result in lower initial costs and earlier IOC for OTV than other concepts. One benefit of this vehicle/brake concept is having no openings or doors for the main engines. In addition, the tankage and structure packages into the brake such that the payload will be relatively close to the brake location and thus keep the C.G. as far in front of the center of pressure as possible. This closeness minimizes the diameter requirement for the brake to avoid impingement heating upon the payload and vehicle tankage.



WEIGHTS	
AEROBRAKE	1794
TANKS	396
STRUCTURE	3350
SUPPORT (ASE)	272
ENVIRONMENTAL CONTROL	185
MAIN PROPULSION	1288
ORIENTATION CONTROL	265
ELECTRIC SYSTEMS	533
G, N&C	160
CONTINGENCY (15%)	1236
DRY WEIGHT	9479
PROPELLANTS, ETC	74000
LOADED WEIGHT	83479

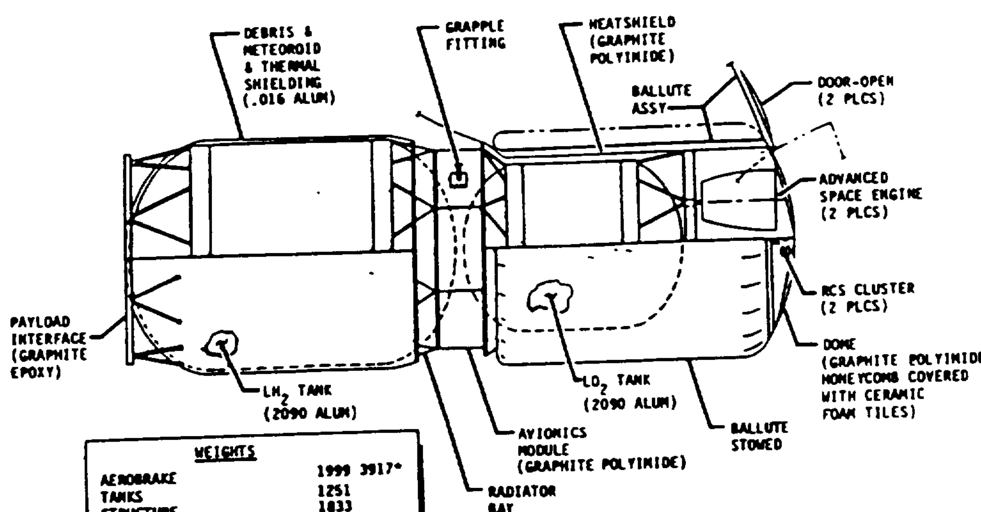
Figure 4.3.2-2 Rigid Aerobrake - Space-Based OTV

4.3.2.3 Ballute Brake OTV

The ballute aeroassisted space-based OTV concept shown in Figure 4.3.2-3 consists of a Shuttle cargo bay deliverable package. The narrow cylindrical shape lends itself to ballute usage because of the packaging concept of the ballute and because of the shape of the inflated ballute following its deployment. Therefore, the tandem propellant tankage with ballute stowage around the LO_2 tank appears to be the optimum ballute/OTV packaging arrangement.

The overall length of the vehicle is driven by the Orbiter cargo bay diameter constraint and by the slender LO_2 tank with cylindrical section in order to package the ballute.

The weights shown are for a vehicle with a ballute with 1500 deg F backwall temperature capability. The vehicle and payload heating consequences of this capability are not well understood. Therefore, the weights are also shown for a ballute with a 600 deg. F. backwall which is a more conservative estimate of material capabilities. However, the more conservative weights make the ballute concept very non-competitive with other vehicle/brake concepts.



WEIGHTS

AEROBRAKE	1999	3917*
TANKS	1251	
STRUCTURE	1833	
SUPPORT (ASE)	272	
ENVIRONMENTAL CONTROL	598	
MAIN PROPULSION	1288	
ORIENTATION CONTROL	265	
ELECTRIC SYSTEMS	533	
G, N&C	160	
CONTINGENCY (15%)	1230	
DRY WEIGHT	9429	11153*
PROPELLANTS, ETC	74000	
LOADED WEIGHT	83429	

*FOR 600°F BACKWALL TEMP

Figure 4.3.2-3 Ballute Aerobrake - Space-Based OTV

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4.3.3 Assumptions

The ground rules and assumptions used for the trade study are as follows:

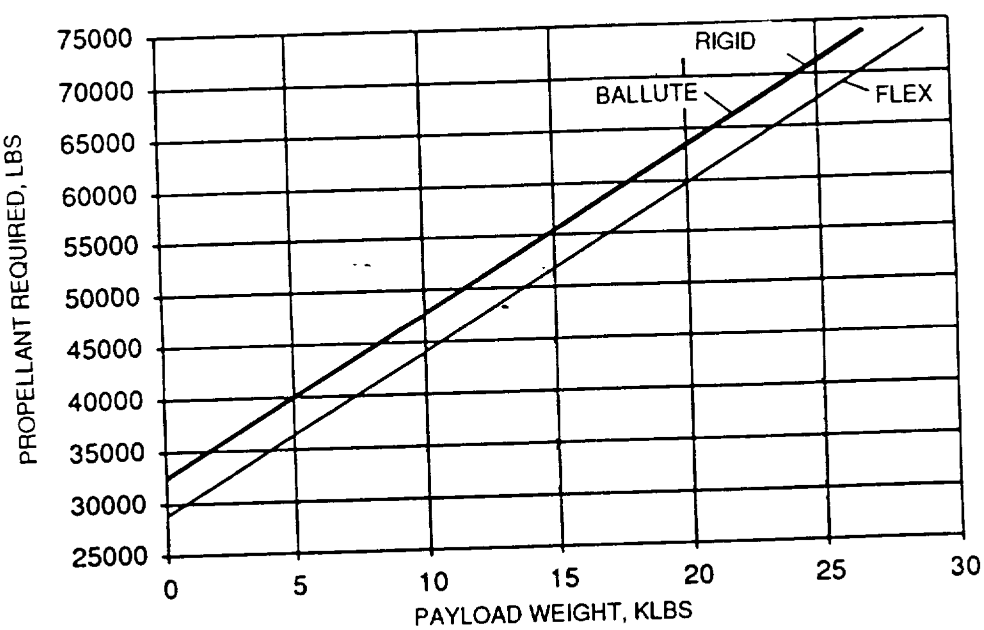
- Vehicle must be man-rated, reusable, space-based
- Deliver 13.3 Klbm to GEO and return 23 ft long 11.3 Klbm payload
- Delivered to orbit and supported by STS
- Single pass aerocapture maneuver

4.3.4 Assessments

4.3.4.1 Performance

Unlike the ground-based systems which are charged for launch costs on the basis of length or gross weight, space-based OTV concepts are primarily assessed by propellant usage. They are delivered once so packaging and manifesting do not present first order impacts. The delivery of propellant to orbit is typically the most important facet. Any concept which can reduce this quantity will be a strong contender.

Figure 4.3.4-1 summarizes the performance in terms of propellant requirements of the three space-based configurations: the rigid brake, the ballute, and the flexible brake styles. Clearly seen in this chart is the significant performance advantage of the flexible design due to its lower weight.



• VEHICLE PERFORMANCE FOR GEO-DELIVERY

Figure 4.3.4-1 Space-Based Aeroassisted OTV Performance Summary

4.3.4.2 STS Support Requirements

A comparison was made of the three space-based candidate aerobrake concepts from an STS support standpoint. To initially deliver the OTV to the Space Station requires two orbiter flights for the flexible brake, one flight for the ballute, and three flights for the rigid brake configuration. The initial OTV mission is then flown. For the remainder of the 39 flights necessary to meet the 40 mission life requirement, the flexible brake is replaced every five flights, and ballute every flight, and the rigid brake twice. Assuming that the flexible brake occupies approximately 1/3 the payload bay, the ballute requires 1/4 of the bay, and the rigid brakes 2/3 of two separate payload bays, the comparison is as shown in Table 4.3.4-1.

Table 4.3.4-1 Orbiter Flight Requirements (40 OTV mission life)

	FLEXIBLE BRAKE	BALLUTE	RIGID BRAKE
INITIAL ASSEMBLY	2	1	3
ADDITIONAL BRAKE DELIVERY DURING 40 MISSIONS *	2.3	9.75	1.3
TOTAL ORBITER FLIGHTS **	4.3	10.75	4.3

* CONSIDERING FLEXIBLE BRAKE REQUIRES 1/3 OF PAYLOAD BAY BALLUTE REQUIRES 1/4 OF PAYLOAD BAY AND RIGID BRAKE REQUIRES 2/3 OF 2 PAYLOAD BAYS

** ENGINE REPLACEMENTS NOT CONSIDERED SINCE THEY SHOULD BE THE SAME IN ALL CASES

4.3.4.3 Mission Support Requirements

A comparison was made of the three space-based candidate aerobrake concepts from a pre- and post-mission IVA operations standpoint (see Table 4.3.4-2). Both the flexible brake and rigid brake configuration require on-orbit assembly of the entire space-based OTV after initial delivery, while the ballute does not. However, this activity occurs only once during the forty mission life of the OTV, the effect on a upper mission basis is very small. Pre-mission and post-mission processing of all three candidates were considered to be the same with the exception of aerobrake inspection and replacement. Since the ballute is jettisoned after each mission the inspection of its inner rigid portion should be less than required for the other concepts. The ballute requires replacement each mission, the flexible brake is replaced after 5 missions, and the rigid brake is replaced after 20 missions. However, when considered from an overall processing flow, no significant difference appears between concepts.

Table 4.3.4-2 IVA Operations Time Comparisons

OPERATION	IVA OPERATIONS TIME (MINUTES) ON A PER MISSION AVERAGE BASIS		
	FLEXIBLE BRAKE	BALLUTE	RIGID BRAKE
ASSEMBLE NEW OTV ON-ORBIT (ONCE PER 40 MISSIONS)	17	0	12
PREMISSION PROCESSING	800	800	800
POSTMISSION PROCESSING	820	820	820
AEROBRAKE INSPECTION	30	10	30
REMOVE / INSTALL BRAKE (EVERY 5, 1, OR 20 MISSIONS)	46 (EVERY 5)	120 (EVERY TIME)	11 (EVERY 20)
TOTAL IVA OPERATION TIME (AVERAGE MINUTES PER MISSION)	1713	1750	1673

4.3.4.4 Cost Comparison

Figure 4.3.4-2 shows the cost data for each of the space-based aeroassisted OTV concepts considered during this trade study. The results indicate that production and development costs are not significant discriminators in comparing the vehicle concepts. The major cost item is the operational cost of providing propellant for the OTV. The range of propellant cost for the three concepts was from $610/lbm to $680/lbm depending upon propellant requirement over and above the propellant available from hitchhiking (see paragraph 3.2.3). Included in the operational costs are the servicing operations of removing and replacing the aeroassist devices on each of the OTV concepts. This particular operation is the only discernable difference in space-based maintenance of the three concepts and is still relatively minute in comparison to the propellant launch costs.

4.3.5 Recommendation

The conclusions from this trade study include the observation that propellant usage for a space-based OTV is the major consideration in selecting an OTV concept. Of course an aeroassisted vehicle design needs to provide for the amount of lift to drag ratio required for adequate control. Since it is now generally accepted that 0.12 L/D is sufficient, the lighter weight flexible fabric aerobraked vehicle is recommended over the other OTV concepts presented here.

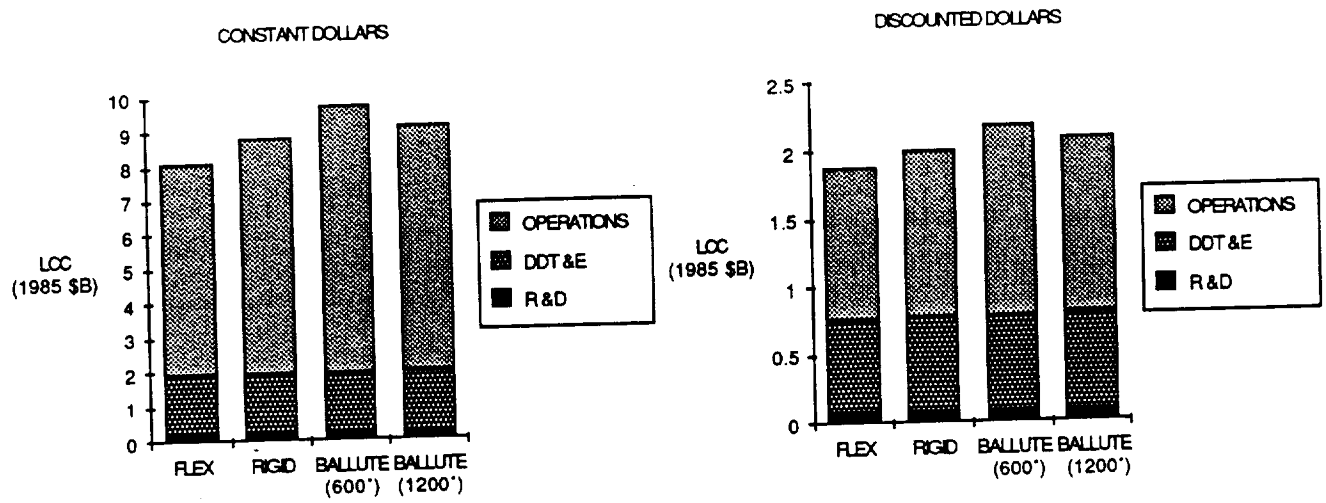


Figure 4.3.4-2 Cost Summary - Space-Based Aeroassisted OTV

4.4 ACC VS CARGO BAY TRADE

The purpose of this trade study was to determine the optimum OTV design concept for STS. An aft cargo carrier (ACC) OTV design concept had been defined in-depth in earlier study effort. Therefore, study effort was spent on determining the best cargo bay concept for comparison with the ACC concept.

Several vehicle candidates were sized to deliver 15 Klbm to GEO and fit within the cargo bay. These candidates were intended to fly all GEO delivery missions in the 1995 - 1997 time frame and be available for flying all the missions not requiring a larger stage in the 1997 - 2010 time frame.

4.4.1 Criteria

The trade study was based on two major criteria; life cycle costs and design flexibility. Safety was a requirement of all designs and therefore not an evaluation criteria. This presumes that the recent NASA decision not to allow the Centaur in the STS payload bay was based on Centaur/ASE design issues, and is not a blanket decision to prohibit all cryogenic stages in the payload bay. The single largest cost in the ground-based LCC is the cost associated with STS launches for the OTVs and payloads. Launch cost is strongly influenced by configuration length and the impact it has on the STS charge algorithm. Other costs are important in understanding the comparisons of various design concepts. These include the development and production costs for each of the concepts. In addition, the operations cost differences between concepts is an important quantity to understand. For instance, the ACC OTV concept requires disassembly and stowage into the orbiter following its mission. These operations costs are a penalty to the ACC OTV concept and are included in comparisons with other concepts.

The most difficult criteria to quantify and assess is flexibility and growth. These can be understood by considering the long term candidate vehicle scenarios and developing cost data commensurate with these scenarios. The initial OTV design is required to grow in later years to accommodate the higher energy mission requirements and to enable conduct of manned missions to GEO (and the moon in Scenario 5).

4.4.2 Concepts

The design concepts considered for the cargo bay trade study included storable and cryogenic propellants and various configurations of each of these propellant types. The cryogenic propellant concepts were sized for the three tankage configurations shown in Figure 4.4.2-1. In addition, the concepts were sized for two aerobrake types (ballute and flexible folding fabric brakes) for each of the tankages. Each of these cryogenic concepts is intended to be fully reusable with the exception of the aerobrakes which must be replaced after each mission since the fabric cannot be refolded after exposure to the aeropass environment.

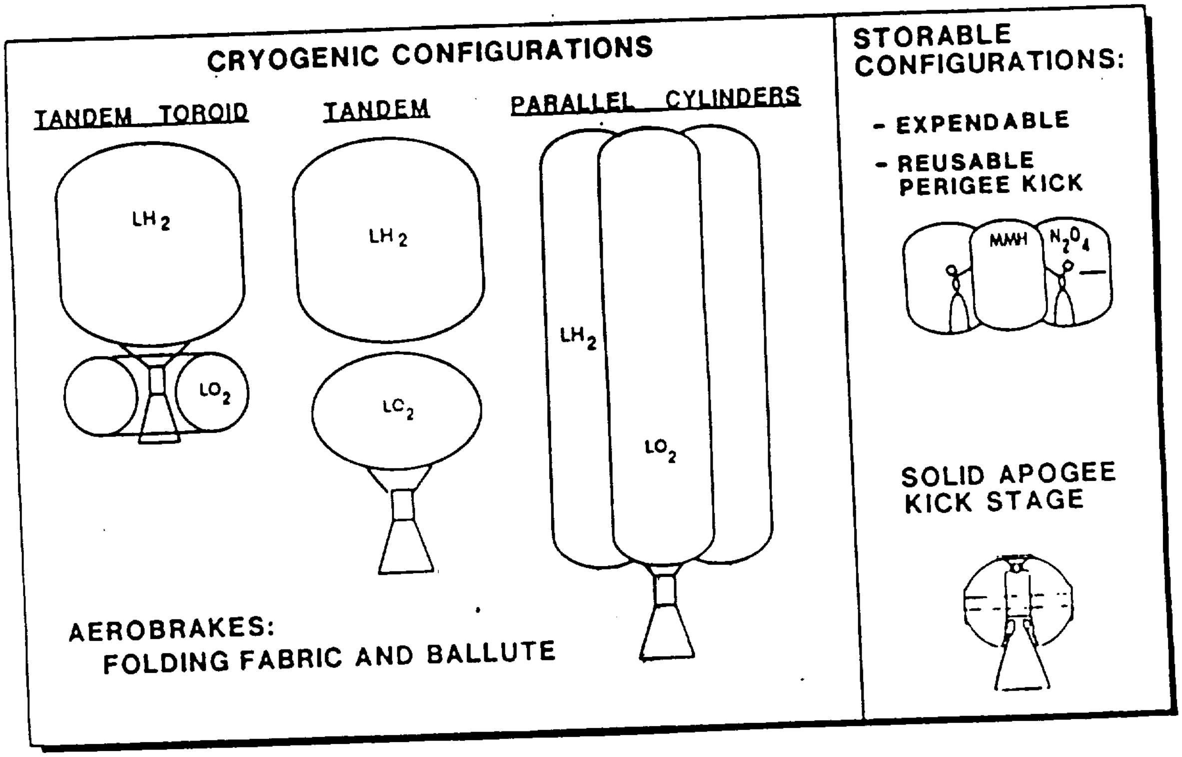


Figure 4.4.2-1 - OTV Design Concepts for STS Cargo Bay

The storable propellant concepts included a liquid expendable stage and a reusable liquid perigee stage with a solid apogee kick stage. These were also sized and priced in order to compare lengths and total launch and operations costs with the most attractive cryogenic configurations.

4.4.3 Assumptions

Cost comparisons were made on the basis of the 31 STAS mission model payloads shown in Table 4.4.3-1. These payloads were multiply manifested (maximum of 4 per launch) into the 60 foot long STS payload by accounting for the length and weight of the ASE, OTV and the payloads. Consequently, longer OTV configurations require more STS launches to accommodate the 58 missions flown by the 31 payloads. In all cases the STS lift capability was assumed to be 72K and the OTV was sized to lift 15K to GEO.

Table 4.4.3-1
Payloads Considered for Cargo Bay Launched OTV

STAS PLD NUMBER	PROGRAM NAME	PAYLOAD DATA				QUANTITY BY YEAR			
		FLIGHT 1	WGT(LB)	LENGTH(FT)	DIAM(FT)	1995	1996	1997	TOTAL
1012	COMM SAT-CLASS I	95	1545	9.8	14.8	1	1	2	4
1013	COMM SAT-CLASS II	95	2975	19.7	14.8	1	2	3	6
1016	COMM SAT-CLASS III	96	4410	29.5	14.8		1	2	3
1032	GSTAR	95	2030	7.9	14.9	1	2		3
1039	SATCOM K F/O	95	2300	12.1	14	2		1	3
2104	GEOSYNC ORBITING ENVIRONMNTL SAT (GOES	95	875	7.9	14.9	1	1		2
2194	HIGH-FREQ DIR BROADCAST SATELLITE (VOA)	96	33070	30	14.9		1		1
2195	MOBILE-SAT-B	95	14550	19.7	13.1	1			1
3446	BS F/O	96	1200	7.9	14.9		1	1	2
3447	COMM SATS INT.-OTHER (ORION,CYGNUS,ISI)	95	1300	8.5	14.9	1			1
3451	DATA RELAY SAT -1,-2,-3	95	1500	7.9	14.9	2			2
3452	DBS	95	1500	12.1	14.9	1	1		2
3453	DBS F/O-UK	95	2000	12.1	14.9	1	1		2
3454	DFS-KOPERNIKUS F/O	96	2400	12.1	14.9		1	1	2
3455	ECS F/O	95	2000	12.1	14.9	1			1
3456	GDL F/O	96	1800	12.1	14.9		1	1	2
3458	GEOSTAR	96	1400	8.5	14.9		1	2	3
3464	INTELSAT VIII	96	3500	24	14.9		2	1	3
3468	KOREASAT	96	1810	10.8	14.9		1		1
3472	NORDCOM	97	3200	20	14.9			1	1
3478	SARIT	96	2645	12.1	14.9		1		1
3479	SBTS-A3 (BRAZIL)	95	1380	7.9	14.9	2			2
3480	STW F/O	95	1900	7.9	14.9	1	1		2
3484	TELESAT CANADA	95	1380	7.9	14.9	1			1
3486	TV-SAT (OPERATIONAL)	95	2700	7.9	14.9	1			1
3487	UNISAT (BRITISH COMM)	95	1870	7.9	14.9	1	1		2
4480	GEOS-2	97	2205	13.1	13.1			1	1
4482	GMS-X	96	1810	13.1	13.1		1		1
4496	METSAT	95	1520	9.8	14.9	1			1
4508	SYNCH SYSTEM-OTHER	97	1200	9.8	14.9			1	1
					TOTALS	20	21	17	58

Table 4.4.3-2 lists the costing ground rules and assumptions used in comparing cargo bay vehicle concepts., existing upper stages, and the ACC OTV concept. In addition, this list indicates the grouping of costs for the comparison data which follows. For instance, the items considered to be operations costs are listed here and the corresponding quantities will be combined under "operations" in the cost comparison assessment.

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Table 4.4.3-2

GROUND-BASED OTV COST GROUND RULES AND ASSUMPTIONS

- ALL COSTS IN 1985 DOLLARS INCLUDING PROFIT, MANAGEMENT RESERVE AND GOVERNMENT SUPPORT

- R&T
  - No differences except for expendable stages without brake AFE

- DDT&E
  - Ground test hardware includes STA, GVTA, MPTA and functional test articles
  - Dedicated flight test article
  - Flight test and GVTA/functional test articles refurbished to operational units

- PRODUCTION
  - Initial operational requirements include one operational unit and one spare (DDT&E units refurbished)
  - Hardware spares included in operations

- OPERATIONS
  - Costs include
    - STS launch costs (for both payload and OTV)
    - Stage operations (spares, ground ops, refurb, IVA, etc.)
    - OMV, ACC, and payload attach when applicable
  - Government supplied cost data for STS launch cost, OMN, IVA/EVA and existing upper stages used when applicable

4.4.4 Assessments

4.4.4.1 IVA Operations Time/Cost Comparison

A comparison was made of the ground-based vehicle candidates from an IVA operations time/cost standpoint. Pre-mission and post-mission operational times were common for all the payload bay concepts with the exception of the fully expendable configuration which obviously does not require any post-mission activity. This data is shown in Table 4.4.4-1.

The aft cargo carrier concept requires considerably more operational time due to the greater complexity involved in grappling after rendezvous and the necessity to mate the OTV with the payload carried within the Orbiter bay. Also post-mission times are longer due to the additional operations required to stow this configuration in the payload bay for return to base.

Table 4.4.4-1
Time/Cost Comparisons for Onorbit Operations

OPERATION	STORABLE: FULLY EXPENDABLE	STORABLE: EXPENDABLE KICK STAGE	CRYO: PAYLOAD BAY TOROID TANK	CRYO: PAYLOAD BAY TANDEM TANK	CRYO: AFT CARGO CARRIER
PREMISSION TIME (MINUTES)					
PAYLOAD CHECKOUT	20	20	20	20	20
GRAPPLE & MATE OTV/ PAYLOAD	--	--	--	--	105
OTV 1/4 CHECKOUT	45	45	45	45	45
DEPLOY	5	5	5	5	5
TOTAL PRE-MISSION TIME	70	70	70	70	175
POST-MISSION TIME (MINUTES)					
GRAPPLE OTV	--	35	35	35	35
STOW IN P/L BAY	--	45	45	45	180
TOTAL POST-MISSION TIME	0	80	80	80	215
TOTAL IVA OPERATIONS (MIN)	70	150	150	150	390
IVA COST, 2-MEN @ $600/MINUTE	$42K	$90K	$90K	$90K	$234K

4.4.4.2 Number of STS Launches

The number of STS launches required to accommodate the 58 payload events is shown as a function of available payload bay length in Figure 4.4.4-1. For reference, a difference of 10 launches impacts the average launch cost of <u>each</u> of the 58 payloads by $12.6M.

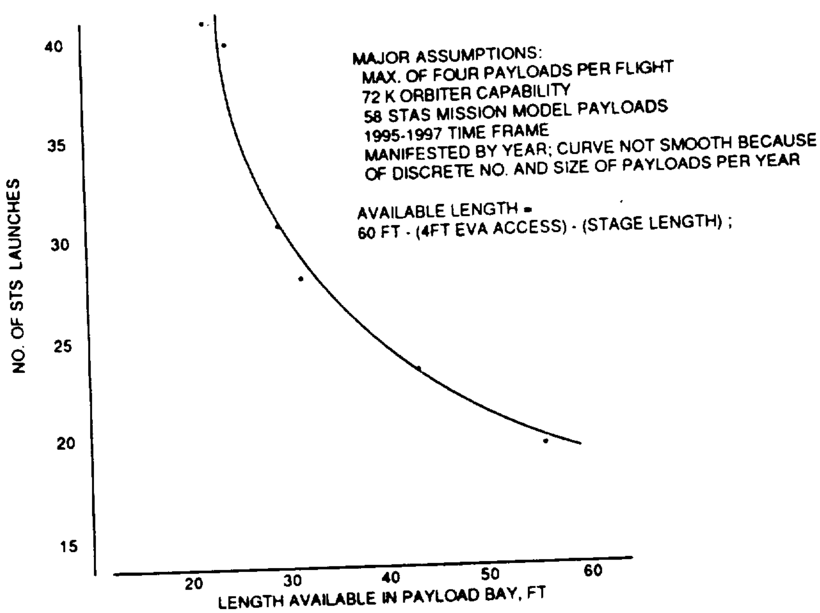


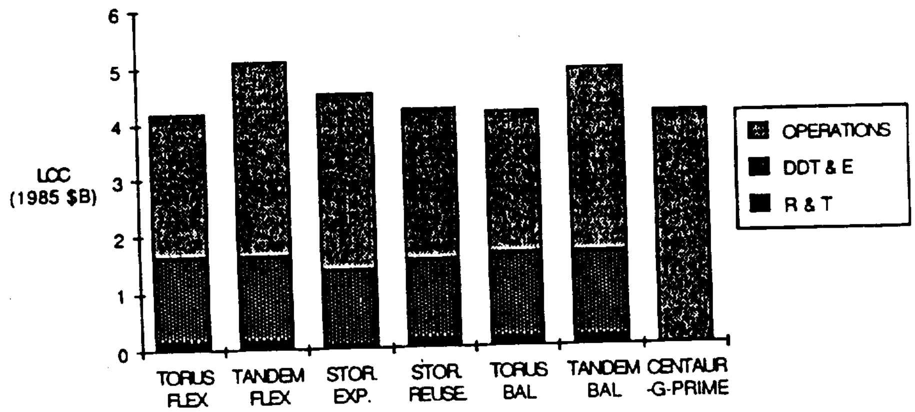
Figure 4.4.4-1 Number of Launches as a Function of Space in Payload Bay

4.4.4.3 Life Cycle Costs

4.4.4.3.1 Cargo Bay Candidates

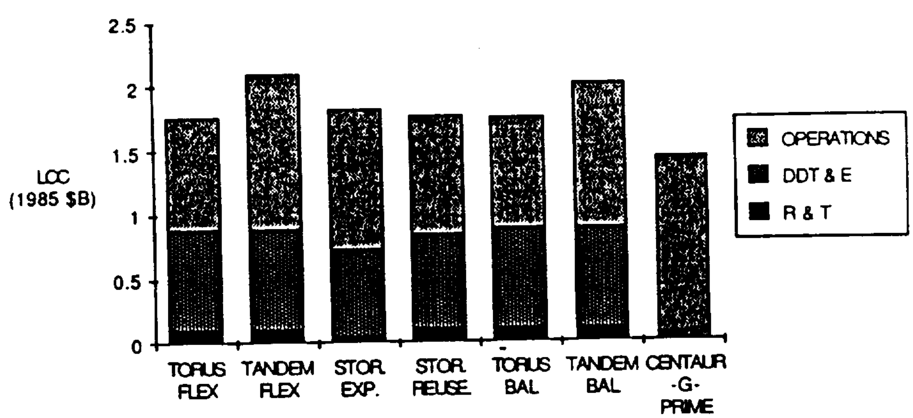
Tables 4.4.4.2 and 4.4.4-3 show the constant and discounted cost data for the ground-based cargo bay OTV candidates. The primary evaluation criteria of interest is that of costs associated with STS flights for OTVs and their payloads. Length of each concept in the cargo bay is, of course, the large driver in determining STS flights required. Production and development costs may not be significant in terms of decision making, but they are accounted for, nonetheless.

Table 4.4.4-2 LCC Comparison, Cargo Bay OTVs (Constant $85)



GROSS WT (WT)	67.3	66.7	77.5	82.9	68.3	67.8	75.0
LENGTH (FT)	26.0	33.5	11.5	24.0	26.5	31.3	26.0
FLIGHTS	31	42	23	28	31	40	31

Table 4.4.4-3 LCC Comparison, Cargo Bay OTVs (10% discounted)



The data shows four of the OTV options are cost competitive.

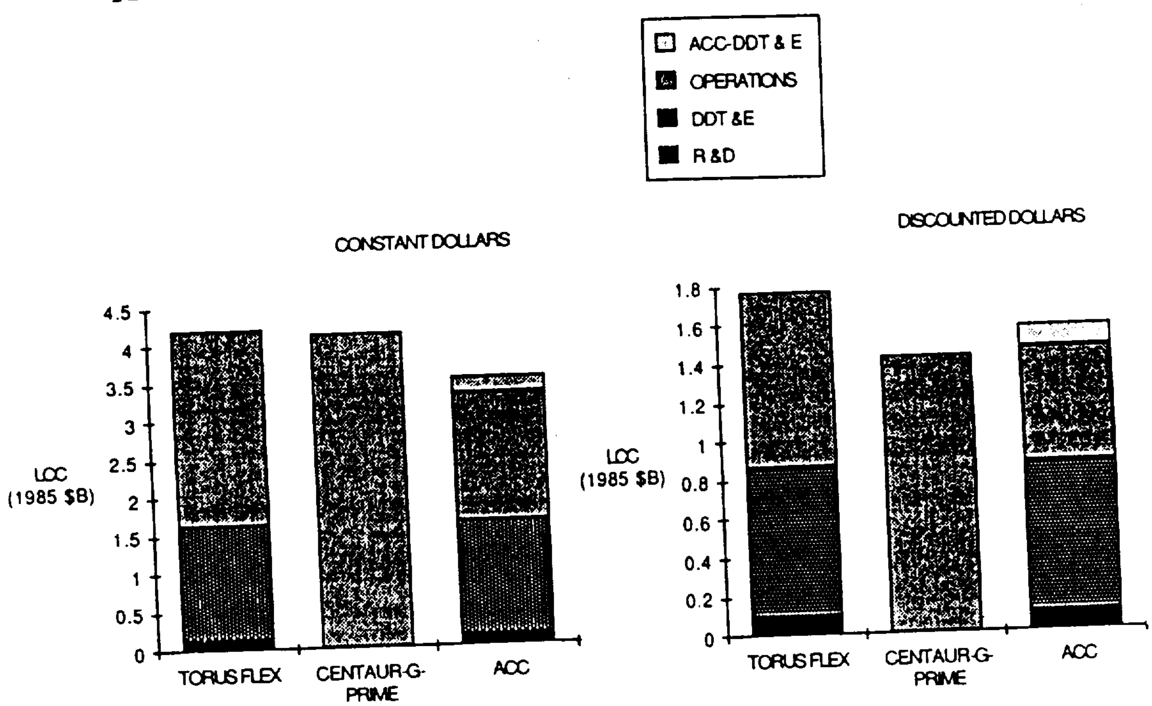
(1) Torus cryogenic with flex brake
(2) Storable reusable
(3) Torus cryogenic with ballute brake
(4) Centaur G prime

Candidates (2) and (4) are not really options: The cargo bay version of the Centaur G prime was recently cancelled and the storable reusable requires an 83K STS capability. The remaining cost competitive candiates are the torus cryogenic stage with either a ballute or a flex brake. The reason for selecting the flex brake over the ballute was discussed in detail in paragraph 4.3.5

4.4.4.3.2 ACC OTV

The ACC OTV concept incurs several costs not associated with the cargo bay versions. On orbit rendezvous and mating with the payloads, and post mission disassembly of the OTV are the major differences. These costs are included in the LCC data shown in Table 4.4.4-4. This data clearly shows that the reduced number of STS launches brought about by stowing the OTV in the ACC more than offsets the unique ACC costs.

Table 4.4.4-4 Cargo Bay Vs ACC OTV LCC Comparison



4.4.5 Sensitivity

The major cost driver in an STS constrained ground-based OTV program is launch costs. Launch costs are minimized by the ACC OTV because the entire payload bay is available for payload placement. If the STS payload data that was shown in Table 4.4.3-1 does not accurately reflect true payload dimensions, the ACC OTV would not show such a strong cost advantage. The average density calculated from the stated payload weights, lengths and diameters is 0.8 lb/ft^3. However, typical small satellites generally have much greater densities. As an example, a 7 foot long, 6 foot diameter payload weighing 1000 lbs has a density of 5 lb/ft^3. If this payload was assigned a diameter of 15 ft (because that is what is currently used in the STS payload bay when the payload is mated with its PAM), the calculated density in reduced to an apparent value of 0.8 lb/ft^3. It is quite possible that many of the small payloads could be situated side-by-side or three in a cluster when multiply manifested for OTV launches. If so, a cargo bay OTV would require fewer launches than indicated to capture the mission model; however, it would still require more launches than the ACC OTV.

4.4.6 Cargo Bay Vs ACC Recommendation

The ACC OTV concept has been selected over the cargo bay concept for several reasons. The primary criteria for this recommendation is the reduction in STS flights (over the cargo bay concept) by carrying the OTV in the ACC. Also, the stowage of the aerobrake is much simpler for the ACC OTV concept than for the cargo bay concept because of the larger diameter package that it is folded around and the larger envelope available around the OTV in the ACC. Further, the growth path to space basing and the flexibility for integration with new launch vehicles is more apparent for the large diameter modular ACC OTV concept than it is for the cargo bay concept.

4.5 DIAMETER OF LARGE CARGO VEHICLE GB OTV

Three trade studies were performed to determine the optimum ground-based OTV configuration for delivery to orbit in a large cargo vehicle. These are: 1) choice of OTV diameter, 2) number of engines, 3) number of vehicles (and their sizes) in the OTV fleet. The first of these trade studies compares two vehicle configurations of different overall diameter in order to select the best concept.

4.5.1 Criteria

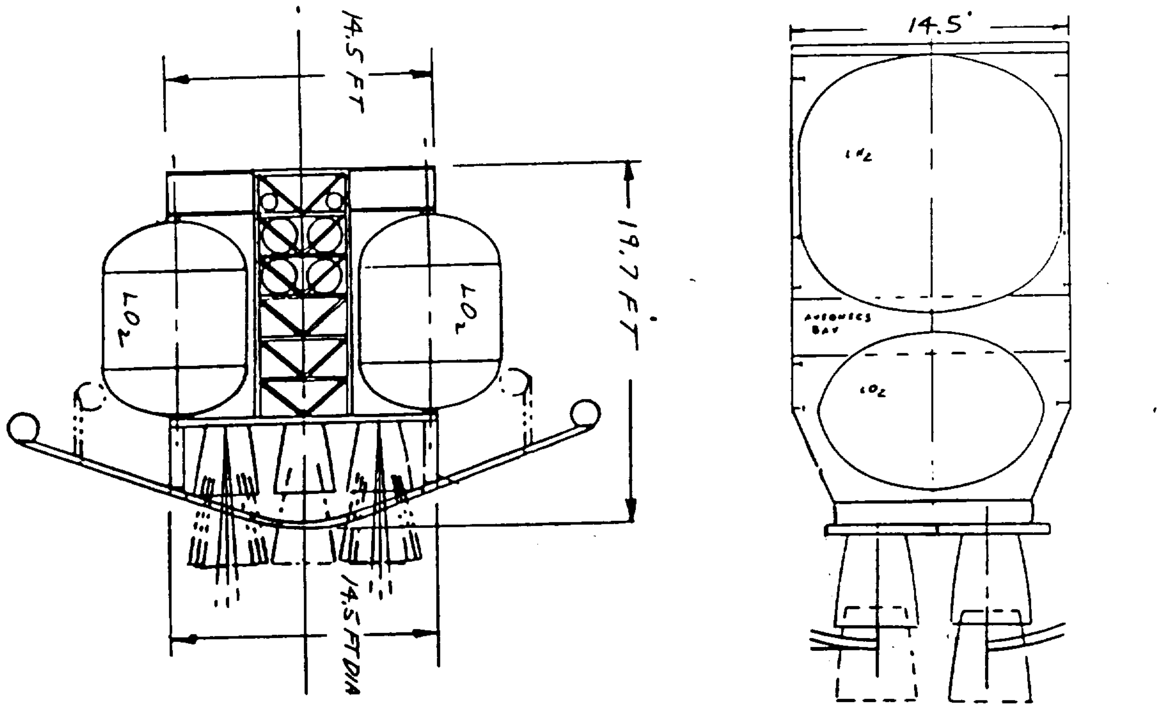
The trade study was based on two major criteria; life cycle costs and design flexibility. Safety was a requirement of all designs and therefore not an evaluation criteria. The single largest cost in the ground-based LCC is the cost associated with cargo vehicle launches for the OTVs and payloads. Other costs are important in understanding the comparisons of various design concepts. These include the development and production costs for each of the concepts. In addition, the operations cost differences between concepts is an important quantity to understand.

The most difficult criteria to quantify and assess is flexibility and growth. These can be understood by considering the long term candidate vehicle scenarios and developing cost data commensurate with these scenarios. The initial OTV design is required to grow in later years to accommodate the higher energy mission requirements and to enable conduct of manned missions to GEO (and the moon in Scenario 5).

4.5.2 Concepts

Figure 4.5.2-1 shows the OTV candidates that were considered for use with the large cargo vehicle. The concept on the left is a 3 engine "wide body" design that is sized such that the vehicle with aerobrake will fit within the 25 ft diameter of the cargo vehicle. Paragraph 4.6 describes the reasons for selecting three engines. The core of the vehicle is sized to 14.5 ft in order to return the expensive parts of the vehicle to earth in the STS (main engines, avionics, RCS systems). The tankage is a large volume, low cost item that may or may not be retrievable to earth in a single STS flight. Therefore, the tanks are intended to be removable from the core following the mission.

The two engine concept on the right is capable of being returned to earth in a single STS flight without any disassembly except for the jettisoning of the aerobrake. This design concept is longer than the wide body concept and therefore is more expensive in terms of launch costs. The man-rating requirement dictates more than one engine. The torus tank concept does not adapt to multiple engines, therefore, the longer but lighter weight tandem ellipsoid tank concept was utilized in this trade.



WIDE BODY　　　　15 FT. COMPATIBLE

Figure 4.5.2-1　Large Cargo Vehicle OTV Candidates

4.5.3 <u>Assumptions</u>

This study assumes that the large cargo vehicle does not have return capability. If it does in fact have the capability to return an entire, fully assembled OTV, the launch cost charging algorithm shows the short, wide-body OTV is the winner. Both concept candidates are assumed to be capable of being man-rated in order to support all types of missions from a ground-based mode of operation.

4.5.4 <u>Assessments</u>

The pie chart of Figure 4.5.4-1 shows the relatively small portion of a reusable aeroassisted OTV that tankage represents. The wide body OTV candidate must jettison and expend at least the LH_2 tanks before subsequent stowage of the core into an orbiter bay for return to earth. One can see that half the tankage cost does not comprise a significant portion of an OTV unit cost and therefore makes expendable tankage a viable concept for a reusable OTV.

FIRST UNIT PRODUCTION COST PERCENTAGES BY SUBSYSTEMS, MMC ESTIMATES

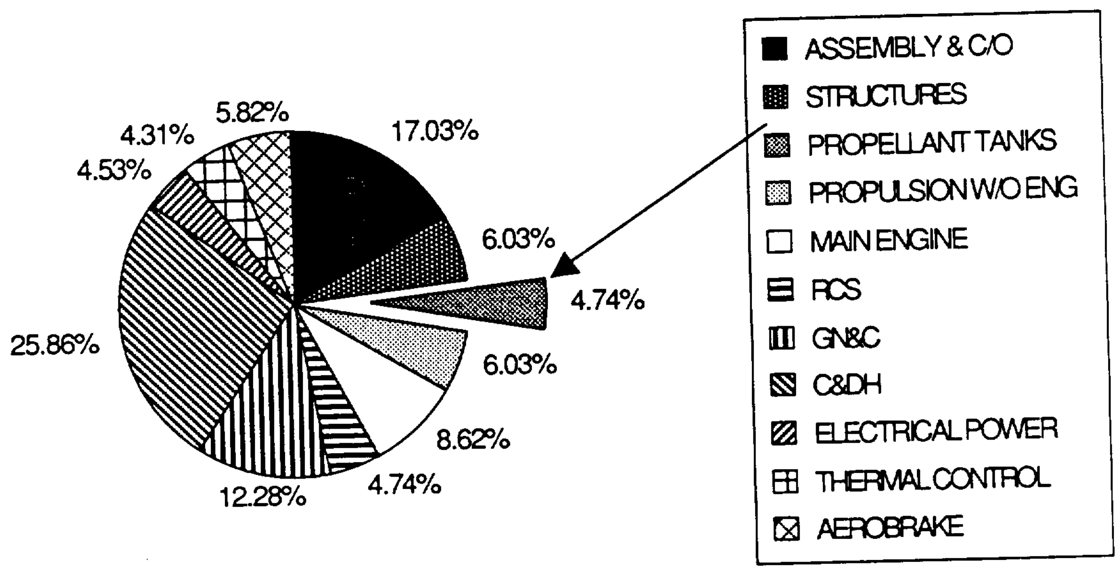


Figure 4.5.4-1 OTV Unit Costs by Subsystem

Other issues to consider besides cost of tankage are the method of disposal for the jettisoned tankage and the loiter time for STS retrieval of the OTV in LEO. The ballistic coefficient of the OTV core with O_2 tanks for the wide body concept is about 10 times that of the H_2 tanks alone. Therefore, the H_2 tanks will deorbit in 1/10 the time. So the loiter time while awaiting STS pickup of the core can be selected depending upon what rendezvous altitude is chosen. The operational aspects of tank disposal are described in paragraph 7.2.4.

Figure 4.5.4-2 shows OTV length vs propellant capacity for concepts that can be returned to earth in STS without disassembly (15 ft compatible) and for a wide diameter OTV concept. All these concepts would fit in a 25 ft diameter large cargo vehicle bay for delivery to LEO. The wide diameter vehicle will require disassembly in low earth orbit following a mission in order to fit within the 15 ft diameter constraint of STS. The three concepts increase in length with increased propellant load at approximately the same rate.

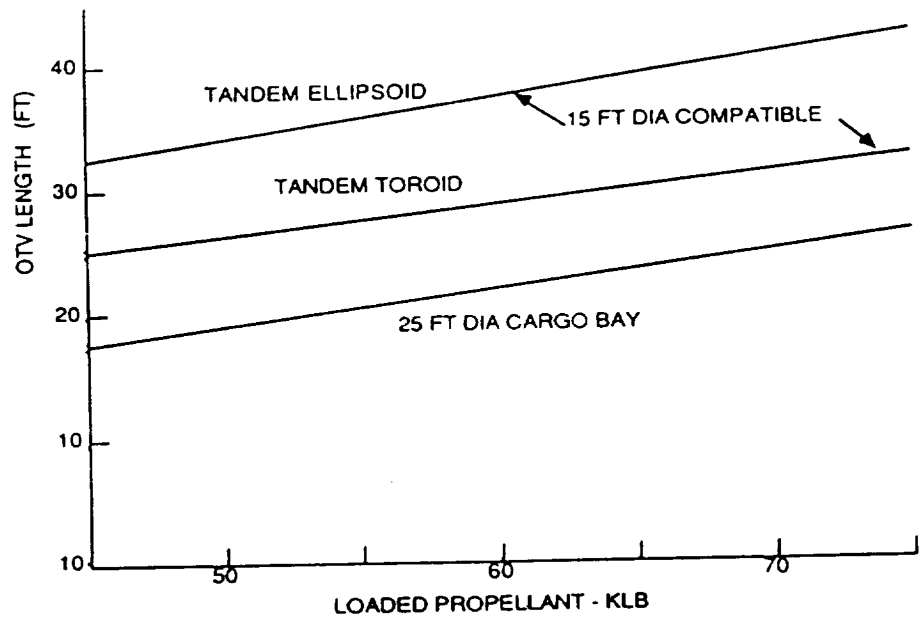


Figure 4.5.4-2 Length Comparisons - Ground-Based OTV

Figure 4.5.4-3 shows the weight comparisons of the wide body vehicle concept and the narrow diameter concept as a function of loaded propellant. Due to the length of the narrow diameter concept the aerobrake diameter is subsequently increased also. This is the main contributor to the increased vehicle dry weight over the wide body configuration.

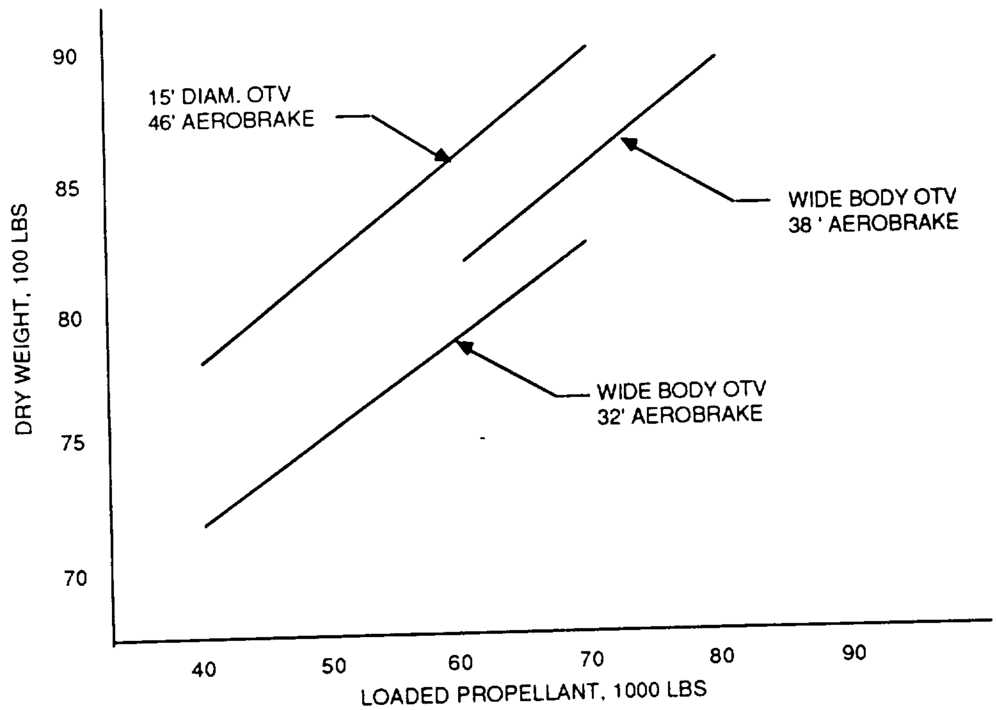


Figure 4.5.4-3 Weight Comparisons - Ground-Based OTV

Figure 4.5.4-4 presents the costs associated with the vehicle concepts under consideration. The development and original production costs are very similar. The same is true for mission loss costs even though the two engine narrow diameter OTV concept is slightly more reliable than the three engine wide body OTV concept.

The largest cost difference between the two concepts is the launch cost which is primarily due to the length differences of the vehicle configurations. The manifesting of the individual payloads in the mission model results in some payloads being charged by length while others are charged on a weight basis. A longer vehicle (with payloads) will be charged more often on length than the shorter vehicle, which increases total launch costs. However, as discussed in Paragraph 2.1.2, side-by-side packaging of payloads might reduce launch costs to the extent that makes the concepts essentially cost the same.

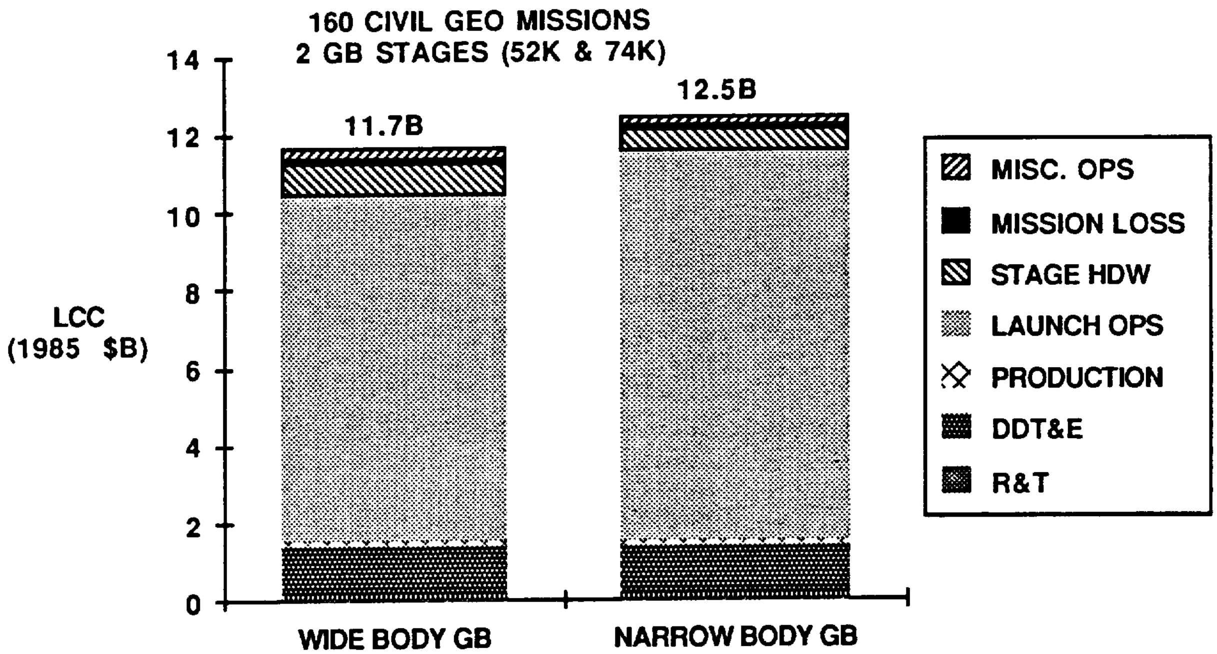


Figure 4.5.4-4 Cost Comparisons - OTV Diameter Trade

The mission loss cost differences between the two vehicle concepts (a function of the main propulsion system differences of three engines vs two) is insignificant compared with the other LCC cost items. Onorbit operations also appears to be a minor contributor to cost differences between the two candidate vehicle concepts. The only cost item that appears to be a noticeable penalty against the wide body concept is the replacement tankage costs. However, this hardware cost is still relatively minor.

4.5.5 Sensitivity

If the large cargo vehicle has return-to-earth capabilities, the wide body OTV still has the same cost advantages. However, if the NASA desires a single upper stage that is capable of being launched on either STS, STS II or the LCV for assured access to space, then the 15 foot diameter OTV would be the obvious selection in spite of the higher costs.

4.5.6 Recommendation

The wide body OTV design is recommended for use in a large cargo vehicle. Its shorter length substantially reduces launch costs when these costs are evaluated using the STS algorithm for shared launch costs. This study should be re-visited when LCV design details are being defined. Parallel payload packaging could possibly make the narrow body OTV equally attractive for LCV usage.

4.6 MAIN PROPULSION SYSTEM TRADE STUDY

The main propulsion system engine arrangement has a first order impact upon the length of an OTV and length is an important commodity in any launch vehicle. Therefore, a ground-based OTV design should be an optimum arrangement of major components with the proper parameters considered in this optimization process.

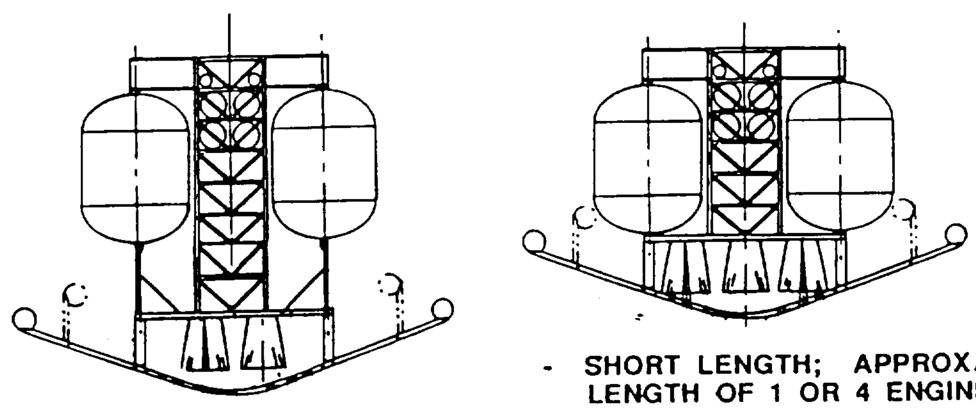
4.6.1 Criteria

The items of major importance in selecting the proper number of engines and their arrangement include length effects on launch costs, performance and gross weight effects on launch costs, reliability and mission loss costs, and unit costs.

4.6.2 Concepts

Two or more engines provide high mission reliability and man ratability for an OTV. A two engine configuration is approximately six feet longer than either one, three or four engine concepts. The length penalty for a two engine OTV is caused by the requirement to operate with loss of one engine. The engine gimbal point must be shifted aft to get the thrust vector through the vehicle c.g. within the gimbal angle limits of approximately 20 degrees. Therefore, for high mission success probability where length may be a large discriminator in terms of launch costs, three or four engines may be attractive.

Figure 4.6.2-1 shows two and three in-line engine concepts along with several of their corresponding attributes.



- REQUIRES LARGE GIMBAL ANGLES FOR ENGINE OUT (20° - 30°)
- ALWAYS RESULTS IN LONGER STAGE THAN WITH 1, 3, OR 4 ENGINES
- MOST RELIABLE ENGINE CONFIGURATION

- SHORT LENGTH; APPROX. LENGTH OF 1 OR 4 ENGINE STAGES
- LARGE MISSION SUCCESS PROBABILITY (RELIABLITY) IMPROVEMENTS OVER 1 ENGINE
- NEGLIGIBLE DRY WEIGHT DIFFERENCE FROM TWO ENGINE CONCEPT

Figure 4.6.2-1 Two Vs Three Engine OTV Configurations

4.6.3 Assumptions

It was assumed that three in-line and four engine concepts will be the same length as a single engine vehicle concept in the calculation of launch costs. This is a reasonable assumption since the gimbal plane location for these concepts is in the same location and only the engine nozzle length will impact overall vehicle length. Three and four engine concepts will have higher dry weights than a single engine concept but this effect may be somewhat offset, since the majority of engine contractor data shows higher Isp for lower thrust engines when length is kept constant. However, in this assessment the Isp was held constant.

Engine-out capability for the two engine version is accomplished by gimballing the remaining engine approximately 20°. For the three in-line engine version, if the center engine goes out, the two outboard engines are still utilized. If one of the outboard engines goes out, the other is automatically shut down. The four engine version operates as two pairs. If one engine goes out, its opposite is shut down.

4.6.4 Assessments

Costs for launch, engine units, and mission losses have been calculated for OTV concepts incorporating one, two, three in-line, and four engines. The costs shown in Table 4.6.4-1 are relative to the single engine case which is used as a reference. The launch costs are for the 160 GEO missions in this Rev. 9 mission model.

The two engine concept length results in the launch costs being assessed primarily on a length basis. Two, three, and four engine concepts offer significant mission success improvement over the single engine vehicle concept and those benefits are shown here. The totals show that a three in-line engine concept offers the best cost compromise of the parameters shown here.

Table 4.6.4-1 Cost Comparisons for Ground-Based OTV Propulsion

	COSTS * BASED ON 160 MISSIONS			
# ENGINES	UNITS	LAUNCH	MISSION LOSS	TOTAL
1	REF.	REF.	REF.	REF.
2	+32	+532	-596	-32
3	+64	+204	-558	-290
4	+96	+252	-537	-189

* MILLIONS OF 1986 $

UNITS ARE $2M/ENGINE
10 MISSIONS/ENGINE
MISSION LOSS IS $160M

4.6.5 Recommendation

This study shows that a wide body ground-based OTV designed for launch in a large cargo vehicle should have 3 engines.

4.7 GB OTV VEHICLE/FLEET SIZING

A trade exists in determining what sizes of OTV are appropriate in optimally capturing the mission model. The large mission payloads require a propellant capacity of 74 Klbm. However, this large size vehicle may not be an efficient way of capturing the small payload missions. Therefore, several vehicle sizes and fleet types were examined to establish the optimum fleet.

4.7.1 Criteria

This study was based just on the 160 civilian payloads going to GEO in the Rev. 9 preliminary mission model. These payloads require a 74K OTV stage in 1999. The evaluation criteria was launch cost savings versus additional DDT&E and production costs.

4.7.2 Concepts

The design concepts all utilized the basic wide body, three engine OTV configuration with folding aerobrake. Linear scaling from the baseline 52K and 74K stages was used to obtain the basic design parameters shown in Table 4.7.2-1.

Table 4.7.2-1 Design Characteristics for Wide Body GB OTV Stages

PROPELLANT CAPACITY, 1000 LBS.	DIAMETER, FT	LENGTH, FT	DRY WEIGHT, LBS
74	24.5	25.5	8795
60	24.5	21.8	8085
52	24.5	19.7	7680
50	24.5	19.2	7579

4.7.3 Assumptions

Launch costs were calculated using the STS shared launch cost charging procedure, assuming the capabilities and costs specified for the large cargo vehicle. Return to earth costs were not assessed in this trade study since there will be no appreciable differences for the vehicle sizes considered.

4.7.4 Assessment

A 74 Klbm size propellant capacity captures the largest payload and is therefore required in all fleet candidates. From there, it is a matter of deciding whether or not to include smaller vehicles in the fleet. Also, the sizes of the smaller vehicle(s) had to be established.

Launch costs have been estimated for the fleet types shown in Table 4.7.4-1. The entire 160 flights being flown with a 74 Klbm stage is the most expensive and the mixed fleets result in reduced launch costs because of the use of a shorter and lighter vehicle.

Table 4.7.4-1 Fleet Candidate Launch Cost Comparisons

FLEET COMPOSITION (PROPELLANT CAPACITY)	FLIGHTS/STAGE	TOTAL LAUNCH COSTS	
		CONSTANT $	DISCOUNTED $
ALL MISSIONS USE 74K STG	160	$8.76B	$3.56B
74K 60K 50K	56 13 91	$8.32B	$3.29B
74K 52K	65 95	$8.36B	$3.31B

4.7.5 Sensitivity

The data shows that a fleet of three different size OTV's has slightly lower total launch costs than a fleet composed only of two vehicles. However, the DDT&E cost of a third vehicle will certainly be greater than the potential savings of $40M. A change in launch costs or in the manner of applying the shared launch cost algorithm will certainly require re-examining this trade study because the candidates are so close in total costs.

4.7.6 Recommendation

Two vehicle sizes of 74 Klbm and 52 Klbm propellant capacity have been chosen for the OTV fleet recommendation. The 74 Klbm size is obviously required for the larger mission payloads. The 52 Klbm size was selected as the IOC OTV size since the larger payloads don't appear for four years after IOC and because the smaller size OTV saves on length, gross weight, and thus launch costs. The small vehicle does not have to be man-rated.

4.8 ALTERNATIVE OTV OPTIONS

Two trade studies have addressed the issues of whether dedicated stages should be developed for capturing the low energy missions and the very high energy missions. Very small (micro) and very large (macro) vehicle design concepts have been defined and compared to the nominal size vehicle fleet (52 Klbm and 74 Klbm propellant capacity stages) in order to make comparisons and assess whether or not either of the dedicated stages are warranted.

4.8.1 Criteria

The baseline OTV fleet consists of two different size vehicle: 52K propellant stage for transportation up to 15K to GEO, and a 74K propellant stage for transporting up to 25K to GEO. The 74K stage is man-rated and capable of performing the 12K up, 10K down manned GEO missions.

All DOD missions could be performed with a 40K propellant OTV stage. Many of the lunar and planetary missions require multiple stages and/or propellant tank sets along with the basic 74K OTV.

The criteria used to evaluate the large and small OTV options is whether or not the reduced launch and onorbit assembly operations costs are adequate to offset the additional DDT&E and production costs.

4.8.2 Concepts

Mission performance requirements analyses and scaling of the baseline OTV designs resulted in the two alternative OTV design concepts summarized in Table 4.8.2-1.

Table 4.8.2-1 Alternative OTV Design Options

FEATURE		MICRO-OTV	MACRO-OTV
Propellant Capacity, lbs		40,000	240,000
Dry Weight, lbs		7,200	17,750
Overall Length, ft		16.5	
Production & DDT&E Costs	(1985 $)	$209M	$570M
Baseline Number of Missions		240	14

4.8.3 Assumptions

The micro OTV (40K propellant) was assumed to be ground-based and LCV launched. It would be utilized for all DOD missions (240 in Scenario 2,3 and 5 and 480 missions in Scenario 4).

The macro OTV (240 K propellant) was assumed to be space-based and consequently was assessed an additional cost of $52M for expansion of Space Station accommodations (hangar, tank farm, and enlarged robotics and checkout systems).

4.8.4 Assessment

4.8.4.1 Micro OTV

The launch costs associated with performing the 240 DOD missions in Scenario 2 with the ground-based 52K stage OTV (propellant off-loaded to reduce launch weights as required) are approximately $8.9B in constant $ 1985. The total LCC is $11.2B. The micro OTV will reduce launch costs. The relative LCC costs are shown in Figure 4.8.4-1. Costs incurred prior to 1995 are the additional DDT&E and production associated with developing the small stage. The reduced launch costs do show a payback of the constant $ 1985 in 2004 for Scenario 2 and in mid-2000 for Scenario 4.

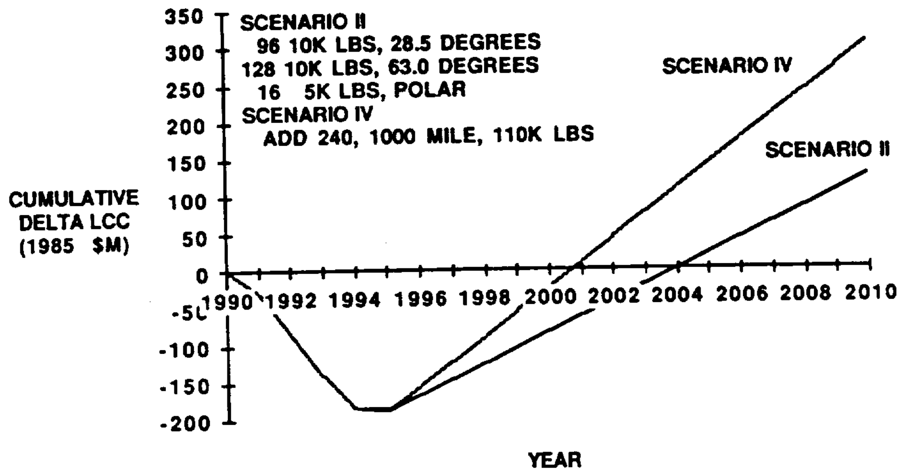


Figure 4.8.4-1 Cost Comparison, 40K vs 52K OTV for DOD Missions (Constant $ 1985)

The discounted cost comparison is shown in Figure 4.8.4-2. As shown, there is no payback for Scenario 2, and Scenario 4 does not recover its costs until the year 2008.

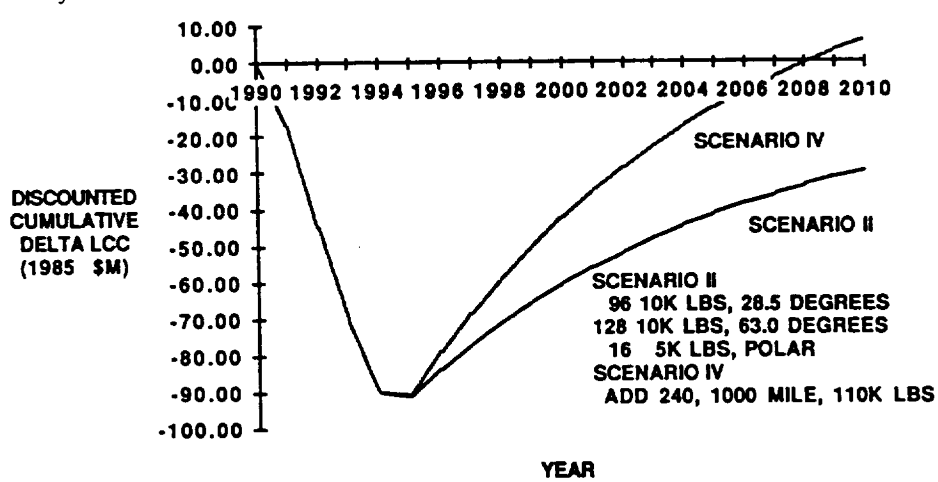


Figure 4.8.4-2 Cost Comparison, 40K vs 52K OTV for DOD Missions (Discounted $ 1985)

4.8.4.2 Macro OTV

This trade determined whether or not a dedicated large stage is justified for capturing the high energy missions. These missions and the number of times they are flown in each of the Scenarios are identified in Table 4.8.4-1.

Table 4.8.4-1 Rev. 9 High Energy Missions

Rev. 9 Missions Number	Scenario 1	2	3	4	5
17088 (planetary)	1	1	1	1	1
17101 (planetary)	0	0	0	0	1
17202 (Lunar)	0	1	1	1	1
17203 (Lunar)	0	4	4	4	1
17206 (Lunar)	0	0	0	0	1
17207 (Lunar)	0	0	0	0	8
16029 (GEO)	0	0	0	0	1
TOTALS	1	6	6	6	14

The first six columns of Table 4.8.4-2 show how these missions would be performed without a large OTV. As shown, expendable kick stages, expendable OTV's and OTV tank sets, and up to 4 stages of OTVs are utilized to perform the missions. The 150K LCV requires 17 launches to perform the 14 missions of Scenario 5. The seventh column of the table shows the propellant required by the 240K OTV to accomplish the missions without multiple tank sets or OTV staging.

Table 4.8.4-2 High Energy Mission Performance Summary

PAYLOAD	PAYLOAD UP (lbm)	EKS WT (lbm)	PROP. REQ'D (lbm)	OTV PROP. CAP. (lbm)	TANK* SET PROP. CAP.	PROP. REQ.-240,000 lbm PROP. CAP. OTV
PLANETARY						
17088	19945	22,235	141,168	74K **	74K	228,100
17101	44100	0	118,401	74K **	52K	114,500
LUNAR						
17202	32850	0	89,992	52K	52K	104,800
17203	72680	0	158,098	2-74K	52K	173,300
17206	93000	0	215,617	2-74K	74K	212,800
17207	72680 (20,000 dn)	0	179,686	2-74K	52K	189,500
TO GEO						
16029	100,000 (4 x 25,000)	0	4 x 69598	4 x 74K	0	234,300 (SINGLE DELIV)

* TANK SET IS ATTACHED TO FIRST STAGE

** OTV IS EXPENDED

Figure 4.8.4-3 shows the LCC cost difference between performing the very large missions with a dedicated large stage versus various staging concepts utilizing the baseline OTVs and tank sets. The data is based on the traffic of Rev. 9, Scenario 5 mission model. The first requirement for the large stage is in the year 1999 (Payload 17088) for all of the Rev. 9 Scenarios. The obvious conclusion is that the investment in the large stage cannot be recovered prior to the mission model cutoff date of 2010.

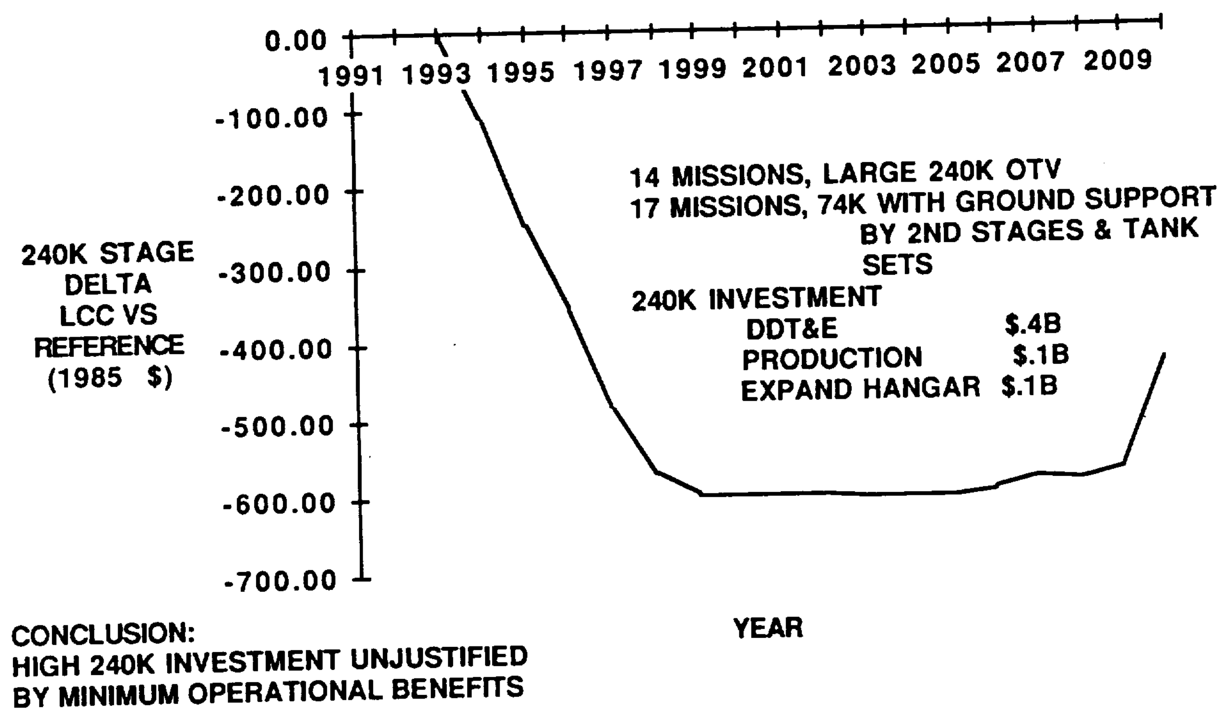


Figure 4.8.4-3 LCC Comparison - Large OTV Versus Staging Baseline OTVs

4.8.5 Sensitivity

4.8.5.1 Micro OTV

The data presented in Section 4.8.4 showed the Sensitivity to Scenario 2 and 4. If launch costs are substantially higher than the projected $70M for the large cargo vehicle, both scenarios would show an increased cost advantage for the 40K OTV.

