MCR-86-2601
NAS8-36108

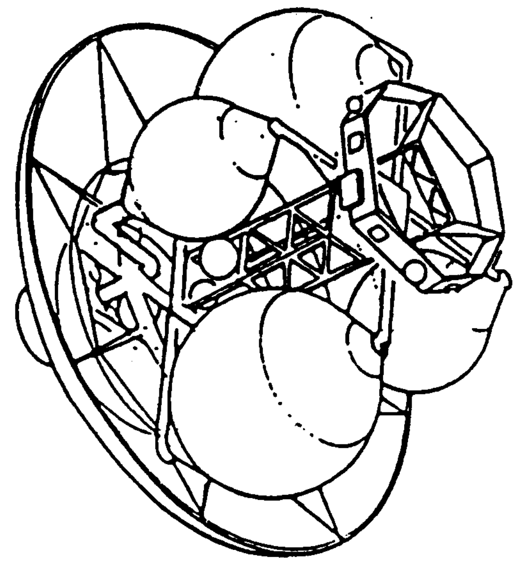
NASA-CR-183552

Volume XI

Study Extension II Results

Orbital Transfer Vehicle Concept Definition and System Analysis Study

1987



183552

(NASA-CR-183552) ORBITAL TRANSFER VEHICLE CONCEPT DEFINITION AND SYSTEMS ANALYSIS STUDY. VOLUME 11: STUDY EXTENSION 2 RESULTS Final Report, Jan. 1987 - Jan. 1988 (Martin Marietta Corp.) 165 p CSCL 22B G3/16

N89-16505

Unclas
0187735

MARTIN MARIETTA

MCR-87-2601
NAS8-36108

ORBITAL TRANSFER VEHICLE
CONCEPT DEFINITION AND SYSTEM ANALYSIS STUDY

VOLUME XI
STUDY EXTENSION II RESULTS

January 1988
Rev 1 - April 1988

Prepared By: W. H. Willcockson
Program Manager

Approved By: for A. C. Wright
Manager, Advanced Programs
NASA Systems

MARTIN MARIETTA
ASTRONAUTICS GROUP
P.O. BOX 179
DENVER, COLORADO 80201

This final report, Volume XI-Study Extension II Results, was prepared by Martin Marietta Astronautics Group for NASA/MSFC in accordance with contract NAS8-36108. The study extension was conducted under the direction of NASA OTV Study Manager, Mr. Donald R. Saxton, during the period from January 1987 to January 1988.

The following personnel were key contributors during this study extension:

Study Manager:	W. H. Willcockson
Denver Engineering Support:	
Propulsion and Design Trades	L. R. Redd
Aerothermal	G. W. Heckel
STS Safety	M. K. Saemisch
Michoud Engineering Support:	
Engineering Manager	W. P. Haese
Cost Analyses	D. R. Callan
Systems Analyses and Mass Properties	A. R. Pagano
Stress Analysis	H. J. Plourde
Environmental Analysis	N. C. Elfer

This report supplements the OTV Phase A Study program results which were presented in Volumes I through X.

Volume	Contents
Volume I	Executive Summary
Volume IA	Executive Summary Supplement
Volume II	OTV Concept Definition and Evaluation
	Book 1 Mission and System Requirements
	Book 2 OTV Concept Definition
	Book 3 Subsystem Trade Studies
	Book 4 Operations
Volume III	System and Program Trades
Volume IV	Space Station Accommodations
Volume V	Work Breakdown Structure and Dictionary
Volume VI	Cost Estimates
Volume VII	Integrated Technology Development Plan
Volume VIII	Environmental Analyses
Volume IX	Study Extension I Results
Volume X	Mars Aerobraking
Volume XI	Study Extension II Results

TABLE OF CONTENTS

VOLUME XI — EXTENSION II STUDY RESULTS

LIST OF FIGURES

LIST OF TABLES

ACRONYMS

ACC	Aft Cargo Carrier
ACS	Attitude Control System
AFE	Aeroassist Flight Experiment
ASE	Airborne Support Equipment
BTU	British Thermal Unit
CNBD	Civil Needs Data Base
CPF	Cost per Flight
CSLI	Civil Space Leadership Initiative
DDT&E	Design, Development, Test & Engineering
DOD	Department of Defense
ET	External Tank
EVA	Extra Vehicular Activity
FSI	Flexible Surface Insulation
GB	Ground-Based
GEO	Geosynchronous Equatorial Orbit
GN&C	Guidance, Navigation and Control
GPS	Global Positioning System
IOC	Initial Operational Capability
IR&D	Independent Research and Development
Isp	Specific Impulse
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
LOI	Lunar Orbit Insertion
LRU	Line Replaceable Unit
MECO	Main Engine Cut-OFF
MLI	Multi-Layer Insulation
MMV	Manned Mars Vehicle
MPS	Main Propulsion System
MSFC	Marshall Space Flight Center
NASA	National Aeronautics and Space Administration
OML	Outer Mold Line
OMV	Orbital Maneuvering Vehicle
OTV	Orbital Transfer Vehicle
PHA	Preliminary Hazards Analysis
PIDA	Payload Installation and Deployment Aid
R&T	Research and Technology
RMS	Remote Manipulator System
RSI	Rigid Surface Insulation
RSS	Root Sum Square
SB	Space-Based
SDV	Shuttle Derived Vehicle
SFE	Smart Front End
SRB	Solid Rocket Booster
SRV	Short Range Vehicle
STS	Space Transportation System

TABI	Tailorable Advanced Blanket Insulation
TEI	Trans-Earth Injection
THIM	Tank Head Idle Mode
TLI	Trans-Lunar Injection
TPS	Thermal Protection System
TVS	Thermodynamic Vent System
W/C_DA	Ballistic Coefficient, pounds/square foot

1.0 INTRODUCTION & SUMMARY

This volume summarizes work conducted in the second extension of the Phase A Orbit Transfer Vehicle Concept Definition and Systems Analysis Study. This study was initiated in 1984 to consider the broad implications and technologies involved with a new advanced upper stage which would represent NASA's workhorse vehicle for orbital transfers in the 1990's and beyond. The initial phase of the study concentrated on a Shuttle-based vehicle delivering primarily geosynchronous payloads. Two vehicle families were derived: ground-based and space-based. These vehicles were all cryogenic, reusable, and aerobraked. The first study extension concentrated on the use of a large cargo vehicle (LCV) for delivering OTV's to low Earth orbit. Here again a ground and space based family was needed with the same general characteristics as before, although somewhat more compact designs were required for LCV packaging.

In this second study extension four major tasks were identified, as follows:

1) Define an initial OTV program consistent with near-term Civil Space Leadership Initiative (CSLI) missions.

2) Develop program evolution to long term advanced missions.

3) Investigate the implications of current STS safety policy on an Aft Cargo Carrier (ACC) based OTV.

4) Expand the analysis of high entry velocity aeroassist.

In general, less emphasis was placed on mission models and life cycle cost analyses than for the previous two phases. An increased emphasis on the breadth of OTV applications was undertaken to show the need for the program on the basis of the expansion of the nation's capabilities in space. Use was made of a driver mission set which specifies various growth options based on the Civil Needs Data Base (CNDB).

Because of uncertainties in the availability of a new large cargo vehicle a program path was derived which utilized the STS for initial flights and then transitioned to the heavy lift booster when available. This path began with an expendable OTV to reduce front end costs and grew in capabilities as required. A Shuttle based system would also give an alternative near-term GEO boost capability for heavy payloads (greater than 5000 lb); thus giving assured access to high energy orbits by matching and exceeding the capability of today's Titan IV. A vehicle compatible with the proposed Shuttle C was also derived, although its subsequent growth path is less clear. Issues involved with the most ambitious application, that of lunar base logistics, were investigated. Although a number of subsystems require modification it appears that a modular OTV can be designed which performs transfers throughout the Earth-Moon space (including lunar landings). A method of utilizing the OTV as the workhorse for buildup and launch of a manned Mars spacecraft was also demonstrated. The use of aeroassist to increase the basic performance of large inclination transfer missions would open polar orbits to visits from low inclination orbital bases.

Safety assessment of the ACC OTV revealed a few new issues but in general confirmed its desirability over the Shuttle cargo bay as the boost location for a new STS cryogenic stage. Finally, a large data base of aeroassisted encounters with the Earth and Mars was developed which should be invaluable in the future to a variety of programs.

The need for such an advanced upper stage is based in the bold new space leadership initiatives that NASA has proposed. Routine round-trip capability beyond low Earth orbit is essential to expanding man's capabilities in the space environment.

2.0 MISSION & PROGRAM OPTIONS

In this second extension of the OTV phase A study, use was made of a discrete driver mission set to define required levels of OTV capability. These missions are derived from the Civil Needs Data Base (CNDB) version 2.0, option 1, with deltas for the Earth, Lunar, and Unmanned Planetary Initiatives. The driver mission set establishes time-points where upgrades in vehicle capabilities are required to accomplish specific mission objectives. Cost analyses were conducted to establish vehicle capability breakpoints where no driving requirements exist. These cost analyses (documented in the Design Analyses section of this report) use a fixed OTV payload size to derive flight rates required to support specific vehicle upgrades.

One of the main drivers for the OTV's future program path is its method of delivery to low Earth orbit (Figure 2.0-1). The availability of a new large cargo vehicle (LCV) by the mid 1990's would almost certainly drive the OTV to be LCV-based exclusively because of the larger lift capability and anticipated lower costs to orbit. The very existence of an LCV, however, would tend to make STS downleg cargo bay space harder to find since many deployed payloads not requiring manned support would be moved off the Shuttle. This would result in LCV-based OTV being either expendable or space based because the LCV gives only a "one-way" ride to orbit.

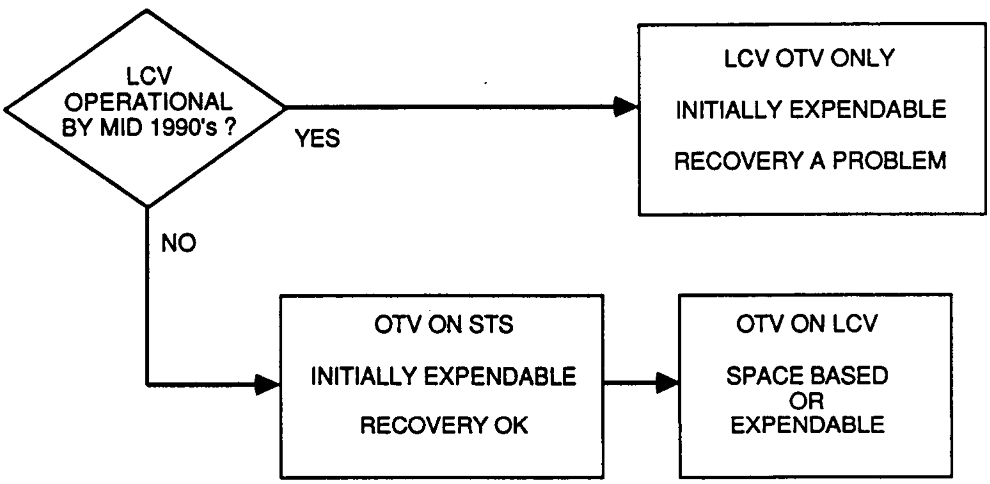


Figure 2.0-1 OTV Boost Options

Two different options currently exist for the LCV's payload bay size. The traditional Shuttle derived vehicle (SDV) approach has a cargo bay diameter of 25 ft. This large size gives an OTV that has good growth characteristics by allowing modular tankage distributed around a structural core (see results of the first study extension contained in Volume IX). More recently, the Shuttle C concept has been proposed with a 15 ft diameter by 60 ft long cargo bay. This LCV would have good lift capability (100klb to low park orbit) but its smaller cargo bay would probably require a more compact stage for volume efficiency.

Growth to higher capability missions would not be as favorable since modular tankage would be more difficult to integrate. A preliminary design for such a Shuttle C OTV is discussed further in the Design Analyses portion of this report (section 4.0).

Another option is presented if the LCV is not operational by the mid 1990's. Rather than delay the startup of a new upper stage, the Shuttle could be used as the delivery vehicle until the LCV is available. Because the Shuttle's cargo bay would be freed up by the deployment of the OTV/payload combined stack, it would be available for the return of a reusable OTV to the ground. The use of the aft cargo carrier (ACC) would allow a wide diameter (up to 27') OTV having good growth capabilities. When the large cargo vehicle became available the OTV could transition over to it. In an expendable mode, such a vehicle could deliver 12.5klb of payload to geosynchronous orbit with a 55klb (to 110 nm) payload capacity Shuttle. This approach would also give two paths to orbit for the OTV: the Shuttle and the LCV as well as filling the gap in the Shuttle's GEO performance brought on by the cancellation of the STS/Centaur.

Thus two OTV options were concentrated on: an LCV-only vehicle and one that begins on the Shuttle and then transitions to the LCV. The initial starting point for the program was chosen to be a near term low-technology expendable vehicle to reduce the program's front-end costs. Such an approach allows the incremental incorporation of vehicle improvements over time in block modifications required by more advanced missons. It was felt that the earliest date that such a vehicle could be made available would be 1993 and so a vehicle with this IOC date was used as a starting point. Later startup dates allow cost-effective incorporation of new technologies into the initial vehicle. This issue is discussed further in the Design Analyses section (4.0).

2.1 DRIVER MISSIONS

Figure 2.1-1 summarizes the driver mission set used in this study. These represent discrete driver missions and not a total mission model. The baseline scenario is derived from NASA's Civil Needs Data Base (CNDB) version 2, option #1. This represents a conservative growth plan with a total of about 16 OTV-class missions spread over a 15 year time frame. This baseline scenario is used by itself and as a core with additions from the three new Civil Space Initiatives. These more aggresive growth options are: 1) Earth Initiative, 2) Unmanned Planetary Initiative, and 3) Lunar Initiative. This gives a total of four driver mission sets.

	1996	1997	1998	1999	2000	2001	2005	2006	2008	2010
BASELINE SCENARIO 1	10K GEO 8.8K PLAN (C3=32)		12/2K GEO					13.2K GEO		22K GEO
EARTH INITIATIVE	25K GEO					16.5/9.5K GEO				
UNMANNED PLANETARY INITIATIVE	21K PLAN (C3=10)		9.9K PLAN (C3=110)							
LUNAR INITIATIVE	8.8K ORB	2.2K SURF			15K SURF (MANNED) 38.5K SURF		40K SURF		40K ORB	80K SURF

OPTION # 1: BASELINE SCENARIO 1

OPTION # 2: BASELINE SCENARIO 1 + EARTH INITIATIVE

OPTION # 3: BASELINE SCENARIO 1 + UNMANNED PLANETARY

OPTION # 4: BASELINE SCENARIO 1 + LUNAR INITIATIVE

Figure 2.1-1 Driver Missions

2.1.1 MISSION CAPTURE - BASELINE MISSION SET

Table 2.1.1-1 lists driver missions for the baseline mission set along with their required propellant quantities for ACC expendable and reuseable vehicles. Also shown are resulting STS and LCV lift requirements for performing the

Table 2.1.1-1 Mission Capture - Baseline Scenario

VEHICLE	PROPELLANT REQUIREMENTS FOR DRIVER MISSIONS				
	GEO DELIVERY 10 K P/L 1996	PLANETARY 8.8 K, C3 = 28-32 1996	MULT. P/L DELIV. 12 K UP/2 K DN 1998	GEO DELIVERY 13.2 K P/L 2006	GEO PLATFORM 22 K P/L 2010
STS LAUNCH, EXPENDABLE, RL10A-3 ENG.	28 KLBM 47.3 K STS LIFT 43.9 K LCV LIFT	27 KLBM 45.3 K STS LIFT 41.9 K LCV LIFT	31.4 KLBM RACK EXPENDED 52.9 K STS LIFT 49.6 K LCV LIFT	33.5 KLBM 56.3 K STS LIFT 53 K LCV LIFT	49.6 KLBM 59.1 K STS LOAD (FOR OTV ONLY) 78.1 K LCV LIFT
STS LAUNCH, AEROBRAKE, REUSABLE, IOC ENGINE	36 KLBM 57.9 K STS LIFT	45 KLBM 65.7 K STS LIFT	42 KLBM 65.9 K STS LIFT	41 KLBM 66.1 K STS LIFT	56 KLBM * 67.9 STS LIFT FOR OTV ONLY

* DEGRADED Isp FOR LOW THRUST OPERATION

missions either in a single or dual launch mode as noted. The expendable vehicle propellant quantities are with respect to use of a RL10A-3 engine (existing Centaur motor at 440 sec ISP); the aeroassisted reusable vehicle concept uses the IOC advanced cryogenic engine (475 sec) which was described in Phase 1 of this study.

The net lift requirements for the STS are identified in the figure and include the weight of the ACC as well as ASE. For the LCV lift requirements only the OTV ASE is included. All the missions were performed with a single launch where possible. Where a dual launch was required (the 22klb GEO platform delivery) the lift requirements for the OTV only are noted.

All the missions can be performed by a 50klb propellant capacity expendable OTV with an RL-10 engine. If the Shuttle is used as the launch vehicle a lift capacity of 59klb (to 110 nm) is required for the 22klb GEO platform delivery mission. This mission must be performed in split fashion with one STS launch carrying the payload and the other carrying the fully fueled OTV. If a reusable OTV is employed, the maximum Shuttle lift requirement is 68klb. If a large cargo vehicle is used as the launch vehicle all missions can be flown intact with a maximum lift capacity of 78.1 klb.

2.1.2 PROGRAM IMPROVEMENTS AND MISSION CAPTURE

Propellant quantities are given in Table 2.1.2-1 for each of the three advanced space initiatives. Program improvements are required in order to accommodate the various civil space initiatives (such as increased propellant capacity, manrating, lunar landing legs,etc.). All propellant quantities are with respect to IOC engine (475 sec) usage unless otherwise noted. Where indicated, the currently planned 55klb STS lift capability (to 110 nm park orbit) is adequate to support a given mission.

Table 2.1.2-1 Mission Capture - Growth Missions

PROGRAM IMPROVEMENT	PROPELLANT REQUIREMENTS FOR DRIVER MISSIONS				
	EARTH INITIATIVE		PLANETARY	LUNAR INITIATIVE	
	25K TO GEO LOW G (0.1)	GEO SERVICING	UNMANNED PLANETARY	LUNAR ORBIT	LUNAR SURFACE
IOC ENGINE (STS LAUNCH) EXPENDABLE	50.4 KLBM 62 K STS LIFT FOR OTV ONLY		10 K P/L, C3 = 80 45 KLBM* 56 K STS LIFT FOR OTV ONLY		
AEROBRAKE/ REUSABILITY (STS LAUNCH)		8.7K UP, 7.9 K DN 45 KLBM 59 K STS LIFT FOR OTV ONLY	21 K, C3 = 10 40 KLBM 55 K STS OK	8.8 K UNMANNED 33 KLBM 55 K STS OK	
LARGE OTV/ MANRATED (SPACE BASED)		16.5 K UP/9.5 K DN 68 KLBM		40 K UNMANNED 94 KLBM	
LUNAR LANDING (4 ENGINES, LANDING LEGS, RADAR)					15 K MANNED 2 @ 85 KLBM 40 K UNMANNED 2 @ 98 KLBM

*NOTE: USING AN RL-10 ENGINE WITH THIS OPTION RESULTS IN 50 KLBM PROPELLANT USAGE

2.2 OTV PHASED GROWTH

Beginning with a relatively modest expendable vehicle the OTV can grow its capabilities as required by the missions planned. Four different scenarios were considered in this study giving dramaticly differing results.

2.2.1 GROWTH PATH DEVELOPMENT PROGRAMS

OTV evolution will consist of vehicle improvements over a period of time as the missions demand and as certain technologies become available. Table 2.2.1-1 lists the possible vehicle improvements in the logical order of evolution and mission need to assess the grouping of subsystem updates. Grouping of these is essential in order to minimize program evolution costs and schedule impacts. This type of "grouping" is intended to minimize test hardware/operations, qualification paperwork, test article and production retooling, demonstration missions, design duplicity, etc. As the vehicle improvements progress, the

Table 2.2.1-1 Phased Growth - Subsystem Groupings

VEHICLE IMPROVEMENTS	AFFECTED SUBSYSTEMS AND IMPACTS				
	AVIONICS	STRUCTURE	TANKAGE	PROPULSION	AEROBRAKE
IOC ENGINE	ENGINE CTRL., TVC,	NEW I/F	PRESSUR. I/F	PROP. ACQ. AND FEED	N/A
2 IOC ENGINES	ENGINE CTRL., TVC, ENG OUT	NEW TRUSS	PRESSUR. I/F	PROP. ACQ. AND FEED	N/A
REUSE	HEALTH MONITORING	FATIGUE TESTING	METEOR., ORU, PRESS. CYCLES	COMPONENT LIFE, ORU'S	N/A
AEROASSIST	GUIDANCE AND CTRL.	AEROBRAKE SUPPORT	INSULATION	RCS THRUSTER #, LOCATION,	INSTALL
LARGE OTV	P.U. SYSTEM, CTRL. SOFT.	NEW OR MODIFIED	NEW LARGER TANKS, P.U.	PROP.ACQ. AND FEED, RCS	LARGER AEROBRAKE
MANRATING	REDUNDANCY, FUEL CELLS	SAFETY FACTORS	METEOROID	REDUNDANCY	LARGER AEROBRAKE
SPACE BASING	MODULAR ORU'S	MODULAR ATTACHMENTS	MODULAR ORU'S	MODULAR ORU'S	DETACHABLE AEROBRAKE
LUNAR LANDING	GUIDANCE & CTRL., RADAR	LANDING LEGS	METEOROID	CONTIN. THROT., ADD ENGINES	LANDING LEG COMPATIBLE

overlap in subsystem development groups help tie the program together into a smooth evolution of continuing enhancements in OTV capabilities. The result of these groupings is that definite "block" changes apply to the evolution of the OTV program and that each subsystem does not have to evolve in small independant steps on its own. Therefore, a vehicle program that provides a range of vehicle improvements can be achieved with a minimum of time and energy spent on incorporating these block changes. For example, when evolving to a large size

OTV the developments required to make the vehicle man-rated and space based should just as well be done all at the same time.

2.2.2 OTV PHASED GROWTH - BASE SCENARIO

Figure 2.2.2-1 summarizes the OTV growth plan for the baseline scenario (Civil Needs Data Base, Version 2, Scenario 1). This scenario is a low growth option with annual OTV flight rates of only one to two per year. The OTV program begins as early as 1993 (earliest initial capability) with a low-technology, low-cost expendable vehicle. As will be discussed in the Design Analyses section, flight rates of 5 to 6 per year are required to justify major system upgrades such as IOC engine and aeroassist. This baseline scenario has requirements for only one to two OTV-class missions per year. These low flight rates do not justify major OTV program improvements and the OTV would remain an expendable vehicle.

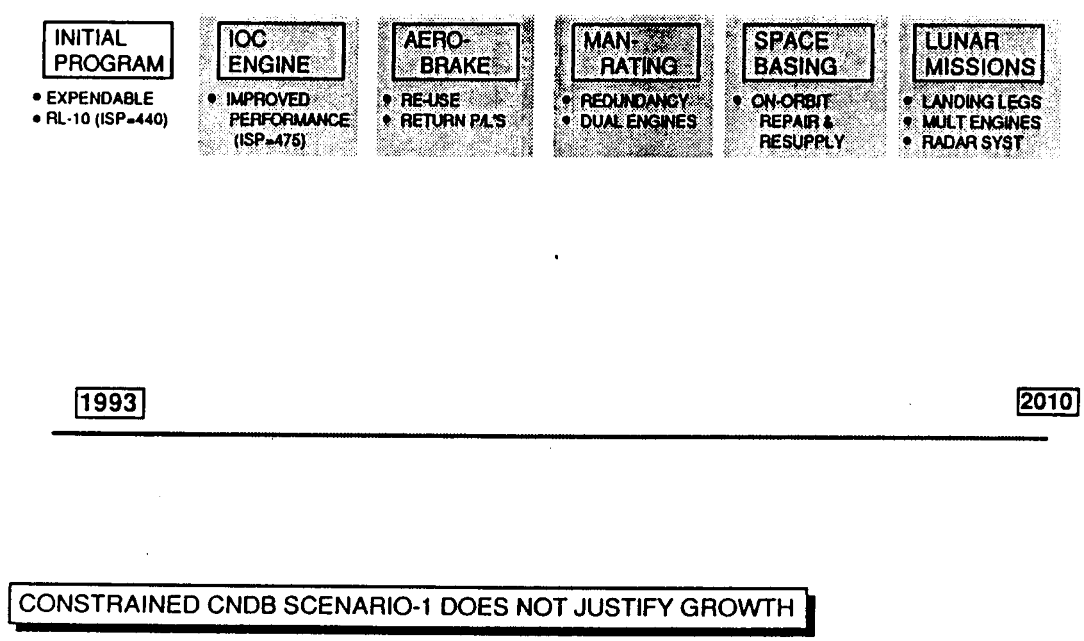


Figure 2.2.2-1 OTV Phased Growth - Baseline Scenario

2.2.3 OTV PHASED GROWTH - EARTH INITIATIVE

Figure 2.2.3-1 summarizes the OTV growth plan for the CSLI Earth Initiative. This initiative contains low-g large platform deployment as well as round-trip GEO-servicing missions. These missions present specific requirements for the OTV which drive hardware development. The first large GEO platform deployment occurs in 1996. Because it is a low-g delivery requiring low thrust capability, the OTV IOC engine must be used, rather than the RL-10. Since aeroassist will be required by 2001 for the GEO servicing mission, and since it is more cost-effective to group IOC and aeroassist block changes together, these modifications are both implemented in 1996. The actual platform deploy mission must use an expendable OTV because of the demanding propellant requirements.

As defined, the baseline GEO servicing mission in 2001 can only be accomplished by a large space-based OTV since it requires in excess of 68klb of propellant on an aerobraked vehicle. This would require OTV space basing capability in 2001, which is probably about the earliest date that it could be available. Two alternate options for this servicing mission were looked at to reduce OTV requirements: splitting the mission and expending the servicer.

In the split mission option, the servicer is delivered by one OTV mission and retrieved by a second one. This reduces propellant capacity requirements of the OTV to 34.2klb (34.2klb to deliver the 16.5klb servicer with an expendable OTV, 33.7klb to retrieve with an aerobraked reusable vehicle). With this split mission option, the need for space basing is eliminated since the smaller OTV can be delivered to orbit fully fueled with a single launch.

In the expended servicer option, the peak propellant requirement of 34.2 klb does not change but the elimination of servicer return means that aeroassist is not needed for recovery. The Earth Initiative model does not contain enough missions to otherwise justify reusability on an economic basis so if this mission requirement drops out so will reuse. This would eliminate efficient round trip missions, however, which may be inconsistent with more general requirements of an expanded near-Earth capability (for example, the retrieval of finished products from manufacturing facilities in solar-synchronous orbits).

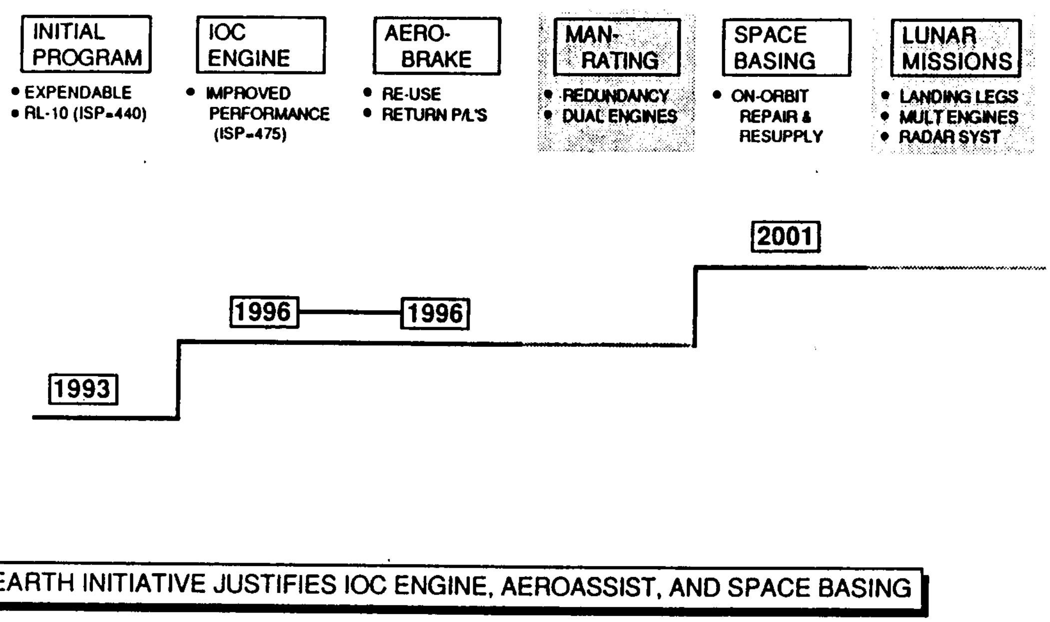


Figure 2.2.3-1 OTV Phased Growth - Earth Initiative

Thus three different development paths are possible for the Earth Initiative. The reference case shown in Figure 2.2.3-1 assumes that the servicer mission is flown as defined in the driver mission set. A single OTV delivers and retrieves the servicer in a single flight which requires the IOC engine, aerobraking capability, and space basing of a 68klb capacity stage. With a split servicing mission a single, smaller vehicle (34.2 klb capacity) would be adequate which would only require the IOC engine and aerobraking capability (no space basing). Finally, if the servicer is not retrieved, the Earth Initiative would require only the development of the advanced IOC engine.

2.2.4 OTV PHASED GROWTH - UNMANNED PLANETARY INITIATIVE

Figure 2.2.4-1 summarizes the OTV growth plan for the Unmanned Planetary Initiative. This initiative does not add a significant number of missions to the base scenario and so is still a low flight rate model. The only driver mission is the 10klb Cassini mission in 1998 which requires a C_3 of 80-110 km^2/sec^2. The C_3 of 80 km^2/sec^2 can be accommodated by a 50klb propellant capacity OTV in an expendable mode using the RL-10 engine. This vehicle requires a net Shuttle lift capability (to 110 nm) of 60klb. If an IOC engine (ISP=475 sec) is utilized with this propellant capacity, a C_3 of about 90 km^2/sec^2 can be achieved instead. If a large cargo vehicle is employed to deliver a 62klb propellant capacity OTV, the full 110 C_3 can be accommodated.

In any event, there is no driving reason, either from a flight rate or requirements standpoint, to add further program improvements. Thus the expendable OTV is the only vehicle required for this initiative.

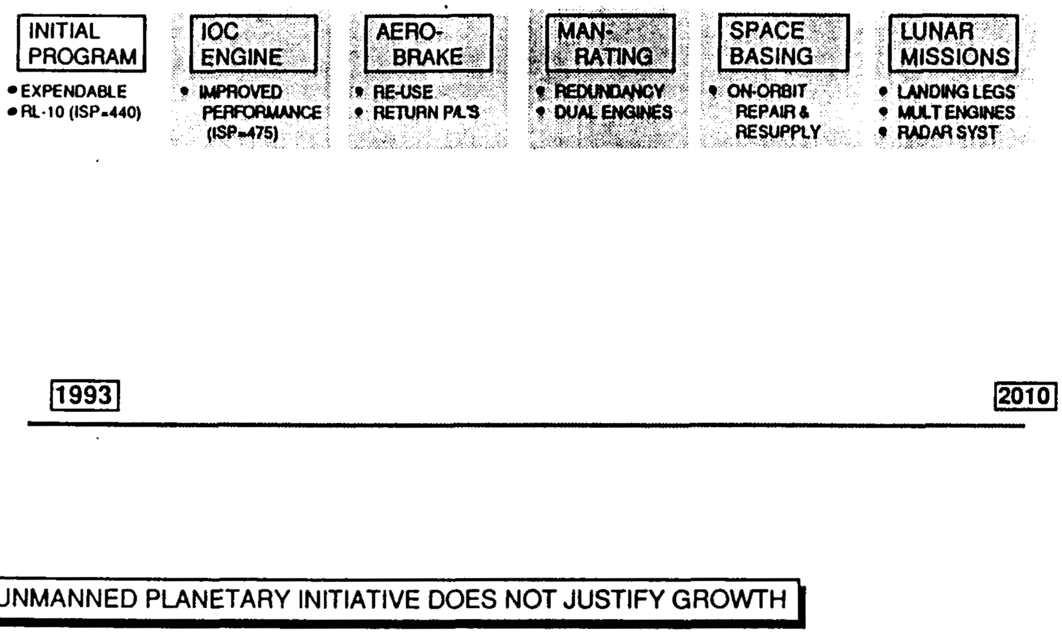


Figure 2.2.4-1 OTV Phased Growth - Unmanned Planetary Initiative

2.2.5 OTV PHASED GROWTH - LUNAR INITIATIVE

The Lunar Initiative has large flight rates and payload sizes which makes it the most demanding of the identified initiatives. The OTV growth plan for this initiative is shown in Figure 2.2.5-1. High traffic rates beginning in the year 2000 will more than justify IOC engine and re-use technology from a cost standpoint. From a requirements standpoint, the round trip manned mission requires man-rating and aeroassist while the 40klb surface delivery mission demands a large propellant capacity reusable stage (98klb) which must be space based. Additionally, landing on the moon requires significant upgrade of OTV subsystems (landing legs, engines, avionics, etc) as is spelled out later in the Design Issues section.

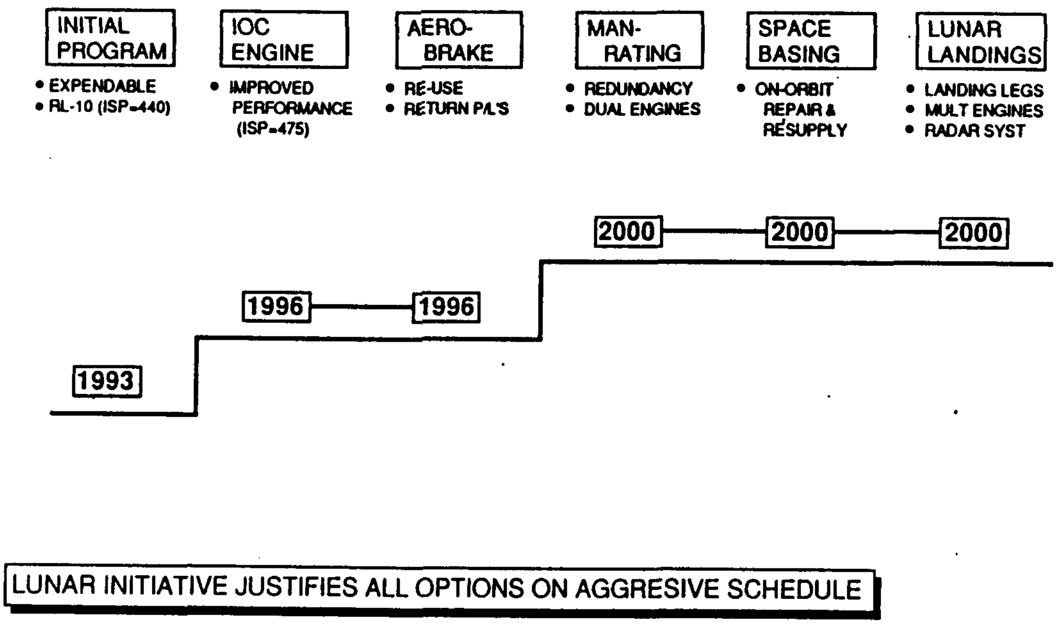


Figure 2.2.5-1 OTV Phased Growth - Lunar Initiative

Thus the Lunar Initiative requires the full range of OTV improvements as is indicated in the chart. Man-rating, space basing and landing capability are all required in 2000 to support both the 15klb round-trip manned mission as well as the 38.5klb delivery mission. This sets a firm date for completion of the program upgrades at the year 2000. It is felt that in the case of the Lunar Initiative, the IOC engine and aeroassist upgrades should be attempted earlier in the schedule to avoid flying too many improvements at once. A reasonable date for achieving these upgrades is 1996 which then allows a four year growth to the ultimate vehicle capability. A small landing mission in 1997 could be accomplished by a ground based 50klb capacity OTV in an expendable mode.

2.2.6 OTV GROWTH SUMMARY

Figure 2.2.6-1 summarizes the OTV growth paths of each of the four mission options covered in this study. Two of the mission scenarios (the baseline set and Unmanned Planetary Initiative) do not require anything more than a 50klb propellant capacity expendable stage. The Earth Initiative mission set, strictly interpreted, requires an IOC engine, aeroassist, and space basing. Redefinition of the unmanned servicer mission in this set could reduce new development down to only an IOC engine, however. The very aggressive Manned Lunar Initiative drives full development of OTV systems including new engines, re-use, space basing, man-rating, and landing capability.

<u>BASE SCENARIO 1</u>

LOW TRAFFIC

EXPENDABLE OTV ONLY

<u>EARTH INITIATIVE</u>

MODERATE TRAFFIC, ROUND TRIP REQUIREMENT

DEVELOP IOC ENGINE & AEROASSIST

<u>UNMANNED PLANETARY</u>

LOW TRAFFIC

EXPENDABLE OTV ONLY

<u>LUNAR INITIATIVE</u>

HIGH TRAFFIC, ROUND TRIP & LANDING REQUIREMENTS

FULL DEVELOPMENT PROGRAM

Figure 2.2.6-1 OTV Growth Summary

2.3 GEO SERVICING OPTIONS FOR OTV

An investigation was conducted into alternate definitions of the unmanned servicing mission using current mass and performance of the orbital maneuvering vehicle (OMV). This gives somewhat different performance figures from that used in the driver mission set discussed in section 2.2.3. Two cases for GEO servicing were investigated for OTV as is illustrated in Figure 2.3-1. The first was to deliver line replaceable units (LRU's) to GEO along with an OMV short range vehicle (SRV) which would separate from the OTV, perform the servicing maneuvers and operations, then rejoin the OTV to be returned to LEO. The maximum delta-V that can be accomodated by the SRV's monopropellant system while separated from the OTV in such a scenario is about 500 fps. The figure shows the weights of the various parts of the stack along with the propellant amounts used by the SRV and OTV.

The second case is for a higher on-orbit servicing delta V than the SRV can accommodate by itself. A working figure from previous mission models is about 800 fps. The options studied include the use of either a complete OMV (SRV with bi-prop module) for the GEO servicing, or the OTV modified for performing the servicing maneuvers on its own. The option using an OMV requires two OTV flights in order to deliver the OMV and LRU's in the first flight and then to retrieve them in the second flight with a 50klb propellant capacity vehicle. The first flight in this option would also correspond to an expended servicer mission where the second flight would not be neccesary.

The second option requires only one flight with the OTV performing the on-orbit maneuvers. Modifications would have to be made to the basic OTV system to incorporate SRV command links as well as cold gas capability. It appears that if the OTV can be made capable of performing the on-orbit maneuvers at reasonable weight impact and development cost, the second option may be worth pursuing.

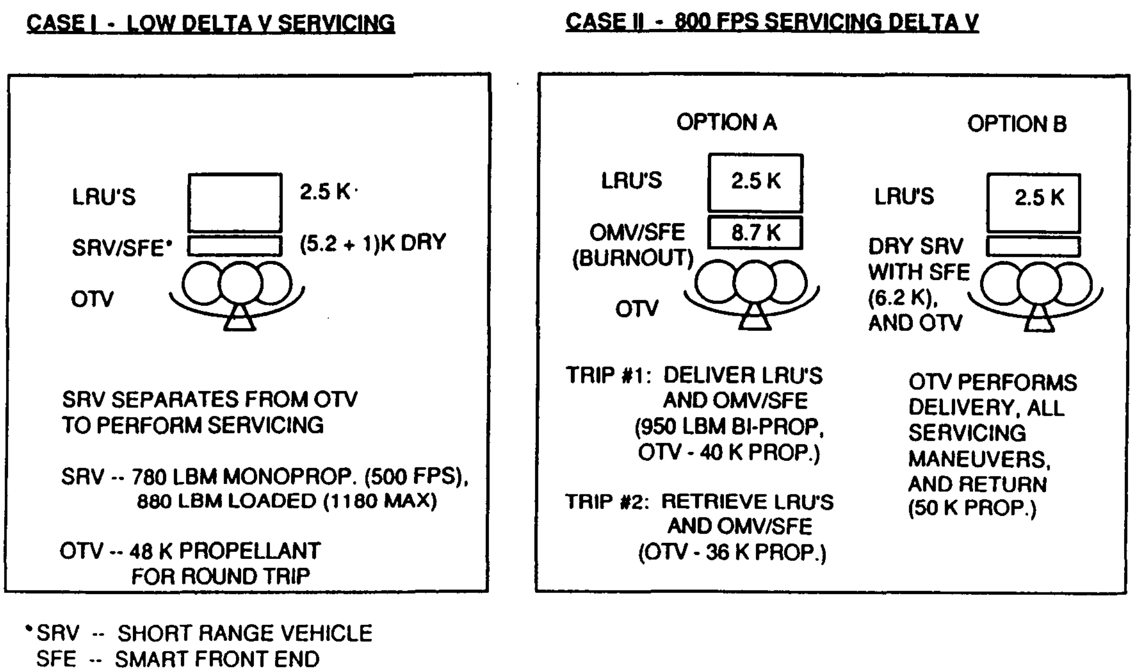


Figure 2.3-1 GEO Servicing Options

The OMV SRV can thus be used as a rendezvous capability kit, however the most weight-efficient package would be a totally redesigned front end for the OTV. This would also be greatly more expensive to implement. In general, the ability to perform rendezvous and docking along with a return capability opens up a variety of missions to the OTV. Besides the already identified satellite servicing; such missions as large space structure assembly, space manufacturing product retrieval from high energy/inclination orbits, satellite inspection, and manned Mars vehicle assembly open new possibilities for space transportation infrastructure.

2.4 SPACE BASE OPTIONS

Figure 2.4-1 shows some of the options for OTV space basing. With the recent problems that Space Station has encountered there are significant questions as to whether the OTV can be based on that facility. As was detailed in the Phase A and Extension #1 studies the Space Station is the most desireable basing location from a program standpoint. It provides a stable base and power supply along with ready access by a servicing crew, as well as being a centralized node for traffic flow.

1) SPACE STATION BASE

- HANGAR ATTACHED TO SPACE STATION
- EASIEST MANNED ACCESS
- COVERED IN PHASE A REPORT

2) FREE FLYER BASE - COORBITAL WITH SPACE STATION

- 30+ MILE SEP FROM STATION, FORMATION FLYING
- MANNED ACCESS VIA OMV AND / OR SHUTTLE
- PROPELLANT SCAVENGING / HITCHIKING - ALL FLIGHTS TO STATION

3) FREE FLYER DECOUPLED FROM SPACE STATION

- NOT DEPENDENT ON SPACE STATION
- MANNED ACCESS VIA SHUTTLE ONLY
- MINIMAL NON-OTV SCAVENGING / HITCHIKING AVAILABLE

4) SPACE TENDED MINIMAL BASE

- SERVICING DIRECTLY FROM SHUTTLE (AVIONICS ONLY)
- DEDICATED EXPENDABLE TANKERS AND / OR STS ACC
- EARLY CAPABILITY OPTION

Figure 2.4-1 Space Base Options

If it is not possible to base the OTV at the Space Station the next best alternative is to deploy a free-flying hangar co-orbital with the Station. A concept layout for such a facility was shown in the Phase A Accomodations report. Because of its proximity to the Station, unplanned servicing calls are still possible, however they are more difficult and require either an OMV + Crew Module or a Shuttle to accomplish. All booster flights to the vicinity of the Station have potential for propellant scavenging and/or hitchiking. The free-flyer obviously has to supply its own attitude control and power.

Along this same line, a free-flyer that is remote from the Station has all the disadvantages of a co-orbiter and few benefits. It does relieve the Station of any support role but manned access and propellant resupply become much more difficult.

Where such a de-coupled free-flyer might be attractive is in a minimum capability space base that is tended by the Shuttle (Figure 2.4-2). This would represent a low-cost approach to space basing. Only limited servicing would be possible, probably avionics changeout only. Propellant re-supply and tanking operations could be performed using unmanned vehicles only (LCV for tankage boost and a dedicated OMV for retrieval and berthing), thus minimizing propellant-handling safety concerns. Problems with this approach include lower operational life for the OTV because of unservicable component failures as well as more limited support for advanced missions.

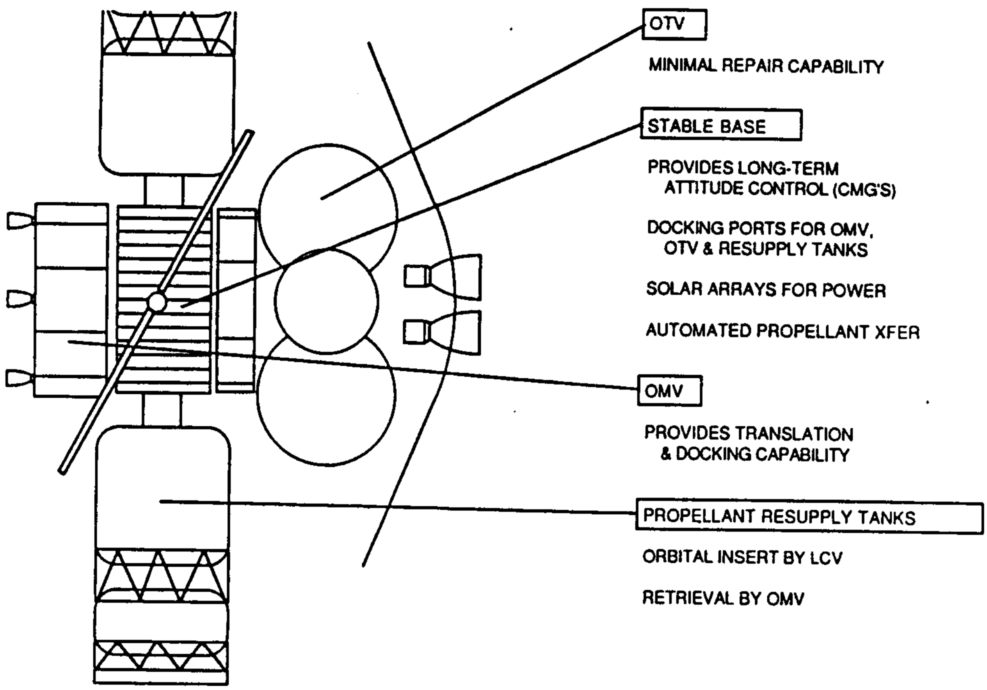


Figure 2.4-2 Minimum Space Base

These last options represent work-arounds to reduced Space Station support capabilities. However they probably represent higher cost options overall than the Station base laid out in earlier portions of this Phase A study.

2.5 LARGE INCLINATION TURNS VIA AEROASSIST

The fact that the OTV has aerobraking capability can be used to improve the performance of missions requiring large plane changes. To achieve a large plane change propulsively requires three burns, in general. The technique is to raise the apogee of the orbit to a sufficiently high altitude where the orbital velocity is low and can easily be changed in direction. This approach uses burn #1 to raise the apogee (as well as performing a small amount of plane change), burn #2 performs the majority of the plane change at apogee, and burn #3 (at perigee) reduces the orbit back to low circular again. The higher the altitude of apogee the better from a performance standpoint, but due to operational considerations it should be limited to 20,000 to 30,000 nm.

With the availability of aeroassist, this same technique can be improved upon by substituting an apogee reducing aeromaneuver for the third burn as is shown in Figure 2.5-1. The same strategy as before is employed for the first burn in raising the apogee, the second burn performs the plane change as well as setting up the perigee targeting for aerobraking. Upon returning to perigee the aero-maneuver reduces the velocity of the vehicle to that required for the final orbit. It must be stessed here that the aeroassist is only used for apogee reduction, no aerodynamic plane change is performed. Because of the heating levels encountered in an aeroassist maneuver, sensitive payloads may require a thermal shroud. A small circularization burn is performed after leaving the atmosphere, typically 250-450 fps depending on the final altitude desired.

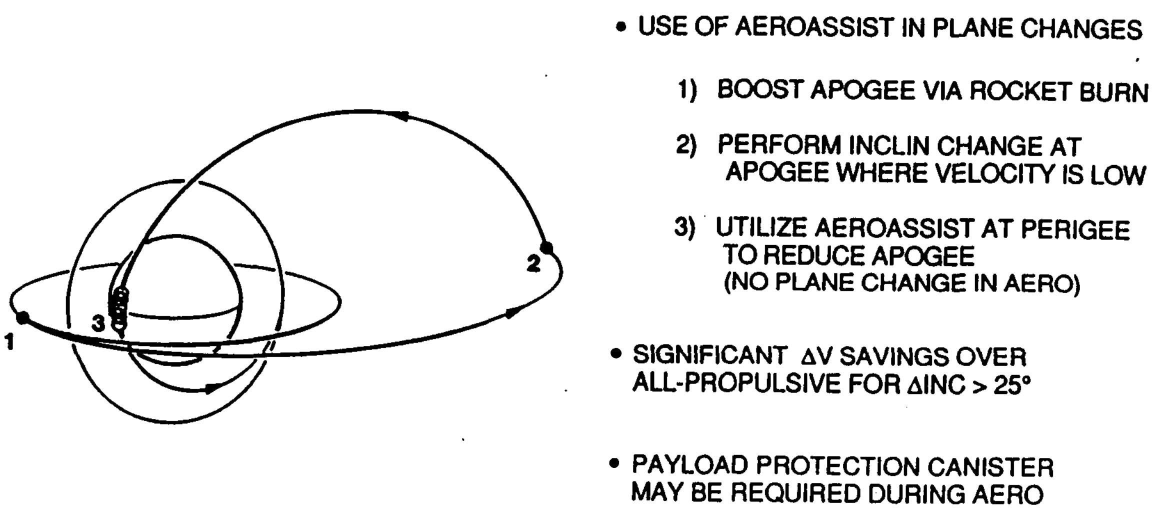


Figure 2.5-1 Large Inclination Change Via Aeroassist

Figure 2.5-2 shows the results of performance comparisons between an optimized all-propulsive plane change and one employing aeroassist. The initial and final orbit is 270 nm circular. The size of the plane change was varied between 0° and 90°. The maximum altitude of apogee was limited to 20,000 nm. It may be seen that for plane changes greater than 25° aeroassist shows significant ΔV savings over the all-propulsive approach. Below 25° it is more efficient to stay with the all-propulsive approach because the intermediate apogee altitude is low.

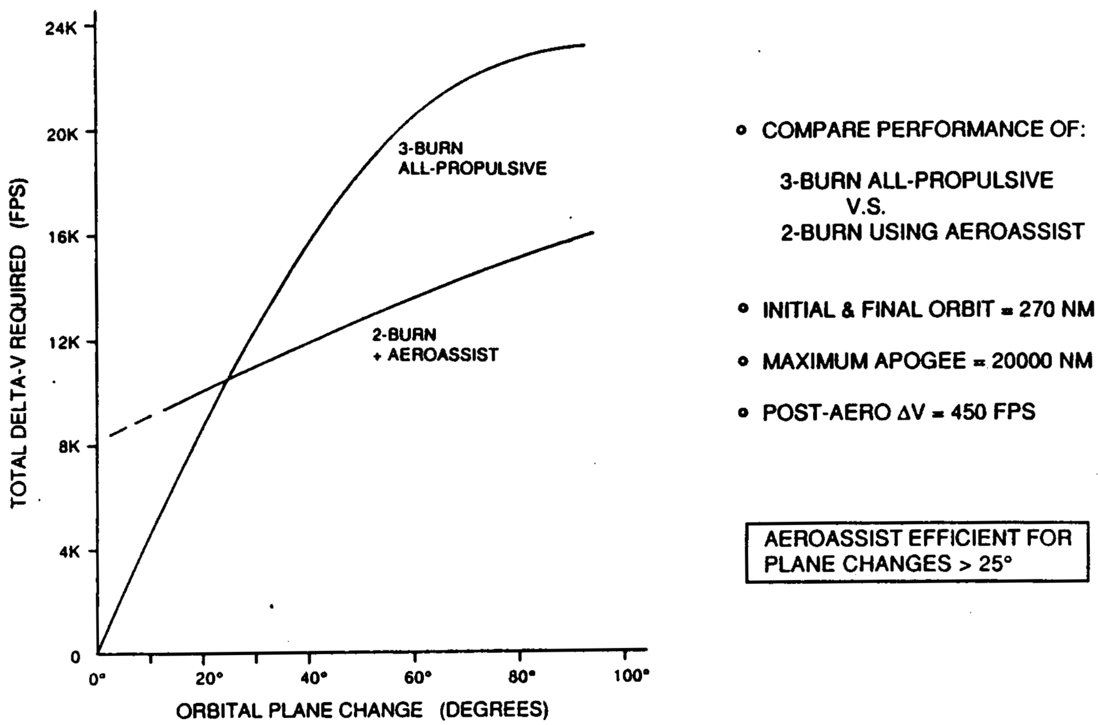


Figure 2.5-2 Large Inclination Change Performance

2.6 LUNAR MISSION PROFILES

A key piece of the lunar initiative analysis is landing mission profile characterization. Three basic types of lunar transfers were investigated as is described below: direct transfer, lunar orbit transfer, and transfer via the L1 libration point.

