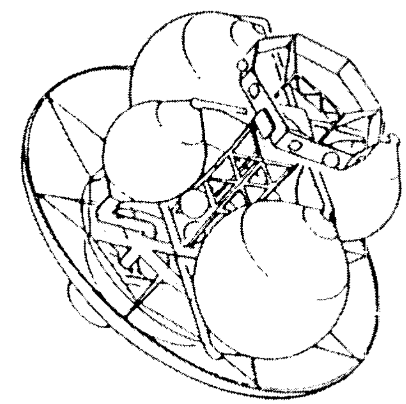
MCR-87-2600/NAS8-36108
DR-3

ORBITAL TRANSFER VEHICLE

CONCEPT DEFINITION
AND
SYSTEM ANALYSIS STUDY



CONTRACT EXTENSION II
FINAL REVIEW

NASA - MSFC
9 DECEMBER 1987

MARTIN MARIETTA

AGENDA

EXECUTIVE SUMMARY	BILL WILLCOCKSON
PROGRAM / MISSION ISSUES	BILL WILLCOCKSON
DESIGN ISSUES	LARRY REDD
STRUCTURAL ISSUES	WALLY HAESE
AEROASSIST	BILL WILLCOCKSON

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EXECUTIVE SUMMARY

PREVIOUS RESULTS

INITIAL OTV PROGRAM

ADVANCED MISSIONS

ACC OTV SAFETY ISSUES

AEROASSIST SUMMARY

EXTENSION II SUMMARY

PROGRAM SUMMARY

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PREVIOUS RESULTS

OTV STUDY OVERVIEW

The current activity is an extension of the Orbital Transfer Vehicle Concept Definition and System Analysis Study that was initially awarded in July, 1984. The viewgraph shows key characteristics of the initial and extension study scenarios. The direction encompassed in items 3 and 4 of the initial study scenario was particularly significant to the results achieved. They were very limiting with respect to the scope of the recommended OTV program.

The extension study opens the scope of potential recommendations by introducing a variety of ambitious programs, and by making the large cargo vehicle recommended by the Space Transportation Architecture Studies available at no acquisition cost to the OTV program. It is a further objective of the extension study to evaluate the sensitivity of OTV program recommendations to scenario variations such as different mission models, different launch vehicle availability, and different space station availability.

OTV STUDY OVERVIEW

1) PHASE A (1984-1985) SCENARIO

ESTABLISH THE OTV DESIGN, OPERATIONS, AND BASING CONCEPTS WHERE:

A) SHUTTLE CAPABILITY IS GROWING AGGRESIVELY (72KLB PAYLOAD)
B) SPACE STATION IS BEING PHASED IN, STRONG BASING OPTION FOR OTV
C) DECISIONS ARE JUSTIFIED BY A CONSERVATIVE MISSION MODEL
D) ANY LARGE CARGO VEHICLE DEVELOPMENT MUST BE JUSTIFIED BY OTV ALONE

2) PHASE A, EXTENSION #1 SCENARIO

ESTABLISH CHANGES TO THE OTV PROGRAM DEFINITION RESULTING FROM:

A) A WIDE VARIETY OF AGGRESIVE MISSION MODELS
B) A LARGE CARGO VEHICLE WHOSE DDT&E IS NOT CHARGED TO THE OTV PROGRAM

3) PHASE A, EXTENSION #2 SCENARIO

INVESTIGATE A PROGRAM TO ACCOMPLISH THE FOLLOWING

A) ESTABLISH AN INITIAL OTV PROGRAM COMPATABLE WITH NEAR-TERM CNDB MISSIONS
B) DEFINE SAFETY IMPACTS TO THE ACC OTV RESULTING FROM THE STS / CENTAUR CANCELLATION
C) INVESTIGATE PROGRAM IMPLICATIONS OF THE ADVANCED MISSIONS PROPOSED IN THE CIVIL SPACE LEADERSHIP INITIATIVE (CSLI) PROGRAM

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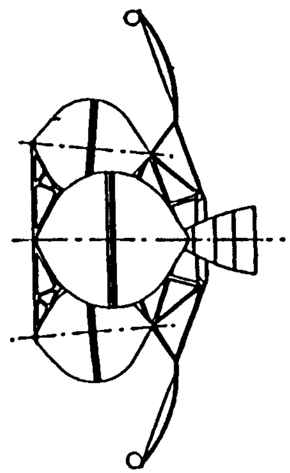
PROGRAM RECOMMENDATIONS (1984/85 STUDY)

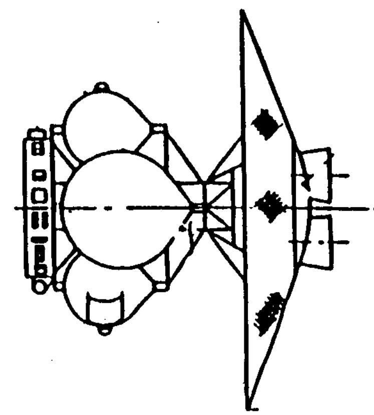
The selected Option I program uses the ground based Aft Cargo Carrier vehicle and the space based vehicle pictured. Both configurations use a four-propellant-tank concept. The ground based configuration is not man-rated and uses one main engine. This engine incorporates new technology, but presses it only to a performance level of 475 seconds specific impulse to reduce development risk. This same engine is used in a dual installation in the space based configuration. The propellant capacity of the ground based configuration is 45,000 pounds. The space based configuration must retrieve the manned capsule, and its propellant capacity is increased to 55,000 pounds and its aerobrake diameter to 44 feet. Both configurations use composite structure -- graphite epoxy for the cool structure and graphite polyimide for the hot aerobrake support structure.

The selected program characteristics, which were justified by the low Revision 8 mission model, are summarized. The IOC for the ground based system is 1984, and the space based IOC is 1999. This scenario justified development of the ACC scavenging system rather than a new large capability propellant tanker. The space based vehicle, although not initially man-rated, has all the equipment installed that is required to make this possible. The only additional requirement is validated flight experience, which is gained during the early unmanned years of space based operation.

PROGRAM RECOMMENDATIONS (1984/85 STUDY)



- ACC CONFIG
- SINGLE ENGINE (475 sec ISP)
- 45KLB PROP
- NON MAN RATED
- INTEG AVIONICS
- 40' AEROBRAKE
- COMPOSITE STRU



- 4 -TANK CONFIG
- DUAL ENGINE (475 sec ISP)
- 55 KLB PROP
- MAN RATED
- AVIONICS RING
- 44' AEROBRAKE
- COMPOSITE STRU

GROUND BASED OTV

SPACE BASED OTV

OPTION I

PROGRAM
- DECISIONS BASED ON LOW REV 8 OTV MISSION MODEL
- ONLY TWO CONFIGURATIONS REQUIRED
- 1994 IOC FOR GROUND BASED SYSTEM, 1999 SPACE BASED
- PREFER ACC OTV & SCAVENGING TO PROP LOGISTICS VEHICLE
- TRANSITION TO MAN RATING AT SPACE BASE IOC

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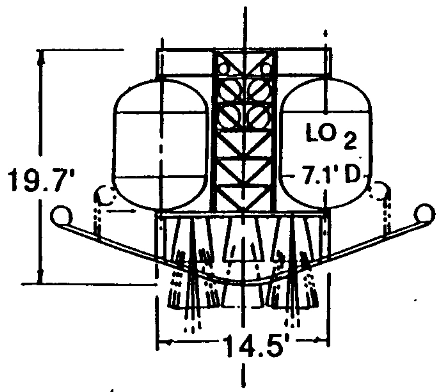
GJD/002

NOMINAL C/V OTV PROGRAM

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

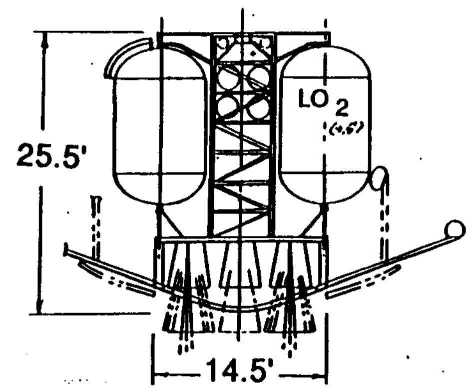
NOMINAL C/V OTV PROGRAM

OPTION 2/2 (SCENARIO 2)



- 4 TANK CONFIG
- THREE ENGINES (475 sec ISP)
- 52 Klb PROP
- NON MAN RATED
- 32' AEROBRAKE
- COMPOSITE STRU

GROUND BASED UNMANNED OTV



- 4-TANK CONFIG
- THREE ENGINES (475 sec ISP)
- 74 Klb PROP
- MAN RATED
- 38' AEROBRAKE
- COMPOSITE STRU

SPACE BASED MANNED OTV

PROGRAM
- DECISIONS BASED ON REV 9, 2/2 MISSION MODEL
- KEY GROUNDRULE: AVAILABLE SHUTTLES TO RECOVER OTV'S
- ONLY TWO OTV CONFIGURATIONS REQUIRED
- 1995 IOC FOR GROUND BASED SYSTEM, 1996 FOR SPACE BASED
- MAN RATED VEHICLE CAN OPERATE FROM GROUND AS WELL AS SPACE WITH MINIMUM DELTAS

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RECOMMENDED AEROASSIST CONCEPT

The preferred flex brake design is summarized in the viewgraph. The central 14.5 foot diameter is fabricated using shuttle tiles set on a graphite polyimide honeycomb cone with engine doors incorporated in it. This structure forms a base for the graphite polyimide ribs that support the flexible portion. The flexible portion is a multi-ply nicalon faced q felt and NEXTEL blanket which is sealed with RTV sealant on the cool (600° F) inside surface. The ribs are glued to the blanket to provide torsional stiffness. An inflated torus provides required curvature at the periphery of the brake, and stiffens the edge. As noted, this is the lightest design approach to a low L/D aerobrake.

This material is in a developmental stage, and its operational characteristics are not well understood. In lieu of definitive data, its operational life has been estimated at five uses -- shorter than the rigid brake at 20 uses, but longer than the single mission life of the ballute which must be repeatedly flexed during use. The data being developed by the Ames Research Center is promising, but needs to be pursued further. Therefore, our recommendation -- use the concept but continue to support the materials technology program.

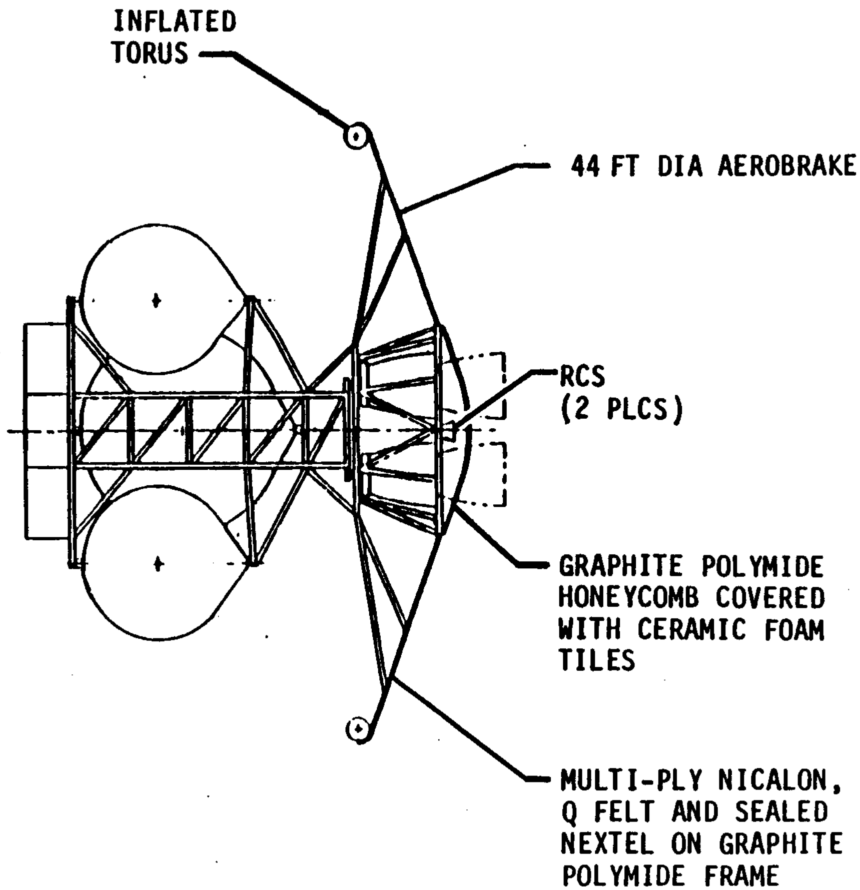
RECOMMENDED AEROASSIST CONCEPT

- FLEX BRAKE PROVIDES LIGHTEST, MOST EFFICIENT APPROACH

- LOW L/D (0.12) GIVES GOOD CONTROL MARGINS WITHOUT OVERDESIGN

- FLEX MATERIAL IS DEVELOPMENTAL AND INVOLVES SOME TECHNICAL RISK

- RECOMMENDATION
  - INCORPORATE FLEX BRAKE AS PROMISING CONCEPT
  - PURSUE MATERIAL TECHNOLOGY DEVELOPMENT



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EXTENSION II RESULTS

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OTV EXTENSION #2 TASKS

DEFINE AN INITIAL OTV PROGRAM CONSISTENT WITH NEAR TERM CSLI MISSIONS

DEVELOP EVOLUTION TO LONG TERM ADVANCED MISSIONS

INVESTIGATE SAFETY IMPLICATIONS OF AN ACC OTV

EXPAND ANALYSIS OF HIGH SPEED AEROASSIST

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INITIAL OTV PROGRAM

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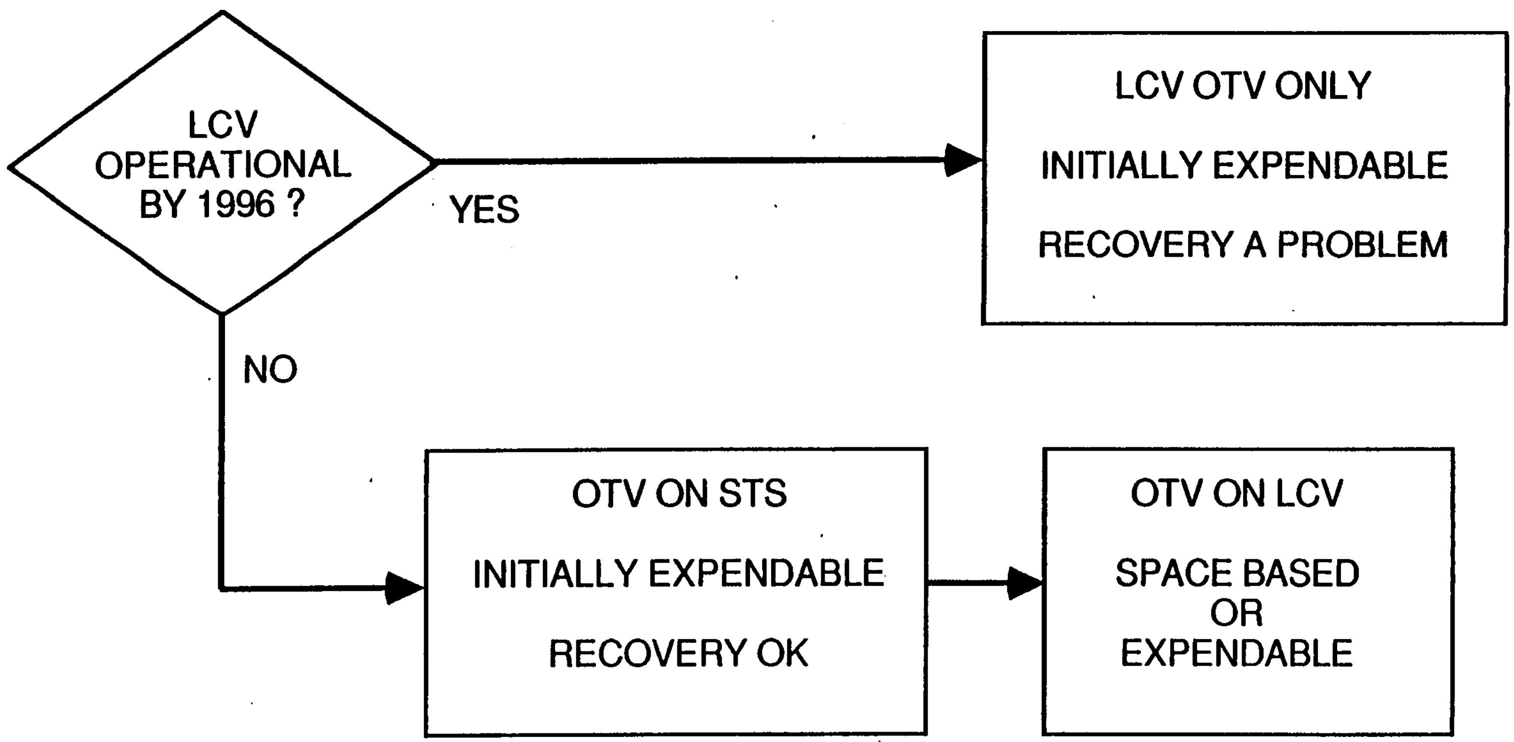
OTV BOOST OPTIONS



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ACC EXPENDABLE OTV

INITIALLY EXPENDABLE OTV

- REDUCES PROGRAM FRONT-END COSTS
- ALLOWS EARLY PROGRAM START
- SAFE ENHANCEMENT OF STS GEO CAPABILITY
- TRANSITION TO HEAVY LIFT LAUNCHER WHEN AVAILABLE
- ORDERLY GROWTH TO REUSEABILITY & SPACE BASING

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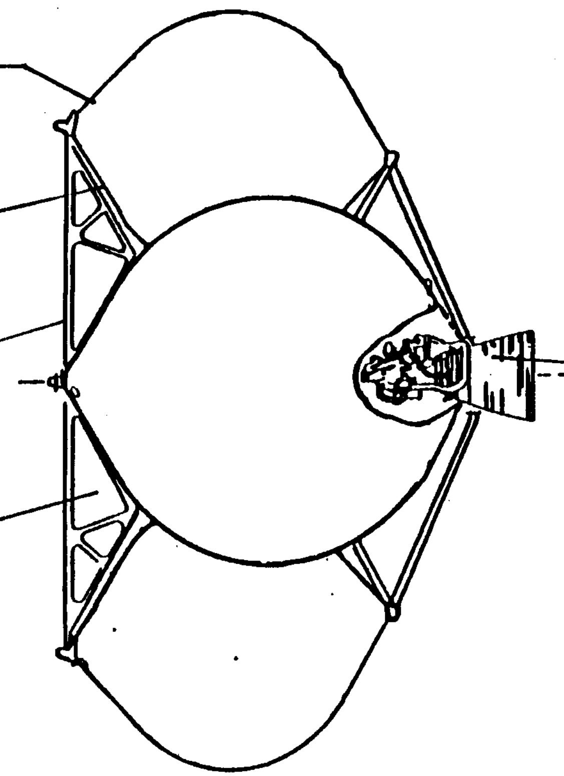
ACC EXPENDABLE OTV BASELINE

The general arrangement and weight breakdown for our selected expendable OTV transported in the ACC are shown on the facing viewgraph. The expendable OTV is based on the same arrangement as the groundbased 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.