4.8.5.2 Macro OTV

This study was performed using the Scenario 5 missions because the total quantity of large missions is so much greater than in Scenario 2. If the study were performed using Scenario 2 mission, the stage size could be reduced from 240K to 230K and the number of missions reduced from 14 to 6. The result would be that there is essentially no change in the fixed costs, while the number of missions that would benefit from the large stage and therefore recover these costs is reduced. Scenario 5 LCC comparisons clearly show that a large stage is not warranted; other scenarios only make the large stage even more unattractive.

4.8.6 Recommendation

A small OTV can be recommended if the DOD traffic is as shown in Scenario 4. At all other scenario traffic levels, off-loading propellant from the 52K OTV stage is more cost effective than designing a new optimized stage.

Even the most ambitious traffic model scenario cannot justify a large size OTV within the constraints of the Rev. 9 mission model which ends in the year 2010.

4.9 GROUND-BASED VS SPACE-BASED TRADE

The objective of the ground-based vs space-based trade was to determine the optimum OTV basing mode for a reusable hardware configuration. In addition, extensive sensitivities to key programmatic inputs were performed to provide a comprehensive set of "what if" scenarios to the reference ground-based/space-based conditions.

In this section the Large Cargo Vehicle is designated as the Unmanned, Partially Reuseable Cargo Vehicle (UPRCV) to emphasize that it does not have the capability to return an OTV to earth. A LCV that has return capability will show an added cost advantage for space-basing since the ground-based OTV costs associated with LCV return are slightly greater than the costs associated with STS return.

4.9.1 Criteria

The requirements for the trade were based on the 160 GEO civil missions identified in Section 3.1.1. UPRCV delivery and STS/STS II ground-based return were employed. Total LCC and total discounted LCC were the prime discriminators between the two concept programs. Second order discriminators other than cost include launch and transfer logistics, mission flexibility and technology advances.

4.9.2 Concepts

The two program concepts are profiled in Figure 4.9.2-1. The reference ground-based program maintains an evolutionary approach to stage development. The 1995 IOC program begins with a 52 klb stage and evolves to a mixed fleet environment in 1996 with the development of a 74 klb stage. As alluded to previously, delivery of payload and stage to LEO is accomplished with the STAS UPRCV. Return to launch site of the ground-based stages is performed by the current STS through 2001. Beginning in 2002 through the end of the analysis timeframe, the STS II performs the return function. Due to the wide diameter of the stages, some tankage and the aeroassist device are expended before the return to launch site (Figure 4.9.2-1).

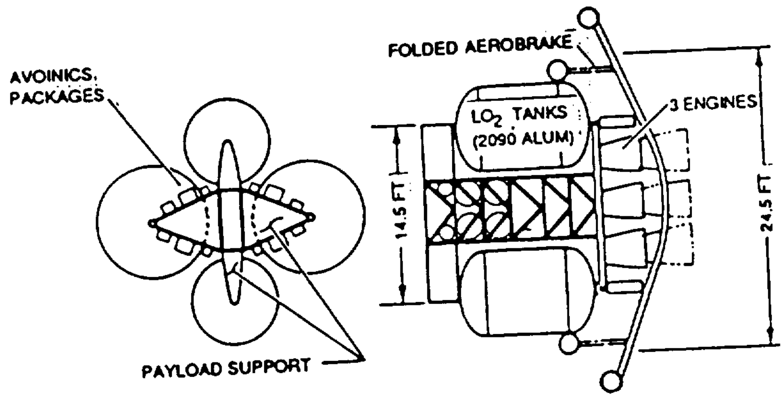
The space-based alternative actually consists of a combined ground-based/space-based capability. The program begins with a 1995 IOC 52 klb ground-based stage. The five GEO civil missions occurring in that year utilize this stage. In 1996 as Space Station accommodations become available, all GEO civil missions are flown in a space-based mode (155 missions through 2010). The 52 klb ground-based stage becomes the dedicated workhorse of the DOD payloads as well as providing limited support to certain lunar and planetary missions. These later missions are not included in this trade.

4.9.3 Ground Rules and Assumptions

The ground rules and assumptions governing the basing trade are consistent with the detailed study ground rules included in Section 8.0. Certain clarifications/additional emphasis to these ground rules are as follows:

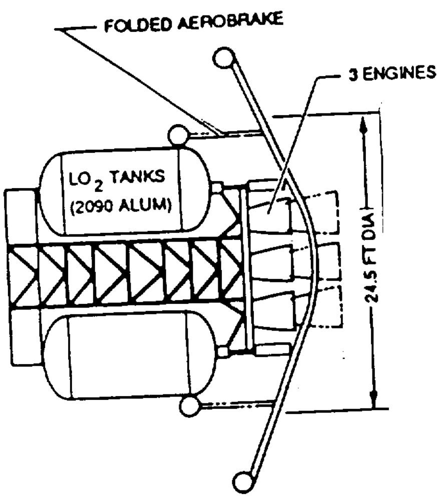
A) The reference ground based return charges (per study ground rules) include the minimum user charge for delivery of return ASE on STS/STS II. The return vehicle is assumed available when required. Disassembly of tanks/stowage IVA is comparable to STS/ACC OTV timelines.

B) Space station accommodations requirements are consistent with those described in Section 7.



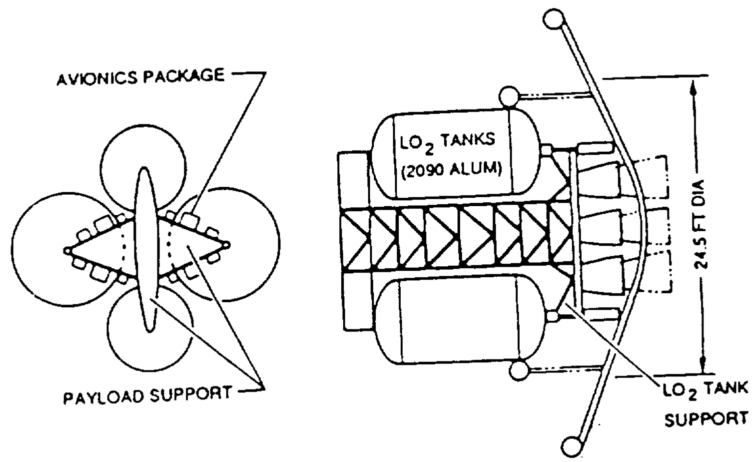
52K GBOTV

DRY WT.	7680 Lbs
LENGTH	19.7 Ft.
TANKAGE EXPENDED	2 LH.
MISSIONS(GB-GB/SB)	95 - 5 (PLUS DOD)



74K GBOTV

DRY WT.	8795 Lbs
LENGTH	25.5 Ft.
TANKAGE EXPENDED	2 LH, 1 LOX
MISSIONS	65



74K SBOTV

DRY WT.	9007 Lbs
LENGTH	25.5 Ft
AIRFRAME DEL. LENGTH	25.5 Ft.
BRAKE DEL. LENGTH	20.0 Ft.
TANKAGE EXPENDED	NONE
MISSIONS	155

Figure 4.9.2-1 Reference GBOTV - GBOTV/SBOTV Characteristics

C) Manifesting of stage hardware and payloads is consistent with length/weight user charge algorithm. Ground-based stage and payload are priced as an integral payload unit.

D) Space-based propellant delivery is performed by a combination of "propellant hitchhiking" (63% of total) and dedicated tanker flights. Section 2.1 contains a discussion of these concepts.

4.9.4 Assessments

As mentioned, the reference GBOTV vs GBOTV/SBOTV basing results are based on the STAS Scenario 2 160 GEO civil missions. The DOD missions for the reference programs are delivered via a GBOTV due to a potential security concern of processing these payloads through the international Space Station. Potentially some of these missions could be serviced by a SBOTV at a lower operational cost if certain security considerations could be alleviated. Additionally, since the DOD payloads were manifested on a weight basis only and demand less performance, the economic advantages of space-basing are not as large.

Figure 4.9.4-1 profiles the LCC of the two candidates by major program phases. The costs are shown for R&T, IOC stage DDT&E, evolutionary stage DDT&E, Space Station accommodations, initial production, launch costs, stage hardware and miscellaneous operations. The two programs are nearly identical through IOC stage DDT&E (R&T $0.2B, DDT&E $1.1B). At this point the impacts of the more sophisticated space-based stage and Space Station accommodations acquisition increase the GBOTV/SBOTV program costs over the GBOTV program by $0.5B (evolutionary stage DDT&E $0.3B vs $0.2B, accommodations $0.4B). Initial production costs for the 74 klb SBOTV are slightly higher than those of the 74 klb GBOTV.

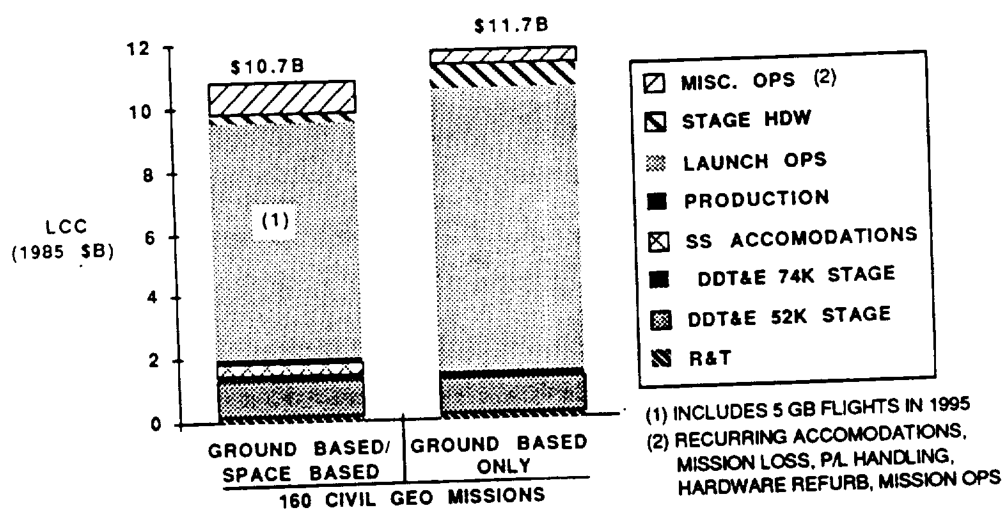


Figure 4.9.4-1 GBOTV-GBOTV/SBOTV Comparison By LCC Phase

Major program costs deltas occur throughout the operational cost elements. Table 4.9.4-1 includes the total top level operations and CPF (Cost per Flight) estimates for the two programs. The SBOTV has a large economic advantage in lower launch costs. This is primarily due to the economic advantage of propellant hitchhiking. The savings are less than they might be because SBOTV payload delivery is always charged on a length basis, whereas, the relative payload delivery costs of the GBOTV are approximately 50% weight charged. This results in a lower overall payload delivery cost for the GBOTV. The payload launch cost difference is best seen by a specific example.

The GBOTV launch cost for the 15 ft 12 klb geoshack logistics payload and stage is $54M. This figure is derived from a GLOW of stage and payload of 87.4 klb (58% of UPRCV performance) versus a gross liftoff length (GLOL) of 42.5 ft (47% of the 90 ft payload envelope). Weight is the maximum constraint. After applying the 75% user charge factor, the launch cost for this mission is calculated at 78% of the UPRCV CPF. The SBOTV launch cost of the payload only is $16M. This figure is based on 15 ft of payload envelope length (17% of the UPRCV capability) versus 12 klb of performance (11% of the UPRCV performance which is 110 klb to the Space Station's altitude). The launch cost for this payload is 22% of the UPRCV CPF. The effective cost per pound to LEO for the payload only is $620/lb for GBOTV delivery ($54.4M/87.4 klb * 12 klb) and $1300/lb for SBOTV delivery ($15.6/12 klb). Section 4.9.5.2.3 provides a sensitivity trade to payload dimensions that normalizes payloads to the 25 ft UPRCV diameter.

Total launch costs for the SBOTV are $7.0B which includes $0.8B for spares delivery, $3.1B for propellant transportation, $3.1B for payload processing/transportation and $0.3B for 1995 GBOTV missions. Total launch costs for the 160 GBOTV missions are $8.8B. This includes $8.3B for delivery of GBOTV stage and payload and $0.5B for GBOTV return from LEO.

Table 4.9.4-1 GBOTV vs GBOTV/SBOTV Operations Cost Comparison

	GBOTV		GBOTV/SBOTV			
	52 & 74 klb Composite (160 Missions)		52 klb GBOTV (5 missions)		74 klb SBOTV (155 missions)	
	Operations	CPF	Operations	CPF	Operations	CPF
Stage Operations	1,070	6.7	40	8.1	559	3.6
Launch/GB Return	8,850	55.3	256	51.2	776	5.0
Propellant	18	0.1	1	0.1	3,075	19.8
SS Accommodations	-	-	-	-	607	3.9
Payload Transportation/Processing[1]	18	0.1	1	0.2	3,137	20.2
Program Support	190	1.2	6	1.3	181	1.2
TOTALS	10,146	63.4	304	60.9	8,335	53.8
	10,146		8,639			

1 GBOTV includes ground processing of payloads only

The impacts of the other two major operational cost groupings nearly offset each other. SBOTV stage hardware ($0.4B) costs are considerably less than that of GBOTV hardware requirements due to the partially expendable aeroassist and expendable tankage of the GBOTV. This SBOTV saving is offset by stage turnaround operations costs (SBOTV at $1.0B vs GBOTV at $0.4B). This impact is caused by SBOTV refurb/accommodations recurring cost differences.

Total LCC savings provided by the GBOTV/SBOTV program are $1.0B for the reference program analyses as was shown in Figure 4.9.4-1.

Discounted LCC is shown in Figure 4.9.4-2 with slightly different cost element groupings. The chart illustrates the impact of the high front end cost requirements of the SBOTV program as compared to ground-basing. This front end penalty is offset by lower operations cost for the SBOTV resulting in a discounted LCC of $2.9B for both the ground and space-based programs.

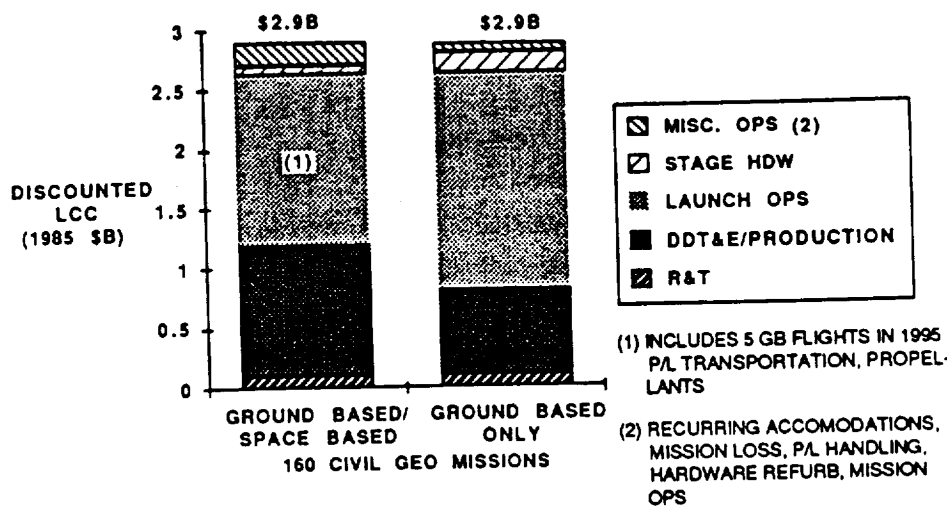


Figure 4.9.4-2 GBOTV-GBOTV/SBOTV Comparison By Discounted LCC Phase

4.9.5 Sensitivities

An extensive set of sensitivities were performed within the context of the basing trade. The sensitivities were designed to address two issues. First, a series of subtrades were performed in order to determine the preferred characteristics of the SBOTV program. A second set of subtrades were performed to allow visibility to the sensitivity of the basing trade to key programmatic inputs.

4.9.5.1 Preferred SBOTV Program

Three subtrades were conducted to answer three basic questions regarding the characteristics of the SBOTV program. The first of these addresses SBOTV IOC sensitivities, the second looks at the economic benefits of a "clean sheet" (optimally designed) versus an evolutionary design approach to SBOTV, and the third issue determines fleet size impacts of SBOTV.

4.9.5.1.1 SBOTV IOC Decision

The decision of when to implement a SBOTV capability is dictated by program requirements combined with an economic justification. Within the reference civil GEO missions, the HF direct broadcast payload with an IOC of 1996 requires a large stage delivery mode. This payload could potentially be split in two which would allow 52 klb stage utilization and postpone the large stage IOC to 1999. The other key program requirement (ground rule) allows a SBOTV IOC in 1996. The economic decision can be based on trading the potential economic advantages of space-based operations beginning in 1996 against deferring onorbit accommodations acquisition spending for a 1999 SBOTV IOC. The impacts of the early IOC penalty can be compared against lower SBOTV operations costs (versus ground-basing) from 1996 through 1998. Discounted LCC provides a valid means of comparison.

Figure 4.9.5-1 shows the cumulative LCC of the two approaches to SBOTV IOC up through 1999. By 1999, the early space-basing IOC has recovered from the early investment penalty of the 1996 IOC and shows a $0.2B LCC advantage. After 1999, the two approaches to space-basing are identical so the charts are truncated.

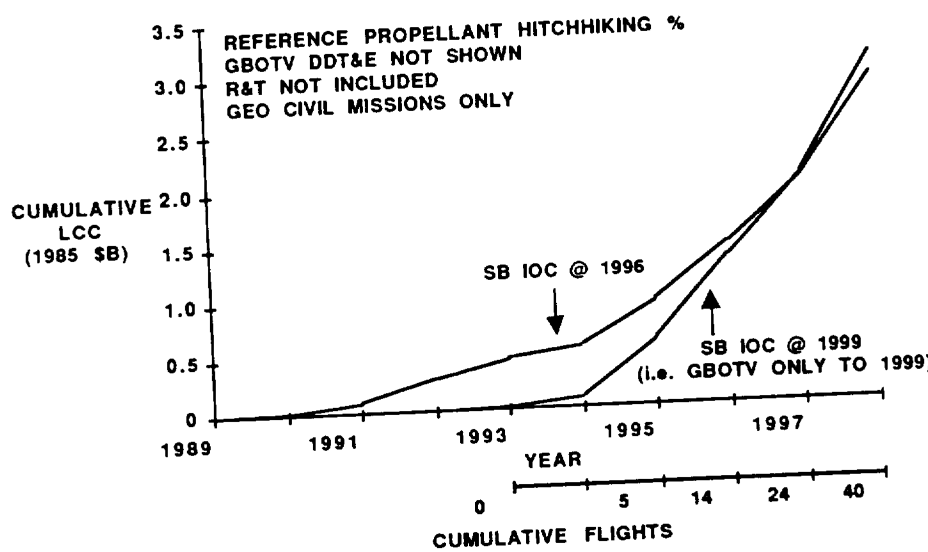


Figure 4.9.5-1 SBOTV IOC Sensitivity, Cumulative LCC (1985 $B)

Figure 4.9.5-2 displays the impacts of an early SBOTV IOC versus delaying accommodations spending by servicing the 1996 through 1998 missions ground-based. The results show the delta program cost in discounted 1985 dollars. The penalty for a 1999 SBOTV is clearly shown up through 1994. Beginning in 1995, two factors contribute to its rapid recovery. First, while early SBOTV IOC accommodations/stage spending is winding down accommodations/stage spending for the 1999 IOC candidate is building up. Second, lower cost SBOTV operations versus a GBOTV from 1996 through 1998 contributes to program savings. The combined effect of these two factors shows that the penalty for early space-basing is minimal (less than $50M discounted 1985 dollars). A net LCC savings for the early IOC is actually $0.2B. This suggests a SBOTV capability should be acquired as early as possible to make maximum use of its economic benefits.

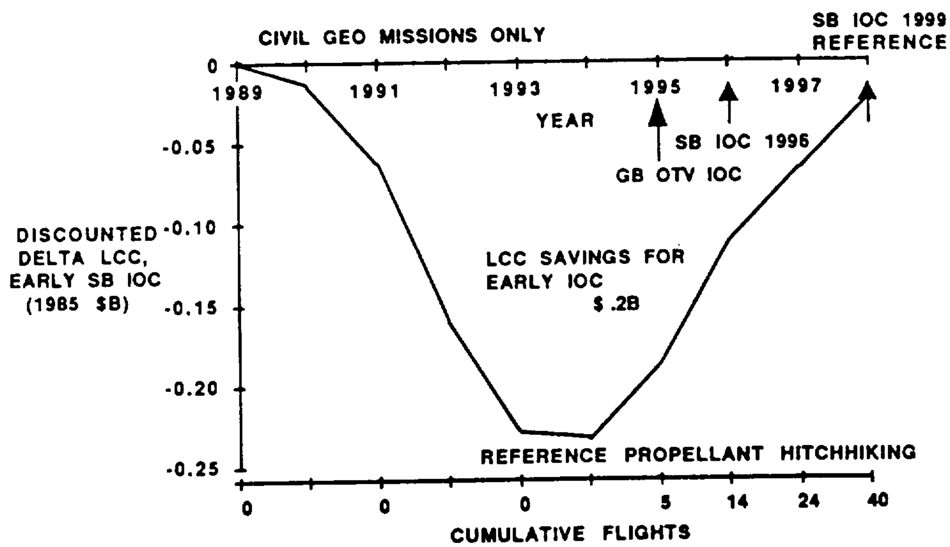


Figure 4.9.5-2 SBOTV IOC Sensitivity, Delta Discounted LCC (1985 $B)

4.9.5.1.2 GBOTV To SBOTV Evolutionary Growth Advantages

The economic viability of a SBOTV program relies on a relatively low cost per pound to LEO of propellant combined with efficient onorbit performance and turnaround stage characteristics. Minimizing the space-based SBOTV propellant requirements through optimization of stage design with respect to performance can be vital to SBOTV economics. However, performance gains through optimum design can provide diminishing returns when front end investment and operations logistics (e.g., spares delivery, turnaround time) are considered. This is especially true if GBOTV development precedes the development of the space-based stage and the space-based stage is not efficiently designed with regard to support launch vehicle delivery constraints.

Figure 4.9.5-3 provides the top level stage characteristics of two space-based stages. Both stages are preceded by a 52 klb GBOTV designed for delivery in the UPRCV. The clean sheet SBOTV is a 74 klb, four ball, tankage concept that was initially designed for space-basing within current STS capabilities. The assembled diameter is greater than 25 ft with a deployed aerobrake diameter of 44 ft. The dry weight is 8378 lb.

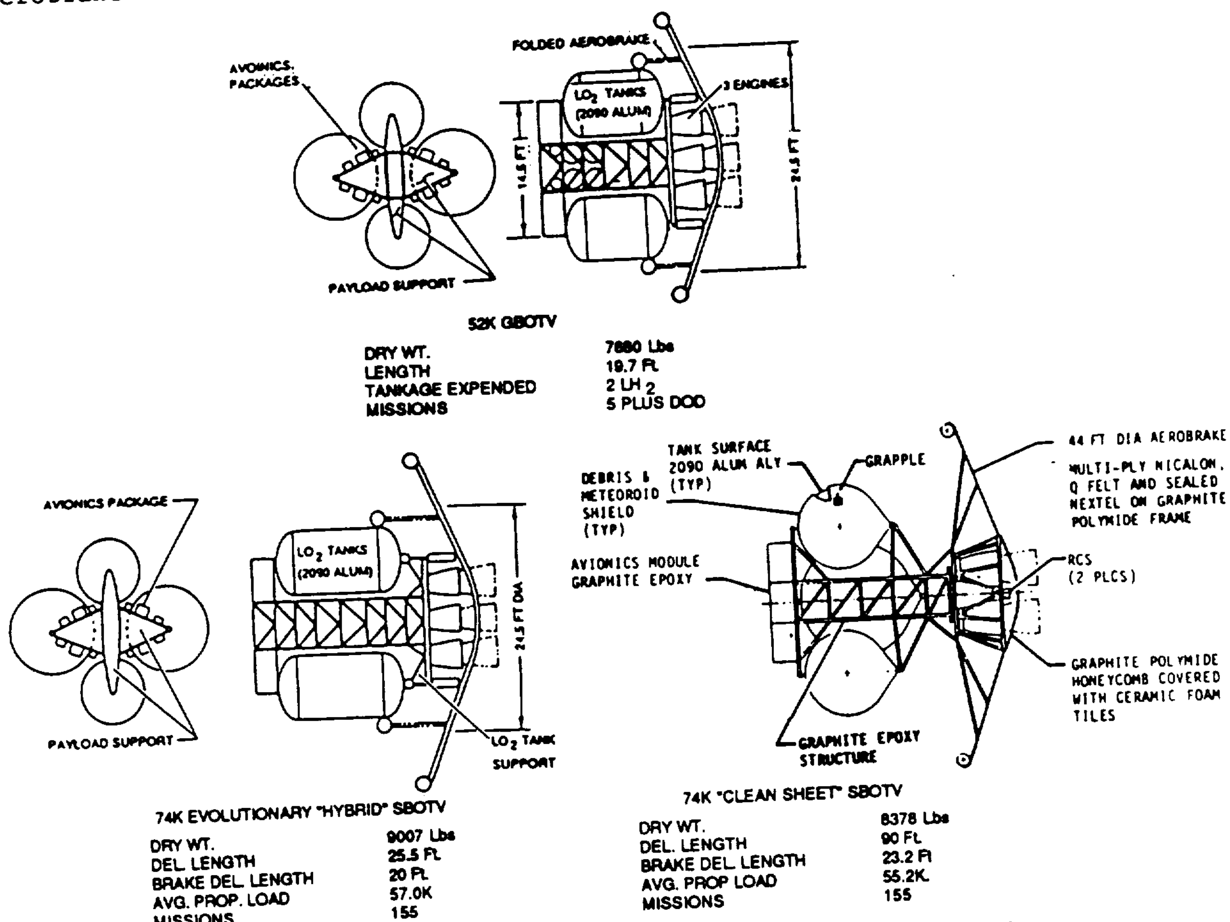


Figure 4.9.5-3 Growth Path SBOTV Stage Characteristics

The hybrid 74 klb SBOTV is an upscaled version of the 52 klb GBOTV. The only major differences (besides resizing) are increased meteroid protection and the addition of quick disconnects for onorbit engine changeout. The aerobrake is 38 ft diameter (deployed) while the stage delivery diameter is 25 ft. The stage dry weight is 9007 lb. Figure 4.9.5-4 shows the cumulative LCC of the evolutionary approaches to space-basing. For two different propellant costs per pound, the chart shows that the overall LCC difference between the hybrid and clean sheet options is minimal although slightly less for the hybrid approach. The following discussion expands on the cost drivers behind the delta costs.

Figures 4.9.5-5 and 4.9.5-6 show the delta constant and discounted LCC for the "clean sheet" approach. The evolutionary program serves as reference. The delta stage DDT&E of $0.2B is the result of major evolutionary design changes in structure/tankage, propulsion and aeroassist subsystems in going from the 52 klb GBOTV to the 74 klb "clean sheet" SBOTV. Operations costs are shown for two different low cost propellant capture ratios in order to emphasize the impact of this key programmatic discriminator. Note that the propellant savings of the "clean sheet" SBOTV become more apparent as propellant costs rise. The contributing factors towards keeping the clean sheet slope negative despite the propellant savings are the delivery costs of spare aerobrakes and airframes (see Length, Figure 4.9.5-3). In spite of this, the 282 klb of propellant saved would not be substantial enough to offset the additional DDT&E expenditures in constant LCC let alone after discounting.

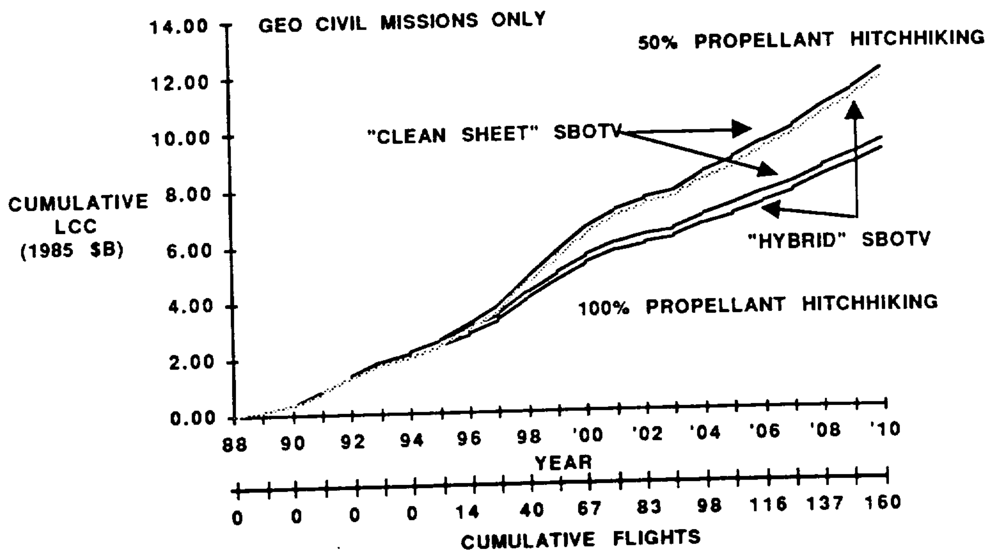


Figure 4.9.5-4 Clean Sheet vs Hybrid SBOTV Cumulative LCC (1985 $B)

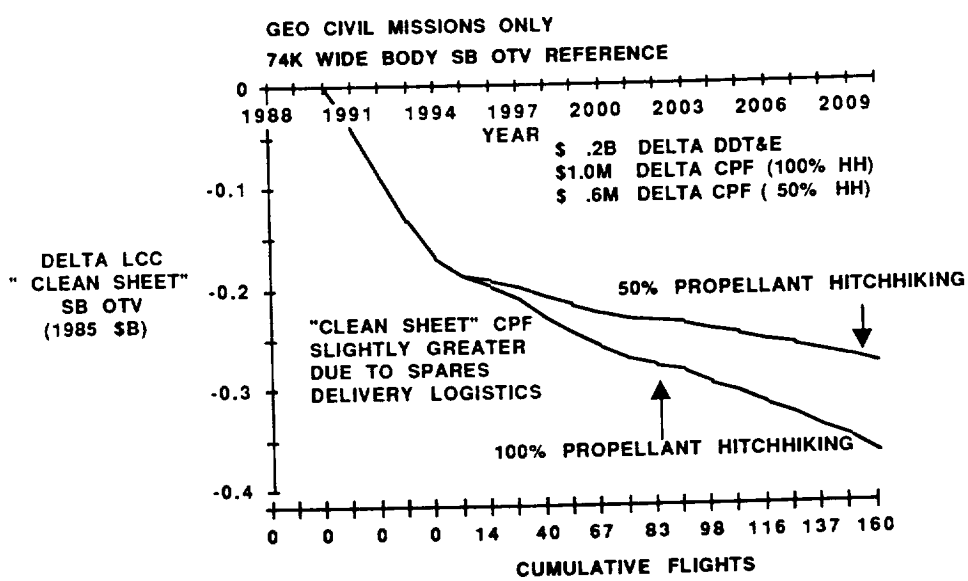


Figure 4.9.5-5 Clean Sheet vs Hybrid SBOTV LCC Comparison (1985 $B)

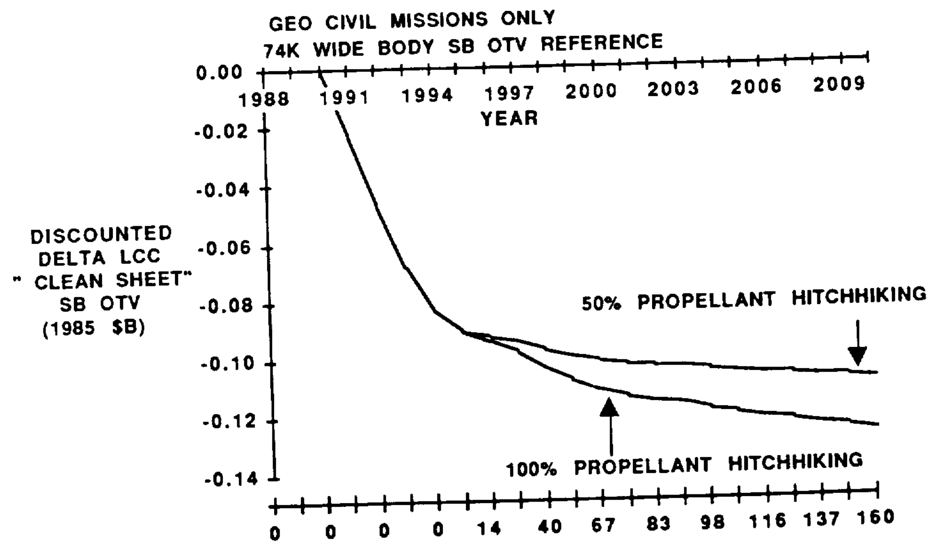


Figure 4.9.5-6 Clean Sheet vs Hybrid SBOTV Discounted Delta LCC

Although the clean sheet SBOTV is not optimally designed for delivery in the UPRCV, the trends show that within a mixed fleet OTV program, major design differences between the ground and space based stages must be significantly justified by performance gains. If significant performance gains coupled with high propellant costs are not present, the evolutionary approach to a GBOTV/SBOTV program is preferred.

4.9.5.1.3 Multi-Fleet Size SBOTV Program

An analysis was performed to determine the economic benefits of providing two different sized stages at space station to determine the reduction in propellant requirements and potential LCC savings. The analysis considered only the operational benefits of implementing a 52 klb SBOTV in combination with the reference 74 klb SBOTV.

The major steps of the trade are outlined in Table 4.9.5-1. The approach to the trade is very simplistic. The 52 klb SBOTV would save approximately 1,115 lb of dry weight over that of the 74 klb SBOTV. This provides an equivalent performance gain for the average mission of approximately 3.3 klb of propellant over the heavier 74 klb SBOTV. Of the 155 GEO civil missions, the 52 klb stage could capture 90 missions which translates to a $90M LCC savings. This savings does not justify the stage upgrade or the accommodations impacts especially when discounted dollars serve as the decision criteria. If propellant costs were to rise to $750/lb (The equivalent UPRCV tanker $/lb) (see Section 8.1.7.5) operational cost savings would still at best just begin to offset the stage and accommodations impacts of two sizes of SBOTV deployed at space station simultaneously. The conclusion is that a small 52 klb SBOTV is not economically justified.

Table 4.9.5-1 Multiple SBOTV Stage Sizes Cost Trade

- o 52 klb OTV approximately 1,115 lb lighter than 74 klb OTV.
- o One pound dry weight requires 3 lb propellant for average GEO mission.
- o Propellant costs = $300/lb.
- o A small OTV could fly 90 of the 155 space-based GEO missions.
- o (3 lb propellant/lb dry weight) x (1,115 lb dry weight) x ($300/lb) x (90 flights) = $90M.
- o $90M when discounted = $30M.
- o This potential savings will not pay for the development of another stage, larger hangar, extra spares, and robotic software modification.
- o Conclusion - Do not develop a 52 klb OTV for space-basing.

4.9.5.2 Basing Sensitivities

Since the basing decision (detailed in Section 4.9.4) depends on a number of uncertain variables, cost sensitivities were performed with regard to key programmatic inputs. The sensitivities include:

A) Space Station accommodations investment/operations;

B) Percentage of SB propellant requirements supplied by low cost means;

C) Basing mode effect on launch vehicle reusable hardware;

D) Launch vehicle cost per flight;

E) GBOTV return from LEO; and

F) Payload manifesting: length vs volume.

In order to maintain a manageable size for the data, a number of the sensitivities will be presented within a single subsection (A, B & C).

4.9.5.2.1 Basing Sensitivity To Accommodations/Propellant and Launch Vehicle Hardware

Any economic advantage of space-basing relies heavily on minimizing Space Station accommodations investment/operations while maintaining low cost methods of delivering propellant to LEO. Major swings in the impacts of these inputs may drastically change basing considerations. A large number of LCC calculations were performed based on the reference GBOTV/SBOTV program outlined in Section 4.9.4 and detailed in Section 8. These calculations include:

A) Accommodations cost growth to more than 200%;

B) Low cost propellant capture (propellant hitchhiking) from 0% to 100% of the program requirements (versus dedicated UPRCV tankers).

In addition, each of these cases is shown without and with potential launch vehicle reusable hardware advantages provided by space-basing. These advantages are derived by determining the equivalent launch vehicle flights delta between the GBOTV and GBOTV/SBOTV programs. The delta flights are then translated into delta service life on the UPRCV booster configuration (based on the Martin Marietta STAS contract service life at 200 flights). The service life savings ratio to total service life is then expressed in terms of hardware unit costs. Equivalent flights for GBOTV is based on the sum of the fractional launch vehicle use for 155 deliveries. Equivalent flights for the SBOTV is the sum of the fractional launch vehicle flights used for dedicated tanker flights and spares delivery. Fractional launch vehicle use for each mission is based on the dominating manifesting constraint, either weight or volume.

Figures 4.9.5-7 through 4.9.5-10 provide cumulative LCC and discounted cost data at the reference Space Station accommodations costs (100%) for varying low cost propellant capture ratios, with and without launch vehicle benefits. Constant dollar payback of SBOTV investments occur within the GEO civil mission model for low cost propellant capture of 50 to 100% of the total required. As the low cost propellant capture ratio goes to 0%, the SBOTV cost curve slope becomes slightly greater than that of the GBOTV and thus diverges. Potential launch vehicle benefits have a minor impact on crossover points. As propellant capture ratios decrease to 0%, the ground and space-based launch vehicle use is nearly identical.

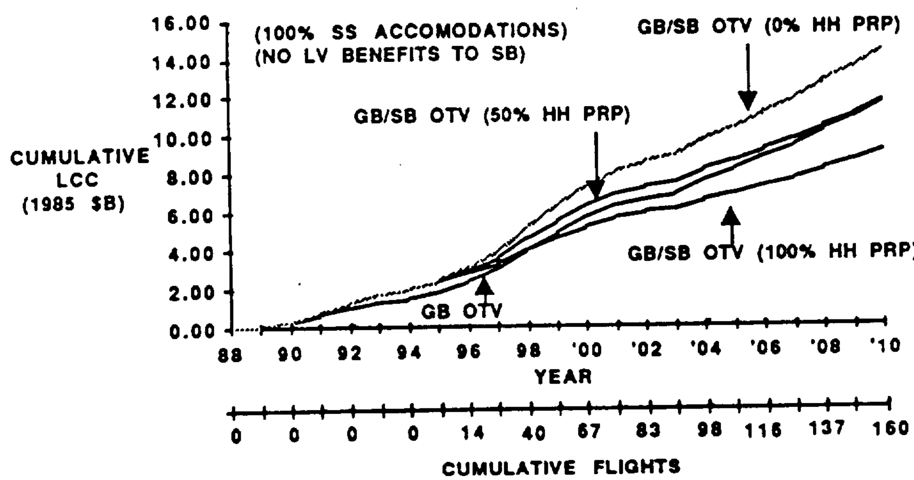


Figure 4.9.5-7 OTV Basing Sensitivity, 100% Accommodations (No LV Benefit) (1985 $B)

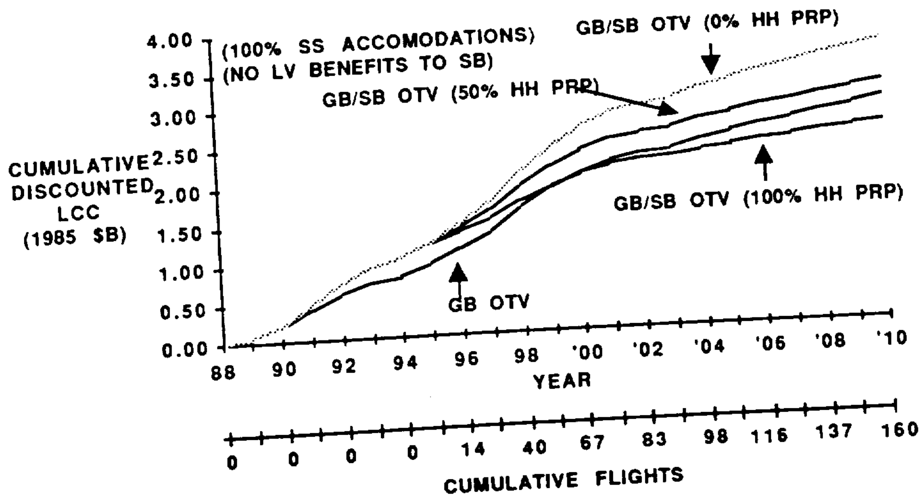


Figure 4.9.5-8 OTV Basing, 100% Accommodations (No LV Benefit) (1985 Discounted $B)

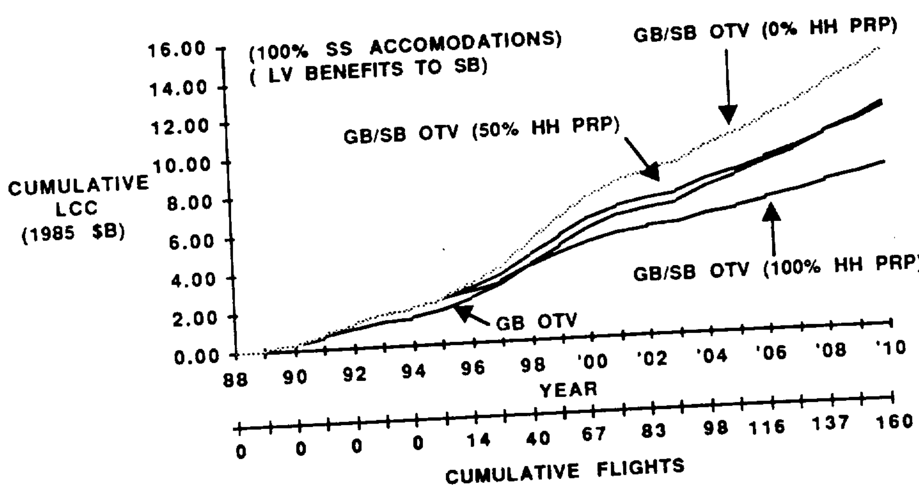


Figure 4.9.5-9 OTV Basing Sensitivity, 100% Accommodations (LV Benefit) (1985 $B)

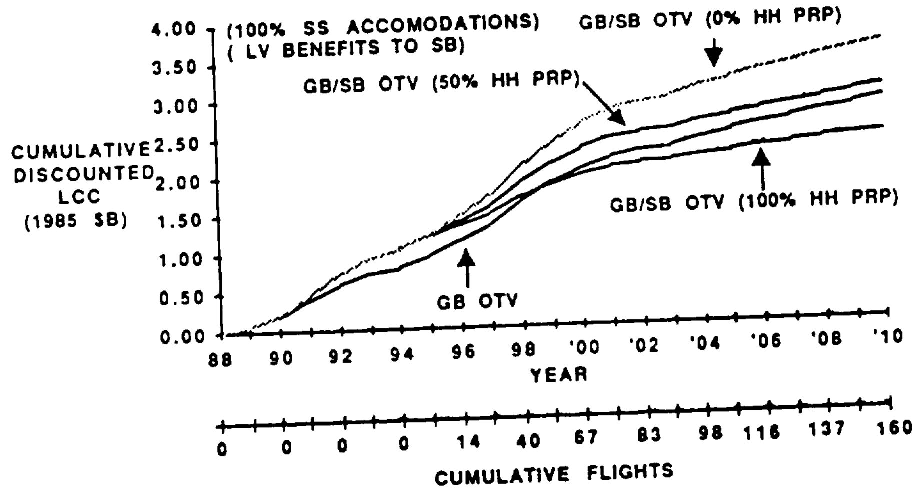


Figure 4.9.5-10 OTV Basing, 100% Accommodations (LV Benefit) (1985 Discounted $B)

Cumulative discounted costs show that only at high level of low cost propellant capture ratios is the space-based investment paid off by lower operations costs. Over 60% of propellants must be delivered via hitch-hiking to realize a discounted LCC payback within the 160 GEO civil missions.

Figures 4.9.5-11 through 4.9.5-14 provide cumulative LCC and discounted LCC cost profiles for similar propellant/launch vehicle cost conditions, but include over 200% growth in nonrecurring/recurring Space Station accommodations costs. The required percentage of SBOTV low cost propellant capture increases by approximately 13% over the above cases in order to provide a SBOTV payback within the GEO civil mission model. Discounting of program costs causes SBOTV payback to require an almost 100% supply of low cost propellant.

This series of sensitivities shows that within the GEO civil mission model and under reference program conditions, at least 50% of on orbit propellant requirements must be met by low cost delivery methods to achieve SBOTV program payback. The higher investment cost of space-based accommodations versus a totally ground-based program increase this requirement to over 60% in discounted dollars. If significant cost growth in SBOTV accommodations investment/turnaround costs occur, the required capture ratios increase by as much as 25%.

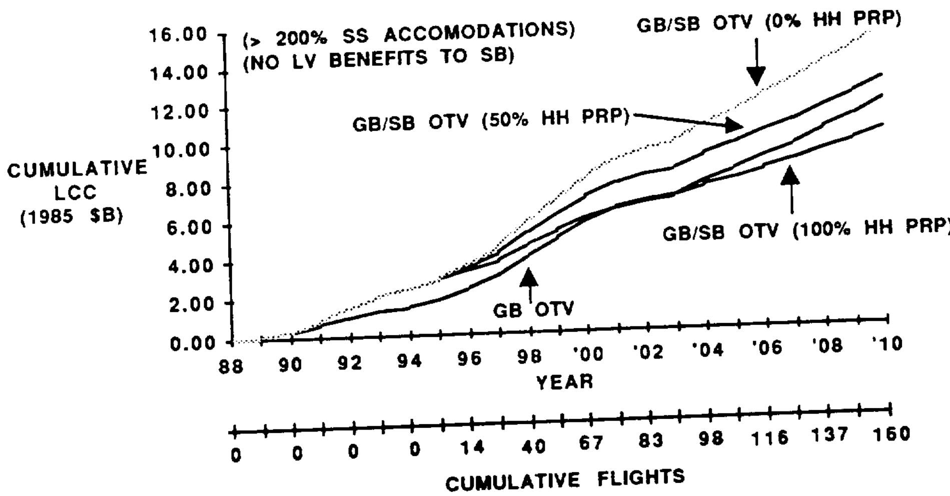


Figure 4.9.5-11 OTV Basing Sensitivity, Growth Accommodations (No LV Benefit) (1985 $B)

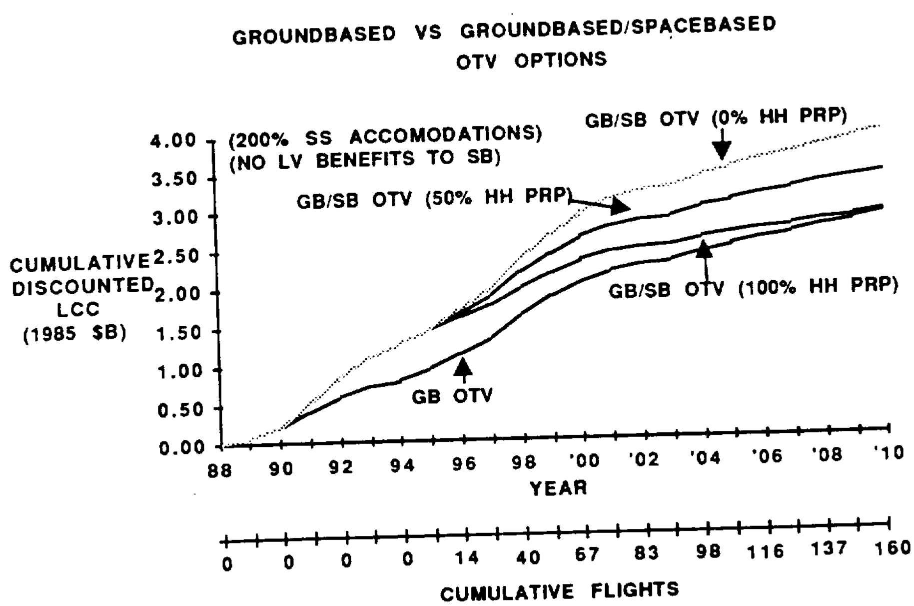


Figure 4.9.5-12 OTV Basing, Growth Accommodations (No LV Benefit) (1985 Discounted $B)

4.9.5.2.2 Basing Sensitivity to UPRCV CPF

The objective of the UPRCV CPF sensitivity analysis was to determine the relative SBOTV payback boundaries as a function of ground ruled UPRCV CPF expectations. To perform this trade the reference UPRCV CPF input of $70M/flight was varied from $50M to $85M per flight. In terms of economic impact, the GBOTV launch cost for each mission was influenced accordingly. The SBOTV impacts include tanker propellant, payload and spares delivery.

Figures 4.9.5-15 and -16 show the cumulative delta LCC and discounted LCC for the reference programs as well as the UPRCV CPF end points. The GBOTV program serves as the reference vehicle while the three plots show relative delta program costs of the GBOTV/SBOTV program. The severe movement of the crossover points as CPF varies illustrates the sensitivity of the GBOTV CPF as compared to the mixed fleet program. Due to this, as launch costs decrease, ground-basing becomes the more economically advantageous program. Alternately, as UPRCV launch costs grow, space-basing becomes more attractive. The relative impacts of discounting show similar trends.

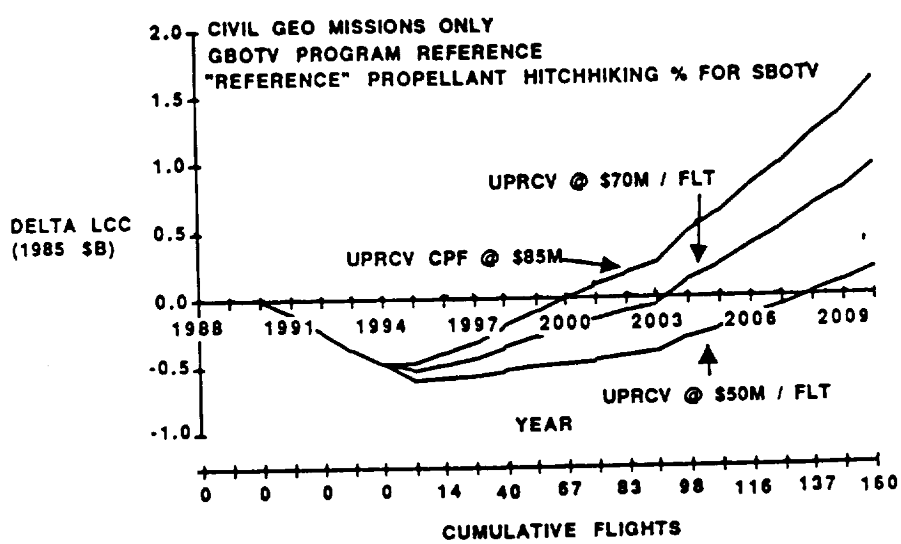


Figure 4.9.5-15 Basing Sensitivity To UPRCV CPF (Constant 1985 $B)

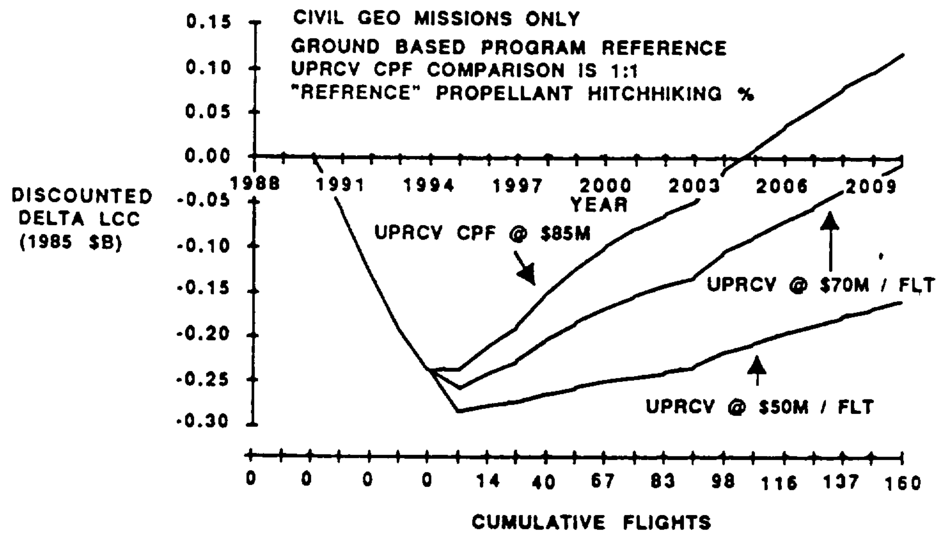


Figure 4.9.5-16 Basing Sensitivity To UPRCV CPF
(Discounted 1985 $B)

Although not explicitly shown in this chart, if space-basing becomes more dependent on tanker propellant its sensitivity to launch cost increases. This condition would essentially negate the impacts of launch cost acting as an economic discriminator between the programs as propellant hitchhiking goes to 0%.

4.9.5.2.3 Basing Sensitivity to payload by Volume

In the course of determining the optimal diameter of a GBOTV, it was found that the preferred vehicle should take maximum advantage of the full UPRCV payload envelope diameter. This provided a minimum length vehicle and resulted in a significant launch cost reduction with respect to length (versus weight) constrained mission deliveries (Section 4.5). In this regard, consideration was given to the payloads and their respective length/diameter characteristics. The objective of this sensitivity is to determine the economic impacts on the basing decision of treating the payloads as a pure volume versus launch vehicle charging by length.

Section 2.1.2 details the user charge alogrithm employed in this study for launch vehicle manifesting. The reference launch cost calculations assume that all payloads (with stages for GBOTV missions) will be charged by the maximum ratio of payload length to launch vehicle length or payload weight to launch vehicle performance. For purposes of this trade the manifesting alogrithm was altered to treat payload as a cylindrical volume and ratioing it to the volume of the UPRCV payload envelope. The rationale behind this exercise is based on the assumption that the availability of a wide diameter payload envelope will influence future payload design, thus causing users to alter the 15 ft payload diameter constraint that predominates within the STAS payload definition.

Figures 4.9.5-17 and 4.9.5-18 show the cumulative delta LCC and discounted LCC impacts on the reference basing cost conditions. If length serves as a constraint in the user charge alogrithm, SBOTV payback occurs after approximately 90 missions (2003). If manifesting is changed to emphasize volume, the economic crossover point occurs 40 flights sooner (1999). The relative average launch cost savings for the two programs is $3M/mission for GBOTV and $8M/mission for SBOTV mission. The $5M/mission SBOTV advantage achieved by treating payloads volumetrically is due to the reference weight/length conditions of payloads and stage. In manifesting only payloads to the Space Statiòn, 100% of them were length charged. By treating payloads volumetrically, the SBOTV payloads launch cost showed large reductions before encountering the weight constraint. On the other hand, the GBOTV manifest of stage and payload was only 50% volume constrained. At the same time the weight user charge factor was much closer to being the dominating constraint for these missions. The combination of these two factors allowed significantly less improvement in GBOTV launch costs.

The cumulative delta discounted LCC trend is similar. The volume impacts are much more pronounced early in the program due to the long, light payloads during that timeframe.

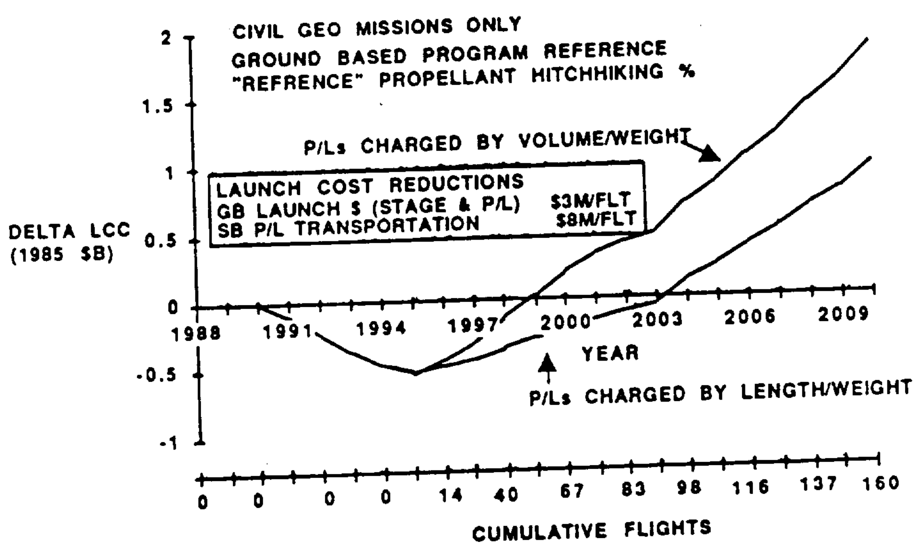


Figure 4.9.5-17 Basing Sensitivity To Payload Volume/Weight (Constant 1985 $B)

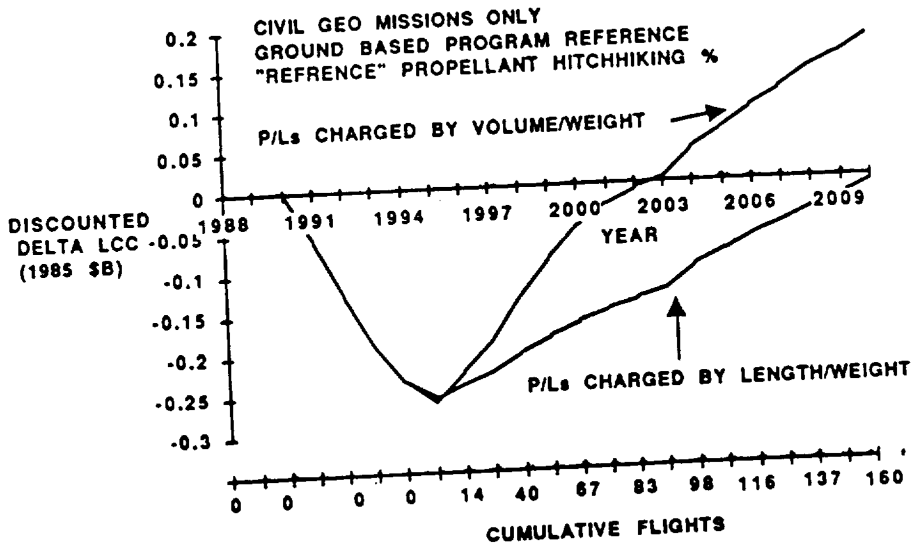


Figure 4.9.5-18 Basing Sensitivity To Payload Volume/Weight
(Discounted 1985 $B)

4.9.5.2.4 GBOTV Return Flight Availability

The last of the major basing sensitivities considers the impacts on ground-basing if the return vehicle availability ground rule becomes less generous. The reference GBOTV program costs includes a minimum user charge to account for the delivery of return ASE on STS/STS II to allow the return of the ground-base stages from LEO. Analysis of the STAS mission model indicates that traffic of return launch vehicle flights may be at a premium and thus could impose a severe cost penalty on GBOTV missions.

The analysis approach used to measure this potential impact includes increasing the GBOTV user charge for ground-based return to 50% (from the reference STS/STS II return of 6.7% which is only charged for ASE delivery) and comparing this to the SBOTV at the reference propellant hitchhiking and completely tanker propellant cases. The composite STS/STS II average return charge CPF increased from $3M/flight to $22M/flight. Figures 4.9.5-19 and 4.9.5-20 show the cumulative delta LCC and discounted LCC impacts of GBOTV return flight availability. At the reference propellant hitchhiking the SBOTV payback occurs in 1997 compared with 2003 (Figure 4.9.5-15) under low cost GBOTV return assumptions. This increases GBOTV delta LCC by $2.9B within the GEO mission model. If SBOTV propellant is completely supplied by tanker SBOTV payback occurs in 2006 (versus no crossover under reference assumptions). Discounted LCC displays the same GBOTV high cost trends.

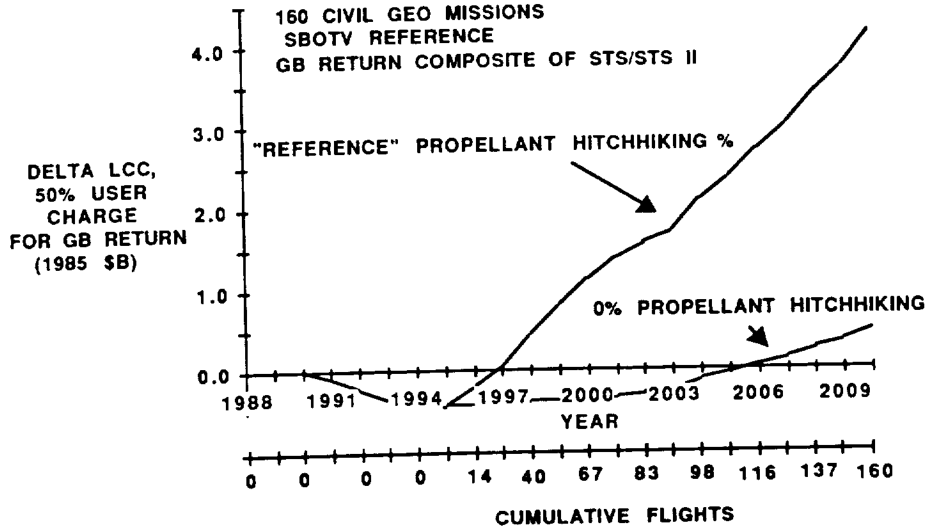


Figure 4.9.5-19 Basing Sensitivity To GBOTV Return Flight (Constant 1985 $B)

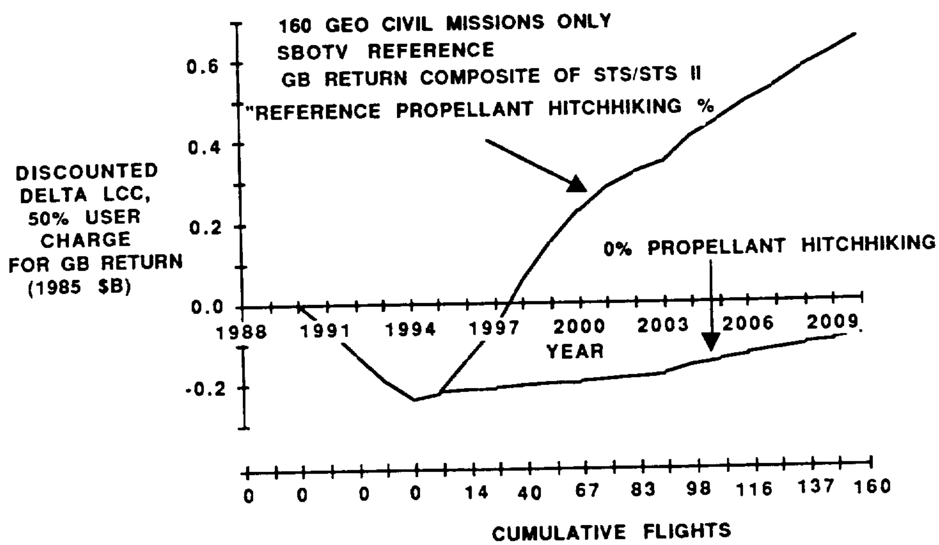


Figure 4.9.5-20 Basing Sensitivity To GBOTV Return Flight (Discounted 1985 $B)

Within the specified study ground rules, launch by LCV and return by STS/STS II is slightly less costly than utilizing the LCV with cargo return capability. The fully reuseable cargo vehicle has a launch cost of $85M/flight vs $70M for the partially reuseable vehicle. Since the "average" OTV/payload combination utilizes 55 percent of LCV capability, the incremented cost per mission is $11M ($15M x .55/.75). The mission costs associated with STS return are $6.1M ($3.1M ASE launch, $2.7M tank replacement, and $0.3M STS operations). Thus, the ground-based program would be assessed an additional cost of $0.7B constant 1985 dollars for utilizing a LCV with return capabilities.

These sensitivities to return flight costs show the potential shortcomings of a reusable GBOTV program. If a minimum cost return flight availability becomes a serious problem even expendable concepts may become preferable over reusable GBOTVs. The SBOTV is not affected by this constraint so that as mission models and future launch vehicle definitions become more defined, realistic GBOTV return scenarios must be seriously considered.

4.9.6 Recommendations

The results of the basing trade clearly show that space-basing is an economically viable augmentation of ground-basing under reference ground rules and analytical findings. Within the 160 GEO civil missions, the analysis shows that space-basing can provide a payback of higher acquisition costs. This payback can be accomplished without relying on DOD missions for increased space-based traffic which results in more efficient utilization of Space Station accommodations. Its feasibility relies heavily on a low cost onorbit propellant supply combined with low investment, operationally efficient Space Station accommodations. The key factors required include a close synergism/development sharing with Space Station and OMV programs, as well as optimizing the use of "leftover" launch vehicle performance and volume capability for propellant delivery. Although propellant scavenging concepts were not included in the context of STAS launch vehicles, any SBOTV propellant support via this method would further enhance space-basing.

The results of the basing sensitivities emphasize the importance of clearer identification of launch vehicle cost and payload definition inputs before a final basing decision can be made. Altering these inputs causes severe swings in LCC results and thus choice of a preferred basing concept.

The basing recommendation at this stage of the OTV Phase A analysis supports a combined GB/SB OTV capability for any new reusable OTV program. Space-basing offers lower operational costs due to deemphasis of transportation costs to LEO. It offers additional benefits in technology advances, large mission capture and mission flexibility. A GBOTV as a supplemental/backup system appears to be a very attractive enhancement and SBOTV predecessor. As launch vehicle/payload definitions become further clarified, the space-based/ground-based program emphasis will again need to be revisited.

5.0 TECHNICAL ANALYSES

This major document section contains subsystem and performance analyses techniques and results that are common to all OTV designs considered in the study effort.

5.1 GN&C ALGORITHM DEVELOPMENT AND ASSESSMENT

A significant development effort was undertaken to define the guidance and navigation characteristics of an aeroassisted OTV. This included entry error analysis, guidance algorithm development and closed loop computer simulations. The purpose of this effort was to help define the maximum efficiency aerobrake by minimizing the control required in the aeropass. This has been successfully accomplished with good results being demonstrated at low L/D's.

5.1.1 OTV Mission Profile

5.1.1.1 Pre-entry Mission Overview

The bread and butter mission for the OTV is currently envisioned to be geosynchronous delivery and retrieval which is thus the primary thrust of current analysis. An overview of this mission is shown in Figure 5.1.1-1. The OTV starts in low earth orbit, having been deployed from the Space Station or shuttle, and initiates transfer to geosynchronous altitude with a perigee burn. In the ensuing coast up to GEO the vehicle performs thermal rolls and any payload peculiar functions such as communication dipouts.

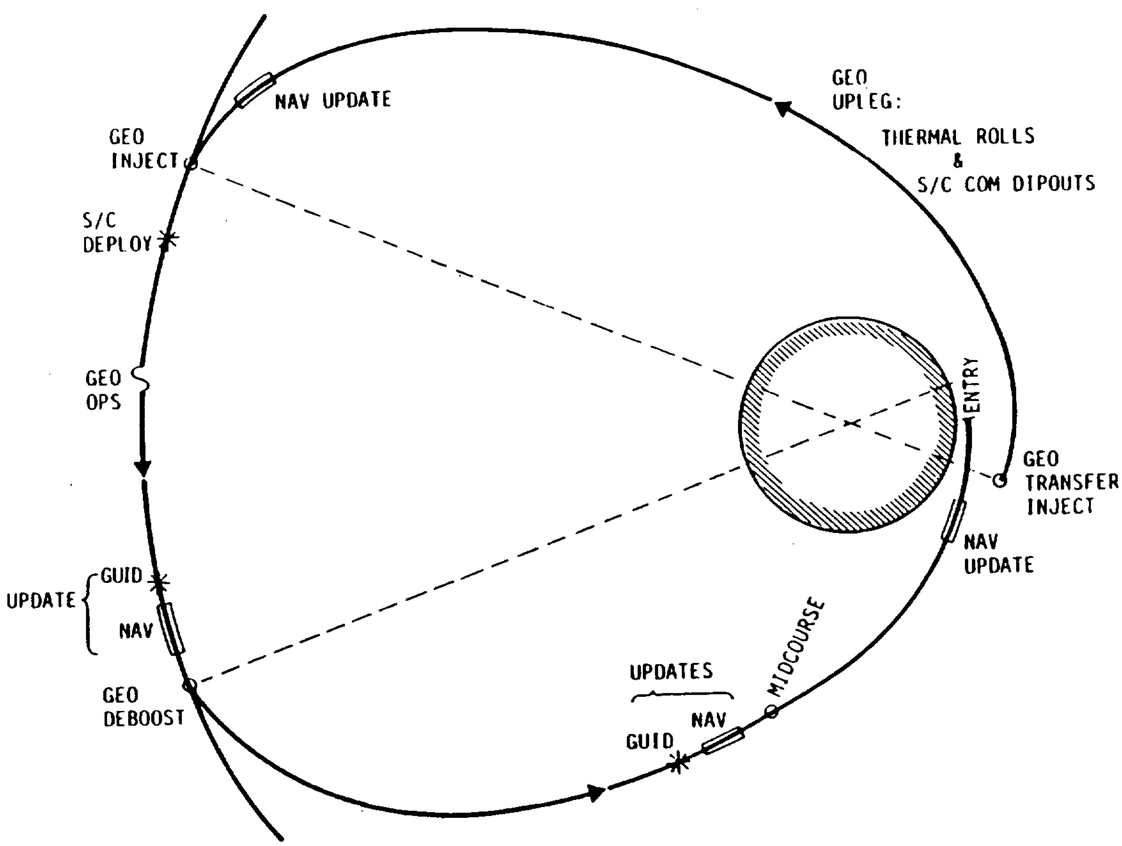


Figure 5.1.1-1 Geosynchronous Mission Overview

Key to the OTV's mission (and in particular, the aerobraking portion) is the autonomous accomplishment of precision navigation. During coast phases the vehicle continuously monitors signals from the Global Positioning System (GPS) satellite network to maintain a precision state vector onboard. This is possible at orbital altitudes below about 8000 nmi through the use of two omni-directional antennae. Above this altitude a medium-gain directional horn system is pointed by the vehicle at the GPS constellation for state vector updates every 12 hours and just prior to any major burn. The other major navigation state, that of inertial alignment, is updated periodically through the use of dual solid-state star trackers. The completion of these two navigation processes, state vector and inertial alignment, are shown on the mission overview as nav updates. Also shown are optional guidance updates which are used to revise pickup vehicle locations or atmospheric status, if necessary.

At the apogee of its transfer orbit the OTV performs a navigation update sequence, and then boosts into geosynchronous orbit via the GEO inject burn. After deploying its payload and performing any other required GEO operations the vehicle executes navigation and guidance updates in preparation for the GEO deboost burn. This burn is targeted to place the orbital perigee within the atmosphere at the proper altitude for aerobraking. Since this is the last major burn, a partial propellant dump may be executed at its conclusion to eliminate excess mission performance reserves and bring the vehicle down to a specific aeroentry mass.