2.6.1 LUNAR PROFILE - DIRECT LANDING

Various modes of lunar transfer were investigated for advanced missions. The first, shown in Figure 2.6.1-1, is a direct transfer from low Earth orbit to the surface of the Moon and back. An aeroassist maneuver is utilized at the end of the mission to brake into a low Earth orbit. Velocities derived for this mission consist of trans-lunar injection (TLI), lunar landing, lunar takeoff and several small midcourse burns.

A three-body integration routine was used to derive velocities required for flight in the combined Earth/Moon system. Earth departure was from a 245 nm circular orbit. By using a minimum TLI ΔV burn of 10035 fps the lunar descent propulsion requirements can be minimized to 8230 fps. This does increase the lunar transit time to 110 hrs. Landing ΔV is the vertical impact velocity derived from these simulations. The vertical landing case gives the highest velocity requirements, thus landings to all other locations on the moon will require less ΔV. No assessment for gravity losses in descent have been attempted at this time since they are a strong function of the mission design which is beyond the scope of the present study.

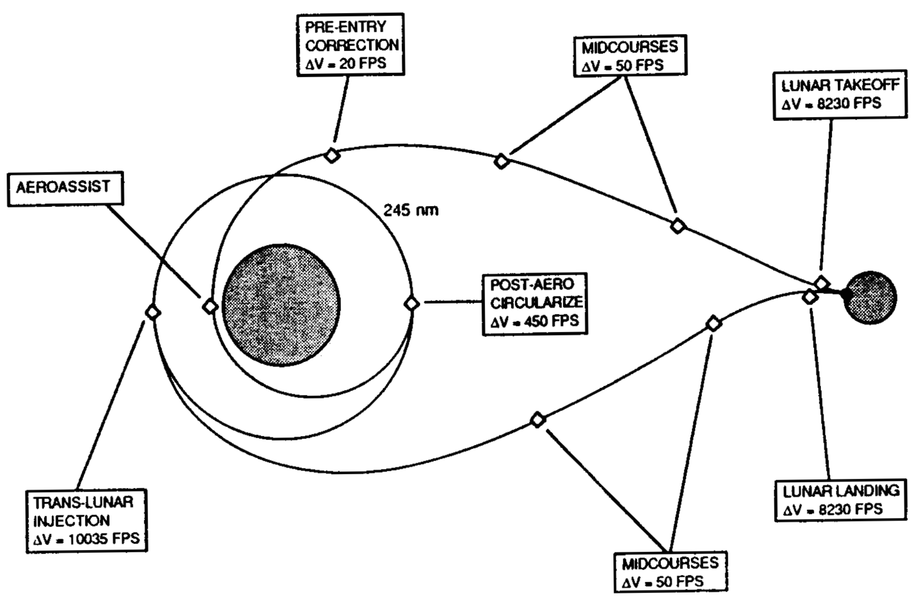


Figure 2.6.1-1 Lunar Profile - Direct Landing

2.6.2 LUNAR PROFILE - LUNAR ORBIT

The lunar orbit profile (Figure 2.6.2-1) uses an intermediate orbit 60 nm above the Moon before descending to the surface. Velocities were derived from Apollo data and three-body integrated trajectories. The major maneuvers are trans-lunar injection (TLI), lunar orbit insertion (LOI), lunar landing, lunar takeoff, and trans-Earth injection (TEI). The trans-lunar trajectory is a "free-return" type which will return to Earth if LOI cannot be achieved. The lunar descent and ascent velocities are smaller than those in the previous direct landing case because the closed lunar orbit has less energy. The lunar orbit mode is probably most appropriate for a mature logistics setup where a permanent lunar orbiting station is in place.

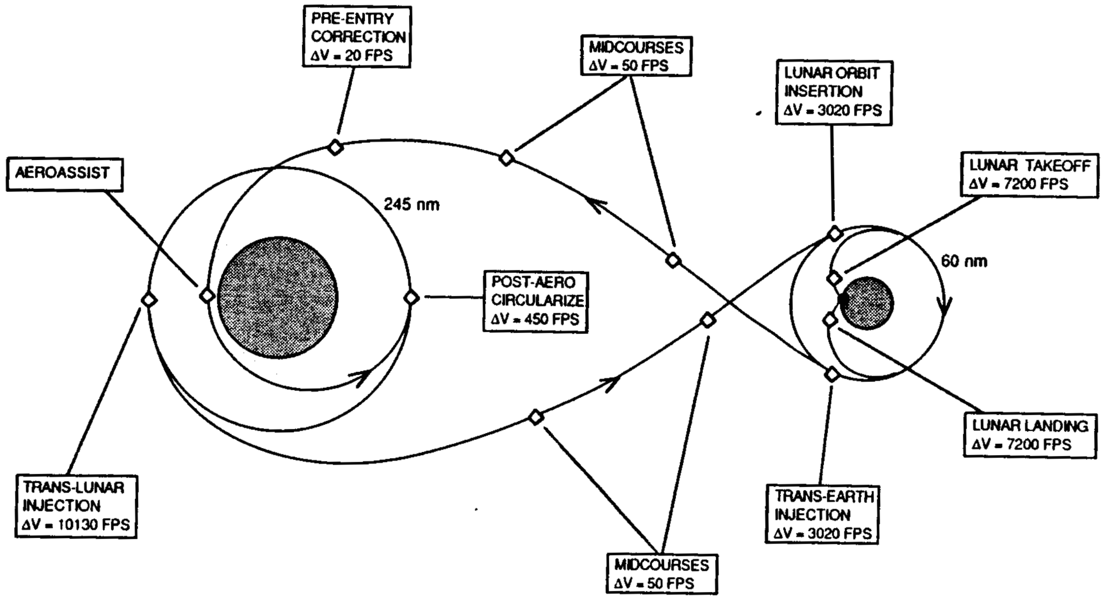


Figure 2.6.2-1 Lunar Profile - Lunar Orbit

2.6.3 LUNAR LIBRATION POINTS

Because of the interaction of the Earth and Moon in an rotating system, gravitationally stable and meta-stable regions are created called the Earth-Moon libration points. There are five of these points as is shown in Figure 2.6.3-1. They are fixed with respect to the Earth-Moon line as shown. Only the L4 and L5 are truely stable points in that an object placed in them will remain without further correction. The rest of the points are meta-stable, they are gravitational saddle points that are stable in only two of three dimensions so an object placed in them will require periodic corrections to stay in place.

The L1 point between the Earth and Moon represents an interesting position for a lunar station. It is close to the Moon and has good access and communication paths with the Earth. Mission profiles have been constructed which travel from the Earth to L1 and then to the Moon as is discussed in the next section.

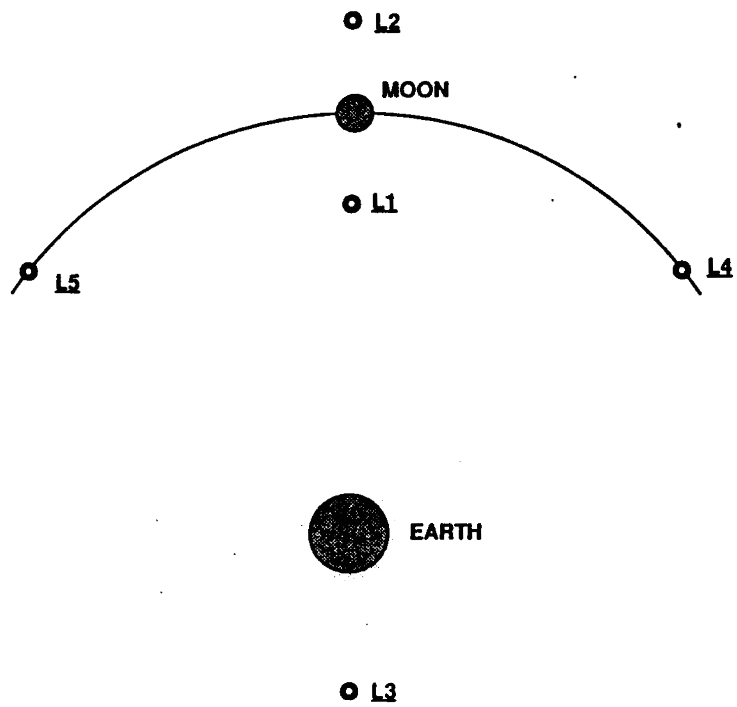


Figure 2.6.3-1 Lunar Libration Points

2.6.4 LUNAR PROFILE - L1 STATION

Figure 2.6.4-1 shows a lunar profile utilizing the L1 libration point as a way station for OTV logistics. This is comparable to the lunar orbit case but has certain advantages in that there is no need for plane alignment since the L1 point is fixed with respect to the Earth and Moon. Such a point could be used for a lunar station with refueling and turnaround facilities or as a more modest transfer point between a dedicated lunar lander (serviced on the lunar surface) and Earth delivery vehicle. The profile shows the Earth to L1 transfer occuring on the left with the L1 to moon transfer on the right. Transfer velocities have been solved for from three-body integration for all but the touchdown/takeoff delta-v's which are derived from Apollo program data.

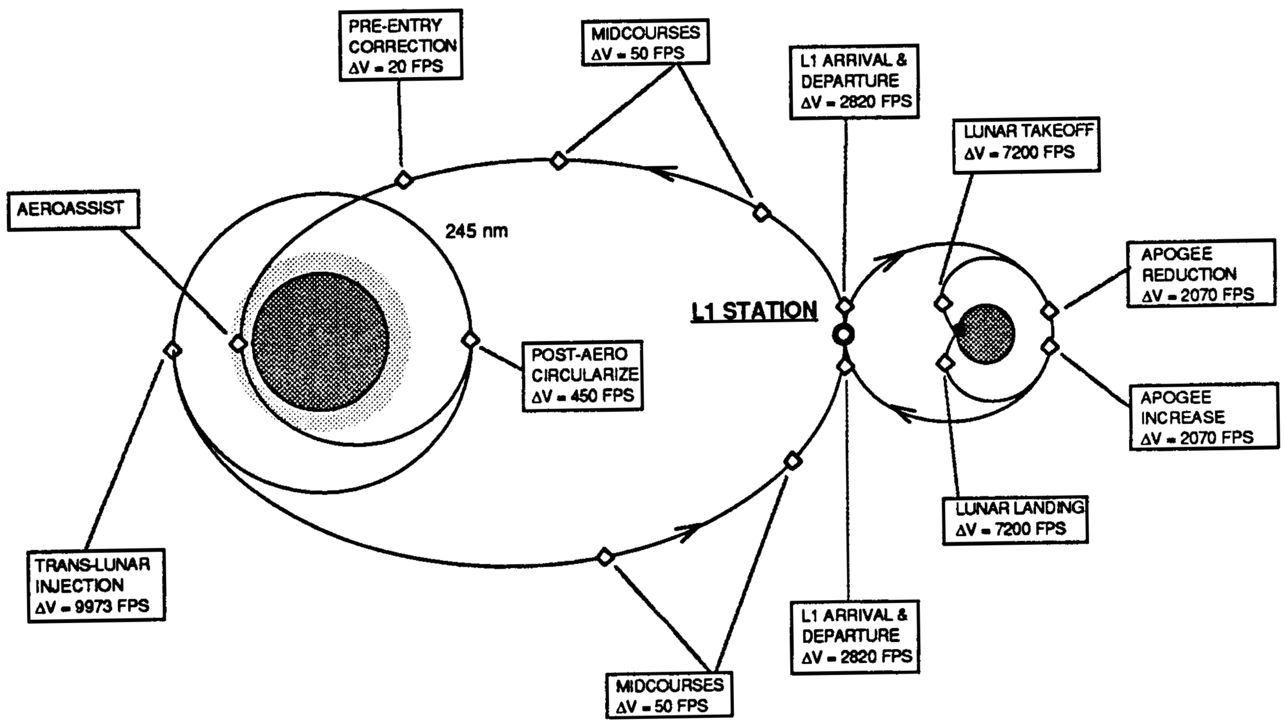


Figure 2.6.4-1 Lunar Profile - L1 Station

2.7 MANNED MARS MISSION LOGISTICS SUPPORT

Planetary boost of a manned spacecraft requires large velocities applied to massive objects. This normally requires a very large upper stage unless the job can be broken up into smaller pieces. Figure 2.7-1 compares three different approaches to boosting a payload into an escape trajectory with a C_3 of 10 km^2/sec^2, which is consistent with a trans-Mars orbit. The first boost technique is to perform a single large burn from an initial low Earth park orbit into the escape trajectory with a required ΔV of 11,900 fps. This is the approach that would require the largest booster because the spacecraft is already assembled and must be injected all at once.

The next two approaches look at delivering the spacecraft in pieces to an energetic assembly orbit. In this fashion, smaller transfer vehicles can be used to build up the interplanetary craft and then, since the craft is in a higher energy orbit, a smaller injection stage can be used for escape. The first option looks at an elliptical assembly orbit with a perigee of 250 nm and an apogee of 100,000 nm. The ΔV required to reach this orbit is 9800 fps, once in it only 2100 fps is required to escape. This orbit gives favorable leverage for an OTV since large modules can be delivered for assembly, the OTV can be retrieved via aeroassist, and an expendable OTV can be used for the escape kick. It must be stressed that this approach does <u>not</u> reduce the overall velocity requirements (and thus net propellant) but <u>does</u> reduce the size of the kick stages required, eliminating the development of a new and huge Earth escape stage. Additionally the highly elliptic Earth park orbit may represent a more favorable departure situation if a low thrust propulsion system is used.

The second option looked at a high altitude circular assembly orbit as opposed to the elliptic one just discussed. By circularizing, a large ΔV penalty is incurred as it takes 13200 fps to reach this orbit. Additionally it takes a large impulse of 7700 fps to escape this orbit. Overall this assembly option is not an optimum approach.

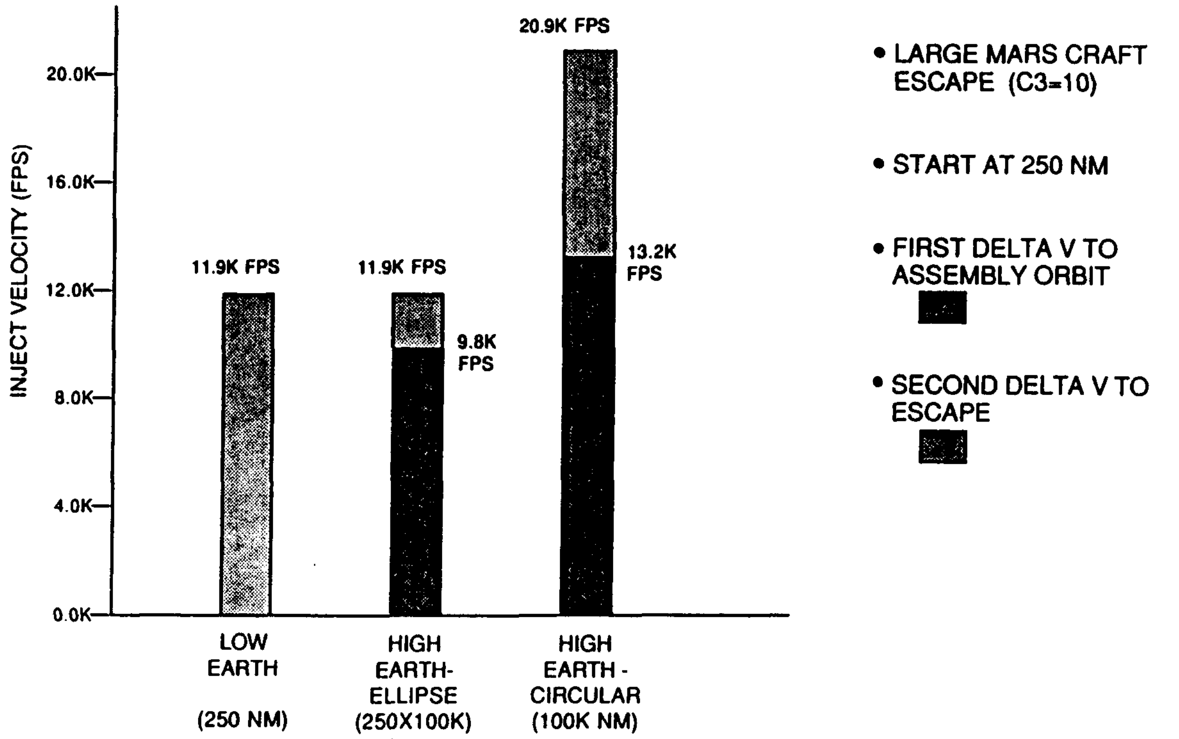


Figure 2.7-1 Earth Escape Velocities

The use of an elliptical assembly orbit for large interplanetary craft appears to have significant benefits. Because large new boost stages will represent substantial development costs it is worthwhile to see whether existing OTV-class vehicles could be utilized instead. Figure 2.7-2 shows a concept for assembling the Mars vehicle in a high energy Earth orbit that then requires a relatively small delta-v for escape. Multiple OTV flights could be utilized to boost Mars spacecraft modules into the high energy assembly orbit where they would be integrated into a main spacecraft. Once the spacecraft was assembled a single OTV, used in an expendable mode, could boost the stack onto a trans-Mars trajectory. This approach maximizes use of existing stages to perform the Mars mission.

The example shown in Figure 2.7-2 is for a 5 times synchronous Earth orbit (250 nm perigee, 126000 nm apogee) where Mars spacecraft assembly takes place. This orbit was selected because it has a high energy state without becoming so elongated that it enters into the lunar sphere of influence. Thus nodal regression rates are low. The perigee is kept at 250 nm for accessibility from the Space Station where modules would be checked out after reaching low Earth orbit. Typical performance figures for a 74Klb propellant capacity OTV are shown. This data shows that a 60.6Klb module could be boosted by a reusable OTV from the Space Station into the 5xSynch assembly orbit. The orbit passes repeatedly, though extremely quickly, through the Van Allen radiation belts.

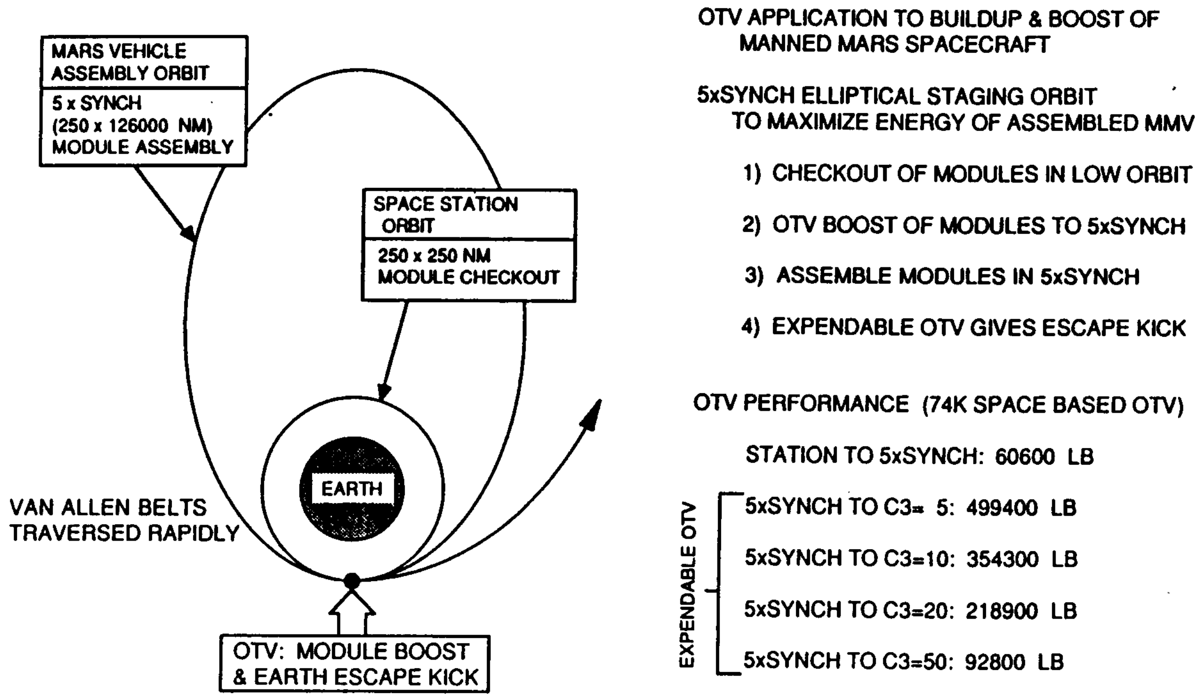


Figure 2.7-2 Manned Mars Mission Logistics Support

Because of the short dwell time, the radiation doses do not appear to represent a major risk for a craft designed for deep space operations. A more detailed assessment of this factor must await further studies, however.

Once the modules have been assembled into the manned Mars vehicle (MMV), an expendable 74Klb OTV can provide the escape kick for various escape energies as shown. For a fairly typical ballistic escape energy of 10 km^2/sec^2 a single OTV can boost a 354300 lb spacecraft into the trans-Mars trajectory. This can be increased substantially by using larger propellant tanks or a two stage OTV approach. It is thus of interest here that a new and very large kick stage need not be developed to enable a manned Mars mission.

3.0 ACC OTV SAFETY ISSUES

The purpose of this task was to examine the key safety issues associated with the aft cargo carrier (ACC) OTV concept. An evaluation of the technical risk in meeting the latest safety requirements of the Space Transportation System (STS) was conducted which contrasted the ACC with the Shuttle cargo bay as a boost location.

The approach was to identify the major hazards apparent in the concept and assess the difficulty in controlling them based on the current hazard control approaches used by the STS and payloads. For the purpose of this assessment, it was assumed that STS payload requirements would be imposed on the OTV as this has been typical of upper stages flown by the STS to-date. They are generally more stringent than STS element requirements. The latest payload requirements were used (as contained in NHB 1700.7a and 1700.b September draft) as well as the draft "return to flight" payload requirements in development by NASA. In addition, comments by members of the STS Payload Safety Review Panel at JSC were incorporated where available.

The assessment was based on the ability of the concept to implement typical hazard control approaches. Each hazard evaluated will be listed on the following figures along with the typical control approach and the technical risk assessment.

This assessment could not consider detailed flight dynamics assessments (ET impact footprint constraints for instance), or critique the STS ACC / OTV structural design. These issues have been considered previously in the OTV phase A study but are outside the scope of the current assessment.

Appendix A contains summary sheets of a preliminary hazard analysis conducted for the ACC OTV. For several key subsystems/operations, hazardous conditions and their causes and effects were identified along with hazard control assessments. Based on this hazard analysis, a set of derived requirements for the STS and ACC OTV were developed and are shown in Appendix B.

3.1 ACC OTV - VEHICLE CONFIGURATION AND FLIGHT PROFILE

Figure 3.1-1 shows the overall launch vehicle configuration for an STS aft cargo carrier (ACC) OTV. The ACC is a hemispheric extension to the aft end of the Shuttle external tank (ET). This provides a large volume approximately 27' in diameter where a payload can be located. For the OTV application the dedicated ACC (or DACC) is used for weight efficiency. The ACC concept has been studied in some detail, as is reported in the "General Purpose Aft Cargo Carrier Study Final Report", May 1985 (NASA contract NAS8-35564).

Figure 3.1-2 shows the boost configuration of the OTV in the dedicated ACC. The OTV has four propellant tanks (2 LOX & 2 LH2) distributed along the longitudinal axis. The aerobrake is folded up along the sides of the vehicle for boost and is deployed shortly after separation. The domed portion of the ACC is jettisoned in ascent, shortly after STS SRB separation.

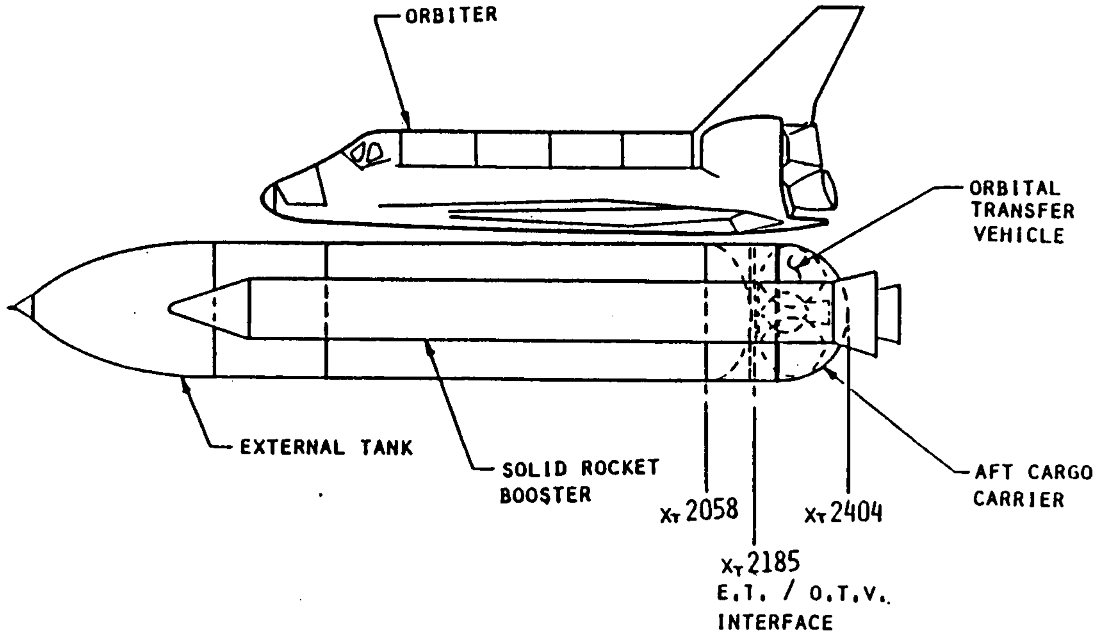


Figure 3.1-1 ACC OTV - Shuttle Boost Configuration

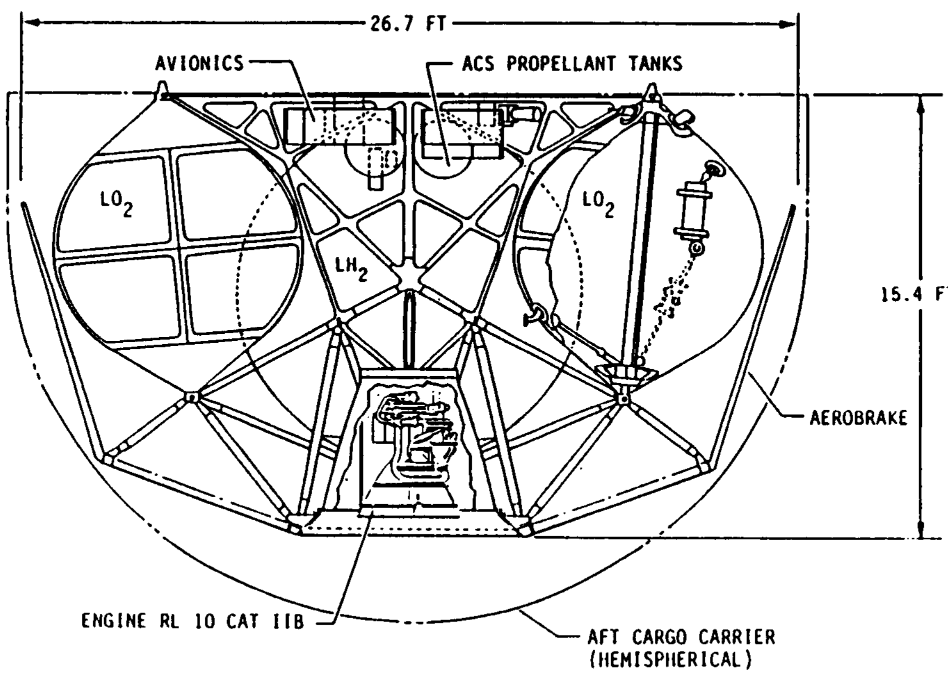


Figure 3.1-2 ACC OTV Boost Packaging

Figure 3.1-3 shows the ascent mission profile for an ACC OTV. The normal shuttle ascent profile is impacted as little as possible, although the vehicle aerodynamics will be somewhat different due to the extension of the ET. Launch, SRB separation, and powered flight to orbit proceed in the same manner as now. ACC shroud separation occurs at T+156 sec, 24 sec after SRB separation. ET disposal targeting at STS main engine cutoff (MECO) is identical to today's requirements. Shortly after MECO the OTV is separated via springs and, after the shuttle has performed OMS-1 and departed the area, the OTV propels itself into a low park orbit. In this orbit, it awaits a rendezvous by the shuttle which then attaches its mission payload (which has been carried to orbit in the shuttle cargo bay).

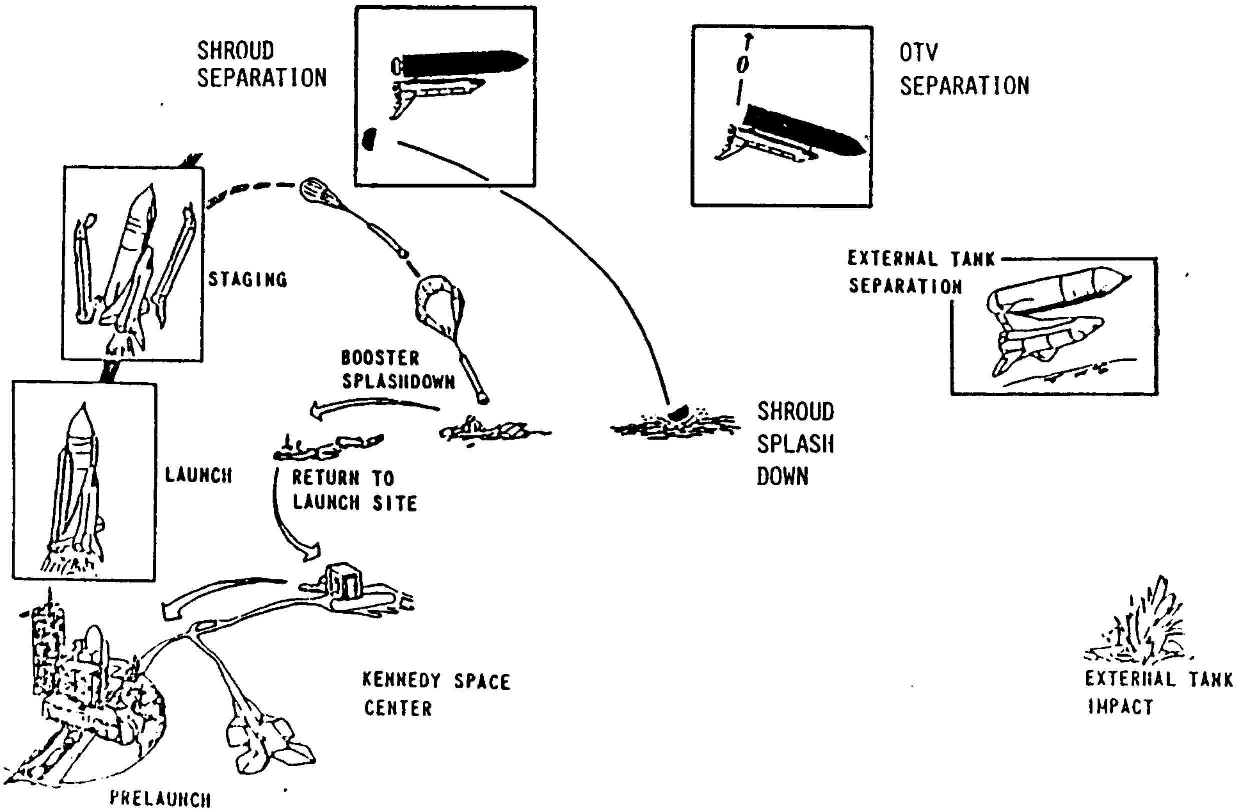


Figure 3.1-3 ACC OTV Ascent Profile

3.2 MAJOR HAZARDS ASSESSMENT

Tables 3.2-1 and 3.2-2 shows the major hazards that were assessed for their safety impacts. Under a hazard group title, the individual hazards are listed with a technical risk assessment for both the ACC and cargo bay approaches. This is the technical risk of the concept's ability to implement the typical control approaches listed. Comments explaining the risk assessment are also provided.

All hazards must be controlled to an acceptable level. A "low" technical risk is considered to present a typical challenge equivalent to other program requirements (i.e. there appears to be nothing unique required to control this hazard). "Medium" risk presents a significant technical challenge and program impacts may result. "High" risk means that there is no known solution for controlling the identified hazard. An "N/A" means the particular design concept does not have this design risk due to the absence of particular hardware (e.g. the risk of premature firing of a destruct system is "N/A" to a stage having none).

Table 3.2-1 Major Hazards - Part 1

HAZARD GROUP / GENERIC HAZARD	ACC RISK	CARGO BAY RISK	TYPICAL CONTROL APPROACH	COMMENTS / CONCLUSIONS
FIRE:				
• PREMATURE MAIN ENGINE FIRING OR INADVERTENT DUMPING OF PROPELLANTS THROUGH MAIN ENGINE	LOW	LOW	THREE SERIES FLOW CONTROL DEVICES CONTROLLED BY ELECTRICAL INHIBITS	AS LONG AS LINES ARE DRY DURING STS MISSION PHASES, THIS HAZARD SHOULD BE CONTROLLABLE
• PREMATURE HYDRAZINE ENGINE FIRING	LOW	LOW	SAME AS ABOVE	MANY ACCEPTABLE DESIGN APPROACHES EXIST
• PROPELLANT LEAKS				
A. TANK SEPARATION POINTS	LOW	N/A	TRIPLE SEALING VALVE	LEAKAGE SHOULD BE CONTROLLABLE BUT SEE OTHER CONCERN UNDER EXPLOSION BELOW
B. VAPOR VENT	LOW	MED	VENT EXTERNALLY - DISCONNECT ON DEPLOYMENT	LEAKAGE CONTROLLABLE - COMPLEX DISCONNECT MECHANISM - PROBABLY DO-ABLE AND STILL MEET REQUIREMENTS. NEED THREE VENT PATHS (VALVES)
C. GROUND / ASCENT	LOW	MED	AS ABOVE	AS ABOVE
D. RETRIEVAL DUMP / FILL DRAIN	LOW	LOW	DRY DURING STS PHASES	WOULD NEED COMPLEX RELIEF MECHANISM IF "WET"
EXPLOSION:				
• PROPELLANT TANK OVERPRESSURE				
A. FAIL TANK SEPARATION VALVES CLOSED	HIGH	N/A	DUAL REDUNDANCY IN OPENING AND CLOSING FUNCTION	PRESENTS POTENTIAL SINGLE POINT FAILURE IN CURRENT CONFIGURATION. FAILING PNEUMATIC VALVE IN VENT LINE WOULD RESULT IN CATASTROPHIC FAILURE. SEE ASCENT VENT REDUNDANCY CHART FOR UPDATED CONFIGURATION.

The fire hazards generally involve release of the propellant into the cargo bay, the ACC carrier, or inadvertent release on the launch pad. The controls for these hazards are rated as low risk since "flow control devices" are used similar to other liquid systems. The only risk assessed as being of concern are associated with the disconnect mechanisms for these systems in the cargo bay. These must assure that no two failures will result in a partially released element. Pyrotechnic release mechanisms (very high reliability) might be used in these systems.

The explosion hazards involve rupture of the propellant tanks from failing to release internal pressure or by overpressurizing. The only concern noted here was with the tank separation valves in the ACC concept. These present potential single failure points should they fail closed by vibration or inadvertent commanding. Because of the lack of dual fault tolerance in the existing concept the design was changed as will be discussed further on.

The fact that the OTV is not dependent on pressure for structural integrity is a positive safety feature of both OTV concepts. This was a major problem with the STS/Centaur system.

Table 3.2-2 shows the conclusion of the explosion hazards and the collision hazards. Collision hazards are associated with structural failures, mechanism failures that interfere with the Orbiter or unacceptable loads impacts on the Orbiter.

Table 3.2-2 Major Hazards - Part 2

HAZARD GROUP / GENERIC HAZARD	ACC RISK	CARGO BAY RISK	TYPICAL CONTROL APPROACH	COMMENTS / CONCLUSIONS
EXPLOSION (CONT): B. PRESSURIZATION SYSTEM OVERPRESSURE	LOW	LOW	DO NOT OPERATE SYSTEM WITHIN SAFE DISTANCE	RESTRICTING ENGINE FIRING TO BE OUTSIDE OF SAFE DISTANCE ELIMINATES CONCERN. OTHERWISE, NEED 2 FT PRESSURIZATION SCHEME
C. LOX COMPATIBILITY	LOW	LOW	USE PROVEN MATERIALS	UNTESTED MATERIALS WILL REQUIRE TESTING
• DESTRUCT SYSTEM	MED	N/A	USE EXISTING TECHNOLOGY	EXACT REQMTS OF ACC SYSTEM TBD
• FAILURE TO DUMP	N/A	N/A OR HIGH	UNKNOWN	IF PROPELLANT MUST BE DUMPED FOR STS ABORT (CARGO BAY ONLY), DESIGN MUST BE 2 F.T. AGAINST PREMATURE DUMP. EXTREME CHALLENGE.
COLLISION:				
• DEPLOYMENT SYSTEM MALFUNCTION (INCOMPLETE SEPARATION / CAPTURE)	LOW	MED	2 FT SCHEMES USING EVA OR JETTISON AS THIRD LEVEL OF REDUNDANCY	MULTIPLE DISCONNECTS (VENTS, ATTACH POINTS PRESENT CONCERN)
• INTERFERE WITH CARGO BAY CLOSURE	LOW	LOW	SEE DEPLOYMENT SYSTEM APPROACH	DO-ABLE
• PREMATURE SEPARATION	LOW	LOW	2 FAILURE TOLERANT SCHEME	MANY ACCEPTED APPROACHES
• STRUCTURAL FAILURE A. VEHICLE	LOW	LOW	1.4 FACTOR OF SAFETY	STANDARD TECHNIQUES
B. COVER	MED - LOW	N/A	SEE ABOVE	LOW RISK IF DESIGN DOES NOT USE PRESSURE. PRESSURE SYSTEM WOULD REQUIRE LAUNCH SEQUENCE TIE-IN.

The need for a destruct system for the ACC OTV is assumed but will be open for further study. If needed, there will be medium technical risk since the destruct system must be dropped (if mounted on the ET) or positively deactivated prior to rendezvous with the Orbiter.

Because of the number of attach points between the cargo bay OTV and the Orbiter, the hazard of deployment system malfunction was rated as medium since developing a two failure tolerant mechanism is extremely difficult and usually requires EVA work-arounds. The ACC configuration is rated as a low risk since two failure tolerance is not required by safety (mechanism failure will not result in Orbiter loss).

The highest risk collision hazard is associated with the failure to dump for the cargo bay configuration should dump be deemed necessary. If required, the OTV interface would have to be both two failure tolerant against failing to dump and two failure tolerant against premature dump. These two constraints directly oppose each other in design implementation. The need to dump is an evolving situation, dependent on CG, landing weight, and/or post-landing cryo inerting concerns. However, it seems unlikely that a cargo bay OTV could, in all instances, avoid the need to perform an in-flight dump. Acceptable solutions to the problems of Orbiter center of gravity, landing weight, and cryo venting must be provided over the full range of vehicle flight envelopes.

In the sections that follow, issues raised here will be discussed along with others dealing with ACC shroud presurization, proximity operations, and LH2 tank jettison.

3.3 ASCENT VENT REDUNDANCY

A problem exists with the current design of the ACC OTV ascent vent as was discussed in the previous section. Figure 3.3-1 shows the baseline propulsion schematic from the 1984-1985 Phase A study. The areas of concern are circled. Three valves in series are used on the GH2 side, and two series valves on the GO2 side to control the ascent venting process. Despite the fact that the valves have twin actuators, the system has only single fault tolerance, instead of dual, to the catastrophic failure of a valve failing closed which would cause tank overpressurization. Previously, the ACC OTV was to be jettisoned if two critical failures occurred in the ascent vent line. This is the only twin failure path that requires such a drastic action and so it was felt that a safer option would be to restructure this system to preclude OTV jettison.

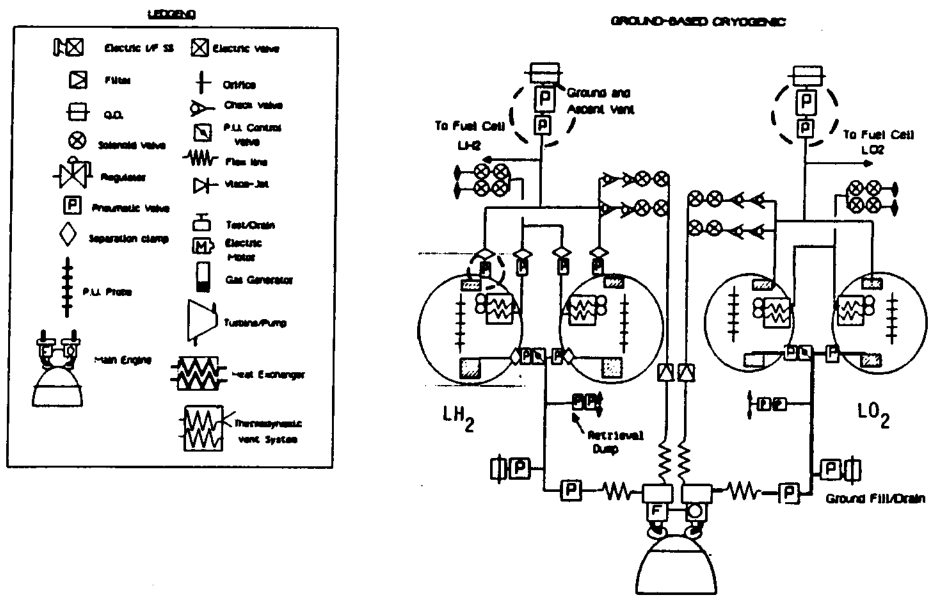


Figure 3.3-1 ACC OTV Propulsion Schematic (1985)

The updated design shown in Figure 3.3-2 cures this problem with parallel pneumatic valves to provide for venting control and a single pyro actuated valve with twin initiators. Under normal operating conditions only the pneumatic

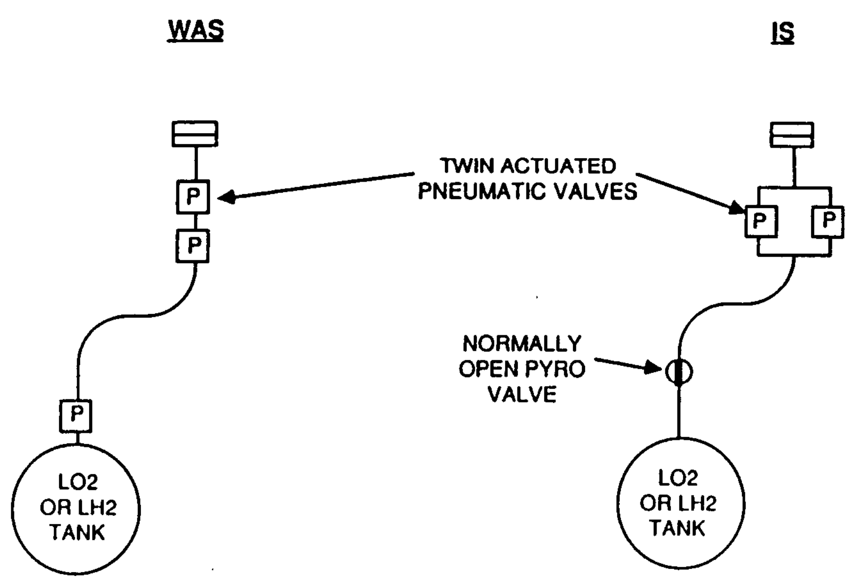


Figure 3.3-2 Ascent Vent Redundancy

valves would be used to close off the system for post-MECO flight. However, if two critical failures occured in the pneumatic system the pyro valve would be fired to seal the line. This system provides for two fault tolerance in the venting system as well as three inhibits for preventing loss of propellant from the tanks.

3.4 ACC PRESSURE STABILIZATION

Currently the dedicated ACC (DACC) uses internal pressure for structural stabilization during the STS SRB ignition overpressure pulse. A review of STS/Centaur Lessons Learned highlights that one of the main problems with the Centaur was its pressure stabilized skin. In this case internal pressure was required throughout the flight to maintain structural integrity. Hence one of the major prohibitions that has resulted from the Centaur cancellation is against pressure stabilized structures. This can be dealt with for the ACC in one of two ways.

The first option would be to use the system as it stands. The argument here is that the ACC pressurization is not required for flight, but only for the extremely brief period of time that the SRB ignition overpressure exists. An adequate pressure in the ACC would then be one of the launch commit criteria to be checked before the SRB ignition command could be issued. Short of a catastrophic rupture of the ACC (which would be a flight critical structural failure anyway), any leak in the system would be slow enough that the countdown could be halted before any ignition-critical pressurization levels were reached. This represents a complication for the Shuttle firing sequence more than a flight-critical safety issue.

An alternate approach was investigated (Figure 3.4-1) that assessed the design impact of making the ACC totally unpressurized for all phases of flight. This

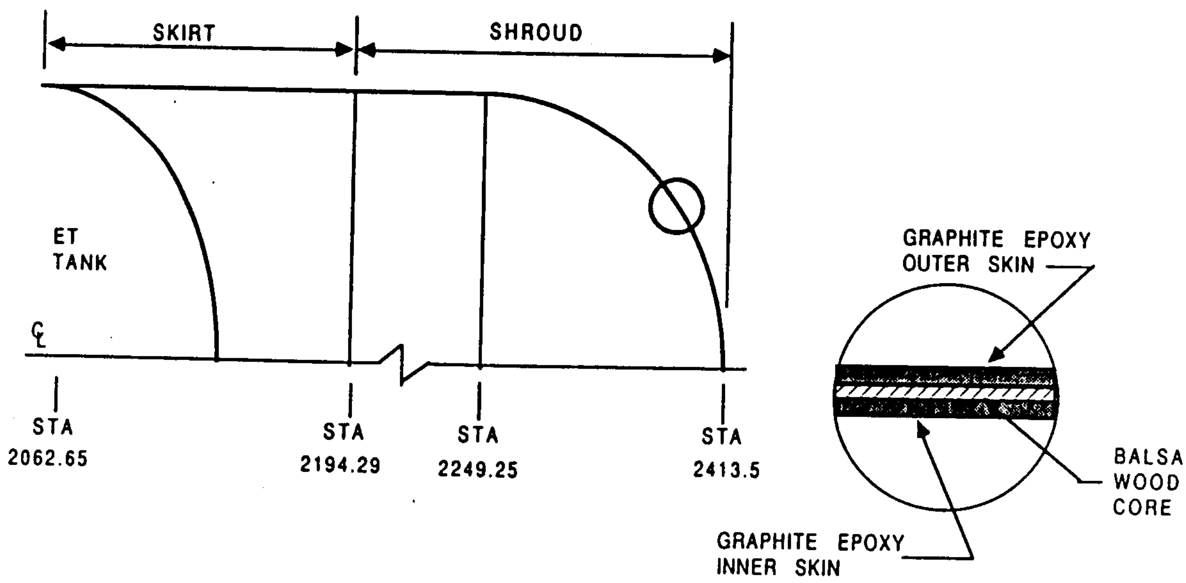


Figure 3.4-1 Dedicated ACC Composite Shroud

approach strengthens the ACC dome structure so that the SRB ignition pulse can be resisted solely with structural stiffness. In order to keep the flight weight manageable, a filament wound approach is neccesary. This approach results in significant manufacturing complication and an increase in weight of 210 lb. Further design details may be found in the Structural Issues section (5.1.4).

Currently it appears that the first option gives an acceptable safety situation for the orbiter with a backout avenue represented by the composite ACC design.

3.5 ACC OTV PROXIMITY OPERATIONS

Because the ACC OTV flies independently to low Earth orbit additional attention must be paid to preventing hazards to Shuttle operations. STS safe separation criteria have been used throughout in designing the ACC OTV flight sequence. Figure 3.5-1 shows the relative motion of the Orbiter, External Tank, and OTV after STS MECO. The OTV separates via springs and coasts backwards in a passive state while the Orbiter performs a normal ET separation sequence. The OTV hydrazine ACS system is turned on at a distance of 200 ft, consistent with STS safe separation criteria.

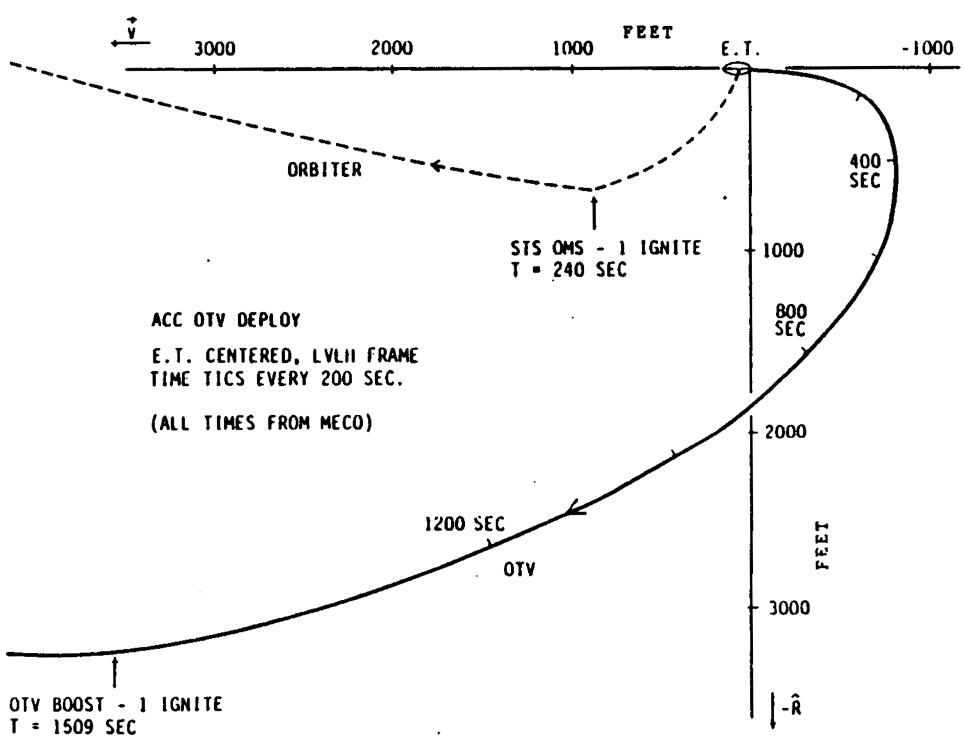


Figure 3.5-1 OTV / Orbiter Separation Profile

When the Shuttle performs its OMS-1 burn the two vehicles are about 1800 ft apart which should be adequate from a plume impingement standpoint. The first

OTV MPS burn occurs at about 25 minutes after MECO at which time the the Orbiter is 52 nm away. The second OTV MPS burn (which injects the OTV into a 140 nm circular orbit) occurs about 77 minutes after MECO at an Orbiter separation of 228 nm. The Orbiter rendezvous sequence commences a few hours after this final OTV main engine burn. A unique concern to the ACC OTV is safing the cryogenic vehicle prior to Shuttle rendezvous and payload mate. Figure 3.5-2 shows the sequence of system safing required to inert the vehicle prior to Shuttle contact. Four primary systems are adressed as follows.

The main propulsion system (MPS) is normally inerted at the end of each burn sequence and will thus not pose a hazard since the final OTV MPS burn is executed at least 200 nmi away. This operation consists of purging the engine of LOX and hydrogen, and removing power from the electronics.

Since water dumps are not desirable in the Shuttle's vicinity the OTV's fuel cell water collection tank will be purged at least 2 hours from docking. The system has a 12 hour capacity so there should be no need for further dumps during the 4 hours the Shuttle and OTV are in close proximity.

The OTV thermodynamic vent system (TVS) will be locked up at a distance of 1000 ft from the orbiter, probably by ground command. Thermal analysis shows a no-vent capability of at least 6 hours if the OTV tank pressure is first reduced to 16 psi. This will eliminate undesirable gaseous venting during the time the two vehicles are in collision range.