The major differences are: aerobrake removal, Al 2219 tanks instead of Al-Li 2090 tanks, a RL10-A engine, and Ag-Zn batteries in place of the fuel cell system. Some GN&C equipment has been removed, or will be, replaced by a smaller system.

The total dry weight of the ACC expendable OTV is 4189 lb.

ACC EXPENDABLE OTV BASELINE



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

EXPENDABLE VEHICLE TRADE SUMMARY

The recommended characteristics of the initial expendable vehicle have been determined based upon cost trade studies. The recommendations 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.

EXPENDABLE VEHICLE TRADE SUMMARY

ALUMINUM
VS
ALUMINUM LITHIUM
TANKAGE → RECOMMEND INCORPORATING ALUMINUM-LITHIUM TANKS AS SOON AS THE MATERIAL IS AVAILABLE

ALUMINUM
VS
COMPOSITE
STRUCTURE → RECOMMEND USING COMPOSITE RATHER THAN ALUMINUM STRUCTURE

RL10A ENGINE
VS
IOC ENGINE → RECOMMEND USING IOC ENGINE AS SOON AS THE ENGINE CAN BE MADE AVAILABLE

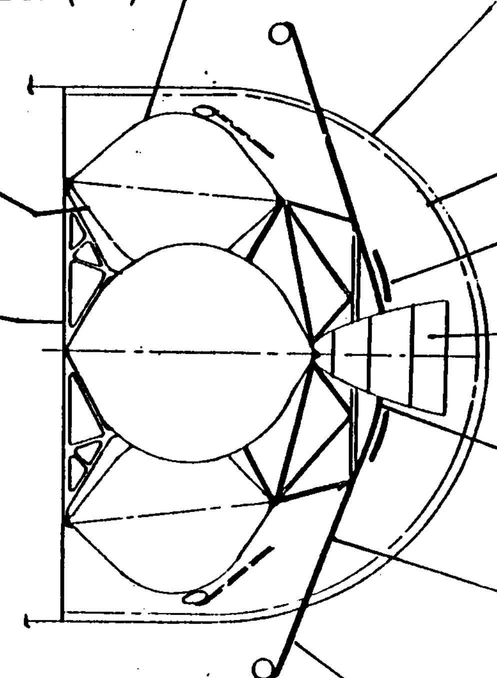
GROUNDBASED CRYOGENIC REUSABLE OTV

This viewgraph shows the general arrangement and weight breakdown of our selected groundbased cryogenic OTV transported in the ACC. 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).

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, along with the core system (LO2 tanks, structure, avionics, and propulsion) are stowed in the Orbiter cargo bay for retrieval after mission completion. The propulsion and avionics subsystems reflect the component count previously considered. The structure is of lightweight graphite/epoxy. The propellant load was selected to enable full use of the projected NSTS lift capability on GEO delivery missions.

GROUND BASED CRYOGENIC REUSABLE OTV



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

PAYLOAD TO GEO WITH STS

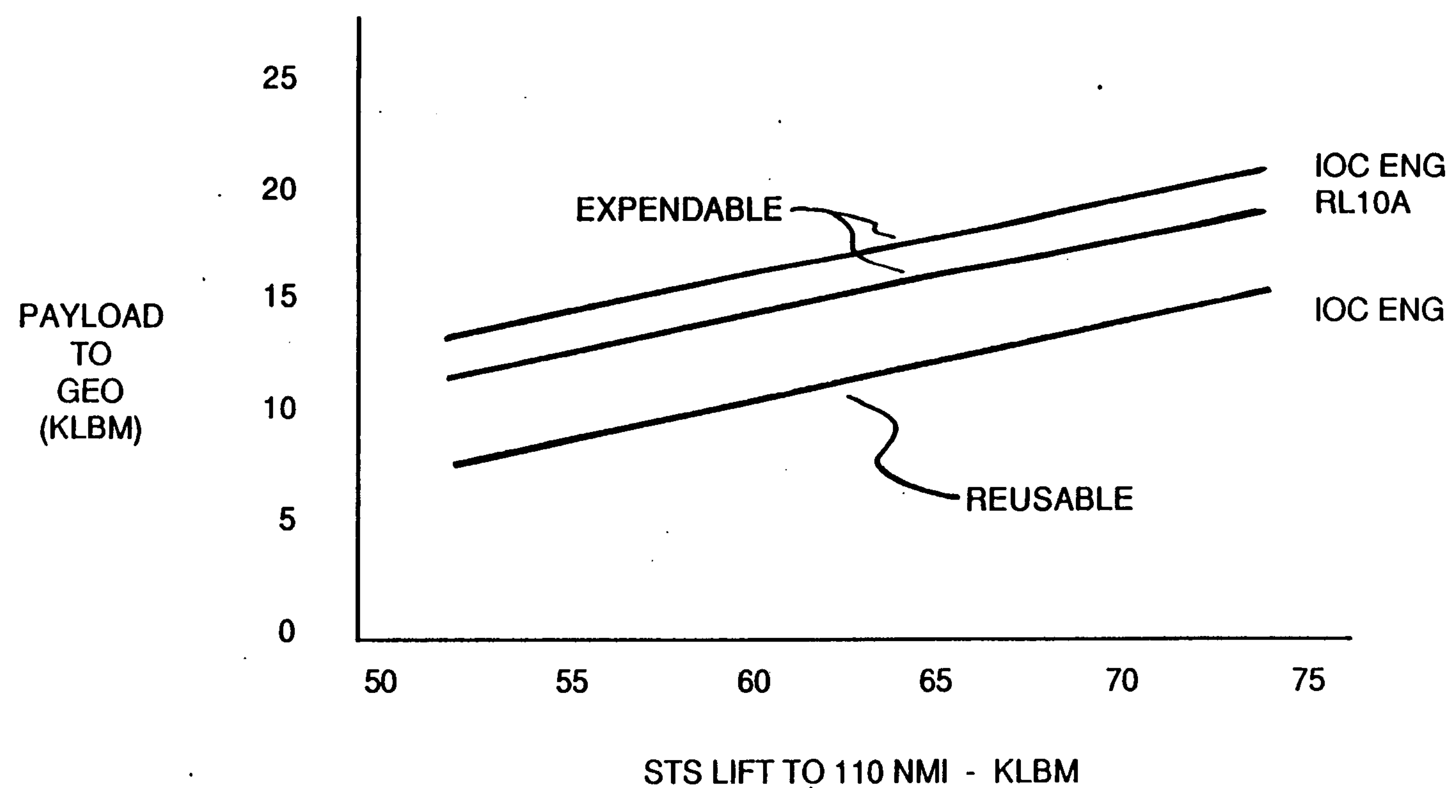
The figure shows OTV payload delivery capability to GEO as a function of STS delivery capability for the reusable and expendable vehicle concepts generated during this study. The STS lift capability shown corresponds to what the Shuttle can deliver to 110 nmi.

The conclusions to be drawn from the figure include the observation that the expendable vehicle concept is capable of delivering significantly greater payload to GEO than with the reusable concept. This may be a crucial realization 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 subsequent 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, could be higher than for the expendable for OTV class payloads.

PAYLOAD TO GEO WITH 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



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LARGE CARGO VEHICLE ISSUES

- EMERGENCE OF A LOW-COST HEAVY LIFT VEHICLE
  - → SHIFT OTV FROM STS TO LCV
  - → STS PERFORMS MANNED AND SPECIAL APPLICATIONS FLIGHTS ONLY
    - →→ VERY LIMITED STS DOWN CAPABILITY
    - →→ DIFFICULT TO RETURN OTV TO GROUND

$\boxed{\text{EXPENDABLE OR SPACE BASED OTV}}$

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LCV EXPENDABLE OTV

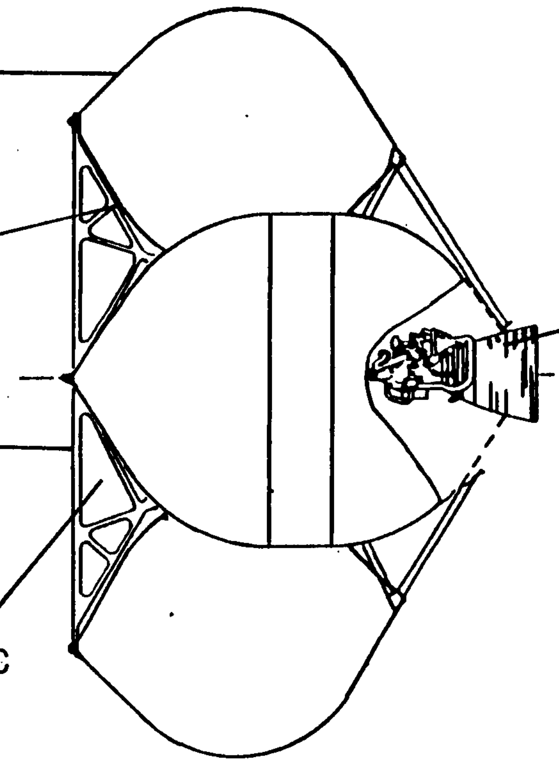
This viewgraph shows the general arrangement and breakdown of our selected expendable configuration which will be used in either a sidemount or inline LCV payload element.

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.

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.

LCV EXPENDABLE OTV



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

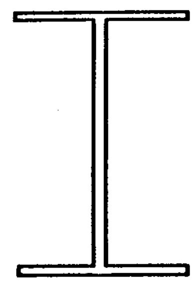
ASE FOR 50K OTV - INLINE CONFIGURATION

This figure 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.

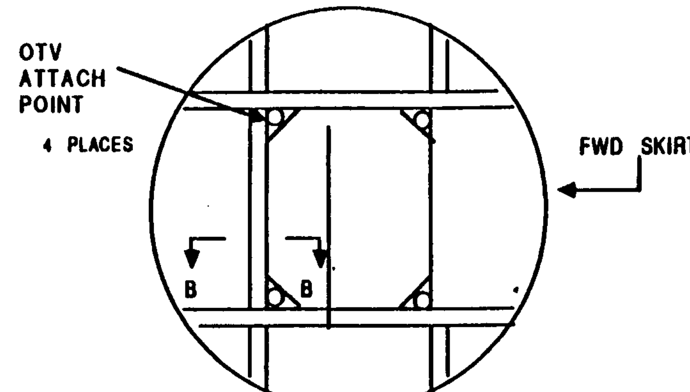
The OTV is mounted from the rear, using the umbilicals and attach points. The shroud (27.5 ft x 90 ft) separates just forward of the OTV support beams. A NASTRAN model was used to check the support beam for sizing.

The total weight of the ASE components is 3409 lb.

ASE for 50K OTV LCV IN-LINE CONFIGURATION

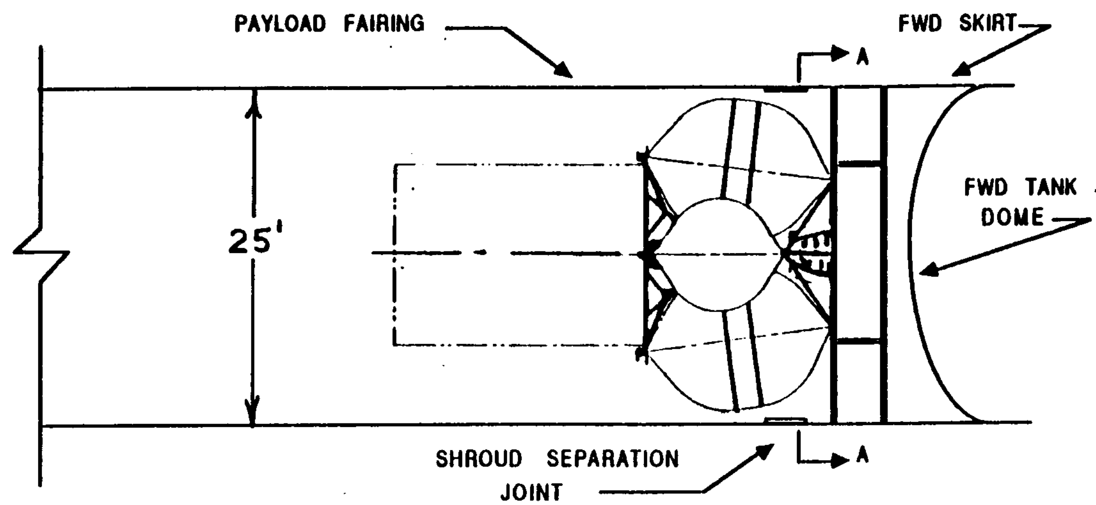


SECTION B-B
SUPPORT BEAM CROSS SECTION



SECTION A-A
OTV ATTACHMENT & SUPPORT BEAMS

ASE	WEIGHT (LB)
SKIRT	1746
FRAMES	810
ATTACH HRDW	108
PROP/MECH	125
AVIONIC/ELEC	152
ORDNANCE	23
CONTINGENCY	445
TOTAL	3409



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MANNED SPACE SYSTEMS

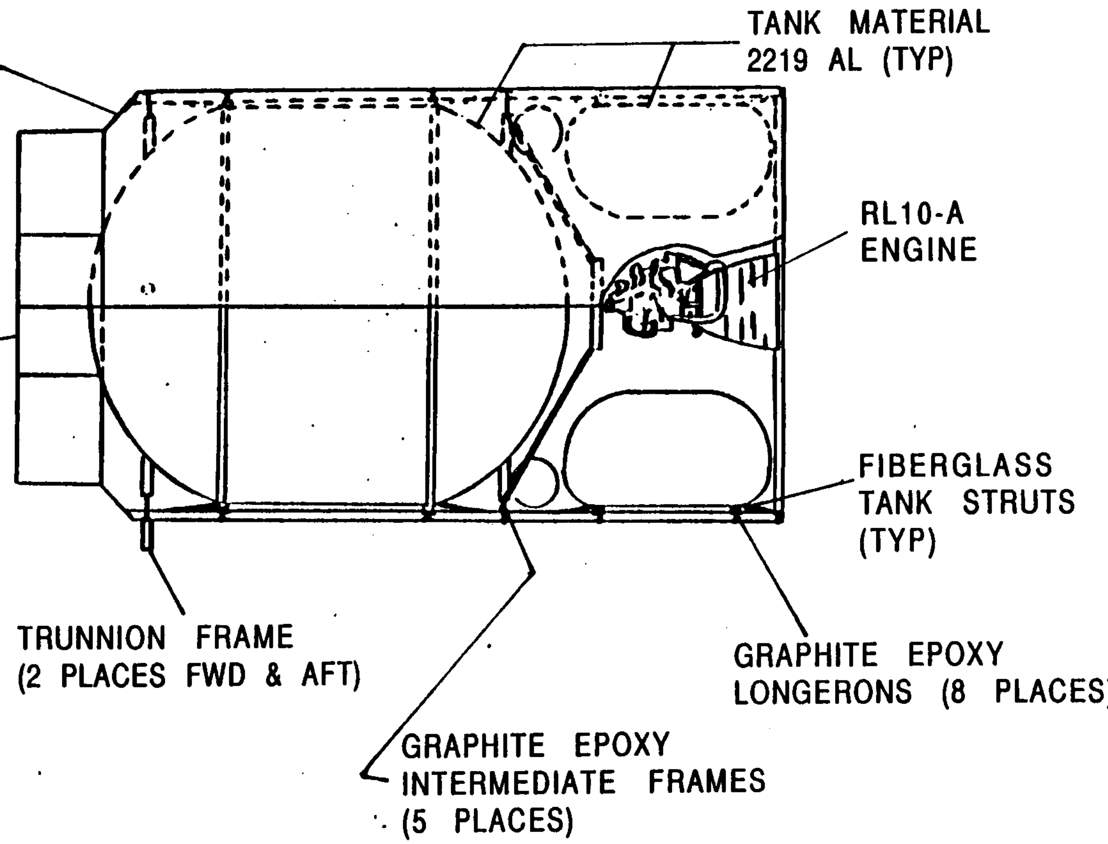
SHUTTLE-C EXPENDABLE OTV

This figure shows a cargo bay expendable OTV capable of derivering 15,000 lbm to GEO from Shuttle-C deployment in LEO. This concept is attractive because of its high performance and the vehicle's short length (compared to other cryogenic configurations).