During the downleg coast the vehicle monitors the accuracy of the deorbit maneuver through the use of GPS and incorporates any last-minute targeting shifts into a midcourse correction which is performed at entry minus one hour. This final trajectory adjustment is used to obtain a very accurate entry point for atmospheric flight (perigee error is less than 0.2 nmi).

5.1.1.2 Aero-Phase Overview (Ground-Based)

The aerobraking trajectory and subsequent orbital maneuvers are shown in Figure 5.1.1-2 for a ground-based OTV. Coasting in on the terminal segment of its downleg trajectory the OTV performs a final navigation update 15 minutes prior to entry. Orbital perigee is targeted to a desired altitude in the atmosphere suitable for aero-entry (typically about 45 nmi).

A single-pass aerobraking maneuver is used to dissipate about 8000 fps of orbital velocity to reduce the OTV apogee down to 140 nmi for Shuttle pickup. Although multi-pass aero-maneuvers have been suggested as a method for reducing aeroassist thermal and g-loads, we feel that single-pass entries represent the most optimum approach. This issue is discussed in more detail in 5.1.1.4.

The aerobraking phase itself lasts a total of 4 to 6 minutes with peak load levels of about 3.2 g's. Upon leaving the atmosphere, the OTV is in a suborbital trajectory since its perigee still lies within the atmosphere. The perigee must be raised to at least 100 nmi to provide a stable orbit. In order to correct for phasing shifts, a single pass in a post-aero-phasing orbit is undertaken. By selecting the perigee of this orbit between 100 nmi and the circularization altitude of 140 nmi, a phasing shift of up to 3.01° can be accommodated which is adequate to correct for atmospheric dispersions. By splitting the circularization burn into two pieces in this fashion, phasing is accomplished with no additional Delta-V penalty.

During the phasing orbit coast, the final orbit plane differences between the OTV and its pickup target orbit are corrected with a small inclination trim burn at the nodal intersection. This burn also acts to null out any residual apogee errors. Finally, upon reaching the phasing orbit apogee, a circularization burn is performed which leaves the OTV in its proper pickup orbit and in the correct relative alignment to its pickup vehicle.

The use of this orbital maneuver sequence allows aerobraking to be accomplished with great precision and with minimum Delta-V. The components of error: apogee, perigee, orbit plane and phasing are very accurately nulled with the aid of GPS. The importance of this accuracy lies in the reduction of shuttle rendezvous complexity which reduces the flight time and pre-flight planning involved, along with associated costs.

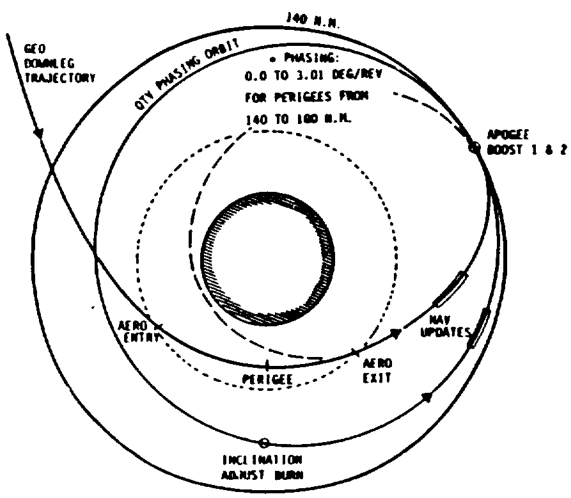


Figure 5.1.1-2
Ground-Based Aerophase

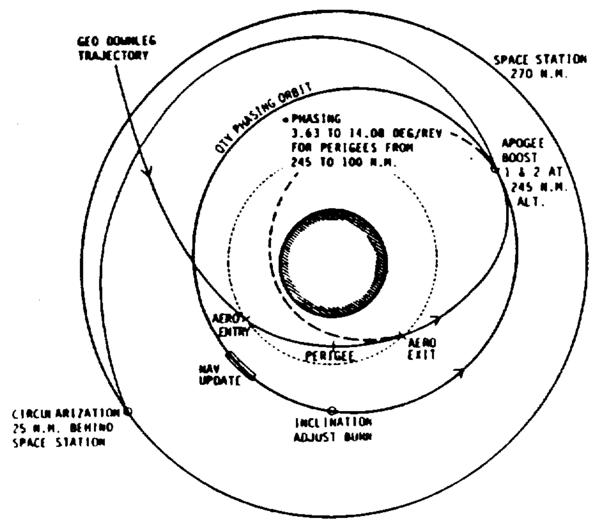


Figure 5.1.1-3
Space-Based Aerophase

5.1.1.3 Aero-Phase Overview (Space-Based)

A space-based entry overview is shown in Figure 5.1.1-3. The aero-phase is very similar to that for ground-based. Because of the higher Space Station altitude the post-aero targeted apogee is correspondingly higher. To maximize the benefit of aero-assist it is most optimum to target the post-aero apogee just below the Space Station orbit. This minimizes the size of the rocket burn required to raise perigee. It is also a better approach from a safety standpoint since it keeps the Space Station well ahead of the OTV during aerobraking, eliminating the possibility of collision due to an off-nominal aeroassist. To avoid interference with the defined Space Station control zones, this initial apogee has been set 25 miles below the 270 nmi station orbit.

The range of OTV phasing orbits available can adjust for up to 10.45° of phase mismatch between the OTV and Space Station. Once the OTV has completed its phasing orbit, a Hohmann transfer is performed which results in the OTV being co-orbital with the station and 25 miles behind it. This transfer process will probably be initiated and monitored by the Space Station for safety.

5.1.1.4 Multi-Pass

Multi-pass aero-assist has the potential for performance improvements by reducing the overall heat load which can lighten up the brake hardware. At the same time the operational impacts of multi-pass make it highly desirable to keep the number of passes down to an absolute minimum.

Figure 5.1.1-4 shows parametric data for aero-assist from GEO with a range of post-aero exit apogees. The curves show control corridor data (perigee altitudes for lift up and lift down limiting conditions) as well as peak stagnation heating (worst case, lift up) and integrated heating (worst case, lift down). The baseline vehicle configuration used has an L/D = 0.116 and a ballistic coefficient of 3.78. The nominal single-pass aeroentry that results in an apogee of 140 nmi for shuttle pickup has the following characteristics:

Lift up perigee altitude	42.29 nmi
Lift down perigee altitude	47.36 nmi
Control corridor width	5.07 nmi
Peak stagnation heating rate	100.58 BTU/FT^2 sec
Peak integrated heating	13960 BTU/FT^2

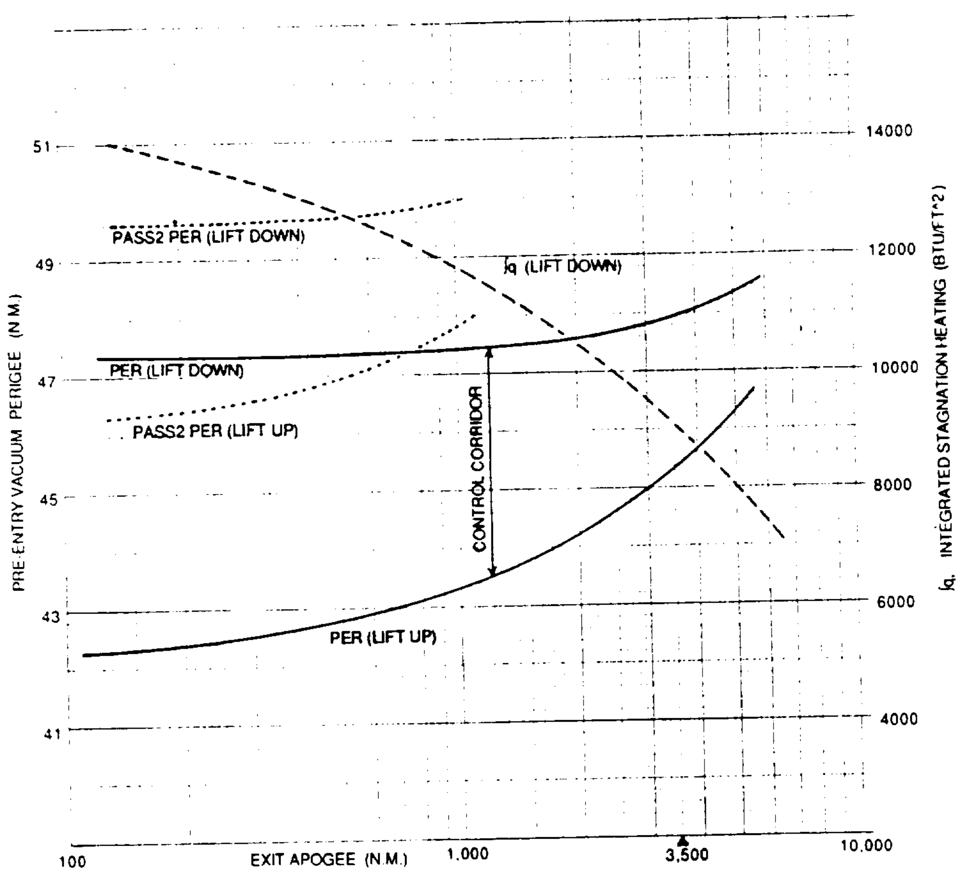


Figure 5.1.1-4 GEO-Return Aeroassist Parameters

With these data as a baseline we then evaluated a two pass GEO return. Each pass was sized to give an equal integrated heat load to minimize TPS thickness and overall brake weight. This resulted in a post aeropass #1 apogee of 3500 nmi and an integrated heat load of 9100 BTU/Ft^2 for each pass (worst case, lift down).

Based on the data from Figure 5.1.1-4, we derived the following benefits for a two-pass GEO return. Because the amount of deceleration is about half that of a single pass GEO return the heating loads are significantly lower: 21% peak stagnation heating reduction and 35% integrated heating reduction. It must be noted that the latter figure can be fully achieved only if the brake cools completely in the 2.6 hours between aero-passes. It is not clear that this will occur since the sole object of the insulating TPS is to slow heat transport. The reduction in peak heat flux would normally allow a reduction in brake diameter which would reduce structural weight, however, the optimum flex brake design is sized by propellant/payload impingement and thus no further reduction is possible. The reduction in integrated heating does allow a 35% reduction in TPS thickness which results in a weight savings of 195 lbs.

A strong penalty associated with multi-pass is a narrowing of the lift control corridor. As is shown in Figure 5.1.1-4, the overall corridor width for pass #1 is 2.6 nmi and for pass #2 it is 3.2 nmi. This represents about a 50% reduction in control capability over the nominal value of 5.0 nmi and is due to the lower aerobraking Delta-V reducing the lateral (control) velocity capability. This will require a doubling of the basic L/D to maintain control margins. Otherwise, the control loss will cause large dispersions in apogee which can increase thermal loads in pass #1 (pushing the TPS weight closer to the single-pass value) and risk vehicle re-entry on pass #2. An alternate solution may be to use very many passes (40 or more) as was proposed for the VOIR Venus orbiter mission. This approach can actually eliminate onboard guidance as well as vehicle lift but requires very many rocket trim burns (one per aeropass). More importantly for the OTV, a 40 pass aero-assist program requires about 6 days to accomplish which is an unacceptable time penalty.

In order to accommodate this doubling of lift, approximately 400 lb of additional aerobrake weight would have to be added. Thus, on a weight basis, two-pass return costs the OTV about 300 lb.

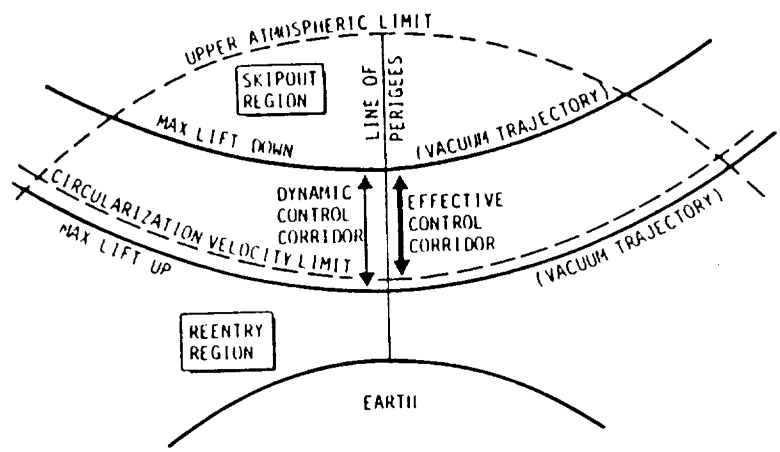
Additional problems associated with multi-pass are the increase in mission duration (consumables) and complexity as well as higher sensitivity to dispersions (pickup vehicle phasing is more difficult, for example).

Thus, it is recommended that the single-pass aero-assist baseline be maintained for OTV return from geosynchronous orbit.

5.1.2 Aero-Assist Control

Trajectory control is extremely critical in the aero-assist phase of flight. For a ballistic entry from geosynchronous orbit, a difference of just 300 feet in perigee altitude results in a 100 mile variation in exit apogee. This can be the difference between aero-braking and "aero-crashing" for missions seeking to return to relatively low shuttle retrieval altitudes (140 nmi). Not only is this kind of perigee accuracy impossible to target, it is overwhelmed by uncertainties in atmospheric density altitude. Thus it is very critical that the OTV have a means of controlling its aero trajectory in real time.

To establish a working concept for control capability the notion of a control corridor was created. As illustrated in Figure 5.1.2-1, the control corridor defines the flight control boundaries of the aeropass. For the case of a lifting vehicle these control boundaries are defined by flying aero-assist trajectories with the lift vector fixed down and fixed up. The two trajectories that hold these attitudes throughout and which also exit the atmosphere with the correct orbital apogee represent the limits of vehicle control. The lift up profile describes the lower boundary and the lift down the upper one. Between these two boundaries, a controllable aeropass is possible, outside of them a skip-out or re-entry results. Thus the control corridor describes that volume where the aero-assist trajectory can be steered to meet the desired apogee exit conditions with the control available.



NOTE: CURVATURE OF TRAJECTORY INVERTED BY VERTICAL EXAGGERATION OF DIAGRAM

Figure 5.1.2-1 Aero-Phase Control Corridor

As a simplification, the corridor is described by the pre-entry vacuum perigees of the two bounding profiles. This is convenient as it allows pre-entry orbital targeting to a well defined entry zone. Targeting of perigee is the most critical orbital parameter, apogee variations by contrast are an order of magnitude less important.

Trajectories that lie near the bottom of the control corridor suffer circularization velocity penalties. Because they go deeper into the atmosphere their exit flight path angles are steeper which drives their exit perigees lower, resulting in a sharp rise in the circularization velocity. To avoid this region the bottom 15% of the corridor is often eliminated, with the remaining volume called an "effective control corridor".

Once an error budget is established, the control corridor is specified which covers it with some margin. This process sizes the adequate amount of control to do the job and will be covered later in this section.

5.1.2.1 Control Options

The two primary methods of aerotrajectory modification are drag control and lift control. Drag control uses direct variation of the vehicle's ballistic coefficient to alter the magnitude of the drag force thereby modulating the deceleration. This is accomplished either by varying the frontal area (as with drag brakes), altering the aerodynamic streamlining (as in the aerospike technique) or via a combination of the two (as is done with the ballute concept). All of these techniques are useful for controlling exit apogee, however, they cannot control orbit inclination.

The lift control technique uses lift inherent in the body to directly alter the vehicle's flight path angle. It can supply control both in the in-plane (to modify exit apogee) and out-of-plane directions (to control inclination). The lift force arises from trim angle of attack which is most easily created in a blunt body by offset c.g. With mid L/D values (of 0.1 to 0.2) significant inclination turns can also be accomplished in the aeropass.

The amount of control achievable with each of these techniques is shown in Figure 5.1.2-2. For each control method a range of control corridor widths is shown as a function of the control capability. Lift control is displayed as a function of L/D, drag control is shown as a function of the drag variation and aerospike (which streamlines the effective entry shape by means of a gas spike created by the main rocket engine) is shown as a function of thrust level.

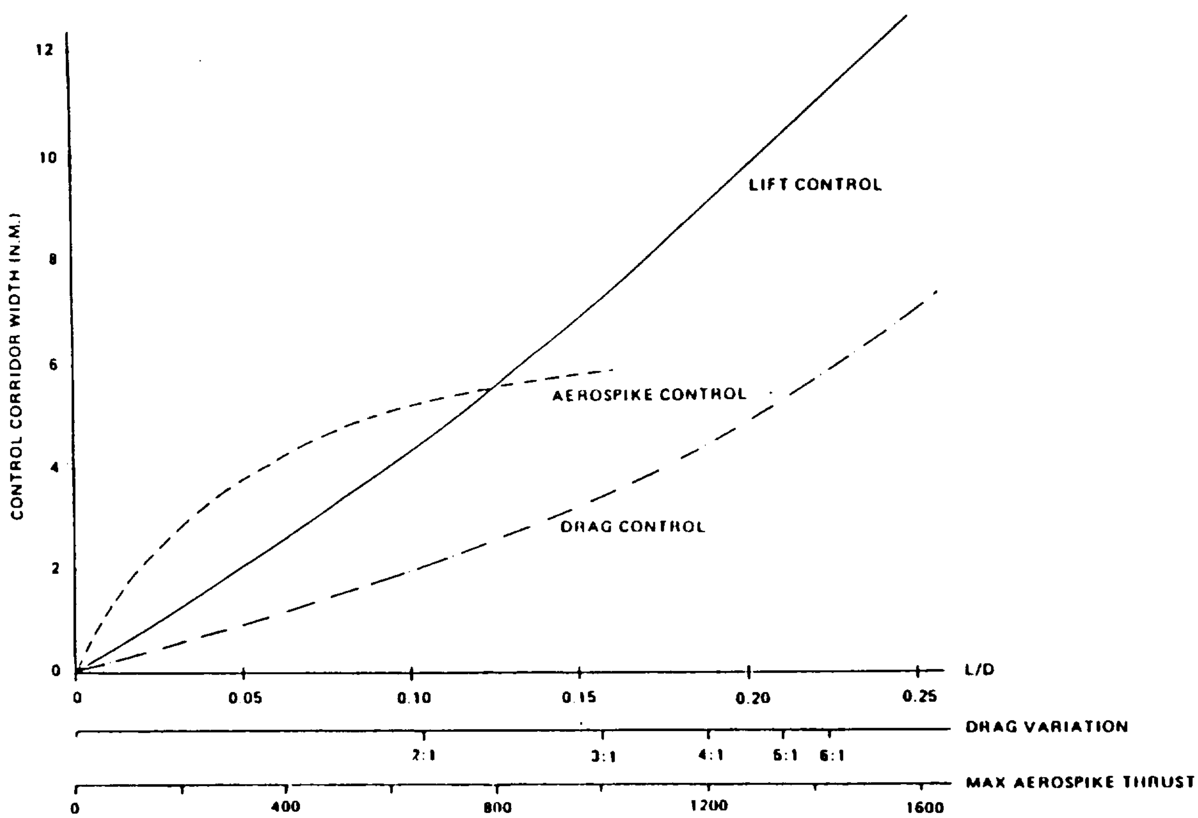


Figure 5.1.2-2 Control Mode Capabilities

For the case of aerospike control, it may be seen from the chart that the control authority is limited to an approximately 5 mile wide corridor with a correspondingly high propellant usage of 420 lb of propellant that is not matched by weight savings in other systems.

The practical geometric constraints of mechanical drag modulation limit its area variation to less than about 3:1. From the chart one can see that this corresponds to a control corridor of 3 nmi or less. This represents a somewhat marginal control situation.

The lift control approach appears to offer the largest amount of control for the smallest vehicle impact. For example, L/D values of 0.12 are easily achievable with the 70 degree Viking aeroshell and result in a control corridor width of 5.0 nmi.

Strictly from a control standpoint, the use of lift appears to have the most promise for the OTV. When other factors are considered such as aerobrake hardware weight*, technical risk, minimization of vehicle impacts, and aero-stability; lift control is clearly the most desirable method of aero-assist control.

(* The common basis design analysis in paragraph 4.3 comparing the lifting brake and ballute drag brake concepts showed the ballute weighs about 1000 lb more than the lifting brake).

5.1.2.2 Low Versus Medium Lift

For a lifting entry vehicle, a range of L/D's are possible depending on the basic aero-assist strategy. The basic options are: 1) Minimizing L/D to cover expected errors only, 2) Increasing L/D to cover errors plus perform a significant out-of-plane maneuver for inclination turns, 3) Increasing L/D to allow flight higher in the atmosphere (via lift down) where loads and heating are lower.

Approach 1) and 2) are compared in Figure 5.1.2-3. Lift can be used to trim the aeromaneuver in the out-of-plane (inclination control) as well as the in-plane (apogee control). The graph shows inclination change capability in the aeropass for deorbit from geosynchronous orbit to a Shuttle recovery orbit of 28.5° inclination and 140 nmi altitude. It may be seen that for an L/D of 1.8 the entire 28.5° plane change can be accomplished in the aeropass.

A comparison was made of the velocity savings to be gained by going from an L/D of 0.25 to 1.00. This represents additional inclination change capability of 11.5° (increasing from 3.5° to 15° delta inclination) which corresponds to a velocity savings of 620 fps. For the ground-based OTV this results in a propellant savings of just 250 lb. Even for the space-based vehicle returning a maximum payload of 14000 lb the savings is only 840 lb. The increase in dry weight necessary to produce the L/D of 1.0 must be less than these propellant savings to realize a net performance benefit. Actual designs undertaken in the course of the Phase A study have shown that this is not the case with an aerobrake dry weight penalty of several thousand pounds being indicated for the 1.0 L/D vehicle. One concludes that adding lift to significantly alter inclination in the aeropass results in an inefficient OTV.

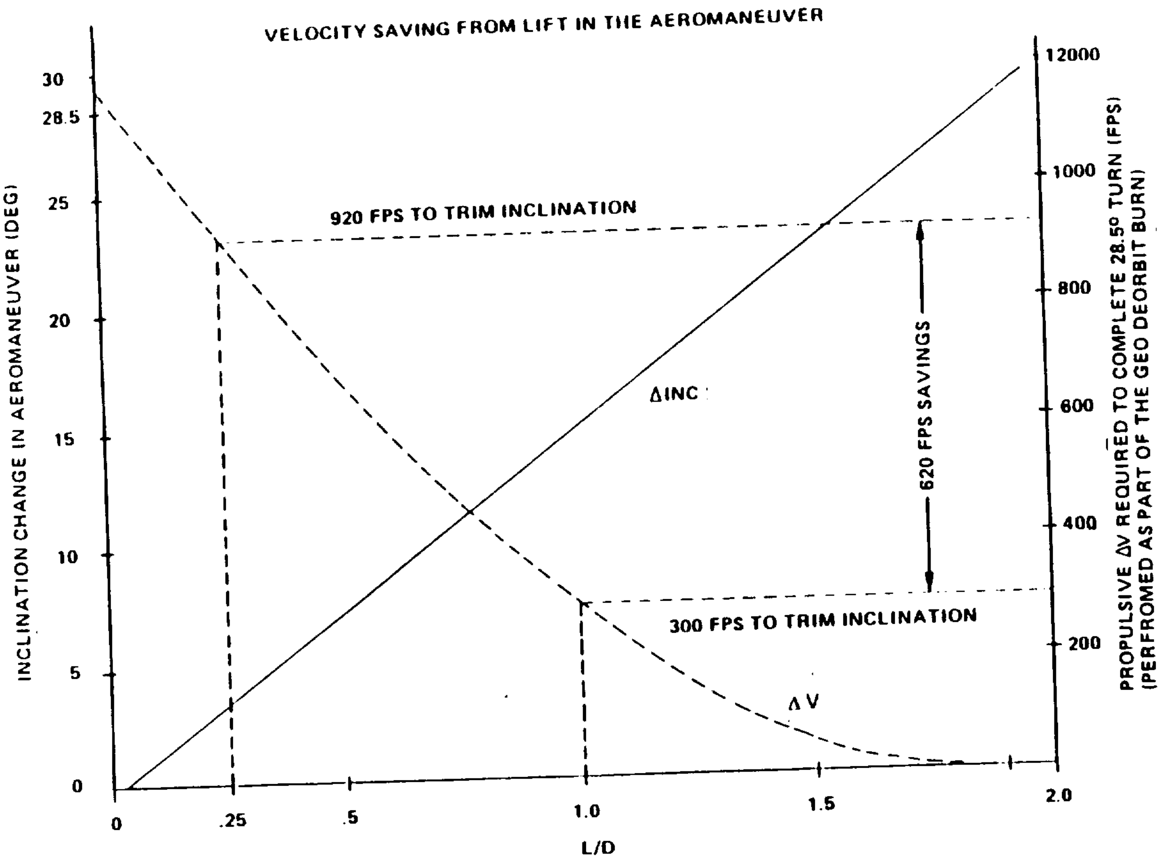


Figure 5.1.2-3 Inclination Turn in the Aero-Phase

Another option for medium L/D vehicles is to use excess lift to fly at a higher location in the atmosphere where g loads and peak heating will be lower. This is accomplished by flying the vehicle with its lift vector primarily down.

A design comparison was made of two 0.3 L/D vehicles (see also paragraph 5.2.2). One was flown at the top 5.0 nmi of its 15.0 nmi capability control corridor and the other was flown in the center. The reduction in g-loading (at the top) resulted in a structural savings of 94 lbs, primarily in the aerobrake. Although the peak heat flux on the vehicle was reduced by 12% the integrated heat flux increased by 21% because the lift down trajectories dwell in the atmosphere for a longer period of time. The net effect was an increase in TPS material weight of 165 lbs. Thus the net effect of flying higher in the 15 mile corridor was to increase vehicle weight by 71 lbs.

This and similar optimization studies show that the best location for the vehicle to fly is near the center of the control corridor. The vehicle weight penalty of an increase in lift is about 210 lb additional dry mass per 0.1 L/D. Thus we conclude that the most weight efficient OTV must fly at the center of its control corridor with the minimum L/D required to safely cover dispersions. The basis for dispersion estimates and our baseline L/D will be covered in the next section.

5.1.3 Aero-Entry Error Analysis

The key to minimizing L/D (and thus the aerobrake weight) is to establish the maximum expected environmental variations which the vehicle's control capability must be able to correct. A number of error sources were considered and their impacts are summarized in Table 5.1.3-1.

Table 5.1.3-1 Aero-Entry Error Analysis

	EQUIVALENT PERIGEE ERROR		
• TARGETING ERRORS (MIDCOURSE)			
. GUIDANCE ERRORS	= 200 FT		
. POINTING ERROR	= 130 FT	± .1 DEG	
. CUTOFF ERROR	= 490 FT	0.33 FPS ACCELEROMETER + 10 MS TIMING ERROR	
. GPS ERROR	= 575 FT	FROM 1020 FT POSITION UNCERTAINTY	
	474 FT	FROM 0.1 FPS VELOCITY UNCERTAINTY	
. NONGRAVITATIONAL	= 320 FT	ACS IMBALANCE	
• AERODYNAMIC VARIATION			
- ATMOSPHERIC UNCERTAINTY	= 5700 FT	± 30% DENSITY	
- L/D UNCERTAINTY	= 9700 FT	± 2° AT 7.2° ANGLE OF ATTACK (± 30% L/D)	
- BALLISTIC UNCERTAINTY	= 1700 FT	WT = ± 150 LB (RESIDUALS)	± 8% W/C_DA
		C_D = ± 5% (STS/VIKING DATA)	
		A = ± 5%	
• RSS	= ± 980 FT	= ± 0.16 N.M. FROM TARGETING	
	= ± 11380 FT	= ± 1.87 N.M. FROM AERODYNAMICS	
	= ± 11420 FT	= ± 1.88 N.M. NET VARIATION	

These uncertainties may be grouped into two categories: targeting errors and aerodynamic variations. One will recall that the mission profile calls for a final trajectory trim correction one hour before entry. Errors in performing this burn will be uncorrected and are referred to as targeting errors. Once the atmosphere is encountered, several factors will cause variations in flight. The most significant of these from a trajectory standpoint have been grouped together under the heading of aerodynamic variations.

The most serious impact to the vehicle of aero errors is the variation in altitude (because of the exponential nature of atmospheric density, undershoots and overshoots are highly self-reinforcing), therefore the error sources are all normalized to their influence on aero-assist altitude (or equivalently density altitude) which is expressed as a variation in vacuum perigee. Because the variables are first-order independent, their individual contributions are RSS'ed to give an overall perigee variation.

The individual targeting errors are summarized as follows. Guidance errors represent the granularity of onboard computations and are estimated to be about 200 ft based on current experience. Pointing errors due to onboard navigation package misalignments will corrupt midcourse burn pointing by about 0.1°. Cutoff errors in executing the burn are due to accelerometer errors (amounting to 0.33 fps) and an assumed 10 millisecond command granularity. GPS state vector error levels at this stage of flight are estimated to be 1020 ft in position and 0.1 fps in velocity. Finally, a non-gravitational term of 320 ft is included to account for trajectory disturbance by the ACS system. When the impact of these errors on perigee shift are computed and their contributions RSS'ed the net effect on perigee altitude due to targeting is found to be 0.16 nmi. This is equivalent to an entry flight path angle variation of 0.02°.

Errors due to aerodynamic variation are summarized as follows. Atmospheric density uncertainty is currently believed to be about plus or minus 30% (representing an uncertainty in day of entry atmosphere, not yearly variations). This shifts the OTV nominal ballistic profile vertically by 5700 ft. Taken another way, an OTV with no knowledge of atmospheric shift could be high or low by 5700 ft with respect to the actual density's nominal aim-point. Better atmospheric sensing and modeling techniques should be able to reduce this uncertainty in the future. Trim angle of attack variation gives rise to the L/D uncertainty shown next. Decreased angles of attack reduce control capability via a decay in lift. The estimated angle of attack uncertainty of 2° can cause a control corridor variation of 9700 ft which must be compensated for by increasing control margin. A derivation of the trim attitude variation is contained in the next section.

Finally, a term for the variability of the vehicle's ballistic coefficient is carried consisting of uncertainties in burnout weight, coefficient of drag (from Viking and Shuttle experience), and platform area (due to brake flexure). The sum total of these factors gives a ballistic coefficient variation of 8% which translates to a (density) altitude uncertainty of 1700 ft. The RSS total of the three aerodynamic parameters gives a net altitude variation of 1.87 nmi which is an order of magnitude more severe than the uncertainty due to targeting.

Combining the variation due to targeting with that for aerodynamics results in a net altitude uncertainty of $\pm$ 1.88 nmi. This represents the overall trajectory uncertainty which must be overcome by the vehicle's control capability. To include some margin this figure was increased by 33% (based on experience) to arrive at a net control corridor width requirement of $\pm$ 2.5 nmi.

Utilizing lift up and lift down aero-assist simulations, various control corridor boundaries were defined for flight through an undispersed atmosphere at various L/D's. The results of this effort is shown in Figure 5.1.3-1 where control corridor width is shown versus L/D. Using this data base an L/D of 0.116 was found to yield the desired control corridor width of 5 miles total.

Using Viking entry vehicle data which had the same aeroshell shape as our proposed OTV, shows a nominal angle of attack of 7.23° is required to achieve the L/D of 0.116.

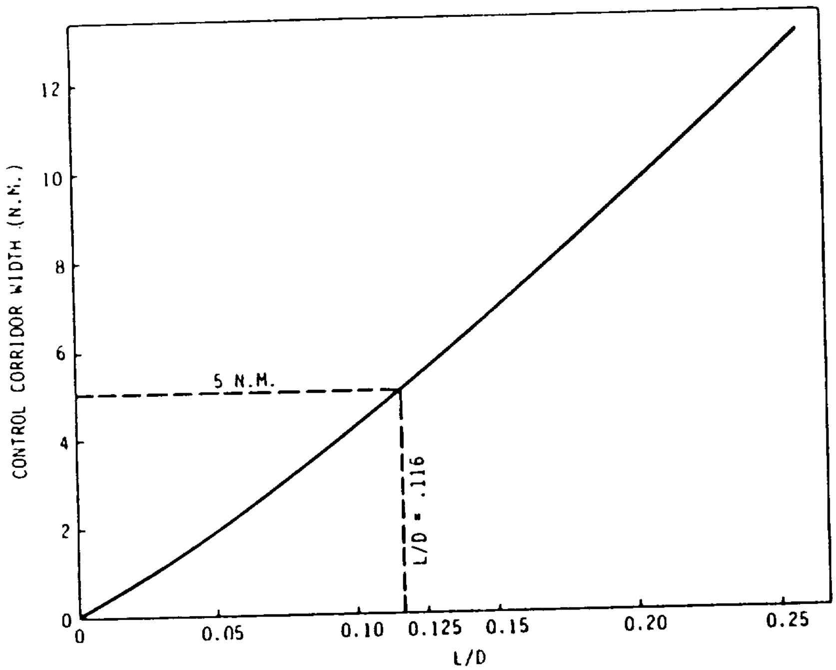


Figure 5.1.3-1 L/D vs Control Corridor

5.1.3.1 Trim Attitude Uncertainty

Variations in the vehicle's trim angle of attack can cause serious problems for a successful aeropass. Decreased trim angles can jeopardize trajectory control through a loss of lift while increased angles can create heating and impingement problems. To understand the magnitude of the problem an analysis was conducted which included center of gravity (c.g.) and aerodynamic variations.

The summary of the c.g. analysis is illustrated in Figure 5.1.3-2. Since the vehicle depends on an offset c.g. to establish a stable trim angle of attack (with no active control surfaces such as flaps for assistance), variations in the lateral location of this c.g. directly impact the desired attitude. A worst case analysis was conducted to define the boundaries of the c.g. envelope and thus the maximum expected attitude variations.

Because a high energy transfer stage requires a great deal of propellant, residuals can play a major role in the final mass properties. Furthermore, our OTV Phase A configuration uses a 4-tank layout so propellant imbalance has a maximum lever arm effect to shift c.g. laterally. Clearly, balancing propellant tank pairs is very important.

The solution chosen to this problem is a refinement of the normal propellant utilization (P.U.) process which all large stages use to achieve simultaneous depletion in all tanks, thereby minimizing wasted propellant. By concentrating point level sensors in the lower 10% of the tank volumes a reasonably precise level difference between tank pairs can be established during the final pre-entry rocket burn to fine tune the P.U. process. Based on Saturn and External Tank data, uncertainty values of 16 lb/tank for LOX and 3 lb/tank for LH_2 are achievable.

These propellant imbalance uncertainties were then RSS'ed together with vehicle dry weight c.g. uncertainties of 0.35 inch to give the overall vehicle c.g. envelope shown in Figure 5.1.3-2. The total c.g. envelope is rectangular, reflecting the greater impact of LOX residuals than LH_2. By aligning the long axis perpendicular to the vehicle centerline as shown, movement of the c.g. within the rectangle has minimum impact on vehicle attitude. The worst case c.g. location is in one of the corners furthest from the centerline. This location gives an angle of attack shift of 0.76° for the ground-based OTV configuration and a 1.12° shift for the space-based vehicle.

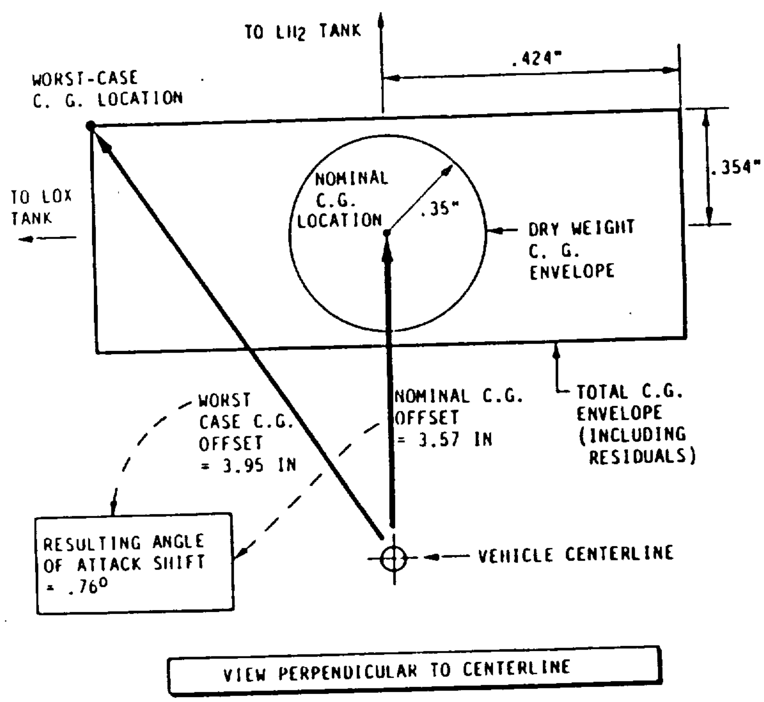


Figure 5.1.3-2 C.G. Impact on Trim Attitude

A further complication to c.g. control is added when the OTV is bringing a payload back via aero-assist. Accurate knowledge of the payload c.g. is practically impossible, in general, which will mandate the use of a moveable payload adapter that is adjustable in-flight. The measurements to drive this adjustment will not be trivial to achieve and might be derived from attitude maneuvers, burn trim pointing or feedback in the aero-assist itself. This area requires further analysis and its uncertainties are not included here.

In addition to c.g. uncertainty, the other key driver for angle of attack variation is the uncertainty in the vehicle's aerodynamic parameters (Cd, Cl, L/D etc.). Work was done in the Viking Project to establish these uncertainties for entry validation. The resulting L/D variation was estimated to be 5%. Similarly, the repeated entries by shuttle vehicles shows a flight-to-flight variation of less than 5% in L/D.

Utilizing this L/D variation we can derive an equivalent angle of attack uncertainty of 0.36° for the OTV, based on its nominal alpha of 7.23°. When this variation is RSS'ed with the previously derived uncertainty due to c.g., we find a total alpha variation of 0.84° for the ground-based ACC OTV and 1.18° for the space-based vehicle.

Because of prediction uncertainties in the entry contours of the flex fabric aerobrake, the initial development flights of the OTV will probably see a higher variation in attitude. The operational vehicle, however, should exhibit a repeatability that will allow the flex distortions to be biased out. The first few flights (development test flights) can be flown on a relatively benign flight profile (via performing partial aero-assist/partial propulsive hybrid trajectories) while the aerodynamics are being calibrated. AFE could also greatly help the situation by flying flex brake test samples.

Primarily because of the uncertainty in the flex brake behavior we chose to increase the max derived angle of attack variation of 1.18°, derived above, to a more conservative 2.0° which is the value used in all error analyses and dispersed closed-loop simulations.

5.1.3.2 Free Molecular Flow Impact

An analysis was undertaken to evaluate free molecular flow impacts. Drag and lift coefficient data for continuum and free molecular flow was implemented into the basic aero simulation. A simple straight-line transition function was used (Figure 5.1.3-3) which is based on Viking test data and computational free molecular data. The results are shown in Figure 5.1.3-4 as

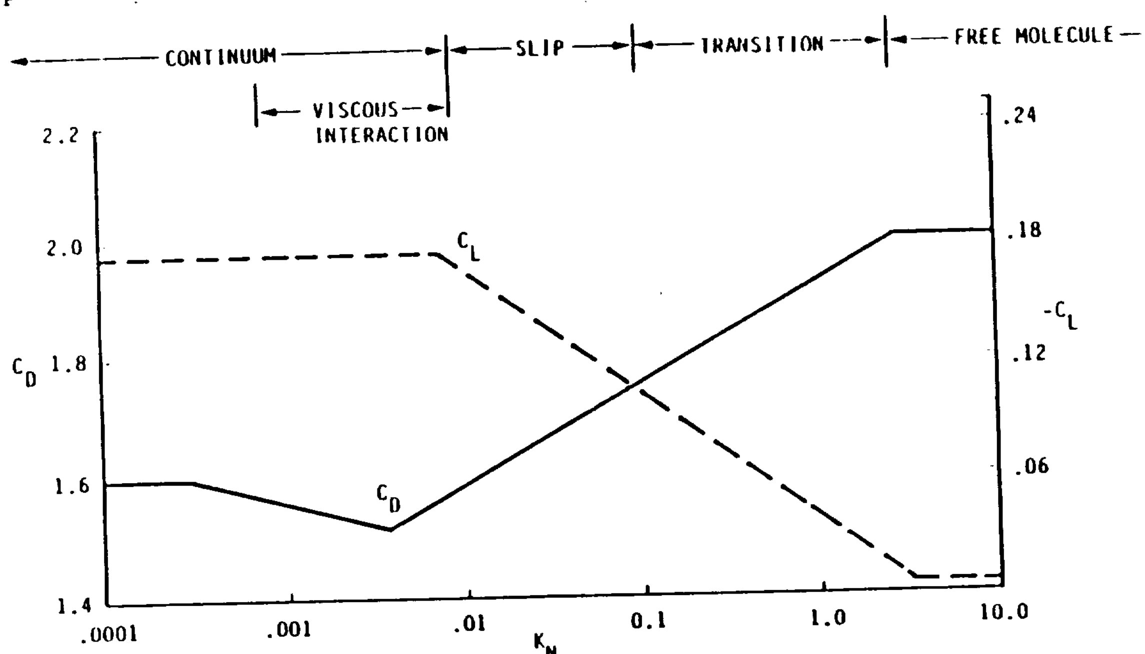


Figure 5.1.3-3 Free Molecular Flow Transition

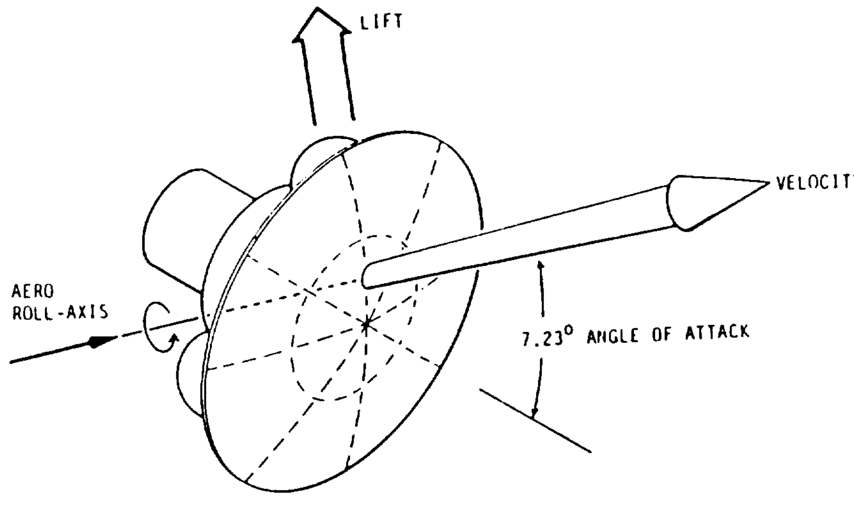


Figure 5.1.4-1 OTV Entry Attitude

Figure 5.1.4-2 shows an overview of the aero-entry process. Again, the entry control corridor is shown as a tunnel bounded on the top by the lift down dynamic limit and on the bottom by the lift up operational limit. By modulating the pointing of its lift vector within these limits the OTV successfully performs aero-assist. The lift vector is controlled by guidance to simultaneously correct exit errors in apogee and orbit plane alignment. Because the OTV's lift is fixed by the vehicle's constant trim angle of attack, the force's effect can only be nulled by integral pointing. This is accomplished via a continuous roll whose integrated lift is approximately zero. A roll rate of only 9° per second (1.5 rpm) is required for trajectory control. Because there is very little roll damping, only the initiation and termination of the roll requires significant RCS fuel. This continuous roll is in contrast to other lift management techniques that require multiple bank angle reversals about the vertical plane, with each oscillation requiring start-stop RCS impulses.

Because of the execution of a pre-entry guidance update the vehicle has attained a stable attitude at entry interface. By holding this attitude for a specified duration, the vehicle will exit the atmosphere with the proper apogee and orbit plane. The combination of these two factors: pre-entry lift targeting and simultaneous nulling of exit apogee and orbit plane errors (made possible by continuous roll) applies the maximum corrective force with the minimum response time.

As the entry proceeds, continuing guidance updates will detect atmospheric density fluctuations and other off-nominal conditions causing subsequent small roll attitude holds (generally pure lift up or down) to tweak the OTV trajectory. Aeropass control terminates, as it began, at the .03 g threshold where vacuum coast begins.

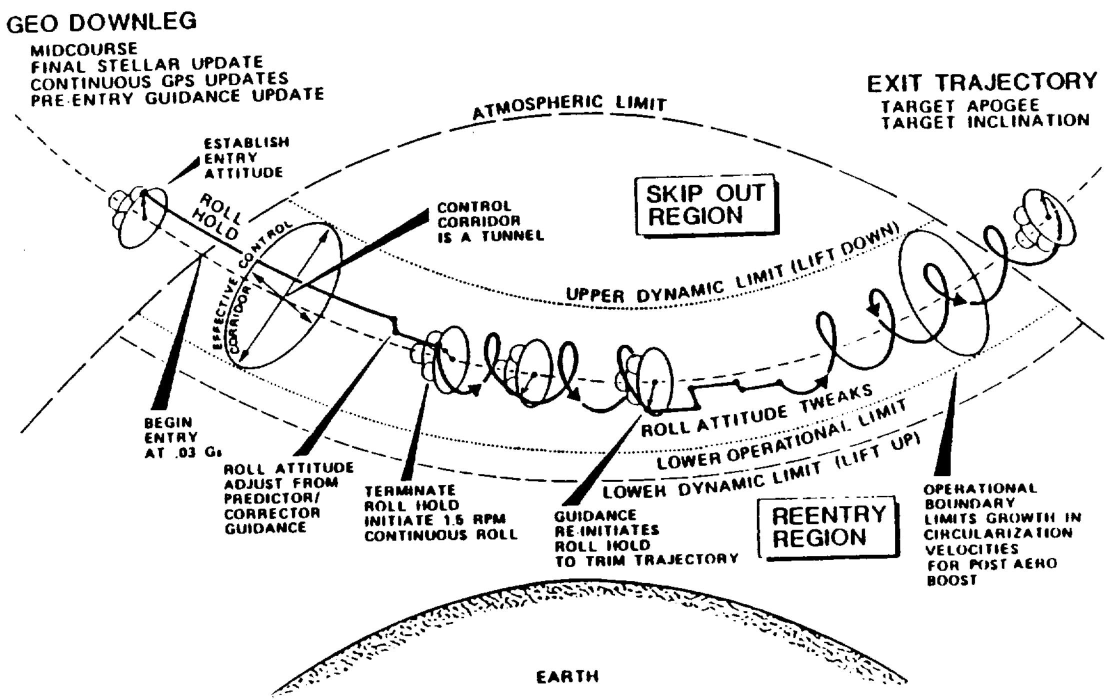


Figure 5.1.4-2 Aero-Entry Overview

5.1.5 Aero-Guidance

The basic aero-guidance scheme is a predictor-corrector which targets to an exit orbit apogee and orbital plane alignment (inclination and ascending node). This guidance technique steers the vehicle by pointing the body-fixed lift vector in a direction which nulls apogee and inclination simultaneously, permitting the most efficient use of the available control and its most rapid application to trajectory correction. After the targets are met the lift vector is nulled via a continuous roll. It should be noted that the lift vector is never perfectly zeroed out by this roll; however, guidance accounts for this by detecting lift residuals in the prediction process. The actual roll hold duration is controlled via a lateral velocity target which is the net sensed velocity in the lift direction that is accumulated during a roll hold. The use of this targeting method reduces the impact of L/D dispersions.

The use of a predictor-corrector provides a good software fit with the OTV orbital guidance package. Because of the variety of missions the vehicle performs, the OTV orbital software is expected to be a menu-driven predictor-corrector type. An important additional feature of the predictor-corrector approach is that it enables a pre-entry prediction to be made. This update bootstraps an initial control set while there is large timing margins for additional computation. It also establishes a nominal entry attitude which reduces the roll response lags by pre-aiming the vehicle.

Because of density dispersions that will always occur in the atmosphere, a feedback routine is included which utilizes sensed accelerations from the navigation package to correct the onboard density model.

5.1.5.1 Guidance Update Cycle

Figure 5.1.5-1 shows the functional flow of an aero-guidance update in block diagram form. Beginning at the left, the guidance function starts with the current navigation state vector plus commanded roll attitude and commanded lateral velocity from the previous update cycle. The navigation state plus sensed decelerations are fed into an atmospheric feedback function which acts to correct the onboard density model for observed fluctuations. The state vector and commanded controls are then fed into the trajectory prediction routine which produces estimated post-aero errors in inclination and apogee.

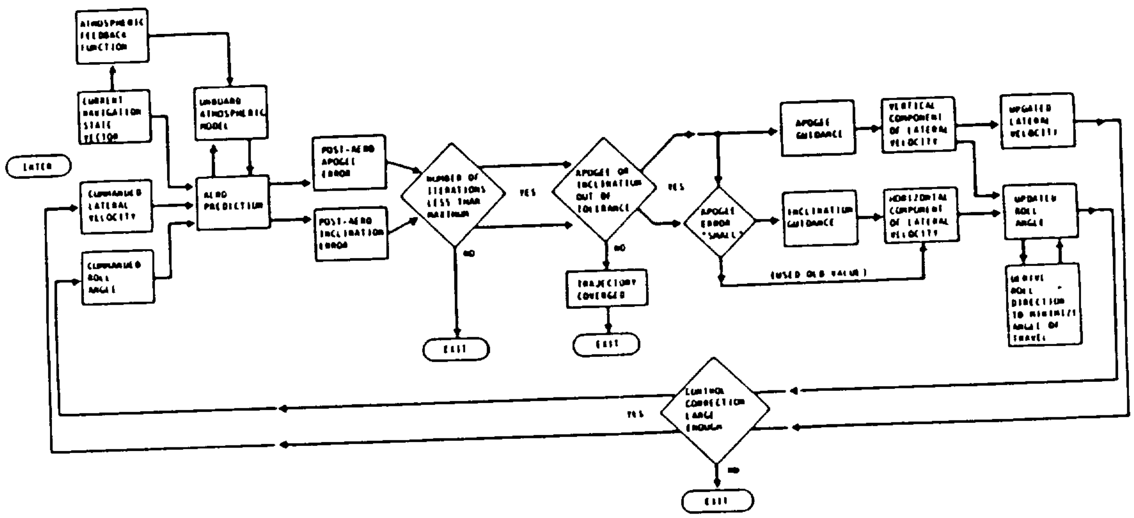


Figure 5.1.5-1 Aero-Guidance Overview

After checking that the maximum iterations for this guidance update have not been exceeded, the predicted errors are compared against mission tolerances. If the errors are both small enough, guidance has converged and the update function is exited. On the other hand, if either exceeds a specified tolerance, the correction portion of the algorithm is entered. When performing corrections, apogee guidance is always executed; however, the inclination correction logic is only performed when apogee errors fall within a specified tolerance band. The reason for this is that trajectories with large apogee errors have false inclination values that will corrupt the inclination steering. If the inclination correction logic is so disabled, a previous output is used instead.

The apogee and inclination guidance functions produce vertical and horizontal components of lateral "velocity to be gained". These two components, when taken together, produce a new target roll attitude for the vehicle. The duration of the new roll hold is determined by the amount of time it takes to accumulate the vertical component of lateral velocity.

These new control variables are compared with the old ones to see if the changes are large enough to be realistically implemented. If not, the update function terminates; otherwise the new control variables are fed back into the prediction routine to start a new guidance iteration.

5.1.5.2 Guidance Predictor

The predictor portion of guidance takes the current vehicle state and propagates it forward through an environmental model, using the current set of control variables, to derive the OTV aero-exit conditions. Guidance utilizes onboard models of the atmosphere, gravity field and vehicle roll dynamics to represent the environment. A fourth order Runga Kutta integrator with a step size of 2.0 seconds is used to propagate the vehicle state.

The onboard roll propagator model, which tracks vehicle attitude, accounts for vehicle inertia. A fairly simple linear rate model is used to describe thruster firings which results in a second order description of the vehicle attitude. This level of fidelity is necessary because the steep rise and decay of aero-assist deceleration can cause significant trajectory residuals if the roll attitude is in error. The relatively small OTV control jets take a few seconds to accelerate the vehicle (angular acceleration is 2.5 deg/sec^2 to achieve a maximum roll rate of 10 deg/sec) which would be a problem for a fixed rate (inertia-less) model.

5.1.5.2.1 Atmospheric Model

The onboard atmospheric model is a simplified version of the 1962 standard atmosphere which gives density as a function of altitude. An oblate earth is used to derive geodetic altitudes. This atmospheric model is scaled up or down globally in response to variations in drag as measured by the onboard accelerometer package. The feedback technique lumps together the indistinguishable effects of ballistic coefficient and atmospheric density variations into one scalar multiplier.

Variations in the earth's upper atmosphere are a strong driver for aero-assist. Random fluctuations observed during shuttle entries show large swings in density occurring over small changes in altitude (Figure 5.1.5-2). The rapidity of these fluctuations can interact strongly with the vehicle's control system rates. In order to damp out the system response, a weighted averaging technique is used to filter the density fluctuations that are fed into guidance. This filter uses a power function of sensed deceleration. Midway through the aeropass outbound leg the filter is switched off and direct drag measurements are used by the density feedback function. The behavior of the filter function is shown in Figure 5.1.5-3.

5.1.5.3 Guidance Corrector

The corrector consists of two pieces, an apogee guidance package issues velocity-to-go targets in the vertical plane while inclination guidance (performing wedge targeting) derives targets for the horizontal plane. These two components are then combined by the roll controller into an attitude pointing command.

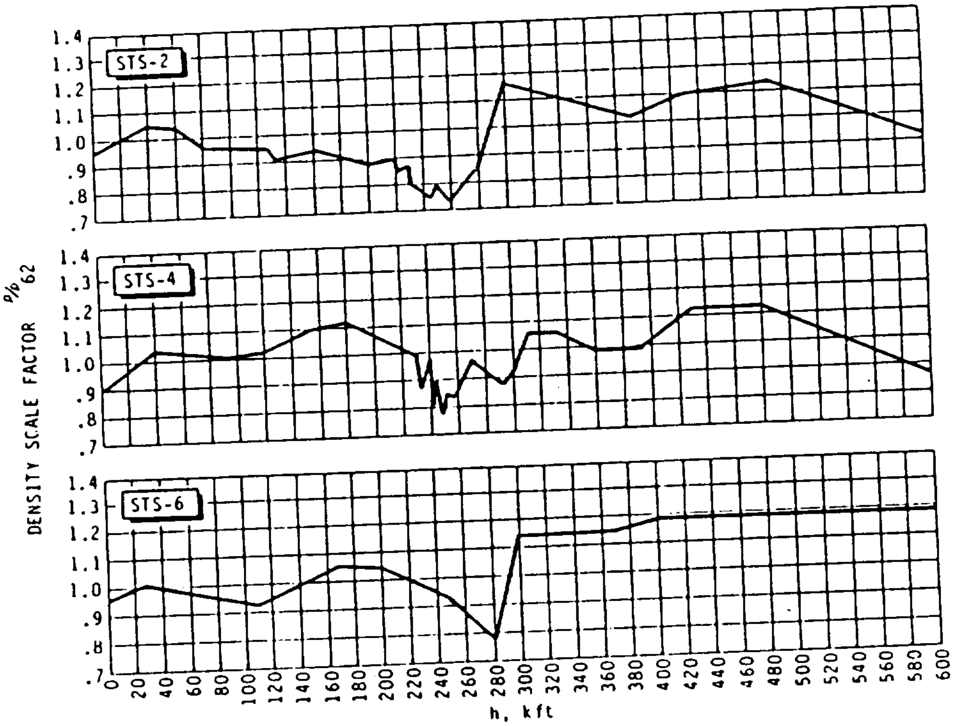


Figure 5.1.5-2 Shuttle Atmospheric Profiles

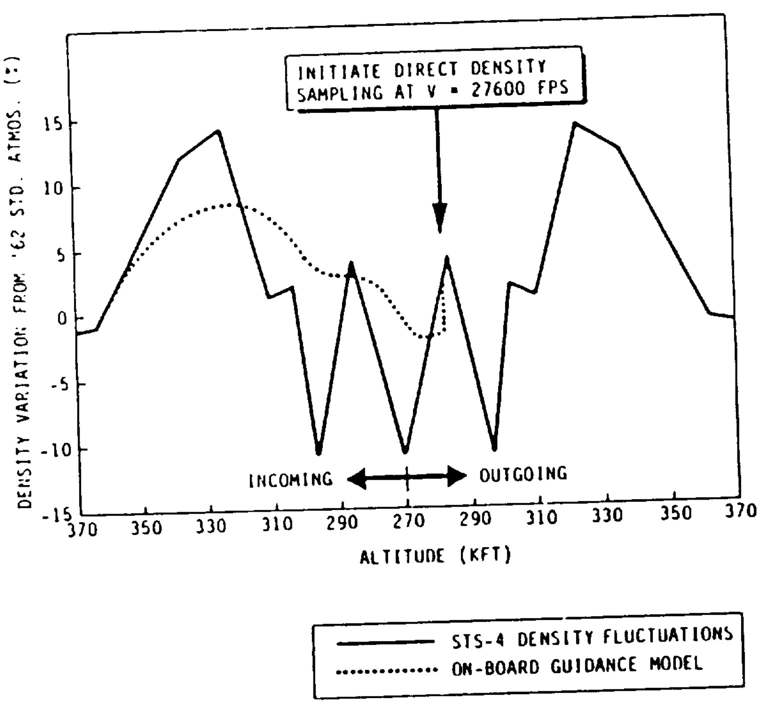


Figure 5.1.5-3 Atmospheric Density Feedback

5.1.5.3.1 Apogee Guidance

Because of the highly non-linear nature of the problem, the basis for apogee targeting is a set of numerical partials derived from previous predictions. Depending on the quantity and freshness of this data set a first order or second order solution is derived which satisfies the target apogee as a function of the vertical component of velocity-to-go. As the vertical component gets very small a threshold test zeroes it out to prevent extraneous roll holds.

5.1.5.3.2 Wedge Angle Targeting

Orbit plane alignment is controlled by steering to a nominal inclination and ascending node simultaneously. This reduces net post-aero plane correction requirements below that required by inclination-only targeting schemes. In essence, the guidance law minimizes the wedge angle between its current orbit plane and a specified target plane.

As illustrated in Figure 5.1.5-4, this targeting scheme works as follows: When guidance predicts a new trajectory the velocity at atmospheric exit is stored. This vector is compared with the desired target orbit plane (specified by inclination and ascending node) and its out of plane component, V_{err}, is computed. This V_{err} is input directly into the lateral guidance loop which attempts to steer it to zero.

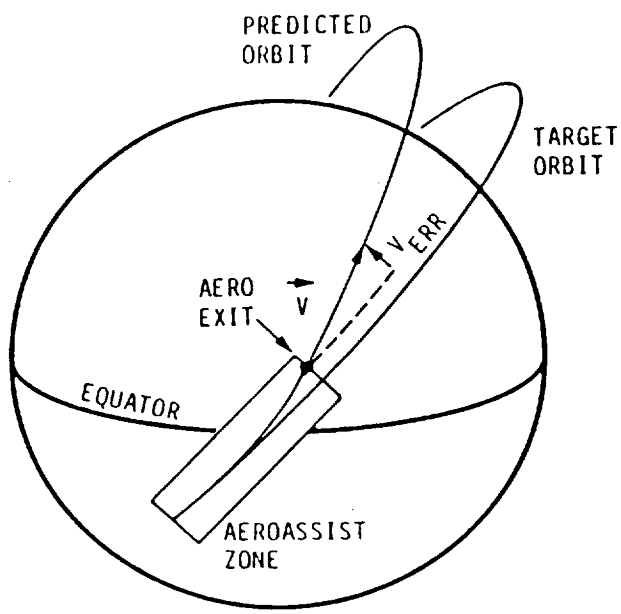


Figure 5.1.5-4 Wedge Angle Targeting

5.1.5.4 Roll Controller

The roll controller integrates velocity targets from guidance along with sensed drag deceleration data to derive vehicle attitude targets and hold durations.

Upon completion of a guidance update cycle the horizontal and vertical velocity-to-go targets are combined vectorally to give a net lift vector target (Figure 5.1.5-5). The orientation of this vector defines the desired attitude of the vehicle's lift vector, its magnitude gives the lateral sensed velocity target.

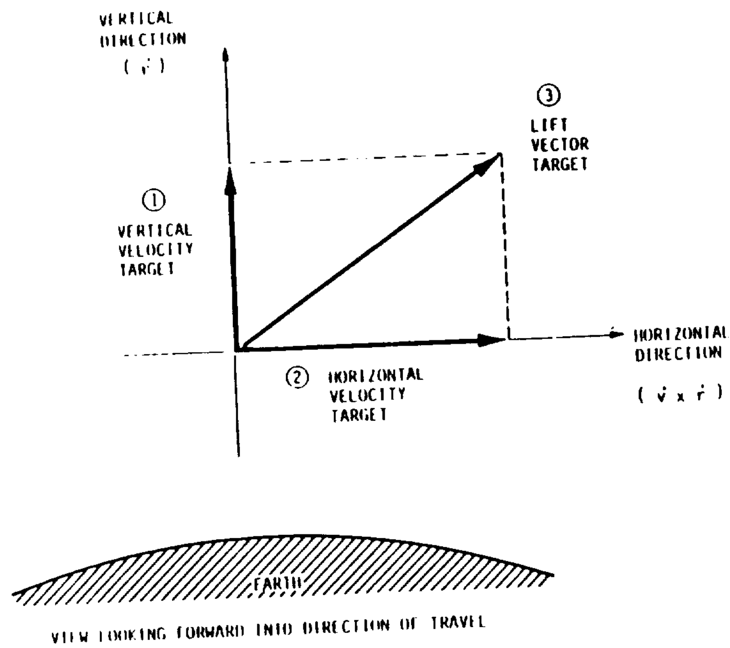


Figure 5.1.5-5 Lift Vector Targeting

Once a commanded roll attitude is computed the software must decide how to get there from its present attitude. If the vehicle is currently holding in roll (zero rate) the shortest path determines which direction to go. If, however, the vehicle is already moving a deadband test is used to determine whether a change of direction (roll reversal) would acquire the target attitude more quickly. Currently an angular value of 110° is used for this roll reversal tolerance.

As the vehicle is rolling to the desired attitude, measurements of drag acceleration by the onboard navigation package are compared against those expected by the guidance model to estimate atmospheric density shifts. Using targets from the last two guidance updates, these observed density shifts are used to adjust the vertical velocity target and, consequently, the target attitude. This simple density tracking function, operating on a 1 second cycle, supplements the more precise guidance update process (operating on a 10 second cycle) to keep the vehicle in step with the quite rapid fluctuations observed in shuttle atmospheres.

Once the vehicle reaches the desired roll attitude, a roll hold is initiated and the onboard accelerometers integrate the sensed lift force into a lateral relative velocity. When this lateral velocity equals the magnitude of the velocity-to-go target the roll hold is terminated and the vehicle initiates its lift-nulling continuous roll which it maintains until a new roll hold attitude is generated by guidance.

5.1.6 Aero-Assist Simulation

To test the performance of the guidance and control scheme described above, the basic package was integrated into a 4 degree of freedom (3 degrees translation, 1 degree rotation in roll) computer simulation. Time history plots of several key trajectory parameters are shown in Figure 5.1.6-1 for an aeroassist simulation utilizing the STS-6 atmosphere.

Different environmental parameters were varied to determine the response of the system. These included 12 shuttle atmosphere profiles to test rapid density fluctuation response, angle of attack errors, position and velocity (targeting) errors and navigation errors. After testing against L/D's of 0.12, 0.08 and 0.06 an extensive data base has been developed which validates the G&N algorithm as well as the general concept of low L/D. The results of individual test cases will be described below.

A summary of the environmental and vehicle characteristics is shown in Table 5.1.6-1.

Table 5.1.6-1 Vehicle Characteristics

ALL VEHICLES

L/D	- 0.116
ANGLE OF ATTACK	- 7.23°
MAX ROLL RATE	- 9°/SEC
ROLL DEADBAND	- 0.2°
TARGET INCLINATION	- 28.5°

VEHICLE UNIQUE	GROUND BASED	SPACE BASED (PAYLOAD - 7.5K CAPSULE)
BALLISTIC COEF.	- 3.78 LB/FT^2	- 6.52 LB/FT^2
RCS THRUST	- 25 LB EACH (3 JETS*)	- 100 LB EACH (3 JETS*)
RCS ISP	- 230 SEC	- 378 SEC
RCS LEVER ARM	- 7.75 FT	- 8.92 FT
ROLL INERTIA	- 13200 SLUG-FT^2	- 23300 SLUG-FT^2
TARGET APOGEE	- 140 NM	- 245 N.M. (25 N.M. BELOW STATION)
ROLL ACCEL.	- 2.52 DEG/SEC^2	- 6.58 DEG/SEC^2

(* NOTE: ONE RCS ROLL JET ASSUMED FAILED OFF)

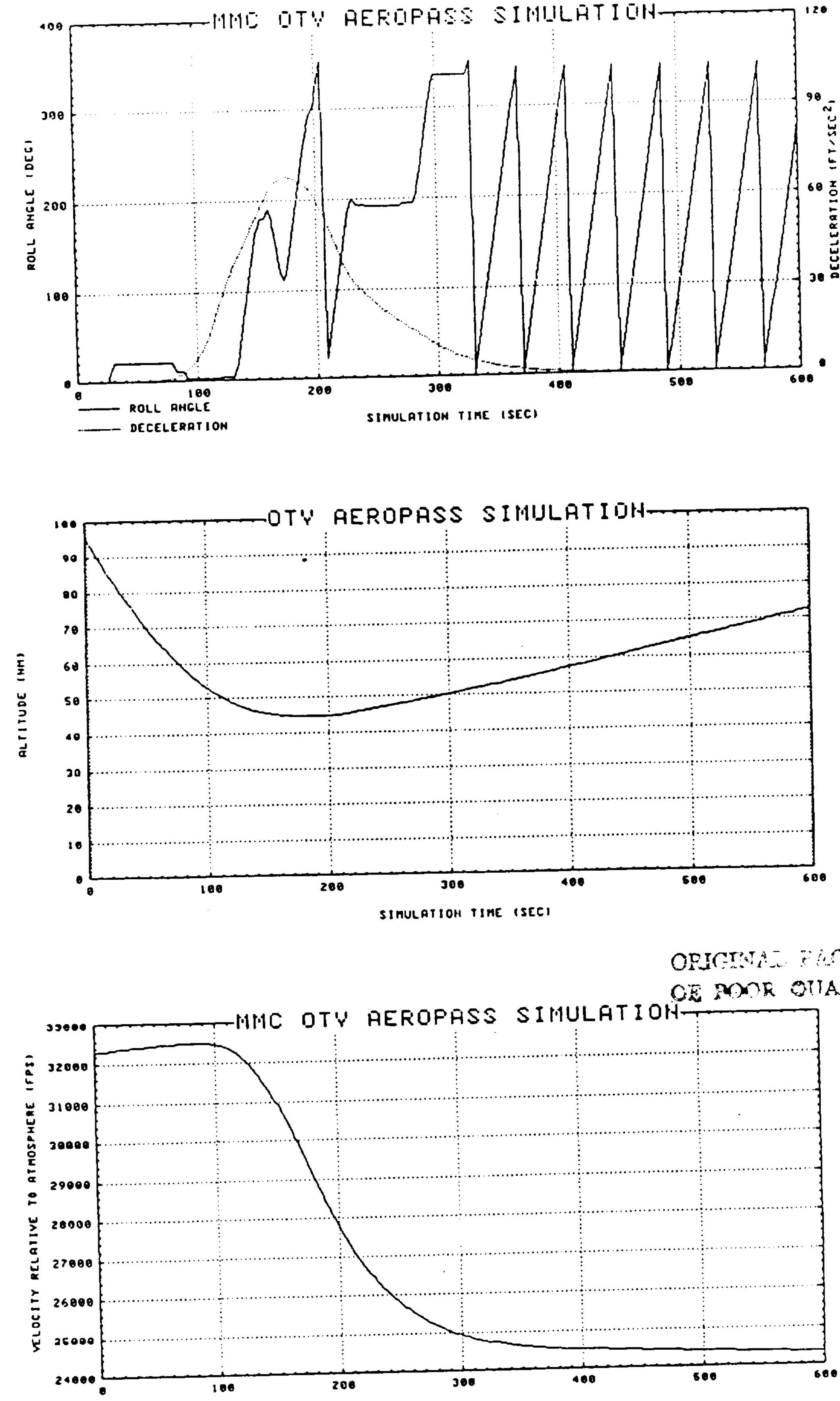


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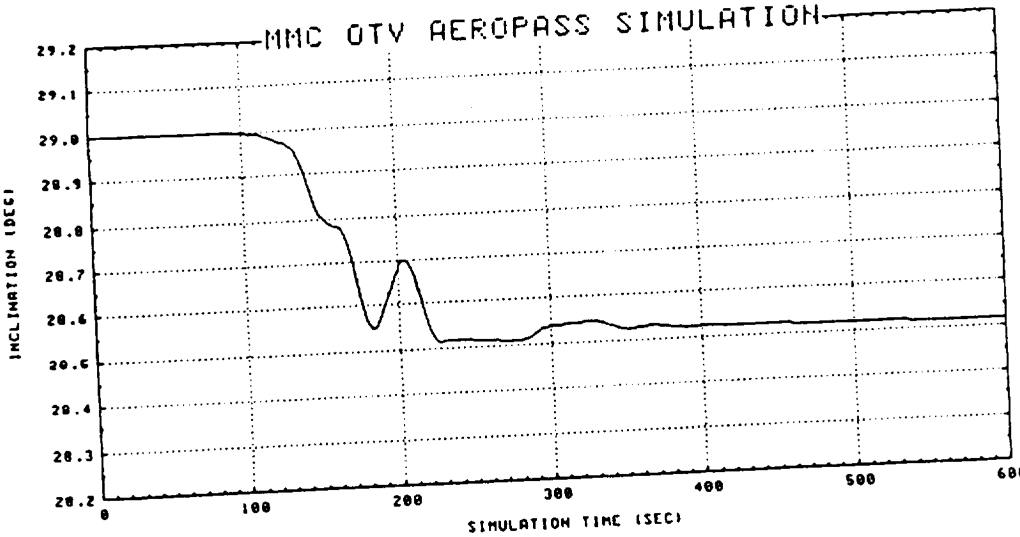
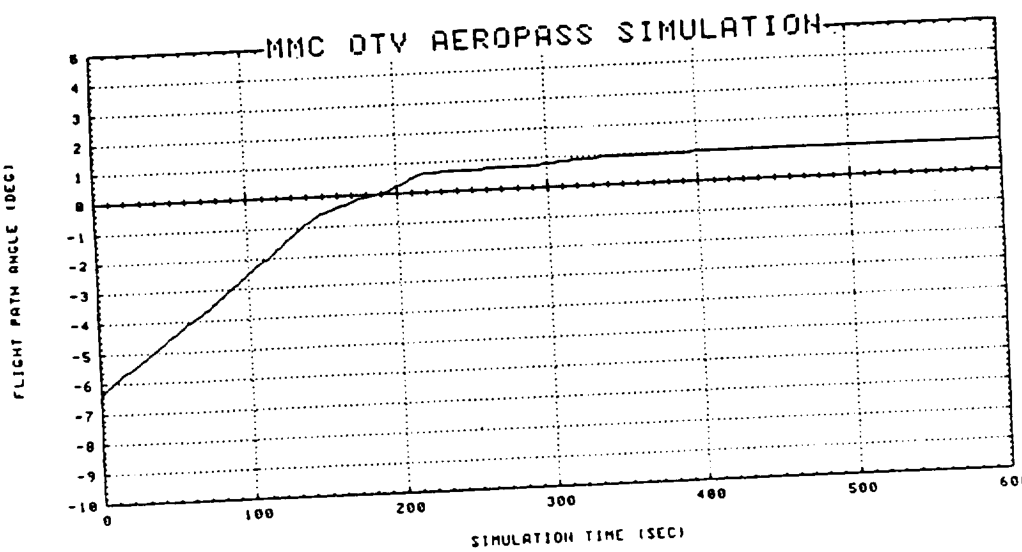
Figure 5.1.6-1 Aeroassist Simulation Time Histories (STS-6 Atmosphere)





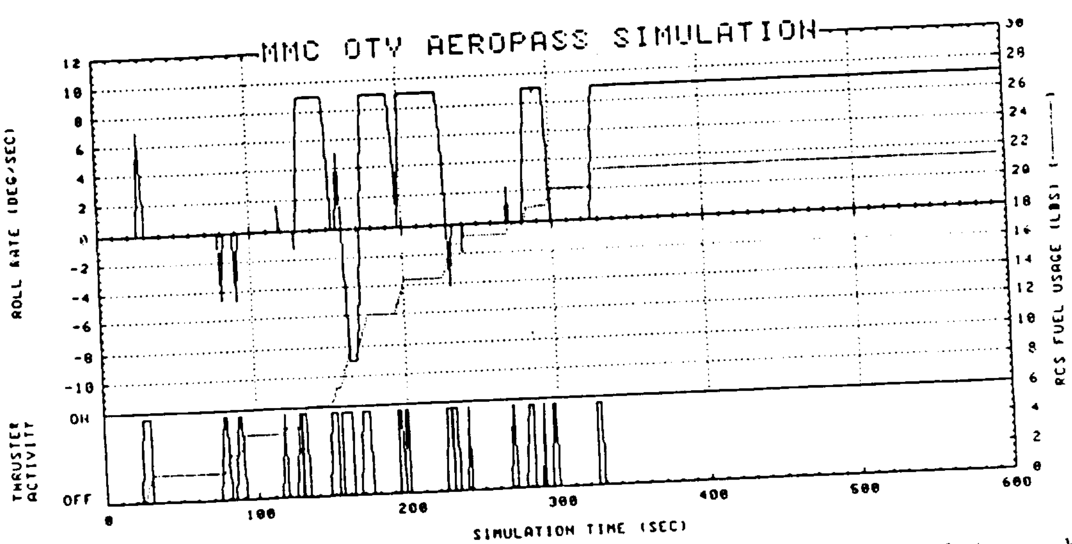


Figure 5.1.6-1 Aeroassist Simulation Time Histories (STS-6 Atmosphere) (Continued)

All simulations shown are for a geosynchronous orbit return to a 140 nmi shuttle retrieval orbit. This ground-based mission is the most demanding because the targeted apogee is much lower than that for space-based (140 vs 245 nmi) leaving a smaller margin of error.