STS APPROACH SAFETY SEQUENCE	RANGE	COMMENTS
1) SAFE MAIN PROPULSION SYSTEM	>200 NM	PURGE ENGINE & LINES REMOVE POWER FROM VALVES & ACTUATORS
2) SAFE FUEL CELL H_2O DUMP SYSTEM	8 NM	PERFORM DUMP 2 HRS FROM DOCK NO DUMP FOR 12 HRS
3) SAFE THERMODYNAMIC VENT SYSTEM	1000 FT	VENT TANKS DOWN TO 16 PSI NO VENT FOR 4 HRS
4) SAFE ATTITUDE CONTROL SYSTEM	TBD	CLOSE VALVES AT ENGINES REMOVE POWER FROM VALVES

MONITOR & CONTROL FUNCTIONS:
(VIA REDUNDANT RF LINK)

TANK TEMPERATURE & PRESURES
ACS STATUS
VALVE STATUS
PAYLOAD LATCHES
AVIONICS SUBSYSTEM STATUS
POWER SUBSYSTEM STATUS

Figure 3.5-2 ACC OTV / STS Prox. Ops. Safing Sequence

The final system to be safed will be the OTV attitude control system (ACS). Safety guidelines for the range at which this must be done are uncertain at present, although it would be desirable to wait until as late as possible to reduce residual attitude rate disturbances.

3.5.1 ACC OTV ON-ORBIT PAYLOAD INTEGRATION

One of the significant complications associated with ACC OTV operations is the need for on-orbit integration of the OTV and spacecraft. Although this operation is normally carried out on the ground it does not represent an insurmountable task if conducted in flight. Many previous U.S. manned spacecraft have utilized on-orbit linking of two modules in their operations including Gemini, Apollo, and Shuttle. The key to these operations is in maintaining a simple, standarized interface. Figure 3.5.1-1 shows a payload adapter concept that has one end standardized to the OTV and the other end designed for the specific payload. The OTV end contains guide pins and electric latches to enable on-orbit docking with the vehicle. The latch system will be commanded by the Shuttle for safety, probably through the RMS. The basic OTV avionics design utilizes a data bus which enables a single electrical command interface to the payload along with a power plug. These features simplify the docking interface. The payload end of the adapter will probably utilize pyrotechnic separation devices for spacecraft deployment. This payload-to-adapter connection will have been built up and verified on the ground before flight.

ON-ORBIT INTEGRATION OF TWO VEHICLES IS NOT A NEW ISSUE

GEMINI, APOLLO, SHUTTLE DOCKING

SIMPLE INTERFACE IS THE KEY

OTV PAYLOAD INTERFACE: POWER, SINGLE DATA BUS TIE, LATCHES

LATCH DRIVES CONTROLLED BY SHUTTLE THROUGH RMS (SAFETY)

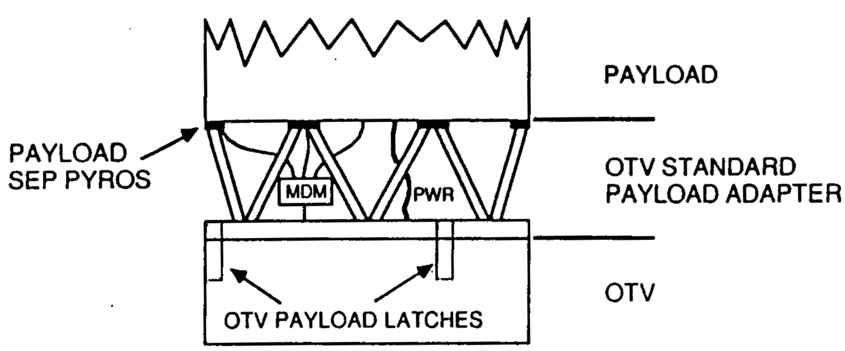


Figure 3.5.1-1 ACC OTV On-orbit Payload Integration

3.6 ACC OTV HARDWARE JETTISON

At the end of its mission the reusable ACC OTV is recovered by the Space Shuttle for return to Earth. In order to fit the vehicle in the Shuttle cargo bay the large liquid hydrogen (LH_2) tanks must first be removed leaving a rack consisting of the main engine, LO_2 tanks, avionics, ACS system and OTV structural core. Originally, the recovery of the ACC OTV included RMS removal of the LH_2 tanks for stowage in the Shuttle and return to Earth. This appeared to be a feasible approach, however it added a very significant number of operations to the normal recovery process. Because these operations have hazards associated with them as well as the fact that stowing tankage in the cargo bay requires hardware (weight penalty) as well as volume that could otherwise be allocated to other payloads, it is now felt that the LH_2 tanks should be jettisoned at the conclusion of MPS operations, as is described below. This will eliminate any reconfiguration operations which will simplify recovery thus increasing safety.

Table 3.6-1 shows the sequence of events required to safely dispose of the OTV's LH_2 tankage as well as the vehicle's aerobrake. Upon exiting the atmosphere after the aeroassist maneuver the aerobrake is jettisoned via springs. Because the trajectory is suborbital at this point, the orbital life of the aerobrake is less than 1 revolution. It is felt that the aerobrake will break up and disintegrate because of the very high heat pulse and aerodynamic loads (peak heat flux of 450 BTU/FT^2, peak load of 40 g's) acting upon an unsupported structure with its engine doors open. This requires much more extensive analysis and test, however, to verify.

Table 3.6-1 ACC OTV Hardware Jettison Sequence

BEGIN AT END OF AEROASSIST PHASE

1) EXIT ATMOSPHERE
2) JETTISON AEROBRAKE, 1 FPS SPRING SEP (ORBIT: 25 X 140 NM)
3) COAST TO APOGEE (140 NM)
4) ORBIT RAISE #1A: MPS BURN TO 100 X 140 NM ORBIT
5) JETTISON LH2 TANKS (ORBIT: 100 X 140 NM)
6) ORBIT RAISE #1B: ACS BURN TO COMPLETE PHASING ORBIT INJECTION, DUMP ALL RESIDUAL MPS PROPELLANTS
7) COAST TO NEXT APOGEE
8) ORBIT RAISE #2: PARK ORBIT INJECT INTO 140 NM CIRCULAR

ALL HARDWARE JETTISONED INTO SHORT DURATION ORBITS

AEROBRAKE - 3/4 REVOLUTION

LH2 TANKS - LESS THAN 1 DAY ORBITAL LIFE

MPS VELOCITY REQUIREMENTS = 280 FPS

ACS VELOCITY REQUIREMENTS = 71 FPS (35 LB PROPELLANT)

After the OTV coasts to its first apogee (Figure 3.6-2), the main propulsion system (MPS) is used to raise the vehicle's perigee out of the atmosphere. When this perigee value reaches 100 nmi, the MPS is shut down and the large LH_2 tanks jettisoned. This leaves the tanks in a 140 by 100 nmi orbit which will decay in less than a day due to the very low ballistic number (about one lb/ft^2) of the tanks. Because of the very thin skin of the tanks (0.025 inch thick), it is very unlikely that anything will reach the ground, thus an uncontrolled decay is acceptable.

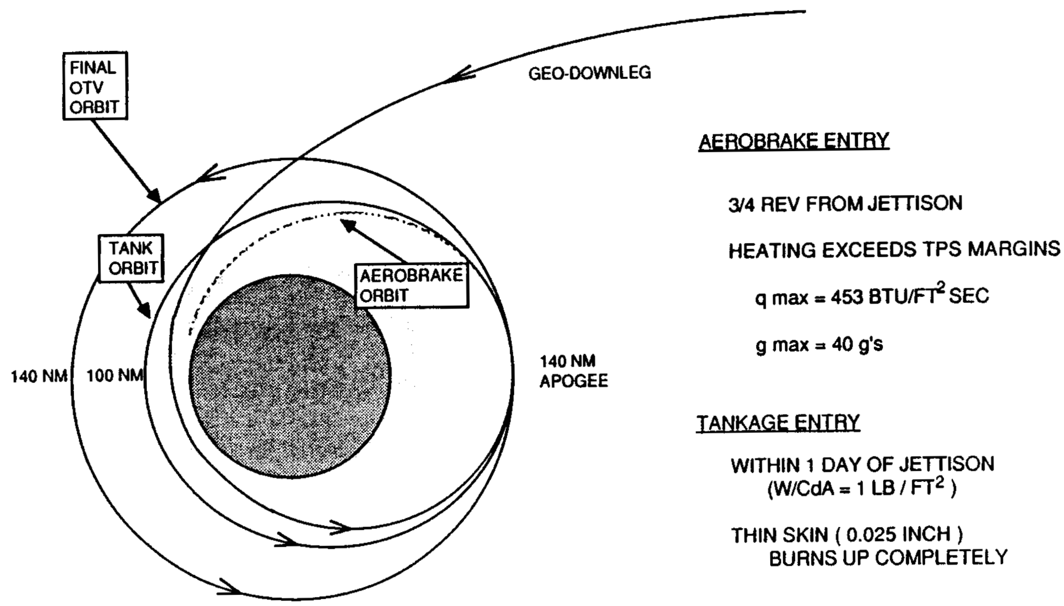


Figure 3.6-2 ACC OTV Hardware Disposal

Upon completing the tank jettison sequence the OTV continues its orbit circularization maneuver using the smaller ACS translation jets. This sequence consists of injection into a phasing orbit with perigee values between 110 and 140 nmi followed, after one to two revolutions, by an orbit cicularization burn into the desired 140 nmi park orbit. The net additional propellant requirement imposed by this jettison maneuver upon the hydrazine ACS system is only 35 lb.

Figure 3.6-3 shows the savings for the Shuttle payload bay volume if the large OTV LH2 tanks are jettisoned rather than being returned to Earth intact. Because of the increasing value of STS down capability (as heightened by Space Station operations assessments) this approach appears to be an attractive one. Not only could additional on-orbit payloads be retrieved by the Shuttle but also low-density payloads could be carried throughout the flight in the STS cargo bay because of the volume savings realized in carrying the OTV in the ACC (at launch, only the OTV's payload and a minimal amount of OTV return ASE is carried in the Shuttle cargo bay, not the OTV itself).

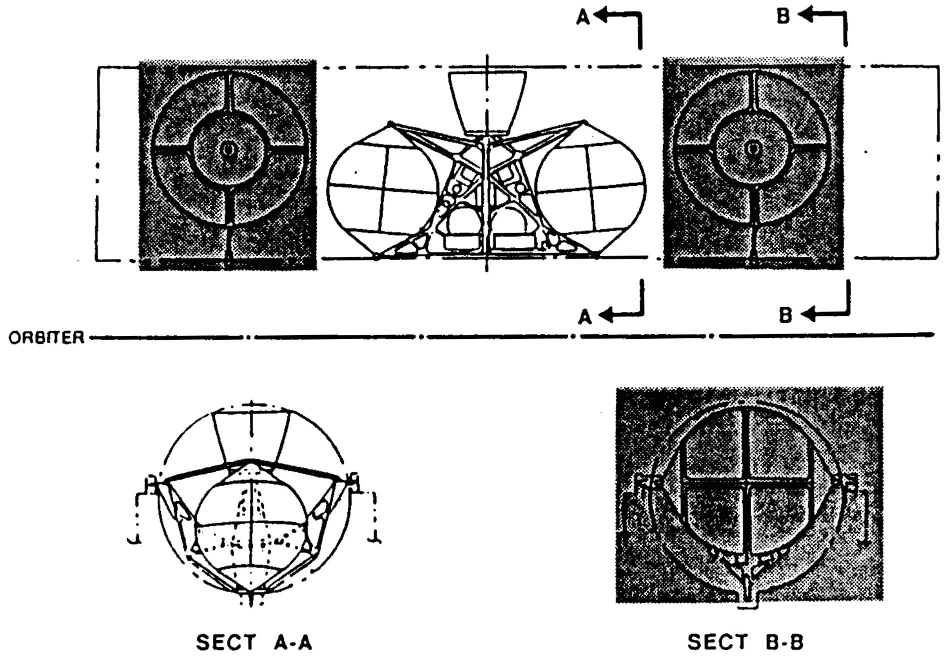


Figure 3.6-3 LH2 Tank Jettison - STS P/L Bay Savings

The reduction in STS payload bay volume required is from 85% (LH2 tank retrieval) down to 40% (LH2 tank jettison. Jettisoning the LH2 tanks also reduces the amount of OTV ASE that the Shuttle must carry, from 2659 lb down to 920 lb. It also reduces the ACC OTV retrieval complexity since no tank removal operations or OTV reconfiguration are required to be performed by the Shuttle prior to berthing in the orbiter bay.

In general, the option of jettisoning the OTV LH2 tanks is an attractive one and is recommended for future ACC OTV assessments.

3.7 ACC OTV SAFETY CONCLUSIONS

Within the constraints of this study, no potential show stoppers were identified for the ACC OTV concept from a safety standpoint. Although the ACC concept does require new Shuttle operating regimes, primarily in the boost phase, their use does not represent high technical risk from a development standpoint. There is a potential show stopper with the cargo bay configuration if an in-flight dump remains a requirement. The safe development of a system that is both dual fault tolerant to dump initiation as well as dump inhibit represents a significant technical challenge.

The ACC OTV has definite safety advantages over the cargo bay configuration as follows:

a. The venting system disconnect mechanisms are not safety critical since the orbiter is not at risk should they fail to operate correctly.

b. The need to dump is not a risk to the Orbiter should it fail (it would most likely not be needed at all).

c. There is no requirement for post-landing inerting of the cryogenic systems in the event of an in-flight abort (the OTV is jettisoned with the ET).

The two medium risk items associated with the ACC configuration are not show-stoppers and are not considered serious disadvantages. The tank separation valve concern has been eliminated completely with an alternate concept. Only the potential new destruct system remains as both a technical and additional safety risk. The safety risk associated with this system should be made to be acceptable since an extensive history in designing these systems exist.

Various members of the STS Payload Safety Review Panel were contacted and asked if there were any lessons learned from the return to flight effort with regard to cryogenic stages in the payload bay. They said that these type of stages were not strictly prohibited but that "all the Centaur problems must be solved" which would involve major modifications to the orbiter for additional venting provisions that were planned for the Centaur and possibly others. The panel members contacted said they have not yet seen a design that meets all of the requirements.

4.0 DESIGN ISSUES

The design issues addressed during this study extension are listed below:

Near term expendable vehicle definition
- performance enhancement options
- cost trade studies

Ground based OTV characteristics
- performance to GEO
- payback for reuse
- technology demonstration opportunities

Lunar mission accommodation
- mission optimization
- Lunar transfer vehicle definition
- Lunar landing considerations
- Cryo engine throttling issues

Optimum cryogenic Shuttle "C" OTV
- vehicle characterization
- performance summary

An "initial" expendable vehicle was defined and performance comparisons were made with the ground based reusable concept developed in earlier Phase A effort. The intent was to provide a vehicle concept that represents a program start in a time frame earlier than for the ground based reusable concept. The main issue that pertains to the near term expendable is in choosing the performance enhancements that fit the time frame of program start. If a somewhat aggressive development was pursued starting in 1988, the earliest IOC date would be 1993. Data for this aggressive date, along with several later ones, is shown in this section.

The Lunar mission optimization, Lunar transfer vehicle and lander definitions, and cryogenic engine implications of Lunar landing constitute a major portion of the design issues. Final subjects include OTV concept definitions that correspond to the Shuttle "C" vehicle concept.

4.1 NEAR TERM EXPENDABLE VEHICLE DEFINITION

The approach taken in defining an early or "initial" expendable vehicle was to start with a concept that would grow into the ground based reusable vehicle. Table 4.1-1 shows the items that would differ between a ground based reusable vehicle and an expendable predecessor. One obvious difference between a reusable aeroassisted vehicle and an expendable version is the aerobrake. But, the brake can be added or removed as a unit without impacting the remaining stage structure. Other dry weight benefits for an expendable stage include less avionics and meteoroid protection requirements. This is primarily due to less time on orbit and no need to return to LEO. Also, an existing RL10A could be used rather than a newly developed engine; once again to provide a vehicle that could be made available at an early date; say, 1993.

Table 4.1-1 Deltas From Ground Based Reusable

ITEM	DELTA WEIGHT (LBM)
- REMOVE AEROBRAKE	-1419
- RL10A-3 VS. IOC ENGINE	+75
- THINNER METEOROID BUMPER	-80
- BATTERIES INSTEAD OF FUEL CELLS	-26
- GROUND UPDATE INSTEAD OF GPS FOR STATE VECTOR	-52
- 2219 AL FOR TANKS	+323

The character of the first OTV design depends upon the year of intended Initial Operational Capability (IOC). This is due to the availability of desirable technologies occurring at different dates. For instance, if 1995 was the target date for IOC rather than 1993, an advanced engine may be available. In addition, aluminum-lithium alloy could perhaps be used for tankage instead of 2219 aluminum for increased performance.

4.1.1 BASELINE EXPENDABLE DEVELOPMENT

The concept shown in Figure 4.1.1-1 illustrates a version of OTV that is possible to develop in the near term (1993) with relatively low risk. For example, the concept incorporates the RL10A which is an existing engine now in production. This vehicle concept is intended to be expendable and not manratable. Additional features include lightweight silver-zinc batteries rather than fuel cells or heavy rechargable batteries. Composite structure was selected over all-aluminum due to the performance advantages, the availability of the composites in the time frame of interest (1993), and the relatively small development cost difference. All-aluminum tanks were selected due to the uncertainty of Al-Li alloy availability in an early timeframe. These tanks were sized to hold 50 Klbm of propellant in order to enable the vehicle to deliver the 22 Klbm platform to GEO. These tanks are also the largest that will fit into the proposed STS aft cargo carrier. The weight statement in the figure shows the vehicle with a nominal propellant load of about 45 Klbm which corresponds to the current lift capability (55 Klbm) of the shuttle.

WEIGHT	
TANKS	1106
STRUCTURE	650
ENVIRONMENTAL CTRL	246
MAIN PROPULSION	944
ORIENTATION CTRL	187
ELECTRICAL SYSTEMS	328
G. N. & C.	182
CONTINGENCY (15%)	540
DRY WEIGHT	4189
PROPELLANTS, ETC	45424
LOADED WEIGHT	49613

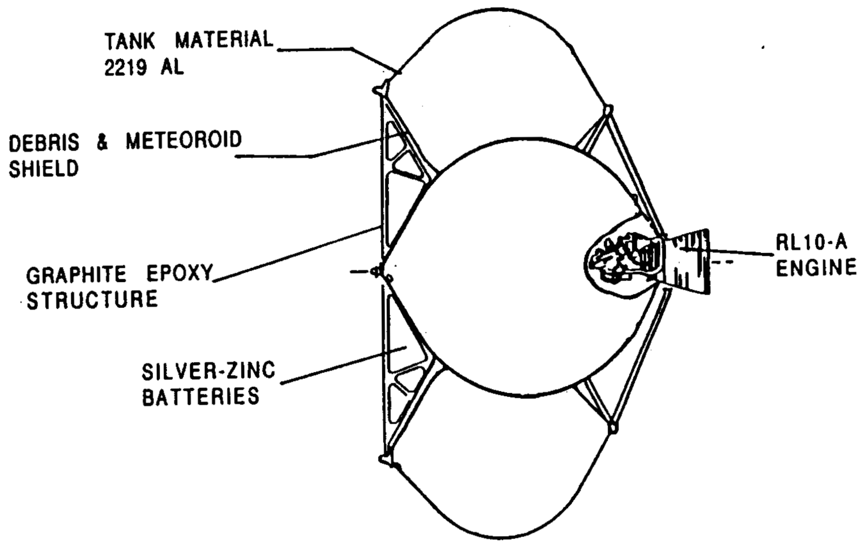


Figure 4.1.1-1 ACC Expendable OTV Baseline

The performance improvements have been calculated for several vehicle enhancements considered for the near term expendable vehicle concept. These performance enhancement deltas shown in Figure 4.1.1-2 are the benefits in GEO payload capability with an STS launch weight constraint of 55klbm. The tankage and structure performance deltas are nearly the same as the dry weight reductions for each of these options. Therefore, these comparisons are relatively independant of STS lift capability. The engine upgrades, however, include both the dry weight differences from the RL10A and the performance improvements due to increases in specific impulse.

4.1.2 COST TRADE STUDIES

In order to compare these performance enhancements for application on the expendable vehicle, a cost assessment of them was made by comparing the benefits of increased payload capabilities. Table 4.1.2-1 highlights the major groundrules and assumptions applied to the three expendable OTV trade studies that follow. All costs are reported in 1985 dollars and exclude fee and contingency. The trade study results report only the affected subsystems and exclude the total stage LCC. This provides visibility to the results within the order of magnitude of the expected cost of the OTV enhancements and precludes them being overwhelmed in the total LCC estimate.

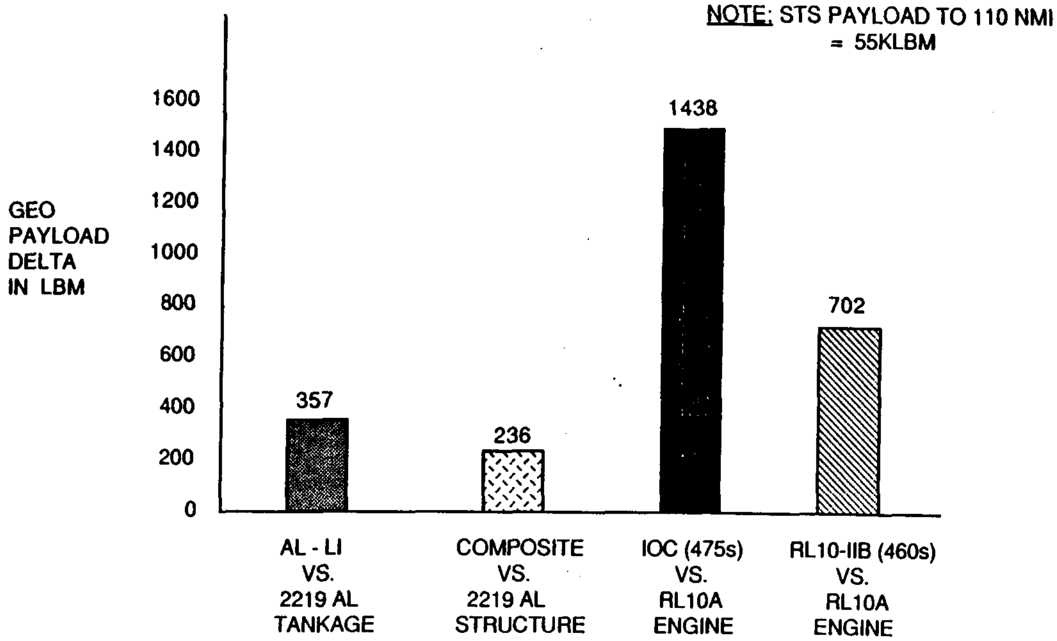


Figure 4.1.1-2 Performance Enhancement Deltas

Table 4.1.2-1 Cost Groundrules: Expendable OTV Trades

- **All Cost Estimates Are In 1985 Dollars And Exclude Fee**
- **Cost Deltas Include Only the Impact Of The Proposed Enhancement**
- **NSTS CPF Assumed At $73M / Flight Per Study Groundrules**
- **Reference Expendable Stage Average Unit Cost At $50M**
  - **Aluminum Structure, Aluminum Tankage, RL-10 Engine**
- **Trade Study Cost Benefits Analysis Include**
  - **Delta DDT&E (Represented By The Y-Axis Offset)**
  - **Delta Unit Cost (Factored Into Recurring Benefits On Per Mission Basis)**
  - **$/LB Impact Based On P/L Lift Differences Between Trade Alternatives**
    - **Benefit Based On $/Lb To Geo Performance Of Reference Candidate**
    - **Includes Delta P/L Only**

The NSTS cost per flight used for purposes of transportation costs to low earth orbit (LEO) was $73M (consistent with the government supplied groundrules). The reference expendable stage average unit cost is $50M. The reference vehicle configuration includes aluminum structure, aluminum tanks, and the RL-10A engine.

The trade study results are presented in the form of cost deltas. These delta costs are derived from the estimates of three major elements of cost. The first cost element is the DDT&E cost estimate of the respective enhancement candidates. In the results presentations that follow, the delta DDT&E costs are represented as the offset on the Y-axis. This offset includes the cost estimate for developing the lighter weight (structures and tanks) or higher performing (IOC engine) trade study candidate. The second element is the unit cost estimates. In the expendable vehicle this is treated as a cost per mission item. The delta unit cost is combined with the third cost element which is the perceived P/L delivery benefit of the lighter weight or higher performing trade candidate. This element of cost represents a measurement of the potential payload benefit of the higher performing trade study candidate. The benefit is calculated on a cost per mission basis. The delta P/L weight is calculated on a per pound basis at the cost of delivering each additional pound at the cost per pound to GEO of the less attractive trade study alternative.

The cost results of replacing the aluminum airframe with composite structure are shown in Figure 4.1.2-1. The DDT&E and unit cost for the aluminum airframe are $21.9M and $1.3M, respectively. The composite airframe exhibits higher DDT&E costs ($27.5M) but slightly lower unit costs ($1.2M). The delta DDT&E cost estimate is represented by the offset on the Y-axis. The additional DDT&E investment required for the composite airframe is $5.9M. There is a slight unit cost benefit due the composite of approximately $0.1M.

	55K Orbiter	72K Orbiter
P/L, Composite Structure	12,245 Lbs	18,127 Lbs
P/L, Aluminum Structure	12,009 Lbs	17,891 Lbs
Delta P/L	236 Lbs	236 Lbs
Composite $/Lb (GEO)	$10,045 $/Lb	$ 6,785 $/Lb
Aluminum $/Lb (GEO)	$10,240 $/Lb	$ 6,875 $/Lb

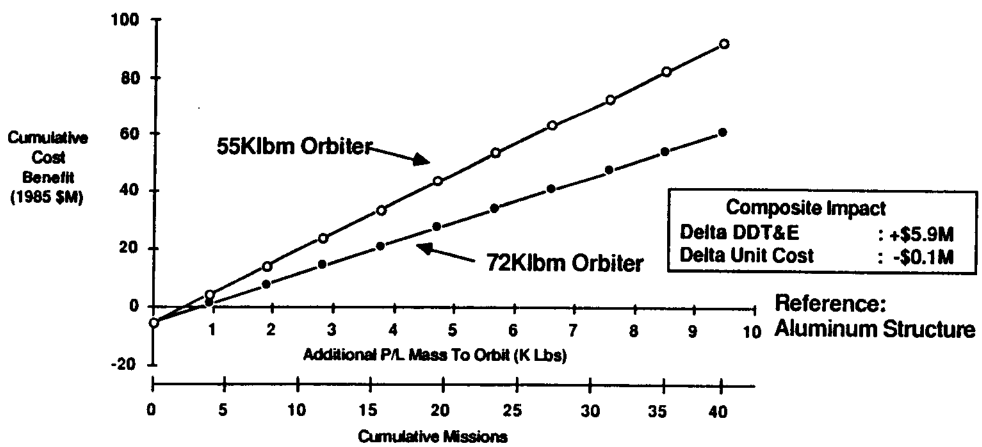


Figure 4.1.2-1 Aluminum vs. Composite Structures Trade

The two plotted lines represent the cumulative cost benefit given a range of Orbiter lift capability of 55K lbs to 72K lbs . The slope of the benefit lines are a combination of the per unit cost difference and the derived P/L benefit of the lighter composite airframe. The stage P/L weight differences (236 lbs per mission) can be translated into deliverable P/L for each of the orbiter performance measures. The additional P/L capability is costed at the cost per pound required to deliver that amount of P/L using the stage with the aluminum airframe ($10.2K /Lb for the 55K Orbiter case and $6.9K /Lb for the 72K Orbiter case). The additional investment in the composite structure is paid back within 3 to 4 missions.

The cost results of replacing the aluminum tanks with aluminum-lithium tanks are shown in Figure 4.1.2-2. The DDT&E and unit cost for the aluminum tanks are $14.6M and $2.4M, respectively. The aluminum lithium tanks exhibit higher DDT&E ($27.5M) and unit costs ($2.9M). The higher DDT&E cost is driven by the probable requirement of performing a dedicated cryogenic proof test with the newer material while avoiding such a test with the aluminum tanks. Additionally, the unit cost difference affects the cost of the ground test hardware. The higher unit cost of the aluminum lithium tanks is due primarily to the higher materials cost. Little difference in fabrication between the two materials is expected at this time.

	55K Orbiter	72K Orbiter
P/L, Aluminum Lithium	12,602 Lbs	18,484 Lbs
P/L, Aluminum	12,245 Lbs	18,127 Lbs
Delta P/L	357 Lbs	357 Lbs
Aluminum Lithium $/Lb (GEO)	$ 9,760 $/Lb	$ 6,654 $/Lb
Aluminum $/Lb (GEO)	$10,045 $/Lb	$ 6,785 $/Lb

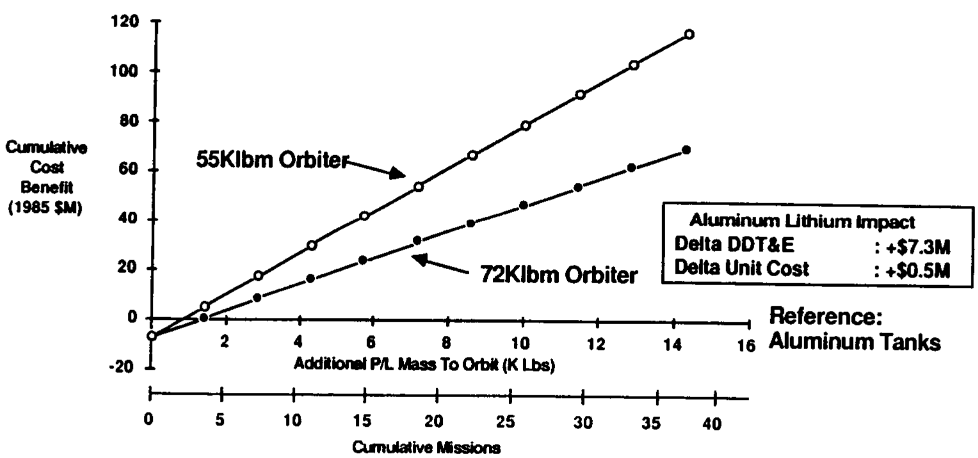


Figure 4.1.2-2 Aluminum vs Al-Li Tanks Trade

The additional DDT&E investment required for the aluminum-lithium tanks is $7.3M. The unit cost delta is approximately $0.5M per set of tanks.

As in the previous trade study results, the two plotted lines represent the cumulative cost benefit given a range of Orbiter lift capability of 55K lbs to 72K lbs. The slope of the benefit lines are a combination of the per unit cost difference and the derived P/L benefit of the lighter aluminum lithium tanks. The stage P/L weight differences (357 lbs per mission) can be translated into deliverable P/L for each of the orbiter performance measures. The additional P/L capability is costed at the cost per pound required to deliver that amount of P/L using the stage with the aluminum tanks ($10.0K/Lb for the 55K Orbiter case and $6.8K/Lb for the 72K Orbiter case). The aluminum-lithium tank payback occurs within 3 to 4 flights.

The cost impact for developing the IOC engine is shown in Figure 4.1.2-3. The DDT&E and unit cost for the RL-10 are $14.8M and $1.7M, respectively. The DDT&E includes primarily ground test hardware and test operations requirements due to integration of the RL-10 to the new expendable stage. The DDT&E cost estimate ($234.8M) for the IOC engine represents a new engine development program. The unit cost estimate for the new engine ($2.2M) is not as significant a cost factor between the two alternative engines.

	55K Orbiter	72K Orbiter
P/L, IOC Engine	13,683 Lbs	19,997 Lbs
P/L, RL-10 Engine	12,245 Lbs	18,127 Lbs
Delta P/L	1,438 Lbs	1,870 Lbs
IOC Engine $/Lb (GEO)	$ 8,990 $/Lb	$ 6,150 $/Lb
RL-10 Engine $/Lb (GEO)	$10,045 $/Lb	$ 6,785 $/Lb

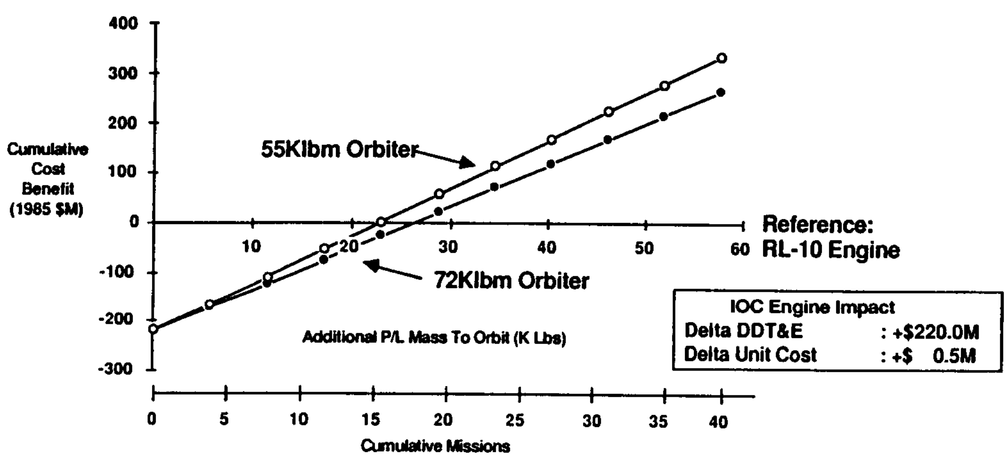


Figure 4.1.2-3 RL10 vs IOC Engine Trade

The additional DDT&E investment required for the IOC engine is $220.0M. The unit cost delta is approximately $0.5M engine.

As in the previous two trade studies, the two plotted lines represent the cumulative cost benefit given a range of Orbiter lift capability of 55K lbs to 72K lbs. The slope of the benefit lines are a combination of the per unit cost difference and the derived

P/L benefit of the performance gains due to the higher Isp of the IOC engine. The stage P/L capability differences (1438 lbs per mission in a 55K Orbiter and 1870 lbs per mission in a 72K Orbiter) can be translated into deliverable P/L for each of the orbiter performance values. The additional P/L capability is costed at the cost per pound required to deliver that amount of P/L using the stage with the RL-10 engine ($10.0K/Lb for the 55K Orbiter case and $6.8K/Lb for the 72K Orbiter case). Due to the higher investment cost in the new engine program the payback of the initial investment is in the 15 to 19 missions range. The overall benefit after 40 missions is much more significant than in the previous trades.

Figure 4.1.2-4 shows the difference in development costs between each of the proposed vehicle enhancements and the existing technology subsystem. Design and qualification of the propellant tanks and the structure will have to be performed independant of the materials used. So, for the tanks and structure the difference in development costs are essentially related to materials characterization and subscale testing. The RL10A already is in production and available; therefore, the IOC engine development cost delta is primarily the development cost of the IOC engine.

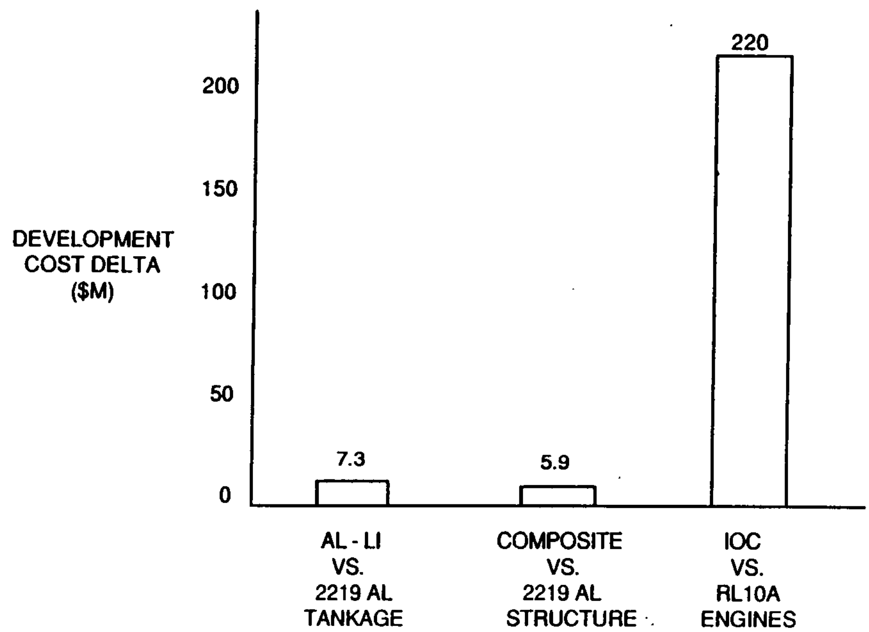


Figure 4.1.2-4 Enhancement Development Cost Deltas

A good indication of the worth of each of the vehicle enhancements is the amount development dollars spent for the performance gained. Figure 4.1.2-5 shows how the enhancements compare on this basis. The best bargain appears to be the Al-Li tanks enhancement. The IOC engine is the highest in terms of cost per lbm of increased performance; however, this enhancement is obviously the single most important upgrade in terms of absolute performance increase.

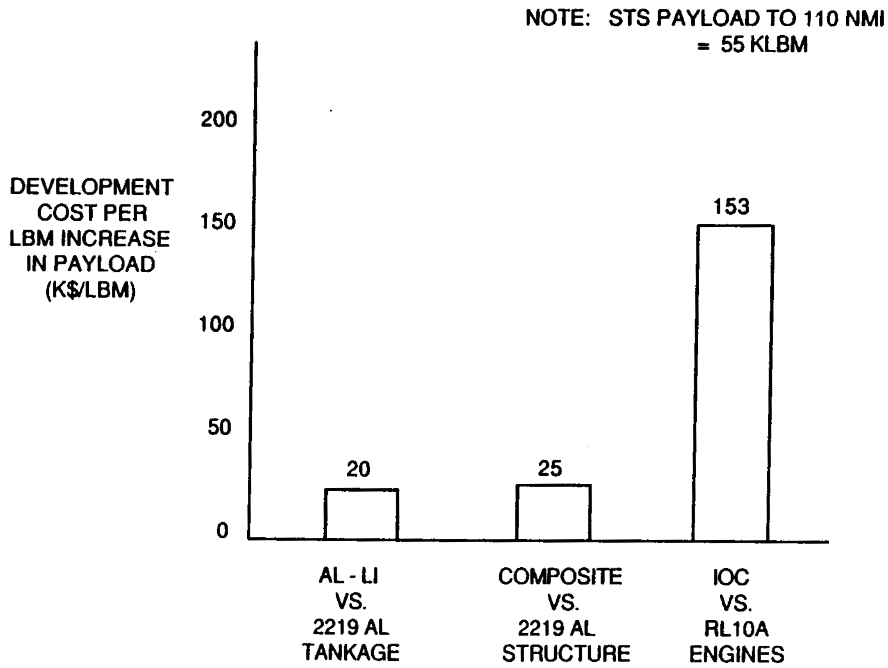


Figure 4.1.2-5 Enhancement Costs Per Lbm P/L Improvement

The conclusions of the cost trade studies on the performance enhancements indicate that the enhancements should be pursued as soon as they are available. The payoff for the IOC engine is in 5 years if the flight rate is 4 per year. The tankage and structure trades both suggest that the enhancements pay for themselves in 5 flights or less and availability of the enhancement is the only other consideration.

The schedule of availability for each of the vehicle enhancements is shown in Figure 4.1.2-6 along with the earlier available subsystem types. Most of the enhancements under consideration could conceivably be made available by 1993 if the go-ahead was in early 1988. The ones in question include the IOC advanced engine and the Al-Li alloy propellant tanks.

The "ultimate" advanced engine would take an estimated 7 1/2 years to fully develop according to Pratt & Whitney; however, presumably an earlier version of this engine (the IOC engine) could be available in 5 years. The new Al-Li alloys under consideration for propellant tanks are presently undergoing materials characterization (which is typically a 5 year period). The final design, development, and qualification of

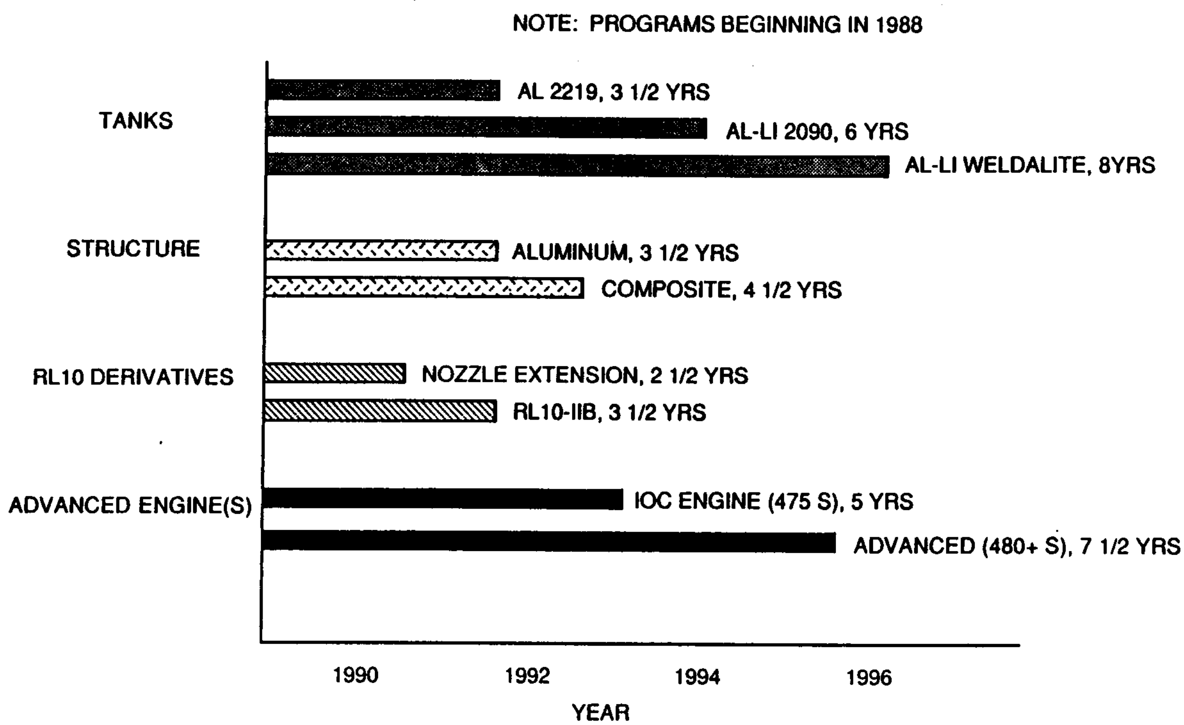


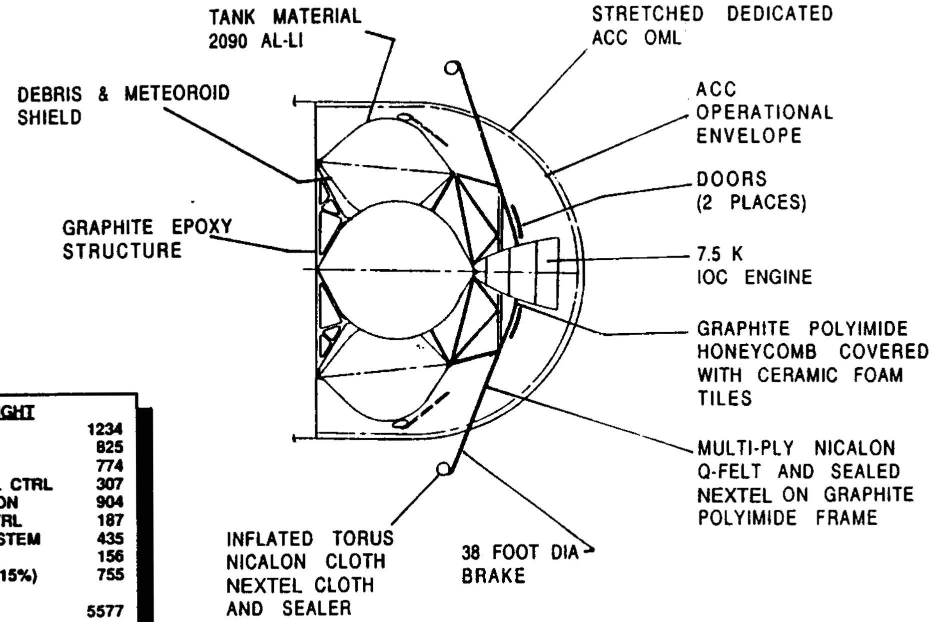
Figure 4.1.2-6 Enhancement Development Times

tanks with these materials must then be performed after the characterizations are complete. These time estimates suggest that these alloys will not be available in 1993.

The recommended conclusions for the initial expendable vehicle are that each of the enhancements examined should be incorporated as soon as possible (depending upon their availability). IOC date, then, determines which enhancements the initial OTV will have.

4.2 GROUND BASED OTV CHARACTERISTICS

During earlier Phase A effort, the ground-based reusable vehicle concept shown in Figure 4.2-1 was developed. The reusable and the expendable concepts have the same structure and same size tanks, but the reusable concept incorporates a new technology, reusable engine. The reusable vehicle offers potential economic advantages over the expendable vehicle, providing that the launch vehicle costs are sufficently low and the launch capability is sufficiently high. In addition, the reusable vehicle could provide an excellent means of demonstrating future technologies such as those required for space basing. These subjects will be discussed in this section.



WEIGHT	
AEROBRAKE	1234
TANKS	825
STRUCTURE	774
ENVIRONMENTAL CTRL	307
MAIN PROPULSION	904
ORIENTATION CTRL	187
ELECTRICAL SYSTEM	435
G. N. & C.	156
CONTINGENCY (15%)	755
DRY WEIGHT	5577
PROPELLANTS, ETC	45424
LOADED WEIGHT	51011

Figure 4.2-1 Ground Based Reusable OTV

4.2.1 REUSABLE VS EXPENDABLE

Weight and performance summaries for both the expendable vehicle baseline and the ground based reusable vehicle concept are shown in Table 4.2.1-1. The performance numbers are given for both a 55 klbm STS and a 65 klbm STS, the latter representing a performance-enhanced vehicle. The standard STS performance to 110 NM had to be adjusted for ACC OTV operations. Because the OTV is deployed at MECO the shuttle's OMS system does not have to inject the stage into park orbit, this is a performance gain for the shuttle (the performance loss to the OTV is accounted for in its mission propellant computations). Shuttle performance groundrules also require sufficient OMS propellant to deorbit the shuttle and cargo bay payload in case of a failed deployment. This groundrule would only apply to the OTV's payload and ASE (since they are the only pieces in the orbiter cargo bay at MECO) which typically totals less than 15 Klb. Finally, the orbiter must fly to the 140 NM OTV park orbit for payload mating which is a performance penalty for the shuttle. When all the above factors are considered for a standard STS lift capability of 55 Klb to 110 NM, the net capacity to MECO is 53460 lb for an ACC OTV mission. For a 65 Klb capacity shuttle, the adjusted ACC OTV lift capability (OTV + payload + ACC + ASE) is 64,290 lb to MECO. The weights remain the same for each stage for the ACC and payload ASE for the two launch weight capabilities. The propellant, payload, and total liftoff weights differ for the two STS capacities.

Figure 4.2.1-1 shows OTV payload delivery capability to GEO as a function of STS delivery capability for the reusable and expendable vehicle concepts generated during this study. By increasing STS lift capability, the OTV concepts can be loaded with more propellant and thus fly heavier payloads to their final destination. The STS lift capability shown corresponds to what the Shuttle can deliver to 110 nmi.

It may be concluded from the figure that the expendable vehicle concept is capable of delivering significantly greater payload to GEO than the reusable concept. This may be crucial if a larger launch vehicle is not available for use with OTV. In other words, large payloads going to GEO may require that the OTV not carry an aerobrake and propellant to return itself to LEO if the mission is constrained by limited STS capacity. Another conclusion is that the cost per pound of payload to GEO for the reusable OTV, including development, production, and operations costs, is higher than for the expendable for OTV class payloads depending upon STS capability. This is discussed in the following paragraph.

Table 4.2.1-1 STS ACC OTV GEO Performance Baseline

WEIGHT SUMMARY IN LBM

	EXPENDABLE (RL10)	REUSABLE (IOC)
ACC	4140	4140
P/L ASE	895	895
OTV ASE	300 (PIDA ONLY)	1333 (EXPEND LH2 TANKS)
OTV DRY	4189	5577
PROPELLANT*	31708 (38678)	33270 (39963)
P/L*	12228 (16088)	8245 (12382)
TOTAL*	53460 (64290)	53460 (64290)

* FOR 55 K STS (65 K STS)

NOTE: OTV MISSION START IS FROM MECO, INITIAL PARK ORBIT IS 140 NMI

OTV + P/L + ASE + ACC = 53460 LBM FOR 55 K ORBITER

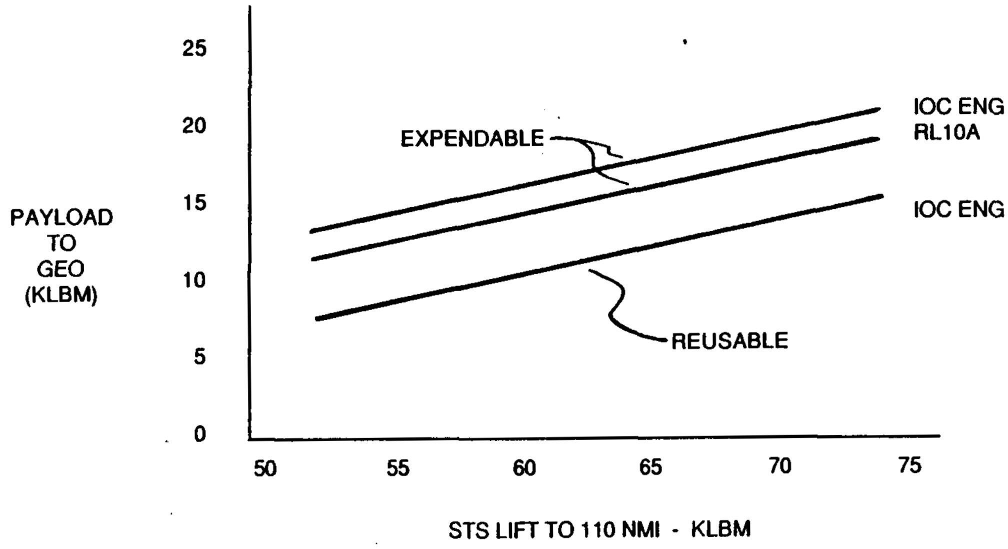


Figure 4.2.1-1 Payload to GEO With STS

The ground based reusable vehicle concept has lower performance in terms of GEO payload than the expendable concept. However, the unit costs associated with expending a vehicle can presumably be significantly reduced on a per-flight basis by the reusable vehicle. Figure 4.2.1-2 shows the payback associated with the reusable concept after the investment is made to develop it. The crossover point compared to the cost of using an expendable vehicle is a function of what the STS lift capability is. In other words, the reusable vehicle carries a proportionately larger payload relative to that of the expendable vehicle for higher STS capacities. The conclusion is that reuse appears attractive if STS capability is greater than 65 Klbm.

4.2.2 TECHNOLOGY DEMONSTRATION OPPORTUNITIES

After initiating the OTV program with perhaps a ground based expendable vehicle there will be opportunities to demonstrate technologies that will be required for the evolution of the OTV to reuse, space basing, aeroassist, etc. These opportunities, as shown in Table 4.2.2-1, will typically be after the completion of a payload delivery mission, for example. The demonstrations will essentially consist of in-space operation of prototype OTV hardware that has "come along for the ride" or other OTV support equipment prototypes that will be used with the post mission OTV for technology demonstration.