The main contributor to the shortened length is incorporation of a toroidal LO2 tank in which the main engine is packaged. This concept was developed to emphasize short length while maintaining high performance, i.e., payload capability at minimum gross weight. According to the mission model assessment, the stage length plus ASE should not exceed 30 ft in order to minimize NSTS launch costs. In other words, the 30 ft payload capability and sufficient performance are the major desirable characteristics for a cargo bay OTV. This stage meets these criteria, i.e., 26.7 ft length, ASE length, and ASE packaging characteristics.

Minimum tank gages are 0.025 for the toroidal LO2 tank and 0.025 for the LH2 tank. The two tanks are protected by a cylindrical debris shield of graphite/epoxy, supported by longerons and ring frames of the same material. Each tank is attached to the longerons and frames by fiberglass/epoxy struts which accommodate the temperature differences. The avionics units have been mounted on an avionics ring that also serves as the payload interface. Ag-Zn batteries provide the power source, and the propulsion unit is a RL10-A engine.

SHUTTLE-C EXPENDABLE OTV (15' DIA)

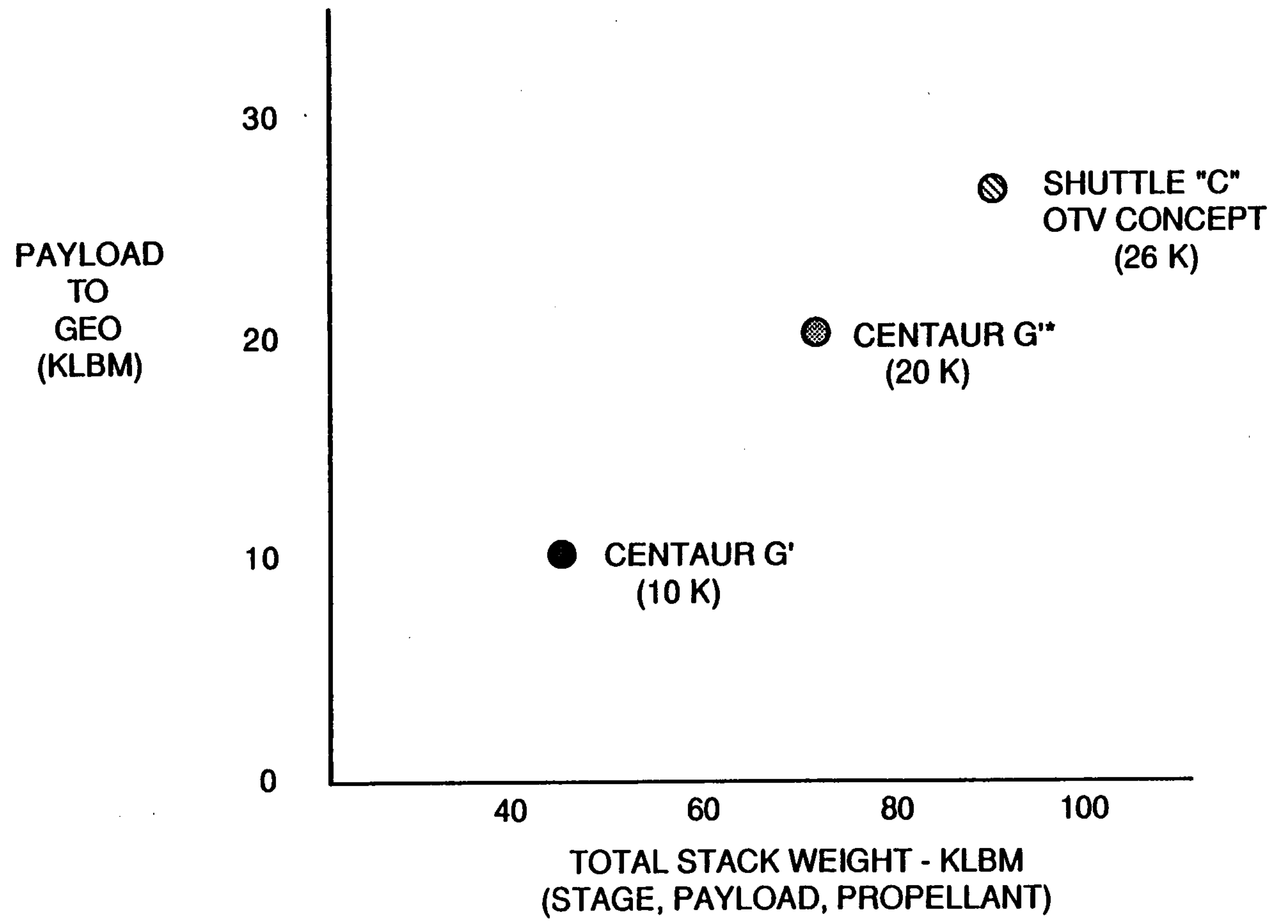


WEIGHT	
TANKS	1399
STRUCTURE	1329
ENVIRONMENTAL CTRL	464
MAIN PROPULSION	1002
ORIENTATION CTRL	187
ELECTRICAL SYSTEM	328
G. N. & C.	182
CONTINGENCY	734
DRY WEIGHT	5625
PROPELLANT, ETC	58924
LOADED WEIGHT	64549

EXPENDABLE VEHICLE COMPARISON

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

EXPENDABLE VEHICLE COMPARISON



* CENTAUR REQUIRES STRUCTURAL MODS
(MAX CAPABILITY TODAY = 10 K P/L)

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OTV BOOST OPTIONS

OTV DEVELOPMENT PATH UNCLEAR UNTIL NEW BOOSTER IS RESOLVED

A) LCV - LARGE DIAMETER CARGO VEHICLE WITH 100K+ CAPABILITY

WIDE DIAMETER GIVES GOOD GROWTH PATHS VIA MODULAR TANKAGE

NO EARTH RETURN - OTV EXPENDABLE UNTIL SPACE BASING POSSIBLE

B) SHUTTLE-C - NARROW DIAMETER (15') CARGO VEHICLE WITH 100K CAPABILITY

SMALLER DIAMETER REQUIRES DENSER PACKAGE, LESS OPTIMUM GROWTH

NO RETURN PATH AS ABOVE

C) SHUTTLE - BEST OPTION IF NO NEW CARGO VEHICLE APPEARS

ACC BEST LOCATION FROM SAFETY STANDPOINT

GIVES EXPENDABLE & REUSABLE PATHS WITH GROWTH CAPABILITY

RE-USE ECONOMICS UNFAVORABLE FOR STS LIFT LESS THAN 60K

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ADVANCED MISSIONS

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.

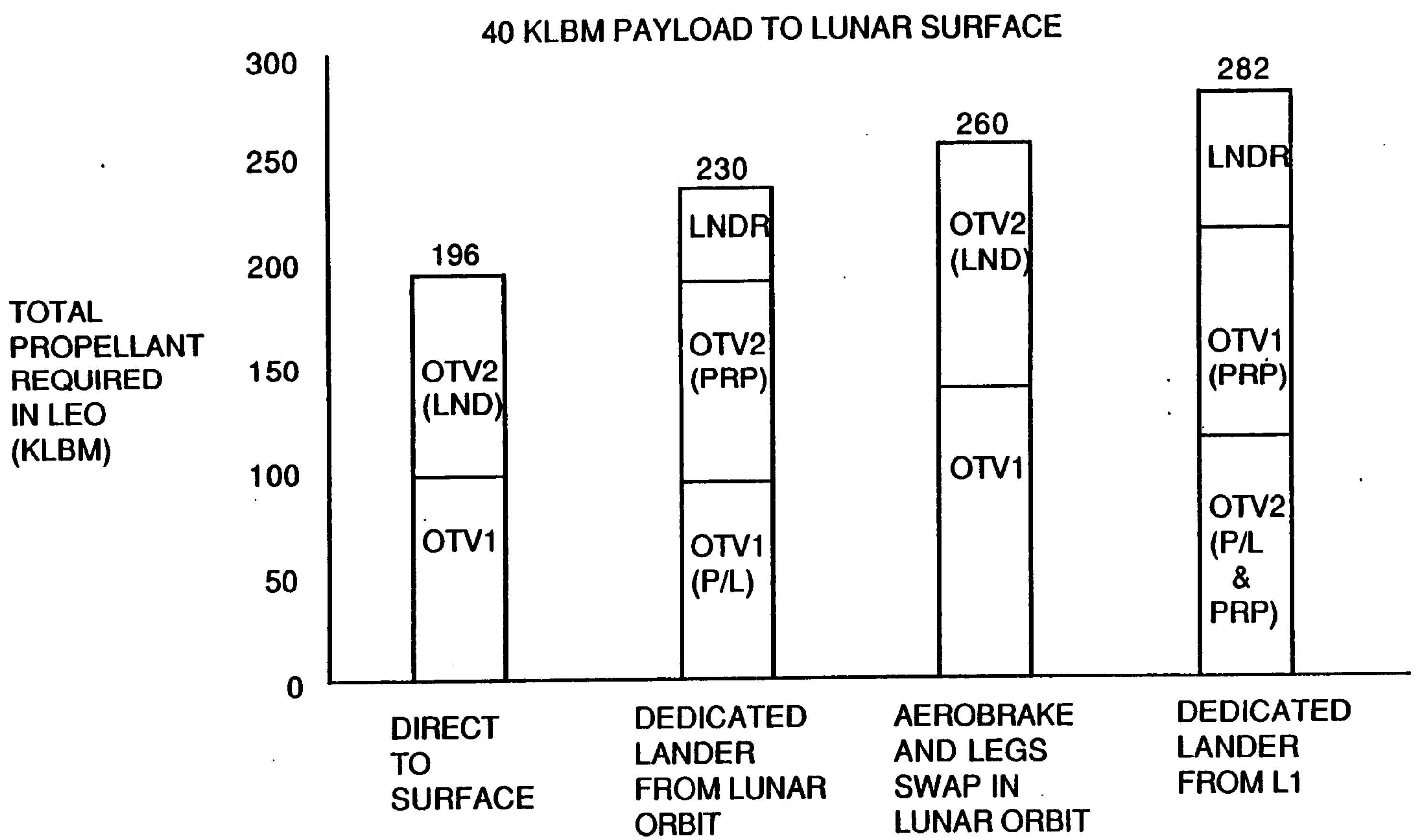
The dedicated lander approach uses two transfer vehicles to deliver the 40 Klbm payload and propellant for the lander to Lunar orbit. Then the propellant is transferred to the lander and the payload is delivered to the surface. The lander then returns to Lunar orbit.

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. Then on the return the landing stage would return to Lunar orbit to swap the landing legs back for its aerobrake and then 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 the figure. The most economical method of payload delivery to the Lunar surface appears to be the direct transfer to the surface. 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.

LUNAR TRANSFER COMPARISONS



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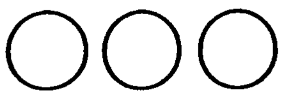
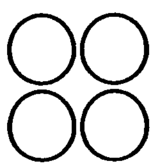
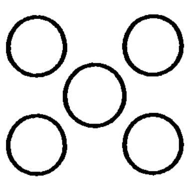
LUNAR LANDING ENGINE CONFIGURATIONS

Three (in-line), four, and five-engine configurations 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.

Four engines were chosen for Lunar landing applications. 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 (within a circular perimeter) and may offer the best growth path from a two engine system.

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
	.9797	0.66K - 11.7KLBF	18:1	- LOWEST RELIABILITY - LARGEST PATTERN - COMPLEX DESIGN AND CONTROL

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LUNAR AEROBRAKE WEIGHTS

This chart 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.

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 2298 lb for an increase of 458 lb over the GEO return brake weight.

LUNAR AEROBRAKE WEIGHTS

	LUNAR BRAKE	° GEO BRAKE
OTV CORE - 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	1046	799
TOTAL AEROBRAKE WEIGHT	2298	1840

ALL WEIGHTS IN POUNDS

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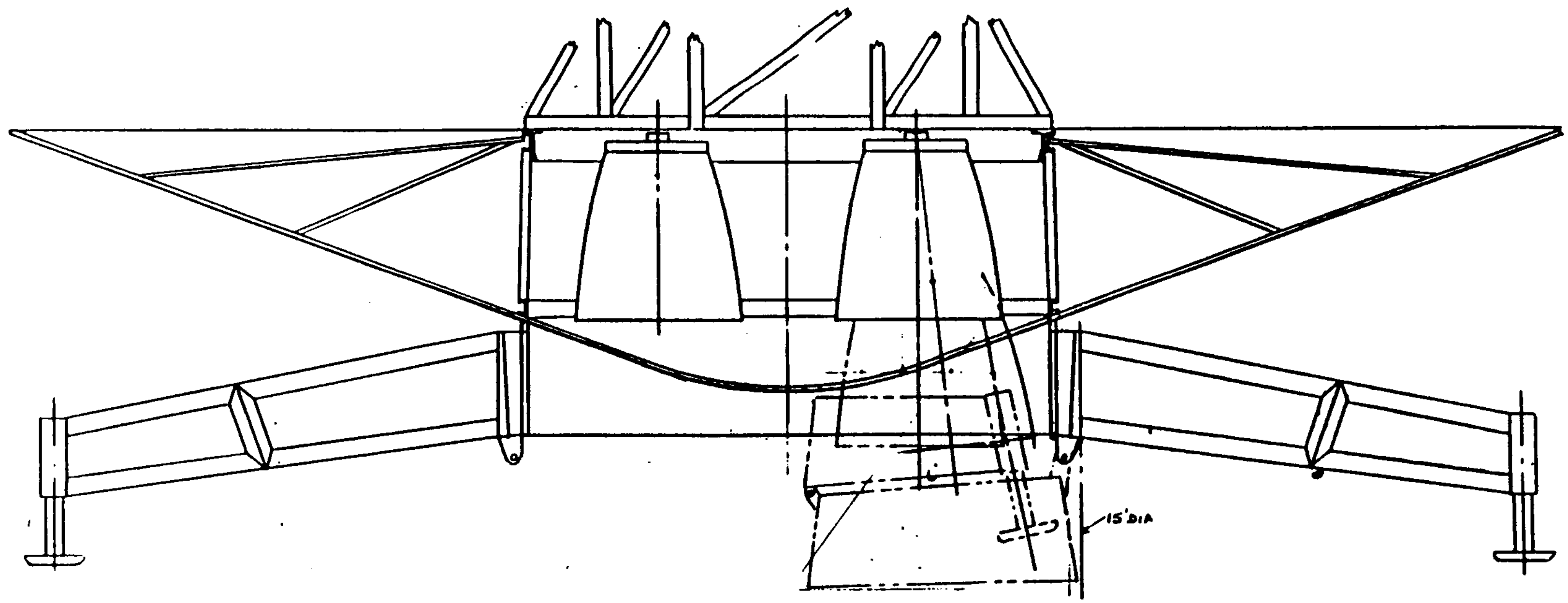
LUNAR LANDING LEGS

The figure shows a possible design for landing legs to accommodate the missions to the lunar surface. The legs fold under the aerobrake hard shell into a diameter compatible with delivery to LEO in the STS cargo bay. Therefore, the leg assembly could be attached to the vehicle after initial launch of both sections.

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.

LUNAR LANDING LEGS

- 4 LEGS, 40 FT PATTERN DIAMETER
- MOUNTING PROVISIONS:
    - A. DIRECTLY TO AEROBRAKE
    - B. MOUNT TO STRUCTURE
- LANDING CONDITIONS:
    - MAX PAYLOAD = 40K
    - ENGINE CUTOFF HEIGHT = 5 FT
    - MAX DECELERATION = 0.5g
- SYSTEM WEIGHT = 1300 LBM



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LRR

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. High traffic rates beginning in the year 2000 will more than justify IOC engine and aeroassist 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 stage (98klb) which must be space based. Finally, landing on the moon requires significant upgrade of the OTV systems (landing legs, engines, avionics, etc) as is spelled out in the Design Issues section.

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 the completion of program upgrades at the year 2000. It is felt that 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 growth to the 2000 targets. A small landing mission in 1997 could be accomplished by a ground based 50klb capacity OTV in an expendable mode.

OTV PHASED GROWTH - LUNAR INITIATIVE

INITIAL PROGRAM	IOC ENGINE	AERO-BRAKE	MAN-RATING	SPACE BASING	LUNAR LANDINGS
• EXPENDABLE • RL-10 (ISP=440)	• IMPROVED PERFORMANCE (ISP=475)	• RE-USE • RETURN P/L'S	• REDUNDANCY • DUAL ENGINES	• ON-ORBIT REPAIR & RESUPPLY	• LANDING LEGS • MULT ENGINES • RADAR SYST
1993	1996	1996	2000	2000	2000

LUNAR INITIATIVE JUSTIFIES ALL OPTIONS ON AGGRESIVE SCHEDULE

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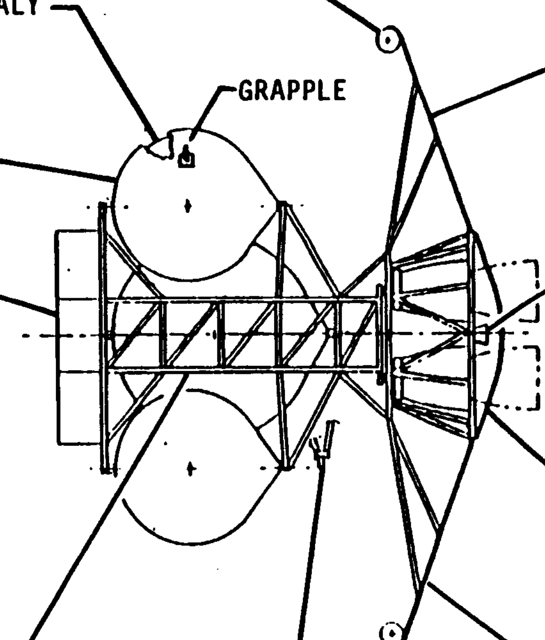
98 KLBM SPACE BASED LUNAR TRANSFER VEHICLE

The figure depicts the workhorse vehicle concept selected for delivering payloads, OTV's + payloads, etc. toward the Moon (the surface, Lunar orbit, or to a libration point). The vehicle was sized such that two stages of this concept (one containing the Lunar landing modifications) could deliver the 40Klbm 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.