5.1.6.1 L/D = 0.12 Run Summaries

Investigation of the baseline 0.12 L/D condition is summarized in Tables 5.1.6-2, 5.1.6-3, and 5.1.6-4. This data base strives to exercise the aero-guidance in the face of the most important environmental variations. Definition of quantities in the tables is contained in Paragraph 5.1.6.4.

Table 5.1.6-2 shows results of flying through 12 shuttle atmospheres. Table 5.1.6-3 uses the 4 worst shuttle atmospheres (STS-1, 7, 9, 13) in combination with angle of attack shifts of plus and minus 2^{o} (consistent with C.G. and aerodynamic uncertainty analysis earlier). Table 5.1.6-4 uses the same 4 shuttle profiles with a flight path angle variation of $\pm.02^{o}$ (equaling the $\pm.16$ nmi variation in perigee altitude variation which was derived in the aero-entry error analysis chart).

Together this set of data shows the robustness of the predictor-corrector guidance scheme and the low lift baseline. The worst circularization Delta-V is 306 FPS which is 65 FPS above the nominal value of 241 fps. This requires 30 lb of additional rocket fuel to correct which is a trivial amount. The largest Delta-V for correction of the wedge angle is 11 fps which is likewise an extremely small quantity. The net phasing variation is 3.21^{o} which slightly exceeds the single pass phasing orbit correction capability of 3.01^{o}. This would leave a phasing residual of $\pm0.1^{o}$ which translates to a 5 nmi in-track error and a shuttle rendezvous timeline variation of ±6 min. This seems like a rather small uncertainty, however an alternate option is to baseline two passes in the post-aero phasing orbit with the resulting correction capability of 6.02^{o} covering completely the derived variation.

Table 5.1.6-2 L/D 0.12 With Shuttle Atmospheres

STS ATMOS	Δ APO (N.M.)	ΔINCLIN (DEG)	Δ WEDGE (DEG)	ΔPHASE (DEG)	ΔV CIRC (FPS)	ΔV WEDGE (FPS)	PERIGEE (N.M.)	MAX LOAD (g's)	Q_{max} (PSF)	HEATING RATE MAX (BTU/FT^2-SEC)	HEATING RATE INTEGRATED (BTU/FT^2)	RCS USAGE (LB)
NOM	0.1	-.0033	.0110	0.00	241	5	6.5	2.60	9.74	86.74	11900	2.9
STS-1	7.7	-.0015	.0155	-0.28	232	7	21.4	2.67	10.02	90.41	12020	12.1
STS-2	-9.6	.0099	.0154	0.65	271	7	-2.3	2.32	8.70	82.73	12590	12.4
STS-3	-2.2	.0093	.0121	0.06	236	5	9.1	2.65	8.94	89.11	11890	24.0
STS-4	-13.2	.0124	.0155	0.62	276	7	-1.7	2.79	10.49	88.61	11980	20.0
STS-5	-0.4	.0549	.0113	0.33	225	5	17.1	2.53	9.50	88.02	12410	21.1
STS-6	-3.7	.0230	.0251	0.34	245	11	6.3	2.60	9.75	86.99	12300	13.3
STS-7	2.8	.0004	.0037	-0.54	263	2	-0.5	2.74	10.27	89.26	11330	19.2
STS-8	-1.2	.0173	.0181	0.19	249	8	3.3	2.46	9.25	86.37	12190	17.8
STS-9	-4.5	.0045	.0141	0.37	266	6	-4.0	2.30	8.64	83.37	12310	17.2
STS-11	12.8	-.0027	.0076	-0.97	257	3	12.5	2.55	9.58	86.22	11650	16.9
STS-13	-16.0	.0208	.0218	0.62	306	10	-15.0	2.82	10.58	89.92	11910	21.0
STS-14	-3.9	.0060	.0217	0.08	257	10	-0.4	2.70	10.12	91.33	11710	28.8
AVERAGE	6.5	.0136	.0152	0.42	257	7	3.8	2.59	9.74	87.70	12024	18.7

Table 5.1.6-3 L/D 0.12 With Angle of Attack Errors

	STS ATMOS.	Δ APO (N.M.)	ΔINCLIN (DEG)	Δ WEDGE (DEG)	ΔPHASE (DEG)	ΔV CIRC (FPS)	ΔV WEDGE (FPS)	PERIGEE (N.M.)	MAX LOAD (g's)	Qmax (PSF)	HEATING RATE MAX (BTU/FT^2-SEC)	HEATING RATE INTEGRATED (BTU/FT^2)	RCS USAGE (LB)
	NOM	0.1	-.0033	.0110	0.00	241	5	6.5	2.60	9.74	86.74	11900	2.9
ΔANGATTK = +2.0°	STS-1	-17.5	.0114	.0213	1.00	285	10	-2.0	2.65	10.02	90.96	12250	18.8
	STS-7	7.5	.0000	.0094	-0.67	252	4	10.1	2.77	10.45	92.92	11440	17.9
	STS-9	-1.5	.0150	.0162	0.17	259	7	-2.7	2.18	8.32	83.35	12320	19.6
	STS-13	-4.3	.0021	.0042	0.15	256	2	0.7	2.82	10.65	88.56	12020	24.8
ΔANGATTK = -2.0°	STS-1	7.0	-.0030	.0159	-.36	239	7	16.7	2.69	10.03	89.55	11690	22.0
	STS-7	25.6	-.0036	.0147	-1.88	281	7	12.0	2.76	10.31	91.60	11220	16.9
	STS-9	-14.6	-.0018	.0212	0.90	292	9	-7.9	2.31	8.64	82.27	12170	18.5
	STS-13	-4.1	.0037	.0106	0.44	238	5	10.4	2.76	10.35	89.18	12180	22.7
	AVERAGE	10.3	.0051	.0142	0.70	263	6	4.7	2.62	9.85	88.55	11936	20.2

Table 5.1.6-4 L/D 0.12 With Flight Path Angle Errors

	STS ATMOS.	Δ APO (N.M.)	ΔINCLIN (DEG)	Δ WEDGE (DEG)	ΔPHASE (DEG)	ΔV CIRC (FPS)	ΔV WEDGE (FPS)	PERIGEE (N.M.)	MAX LOAD (g's)	Qmax (PSF)	HEATING RATE MAX (BTU/FT^2-SEC)	HEATING RATE INTEGRATED (BTU/FT^2)	RCS USAGE (LB)
	NOM	0.1	-.0033	.0110	0.00	241	5	6.5	2.60	9.74	86.74	11900	2.9
ΔFLT PATH = +.02°	STS-1	17.4	.0080	.0221	-1.15	263	10	13.8	2.68	10.07	90.75	11720	12.8
	STS-7	23.0	.0049	.0103	-1.58	267	5	17.0	2.75	10.31	91.90	11350	16.3
	STS-9	-20.2	.0232	.0248	1.33	302	11	-7.8	2.33	8.73	83.10	12470	17.9
	STS-13	-6.5	.0064	.0131	0.42	251	6	5.7	2.78	10.44	89.88	12050	23.8
ΔFLT PATH = -.02°	STS-1	21.7	-.0017	.0109	-1.29	255	5	22.1	2.72	10.19	89.82	11880	20.6
	STS-7	-2.3	-.0059	.0059	-0.22	254	3	-0.1	2.67	10.04	88.79	11410	22.0
	STS-9	-11.5	.0061	.0158	0.69	287	7	-8.4	2.37	8.91	84.02	12240	20.6
	STS-13	-17.6	-.0052	.0099	0.87	298	4	-9.0	2.79	10.47	89.85	12150	20.6
	AVERAGE	15.0	.0077	.0141	0.94	272	6	4.2	2.64	9.90	88.51	11909	19.3

The remaining trajectory parameters (g-loading, dynamic pressure and aero-heating) all lie within basic design limits used to size the aerobrake. The maximum quantity of RCS hydrazine fuel required to perform aero roll maneuvers reaches a high value of 29 lb. This represents a fairly small requirement for a 6000 lb vehicle and shows the efficiency of the continuous roll concept.

Overall, the simulation data base shows the soundness of the guidance algorithm and the workability of the 0.12 L/D.

5.1.6.2 L/D = 0.08 Run Summaries

Because the L/D = 0.12 requirement was derived on the basis of a 33% margin on the aero-entry error analysis, a natural question to ask is what L/D results from a zero-margin analysis. This form of error assessment results in a L/D = 0.08. Several runs were made at this L/D as summarized in Tables 5.1.6-5 and 5.1.6-6.

The first set shows results of the 12 shuttle atmospheres, the second combines the 4 worst shuttle atmospheres with minus angle of attack and entry flight path variations (worst directions).

Table 5.1.6-5 L/D 0.08 With Shuttle Atmospheres Only

STS ATMOS.	Δ APO (N.M.)	ΔINCLIN (DEG)	Δ WEDGE (DEG)	ΔPHASE (DEG)	ΔV CIRC (FPS)	ΔV WEDGE (FPS)	PERIGEE (N.M.)	MAX LOAD (g's)	Qmax (PSF)	HEATING RATE MAX (BTU/FT^2-SEC)	HEATING RATE INTEGRATED (BTU/FT^2)	RCS USAGE (LB)
NOM	-2.2	-.0039	.0116	0.00	241	5	6.6	2.57	9.68	86.83	11930	8.1
STS-1	19.3	-.0057	.0193	-1.11	255	9	19.7	2.66	10.01	89.99	11940	18.6
STS-2	-14.0	.0002	.0227	0.59	304	10	-16.0	2.30	8.68	83.00	12220	24.9
STS-3	2.9	.0142	.0166	-0.27	243	7	10.7	2.56	9.63	88.33	11860	17.7
STS-4	-10.8	.0017	.0100	0.74	251	4	9.6	2.68	10.07	88.18	12240	17.4
STS-5	13.6	.0271	.0284	-0.52	240	13	22.5	2.51	9.45	88.06	12370	15.4
STS-6	9.6	-.0105	.0185	-0.28	235	8	21.3	2.56	9.65	85.27	12260	14.1
STS-7	32.4	.0000	.0148	-2.24	284	7	17.0	2.56	9.62	89.08	11390	23.9
STS-8	-1.9	.0015	.0174	0.11	249	8	2.5	2.42	9.11	85.98	12120	23.2
STS-9	-15.9	.0004	.0188	0.57	260	8	0.8	2.31	8.71	81.63	12390	18.8
STS-11	-2.4	-.0046	.0082	-0.09	248	4	2.7	2.43	9.15	85.68	11860	17.6
STS-13	-6.5	-.0023	.0193	-1.11	259	9	19.7	2.66	10.01	89.99	11940	18.6
STS-14	-2.1	.0061	.0156	0.12	239	7	8.3	2.84	10.68	90.97	11770	26.5
AVERAGE	11.0	.0062	.0175	0.65	256	8	9.9	2.54	9.56	87.18	12030	19.7

Table 5.1.6-6 L/D 0.08 With Angle of Attack and Flight Path Angle Errors

	STS ATMOS.	Δ APO (N.M.)	ΔINCLIN (DEG)	Δ WEDGE (DEG)	ΔPHASE (DEG)	ΔV CIRC (FPS)	ΔV WEDGE (FPS)	PERIGEE (N.M.)	MAX LOAD (g's)	Qmax (PSF)	HEATING RATE MAX (BTU/FT^2-SEC)	HEATING RATE INTEGRATED (BTU/FT^2)	RCS USAGE (LB)
	NOM	-2.2	-.0039	.0116	0.00	241	5	6.6	2.57	9.68	86.83	11930	8.1
ΔANGATTK = -2.0°	STS-1	43.7	.0014	.0063	-2.85	299	3	19.4	2.58	9.66	88.65	11800	23.6
	STS-7	43.6	.0194	.0211	-2.98	302	9	17.9	2.57	9.63	89.11	11350	20.6
	STS-9	6.9	-.0019	.0218	0.23	231	10	20.1	2.10	7.87	80.24	12650	20.5
	STS-13	17.0	.0620	.0629	-0.89	251	28	19.6	2.64	9.91	86.73	12060	25.0
ΔFLT PATH = -.02°	STS-1	-0.9	.0017	.0085	0.12	229	4	14.2	2.65	9.96	89.23	12040	23.6
	STS-7	37.5	.0148	.0149	-2.67	296	7	15.4	2.64	9.93	89.20	11260	17.8
	STS-9	3.8	.0013	.0139	0.23	230	6	17.8	2.30	8.67	81.91	12530	19.6
	STS-13	-7.2	-.0070	.0142	0.32	260	6	1.4	2.74	10.30	88.51	12050	28.6
	AVERAGE	20.1	.0137	.0205	1.29	262	9	15.7	2.53	9.49	86.70	11968	22.4

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The results are generally the same as with the L/D = 0.12 case with most errors being slightly larger but still manageable. This shows that the L/D = 0.08 case would probably be acceptable for OTV operations. However, since weight savings in going to the lower L/D would be marginal (analysis indicates we are near the bucket of the weight vs L/D curve), the baseline L/D will remain at 0.12. The favorable 0.08 results do demonstrate a very healthy margin in the baseline L/D.

5.1.6.3 L/D = 0.06 Run Summaries

A limited set of shuttle atmospheres was run with an L/D of 0.06 illustrated in Table 5.1.6-7. These consisted of the 4 worst case shuttle atmospheres in conjunction with negative dispersions on angle of attack and flight path angle (most sensitive direction). As can be seen in the chart the atmosphere-only and delta flight path angle runs were successful but one angle of attack run skipped out with an apogee error of 1500 nmi.

Table 5.1.6-7 L/D 0.06 Results

	STS ATMOS.	Δ APO (N.M.)	ΔINCLIN (DEG)	Δ WEDGE (DEG)	ΔPHASE (DEG)	ΔV CIRC (FPS)	ΔV WEDGE (FPS)	PERIGEE (N.M.)	MAX LOAD (g's)	Qmax (PSF)	HEATING RATE MAX (BTU/FT^2-SEC)	HEATING RATE INTEGRATED (BTU/FT^2)	RCS USAGE (LB)
	NOM	-2.3	-.0078	.0142	0.00	241	6	6.7	2.54	9.59	86.48	11940	3.5
	STS-1	17.0	-.0021	.0180	-0.98	254	8	18.2	2.60	9.79	88.38	11200	20.2
	STS-7	-3.4	.0006	.0406	-0.12	265	18	-44.0	2.69	10.13	90.29	11470	22.7
	STS-9	6.6	-.0093	.0524	0.37	225	23	23.0	2.05	7.72	79.85	12820	17.0
	STS-13	4.6	.0066	.0150	-0.09	232	7	18.1	2.73	10.29	88.19	12130	21.7
ΔANGATTK = -2.0°	STS-1	44.8	-.0163	.0189	-2.82	295	8	22.9	2.54	9.54	87.66	11970	29.6
	STS-7	-39.3	.0063	.0176	1.69	371	8	-26.6	2.60	9.80	89.14	11740	18.5
	STS-9	1511.0	.2232	.2235	-117.80	2285	99	44.1	1.90	7.15	78.19	10210	4.6
	STS-13	51.7	.0079	.0224	-3.12	302	10	25.1	2.46	9.25	85.36	12160	22.0
ΔFLT PATH = -.02°	STS-1	-1.1	.0020	.0251	0.08	234	11	11.6	2.61	9.86	89.44	11990	13.9
	STS-7	30.2	.0116	.0120	-2.14	287	9	13.4	2.58	9.72	89.04	11400	25.5
	STS-9	-20.5	-.0017	.0189	1.36	298	8	-5.1	2.15	8.12	80.69	12500	17.0
	STS-13	43.3	-.0186	.0247	-2.68	290	11	24.0	2.68	10.12	87.40	11960	18.6
	AVERAGE	147.8	.0255	.0408	11.10	445	18	10.4	2.47	9.29	86.14	11796	19.3

On the basis of this, the L/D = 0.06 cannot be recommended for use on the OTV but its success in an undispersed (shuttle atmosphere only) environment indicates that an L/D equivalent to 0.06 represents the bottom end of a dispersed control condition. For example, an L/D that had a nominal value of 0.08 with dispersed extremes (due to angle of attack) of 0.06 to 0.10 would be quite acceptable by these results.

5.1.6.4 Aero-Results Definitions

The following list contains definitions of the trajectory parameters shown in the aero-assist run summaries.

DELTA-APO	Error in post-aero apogee (nmi). Nominal apogee is 140 nmi.
DELTA-INCLIN	Error in inclination (deg). Nominal value is 28.5°.
DELTA-WEDGE	Wedge angle measured between nominal target plane and actual exit orbital plane (deg).
DELTA-PHASE	Phase shift (in degrees) of the OTV after circularizing at the nominal target altitude. This is computed with respect to the nominal (undispersed) profile and is a measure of the conditions for the pickup vehicle (shuttle or OMV).
DELTA-V CIRC	This is the net velocity (in fps) required to perform a Hohmann transfer from the apogee of the exit orbit to the desired circular target orbit.
DELTA-V WEDGE	The net velocity required to null the wedge angle error (in fps).
PERIGEE	The altitude of perigee of the exit orbit (nmi).
MAX LOAD	The maximum value of net deceleration encountered in the aeropass, measured in g's.
Q MAX	The maximum value of dynamic pressure encountered in the aeropass, measured in pounds per square foot.
MAX HEAT RATE	The maximum value of stagnation heating (referenced to a 1 ft sphere) encountered in the aeropass, measured in BTU/Ft^2 sec.
INTEGRATED HEAT FLUX	The value of stagnation heating integrated over the entire aeropass (BTU/Ft^2).
RCS USAGE	The amount of RCS propellant (in pounds) expended to perform all roll maneuvers in the aeropass. Vehicle roll inertia is accounted for, pitch and yaw damping requirements are not.

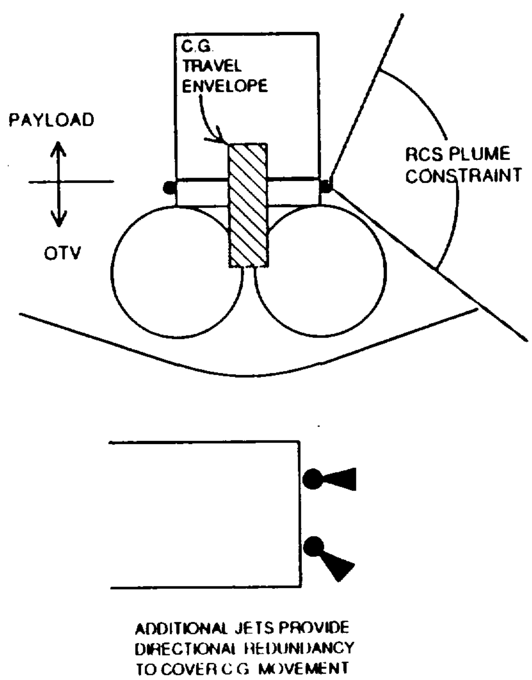
5.1.7 RCS Control Jet Location

Location of the RCS system on the OTV presents problems not encountered by traditional upper stages. Unique aspects of the OTV which impact the RCS system are the ability to accommodate a wide variety of payload shapes (including extended space structures) and the need to control the vehicle in the aerobraking phase.

In order to keep development costs down the OTV has been simplified wherever possible. Because there is no absolute requirement for rendezvous and docking, this capability (and its associated cost) has not been included in the basic design. Thus the only RCS requirement is to provide 3-DOF control and +x translation (the latter provides vernier trim on critical burns and settling thrust for propellant dumps).

Other desirable features are that the RCS system minimize OTV/payload contamination, minimize weight impacts to the vehicle and minimize development costs.

Figure 5.1.7-1 shows jet location Option #1. In this option the RCS jets are mounted in the vicinity of the payload adapter ring on the front of the vehicle. Because of the large c.g. travel which the vehicle experiences and because the RCS jet firing direction is constrained by payload and aerobrake impingement constraints, the vehicle can experience control loss due to the C.G. traveling into the line of action of the jets. This situation is best corrected by adding directionally biased pitch and yaw jets (8 total, which includes redundancy) that the vehicle can switch to if the primary set becomes ineffective.

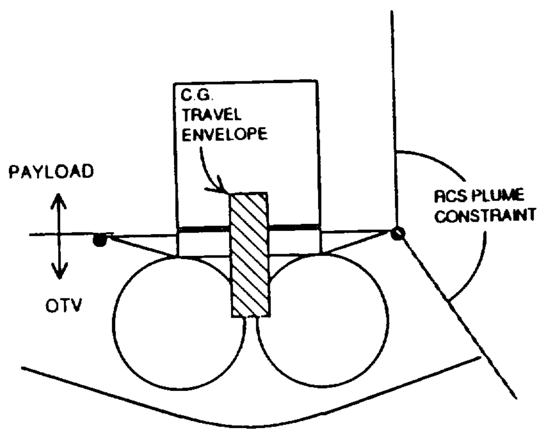


- RCS MOUNTED ON PAYLOAD INTERFACE RING
- RCS JET POINTING LIMITED BY AEROBRAKE AND PAYLOAD IMPINGEMENT
- C.G. TRAVEL ENVELOPE INCLUDES RCS PLANE, CONTROL LOSS IMPACTS
- FIX VIA ADDITION OF 8 JETS TO SPAN C.G. ENVELOPE (SEE LOWER LEFT)
- RECIRCULATION REGION BEHIND AEROBRAKE COMPLICATES ANALYSIS
- PAYLOAD/OTV CONTAMINATION DUE TO RECIRCULATION

Figure 5.1.7-1 RCS Jet Locations - Option #1

This location will require a wind-tunnel test program because of the complicated nature of the recirculation flow behind the aerobrake during aeropass. In addition, this recirculation will cause contamination of the OTV and payload from jet exhaust products being trapped.

Figure 5.1.7-2 shows RCS jet location Option #1A. This option is similar to Option #1, but makes use of struts to move the RCS jets away from the body of the OTV. This increases the directions which the jets can be fired into by reducing the geometric impingement constraint such that single direction jets, firing in a generally forward direction, can be used. This eliminates the need for additional jets as in Option #1, but at a cost of more structural weight for the struts as well as the complexity involved with deployment upon reaching orbit. Overall, this option does not represent an improvement over Option #1.

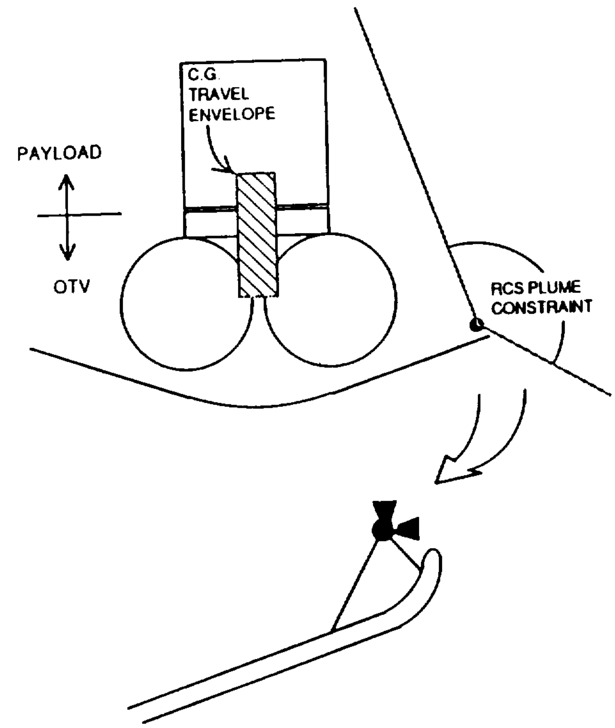


- SIMILAR MOUNTING LOCATION AS OPTION 1
- STRUT STANDOFF LESSENS PAYLOAD IMPINGEMENT CONSTRAINT AND IMPROVES C.G. TRAVEL IMPACTS
- STRUT IMPOSES ADDITIONAL WEIGHT AND COMPLEXITY FOR INITIAL DEPLOYMENT
- RECIRCULATION COMPLICATES CONTROL ANALYSIS
- PAYLOAD/OTV CONTAMINATION

Figure 5.1.7-2 RCS Jet Locations - Option #1A

Figure 5.1.7-3 shows RCS location Option #2. This option mounts the RCS jets on the outer perimeter of the aerobrake. Because of the increased moment arm the torque efficiency allows reduction in the thrust level of the jets. This location has no trouble with c.g. travel. The general brake stiffness looks good for stability purposes. However, the twisting of the aerobrake ribs will require deployable struts that connect the RCS rib with its two neighbors. This, in conjunction with flexible lines and the volume of the jet package itself will make folding the aerobrake a very complicated problem. In addition, when the aerobrake is expended the RCS jets are lost as well.

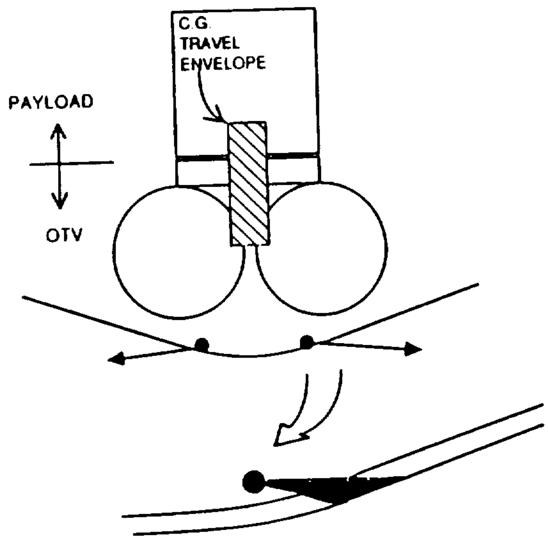
These complications make this option undesirable.



- RCS MOUNTED ON OUTER DIAMETER OF AEROBRAKE
- GOOD TORQUE EFFICIENCY DUE TO LARGE MOMENT ARM
- TIP DEFLECTIONS MANAGEABLE (<2/10 IN)
- MORE DIFFICULT AEROBRAKE DEPLOY:
  - FLEXIBLE LINES
  - DEPLOYABLE STRUTS
  - JET VOLUME CONSTRAINT
- RCS EXPENDED WITH AEROBRAKE
- NO C.G. TRAVEL IMPACTS

Figure 5.1.7-3 RCS Jet Locations - Option #2

Figure 5.1.7-4 shows RCS jet location Option #3. This option mounts the RCS jets into the nose of the aerobrake. The jets are mounted flush with the aerobrake, utilizing scarfed nozzles as in the Space Shuttle and Apollo vehicles. The c.g. travel issue presents no problem for this location. The currently estimated aeropass temperatures are within hardware limits currently designed into the shuttle jets. The major problem with this location is the impact of free stream flow disruption on vehicle stability and control, probably requiring a larger flow test program than any of the other options. This is contrasted against the most viable other option (#1) which itself requires a significant test program due to recirculation effects. Option #3 eliminates the payload plume impingement constraints presented by Option #1 and presents the lowest contamination level of any option. This option would best accommodate the widest variety of payload configurations, such as extended space structures. It also has less hardware than Option #1 (no alternate jets) and thus represents a lighter system. Additionally, it appears very advantageous to provide +x translation capability to the OTV for precision vernier shutdowns on main engine burns as well as providing settling thrust for propellant dumps. These translation jets must fire aft, most probably through the aerobrake, and thus integrate much better with Option #3 than #1.



- RCS JETS MOUNTED IN AEROBRAKE NOSE
- NOZZLES SCARFED TO SURFACE OF BRAKE (NO PROTUBERANCES)
- MAX.TEMP. (2500 DEG) WITHIN CAPABILITY OF JETS
- DISRUPTION OF FREE STREAM FLOW COMPLICATES ANALYSIS
- MINIMIZES PAYLOAD/OTV CONTAMINATION
- NO C.G. TRAVEL IMPACTS
- MAXIMIZES PAYLOAD ENVELOPE (MIN PLUME IMPACT)

RECOMMENDED CONFIGURATION

Figure 5.1.7-4 RCS Jet Locations - Option #3

Overall, Option #3 shows enough promise to be carried as our baseline configuration. In subsequent design analysis efforts, however, flow disruption needs to be tested in a wind tunnel to better evaluate its impacts.

5.1.8 Conclusions

The goal of making the OTV efficient and cost-effective has been addressed for the aero-braking portion of the mission. The best method for controlling the trajectory in this phase is through the use of a lifting brake. The use of entry error analysis has been used to derive an L/D requirement of 0.12. A predictor-corrector guidance scheme was developed which controls exit apogee and orbital plane geometry in the aero-assist. The guidance incorporates density feedback functions to compensate for large atmospheric fluctuations observed in shuttle entries. The overall sizing and timing of guidance is similar to software flying today. Lift management in the aero-phase utilizes continuous roll which results in speedy and efficient trajectory corrections as well as a minimization of RCS propellant requirements. Results of extensive aeropass simulations confirm the robustness of the 0.12 L/D and the aero-guidance scheme. Very favorable results are also indicated at a lower L/D of 0.08.

The most weight optimum solution to the problem of RCS jet location is to locate the jets in the nose of the aerobrake, though wind tunnel work is required to verify this conclusion.

5.2 AEROTHERMAL ANALYSIS

Aerothermal trade studies presented here encompass three major areas. These areas include: 1) the comparison of aerobrake configurations, 2) the examination of using a higher L/D to reduce TPS weight, and 3) the optimum zone for the control corridor within the aeropass envelope.

5.2.1 Aerobrake Design Concepts

As a result of the Phase A study, three aerobrake candidates presented in Figure 5.2.1-1 emerged. The first candidate is a 40 foot rigid tile TPS shaped lifting brake. The second and third candidates employ a TPS combining the rigid tiles on the nose section (the high heating environment) with a flexible fabric skirt. These two configurations are the 44 foot symmetric lifting brake and the 50 foot ballute modulated-drag brake. With the data base developed in Phase A and previous studies, the first objective was to determine the impact of the Rev. 9 mission model on the aforementioned brake sizes. Based on the original STAS mission model, the number of return payloads versus payload length is presented in Figure 5.2.1-2. The symmetric aerobrake was sized to protect the return payloads from direct flow impingement and Figure 5.2.1-2 clearly shows that aerobrake sizing rationale is strongly dependant on the mission model.

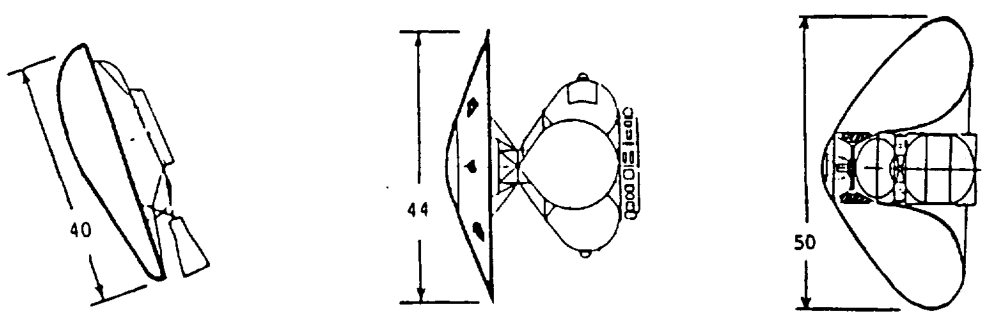
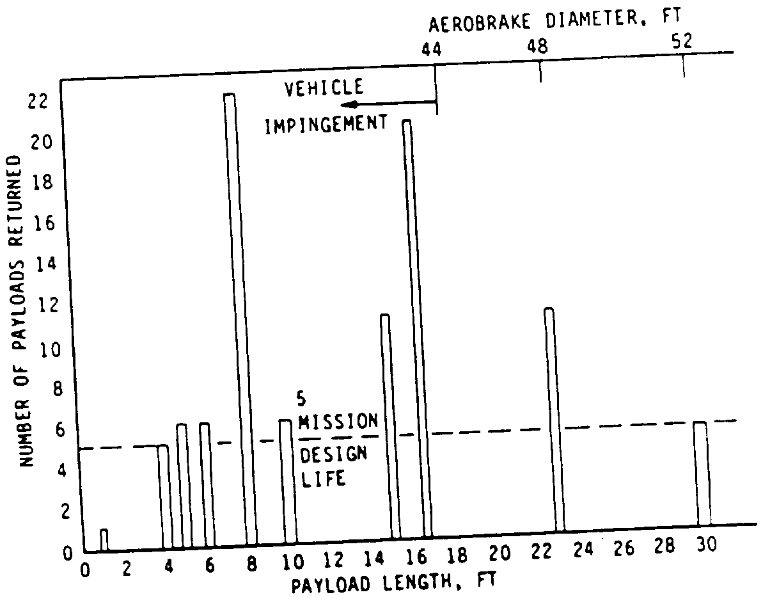


Figure 5.2.1-1 Aerobrake Design Concepts

The information presented in Figure 5.2.1-2 demonstrates that return payload capability sizes the aerobrake. A 30 foot long payload requires a 52 foot diameter brake while a 23 foot long payload requires a 48 foot diameter. The figure shows four design options available for capturing the mission model. The three driver payloads along with their return weights and dimensions are shown in Figure 5.2.1-3. The 30 foot long COMSAT Class IV payload has deployed solar panels of limited strength (0.1 g). As shown in Table 5.2.3-2 aerobraking results in peak decelerations of approximately 3.5 g's. Our analysis shows that this payload can be returned all-propulsive, with the engines operating in the pumped idle mode (thrust $\sim$ 750 lbs. Isp $\sim$ 440 sec.) Thus, no aerobrake would be used and this 30 foot long payload is not an aerobrake design driver. Of the other two missions, the 23 foot long, unmanned servicing mission was selected as the driver mission for aerobrake sizing due to its weight and length.

ORIGINAL PAGE IS
OF POOR QUALITY

- DATA FOR STAS CORE MODEL ONLY
- DATA FOR SYMMETRIC CONICAL BRAKE CONCEPT
- AEROBRAKE SIZED BY FLOW IMPINGEMENT

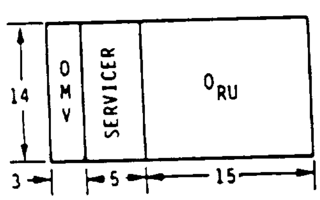


AEROBRAKE SIZE OPTIONS

1) CAPTURE ENTIRE MISSION MODEL WITH ONE 52 FT. DIAM. BRAKE.
2) USE ONE 52 FT., TWO 48 FT. AND USE 44 FT. FOR REMAINDER OF MISSIONS.
3) USE THREE 52 FT. ALONG WITH 44 FT. FOR REMAINDER OF MISSIONS.
4) USE 44 FT. AND 48 FT. BRAKES ONLY, RETURN THE 30 FT. PAYLOADS WITH ALL-PROPULSIVE MANUEVERS.

Figure 5.2.1-2 Return Payload Length

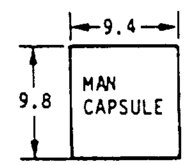
- UNMANNED SERVICING MISSION 2117



WEIGHT:

OMV	4,510 LB
SERVICER	1,290 LB
RETURN P/L	5,500 LB
TOTAL	11,300 LB

- MAN SORTIE (GEO SHACK)
  PAYLOAD 15010
  RETURN WT = 10,000 LB



- COMSAT CLIV RETRIEVAL
  MISSION 1020
  RETURN WT. 10,030 LB
  0.1G REQUIREMENT → PROPULSIVE RETURN

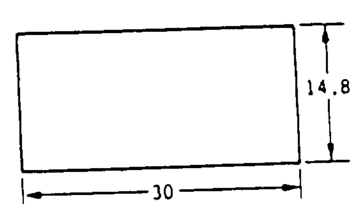
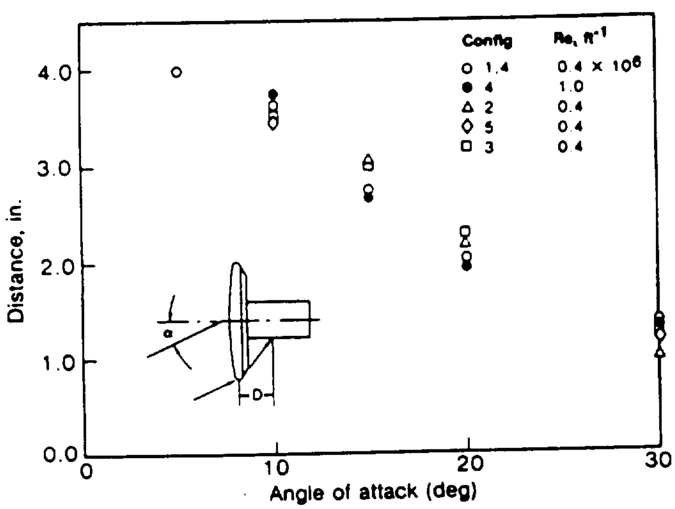


Figure 5.2.1-3 Aerobrake Design Driver Payloads

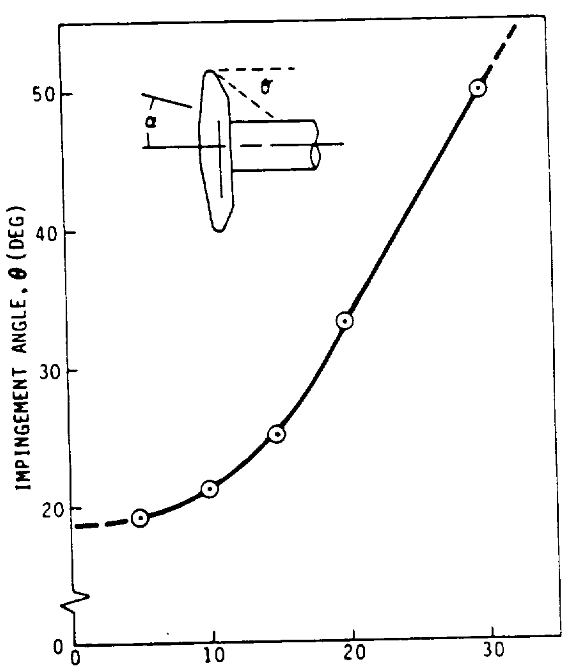
Aerobrake sizing criteria, of which no direct flow impingement is just a part, encompasses many different aspects of vehicle and payload constraints. The "no direct flow impingement" criteria makes use of a single stage, 74K propellant load vehicle with return payload lengths previously discussed. Additionally, two aerobrake sizing requirements address TPS heat flux limits and one addresses TPS packaging constraints. The thermal protection system, TPS, is composed of rigid surface insulation, RSI and/or flexible surface insulation, FSI, having heat flux limits of 50 and 30 BTU/Ft^2-sec, respectively. The RSI/FSI is bonded to RTV which in turn is attached to the vehicle structure. RSI/FSI thickness requirements develop from the limitation of 600°F for the RTV material and a single perigee post pass burn of the returning vehicle. The entire TPS must also be packaged so as to fit in the Shuttle cargo bay, SCB, or the aft cargo carrier, ACC.

Aerodynamic stability, upon returning from GEO, is also a factor in sizing the aerobrake. Center of pressure and center of gravity locations, including a payload center of gravity at the midpoint of the payload, relative to each other must provide sufficient stability for a controlled aeroassist return.

The aerobrake is currently sized to avoid shock impingement at the afterbody. General Dynamics has performed lifting brake wind tunnel tests which included investigation of shock impingement. Results of these tests are presented in Figure 5.2.1-4. For the three candidate aerobrakes flying at angles of attack of 17°, 7.5°, and 0° the flow impingement angles are 28.5°, 20°, and 19°, respectively.



FLOW REATTACHMENT DISTANCE
(REF PAPER AIAA-84-0309)



FLOW TURNING ANGLE

Figure 5.2.1-4 Flow Impingement Data

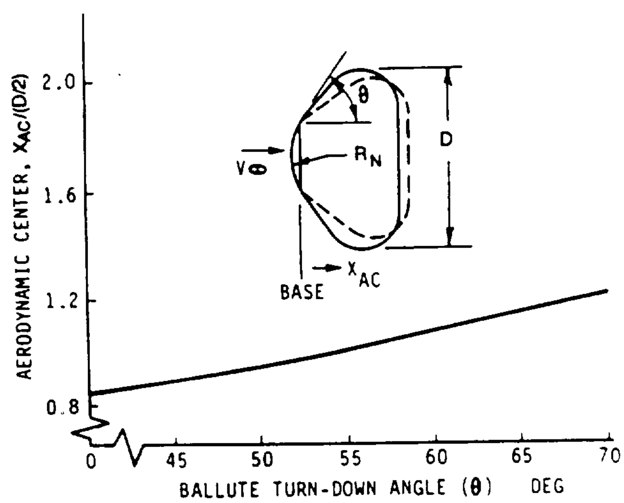
The Phase A studies show that the driving factor in sizing low L/D lifting brakes is flow impingement. For the two candidate lifting brakes, reentry weights, trim conditions, and other parameters necessary for sizing are presented in Table 5.2.1-1. The ballute modulated drag brake must also protect the payload from direct flow impingement, and the data base needed for sizing the ballute aerobrake is presented in Figure 5.2.1-5. Using the information presented here, coupled with the STAS mission model of returning a 23 foot, 11,300 lb payload from GEO, a comparison of the three aerobrake candidates has been prepared.

Table 5.2.1-1 Lifting Brake Sizing Data

	SHAPED BRAKE	SYMMETRIC BRAKE
DRY WEIGHT	9775	7600
RESIDUALS	1025	1025
ϕ_{IMP}	28.5°	20.0°
α	17°	7.5°
C_L	-0.453	0.189
C_D	1.530	1.578
L/D	-0.296	-0.120
c.p.	1.86R	2.07R

- BRAKE DIAMETER SIZING LIMITED BY AERO-STABILITY LIMITS WITH RETURN PAYLOAD
- RETURN WEIGHT, W_{DRY} = 10250
  W_{RESID} = 1025
  $W_{P/L}$ = 11300
  WT = 22575 LBS
- TURN DOWN RATIO = 1.5*
  DRAG MODULATION = 1.271
  TURN DOWN AREA = 1.180
- R_N = 12 FT, θ = 70°
- BALLUTE SIZED SO C.P. - C.G. MARGIN IN MAXIMUM TURNED-DOWN CONDITION IS 5% OF D (BEFORE TURN DOWN)

*TURN DOWN RATIO IS FULLY INFLATED FRONTAL AREA OVER DEFLATED FRONTAL AREA



C.P. AS A FUNCTION OF BALLUTE CONE ANGLE OR TURNDOWN

Figure 5.2.1-5 Ballute Brake Sizing Data

The two lifting brake candidates differ from the ballute candidate in that the driving factor in sizing the aerobrake is flow impingement. In the case of the shaped brake, to return the 23 foot long payload mounted perpendicular to the brake base, the 40 foot base diameter grows to 44 feet. This growth creates a major impact to this vehicle's integral design and delivery to orbit operations. By using a payload adapter which allows the return payload to be canted out of the flow impingement regions, Figure 5.2.1-6, the original 40 foot diameter shaped brake can be used. This results in no brake growth or change to the original design and the payload c.g. can still be positioned to maintain trim conditions. The payload adapter required to cant the payload is more complex than a no-cant adapter, and is likely to weight more. However, it is unlikely to weigh as much as a 4 foot larger aeroshell. As with the shaped brake, the symmetric brake must grow from 44 to 48 feet in diameter to return a 23 foot long payload when mounted in-line with the stage. Although brake growth with this concept does not result in redesign of the stage (as does the shaped brake), it would be desired to maintain a single universal brake size. Following the approach used for the shaped brake, it can be seen, Figure 5.2.1-7, that by using a canted payload interface the original 44 foot symmetric aerobrake is capable of returning the 23 foot payload without causing impingement. Thus, both lifting brakes with the use of a cantable payload mount are capable of returning the 23 foot payload using their original, Phase A brake sizes (i.e., no growth requirement).

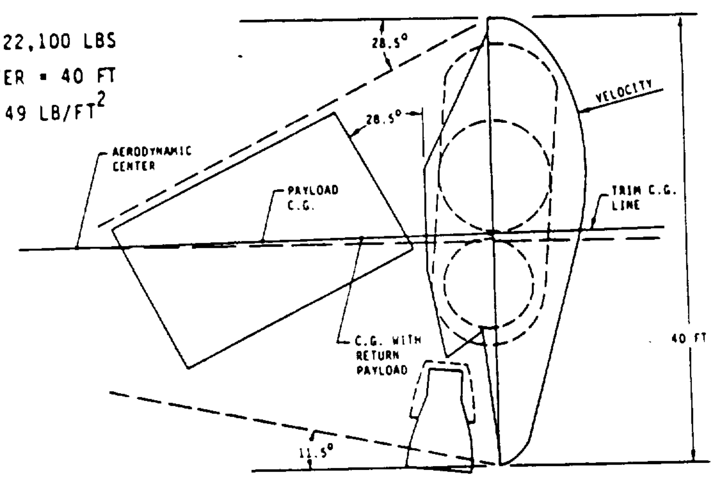


Figure 5.2.1-6 Rigid Brake Canted Payloads

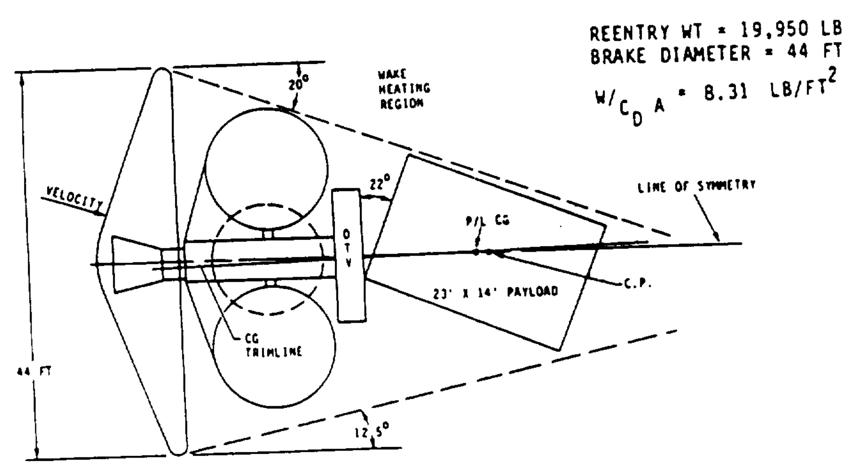


Figure 5.2.1-7 Symmetric Brake Canted Payloads

Unlike the lifting brakes, the ballutes driving factor in sizing the aerobrake are the relative positions of the center of pressure and center of gravity. The ballute has been sized such that the C.P. - C.G. margin during the maximum turned-down condition is 5% of the aerodynamic length. The ballute is inflated with GN_2 during the aeromaneuver and modulation of the internal pressure controls the shape of the ballute and thereby, its drag. The ballute aerodynamic stability is important because a positive static stability margin requirement sizes the ballute diameter and therefore has a large weight impact. To provide a positive static margin, a minimum static margin was selected as 5% of the ballute length based on Phase A results for an aerodynamically stabilized vehicle. Using the parameters presented in Figure 5.2.1-5, Figure 5.2.1-8 shows the c.g. locations of the return payload and stage. The combined c.g. location establishes the minimum ballute size for aero-stability. This location along with the 5% margin gives the desired aerodynamic center which can then be related to the necessary ballute diameter. The resultant ballute diameter required for a 11.3Klb, 23 foot payload return is 69 feet. Its nominal and turned down profile are shown in Figure 5.2.1-9. This diameter increase corresponds to a 38% increase from the 50 foot diameter shown in Figure 5.2.1-1 as the initial point in the study comparisons. The main impact of this brake size is in weight since this is proportional to area which is increased by a factor of approximately 1.9.

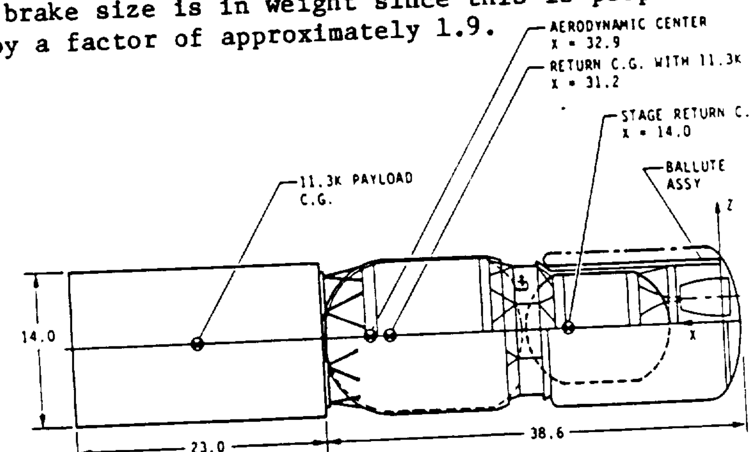


Figure 5.2.1-8 C.G. Locations, Ballute Braked OTV

A summary of the resultant aerobrake diameters to return a 23 foot, 11,300 lb payload for the three aeroassist devices is presented in Table 5.2.1-2. The resultant surface area of both the rigid and flexible TPS and the mean TPS thicknesses for use in the weight trades are also given.

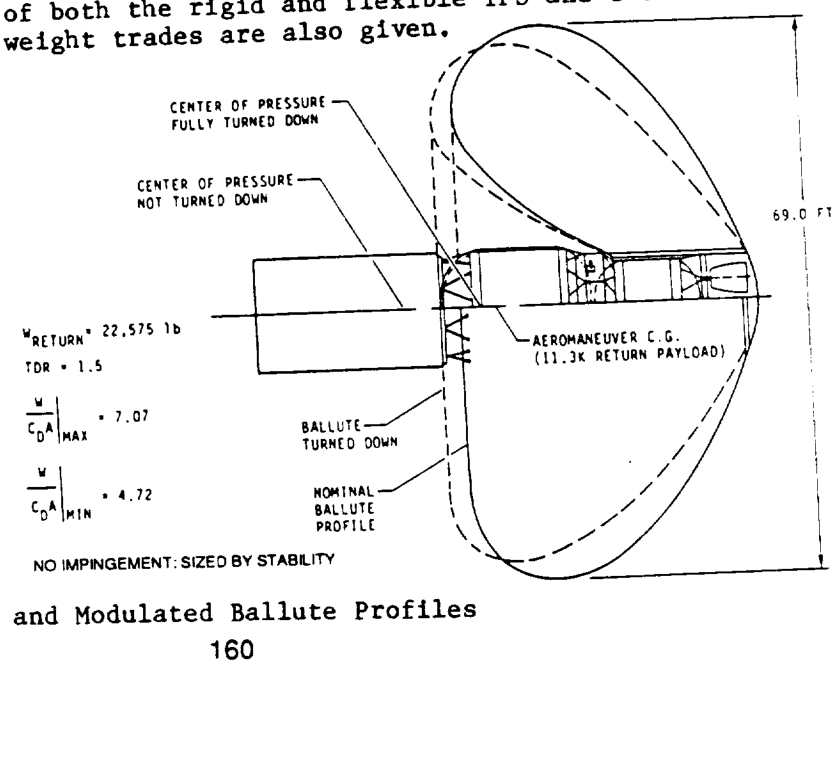


Figure 5.2.1-9 Inflated and Modulated Ballute Profiles

Table 5.2.1-2 Brake Size Comparison

PARAMETER	SHAPED BRAKE	SYMMETRIC BRAKE	BALLUTE BRAKE
BRAKE DIAM (FT)	40	44	69
NOSE RADIUS (FT)	24	11	12
SURFACE AREA, RSI (FT^2)	1570	149	160
SURFACE AREA, FSI (FT^2)	---	1553	4375
MEAN TPS THICKNESS, RSI	t max	t max	t max
MEAN TPS THICKNESS, FSI	t max	t max	0.8t max

In sizing the TPS of the aeroshields, heat flux distributions on the brakes was based on wind tunnel data for 70° conically blunt aeroshells and are presented in Figure 5.2.1-10. In the aeroheating analysis, peak heat fluxes were used to select and evaluate TPS materials while integrated heat loads were used to establish the TPS thickness.

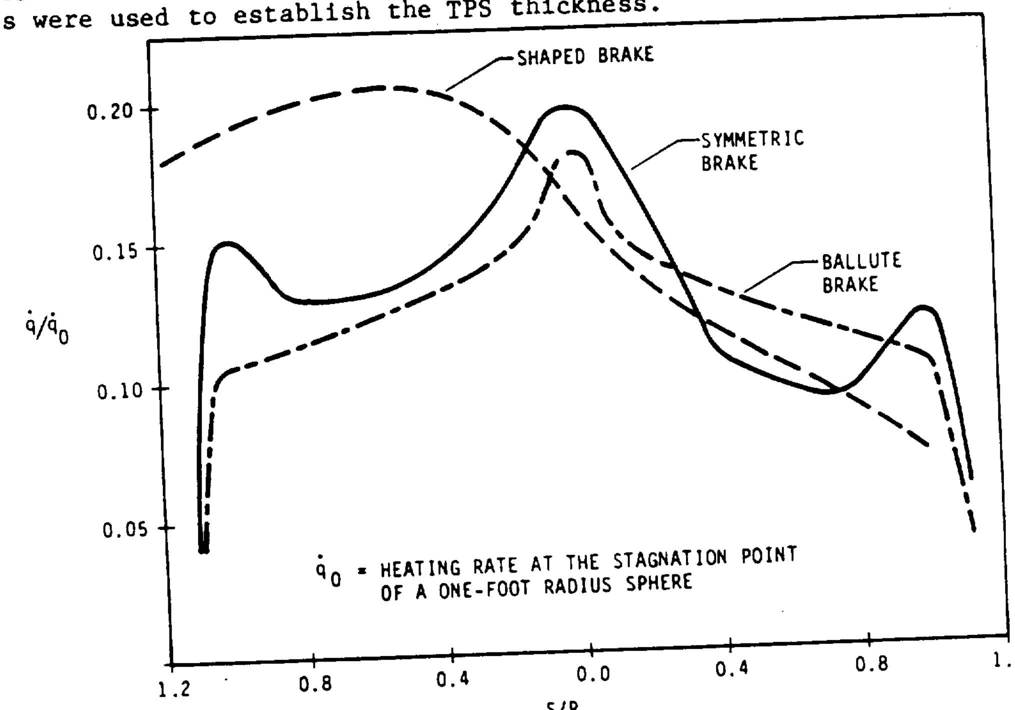


Figure 5.2.1-10 Heat Flux Distribution

The heating factors used for aerothermal predictions of each vehicle concept are shown in Figure 5.2.1-11. For the rigid/flex TPS concepts, a rigid nose section measuring 13.5 feet in diameter was used in determining the S/R value for the flex TPS. In addition to the convective heating, a non-equilibrium radiation component is added to give the total incident heat flux. The heat flux history used for the non-equalibrium radiation component is also shown.

TPS DESIGN PARAMETERS

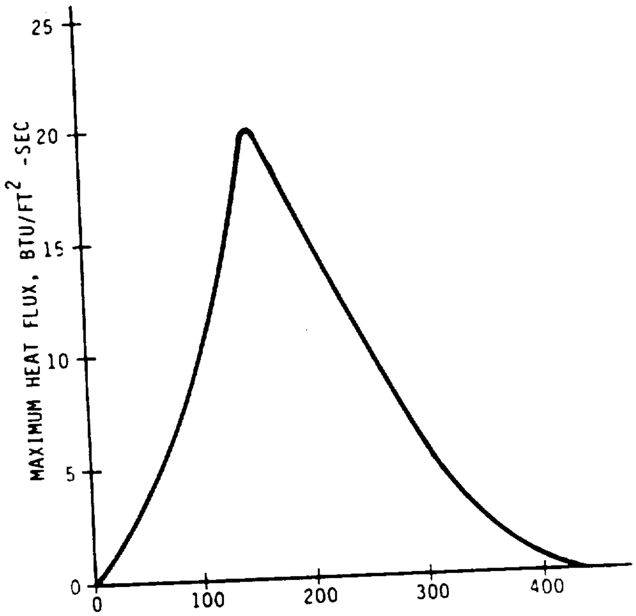
	SHAPED BRAKE	SYMMETRIC BRAKE		BALLUTE BRAKE	
	RSI	FSI	RSI	FSI	RSI
S/R	0.54	0.32	0.0	0.22	0.0
$\dot{q}/\dot{q}_o$	.204	.151	.196	.140	.180
F_S	1.0	1.15	1.0	1.1	1.0
K	0.7	0.7	0.7	0.7	0.7
$\alpha\lambda$	0.3	0.3	0.3	0.3	0.3

F_S = HEATING FACTOR DUE TO SURFACE ROUGHNESS

K = SURFACE CATALYSIS FACTOR

$\alpha\lambda$ = SPECTRAL ABSORPTION COEFFICIENT

$\dot{q}_{TOTAL} = K F_S \dot{q}_{CONV} + \alpha\lambda \dot{q}_{RAD}$



NON-EQUILIBRIUM RADIATION HEATING PROFILE

Figure 5.2.1-11 Heating Factors For Design Environments

The peak heat flux, heat load, and resultant TPS thickness are shown for each concept in Table 5.2.1-3. Heat shield weights reflect protection against both convective and radiative heating. Ballute insulation thickness was tailored with radius, while rigid and flex brakes used constant insulation thickness. More detailed weight breakdowns were presented in MCR-86/NAS8-36108, Contract Extension Final Review. Also included is the heatshield TPS weights, support structure weight, and the total aerobrake weight. To determine the optimum aeroassist device, percent of brake weight compared to the retrieved weight is tabulated. It can be seen that the symmetric brake provides the lightest aeroassist device.

Table 5.2.1-3 Design Criteria Comparison

	SHAPED BRAKE	SYMMETRIC BRAKE	BALLUTE 600°F B.W.	BALLUTE 1500°F B.W.
DIAMETER (FT)	40.0	44	69	69
PEAK STAG q (BTU/FT^2-SEC)	31.5	26.4	20.8	20.8
TOTAL HEAT LOAD (BTU/FT^2)	3926	3805	3049	3049
TPS THICKNESS (IN)				
RSI	0.78	0.77	0.67	0.67
FSI	0.00	0.45	0.50	0.10
TPS WEIGHT (LB)				
RSI	1432	134	127	127
FSI	0	894	2682	1193
STRUCTURE WT (LB)	2239	812	1107	1107
TOTAL AEROBRAKE WT (LB)	3671	1840	3916	2427
BRAKE WT / RETURN WT	0.166	0.092	0.173	0.108

Results of this common basis aeroassist trade is summarized below:

- o Ballute diameter of 69 feet required for return payloads
- o Rigid shaped and 600°F ballute brakes are not weight competitive
- o Stability considerations size ballutes and result in a weight penalty
- o Rigid brake overdesigned for most missions
- o Symmetric brake design gives lowest structural support weight and combined with use of flexible TPS (TABI) for the heatshield is the optimum aerobrake concept

Based upon the above results, it can be seen that the symmetric brake is preferred because it is lightest. Major features it possesses over the other concepts are:

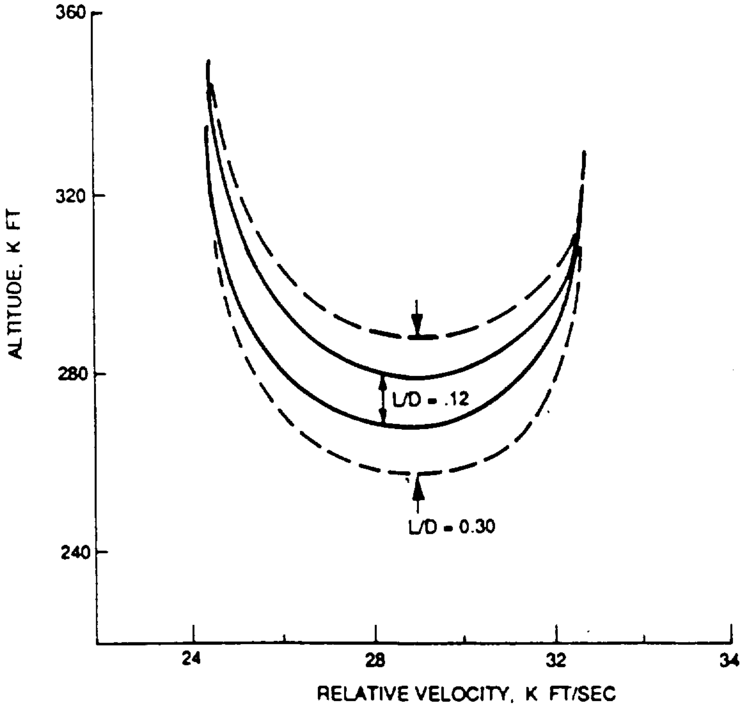
- o VS RIGID SHAPED BRAKE
  - Excess L/D and 100% usage of rigid doubles its aeroassist weight
  - Placing tankage and stage in aeroshell greatly reduces backwall view to space increasing TPS requirements
  - Cannot be ground-based
- o VS BALLUTE DRAG BRAKE
  - Lower controlability
  - 1500°F ballute thermal control and TPS requirements on stage and payload not desirable
  - Not reusable
  - Higher reliability risk

Following completion of this trade study, the OTV Rev. 9 mission model was released. The driver mission in this model was a return payload 15 ft long and weighing 10,000 lb. It should be emphasized that this new driver mission does not affect the above results and conclusions. However, two minor impacts can be noted: 1) a canted payload adapter kit is not required on the lifting brake concepts, and 2) the ballute diameter could be reduced to 62.6 feet and its weight reduced by 200 pounds.

5.2.2 Optimum Zone In Aeropass Envelope

The objective of the following trade is to determine the potential benefits of increasing L/D beyond the minimum required for control. Based on a guidance and navigational error analysis, a 5 nmi control corridor width is adequate to control the OTV. Results show that an L/D of 0.12 gives the desired 5 nmi corridor. By increasing L/D the operational corridor width increases. Flying a continuous lift-down GEO return trajectory enables the OTV to pass at higher altitudes. This results in lower heating rates and g-loads. By flying a continuous lift-up trajectory the OTV flies through the bottom of its operational corridor which decreases the time duration of aeroheating.

The effect of L/D on flight corridor width is shown in Figure 5.2.2-1 for an L/D of 0.12 and 0.30. With an L/D of 0.30, the 5 nmi corridor required for control can be flown at various altitudes throughout the 15 nmi aeropass envelope achievable. Until now we have based aerobrake design for the L/D = 0.3 shaped brake on the thermal environment of a $\pm$ 2.5 nmi corridor about the aeropass midpoint. This study investigates the possible benefit of flying either high or low in the achievable corridor.



L/D	ENVELOPE (N. MILES)
0.12	5.0
0.30	15.0

W/C_D A CONSTANT

Figure 5.2.2-1 Aeropass Envelopes

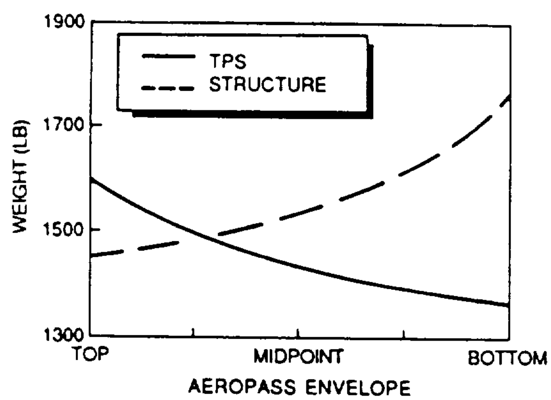
The rigid shaped brake (having the highest support structure weight) was selected for this trade since it would provide the greatest weight savings from the reduced g-loads. Thermal and structural design data from trajectory simulations are tabulated in Table 5.2.2-1. It can be seen that heat fluxes and loads increase as the aeropass is performed deeper in the atmosphere. However, total heat loads are reduced because of the shorter duraton in the atmosphere. The required TPS thickness based on the total heat load and resultant aeroshield structural weight based on g-loads are also shown. When using downward lift from additional L/D to aerobrake at a higher altitude, the vehicle spends a longer time in the atmosphere to achieve the same V. This longer duration results in a higher heat loads which requires thicker TPS to maintain backwall temperatures below 600°F. Although g-loads and hence support structure weight are reduced due to the thinner atmosphere, this weight savings does not offset the increased TPS weight.

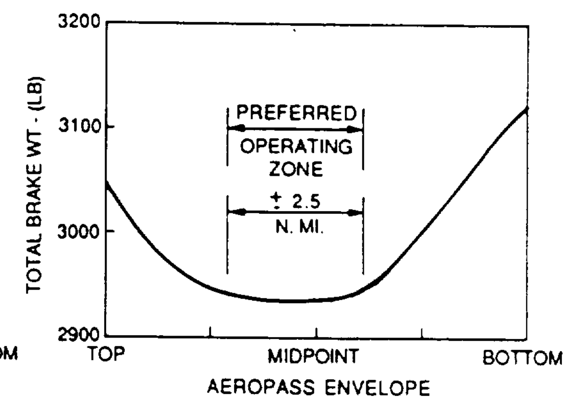
Aerobrake weight variations through the aeropass corridor and a breakdown of the TPS and support structure are shown in Figure 5.2.2-2. The minimum total brake weight occurs about the midpoint of aeropass corridor. It should also be noted that for a $\pm$ 2.5 nmi control zone about the midpoint, the brakes weight remains basically constant.

Table 5.2.2-1 Rigid Brake Design Data

RIDGID BRAKE	CORRIDOR TOP	CORRIDOR MIDPOINT	CORRIDOR BOTTOM
PEAK HEAT FLUX	27.8	31.6	36.0
HEAT LOAD	4,743	3,923	3,648
TILE THICKNESS (IN)	0.87	0.78	0.74
PEAK G-LOAD	2.66	3.58	5.37
TILE WT (LB)	1597	1432	1359
SUBSTRUCTURE WT (LB)	2155	2239	2469
TOTAL BRAKE WT (LB)	3752	3671	3828

- WEIGHT OF A/B SHELL (SKIN = 710 LBS) NOT INCLUDED IN FOLLOWING COMPARISON AT 1/4 IN. MINIMUM FABRICATION THICKNESS





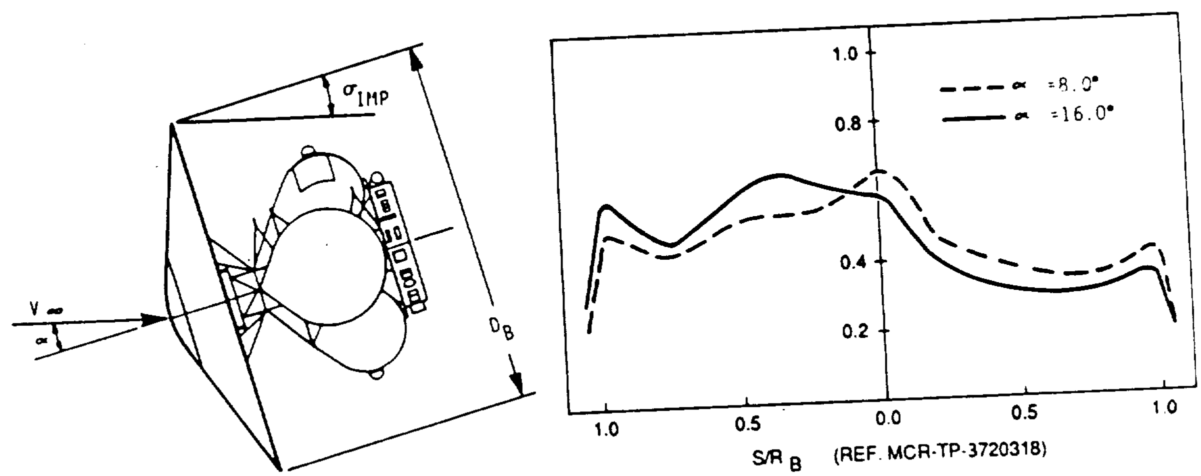
- THE OPTIMUM LOCATION OF THE +/- 2.5 NAUTICAL MILE CONTROL CORRIDOR IS 0.5 NAUTICAL MILE ABOVE THE MIDPOINT OF THE 15 NAUTICAL MILE AEROPASS ENVELOPE

Figure 5.2.2-2 Optimum Corridor Location

5.2.3 Optimum L/D

Knowing the optimum operating zone for an OTV with additional L/D, an aerobrake weight trade to evaluate the optimum L/D can be made. With the preferred operation zone of higher L/D lying in the altitude region of the minimum required L/D of 0.12 and the shown weight advantages of flex TPS, the 70° symmetric rigid/flex aerobrake was selected for this analysis. The current trade study is to establish how increasing L/D to 0.30 affects the aerobrake and if redesign of the brake for a L/D = 0.30, instead of 0.12 causes brake growth.

Stable trim is maintained by an offset center-of-gravity location. The offset is selected to provide the desired trim L/D, and thus, sets the vehicle's angle of attack. As L/D or angle of attack increases so does the flow impingement angle. This results in an increase of the aerobrake diameter to prevent flow impingement onto the vehicle. For the two L/D's of interest the vehicle's flow impingement angles, resultant brake diameters, and heating profiles across the brake are shown in Figure 5.2.3-1. Note the increase in edge heating and the shift in the peak heating region from the rigid nose to the flexible portion of the brake as L/D increases to 0.30.