- EQUAL CUMULATIVE MASS TO GEO
- $10 M/FLT COST FOR USING REUSABLE VEHICLE (SEE VOL. IX, TABLE 8.2.4-1)
- DELTA DDT & E = $434M ($164M-AERO, $220M-ENGINE, $50M-OTHER SUBSYSTEMS)

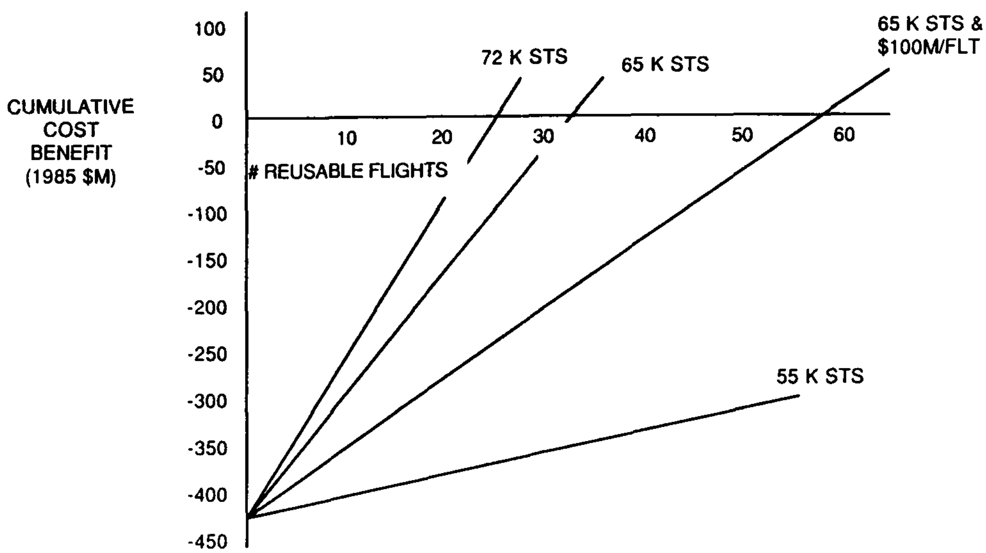


Figure 4.2.1-2 Reusable Vehicle Payback Over Expendable

Table 4.2.2-1 Technology Demonstration Opportunities

ADVANCED MISSION TECHNOLOGIES	LOW RISK VALIDATION METHODS
AEROASSIST	EQUIP EXPENDABLE VEHICLE WITH AEROBRAKE AND GUIDANCE PACKAGE FOR RETURN FOLLOWING DELIVERY MISSION
LONG TERM CRYOGENIC STORAGE	EQUIP EXPENDABLE OR REUSABLE VEHICLES WITH VARIOUS THERMAL CONTROL SYSTEMS AND INSTRUMENTATION FOR POST MISSION LONG TERM SYSTEM EVALUATIONS
FAILURE DETECTION AND ISOLATION	INSTRUMENTED VEHICLE RECOVERED AND RETURNED TO GROUND FOR INSPECTION TO CORRELATE DEGRADATION TRENDS
ON - ORBIT SERVICING	EQUIP G.B. OTV WITH ORU'S (ORBITAL RE-PLACEABLE UNITS) FOR SERVICING DEMON-STRATION USING STS AS PLATFORM WITH EVA AND/OR ROBOTICS/TELEOPS
SPACE BASED REFUELING	RETURN EXPENDABLE TO LEO OR USE G.B. REUSABLE (BEFORE RETURNING TO EARTH) FOR ON-ORBIT REFUELING DEMONSTRATION

4.3 LUNAR VEHICLE DESIGN ISSUES

4.3.1 LUNAR TRANSFER COMPARISONS

A study was performed in order to determine the optimum strategy for delivering payloads to the Lunar surface. Performance calculations were conducted for candidate mission scenarios for the 40 Klbm payload delivery mission.

The direct to surface method consists of using two stages (one of which contains landing legs, radar, etc.) to do a Surveyor type of landing on the Moon without first going into Lunar orbit. The first stage does the first kick from LEO and then returns itself to LEO via aerocapture. The second stage then finishes the transfer, performs the landing, then ascends from the Moon and returns itself to LEO.

The dedicated lander approach uses two transfer vehicles to deliver the 40 Klbm payload and propellant for the lander to Lunar orbit. Then the propellant is transferred to the lander and the payload is delivered to the surface. The lander then returns to Lunar orbit.

A mission scenario was examined that considered a two stage approach in which aerobrake and landing legs would be swapped in Lunar orbit. The first stage would do the initial kick in LEO and the second stage would complete the transfer to Lunar orbit for the swap and subsequent completion of the payload delivery to the Lunar surface. For the return leg, the landing stage would return to Lunar orbit to swap the landing legs back for its aerobrake and return to earth.

The dedicated lander scenario was also examined for use from the Earth-Moon libration point L1. This scenario is identical to the dedicated lander operation described earlier but for lander basing at L1 instead of in Lunar orbit.

The resulting propellant quantities required for each of the mission scenarios are shown in Figure 4.3.1-1. The most economical method of payload delivery to the Lunar surface appears to be the direct transfer to the surface which is depicted in Figure 4.3.1-2. This mission option avoids the logistics problems associated with maintaining a dedicated lander in either Lunar orbit or L1. It also avoids the operations associated with equipment changeout going to and from the Moon.

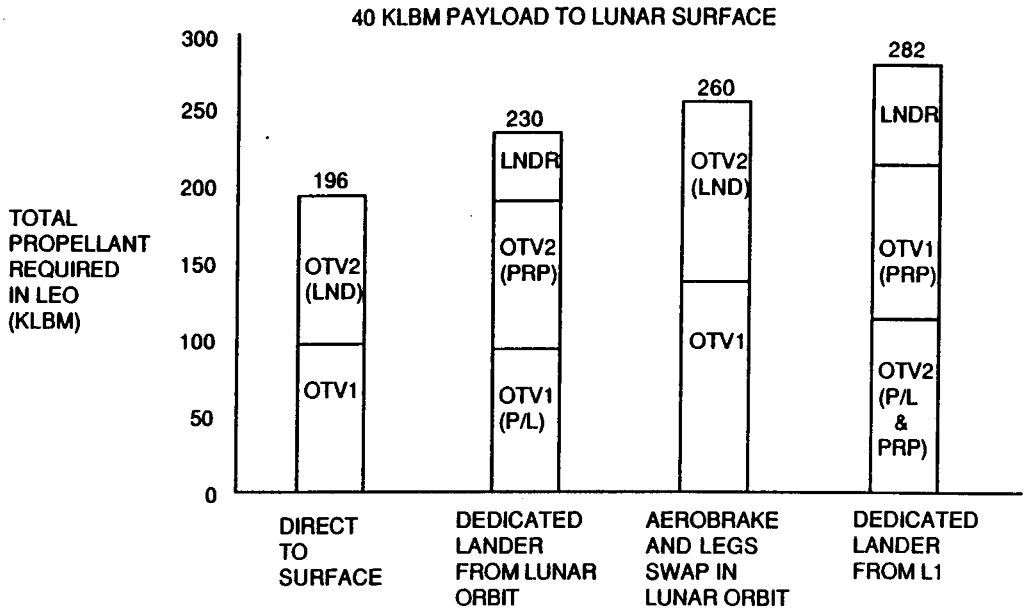


Figure 4.3.1-1 Lunar Transfer Comparisons

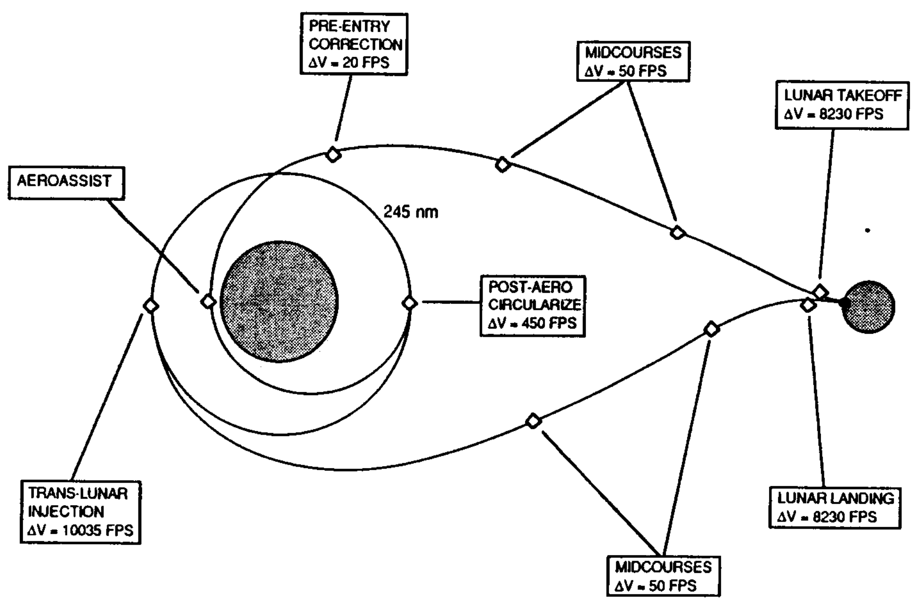


Figure 4.3.1-2 Lunar Profile - Direct Ascent

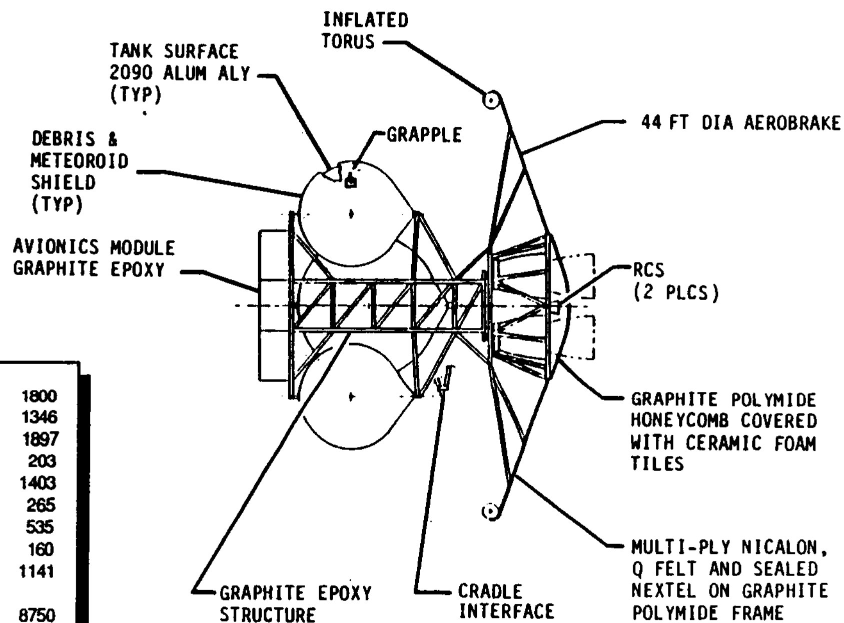
4.3.2 LUNAR TRANSFER VEHICLE DEFINITION

Figure 4.3.2-1 depicts the transfer vehicle concept selected for delivering payloads, OTV's + payloads, etc. toward the Lunar surface, Lunar orbit, or to a libration point. The vehicle was sized such that two stages of this concept (one containing Lunar landing modifications) could deliver the 40 Klbm payload to the Lunar surface and return themselves to LEO. The vehicle is essentially a larger version of the 74 k space based vehicle that was recommended for routine GEO delivery missions. Only the tanks have been upsized for the larger propellant loads. With further vehicle optimization, however, the thrust levels of the engines may need to be uprated for better overall vehicle performance.

4.3.3 LUNAR LANDING CONSIDERATIONS

Several groundrules were assumed to apply to a Lunar landing scenario with an OTV. Some Lunar landings will be manned, thus engine out capability was imposed upon the configuration candidates. In addition, attitude misalignments were not allowed because of the need to descend and land in an upright orientation. For instance, two engines with one engine out would experience an attitude misalignment due to the thrust vector not coinciding with the axis of symmetry.

98000 lbm PROPELLANT CAPACITY



WEIGHT	
AEROBRAKE	1800
TANKS	1346
STRUCTURE	1897
ENVIRONMENTAL CTRL	203
MAIN PROPULSION	1403
ORIENTATION CONTROL	265
ELECTRIC SYSTEMS	535
G,N&C	160
CONTINGENCY (15%)	1141
DRY WEIGHT	8750
PROPELLANTS, ETC	98000
LOADED WEIGHT	106750

Figure 4.3.2-1 98 Klbm Lunar Transfer Vehicle

The thrust level requirements associated with Apollo landings were adopted as groundrules for this study. These included thrust level variation during the landing sequence in order to provide 0.31g at descent ignition to 0.065g at touchdown. Therefore, continuous throttling capability of the main engines is a necessity.

Table 4.3.3-1 shows the weights of OTV, payloads, and propellants at Lunar touchdown for two different missions. Using these weights and the suggested g-level at touchdown from the Apollo landing thrust requirements (0.065g), the minimum thrust levels for Lunar landing vehicle were derived. Likewise, the descent ignition weights and 0.31g were used to obtain the maximum thrust levels.

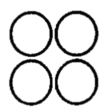
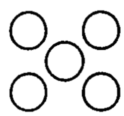
In order to accommodate these thrust level requirements, three (in-line), four, and five-engine configuration candidates were considered for Lunar landing missions. A single engine cannot meet the engine out requirement and two and three (cluster) engine configurations would cause an attitude misalignment upon engine-out. Engine systems with greater than five engines were not considered because of increased weight, decreased reliability, large engine pattern, increased costs, and increased complexity.

Table 4.3.3-1 Thrust Levels for Lunar Landing

	15K MANNED	40K DELIVERY
OTV AND PROPELLANT WEIGHT AT TOUCHDOWN	11.7K + 24.8K = 36.5KLBM	11.7K + 13.2K = 24.9KLBM
TOTAL TOUCHDOWN WT.	15K + 36.5K = 51.5KLBM	40K + 24.9K = 64.9KLBM
MINIMUM THRUST	0.065g(51.5K) = 3.3KLBF	0.065g(64.9K) = 4.2KLBF
DESCENT IGNITION WT.	89.5KLBM	112.8KLBM
MAXIMUM THRUST	0.31g(89.5) = 27.7KLBF	0.31g(112.8) = 35KLBF

RESULTS: CONTINUOUS THRUST RANGE REQUIREMENTS = 3.3KLBF TO 35KLBF

Table 4.3.3-2 Lunar Landing Engine Configurations

MAIN ENGINE CONFIGURATION	MISSION RELIABILITY (10 BURNS)	THRUST RANGE PER ENGINE	THROTTLING RATIO	REMARKS
	.9919	1.1K - 35KLBF	32:1	- HIGH THRUST REQUIRED - LARGE THROTTLING RATIO - WIDE PATTERN
	.9864	0.8K - 17.5 KLBF	21:1	- SMALLEST PATTERN - GOOD RELIABILITY - GROWTH FROM TWO ENGINES - SMALLER THROTT. RATIO
	.9797	0.66K - 11.7KLBF	18:1 (10:1)*	- LOWEST RELIABILITY - LARGEST PATTERN - COMPLEX DESIGN AND CONTROL - SMALLEST THROTT. RATIO

* FINAL DESCENT AND LANDING WITH THREE ENGINES

Four engines were chosen for Lunar landing applications based upon the assessment results presented in Table 4.3.3-2. The system reliability of four engines is between that of three and five engine systems. However, the maximum thrust requirement and throttling ratio are much reduced from those of the three engine system and not significantly larger than those of the five engine system. The four engine system was also chosen because it has the smallest pattern for packaging within a circular perimeter and may offer the best growth path from an existing two engine system.

A five engine configuration would have the lowest required throttling ratio if one of the landing ground rules was changed to allow operating engines to be shutdown. If two opposing outboard engines of the five engine pattern were shutdown during the Lunar descent, the throttling range of the system would be reduced to 10:1 and there would still be engine out capability with the remaining three engines. The safety and reliability implications of shutting down operating engines should be assessed, however, in choosing five engines rather than four.

The modifications shown in Table 4.3.3-3 must be made in converting a space based OTV from orbit-to-orbit delivery capability to delivering payloads to the Lunar surface. The two additional engines with increased thrust and continuous throttleability are needed for a lunar landing. In addition, landing legs, radar, and landing software must be added in order to accommodate the landing scenario. For the return to LEO from the moon, slightly beefed up structure and thicker TPS on the aerobrake are required compared to the vehicle only returning from GEO or an initial kick towards the moon (see Section 6.6). Meteoroid protection requirements are not presently thought to differ much from those for LEO-GEO transfer.

Table 4.3.3-3 Lunar Landing Deltas

ITEM	DELTAS (LBM)
ADD 2 ENGINES + PLUMBING	782
AEROBRAKE	573
RADAR	69
LANDING LEGS	1495
METEOROID SHIELDING	0
LANDING SOFTWARE	SMALL
PRIMARY STRUCTURE	64
	2983

The concept shown in Figure 4.3.3-1 was created by incorporating the Lunar landing modifications to the 98 Klbm Lunar transfer vehicle. The 98 Klbm transfer vehicle and this lander concept would together be capable of delivering 40 Klbm to the Lunar surface, then both vehicles would return themselves to LEO. The figure also shows a design concept for landing legs that fold under the aerobrake hard shell into a diameter compatible with delivery to LEO in the STS

WEIGHT	
AEROBRAKE	2298
TANKS	1873
STRUCTURE	2726
ENVIRONMENTAL CTRL	203
MAIN PROPULSION	2083
ORIENTATION CONTROL	265
ELECTRIC SYSTEMS	535
G, N & C	219
CONTINGENCY (15%)	1530
DRY WEIGHT	11732
PROPELLANTS, ETC	98000
LOADED WEIGHT	109732

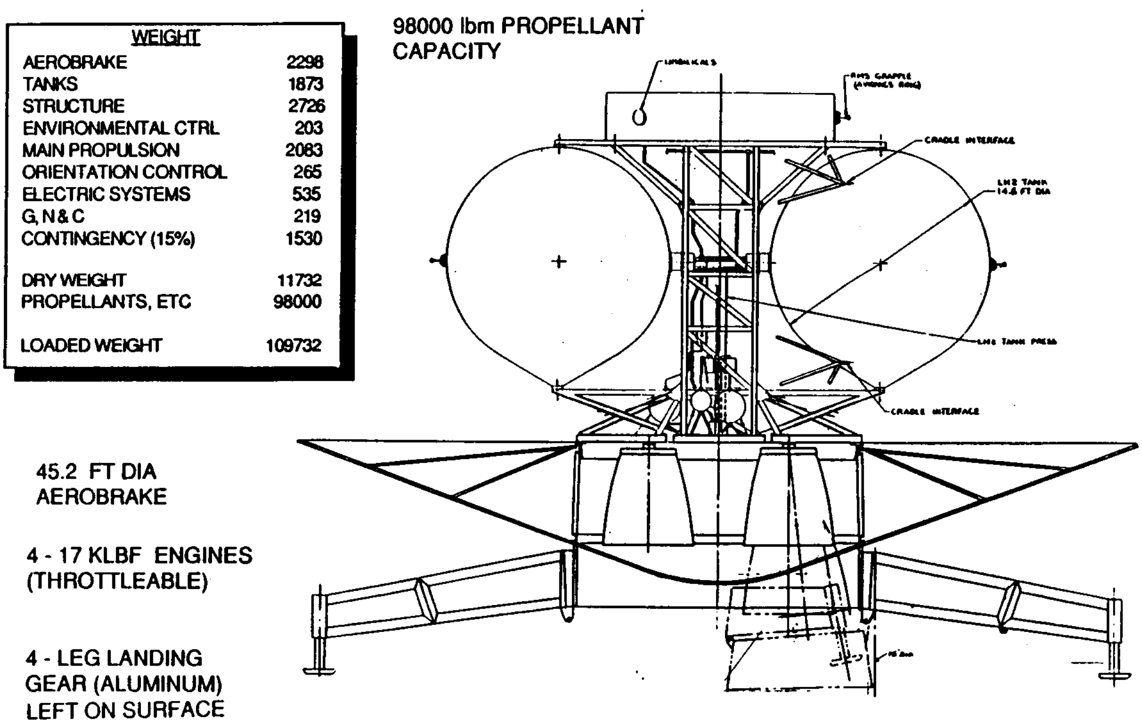


Figure 4.3.3-1 98 Klbm Lunar Lander

cargo bay. Therefore, the leg assembly could be attached to the vehicle after initial launch of both. The aluminum structure of the four legs was designed to support the landing of the heaviest payload (40 klbm). The leg assembly could be fashioned to be attachable to the aerobrake structural ring or through the aerobrake directly to the stage structure.

Figure 4.3.3-2 shows the arrangement of the four engines recommended for lunar landings. The aerobrake doors are intended to rotate open to positions parallel to the engines' axes, and then withdrawn into the engine compartment alongside the engines during engine nozzle extension, engine operation, and nozzle retraction.

A dedicated Lunar lander concept, shown in Figure 4.3.3-3, was sized for the purpose of remaining in Lunar orbit and delivering to the surface the payload that the 98 Klbm vehicle could deliver to Lunar orbit. In other words the dedicated lander would be placed into Lunar orbit and serviced there (or perhaps on the surface) for use in transferring payloads between Lunar orbit and the Lunar surface. This scenario implies that the dedicated lander is refueled in either Lunar orbit or on the surface of the moon. The 98 Klbm transfer vehicle is capable of delivering about 42 Klbm from LEO to Lunar orbit; therefore, the dedicated lander was sized to deliver this size payload to the Lunar surface and then return itself to Lunar orbit.

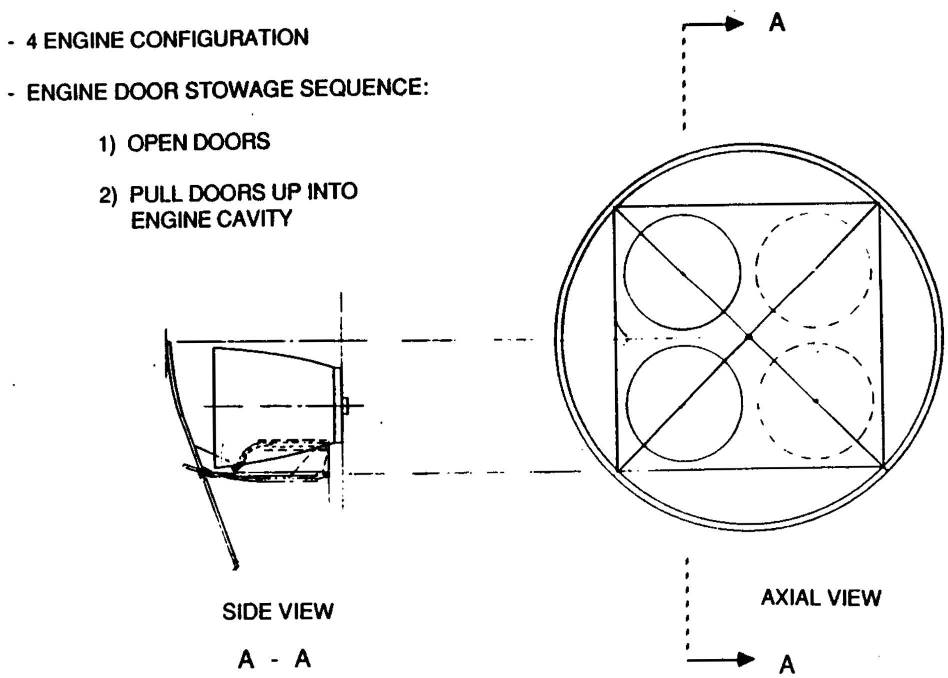
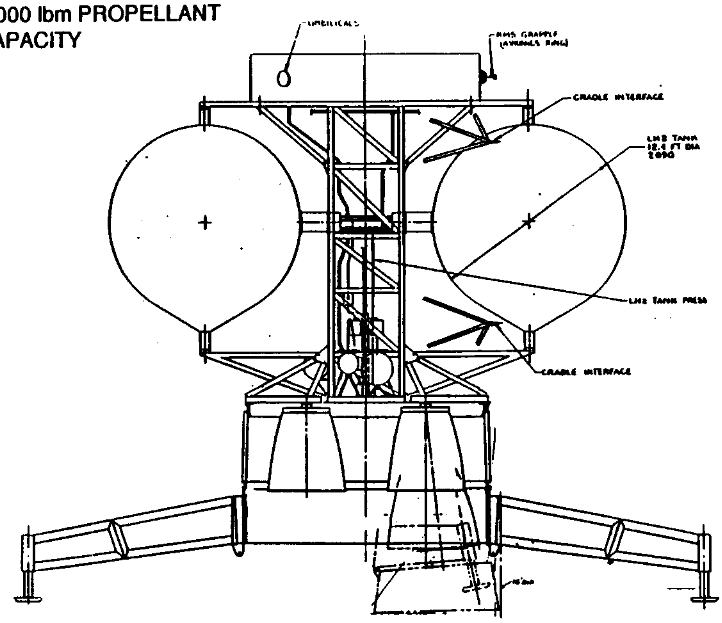


Figure 4.3.3-2 Lunar Lander Engine Compartment

WEIGHT	
TANKS	1087
STRUCTURE	2726
ENVIRONMENTAL CTRL	203
MAIN PROPULSION	2083
ORIENTATION CONTROL	265
ELECTRIC SYSTEMS	535
G,N&C	219
CONTINGENCY (15%)	1068
DRY WEIGHT	8186
PROPELLANTS, ETC	44000
LOADED WEIGHT	52186

44000 lbm PROPELLANT
CAPACITY



4 - 17 K ENGINES
(THROTTLEABLE)

4 - LEG LANDING
GEAR (ALUMINUM)

Figure 4.3.3-3 Dedicated Lunar Lander

The selected baseline Lunar transfer vehicle (with 98Klbm loaded propellant) was used in determining payload capabilities in performing Lunar missions in various ways. These options are shown in Figure 4.3.3-4 along with the payload amounts to the surface that correspond to each of these options. In each case, the OTV's providing transportation return themselves to LEO. Wherever a refueling quantity is shown, this amount of propellant was assumed to be available at the location indicated, either via propellant hitchhiking on another flight, scavenging unused propellant from a previous OTV, etc. In addition to the usage of the 98 klbm size transfer vehicle and 98 klbm lander, dedicated lander concepts are shown delivering payloads to the surface (from Lunar orbit or L1) and then returning to their basing location.

Performance parametrics for the 98 klbm transfer vehicle and 98 klbm lander are shown in Figure 4.3.3-5. The payload weights are given as a function of loaded propellant for the 98 klbm capacity vehicle. Two cases are shown for delivery to the Lunar surface. One case is for round trip of the payload to and from the surface back to LEO. The other case is for payload delivery to the surface and return of the OTV to LEO. Both options use one transfer vehicle and one landing vehicle. The third case is for delivery capability of one 98 klbm transfer vehicle from LEO to Lunar orbit.

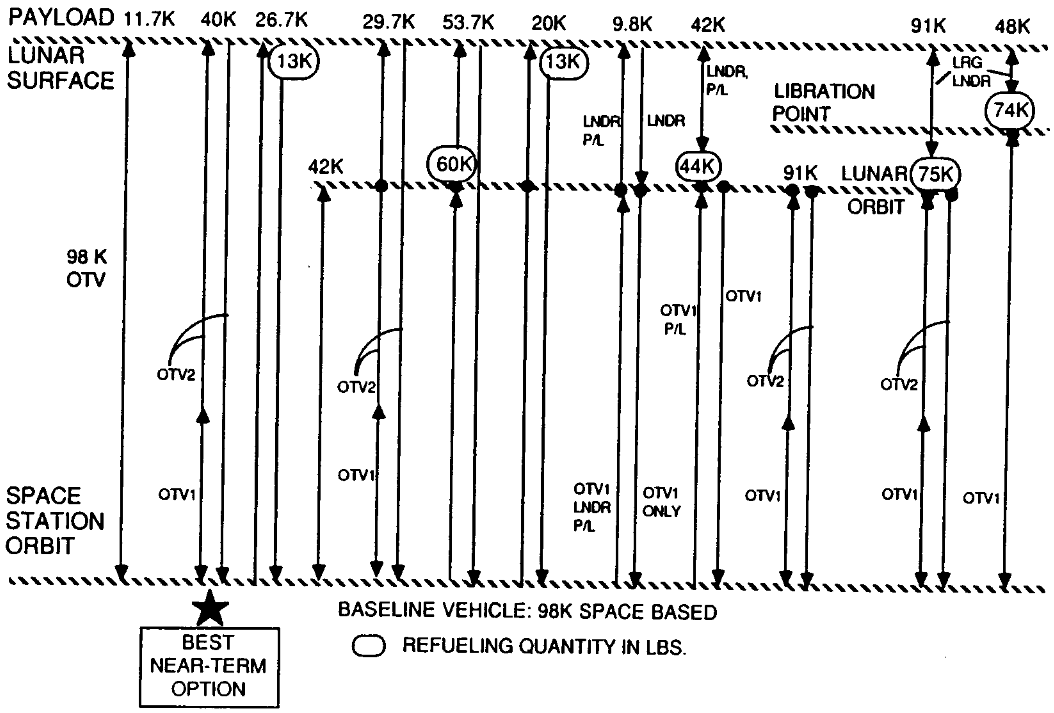


Figure 4.3.3-4 Lunar Delivery Options

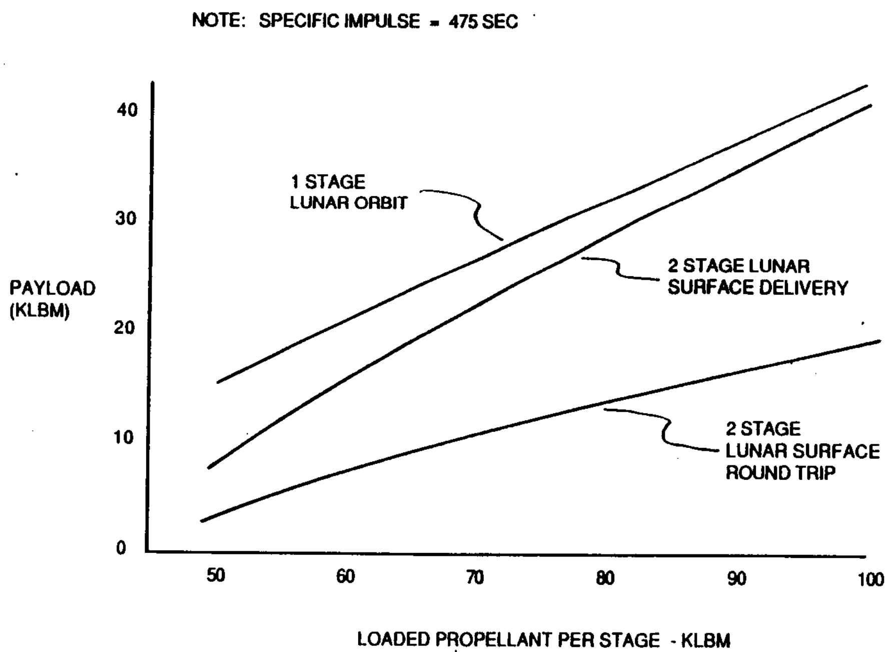
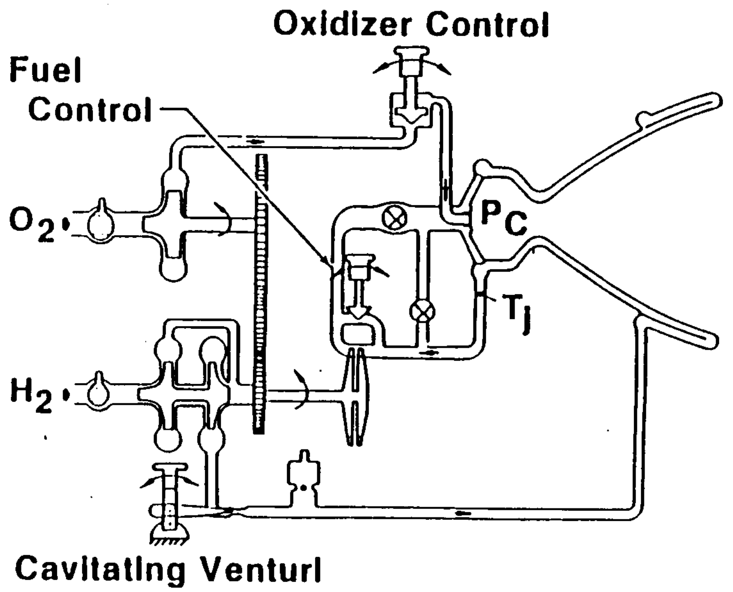


Figure 4.3.3-5 Lunar OTV Performance

4.3.4 CRYO ENGINE THROTTLING FOR LUNAR LANDING

Cryogenic engine technology should not be taken for granted for the Lunar landing mission. The engine configuration trade study suggests that for an engine pattern that meets the ground rules a throttling range of about 20:1 is required (18:1 for five engines, 21:1 for four). Pratt & Whitney has successfully demonstrated a 10:1 throttling range with an RL10A-3-7 with no major engine modifications required. However the 20:1 range would require changes to this engine configuration in order to provide for smooth combustion throughout the range of thrust. For throttling ratios of greater than 10:1 a heat exchanger is likely to be required in order to gasify the oxygen before it reaches the injector (downstream of the oxygen turbopump shown in Figure 4.3.4-1) in order to prevent instabilities in combustion. With low thrust operation of the engine, the pump discharge pressure is relatively low. Thus, the delta P across the injector may be too low to prevent feedback from the combustion chamber (pressure fluctuations propagating upstream into the feed system); therefore the need to gasify it upstream of the injector.



Oxidizer Control
f (Throttle - Pumped Operation)
f (P_C and T_j in THI)
Fuel Control
f (Throttle - Pumped Operation)
(Closed in THI)
Cavitating Venturi
f (Throttle - Pumped Operation)
(Open in THI)

Figure 4.3.4-1 RL10A-3-7 Propellant Flow Schematic

So, the current RL10 engine cycle is capable of modification to perform 20:1 throttling. But, this throttling range may not be necessarily of a purely continuous nature. Due to a thrust range discontinuity caused by the required oxygen phase change, the cycle will not allow unlimited up and down throttling through this discontinuity. For example, between 25 and 100% of full thrust, liquid oxygen is supplied to the injector and sufficient upstream pressure is provided by the turbine discharge for stable combustion. Between 5 and 20% of full thrust the turbine discharge pressure is too low to provide stable combustion with liquid oxygen, therefore a heat exchanger is needed to provide gaseous oxygen to the injector and the combustion chamber. In the region between 20 and 25% of full thrust a discontinuity exists due to the phase change of oxygen. Operation in this range, either continuous or repeated, is not recommended since damage to the engine could occur due to the unstable nature of the combustion.

For the Lunar landing scenario with all four engines operating, no problem exists with this thrust range discontinuity because once the initial descent burns (relatively high thrust) are completed, the engines' thrust level is dropped to a range that would accommodate hover and final descent. This throttling down corresponds to passing through the phase change discontinuity and into the gaseous oxygen operation range (the 5 to 20% range) as shown in Figure 4.3.4-2. The problem results when an engine-out condition occurs and the hover/final descent thrust range for the remaining engines spans the thrust discontinuity region. This is unacceptable from an engine life and reliability standpoint since the ability to throttle up and down through this thrust range repeatedly is important in a controlled landing.

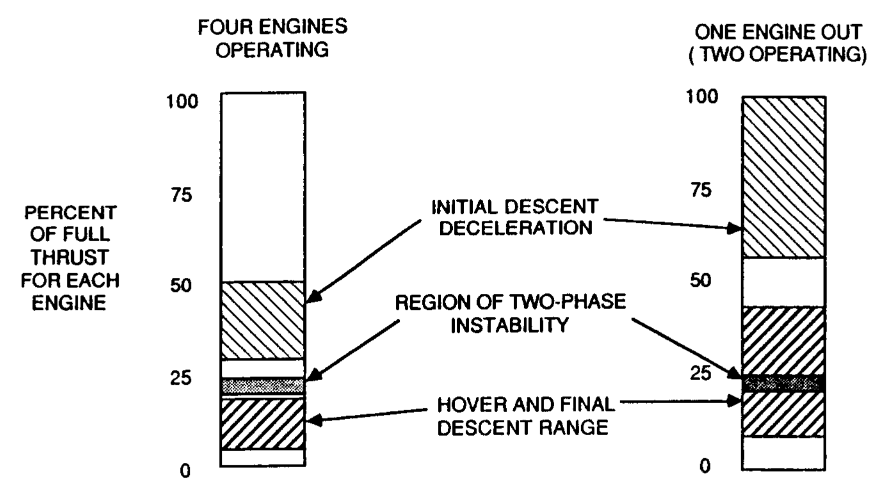


Figure 4.3.4-2 Expander Cycle Throttling Discontinuity

The possible solutions to the thrust range discontinuity problem are as follows:

a. Modify the heat exchanger circuit and engine control system to accommodate throttling through the thrust discontinuity without causing unacceptable instabilities and chugging. This implies perhaps a dual path for the oxygen between the turbopump and the injector -- one for liquid oxygen, and one for the heat exchanger loop with gaseous oxygen.

b. Design the mission operations so that the need to pass through the thrust discontinuity repeatedly in an engine out condition can be avoided or minimized (essentially restricting the landing thrust range flexibility). This would probably mean a performance degradation should be expected and additional propellant (contingency) may be required.

c. Change the groundrules on engine-out so that when it occurs the contingency operation requires return to Earth or abort to Lunar orbit, and not successful landing on the moon. (This would also relieve the no-attitude-misalignment criteria upon engine-out and then perhaps drive the engine configuration design back to two engines). Abort to the Lunar surface may be an alternative for an unmanned mission.

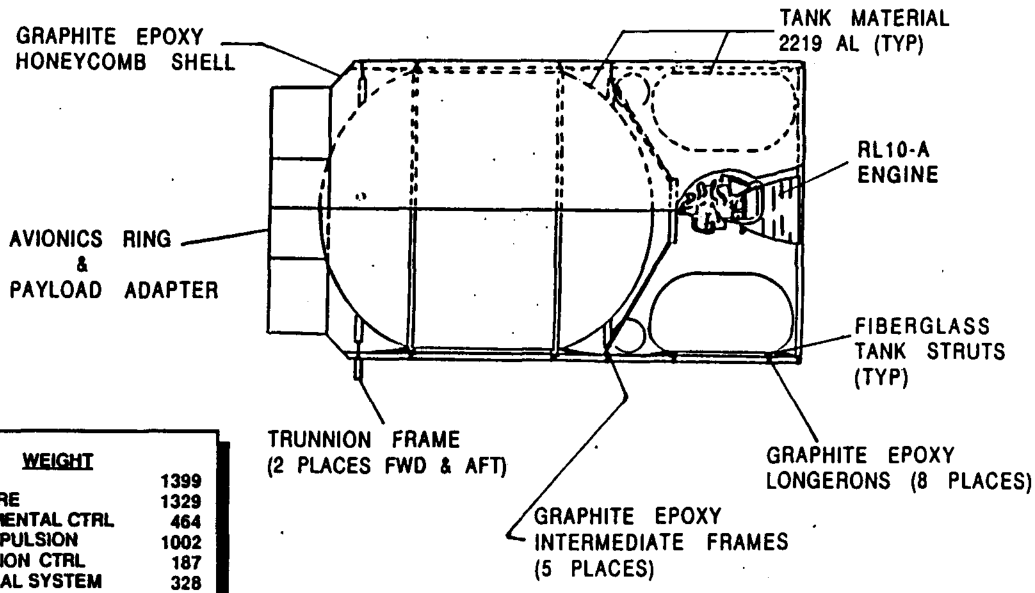
d. Develop an advanced engine cycle (such as Aerojet TechSystems has proposed) that gasifies oxygen at all thrust levels and thus provides full thrust range continuous throttling. This requires an advanced engine development for a cryogenic space engine cycle that is significantly altered from the RL10 cycle that exists today.

e. Use six main engines (instead of four) in order to provide for engine-out capability, remain between 5 and 20% of full thrust for hover and final descent, and to keep individual engine thrust level less than 17kbf.

4.4 SHUTTLE "C" EXPENDABLE OTV

In the event that a large cryogenic upper stage is required to be launched from the ground in an expendable launch vehicle with a 15 ft diameter constraint (e.g. Shuttle "C"), the concept shown in Figure 4.4-1 may be optimum. The tandem toroid configuration (LOX contained in the toroidal tank) is the shortest arrangement that can be achieved with LOX and hydrogen in a 15 ft cargo bay. Short length is even more essential in an increased payload capacity launch vehicle (with 15 ft diameter and 60 ft length constraints) than it is in the Orbiter bay since volume constraints are more pronounced with the increased payload capability. Length is the most important cost driver in terms of the resulting number of required STS flights, etc. from previous mission capture analyses (see Vol. IX). Therefore, the emphasis upon short length is necessary in this situation.

A single engine was chosen for this unmanned, expendable vehicle application. Not only does the single engine fit into the minimum length configuration, but it provides the maximum performance of any engine configuration.



WEIGHT	
TANKS	1399
STRUCTURE	1329
ENVIRONMENTAL CTRL	464
MAIN PROPULSION	1002
ORIENTATION CTRL	187
ELECTRICAL SYSTEM	328
G. N. & C.	182
CONTINGENCY	734
DRY WEIGHT	5625
PROPELLANT, ETC	58924
LOADED WEIGHT	64549

Figure 4.4-1 Shuttle"C" Expendable OTV (15ft dia)

With a 100 Klbm launch vehicle payload capability to LEO, the concept is capable of delivering 26000 lbm to GEO with an RL10A engine. This performance includes 10 klbm for ASE. Unless the vehicle would ever need to carry men and therefore be man-rated, the single engine arrangement is the highest performance candidate.

If Shuttle "C" comes into existance, it will provide a much larger payload capability to LEO than is presently available. Current estimate is 100 klbm. With this in mind, expendable upper stages that match this lift capability may be highly desirable. Figure 4.4-2 shows the payload to GEO as a function of stack weight for both the Centaur G' and the Shuttle "C" OTV concept.

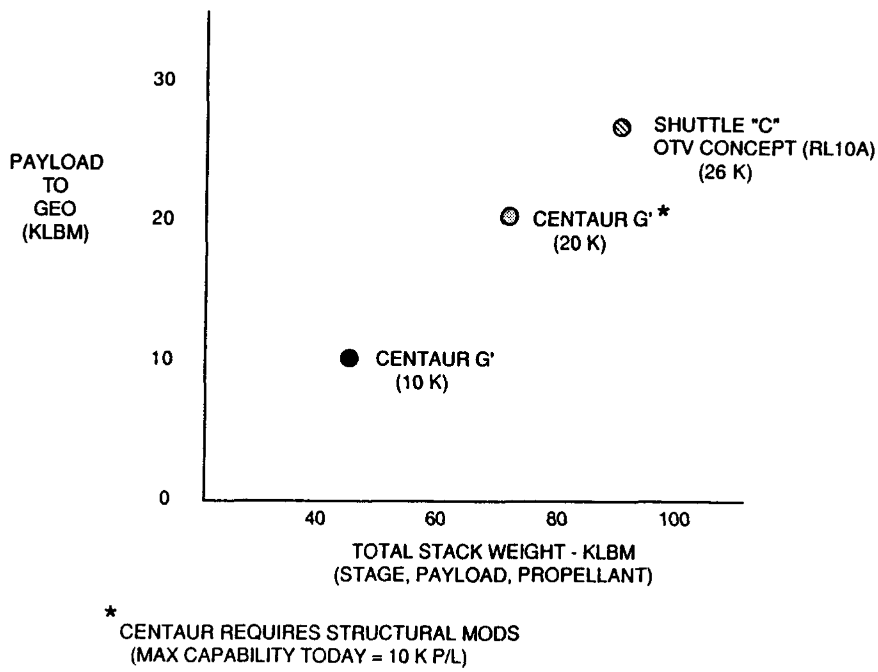


Figure 4.4-2 Expendable Vehicle Comparison

5.0 STRUCTURAL ISSUES

The structural issues addressed during this study extension are listed below:

ACC Expendable OTV Definition
- Airframe
- Enhancements - tanks, engine
- Meteoroid shielding
- Composite ACC
- Battery selection

Ground Based Reusable OTV Updates
- Aerobrake
- Meteoroid shield

LCV Expendable OTV Issues
- LCV OTV Concept Definition
- ASE for LCV (Side mount and In-line)
- Airframe Analysis

5.1 ACC EXPENDABLE OTV DEFINITION

The general arrangement and weight breakdown for our selected expendable OTV transported in the ACC are shown in Figure 5.1-1. Table 5.1-1 shows additional detail on the stage weights. The expendable OTV is based on the same

WEIGHT	
TANKS	1106
STRUCTURE	650
ENVIRONMENTAL CTRL	246
MAIN PROPULSION	944
ORIENTATION CTRL	187
ELECTRICAL SYSTEMS	328
G. N. & C.	182
CONTINGENCY (15%)	540
DRY WEIGHT	4189
PROPELLANTS, ETC	45424
LOADED WEIGHT	49613

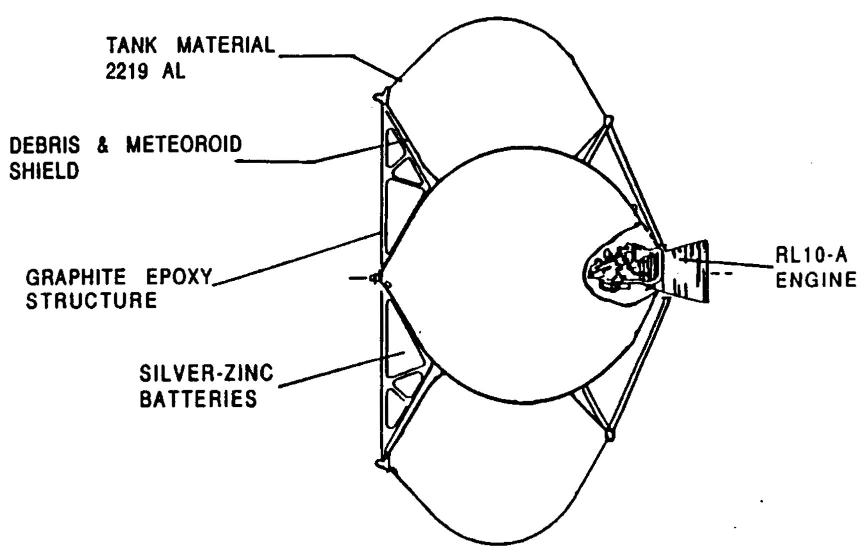


Figure 5.1-1 ACC Expendable OTV Baseline

arrangement as the ground based reusable OTV, i.e., four-tank cryogenic single engine configuration. Where applicable, many of the same components from the reusable OTV are used on the expendable vehicle, e.g., composite airframe, propulsion feed system, avionics equipment, and thermal control.

WBS GROUP	WEIGHT (LB)
2 STRUCTURES	878
3 PROPELLANT TANKS	1272
4 PROPULSION FEED SYS	698
5 MAIN ENGINES	388
6 REACTION CONTROL SYSTEM	215
7 G. N. & C.	128
8 COMM & DATA HNDLG	81
9 ELECTRICAL PWR	377
10 THERMAL CONTROL	153
11 AEROBRAKE	000
DRY WEIGHT	4189
12 FLUIDS	
RESIDUAL-LH2	96
RESIDUAL-LO2	579
HYDRAZINE	400
PRESSURANT	14
COOLANT	00
INERT WEIGHT	5278
USABLE MAIN PROP.	
FUEL-LH2 w/FPR	6342
OXIDIZER-LO2 w/FPR	37993
IGNITION WEIGHT	49613
MASS FRACTION	
44335 MAIN PROP W/FPR / 49613 IGNITION WEIGHT	.89

Table 5.1-1 ACC Expendable Weight Summary

The major differences from the ground based reusable concept are: no aerobrake, Al 2219 tanks instead of Al-Li 2090 tanks, an RL10-A engine, and Ag-Zn batteries in place of the fuel cell system. Some GN&C equipment has been removed, or replaced by a smaller system. The total dry weight of the ACC expendable OTV is 4189 lb.

5.1.1 AIRFRAME CONCEPTS

The original 2219 aluminum airframe concept is a multi-member truss work based on the volume and weight efficient principals suggested by Larry Edwards (NASA HQ). Figure 5.1.1-1 shows a view of the airframe and some typical cross sectional views of the builtup members. Each member has been sized by a NASTRAN model based on the loading conditions and a FS of 1.4, and then checked for buckling and deflection. The truss work consists of individual builtup sections composed of "T's" and a web plate which are fastened together by rivets. The sections are then joined by splice plates and welded to form the entire structure. The airframe weighs 684 lb, including fittings and attachments.

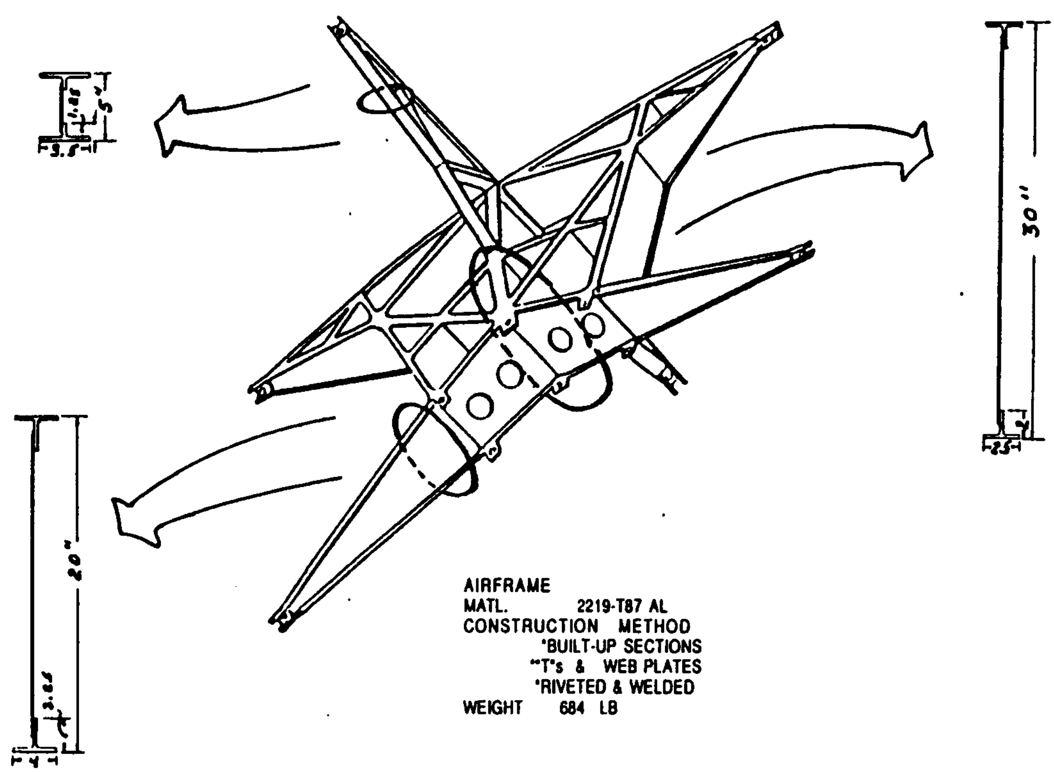


Figure 5.1.1-1 Original Aluminum Airframe Design

As part of the weight optimization effort, the airframe structural analysis was recalculated using Graphite/Polyimide (Gr/Pi) and Polymethacrylimide foam. The analysis was based on the same NASTRAN model, loading conditions, and SF as the aluminum airframe, and utilized the Gr/Pi and foam material properties. Figure 5.1.1-2 shows a view of the airframe and some typical cross sectional views of the builtup members. The truss work consists of individual builtup sections composed of a foam core and bonded face sheets. To form the entire structure, the sections are joined together by overlaid and bonded Gr/Pi splice plates. The airframe weighs 454 lb, including fittings and attachment. 230 lb are saved by using a composite structure instead of the similar aluminum structure. Since the cost and schedule impacts of using the composite structure are minimal, it was chosen over the aluminum.

5.1.2 TANKAGE AND ENGINE ENHANCEMENTS

Table 5.1.2-1 shows the tankage and engine enhancement candidates and weight breakdown for the ACC expendable OTV. The first modification to be made to the vehicle is the replacement of the Al 2219 tanks with Al-Li 2090 tanks which results in a dry weight saving of 349 lb. The second modification incorporates an IOC engine into the propulsion system which saves an additional 78 lb dry weight from the baseline vehicle in addition to increasing Isp considerably.