98 KLBM LUNAR TRANSFER VEHICLE

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

98 KLBM SPACE BASED LUNAR LANDING VEHICLE

The concept shown in the figure 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.

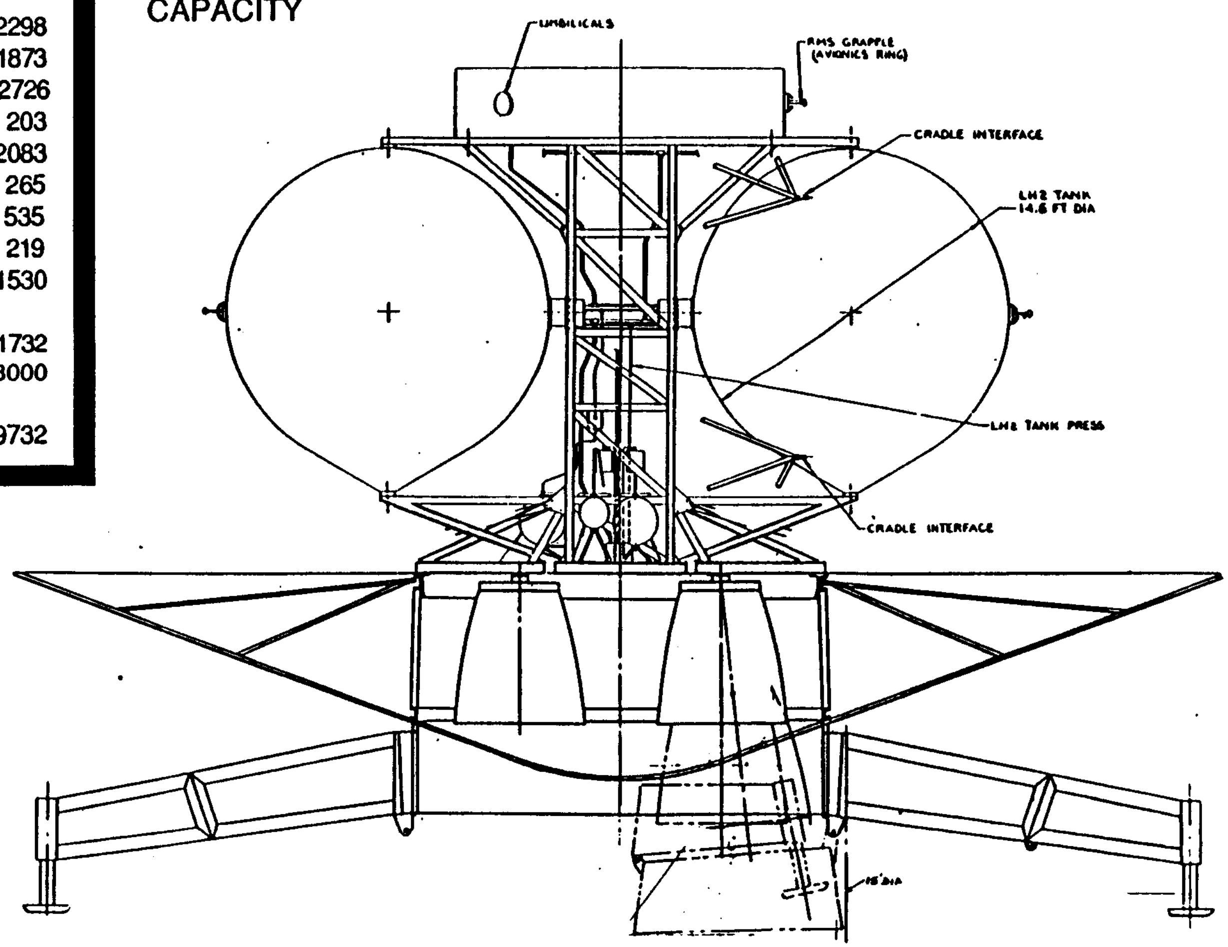
The figure shows a design concept for landing legs to accommodate the missions to the lunar surface. The legs fold under the aerobrake hard shell into a diameter compatible with delivery to LEO in the STS cargo bay. Therefore, the leg assembly could be attached to the vehicle after initial launch of 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.

98 KLBM LUNAR LANDER AND EARTH RETURN VEHICLE

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

98000 lbm PROPELLANT CAPACITY



45.2 FT DIA AEROBRAKE

4 - 17 KLBF ENGINES (THROTTLEABLE)

4 - LEG LANDING GEAR (ALUMINUM) LEFT ON SURFACE

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DEDICATED LUNAR LANDER

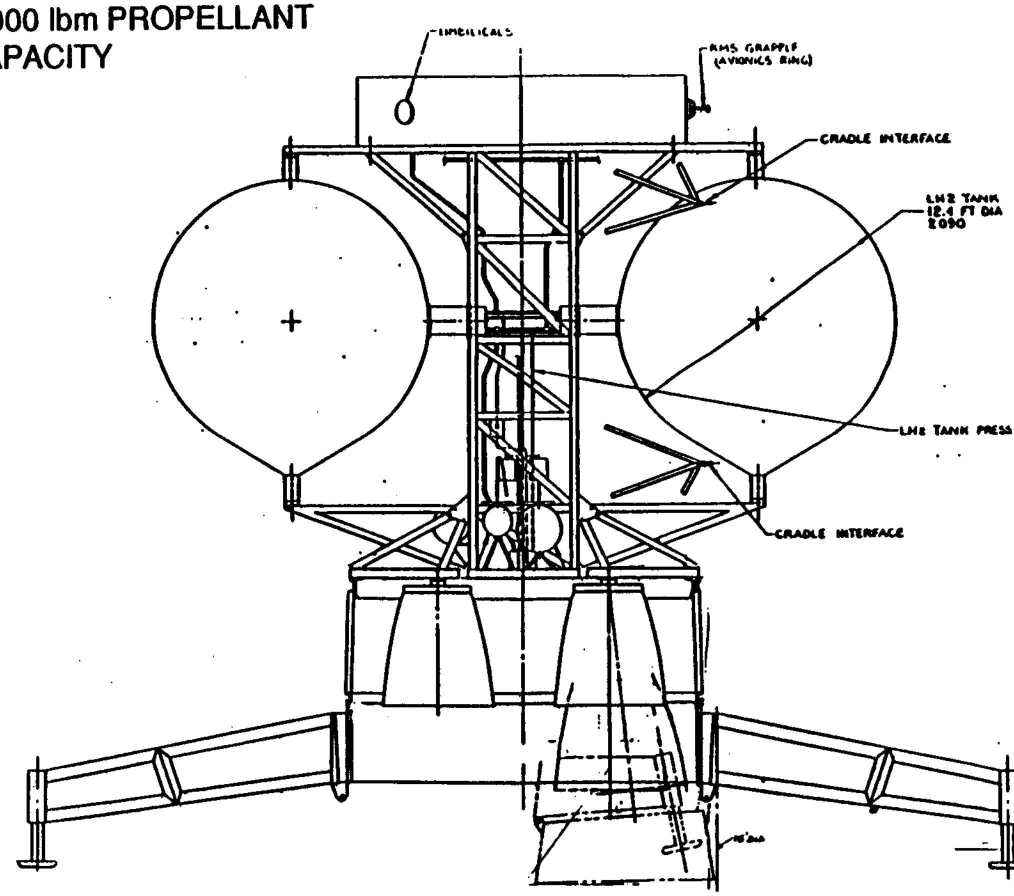
A Lunar lander concept was sized for the purpose of remaining in Lunar orbit and delivering to the surface the payload that the 98 Klbm vehicle could deliver to Lunar orbit. In other words the dedicated lander would be placed into Lunar orbit and serviced there (or perhaps on the surface) for use in 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.

DEDICATED LUNAR LANDER

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)

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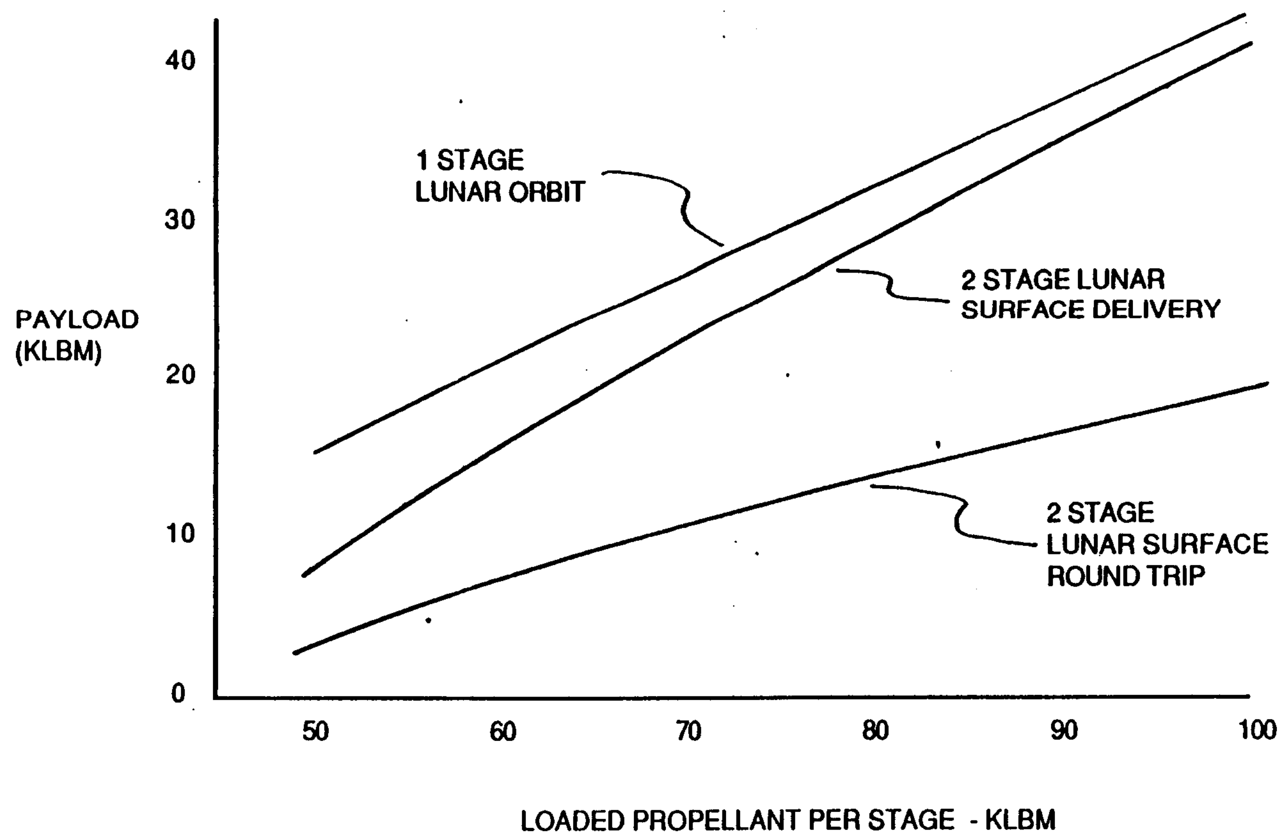
LUNAR OTV PERFORMANCE

Performance parametrics for the 98 klbm transfer vehicle and 98 klbm lander are shown in the figure. 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 using one transfer vehicle and one landing vehicle. 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. The third case is for delivery capability of one 98 klbm transfer vehicle from LEO to Lunar orbit.

LUNAR OTV PERFORMANCE

NOTE: SPECIFIC IMPULSE = 475 SEC



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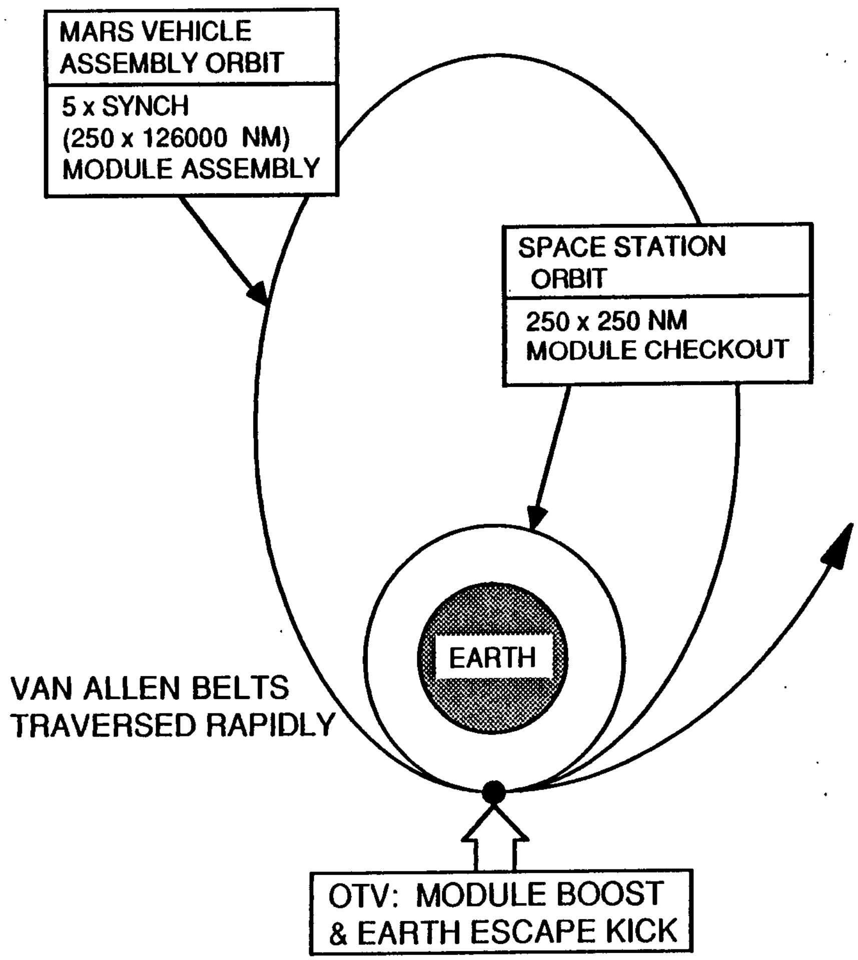
MANNED MARS MISSION LOGISTICS SUPPORT

The flight of a manned Mars mission will involve some extremely large spacecraft which are generally thought to require kick stages much larger than current OTV class. The need for such large stages is based on a direct departure from low Earth orbit. Because new boost stages will represent substantial development costs it is worthwhile to see whether existing OTV-class vehicles could be utilized instead. Shown on this chart is 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 which would be assembled into the 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 here 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. 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. The radiation doses do not appear to represent a major risk for a craft designed for deep space operations, though a more detailed assessment of this factor must await further studies.

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 kick stage need not be developed to enable a manned mars mission.

MANNED MARS MISSION LOGISTICS SUPPORT



OTV APPLICATION TO BUILDUP & BOOST OF MANNED MARS SPACECRAFT

5xSYNCH ELLIPTICAL STAGING ORBIT TO MAXIMIZE ENERGY OF ASSEMBLED MMV

1) CHECKOUT OF MODULES IN LOW ORBIT

2) OTV BOOST OF MODULES TO 5xSYNCH

3) ASSEMBLE MODULES IN 5xSYNCH

4) EXPENDABLE OTV GIVES ESCAPE KICK

OTV PERFORMANCE (74K SPACE BASED OTV)

STATION TO 5xSYNCH: 60600 LB

EXPENDABLE OTV:

- 5xSYNCH TO C3= 5: 499400 LB
- 5xSYNCH TO C3=10: 354300 LB
- 5xSYNCH TO C3=20: 218900 LB
- 5xSYNCH TO C3=50: 92800 LB

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OTV GROWTH SUMMARY

BASE SCENARIO 1

LOW TRAFFIC

NO OTV GROWTH, EXPENDABLE ONLY

EARTH INITIATIVE

MODERATE TRAFFIC, ROUND TRIP REQUIREMENT

DEVELOP IOC ENGINE & AEROASSIST

UNMANNED PLANETARY

LOW TRAFFIC

NO OTV GROWTH, EXPENDABLE ONLY

LUNAR INITIATIVE

HIGH TRAFFIC, ROUND TRIP & LANDING REQUIREMENTS

FULL DEVELOPMENT PROGRAM

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ACC OTV SAFETY ISSUES

DACC COMPOSITE SHROUD

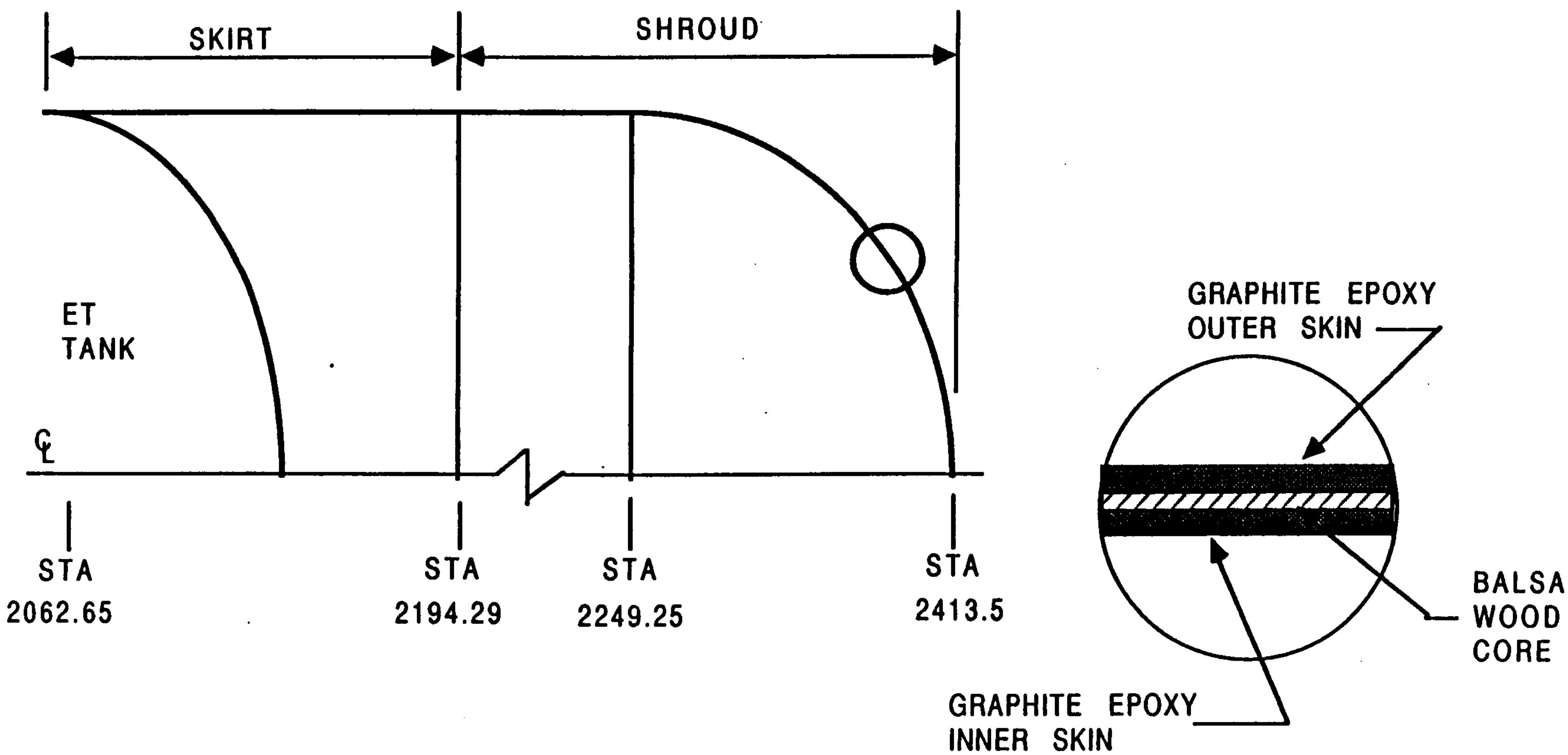
In the baseline design, the skins will be a sandwich structure. The inner and outer skins will be filament wound AS4W-12K graphite fiber using HBRF 55A epoxy resin. This composite will have 50% fiber by volume. The lamina properties for this composite are: the modulus in the fiber direction is 17.21×10^6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

The baseline 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 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.