- ANGLE OF ATTACK SET BY OFFSETTING C.G.
- ANGLE OF ATTACK SETS FLOW IMPINGEMENT ANGLE
- FLOW IMPINGEMENT ANGLE SIZES BRAKE DIAMETER

L/D	=	0.30	0.12
α	=	18.5°	7.2°
σ_{IMP}	=	30.7°	20.0°
D_B	=	50'	44'
$W/C_D A$	=	7.10	8.02

Figure 5.2.3-1 Angle of Attack Effects

Thermal and structural design data from trajectory simulations are tabulated in Table 5.2.3-1 for the two L/D values of interest. The resulting aerobrake heating environment and TPS requirements are also shown. In Table 5.2.3-2 a TPS and structural weight breakdown for the two L/D brakes is presented. Also it can be seen that there is a weight increase to the brake when going from the L/D of 0.12 to 0.30 of 379 lbs or 21%.

Table 5.2.3-1 Symmetric Brake Design Criteria

L/D	D_B (FT)	$\frac{W}{C_D A}$	CORRIDOR MIDPOINT	$\dot{q}_{R=1}$ (BTU/ft^2-SEC)	$\int q\,dt$ (BTU/ft^2)	g - LOAD
0.12	44	8.02	+2.5 n.m.	110.0	19,865	1.83
0.12	44	8.02	-2.5 n.m.	144.3	14,983	3.37
0.30	50	7.10	+2.5 n.m.	119.9	16,039	2.79
0.30	50	7.10	-2.5 n.m.	144.5	14,452	3.76

L/D	TPS	$\dot{q}_{max}$ (BTU/ft^2-sec)	$\int \dot{q}\,dt$ (BTU/ft^2)	THICKNESS (INCHES)	AREA (FT^2)	WEIGHT (LB)	TOTAL TPS WEIGHT
0.12	RSI	25.8	3731.4	0.73	149	127	1001
	FSI	23.5	3420.7	0.44	1553	874	
0.30	RSI	25.8	3206.6	0.68	149	118	1355
	FSI	28.8	3536.6	0.46	2123	1237	

Table 5.2.3-2 Aerobrake Data Versus L/D

L/D	0.12	0.30
BRAKE DIAMETER	44	50
W/C A	8.0	7.1
PEAK HET FLUX	25.8	18.8
G-LOADS	3.4	3.8
CENTRAL TILE WT	127	118
FLEX INS. WT	874	1237
STRUCTURAL WT	812	837
TOTAL WT	1813	2192

Based upon the above trades the L/D of 0.12 symmetric flex brake is superior. The lower L/D provides a lighter and smaller brake with the heatshield TPS weight being reduced 26% and support structure reduced 17% when compared to a L/D of 0.30. This is due to a less severe heating environment and slightly lower loads associated with the lower L/D aeropass.

In conclusion, the selected aeroassist device for OTV is a 70 degree conical lifting brake, which is a constant drag concept with small lift capability that provides the maneuverability to compensate for atmospheric dispersions. The configuration is based on the Viking aeroshell shape which provides the concept with ground and flight test data and verification within analytical code potential. Major features of this aeroshell concept include: inherent stability compared to other forecone angles; simple design and passive structure; its geometry incorporates symmetry which overcomes the rolling instability found in non-symmetric shapes; and the flexible TPS offers significant weight reduction and does not limit OTV basing options. In addition, the flexible TPS reduces support structure weight and allows the brake to be folded for transporting. The brake is weight optimum at an L/D of 0.12, which has been shown to provide adequate margin for guidance dispersions and upper atmospheric variations. It has also been shown that the L/D = 0.12 aeropass envelope is in the optimum operating zone for higher L/D vehicles but at a reduced heatshield weight.

5.3 STRUCTURAL ANALYSIS

5.3.1 Meteoroid and Space Debris Shielding

The space-based 74 klb (four tank) and the ground-based 45 klb and 52 klb OTVs were assessed for the shielding required to survive the meteoroid and space debris environments to the defined requirements. These requirements are defined as 0.999 probability of no penetration per mission, where no penetration is defined as no impact on the aluminum propellant tank pressure wall. The perigee inclination is assumed to be 28.5^{o} for both the vehicles and altitudes of 270 nm and 140 nm for the space-based and ground-based OTV, respectively. The space-based OTV is assumed to be shielded from the two environments while it resides at the Space Station. The payloads and their required shielding were not addressed in the OTV analysis.

5.3.1.1 Meteoroid Environment

The average total meteoroid model defined in document NASA SP-8013 was used. This model is consistent with that used for the Space Station Phase B contract. The document defines the threat as being predominantly of cometary origin with an average velocity of 20 km/sec and an average density of 0.5 g/cc (equivalent to lightly crushed ice).

As the definition of the payload delivery orbits are not well defined at this time, assumptions of the Earth's shielding and defocusing factors are made. The Earth's shielding factor varies from 0.68 at low Earth orbit (LEO) to 1.0 at geosynchronous Earth orbit (GEO) and the Defocusing factor varies from 1.0 at LEO to 0.65 at GEO. Consequently, to ensure all situations are accounted for, the values were assumed to be 1.0 for both factors on both vehicles. With better definition of the mission models, less conservative factors can be adopted.

5.3.1.2 Space Debris Environment

The space debris environment is defined in the document JSC20001 by D. Kessler. The document defines the average impacting velocity as 9 km/sec and an average density of 2.8 g/cc (that of aluminum). The flux-diameter relation is defined at two altitudes, 270 nm, which was used for space-based OTV, and for 220 nm. For the ground-based OTV the environment was adjusted from the 220 nm definition to 140 nm based on Figure 5.3.1-1 (which was given in a presentation by D. Kessler of NASA in June 1984). As shown there is a significant reduction in the space debris environment, a factor of 0.5 on the flux, due to the reduced altitude and the resultant flux-diameter relation at 140 nm is:

$$LOG_{10}(0.5 \times F_{sd}) = -5.82 - 2.42LOG_{10}(D_{sd}) \qquad (1)$$

Where: F_{sd} = Flux (impacts/sq meter/year of diameter D_{sd} or greater)

D_{sd} = Diameter of space debris (cms)

The space debris threat is assumed to be only present at LEO. At GEO all the satellites and debris are orbiting in the equatorial plane, their velocities are all equivalent and in the same direction, hence their relative velocities are zero and there is a negligible chance of collision.

The time lines for the space-based and ground-based OTV mission models were used to establish the space debris exposure time. Included in this was the time for the transfer orbit to 1620 nm (where the space debris environment drops off) and aerobraking in the atmosphere.

Altitude Distribution

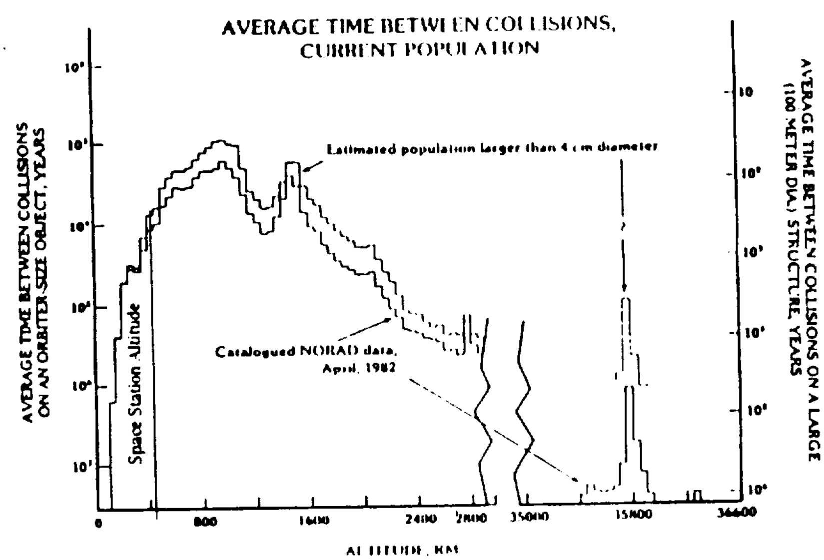


Figure 5.3.1-1 Space Debris Population With Altitude

Following is the summary of the exposure times used:

	EXPOSURE TIME TO METEOROIDS	EXPOSURE TIME TO SPACE DEBRIS
Ground-Based 45 klb & 52 klb	68 hours	18 hours
Space-Based 74 klb	15 days	11 hours

When a better knowledge of the vehicle's orientation at LEO is available, more use can be made of the two dimensional nature of the space debris environment on the effective exposure area. For this analysis the space debris exposure area was taken as 140 sq meter for both vehicles.

5.3.1.3 Combining the Environments

To combine the two environments and solve for the space debris and meteoroid fluxes an assumption has to be made relating the two particle types. That assumption is that space debris and meteoroid particles with equivalent kinetic energy will just penetrate the same shield design.

$$1/2 \times M_{sd} \times V_{sd}^2 = 1/2 \times M_{met} \times V_{met}^2 = K.\ E. \tag{2}$$

The relation between the probability and the flux is as follows:

$$P = e^{-(A + B)} \tag{3}$$

Where:

$A = F_{sd} \times A_{sd} \times T_{sd}$

F_{sd} = Flux space debris (impacts/sq meter/year of diameter D or greater)
A_{sd} = Space debris exposure area (sq meters)
T_{sd} = Exposure time (years)

$B = F_{met} \times A_{met} \times T_{met}$ x Earth shielding factor x Defocusing factor

F_{met} = Flux meteoroids (impacts/sq meter/year of mass M or greater)
A_{met} = Meteoroid exposure area (sq meters)
T_{met} = Exposure time (years)

P = Probability of no penetration

Using the assumptions and equations mentioned previously the following table gives the design particle sizes the shield has to stop to meet the 0.999 probability of no penetrations per mission.

	METEOROID DIAMETER	MASS	SPACE DEBRIS DIAMETER	MASS
Ground-Based 45 klb & 52 klb	0.139cm	0.0007g	0.134cm	0.0035g
Space-Based 74 klb	0.217cm	0.0027g	0.209cm	0.0132g

The above requirement results in the individual environment probabilities and close proximity (to either the Shuttle or Space Station) probability as follows.

	Close Proximity Duration	Close Proximity Probability	Mission Prob Breakdown MET Prob	Mission Prob Breakdown S D Prob.
Ground-Based 45k & 52k	4.0 hr	0.999937	0.999028	0.999972
Space-Based 74k	3.5 hr	0.999980	0.999031	0.999968

5.3.1.4 Shield Sizing

The previous shield design used on other space vehicles has been the Whipple bumper system shown in Figure 5.3.1-2. This shield system is only used if there is a weight penalty in protecting the fuel tanks from penetration by thermal insulation only. The Whipple bumper system provides a low weight effective shield against hypervelocity impacts (>5 km/sec). The function of the bumper is to shock the incoming particle which then fragments and vaporizes. The result is an expanding vapor cloud including molten fragments of the bumper and particle. The gap between the bumper and rear wall allows this cloud to expand and disperse and consequently the impacting energy is deposited over a large area on the rear wall. The rear wall is then designed to resist the pressure pulse and the cratering made by the impacting of the small fragments.

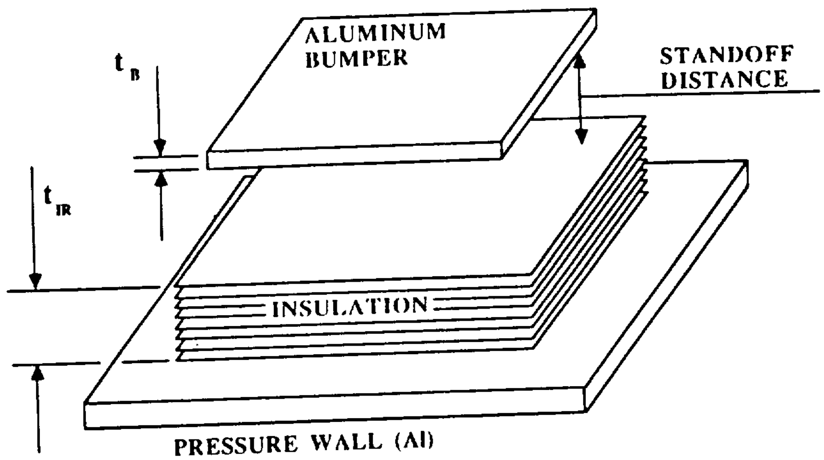


FIGURE 5.3.1-2 - WHIPPLE BUMPER SHIELD SYSTEM

The AIAA paper 69-372 "Meteoroid Protection by Multi-wall Structures" by Burton Cour-Palais was used to determine the shield sizing required. Figure 5.3.1-3 shows the effective bumper thickness required to fragment the impacting particle as a function of velocity. For an average velocity of 20 km/sec for meteoroids, and 9 km/sec for space debris, the bumper thickness should be set at 0.04 and 0.16 x diameter of the impacting particle respectively. This shows that the space debris particle designs the bumper.

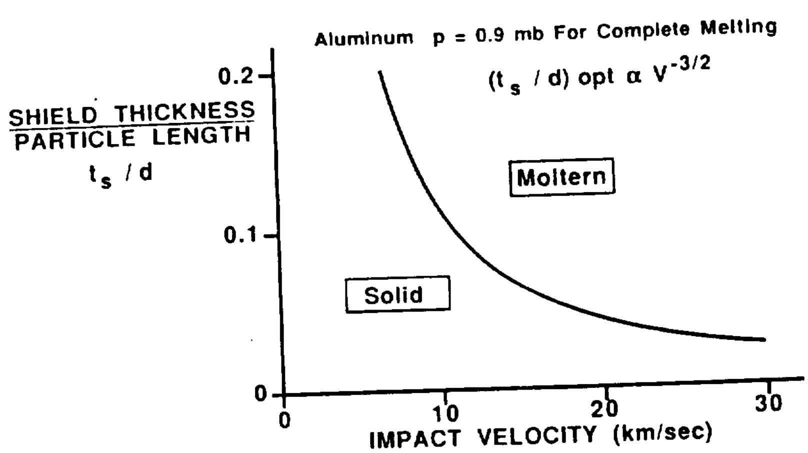


FIGURE 5.3.1-3 - OPTIMUM t_s/d VERSUS IMPACT VELOCITY

The paper also defines that the maximum effective gap is 30 x diameter of the particle, above this value the failure is dominated by the individual fragments and additional gap does not increase penetration resistance. The size required for an aluminum rear wall, at the 30 x diameter gap is defined as:

$$t_R = 0.055 \times (p_p \times p_r)^{1/6} \times M^{1/3} \times V \qquad (4)$$

Where: t_R = Aluminum thickness required (cm)
p_p = Impacting particle density (g/cc)
p_r = Rear wall density (g/cc)
M = Impacting particle mass (g)
V = Impacting particle velocity (km/sec)

The required rear wall aluminum thickness is translated to a required insulation thickness using data from Figure 19 of the document NASA TMX-53955, "Meteoroid Physics Research at MSFC", June 1969. This is reproduced in this report as Figure 5.3.1-4. This data is for individual projectiles impacting low density materials and, at a gap of 30 x diameter, is the probable rear wall failure mode. An aluminum plate thickness designed to stop the test particle used in NASA TMX-53955 was calculated from Equation 3 in the document NASA SP-8042:

$$t_{TAP} = 0.42 \times M^{0.352} \times p^{1/6} \times V^{0.875} \qquad (5)$$

Where: t_{TAP} = Thickness of plate penetrated (cm)
M = Mass of projectile (g)
p = Projectile density (g/cc)
V = Impact velocity (km/sec)

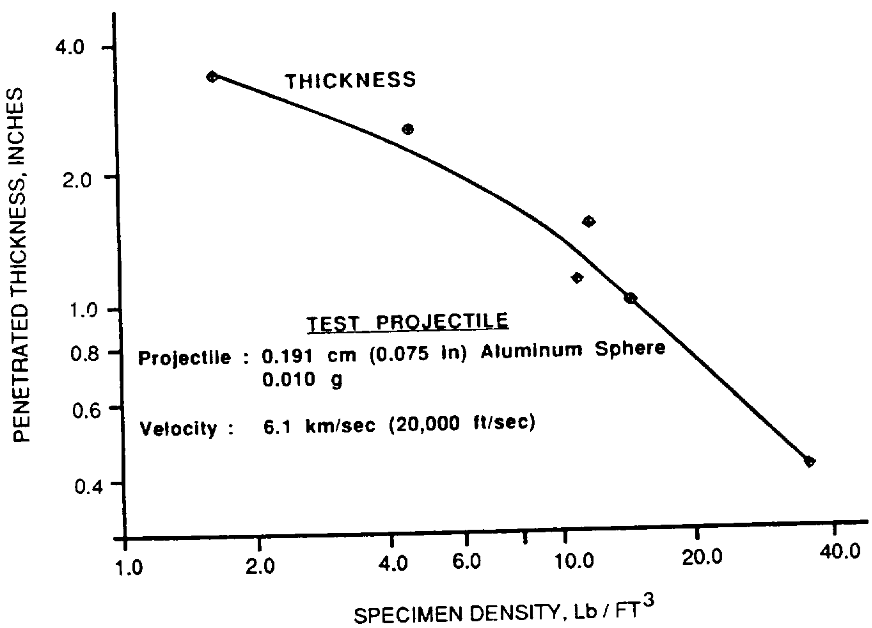


FIGURE 5.3.1-4 TARGET DENSITY VERSUS PENETRATED THICKNESS

This value was calculated as 0.645 cm. The aluminum rear wall thickness was then ratioed to an equivalent insulation thickness as follows:

$$t_{IR} = t_R \times (t_{TDI}/t_{TAP}) = t_R \times (t_{TDI}/0.645) \tag{6}$$

Where:

t_{IR}	=	Insulation thickness required (cm)
t_R	=	Aluminum thickness (cm) from Equation (4)
t_{TDI}	=	Test demonstrated insulation thickness (cm) from Figure 5.3.1-4

The density of the MLI was taken as 0.788 lb/ft^3 and 2.0 lb/ft^3 for the foam insulation. The resultant thicknesses required to meet the design requirement of 0.999 probability of no penetration is tabulated below:

Shield Description	Ground-Based 45k (& 51k)	Space-Based 74k
Bumper (in)	0.009	0.014
Gap (in)	1.5	3.0
Effective Al Rear Wall (in)	0.0417	0.0649
Insulation (in)	0.62	1.0

5.3.1.5 Recommendations

Certain assumptions have been made in the analysis which can be improved upon later in the OTV program as more data becomes available. Similarly as the space debris environmental effects are better understood and analysis techniques improved, more accurate shield sizings can be performed. The proposed shielding configurations will have to be tested to verify the analysis made here. Currently the analysis does not account for the velocity spectrum of the two environments (the average velocities of the two environments were used), or the angular distribution of the space debris environment.

5.3.2 Canted Payload Adapter - 74K Space-Based Cryogenic OTV

Figure 5.3.2-1 shows a canted payload adapter between the 74 klb space-based cryo OTV avionics ring and a 15 ft dia x 23 ft payload. The 38 ft brake of the 74k space-based OTV allows return of a 15 ft long payload which meets current requirements. The 38 ft brake with a canted adapter allows return of a 23 ft long payload. Without the canted adapter a 44 ft brake would be required.

The canted adapter shown in Figure 5.3.2-1 is made of 6-inch aluminum channel welded together to form a truss assembly, the lower side of the truss attaches to the three adjustable payload support points inside the OTV avionics ring. The canted side of the truss will attach to the payload as shown. This will require mating structural support to be designed into the payload aft structure.

Analysis of the adapter is shown on the next page.

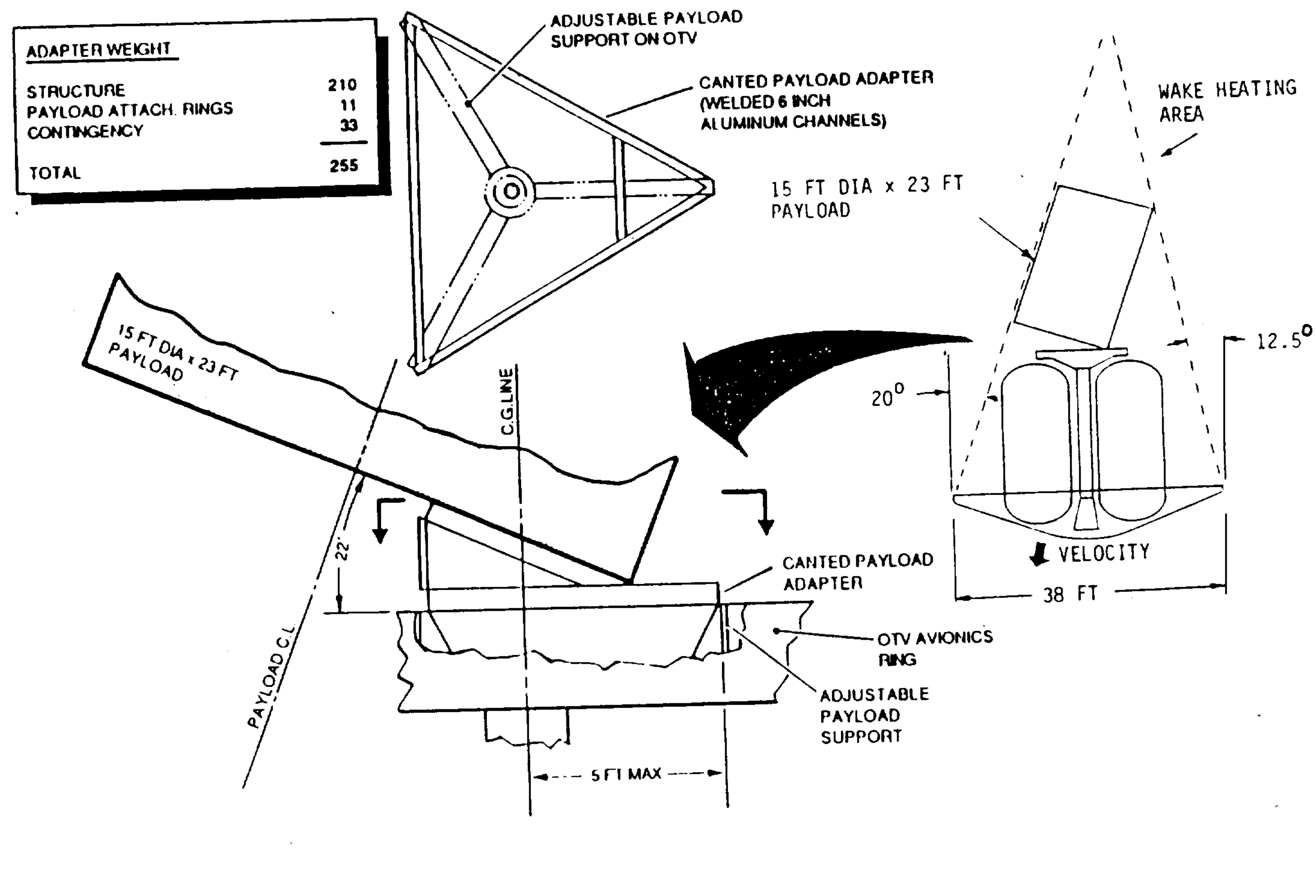
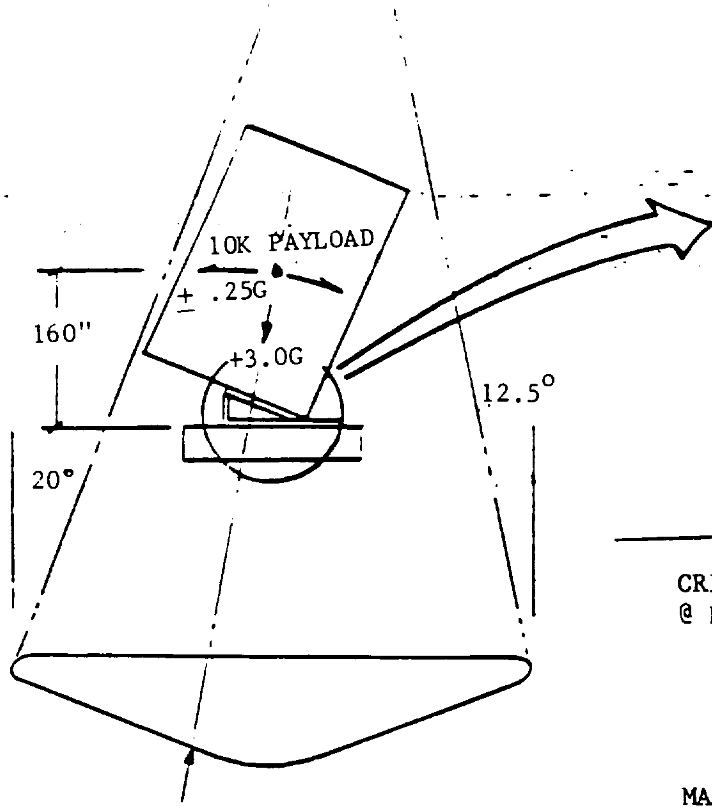
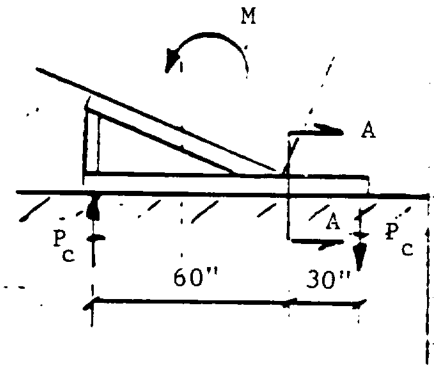


Figure 5.3.2-1 - CANTED PAYLOAD ADAPTER





CRITICAL LOAD CONDITIONS:

@ RE-ENTRY: LONG. = +3.0 G
LAT. = ± 0.25G

OR

LONG. = 0 G
(DYNAMIC) LAT. = ± 0.25G

MAX BENDING ON SEC A-A OCCURS
WHEN LONG. = 0 G
LAT. = ± 0.25G

M = PAYLOAD MOMENT
= 0.25G (10KLB) (160") = 400"KLB

COUPLE LOAD Pc = M/90"
= 4.44KLB (limit)

BEND MOM @ A-A:
30Pc = 133.3"KLB (limit)

SECT. MOD S_x = 5.06 in^3

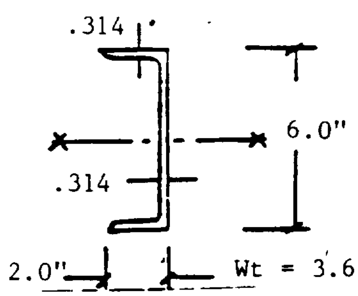
MAX BEND STRESS ON A-A:

(limit) $b_y = \frac{M_x}{S_x} = \frac{133.3}{5.06}$
= 26.3 KSI

(ult) b_y = 1.4 (26.3)

= 36.9 KSI

SECT A-A



Wt = 3.63 lb/ft

Mat'l: 2219-T8511 Extr.
annealed at weld zone
@ Rm temp F_{tu} = 31 ksi
@-300°F F_{tu} = 38 ksi
F_{ty} = 30 ksi

FIGURE 5.3.2-2 - LOADS SCHEMATIC

5.4 AVIONICS SELECTION REVIEW

The avionics system, Figure 5.4-1, is a modular design that supports technology insertion as well as redundancy enhancement. A significant feature is its distributed computer architecture with a flexible executive operating system that facilitates performance enhancement and permits affordable software development. The design is generally dual fault tolerant through internal component redundancy for mission success and for critical operations in the vicinity of the Orbiter. An avionics component list and physical description is presented in Table 5.4-1.

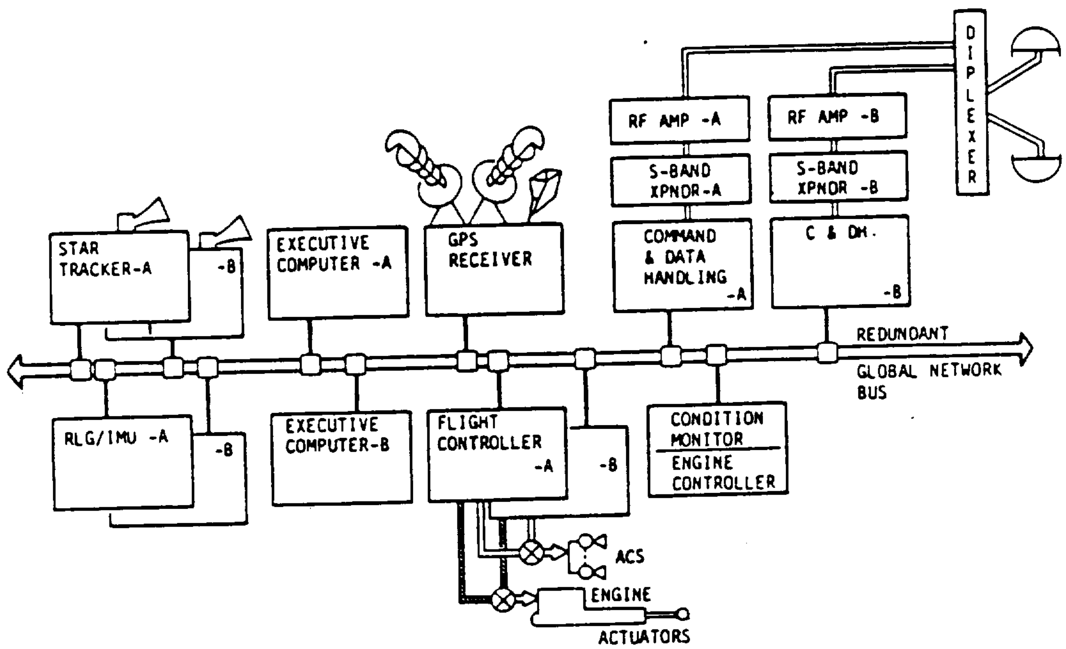


Figure 5.4-1 OTV Avionics Block Diagram

5.4.1 Guidance, Navigation and Control (GN&C)

The GN&C hardware consists of the following:

a. Dual redundant Ring Laser Gyro (RLG) Inertial Measurement Unit(s) (IMU)
b. Dual star trackers
c. GPS receiver/processor and high and low-altitude antennas
d. Dual majority vote flight controllers

Two RLG IMUs were selected because of good stability for long missions as well as low recalibration requirements from mission to mission. Each IMU included three (3) ring laser tyros (RLGs) and three (3) pendulous mass accelerometers and required computers and power supplies. A star tracker was selected instead of a scanner to take advantage of increased sensitivity of trackers and to minimize required maneuvers.

Table 5.4-1 OTV Avionics Equipment List (Sheet 1 of 2)

Subsystem / Equipment	Weight (lbs)	Power (watts)	Size (in.) H W L	Total Qty.	Total Wt.	Power Max.	Power Avg.
GN&C							
Star Scanner	11	10	7x 7x20	2	22	20	10
IMU	24	40	8x 8x12	2	48	80	80
GPS Receiver	20	30	8x 8x 9	1	20	30	10
GPS Antenna-Low Alt	5		6x 6x10	2	10		
GPS Antenna-Hi Alt	5		18x18x26	1	5		
Flight Controller	30	90	8x 8x16	2	60	180	120
Engine Thrust Controller	10	60	8x10x 9	1	10	60	60
			Subsystem Total		175	370	214
Data Management							
Executive Computer & Mass Memory	10	60	6x 8 x 9	2	20	120	120
			Subsystem Total		20	120	120
Telemetry and Command							
Command & Data Handling	15	35	6x 8x10	2	30	45	22
TLM Power Supply	7	10	4x 7x 7	2	14	20	5
			Subsystem Total		44	65	27

5.4.2 Data Management

The OTV data management subsystem is configured in a distributed architecture that includes two Executive Computers (dual-CPU type), each with large shareable mass memories and local memories. Key functional areas under Executive Computer software control are the Executive Operating System, attitude, guidance and navigation management, sequence control, power management, and test and checkout. The Executive and all of the other intelligent avionics subsystems are interconnected via a global network bus. This global network can support a throughput of from 10 to 20 Mbps via fiber optic cable. The network structure permits each subsystem to access the bus using an intelligent, standard protocol interface.

Table 5.4-1 OTV Avionics Equipment List (Sheet 2 of 2)

Subsystem / Equipment	Weight (lbs)	Power (watts)	Size (in.) H W L	Total Qty.	Total Wt.	Power Max.	Power Avg.
Communications and Tracking							
STDN/TDRS Xponder	16	55	6x 6x14	2	32	65	65
20w RF Power Amp	6	125	3x 6x10	2	12	125	40
S-Band RF System	50	20		2	100	40	20
		Subsystem Total			144	230	125
EPS							
Fuel Cell (FC)	45		11x12x12	2	90		
FC Radiators	25		$25ft^2$x2"	2	50		
FC Plumbing	25				25		
FC Coolant	15				15		
FC Water Storage	15				15		
Power Control & & Distribution	27	10	6x 8x12	2	54	20	20
Engine Power		600				600	
		Subsystem Total			249	620	20
		System Total			632	1405	506

5.4.3 Telemetry and Command (T&C)

The telemetry and command subsystem is designed around a basic SCI Data Acquisition and Control System (DACS) having a single control and I/O interface unit. The central unit consists of an 80C86 CMOS microprocessor-based system with local RAM (32K) and ROM (8K) for conducting telemetry and command processing independent of the executive computer. Command decoding and authentication, time tagging and command override services are provided.

5.4.4 Communication and Tracking (C&T)

The C&T subsystem provides both direct and relay communication with the ground. Communication with the Orbiter is either direct or through a ground station. The C&T subsystem operates at S-band and is compatible with STDN/TDRSS and SGLS depending upon the specific mission. Provisions have been incorporated for redundant transponders, RF power amplifiers and COMSEC equipment. Two electronically switched steerable array antennas provide hemispheric coverage. Each antenna includes a redundant microprocessor and redundant switching power divider. The other major components are inherently redundant, i.e., 145 passive elements with associated power drivers. Each antenna also includes an integrated preamplifier to facilitate parallel operation of two receivers (for fault tolerant reception) with minimal RF distribution losses. The direct/relay feature provides maximum flexibility from low earth orbit to GEO in terms of coverage and link margins for the various OTV missions. Relay C&T via TDRSS provides the primary communications for OTV operations below 10,000 Km altitude. Direct C&T is the primary mode for higher OTV altitudes, with TDRSS as a backup where coverage is available. The heart of the C&T subsystem is a dual mode TDRSS/STDN transponder and 20 watt RF amplifier (such as the existing Motorola packages) combined with the Ball Aerospace ESSA. This combination provides the flexibility in spatial coverage and the necessary link margins for the various OTV missions.

5.4.5 Electrical Power Subsystem (EPS)

The OTV Electrical Power Subsystem, Figure 5.4.5-1 consists of redundant fuel cells, vehicle cabling, power distribution and control, reactants, plumbing, and radiators. Power is distributed through redundant buses to the OTV subsystems. The Power Control and Distribution Assembly (PCDA) contains motor driven switches and relays needed to provide load control and fault protection circuitry. The PCDA also interfaces the command and data systems where commands are received from the OTV data bus, and health and status are passed to the data management subsystem. Each of the OTV fuel cells is sized to delivery 1.7 KW peak which includes 20% design margin. The fuel cells are also sized to provide coarse bus voltage regulation (28 $\pm$ 4 VDC) during worst case operation at the end of a five year life. This eliminates the requirement for active power conditioning. An active coolant loop and radiator system are used to reject fuel cell waste heat. Two 25 sq ft radiators are sized to reject the fuel cell waste heat. Reactants are taken from the main propellant system. Redundant fuel cells and plumbing allow the EPS to meet system reliability requirements without battery backup. There is no safety issue associated with this type of a fuel cell application because it is an extension of the STS design. System power up is also simplified because fuel cell initialization consists of warming the catalysts to operating temperature and supplying reactants.

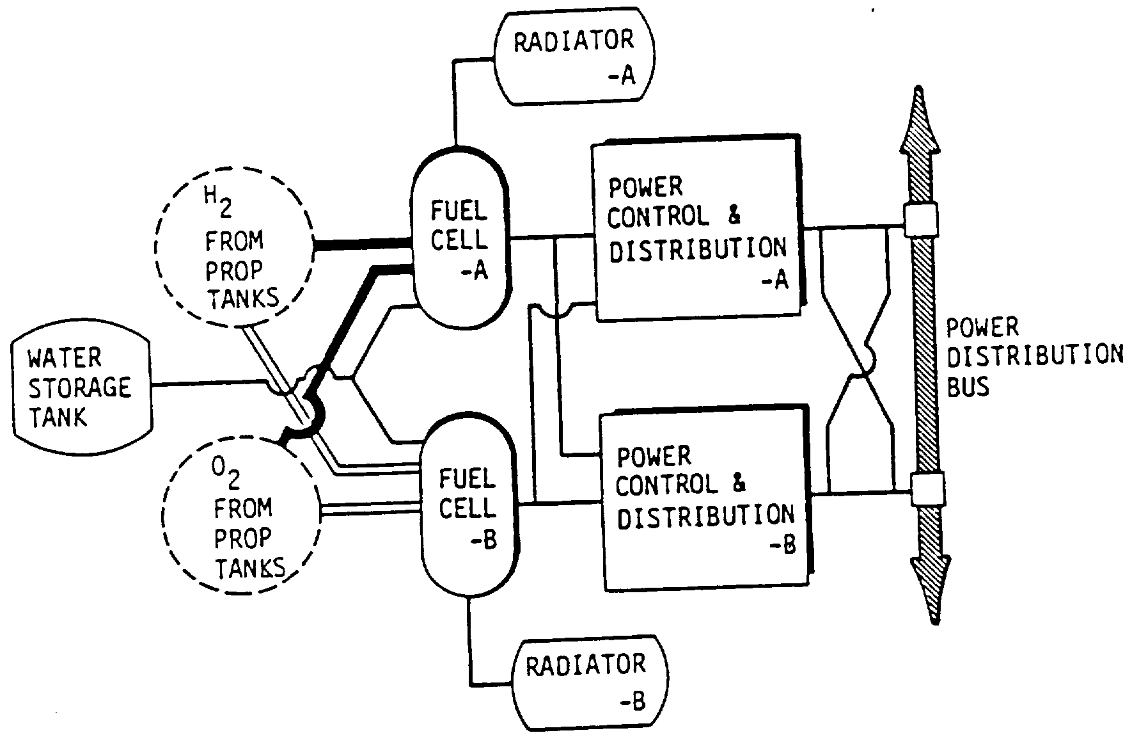


Figure 5.4.5-1 EPS Configuration

5.4.6 Thermal Control

The avionics are mounted circumferentially and outboard on the avionics ring located at the payload/OTV interface. The outboard side of the ring is painted with a low alpha over epsilon paint. The avionics are housed in MMS-type boxes. The avionics components are mounted to the skirt in a manner which allows component waste heat to travel freely to the skirt. The location of the avionics on the ring will allow for the component waste heat to be evenly distributed among all the avionics. This reduces supplemental heater power requirements.

The fuel cell TCS is sized for a nominal 25-day OTV flight duration which requires two 25-ft^2 radiators to dissipate fuel cell waste heat. The radiators are located on the avionics ring simplifying the cooling loop system and reducing its weight. The two radiators are mounted on opposite sides of the vehicle to accommodate long duration fixed OTV orientation with respect to the sun vector, thus preventing fuel cell overheating.

All H_2 and O_2 cryo tanks are insulated with 1.0 inch (50 layers) of MLI. The main propellant feedline insulation consists of 2 layers of gold foil.

Meteoroid shielding is provided on propellant tankage with stand-off thin wall bumpers.

5.5 TETHER UTILIZATION

5.5.1 Tether Deployment Operations Concept

Figure 5.5.1-1 depicts the general procedure for deploying OTV using a tether. The OTV is deployed vertically in a gravity gradient stabilized mode using an 81 nautical mile long tether. When the OTV has reached its maximum deployment distance, it is released. Since it is stabilized along the local vertical, it is traveling at super-orbital velocity, and has achieved a significant orbital momentum which is extracted from the Space Station. During the tether maneuver, the micro-g environment at the Space Station is disturbed. After release, the Space Station orbit perigee is significantly reduced. Orbital makeup using Space Station propulsion would be a poor trade, since its Isp is lower than OTV Isp. Therefore, a companion program of deploying Orbiters towards reentry is required to maintain a Space Station momentum balance. This complementary procedure also saves propellant from the Orbiter OMS propellant budget. These procedures involve operations complexities that must be balanced against propellant savings to determine if the approach should be pursued.

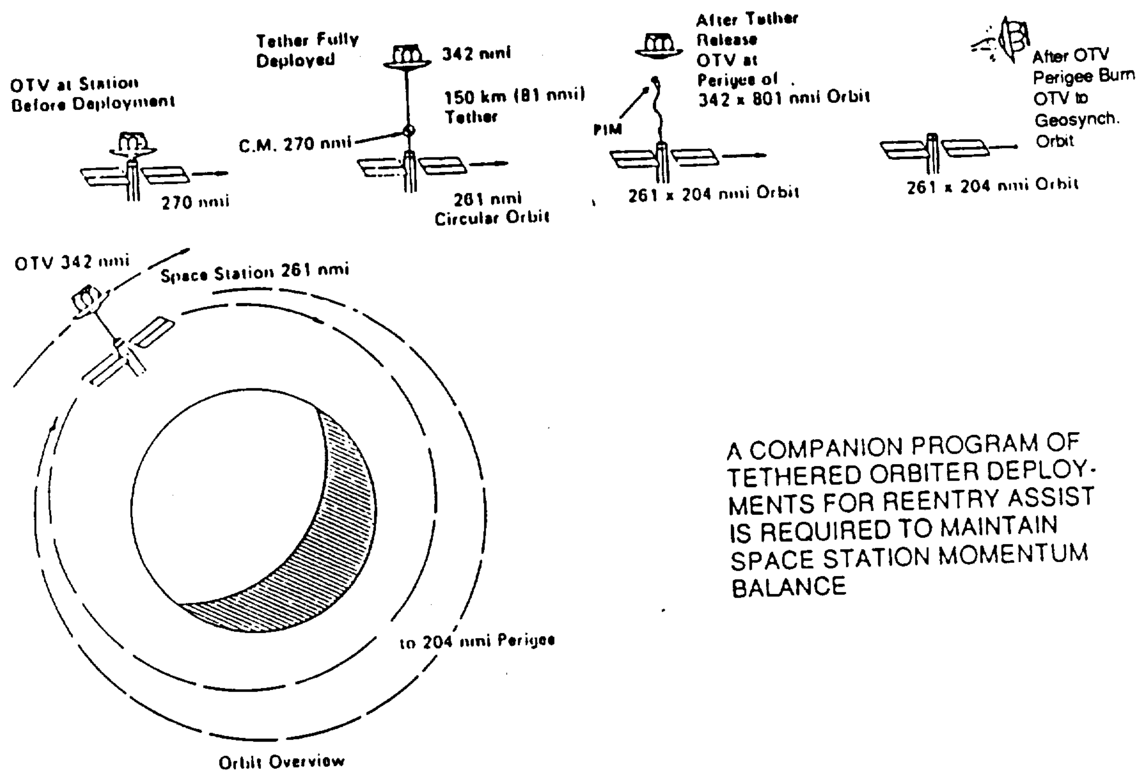


Figure 5.5.1-1 Tether Deployment Operations Concept

5.5.2 Tether Deployment Evaluation

Tether launch of the Orbital Transfer Vehicle offers a significant benefit in performing geostationary missions from a space-base. This benefit is a function of tether length, as shown in Figure 5.5.2-1. The velocity reduction provided by an 80 nautical mile tether reduces the propellant required for a cryogenic OTV to perform a 20 Klb GEO delivery mission by 8.8 percent. The companion tether deorbit of a Shuttle that is required to maintain the momentum balance of the Space Station, reduces the required OMS budget of the Orbiter by a related amount -- numerically equal to 13.5 percent of the OTV propellant requirement. These propellant reductions offer a cost benefit that can be balanced against the operations costs associated with tether operations and tether system acquisiotn costs. These costs have been estimated by our tether applications personnel at \$2.7M per OTV operation (including OTV deployment and companion Orbiter deorbit), and \$90M delta cost to acquire an OTV tether operations capability (cost beyond an Orbiter operations capability). The resulting net life cycle cost advantage is \$572M in constant '85 \$, and \$90M in 10 percent discounted dollars. This corresponds to a 3% discounted cost benefit due to tether use for a space-based program.

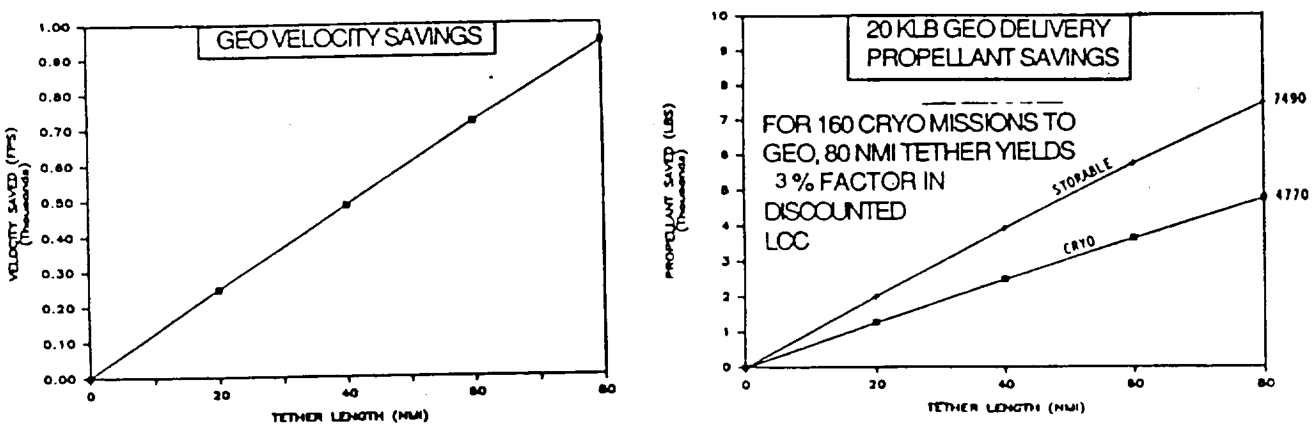


Figure 5.5.2-1 Tether Benefits

The tether deployment mechanism is a portable device that moves along the MRMS track to the desired deployment location. The OTV/payload is moved from its hangar to the deployment mechanism by MRMS and mated with the payload interface module (PIM) of the tether unit. The PIM then imparts the initial separation velocity to the OTV/payload. As the tether reels out, it must be braked which allows an added potential benefit of generating electrical power.

Deployment operations require a total of 16 hours of tether operation, during which time the micro-g environment of the station is disturbed. This activity should be scheduled to coincide with other Space Station operations that disturb the micro-g environment. Similarly, OTV deployment and Orbiter deployment operations must be scheduled within limits of acceptable station altitude excursion.

OTV retreival operations have been investigated, and they offer considerably less benefit than deployment operations, at a significant increase in operational complexity.

5.5.3 Tether Recommendation

Tether operations for deployment of OTV and Shuttle Orbiters offer a significant reduction in OTV operations cost and propellant requirements, but cause certain operations disadvantages that we believe can be adequately mitigated. It is clear that the issues involved must be worked off with other Space Station users before resolution is possible. We believe this resolution should be pursued, and the net benefit of tether use validated.

5.6 PERFORMANCE ASSESSMENT METHODOLOGY

OTV performance was assessed using basic vehicle characteristics operating in the framework of the mission designs described in section 3.3.

5.6.1 Performance Software

In order to automate the performance assessment process a spreadsheet type program was developed on the Apple Macintosh (Figure 5.6.1-1). This spreadsheet is structured into an upper half, containing mission data, and a lower, containing OTV configuration data. The mission data is broken up into rows, each row representing a mission segment (see section 3.3). For each segment the spreadsheet calculates propellant requirements based on the impulsive Delta-V (column E) plus gravity loss correction * (column F), the OTV weight at the end of the last segment (column B) plus the payload weight (column G), and the OTV Isp and thrust level (lower half, column G). All Delta-V's are scaled up by the flight performance reserve (FPR) factor contained in column G, lower half. This performance reserve has been ground ruled at 2% for all configurations. The ideal rocket equation is used to compute MPS propellant requirements which are displayed in column K and used to update propellant remaining (column C) and OTV weight (column B). These last two columns display status that is effective at the end of that particular segment. In addition, the program estimates consumables usage in cryo propellant boiloff (column H), fuel cell usage (column I), and ACS propellant usage (column J) which are included in vehicle mass calculations. These quantities are calculated as a function of segment duration (column D, in hours) as applied to configuration-dependent rate data contained in column G, lower half.

Other OTV configuration data contained in the lower half of the spreadsheet includes a dry weight statement (column D), a propellant capacity (column J), and a trapped propellant allocation (column G) which when multiplied by the propellant capacity gives the unusable propellant left in the vehicle at the end of the mission.

	A	B	C	D	E	F	G	H	I	J	K
1	MISSION	TOTAL OTV	MPS PROP	SEGMENT	GEO	DELTA-V	PAYLOAD	CRYO	FUEL CELL	ACS PROP	MPS PROP
2	SEGMENT	WEIGHT	REMAINING	DURATION	DELTA-V	GRAVLOSS	WEIGHT	BOILOFF	USAGE	USAGE	USAGE
3	NUMBER	(LBS)	(LBS)	(HRS)	(FPS)	(FPS)	(LBS)	(LBS)	(LBS)	(LBS)	(LBS)
4											
5	0	49009	41392	0	0	0	10000	0	0	0	0
6	1	24100	16483	2.00	8073.99	139	10000	9	1	4	24895
7	2	13118	5501	5.30	5855.68	-49	10000	23	4	12	10943
8	3	8701	1084	12.00	6051.61	0	0	53	8	26	4330
9	4	8658	1041	4.20	20.00	0	0	18	3	9	12
10	5	8437	820	3.00	350.00	0	0	13	2	7	199
11	6	8397	780	5.50	0.00	0	0	24	4	12	0
12											
13											
14			52K OTV		OTV1						
15			AVIONICS	926		ISP	475				
16			STRUCT.	1649		THRUST	15000				
17			THERMAL	271		TRAPPED	0.015		SIZE TANK	0	
18			AEROBRAKE	1492		FPR	0.02		SIZE BRAKE	0	
19			PROPULSION	1932		BOILOFF	4.40		PROP CAP.	52000	
20			TANKAGE	1348		FUEL CELL	0.69				
21			TOTAL SUM	7617		ACS RATE	2.20				

Figure 5.6.1-1 Performance Spreadsheet Program

(* See "Design Driver Mission Analysis, Section 3.3, for Gravity Loss Equations)

5.6.2 Geosynchronous Propellant Requirements

When the performance program described above is linked with the mission model data base a propellant map by mission and by year is generated. Figure 6.2.3-3 shows such a map for the ground-based option, flying scenario #2 geosynchronous missions. Included in the database is mission data (payload description, weight and orbit characteristics), OTV type and dry weight, a mission frequency map and a yearly OTV propellant map.

Several such data maps were generated for various scenarios, OTV configurations, and basing options for use in the cost trades.

Lunar and planetary propellant requirements will be covered in the following two sections.

5.6.3 Lunar Propellant Requirements

Figure 5.6.3-1 summarizes results of analyzing the 6 lunar missions. In all cases all of the OTV hardware is reusable.

Two of the missions (17200 & 17201) are accomplished with a solo 52 Klb propellant capacity OTV. Mission #17202 is handled with a 52 Klb OTV and 52 Klb capacity tankset. Mission #17203 is performed with a 2-stage OTV, both of whose stages are 74 Klb propellant capacity, and a 52 Klb capacity tankset bolted to Stage 1. This same configuration performs mission #17207. Finally, a very similar 2-stage stack performs mission #17206, the only difference being that the tank set on Stage 1 is of 74 Klb capacity.

PLD. NO.	PAYLOAD UP	PAYLOAD DOWN	PROPELLANT REQUIRED	STAGE 1 OTV PROP CAP/DRY WT	STAGE 2 OTV PROP CAP/DRY WT	TANK SET* PROP CAP/DRY WT
17200	5072	0	36523	52K / 7617	0 / 0	0 / 0
17201	5072	0	31785	52K / 7617	0 / 0	0 / 0
17202	32850	0	89992	52K / 7617	0 / 0	52K / 4003
17203	72680	0	158098	74K / 8732	74K / 8732	52K / 4003
17206	93000	0	215617	74K / 8732	74K / 8732	74K / 4805
17207	72680	20000	179686	74K / 8732	74K / 8732	52K / 4003

* NOTE: TANK SET IS ATTACHED TO FIRST STAGE

Figure 5.6.3-1 Lunar Performance Summary

5.6.4 Planetary Propellant Requirements

Figure 5.6.4-1 displays analysis of the 24 planetary missions in the Rev 9 model. The basic mission profile is as follows: The OTV injects the payload into a hyperbolic orbit which may or may not be at the target C_3, depending on whether the payload carries a kick stage. The OTV separates and after a one hour coast performs a retro burn to put it into a highly elliptical orbit around the earth. After a pair of plane adjustment burns near apogee the OTV aerobrakes back into a low park orbit for retrieval. All planetary missions were assumed to begin coplanar with their outgoing C_3 vector.

In addition, a special study was performed to analyze the application of aeroassist to a manned Mars Mission. The results of this study are contained in Volume X - Aerocapture for Manned Mars Missions.

A performance program was written which accounts for the above mission profile and attempts to minimize the launch stack weight. If the mission cannot be accomplished by the OTV alone a 52 Klb or 74 Klb capacity tank set is added to the stage. If this is not sufficient an expendable kick stage is added to the payload. This kick stage is assumed to be a solid fuel device with an Isp of 310 sec and a 0.9 mass fraction. If this still doesn't work, the program expends the OTV. The program also assumes a 2% flight performance reserve on all burns and a 1.5% trapped propellant allocation.

For a further discussion, see section 3.3 - Design Driver Mission Analyses, and MMC OTV TM I.1.2.0.0-1.

As the table indicates there are 5 missions that require tanksets (4 of these require kick stages as well), without these additions the OTV would have been expended on these flights. Of the rest of the missions, 5 require kickstages, 2 expend their OTV and one (17095, the Pluto Orbiter) uses a reusable 74 Klb OTV Stage 1 and an expendable 74 Klb OTV Stage 2. In summary, the planetary program requires the following:

10 52 Klb OTV Flights (Reusable)
12 74 Klb OTV Flights (Reusable)
3 74 Klb OTV Flights (Expendable)
5 OTV Tanksets (Reusable)
9 Expendable Solid Kick Stages

PLD NO	PAYLOAD UP	EKS WEIGHT	PROPELLANT REQUIRED	OTV PROP CAP/DRY WT	TANK SET PROP CAP/DRY WT	
17085	6615	0	49919	52K / 7617	0 / 0	
17086	3300	20273	104822	52K / 7617	52K / 4003	[1]
17087	3310	0	31336	52K / 7617	0 / 0	
17088	19945	22235	141168	74K / 8732	74K / 4805	[1]
17089	4475	25833	73317	74K / 8732	0 / 0	
17090	9600	0	40706	74K / 6947*	0 / 0	
17091	9810	20218	95498	52K / 7617	52K / 4003	[1]
17092	12130	9277	50347	52K / 7617	0 / 0	
17093	2865	18936	95792	52K / 7617	52K / 4003	[1]
17094	14110	0	68547	74K / 8732	0 / 0	
17095	32305	**	103600	74K / 8732	0 / 0	
17096	20811	0	57413	74K / 8732	0 / 0	
17097	11555	0	71264	74K / 8732	0 / 0	
17098	2865	13005	60747	74K / 8732	0 / 0	
17099	14771	0	63499	74K / 8732	0 / 0	
17100	6614	0	60132	74K / 6947*	0 / 0	
17101	44100	0	118401	74K / 8732	52K / 4003	[1]
17102	2865	0	47610	52K / 7617	0 / 0	
17103	4960	0	51296	52K / 7617	0 / 0	
17104	5955	0	60644	74K / 8732	0 / 0	
17300	10000	0	35693	52K / 7617	0 / 0	
17500	1000	6395	32571	52K / 7617	0 / 0	
17501	2500	0	63227	74K / 8732	0 / 0	
17502	1500	13406	73390	74K / 8732	0 / 0	

* NOTE: NO AEROBRAKE, THE OTV IS EXPENDED ON THESE MISSIONS
**NOTE: EKS IS AN OTV WITHOUT AN AEROBRAKE, DRY WEIGHT = 8057 LB
[1] Without tankset and EKS, OTV is not reusable

Figure 5.6.4-1 Planetary Performance Summary

5.6.5 DOD Propellant Requirements

Figure 5.6.5-1 summarizes the propellant requirements of the four generic DOD missions using the two final OTV configurations (52 Klb and 74 Klb propellant capacity). Two things to note are that all missions but the mid-inclination low can be performed space-based and that all the ground-based missions can be performed by the small (52 Klb) OTV.

PAYLOAD NUMBER	MISSION NAME (INCLINATION, ALTITUDE)	PAYLOAD (UP/DOWN)	SPACED BASED PROP. USAGE, (OTV)	GROUND BASED PROP. USAGE, (OTV)
19036	MID-INCLINATION (63°, 193000 NM)	10,000 / 0	43,900 LB (52K)	38,800 LB (52K)
19037	MID-INCLINATION, LOW (63°, 1000 NM)	110,000 / 0	(216,100 LB)*	33,900 LB (52K)
19517	POLAR (90°, 4000 NM)	5,000 / 0	74,100 LB (74K)	25,327 LB (52K)
19035	GEO (0°, 19300 NM)	10,000 / 0	41,000 LB (52K)	41,400 LB (52K)

* PROPELLANT CAPACITY EXCEEDED

Figure 5.6.5-1 DOD Mission Performance

5.6.6 Other Performance Analyses

Several performance analysis tasks were completed in support of auxiliary trades and are summarized in the following paragraphs.

5.6.6.1 DOD Small Stage

A trade was conducted to see if a small custom OTV could be competitive for low-energy missions. A 40 Klb propellant capacity stage with a dry weight of 7200 lb was sufficient for this application and its performance is shown for the four generic DOD missions in Figure 5.6.6-1. All missions are launched into the correct orbital plane by the LCV with the OTV returning to a 28.5° inclination, high traffic orbit, for retrieval. For further details see section 3.3 - Design Driver Mission Analyses.

PAYLOAD NO.	MISSION NAME	PAYLOAD (UP/DOWN)	PROP. USAGE (LB)
19036	MID-INCLINATION (63°, 19300 NM)	10,000 / 0	37,000
19037	MID-INCLINATION, LOW (63°, 1000 NM)	110,000 / 0	32,900
19517	POLAR (90°, 4000 NM)	5,000 / 0	23,700
19035	GEO (0°, 19300 NM)	10,000 / 0	39,500

NOTE: ALL MISSIONS ARE GROUND BASED, RETURN TO SPACE STATION FOR RETRIEVAL.

Figure 5.6.6-1 DOD 40K OTV Performance

5.6.6.2 Stretch Centaur

To evaluate the reusable vs expendable trade a stretched Centaur was created which could fly the entire mission model. The basic driver missions sized the vehicle, the 25 Klb GEO delivery missions and the 12 Klb up/10 Klb down GEO shack logistics mission. Two sizes of expendable Centaur were required: a 60 Klb version which could perform the delivery mission and a 70 Klb version which, when staged, handled the logistics mission. Dry weights were based on a linear extrapolation of today's Centaur weights using OTV weight trends. Propellant requirements are summarized in Figure 5.6.6-2.

	60K CENTAUR	70K CENTAUR
PROP. CAPACITY	60,000	70,000
DRY WEIGHT	8350	8875
THRUST	33,000	33,000
NO. OF STAGES	1	2
PAYLOAD NO.	15009	15011
MISSION NAME	MANNED GEO SHACK	GEO SHACK LOGISTICS
PAYLOAD (UP/DOWN)	25,080 / 0	12,000 / 10,000
PROP. USAGE (LB)	59,429	133,052

Figure 5.6.6-2 Stretch Centaur Performance

5.6.6.3 Extra Large OTV

Figure 5.6.6-3 summarizes the performance of the very large 240K propellant capacity OTV. This vehicle has a dry weight of 17,740 lb and a thrust level of 30,000 lb. The objective of this vehicle is to eliminate two-stage and tankset operations. The missions displayed are the only ones for which this can be done. Several planetary missions remain with tanksets because their high velocity requirements cannot be supplied by a large dry weight OTV. Contrast these propellant requirements with those for the baseline OTVs as shown in the lunar and planetary propellant requirement sections above.

PAYLOAD NO.	MISSION NAME	PAYLOAD (UP/DOWN)	PROP. USAGE (LB)
-	MAX GEO DELIVERY	100,000 / 0	234,300
17202	LUNAR SURFACE EXPLORER	32,850 / 0	104,800
17203	UNMANNED LUNAR SURF. DELIVERY	72,680 / 0	173,300
17206	LUNAR ORBIT STATION	93,000 / 0	212,800
17207	LUNAR SURFACE SORTIE CAMP	72,680 / 20,000	189,500
17088	COMET NUCL. SAMPLE RETURN	19,945 / 0	228,100
17101	VENUS SAMPLE RETURN	44,100 / 0	114,500

Figure 5.6.6-3 Very Large OTV Performance

6.0 SELECTED DESIGNS

The following paragraphs describe the recommended OTV concepts capable of performing the Rev. 9 missions. Paragraph 6.1 shows the ground and space vehicles recommended for an STS constrained launch environment. Paragraph 6.2 shows similar data for the OTV designs that are optimum when a Large Cargo Vehicle is available for launch.

6.1 UPDATED STS/LAUNCH OTVs

6.1.1 Descriptions

6.1.1.1 Updated STS/ACC Launched, Ground-Based OTV

Figure 6.1.1-1 shows an updated version of the recommended ground-based OTV from the 1984/85 study effort. The major updates are as follows: Beefed up structure to provide a margin for the vibration environment anticipated in the ACC; the addition of debris shielding; and a redesign of the aerobrake to move the rib fold outboard and straighten the ribs. This vehicle is not manrated and utilizes a 38 ft aerobrake. It is capable of delivering 15 Klb to GEO and also capable of performing the multiple payload delivery missions consisting of a 12 Klb delivery and a 2 Klb return (Rev. 9 early requirements).

WEIGHT (Lbs)	
AEROBRAKE	1566
TANKS	524
STRUCTURE	774
ENVIRONMENTAL CONTROL	424
MAIN PROPULSION	904
ORIENTATION CONTROL	187
ELECTRIC SYSTEMS	613
G, N & C	148
CONTINGENCY 15%	772
DRY WEIGHT	5920
PROPELLANTS, ETC.	45434
LOADED WEIGHT	51354

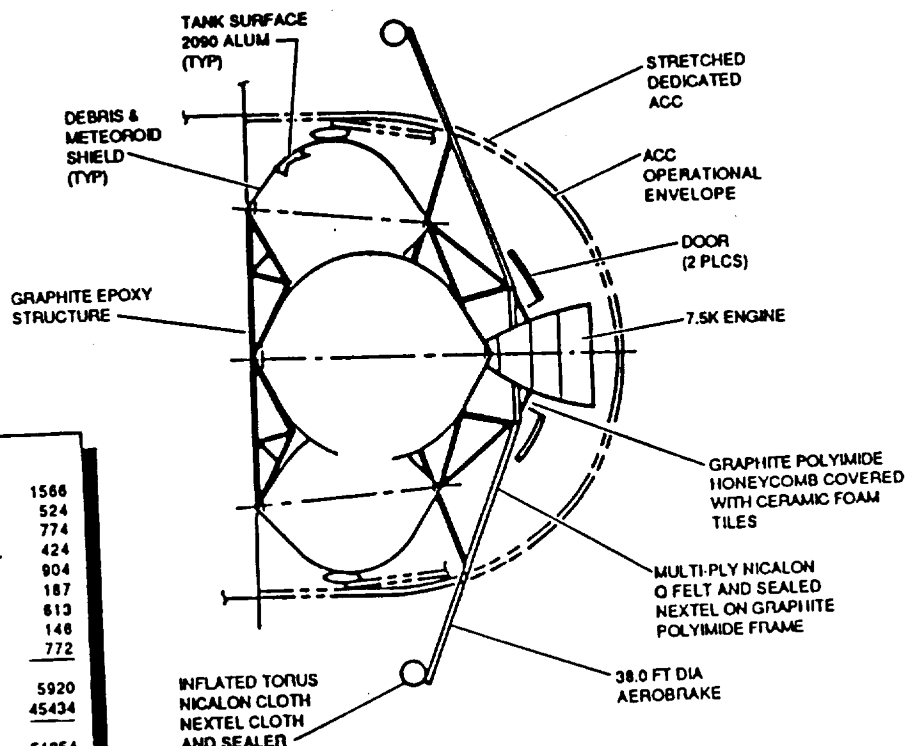


Figure 6.1.1-1 Updated STS GB OTV

6.1.1.2 Updated STS/Cargo Bay Launched, Space-Based OTV

The primary updates to the space-based cryogenic OTV concept developed in the 1984/85 study effort are with regard to overall sizing and additional meteoroid and debris protection. The revision in overall sizing results from the updated mission model being used for this study (Rev. 9). This mission model requires a 74 Klb propellant capacity OTV to perform the 12,000 pounds up, 10,000 pounds back manned GEO Sortie and geoshack Logistics missions. Therefore, the vehicle has been scaled up in size accordingly from the 55 Klb propellant capacity required in the earlier effort. This vehicle is called the "clean-sheet" space-based OTV. It is designed to be launched in the STS cargo bay and robotically assembled at the Space Station.

WEIGHT (Lbs)	
AEROBRAKE	1800
TANKS	1025
STRUCTURE	1370
SUPPORT (ASE)	114
ENVIRONMENTAL CONTROL	730
MAIN PROPULSION	1288
ORIENTATION CONTROL	265
ELECTRIC SYSTEMS	533
G, N & C	160
CONTINGENCY 15%	1093
DRY WEIGHT	8378
PROPELLANTS, ETC.	74015
LOADED WEIGHT	82393

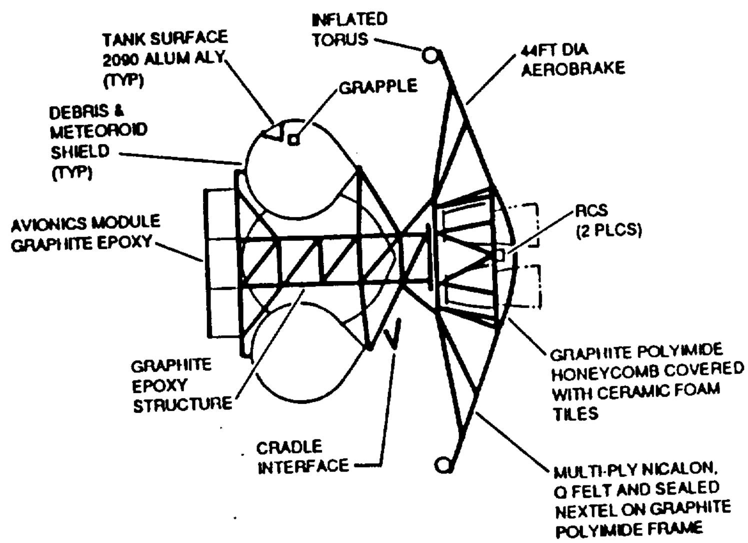


Figure 6.1.1-2 "Clean-Sheet" Space-Based OTV

6.1.2 Weight Statements - STS Launched OTVs

6.1.2.1 45K - ACC OTV Weights

Total flight vehicle weight for the ground-based, 45 Klb propellant, ACC launched OTV is summarized in Table 6.1.2-1. Dry weight, non-propulsive fluids and useable propellants are shown. Individual items include a 15% contingency allowance. Table 6.1.2-2 shows the detailed dry weight breakdown by WBS element.

Table 6.1.2-1 Stage Weight Summary - Ground-Based Cryo 45 Klb Propellant Load

WBS Group	Weight (lb)
Structures	1223
Propellant Tanks	603
Propulsion	726
Main Engines	313
Reaction Control System	215
GN&C	180
Comm & Data Handling	303
Electrical Power	403
Thermal Control	153
Aerobrake	1801
DRY WEIGHT	5920
Fluids	
Residual - LH2	96
Residual - LO2	579
Coolant	10
Hydrazine	400
Pressurant	14
INERT WEIGHT	7019
USABLE MN. PROP	
Fuel - LH2 w/FPR	6332
Oxidizer - LO2 w/FPR	37993
IGNITION WEIGHT	51354
MASS FRACTION $\frac{44335 \text{ Main Prop w/FPR}}{51354 \text{ Ignition Weight}}$	0.86

Table 6.1.2-2 Detailed Dry Weight Breakdown
Ground-Based Cryo. 45 Klb Propellant Load

WBS Group	Element	Weight	
2.0	Structures		
2.1	Airframe		634
	Truss Work	552	
	Contingency	82	
2.2	Thrust Structure		29
	Engine Truss	25	
	Contingency	4	
2.3	Equipment Mounts		111
	Rems & Hydrazine Tank	12	
	Electrical Equip.	46	
	Avionic Equipment	39	
	Contingency	14	
2.4	Payload Attachment		46
	Adapter Attachment	40	
	Contingency	6	
2.5	Micrometeoroid Shield		334
	Bumper	261	
	Standoff	30	
	Contingency	43	
2.6	Handling & Storage		69
	PIDA Fixtures	30	
	RMS Fixtures	30	
	Contingency	9	
	GROUP 2 TOTAL		1223
3.0	Propellant Tanks		
3.1	Tank Structure		483
	LH2 (2)	242	
	LO2 (2)	178	
	Contingency	63	
3.2	Tank Mounts		120
	LH2	52	
	LO2	52	
	Contingency	16	
	GROUP 3 TOTAL		603
4.0	Propulsion		
4.1	Pressurant & Pneumatic System	131	
	Lines, Valve, X-Ducer	114	
	Contingency	17	
4.2	Prop, FV&D System - Fuel		234
	Feed	73	
	Vent & Drain	100	
	Pressurization	31	
	Contingency	31	
4.3	Prop., FV&D System - Ox		204
	Feed	65	
	Drain & Vent	82	
	Pressurization	31	
	Contingency	26	

Table 6.1.2-2 Detailed Dry Weight Breakdown
Ground-Based Cryo. 45 Klb Propellant Load
(Continued)

WBS Group	Element	Weight	
4.4	Prop. Utilization System		129
	Probes	44	
	Computer	68	
	Contingency	17	
4.5	Misc. System		28
	Pyro Cable Cutter	24	
	Contingency	4	
	GROUP 4 TOTAL		726
5.0	Main Engines		276
5.1	Engine		
	Engine	240	
	Contingency	36	
5.2	Actuators		37
	Actuator	32	
	Contingency	5	
	GROUP 5 TOTAL		313
6.0	Reaction Control System		43
6.1	REM Assy		
	REMS	37	
	Contingency	6	
6.2	Tank		94
	Hydrazine	82	
	Contingency	12	
6.3	Plumbing & Installation		78
	Line, Valves, X-ducers	68	
	Contingency	10	
	GROUP 6 TOTAL		215
7.0	GN&C		166
7.1	Control & Guidance		
	Flight Controller & TLM	52	
	IMU Processor	37	
	GPS Receiver	45	
	Thrust Controller	10	
	Contingency	22	
7.2	Navigation		14
	Star Scanner	12	
	Contingency	2	
	GROUP 7 TOTAL		180

Table 6.1.2-2 Detailed Dry Weight Breakdown
Ground-Based Cryo. 45 Klb Propellant Load
(Continued)

WBS Group	Element	Weight	
8.0	Communications & Data Handling		
8.1	Communications		263
	GPS Antenna System	15	
	STDN/TDRS X-ponders	16	
	20w RF Power Amp	6	
	S Band RF System	180	
	Deploy Timer	12	
	Contingency	34	
8.2	Data Management		40
	Central Computer	20	
	CMD & Data Handling	15	
	Contingency	5	
8.3	Video		-0-
	N/A	-0-	
	GROUP 8 TOTAL		303
9.0	Electrical Power		
9.1	Fuel Cell System		109
	Fuel Cell	70	
	Fuel Cell Plumbing	25	
	Contingency	14	
9.2	Radiator System		52
	Radiator	35	
	Plumbing	10	
	Contingency	7	
9.3	Residual H_2O System		15
	Tank	8	
	Plumbing	5	
	Contingency	2	
9.4	Reactant Tank & Plumbing		94
	LH2	9	
	LO2	7	
	LH2 Plumbing	33	
	LO2 Plumbing	31	
	Contingency	14	
9.5	Power Distribution		133
	Wire, Harness Connectors	116	
	Contingency	17	
	GROUP 9 TOTAL		403
10.0	Thermal Control		
10.1	Insulation		109
	LH2 Tank	61	
	LO2 Tank	32	
	ACS Tank	2	
	Contingency	14	

Table 6.1.2-2 Detailed Dry Weight Breakdown
Ground-Based Cryo. 45 Klb Propellant Load
(Continued)

WBS Group	Element	Weight	
10.2	Thermal Control		44
	ACS (Htr.Tape)	3	
	FC System (Htr Tape)	3	
	Prop.Line, F/E Sys.	16	
	Engine Compt	10	
	Electrical System	6	
	Contingency	6	
	GROUP 10 TOTAL		153
11.0	Aerobrake		990
11.1	Heat Shield		
	Hardshell w/TPS	531	
	TPS Flex Quilt	330	
	Contingency	129	
11.2	Mechanical System		268
	Doors w/Motor	85	
	Torus System	112	
	Springs	36	
	Contingency	35	
11.3	Support Structure		543
	Ribs	249	
	Ring Frames	223	
	Contingency	71	
	GROUP 11 TOTAL		1801
15.0	Propellants		45010
15.1	Main		
	Usable LH2 incl.FPR	6332	
	Usable LO2 incl.FPR	37993	
	Residual LH2	96	
	Residual LO2	579	
	Press.Pneum.(He)	10	
15.2	FC Coolant & Reactants		10
	Coolant	10	
15.3	ACS		414
	Hydrazine	400	
	Pressurant - GH2	14	
	GROUP 15 TOTAL		45434

6.1.2.2 74 Klb "Clean Sheet" Space-Based OTV

Total flight vehicle weight for the "clean sheet" space-based, 74 Klb propellant, STS launched OTV is summarized in Table 6.1.2-3. Dry weight, non-propulsive fluids and useable propellants are shown. Individual items include a 15% contingency allowance. Table 6.1.2-4 shows the detailed dry weight breakdown by WBS element.