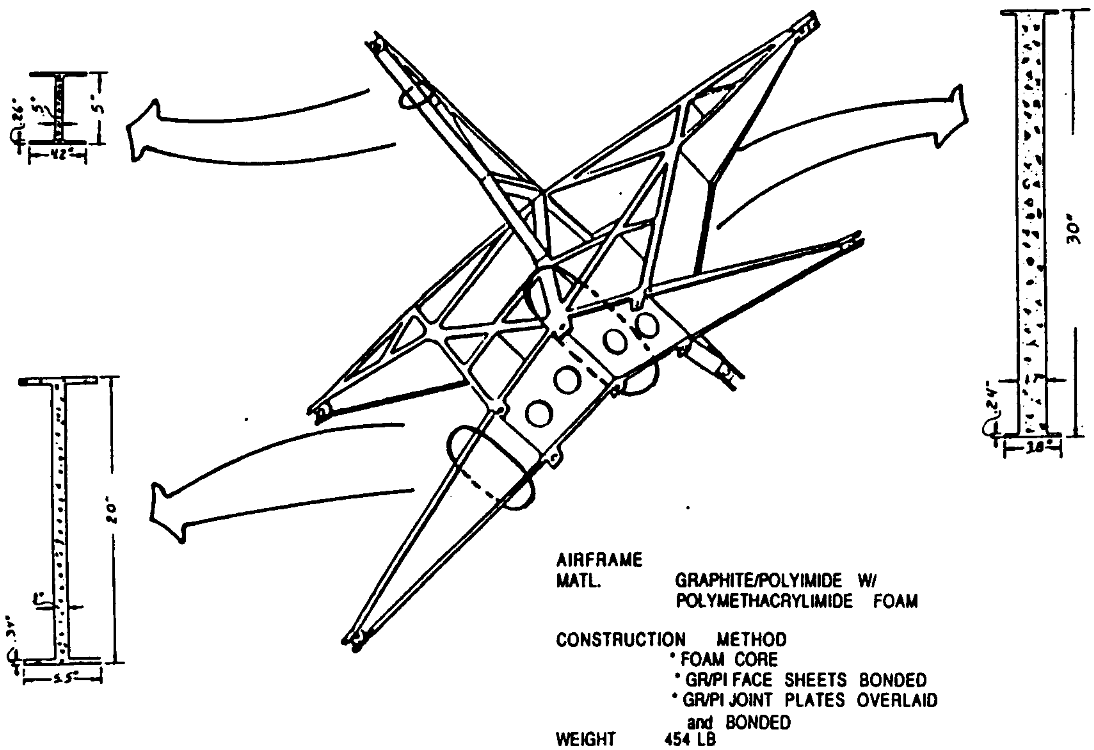


Figure 5.1.1-2 Composite Airframe Design Concept

Table 5.1.2-1 ACC Expendable Enhancements

COMPONENTS	BASELINE WEIGHT (LB)	AL-LI TANKS WEIGHT (LB)	IOC ENGINE AL-LI TANKS WEIGHT (LB)
TANKS	1106	799	799
STRUCTURES	650	650	650
ENVIRONMENTAL CTRL	246	246	246
PROP. w/o ENGINE	607	607	607
MAIN ENGINE	337	337	272
ORIENTATION CTRL	187	187	187
ELECTRICAL SYSTEM	328	328	328
G. N. & C.	182	182	182
CONTINGENCY	546	504	494
DRY WEIGHT	4189	3840	3762
DELTA		-349	-424

REMARKS:

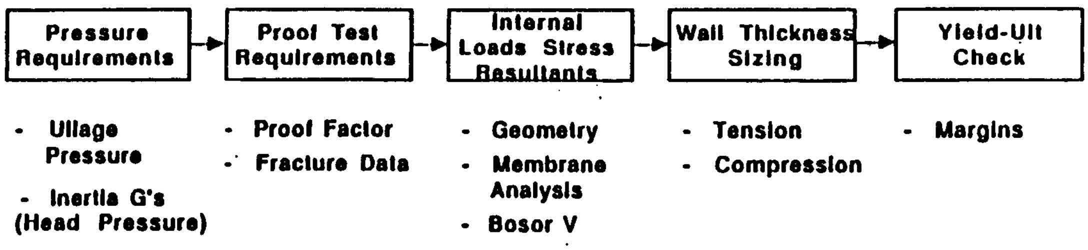
BASELINE COMPOSITE AIRFRAME
2219 AL TANKS
RL10-A ENGINE

ENHANCEMENT #1 - REPLACE 2219 AL TANKS WITH 2090 AL-LI TANKS
NO OTHER CHANGES

ENHANCEMENT #2 - REPLACE RL10-A ENGINE WITH IOC ENGINE
REPLACE 2219 AL TANKS WITH 2090 AL-LI TANKS

The procedure for determining propellant tank wall thickness is shown in Figure 5.1.2-1. The tank maximum operating pressure (consisting of ullage and inertial head) are multiplied by the proof test factors and divided by the fracture toughness ratio (FTR). The proof test factor is adjusted for temperature effects and the specified number of cycles while the FTR is adjusted for temperature. Figure 5.1.2-1 also shows the calculation results for the required proof test pressures.

- **Design Process**



- **Proof Pressure (Pp in Psig)**

Tank	Pp	= P (Limit Flight)	X Proof Factor	÷ Fracture Toughness Ratio
LO2	49	39	1.42	1.12
LH2	26	22	1.42	1.20

Figure 5.1.2-1 Main Propellant Tanks

Figure 5.1.2-2 shows the results of the LO2 tank stress analysis (using the BOSOR shell program) including capability margin, membrane force, and wall thickness. The tank was originally sized using AL 2219 and a 0.025-in. minimum gage was recommended. As a weight optimization alternative, Al-Li 2090 was considered and the minimum gage was reduced to 0.018-in.

Figure 5.1.2-3 shows the results of the LH2 tank stress analysis (also using the BOSOR shell program). Like the LO2 tank analysis, the tank was originally sized using AL 2219 and a 0.025-in. minimum gage was recommended. When Al-Li 2090 was considered the minimum gage was reduced to 0.015-in.

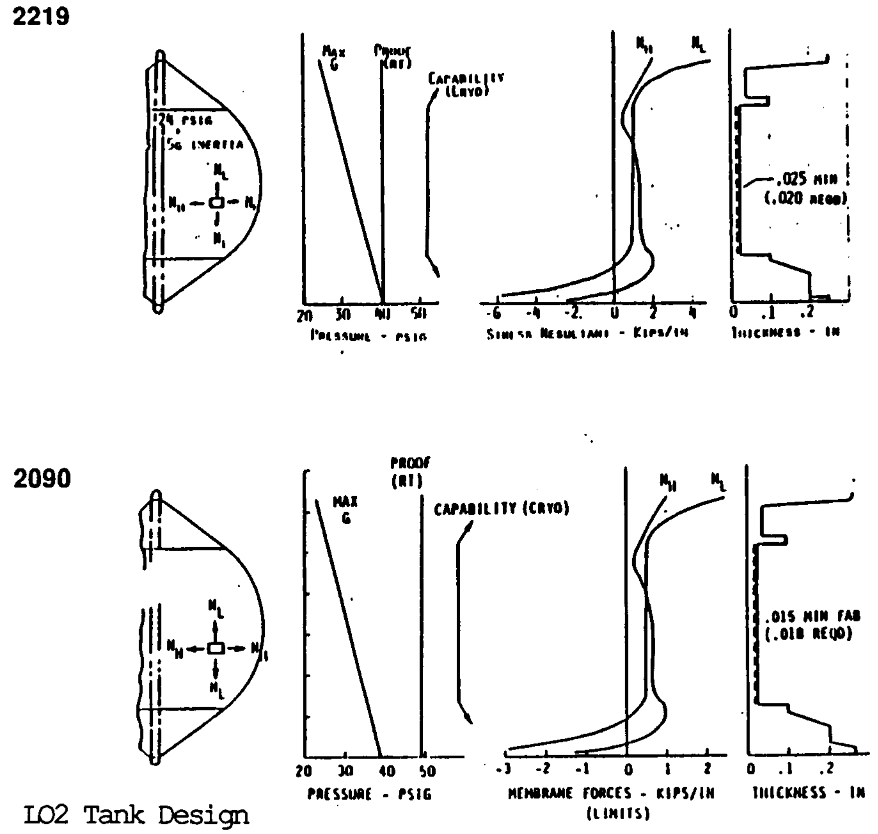


Figure 5.1.2-2 LO2 Tank Design

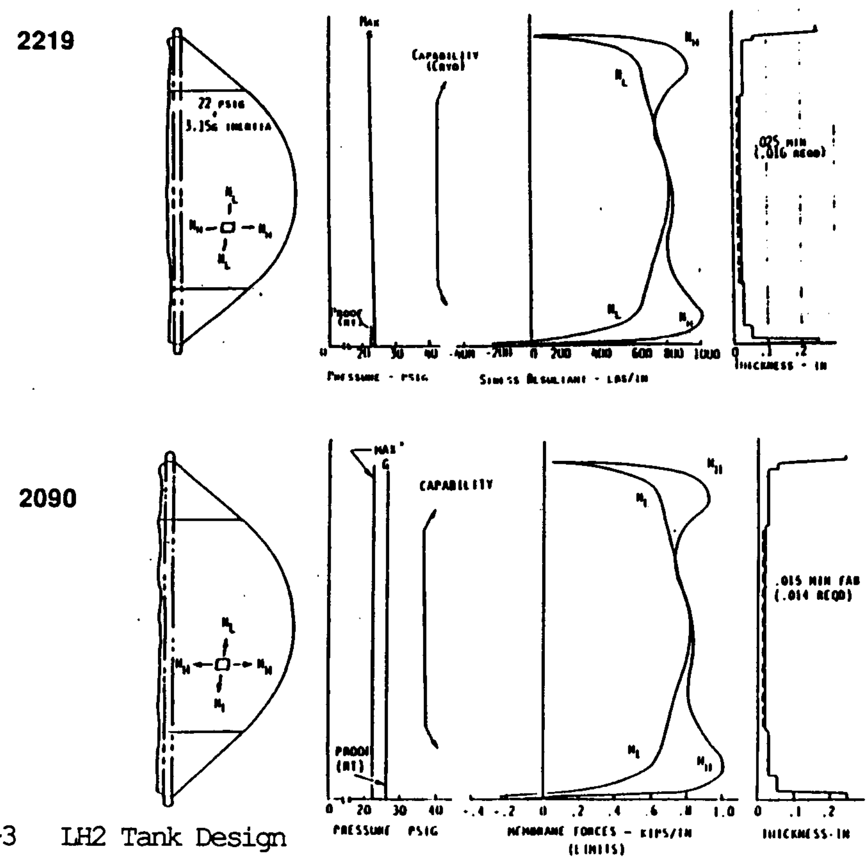


Figure 5.1.2-3 LH2 Tank Design

5.1.3 OTV DEBRIS/METEOROID ASSESSMENT

Debris and meteoroid protection was sized for the ground based vehicle concepts. To meet a proposed 0.999 probability of no damage per mission from space debris or meteoroids, the OTV will require a bumper at some spacing from the pressure wall. With a minimum Al-Li alloy pressure wall thickness of 0.015-in. for structural and/or fabrication requirements, and 0.5-in. of 0.788 lb/cu ft MLI to meet thermal requirements, only the bumper thickness and its location were varied to achieve appropriate levels of penetration resistance. Additional thickness of the pressure wall or thermal blanket was not analyzed as part of a total system trade.

A parametric study was performed using different bumper thicknesses and spacings. The probability of penetration was calculated from the particle diameter to penetrate each design. Penetration may occur by several of the following mechanisms. (1) If the weight per unit area (areal density) of the bumper is insufficient to fragment the projectile, then penetration will occur. This is assumed to be 15% of the particle's areal density. (2) Even if the bumper and MLI stop all fragments from reaching the rear wall, that wall must absorb all the momentum. The Rockwell equation for no yield of the pressure wall was used for this failure mode. (3) Since space debris impact at 3 km/s will not shock the debris enough to vaporize it, the critical debris diameter was 1.2 times the combined thickness of the bumper and the effective MLI thickness. The equivalent aluminum thickness of the MLI was calculated from the penetration of low density materials in NASA TMX-53955, in comparison to the penetration of the aluminum sheet in NASA 8042.

The probability calculation was based on an exposure area of $140m^2$, space debris flux from JSC 20001, and a meteoroid flux from NASA SP 8012. The altitude profile of the OTV was used to calculate effective exposure times at 400 km based on: (1) the density of space debris tracked by NORAD as a function of altitude; (2) the meteoroid shadowing of the OTV by the Earth; and (3) a defocusing factor for the attraction of the Earth's gravity on meteoroids. Table 5.1.3-1 lists the assumptions used during the study.

Bumper thickess has a strong influence on the probability of penetration for thin bumpers as shown in Figure 5.1.3-1. If the incident particle is not broken up by the bumper, then cratering of the rear wall will occur. However, as bumper thickness increases, the rear wall can no longer absorb the momentum of the impact. Increasing the spacing spreads the momentum over a larger area and a larger mass projectile can be stopped.

Table 5.1.3-1 OTV Debris/Meteoroid Assumptions

ASSUMPTIONS:

- MINIMUM OF 0.5" THICKNESS OF MLI USED FOR THERMAL REQUIREMENTS
  - 0.788 lb/ft^3
- MINIMUM AL-LI PRESSURE WALL THICKNESS 0.015" FOR STRUCTURE/FABRICATION

MINIMUM DIAMETER PARTICLE TO PENETRATE CHOSEN FROM

- PROJECTILES NOT SHATTERED BY BUMPER WILL PENETRATE
  - BUMPER AREAL DENSITY ≥ 0.15 x PROJECTILE DIAMETER x DENSITY
  - NO BENEFIT FROM MLI ASSUMED
- PRESSURE WALL MUST ABSORB ALL MOMENTUM (RI APOLLO EQUATION)
  - NO BENEFIT FROM MLI ASSUMED
- LOW VELOCITY DEBRIS WILL BE STOPPED BY BUMPER + MLI ONLY
  - MLI FRAGMENT PENETRATION RESISTANCE EQUIVALENT TO 0.032"AL
  - CRITICAL DEBRIS DIAMETER = 1.2 x TOTAL THICKNESS OF BUMPER + MLI

EXPOSURE TIMES RATIOED TO 400 KM ALTITUDE

- JSC 20001 USED FOR DEBRIS FLUX AT 400 KM
- 140 m^2 EXPOSURE AREA

	DEBRIS TIME hrs	METEOROID TIME hrs
EXPENDABLE	15	30
REUSABLE	112	210

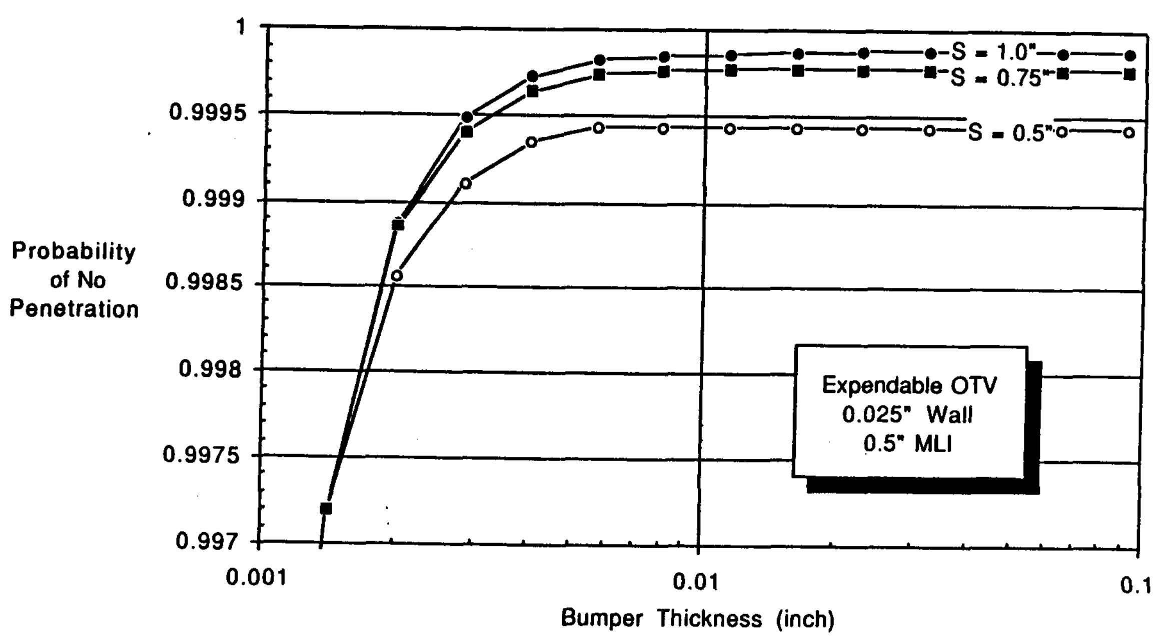


Figure 5.1.3-1 OTV Debris/Meteoroid Bumper - Expendable

The size of a meteoroid and the size of debris which can be stopped by each design is used to calculate a flux of each size particle (or larger) from NASA TMX-8013 or NASA JSC 20001, respectively. Each flux is used with the appropriate exposure time and area to calculate a probability of no penetration.

For an expendable vehicle, a layer of Beta Cloth will suffice as a bumper with a 0.6-in. standoff as shown in Table 5.1.3-2. Expected increases in the space debris and meteoroid environment will affect these numbers, and changes to the environment over the lifetime of the program must be considered. With a worse environment, the expendable vehicle would be modified closer to the proposed reusable vehicle design. The reusable design would be modified for a worse environment by using a 4-in. standoff, increasing the bumper thickness, and adding beta cloth or kevlar cloth on top of the MLI for increased fragment protection. Increases in the environment should be watched closely to determine the need for increased protection, and the design should allow for the larger standoffs that might be required.

Table 5.1.3-2 OTV Debris/Meteoroid Bumper Sizing

RECOMMENDATIONS

- **BUMPER SIZED TO MEET 0.999 PROBABILITY OF NO PENETRATION PER MISSION:**

OTV \ DEBRIS	BUMPER THICKNESS [inch]	MIN BUMPER SPACE TO WALL [inch]
EXPENDABLE	0.003	0.6
REUSABLE	0.006	1.5

- **USE BETA CLOTH WITH AN AREAL DENSITY EQUIVALENT TO THESE THICKNESSES OF ALUMINUM**

5.1.4 DACC COMPOSITE SHROUD

The effects of the overpressure at SRB ignition on the ACC shroud have been assessed and a composite material ACC shroud (unpressurized) was compared to the all aluminum pressurized ACC shroud. The worst case for pressure loading occurs at ignition when an overpressure exists on the shroud which varies from 0.5115 psia at the connecting ring and increases with axial distance to 0.900 psi at the dome center. Figure 5.1.4-1 shows the external pressure distribution on the shroud.

Structural requirements are outlined in Table 5.1.4-1. The major load is the overpressure at ignition which makes the structure buckling critical. The shroud is designed to withstand accelerations up to 3.15g in the axial direction and up to 2.5g in the radial/normal direction. Although not a specific requirement, a FS of 1.4 was used for all internal and external loads, and a FS of 2.0 was used for all buckling critical loads. the higher FS for buckling accounts for the uncertainty between the design and test data.

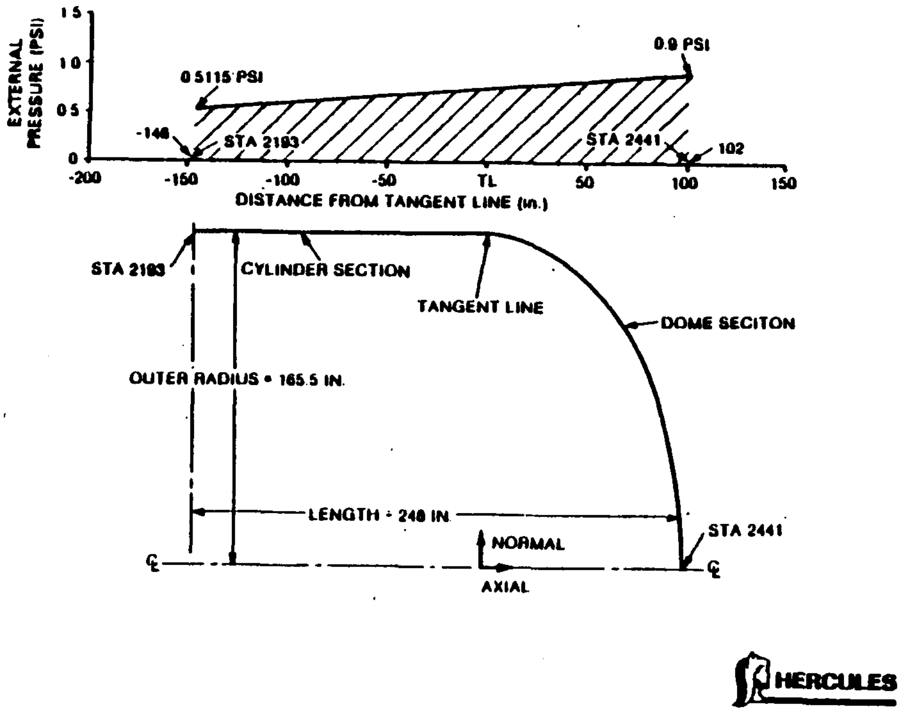


Figure 5.1.4-1 Pressure at Ignition and Shroud Geometry

Table 5.1.4-1 STS Structural Design Requirements

- **Factor of Safety**
  - **1.4 for all internal & external loads**
  - **2.0 for buckling**
- **Ignition overpressure = 0.9 psi (max.)**
- **Acceleration**
  - **Liftoff: Axial = +2.49G**
    - 0.37G
    Normal = 0.82G
  - **Meco: Axial = 3.15G**
    Normal = 0.81G
  - **Handling: 2.5G**

Detailed preliminary structural analyses were performed on the baseline shroud design. A finite-difference computer code (BOSOR) evaluated the buckling stability of the shroud under external pressure loading. Classical closed form methods were used to evaluate the structural integrity of the shroud under acceleration and handling loads. Analyses results indicate that the composite shroud is structurally adequate under the specified structural loading conditions. Table 5.1.4-2 shows that the minimum FS is 1.4 at the shroud-to-skirt joint under acceleration loading and in the shroud cylinder under external pressure.

Table 5.1.4-2 Structural Analysis Summary

COMPONENT	LOADING CONDITION	ANALYSIS METHOD	FACTOR OF SAFETY
CYLINDER	EXTERNAL PRESSURE	BOSOR 4 (1)	1.42
	ACCELERATION	CLOSED FORM	>10
	HANDLING-BENDING MOMENT DURING ROTATION	BOSOR 4 (1) & CLOSED FORM	>10
DOME	EXTERNAL PRESSURE	BOSOR 4 (1)	6.33
	HANDLING-AXIAL PULL ON PORT OPENING	BOSOR 4 (1)	>10
JOINT	ACCELERATION	CLOSED FORM	1.4

(1) BUSHNELL D., "STRESS, STABILITY AND VIBRATION OF COMPLEX BRANCHED SHELLS OF REVOLUTION," NASA CR-2116, OCTOBER 1972

(2) FACTOR OF SAFETY = ALLOWABLE VALUE / ACTUAL VALUE

In the composite shroud design shown in Figure 5.1.4-2, the inner and outer skins will be a sandwich structure. The skins will be filament wound AS4W-12K graphite fiber using HBRF 55A epoxy resin. This composite will have 50% fiber by volume. The lamina properties for this composite are: the modulus in the fiber direction is 17.21×10^6 psi; the modulus across the fibers is 9.662×10^5 psi; and the Poisson's ratio is 0.275.

The composite design core is composed of balsa wood with the grain perpendicular to the skins. The balsa has a modulus perpendicular to the grain of 16,000 psi, a modulus parallel to the grain of 330,000 psi, and a shear modulus of 14,450 psi.

In constructing this sandwich skin, the AS4W/55A composite will be wound onto the mandrel at an angle of $\pm 10°$ and a thickness of 0.04-in. at the tangent line. To complete the inner skin, a 0.02-in. thick hoop ply will be wound from

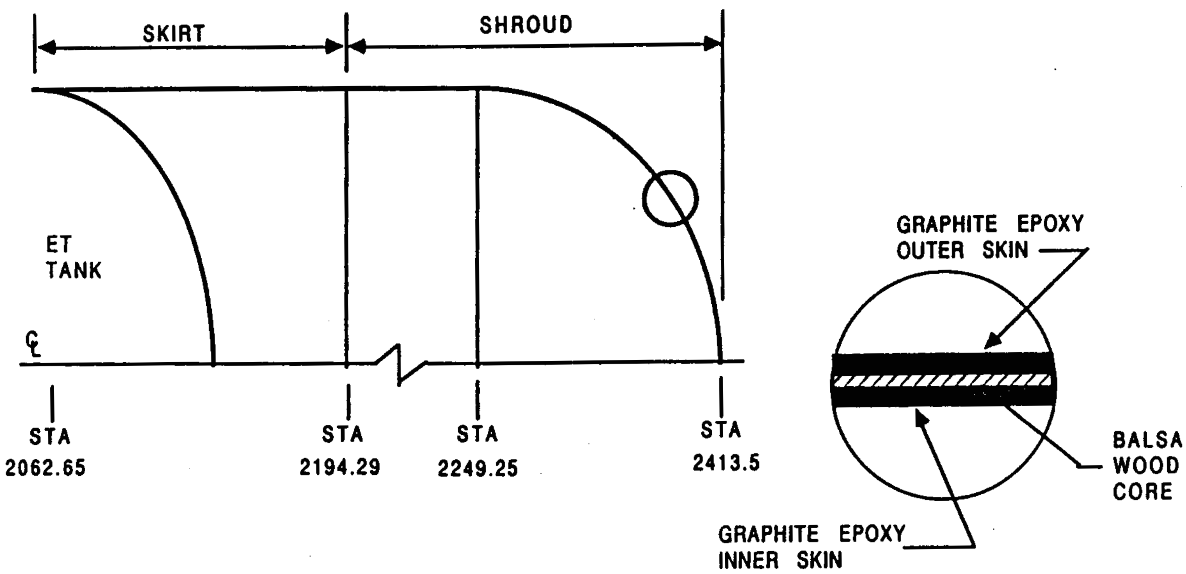


Figure 5.1.4-2 DACC Composite Shroud

tangent line-to-tangent line on the cylinder. Then a 0.625-in. layer of balsa core will be applied to the inner skin. Once the core has been applied, an outer skin will be wound on top of it which has the same layup and thicknesses as the inner skin. This type of construction results in a shroud capable of withstanding the specified buckling loads.

Table 5.1.4-3 shows the weight breakdown and comparison of the unpressurized composite shroud and the pressurized metal shroud. The aluminum forward skirt and payload support beams were baselined for both concepts. Both designs used the same structural requirements in developing the concept configurations.

The metal pressurized shroud consists of riveted chem milled gore panels, a dome cap, and a riveted chem milled barrel structure. To optimize the weight, the panel gage was reduced. This approach necessitated pressurizing the shroud at ignition to counteract the oil-canning effect of overpressurization on the thinner panels.

As discussed earlier, the composite shroud configuration is a sandwich structure consisting of an inner and outer skin made of Graphite/Epoxy and a core of balsa wood. The dome and barrel integral structure is designed to accommodate overpressurization at ignition without pressurizing the shroud. The composite sandwich also serves as part of the thermal control system.

Table 5.1.4-3 DACC Shroud Weight Comparison

	METAL PRESSURIZED WEIGHT (LB)	COMPOSITE UNPRESSURIZED WEIGHT (LB)	DETLAS WEIGHT (LB)
SKIRT			
STRUCTURE	2556	2556	0
THERMAL PROTECTION	173	173	
AVIONICS/ELECTRICAL	152	152	
PROP/MECH	125	125	
ORDNANCE	23	23	
CONTINGENCY	454	454	
SUBTOTAL	3483	3483	0
SHROUD			
DOME	781	1248	+467
ATTACH FLANGE	62	62	0
SEPARATION ASSY	191	211	+20
THERMAL PROTECTION	858	554	-304
PROP/MECH	9	9	0
ORDNANCE	74	74	0
ATTACH HRDW	20	20	0
CONTINGENCY (15%)	299	326	+27
SUBTOTAL	2294	2497	+210
TOTAL	5777	5980	+210

Translating the two different design concepts into a weight difference produces a net weight increase of over 200 lb for the composite shroud. Although the composite structure is 467 lb heavier than the metal shroud, a 304 lb weight saving is realized in the thermal control. An advantage is gained by eliminating the need for pressurization. with the composite shroud.

5.1.5 BATTERY SELECTION

Table 5.1.5-1 lists the five batteries considered to replace the OTV fuel cell power system. Each candidate's characteristics are listed with their advantages and disadvantages.

The Ag-Zn alkaline batteries are cycle-limited secondaries that are used in many primary applications. They have high-energy density, a relatively poor cycle life, little loss of capacity during dry storage, high reliability, and storage capacity. Although they have a narrower operating temperature range, the Ag-Zn batteries—when discharged at high rates to obtain maximum output, and by using their self-heating capability—can supplement battery heaters.

The Ni-Cd alkaline batteries are used when long-life secondary batteries are required. These batteries have low energy density, high cycle life, a relatively low discharge rate (less than 40% of storage capacity), and medium reliability.

Table 5.1.5-1 Battery Candidates

CHARACTERISTICS					ADVANTAGES / DISADVANTAGES							
Type	Pri/Sec	Energy Density WH/LB	Cell Voltage (nominal)	Temp Range C Oper/Strg	Technology Risk	Discharge Rate	Shelf Life Year	Capacity (loss per yr) Storage O/D @ C	Reliability	Safety Factor	Cycle Life
SILVER-ZINC Ag-Zn	S	50 - 120 58	1.5	0 TO 55 -50 TO 80	LOW	HIGH	2 - 5	10	HIGH	HIGH	7500
NICKEL-CADMIUM Ni-Cad	S	8 - 20 12	1.25	-10 TO 30 -40 TO 60	LOW	LOW	5.0 MIN	30	MED	HIGH	10000
NICKEL-HYDROGEN Ni-H2	S	25 - 30 22	1.25	-20 TO 40	MED	LOW	15	30	HIGH	HIGH	30000
LITHIUM-THIONYL CHLORIDE Li-SoCl2	P	150 (650)	3.6	-40 TO 70	MED	LOW	10	1 - 2	NEW	NEW	N/A
LITHIUM-SULPHUR DIOXIDE Li-So2	P	150 (440)	3.0	-55 TO 70	MED	LOW	10	1 - 2	NEW	NEW	N/A

The Ni-H battery is a hybrid system utilizing the hydrogen electrode from the fuel cell and the nickel electrode from the Ni-Cd cell. This battery has a higher energy density and cycle life than the Ni-CD secondary batteries. It also has a recharge fraction of 1.06, a 65% depth of discharge, a low discharge rate, and high SF. However, due to the presence of extremely flammable hydrogen gas, controls must be implemented to constrain cell pressure within safety limits.

The two Li batteries (i.e., Lithium-Thionyl Chloride and Lithium-Sulphur Dioxide) have the highest energy density of all the primary and secondary power sources. They have a long shelf life, high cell voltages, and a wide range of operating temperatures. They also have a low discharge rate, low capacity, and potential danger to humans and equipment due to the explosive nature of Li compounds. Since these batteries are relatively new, their reliability and SF are yet to be determined. Testing is being performed and their use is proposed on the Jupiter Galileo Probe.

The mission requirements for the expendable OTV power source are: a single use system with an operational time of 33 hours; an average watt use of 446 watts; a maximum use of 964 watts; and a voltage of 28 volts (nominal).

In addition to meeting the mission requirements, the selected battery must meet the five design criteria shown in Table 5.1.5-2: (1) medium to high energy density; (2) low technology risk; (3) high degree of reliability; (4) high factor of safety (FS); and (5) a lightweight system, i.e., less than or equal to the 270 lb fuel cell system.

Table 5.1.5-2 Battery Selection

MISSION REQUIREMENTS

SINGLE USE	33 HR DURATION
AVG WATTS	446 WATTS
MAX WATTS	964 WATTS
VOLTAGE	28 V (nominal)
WATT-HR	14718

BATTERY REQUIREMENTS

ENERGY DENSITY	MED-HIGH
TECHNOLOGY RISK	LOW
RELIABILITY	HIGH
SAFETY FACTOR	HIGH
WEIGHT	LIGHT

BATTERY WEIGHT based on WH/LB

	WATT-HR	WH/LB	WEIGHT
SILVER-ZINC	14718	58	254
NICKEL-CADMIUM	14718	12	1227
NICKEL-HYDROGEN	14718	25	589
LITHIUM-THIONYL-CHLORIDE	14718	150	98
LITHIUM-SULPHUR DIOXIDE	14718	150	98

BATTERY SELECTED SILVER-ZINC

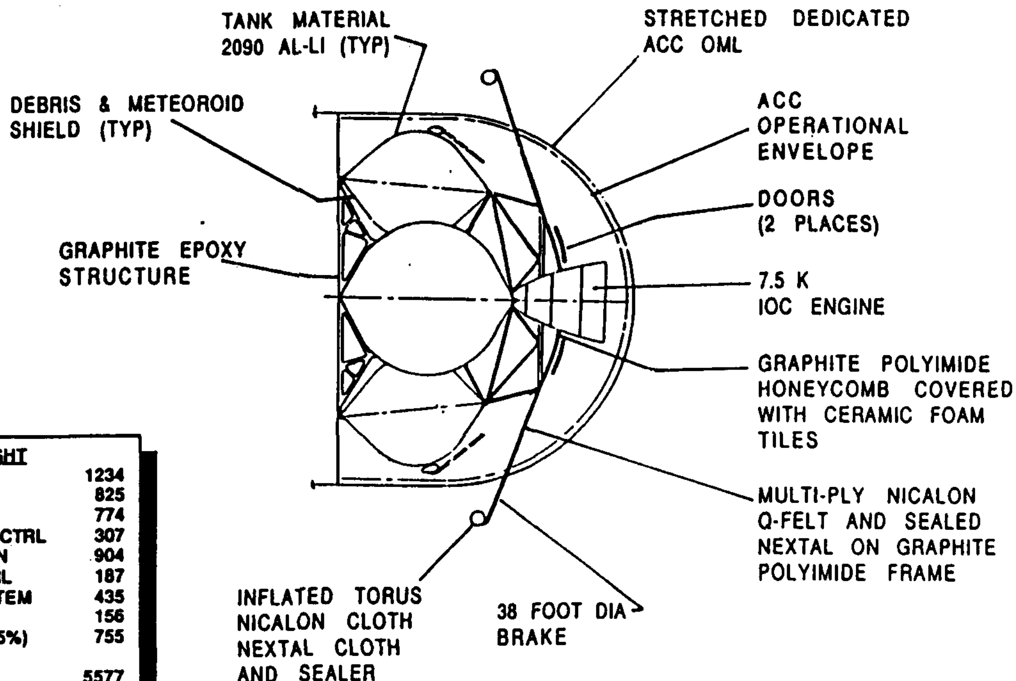
ENERGY DENSITY	58 WH/LB
TECHNOLOGY RISK	LOW (In service now)
RELIABILTY	HIGH (Mission success)
SAFETY FACTOR	HIGH (No incidents)
WEIGHT	LOW (254 LB)

Each battery being considered shall meet the mission requirements for a power source on an expendable OTV. When judged on the battery requirements, the Silver-Zinc (Ag-Zn) batteries meet each of the five criteria. Due to their low energy-density and corresponding high systems weight, the Nickel-Cadmium (Ni-Cd) and Nickel-Hydrogen (Ni-H) batteries are eliminated. Although Lithium (Li) batteries have a higher energy density and low systems weight, they have a high technology risk since their reliability and FS have yet to be fully determined.

The Ag-Zn batteries have an energy density of 58 WH/lb and a system's weight of 254 lb (≤270 lb). They are currently in service on a number of space vehicles, including Titan and Transtage. Moreover, there have been no safety incidents associated with these batteries. The fact that they are currently in service, are highly reliable with a high FS, gives the Ag-Zn batteries a desirable low technology risk rating.

5.2 GROUND BASED CRYOGENIC REUSABLE OTV UPDATES

Figure 5.2-1 shows the general arrangement and weight breakdown of our selected groundbased cryogenic OTV transported in the ACC. Table 5.2-1 shows additional detail on the stage weights. The four-tank single advanced technology engine configuration uses the volume and weight efficient principals (suggested by Larry Edwards) to fit into the stretched version of the ACC (42-in. stretch).



WEIGHT	
AEROBRAKE	1234
TANKS	825
STRUCTURE	774
ENVIRONMENTAL CTRL	307
MAIN PROPULSION	904
ORIENTATION CTRL	187
ELECTRICAL SYSTEM	435
G. N. & C.	156
CONTINGENCY (15%)	755
DRY WEIGHT	5577
PROPELLANTS, ETC	45424
LOADED WEIGHT	51011

Figure 5.2-1 Ground Based Reusable OTV

Table 5.2-1 Ground Based Reusable OTV Weight Summary

WBS GROUP	WEIGHT (LB)
2 STRUCTURES	1089
3 PROPELLANT TANKS	949
4 PROPULSION FEED SYS	727
5 MAIN ENGINES	313
6 REACTION CONTROL SYSTEM	215
7 G. N. & C.	180
8 COMM & DATA HNDLG	99
9 ELECTRICAL PWR	433
10 THERMAL CONTROL	153
11 AEROBRAKE	1419
DRY WEIGHT	5577
12 FLUIDS	
RESIDUAL-LH2	96
RESIDUAL-LO2	579
HYDRAZINE	400
PRESSURANT	14
COOLANT	10
INERT WEIGHT	6676
USABLE MAIN PROP.	
FUEL-LH2 w/FPR	6342
OXIDIZER-LO2 w/FPR	37993
IGNITION WEIGHT	51011
MASS FRACTION	
44335 MAIN PROP W/FPR / 51011 IGNITION WEIGHT	.87

The 38 ft diameter aerobrake folds forward when stowed in the ACC. The aerobrake is discarded after flight and is not stowed in the Orbiter for retrieval. The Aluminum Lithium (Al-Li) propellant tanks are designed by engine inlet pressure requirements. The LO2 tank minimum gage is 0.018-in. and the LH2 tank minimum gage is 0.015-in. The tanks are insulated with Multilayered Insulation (MLI).

The LH2 tanks are removed on orbit and are discarded and allowed to re-enter the atmosphere and vaporize. The core system (LO2 tanks, structure, avionics, and propulsion) are stowed in the Orbiter cargo bay for retrieval after mission completion. The structure is of lightweight graphite/epoxy. The propellant load was selected to enable full use of the projected 72 Klbm NSTS lift capability on GEO delivery missions.

Table 5.2.1-1 shows the latest weight changes to the recommended ground-based OTV.

Table 5.2.1-1 Ground Based OTV Weight Change Summary (lb)

COMPONENTS	OCT '86	DEC '87
AEROBRAKE	1566	1234 (1)
TANKS	524	825 (2)
STRUCTURE	774	774
ENVIRONMENTAL CTRL	424	307 (3)
MAIN PROPULSION	904	904
ORIENTATION CTRL	187	187
ELECTRICAL SYS	613	435 (4)
G. N. & C.	156	156
CONTINGENCY	772	755
DRY WEIGHT	5920	5577
DELTA		-343

(1) The aerobrake's hardcore center has been modified from 25.5 ft to 13.5 ft, and the 25.5 ft support frame removed resulting in a decrease of 332 lb.

(2) The LH2 and LO2 tanks were reanalyzed per the latest property information for the Al-Li 2090. This analysis required an increase in weld land and membrane thickness for the gore panels on both tanks. The conical ends were also reanalyzed and their thickness increased. This result of this re-evaluation was a weight increase of 301 lb.

(3) Environmental Control - the debris/meteoroid shield was recalculated based on data developed during the Space Station study program, allowing a much thinner bumper which produces a weight saving of 117 lb.

(4) Electrical System - the S-Band system was replaced with a lighter system.

5.2.1 AEROBRAKE MODIFICATIONS

Table 5.2.1-2 shows the weight changes in the 38 ft diameter aerobrake that occur due to a reduction in the diameter of the hard shell center. Originally, a 25.5 ft diameter hard shell center was used to allow simple folding and stowage of the aerobrake flex section ribs. A new folding technique has been developed (see below) which allows a smaller and lighter center section. Figure 5.2.1-1 shows the intended reduction in hard shell diameter. The center, with a surface density of 1.05 lb/sq ft was reduced from 25.5 ft to 13.5 ft. The area (~400 sq ft) was covered with Flexquilt TPS at 0.49 lb/sq ft. The net result was a weight saving of 215 lb.

Table 5.2.1-2 Aerobrake Weight Changes

	WEIGHTS (LB)		
COMPONENTS	**WAS**	**IS**	**DELTAS**
HEAT SHIELD			
HARD SHELL w/TPS	**531**	**120**	**-411**
TPS w/FLEX QUILT	**330**	**526**	**+196**
MECHANICAL SYSTEM			
DOORS w/ MOTORS	**85**	**70**	**-15**
TORUS SYSTEM	**112**	**112**	**0**
SPRINGS	**36**	**36**	**0**
SUPPORTS STRUCTURE			
RIBS	**249**	**249**	**0**
RING FRAMES	**223**	**121**	**-102**
CONTINGENCY	**235**	**185**	**-50**
TOTAL	**1801**	**1419**	**-382**

A secondary effect occurs from removing the rib supported at 25.5 ft and using the attachment frame at 13.5 ft to support the ribs. This modification results in a 102 lb weight saving. Including contingencies, the total weight saving is 382 lb.

Figure 5.2.1-2 shows the stowage arrangement for the 38 ft aerobrake. To accommodate the aerobrake with a 13.5 ft diameter hard shell center, the ribs have been clocked off the centerline of the tanks by 15°, with a 30° typical spacing. However, the ribs on either side of the LH2 were clocked 20° off the tank centerline with a spacing of 40°. This allows the ribs to fold within the operational envelope and avoid interfering with the LH2 tank.

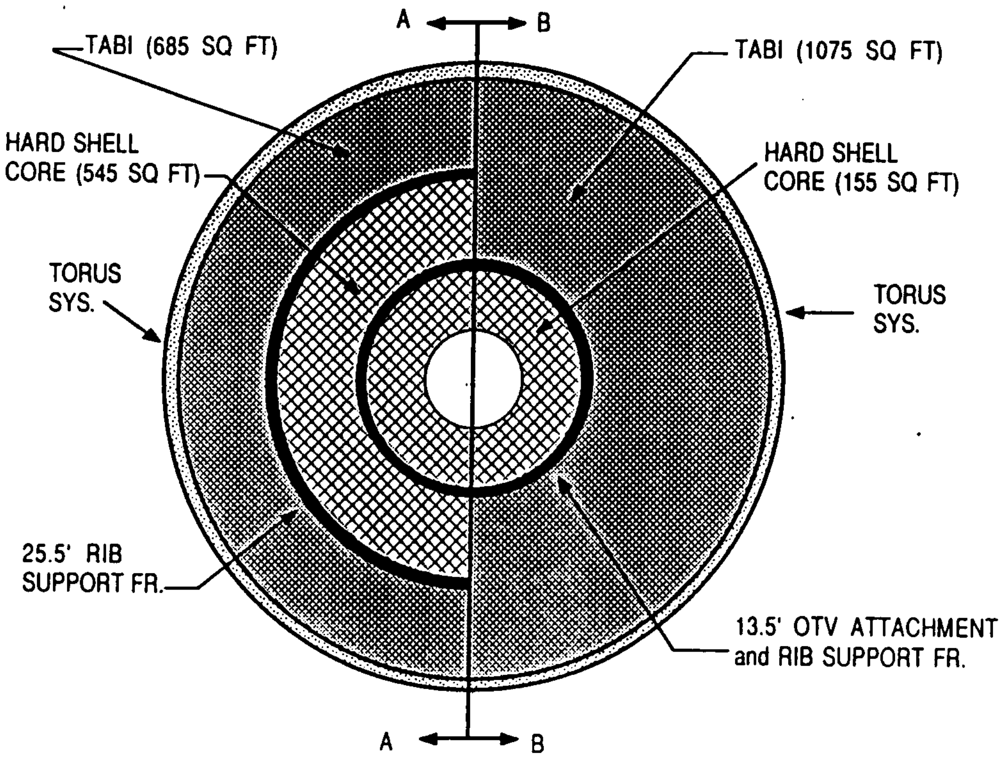


Figure 5.2.1-1 Aerobrake Design Changes

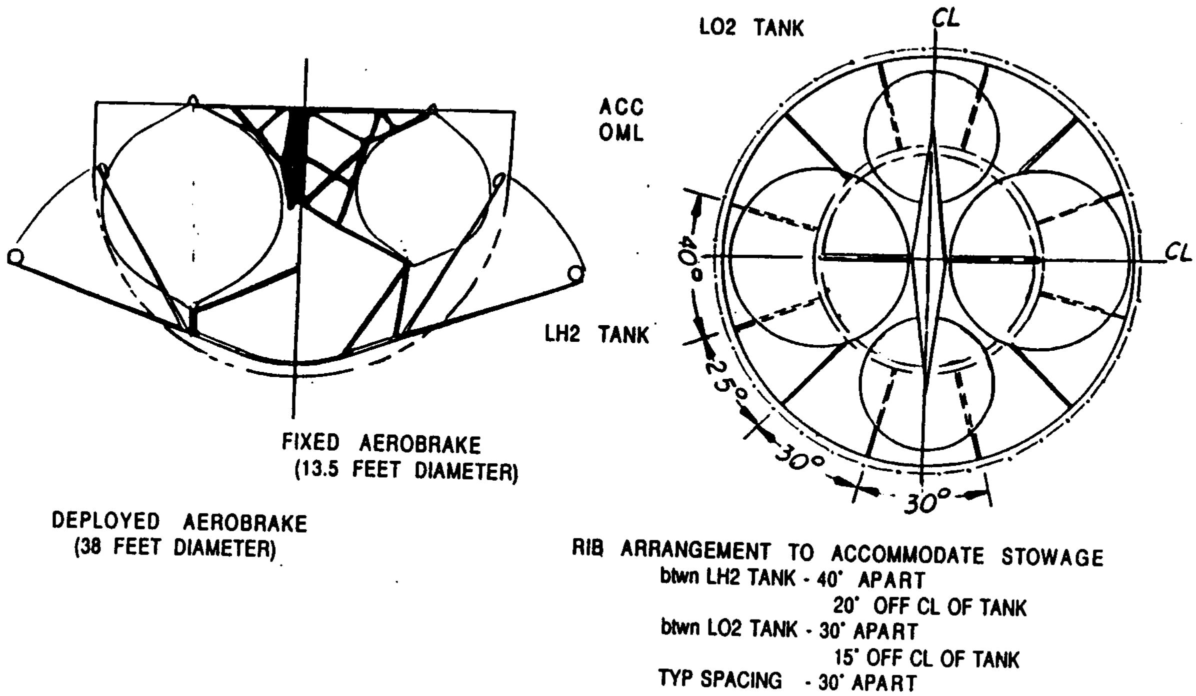


Figure 5.2.1-2 Aerobrake Stowage Arrangement

5.2.2 REUSABLE OTV METEOROID PROTECTION

For a reusable OTV, at least two layers of Beta Cloth should be used with at least a 1.5-in. standoff (shown earlier in Table 5.1.3-2) and with a bumper thickness of at least 0.006in as determined from Figure 5.2.2-1. Expected increases in the space debris and meteoroid environment will affect these numbers and changes to the environment over the lifetime of the program must be considered. The reusable design would be modified for a worse environment by using a 4-in. standoff, increasing the bumper thickness, and/or adding beta cloth or kevlar cloth on top of the MLI for increased protection from fragmentation.

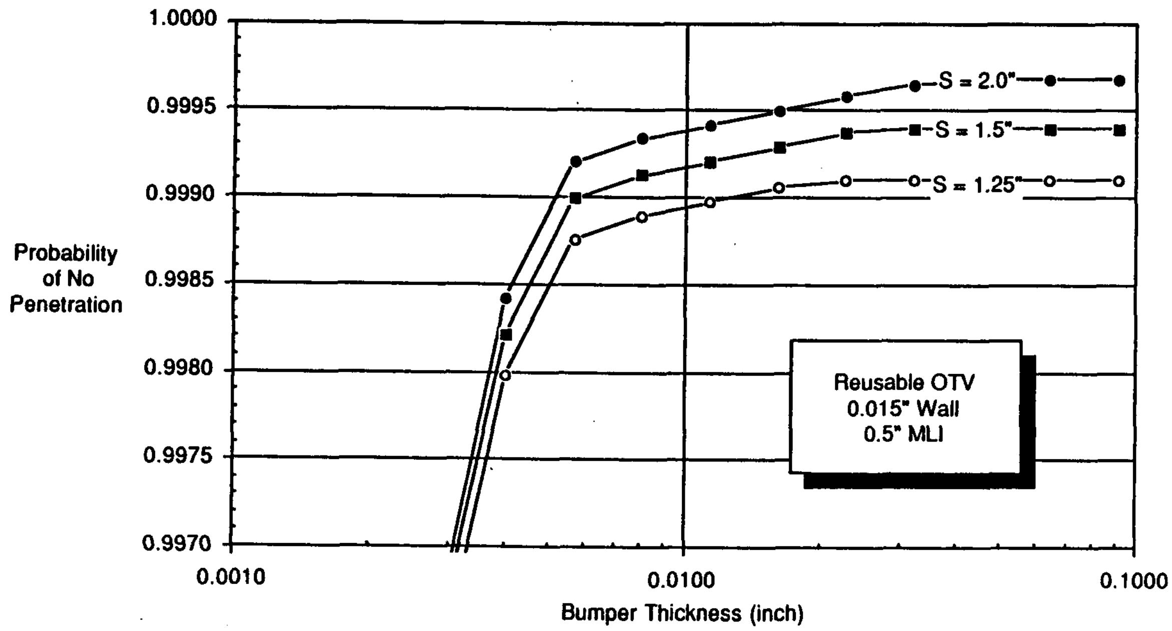


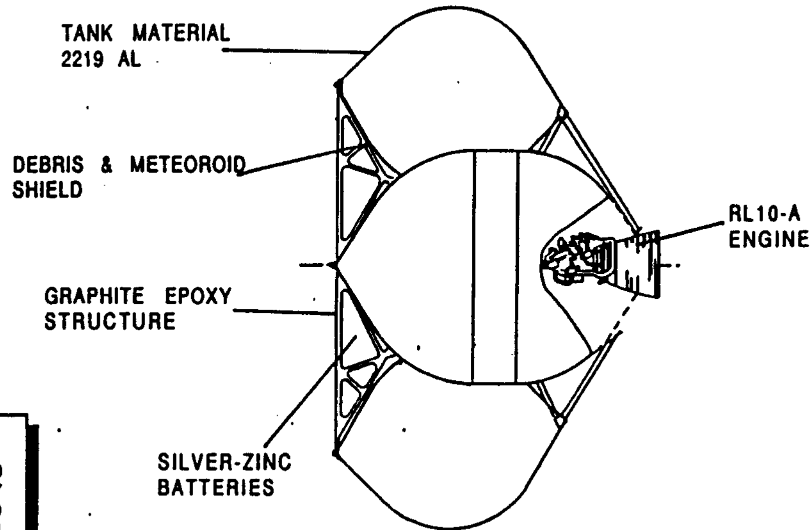
Figure 5.2.2-1 OTV Debris/Meteoroid Bumper - Reusable

5.3 LARGE CARGO VEHICLE (LCV) EXPENDABLE OTV

5.3.1 LCV OTV CONCEPT DEFINITION

Figure 5.3.1-1 shows the general arrangement and breakdown of our selected expendable configuration which will be used in either a sidemount or inline LCV payload element. Table 5.3.1-1 shows additional detail for the stage weights.

The LCV expendable concept uses the same features as the ACC expendable baseline OTV, i.e., composite airframe, Al 2219 tanks, Ag-Zn batteries, RL10-A engine, avionics equipment, and the same propulsion feed system.



WEIGHT	
TANKS	1150
STRUCTURE	667
ENVIROMENTAL CONTROL	259
MAIN PROPULSION	944
ORIENTATION CONTROL	187
ELECTRICAL SYSTEMS	328
G. N. & C.	182
CONTINGENCY (15%)	556
DRY WEIGHT	4273
PROPELLANTS, ETC	50424
LOADED WEIGHT	54697

Figure 5.3.1-1 LCV Expendable OTV

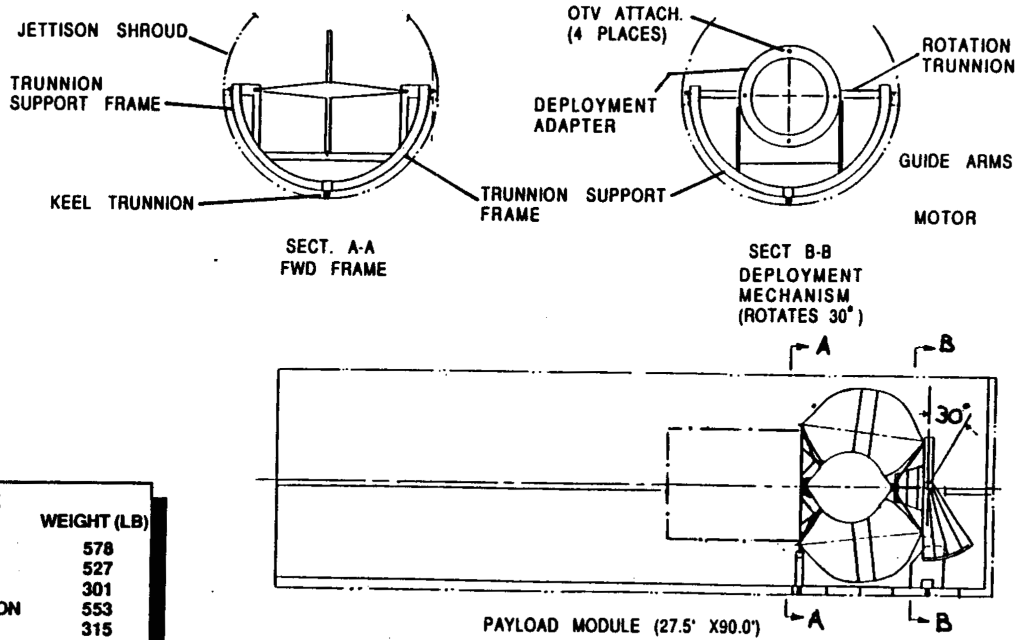
Table 5.3.1-1 LCV OTV Weight Summary

WBS GROUP	WEIGHT (LB)
2 STRUCTURES	905
3 PROPELLANT TANKS	1323
4 PROPULSION FEED SYS	698
5 MAIN ENGINES	388
6 REACTION CONTROL SYSTEM	215
7 G. N. & C.	128
8 COMM & DATA HNDLG	81
9 ELECTRICAL PWR	377
10 THERMAL CONTROL	160
11 AEROBRAKE	000
DRY WEIGHT	4273
12 FLUIDS	
RESIDUAL-LH2	107
RESIDUAL-LO2	643
HYDRAZINE	400
PRESSURANT	10
COOLANT	00
INERT WEIGHT	5433
USABLE MAIN PROP.	
FUEL-LH2 w/FPR	7036
OXIDIZER-LO2 w/FPR	42214
IGNITION WEIGHT	54697
MASS FRACTION	
49250 MAIN PROP W/FPR / 54697 IGNITION WEIGHT	.90

The major difference between the two vehicles is the LH2 tank configuration. The LH2 tank diameter was reduced and a barrel section added because the payload element enveloped (25 ft diameter) is smaller than the ACC envelope. Also, the vehicle is rear-mounted on the airframe instead of top-mounted. Some additional support struts were required. The total dry weight of the LCV expendable OTV is 4273 lb.