DACC COMPOSITE SHROUD



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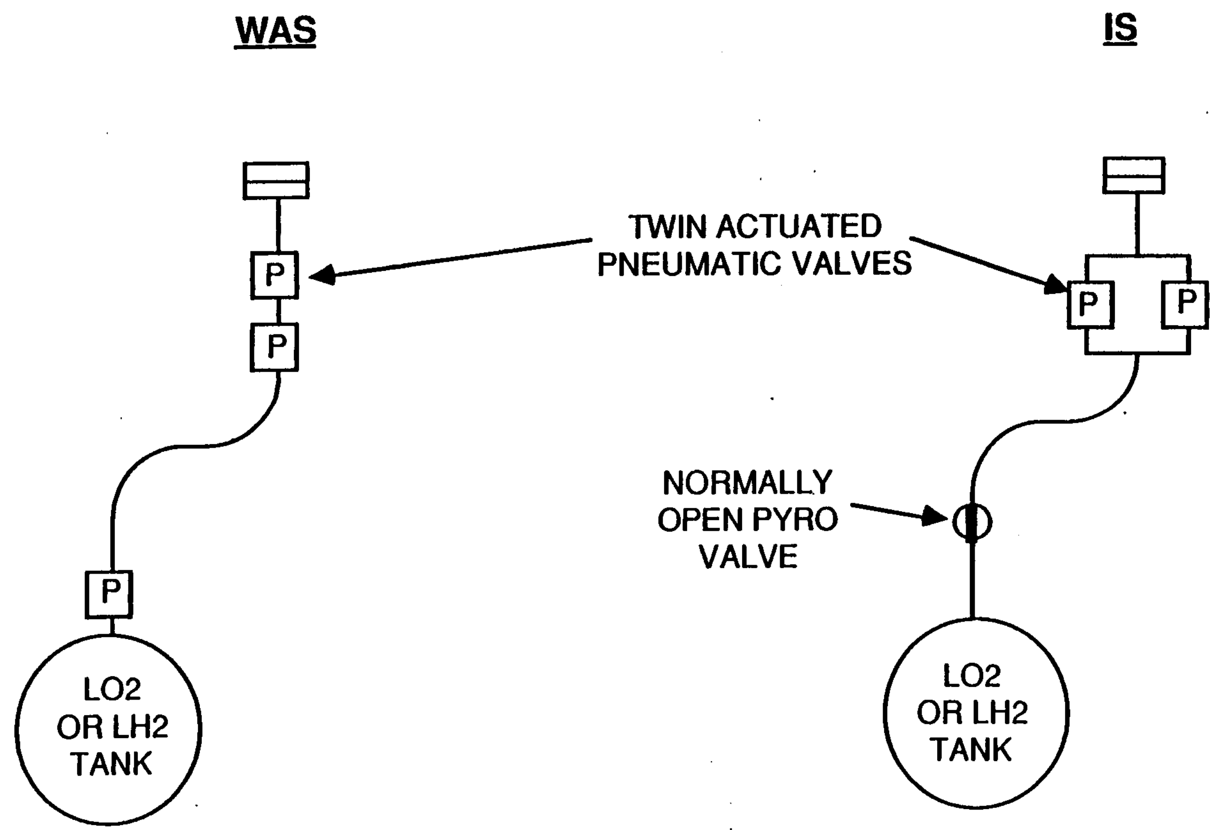
MANNED SPACE SYSTEMS

ASCENT VENT REDUNDANCY

The previous design of the ascent vent system consisted of three pneumatically actuated valves located in series in order to provide three inhibits downstream of the propellant tank. The pneumatic valves are intended to have twin actuators so that each valve can have one failure and continue to operate. The problem with the previous system design, however, is that if a valve was to have two failures, the vent system would not function, therefore creating a catastrophic condition of not relieving tank pressure.

The updated design cures this problem with parallel pneumatic valves to provide for venting control and a single pyro actuated valve with twin initiators. This system provides for two fault tolerance in the venting system as well as three inhibits for preventing loss of propellant from the tanks.

ASCENT VENT REDUNDANCY



ACC OTV PROX OPS SAFETY SEQUENCE

A unique concern to an ACC OTV is vehicle safing for Shuttle rendezvous and payload mate. This figure 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 desireable 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. Analysis shows a capability for 6 hours of no-vent if the OTV tanks are first reduced to 16 psi. This will eliminate undesireable gaseous venting during the time the two vehicles are in collision range.

The final system to be safed will be the OTV Attitude Control System (ACS). The range at which this must be done is uncertain at present, it would be desireable to wait until as late as possible to reduce residual attitude rate disturbances.

ACC OTV PROX OPS SAFETY SEQUENCE

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_20 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

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CONCLUSIONS

Within the constraints of this study, two major conclusions were reached.

The first is that there were no potential show stoppers identified for the ACC OTV concept and there is a potential show stopper with the cargo bay configuration (the need to dump would be directly opposing the controls to prevent premature dumping).

Secondly, it was concluded that the ACC OTV has definite safety advantages over the cargo bay configuration:

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).

The two medium risk items associated with the ACC configuration are not show-stoppers. The tank separation valve concern could be eliminated completely with other concepts. This leaves only the potential new destruct system as both a technical and additional safety risk. The safety risk associated with this system should be made to be acceptable since history in designing these systems exist.

JSC was 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 this is not prohibited but "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 JSC safety panel members contacted said they havew not seen a design that meets all of the requirements but would not project that it could not ever be done. A system requiring pressure for structural integrity would be the biggest challange and is probably not do-able without major safety compromises.

CONCLUSIONS

- THE ACC OTV CONCEPT CAN MOST LIKELY BE MADE TO MEET THE CURRENT REQUIREMENTS
    - NO SHOW STOPPERS SEEN
    - SAFETY ADVANTAGES OVER IN-BAY APPROACH
    - NEED TO DELETE TANK SEPARATION VALVES OR FIND DIFFERENT APPROACH

- THE CARGO BAY CONFIGURATION HAS POTENTIAL SHOWSTOPPERS

- NEED TO DUMP PRIOR TO RETURN WOULD REQUIRE A SYSTEM THAT IS BOTH TWO-FAILURE TOLERANT TO PREMATURE DUMPING AND AGAINST FAILURE TO
    - NO DESIGN HAS ACCOMPLISHED THIS
    - NEED TO DUMP IS TBD - WOULD BE DRIVEN BY:
        - NEED TO CHANGE CG; OR,
        - NEED TO DECREASE WEIGHT; OR,
        - NEED TO ELIMINATE CRYOS IF ALL LANDING SAFETY ISSUES ARE NOT SOLVED (FURTHER ANALYSIS REQUIRED)
    - EXTENSIVE MODIFICATIONS TO THE ORBITER REQUIRED PER JSC (VENTING)
    - "ALL THE CENTAUR ISSUES MUST BE RESOLVED" PER JSC

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AEROASSIST

AEROASSIST CLASSES

Several different classes of entries have been studied in the course of this contract as is summarized in this figure. Earth return missions utilize aeroassist to reduce the energy of an existing elliptical Earth orbit. The desired end condition is 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. Cases consistant with return from Mars have been investigated with encounter C_3's ranging from 8.0 to 68 km^2/sec^2. The third class of missions are those of Mars capture. 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).

AEROASSIST CLASSES

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

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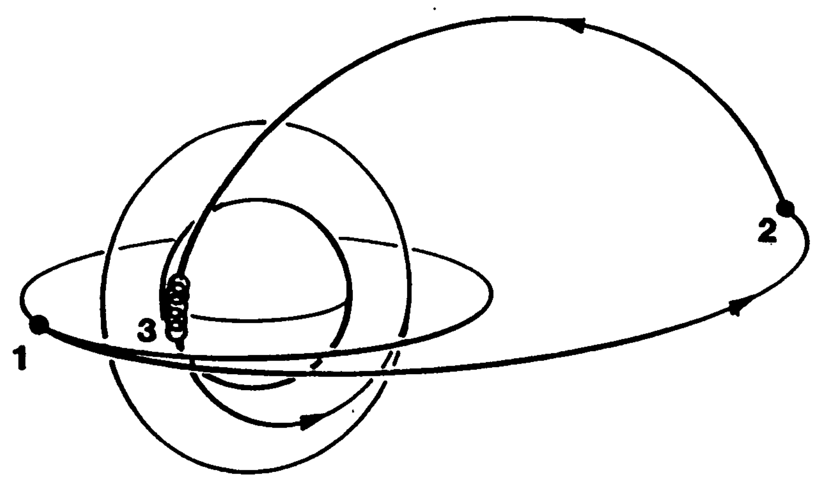
LARGE INCLINATION TURNS VIA AEROASSIST

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. Thus the all-propulsive approach is to use 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 a low circular one 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. The same strategy 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, sensitive payloads may require a thermal shroud for the aero phase. A small circularization burn is performed after leaving the atmosphere, typically 250-450 fps depending on the final altitude desired.

The next chart 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.

LARGE INCLINATION CHANGES VIA AEROASSIST



- USE OF AEROASSIST IN PLANE CHANGES

  1) BOOST APOGEE VIA ROCKET BURN

  2) PERFORM INCLIN CHANGE AT APOGEE WHERE VELOCITY IS LOW

  3) UTILIZE AEROASSIST AT PERIGEE TO REDUCE APOGEE (NO PLANE CHANGE IN AERO)

- SIGNIFICANT ΔV SAVINGS OVER ALL-PROPULSIVE FOR $\Delta INC > 25°$

- PAYLOAD PROTECTION CANISTER MAY BE REQUIRED DURING AERO

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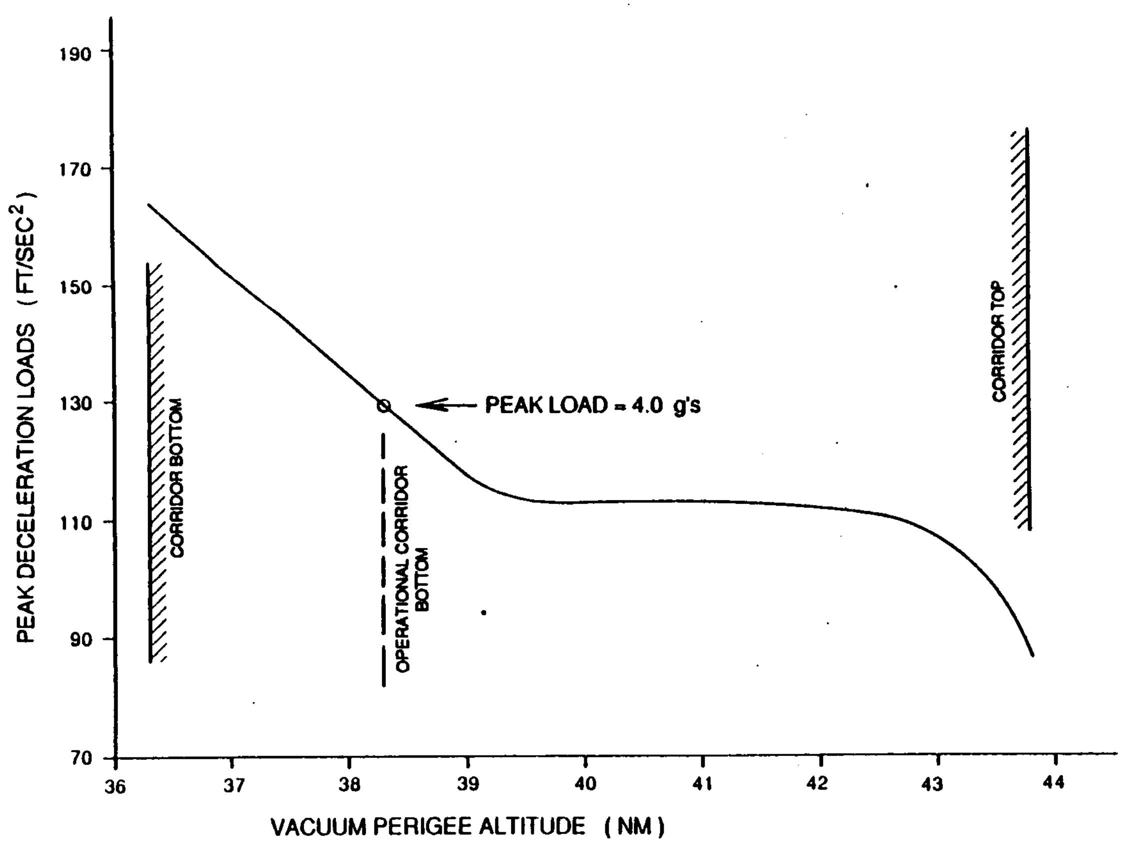
LUNAR LOADs, L/D = 0.14

This chart 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 utilizing the upper 5.5 nm for flight, peak loads are reduced to 4.0 g's.

LUNAR LOADS, g-RELIEF: L/D = 0.14



- 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

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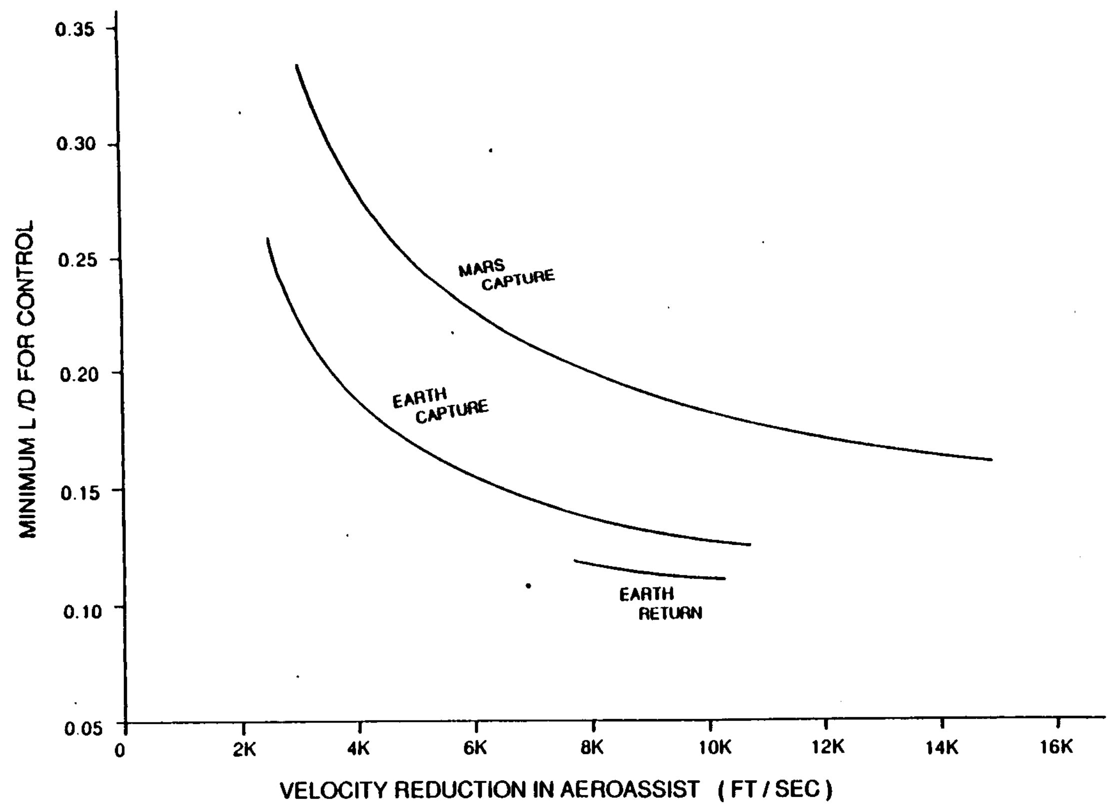
MINIMUM L/D REQUIREMENTS FOR AEROASSIST

This figure shows the decreasing L/D requirements for increasingly energetic aeroassist maneuvers. As the three previous charts 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 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.