Table 6.1.2-3 Stage Weight Summary - Space-Based Cryo 74 Klb Propellant Load

WBS Group	Weight (lb)
Structures	2182
Propellant Tanks	1178
Propulsion	986
Main Engines	625
Reaction Control System	305
GN&C	184
Comm & Data Handling	257
Electrical Power	357
Thermal Control	234
Aerobrake	2070
DRY WEIGHT	8378
Fluids	
Residual - LH2	159
Residual - LO2	954
Coolant	15
INERT WEIGHT	9506
USABLE MN. PROP	
Fuel - LH2 w/FPR	10412
Oxidizer - LO2 w/FPR	62475
IGNITION WEIGHT	82393
MASS FRACTION	
72887 Main Prop w/FPR / 82393 Ignition Weight	0.88

Table 6.1.2-4 Detailed Weight Breakdown
Space-Based Cryo. 74 Klb Propellant Load

WBS Group	Element	Weight	
2.0	Structures		
2.1	Airframe		1015
	Center Truss	485	
	Fwd Truss	244	
	Aft Truss	98	
	Fittings	56	
	Contingency	132	
2.2	Thrust Structure		97
	Engine Truss	84	
	Contingency	13	
2.3	Equipment Mounts		128
	REMS	7	
	Accummulators	15	
	Electrical	37	
	Avionic	52	
	Contingency	17	
2.4	Payload/Avionics Ring		198
	Avionic Ring	142	
	Payload Adapter	30	
	Contingency	26	
2.5	Micrometeoroid Shield		606
	Bumper	487	
	Standoffs	40	
	Contingency	79	
2.6	Handling & Storage		138
	Crane Interface	90	
	RMS Grapple Fixture	30	
	Contingency	13	
	GROUP 2 TOTAL		2182
3.0	Propellant Tank		
3.1	Tank Structure		902
	LH2 (2)	294	
	LO2 (2)	128	
	Center Post	363	
	Contingency	117	
3.2	Tank Mounts		276
	LH2 (4)	120	
	LO2 (4)	120	
	Contingency	36	
	GROUP 3 TOTAL		1178
4.0	Propulsion		
4.1	Pressurant & Pneumatic System		48
	Lines, Valve, Transducer	42	
	Contingency	6	

Table 6.1.2-4 Detailed Weight Breakdown
Space-Based Cryo. 74 Klb Propellant Load
(Continued)

WBS Group	Element	Weight	
4.2	Prop. FV&D System Fuel		265
	Feed	108	
	Vent & Drain	91	
	Press.	31	
	Contingency	35	
4.3	Prop. FV&D System Ox		264
	Feed	107	
	Vent & Drain	91	
	Press.	31	
	Contingency	35	
4.4	Prop. Utilization System		279
	Probes	83	
	Computer	160	
	Contingency	36	
4.5	Miscellaneous System		131
	Eng. Removal Q/D	114	
	Contingency	17	
	GROUP 2 TOTAL		987
5.0	Main Engines		
5.1	Engines		552
	Engines	480	
	Contingency	72	
5.2	Actuators		74
	Actuators	64	
	Contingency	10	
	GROUP 5 TOTAL		626
6.0	Reaction Control System		
6.1	Thrusters		69
	REM	60	
	Contingency	9	
6.2	Accummulators		71
	Tank	62	
	Contingency	9	
6.3	Plumbing		96
	Valves & Lines	83	
	Contingency	3	
6.4	Conditioning Units		69
	Units	60	
	Contingency	9	
	GROUP 6 TOTAL		305

Table 6.1.2-4 Detailed Weight Breakdown
Space-Based Cryo. 74 Klb Propellant Load
(Continued)

WBS Group	Element	Weight	
7.0	GN&C		
7.1	Guidance & Control		159
	Flt.Controllers & TLM	60	
	IMU Processor	48	
	GPS Receiver	20	
	Thrust Controller	10	
	Contingency	21	
7.2	Navigation		25
	STAR Scanner	22	
	Contingency	3	
	GROUP 7 TOTAL		184
8.0	Communication & Data Handling		
8.1	Communications		199
	GPS Antenna System	15	
	STDN/TDRS X-Ponder	32	
	20W RF Power Amp	12	
	S-Band RF System	100	
	TLM Power Supply	14	
	Contingency	26	
8.2	Data Handling		58
	Central Computer	20	
	CMD & Data Management	30	
	Contingency	8	
8.3	Video System		-0-
	GROUP 8 TOTAL		257
9.0	Electrical Power		
9.1	Fuel Cell System		103
	Fuel Cell	70	
	Plumbing	20	
	Contingency	13	
9.2	Radiator System		58
	Radiator	35	
	Plumbing	15	
	Contingency	8	
9.3	Residual H_2O System		17
	Accummulator Tanks	10	
	Plumbing	5	
	Contingency	2	
9.4	Reactant Plumbing		29
	Plumbing	25	
	Contingency	4	
9.5	Power Distribution		150
	Wire, Harness, Connector	130	
	Contingency	20	
	GROUP 9 TOTAL		357

Table 6.1.2-4 Detailed Weight Breakdown
Space-Based Cryo. 74 Klb Propellant Load
(Continued)

WBS Group	Element	Weight	
10.0	Thermal Control		176
10.1	Insulation		
	MPS Tanks	146	
	ACS Tanks	5	
	FC Tanks	2	
	Contingency	23	
10.7	Thermal Control		58
	Engine Thrust Comp.	16	
	Prop.Lines & F/C Sys.	24	
	Electrical & Plumbing	10	
	Contingency	8	
	GROUP 10 TOTAL		234
11.0	Aerobrake		1008
11.1	Heat Shield		
	Hardshell w/TPS	101	
	Flex.TABI	776	
	Contingency	131	
11.2	Mechanical System		328
	Doors	133	
	Torus System	152	
	Contingency	43	
11.3	Support Structure		734
	Ribs & Struts	417	
	Center Structure	221	
	Contingency	96	
	GROUP 11 TOTAL		2070
15.0	Propellants		74000
15.1	Main Propellants		
	Usable FU LH2 w/FPR	10412	
	Usable OX LO2 w/FPR	62475	
	Residual FU LH2	159	
	Residual OX LO2	954	
15.2	FC Coolant & Reactant		15
	Coolant	15	
	GROUP 15 TOTAL		74015

6.1.3 Mission Applications

Basic performance data for the ground and space-based configurations is shown in Figures 6.1.3-1 and 6.1.3-3. These graphs show propellant requirements as a function of payload weight for three different types of geosynchronous missions: delivery, retrieval, and round trip.

The ground-based ACC OTV (Figure 6.1.3.1) is capable of delivering a 15K payload to geo, retrieving a 17.6K payload from GEO, or taking a 8K payload to GEO and back. The 72 K maximum lift capability of the Shuttle is required to perform the 15K delivery mission. The weight summary for this mission is shown in Table 6.1.3-2. Allocations are shown for OTV/payload, ACC effective weight, Orbiter delta OMS propellant and OTV retrieval ASE. The ACC drag adjustment is a streamlining effect on the STS boost stack due to the presence of the ACC on the bottom of the ET. The delta Orbiter OMS propellant arises because the Shuttle OMS-1 &2 orbit insertion burns are performed with the Orbiter and the 15K spacecraft only, and not the 50K OTV which delivers itself to orbit after separating at MECO.

The space-based 74K OTV performance is shown in Figure 6.1.3-3. It is capable of delivering a 29.1K payload to GEO, retrieving a 32.2K spacecraft from GEO, and taking a 15.3K payload to GEO and back.

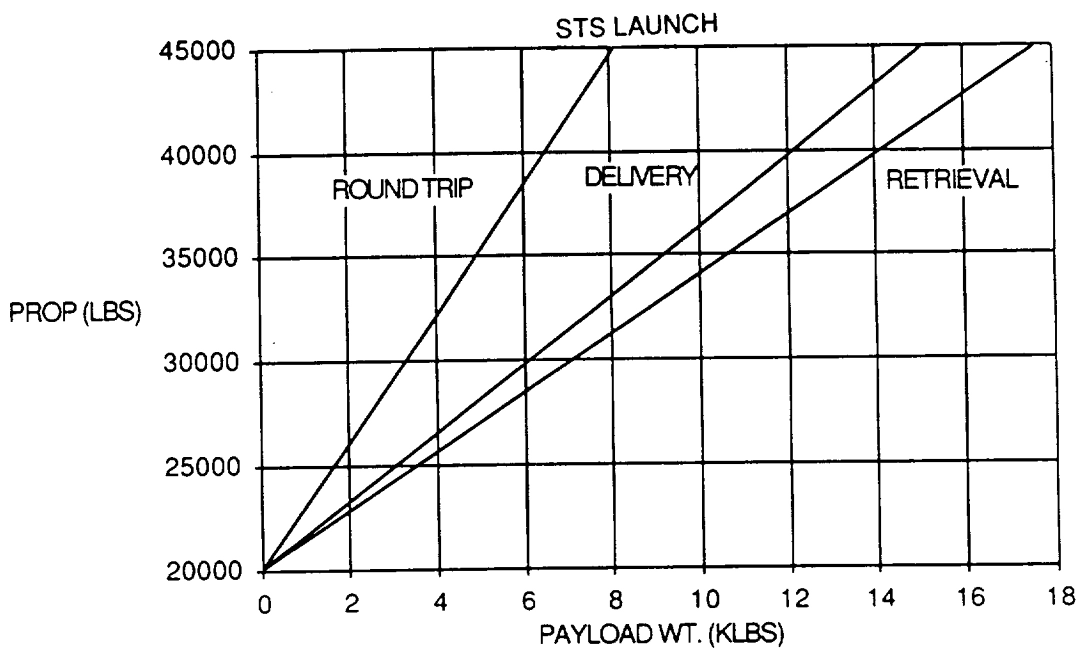


Figure 6.1.3-1 ACC OTV GEO Performance

OTV	5920 LB	STAGE DRY WEIGHT
OTV PROP.	45000 LB	MAX PROPELLANT CAPACITY
PAYLOAD ASE	1570 LB	
STS FITTINGS	930 LB	5 FITTINGS FOR ST TYPE MOUNTING
STS UMBILICAL	640 LB	ST (SPACE TELESCOPE) TYPE MOUNTING
PAYLOAD	15040 LB	MAX GEO P/L CAPABILITY
ACC	3690 LB	
WT. AT MECO	4140 LB	NON-STAGED PORTION
DRAG ADJUST	-450 LB	DRAG STREAMLINING EFFECT
Δ OMS PROP.	-1650 LB	OMS 1 & 2 DO NOT BOOST OTV
OTV RETURN ASE	2659 LB	
FITTINGS	1033 LB	FITTINGS TO RECEIVE OTV AT EOM
CRADLE SUPPT.	796 LB	EXTRA PIECES TO COMPLETE OTV ASE STRUCTURE
PURGE/PRESS.	530 LB	MAINTAIN TANKAGE INTEGRITY
PIDA	300 LB	STUB FIXTURE TO HOLD OTV FOR DISASSEMBLY
STS P/L AT MECO	71700 LB	CONSISTENT WITH GROUNDRULED 72K LIFT CAPABILITY

Figure 6.1.3-2 ACC OTV Mission Weight Summary

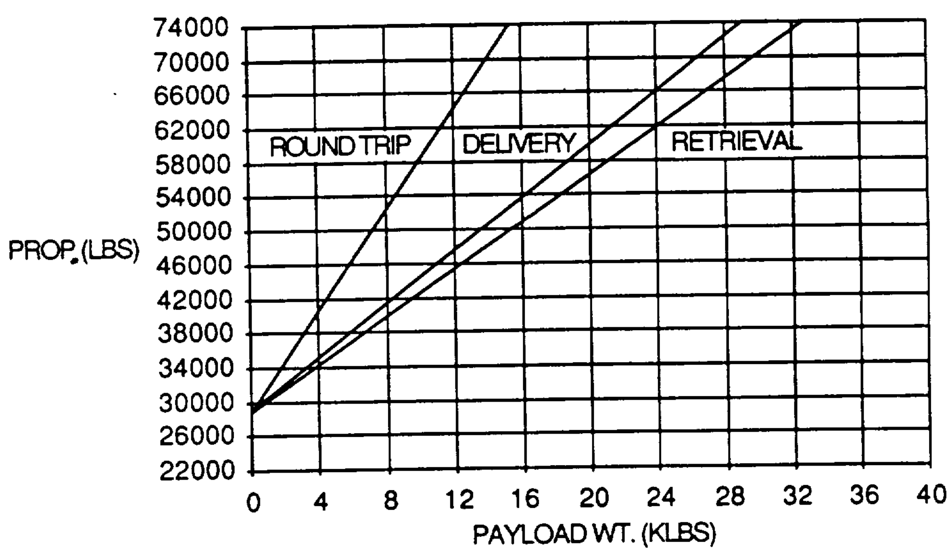


Figure 6.1.3-3 74K Optimized Space-Based GEO Performance

6.2 LARGE CARGO VEHICLE LAUNCHED OTVs

This section describes the preferred Orbital Transfer Vehicle vehicles in the era where a large cargo vehicle is available and Scenario 2 missions are to be performed. It will comprise two types of orbital transfer vehicles. A three in-line engine, four side-by-side tank, unmanned, ground-based vehicle with a 52,000 pound propellant capacity will support initial missions. This vehicle will be used throughout the operational period. A generally similar manned, space-based vehicle with a 74,000 pound propellant capacity will be made operational as soon as it can be supported by the Space Station. All manned missions will be launched from a space-base, but the space-based vehicle can be launched from the ground as well. Its initial mission will be ground-based -- returning to residence at the Space Station upon completion of the mission.

6.2.1 Descriptions

6.2.1.1 Ground-Based - Unmanned OTV

The ground-based OTV is shown in Figure 6.2.1-1. The 25 foot diameter was selected to minimize the length occupied in the LCV. For return in an STS, the hydrogen tanks are expended. The 14 1/2 foot diameter core section containing propulsion, avionics, structure, and the hard reuseable portion of the aerobrake along with the oxygen tanks fit inside the STS payload bay.

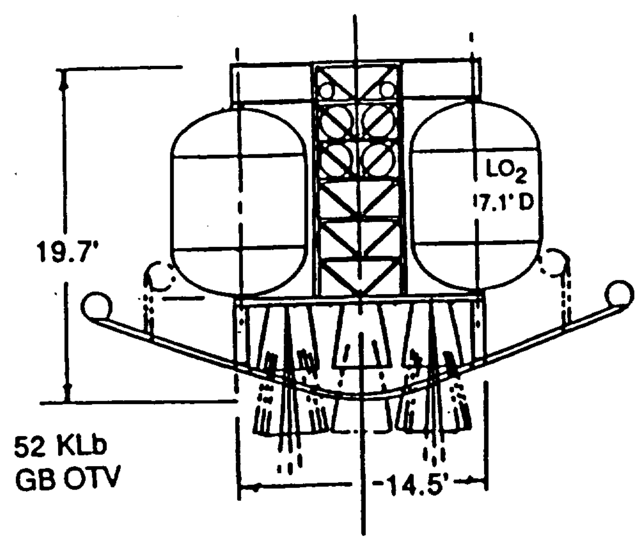


Figure 6.2.1-1 52 Klb GB OTV

The major features of this vehicle are as follows:

- o 25 foot diameter
- o Less than 20 foot long
- o 4 cylindrical propellant tanks
- o Three in-line engines (Isp 475)
- o Non-manrated
- o 32 foot diameter aerobrake
- o Composite structures
- o IOC of 1995
- o Minimal changes required for manrating/space-basing
- o Propellant capacity of 52 Klb
- o Sized for 15 Klb payload delivery to GEO

6.2.1.2 Space-Based - Man-Rated OTV

The space-based manrated OTV is shown in Figure 6.2.1-2. The major physical differences between this vehicle and the 52K stage are:

- o 74K propellant
- o Sized to deliver 25K to GEO (and 12K delivery, 10K return)
- o Manrated
- o 38 foot diameter aerobrake
- o 25 1/2 foot length
- o Additional meteoroid shielding
- o SOFI insulation on LH_2 tanks replaced with MLI
- o Quick disconnects in propulsion system for robotic changeout
- o For return to earth by STS, both hydrogen and one oxygen tank are expended
- o IOC of 1996 (as soon as SS available)

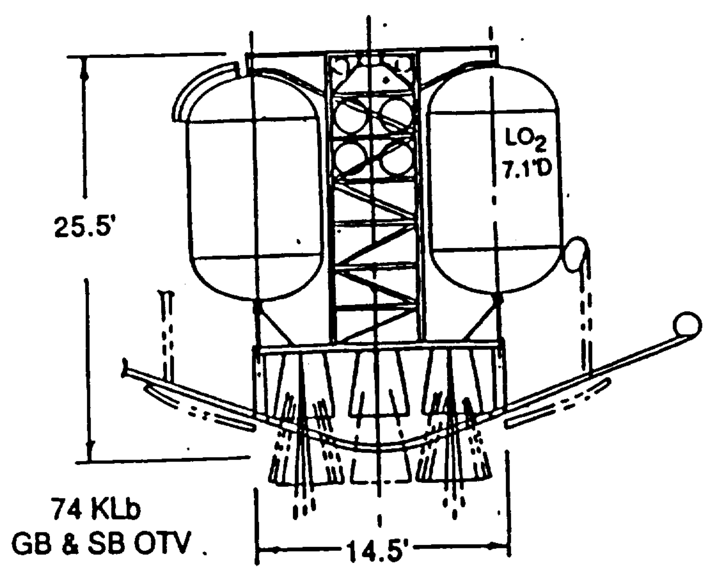


Figure 6.2.1-2 74K SB OTV

6.2.2 Weight Statements - LCV Launched OTVs

6.2.2.1 52 Klb LCV OTV Weights

Total flight vehicle weight for the ground-based, 52 Klb propellant, Large Cargo Vehicle (LCV) launched OTV is summarized in Table 6.2.2-1. Dry weight, nonpropulsive fluids and useable propellants are shown. Individual items include a 15% contingency allowance. Table 6.2.2-2 shows the detailed dry weight breakdown by WBS element.

Table 6.2.2-1 Stage Weight Summary
Ground-Based Cryo 52 Klb Propellant Load
Wide Body Transport

WBS Group	Weight (lb)
Structures	1488
Propellant Tanks	1509
Propulsion	896
Main Engines	793
Reaction Control System	305
GN&C	180
Comm & Data Handling	303
Electrical Power	444
Thermal Control	271
Aerobrake	1491
DRY WEIGHT	7680
Fluids	
Residual - LH2	111
Residual - LO2	669
Coolant	15
INERT WEIGHT	8475
USABLE MN. PROP	
Fuel - LH2	7317
Oxidizer - LO2	43903
IGNITION WEIGHT	59695
MASS FRACTION	
51220 Main Prop w/FPR / 59695 Ignition Weight	0.86

Table 6.2.2-2 Detailed Weight Breakdown
Ground-Based Cryo 52 Klb Propellant Load
Wide Body Transport

WBS Group	Element	Weight	
2.0	Structures		
2.1	Airframe		720
	Center Truss	285	
	LH2 Truss Support	181	
	LO2 Truss Support	133	
	Misc. Attachments	28	
	Contingency	93	
2.2	Thrust Structure		99
	Engine Truss	86	
	Contingency	13	
2.3	Equipment Mounts		128
	REMS	7	
	Accummulators	15	
	Electrical	37	
	Avionics	52	
	Contingency	17	
2.4	Payload Adapter		46
	Adapter Attachment	40	
	Contingency	6	
2.5	Micrometeoroid Shield		357
	Bumper	279	
	Standoff	31	
	Contingency	47	
2.6	Handling & Storage		138
	Grapple Fixture	120	
	Contingency	18	
	GROUP 2 TOTAL		1488
3.0	Propellant Tanks		
3.1	Tank Structure		1348
	LH2 (2)	458	
	LO2 (2)	325	
	Center Post LH2	222	
	Center Post LO2	167	
	Contingency	176	
3.2	Tank Mounts		161
	LH2	70	
	LO2	70	
	Contingency	21	
	GROUP 3 TOTAL		1509
4.0	Propulsion		
4.1	Pressurant & Pneumatic System	54	
	Line Valves X-Ducer	47	
	Contingency	7	

Table 6.2.2-2 Detailed Weight Breakdown
Ground-Based Cryo. 52 Klb Propellant Load
Wide Body Transport
(Continued)

WBS Group	Element	Weight	
4.2	Prop FV&D System - FU		281
	Feed	113	
	Vent &* Drain	98	
	Press.	34	
	Contingency	36	
4.3	Propellant FV&D System - OX	231	
	Feed	113	
	Vent & Drain	98	
	Press.	34	
	Contingency	36	
4.4	Prop. Utilization System		280
	Probe	83	
	Computer	160	
	Contingency	37	
	GROUP 4 TOTAL		896
5.0	Main Engines		
5.1	Engines		683
	Engines	594	
	Contingency	89	
5.2	Actuators		110
	Actuators	76	
	Contingency	14	
	GROUP 5 TOTAL		793
6.0	Reaction Control System		
6.1	Thrusters		69
	REM	60	
	Contingency	9	
6.2	Accummulations		71
	Tanks	62	
	Contingency	9	
6.3	Plumbing		96
	Valves & Lines	83	
	Contingency	13	
6.4	Conditioning Units		69
	Turbo Pump Assy	35	
	Gas Generators	5	
	Heat Exchanger	20	
	Contingency	9	
	GROUP 6 TOTAL		305
7.0	Guidance, Navigation & Control		
7.1	Guidance & Control		166
	Flight Controller	52	
	IMU Processor	37	
	GPS Receiver	45	
	Thrust Controller	10	
	Contingency	22	

Table 6.2.2-2 · Detailed Weight Breakdown
Ground-Based Cryo. 52 Klb Propellant Load
Wide Body Transport
(Continued)

WBS Group	Element	Weight	
7.2	Navigation		14
	STAR Scanner	12	
	Contingency	2	
	GROUP 7 TOTAL		180
8.0	Communication & Data Handling		
8.1	Communications		263
	GPS Antenna System	15	
	SIDN/TDRS X-Ponder	16	
	20W RF Power Amp	6	
	S Band RF System	180	
	Deploy Timer	12	
	Contingency	34	
8.2	Data Management		40
	Central Computer	20	
	CMD & Data Handling	15	
	Contingency	5	
8.3	Video		-0
	N/A	-0	
	GROUP 8 TOTAL		303
9.0	Electrical Power		
9.1	Fuel Cell System		109
	Fuel Cell	70	
	Plumbing	25	
	Contingency	14	
9.2	Radiator System		52
	Radiator	35	
	Plumbing	10	
	Contingency	7	
9.3	Residual H_2O System		15
	Tank	8	
	Plumbing	5	
	Contingency	2	
9.4	Reactant Tank & Plumbing		94
	LH2	9	
	LO2	7	
	LH2 Plumbing	33	
	LO2 Plumbing	31	
	Contingency	14	
9.5	Power Distribution		174
	Wire & Harness	151	
	Contingency	23	
	GROUP 9 TOTAL		444

Table 6.2.2-2 Detailed Weight Breakdown
Ground-Based Cryo. 52 Klb Propellant Load
Wide Body Transport
(Continued)

WBS Group	Element	Weight	
10.0	Thermal Control		
10.1	Insulation		192
	LH2	132	
	LO2	33	
	ACS Tank	2	
	Contingency	25	
10.2	Thermal Control		79
	ACS	4	
	FC System (Htr Tape)	5	
	Prop.Lines	18	
	Engine Compt.	24	
	Electrical System	18	
	Contingency	10	
	GROUP 10 TOTAL		271
11.0	Aerobrake		
11.1	Heat Shield		557
	Hard Shell w/TPS	130	
	TPS TABI	354	
	Contingency	73	
11.2	Mechanical System		253
	Doors w/Motors	93	
	Torus System	91	
	Springs	36	
	Contingency	33	
11.3	Support Structure		681
	Ribs Fixed & Hinged	170	
	Support Structure	423	
	Contingency	88	
	GROUP 11 TOTAL		1491
15.0	Propellants		
15.1	Main		52000
	Usable LH2 inch FPR	7317	
	Usable LO2 incl FPR	43903	
	Residual LH2	111	
	Residual LO2	669	
15.2	FC Coolant & Reactant		15
	Coolant	15	
	GROUP 15 TOTAL		52015

6.2.2.2 74 Klb Wide Body, Space-Based OTV

Total flight vehicle weight for the space-based, 74 Klb propellant, LCV launched OTV is summarized in Table 6.2.2-3. Dry weight, nonpropulsive fluids and useable propellants are shown. Individual items include a 15% contingency allowance. Table 6.2.2-4 shows the detailed dry weight breakdown by WBS element. This is the "hybrid" OTV which results from first "stretching" the propellant tanks and structure which yields a 74 Klb ground-based OTV. The 74 Klb ground-based vehicle is then man rated and modified by kits for space-based debris environments and serviceability requirements. Table 6.2.2-5 shows how this vehicle weighs 185 lbs more than its equivalent ground-based version.

Table 6.2.2-3 Stage Weight Summary
Space-Based Cryo. 74 Klb Propellant Load
Wide Body Transport

WBS Group	Weight (lb)
Structures	1804
Propellant Tanks	1941
Propulsion	1254
Main Engines	792
Reaction Control System	305
GN&C	184
Comm. & Data Handling	257
Electrical Power	458
Thermal Control	229
Aerobrake	1783
DRY WEIGHT	9009
Fluids	
Residual - LH2	159
Residual - LO2	951
Coolant	15
INERT WEIGHT	10132
USABLE MN. PROP.	
Fuel	10413
Oxidizer	62477
IGNITION WEIGHT	83022
MASS FRACTION	
72890 Main Prop.w/FPR / 83022 Ignition Weight	0.88

Table 6.2.2-4 Detailed Weight Breakdown
Space-Based Cryo. 74 Klb Propellant Load
Wide Body Transport

WBS Group	Element	Weight	
2.0	Structures		855
2.1	Airframe		
	Center Truss	405	
	LH2 Truss Supports	180	
	LO2 Truss Supports	131	
	Misc. Attachments	28	
	Contingency	111	
2.2	Thrust Structure		99
	Engine Truss	86	
	Contingency	13	
2.3	Equipment Mounts		128
	REMS	7	
	Accummulators	15	
	Electrical	37	
	Avionics	52	
	Contingency	17	
2.4	Payload Adapter		35
	Adapter Attachment	40	
	Contingency	6	
2.5	Micrometeoroid Shield		548
	Bumper	434	
	Standoff	43	
	Contingency	71	
2.6	Handling & Storage		128
	Grapple Fixtures	120	
	Contingency	8	
	GROUP 2 TOTAL		1804
3.0	Propellant Tanks		1780
3.1	Tank Structure		
	LH2 (2)	628	
	LO2 (2)	445	
	Center Post LH2	275	
	Center Post LO2	197	
	Contingency	232	
3.2	Tank Mounts		161
	LH2	70	
	LO2	70	
	Contingency	21	
	GROUP 3 TOTAL		1941
4.0	Propulsion		
4.1	Pressurant & Pneumatic System	69	
	Line Valves X-Ducer	60	
	Contingency	9	

Table 6.2.2-4 Detailed Weight Breakdown
Space-Based Cryo. 74 Klb Propellant Load
Wide Body Transport
(Continued)

WBS Group	Element	Weight	
4.2	Prop FV&D System - FU		354
	Feed	140	
	Vent &* Drain	125	
	Press	43	
	Contingency	46	
4.3	Propellant FV&D System - OX	354	
	Feed	140	
	Vent & Drain	125	
	Press	43	
	Contingency	46	
4.4	Prop. Utilization System		280
	Probe	83	
	Computer	160	
	Contingency	37	
4.5	Mics. System		197
	Engine Q/D	171	
	Contingency	26	
	GROUP 4 TOTAL		1254
5.0	Main Engines		
5.1	Engines		683
	Engines	594	
	Contingency	89	
5.2	Actuators		110
	Actuators	96	
	Contingency	14	
	GROUP 5 TOTAL		792
6.0	Reaction Control System		
6.1	Thrusters		69
	REM	60	
	Contingency	9	
6.2	Accumulations		71
	Tanks	62	
	Contingency	9	
6.3	Plumbing		96
	Valves & Lines	83	
	Contingency	13	
6.4	Conditioning Units		69
	Turbo Pump Assy	35	
	Gas Generators	5	
	Heat Exchanger	20	
	Contingency	9	
	GROUP 6 TOTAL		305

Table 6.2.2-4 Detailed Weight Breakdown
Space-Based Cryo. 74 Klb Propellant Load
Wide Body Transport
(Continued)

WBS Group	Element	Weight	
7.0	Guidance, Navigation & Control		159
7.1	Guidance & Control		
	Flight Controller	60	
	IMU Processor	48	
	GPS Receiver	20	
	Thrust Controller	10	
	Contingency	21	
7.2	Navigation		25
	STAR Scanner	22	
	Contingency	3	
	GROUP 7 TOTAL		184
8.0	Communication & Data Management		199
8.1	Communications		
	GPS Antenna System	15	
	SIDN/TDRS X-Ponder	32	
	20W RF Power Amp.	12	
	S Band RF System	100	
	Deploy Timer	14	
	Contingency	26	
8.2	Data Management		58
	Central Computer	20	
	CMD & Data Handling	30	
	Contingency	8	
8.3	Video		-0
	N/A	-0	
	GROUP 8 TOTAL		257
9.0	Electrical Power		103
9.1	Fuel Cell System		
	Fuel Cell	70	
	Plumbing	20	
	Contingency	13	
9.2	Radiator System		57
	Radiator	35	
	Plumbing	15	
	Contingency	7	
9.3	Residual H_2O System		17
	Tank	10	
	Plumbing	5	
	Contingency	2	

Table 6.2.2-4 Detailed Weight Breakdown
Space-Based Cryo. 74 Klb Propellant Load
Wide Body Transport
(Continued)

WBS Group	Element	Weight	
9.4	Reactant Tank & Plumbing		92
	LH2	9	
	LO2	7	
	LH2 Plumbing	33	
	LO2 Plumbing	31	
	Contingency	12	
9.5	Power Distribution		189
	Wire & Harness	165	
	Contingency	24	
	GROUP 9 TOTAL		458
10.0	Thermal Control		
10.1	Insulation		150
	LH2	86	
	LO2	43	
	ACS Tank	2	
	Contingency	19	
10.2	Thermal Control		79
	ACS	4	
	FC System (Htr Tape)	5	
	Prop Lines	18	
	Engine Compt.	24	
	Electrical System	18	
	Contingency	10	
	GROUP 10 TOTAL		229
11.0	Aerobrake		
11.1	Heat Shield		765
	Hard Shell w/TPS	130	
	TPS TABI	536	
	Contingency	99	
11.2	Mechanical System		277
	Doors w/Motors	93	
	Torus System	112	
	Springs	36	
	Contingency	36	
11.3	Support Structure		741
	Ribs Fixed & Hinged	222	
	Support Structure	423	
	Contingency	96	
	GROUP 11 TOTAL		1783

Table 6.2.2-4 Detailed Weight Breakdown
Space-Based Cryo. 74 Klb Propellant Load
Wide Body Transport
(Continued)

WBS Group	Element	Weight	
15.0	Propellants		74000
15.1	Main		
	Usable LH2 inch FPR	10413	
	Usable LO2 incl FPR	62477	
	Residual LH2	159	
	Residual LO2	951	
15.2	FC Coolant & Reactant		15
	Coolant	15	
	GROUP 15 TOTAL		74015

Table 6.2.2-5 Modifications to 74 Klb OTV for Ground to Space-Basing

ITEM	WT.CHANGE (LBM)	REASON
Debris Shield	+ 104	Increased Meteoroid Exposure Time
Engine Q/D	+ 171	Not on GB
Thermal - LH2	- 90	Replace 1/2 in SOFI With MLI for 1 in Total
Net Difference	+ 185	

6.2.3 Mission Applications

Performance data for the 52K and 74K ground-based LCV launched OTVs is shown in Figures 6.2.3-1 and 6.2.3-2. These charts show propellant requirements vs payload delivered for three different types of geosynchronous missions: delivery, retrieval and rountrip.

Using the 52K OTV where possible and the 74K vehicle where needed the entire Rev.9 mission model is covered. The geosynchronous and DOD portion are shown in Figure 6.2.3-3 for Scenario #2. This figure shows payload requirements (mission orbit, size and weight), OTV characteristics and propellant requirements per mission, payload flight distribution schedule, and OTV propellant requirements per year.

The space-based 74K hybrid OTV performance graph is shown in Figure 6.2.3-4. It is capable of delivering 27.6K to GEO as well as retrieving 30.6K and taking 14.5K on a rountrip mission. When this vehicle is used to fly the Rev.9 mission model its propellant requirements are summarized in Figure 6.2.3-5.

The planetary and lunar performance is summarized in Section 5.6, Performance Assessment Methodology.

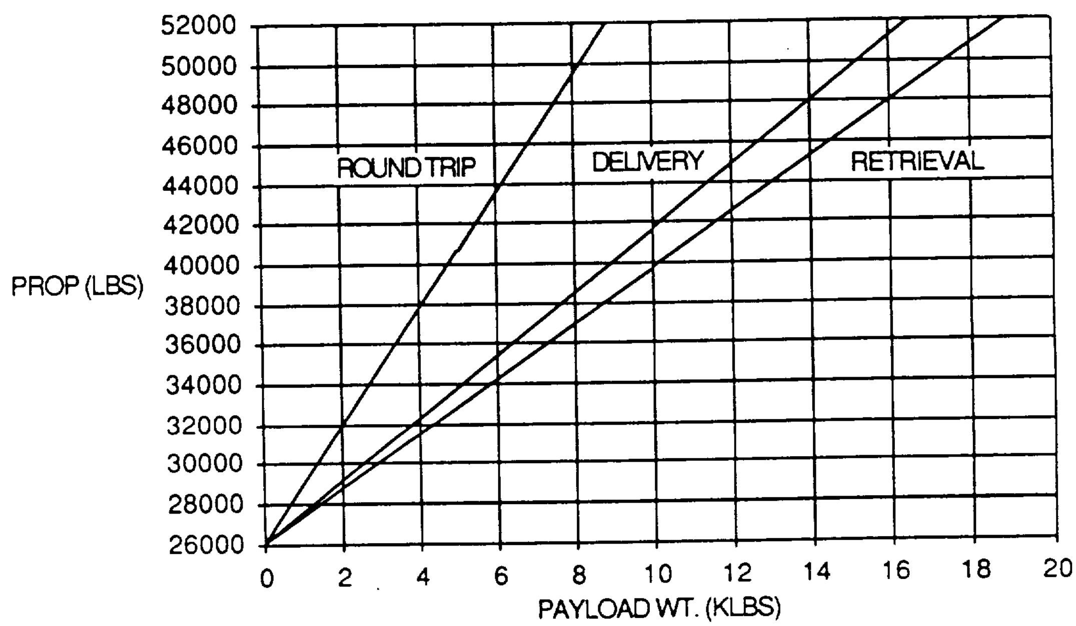


Figure 6.2.3-1 52K Ground-Based OTV GEO Performance (LCV Launch)

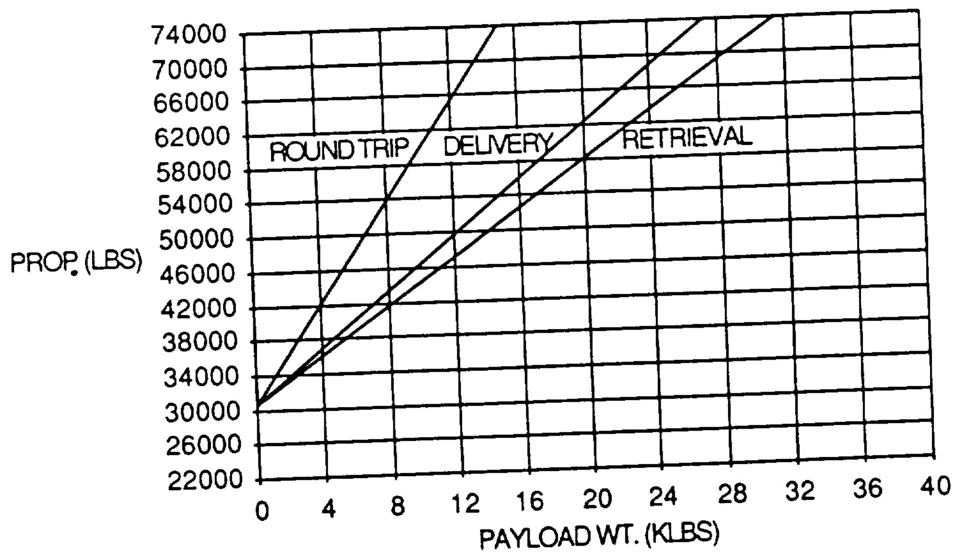


Figure 6.2.3-2 74K Ground-Based OTV Performance

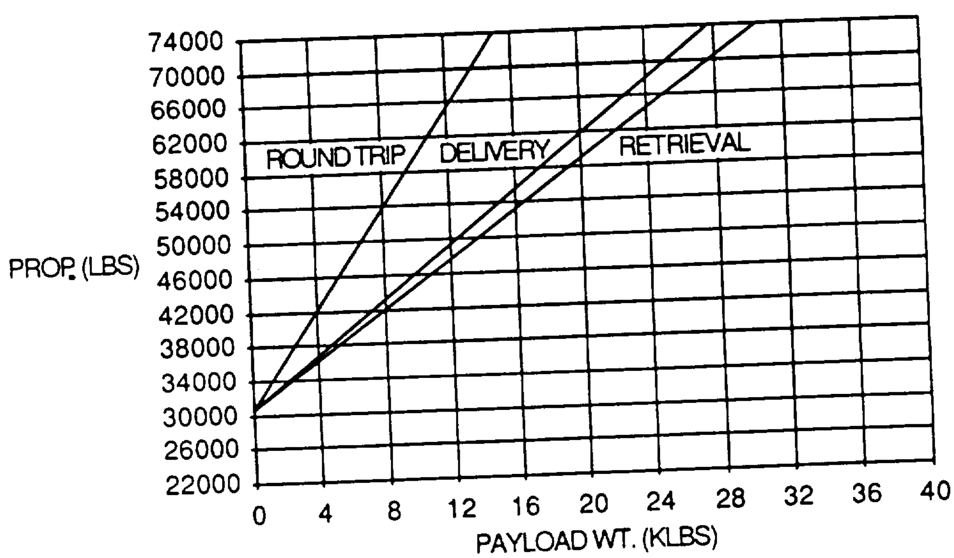


Figure 6.2.3-4 74K Hybrid Space-Based GEO Performance

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	A	C	D	E	F	G	H	I	J	K	L	M	N	O
2	PLD NO	NAME	S/C ORB	S/C ORB	LENGTH	S/C DIAM	S/C WGHT	LENGTH	S/C DIAM	S/C WGHT	PROP CAP	DRY WT	OTV TYPE	TEMPLATE
3			INCL (DEG)	ALT (NM)	(FT) UP	(FT) UP	(LB) UP	(FT) DN	(FT) DWN	(LB) DWN				
4	10100	REFLIGHTS	0	GEO	20	15	20000	0	0	0	62004	8732	74K	GEO
5	13006	EXPERIMENTAL GEOPLATFORM	0	GEO	49.2	6.6	14550	0	0	0	48589	7617	52K	GEO
6	15008	UNMANNED GEO SHACK	0	GEO	13.2	14.9	16720	0	0	0	52048	7617	52K	GEO
7	15009	MANNED GEO SHACK	0	GEO	19.8	14.9	25080	0	0	0	70317	8732	74K	GEO
8	15010	MANNED GEO SORTIE	0	GEO	9.8	14.8	12000	9.8	14.8	10000	70768	8732	74K	GEOMAN
9	15011	GEO SHACK LOGISTICS	0	GEO	22	15	12000	22	15	10000	63072	8732	74K	GEO
10														
11														
12	18072	MOBILE SAT-B	0	GEO	19.7	13.1	14550	0	0	0	48589	7617	52K	GEO
13	18073	MOBILE SAT-C	0	GEO	55	13.1	13230	0	0	0	46494	7617	52K	GEO
14	18074	SETI GEO ANTENNA A	0	GEO	19.8	14.9	21826	0	0	0	64976	8732	74K	GEO
15	18075	SETI GEO ANTENNA B	0	GEO	13.2	14.9	14551	0	0	0	48591	7617	52K	GEO
16	18076	SOLAR TERR GEO EXP	0	GEO	16.4	13.1	7055	0	0	0	36769	7617	52K	GEO
17	18308	VOA A, HF DIR. BROADCAST	0	GEO	19.8	14.9	21826	0	0	0	64976	8732	74K	GEO
18	18309	VOA B, HF DIR. BROADCAST	0	GEO	13.2	14.9	14551	0	0	0	48591	7617	52K	GEO
19	18542	TD OF LARGE GEO SAT	0	GEO	19.7	13.1	13670	0	0	0	47192	7617	52K	GEO
20	18750	COMSAT CLIV RETRIEVAL	0	GEO	0	0	0	30	14.8	10030	39498	7617	52K	GEO
21	18751	COMSAT CLIV DELIVERY	0	GEO	30	14.8	10030	0	0	0	41440	7617	52K	GEO
22														
23	18912	MULTIPLE PLD DELIVERY	0	GEO	30	15	12000	10	15	2000	47311	7617	52K	GEO
24														
25	19035	DOD GENERIC GEO	0	GEO	0	0	10000	0	0	0	41392	7617	52K	GEO
26	19036	DOD GENERIC MID INCLIN	63	19300	0	0	10000	0	0	0	38841	7617	52K	DODINC3
27	19517	DOD GENERIC POLAR	90	4000	0	0	5000	0	0	0	25327	7617	52K	DODPOLAI
28														
29														
30														

REV. 9 MISSION MODEL, SCENARIO 2

	P	Q	R	S	T	U	V	W	X	Y	Z	AA	AB	AC	AD	AE	AF
2						MISSIONS/YEAR											
3	1995	1996	1997	1998	1999	2000	2001	2002	2003	2004	2005	2006	2007	2008	2009	2010	TOTAL
4			1	1		1		1		1		1		1		1	8
5				1													1
6				1													1
7										1							1
8								1	1	2	2	2	2	2	2	2	16
9					1	1	2	1	2	4	4	4	4	4	4	6	37
10																	0
11																	0
12	1																1
13					1												1
14					1												1
15					1												1
16					1												1
17		1															1
18		1															1
19				1													1
20				1			1										2
21				1			1										2
22																	0
23	4	7	9	10	9	11	6	3	1	3	2	3	3	5	4	4	84
24																	0
25	6	6	6	6	6	6	6	6	6	6	6	6	6	6	6	6	96
26	8	8	8	8	8	8	8	8	8	8	8	8	8	8	8	8	128
27	1	1	1	1	1	1	1	1	1	1	1	1	1	1	1	1	16
28	TOTAL MISSIONS / YEAR																
29	20	24	25	31	29	27	53	21	19	26	48	24	24	27	25	28	400
30																	

	AG	AH	AI	AJ	AK	AL	AM	AN	AO	AP	AQ	AR	AS	AT	AU	AV	AW
2								PROPELLANT /YEAR									
3	1995	1996	1997	1998	1999	2000	2001	2002	2003	2004	2005	2006	2007	2008	2009	2010	TOTAL
4	0	0	62004	62004	0	62004	62004	62004	0	62004	62004	62004	0	62004	0	62004	620,038
5	0	0	0	48589	0	0	0	0	0	0	0	0	0	0	0	0	48,589
6	0	0	0	52048	0	0	0	0	0	0	0	0	0	0	0	0	52,048
7	0	0	0	0	0	0	0	0	0	70317	0	0	0	0	0	0	70,317
8	0	0	0	0	0	0	0	70768	70768	141537	141537	141537	141537	141537	141537	141537	1,132,293
9	0	0	0	0	63072	63072	126145	63072	126145	252290	252290	252290	252290	252290	252290	378434	2,333,678
10																	0
11																	0
12	48589	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	48,589
13	0	0	0	0	46494	0	0	0	0	0	0	0	0	0	0	0	46,494
14	0	0	0	0	64976	0	0	0	0	0	0	0	0	0	0	0	64,976
15	0	0	0	0	48591	0	0	0	0	0	0	0	0	0	0	0	48,591
16	0	0	0	0	36769	0	0	0	0	0	0	0	0	0	0	0	36,769
17	0	64976	0	0	0	0	0	0	0	0	0	0	0	0	0	0	64,976
18	0	48591	0	0	0	0	0	0	0	0	0	0	0	0	0	0	48,591
19	0	0	0	47192	0	0	0	0	0	0	0	0	0	0	0	0	47,192
20	0	0	0	39498	0	0	39498	0	0	0	0	0	0	0	0	0	78,995
21	0	0	0	41440	0	0	41440	0	0	0	0	0	0	0	0	0	82,879
22																	0
23	189243	331176	425797	473108	425797	520419	283865	141932	47311	141932	94622	141932	141932	236554	189243	189243	3,974,110
24																	0
25	248355	248355	248355	248355	248355	248355	248355	248355	248355	248355	248355	248355	248355	248355	248355	248355	3,973,676
26	310730	310730	310730	310730	310730	310730	310730	310730	310730	310730	310730	310730	310730	310730	310730	310730	4,971,683
27	25327	25327	25327	25327	25327	25327	25327	25327	25327	25327	25327	25327	25327	25327	25327	25327	405,225
28																	
29	TOTAL PROPELLANT / YEAR																
30	822244	1029154	1072213	1348289	1270111	1229907	1137362	922189	828635	1252491	1134863	1182174	1120170	1276796	1167481	1355630	18,149,709

Figure 6.2.3-3 Ground-Based OTV Performance Data (GEO & DOD, Rev.9, Scenario #2)

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	A	C	D	E	F	G	H	I	J	K	L	M	N	O
2	PLD NO	NAME	S/C ORB	S/C ORB	LENGTH	S/C DIAM	S C WGHT	LENGTH	S/C DIAM	S C WGHT	PROP CAP	DRY WT	OTV TYPE	TEMPLATE
3			INCL (DEG)	ALT (NM)	(FT) UP	(FT) UP	(LB) UP	(FT) DN	(FT) DWN	(LB) DWN				
4	10100	PER FLIGHTS	0	GEO	20	15	20000	0	0	0	61677	8945	74K SB	GEOSB
5	13006	EXPERIMENTAL GEOPLATFORM	0	GEO	49.2	6.6	14550	0	0	0	53175	8945	74K SB	GEOLB
6	15008	UNMANNED GEO SHACK	0	GEO	13.2	14.9	16720	0	0	0	56545	8945	74K SB	GEOLB
7	15009	MANNED GEO SHACK	0	GEO	19.8	14.9	25080	0	0	0	69726	8945	74K SB	GEOLB
8	15010	MANNED GEO SORTIE	0	GEO	9.8	14.8	12000	9.8	14.8	10000	68239	8945	74K SB	GEOMANSB
9	15011	GEO SHACK LOGISTICS	0	GEO	22	15	12000	22	15	10000	63367	8945	74K SB	GEOLB
10														
11														
12	18072	MOBILE SAT B	0	GEO	19.7	13.1	14550	0	0	0	53175	8945	74K SB	GEOSB
13	18073	MOBILE SAT C	0	GEO	55	13.1	13230	0	0	0	51133	8945	74K SB	GEOLB
14	18074	SETI GEO ANTENNA A	0	GEO	19.8	14.9	21826	0	0	0	64555	8945	74K SB	GEOLB
15	18075	SETI GEO ANTENNA B	0	GEO	13.2	14.9	14551	0	0	0	53176	8945	74K SB	GEOLB
16	18076	SOLAR TERR GEO EXP	0	GEO	16.4	13.1	7055	0	0	0	41661	8945	74K SB	GEOLB
17	18308	VOA A, HF DIR BROADCAST	0	GEO	19.8	14.9	21826	0	0	0	64555	8945	74K SB	GEOLB
18	18309	VOA B, HF DIR BROADCAST	0	GEO	13.2	14.9	14551	0	0	0	53176	8945	74K SB	GEOLB
19	18542	TD OF LARGE GEO SAT	0	GEO	19.7	13.1	13670	0	0	0	51813	8945	74K SB	GEOLB
20	18750	COMSAT CLV RETRIEVAL	0	GEO	0	0	0	30	14.8	10030	44940	8945	74K SB	GEOLB
21	18751	COMSAT CLV DELIVERY	0	GEO	30	14.8	10030	0	0	0	46209	8945	74K SB	GEOLB
22														
23	18912	MULTIPLE PLD DELIVERY	0	GEO	30	15	12000	10	15	2000	52053	8945	74K SB	GEOLB
24														
25	19035	DOD GENERIC GEO	0	GEO	0	0	10000	0	0	0	41392	7617	52K	GEO
26	19036	DOD GENERIC MID INCLIN	63	19300	0	0	10000	0	0	0	38841	7617	52K	DODINCL
27	19517	DOD GENERIC POLAR	90	4000	0	0	5000	0	0	0	25327	7617	52K	DODPOLAR
28														
29														
30														

REV. 9 MISSION MODEL, SCENARIO 2

	P	Q	R	S	T	U	V	W	X	Y	Z	AA	AB	AC	AD	AE	AF
2						MISSIONS/YEAR											
3	1995	1996	1997	1998	1999	2000	2001	2002	2003	2004	2005	2006	2007	2008	2009	2010	TOTAL
4			1	1		1		1		1		1		1		1	8
5				1													1
6				1													1
7										1							1
8								1	1	2	2	2	2	2	2	2	16
9					1	1	2	1	2	4	4	4	4	4	4	6	37
10																	0
11																	0
12	1																1
13					1												1
14					1												1
15					1												1
16					1												1
17		1															1
18		1															1
19				1													1
20				1			1										2
21				1			1										2
22																	0
23	4	7	9	10	9	11	6	3	1	3	2	3	3	5	4	4	84
24																	0
25	6	6	6	6	6	6	6	6	6	6	6	6	6	6	6	6	96
26	8	8	8	8	8	8	8	8	8	8	8	8	8	8	8	8	128
27	1	1	1	1	1	1	1	1	1	1	1	1	1	1	1	1	16
28	TOTAL MISSIONS/YEAR																
29	20	24	25	31	29	27	53	21	19	26	48	24	24	27	25	28	400
30																	

	AG	AH	AI	AJ	AK	AL	AM	AN	AO	AP	AQ	AR	AS	AT	AU	AV	AW
2								PROPELLANT /YEAR									
3	1995	1996	1997	1998	1999	2000	2001	2002	2003	2004	2005	2006	2007	2008	2009	2010	TOTAL
4	0	0	61677	61677	0	61677	61677	61677	0	61677	61677	61677	0	61677	0	61677	616,771
5	0	0	0	53175	0	0	0	0	0	0	0	0	0	0	0	0	53,175
6	0	0	0	56545	0	0	0	0	0	0	0	0	0	0	0	0	56,545
7	0	0	0	0	0	0	0	0	0	69726	0	0	0	0	0	0	69,726
8	0	0	0	0	0	0	0	68239	68239	136478	136478	136478	136478	136478	136478	136478	1,091,824
9	0	0	0	0	63367	63367	126733	63367	126733	253467	253467	253467	253467	253467	253467	380200	2,344,568
10																	0
11																	0
12	53175	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	53,175
13	0	0	0	0	51133	0	0	0	0	0	0	0	0	0	0	0	51,133
14	0	0	0	0	64555	0	0	0	0	0	0	0	0	0	0	0	64,555
15	0	0	0	0	53176	0	0	0	0	0	0	0	0	0	0	0	53,176
16	0	0	0	0	41661	0	0	0	0	0	0	0	0	0	0	0	41,661
17	0	64555	0	0	0	0	0	0	0	0	0	0	0	0	0	0	64,555
18	0	53176	0	0	0	0	0	0	0	0	0	0	0	0	0	0	53,176
19	0	0	0	51813	0	0	0	0	0	0	0	0	0	0	0	0	51,813
20	0	0	0	44940	0	0	44940	0	0	0	0	0	0	0	0	0	89,879
21	0	0	0	46209	0	0	46209	0	0	0	0	0	0	0	0	0	92,419
22																	0
23	208213	364373	468479	520532	468479	572586	312319	156160	52053	156160	104106	156160	156160	260266	208213	208213	4,372,473
24																	0
25	248355	248355	248355	248355	248355	248355	248355	248355	248355	248355	248355	248355	248355	248355	248355	248355	3,973,670
26	310730	310730	310730	310730	310730	310730	310730	310730	310730	310730	310730	310730	310730	310730	310730	310730	4,971,683
27	25327	25327	25327	25327	25327	25327	25327	25327	25327	25327	25327	25327	25327	25327	25327	25327	405,225
28																	
29	TOTAL PROPELLANT / YEAR																
30	845799	1066516	1114568	1419302	1326783	1282041	1176290	933854	831437	1261919	1140140	1192193	1130516	1296300	1182569	1370980	18,571,207

Figure 6.2.3-5 Space-Based OTV Performance Data (GEO & DOD, Rev.9, Scenario #2)

7.0 OPERATIONS AND ACCOMMODATIONS

Operations and accommodations issues were reviewed to assess the impact of the Revision 9 mission model, design of the wide body OTV, and delivery to LEO by a LCV.

Proximity operations problems near the Space Station were analyzed and three possible work-around solutions investigated. It is recommended that a joint working group representing Space Station, OMV, and OTV review these proposals and designate the best solution. Operational time lines were reviewed and event times substantiated for GEO, Lunar, and Planetary type missions. A review of the Ford Aerospace and LMSC documentation on geostationary platforms proposed for the 1995 - 2000 time period show that the OTV system can meet all performance and support requirements for delivery of either type system to orbit. Flight Operations and Ground Operations were analyzed and requirements defined for ACC, Shuttle Payload Bay, and LCV delivery of an OTV system. Operational requirements in support of the various aerobrake configurations for both space-based and ground-based OTV were defined and methodology developed. Aerobrake TPS inspection techniques were evaluated and recommendations made for inspection aids. A number of trade studies were also performed, including: an operational comparison of the flexible brake, ballute, and shaped brake; comparison of methods to deorbit expended propellant tanks; and change out methodology for the 3 engine wide body OTV. Turnaround times needed for space-based and ground-based OTVs were determined, minimum fleet size and production rates required were established for the OTV system and for the major replaceable components.

Space Station accommodations were reviewed and changes are recommended from the initial study phase. Changes include a smaller hangar, a smaller propellant storage facility, and a re-estimate of robotic software and hardware requirements. Total reduction in requirements lowered the estimated cost of IOC accommodation to 45% to that proposed in the initial study phase. A trade study analysis of EVA/IVA requirements was conducted with the resultant recommendation that processing and servicing be performed by IVA supervisory control using a robotic manipulator arm.

7.1 SPACE STATION ACCOMMODATIONS

Space Station accommodations specified in the initial study phase were reviewed and revised for compatibility to the requirements of the Revision 9 mission model and the wide body OTV designed for LCV delivery. As part of this assessment, changes were made to the hangar layout, propellant storage requirements, OTV servicing by EVA/IVA, and the robotics software requirements.

7.1.1 OTV Hangar

An end view of the Space Station OTV hangar is shown below. The internal cross sectional area has been reduced 1596 ft^2 from the hangar proposed in the initial study. This was made possible primarily by the reduced diameter of the aerobrake. The OTV stack is rotated on the cradle allowing accessibility to all components from the overhead manipulator. Recommended hangar skin is the Goodyear inflatable material proposed in the initial study and described in NASA CR-66948 Summary Report.

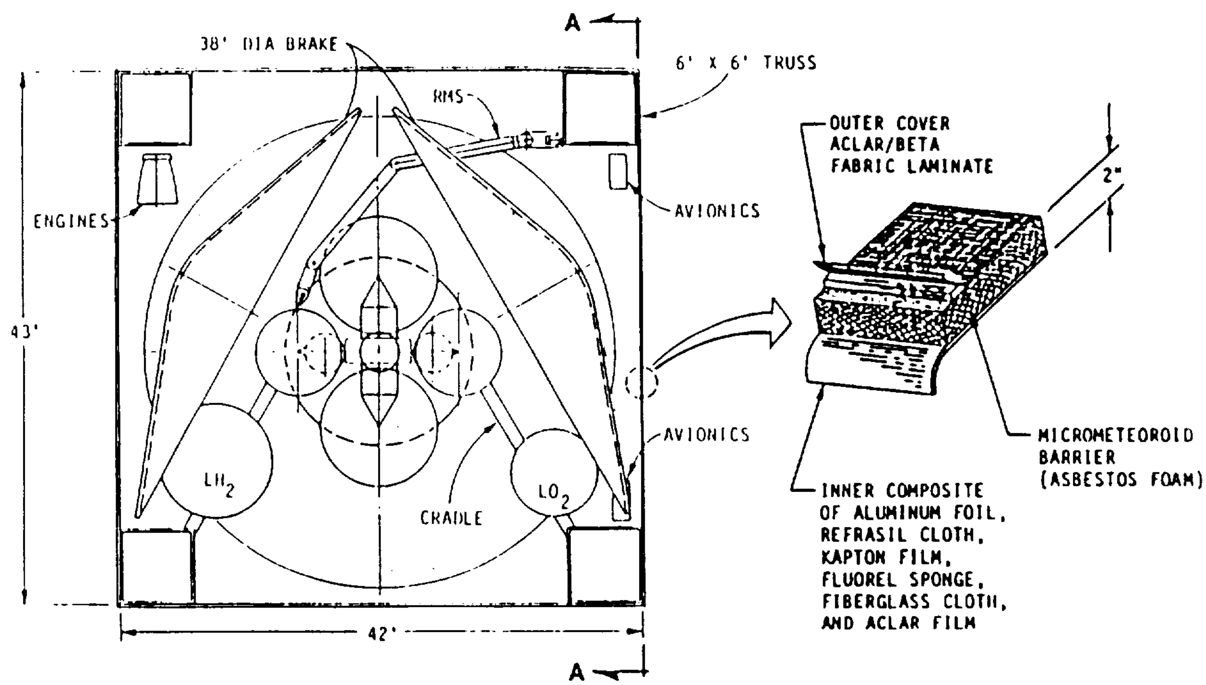


Figure 7.1.1-1 OTV Hangar Cross Sectional View

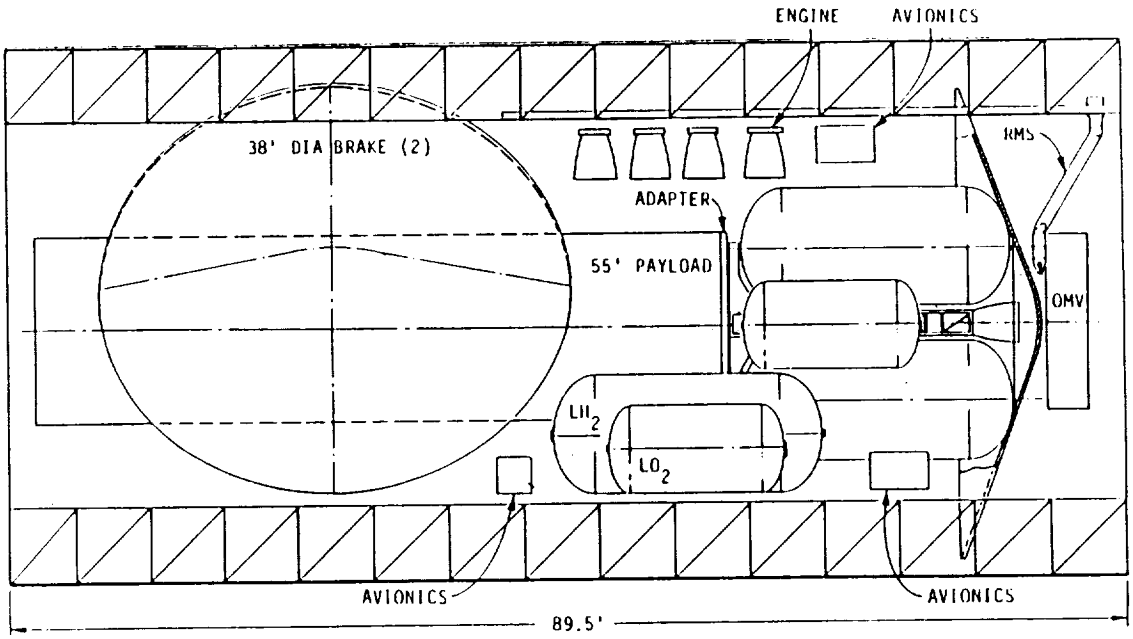
Hangar length requirements, as driven by the overall length of the OTV stage(s) and payloads, as a function of year and mision model scenario are summarized in Table 7.1.1-1.

The longest payload in scenario #2 is the Mobile Sat C (55 ft) scheduled for launch in 1999. The longest scenario #2 stack is the Pluto Orbiter which requires 104.5 feet for assembly. This payload plus that of the Unmanned Lunar Surface Mission can be accommodated in a hangar of 114 feet in length. The longest payload/stack in the mission model is the scenario #5 Surface Sortie/Camp which has an overall length of 139.5 feet.

Table 7.1.1-1 OTV Hangar Length Driver Missions

	A	B	C	D	E	F	G	H	I	J
1	OTV			LENGTH REQUIRED IN FEET						
2	MISSION	DESIGNATION	PAYLOAD	1ST STAGE	2ND STAGE	TANK SET	OMV	TOTAL	SCENARIO	1ST FLIGHT
3	18073	MOBILE SAT C	55.0	25.5			3.5	84.0	2	1999
4	17203	UNMAN LUN SUFR	32.0	25.5	25.5	17.5	3.5	104.0	2	2000
5	17207	SURF SORT/CAMP	67.5	25.5	25.5	17.5	3.5	139.5	5	2006
6	17095	PLUTO ORBITER	50.0	25.5	25.5		3.5	104.5	2	2007
7	17026	LUNAR ORB STA	12.0	25.5	25.5	23.5	3.5	90.0	5	2008

The 90 foot hangar shown in Figure 7.1.1-2 is of sufficient length to accommodate scenario #2 payloads up to the year 2000. At that time the hangar will be extended to 114 feet.If the scenario 5 lunar missions become a reality, the hangar could be extended to 150 foot length in the year 2006.



SECTION A-A

Figure 7.1.1-2 OTV Hangar, Initial Requirement

7.1.2 Propellant Storage

It is proposed that the propellant storage tank farm be reduced in size from that previously identified in the initial study effort. At that time a 200,000 lbm tank farm was recommended. It is now felt that, at least initially, a smaller tank farm will suffice. Prior to the year 2000, there are only 2 missions in scenario #2 that require 2 stages, one takes place in 1998 and the other in 1999. These secondary stages can be delivered to the Space Station fully fueled. Table 7.1.2-1 shows that considering the maximum propellant load for the SBOTV and the anticipated amount of propellant arriving as hitchhiked fuel during any month, a total storage capacity of 100,000 lbm will be sufficient for the early years of space-based OTV operation. The tank farm will be scarred for expansion as future requirements dictate.

Table 7.1.2-1 Propellant Storage Requirements, IOC

	PROPELLANT STORAGE REQUIREMENTS		
	LO2 (LBM)	LH2 (LBM)	TOTAL (LBM)
MAXIMUM PROPELLANT LOAD PER OTV (SINGLE STAGE)	63,430	10,570	74,000
AVERAGE HITCH HIKED LOAD ARRIVING SS	16,372	2,728	19,100
REPLACEMENT FOR 30 DAY BOIL-OFF @ 0.3LBM / HR.		216	216
CONTINGENCY (7%)	5,729	955	6,684
TOTAL STORAGE REQUIREMENTS	85,531	14,469	100,000

The two-tank system consists of a large LH_2 tank capable of handling 3500 ft^3 and a smaller LO_2 tank capable of holding 1250 ft^3. As the need requires, additional tanks can be added to the propellant storage facility.

7.1.3 Degree of Automation

When considering OTV processing operations at Space Station by EVA or IVA, it is not just a decision between robotics and manual EVA. Automation is a continuum stretching from hands-on operations through to autonomous robotics. Level of complexity and development costs soar as operations are made completely automated. A degree of manual intervention tends to keep cost down by allowing human decision making to determine what to do next, and then have the robot do a limited set of tasks. This is referred to as supervisory control. The trends are indicated in Figure 7.1.3-1.

For OTV processing support from the Space Station, the availability of personnel for OTV related activities must also be considered. By utilizing an IVA astronaut, supervisory control, and a RMS robotic arm demands made on the astronaut and the time necessary for turnaround of an OTV mission are minimized.

An in-depth trade study was conducted to assess the level of automation that should be incorporated in space based OTV support operations. This assessment included evaluation of the parameters listed in Table 7.1.3-1. Consideration was given to performing specific operations with EVA, remote operations with an IVA crew member providing control, and fully automated robotic operation. It was found that remote operations were preferable to fully automated operations in most cases, although the precise level of automation depends on the specific task. The numerical ranking shown in the chart below is generically indicative of the preferred approach, with the highest number being the most desirable.

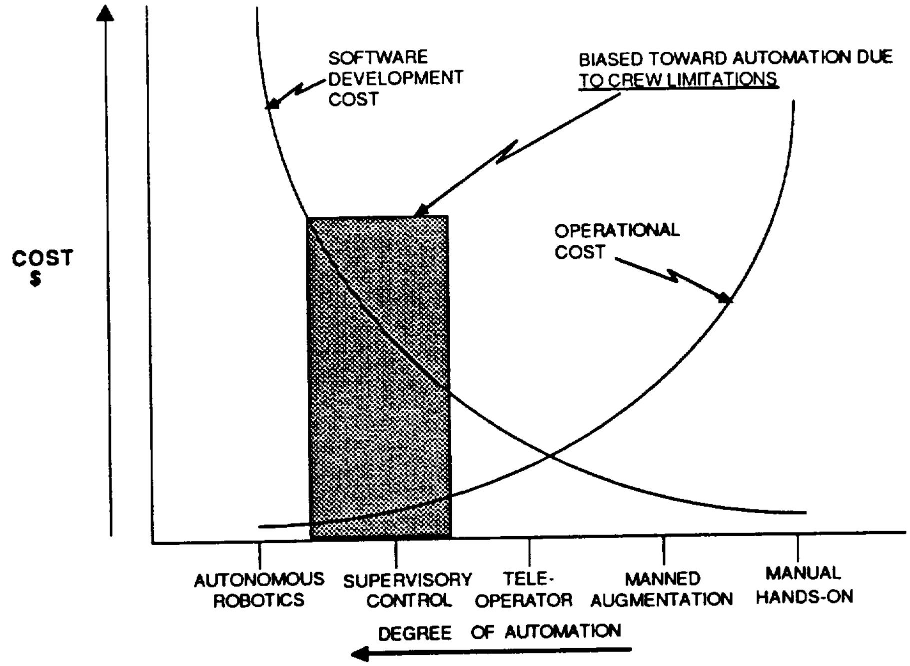


Figure 7.1.3-1 Level of Automation Versus Costs

Table 7.1.3-1 EVA/IVA Trade Study Results

PARAMETER	EVA	RMS (TELEOP)	AUTO ROBOTICS
OPERATIONAL CREW REQUIREMENTS	1	5	10
MAINTENANCE CREW REQUIREMENTS	10	5	1
DEVELOPMENT COST	10	8	1
OTV DESIGN DRIVERS	10	9	8
TPS INSPECTION AND REPAIR	5	4	2
PROPELLANT LOADING	1	8	10
OPERATIONAL COST	1	7	10
PAYLOAD MATING	1	10	6
PRE-LAUNCH TESTING	1	10	9
SCHEDULED/UNSCHEDULED MAINTENANCE	1	9	10
TOTALS	41	75	67

7.1.4 SBOTV IVA/Robotics Software Requirements

The code required to develop the robotics for the full up system has been conservative estimated at 400,000 lines. This is based, to a large extent, on a test bed intelligent robot being develop by MMC under Air Force contract F33615-82-C-5139. Additional estimates were developed for the generic control of a manipulator system, specific operations involved in the OTV processing and maintenance activity, OTV system checkout, and propellant farm management. A breakout of the various subsystem code requirements is shown in Table 7.1.4-1.

Table 7.1.4-1 Robotic Software Line of Code Requirements

	LINES OF CODE
• MANIPULATOR CONTROL	20K
• TRANSPORT, REMOVE & REPLACE OPERATIONS	50K
• DIAGNOSTICS & CHECKOUT	35K
• PROPELLANT FARM MANAGEMENT & PROPELLANT TRANSFER	20K
• AI, PATH PLANNING, POSITION SCANNER, GEOMETRIC REASONER, EXCEPTION HANDLING, PROCESSING ARCHITECTURE.	75K
• CONTINGENCY FACTOR (100%)	200K
TOTAL	400K

7.1.5 Space Station Accommodations Cost Revision

Based on data presented in this section, a revised cost estimate was generated for use in the cost trades being performed as part of the study effort. As can be seen, the revised cost figures are significantly lower than those used during the initial study phase. It had been initially assumed that the OTV program would have to bear the entire development cost of robotic hardware. It is now felt that this cost should drop drastically due to two separate factors: first, that Space Station and OMV have an equal need for the development of this hardware and should share the cost. Second, with the many advances currently occurring in this field, cost will be dropping. Imaging system requirements for OTV can be adapted from that developed for OMV to meet the needs for onorbit satellite servicing. Software requirements, hangar size and tank farm needs have been previously discussed. Transportation costs represent the difference between the Shuttle and the LCV. A comparison of the IOC accommodation costs is shown in Table 7.1.5-1.