5.3.2 ASE FOR LCV (SIDE MOUNT AND IN-LINE)

Figure 5.3.2-1 shows the ASE components and weight breakdown for the LCV expendable OTV sidemount configuration. The ASE is designed to support and launch the OTV from a 27.5 ft x 90 ft unmanned Payload (P/L) Module. The OTV is rear-mounted on a tilt table deployment mechanism and rotated into a launch angle. The OTV forward end is supported by an adapter frame. The loads and deflections have been checked using a NASTRAN model. The total weight of the ASE components is 2904 lb.



ASE	WEIGHT (LB)
FWD FRAME	578
AFT FRAME	527
DEPLOY. ADAPTER	301
ROTATION TRUNNION	553
MOTOR & ARMS	315
SUBSYSTEMS	100
PROP/MECH	128
ORDNANCE	23
CONTINGENCY	379
TOTAL	2904

Figure 5.3.2-1 ASE for LCV Side-Mount of OTV

Figure 5.3.2-2 shows the ASE components and weight breakdown for the LCV expendable OTV inline configuration. The ASE equipment (skirt, support beams, and hardware) is the same structure as on the ACC. The OTV is mounted from the rear, using the umbilicals and attach points. The shroud (27.5 ft x 90 ft) separates just forward of the OTV support beams. A NASTRAN model was used to check the support beam for sizing. The total weight of the ASE components is 3409 lb.

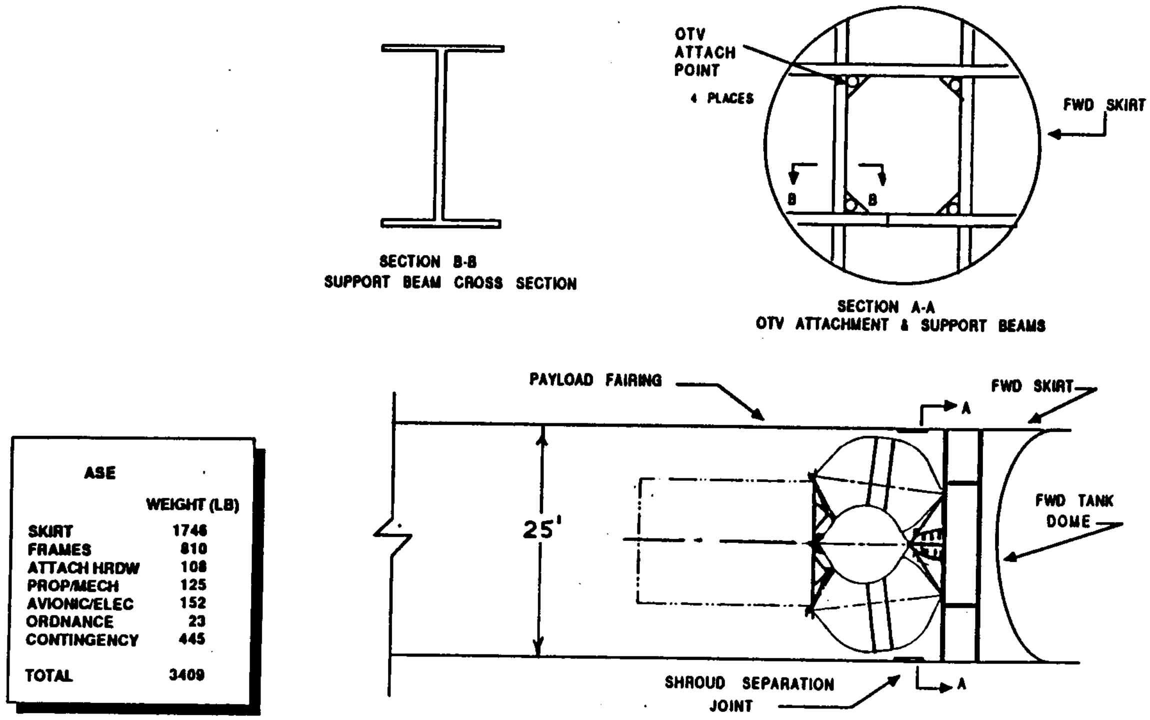


Figure 5.3.2-2 ASE for LCV In-Line Mount of OTV

5.3.3 AIRFRAME ANALYSIS

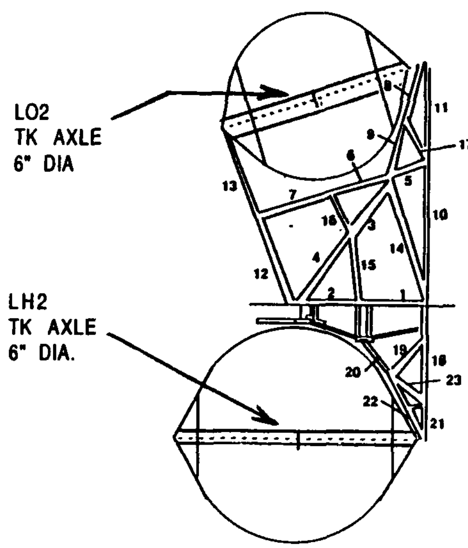
Figure 5.3.3-1 shows a tabulation of the cap loads for the LCV vs. ACC launch. To maintain the structural capability of the rack, LCV loads were designated to transfer the payload axial (X) and Y and Z moment loads directly into the rack support structure. This is accomplished by placing a 6-in. diameter axle tube along the axis of the propellant tanks.

The rack support beams are simply supported at the vehicle wall. Although several computer runs were made with the fuel tank struts both fixed and free, no significant load difference or deflection was found.

A stress analysis of the new/modified OTV rack and support structure was made to substantiate the integrity of the structure. The two principal requirements of the new design are:

(1) The new/modified rack must react the payload (14 Klb) and fuel tank loads, whereas the current rack is designed to react only fuel tank loads; and

(2) The modified rack is supported by a grillage of deep I-Beams located aft of the fuel tanks, whereas in the current design the rack support structure is located at the forward end of the rack.



REMARKS:
MAIN LOAD PATH
--- REAR MOUNTED -- TANK AXLE
--- FRONT MOUNTED -- AIRFRAME
MAJOR MODIFICATIONS
--- TANK AXLES 3" TO 6" IN DIAMETER
--- ADDITION TANK SUPPORT STRUTS
--- AIRFRAME -- NONE

Member Number	CAP LOADS (kips)		CAP AREA (sq in.)	
	Was	Now	Was	Now
1	26.7	12.70	0.478	0.478
2	44.1	14.71	0.807	0.807
3	82.0	10.36	1.513	1.513
4	87.9	14.16	1.513	1.513
5	13.4	01.76	0.231	0.231
6	17.7	00.62	0.334	0.334
7	15.0	04.34	0.334	0.334
8	97.8	44.90	1.743	1.743
9	73.3	32.60	1.743	1.743
10	50.6	12.30	1.470	1.470
11	74.1	19.40	1.470	1.470
12	58.1	34.70	0.995	0.995
13	66.6	80.33	1.220	2.260
14	21.1	11.99	0.810	0.810
15	2.5	00.80	0.067	0.067
16	8.2	00.44	0.151	0.151
17	15.1	04.01	0.263	0.263
18	33.0	03.88	0.858	0.925
19	13.3	01.00	0.240	0.500
20	36.7	Deleted	0.858	Deleted
21	29.1	05.24	0.858	0.925
22	33.5	06.96	0.858	0.925
23	8.7	01.84	0.373	0.925

Figure 5.3.3-1 LCC VS ACC OTV Cap Loads

Preliminary beam sizes were calculated by hand based on the existing rack geometry. The rack FEM was then revised using these new section properties, and the element loads were determined by NASTRAN. Based on the NASTRAN results, MS for the structural elements were found. In general, the critical failure mode was column buckling. In addition to the stress requirements, the maximum deflection in any direction at ultimate load was limited to 3-in. to satisfy the stiffness requirements.

Other major modifications to the rack include tying the forward outboard ends of the rack together with four 3-in. diameter tubes. These tubes could be part of the payload support structure. The aft ends of the fuel tanks are also tied together with four struts. These struts remain with the OTV to stabilize the fuel tanks.

6.0 HIGH SPEED AEROASSIST

Several different classes of entries have been studied in the course of this contract as is summarized in Figure 6.0-1. Earth return class missions utilize aeroassist to reduce an existing high-energy elliptical Earth orbit down to a low park orbit suitable for Shuttle or Space Station retrieval. There are three missions in this class: geosynchronous return, lunar return, and planetary boost return. The second class of missions is that of Earth capture. Here aeroassist is used to capture an existing hyperbolic flyby into a highly elliptical Earth orbit for later retrieval. Encounter C_3's ranging from 8.0 to 68 km^2/sec^2 have been investigated, consistent with return from Mars. The third class of missions are those representing capture into Mars orbit. These are similar to the Earth capture cases but for a different parent body; the C_3 range is from 8.2 to 60.0 km^2/sec^2.

For each aeroassist condition, three different sets of data have been prepared. First, an aero-entry error analysis derives the level of uncertainty associated with the particular entry. This analysis is critical to establishing trajectory control and vehicle lift requirements. Second, an entry control and loads parametric graph shows control corridor and deceleration loads sensitivities. This data is used to establish vehicle L/D and structural sizing. The third chart in each set shows peak stagnation heating and integrated heating data which is used to size the thermal protection system (TPS).

THE FOLLOWING CLASSES OF ENTRIES ARE SUMMARIZED:

1) GEOSYNCHRONOUS ORBIT RETURN

2) LUNAR RETURN

3) PLANETARY BOOST RETURN

4) EARTH CAPTURE	C3 =	8.0	16.0	32.0	68.0	KM^2/SEC^2
5) MARS CAPTURE	C3 =	8.23	13.0	31.0	60.0	KM^2/SEC^2

FOR EACH ENTRY THE FOLLOWING DATA IS CONTAINED
1) AEROENTRY ERROR ASSESSMENT
2) CONTROL & LOADS DATA CHART
3) HEATING DATA CHART

Figure 6.0-1 Aeroassist Classes

6.1 AEROASSIST OVERVIEW

Figure 6.1-1 illustrates an aerobraking maneuver from a highly elliptic Earth orbit down to a lower one. The initial entry orbit's perigee is carefully targeted to a desired location in the Earth's atmosphere. The aeroassist phase occurs while the vehicle is in the sensible atmosphere. Its object is to perform a controlled velocity reduction such that the vehicle has the desired apogee upon exit from the atmosphere. This apogee is generally at the same height as the desired final park orbit which is achieved by a post-aero apogee boost.

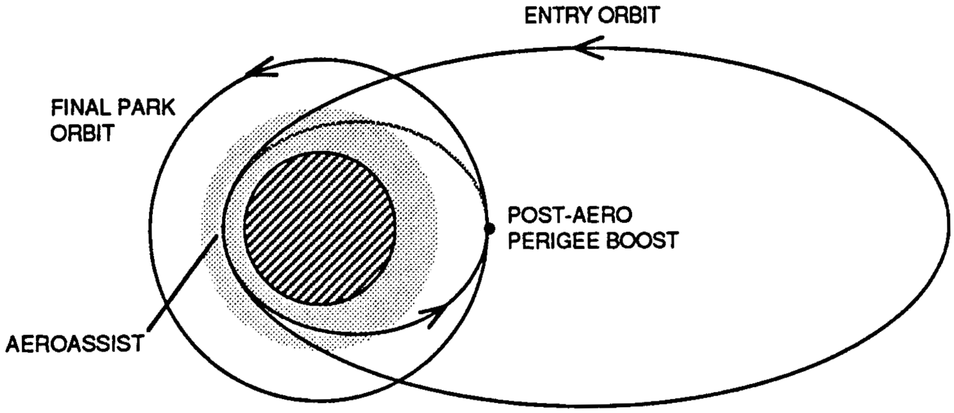


Figure 6.1-1 Earth Return to Low Orbit

The process of aero-capture, shown in Figure 6.1-2, is very similar to that of aeroassist except that the incoming trajectory is hyperbolic. This means that without the aero-maneuver the vehicle would escape the planet, hence the term "aerocapture". Otherwise the principal is the same with an aero phase followed by a perigee raise maneuver, performed at apogee. Also shown is a lander option which would deploy an entry capsule to the surface after a stable park orbit is achieved.

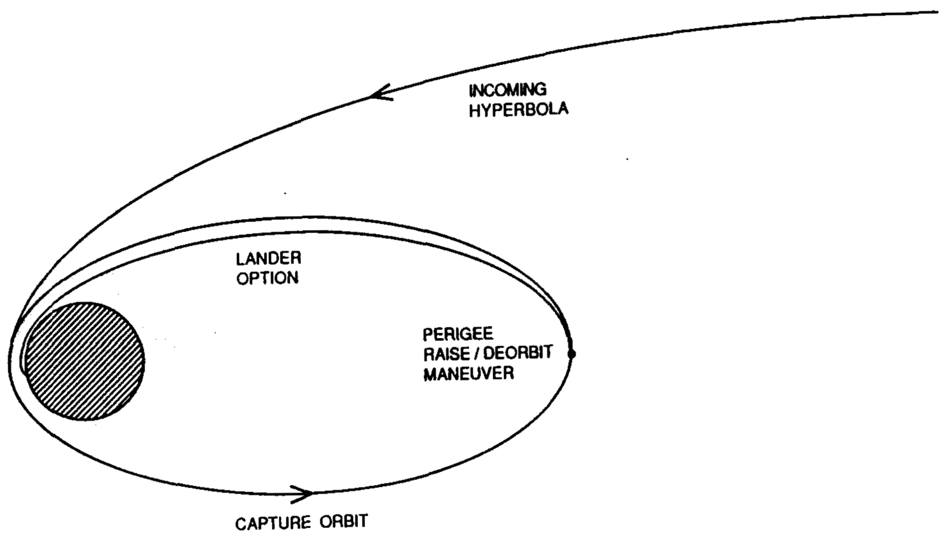


Figure 6.1-2 Planetary Aero-Capture

6.1.1 AEROASSIST CONDITIONS

Table 6.1.1-1 and 6.1.1-2 summarize several important pre- and post-entry parameters for the aeroassist maneuvers studied. Table 6.1.1-1 shows the Earth return and capture missions discussed previously. The initial semi-major axis is for the pre-entry orbit and is a measure of the entry interface energy state. The aero velocity reduction is the amount of inertial velocity that is removed from the body by the aeroassist maneuver. Finally the exit orbit apogee is the target that the aeromaneuver has achieved when the vehicle leaves the atmosphere.

The Earth return aeromaneuvers all use an exit orbit apogee target of 245 nm which is consistent with return to the Space Station. The Earth capture maneuvers use an exit target of 38485 nm which represents an Earth-synchronous orbit when the perigee is raised to 250 nm. This elliptic orbit must be used because of the excessive energies involved in the higher C_3 Earth encounters.

Table 6.1.1-2 summarizes the same information as above for the Mars capture missions. The exit apogee target is for a Mars synchronous orbit that has a final perigee altitude of 270 nm. This orbit is of strong interest because of its combination of favorable site reconnaissance and communication relay links.

Table 6.1.1-1 Aeroassist Conditions - Earth Entries

EARTH RETURNS

CASE	INITIAL SEMIMAJOR AXIS	AERO VELOCITY REDUCTION	EXIT ORBIT APOGEE
GEO RETURN	7.97513 E7 FT	7809.3 FPS	245 NM
LUNAR RETURN	8.95096 E8 FT	10099.1 FPS	245 NM
PLANET. BOOST	4.18627 E8 FT	9851.4 FPS	245 NM

EARTH CAPTURES

C3	INITIAL SEMIMAJOR AXIS	AERO VELOCITY REDUCTION	EXIT ORBIT APOGEE
8 KM^2/SEC^2	-1.63468 E8 FT	2588.9 FPS	38485 NM
16 KM^2/SEC^2	-8.17341 E7 FT	3716.2 FPS	38485 NM
32 KM^2/SEC^2	-4.08671 E7 FT	5877.7 FPS	38485 NM
68 KM^2/SEC^2	-1.92316 E7 FT	10366.5 FPS	38485 NM

Table 6.1.1-2 Aeroassist Conditions - Mars Entries

MARS CAPTURES

C3	INITIAL SEMIMAJOR AXIS	AERO VELOCITY REDUCTION	EXIT ORBIT APOGEE
8.23 KM^2/SEC^2	- 1.70712 E7 FT	3223.6 FPS	18108 NM
13 KM^2/SEC^2	-1.08087 E7 FT	4536.3 FPS	18108 NM
31 KM^2/SEC^2	-4.53267 E6 FT	8866.2 FPS	18108 NM
60 KM^2/SEC^2	-2.34188 E6 FT	14564.8 FPS	18108 NM

6.1.2 PLANETARY DATA

Table 6.1.2-1 summarizes the key data for Earth and Mars used in the analysis of the various aeroentries described. This includes information on planet shapes and sizes, spin rates, gravitational constants, and atmospheres.

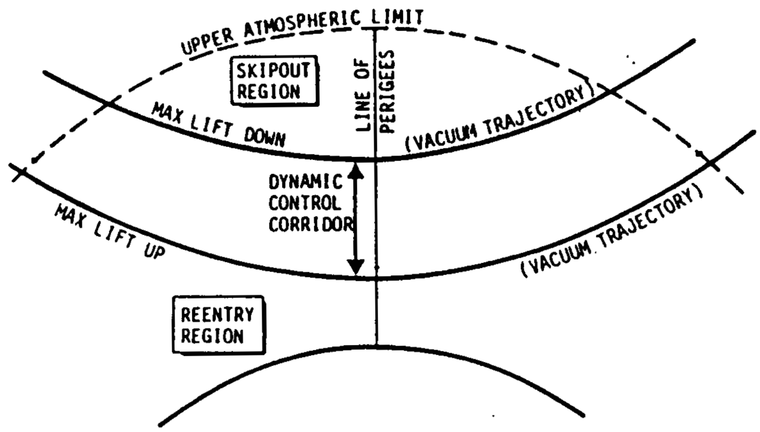
Table 6.1.2-1 Planetary Data

	EARTH	MARS
EQUATORIAL RADIUS	2.09256627E7 FT	1.114567E7 FT
POLAR RADIUS	2.08555024E7 FT	1.107448E7 FT
SPIN RATE	7.292115146E-5 RAD/SEC	7.0882181E-5 RADIAN/SEC
GRAVITY CONSTANT (MU)	1.407645794E16 FT3/SEC2	1.512468E15 FT3/SEC2
GRAVITY: J2 TERM	0.0010826	0.001965
GRAVITY: J3 TERM	-0.000002565	0
GRAVITY: J4 TERM	-0.000001608	0
ATMOSPHERE (NOMINAL)	1962 STANDARD	NORTH SUMMER NOMINAL (MARS REFERENCE ATMOS.)

6.1.3 CONTROL CORRIDOR DEFINITION

Safe flight through the atmosphere is restricted to a region which can be controlled with the lift available to the vehicle. The entry vehicle uses lift vector pointing to control its trajectory. The limits of this control are continuous lift up and continuous lift down. Trajectories run with these two limiting conditions define lower and upper (respectively) boundaries for vehicle flight. Conditions which exceed these boundaries will result in either re-entry or skipout.

For the purposes of establishing a working concept (Figure 6.1.3-1), these boundary profiles are characterized by their pre-entry vacuum perigee altitudes. The difference in the perigee altitudes for the two limiting conditions is known as the dynamic control corridor. This corridor represents the zone within which an orbital targeting routine must aim the vehicle for a successful aeropass. The size of this control corridor is established by error analysis (next section).



• CONTROL CORRIDOR BOUNDED BY:

CONTINUOUS LIFT UP CASE (LOWER BOUNDARY)

CONTINUOUS LIFT DOWN CASE (UPPER BOUNDARY)

• RESULTING CORRIDOR IS EXPRESSED AS THE PERIGEE ALTITUDE SEPARATION OF THE VACUUM TRAJECTORIES. USE OF VACUUM ORBITS EASES ORBITAL GUIDANCE TARGETING.

NOTE: CURVATURE OF TRAJECTORY INVERTED BY VERTICAL EXAGGERATION OF DIAGRAM

Figure 6.1.3-1 Control Corridor Definition

6.1.4 AERO ERROR ANALYSIS ASSUMPTIONS

An error analysis was conducted for each of the aeroassist entry conditions to determine levels of trajectory control required. This error analysis evaluates the uncertainties in variables of the entry process. By sizing the level of aerodynamic control required, an estimate of each vehicle's L/D can be made once control corridor sensitivities have been derived. Table 6.1.4-1 summarizes error analysis assumptions that are common to all entries. The uncertainties that were accounted for include: navigation errors, final midcourse correction burn uncertainties, atmospheric density variability, vehicle angle of attack and ballistic coefficient variations. These variables are discussed in greater detail in each error analysis page that follows.

6.2 EARTH RETURN RESULTS

The following sections summarize the data and results for Earth return aeroassists. These include return from GEO, from the Moon, and from a planetary boost mission.

Table 6.1.4-1 Aero Error Analysis Assumptions

ASSESSMENT OF ENTRY ERRORS SETS CONTROL CORRIDOR SIZE AND L/D

THE FOLLOWING ASSUMPTIONS ARE COMMON

NAVIGATION:

1) EARTH AEROBRAKING UTILIZES GPS SYSTEM YIELDING 1020 FT AND 0.1 FPS NAV STATE ACCURACY

2) MARS AEROBRAKING UTILIZES OPTICAL NAV YIELDING 1.0 NM AND O.12 FPS ACCURACY PER 10000 NM SEP FROM MARS

FINAL NAV UPDATE FOR MIDCOURSE AT 1.5 HR FROM MARS ENTRY

MIDCOURSE: FINAL MIDCOURSE CORRECTION AT ENTRY MINUS 1.0 HOUR

ATMOSPHERE:

1) EARTH DENSITY VARIABILITY = ± 30 %

2) MARS DENSITY VARIABILITY = ± 50 %

ANGLE OF ATTACK UNCERTAINTY: ± 2.0° ON 9.0° (EARTH) OR ±2.0° ON 12.0° (MARS)

BALLISTIC COEFFICIENT UNCERTAINTY: ± 8% ON W/CDA

IMPACT OF ALL ERRORS EXPRESSED IN THE EQUIVALENT VARIATION IN PERIGEE ALTITUDE

6.2.1 GEO RETURN AEROASSIST- ERROR ANALYSIS

Table 6.2.1-1 summarizes the aeroassist error analysis conducted for the GEO return case. A series of error sources was considered with their impacts being normalized to an equivalent variation in vacuum perigee. The RSS total of these effects was then used to size the aero-control corridor and L/D of the vehicle. The sources were grouped into two categories: 1) targeting errors which cause the vehicle to miss its desired entry aim-point and 2) aerodynamic variations which cause the vehicle to fly a different atmospheric trajectory than expected.

Targeting Errors - The last opportunity to correct the vehicle's incoming trajectory occurs one hour before entry with a final midcourse correction burn. All errors prior to this point are nulled out and only those factors that disturb the burn and subsequent flight are considered.

a) Pointing Errors - Midcourse burn attitude errors due to IMU misalignment amount to about 0.1° based on current star tracker and IMU drift assessments. This translates to a 140 ft error in vacuum perigee altitude.

b) Cutoff Errors - Accelerometer error for a 20 fps correction burn.

c) Navigation Error - Earth aeroassist can make use of the Global Positioning System (GPS) which is a set of highly accurate navigation satellites. Estimates of the GPS error at this stage are 1020 ft in position and 0.1 fps in velocity. This leads to perigee errors of 1044 ft and 404 ft, respectively.

Table 6.2.1-1 GEO Aero-entry Error Analysis

		EQUIVALENT PERIGEE ERROR	
• TARGETING ERRORS (FINAL CORRECTION BURN AT ENTRY MINUS 1 HR)			
	- POINTING ERROR	= 140 FT	± .1 DEG
	- CUTOFF ERROR	= 1333 FT	.33 FPS ACCELEROMETER
	- NAV ERROR	= 1044 FT	FROM 1020 FT POSITION UNCERTAINTY
		404 FT	FROM 0.1 FPS VELOCITY UNCERTAINTY
• 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	± 8% W/C_DA
• RSS		= ± 1780 FT = ± 0.29 NM FROM TARGETING	
		= ± 11400 FT = ± 1.87 NM FROM AERODYNAMICS	
		= ± 11500 FT = ± 1.90 NM NET VARIATION	

CONCLUSION: 5.04 N.M. CONTROL CORRIDOR REQUIRED TO COVER ERRORS WITH 33% MARGIN

<u>Aerodynamic Variations</u> - No two aero-entries will be exactly alike. The impact of variations in the atmosphere and the vehicle are accounted for here.

a) Atmospheric Uncertainty - The unknown component of the Earth's atmospheric density variation is currently estimated to be about 30%.

b) L/D Uncertainty - An angle-of-attack variation of 2° due to variations in the entry location and aerodynamics consistent with Viking and Shuttle data.

c) Ballistic Uncertainty - Weight uncertainty = 150 lbs (propellant residual uncertanty), coefficient of drag (Cd) variation = 5% (Shuttle and Viking experience), and brake area variation = 3% (to cover uncertainties in flexible brake geometry). The RSS effect of these factors on ballistic coefficient is 8%.

Because all the above factors are independent their effects are RSS'ed together to yield a net variation in perigee of ±1.90 nm. This figure is increase 33% to account for control lags and other dynamic effects (based on aero-guidance experience) which gives a net control corridor requirement of ±2.52 nm, or a net width of 5.04 nm. This size control corridor sets an L/D of 0.12 for the entry vehicle.

6.2.1.1 GEO RETURN - CONTROL & LOADS

Figure 6.2.1.1-1 summarizes aeroassist control corridor growth and vehicle deceleration loads as a function of L/D. Various entry trajectories were generated utilizing a pre-entry ellipse with an apogee of 19323 nm that is consistent with return from a geosynchronous orbit. The post-aero exit orbit is targeted to an apogee of 245 nm which represents return to the Space Station (however return to a lower park orbit will not significantly change the results). Aerodynamic L/D and ballistic coefficient were varied for continuous lift up and lift down trajectories to generate the parametric data base. Because of natural sensitivities, this data is shown as a function of L/D.

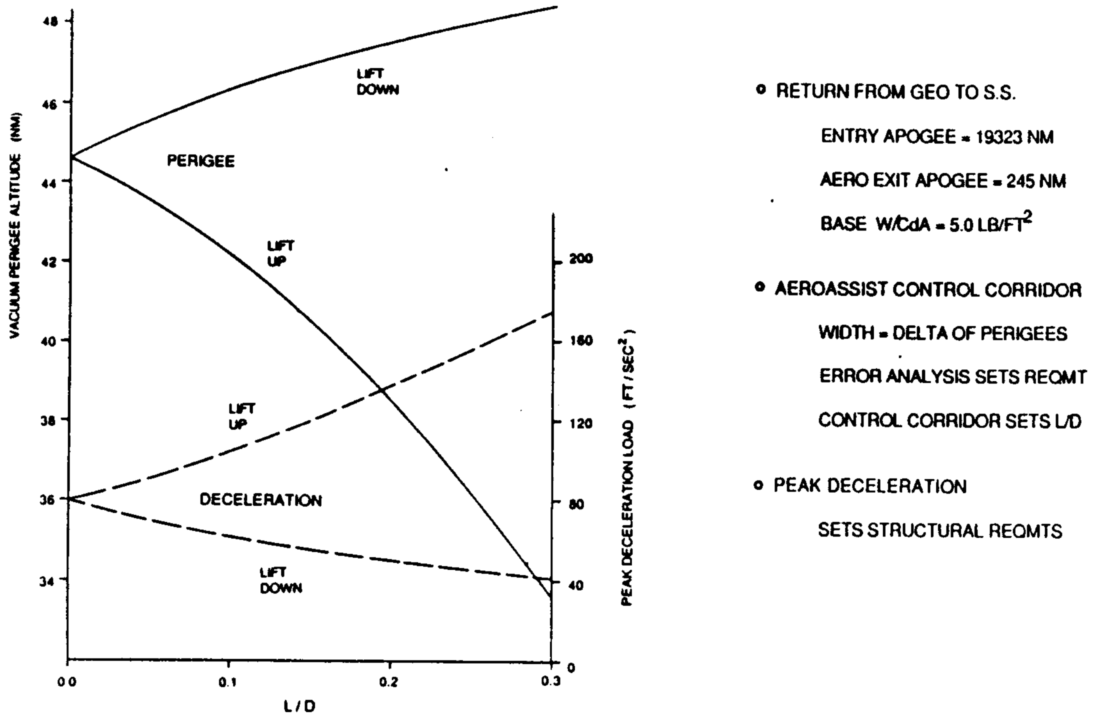


Figure 6.2.1.1-1 GEO Return Control & Loads

The difference between the pre-entry vacuum perigees for lift up and lift down aero-trajectories defines a control corridor width which represents the region in which the vehicle can be steered to the desired exit conditions with the available lift. Since the error analysis of the previous section has defined the magnitude of this control corridor, the vehicle's required L/D is set. For a control corridor width of 5.04 nm an L/D of 0.12 is required for GEO return.

Peak entry deceleration is shown for continuous lift up and lift down trajectories. The highest values of deceleration are always encountered in the lift up case which is thus used as a worst case loading condition for structural sizing. For this case, an L/D of 0.12 results in peak loads of 3.5 g's.

6.2.1.2 GEO RETURN - HEATING

Figure 6.2.1.2-1 shows heating information for return from GEO to the Space Station. Stagnation point convective heating values are calculated using a modified Fay-Riddell method normalized to a 1.0 ft. radius sphere. When this convective heating is combined with an estimate of non-equilibrium heating the net heat flux on the brake can be computed. The data shown in the charts is the convective heating only. Heating data has been shown as a function of ballistic coefficient which is its principal sensitivity over the range of the study space.

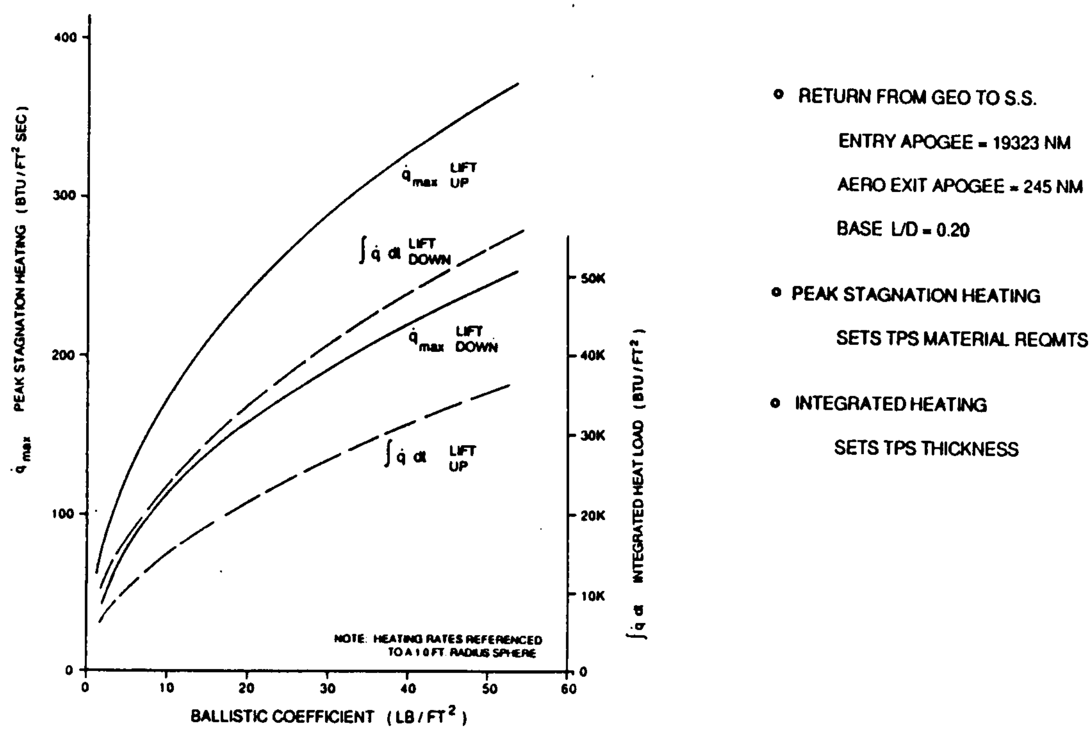


Figure 6.2.1.2-1 GEO Return Heating

Peak stagnation heating determines which TPS materials are acceptable for the aerobrake. The lift up condition generates maximal peak heating values. Integrated stagnation heating is shown for the lift down maximal condition. This parameter determines the required thickness of the aerobrake's insulating TPS.

6.2.2 LUNAR RETURN AEROASSIST

Table 6.2.2-1 shows entry error analysis for the lunar return condition. The primary difference between the lunar entry error analysis and that conducted for the GEO return is in the sensitivity of the incoming trajectory to dispersions. The lunar entry condition is faster because of the much higher apogee of the incoming orbit (287700 nm), consistent with a lunar free return. The actual dispersions are the same because of a common Earth environment for entry.

Table 6.2.2-1 Lunar Return Aero-entry Error Analysis

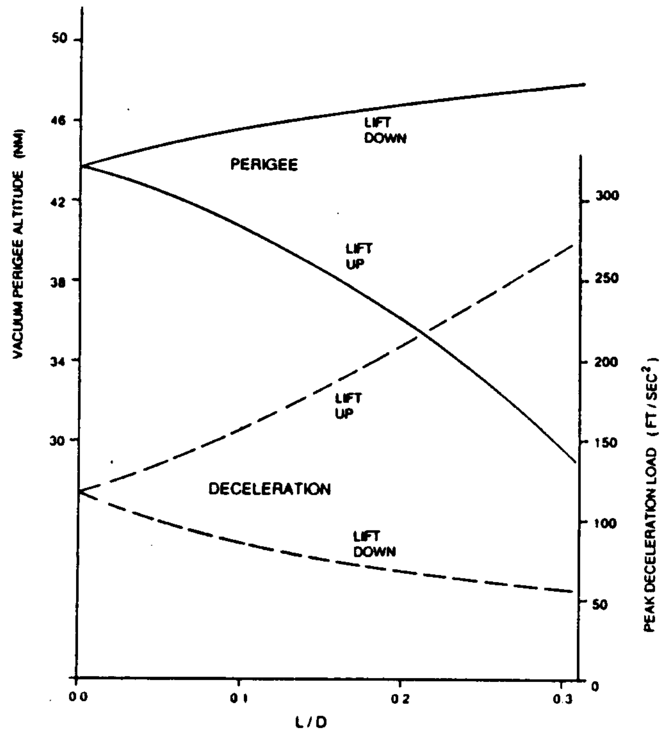
	EQUIVALENT PERIGEE ERROR	
• TARGETING ERRORS (FINAL CORRECTION BURN AT ENTRY MINUS 1 HR)		
- POINTING ERROR	= 140 FT	± .1 DEG
- CUTOFF ERROR	= 1320 FT	.33 FPS ACCELEROMETER
- NAV ERROR	= 1030 FT	FROM 1020 FT POSITION UNCERTAINTY
	400 FT	FROM 0.1 FPS VELOCITY UNCERTAINTY
• AERODYNAMIC VARIATION		
- ATMOSPHERIC UNCERTAINTY	= 18800 FT	± 30% DENSITY
- L/D UNCERTAINTY	= 10900 FT	± 2° AT 8° ANGLE OF ATTACK (± 30% L/D)
- BALLISTIC UNCERTAINTY	= 1600 FT	± 8% W/C_DA
• RSS	= ± 1720 FT = ± 0.28 NM FROM TARGETING	
	= ± 12500 FT = ± 2.06 NM FROM AERODYNAMICS	
	= ± 12600 FT = ± 2.08 NM NET VARIATION	

CONCLUSION: 5.53 N.M. CONTROL CORRIDOR REQUIRED TO COVER ERRORS WITH 33% MARGIN

The 5.53 nm net control corridor size sets a minimum L/D requirement of 0.11 for the entry vehicle based on the control parametrics in the next section. An analysis of aerobrake sizing actually increased this L/D for load relief peculiar to the lunar vehicle application. This issue is discussed in detail later on in this report.

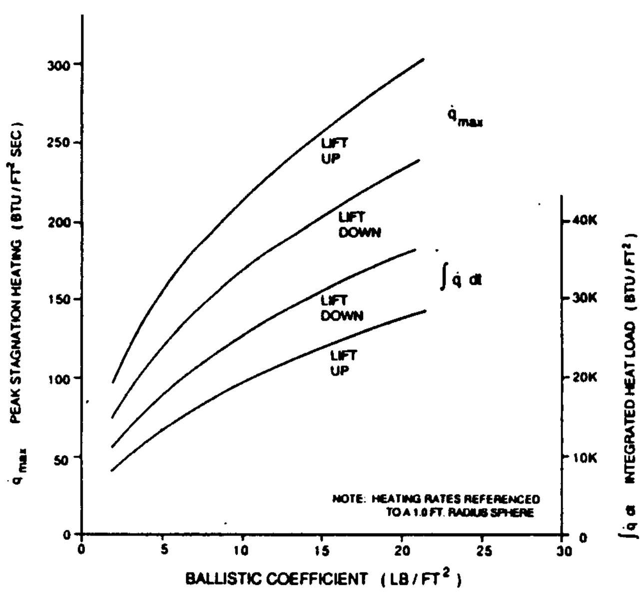
Figure 6.2.2-1 shows control corridor and deceleration loads data for Lunar return. Control corridor data is derived by differencing the vacuum perigee curves for lift up and lift down conditions. The peak deceleration level is used to size structural elements.

Figure 6.2.2-2 shows heating data for the Lunar return case. Peak stagnation heating determines which materials are thermally suitable for brake construction while integrated heating sets the required TPS thickness.



- RETURN FROM MOON TO S.S.
  - ENTRY APOGEE = 287700 NM
  - AERO EXIT APOGEE = 245 NM
  - BASE W/CdA = 5.0 LB/FT^2
- AEROASSIST CONTROL CORRIDOR
  - WIDTH = DELTA OF PERIGEES
  - ERROR ANALYSIS SETS REQMT
  - CONTROL CORRIDOR SETS L/D
- PEAK DECELERATION
  - SETS STRUCTURAL REQMTS

Figure 6.2.2-1 Lunar Return Controls & Loads



- RETURN FROM MOON TO S.S.
  - ENTRY APOGEE = 287700 NM
  - AERO EXIT APOGEE = 245 NM
  - BASE L/D = 0.10
- PEAK STAGNATION HEATING
  - SETS TPS MATERIAL REQMTS
- INTEGRATED HEATING
  - SETS TPS THICKNESS

Figure 6.2.2-2 Lunar Return Heating

6.2.3 PLANETARY BOOST RETURN AEROASSIST

Figure 6.2.3-1 shows the control and loads data for return from a worst case planetary boost mission. Initial entry orbit has an apogee of 130900 nm which results from a very energetic planetary deploy mission (#17500, Planet B & C). Because the energy of this return is very close to that for the lunar return case the error analysis is not shown but would be almost identical to that shown in the lunar return section above.

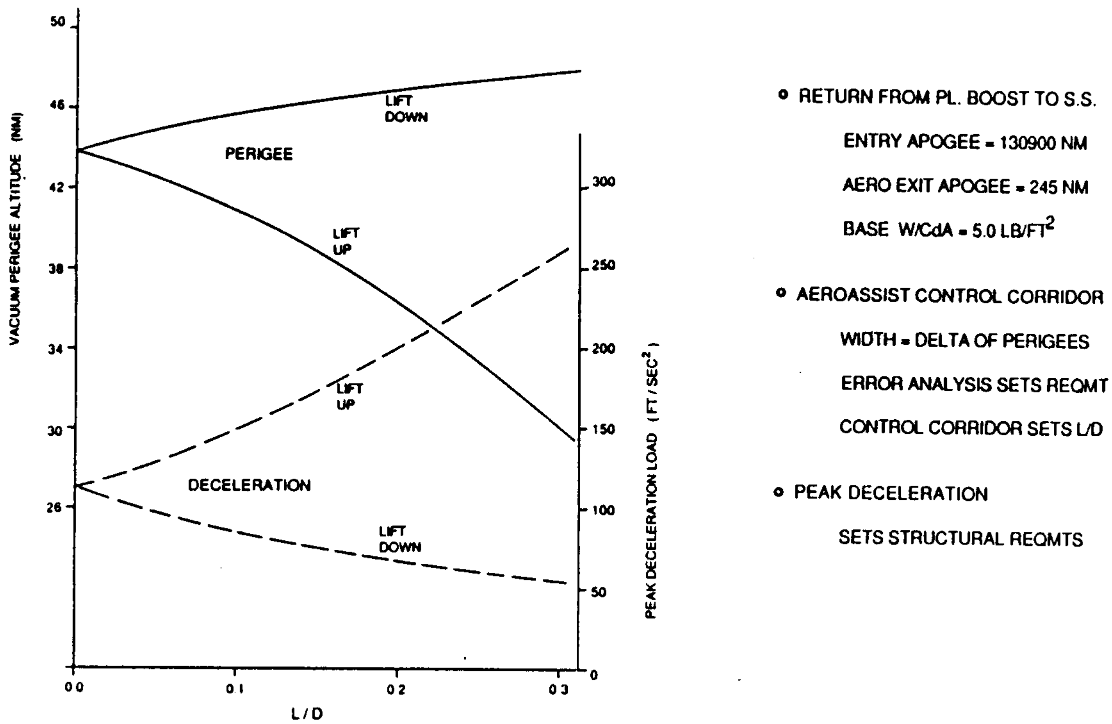
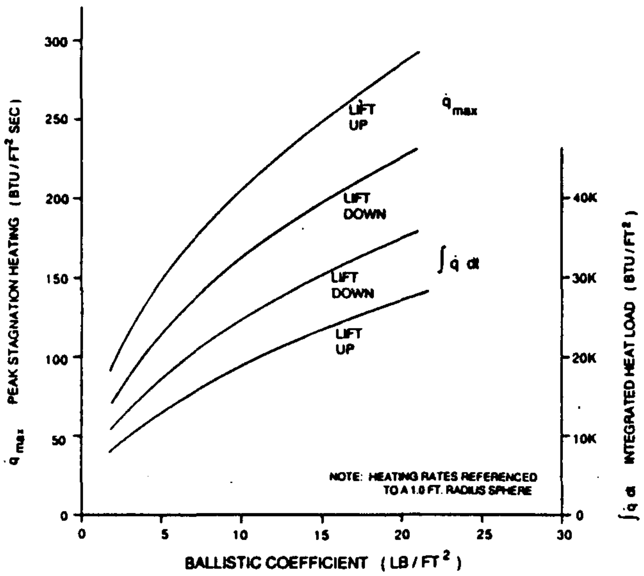


Figure 6.2.3-1 Planetary Boost Return - Control & Loads

Planetary boost convection heating data is shown in Figure 6.2.3-2.



- RETURN FROM PL. BOOST TO S.S.
  - ENTRY APOGEE = 130900 NM
  - AERO EXIT APOGEE = 245 NM
  - BASE L/D = 0.10
- PEAK STAGNATION HEATING
  - SETS TPS MATERIAL REQMTS
- INTEGRATED HEATING
  - SETS TPS THICKNESS

Figure 6.2.3-2 Planetary Boost Return Heating

6.3 MARS AEROCAPTURE RESULTS

The following sections summarize the data and results for Mars aerocaptures. These represent missions where an initially hyperbolic encounter trajectory with Mars is modified into a closed orbit about the planet by means of an aeroassist maneuver. Because the aeroassist maneuver captures the vehicle into Mars orbit it is termed an "aerocapture". Four different hyperbolic encounter energies were considered: C_3 = 8.2, 13, 32, and 60 km^2/sec^2. The aero-exit apogees were targeted to 18108 nm which corresponds to a Mars synchronous condition.

6.3.1 MARS AEROCAPTURE, C3=8.2 - ERROR ANALYSIS

Table 6.3.1-1 summarizes the error analysis conducted to derive Mars capture control requirements. All errors are normalized into equivalent variations in perigee altitude which is the strongest driver to aeroentry uncertainty. The variables are categorized into targeting errors and aerodynamic uncertainties.

Table 6.3.1-1 Mars Capture Aero-entry Error Analysis: C3=8.2 km^2/sec^2

	EQUIVALENT PERIGEE ERROR	
• TARGETING ERRORS (FINAL CORRECTION BURN AT ENTRY MINUS 1 HR)		
- POINTING ERROR	= 138 FT	± .1 DEG
- CUTOFF ERROR	= 1300 FT	.33 FPS ACCELEROMETER
- NAV ERROR	= 6694 FT	FROM 6883 FT POSITION UNCERTAINTY
	776 FT	FROM 0.136 FPS VELOCITY UNCERTAINTY
• AERODYNAMIC VARIATION		
- ATMOSPHERIC UNCERTAINTY	= 14900 FT	± 50% DENSITY
- L/D UNCERTAINTY	= 5200 FT	± 2° AT 12° ANGLE OF ATTACK (± 17% L/D)
- BALLISTIC UNCERTAINTY	= 2400 FT	± 8% W/C_DA
• RSS	= ± 6860 FT = ± 1.13 NM FROM TARGETING	
	= ± 16000 FT = ± 2.63 NM FROM AERODYNAMICS	
	= ± 17900 FT = ± 2.94 NM NET VARIATION	

CONCLUSION: 7.82 N.M. CONTROL CORRIDOR REQUIRED TO COVER ERRORS WITH 33% MARGIN

The targeting errors result from inaccuracies in the execution of the final correction burn one hour before entry and include allocations for pointing error, cutoff error and navigation error. The pointing error of 0.1° results from stellar update alignment errors and subsequent IMU drift which corrupts the desired pointing of the final correction. The velocity cutoff error of 0.33 fps results from onboard accelerometer errors and is a working figure derived from the OTV configuration. The navigation error is representative of video navigation capabilities and represents a final onboard trajectory solution 1.5 hours before entry (half an hour before the final midcourse). These independent error contributions are RSS'ed together to yield a net perigee variation due to targeting errors of ± 1.13 nmi.

The aerodynamic errors result from variations in the Mars atmospheric density as well as in vehicle aerodynamic properties during the entry phase. A Martian atmospheric variation of ± 50% in density is assumed (as compared wit Earth applications) which is derived from the cool versus warm density models contained in the Mars Reference Atmosphere (Kliore, 1982). The L/D uncertainty results from a vehicle trim attitude variability of ± 2° in the continuum flo region of entry. The size of the variation is that derived for the OTV, when the Mars vehicle becomes better defined a similar derivation will be possible for its specific configuration. Finally, a ballistic uncertainty of ± 8% is carried which also represents a quantity derived from the OTV. The RSS of the arodynamic variations is ± 2.63 nm in nominal perigee altitude.

When the targeting and aerodynamic errors are combined a net perigee variation of ± 2.94 nmi. results. This variation in the aeroentry traj covered by the control capability of the vehicle in order to successfully accomplish the aeroassist. From experience with the OTV aeroentry process a 33% margin is added to the net variation to account for control lags. This results in a net control corridor requirement of 7.82 nm which then sets the L/D of a Mars entry vehicle with this hyperbolic encounter C_3 at 0.32 using the control sensitivity data contained in the next section.

6.3.1.1 MARS AEROCAPTURE, C3=8.2 - CONTROL & LOADS

Figure 6.3.1.1-1 summarizes the growth in control corridor and deceleration loads as a function of L/D. Various entry trajectories were generated utilizing a pre-entry hyperbolic C_3 of 8.2 km^2/sec^2 and a Mars capture apogee of 18108 nm (post-aero). Aerodynamic L/D and ballistic coefficient were varied for continuous lift up and lift down trajectories to generate the parametric data base. Because of natural sensitivities the data on pre-entry perigee altitude and peak deceleration is shown as a function of L/D.

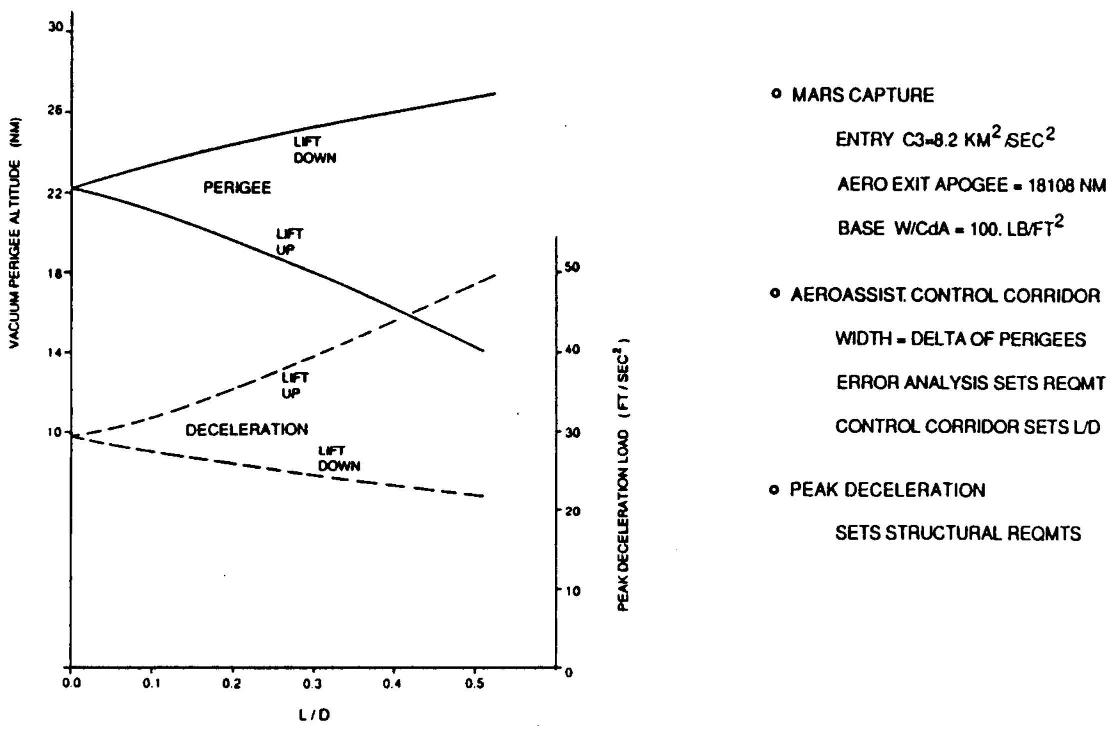


Figure 6.3.1.1-1 Mars Capture, C3=8.2 - Control & Loads

The difference between the pre-entry vacuum perigees for lift up and lift down aero-trajectories defines a control corridor width which represents the region in which the vehicle can be steered to the desired exit conditions with the available lift. With error analysis having defined the magnitude of this control corridor, the vehicle's required L/D is set. For a control corridor width of 7.82 nm, an L/D of 0.32 is required for Mars capture at this C_3.

Peak entry deceleration is shown for lift up and lift down trajectories. The highest values of deceleration are always encountered in the continuous lift up case which is thus used as a worst case loading condition for structural sizing.

6.3.1.2 MARS AEROCAPTURE, C3=8.2 - HEATING

Figure 6.3.1.2-1 shows heating information for the Mars capture with an encounter C_3 of 8.2 km^2/sec^2. Stagnation point convective heating values are calculated using a modified Fay-Riddell method normalized to a 1.0 ft. radius sphere. When this convective heating is combined with an estimate of non-equilibrium heating the net heat flux on the aerobrake can be computed. The data shown in the charts is the convective heating only. Heating data has been shown as a function of ballistic coefficient which is its principal sensitivity over the range of the study space.