MINIMUM L/D REQUIREMENTS FOR AEROASSIST



- L/D REQMTS VS AERO DELTA-V
- EARTH RETURN
  EXIT APOGEE = 245 NM
- EARTH CAPTURE
  EXIT APOGEE = 38485 NM
- MARS CAPTURE
  EXIT APOGEE = 18108 NM
- CONTROL REQUIREMENTS
  FROM ERROR ANALYSIS

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PROGRAM SUMMARY

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EXTENSION II RESULTS

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INITIAL PROGRAM

- INITIALLY EXPENDABLE VEHICLE CAN GET PROGRAM STARTED
- STS / ACC ALLOWS FLIGHTS TO BEGIN WHILE LCV BEING DEVELOPED

ADVANCED MISSIONS

- LUNAR LANDER CAN BE ADAPTED FROM SPACE BASED OTV
- 98K PROPELLANT CAPACITY REQD. FOR CSLI LUNAR INITIATIVE

ACC SAFETY

- NO SHOW STOPPERS FOR ACC OTV
- ACC SAFEST LOCATION FOR STS BOOST

AEROASSIST

- USE LOAD RELIEF FOR LUNAR RETURNS

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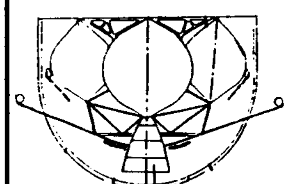
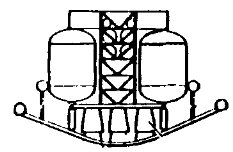
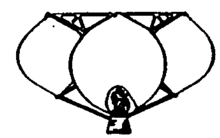
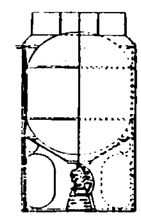
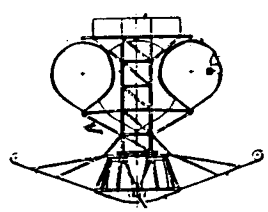
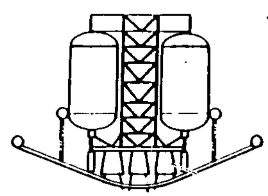
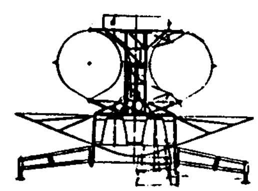
OTV STUDY SUMMARY

	INITIAL CONTRACT	EXTENSION #1	EXTENSION #2	
LAUNCH VEHICLE	STS	LARGE DIAMETER CARGO VEHICLE (LCV)	STS + LCV	SHUTTLE C
INITIAL OTV	 ACC BASED	 GOOD GROWTH STS RETURN	 ACC BASED INITIALLY GROW TO LCV	 EXPENDABLE VOLUME EFFICIENT
GROWTH OTV	 STS DELIVERY SPACE BASED	 SPACE BASED	 STS OR LCV DELIVERY SPACE BASED	? MAJOR BLOCK CHANGE

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OTV FOR ADVANCED MISSIONS

OTV IS NEEDED FOR NASA'S ADVANCED MISSIONS (CIVIL SPACE LEADERSHIP INITIATIVE)

EARTH INITIATIVE:	LOW THRUST DELIVERY OF LARGE SPACE STRUCTURES
	SATELLITE SERVICING MISSIONS (ROUND TRIP)
	TWO-WAY LOGISTICS SUPPORT FOR HIGH ALTITUDE / POLAR ORBITS
LUNAR INITIATIVE:	HEAVY LIFT CAPABILITY FOR LUNAR ORBIT / SURFACE BASE
	EFFICIENT RETURN USING AEROASSIST
	LOGICAL GROWTH TO LANDING CAPABILITY
MANNED PLANET.:	HEAVY UNMANNED PRECURSOR MISSION DELIVERY
	AEROASSIST TECHNOLOGY EXTENDED TO PLANETARY AEROCAPTURE
	BUILD-UP & BOOST OF MANNED PLANETARY CRAFT

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CAPABILITY ENHANCEMENTS FROM OTV

- HIGH PERFORMANCE, GROWTH-ORIENTED CRYO SYSTEM
  - DUAL COMPATABILITY WITH STS & LCV
  - LOW-TECH EXPENDABLE CAN DELIVER 12.2K TO GEO (WITH 55K STS)
  - ALTERNATIVE TO CENTAUR GIVES ASSURED ACCESS
  - ADVANCED UPPER STAGE ENABLING FOR CSLI MISSIONS
- RE-USABLE SYSTEM
  - PAYLOAD RETRIEVAL CAPABILITY
  - LOWER COSTS THROUGH RE-USE
- AEROASSIST
  - EFFICIENT RETURN CAPABILITY
  - LARGE INCLINATION TURNS
  - AERO-TESTBED VEHICLE

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TECHNOLOGY ENHANCEMENTS

- AEROASSIST
- NEW HIGH PERFORMANCE SPACE CRYO ENGINE
- ADVANCED MATERIALS (COMPOSITES, AL-LI, ETC)
- ADAPTABLE SOFTWARE REDUCES RECURRING COSTS
- EFFICIENT TURNAROUND TECHNIQUES (GROUND & SPACE BASING)
  - VEHICLE SELF-CHECKOUT & DIAGNOSTICS
  - SUBSYSTEM HEALTH MONITORING (ENGINES, MECHANISMS, ETC)
  - ROBOTIC CHANGEOUT OF HARDWARE

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SUMMARY

- INITIALLY EXPENDABLE OTV
  - REDUCES PROGRAM STARTUP COSTS
  - ALLOWS EARLIER OTV START (FILL NEAR-TERM NEEDS)
  - PHASED GROWTH TO RE-USABLE VEHICLE

- LAUNCH MODE
  - o LCV BOOST
    - MORE MISSIONS CAN BE FLOWN INTACT
    - RETURN TO GROUND MAY BE DIFFICULT
  - o STS BOOST
    - ACC IS BEST LOCATION (SAFETY)
    - VIABILITY STRONGLY DRIVEN BY LCV AVAILABILITY DATE

- ADVANCED MISSIONS
  - ADVANCED CSLI MISSIONS REQUIRE OTV
  - LUNAR LOGISTICS IS THE PRIMARY DRIVER
  - TWO 98KLB STAGES CAN DELIVER 40KLB TO LUNAR SURFACE

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PROGRAM & MISSION OPTIONS

- DRIVER MISSIONS
- OTV PHASED GROWTH
- GEO SERVICING
- LARGE INCLINATION CHANGES
- LUNAR MISSION PROFILES
- MANNED PLANETARY SUPPORT
- ACC OTV SAFETY

DRIVER MISSIONS

This figure summarizes the current set of driver missions supplied by MSFC. These are used as driver missions and do not represent a total mission model. The baseline scenario is NASA's Civil Needs Data Base (CNDB) Rev. 2, Scenario #1. This represents a conservative growth plan for future missions. This baseline scenario is used by itself and as a core with additions from the three new 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 mission options.

DRIVER MISSIONS

	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 (MAN) 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

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MISSION CAPTURE - CORE

The table lists the driver missions for the core mission model. The required propellant quantities are shown for ground based expendable and reusable OTV concepts -- both intended to be launched via the STS. As noted, the expendable vehicle propellant quantities are with respect to use of a RL10A-3 (existing, 440 sec) engine and the aeroassisted reusable vehicle concept propellant quantities are with respect to use of the IOC (475 sec) engine. Cases where a different engine's performance and weight were used in the propellant calculation are also provided with the engine identified in parenthesis. The propellant quantities that exceed the capability of the STS (with OTV and ASE weights included) are noted.

MISSION CAPTURE - CORE

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 47.5 K STS LIFT	27 KLBM 45.3 K STS LIFT	31 KLBM RACK EXPENDED 50.5 K STS LIFT	33 KLBM 55.7 K STS LIFT	49 KLBM 58.5 K STS LOAD FOR OTV ONLY
STS LAUNCH, AEROBRAKE, REUSABLE,	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

PROGRAM IMPROVEMENTS AND MISSION CAPTURE

Propellant quantities are given in the table for each of the three space initiatives in addition to the core mission model. Program improvements are required in order to accommodate the various 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 in parenthesis.

PROGRAM IMPROVEMENTS AND MISSION CAPTURE

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

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GROWTH PATH DEVELOPMENT PROGRAMS - GROUPED

The table shows groupings of subsystem developments which correspond to overall vehicle improvements that are required by the various missions. These groupings were arrived at by attempting to minimize the program schedule and cost impacts with attention given to preserving vehicle performance and flexibility at each step in the evolution. 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.

GROWTH PATH DEVELOPMENT PROGRAMS - GROUPED

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

OTV PHASED GROWTH - BASE SCENARIO

This figure 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 one to two per year. The OTV program begins as early as 1993 with a low-tech., low-cost expendable vehicle. As will be discussed in the Design Issues section, flight rates of 5 to 6 per year are required to justify major system upgrades such as IOC engine and aeroassist. Thus, this low flight rate scenario does not justify major OTV program improvements and the OTV would remain an expendable vehicle through 2010.

OTV PHASED GROWTH - BASE SCENARIO

INITIAL PROGRAM	IOC ENGINE	AERO-BRAKE	MAN-RATING	SPACE BASING	LUNAR MISSIONS
• EXPENDABLE • RL-10 (ISP=440)	• IMPROVED PERFORMANCE (ISP=475)	• RE-USE • RETURN P/L'S	• REDUNDANCY • DUAL ENGINES	• ON-ORBIT REPAIR & RESUPPLY	• LANDING LEGS • MULT ENGINES • RADAR SYST

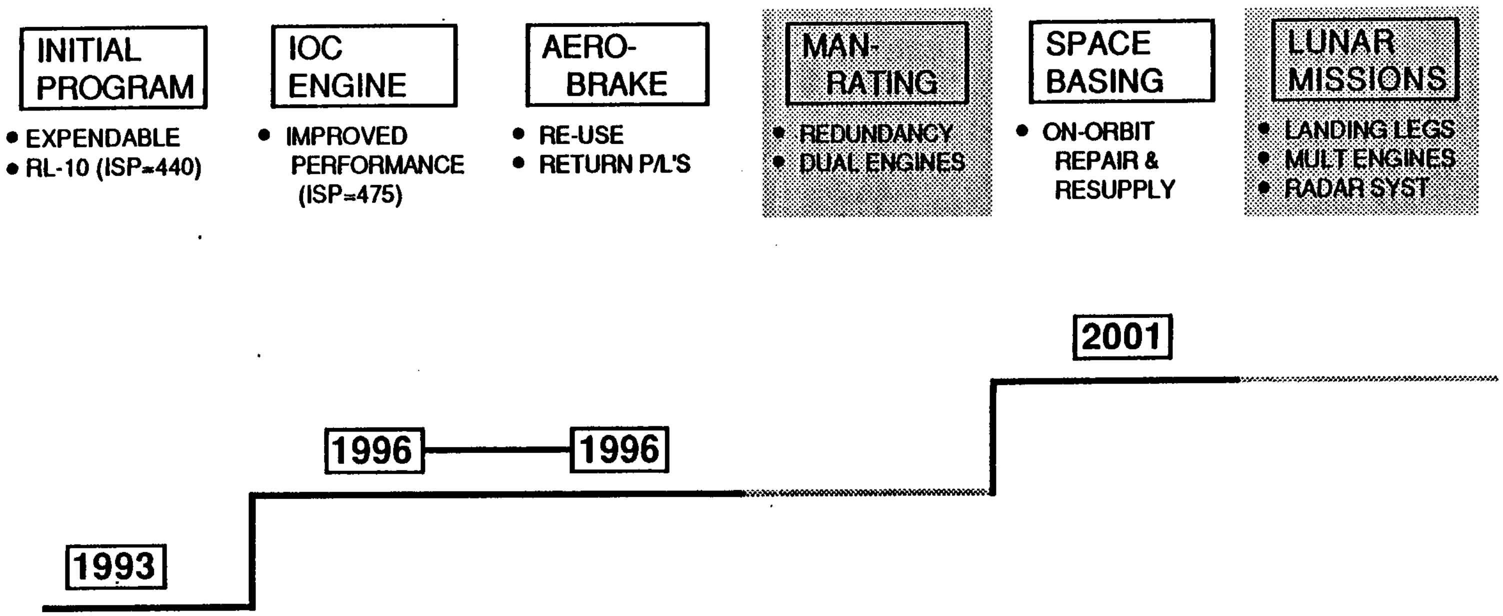
1993

CONSTRAINED CNDB SCENARIO-1 DOES NOT JUSTIFY GROWTH

OTV PHASED GROWTH - EARTH INITIATIVE

This figure summarizes the OTV growth plan for the CLSI 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, the OTV IOC engine will need to be used, rather than the RL-10, because it has low thrust capability. Since aeroassist will be required down the road 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 demanding propellant requirements. The GEO servicing mission in 2001 can only be accomplished by a large space-based OTV since it requires in excess of 68klb of propellant. Thus, the OTV moves to space basing capability in 2001, which is probably about the earliest that it could be available.

OTV PHASED GROWTH - EARTH INITIATIVE



EARTH INITIATIVE JUSTIFIES IOC ENGINE, AEROASSIST, AND SPACE BASING

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OTV PHASED GROWTH - UNMANNED PLANETARY INITIATIVE

This figure 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 110 km^2/sec^2. A C_3 of up to 80 km^2/sec^2 can be accomodated by the 50klb propellant capacity OTV in an expendable mode. This is the largest vehicle that can be boosted by the current shuttle. If a large cargo vehicle is employed to deliver a 62klb propellant capacity OTV, the full 110 C_3 can be accomodated.

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.

OTV PHASED GROWTH - UNMANNED PLANETARY INITIATIVE

INITIAL PROGRAM	IOC ENGINE	AERO-BRAKE	MAN-RATING	SPACE BASING	LUNAR MISSIONS
• EXPENDABLE • RL-10 (ISP=440)	• IMPROVED PERFORMANCE (ISP=475)	• RE-USE • RETURN P/L'S	• REDUNDANCY • DUAL ENGINES	• ON-ORBIT REPAIR & RESUPPLY	• LANDING LEGS • MULT ENGINES • RADAR SYST

1993

UNMANNED PLANETARY INITIATIVE DOES NOT JUSTIFY GROWTH

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. High traffic rates beginning in the year 2000 will more than justify IOC engine and aeroassist 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 stage (98klb) which must be space based. Finally, landing on the moon requires significant upgrade of the OTV systems (landing legs, engines, avionics, etc) as is spelled out in the Design Issues section.

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 the completion of program upgrades at the year 2000. It is felt that 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 growth to the 2000 targets. A small landing mission in 1997 could be accomplished by a ground based 50klb capacity OTV in an expendable mode.

C-2

OTV PHASED GROWTH - LUNAR INITIATIVE

INITIAL PROGRAM	IOC ENGINE	AERO-BRAKE	MAN-RATING	SPACE BASING	LUNAR LANDINGS
• EXPENDABLE • RL-10 (ISP=440)	• IMPROVED PERFORMANCE (ISP=475)	• RE-USE • RETURN P/L'S	• REDUNDANCY • DUAL ENGINES	• ON-ORBIT REPAIR & RESUPPLY	• LANDING LEGS • MULT ENGINES • RADAR SYST
1993	1996	1996	2000	2000	2000

LUNAR INITIATIVE JUSTIFIES ALL OPTIONS ON AGGRESIVE SCHEDULE

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OTV GROWTH SUMMARY

BASE SCENARIO 1

- LOW TRAFFIC
- EXPENDABLE OTV ONLY

EARTH INITIATIVE

- MODERATE TRAFFIC, ROUND TRIP REQUIREMENT
- DEVELOP IOC ENGINE & AEROASSIST

UNMANNED PLANETARY

- LOW TRAFFIC
- EXPENDABLE OTV ONLY

LUNAR INITIATIVE

- HIGH TRAFFIC, ROUND TRIP & LANDING REQUIREMENTS
- FULL DEVELOPMENT PROGRAM

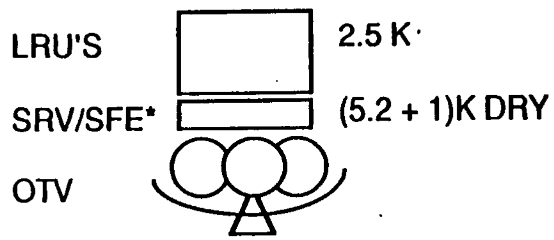
GEO SERVICING CONFIGURATIONS FOR OTV

Two cases for GEO servicing were investigated for OTV. The first was for delivering LRU's (line replaceable units) 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 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. Therefore the options studied include the use of either an OMV (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. The second option requires only one flight with the OTV performing the on-orbit maneuvers. It appears that if the OTV can be made capable of performing the on-orbit maneuvers at reasonable weight impact (such as that shown) and development cost, the second option may be worth pursuing.