Table 7.1.5-1 IOC Accommodations Costs for OTV

ITEM	PHASE A COST $M	REVISED COST $M	COMMENTS
ROBOTIC HARDWARE	165	96	SHARED COST ITEM (OTV, OMV, & SS)
STEREO-VISON IMAGE SYSTEM	100	30	ADAPTATION OF OMV SYSTEM
SOFTWARE	285	57	RE-ASSESSMENT OF REQUIREMENTS REDUCES LOC FROM 2M TO 400K
HANGAR	76	65	43X42X90 FT 1 OTV + 55 FT PL SIZED FOR GEO MISSIONS
TANK FARM	170	120	100 LBS PROP CAPACITY
TRANSPORTATION	140	50	UPRCV LAUNCH COST
TOTAL	936	418	

7.2 FLIGHT OPERATIONS

Flight Operations analysis conducted during the initial study was extended to encompass the new mission requirements and reflect the delivery of the wide body OTV by a UPRCV. Proximity operations near the Space Station were analyzed and flight operations requirements established for various mission and basing concepts. Operational impacts of aerobrake handling and servicing were evaluated and a trade study conducted to determine the preferred method of deorbiting expended propellant tanks, assuming that the return-to-earth vehicle for a ground-based OTV was limited to 15 foot diameter.

7.2.1 Proximity Operations, OTV - Payload Retrieval

Further study is necessary to determine the best approach to proximity operations involving a returning OTV with payload attached. Because these proximity operations affect the OMV and Space Station, as well as the OTV, a solution must involve representatives of all these programs.

Initial departure from the station is straightforward. The main area of concern is the last 1000 feet of retrieval through handoff to the Space Station remote manipulator. Three options for these proximity operations are shown in Figure 7.2.1-1.

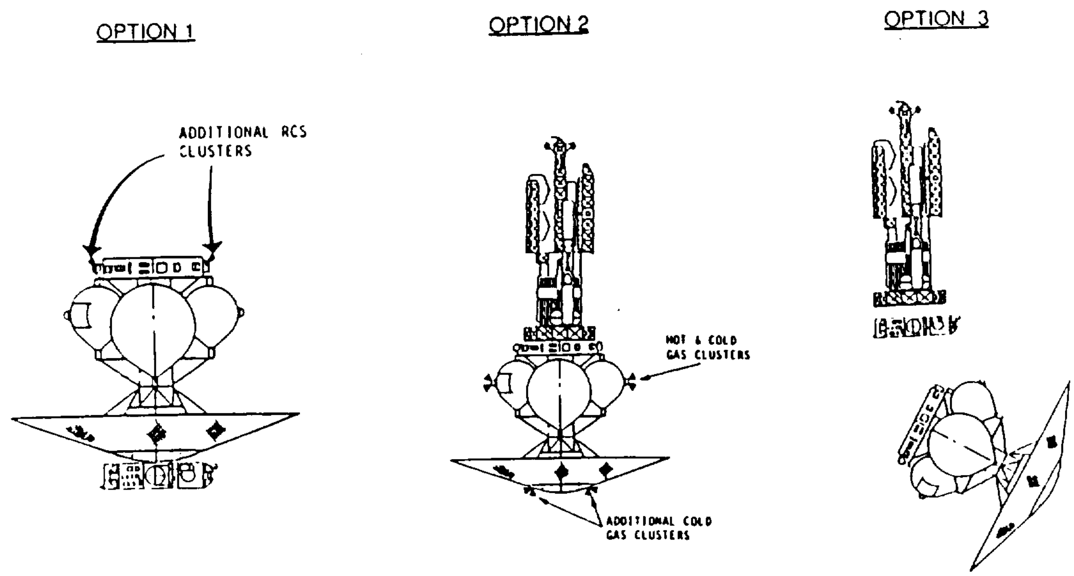


Figure 7.2.1-1 OTV - Payload Retrieval Options

OTV Payload Retrieval - Option 1

The OTV RCS system is controlled by commands from the OMV logic and command system. Two complete sets (both hot and cold gas) of RCS clusters would be installed on the avionic ring payload adapter to provide maneuvering capability lost to the OMV due to aerobrake interference and to overcome the C.G. offset resulting from the attached payload. It would be necessary to develop an OMV/OTV RCS interconnect logic system that would be provided as part of the OTV avionics subsystem.

The OTV/OMV docking adapter would need to incorporate an automatic RCS interconnect in order that total RCS control could be under OMV authority once docking had taken place.

OTV Payload Retrieval - Option 2

The OTV RCS system is controlled by commands from the Space Station Control Station. Expand the planned OTV RCS system to include both a hot and cold gas system. This involves the placement of additional cold gas RCS clusters next to the hot gas clusters currently positioned within the rigid brake area. Additionally, two clusters of each of the hot and cold gas jets would be installed on the avionic ring payload adapter interface. It would also be necessary to add a command data link so that the OTV could be controlled from the Space Station control station during proximity operations.

OTV Payload Retrieval - Option 3

With this option no changes would need to be made to the OTV RCS system. Returning from a mission with payload attached, the OTV will approach to within 8 nm of the Space Station on the -V bar. Just prior to the OMV's final approach to the area, OTV will separate from the payload to allow the OMV to mate with the payload for return to Space Station. After delivering the payload, the OMV would return, dock with OTV at the payload adapter interface, and return the OTV to the Space Station.

There is some concern that the payload, after separation from the OTV, could become unstable and cause difficulty for an OMV dock. Also, even with the OMV docked to the opposite end of the now payload-free OTV, some degree of plume impingement effect may still exist.

7.2.2 Flight Operations Requirements

7.2.2.1 LCV Delivery of Wide Body OTV, Ground-Based, Unmanned

Premission operations: The OTV and payload will be delivered to LEO fully assembled and intact. The OTV/Payload will be released from the LCV and allowed to coast for up to 12 hours for prepositioning prior to launch. Ground control will conduct checkout of both the OTV and payload prior to initiating an engine burn.

Launch-from-LEO operations will be conducted, the mission performed, and the returning OTV will execute the aeropass maneuver.

Postmission operations: at the end of the aeropass maneuver, the OTV will jettison the flexible portion of the aerobrake. The OTV is then injected into a low circular orbit in the range of 100 - 150 nmi. As the OTV reaches its desired orbit, the accumulators are fully charged and the LH_2 tanks are jettisoned. In the case of the larger OTV (74K), one of the LO_2 tanks will also be jettisoned. The OTV will then perform an ignition

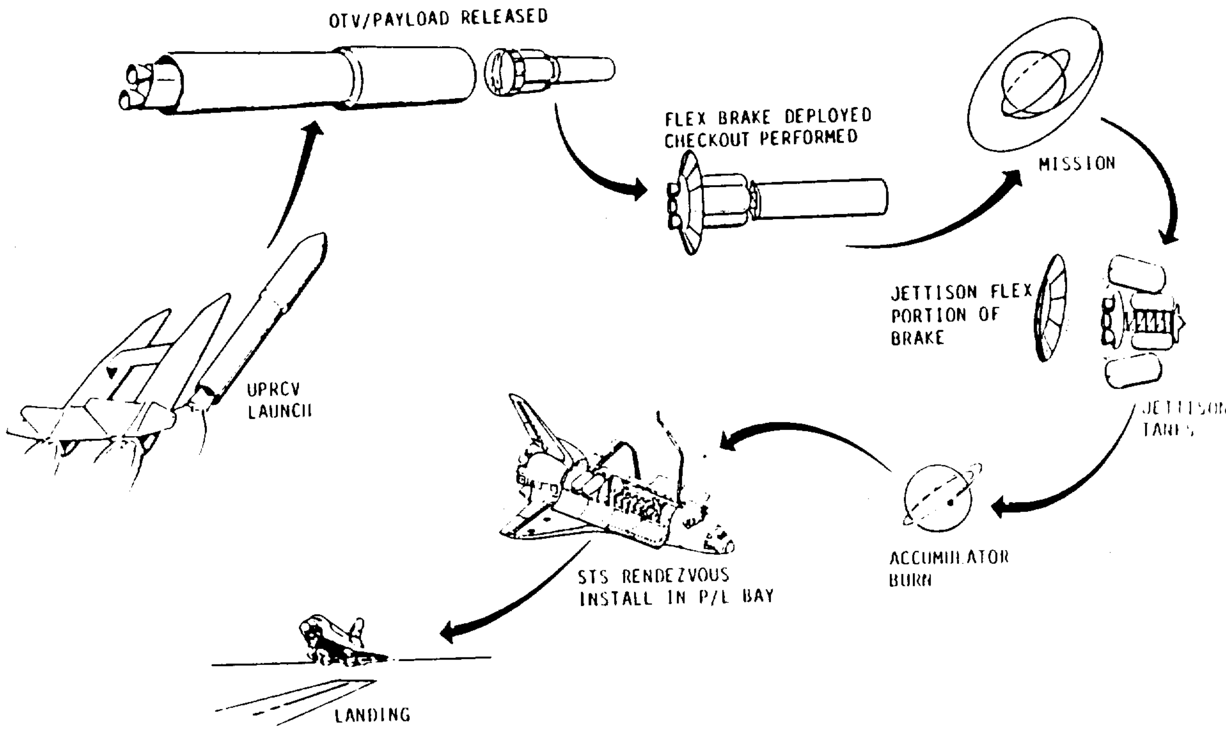


Figure 7.2.2-1 UPRCV Delivery, Unmanned GBOTV

burn utilizing the accumulator gases to gain a higher orbit. Once there, all systems will be shut down and the inert OTV will await STS rendezvous. The STS will rendezvous with the OTV, grapple it, and secure it to the Payload Installation and Deployment Aid (PIDA). Using the RMS the LO_2 tank(s) will be removed and installed in the payload bay. The remaining core structure with engines retracted and rigid brake attached will then be loaded into the bay.

7.2.2.2 LCV Delivery of Wide Body OTV, Ground-Based, Manned Mission

Premission operations: the OTV and the empty crew capsule (CC) will be delivered to LEO fully assembled and intact. The OTV/crew capsule will be released from the LCV. STS with the OTV crew on board is launched and rendezvous with the OTV. STS then docks with the capsule and the OTV crew transfers to the Manned Capsule and checkout is performed. STS undocks and allows the OTV/CC to coast for up to 12 hours for prepositioning prior to launch. Launch from LEO can be conducted by ground control or by the CC crew.

Launch-from-LEO operations will be conducted, the mission performed, and the returning OTV will execute the aeropass maneuver.

Postmission operations: at the end of the aeropass maneuver, the OTV will jettison the flexible portion of the aerobrake. At this point the

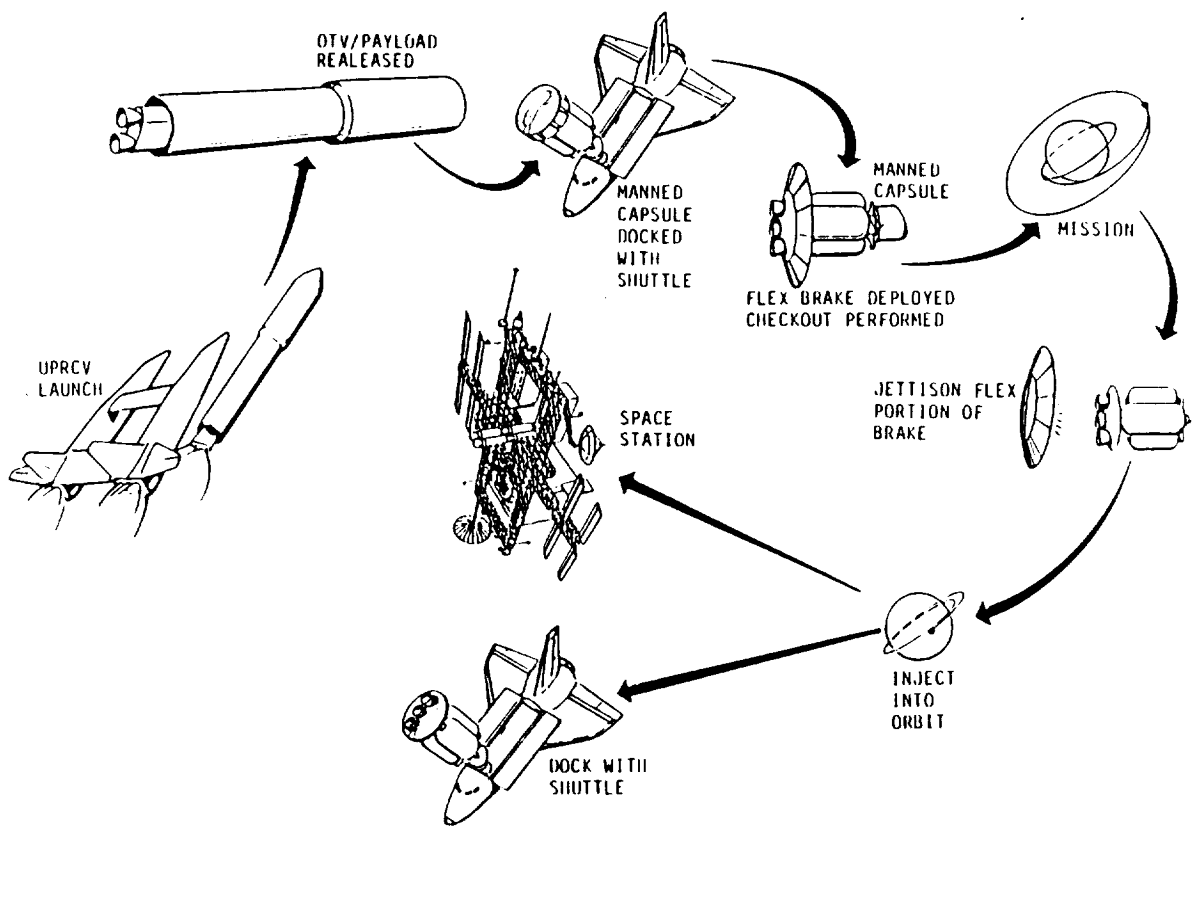


Figure 7.2.2-2 UPRCV Delivery, Manned GBOTV

crew can place the OTV in position for rendezvous with STS or possibly could return to Space Station to await pickup. If the return was to rendezvous with STS, OTV will be injected into a low circular orbit in the range of 150 nmi, and await the shuttle. When STS arrives it will dock with the CC and the crew will transfer to the orbiter. The STS will grapple the OTV/CC and secure it to the PIDA. Using the RMS, the LH_2 and LO_2 tank(s) will be removed and released to deorbit. The crew capsule and remaining core structure with engines and rigid brake attached will then be loaded into the bay. If the return was to Space Station, the aerobrake will not be jettisoned, the OTV will be injected into orbit behind Space Station at the designated pickup point to await rendezvous with the OMV to be ferried to Space Station.

7.2.2.3 LCV With Return Capability Delivery of Wide Body OTV, Ground Based System

Premission operations: The OTV and payload will be delivered to LEO fully assembled and intact. The OTV/Payload will be released from the LCV and allowed to coast for up to 12 hours for prepositioning prior to launch. Ground control will conduct checkout of both the OTV and payload prior to initiating an engine burn.

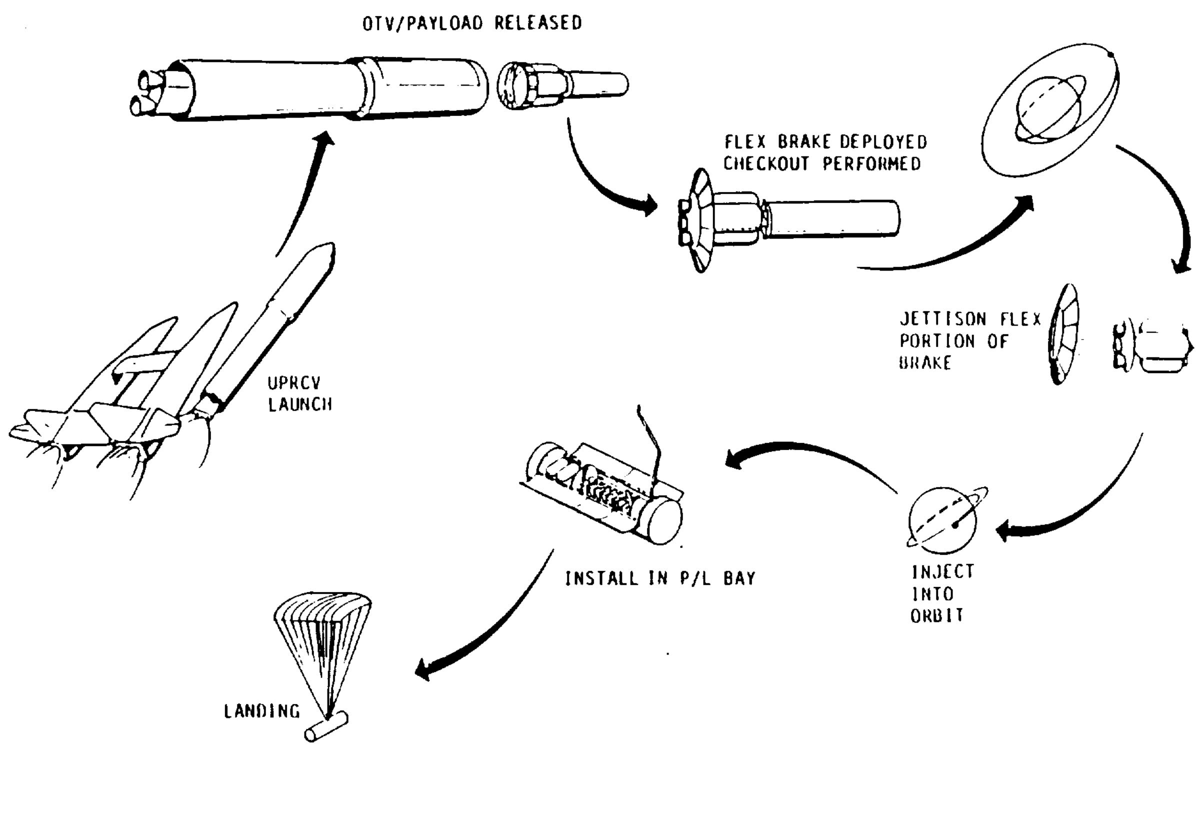


Figure 7.2.2-3 UPRCV with Return Capability, GBOTV

Launch-from-LEO operations will be conducted, the mission performed, and the returning OTV will execute the aeropass maneuver.

Postmission operations: at the end of the aeropass maneuver, the OTV will jettison the flexible portion of the aerobrake. The OTV is then injected into an appropriate orbit to rendezvous with the LCV. Using it's RCS, the OTV will approach within grapple distance of the LCV and shut down. The LCV will then use its RMS to grapple the OTV and load it into the cargo bay. This scenario would justify the OTV control option described as Option 2 in paragraph 7.2.1.

7.2.2.4 LCV Delivery of Wide Body OTV, Space-Based System

For the space-based Wide Body OTV, each new OTV delivery will be handled as a GBOTV launch. Subsequent delivery of payloads and OTV spare components by LCV will be to ZONE 4 behind the Space Station. OMV will rendezvous with the LCV and ferry the payload and/or component spares to Space Station. At Space Station, for each subsequent mission beyond the initial delivery of each OTV, payload mating, propellant loading, checkout, and deployment from the station will be performed.

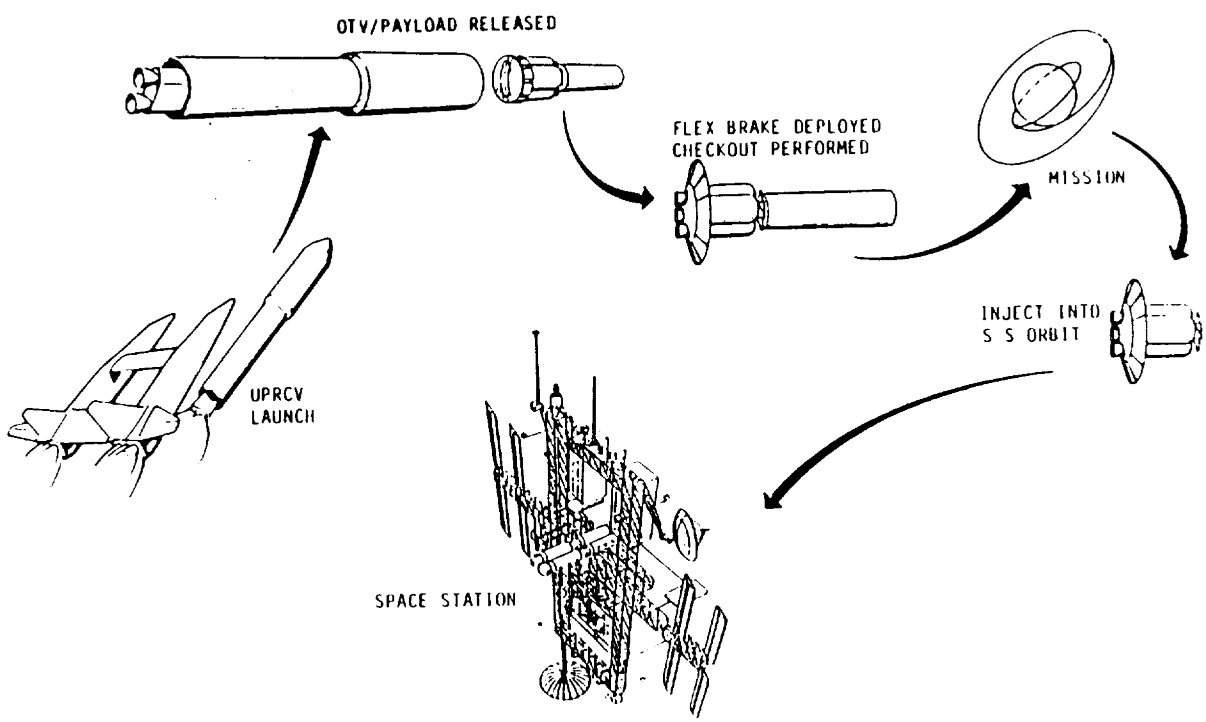


Figure 7.2.2-4 UPRCV Delivery, SBOTV

Ground control will conduct Launch-from LEO operations, the mission performed, and the returning OTV will execute the aeropass maneuver.

OTV will be injected into orbit behind Space Station at the designated pickup point to await rendezvous with the OMV to be ferried to Space Station. Once at Space Station, propellant detanking will be performed and inspection of the returned OTV will take place. Diagnostic testing will be performed and any necessary maintenance action taken. The OTV will then be placed in storage to await the next mission.

7.2.3 Aerobrake Operations Comparison

A comparison of operations required to support the various aerobrake configurations is shown in Table 7.2.3-1. The least human involvement occurs with the ground-based ACC version because the entire aerobrake is jettisoned at the completion of the mission and no further handling or refurbishment is required. The most demanding of the aerobrake configurations are the two space-based versions; since inspection, replacement, and possibly limited repair are performed at the Space Station. The ground-based STS payload bay version and the two wide body LCV versions are essentially the same from the an operations standpoint.

Table 7.2.3-1 Aerobrake Operations Comparison

	GROUND BASED OTV				SPACE BASED	
	GROUND BASED PAYLOAD BAY VERSION	GROUND BASED ACC VERSION	52 K WIDE BODY	74 K WIDE BODY	2-ENGINE OTV	74 K WIDE BODY
BRAKE SIZE	38 FT	38 FT	32 FT	38 FT	44 FT	38 FT
OTV LAUNCH LOCATION	STS PAYLOAD BAY	STS AFT CARGO CARRIER	UPRCV CARGO BAY	UPRCV CARGO BAY	STS AFT CARGO CARRIER	UPRCV CARGO BAY
BRAKE CONFIGURATION @ LAUNCH	ATTACHED TO OTV, OUTER 12' FOLDED AROUND TANKS	ATTACHED TO OTV, OUTER 12' FOLDED AROUND TANKS	ATTACHED TO OTV, OUTER 3.75 FT. FOLDED AT TANK	ATTACHED TO OTV, OUTER 6.75 FT. FOLDED AT TANK	FOLDED UNATTACHED	ATTACHED TO OTV, OUTER 6.75 FT. FOLDED AT TANK
DEPLOYMENT @ LEO	AUTOMATIC	AUTOMATIC	AUTOMATIC	AUTOMATIC	USING RMS AND ASE	USING RMS AND ASE
PREMISSION ON-ORBIT OPERATIONS	NONE	NONE	NONE	NONE	INSTALL TO OTV. R&R EVERY 5 FLTS.	REMOVE & REPLACE EVERY 5 FLIGHTS
POST MISSION @ LEO	JETTISON FLEX PORTION OF BRAKE	JETTISON BRAKE	JETTISON FLEX PORTION OF BRAKE	JETTISON FLEX PORTION OF BRAKE	NONE	NONE
POSTMISSION ON-ORBIT OPERATIONS	INSTALL OTV WITH RIGID BRAKE ATTACHED IN STS BAY	NONE	INSTALL OTV WITH RIGID BRAKE ATTACHED IN STS BAY	INSTALL OTV WITH RIGID BRAKE ATTACHED IN STS BAY	INSPECT FOR DAMAGE	INSPECT FOR DAMAGE
REFURBISHMENT REQUIREMENTS	REFURB RETURNED RIGID PORTION	NONE	REFURB RETURNED RIGID PORTION	REFURB RETURNED RIGID PORTION	REFURBISHMENT MAY BE PRACTICAL	REFURBISHMENT MAY BE PRACTICAL
GROUND OPERATIONS	FIT NEW FLEX MATERIAL TO RIGID CENTER. INSTALL BRAKE ON OTV, FOLD, SECURE.	INSTALL NEW BRAKE ON OTV, FOLD, SECURE.	FIT NEW FLEX MATERIAL TO RIGID CENTER. INSTALL BRAKE ON OTV, FOLD, SECURE.	FIT NEW FLEX MATERIAL TO RIGID CENTER. INSTALL BRAKE ON OTV, FOLD, SECURE.	INSTALL NEW BRAKE IN PAYLOAD BAY	INSTALL NEW BRAKE IN PAYLOAD BAY

When a new or replacement aerobrake is brought to the Space Station, it can be removed from the payload bay and placed in storage or it could be readied for immediate use.

For use, it is necessary to affix the ASE deployment mechanism to the aerobrake structure and actuate the telescoping members in order to deploy the flexible portion of the brake. Utilizing the remote manipulator arm, the old brake is released and removed. The new brake is then positioned and fixed into place.

Post mission inspection of a returned aerobrake will most likely be performed with the aid of a CCTV camera mounted to a manipulator arm. A thorough inspection has to be made of the surface area with major concentration given to all interface areas. These include the rigid to flexible interface, the openings within the rigid surface for doors and RCS jets, and the interfaces within the flexible portion of the brake where the gore panels were sewn together as well as the flutings within the panels themselves. Possible inspection aids are listed in Table 7.2.3-2.

Table 7.2.3-2 Onorbit Aerobrake Inspection

VISUAL (CCTV) INSPECTION	POSSIBLE INSPECTION AIDS
RIGID BRAKE INSPECTION	ACOUSTIC
BROKEN TILES LOOSE TILES OUTER COATING DAMAGE INTERFACE AREAS AT DOORS & RCS CLUSTERS	ACOUSTO-ULTRASONIC DEVICE (NASA LEWIS)
	OPTICAL
	LASER INTERFEROMETER
FLEXIBLE BRAKE INSPECTION	RADIOGRAPHIC
WEAR BURNS FRAYED AREAS DETERIORATION DISCONTINUITY GORE PANEL BREAKAGE	ISOTOPE WEAR DETECTOR (ROCKETDYNE)
	ELECTRICAL
INTERFACE INSPECTION	EXO-ELECTRON EMISSION DETECTOR (ROCKETDYNE)
BREAKS MISALIGNMENTS	

7.2.4 Propellant Tank Deorbit Trade Study

With LCV delivery of the wide body OTV, expending the propellant tanks so that the core vehicle can be returned on the STS presents somewhat of a challenge. Due to restrictions within the Orbiter bay as to where equipment can be secured for the return trip, it becomes necessary to expend propellant tanks. With the 52K OTV; the core vehicle, structure, rigid portion of the aerobrake, avionics, and the two LO_2 tanks can be returned in the STS payload bay. The two LH_2 tanks must be expended. With the 75K OTV, both LH_2 tanks and one LO_2 tanks are unable to be returned and must be expended.

The area of concern is keeping the core OTV in an orbit stable enough to await the next return STS flight, and, at the same time, ensuring that the jettisoned tanks do not contribute to the space debris problem.

7.2.4.1 Evaluation

An analysis was conducted to determine the most cost effective method of disposing of those tanks that could not be returned in the payload bay. The four methods shown below were considered as possible candidates in the trade study that is documented on the succeeding pages.

The "OMV DEORBIT" requires the returning OTV to inject into a circular orbit 25 nm beneath the Space Station. With Space Station at 250 nm, orbital phasing would place the OTV within prime position for rendezvous approximately every 6 1/2 days. Both STS and OMV would need to rendezvous with the OTV to perform the retrieval operation. OMV could either deorbit the expendable tanks or return them to Space Station for storage.

The "STS DEORBIT" requires essentially the same operations as does the "OMV DEORBIT" method. Additionally, however, it would also require the STS to maneuver to a lower altitude to release the expendable tanks. This would require development of a special holding fixture to which the tanks can be secured and then release upon command.

The "OTV AUXILIARY PROPELLANT" requires the addition of a secondary set of tanks to be used after the main tanks have been jettisoned. This would require an additional development effort and would also add weight to the OTV.

The "NORMAL DECAY" presents the least impact to the system since the only additional mechanism required is that for jettisoning the propellant tanks upon command. For both the 52K and the 74K vehicles, the ballistic coefficient ratios between the core vehicle and tanks are approximately 7 to 1 for the LO_2 tank and 9 to 1 for the LH_2 tank. This, in addition to the accumulator burn that provides an altitude increase in excess of 25 nmi, combine to provide an OTV to LO_2 tank orbit lifetime ratio of 30 to 1. This means that for an orbital life of one day for the LO_2 tanks, the OTV core will stay in orbit for 30 days. With regard to the LH_2 tank, the ratio is almost 40 to 1.

Table 7.2.4-1 Tank De-Orbit Candidate Evaluation

PARAMETER	CANDIDATE METHOD FOR TANK DE-ORBIT			
	OMV DEORBIT	STS DEORBIT	AUX TANKS	NORMAL ORBIT DECAY
ORBIT STABILITY	STABLE	STABLE	STABLE	OTV - LO2 TANK HAVE BALLISTIC COEFFICIENT RATIO >7 TO 1. FINAL OTV ORBIT CAN BE DETERMINED BY NEED.*
DEVELOPMENT REQUIREMENTS	OMV / TANK INTERFACE (MINIMAL COST)	STS PAYLOAD BAY TANK HOLDING FIXTURE	AUXILIARY TANKS, PROP LINES & VALVES, PROP TANK JETTISON MECH.	PROPELLANT TANKS JETTISON MECHANISM
WEIGHT IMPACT ON OTV	MINIMAL	MINIMAL	~ 800 LBS	~ 140 LBS
RECURRING COST	OMV CHARGE ($500K)	EXTRA 1/2 DAY STS CHARGE ($325K)	ADDITIONAL PROP COST FOR OTV	ADDITIONAL PROP COST FOR OTV

* COMBINATION OF BALLISTIC COEFFICIENT DIFFERENCE AND ALTITUDE BOOST COULD RESULT IN A RELATIVE OTV - LO2 LIFETIME OF 30 TO 1, I.E., A REQUIREMENT FOR A 30 DAY OTV ORBIT WOULD RESULT IN A ONE DAY TIME PERIOD FOR LO2 TANK DEORBIT.

7.2.4.2 Cost Comparison

There is no development cost associated with the "OMV DEORBIT" candidate and the others all represent modest costs with the "AUXILIARY TANKS" being the most expensive. However, the vast preponderance of increased costs is that which reoccurs each flight over the life of the program. "NORMAL DECAY" is the obvious winner on cost, it being only 1/4 of the closest competitor, "STS DEORBIT".

Table 7.2.4-2 Candidate Cost Comparison

PARAMETER	CANDIDATE			
	OMV DEORBIT	STS DEORBIT	AUX TANKS	NORMAL DECAY
DEVELOPMENT COSTS	NONE	$6M	$16M	$2M
RECURRING COST ITEMS:				
● OMV CHARGE	$500K			
● STS CHARGE		$325K		
● OTV ADDED PROP COST			$480K	$84K
(422 MISSIONS)	$211M	$137M	$203M	$35M
TOTAL CONSTANT 85 $ COST	$211M	$143M	$219M	$37M

7.2.4.3 Solution

It is recommended that the "NORMAL DECAY" option be selected as the preferred method of deorbiting expendable tanks for the Wide Body GBOTV. With a 30 to 1 decay ratio it seems reasonable that an OTV return orbit can be selected that will provide the desired stability for an inert OTV while still insuring a rapid reentry of the jettisoned propellant tanks.

7.2.5 Geostationary Platform Support Requirements

A review of the Ford Aerospace (WDL TR10623/NAS8-36104) and LMSC (LMSC D060799/9NAS8-36103) documentation has shown the OTV system, as proposed, should be capable of meeting all performance and support requirements imposed for the delivery of candidate geostationary platforms to orbit. The NASA provided these reports for OTV contractor review so that a realistic assessment could be made of the requirements being imposed on an Orbital Transfer Vehicle by platforms under study.

7.2.5.1 Geostationary Platform 6L-R2, LMSC Study

LMSC evaluated 8 platform configurations from which two were selected for a further in-depth study of the type of GEO mission that would be required in the 1995 - 1998 time period. The 6L-R2 shown in Figure 7.2.5-1 represents the low-end mission that could be carried up in STS payload bay and launched from the Orbiter. The platform weighs 10,000 lbs, measures 40' x 14.8', and is designed for a 10 year life. This mission would be suitable for GBOTV, either with a storable in the payload bay or a cryo stage in the ACC. If a ground-based payload bay cryogenic OTV were used, this platform would be a candidate for dual payload manifest/delivery.

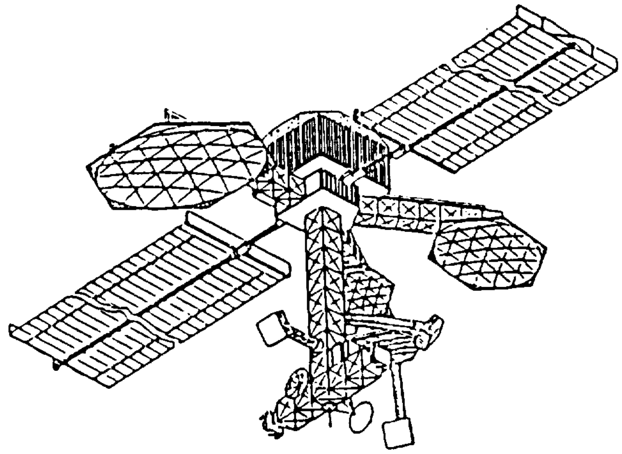


Figure 7.2.5-1 Geostationary Platform 6L-R2

7.2.5.2 Geostationary Platform 7L-R3, LMSC Study

The 7L-R3 in Figure 7.2.5-2 represents the high-end mission suitable to a SB/OTV launch from Space Station. The payload requires the full STS cargo bay for delivery to LEO. Once deployed from the STS, it is reconfigured twice onorbit. Once in LEO to a configuration designed to withstand OTV thrust, and once in GEO to its operations use profile. Designed for a 10 - 15 year life, servicing would be performed by an OTV/OMV mission to GEO. The platform weighs 21,000 lbs and measures 60' x 14.9'. Although no acceleration limits are set, it is assumed to be limited to 0.1G.

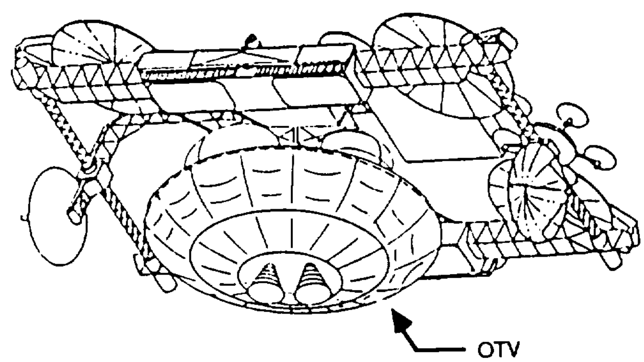


Figure 7.2.5-2 Geostationary Platform 7L-R3

7.2.5.3 Geostationary Platform F6A, Ford Aerospace Study

Ford Aerospace also studied a number of candidates and selected one for further analysis. The F6A shown in Figure 7.2.5-3 is a high-end platform that requires Space Station support for assembly and checkout prior to transfer by OTV to GEO stationary orbit. The 500 watt power demand can be met by the main tank fed fuel cell power approach proposed for OTV. The communications support is also within the OTV system design capability.

F6A/MODULAR PLATFORM

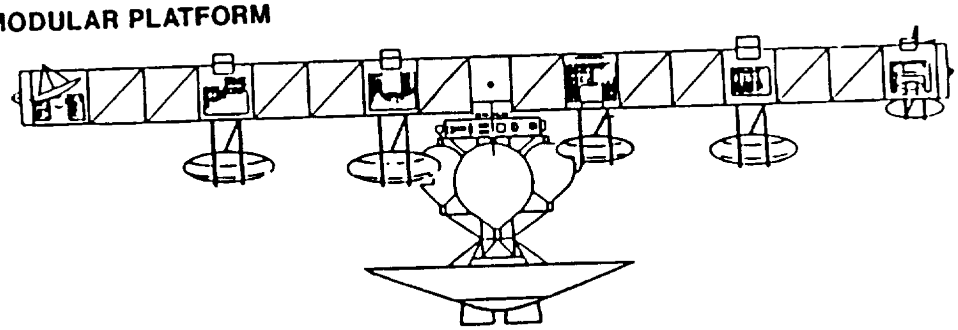


Figure 7.2.5-3 Geostationary Platform F6A

7.2.5.4 Low Thrust Transfer to GEO

A review of the Ford Aerospace and LMSC documentation has shown the OTV system, as proposed, is capable of meeting all performance and support

Table 7.2.5-1 Multiple Burn Transfer to GEO

BURN #	ORBIT	APPROXIMATE BURN TIME (MINUTES)	TIME TO NEXT BURN (HOURS)
1	292 X 1420	16	1.8
2	324 X 3317	16	2.4
3	348 X 7177	16	4.0
4	371 X 19353	16	4.9
5	19323 X 19323	40	
TOTAL			14.8 HOURS

requirements imposed for the delivery of candidate geostationary platforms to orbit.

Low thrust level requirements such as stipulated for the F6A will require a pump idle mode burn sequence resulting in 4 perigee burns of approximately 16 minutes each to obtain an orbit of 371 x 19353 nmi. One additional burn of approximately 40 minutes will be used to circularize the orbit at GEO. The entire flight duration will require just under 15 hours as indicated in Table 7.2.5-1.

7.3 GROUND OPERATIONS FLOW

The pictorial representation in Figure 7.3-1 is a top level sequence of operations from landing of the GBOTV return ferry flight aboard the Shuttle (I or II) through integration with and the launch on an unmanned partially reusable cargo vehicle (UPRCV or (LCV). The operations required for preparations for the next flight are divided into seven discrete tasks as summarized below.

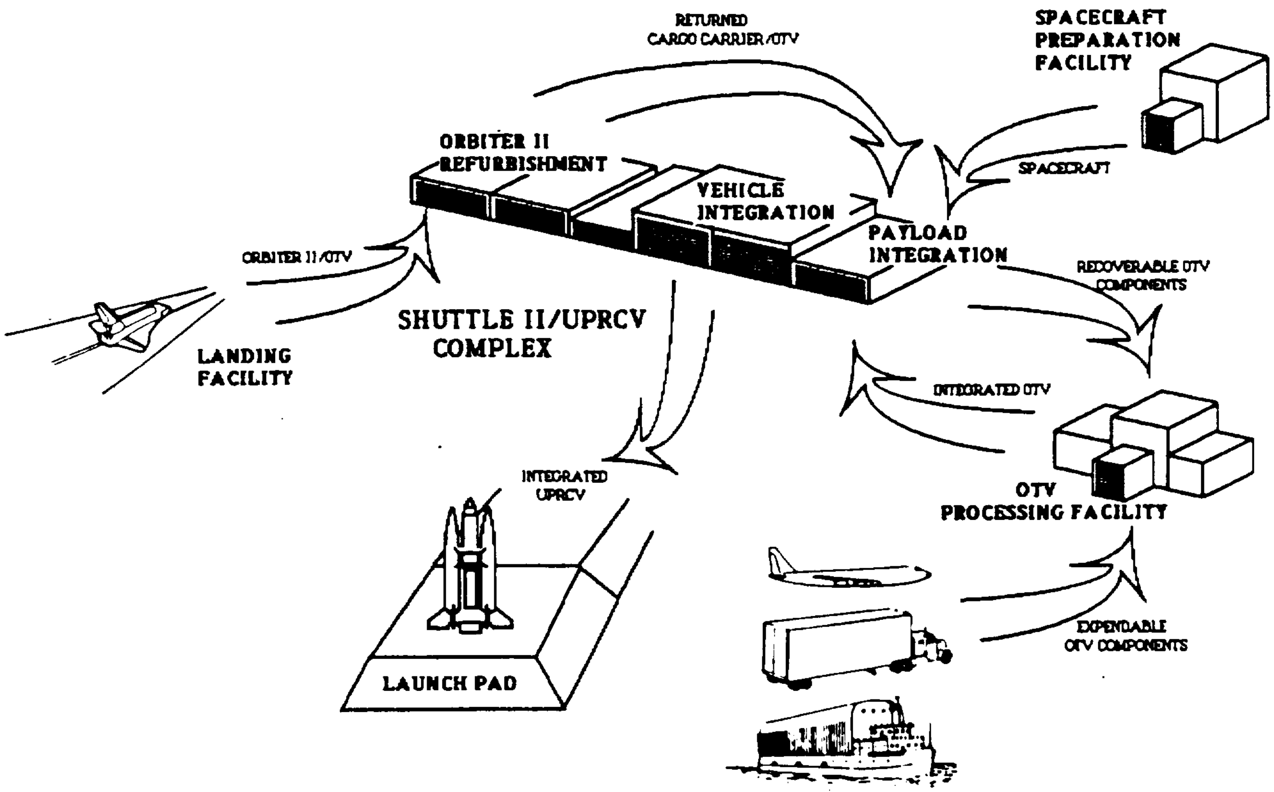


Figure 7.3-1 Ground Operations Flow

The Task 1 activities (Return to Launch Site and Recover OTV) begin with the Orbiter touchdown. The Shuttle I/II is towed to the Shuttle LCV complex and brought into the Orbiter processing facility. Here the OTV core and LO_2 tank(s) are removed and transferred to the OTV processing facility (OTVPF). While in the OTVPF, Task 2 (Postflight Maintenance and Refurbishment) and Task 3 (Assembly and Test) are completed resulting in a completely reintegrated OTV which is then transported to the payload integration cell. Here the OTV and spacecraft are mated (Task 4) and integrated with the LCV cargo carrier (Task 5). The integrated LCV cargo carrier is transferred to the launch vehicle integration cell and mated with the previously integrated vehicle booster, tank, and propulsion/avionics module (Task 6). The integrated LCV/OTV/spacecraft is then transported to the pad. Launch preps (Task 7) include parallel vehicle/OTV cryogenic loading.

7.3.1 Ground Facilities Summary

Table 7.3.1-1 summarizes the facility requirements for the GBOTV at the STS/LCV launch site. A dedicated OTVPF is required. All other capabilities necessary for the OTV operational turnaround are within the scope of general launch site requirements and will be provided by the STS/LCV facilities.

Table 7.3.1-1 Ground Facilities Requirements

- DEDICATED OTV PROCESSING FACILITY (OTVPF)
  - AIRLOCK
  - HIGH BAY
  - MPS/RCS TANK LAB/SHOP
  - AEROBRAKE CHECKOUT CELL

- ON LINE SHUTTLE/SHUTTLE II/UPRCV FACILITIES
  - SHUTTLE/SHUTTLE II LANDING FACILITIES
  - ORBITER PROCESSING/ORBITER II REFURBISHMENT FACILITIES
  - UPRCV PAYLOAD INTEGRATION FACILITY
  - UPRCV VEHICLE INTEGRATION FACILITY
  - UPRCV LAUNCH PAD

- OFF LINE SUPPORT FACILITIES/AREAS
  - BATTERY LAB.
  - ORDNANCE STORAGE/TEST
  - SPARES STORAGE
  - CALIBRATION LAB.
  - CLEANING LAB.
  - FMEA LAB.

7.3.2 OTV Processing Facility (OTVPF) Requirements

Top level requirements for a dedicated facility for stand-alone OTV turnaround operations are shown in Table 7.3.2-1. This facility will support the Task 2 (Maintenance and Refurbishment) and Task 3 (Assembly & Test) activities. The high bay area will support all OTV core activities and the integration of the recoverable and expendable components of the OTV. The MPS/RCS tank lab will support the maintenance and preflight preparation of the recoverable tank(s) and the receiving and checkout of the expendable tanks. The aerobrake checkout cell is required for the receiving and preparation of the expendable aerobrake components for installation on the OTV core.

Table 7.3.2-1 OTV Processing Facility

AIRLOCK

45' x 45' x 50' high
40' x 40' doors
10 ton overhead crane

HIGH BAY

80' x 80' x 85' high
20 ton overhead crane
70' hook height

MPS/RCS TANK LAB

50' x 75' x 50' high
10 ton overhead crane
35' hook height

AEROBRAKE CHECKOUT CELL

80' x 80' x 50' high
10 ton overhead crane
35' hook height

GENERAL REQUIREMENTS

STANDARD COMMERCIAL POWER
UNINTERRUPTED INSTRUMENTATION POWER
EMERGENCY POWER SYSTEM
CLEANLINESS: 100K
TEMPERATURE: 70 +/- 5 F
RELATIVE HUMIDITY: 30-50%
HIGH VOLUME AIR HANDLERS
CCTV/OIS/PAGING/TELEPHONE COMM
R.F. SYSTEM
EMERGENCY WARNING SYSTEM
SHOP AIR
FACILITY GN2/GHe
POTABLE WATER
GROUNDING
VACUUM SYSTEM
EMERGENCY EYE WASH/SHOWERS
LIGHTNING PROTECTION
FIRE DETECTION/DELUGE
HAZ.GAS DETECTION

7.3.3 OTV GSE Requirements

Table 7.3.3-1 provides a descriptive listing of the GSE requirements which have been identified by the definition of the processing activities. The listing does not include common items which are considered launch site GFE.

Table 7.3.3-1 GSE Requirements

TRANSPORTION, HANDLING AND ACCESS EQUIPMENT

OTV TRANSPORTER / COVER
OTV TRANSPORTER COVER LIFTING SLING / FIXTURE
OTV LIFTING SLING SET
OTV ASSEMBLY / TEST STAND
OTV ASSEMBLY / TEST STAND ACCESS EQUIPMENT
AEROBRAKE SHIPPING CONTAINER / COVER
AEROBRAKE SHIPPING CONTAINER COVER LIFTING SLING / FIXTURE
AEROBRAKE LIFTING / INSTALLATION EQUIPMENT
AEROBRAKE ASSEMBLY / TEST STAND
RECOVERABLE MPS TANK(S) TRANSPORTER / COVER
EXPENDABLE MPS TANKS SHIPPING CONTAINER
MPS TANK TRANSPORTER / CONTAINER LIFTING SLING / FIXTURE
MPS TANK LIFTING / INSTALLATION EQUIPMENT
ORDNANCE CARRYING CASE
ORDNANCE INSTALLATION EQUIPMENT
BATTERY CART
BATTERY INSTALLATION EQUIPMENT
STANDARD INSTALLATION EQUIPMENT
STANDARD TOOL KITS
MISC. EQUIPMENT SHIPPING CONTAINERS
EQUIPMENT DOLLIES / CARTS

TEST AND CHECKOUT EQUIPMENT

PROPULSION / AVIONICS SYSTEMS COMMAND, CONTROL, CHECKOUT, AND MONITORING CONSOLE SET
DATA RECORDING SYSTEM
BATTERY ACTIVATION AND TEST SET
STRAY VOLTAGE TEST SET
ORDNANCE CIRCUIT TEST SET
ALIGNMENT EQUIPMENT
STANDARD ELECTRONIC TEST EQUIPMENT

SYSTEM / INTERFACE SIMULATION EQUIPMENT

SPACECRAFT INTERFACE TEST EQUIPMENT
LAUNCH / LANDING VEHICLE INTERFACE TEST EQUIPMENT
SIMULATED MISSION SEQUENCE TEST EQUIPMENT

7.3.4 Criteria For Operational Objectives

Criteria for OTV design, technological advancements, and launch site test philosophy needs to be met to guarantee that the turnaround assessment of the ground based OTV will be achieved. Each criteria results in improved operations from current processing techniques. These improvements are realized in reduced times and manpower, and ultimately in significantly decreased operational contributions to life cycle costs. The criteria is presented in tabular form in Table 7.3.4-1.

Table 7.3.4-1 - Criteria For Operational Objectives

DESIGN FEATURES
- -AUTOMATED LEAK DETECTION
- -NO POST MISSION DRAIN/PURGE REQUIREMENTS
- -MINIMAL OTV/SPACECRAFT INTERFACES
- -MINIMAL OTV/LAUNCH & LANDING VEHICLE INTERFACES
- -HIGH ACCESSIBILITY AND QUICK FASTEN/RELEASE ORU's

TECHNOLOGIES
- -ELIMINATE ORDNANCE
- -NO PLANNED TPS TURNAROUND REFURB/EASE OF REPAIR & INSPECTION
- -FAULT DETECTION/FAULT ISOLATION TO ORU LEVEL
- -SELF ALIGNMENT AND AUTO MATE/DEMATE MECHANICAL INTERFACES
- -SELF MONITORING COMPONENTS THAT USE FLIGHT DATA TO DETERMINE HEALTH STATUS AND MAINTENANCE REQUIREMENTS

TEST PHILOSOPHY
- -MINIMAL ON-LINE OPERATIONS
- -TEST AT SYSTEM LEVEL ONLY
- -NO REPETITION OF TEST DUE TO FACILITY TRANSFERS

7.4 TURNAROUND TIME AND FLEET SIZING

An analysis was made of the turnaround times required at Space Station and at the ground operations center. Average turnaround times were established, minimum fleet size determined, and production requirements to support the various mission scenarios determined.

7.4.1 Space-Based OTV Processing Operations

Operational Flow Charts developed during the initial study phase were reviewed to determine average turnaround time required for a Space-Based OTV. The Phase A functional flow operations data considered an IVA astronaut and a programmed robotic arm performing all servicing, checkout, remove and replace tasks. Task times were developed by using a robotic simulation to establish times required for transport, inspection, assembly, and disassembly. These times were then used as building blocks to establish overall operations times.

During the follow-on effort, expected frequency of operations were determined based on anticipated component life and normal servicing functions. An average time per mission was then computed for each activity and an accumulated serial time for an average turnaround developed at 3 1/4 days. This information is presented in Table 7.4.1-1.

Table 7.4.1-1 SBOTV Average Processing Times

	A	B	C	D	E	F
1	FUNCTIONAL		TIME		AVERAGE	ACCUMULATIVE
2	FLOW	OPERATION	REQUIRED	FREQUENCY	TIME PER	TIME
3	CODE		(HOURS)		MISSION	
4						
5	1.1	PRELAUNCH PROCESSING	5.58	1	5.58	5.58
6	2.1	OTV MATING	15.50	1	15.5	21.08
7	3.1	DEPLOY FROM SPACE STATION	2.67	1	2.67	23.75
8	4.1.1	BERTH IN HANGAR	2.17	1	2.17	25.92
9	4.1.2	INSPECTION	6.50	1	6.5	32.42
10	4.1.3	COMPLETE SAFING	0.67	1	0.67	33.09
11	4.1.4.1	PROPELLANT TANK SCH MAINT	2.25	1/5	0.45	33.54
12	4.1.4.2.1	AVIONICS MODULE TEST	2.08	1	2.08	35.62
13	4.1.4.2.2	AVIONICS MODULE R&R	1.83	1/80	0.02	35.64
14	4.1.4.2.3	AVIONICS ACS UPDATE	0.75	1/5	0.15	35.79
15	4.1.4.3.1	RCS LEAK CHECK	0.83	1	0.83	36.62
16	4.1.4.3.2	RCS TRANSDUCER CHECK	0.92	1	0.92	37.54
17	4.1.4.3.3	RCS RESUPPLY	1.83	1/5	0.37	37.91
18	4.1.4.3.4.1	RCS TRANSDUCER R&R	2.33	1/10	0.23	38.14
19	4.1.4.3.4.2	RCS THRUSTER JET R&R	2.92	1/5	0.58	38.72
20	4.1.4.4.1	ENGINE POST FLIGHT MAINTENANCE	4.25	1	4.25	42.97
21	4.1.4.4.2	ENGINE PERIODIC MAINTENANCE	2.00	1/3	0.67	43.64
22	4.1.4.4.3	ENGINE R&R	5.50	1/10	0.55	44.19
23	4.1.4.5.1	AEROBRAKE INSPECTION	1.17	1	1.17	45.36
24	4.1.4.5.2	AEROBRAKE R&R	4.42	1/5	0.88	46.24
25	4.1.5.1.1	PROPELLANT TANK R&R	3.00	1/50	0.06	46.30
26	4.1.5.1.2	PROP TANK INSULATION REPAIR	2.00	1/3	0.67	46.97
27	4.1.5.1.3	PROP SYSTEM TRANSDUCER R&R	2.08	1/5	0.42	47.39
28	4.1.5.1.4.1	PROP UTILIZATION SYSTEM R&R	2.17	1/80	0.03	47.42
29	4.1.5.1.4.2	THERMAL DYNAMIC VENT SYS R&R	2.17	1/80	0.03	47.45
30	4.1.5.1.5	RECONFIGURE TANKS	12.00	1/80	0.15	47.57
31	4.1.5.2	AVIONICS ANTENNA R&R	2.58	1/20	0.13	47.70
32	4.1.5.5	TURBOPUMP R&R	4.67	1/20	0.23	47.93
33	4.1.5.6	AEROBRAKE TPS REPAIR	3.08	1/3	1.03	48.96
34	4.1.5.7	GHe, GO2 OR GH2 REGULATOR R&R	2.50	1/10	0.25	49.21
35	4.1.5.8	GHe, GO2 OR GH2 SPHERE R&R	2.17	1/50	0.04	49.25
36	4.1.5.9	PAYLOAD INTERFACE R&R	4.00	1/50	0.08	49.33
37	4.1.5.10	LO2 OR LH2 PRESS SYS R&R	2.58	1/20	0.13	49.46
38	4.1.5.11	CORE STRUCTURE R&R	81.92	1/200	0.41	49.87
39	5.1.1	LOAD ASSEMBLY PROGRAM	0.50	1/40	0.01	49.88
40	5.1.2	POSITION CORE IN CRADLE	1.00	1/40	0.02	49.90
41	5.1.3	INSTALL LOOSE HARDWARE	2.75	1/40	0.07	49.97
42	5.1.5	INSTALL BRAKE SUPT ASSEMBLY	2.58	1/40	0.06	50.03
43	5.1.6	INSTALL GPS ANTENNA	0.75	1/40	0.02	50.05
44	5.1.7	ATTACH UMBILICALS	11.08	1/40	0.28	50.33
45	6.1.2	OTV PROPELLANT LOADING	9.92	1	9.92	60.25
46	6.1.3	OTV DETANKING	5.10	1	5.1	65.35

7.4.2 Ground-Based OTV Processing Operations

A turnaround time for the ground-based vehicles was determined by utilizing data prepared by Boeing under a NASA KSC Study (NAS10-11168). The Boeing OTV Launch Operations Study was performed using a generic OTV. Table 7.4.2-1 is taken from the "Recurring Nominal Flow" presented by Boeing, but

Table 7.4.2-1 GBOTV Average Processing Times

MAINTENANCE & REFURBISHMENT

NO.	SER TIME	MH	TASK
1	19	120	MOVE TO OTVPF
2	26	260	MAINTENANCE
3	0	0	UNPLANNED MAINT.
4	0	0	MODS
5	3	30	RETEST VERIF.
6	7	40	STORAGE

PREPARATIONS

NO.	SER TIME	MH	TASK
7	56	480	MECHANICAL ASSEMBLY
8	27	135	ELECTIRCAL ASSEMBLY
9	50	590	OTV INTEG SYS TEST

OTV / SC INTEGRATION

NO.	SER TIME	MH	TASK
10	12	88	OTV /SC MECH/ELECT MATE
11	14	116	OTV / SC INTEG TEST

LAUNCH PREPARATION

NO.	SER TIME	MH	TASK
12	12	96	PREPS TO MOVE
13	8	52	INSTALL IN CAN
14	10	80	INSTALL IN RSS
15	14	104	ADDN'L SUBSYS INSTL
16	14	80	LOAD OTV RCS
17	7	53	INSTALL IN ORBITER
18	10	91	PL/ORB INTFC TEST
19	7	70	SC POCC TEST
20	11	21	FINAL PL CLOSEOUT
21	9	62	LAUNCH PREPS
22	13	79	DEPLOY OTV/SC

MISSION & RECOVERY

NO.	SER.TIME	MH	TASK
23	7	40	MOVE ORB TO OPF
24	0	0	MOVE TO CRYO FACL
25	0	0	VENT TANKS

	SER.TIME	MH
TOTALS	336	2687

(1) TAKEN FROM BOEING GBOTV MANPOWER FLOW, PRESENTED AT KSC ON 1-31-86.

has been slightly altered to better portray the MMC concept of OTV as opposed to the Boeing concept. The resultant turnaround time for the ground-based Vehicle is 336 hours or 2 weeks.

There are several reasons why the time requried for ground-based turnaround is considerably longer than space-based time. Firstly, all ground operations have to be integrated with (and secondary to) launch vehicle preparations; secondly, a greater amount of refurbishment, testing, and integration takes place on the ground; and thirdly, our concept for space-basing is to use as much automation as possible with EVA used only as a last contingency.

7.4.3 Turnaround Labor Cost

Although the man-hours of space-based labor is only a fraction of that needed for ground operations, its cost is considerably more. The high cost of space-based labor is another reason for providing a degree of automation at Space Station.

Table 7.4.3-1 Turnaround Labor Costs

LABOR	SPACE BASED	GROUND BASED
MANHOURS REQUIRED FOR NORMAL TURN-AROUND	65.4(1)	2687(2)
AVERAGE LABOR COST PER TURN-AROUND	$1,224,484(3)	$64,488(4)

(1) FROM SPACE BASED PROCESSING OPERATIONS TABLE, CONSIDERING A DEDICATED IVA ASTRONAUT TO OPERATE RMS AND MONITOR ALL TASK PERFORMED, EITHER MANUAL OR AUTOMATIC.

(2) FROM GROUND BASED PROCESSING OPERATIONS TABLE.

(3) IVA LABOR COSTS @ $18,732/HOUR FROM REVISED GROUND RULES, D.R. SAXTON TRANSMITTAL PF20(86-50), MARCH 20, 1986.

(4) LAUNCH SERVICE CREW PERSONNEL AVERAGE LABOR COST @ $185/DAY, FROM NASA COMPTROLLER OFFICE, SYMPHONY MODEL.

7.4.4 Turnaround Time Available and Fleet Size Required, Scenario #2

An analysis of the five mission model scenarios was made to determine the maximum turnaround time available on a per-mission basis. Mission durations were determined by taking the stay time given in the March 14, 1986 Ground Rules and adding times required for launch and return. For space-based operations, all missions were considered to require one day up and one day back, with the exception of the Lunar Missions which were considered to require 3 days up and 3 days back. For the ground-based vehicle, 2 days were added to each mission to cover LEO phasing, checkout and launch; rendezvous, disassembly and loading of the vehicle into the cargo bay; and scheduling and time delays associated with STS return to earth. Considering only one OTV in the fleet, total mission days per year were then determined and the average turnaround time available computed.

For the Scenario #2 space-based vehicle concept, sufficient time exists to turn around an OTV and complete all mission requirements with only one OTV in the fleet at a given time as shown in Table 7.4.4-1.

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Table 7.4.4-1 Average Turnaround Time Available, Scenario #2

	A	B	C	D	E	F	G	H	I	J	K	L	M	N	O	P	Q	R	S
1		MISS	ION																
2		DURA	TION																
3	MISSIONS	(DA	YS)					OTV	FLIGH	TS	PER	FY							
4		GBOTV	SBOTV	95	96	97	98	99	00	01	02	03	04	05	06	07	08	09	10
5	GEO PLATFORM	5	3				1												
6	PLANETARY	4	2	1		1	3	2	3		1		1			2			
7	MULT-PLATFORM DELIVERY	5	3	4	7	9	10	9	11	6	3	1	3	2	3	3	5	4	4
8	INDIVIDUAL GEO STATION	5	3	1	2		2	4		1									
9	UNMANNED SERVICING	6	4																
10	SATELLITE RETRIEVAL	6	4				1			1									
11	MANNED GEO SORTIE	10	8								1	1	2	2	2	2	2	2	2
12	GEO SHACK	14	12				1						1						
13	GEO SHACK LOGISTICS	6	4					1	1	2	1	2	4	4	4	4	4	4	6
14	UNMANNED LUNAR	15	13					2				1							1
15	UNMANNED SURFACE DELIVERY	15	13															3	1
16	LUNAR ORBIT STATION	24	22																
17	LUNAR SURFACE SORTIE	24	22																
18	DOD	5	3	15	15	15	15	15	15	15	15	15	15	15	15	15	15	15	15
19																			
20	SUBTOTALS			21	24	25	33	33	30	25	21	20	26	23	24	26	26	28	29
21																			
22	REFLIGHTS	5	3			1	1		1		1		1		1		1		1
23																			
24	TOTALS			21	24	26	34	33	31	25	22	20	27	23	25	26	27	28	30
25																			
26	GROUND BASED VEHICLE																		
27	TOTAL MISSION DAYS PER YEAR			104	120	129	177	180	153	128	115	117	157	119	139	142	149	184	186
28																			
29	AVE. TURNAROUND TIME AVAIL	ABLE	(DAYS)	12.4	10.2	9.1	5.5	5.6	6.8	9.5	11.4	12.4	7.7	10.7	9.0	8.6	8.0	6.5	6.1
30																			
31	NUMBER OF OTVs REQUIRED			2	2	2	3	3	2	2	2	2	2	2	2	2	2	3	3
32																			
33	SPACE BASED VEHICLE																		
34	TOTAL MISSION DAYS PER YEAR				72	77	109	118	91	78	71	77	103	83	89	90	95	128	126
35																			
36	AVE. TURNAROUND TIME AVAIL	ABLE	(DAYS)		10.2	9.3	7.5	7.5	8.8	11.5	13.4	14.4	9.7	12.3	11.0	10.7	10.0	8.5	8.0
37																			
38	NUMBER OF OTVs REQUIRED				1	1	1	1	1	1	1	1	1	1	1	1	1	1	1
39																			

For the ground-based vehicle concept in Scenario #2, at least two OTVs are needed in the fleet during most years and a fleet of three is needed during the years 1998, 1999, 2009, & 2010.

7.4.5 Turnaround Time Available and Fleet Size Required, Scenario #5

The Scenario #5 mission model has a much higher flight rate than Scenario #2. For the ground-based vehicle concept, with total mission time surpassing 365 days in years 2007 and beyond, fleet size grows from two in the early years to six in 2010. For a space-based vehicle concept, a fleet of one will still suffice until the heavy traffic years, starting in 2006. The negative values for "average turnaround time available" for the ground-based OTV for years 2007 - 2010 mean that two separate parallel ground processing facilities will be required.

Table 7.4.5-1 Average Turnaround Time Available, Scenario #5

	A	B	C	D	E	F	G	H	I	J	K	L	M	N	O	P	Q	R	S
1–3	MISSIONS	MISSION DURATION (DAYS)						OTV FLIGHTS PER FY											
4		GBOTV	SBOTV	95	96	97	98	99	00	01	02	03	04	05	06	07	08	09	10
5	GEO PLATFORM	5	3				1												
6	PLANETARY	4	2	1	2	2	6	3	3		1		3		1	3			
7	MULT-PLATFORM DELIVERY	5	3	4	7	9	10	9	11	6	3	1	3	2	3	4	6	5	5
8	INDIVIDUAL GEO STATION	5	3	1	2		2	4		2	1	1		2					4
9	UNMANNED SERVICING	6	4																
10	SATELLITE RETRIEVAL	6	4				1			1									
11	MANNED GEO SORTIE	10	8					1	1	2	2	2	2	2	2	2	2	2	2
12	GEO SHACK	14	12				1	1											
13	GEO SHACK LOGISTICS	6	4					1	1	5	4	4	4	4	4	6	6	6	6
14	UNMANNED LUNAR	15	13		2				1							1			1
15	UNMANNED SURFACE DELIVERY	15	13																
16	LUNAR ORBIT STATION	24	22														1		
17	LUNAR SURFACE SORTIE	24	22												1	1	2	2	2
18	DOD	5	3	15	15	15	15	15	15	15	15	15	15	15	15	15	15	15	15
19	NUCLEAR WASTE DISPOSAL	4	2				5	18	19	22	24	25	26	28	34	38	44	51	57
20–21	SUBTOTALS			21	28	26	41	52	51	53	50	48	53	53	60	70	76	81	92
22–23	REFLIGHTS	5	3			1	1	1	1	1	1	1	1	1	1	2	1	2	2
24–25	TOTALS			21	28	27	42	53	52	54	51	49	54	54	61	72	77	83	94
26																			
27	GROUND BASED VEHICLE																		
28	TOTAL MISSION DAYS PER YEAR			104	158	133	209	259	254	264	244	234	255	256	303	376	414	418	502
29–30	AVE TURN AROUND TIME AVAILABLE	(DAYS)		12.4	7.4	8.8	3.7	2.0	2.1	2.0	2.4	2.7	2.0	2.0	1.0	-0.1	-0.6	-0.6	-1.5
31–32	NUMBER OF OTVs REQUIRED			2	2	2	3	4	4	4	4	4	4	4	5	5	5	5	6
33																			
34	SPACE BASED VEHICLE																		
35	TOTAL MISSION DAYS PER YEAR				102	79	125	123	150	156	142	106	147	148	181	220	260	252	289
36–37	AVE TURN AROUND TIME AVAILABLE	(DAYS)			9.4	10.6	5.7	4.6	4.1	3.9	4.4	5.3	4.0	4.0	3.0	2.0	1.4	1.4	0.9
38–39	NUMBER OF OTVs REQUIRED				1	1	1	1	1	1	1	1	1	1	2	2	3	3	3

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7.4.6 OTV Production Requirements, Ground-Based Fleet

The analysis summarized in Table 7.4.6-1 was conducted to determine production requirements necessary to meet the needs of an all ground-based fleet of OTVs for the Scenario #2 mission model. Since two different size vehicles are used in the ground-based scheme, the fleet sizing was done after assigning the various missions to either the small or large vehicle as determined by performance requirements.

Table 7.4.6-1 Production Requirements for All Ground-Based Fleet

	A	B	C	D	E	F	G	H	I	J	K	L	M	N	O	P	Q	R	T
1	SCENARIO 2	1994	1995	1996	1997	1998	1999	2000	2001	2002	2003	2004	2005	2006	2007	2008	2009	2010	TOTAL
2	52K MISSIONS	0	21	24	25	30	30	26	23	18	16	18	17	18	18	20	19	20	343
3	74K MISSIONS	0	0	0	1	4	3	5	2	4	4	9	6	7	8	7	9	10	79
4	2-STAGE MISSIONS (74K)	0	0	0	0	1	1	3	0	1	1	2	0	0	1	0	3	1	14
5																			
6	MIN 52K FLEET SIZE	0	2	2	2	2	2	2	2	1	1	1	1	1	1	2	1	2	
7	MIN. 74K FLEET SIZE	0	0	0	1	1	1	1	1	1	1	1	1	1	1	1	1	1	
8																			
9	PRODUCTION REQ'D (52K):																		
10	OTV VEHICLE	2	0	0	1	1	0	1	0	1	0	1	0	1	1	0	1	0	10
11	AEROBRAKE	4	1	5	5	5	5	4	4	3	3	2	4	3	2	4	3	4	61
12	ENGINE SET	0	1	1	2	0	2	1	1	0	1	0	2	0	0	1	1	1	14
13																			
14	USAGES REMAINING (52K)																		
15	OTV VEHICLE	80	59	35	50	60	30	44	21	43	27	49	32	54	76	56	77	57	
16	AEROBRAKE	30	14	15	20	20	15	14	11	13	12	9	12	14	11	11	12	12	
17	ENGINE SET	30	24	15	35	20	20	24	16	13	12	9	22	19	16	11	22	17	
18																			
19	PRODUCTION REQ'D (74K):																		
20	OTV VEHICLE	0	0	1	0	0	0	0	0	0	1	0	0	0	0	1	0	0	3
21	AEROBRAKE	0	0	0	0	1	1	1	1	1	1	1	2	1	2	1	2	2	17
22	ENGINE SET	0	0	0	0	0	0	1	0	0	0	0	1	0	1	0	0	1	4
23																			
24	USAGES REMAINING (74K)																		
25	OTV VEHICLE	0	0	40	39	34	30	22	20	15	50	39	33	26	17	50	38	27	
26	AEROBRAKE	0	0	5	4	4	5	2	5	5	10	4	8	6	7	10	8	7	
27	ENGINE SET	0	0	15	14	9	5	12	10	5	15	4	13	6	12	20	8	12	

Ground Rules applied to this analysis assumed the basic vehicle would serve for 40 missions, aerobrake for 5 missions, and engines for 15 missions. The term "engine set" was used rather than "engine" to accommodate the different configurations under study. In the case of the 3 engine configuration, engine sets can be multiplied by 3 to determine total engine needs.

Production requirements for a current year were determined by examining the total flights needed during the next year plus 1/2 the flights needed during the subsequent year less the usages remaining from the previous year.

7.4.7 OTV Production Requirements, Space-Based Fleet

Production requirements for an all space-based fleet to meet Scenario #2 needs, are much the same as the ground-based concept with two exceptions: with the space-based scheme, only one size vehicle is used and the total program length is a year shorter since space-basing was not assumed to start until 1996.