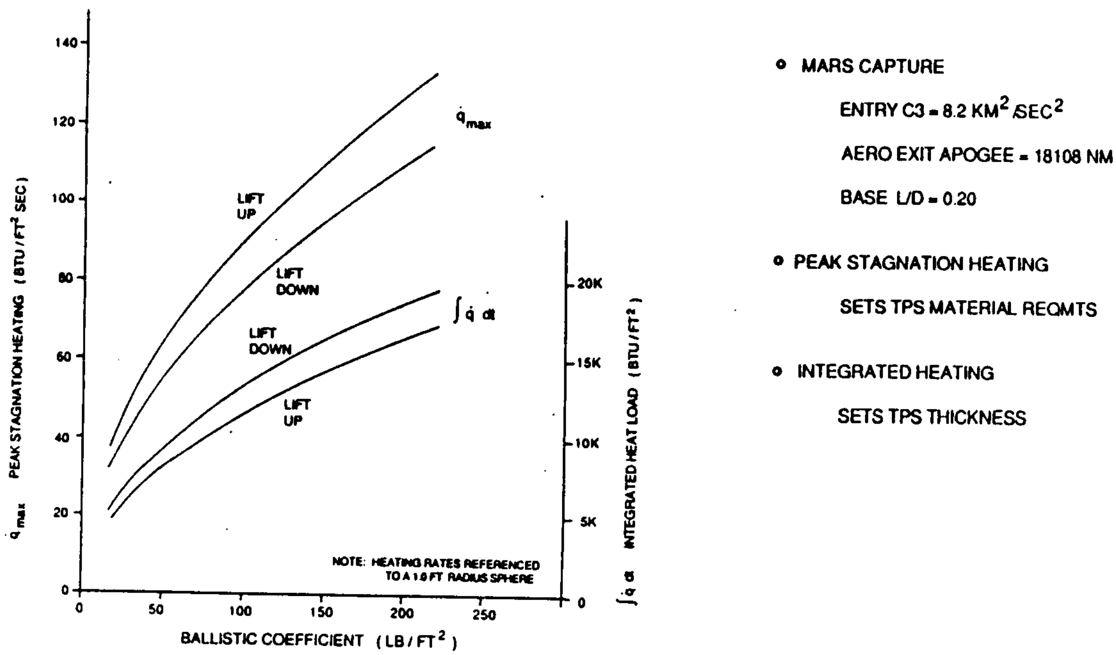


Figure 6.3.1.2-1 Mars Capture, C3=8.2 - Heating

Peak stagnation heating determines which TPS materials are acceptable for the aerobrake. The lift up condition shown generates maximal peak heating values. Integrated stagnation heating is shown for the lift down maximal condition. This parameter determines the required thickness of the aerobrake's insulating TPS.

6.3.2 MARS AEROCAPTURE, C3=13

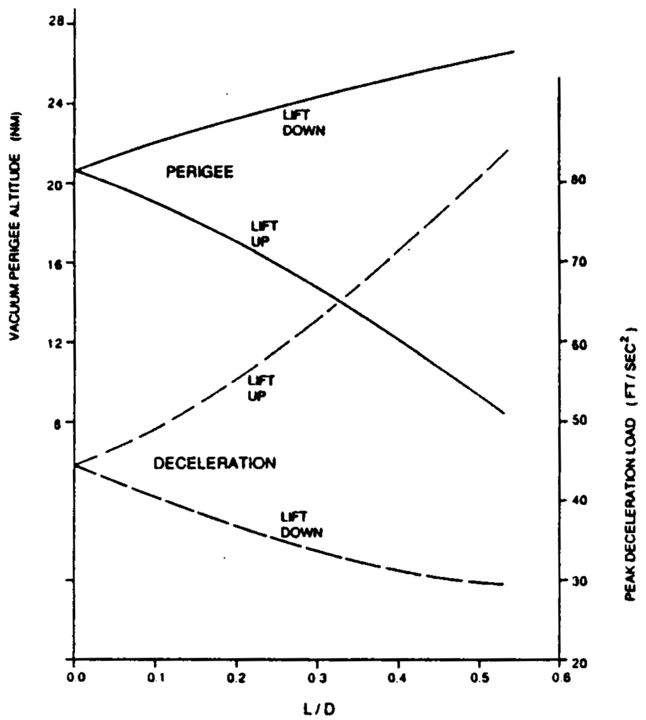
Table 6.3.2-1 summarizes the error analysis conducted for a Mars capture with an encounter C_3 of 13 km^2/sec^2. The primary difference between this analysis and that conducted for the previous 8.2 km^2/sec^2 Mars capture is in the dispersion sensitivity of the faster incoming trajectory. In addition the final navigation solution at entry minus 1.5 hours occurs further out which increases the state vector error to 7824 ft in position and 0.155 fps in velocity. The other dispersions are the same because of a common Mars environment for entry.

Table 6.3.2-1 Mars Capture Aero-entry Error Analysis: C3=13 km^2/sec^2

	EQUIVALENT PERIGEE ERROR	
• TARGETING ERRORS (FINAL CORRECTION BURN AT ENTRY MINUS 1 HR)		
- POINTING ERROR	= 136 FT	± .1 DEG
- CUTOFF ERROR	= 1282 FT	.33 FPS ACCELEROMETER
- NAV ERROR	= 7688 FT	FROM 7824 FT POSITION UNCERTAINTY
	880 FT	FROM 0.155 FPS VELOCITY UNCERTAINTY
• AERODYNAMIC VARIATION		
- ATMOSPHERIC UNCERTAINTY	= 15200 FT	± 50% DENSITY
- L/D UNCERTAINTY	= 6700 FT	± 2° AT 12° ANGLE OF ATTACK (± 17% L/D)
- BALLISTIC UNCERTAINTY	= 2400 FT	± 8% W/C_DA
• RSS	= ± 7850 FT	= ± 1.29 NM FROM TARGETING
	= ± 16800 FT	= ± 2.76 NM FROM AERODYNAMICS
	= ± 18500 FT	= ± 3.05 NM NET VARIATION

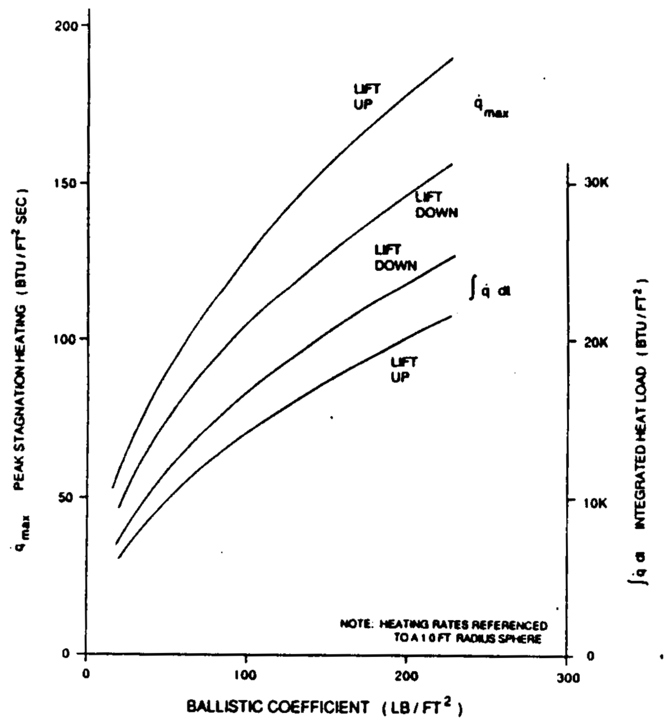
CONCLUSION: 8.12 N.M. CONTROL CORRIDOR REQUIRED TO COVER ERRORS WITH 33% MARGIN

The 8.12 nm net control corridor size sets a minimum L/D requirement of 0.26 for the entry vehicle when control parametrics (Figure 6.3.2-1) are utilized. Figure 6.3.2-2 summarizes the aerocapture heating environment for this encounter condition.



- MARS CAPTURE
  - ENTRY C3=13 KM²/SEC²
  - AERO EXIT APOGEE = 18108 NM
  - BASE W/CdA = 100. LB/FT²
- AEROASSIST CONTROL CORRIDOR
  - WIDTH = DELTA OF PERIGEES
  - ERROR ANALYSIS SETS REQMT
  - CONTROL CORRIDOR SETS L/D
- PEAK DECELERATION
  - SETS STRUCTURAL REQMTS

Figure 6.3.2-1 Mars Capture, C3=13 - Control & Loads



- MARS CAPTURE
  - ENTRY C3 = 13 KM²/SEC²
  - AERO EXIT APOGEE = 18108 NM
  - BASE L/D = 0.20
- PEAK STAGNATION HEATING
  - SETS TPS MATERIAL REQMTS
- INTEGRATED HEATING
  - SETS TPS THICKNESS

Figure 6.3.2-2 Mars Capture, C3=13 - Heating

6.3.3 MARS AEROCAPTURE, C3=31

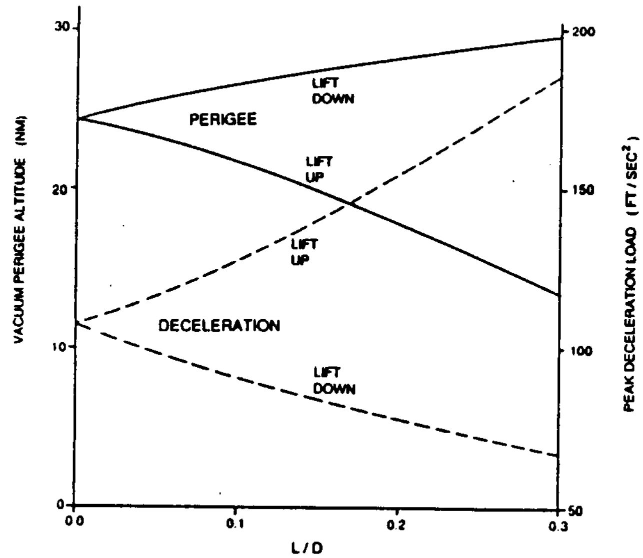
Table 6.3.3-1 summarizes the error analysis conducted for a Mars capture with an encounter C_3 of 31 km^2/sec^2. The primary difference between this analysis and that conducted for the 8.2 km^2/sec^2 Mars capture is in the dispersion sensitivity of the faster incoming trajectory. In addition the final navigation solution at entry minus 1.5 hours occurs further out which increases the state vector error to 10720 ft in position and 0.212 fps in velocity. The other dispersions are the same because of a common Mars environment for entry.

Table 6.3.3-1 Mars Capture Aero-entry Error Analysis: C3=31 km^2/sec^2

	EQUIVALENT PERIGEE ERROR	
• TARGETING ERRORS (FINAL CORRECTION BURN AT ENTRY MINUS 1 HR)		
- POINTING ERROR	= 132 FT	± .1 DEG
- CUTOFF ERROR	= 1245 FT	.33 FPS ACCELEROMETER
- NAV ERROR	= 10684 FT	FROM 10720 FT POSITION UNCERTAINTY
	1181 FT	FROM 0.212 FPS VELOCITY UNCERTAINTY
• AERODYNAMIC VARIATION		
- ATMOSPHERIC UNCERTAINTY	= 15600 FT	± 50% DENSITY
- L/D UNCERTAINTY	= 11500 FT	± 2° AT 12° ANGLE OF ATTACK (± 17% L/D)
- BALLISTIC UNCERTAINTY	= 2500 FT	± 8% W/C_DA
• RSS	= ± 10820 FT = ± 1.78 NM FROM TARGETING	
	= ± 19500 FT = ± 3.22 NM FROM AERODYNAMICS	
	= ± 22300 FT = ± 3.67 NM NET VARIATION	

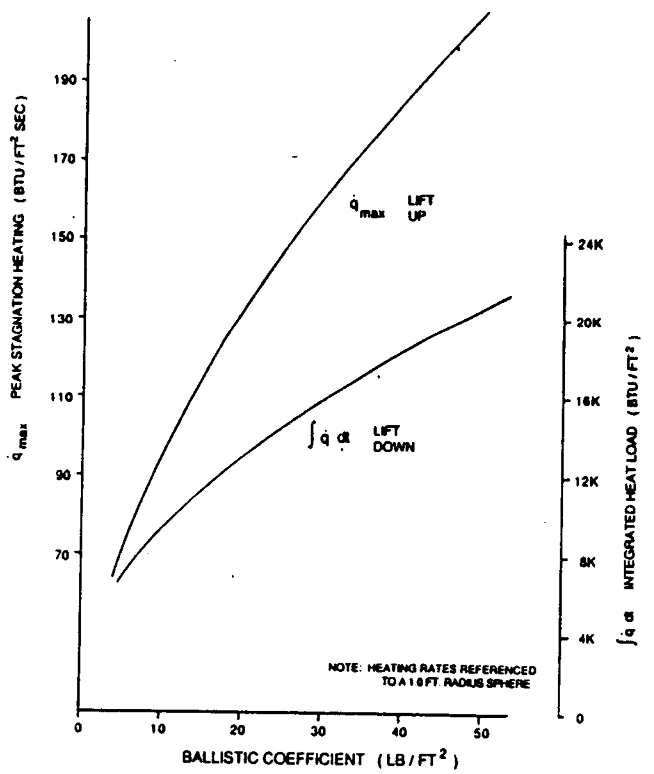
CONCLUSION: 9.76 N.M. CONTROL CORRIDOR REQUIRED TO COVER ERRORS WITH 33% MARGIN

The 9.76 nm net control corridor size sets a minimum L/D requirement of 0.19 for the entry vehicle when control parametrics (Figure 6.3.3-1) are utilized. Figure 6.3.3-2 summarizes the aerocapture heating environment for this encounter condition.



- MARS CAPTURE
  - ENTRY C3=31 KM^2/SEC^2
  - AERO EXIT APOGEE = 18108 NM
  - BASE W/CdA = 25. LB/FT^2
- AEROASSIST CONTROL CORRIDOR
  - WIDTH = DELTA OF PERIGEES
  - ERROR ANALYSIS SETS REQMT
  - CONTROL CORRIDOR SETS L/D
- PEAK DECELERATION
  - SETS STRUCTURAL REQMTS

Figure 6.3.3-1 Mars Capture, C3=31 - Control & Loads



- MARS CAPTURE
  - ENTRY C3 = 31 KM^2/SEC^2
  - AERO EXIT APOGEE = 18108 NM
  - BASE L/D = 0.20
- PEAK STAGNATION HEATING
  - SETS TPS MATERIAL REQMTS
- INTEGRATED HEATING
  - SETS TPS THICKNESS

Figure 6.3.3-2 Mars Capture, C3=31 - Heating

6.3.4 MARS AEROCAPTURE, C3=60

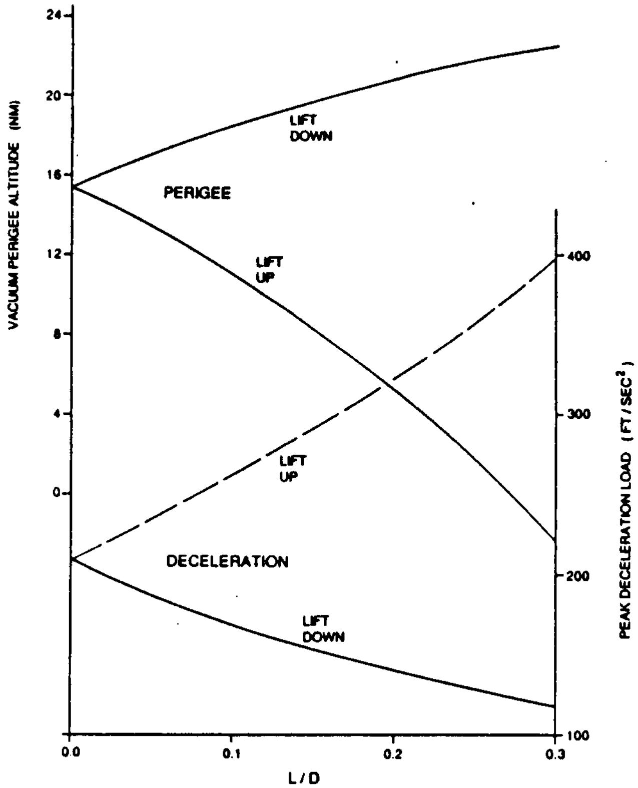
Table 6.3.4-1 summarizes the error analysis conducted for a Mars capture with an encounter C_3 of 60 km^2/sec^2. The primary difference between this analysis and that conducted for the 8.2 km^2/sec^2 Mars capture is in the dispersion sensitivity of the faster incoming trajectory. In addition the final navigation solution at entry minus 1.5 hours occurs further out which increases the state vector error to 14270 ft in position and 0.282 fps in velocity. The other dispersions are the same because of a common Mars environment for entry.

Table 6.3.4-1 Mars Capture Aero-entry Error Analysis: C3=60 km^2/sec^2

	EQUIVALENT PERIGEE ERROR	
• TARGETING ERRORS (FINAL CORRECTION BURN AT ENTRY MINUS 1 HR)		
- POINTING ERROR	= 130 FT	± .1 DEG
- CUTOFF ERROR	= 1222 FT	.33 FPS ACCELEROMETER
- NAV ERROR	= 14284 FT	FROM 14270 FT POSITION UNCERTAINTY
	1552 FT	FROM 0.282 FPS VELOCITY UNCERTAINTY
• AERODYNAMIC VARIATION		
- ATMOSPHERIC UNCERTAINTY	= 16100 FT	± 50% DENSITY
- L/D UNCERTAINTY	= 17300 FT	± 2° AT 12° ANGLE OF ATTACK (± 17% L/D)
- BALLISTIC UNCERTAINTY	= 2600 FT	± 8% W/C_DA
• RSS	= ± 14420 FT = ± 2.37 NM FROM TARGETING	
	= ± 23800 FT = ± 3.91 NM FROM AERODYNAMICS	
	= ± 27800 FT = ± 4.58 NM NET VARIATION	

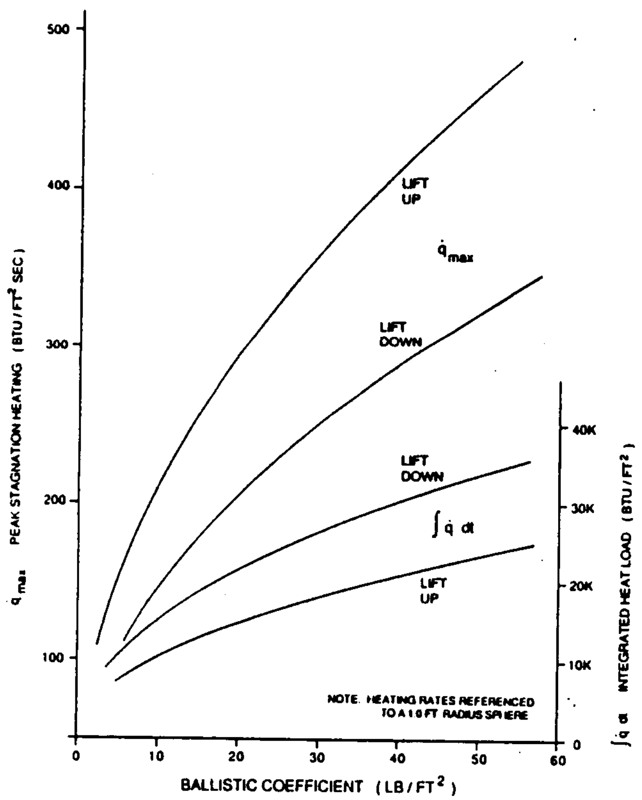
CONCLUSION: 12.18 N.M. CONTROL CORRIDOR REQUIRED TO COVER ERRORS WITH 33% MARGIN

The 12.18 nm net control corridor size sets a minimum L/D requirement of 0.16 for the entry vehicle when control parametrics (Figure 6.3.4-1) are utilized. Figure 6.3.4-2 summarizes the aerocapture heating environment for this encounter condition.



- MARS CAPTURE
  - ENTRY C3=60 KM^2/SEC^2
  - AERO EXIT APOGEE = 18108 NM
  - BASE W/CdA = 100. LB/FT^2
- AEROASSIST CONTROL CORRIDOR
  - WIDTH = DELTA OF PERIGEES
  - ERROR ANALYSIS SETS REQMT
  - CONTROL CORRIDOR SETS L/D
- PEAK DECELERATION
  - SETS STRUCTURAL REQMTS

Figure 6.3.4-1 Mars Capture, C3=60 - Control & Loads



- MARS CAPTURE
  - ENTRY C3 = 60 KM^2/SEC^2
  - AERO EXIT APOGEE = 18108 NM
  - BASE L/D = 0.20
- PEAK STAGNATION HEATING
  - SETS TPS MATERIAL REQMTS
- INTEGRATED HEATING
  - SETS TPS THICKNESS

Figure 6.3.4-2 Mars Capture, C3=60 - Heating

6.4 EARTH AEROCAPTURE RESULTS

The following sections summarize the data and results for Earth aerocaptures. These represent missions where an initially hyperbolic encounter trajectory with the Earth is modified into a closed orbit about the planet by means of an aeroassist maneuver. Because the aeroassist maneuver captures the vehicle into Earth orbit it is termed an "aerocapture". These missions differ in aero-exit conditions from the Earth return cases shown earlier in that their apogees are very high (38485 nm, Earth synchronous) to reduce heating and aerodynamic loads. Four different hyperbolic encounter energies were considered: C_3 = 8, 16, 32, and 68 km^2/sec^2.

6.4.1 EARTH AEROCAPTURE, C3=8.0 - ERROR ANALYSIS

Table 6.4.1-1 shows the results of entry error analysis conducted for the C_3 = 8.0 km^2/sec^2 Earth capture mission. Use of the GPS navigation system is baselined as in the Earth return cases. Also a somewhat higher base angle of attack (9°, consistent with generally higher L/D requirements) is used. The 2° variation in this higher angle of attack actually results in a somewhat lower L/D dispersion than for the Earth return error analyses. For a more extensive discussion of Earth aeroassist error analysis see "GEO Return Aeroassist - Error Analysis" (section 6.2.1).

Table 6.4.1-1 Earth Capture Aero-entry Error Analysis: C3=8 km^2/sec^2

	EQUIVALENT PERIGEE ERROR	
• TARGETING ERRORS (FINAL CORRECTION BURN AT ENTRY MINUS 1 HR)		
. POINTING ERROR	= 139 FT	± .1 DEG
. CUTOFF ERROR	= 1309 FT	.33 FPS ACCELEROMETER
. NAV ERROR	= 1025 FT	FROM 1020 FT POSITION UNCERTAINTY
	397 FT	FROM 0.1 FPS VELOCITY UNCERTAINTY
• AERODYNAMIC VARIATION		
· ATMOSPHERIC UNCERTAINTY	= 5600 FT	± 30% DENSITY
· L/D UNCERTAINTY	= 2300 FT	± 2° AT 9° ANGLE OF ATTACK (± 22% L/D)
· BALLISTIC UNCERTAINTY	= 1500 FT	± 8% W/C_DA
• RSS	= ± 1720 FT	= ± 0.28 NM FROM TARGETING
	= ± 6200 FT	= ± 1.03 NM FROM AERODYNAMICS
	= ± 6500 FT	= ± 1.06 NM NET VARIATION

CONCLUSION: 2.83 N.M. CONTROL CORRIDOR REQUIRED TO COVER ERRORS WITH 33% MARGIN

The net result of this error analysis for the entry C_3 is a 2.83 nmi control corridor requirement. This control corridor requirement translates to a vehicle L/D of 0.25 using the control parametrics presented in the next section.

6.4.1.1 EARTH AEROCAPTURE, C3=8.0 - CONTROL & LOADS

Figure 6.4.1.1-1 summarizes the growth in control corridor and deceleration loads as a function of L/D. Various entry trajectories were generated utilizing a pre-entry hyperbolic C_3 of 8.0 km^2/sec^2 and an Earth capture apogee of 38485 nm (post-aero). Aerodynamic L/D and ballistic coefficient were varied for continuous lift up and lift down trajectories to generate the parametric data base. Because of natural sensitivities the data on pre-entry perigee altitude and peak deceleration is shown as a function of L/D.

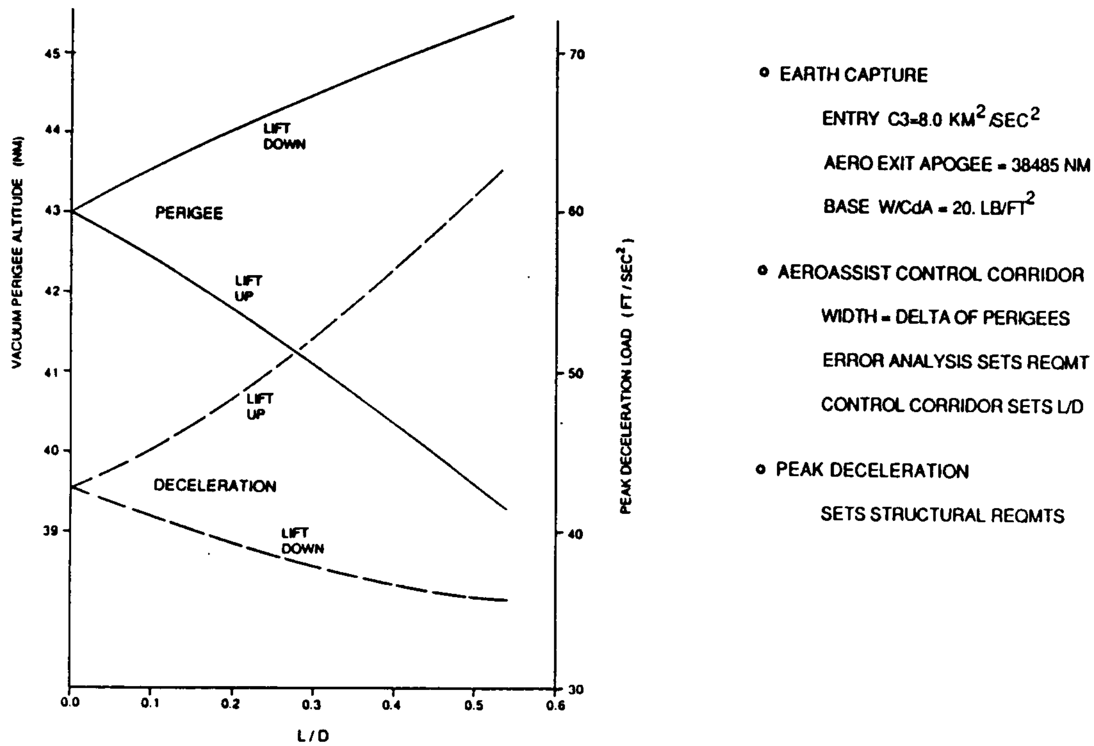


Figure 6.4.1.1-1 Earth Capture, C3=8 - Control & Loads

The difference between the pre-entry vacuum perigees for lift up and lift down aero-trajectories defines a control corridor width which represents the region in which the vehicle can be steered to the desired exit conditions with the available lift. With error analysis having defined the magnitude of this control corridor, the vehicle's required L/D is set. For a control corridor width of 2.83 nm an L/D of 0.25 is required for Earth capture from an encounter C_3 of 8.0 km^2/sec^2.

Peak entry deceleration is shown for lift up and lift down trajectories. The highest values of deceleration are always encountered in the continuous lift up case which is thus used as a worst case loading condition for structural sizing.

6.4.1.2 EARTH AEROCAPTURE, C3=8.0 - HEATING

Figure 6.4.1.2-1 shows heating information for Earth capture with an encounter C_3 of 8.0 km^2/sec^2. Stagnation point convective heating values are calculated using a modified Fay-Riddell method normalized to a 1.0 ft. radius sphere. When this convective heating is combined with an estimate of non-equilibrium heating the net heat flux on the aerobrake can be computed. The data shown in the charts is the convective heating only. Heating data has been shown as a function of ballistic coefficient which is its principal sensitivity over the range of the study space.

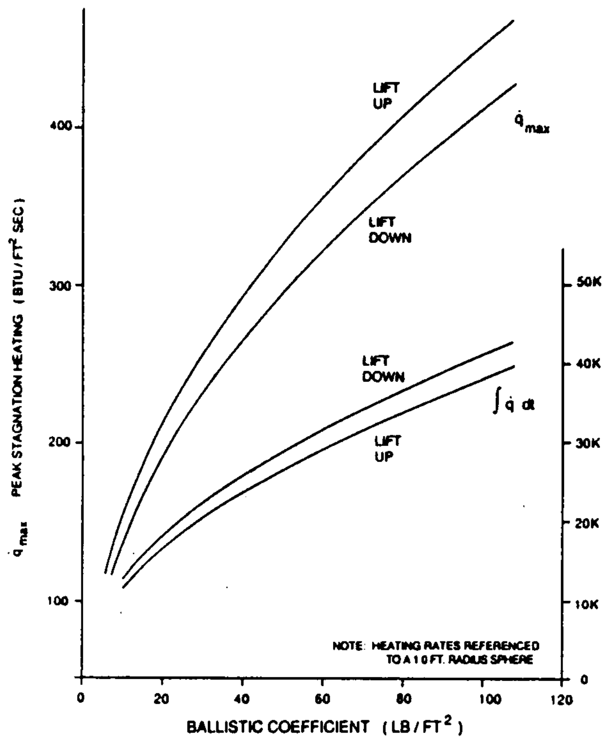


Figure 6.4.1.2-1 Earth Capture, C3=8 - Heating

Peak stagnation heating determines which TPS materials are acceptable for the aerobrake. The lift up condition shown generates maximal peak heating values. Integrated stagnation heating is shown for the lift down maximal condition. This parameter determines the required thickness of the aerobrake's insulating TPS.

6.4.2 EARTH AEROCAPTURE, C3=16

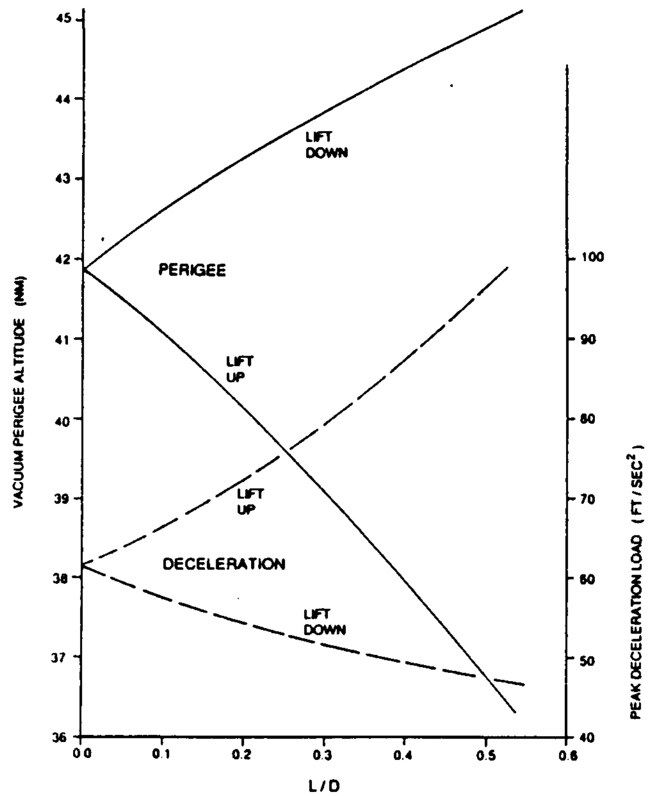
Table 6.4.2-1 summarizes the error analysis conducted for an Earth capture with an encounter C_3 of 16 km^2/sec^2. The primary difference between this analysis and that conducted for the previous 8.0 km^2/sec^2 capture is in the dispersion sensitivity of the faster incoming trajectory. The other dispersions are the same because of a common Earth environment for entry.

Table 6.4.2-1 Earth Capture Aero-entry Error Analysis: C3=16 km^2/sec^2

	EQUIVALENT PERIGEE ERROR	
• TARGETING ERRORS (FINAL CORRECTION BURN AT ENTRY MINUS 1 HR)		
. POINTING ERROR	= 138 FT	± .1 DEG
. CUTOFF ERROR	= 1301 FT	.33 FPS ACCELEROMETER
. NAV ERROR	= 1024 FT	FROM 1020 FT POSITION UNCERTAINTY
	394 FT	FROM 0.1 FPS VELOCITY UNCERTAINTY
• AERODYNAMIC VARIATION		
- ATMOSPHERIC UNCERTAINTY	= 5700 FT	± 30% DENSITY
- L/D UNCERTAINTY	= 3300 FT	± 2° AT 9° ANGLE OF ATTACK (± 22% L/D)
- BALLISTIC UNCERTAINTY	= 1500 FT	± 8% W/C_DA
• RSS	= ± 1700 FT	= ± 0.28 NM FROM TARGETING
	= ± 6800 FT	= ± 1.11 NM FROM AERODYNAMICS
	= ± 7000 FT	= ± 1.15 NM NET VARIATION

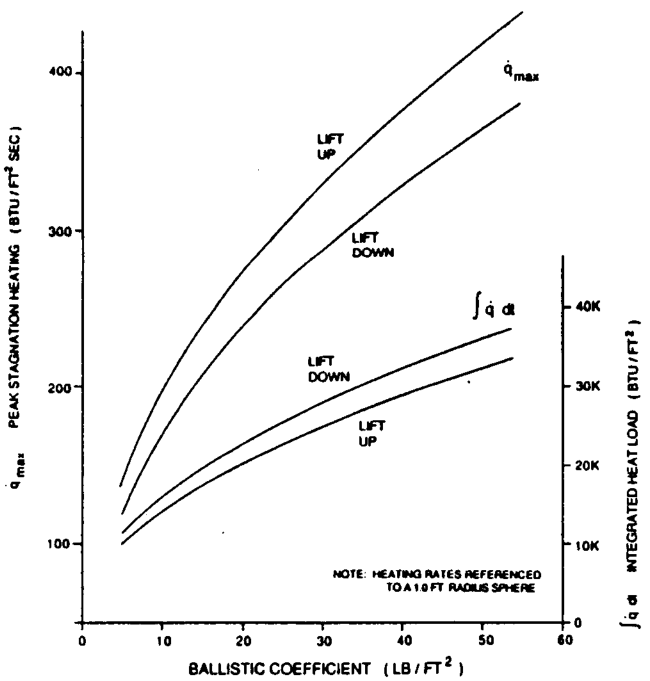
CONCLUSION: 3.05 N.M. CONTROL CORRIDOR REQUIRED TO COVER ERRORS WITH 33% MARGIN

The 3.05 nm net control corridor size sets a minimum L/D requirement of 0.195 for the entry vehicle when control parametrics (Figure 6.4.2-1) are utilized. Figure 6.4.2-2 summarizes the aerocapture heating environment for this encounter condition.



- EARTH CAPTURE
  - ENTRY C3=16 KM^2/SEC^2
  - AERO EXIT APOGEE = 38485 NM
  - BASE W/CdA = 20. LB/FT^2
- AEROASSIST CONTROL CORRIDOR
  - WIDTH = DELTA OF PERIGEES
  - ERROR ANALYSIS SETS REQMT
  - CONTROL CORRIDOR SETS L/D
- PEAK DECELERATION
  - SETS STRUCTURAL REQMTS

Figure 6.4.2-1 Earth Capture, C3=16 - Control & Loads



- MARS CAPTURE
  - ENTRY C3 = 16 KM^2/SEC^2
  - AERO EXIT APOGEE = 38485 NM
  - BASE L/D = 0.20
- PEAK STAGNATION HEATING
  - SETS TPS MATERIAL REQMTS
- INTEGRATED HEATING
  - SETS TPS THICKNESS

Figure 6.4.2-2 Earth Capture, C3=16 - Heating

6.4.3 EARTH AEROCAPTURE, C3=32

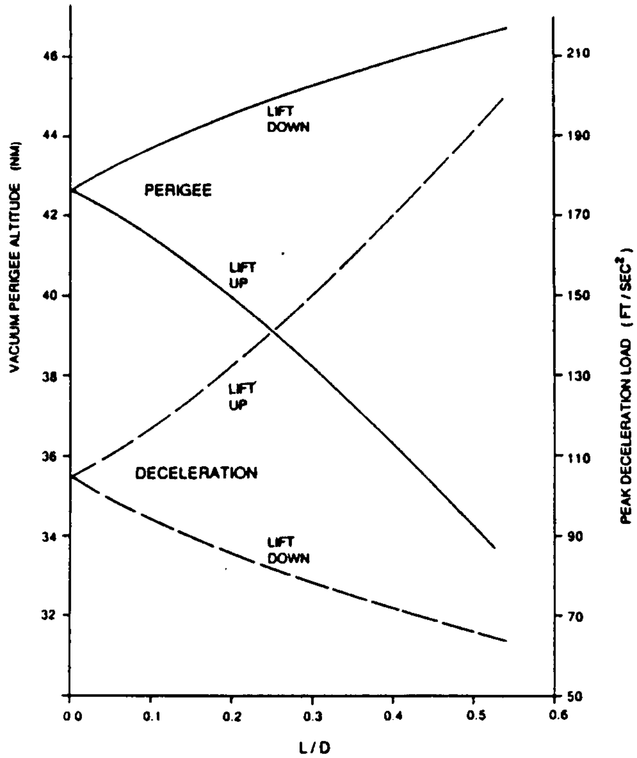
Table 6.4.3-1 summarizes the error analysis conducted for an Earth capture with an encounter C_3 of 32 km^2/sec^2. The primary difference between this analysis and that conducted for the 8.0 km^2/sec^2 Earth capture is in the dispersion sensitivity of the faster incoming trajectory. The other dispersions are the same because of a common Earth environment for entry.

Table 6.4.3-1 Earth Capture Aero-entry Error Analysis: C3=32 km^2/sec^2

	EQUIVALENT PERIGEE ERROR	
• TARGETING ERRORS (FINAL CORRECTION BURN AT ENTRY MINUS 1 HR)		
- POINTING ERROR	= 137 FT	± .1 DEG
- CUTOFF ERROR	= 1288 FT	.33 FPS ACCELEROMETER
- NAV ERROR	= 1024 FT	FROM 1020 FT POSITION UNCERTAINTY
	390 FT	FROM 0.1 FPS VELOCITY UNCERTAINTY
• AERODYNAMIC VARIATION		
- ATMOSPHERIC UNCERTAINTY	= 6000 FT	± 30% DENSITY
- L/D UNCERTAINTY	= 4900 FT	± 2° AT 9° ANGLE OF ATTACK (± 22% L/D)
- BALLISTIC UNCERTAINTY	= 1600 FT	± 8% W/C_DA
• RSS	= ± 1700 FT	= ± 0.28 NM FROM TARGETING
	= ± 7900 FT	= ± 1.30 NM FROM AERODYNAMICS
	= ± 8100 FT	= ± 1.33 NM NET VARIATION

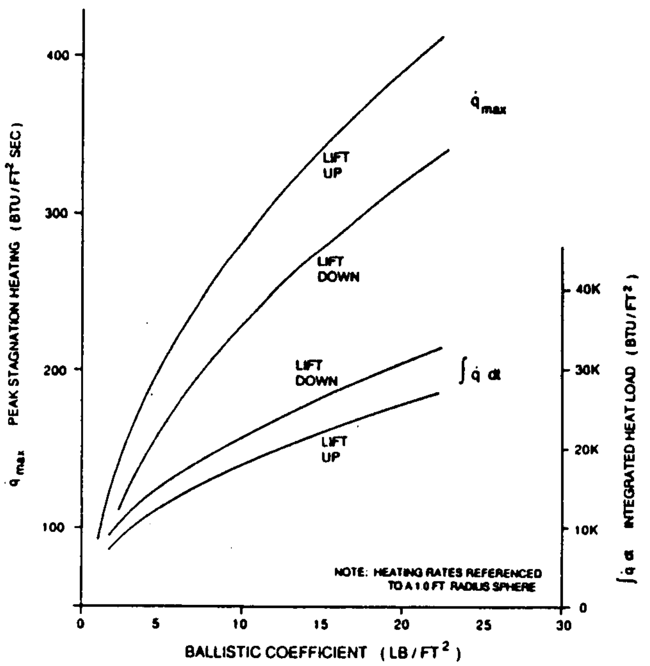
CONCLUSION: 3.54 N.M. CONTROL CORRIDOR REQUIRED TO COVER ERRORS WITH 33% MARGIN

The 3.54 nm net control corridor size sets a minimum L/D requirement of 0.155 for the entry vehicle when control parametrics (Figure 6.4.3-1) are utilized. Figure 6.4.3-2 summarizes the aerocapture heating environment for this encounter condition.



- EARTH CAPTURE
  - ENTRY C3=32 KM^2/SEC^2
  - AERO EXIT APOGEE = 38485 NM
  - BASE W/CdA = 10. LB/FT^2
- AEROASSIST CONTROL CORRIDOR
  - WIDTH = DELTA OF PERIGEES
  - ERROR ANALYSIS SETS REQMT
  - CONTROL CORRIDOR SETS L/D
- PEAK DECELERATION
  - SETS STRUCTURAL REQMTS

Figure 6.4.3-1 Earth Capture, C3=32 - Control & Loads



- MARS CAPTURE
  - ENTRY C3 = 32 KM^2/SEC^2
  - AERO EXIT APOGEE = 38485 NM
  - BASE L/D = 0.20
- PEAK STAGNATION HEATING
  - SETS TPS MATERIAL REQMTS
- INTEGRATED HEATING
  - SETS TPS THICKNESS

Figure 6.4.3-2 Earth Capture, C3=32 - Heating

6.4.4 EARTH AEROCAPTURE, C3=68

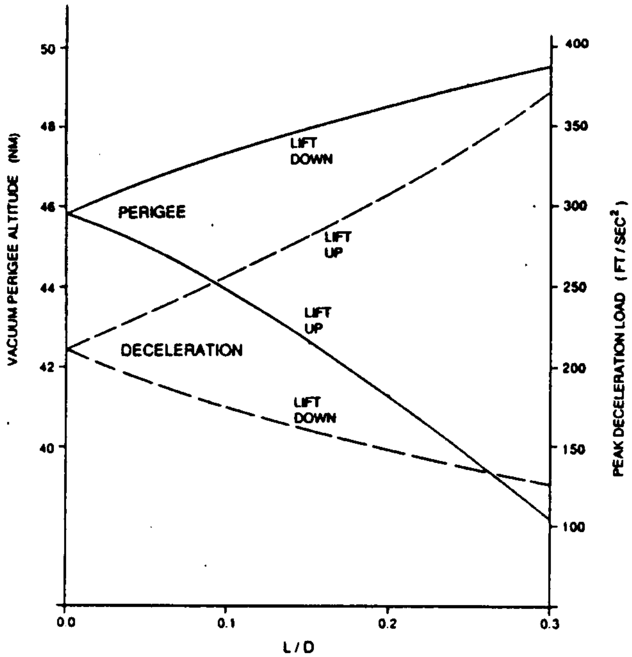
Table 6.4.4-1 summarizes the error analysis conducted for an Earth capture with an encounter C_3 of 68 km^2/sec^2. The primary difference between this analysis and that conducted for the 8.0 km^2/sec^2 Earth capture is in the dispersion sensitivity of the faster incoming trajectory. The other dispersions are the same because of a common Earth environment for entry.

Table 6.4.4-1 Earth Capture Aero-entry Error Analysis: C3=68 km^2/sec^2

	EQUIVALENT PERIGEE ERROR	
• TARGETING ERRORS (FINAL CORRECTION BURN AT ENTRY MINUS 1 HR)		
- POINTING ERROR	= 134 FT	± .1 DEG
- CUTOFF ERROR	= 1266 FT	.33 FPS ACCELEROMETER
- NAV ERROR	= 1026 FT	FROM 1020 FT POSITION UNCERTAINTY
	384 FT	FROM 0.1 FPS VELOCITY UNCERTAINTY
• AERODYNAMIC VARIATION		
- ATMOSPHERIC UNCERTAINTY	= 6300 FT	± 30% DENSITY
- L/D UNCERTAINTY	= 7300 FT	± 2° AT 9° ANGLE OF ATTACK (± 22% L/D)
- BALLISTIC UNCERTAINTY	= 1700 FT	± 8% W/C_DA
• RSS	= ± 1700 FT	= ± 0.28 NM FROM TARGETING
	= ± 9800 FT	= ± 1.61 NM FROM AERODYNAMICS
	= ± 9900 FT	= ± 1.64 NM NET VARIATION

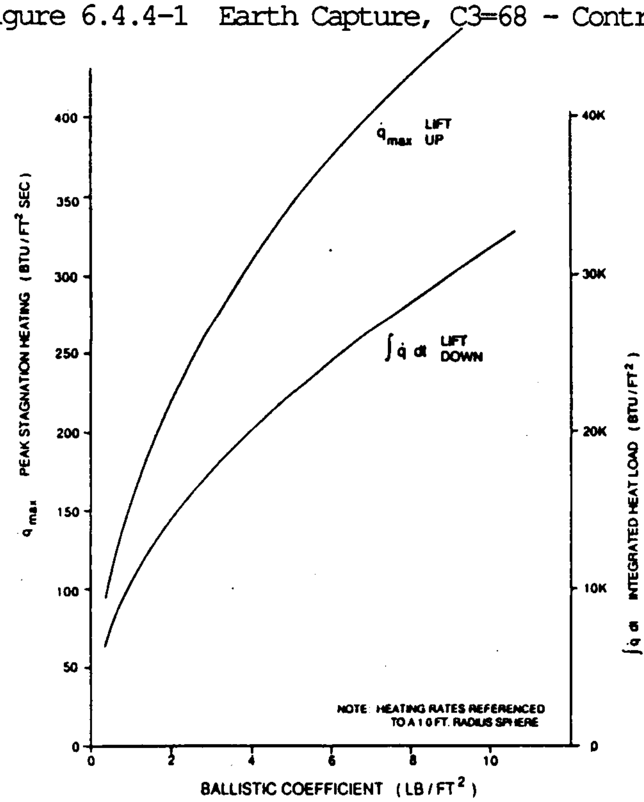
CONCLUSION: 4.35 N.M. CONTROL CORRIDOR REQUIRED TO COVER ERRORS WITH 33% MARGIN

The 4.35 nm net control corridor size sets a minimum L/D requirement of 0.13 for the entry vehicle when control parametrics (Figure 6.4.4-1) are utilized. Figure 6.4.4-2 summarizes the aerocapture heating environment for this encounter condition.



- EARTH CAPTURE
  - ENTRY C3=68 KM^2/SEC^2
  - AERO EXIT APOGEE = 38485 NM
  - BASE W/CdA = 2.0 LB/FT^2
- AEROASSIST CONTROL CORRIDOR
  - WIDTH = DELTA OF PERIGEES
  - ERROR ANALYSIS SETS REQMT
  - CONTROL CORRIDOR SETS L/D
- PEAK DECELERATION
  - SETS STRUCTURAL REQMTS

Figure 6.4.4-1 Earth Capture, C3=68 - Control & Loads



- MARS CAPTURE
  - ENTRY C3 = 68 KM^2/SEC^2
  - AERO EXIT APOGEE = 38485 NM
  - BASE L/D = 0.15
- PEAK STAGNATION HEATING
  - SETS TPS MATERIAL REQMTS
- INTEGRATED HEATING
  - SETS TPS THICKNESS

Figure 6.4.4-2 Earth Capture, C3=68 - Heating

6.5 AEROASSIST CONTROL SUMMARY

The amount of velocity reduction accomplished in an aeroassist has a direct impact on the amount of lift control available. Since the lift force is a function of the drag force for a fixed L/D, a larger aero-deceleration (drag directed) results in a larger cross component of lift. This is illustrated in Figure 6.5-1 which plots control corridor magnitudes for given L/D values vs aeroassist velocity reduction for the Earth return case.

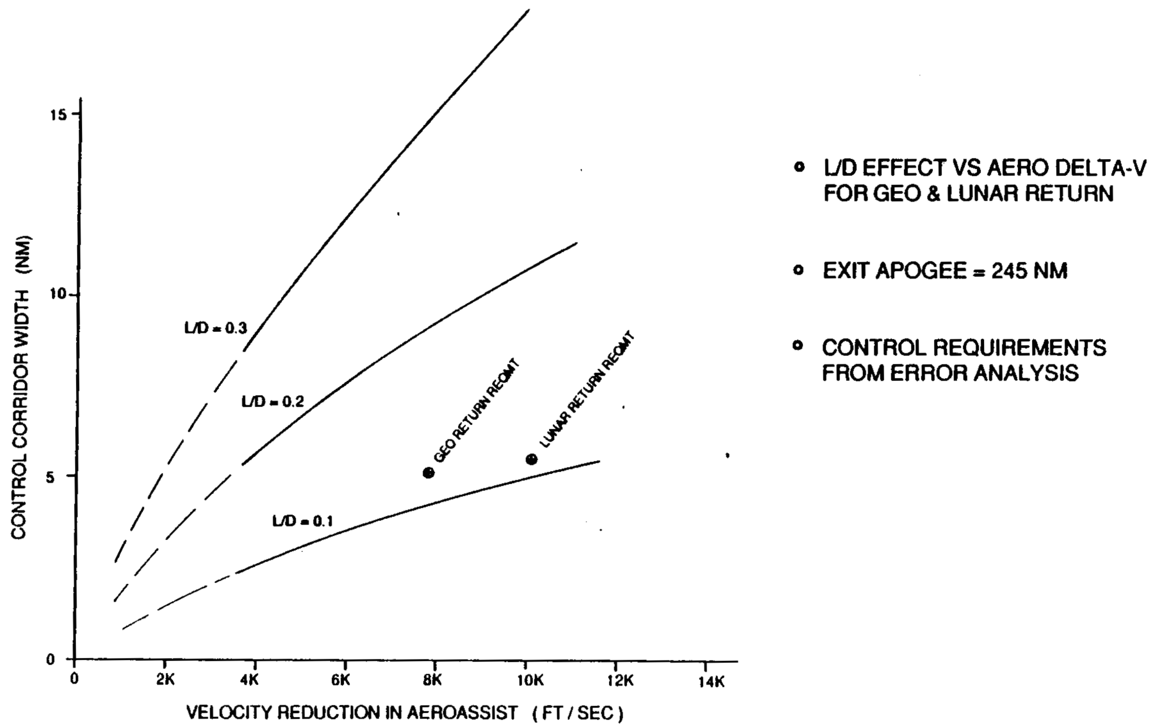


Figure 6.5-1 Control vs Aero Delta-V: Earth Return

The higher the aero ΔV the larger the control corridor (i.e. the larger the amount of trajectory control available). Figure 6.5-1 shows these trends for Earth return type missions, that is those which return to a low Earth park orbit (245 nm). The two missions for which error analysis (sizing the control corridor) have been conducted are indicated: GEO return and lunar return. These points indicate that while the control corridor requirements are growing for higher energy missions, the control capability from a given L/D grows at a faster rate. Thus the required L/D declines with increasingly energetic aeroassists.

Figure 6.5-2 summarizes the growth in control corridor capability for Earth capture missions (those which capture an incoming vehicle into a 245 x 38485 nm park orbit). As with the previous graph for the Earth return case, control capability grows steadily with increased aeroassist ΔV. Also shown are the control corridor requirements for the four capture conditions analysed (C_3= 8, 16, 32, and 68 km^2/sec^2). Again, as with the Earth return case the growth in control requirements with increasingly energetic missions is outstripped by the growth in control capability resulting in a net decrease in L/D requirements.

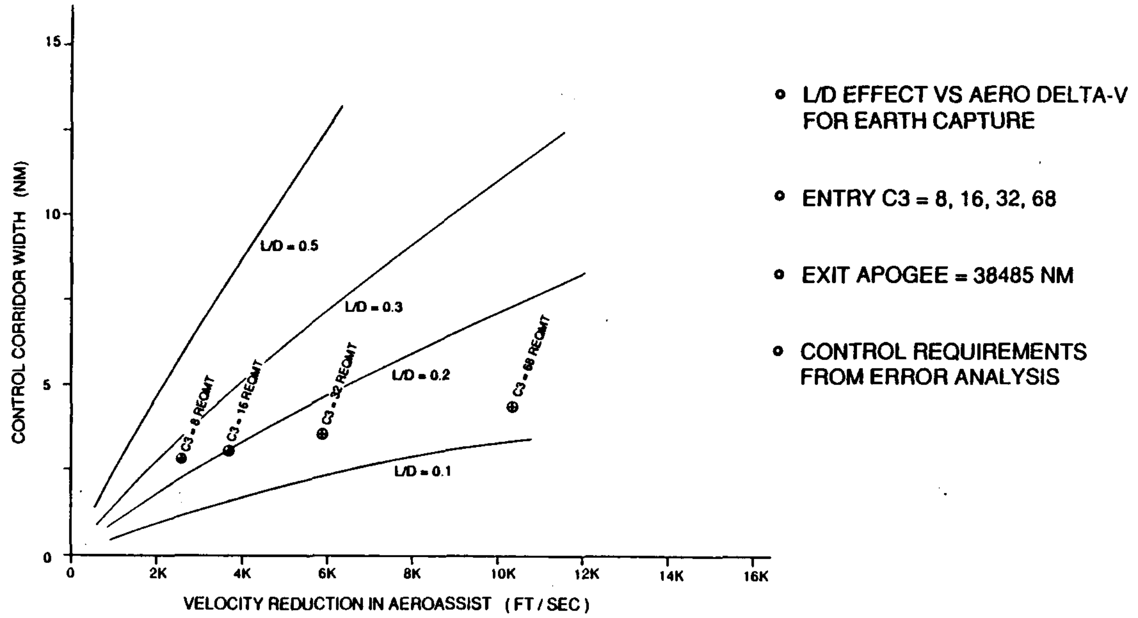


Figure 6.5-2 Control vs Aero Delta-V: Earth Capture

Figure 6.5-3 summarizes the growth in control corridor capability for Mars capture missions (those which capture an incoming vehicle into a 270 x 18108 nm park orbit). As before, control capability grows steadily with increased aeroassist ΔV. Also shown are the control corridor requirements for the four capture conditions analysed (C_3= 8.2, 13, 31, and 60 km^2/sec^2). As before, the growth in control requirements with increasingly energetic missions is outstripped by the growth in control capability resulting in a net decrease in L/D requirements.