GEO SERVICING CONFIGURATIONS FOR OTV

CASE I - LOW DELTA V SERVICING



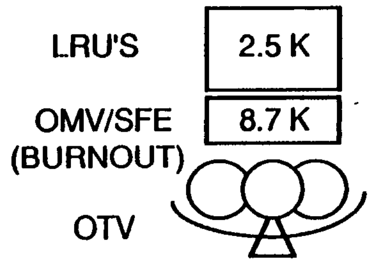
SRV SEPARATES FROM OTV TO PERFORM SERVICING

SRV -- 780 LBM MONOPROP. (500 FPS), 880 LBM LOADED (1180 MAX)

OTV -- 48 K PROPELLANT FOR ROUND TRIP

CASE II - 800 FPS SERVICING DELTA V

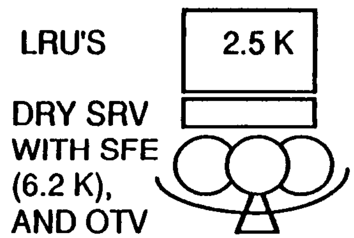
OPTION A



TRIP #1: DELIVER LRU'S AND OMV/SFE (950 LBM BI-PROP, OTV - 40 K PROP.)

TRIP #2: RETRIEVE LRU'S AND OMV/SFE (OTV - 36 K PROP.)

OPTION B



OTV PERFORMS DELIVERY, ALL SERVICING MANEUVERS, AND RETURN (50 K PROP.)

* SRV -- SHORT RANGE VEHICLE
SFE -- SMART FRONT END

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OTV WITH OMV SRV

OMV SHORT RANGE VEHICLE (SRV) CAN BE USED AS A RENDEZVOUS CAPABILITY KIT

OMV SRV CAN FLY INDEPENDENTLY OR BE LINKED TO OTV
(LATTER REQUIRES OTV TO SRV COMMAND LINK + OTV RCS COLD GAS MODS)

MISSION APPLICABILITY:

- SATELLITE SERVICING
- LARGE SPACE STRUCTURE ASSEMBLY
- SPACE MANUFACTURING PRODUCT RETRIEVAL
  (HIGH INCLINATION OR HIGH ENERGY ORBITS)
- SATELLITE INSPECTION
- MANNED MARS VEHICLE ASSEMBLY

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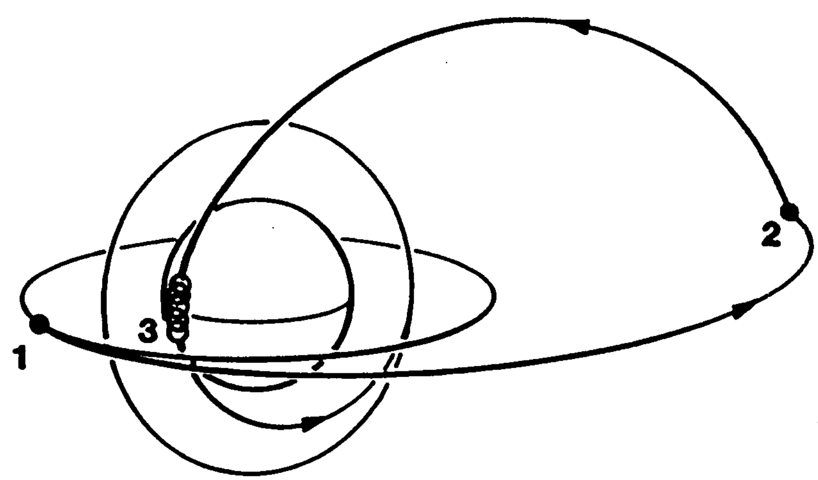
LARGE INCLINATION TURNS VIA AEROASSIST

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. Thus the all-propulsive approach is to use 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 a low circular one 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. The same strategy 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, sensitive payloads may require a thermal shroud for the aero phase. A small circularization burn is performed after leaving the atmosphere, typically 250-450 fps depending on the final altitude desired.

The next chart 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.

LARGE INCLINATION CHANGES VIA AEROASSIST



- USE OF AEROASSIST IN PLANE CHANGES
  1) BOOST APOGEE VIA ROCKET BURN
  2) PERFORM INCLIN CHANGE AT APOGEE WHERE VELOCITY IS LOW
  3) UTILIZE AEROASSIST AT PERIGEE TO REDUCE APOGEE (NO PLANE CHANGE IN AERO)
- SIGNIFICANT ΔV SAVINGS OVER ALL-PROPULSIVE FOR $\Delta INC > 25°$
- PAYLOAD PROTECTION CANISTER MAY BE REQUIRED DURING AERO

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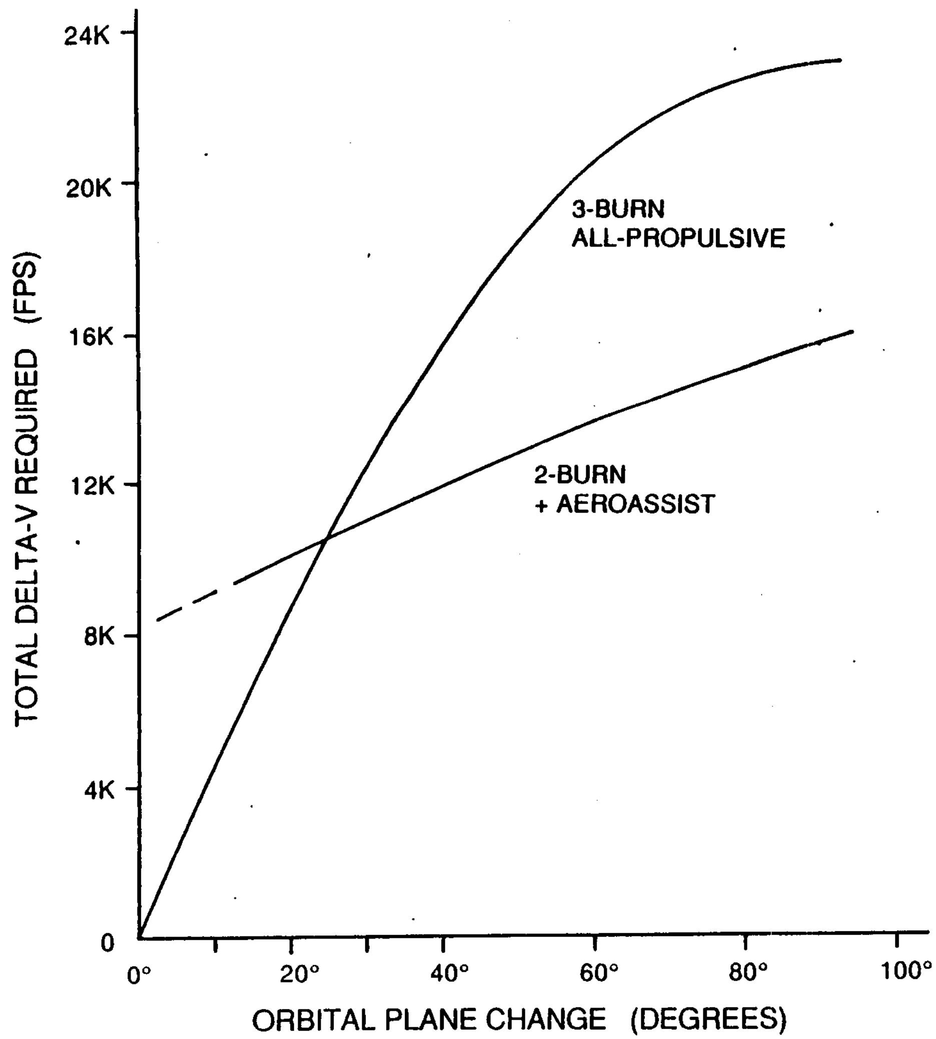
LARGE INCLINATION CHANGES VIA AEROASSIST - PERFORMANCE

This chart summarizes the performance of all-propulsive vs aeroassisted plane change maneuvers. See the previous facing page for a more detailed discussion.

LARGE INCLINATION CHANGES VIA AEROASSIST - PERFORMANCE



- COMPARE PERFORMANCE OF:

  3-BURN ALL-PROPULSIVE
  V.S.
  2-BURN USING AEROASSIST

- INITIAL & FINAL ORBIT = 270 NM
- MAXIMUM APOGEE = 20000 NM
- POST-AERO ΔV = 450 FPS

AEROASSIST EFFICIENT FOR PLANE CHANGES > 25°

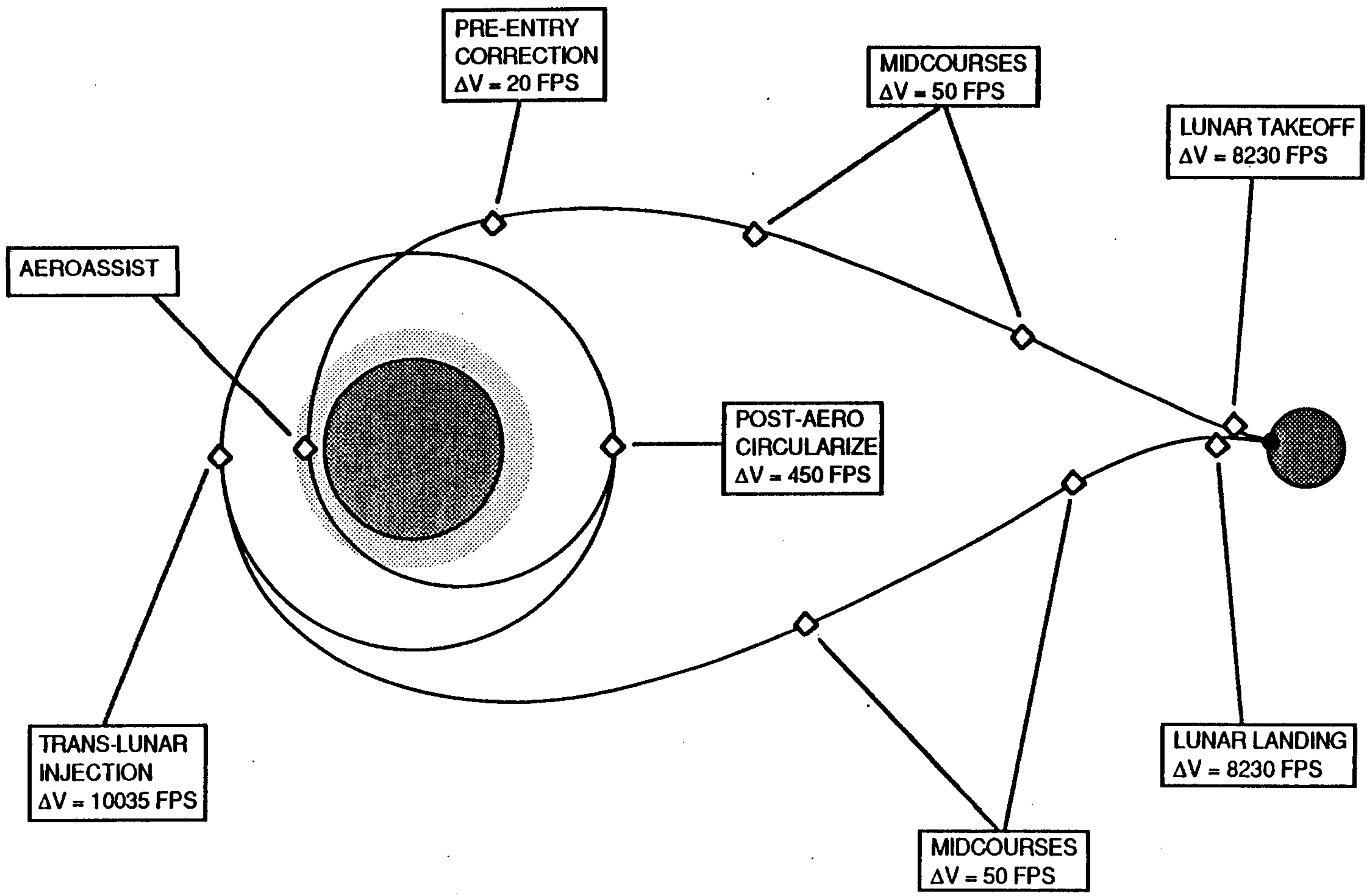
MARTIN MARIETTA

LUNAR PROFILE - DIRECT ASCENT

Various modes of lunar transfer were investigated for advanced missions. The first, shown here, is a direct transfer from low Earth orbit to the surface of the Moon followed by takeoff and direct injection into a trans-Earth trajectory. 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 Earth-moon flight. 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, with no assessment for gravity losses in descent.

LUNAR PROFILE - DIRECT ASCENT



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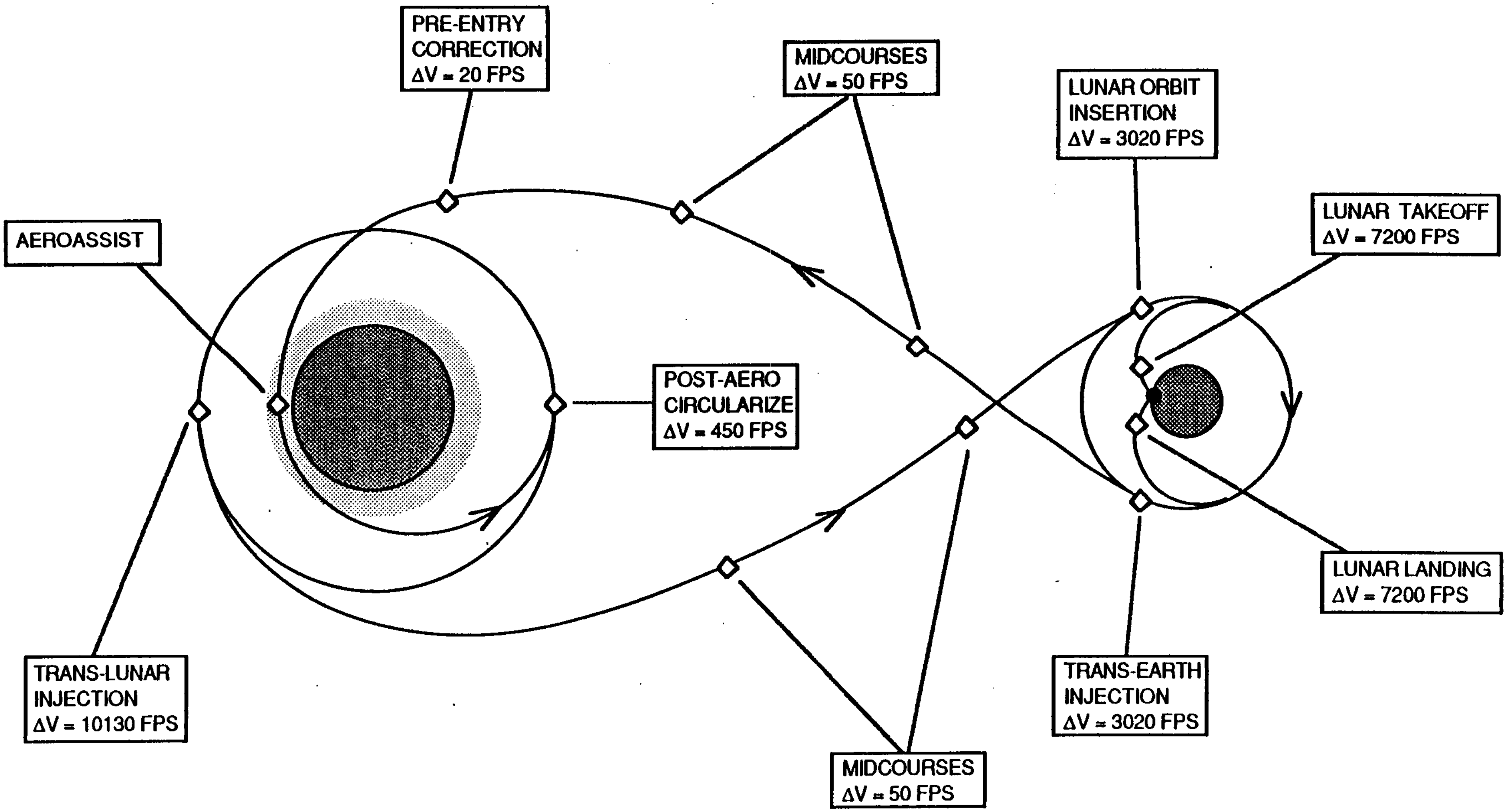
LUNAR PROFILE - LUNAR ORBIT

This Lunar profile uses an intermediate orbit about the Moon before descending to the surface. Velocities were derived from Apollo data and three-body integrated trajectories. These 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 Ascent 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.