Table 7.4.7-1 Production Requirements for All Space Based Fleet

	A	B	C	D	E	F	G	H	I	J	K	L	M	N	O	P	Q	S
1	SCENARIO 2	1995	1996	1997	1998	1999	2000	2001	2002	2003	2004	2005	2006	2007	2008	2009	2010	TOTAL
2																		
3	TOTAL MISSIONS	0	24	26	34	33	31	25	22	20	27	23	25	26	27	28	30	401
4																		
5	TWO STAGE MISSIONS	0	0	0	1	1	3	0	1	1	2	0	0	1	0	3	1	14
6																		
7	TOTAL MISSIONS	0	24	26	35	34	34	25	23	21	29	23	25	27	27	31	31	415
8																		
9	MIN. ACTIVE FLEET SIZE	0	1	1	1	1	1	1	1	1	1	1	1	1	1	1	1	
10																		
11	PRODUCTION REQD:																	
12	OTV VEHICLE	1	1	1	0	1	1	1	0	1	0	1	1	0	1	1	1	12
13																		
14	AEROBRAKE	3	3	5	7	6	5	3	5	4	5	4	4	5	5	5	6	75
15																		
16	ENGINE SET	1	0	2	1	2	1	0	2	0	2	1	0	2	2	0	2	18
17																		
18	USAGES REMAINING:																	
19	OTV VEHICLE	40	56	70	35	41	47	62	39	58	29	46	61	34	47	56	65	
20																		
21	AEROBRAKE	20	16	20	20	21	17	12	14	18	14	16	16	14	17	16	20	
22																		
23	ENGINE SET	30	21	40	20	31	27	17	24	18	19	26	16	19	37	21	35	

7.4.8 OTV Production Requirements

Overall production required for either a ground-based or space-based program is summarized in Table 7.4.8-1. If a combination of ground-based/space-based were used, these results would be somewhat different. Over time, however, production requirements are most closely related to mission model, not basing concept.

Table 7.4.8-1 Production Requirements Comparison

- GROUND BASED 16 YEAR PROGRAM (1995 -2010)
  - TOTAL PRODUCTION REQUIREMENTS

74K OTV		52K OTV	
LARGE VEHICLE	3	SMALL VEHICLE	10
AEROBRAKE	17	AEROBRAKE	61
ENGINE SETS	4	ENGINE SETS	14

- SPACE BASED 15 YEAR PROGRAM (1996 - 2010)
  - TOTAL PRODUCTION REQUIREMENTS

74K OTV	
LARGE VEHICLE	12
AEROBRAKE	75
ENGINE SETS	18

7.5 DESIRED SPACE STATION SUPPORT TO GBOTV

Support from Space Station would be desirable, improve efficiency, and increase the flexibility of operations of a ground-based vehicle. The amount of support desired is somewhat dependent upon the launch vehicle utilized.

7.5.1 Large Cargo Vehicle Delivery to LEO

Large cargo vehicle delivery to LEO: for the LCV delivery of OTV, it is assumed that the OTV and payload will be delivered to LEO fully assembled, fueled and intact, ready to launch. Pre-mission support from Space Station would be limited to temporary storage/repair should a payload fail during ground launch. Post mission support would be the provision of a berthing area for OTV to await the arrival of the STS and to provide assistance in disassembly and installation into the Shuttle bay.

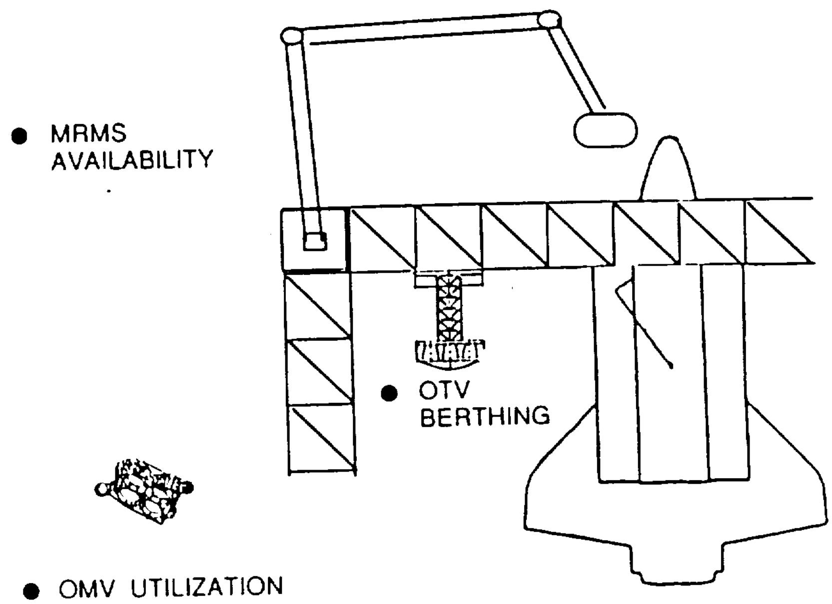

Figure 7.5-1 Space Station Support to GBOTV

7.5.2 STS Launch or Other Launch Vehicles

Considering an STS launch, space-base support would be very useful for OTV/payload mating operations, mating the 2nd stage OTV with the OTV/payload, performing onorbit checkout of the system, and providing temporary storage to payloads sent ahead of the OTV in order to accommodate manifesting or to increase the launch window. Post mission support would be similar to that needed for the large cargo delivery and would also be useful as a storage place for the multiple payload adapter. Similar support activities could also be provided to the integration of an OTV with payloads delivered to LEO by other means; such as by CELV, or even launch vehicles from the other countries (European, Japanese, Chinese).

During the early stages of the GBOTV program, the Space Station facilities and personnel could be used effectively to prove out, demonstrate, and develop concepts to be utilized on the SBOTV at some future date. Inspection procedures, diagnostic checkout, limited remove and replace functions, utilization of a rudimentary RMS, demonstration of aerobrake reusability, and EVA/IVA timelines could all be evaluated and analyzed. Additionally, procedures tools and techniques could be developed and evaluated, and demonstrations performed of propellant transfer and storage (including propellant hitchhiking), adequacy of meteoroid and debris shielding, traffic control, communications, and OMV utilization.

8.0 PREFERRED STAS REUSABLE OTV PROGRAM COSTS

This section presents the criteria, cost analysis methodology and total program costs by WBS for the preferred GBOTV/SBOTV program within the context of the STAS era launch vehicles. The trade study results included in Section 4.0 of this volume led to the selection based on the lowest constant/ discounted LCC. This section will present a comprehensive outline of the cost methodology, ground rules and assumptions governing both the trade study efforts as well as the preferred program cost. In general, the trade study cost estimates for alternative concepts were reported to the same work breakdown structure (WBS). This permitted visibility to every effort of LCC and allowed annual fiscal year funding projections for budget and discounted LCC comparisons. The presentation of the selected program cost by this method should serve as a guide in providing more clarification of the methodology/results of the trades.

The scope of the cost analysis effort includes all costs directly incurred due to upper stage requirements and other supporting programs. Besides cost impacts directly related to stage requirements, peripheral cost elements, such as launch costs, Space Station and OMV support, and propellant logistics are also included.

This section is organized to document the methodology, reporting structure, schedule, test/operational/spares philosophy and cost ground rule and assumptions. Because the intent of this section is nearly identical in structure to Volume VI, Cost Analysis, of the phase A contract's final report, references to that volume will be made. This will be especially true in the methodology area, since a consistent approach from the STS constrained OTV results has generally been maintained.

8.1 COSTING APPROACH AND RATIONALE

8.1.1 Methodology

The WBS and WBS Dictionary were developed in conjunction with the Marshall Space Flight Center (MSFC) engineering cost group during the early phases of contract performance. The resultant WBS structure provided a consistent and thorough format for reporting all OTV and related programmatic cost impacts However, the WBS structure was later updated (with MSFC concurrence) to include the cost impacts for other programs supporting the OTV.

The mechanism for estimating and reporting costs to the WBS is an automated LCC computer model developed by Martin Marietta with corporate funding. The model calculates all phases of costs based on the technical description of the OTV, the operational scenarios and the requirements of any supporting programs, e.g., Space Station, LCV.

Typical inputs to the LCC model include the following:

- o OTV stage weight at the subsystem component level;
- o Test hardware requirements;
- o Annual mission and propellant requirements;
- o Operational turnaround times;
- o Intravehicular activity (IVA) and extravehicular activity (EVA) requirements;

- o Key implementation schedule dates;
- o Supporting program data; and
- o Specific payload transportation requirements.

The cost WBS reporting structure is consistant with the mass properties data given in Tables 6.2.2-2 and -4. These mass properties are the basis of OTV development and unit cost estimates. The LCC model aggregates the costs by phase and hardware elements to produce a hierarchy of cost reports by WBS. This model is a proven effective tool for assessing the impact of design/operational sensitivities and for displaying the resultant cost estimates in a concise format.

The key to our cost estimating methodology is the Martin Marietta cost analysis database (CADB). The CADB, which is consistent with government and industrywide historical experience, contains cost data for previous Martin Marietta programs (e.g., Viking, Titan transtage) in the form of cost estimating relationships (CER). The CERs provide the basis for estimating the cost of generic hardware/software development and production efforts. Additional cost model CERs (e.g., Space Station, SAMSO spacecraft), were often used as secondary parametric cost estimating resources.

These CERs are organized so the cost analyst may focus the cost estimates towards programs that are most similar to the OTV. For example, as a test of reasonableness, aerobrake estimates were checked against Viking aeroassist cost data with proper complexity normalization. Similarly, specific data points from Martin Marietta's propellant tankage experience were used to refine the nonrecurring cost and unit cost tankage concepts.

To complement our historical cost data, vendor and government quotes were used to develop certain key cost impacts. The most significant areas where this practice was applied were engine design and development and unit cost impacts.

Operations cost impacts were developed by incorporating operational definitions and inputs provided by the MSFC study ground rules. Martin Marietta supported these data with analyses and historical data gained from previous space programs. The annual operations fixed costs, variable cost per flight (CPF) and learning curves were based on the aggregate impacts of the above inputs. The primary drivers in operations cost inputs were the Rev.9 mission model payload requirements and the integration of supporting programs with OTV operational requirements. The operational cost elements identified include the following:

- o The annual propellant and IVA/EVA;
- o LCV integration and launch of OTV hardware and payloads;
- o Hardware operational spares and stage hardware refurbishment;
- o Expected mission losses;
- o Orbital Maneuvering Vehicle (OMV) use;
- o Mission Control; and
- o Program support.

Inputs for each of these elements were developed in relation to the specifics of the OTV mission model, study ground rules and Martin Marietta analyses. The primary focus of the analyses is based on the requirements of Scenario 2 of the Rev. 9 mission model.

8.1.2 Master Schedule

A set of OTV programmatic schedules was developed to assist the MSFC Phase C/D OTV implementation planning and to identify the time phasing of OTV support programs. The schedules consist of a detailed plan for each of the program's lower level efforts. The schedules are laid out to clearly identify all major programmatic efforts leading to the OTV initial operational capability (IOC). These schedules were also used to prepare OTV funding profiles and present value evaluations.

Figure 8.1.2-1 highlights the DDT&E schedule of the ground-based OTV nonrecurring efforts for engineering, tooling, test article fabrication, test operations and Kennedy Space Center (KSC) facility efforts.

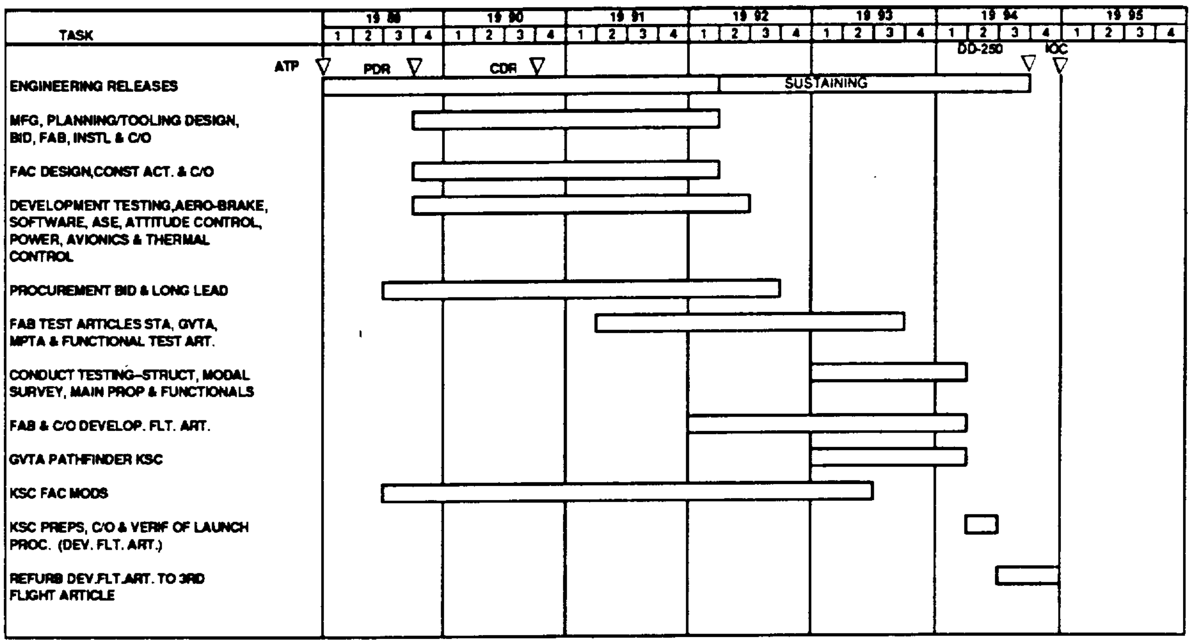

Figure 8.1.2-1 Ground-Based OTV (52 Klb) Implementation Schedule

Figure 8.1.2-2 highlights the DDT&E schedule of the space-based OTV and provides a schedule similar to the ground-based OTV DDT&E. In addition, this figure highlights the nonrecurring and manufacturing activities included in OTV Space Station accommodations. Due to the evolutionary approach of the preferred OTV program, the ground test article fabrication and test operations represent only those efforts uniquely defined by the space-based requirements. Justification for this approach is provided by the similarities in stage design as well as the opportunity to employ the ground-based stage as a testbed in many key test areas.

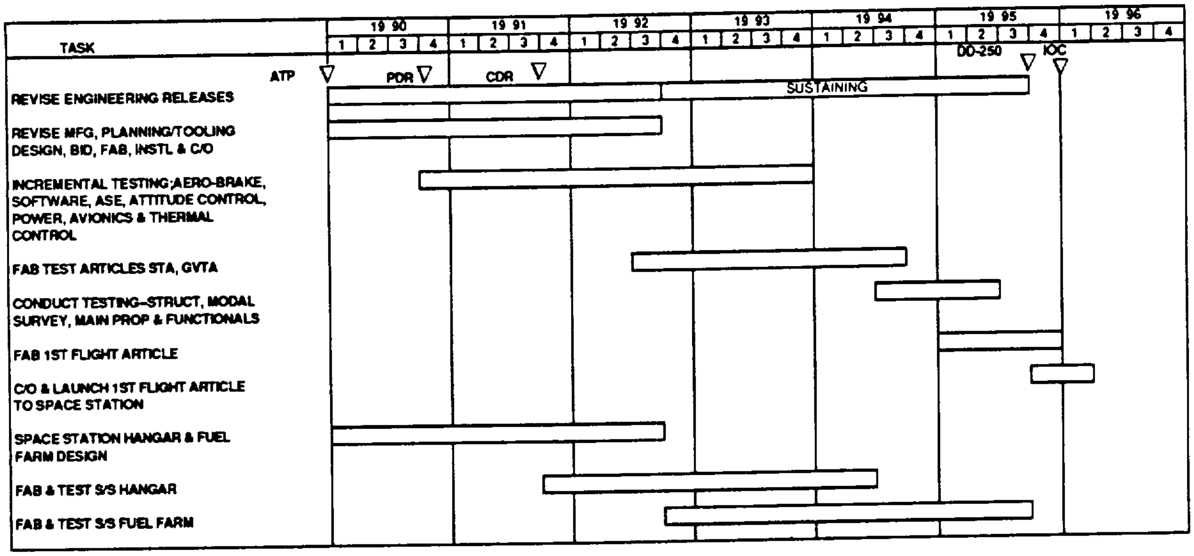

Figure 8.1.2-2 Space-Based OTV (74 Klb) Implementation Schedule

8.1.3 Test Philosophy

The test philosophy adopted for the OTV DDT&E is consistent with past Martin Marietta experiences in similar programs and designed to take advantage of the evolutionary approach to implementation of our preferred concept.

The initial ground-based OTV will require a comprehensive test program that can be roughly divided into three parts.

The first portion of the OTV test program is included in the research and technology (R&T) effort identified by the study ground rules. This includes efforts preceding DDT&E that are involved with two key technology areas: the creation of an advanced engine technology base and an Aeroassist Flight Experiment (AFE).

The second portion of the test requirements occurs during the DDT&E program phase. This effort includes the following: all lower level structural, thermal, stress, etc., testing; a full-scale ground vibration test article (GVTA); a structural test article (STA); a functional test article (FUTA); a main propulsion test article (MPTA) manufacture; and test operations. Table 8.1.3-1 is a matrix of subsystems components included in each of the test articles. The refurbishment hardware column corresponds to the level of effort required to manufacture an operational stage from test hardware subsystems. Major assembly and checkout costs are included as applicable.

Table 8.1.3-1 Ground Test Article Subsystem Requirements

Subsystem	STA	GVTA	MPTA	Func. Test	Refurb.	Total
Structures	1	1			0.5	2.5
Tanks	0.5	1	1			2.5
Main Prop (less Eng.)			1.5		0.6	2.5
Main engine			1		1	2
RCS			1	1(2)	0.1	2
GN&C				1.1	1	2
C&DH				1	1	2
Electrical Power			1	1(2)	0.5	2.4
Environmental Control				1	1	2
Aerobrake	1(1)	1				2

(1) Separate test from structures Structural Test Article (STA)
(2) Separate test avionics FTA

The third phase of the test program for the ground-based OTV is the manufacture and operations of the Flight Test Article (FTA), including: the cost of a fully operational ground-based stage, KSC pathfinder and LCV integration costs, and partial costs of LCV launch. The ground-based FTA and GVTA are refurbished to operational units to support the ground-based mission requirements.

Due to the operational experience obtained during the ground-based period, our test philosophy for the evolutionary space-based program is to minimize test hardware and operations requirements. This experience includes initial LCV delivery of operational hardware, payload mating, stage hardware characteristics other than hardware unique to the space-based stage, and geosynchronous Earth orbit (GEO) delivery scenarios.

The primary test impacts encountered occur in the man/OTV interface, Space Station stage refurbishment techniques, onorbit propellant transfer, and OMV logistics.

The requirement for a dedicated space-based test flight was assumed to be unnecessary. The justification for this assumption is based on previous experience obtained during 1995 ground-based operations.

8.1.4 Operational Philosophy

The details of the ground-based and space-based OTV operational scenarios are presented throughout this volume. This section will not attempt to review all of these discussions, but will limit the discussion to how these operational scenarios were translated into operations costs.

Table 8.1.4-1 highlights the ground-based OTV operations cost elements by function and provides the basis of estimate for each element. Comments are provided for certain elements where further explanation is required.

Table 8.1.4-1 Ground-Based OTV Operations Cost Rationale

Operations Cost Element	Function	Basis of Estimate
Mission Operations	Mission Control	35 man-year/year effort
Program Support	Program Mgt. Sus. Eng., miscellaneous ground-based support	Historical program factors
Airframe Spares	H/W spares, prod. suppt, expendable tanks/brakes as required; ASE & GSE	% of unit cost & service life
Hardware IVA	Disassembly of tanks/ Brake; stage stowage	IVA/mission
Aerobrake Spares	H/W spares, prod. suppt	% of unit cost & service life; 92% learning
Engine Spares	H/W spares, prod. suppt	Unit $s & service life
H/W Refurb/Misc. Spares	GBOTV H/W, GSE, ASE spares ground process & refurbishment	% of unit $ & mission rate; crew size
Expected Mission Losses	Reliability based expectation of mission loss	(1-rel) * missions * expected value of an average mission
Propellant	Ground-based propellant cost and loading	Ground-based propellant @ $2/lb
Payload Clustering Structure	Multiple payload carrier refurbishment	% of unit cost & mission rate
LCV Launch Cost/ STS/STS II Return	LCV launch of OTV H/W, payloads & propellant; hardware return flight via STS/STS II	Study Ground Rule CPF; cost prorated by weight/volume user charge algorithm; minimum STS/STS II CPF return to launch site
Payload Transportation		Payloads manifested with OTV H/W & propellant; includes only ground processing crew

The overwhelming operations cost impact of the ground-based OTV is the LCV launch costs of combined stage hardware, propellant and payload to low Earth orbit (LEO). The launch vehicle manifesting philosophy applied in determining the user charge is consistent with the guidelines provided by JSC-11802, "STS Reimbursement Guide". Most of the ground-based OTV missions, except some support of the 22 lunar/planetary missions, were within the LCV performance and volume constraints provided in the study ground rules. Therefore, a single LCV launch was sufficient for all missions. To establish user charges, the OTV, propellant and payload were treated as an integral payload unit. A minimum STS/ STS II return flight charge of 6.7% of the user charge was uniformly applied to each GBOTV mission. This percentage represents the minimum STS/STS II user charge for carrying return ASE and is consistent with study ground rules. Sensitivities to this ground rule are documented in Section 4.9.

The other OTV operations cost elements are fairly well defined. The next most significant item is hardware spares. For ground-based missions the brake is treated as an expendable item that is replaced after each flight. Additional hardware impacts due to partially expendable tankage were included. Engine spares are based on service life replacement after initial operations items are expended. The initial refurbished FTA is sufficient to satisfy the first year mission model airframe requirements while the refurbished GVTA serves as an operational spare. Subsequent airframe spares are prorated on a per flight basis.

The program support impacts include program management, sustaining engineering and miscellaneous launch operations personnel costs. Flight hardware refurbishment includes a fixed work force dedicated to stage turnaround between missions. Crew size was based on turnaround scenarios identified in Section 7.1. The ground support equipment (GSE) and airborne support equipment (ASE) spares are also included in airframe spares costs.

Expected mission losses are a function of stage reliability and the expected cost of an average mission including stage hardware and payload. As defined by the study ground rules, the ground-based missions operations element includes a 35 man-year per year effort.

The payload clustering structure includes operational refurbishment costs for the multiple payload carrier and supporting ASE. The remaining operational cost elements consist of IVA time associated with hardware disassembly for STS/ STS II return.

The initial portion of the first mission (out of a useful life of 40 missions) of a space-based OTV flies as if it were ground-based. After payload delivery, the OTV returns to the Space Station where turn-around activities commence.

Table 8.1.4-2 shows the operations cost philosophy for the space-based OTV portion of the mission model. Many of the operations cost elements function in a similar manner to their ground-based counterparts. However, there are significant differences between ground-based and space-based OTVs that merit discussion.

Table 8.1.4-2 Space-Based OTV Operations Cost Rationale

Operations Cost Element	Function	Basis of Estimate
Mission Operations	Mission Control	35 man-year/year effort
Space Station Accommodations	IVA/maintenance activities of tank farm, hangar, maintenance of hardware/software	IVA estimates/annual facilities maintenance definition
Program Support	Program Mgt. Sus. Eng., miscellaneous support labor	Historical program factors
Airframe Spares	H/W spares, prod. suppt	Unit $ & service life
Aerobrake Spares	H/W spares, prod. suppt	Unit $ & service life
Engine Spares	H/W spares, prod. suppt	Unit $ & service life
H/W Spares Processing/ H/W IVA	Ground process of H/W spares; onorbit H/W IVA	Crew sizing, IVA times
Expected Mission Losses	Reliability based expectation of mission loss	(1-rel) * missions * expected value of an average mission
Propellant	Composite hitchhiked/ tanker cost per lb	63% hitchhiked, 37% dedicated tanker
Payload Clustering Structure	Multiple payload carrier refurbishment	% of unit cost & mission rate
OMV Usage	OMV deployment/retrieval during Space Station proximity operations	Per study ground rules; average propellant use of 1000 lb; 2 hours out, 1.5 hours return
LCV Launch Cost	LCV launch of initial operational stage & replacement H/W spares	Study Ground Rule CPF; Manifested consistent with STS reimbursement guide & H/W size/weight
Payload Transportation	LCV launch of payloads to Space Station	Manifested on LCV consistent with STS reimbursement guide by weight and volume constraints, payload mate IVA, Space Station user charge, payload ground processing

The primary difference between the operational philosophies of ground and space-basing and resulting cost impacts occurs in the switch in emphasis from the LCV launch costs of OTV stage, propellants and payload to onorbit propellant payload delivery to LEO. The LCV launch costs for SBOTV hardware include only the initial deployment of the operational space-based stage and subsequent delivery of operations spares. The LCV payload transportation costs are now treated as an independent cost element. On the other hand, the propellant requirements are now satisfied predominantly by propellant hitchhiking (over 63%) at a lower cost per pound rather than the LCV launch with hardware. The remainder of the propellant was provided by dedicated tanker flights. Alternatively, the IVA increases significantly and OMV use becomes an active operations cost element for the SBOTV.

Although the annual mission rates were similar, the ground-based manpower efforts (i.e., program support and hardware refurbishment), were significantly reduced during space-basing. This is due to the extensive robotics and imaging hardware/software developed for these functions onorbit at the Space Station. Aerobrake and tank spares were reduced due to reusability implementations while the engine and airframe spares were treated in the same manner as the ground-based OTV (other than onorbit changeout).

8.1.5 Spares Philosophy

Operational spares requirements are based on a combination of service life expectations and historical spares factors for aerospace programs. Initial hardware requirements at the IOC for both the ground-based and space-based stages are two complete units: one operational unit and one operational spare. This is the minimum constraint active throughout the period of operations.

Service life replacement begins as initial parts on the operational stage reach their expected life limits. Table 8.1.5-1 highlights those components affected by service life.

The multiple payload carrier, GSE, ASE and space support equipment (SSE) spares were calculated by historical program factors as a function of unit cost and mission model requirements. For funding purposes spares cost were allocated on an annual basis.

Table 8.1.5-1 Operational Spares Philosophy

Subsystem	Service Life (1)	LCV User Charge Per Delivery (2)
Aerobrake	$5^{(3)}$ missions	30%
Engine	10 missions	3%
Airframe, Avionics, etc.	$40^{(4)}$ missions	37%

(1) Initial IOC hardware provides service life performance
(2) Not applicable to ground-based operations
(3) Treated as an expendable subsystem during ground-based operations
(4) Expendable GBOTV Tankage replaced as required

8.1.6 Work Breakdown Structure

The WBS used to report OTV cost estimates was developed from the general WBS structure used by Martin Marietta on previous NASA studies. This WBS matrix format provides the flexibility to accommodate a variety of OTV stage configurations and supporting programs. At the same time, the format conforms to the LCC methodology and displays the cost estimates in a consistent manner.

The OTV WBS matrix (Figure 8.1.6-1) and its relationship to the Space Transportation WBS (Figure 8.1.6-2) are arranged to provide visibility to major OTV hardware elements, the major phases of program cost and the OTV impact to the space transportation architecture system. Volume V contains the complete WBS Dictionary definition.

8.1.7 Ground Rules and Assumptions

The following ground rules and assumptions were used and applied in a consistant manner to develop the OTV LCC estimate. They are grouped by programmatics, R&T, DDT&E, production, operations and facilities.

8.1.7.1 Programmatics

A) All costs are shown in constant fiscal year 1985 dollars and are exclusive of fees and contingencies.

B) The NASA study ground rules have been followed as applicable; exceptions are noted within the discussion.

8.1.7.2 R&T

The R&T cost impacts reflect study ground rule costs of $100M for the AFE and $53M for an advanced engine technology base.

8.1.7.3 DDT&E

A) Ground test hardware for the initial ground-based stage include a complete STA, GVTA, MPTA and functional test article. The follow-on space-based ground test hardware includes additional hardware as required.

B) The initial ground-based stage requires a dedicated FTA and LCV launch operations and STS return. The dedicated flight test was waived for the space-based stage.

C) The GBOTV GVTA and FTA are refurbished to meet initial operational hardware requirements.

D) Space-based OTV DDT&E efforts assume maximum sharing of previous ground-based experience (evolutionary approach).

E) DDT&E for the multiple payload carrier is included in the ground-based DDT&E.

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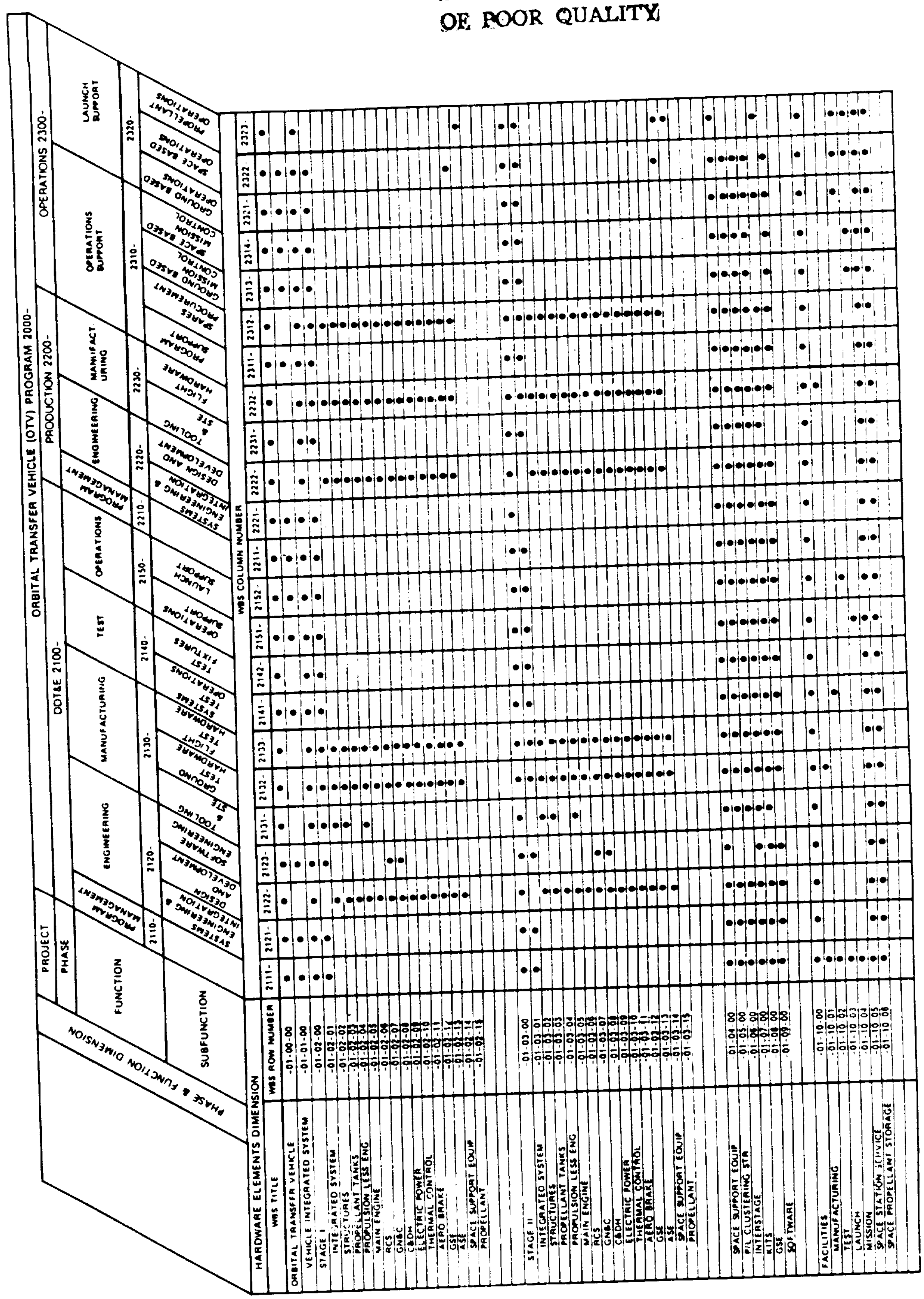

Figure 8.1.6-1 OTV Program Work Breakdown Schedule

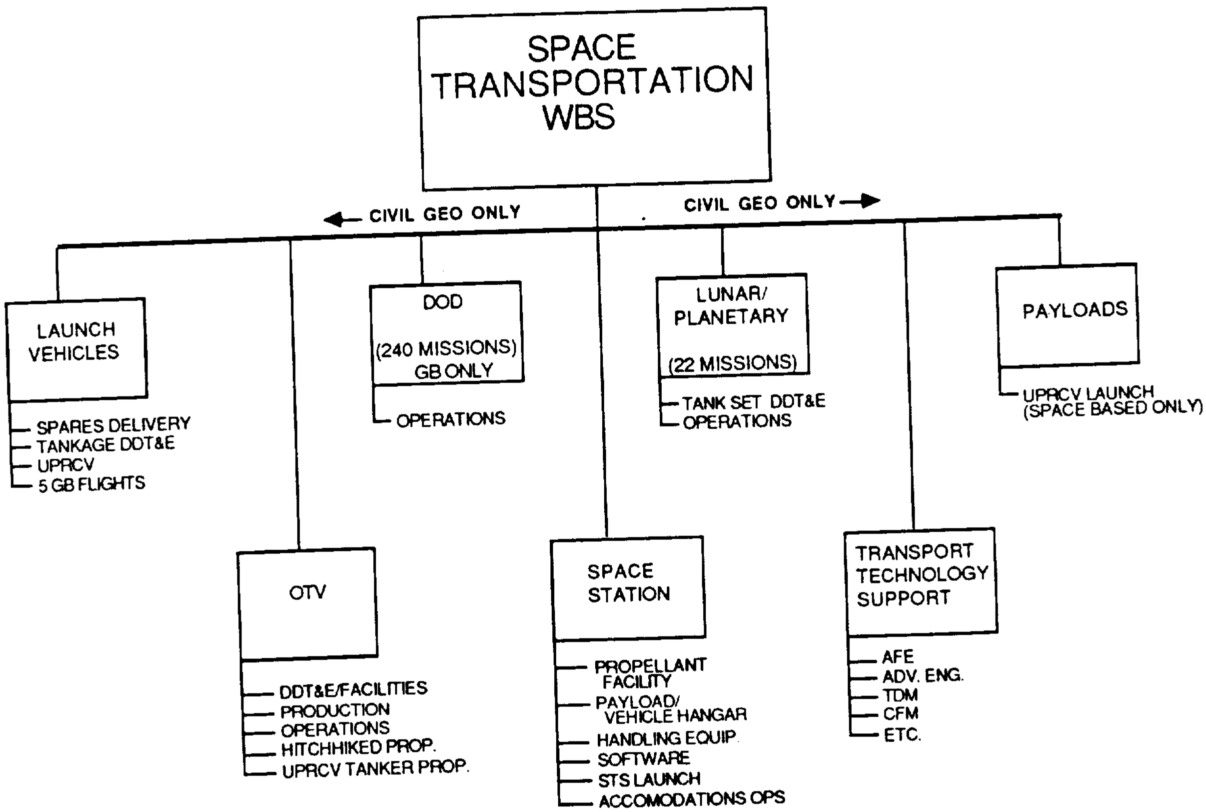

Figure 8.1.6-2 Space Transportation Work Breakdown Schedule

F) The Level II systems integration costs include the additional efforts required to interface stage hardware with other related program elements, e.g., LCV, STS/STS II return, hangar, refurbishment robotics, tank farm).

8.1.7.4 Initial Production

A) During operations, both the ground-based and space-based portions of the mission model require a minimum of two operational stages at all times (one operational unit; one spare). Refurbished DDT&E hardware satisfies ground-based IOC requirements. Production of two space-based stages prior to IOC is required to meet the space-based IOC requirements.

B) No production learning was applied to initial stage manufacture due to small production run.

C) The launch vehicle transportation charges of initial space-based production hardware are included in operations.

8.1.7.5 Operations

A) The Rev. 9 Scenario 2 mission model was used in determining reference operations costs.

B) A cost per LCV flight of $70M was used in determining operations costs. Launch performance to LEO (approximately 160 nm) was assumed to be 150,000 lb with a 25 ft diameter by 90 ft length payload envelope. Performance to Space Station altitudes (approximately 250 - 270 nm) was assumed to degrade to 109,000 lb. The performance degradation primarily influenced SBOTV propellant cost/lb as spares and payload delivery were volume constrained. ASE weight/length was included in manifesting. The STS/STS II return costs of GBOTV hardware were based on a CPF of $73M/ $20M, respectively. The estimates were derived from minimum return ASE weight/volume delivery constraints per study ground rules.

C) The cost estimate for the mission operations function was based on a fixed 35 man-year per year level of effort per basing mode.

D) Payload transportation costs were determined according to STS program user charge guidelines:
   1) Ground-based OTV payloads were manifested with OTV stage hardware as an integral payload unit.
   2) Space-based OTV payloads were charged according to volume/length constraints and reimbursement guide break points. A $250K Space Station user charge per payload was applied per study ground rules.

E) IVA time was charged at $18K per hour. EVA time was identified as a contingency function and not included in cost estimates.

F) A return flight charge was applied to ground-based missions at 0.067 of the STS user charge to pay for return ASE delivery on a nondedicated STS/STS II return flight.

G) Ground rules unique to space-based operations are:

1) Two OMV uses per mission were required for stage deployment. They were estimated according to the study ground rules at 2 hours out and 1.5 hours back, and an average of 1000 lb of propellant per OTV mission.
2) LCV launch costs include delivery of initial operational stage and operational spares as required.
3) Onorbit propellant costs were determined as a composite of hitchhiked propellant and dedicated LCV tanker delivery. Propellant hitchhiking supplied approximately 63% of the propellant required for the 155 civil GEO missions (1996 - 2010). The cost estimate of approximately $200/lb includes delivery tanks, OMV use and tank farm operations. Without tank farm operations, the cost per pound was approximately $170/lb. Dedicated tanker propellant costs were $750 and included tankage, OMV use and LCV launch cost. Approximately 100,000 lb propellant could be delivered to the Space Station per tanker event.

H) Operational spares cost estimates were developed according to the following guidelines:

1) Reference LCV transportation costs of $70M/flight to LEO were used, partial charges were based on the STS length charging algorithm.
2. Hardware service life and transportation charges are as follows:
   a) Aeroassist life, five flights; each brake delivered for 30% of a LCV charge;
   b) Engine life, 10 flights; replacement engine sets delivered for 3% of a full LCV charge; and
   c) Avionics, EPS, structures, 40 flights; spares delivered for 37% of LCV charge.

8.1.7.6 Facilities

Facilities cost impacts were based on new or modified square footage requirements and include the following:

A) Provisions for manufacturing floor space for DDT&e, initial production and operational spares hardware;

B) A dedicated OTV launch processing facility (KSC); and

C) Missions operations floor space and equipment at an existing facility.

8.2 SUMMARY COST PRESENTATION

The cost analysis task was conducted to provide NASA with economic justification and visibility into potential OTV program cost drivers and to determine the preferred OTV program approach that minimizes LCC. The OTV cost estimates were developed by LCC phase (i.e. DDT&E, Production and Operations) and include cost estimates for the impacts of other programs required to support OTV capability. In order to provide greater credibility to the cost analysis results, the detailed results of our preferred GBOTV/SBOTV program have been prepared.

The preferred OTV program for STAS era launch vehicles combines a dual basing capability approach to satisfying future upper stage transportation requirements. Section 4.9.2 provides top level vehicle characteristics of the ground-based and SBOTV stages. Section 6.2 includes the selected design concepts overview including weight statements and mission application descriptions.

Table 8.2-1 provides a brief overview of which stages and basing mode are applied to respective classes of missions. A description of the lunar and planetary missions is included in Section 6.2. The basic program approach includes a 1995 GBOTV IOC followed by a SBOTV IOC in 1996. The SBOTV provides the primary support to the civil GEO missions from that point on. Additionally, it serves as the basis for the majority of the lunar and planetary Scenario II missions. The GBOTV is used nearly exclusively for DOD payloads with limited support to lunar/planetary missions.

Table 8.2-1 Preferred Program Mission Application Overview

	TIMEFRAME	BASING MODE	STAGE APPLICATION	P/L WEIGHT CLASS	TOTAL MISSION
Civil GEO Missions I	1995	GB	52K GBOTV	14.6K	5
Civil GEO Missions II	1996-2010	SB	74K SBOTV	25.1K	155
DOD 28°	1995-2010	GB	52K GBOTV	10K	96
DOD Mid-Inclination	1995-2010	GB	52K GBOTV	10K	128
DOD Polar	1995--2010	GB	52K GBOTV	5K	16
Lunar/Planetary	1997-2010	SB/GB	74K SBOTV 52K GBOTV Aux. Tanks Solids	See Section 6.2	22

Figure 8.2-1 shows the total OTV program LCC by major program element and phase of $24.1B. The cost presentation is intended to emphasize the civilian GEO portion of the Scenario II mission model while showing additional cost requirements for DOD and lunar/planetary missions. The cost estimates for these later two classes of missions include only operations costs and unique DDT&E requirements (e.g. auxiliary tank set development). All other nonrecurring impacts are identified within separate categories. Operations cost elements listed outside of the DOD and lunar/planetary areas of the WBS are exclusive to civil GEO missions.

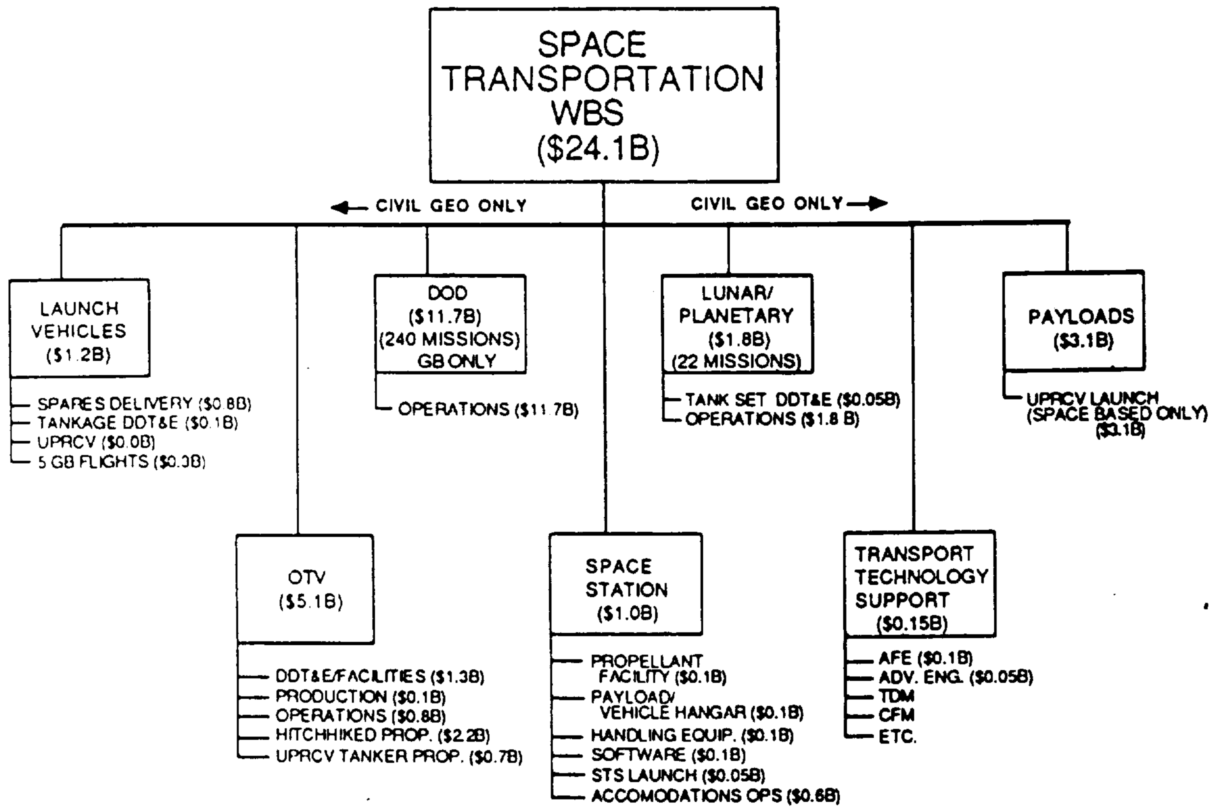

Figure 8.2-1 OTV Program LCC (1985 $B)

Figure 8.2-2 collects the data from the previous figure and highlights the relative OTV program impacts of nonrecurring versus the operations costs of the three respective classes of missions. The OTV acquisition costs represent less than 10% of total LCC while the DOD operations cost estimate is almost 50% of LCC due to including nearly 60% of the 422 missions. The higher operating costs of the more demanding civil GEO and lunar/planetary missions is reflected in their respective percentages of program LCC.

8.2.1 Research and Technology

The R&T costs identified in the study ground rules were included in the OTV program LCC. These costs consist of $53M for the development of the advanced engine technology base and $100M for the aeroassist flight experiment.

8.2.2 DDT&E

The OTV program DDT&E cost estimates include the total nonrecurring costs to develop, integrate and test the OTV ground and space-based capabilities.. In addition, integration and test of OTV and supporting program interfaces (launch vehicle, Space Station, OMV, etc.) are included in Level II systems engineering and integration (SE&I, test operations and program management).

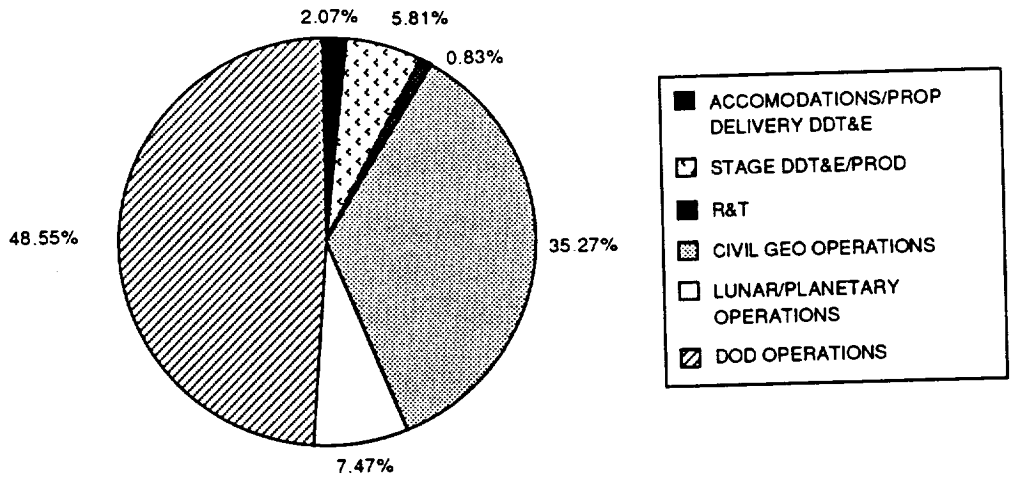

Figure 8.2-2 OTV Costs By Percentage Of Total Program LCC

The initial GBOTV DDT&E cost estimates are based on a new start, clean sheet estimating philosophy. Subsequent SBOTV DDT&E cost estimates are treated as a follow-on evolutionary program to the GBOTV. Launch and manufacturing facility costs are not included in DDT&E and reported separately. The cost estimates for stage GSE, ASE and SSE are included in DDT&E.

8.2.2.1 GBOTV Stage DDT&E

Table 8.2.2-1 shows the DDT&E estimate of the GBOTV, the multiple payload carrier and Level II program costs. The total DDT&E estimate is $1.1B. This includes $0.9B for GBOTV stage and multiple payload carrier, and $0.2B for Level II systems integration costs. A dedicated LCV test operation launch and hardware return is included in Level II estimates.

The stage design and development cost estimate of $442M is dominated by engine, avionics and aeroassist impacts. These subsystems account for over 75% of total engineering. Test hardware includes the production of the dedicated flight test unit as well as stage ground test hardware and two sets of GSE and ASE. This element also includes refurbishment costs of the GVTA/functional test and flight test article for use as operational stages. Stage SE&I and flight software are other significant cost drivers. Total stage DDT&E is $850M.

The multiple payload carrier DDT&E cost estimate of $30.1M is driven by ground and flight test hardware acquisition. In order to support stage test and payload interface requirements, multiple test articles will be manufactured.

The Level II program integration cost estimate is dominated by SE&I ($95M). This effort includes the integration effort of the GBOTV with payload, LCV, return vehicles and ground processing requirements and interfaces.

System test operations costs of $26M include launch vehicle/stage, generic payload and dedicated Pathfinder testing operations. Overall program management costs include support costs incurred to oversee total stage, payload and launch vehicle/ facility integration efforts ($56M). An additional $30M for dedicated flight test launch costs for LCV delivery and STS return ($5.0M) is included. The total Level II integration and test cost estimate is $212M.

Table 8.2.2-1 OTV DDT&E Cost Estimate (1985 $M)

	GB 52K Stage		SB 74K Stage	
Design & Development	$ 442.0		$ 99.0	
Structures		24.8		12.7
Propellant tanks		17.0		12.0
Propulsion Less Engines		12.6		1.9
Main Engine		175.0		3.5
RCS		11.6		4.7
GN&C		81.5		9.0
C&DH		39.4		4.3
Electrical Power		16.6		1.9
Environmental Control		11.7		13.5
Aerobrake		36.8		10.7
GSE		5.2		0.5
ASE		10.3		1.0
SSE		-		22.9
Software	63.0		7.0	
Tooling	27.0		5.0	
Ground & Flight Test Hardware	142.0		31.0	
System Test Ops/Fixtures	27.0		6.0	
Systems Engr. & Integration	101.0		21.0	
Program Management	48.0		10.0	
Subtotal	$ 850.0		$ 179.0	
Multiple Payload Carrier	30.0			
Auxiliary Tankage/ASE			60.0	
Subtotal	$ 880.0		$ 239.0	
Level II Program Costs				
Systems Engr. & Integration	$ 95.0		$ 19.0	
Test Operations	26.0		31.0	
Flight Test Launch	35.0		-	
Program Support	56.0		12.0	
Subtotal	$ 212.0		$ 62.0	
OTV DDT&E Subtotal	1,092.0		301.0	
OTV DDT&E	$1,393.0			

8.2.2.2 SBOTV Stage DDT&E

Table 8.2.2-1 shows the transitional DDT&E cost estimates for the evolutionary SBOTV and associated Level II systems integration costs. These estimates represent the trailing nonrecurring effort in acquiring a SBOTV capability in 1996. The philosophy reflects treating SBOTV DDT&E as a follow-on type program although some efforts of the GBOTV and SBOTV effort are nearly concurrent. The total DDT&E cost estimate is $0.3B. This includes $0.2B for followon SBOTV stage development and auxiliary propellant tankage systems, and $0.1B for Level II. Multiple payload carrier DDT&E is accounted for in the GBOTV DDT&E cost estimate while other space-based related program costs (space-based accommodations, propellant delivery tankage, etc.) are detailed in Section 8.2.2.3).

The SBOTV stage DDT&E cost estimate of $179M reflects the preceding GBOTV development experience. Major cost impacts are transitional engineering, ground test hardware and SE&I. The major hardware and operational requirements behind these impacts include a combination of hardware resizing, subsystem repackaging and space-based integration and test requirements. The primary subsystems impacts occur in structures/tankage, aeroassist and TPS/meteoroid shield. Additional impacts for space support equipment are included.

Auxiliary propellant tankage DDT&E of $60M includes the tanks and associated structure development of tank systems for support of the more demanding lunar and planetary missions. This effort includes development of both a 52 klb and 74 klb tank set.

SBOTV Level II DDT&E consists primarily of the integration and test efforts required due to space-basing. These impacts have been minimized by waiving the requirement for a dedicated test flight. The decision was made based on the potential use of the GBOTV as a test bed for certain onorbit procedures during 1995 operations. The total Level II DDT&E is $62M.

8.2.2.3 Other Related Programs

Table 8.2.2-2 shows the acquisition costs of Space Station accommodations and propellant delivery tankage for the SBOTV. The Space Station accommodations cost estimate of $0.4B includes the following nonrecurring costs required to support the SBOTV: Robotics and imaging hardware; software; frame; hanger and delivery launch cost. This investment provides a SBOTV turnaround facility that is semiautonomous and can be supported by minimum IVA monitoring effort.

Propellant delivery tankage is required for the SBOTV to support two space-based propellant acquisition schemes; propellant delivered via the propellant hitchhiking scheme and propellant delivered via dedicated launch vehicle tanker flight. Tanksets delivered via LCV are expendable while those used with STS/STS II may be recovered. Rough order of magnitude DDT&E based on preliminary design concepts are $20M for hitchhiking tanks and $40M for tanker tankage. Total DDT&E for propellant delivery tankage is $60M.

Table 8.2.2-2 Space Station Accommodations Cost Estimate (1985 $M)

DDT&E & Production	Cost	Comments
Robot Hardware	$ 96M	
- 2 zero-g manipulator arms (6 joint arms with controller)		MMC Robot Arm Study Analogy
- End effectors		" " " " "
- Mobility Fixtures		" " " " "
- A/B & Ground Control Stations		" " " " "
- Offline Programmer Station		" " " " "
Image System - Enhancement/Stereo Vision	30M	" " " " "; adaptation of OMV system
Software	57M	400K lines of code
Hangar	65M	43 ft x 42 ft x 90 ft; one OTV and 55 ft payload
Tank Farm	120M	Equipment List Including Delivery & A&CO; 100K lb capacity
Transportation	50M	LCV Charging Policy
Total Constant Dollars	$418M	

8.2.3 Initial Stage Hardware Production

The recurring production costs for OTV hardware include only the cost estimates for IOC hardware. Subsequent hardware requirements are satisfied by operational spares and reported to operations. Also excluded are refurbishment costs for any DDT&E hardware refurbished to operational units (included in DDT&E), launch costs (included in operations) and two sets of GSE/ASE/SSE (included in DDT&E ground test hardware).

Tables 8.2.3-1 and -2 present unit and initial production cost estimates for the 52K GBOTV and 74K SBOTV stages. The total production cost estimate includes the production of the two SBOTV stages. Initial GBOTV operations stage hardware requirements include one operational stage and one spare (1995), This constraint remains active throughout the operations period and is supplemented as mission rates increase by operations spares. Due to refurbishment of the GVTA and flight test articles from GBOTV DDT&E, the GBOTV has no initial recurring stage production costs. The GBOTV unit cost of $62.4 is shown in Table 8.3.1-1 for comparison with the SBOTV unit cost.

Table 8.2.3-1 GBOTV Unit/Production Cost Estimates (1985 $M)

	Unit		Production
Flight Hardware	$48.0		GVTA & FTA Refurbed to Operational Units
Structures		2.1	
Propellant Tanks		2.7	
MPS (without Engines)		2.8	
Main Engine		6.0	
ACS		2.2	
GN&C		6.0	
C&DH		12.0	
Electrical Power		2.1	
Thermal/Meteor Shield		1.4	
Aerobrake		2.7	
A&CO		8.0	
STE & Tooling	4.8		
Sustaining Engineering	4.8		
SE&I	1.4		
Program Management	3.4		
Total	$62.4		

Table 8.2.3-2 SBOTV Unit/Production Cost Estimates (1985 $M)

	Unit		Production
Flight Hardware	$51.4		$102.8
Structures		2.5	
Propellant Tanks		3.1	
MPS (without Engines)		2.8	
Main Engine		6.0	
ACS		3.0	
GN&C		6.4	
C&DH		12.0	
Electrical Power		2.2	
Thermal/Meteor Shield		1.8	
Aerobrake		3.0	
A&CO		8.6	
STE & Tooling	5.1		10.2
Sustaining Engineering	5.1		10.2
SE&I	1.5		3.0
Program Management	3.6		7.2
Total	$66.7		$133.4

Due to reduced DDT&E requirements for the SBOTV test hardware, the opportunity for DDT&E test hardware refurbishment to operational units was lost. A production cost of $133.4M for two initial SBOTV units is included to meet initial operational hardware requirements. Learning was not applied on the initial two stages, being reserved for operational spares production included in operations. The average unit cost of the two 74K SBOTV stages therefore reflects first unit production estimates.

The total nonrecurring production cost required to meet initial ground and space-based IOC hardware requirements is $135M.

8.2.4 Operations

The OTV program operations cost estimates include all the reusable OTV stage turnaround and hardware related costs, propellant costs, LCV launch of hardware and payloads, Space Station accommodations, onorbit activities and OMV use. Section 8.1.4 details the particular cost components of both ground and space-based servicing of payloads and OTV hardware. The relative operations and cost per flight of four classes of missions are shown in Figure 8.2.4-1. A composite cost per flight for the 422 missions is misleading due to the wide variation in payload characteristics between the DOD, civil GEO and lunar/planetary missions.

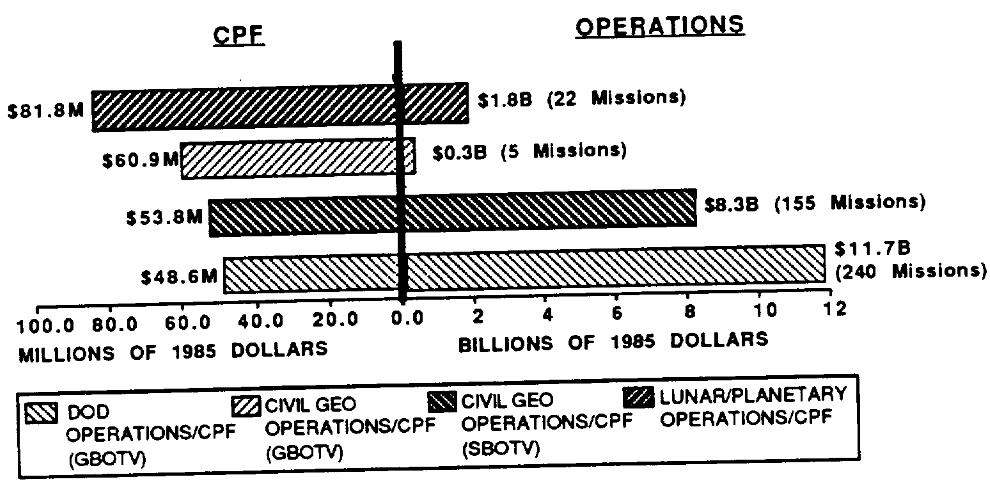

Figure 8.2.4-1 OTV Program Operations/CPF By Mission Type (1985 $B)

The missions classes are ranked in descending order by cost per flight to the left of the center line. To the right of the center line, resulting operations costs and total missions are presented. The 240 ground-based DOD missions show the least CPF ($48M) due to the low average payload weight and resulting propellant and launch cost impacts. These payloads were manifested with stage on the LCV by mass only, thus no volume impacts for launch costs are included. The increased space-based civil GEO mission CPF ($53.8M) is due

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primarily to the increased propellant demands of these payloads. An average increase of greater than 15 klb per mission over DOD payload servicing demands is present. Five GEO missions serviced by the GBOTV in 1995 illustrate the SBOTV savings over ground-basing with respect to the more demanding payloads. These five missions are some of the least demanding civilian payloads, yet CPF exhibits a $7.1M/mission increase ($53.8 vs $60.9M). The lunar/planetary missions have the highest CPF ($81.8M). Additional hardware (via staging and auxiliary tank sets), solid kick stages and propellant demands for the larger missions within this class are the main cost drivers.

Table 8.2.4-1 presents the operations and cost per flight for three of the four mission classes by major operations categories: stage hardware/refurb, mission operations, mission loss, launch/GB return, propellant, Space Station accommodations, space-based payload transportation and program support. Lunar/planetary is not shown because mission requirements are so unique for the 22 missions (individual mission CPF ranged from $51M to $181M per mission). The comparisons to be made from this data include the differences between the two ground-based classes of missions and the space-based vs ground-based civil GEO missions. The most significant difference between the two ground-based missions occurs in launch cost. The DOD payloads averaged less than 10,000 lbs and were manifested only on a weight basis because the Rev. 9 mission model provided no dimensional data. These were also "delivery only" missions. Of the five GEO missions flown ground-based (prior to Space Station availability), four were multiple payload missions at 12 klb. Two of the four were heavily volume constrained. The fifth payload of this group was over 14.5 klb. The combination of additional propellant requirements and volume/length impacts contributed to the $10M/mission launch cost delta. The differences in stage operations and program support is caused by rate impacts on fixed costs and production learning.

Table 8.2.4-1 OTV Operations/CPF (1985 $M)

	GBOTV Civil GEO (5 Missions)		SBOTV Civil GEO (155 Missions)		GBOTV DOD (240 Missions)	
	Operations	CPF	Operations	CPF	Operations	CPF
Stage Operations	35	7.0	466	3.0	1,357	5.6
Mission Operations	3	0.6	40	0.3	40	0.2
Mission Loss	2	0.5	53	0.3	67	0.3
Launch/GB Return (1)	256	51.2	776	5.0	9,888	41.2
Propellant (2)	1	0.1	3,075	19.8	27	0.1
SS Accommodations	-	-	607	3.9	-	-
Payload Transportation/ Processing	1	0.2	3,137	20.2	18	0.1
Program Support	6	1.3	181	1.2	264	1.1
	304	60.9	8,335	53.8	11,661	48.6

(1) Ground-based includes stage, propellant and payload transportation and stage return from LEO; Space-Based includes spares delivery

(2) Includes Ground-based propellant acquisition cost

The differences between ground and space-based operations costs is best seen by first comparing ground-based launch costs against space-based launch, propellant and payload transportation costs and then determining the impacts of other operational elements. The GBOTV launch/return CPF is \$51.2M and includes payload, stage and propellant delivery and inert stage return from LEO via STS. This compares to a space-based CPF of \$45.0M for spares delivery (\$5.0M), propellants (\$19.8M), and payload transportation (\$20.2M). The \$6.2M delta is primarily due to the savings provided by low cost propellant delivery to LEO via propellant hitchhiking combined with the weight/volume penalty of stage hardware delivery of each ground-based mission. This savings could be greater except for the launch cost penalty SBOTV missions incur in 100% volume constrained payload manifesting (see Sections 2.1.2 and 4.9.5.2.3).

The other operational differences between ground-based and space-based missions occur in stage operations and Space Station accommodations costs. GBOTV stage operations costs are higher due to expendable aerobrakes and partially expendable tankage although the delta is reduced by higher SBOTV turnaround costs. The SBOTV accommodations cost delta is self-explanatory.

8.3 TOTAL PROGRAM FUNDING

This section presents the program funding data for the total acquisition and operations cost for the OTV and other related programs. These data will assist in forecasting Phase C/D planning for the OTV program.

The funding streams are first presented for the OTV program without other related program costs. The OTV program funding stream is then merged with the other related program funding streams in order to present a total view of NASA funding impacts pertaining to OTV acquisition and operations. The funding streams include expenditures for all phases of LCC.

8.3.1 Ground Rules and Assumptions

The following ground rules and assumptions were used to develop the OTV and related program funding streams:

A) Program funding is shown for the fiscal year and is based on OTV hardware and facilities schedules (see Section 8.1.3).

B) Annual DDT&E funding was based on historical funding curves with exceptions made for flight test impacts. DDT&E costs include stage, multiple payload carrier and Level II impacts, facilities impacts are included.

C) Reusable hardware production costs include the total production expenditures for two space-based stages. Production costs for the multiple payload carrier is included in OTV DDT&E. Funding was developed to ensure hardware availability at IOC.

D) Operations cost were funded based on the annual mission rate for a particular year for each class of mission.

8.3.2 Selected OTV Program Summary

8.3.2.1 Program Schedule

Figure 8.3.2-1 presents the top level development schedule for OTV and other related program acquisition efforts. The schedule was developed to ensure ground-based operational capability in 1995 and space-based operational capability in 1996.

8.3.2.2 Program Funding

Figure 8.3.2-2 presents the total program funding for the preferred OTV concept LCC. Annual funding levels were developed and are shown by LCC phase for both the ground-based and space-based program costs and four classes of mission operations.

The R&T funding reflects and anticipated spending start in 1988 with the major portion of the costs occurring in 1990 & 1991 due to AFE requirements. Peak funding is $46M.

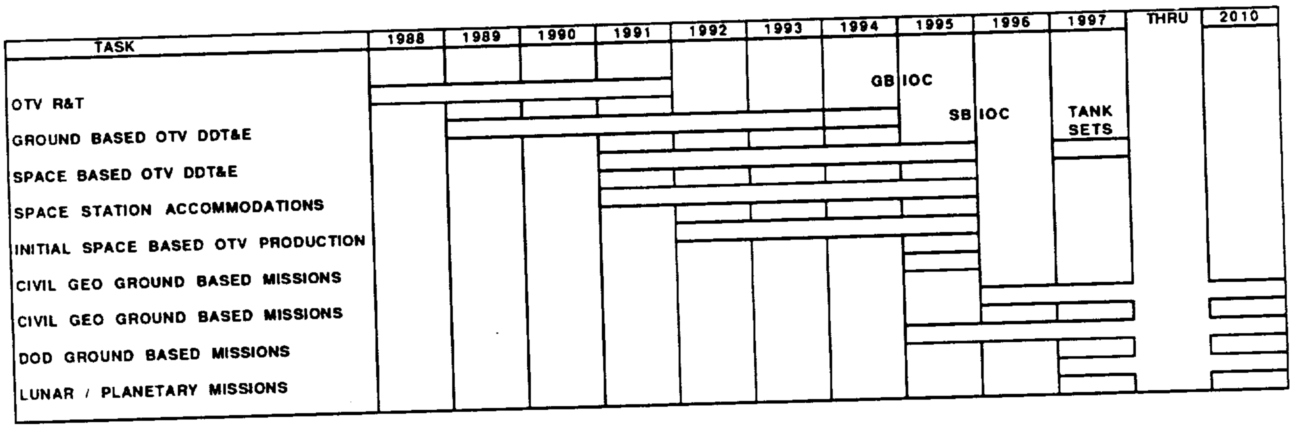

Figure 8.3.2-1 Top Level Development Schedule

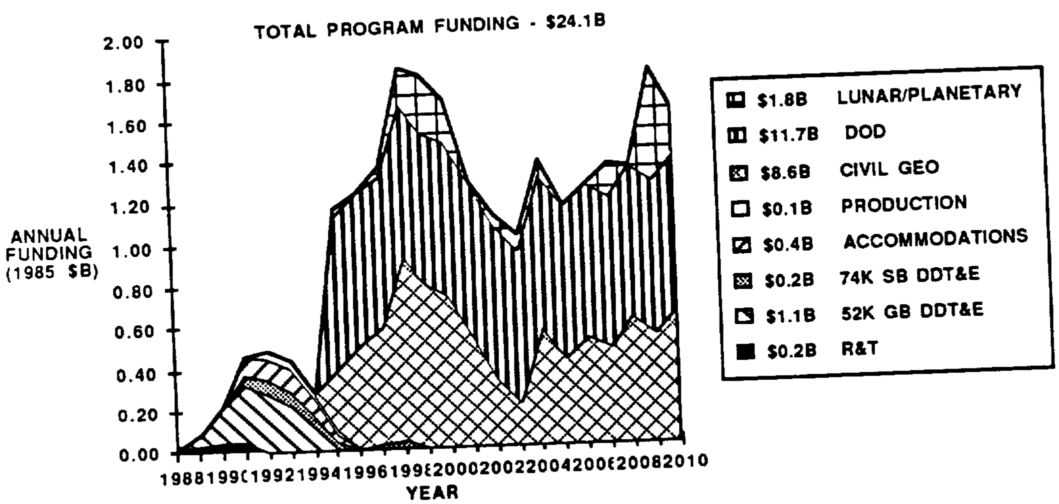

Figure 8.3.2-2 OTV Program Funding

Ground-based DDT&E funding for the 52 Klb GBOTV stage (Figure 8.3.2-3) begins in 1989 culminating at IOC in 1995. Included are the estimate for stage, multiple payload carrier, and Level II DDT&E. Due to the magnitude of scale, facilities costs ($20M) and payload carrier DDT&E ($30M) have been included in ground-based DDT&E. Peak funding for ground-based DDT&E occurs in 1991 and 1992 at $278M.

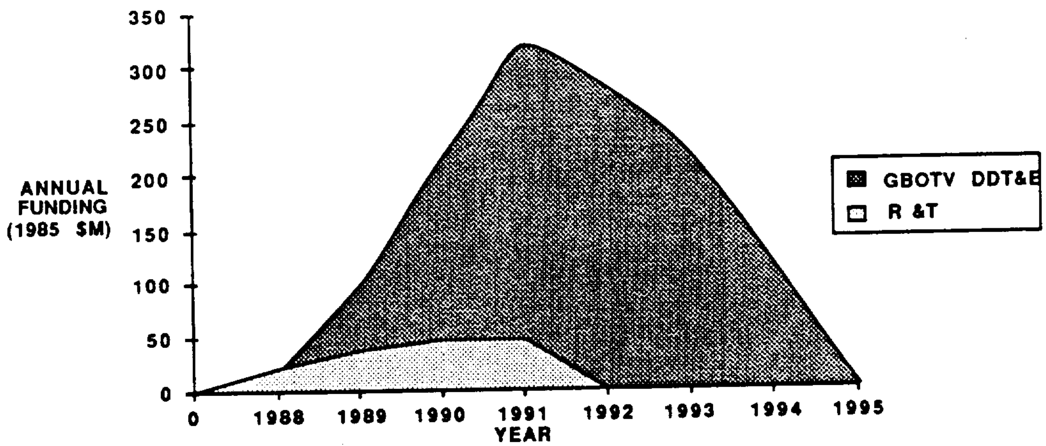

Figure 8.3.2-3 GBOTV Nonrecurring Program Funding

No initial ground-based OTV production is required due to refurbishment of DDT&E test articles.

Ground-based operations costs include OTV specific turnaround costs, and LCV launch costs. The ground-based vehicle with payload was manifested on a single LCV flight therefore these costs include payload transportation.

Annual DOD ground-based OTV operations costs are based on a uniform flight rate of fifteen flights a year for the sixteen year period of operations as specified in the mission model. Peak operation funding reflects the uniform flight rate and remains at a fairly constant level of $0.7B/year. Five GBOTV flights are included in civil GEO operations in 1995.

Space-based DDT&E funding (Figure 8.3.2-4) begins in 1991 culminating at IOC in 1996. Included are the estimates for the space-based stage and Level II DDT&E. Peak funding for space-based DDT&E is $70M in 1993. Concurrent Space Station accommodations acquisition begins in 1991 and culminates in 1995. Peak annual funding is $105M.

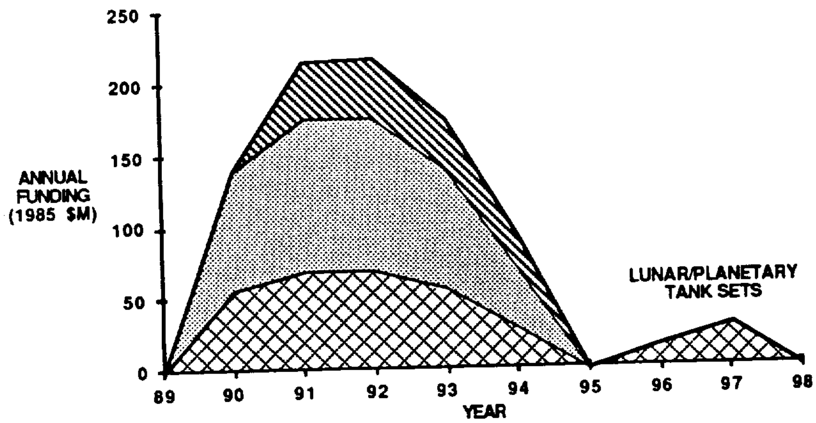

Figure 8.3.2-4 SBOTV Nonrecurring Program Funding

Space-based OTV production includes the cost for manufacturing two operational space-based stages. Funding occurs over the four year period (1992 - 1996) prior to space-based IOC (1996). Peak annual funding is $41M and occurs in 1994.

Annual space-based operations costs are based on the annual flight rate of the scenario II civil GEO mission model. Flight rates vary from four in 2003 to sixteen in 1998. Operations occurs from 1996 through 2010. Peak annual funding for space-based operations occurs in 1998 at $905M.

Lunar and planetary missions occur sporadically throughout the 1995 to 2010 time frame. Peak levels of operations cost occur in 1999 ($280M) and 2009 ($545M).

Peak annual funding for the OTV program occurs in 1998 at $1.8B.