6.5.1 MINIMUM L/D REQUIREMENTS FOR AEROASSIST

Figure 6.5.1-1 shows the decreasing L/D requirements for increasingly energetic aeroassist maneuvers covering the three different entry missions. As the three previous figures have shown, the growth in control capability is faster than the growth in control requirements for larger aeroassist ΔV's. All three aeroassist mission types are shown on this graph: Earth return, Earth capture, and Mars

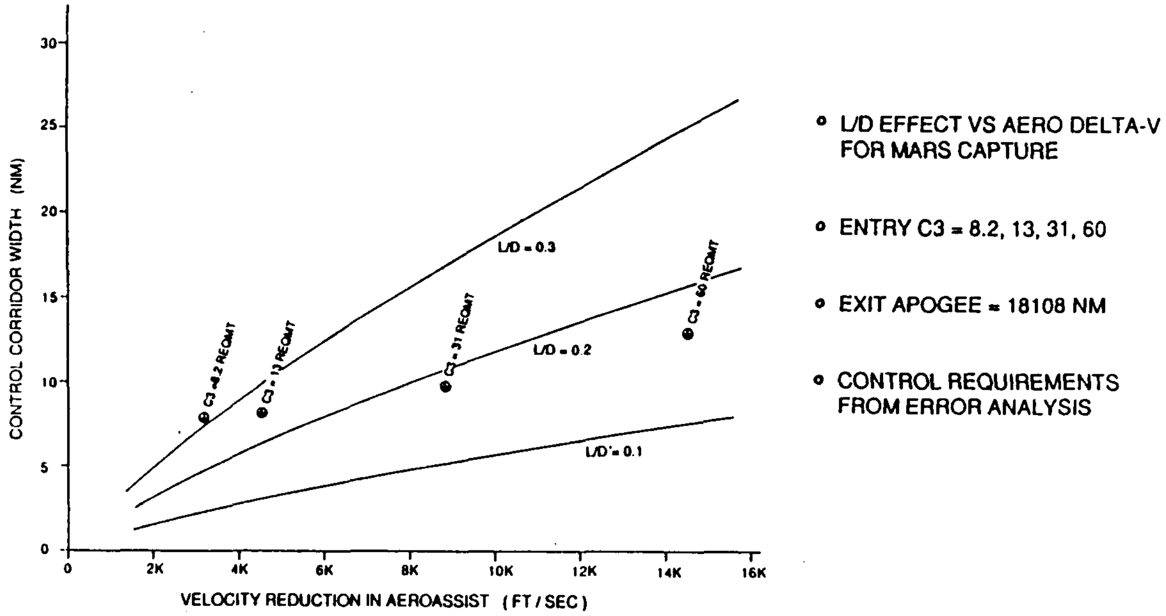


Figure 6.5-3 Control vs Aero Delta-V: Mars Capture

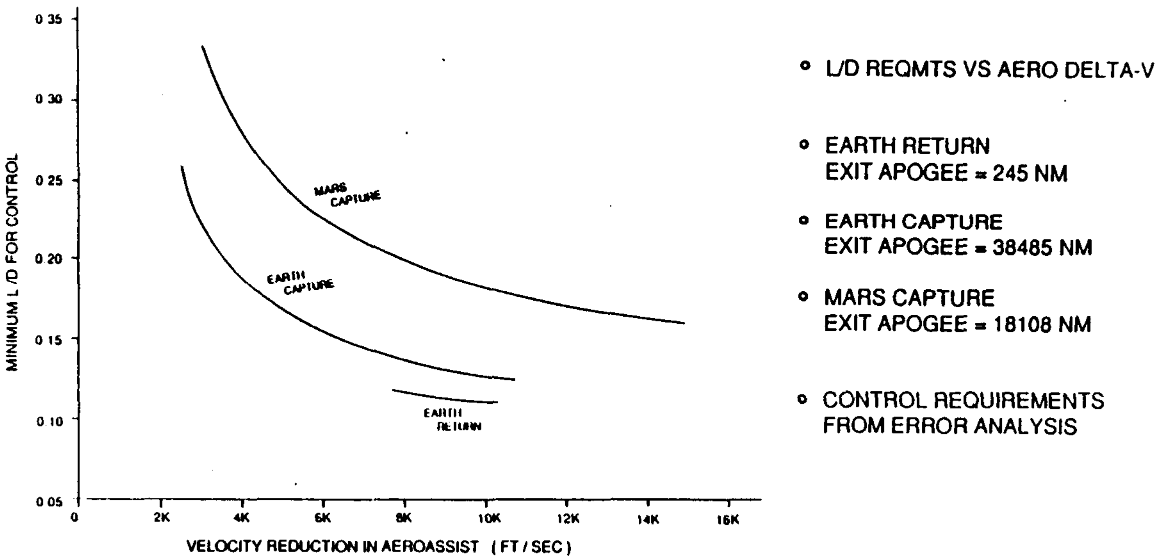


Figure 6.5.1-1 Minimum L/D Requirements For Aeroassist

capture. Each of the mission classes shows the same trends with vertical offsets due to dynamic rate differences in the aeroassist processes. From this data one can see that it is the less energetic entries that will be the most difficult to control. Fortunately, these are also the type of velocity reduction maneuvers that are more efficiently conducted propulsively.

6.6 LUNAR AEROBRAKE DESIGN

In order to more accurately characterize the performance of the OTV lunar logistics vehicle a design was undertaken of the lunar aerobrake. This design is based on lunar return parametrics, presented earlier, which are consistent with return from low lunar orbit on a "free-return" type trajectory. Direct landing and L1 libration point returns will not vary significantly from these results.

6.6.1 LUNAR LOAD RELIEF

After performing the lunar aeroentry error analysis and comparing it against the applicable control parametrics it was found that an L/D of 0.11 was required to maintain acceptable control margins. Unfortunately, this L/D level also results in significantly higher levels of peak deceleration than are encountered in typical GEO returns (4.8 g v.s. 3.5 g). Since an implicit goal is to produce the lunar logistics vehicle by a minimum number of modifications to the baseline space based OTV, alternative aeroassist approaches were investigated for load relief.

By analysing the load profile for a nominal GEO vehicle when flown through a lunar return (next section), it was found that the lower 25% of the control corridor contains a steeply rising peak load. Trajectories in this region dive steeply into the atmosphere and, through the use of a predominantly lift up condition, exit steeply out. Such an entry will go deeper and encounter a faster onset of aero-loads than do entries which occur higher in the corridor. By removing this lower 25% of the corridor these higher load levels can be eliminated. Since the basic control corridor requirement remains it is necessary to expand the control capability such that when 25% of it is eliminated, the remaining piece still spans the requirement.

When this control corridor expansion was performed it resulted in a new corridor requirement of 7.3 nm which equates to a new L/D of 0.14. When this higher L/D is used in lunar entries the load profile shown in Fig. 6.6.2-2 results. By flying in the upper 5.5 nm of the corridor (the requirement from error analysis), peak loads of 4.0 g's result. These loads result in substantially lesser OTV core structure modifications of only 64 lb. This technique does result in higher aerobrake weights due to higher integrated heating. The overall vehicle weighs slightly more, consistent with results presented in the first extension of this study. Since the aerobrake would have to be redesigned anyway for lunar return the amount of vehicle redesign is minimized by keeping the core relatively unchanged.

6.6.2 LUNAR RETURN LOADS, L/D = 0.12 & 0.14

Figure 6.6.2-1 shows the peak load profile spanning the control corridor for a vehicle returning from the moon with an L/D of 0.12 (baseline GEO return vehicle value) to a Space Station pickup orbit at an altitude of 245 nm.

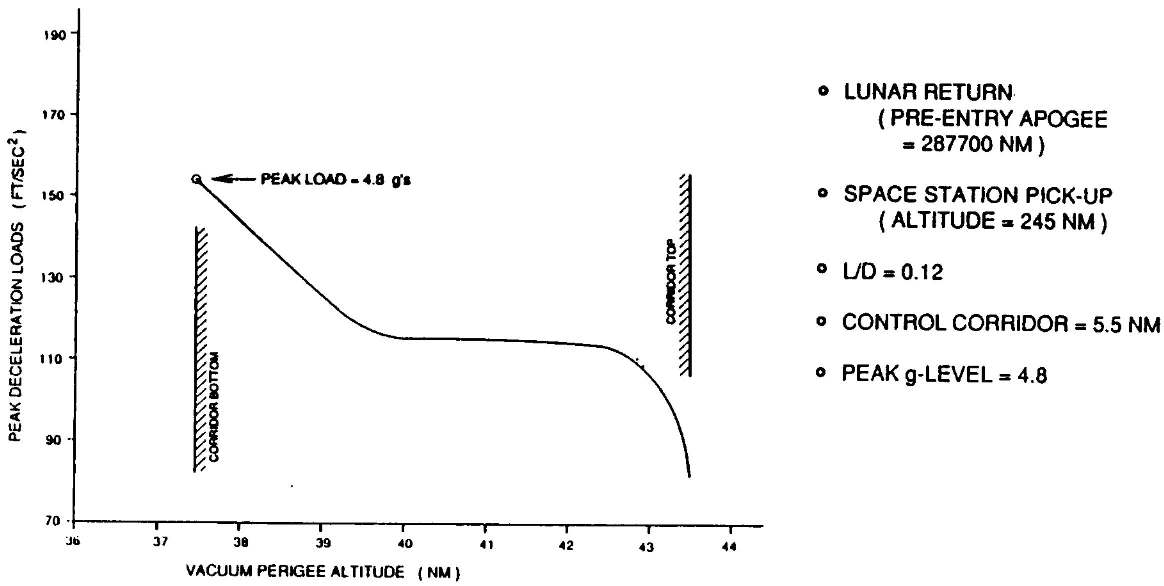
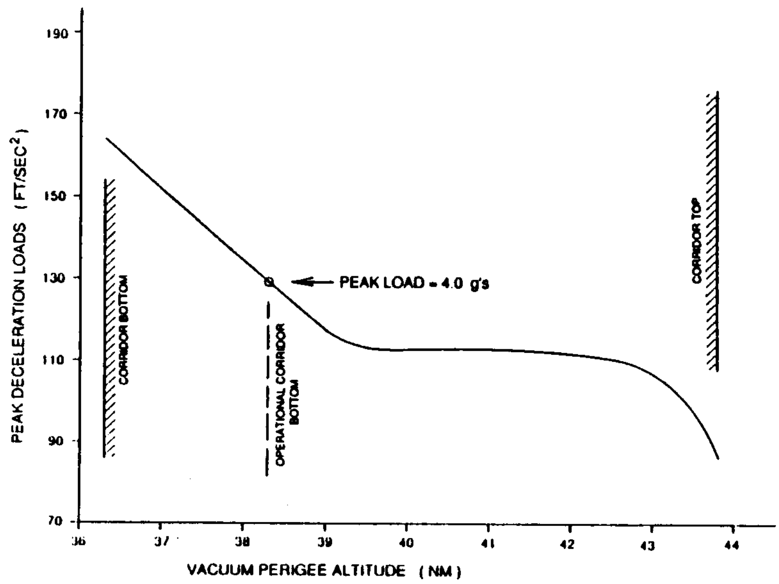


Figure 6.6.2-1 Lunar Loads, L/D = 0.12

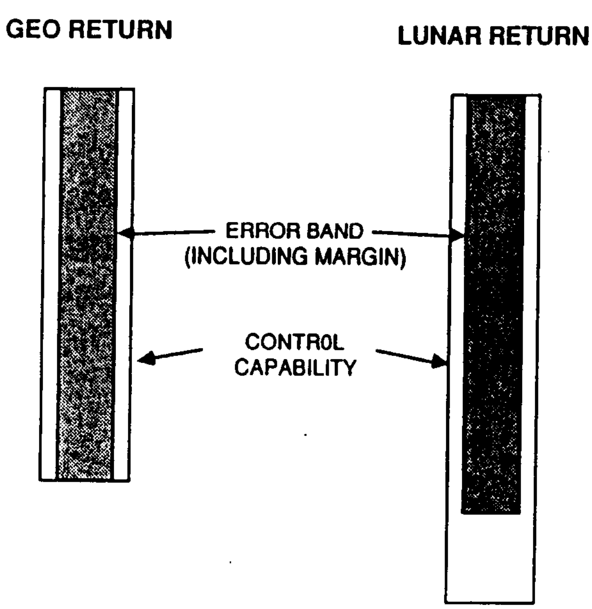
Figure 6.6.2-2 shows the peak load profile spanning the control corridor for a vehicle returning from the moon with an L/D of 0.14 to a Space Station pickup orbit at an altitude of 245 nm. By restricting flight to the upper 5.5 nm of the corridor, peak loads are reduced to 4.0 g's.

These two figures illustrate the amount of load relief achievable with fairly modest increases in L/D. Figure 6.6.2-3 summarizes the basic principal of using excess control for load relief. The minimum control requirement is derived from error analysis and is about the same for both GEO and lunar returns. By oversizing the control capability in the lunar case the upper portion of the corridor can be used as the operating flight envelope since it has more benign vehicle loading.



- LUNAR RETURN
  (PRE-ENTRY APOGEE
  = 287700 NM)
- SPACE STATION PICK-UP
  (ALTITUDE = 245 NM)
- L/D = 0.14
- CONTROL CORRIDOR = 7.3 NM
- PEAK g-LEVEL = 4.0

Figure 6.6.2-2 Lunar Loads, G-Relief: L/D = 0.14



GEO RETURN

ACCEPTABLE AERO LOADS

CENTERING ERROR BAND IN
CONTROL CAPABILITY
GIVES WEIGHT OPTIMUM
AEROBRAKE

LUNAR RETURN

OVERSIZE CONTROL
CAPABILITY

BIAS ERROR BAND
TO TOP OF CONTROL
BAND FOR LOAD RELIEF

GIVES ACCEPTABLE LOADS

Figure 6.6.2-3 Lunar Aero Load Relief

6.6.3 LUNAR AEROBRAKE CHARACTERISTICS

Table 6.6.3-1 summarizes the important features of the lunar aerobrake. The heaviest return payload was used which is the 15000 lb crew module. Load relief, discussed previously, was used to reduce the peak deceleration loads to 4.0 g's.

Table 6.6.3-1 Lunar Aerobrake Characteristics

	LUNAR BRAKE	GEO BRAKE
DIAMETER, FT	45.2	44.0
W / CDA, LB/FT^2	10.8	8.0
L / D	0.14	0.12
ANGLE OF ATTACK, DEG	8.83°	7.23°
PEAK g-LOAD	4.0 g	3.5 g
TPS AREA, FT^2		
RSI	149	149
FSI	1641	1553
PEAK STAGN. HEAT, BTU/FT^2-SEC	36.9	26.4
TOTAL HEAT LOAD, BTU/FT^2-SEC	4802	3805
TPS THICKNESS, INCH		
RSI	0.92	0.77
FSI	0.52	0.45

Because the angle of attack is somewhat higher than for the GEO return case, the aerobrake diameter must be increased to compensate for the increased impingement angle. This results in the brake being 45.2 ft. in diameter. The hard shell center core portion of the brake is the same size as the GEO brake, with the outer flex fabric annulus being increased in size for the larger diameter. The peak stagnation heating is significantly higher than for the GEO brake but the flux is still within the capabilities of both the rigid and flexible surface insulation (RSI & FSI). The increase in TPS thickness to protect against the higher heat loads is shown.

Table 6.6.3-2 summarizes the basic subsystem weights for the lunar and GEO return aerobrakes used on the space based OTV. The lunar brake weight was then used in performance assessments of OTV lunar logistics.

Table 6.6.3-2 Lunar Aerobrake Weights

	LUNAR BRAKE	GEO BRAKE
OTV CORE - Δ STRUCTURE CHANGES	+64	-
TPS WEIGHTS		
RSI	160	147
FSI	1092	894
AEROBRAKE STRUCTURE		
RSI HONEYCOMB SUBSTRATE	78	73
INTERFACE RING	264	217
RADIAL BEAMS (12)	152	120
SUPPORT STRUTS	283	220
DOORS & ATTACH HARDWARE	270	169
STRUCTURE TOTAL	1047	799
TOTAL AEROBRAKE WEIGHT	2299	1840

ALL WEIGHTS IN POUNDS

The core of the OTV increases by 64 lb over the basic GEO return vehicle due to the higher aerodynamic loads encountered in lunar return. TPS weights increase because of higher heating but also because of the larger diameter of this aerobrake. The increased peak loads scale up the supporting stucture of the brake. In the case of the radial beams and support struts the increased brake diameter also contributes to higher weights. Finally an allocation of 100 lb was made for the more complex door mechanisms required to protect the 4-engine landing cluster. Overall, the lunar aerobrake weighs 2299 lb for an increase of 458 lb over the GEO return brake weight.

APPENDICES - PRELIMINARY OTV SAFETY ANALYSIS

The following two appendices contain the results of a preliminary safety analysis conducted for the ACC OTV system. Appendix A contains summary level OTV preliminary hazard analysis charts. These charts address safety related issues for the ACC OTV mission phases: Prelaunch, STS Ascent, OTV/STS Rendezvous Operations, OTV Mission (payload delivery and return), STS/OTV Descent, and Postlanding Activities. Appendix B contains the safety requirements derived to date from the preliminary safety analysis. These requirements are generally allocations of top level safety requirements to the various subsystems. These derived requirements will be used as inputs for developing design and operations specifications and concepts. The attached requirements should be used in addition to the existing design, integration, and operations requirements developed for the OTV.

Following each requirement is a referenced Preliminary Hazard Analysis Number (PHA). The PHA number refers to the hazard analysis that was the basis for the requirement derivation. The preliminary safety analysis included the research and evaluation of work accomplished during the initial Phase A period of this contract. It also included an analysis of the work performed by NASA and the advanced program study contractors supporting Aft Cargo Carrier (ACC), Orbital Transfer Vehicle (OTV), and the STS Cargo Bay integration of Upper Stage Vehicle Analyses. While the current safety analysis is not complete, it does provide a base line for: (1) Future analyses, (2) Criteria for preliminary hardware and software designs, (3) Systems and vehicle integration, and (4) The development of operations concepts, plans and reference missions.

APPENDIX A

ACC OTV
PRELIMINARY
HAZARDS ASSESSMENT

OTV PRELIMINARY HAZARD ANALYSIS

PHA No.: ACC-1-1

MISSION PHASE: I - Prelaunch

SUBSYSTEM OR OPERATION: OTV Pad Operations

EFFECTIVITY: Mating the OTV to the ACC/ET

ENGINEER: A.W. Bowman

DATE: 24 April 1987

SHEET 1 OF 1

HAZARDOUS CONDITION	HAZARD CAUSE	HAZARD EFFECT	HAZARD LEVEL	SAFETY REQUIREMENTS	HAZARD CONTROL
1. Fuel Leakage From the OTV While On the Pad	1. Over Pressure of the Fuel Tanks or Lines 2. Damage to Fuel Tanks or Associated Hardware	1.Contamination of : (a) Pad, (b) Orbiter, (c) Personnel, (d) Launch Support Equipment 2. Loss of Flight 3. Loss of OTV 4. Loss of Mission 5. Loss of Orbiter	1. Catastrophic 2. Controlled 3. Critical 4. Critical 5. Catastrophic	1. Launch Support Hardware, Software & Procedures Must Ensure Protection of Hazardous Material From Inadvertent Damage 2. Maintain a System to Capture Spilled Fuels and Associated Gases 3. Control the Number of Materials That Would React With Hazardous Fluids and Gases. 4. Provide Thermal Control	1. Develop Procedures and Methods For Safe Handling of Hazardous Liquids and Gases at the Pad (Small Volume) 2. Verify all S/W For Safe Control of Fluids and Gases 3. Verify all Control and Handling Procedures and Methods Prior to Use.
2. Damage To OTV/ACC During On-Pad Mating Operations	1. Improper Handling of The OTV During Transporation And/or Mating to The ACC	1. OTV Unable to Deploy From ACC. 2. Loss of OTV During Assent (Premature Deployment) 3. Damage to ACC/ET 4. Damage to Launch Pad And/or Launch Support Equipment or Personnel	1. Critical 2. Catastrophic 3. Catastrophic 4. Catastrophic	1. Handholds/Fixtures marked. 2. C.G. And Mating Locations/Interfaces Marked And Accessible 3. Transportation and Mating Procedures Developed and Tested 4. Positive Control Over All OTV Pad Operations And Movement Must Be Maintained 5. No Materials Shall Be Used That Produce Arcing or Sparking	1. Train Ground Crew Prior to Handling OTV 2. Verify All OTV Ground Operations Procedures 3. Validate All Ground Hardware/Software Prior to Use.

OTVHAZ-3

PHA No.: ACC-2-1

OTV PRELIMINARY HAZARD ANALYSIS

MISSION PHASE: II - Orbiter Ascent

SUBSYSTEM OR OPERATION: ACC/OTV Deployment From ET

EFFECTIVITY: OTV Rendezvous Operations with the STS

ENGINEER: A. W. Bowman

DATE: 24 April 1987

SHEET 1 OF 1

HAZARDOUS CONDITION	HAZARD CAUSE	HAZARD EFFECT	HAZARD LEVEL	SAFETY REQUIREMENTS	HAZARD CONTROL
1. Premature Separation of ACC From ET (OTV From ACC)	1. Cmd Error from the Gnd or On-board Event Sequencer 2. Premature Pyrotechnic Detonation of Explosive Devices 3. Malfunction in ET/ACC Separation Control System	1. Loss of OTV 2. Unable to Deploy STS P/L 3. Damage to STS 4. Damage to ET 5. Premature Separtion of OTV From ET (Bad Orbit)	1. Catastrophic 2. Controlled 3. Catastrophic 4. Catastrophic 5. Critical	1. Fail Safe Command, Control and Verification 2. Pyro System Grounding and Radiation Sheilding 3. Design All Circiuts for Fail-Safe- Redundant Operations 4. Develop Mission Abort Procedures 5. ACC/OTV/ET/STS ICD'S: Development & Validation By Test 6. Orbiter Caution & Warning System Shall Monitor All OTV Status	1. Review & Test Design & Sim Proc's. 2. Pyro & Assoc Elec Cir shall Conform to STS Spec JSC 08060 3. Review all Designs 4. Review and Sim Procedures 5. Review/Update ICD's
2. Premature Seperation of OTV From ACC	1. Same as Above 2. Shroud Deployment Malfunction 3. Malfunction in OTV Deployment Control System	1. Same as Above 2. Loss of OTV Mission	1. Catastrophic 2. Controlled	1. OTV Fuel Tanks Must Not Survive Re-entry 2. OTV/ACC Must Maintain The Smallest Re-entry Footprint Possible	Destroy ACC/OTV Prior to Re-entry

OTVHAZ-1

PHA No.: ACC-3-1

OTV PRELIMINARY HAZARD ANALYSIS

MISSION PHASE: III - Low Earth Orbit Operations

SUBSYSTEM OR OPERATION: OTV Manuvering And Boost Orbit Ops

EFFECTIVITY: OTV/Orbiter Rendezvous Preparation

ENGINEER: A.W. Bowman

DATE: 24 April 1987

SHEET 1 OF 1

HAZARDOUS CONDITION	HAZARD CAUSE	HAZARD EFFECT	HAZARD LEVEL	SAFETY REQUIREMENTS	HAZARD CONTROL
1. No OTV Safety Issues Identified For This Phase					

OTVHAZ-4

PHA No.: ACC-4/6-1

OTV PRELIMINARY HAZARD ANALYSIS

MISSION PHASE: IV and VI : OTV/STS Rendezvous Operations 1 & 2

SUBSYSTEM OR OPERATION: OTV/Orbiter Proximity Operations

EFFECTIVITY: On-Orbit Operations : OTV Approach to STS

ENGINEER: A.W. Bowman

DATE: 24 April 1987

SHEET 1 OF 2

HAZARDOUS CONDITION	HAZARD CAUSE	HAZARD EFFECT	HAZARD LEVEL	SAFETY REQUIREMENTS	HAZARD CONTROL
1. Orbiter Contamination From OTV Fuel	1. Inadvertent OTV ACS/MPS Motor Firing 2. Fuel Venting or Dumping 3. Fuel Leak (Tanks, Lines or Valves)	1. Damage to Orbiter Skin/Tile 2. Damage to Orbiter Cargo Bay Doors 3. Chemical Reaction with Cargo Bay Contents/Sys 4. Contamination of Crew EVA Systems 5. Contamination of Ground Crews	1. Catastrophic 2. Catastrophic 3. Catastrophic 4. Critical 5. Critical	1. Fail-Safe Fuel Systems 2. Continuous OTV CMD & Control Capabilities 3. OTV Function Override capabilities 4. Provide Positive Thermal/Pressure Monitor & Control	1. Operations Abort Procedures 2. Continuous Monitor and Control of All On-board OTV Operations and Functional Status
2. OTV/Orbiter Collision	1. Inadvertent ACS Firing 2. Inadvertent MPS Firing 3. Malfunction of OTV ACS	1. Loss or Damage to STS, OTV or P/L 2. Loss of STS Habitable Area 3. Loss of Mission	1. Catastrophic 2. Catastrophic 3. Critical	1. Safe all OTV Systems Prior to STS Rendezvous 2. Maintain Continuous Monitor & Control of OTV From Both the STS and Ground Control Center(s) 3. The Crew Shall Be Trained In All Phases of OTV Operations	1. Perform OTV Venting Prior to STS Rendezvous 2. Maintain 100 % C&T Coverage by Ground Control Center(s) During All Proximity Operations 3. Orbiter Crew Shall Maintain Positive Control of the OTV at All Times

OTVHAZ-2

OTV PRELIMINARY HAZARD ANALYSIS

PHA No.: ACC-4/6-2

MISSION PHASE: IV and VI: OTV/STS Rendezvous Operations 1 & 2

ENGINEER: A.W. Bowman

SUBSYSTEM OR OPERATION: OTV/Orbiter Proximity Operations

DATE: 24 April 1987

EFFECTIVITY: On-Orbit Operations: OTV Approach/Departure at Orbiter

SHEET 2 OF 2

HAZARDOUS CONDITION	HAZARD CAUSE	HAZARD EFFECT	HAZARD LEVEL	SAFETY REQUIREMENTS	HAZARD CONTROL
3. Orbiter/OTV Docking & Undocking	1. Docking Mechanism Jammed 2. Docking Mechanism Malfunction 3. STS/RMS Stressed	1. Unable to Dock OTV 2. Unable to Release OTV 3. Unable to Mate P/L to OTV 4. Damage to (a) OTV, (b) P/L, (c) Orbiter 5. Unable to Control OTV or Payload	1. Critical 2. Catastrophic 3. Critical 4. Catastrophic 5. Catastrophic	1. Docking Mechanisms Shall be: (a) Jam Resistant (b) Engagement Locks Shall Have Multiple Load Paths 2. Emergency Unlock & Disconnect Capability 3. Use Two STS/RMS Mechanisms And/Or Other Mechnical Aids	1. H/W & S/W Shall be Validated Prior To Use. 2. Orbiter Crew Training Performed Prior To On-Orbit Operations 3. Reduce EVA Rqmt's

OTVHAZ-2b

OTV PRELIMINARY HAZARD ANALYSIS

PHA No.: ACC-5-1

MISSION PHASE: V - OTV Mission Operations

SUBSYSTEM OR OPERATION: High Earth Orbit And/Or Planetary Operations

EFFECTIVITY: OTV/Payload Operations

ENGINEER: A. W. Bowman

DATE: 24 April 1987

SHEET 1 OF 1

HAZARDOUS CONDITION	HAZARD CAUSE	HAZARD EFFECT	HAZARD LEVEL	SAFETY REQUIREMENTS	HAZARD CONTROL
1. No OTV Safety Issues Identified For This Phase					

OTVHAZ-5

OTV PRELIMINARY HAZARD ANALYSIS

PHA No.: ACC-7-1

MISSION PHASE: VII - OTV/Orbiter Return (Re-entry) Operations

SUBSYSTEM OR OPERATION: Return Operations

EFFECTIVITY: Orbiter Cargo Bay Operations

ENGINEER: A. W. Bowman

DATE: 24 April 1987

SHEET 1 OF 1

HAZARDOUS CONDITION	HAZARD CAUSE	HAZARD EFFECT	HAZARD LEVEL	SAFETY REQUIREMENTS	HAZARD CONTROL
1. Contamination of Orbiter Cargo Bay	1. Hazardous Liquids And/or Gases Leaking From OTV Fuel System	1. Damage to Cargo Bay Contents 2. Personnel Exposure to Toxic Substance	1. Catastrophic 2. Catastrophic	1. OTV Fuel System Must Be Safed Prior to Mating or Placing in the Cargo Bay 2. All ACS and MPS Control Systems Shall Be Safed 3. OTV Power System Shall Be Safed 4. The OTV Shall Not Impact Other P/L Elements (Orbiter Resources or Other P/L Envelopes)	1. Fuel System Shall Be Purged Prior to Placing OTV in The Cargo Bay 2. All OTV Liquids And Gases Shall Be Expelled Prior to Entering Cargo Bay 3. The OTV Shall Be Powered Down And Inactive

OTVHAZ-6

OTV PRELIMINARY HAZARD ANALYSIS

PHA No.: ACC-8-1

MISSION PHASE: VIII - Postlanding Activities

SUBSYSTEM OR OPERATION: OTV/Orbiter Demate & Turnaround Activities

EFFECTIVITY: OTV Refurbishment

ENGINEER: A.W. Bowman

DATE: 24 April 1987

SHEET 1 OF 1

HAZARDOUS CONDITION	HAZARD CAUSE	HAZARD EFFECT	HAZARD LEVEL	SAFETY REQUIREMENTS	HAZARD CONTROL
1. No OTV Safety Issues Identified For This Phase (See STS Safety Requirements)					

OTVHAZ-7

APPENDIX B

STS & ACC OTV DERIVED SAFETY REQUIREMENTS

DERIVED REQUIREMENTS FOR THE ACC OTV

(REQUIREMENTS ARE BASED ON PRELIMINARY
OTV HAZARD ANALYSIS, PHA #'S INDICATED)

	REQUIREMENTS *	PHA No.
A	**STRUCTURES AND MATERIALS**	
1.	The Orbiter Caution and Warning (C&W) system shall include a rapid OTV depressurization warning device.	ACC-2-1
2.	The pressurization system shall be designed so that no two failures results in a catastrophic overpressurization of the OTV volume.	ACC-1-1
3.	The OTV pressurized volume shall be designed to be Fail Safe.	ACC-1-1
4.	No two mechnical, electrical, or operator errors shall result in a catastrophic loss of OTV volume pressurization.	ACC-1-1
5.	No two failures shall result in failed ON commands to valves in pressure systems.	ACC-4/6-1
6.	Thermal control systems shall provide a two failure tolerance against freezing of liquid lines.	ACC-4/6-1
7.	No two instrumentation failures shall result in OTV fuel tank or associated hardware over/under pressurization.	ACC-1-1
8.	All materials including seals, gaskets and lubricants used in flight equipment shall be compatible with system commodities.	ACC-1-1
9.	Pressure excursions caused by anomalous leakage rates shall be announced by the Orbiter C & W system.	ACC-4/6-1
10.	External structure shall be designed so that worst-case leakage shall not result in structural failure.	ACC-1-1
11.	Equipment which may be damaged during installation shall be equipped with suitable guards, cushions or other protective devices as appropriate.	ACC-1-1
12.	Equipment, systems, subsystems and fittings shall be designed to accommodate the sequence of their installation/attachment to a STS system or structure.	ACC-1-1

* Note: All safety requirements include both manned, man-tended, and unmanned operations applications. It is assumed that any OTV may be called upon to support a manned operation.

13. Sensitive switches/controls shall be physically protected from inadvertent activation by the use of guards, covers or other suitable means and shall be clearly marked, visible and remain accessible to the crew. ACC-4/6-1

14. Warning placards or labels shall be provided on all controls which are not to be operated during ground operations. ACC-1-1

15. Venting systems shall not vent incompatible substances at the same time and shall be purged, if necessary, prior to or after venting a known reactive substance. ACC-4/6-1

16. Venting systems shall be single failure tolerant against venting incompatible substances at the same time. ACC-4/6-1

B. MECHANISMS

1. No single mechanical failure shall result in the loss of the ability to vent. ACC-1-1

2. No single electrical component failure shall result in the loss of the ability to vent. ACC-1-1

3. Docking/payload mating mechanisms shall be jam resistant. ACC-4/6-2

4. The engagement locks on docking mechanisms shall incorporate multiple load paths to insure safe engagement with a TBD margin of safety with one broken locking member. ACC-4/6-2

5. A means shall be provided for emergency unlock and disconnect from a damaged docking mechanism. ACC-4/6-2

6. Redundant power distribution buses shall not be routed through the same connector. ACC-2-1

7. Systems shall be designed with overload protection. ACC-2-1

8. All S/W used to control hardware movement or initiate hardware action shall be verified through analysis and/or test prior to acceptance. ACC-1-1

9. TBD factors of safety shall be incorpoprated into design. Conservative factors of safety shall be provided where critical-failure point modes of operation cannot be eliminated. ACC-1-1

10 No single failure or operator error shall result in premature/inadvertent deployment of the OTV. ACC-2-1

11.	There shall be no devices that create arcing or sparking during normal operations. Devices capable of producing arcing or sparking shall be single failure tolerant against such occurrences. Notification of the failure conditions shall be provided to the crew.	ACC-1-1
12.	Devices capable of producing hot spots that exceed the temperatures specified in the OTV PDRD shall be single failure tolerant against such occurrences. Notification of the failure condition shall be provided to the crew.	ACC-1-1
13.	Fuse/wire compatibility requirements shall be applied to prohibit the possibility of wire segments exceeding critical temperatures from all possible shorts.	ACC-2-1
14.	Fault isolation techniques shall be used to prohibit the possibility of shorting unfused circuitry to the ground.	ACC-2-1
15.	Where the return of a circuit is switched, both the feed and the return must be switched at the same time.	ACC-2-1
16.	No two events or operator errors shall result in irreversible/ complete loss of power.	ACC-2-1
17.	Power consuming assemblies shall be protected from power surges on the main feed lines.	ACC-2-1
18.	All S/W used to control the electrical system shall be verified through analysis and/or testing prior to acceptance.	ACC-2-1
19.	Multiple operator/control center actions shall be required to initiate discharge of pyrotechnic devices. Operator feedback shall be provided to indicate successful completion of actions preparatory to pyrotechnic discharge.	ACC-2-1
20.	No combination of two failures (including operator error) shall result in initiation of pyrotechnics devices where the results are potentially catastrophic.	ACC-2-1
21.	Where pyrotechnic devices can not be avoided, the NASA Standard Initiator (NSI) shall be the preferred device.	ACC-2-1
22.	Firing circuitry employed for pyrotechnic devices shall provide a minimum of 20 dB with (TBD) safety margin on electromagnetic interference (EMI).	ACC-2-1
23.	Hazardous gas detectors shall be provided in locations where release of a hazardous gas would pose a hazard.	ACC-4/6-1
24.	Orifice purge flows shall restrict hydrogen/air ratios below explosive limits.	ACC-4/6-1

25.	No single failure shall result in the presence of a potential ignition source.	ACC-1-1
26.	Where hazardous gases may pose a potential combustion or explosive threat, electrical equipment shall be designed to explosion proof standards or "intrinsically safe" standards.	ACC-1-1
27.	Electronic components that require power during ascent shall be designed and/or qualified to the criteria of NS2/81-M082.	ACC-2-1

C. <u>THERMAL CONTROL SYSTEM</u>

1.	OTV heat rejection capabilities shall be single failure tolerant.	ACC-2-1
2.	All S/W used to control the thermal subsystem shall be verified through analysis and/or test prior to acceptance.	ACC-1-1

D. <u>OTV/ORBITER AND OTV/PAYLOAD INFORMATION SYSTEMS</u>

1.	No single sensor failure shall result in premature Caution and Warning (C&W) activation.	ACC-1-1
2.	No single electrical failure shall result in premature C&W activation.	ACC-1-1
3.	The alarm limits to which the sensors are set shall reflect the environment in which they are operated and the degree to which they will monitor for safe conditions.	ACC-4/6-1
4.	All S/W used to control or monitor OTV information systems and/or interfaces shall be verifed by analysis and/or test prior to acceptance.	ACC-4/6-1
5.	No single sensor failure shall result in the loss of the C&W system's ability to detect a hazardous condition.	ACC-1-1
6.	No single electronic failure shall result in loss of the C&W system's ability to function.	ACC-1-1

E <u>COMMUNICATIONS AND TRACKING</u>

1.	Procedures (including checklists) and crew training will be developed and implemented to insure all crew members are familiarized with the correct operating and safety procedures. Specific controls are TBD.	ACC-4/6-1
2.	All S/W used to control or initiate communications and/or tracking systems action shall be verified through analysis and/or test prior to acceptance.	ACC-4/6-1

3. Loss of optical or sensor reference or alignment capability in general will require that all engagement or proximity operations be stopped/suspended and equipment/platforms/payloads/OTV be brought into equilibrium until problem has been corrected. — ACC-4/6-1

4. The OTV shall have continunous ground command and control capabilities during all Orbiter proximity operations — ACC-4/6-1

F. <u>PROPULSION SYSTEM</u>

1. All S/W used to control the propulsion system shall be verified through analysis and/or testing prior to acceptance. — ACC-4/6-1

2. No two mechanical component failures shall result in premature/accidental engine firing. — ACC-4/6-1

3. No two electrical component failures shall result in premature/accidental engine firing. — ACC-4/6-1

4. No TBD number of operator errors shall result in premature/accidental engine firing. — ACC-4/6-1

5. No two mechanical, electrical, or operator failures/errors shall result in the loss of the ability to perform collision avoidance maneuver(s). — ACC-4/6-1

6. The electrical control system shall be able to diagnose electrical failures that will cause failure to fire or improper firing and reroute signals, etc., in order to perform the required maneuver(s) in the required time. — ACC-2-1

G. <u>ENVIRONMENTAL CONTROL SYSTEM</u>

1. No single electrical component failure shall result in loss of the ability to detect a fire given the condition that the fire has started and may have affected the fire detection system at any location. — ACC-1-1

2. No single sensor failure shall result in premature fire suppression system activation. — ACC-1-1

3. The fire suppression system shall be an arm/fire system. — ACC-1-1

4. No single equipment failure shall result in a critical hazard. — ACC-1-1

5. No single electrical, mechanical, or operator failure/error shall result in premature fire suppression system activation. — ACC-1-1

6, All S/W used to control the fire detection and suppression system shall be verified through analysis and/or testing prior to acceptance. — ACC-1-1

7. Cryogenic surfaces shall be insulated to preclude condensation of air. ACC-1-1

8. Controller/valves shall be fail safe such that the STS crew and/or OTV payload are not subjected to any increased hazardous risk. ACC-4/6-1

9. The amount of any reactive gases released into the OTV compartment shall not result in an average concentration greater than 25% of the lower explosion limit. ACC-1-1

10. The total available volume of combustible gas released into the an enclosure and ignited in the worst case concentration should not cause the structural damage of the enclosure by exceeding the compartment proof pressure. ACC-1-1

11. Provide for containment of shrapnel within the package interface of any major assembly which may require an enclosure with an explosive mixture of hydrogen (or other hazardous gases) leakage. ACC-1-1

12. Sizing of tubing or connectors shall be such that they are impossible to cross-connect. ACC-1-1

13. No two credible seal failures shall result in the release of hazardous gases or fluids. ACC-1-1

H. MAN-SYSTEMS

1. Radiation calculation baselines shall include:
   a. The galactic cosmic radiation environment shall be as defined by Adams, et.al., in NRL Memorandum 4506. TBD
   b. A reference orbit of 140 NM shall be the OTV park orbit. TBD
   c. Uniform shielding of 100 mils aluminum from internal equipment shall be assumed and factored into dose calculations for pressurized volume.
   d. No uncertainty factor shall be applied to the proton and electron spectra. TBD
   e. The trapped electron spectrum shall be calculated using NSSDC AE-8, May 1985, and magnetic field values for 1970 TBD
   f. The trapped proton spectrum shall be calculated using NSSDC/WDC-A-R&S 76-06, AP-8-Trapped Proton Environment for Solar Mazimum and Solar Minimum's, and magnetic field valure for 1970 (Epoch 1970). TBD

2. Equipment having an EVA interface shall meet JSC 10615, "Shuttle EVA Description and Design Criteria." ACC-4/6-1

3. All other *man-system* requirements are *TBD*. TBD

I. FLUID MANAGEMENT SYSTEM

1.	System connectors shall be keyed or sized so that it is physically impossible to connect an incompatible commodity or pressure level/vessel.	ACC-1-1
2.	Color coding of pressure vessels, pipes, tubing and connectors shall conform to TBD upon delivery of articles.	ACC-1-1
3.	All liquid and gas systems shall be designed to permit leak testing after installation.	ACC-1-1
4.	An isolation shutoff valve shall be installed in each system supplied from a common liquid or gas pressure source.	ACC-4/6-1
5.	All materials including seals, gaskets and lubricants used in flight equipment shall be compatible with the system commodity.	ACC-1-1
6.	Perform off-gas testing prior to space flight use.	ACC-1-1

J. SOFTWARE

1.	Perform analysis to identify areas of sensitivity to hardware-induced software errors (bit-changes, errors, etc.). Where sensitivity is identified, implement verification measures (such as command redundancy, command/compliment schemes, etc.) commensurate with the controlled function criticality.	ACC-4/6-1
2.	Provide S/W pre-use checkout capability.	ACC-4/6-1
3.	Provide redundancy for critical software functions (i.e., function capability simultaneously present in two or more processors).	ACC-4/6-1
7.	Implement S/W such that two or more processors are required to initiate a potentially hazardous event sequence.	ACC-2-1
8.	Utilize modular S/W design and structure to enhance comprehension of decision logic.	ACC-4/6-1
9.	Utilize write protected memory locations for critical software.	ACC-4/6-1
10.	Protect S/W that controls interrupt priorties against inadvertent overwrite.	ACC-4/6-1
11.	Initilize all unused memory locations to a pattern that, if executed as an instruction, will cause the system to revert to a known safe state.	ACC-2-1
12.	Evaluate all software interrupt priorities for safety impact.	ACC-1-1

13. Identify singularities (potential division by zero, etc.) associated with critical S/W modules. Verify that potential singularity occurences will return the system to a known safe state. — ACC-2-1

14. Verify critical S/W modules by test. — ACC-1-1

15. Incorporate provisions in safety critical software modules to ensure that errors resulting from on-orbit compilation of additional S/W does not overwrite, invalidate, or otherwise render the critical S/W ineffective. — ACC-4/6-1

16. Provide S/W control where system response is time critical. — ACC-2-1

17. Implement initiation of potentially hazardous event sequences such that:
    a. No hazardous sequence can be initiated without operator intervention. — ACC-4/6-1
    b. Two or more operator actions are required to initiate any potentially hazardous sequence. — ACC-4/6-1

18. Provide capability for operational checkout to testable critical system elements prior to exceution of a potentially hazardous sequence. — ACC-4/6-1

19. Protect S/W against hardware induced errors. — ACC-4/6-1

K. <u>MISCELLANEOUS</u>

1. Provide redundant source of power for critical systems/equipment/components. — ACC-2-1

2. Utilize redundancy via other OTV/STS processors for critical functions. — ACC-4/6-1

3. Provide processor self-test capability to verify processor integrity prior to initiation of potentially hazardous event sequences. — ACC-4/6-1

4. Subsystems or materials subject to degraded performance or failure due to environmental extremes shall be provided with active and/or passive thermal control with failure tolerance levels consistent with hazard potential. — ACC-2-1

L. <u>MECHANICAL GROUND SUPPORT EQUIPMENT</u>

1. Design shall include a scupper to catch leaking or spilled transfer fluid. — ACC-1-1

2. Service lines/hoses shall be of sufficient length to provide remote operation of pressure control panel. — ACC-1-1

3.	Lines and fittings shall be designed to withstand pressures at least four times maximum working pressure without rupture or burst.	ACC-1-1
4.	Tubing shall be stainless steel per KSC-SPEC-0007, fabricated and installed per KSC-SPEC-0008.	ACC-1-1
5.	Pressure connections for tubing shall be in accordance with MC240 or MS33649.	ACC-1-1
6.	Tubing shall be proof pressure tested to 1-1/2 times maximum working pressure by hydrostat or 1-1/4 times by pneumatic.	ACC-1-1
7.	Material in contact with fuels, oxidizers or combustible gases shall be selected, tested and certified per NHB 8060.1.	ACC-1-1
8.	Control system shall be equipped with an emergency stop switch.	ACC-1-1
9.	All components including structures should be constructed of compatible material that is not subject to oxidation.	ACC-1-1
10.	Control stations should be designed to conform to MIL-STD-1472, Chapter 5.	ACC-1-1
11.	Controls for critical functions should be designed and located in a manner not susceptible to inadvertent operation.	ACC-1-1
12.	Provide locking and tested tiedowns capable of restraining anticipated loads with a satety factor of 5:1 ultimate. NOTE: Special requirements for air transport of pressurized vessels and hazardous chemicals are listed in AFR71-4 and U.S. Code Title 49, Exemptions and waivers may be required.	ACC-1-1
13.	OTV mating to launch vehicle design shall provide for handling by two or more personnel and/or mechanical means.	ACC-1-1
14.	Hoist/crane design shall include positive failsafe braking system, finite controls and capability to lift a minimum of five (5) times maximum anticipated load.	ACC-1-1
15.	Hoist/crane line shall be sized to carry a suspended load of at least five (5) times maximum anticipated load.	ACC-1-1
16.	Positioning of loads shall be facilitated through use of center of gravity identification, matching guidelines, identification of attaching points, etc.	ACC-1-1
17.	All designs shall avoid the use of carbon based lubricants and minimize friction points.	ACC-1-1

18.	Test sets should be cleaned for oxygen service per MSFC-SPEC-164.	ACC-1-1
19.	All valves and controls should conform in shape, size, and mode of operation as outlined in MIL-STD-1472.	ACC-1-1

M ELECTRICAL GROUND SUPPORT EQUIPMENT

1.	Provide ventilation and/or air conditioning commensurate with component environmental requirements.	ACC-1-1
2.	System controls/panel design shall conform to MIL-STD-1472, Section 5.4.	ACC-1-1

N. OTV OPERATIONS

1.	OTV shall be shut down, except for required avonics, power, and command and control systems, and be orbit/attitude stabilized prior to TBD feet of Orbiter rendezvous.	ACC-4/6-1
2.	During Orbiter proximity operations, there shall be continunous Ground Control Center monitoring and control capabilities of the OTV.	ACC-4/6-1
3.	The OTV shall have all fuel tanks vented, sealed, and all associated lines purged prior to Orbiter proximity operations.	ACC-4/6-1
4.	The Orbiter Crew shall assume control of the OTV for all proximity operations. The control zone shall be TBD NM in any direction of the Orbiter.	ACC-4/6-1
5.	The OTV water collection tanks shall be dumped no later than TBD hours prior to Orbiter rendezvous.	ACC-4/6-1
6.	The OTV shall not impact any other element envelopes within the Orbiter Cargo Bay.	ACC-7-1
7.	The Orbiter shall be required to approach/depart the OTV. The OTV shall not fire its ACS or MPS unless necessary to protect the Orbiter and/or its crew.	ACC-4/6-1
8.	The crew shall have direct line-of-sight viewing of all proximity operations which involve docking or berthing, including an unobstructed view of the approach and departure paths.	ACC-4/6-1
9.	For all proximity operations a crew member shall be required to actively monitor and exercise command and control of the OTV.	ACC-4/6-1
10.	The OTV shall be equipped with visual ranging cues (markings/targets of known dimension)visible to the Orbiter crew along the OTV's normal approach path.	ACC-4/6-1

11.	The OTV shall provide for nighttime visibility for proximity ops.	ACC-4/6-1
12.	The Orbiter crew shall be trained in all aspects of OTV operations prior to conducting on-orbit operations.	ACC-4/6-1
13.	Operations for dealing with potential collisions with orbital debri are TBD.	TBD
14.	Operations for dealing with potential Orbiter collisions with the OTV are TBD.	ACC-4/6-1
15.	A designated STS center shall be responsible for coordination and integration of proposed OTV missions into the Orbiter's long-term planning effort.	ACC-4/6-1

OTV/AFT CARGO CARRIER REQUIREMENTS

(THESE REQUIREMENTS ARE ALSO APPLICABLE
TO STS AND SPACE STATION OPERATIONS)

	REQUIREMENTS	PHA No.
1.	OTV shall not impact other element envelopes assigned within the ACC (if any).	ACC-2-1
2.	ACC shall withstand natural and induced thermal and external acoustic environments.	ACC-2-1
3.	ACC shall withstand structural loads.	ACC-2-1
4.	ACC shall withstand aero loads.	ACC-2-1
5.	Compartmental hazardous gas content shall be less than 4% (safe compartments).	ACC-1-1
6.	ACC shall provide a breathable air purge during ingress of ground crew.	ACC-7-1
7.	ACC shall be purged prior to, during, and after ET cryogenic tanking.	ACC-1-1
8.	ACC shall provide capability to drain propellants at the pad.	ACC-1-1
9.	ACC flight subsystems' redundancy shall not be less than fail safe.	ACC-2-1
10.	Redundant components shall be physicialy oriented or separated to reduce the chance of multiple faillures from the same cause.	ACC-2-1
11.	Explosive devices shall be armed as near the time of use as is feasible with provisions for disarming.	ACC-2-1
12.	Pytotechnics and associated electrical circuits and electronics shall conform to STS Spec. JSC 08060.	ACC-2-1
13.	All ACC LRUs shall be accessible.	ACC-1-1
14.	ACC shall not violate the LH2 tank aft dome pressure requirement of 0.19 psi when LH2 tank is depressurized prior to loading.	ACC-1-1
15.	Power provided by the Orbiter (STS) shall not exceed 50 KWh.	ACC-1-1

16.	ACC/PL mass properties shall be such that the mated orbiter and ET/ACC is controllable with the orbiter flight control system during all mission phases (while the ET is attached to the STS).	ACC-2-1
17.	ACC shall accommodate the OTV with the environments (acceleration, vibration, acoustic, thermal, and pressure) specified in ICD 2-19001.	ACC-2-1
18.	ACC shall be able to carry a payload (OTV) mass of 55 Klb.	ACC-1-1
19.	ACC shall provide the capability to carry a cargo with the following dimensions: 25' Diameter, 15' Length	ACC-1-1
20.	ACC shall comply with the contamination criteria specified in ICD 2-19001.	ACC-1-1
21.	The OTV shall be accessible for LRU replacement on the pad, while mated to the ET/ACC.	ACC-1-1
22.	The ACC shall provide the capability for the following:	
	a. Ground checkout and status of the OTV.	ACC-1-1
	b. Flight status of the OTV	ACC-2-1
	c. On-orbit predeployment checkout of the OTV.	ACC-4/6-1
23.	The ACC environmental protection system (EPS) shall:	
	a. Protect the ACC structure from ascent heating of TBD BTU/ft^2 sec.	ACC-2-1
	b. Satisfy ice/frost accumulation limit requirement of 1/16".	ACC-1-1
	c. Maintain P/L compartment within allowable temperature limits during ascent of TBD degrees.	ACC-2-1
	d. Maintain P/L compartment within allowable acoustic limits during ascent of TBD db.	ACC-2-1
	e. Maintain subsystem components allowable temperature limits of TBD degrees.	ACC-2-1
24.	Purge, Vent, Hazardous Gas Detection Systems shall provide the following:	ACC-1-1
	a. Prevent/monitor hazardous gas accumulation.	
	b. Condition payloads (OTV).	
	c. Maintain compartments within allowable pressure/temperatures.	
	d. Reduce acoustic levels in compartments.	

25.	Shround Separation System (SSS) shall: a. Sever the ACC shroud at the separation plane. b. Maintain positive control of the shroud during separation. c. Receive separation signal from and confirm separation to the STS Orbiter.	ACC-2-1
26.	Payload Accommodation System (PAS) shall:	
	a. Provide ACC/payload interface (structural, electrical, fluid).	ACC-1-1
	b. Support cargo during ground processing, ascent, and on-orbit operations.	ACC-1-1
	c. Deploy payload.	ACC-2-1
27.	ACC Avionics Subsystem (AS) shall provide electrical power, data transfer, and command/control for all ACC subsystems.	ACC-2-1
28.	ACC will not affect ET break-up altitude requirement.	ACC-2-1