LUNAR PROFILE - LUNAR ORBIT

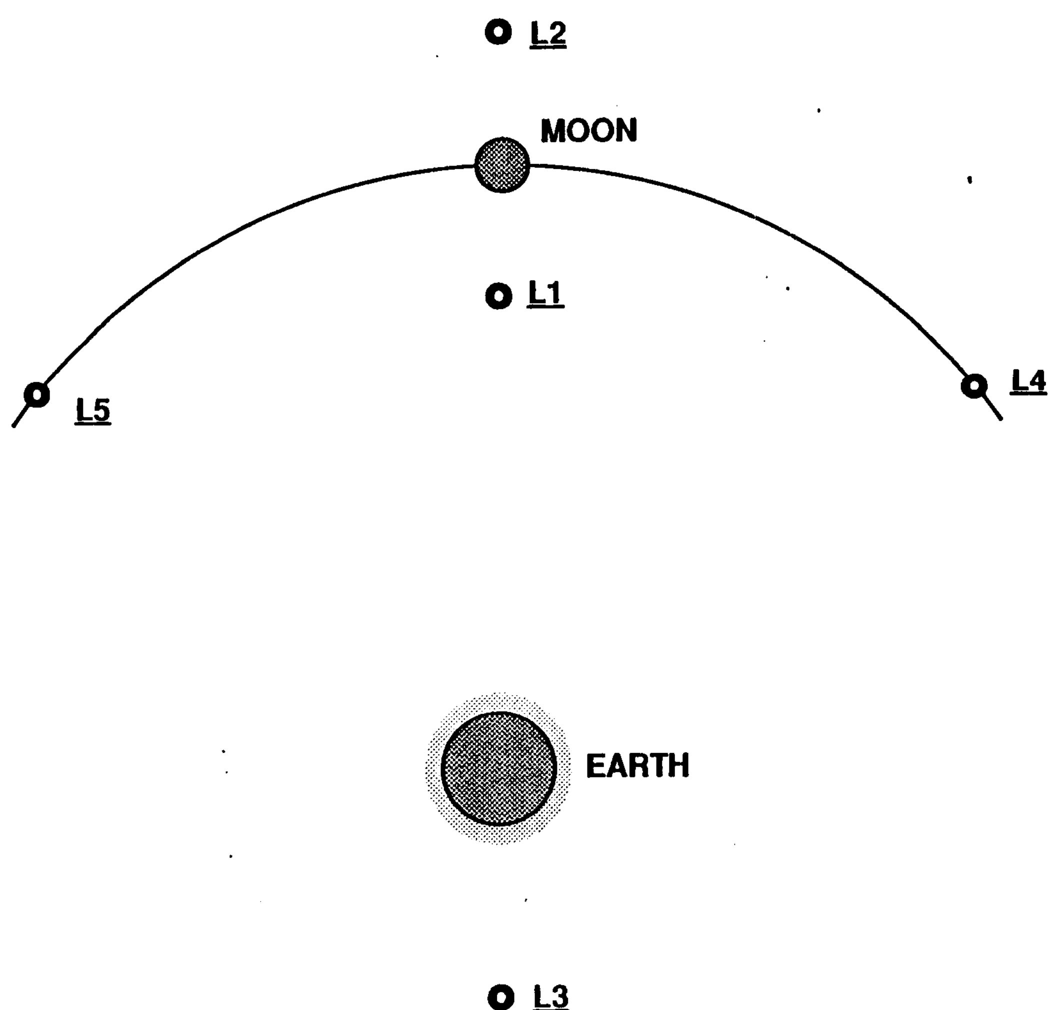


LUNAR LIBRATION POINTS

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 this figure and 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 gravitaional 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 go from the Earth to L1 and then to the Moon. These are summarized in the next chart.

LUNAR LIBRATION POINTS



- LIBRATION POINTS ARE GRAVITATIONALLY STABLE REGIONS
- L1 MAY REPRESENT ATTRACTIVE STATION LOCATION

  CLOSE TO EARTH & MOON

  GOOD COMMUNICATIONS

  HOVERS OVER NEAR SIDE

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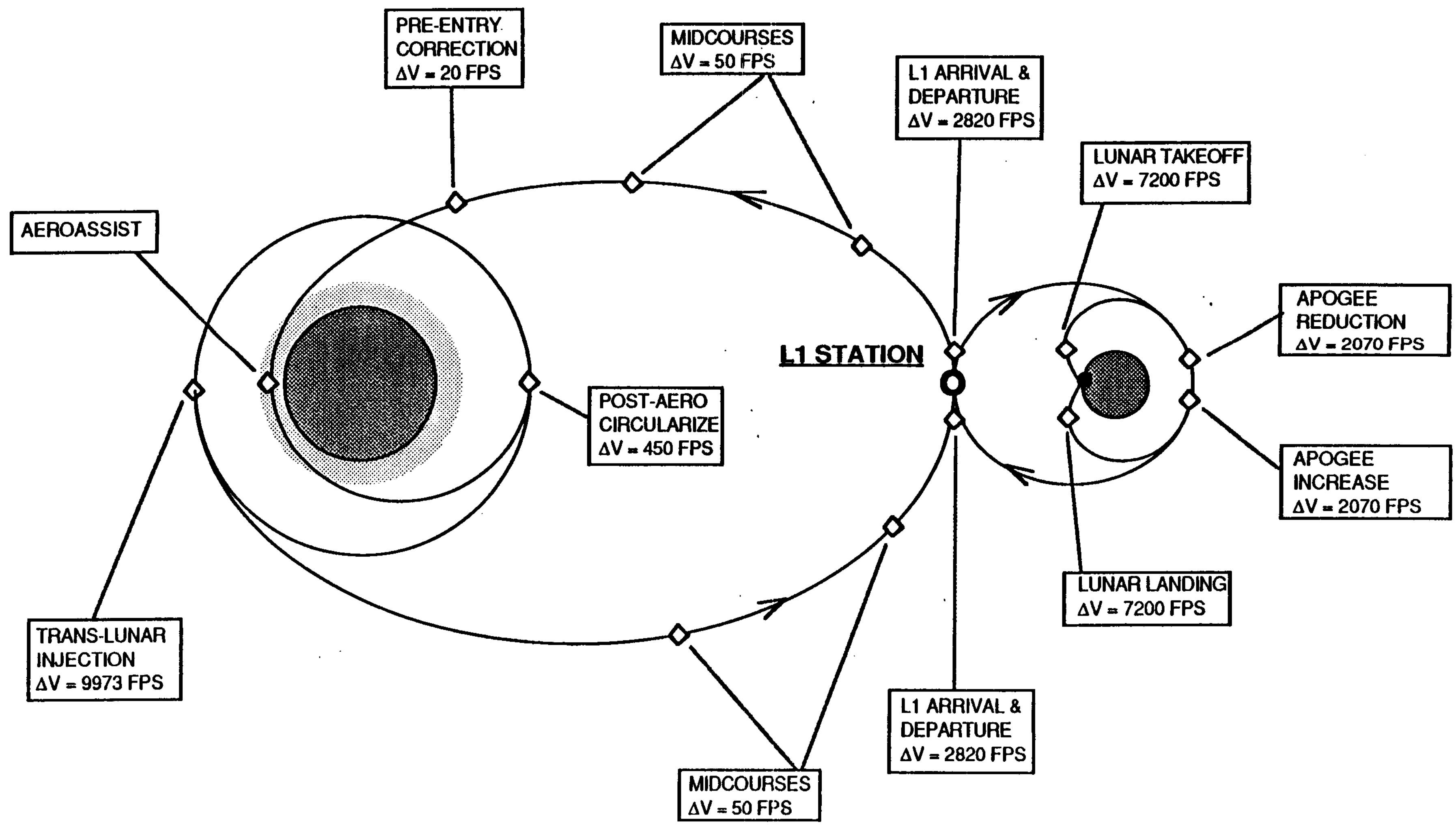
LUNAR PROFILE - L1 STATION

This Lunar profile utilizes 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.

LUNAR PROFILE - L1 STATION

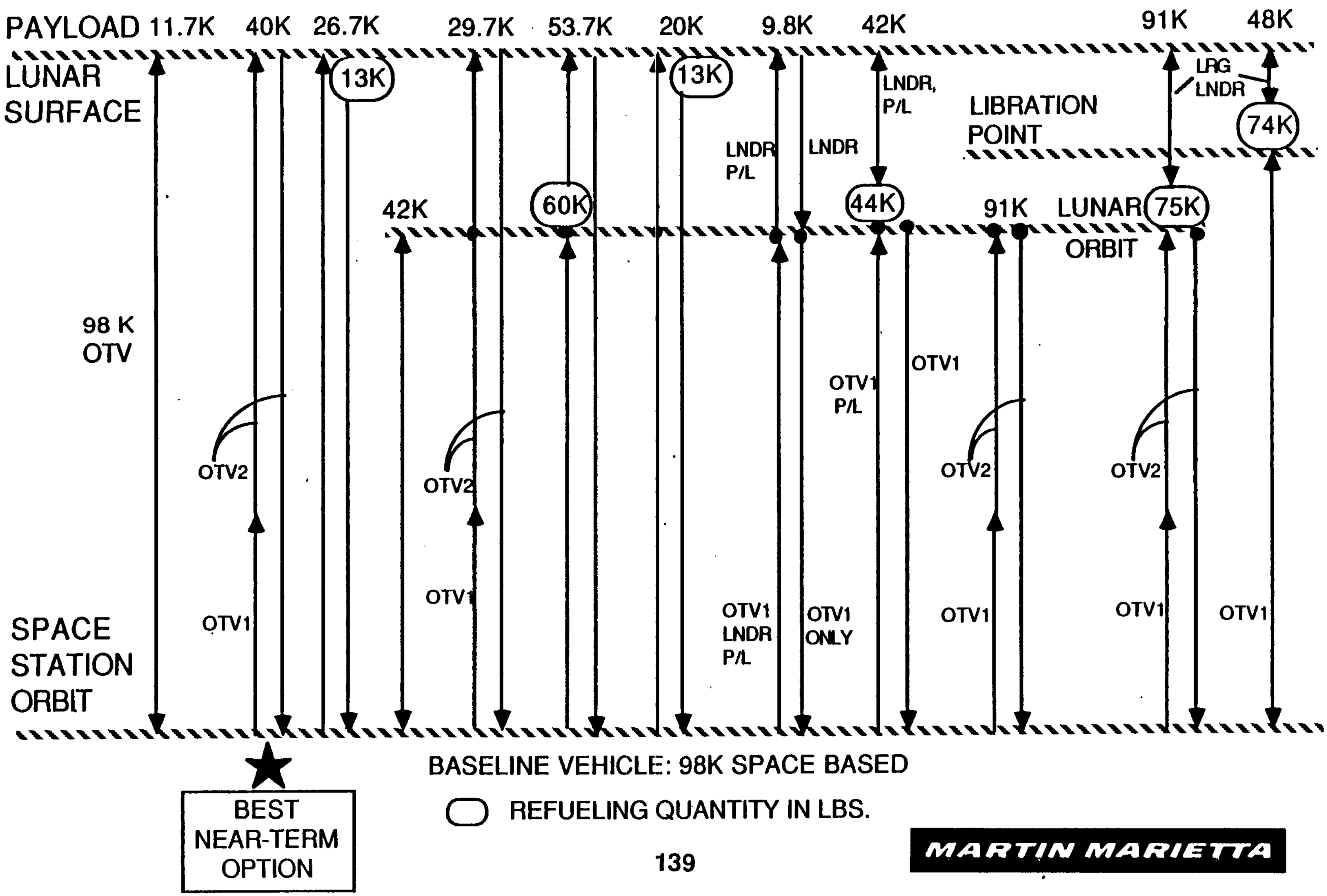


LUNAR DELIVERY OPTIONS

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 the figure along with the payload amounts to the surface that correspond to each of these options. 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 lander, a dedicated lander concept is shown with its function of delivering to the surface (from Lunar orbit or L1) a payload and then returning itself to its basing location.

LUNAR DELIVERY OPTIONS / PAYLOAD CAPABILITIES



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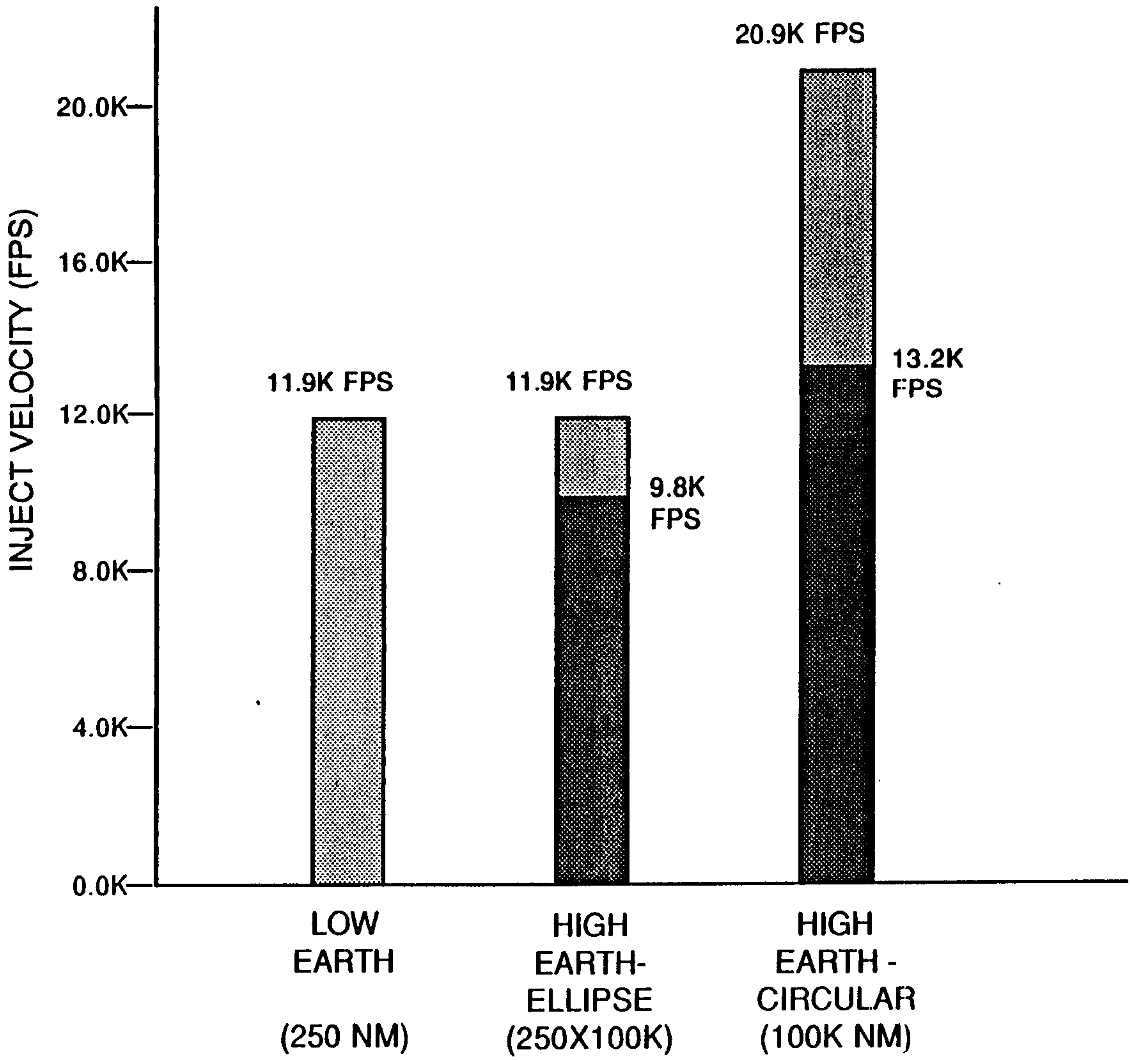
EARTH ESCAPE VELOCITIES

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 peices. This chart 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 the 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.

The next two approaches look at delivering the spacecraft in peices to an energetic assembly orbit. In this fashion, smallér 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. The second option looked at a high altitude circular assembly orbit. 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.

The use of an elliptical assembly orbit for large interplanetary craft appears to have significant benefits and will be explored further in the next chart.

EARTH ESCAPE VELOCITIES



- LARGE MARS CRAFT ESCAPE (C3=10)
- START AT 250 NM
- FIRST DELTA V TO ASSEMBLY ORBIT
- SECOND DELTA V TO ESCAPE

MARTIN MARIETTA

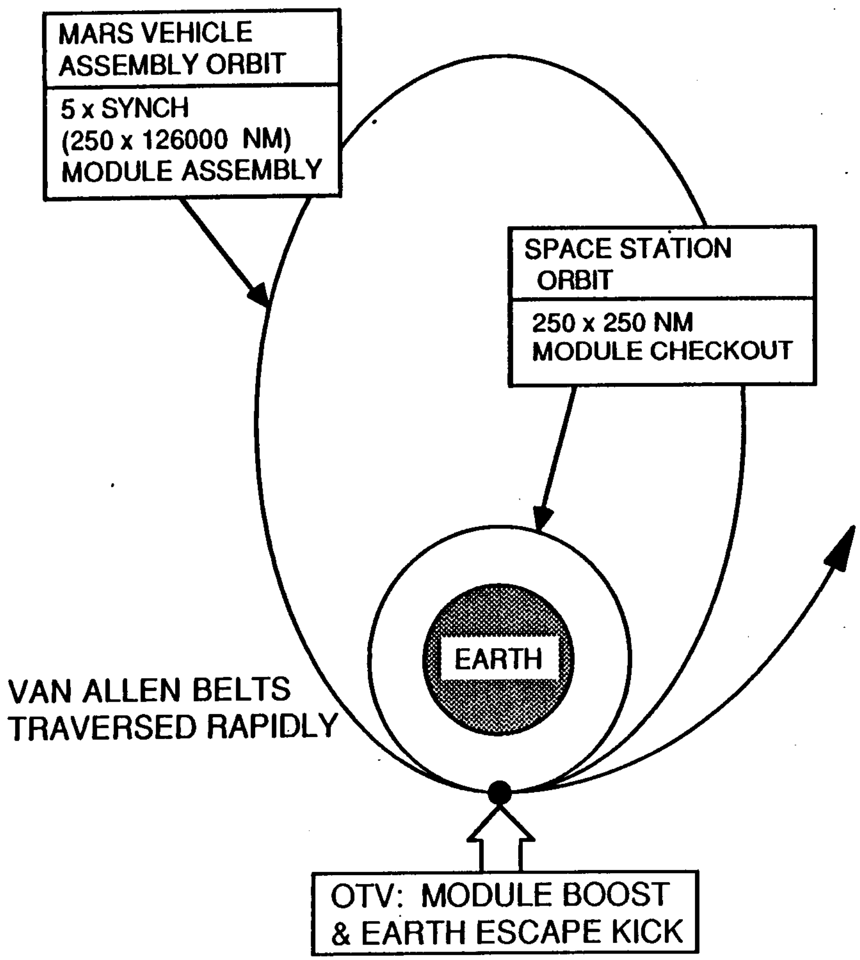
MANNED MARS MISSION LOGISTICS SUPPORT

The flight of a manned Mars mission will involve some extremely large spacecraft which are generally thought to require kick stages much larger than current OTV class. The need for such large stages is based on a direct departure from low Earth orbit. Because new boost stages will represent substantial development costs it is worthwhile to see whether existing OTV-class vehicles could be utilized instead. Shown on this chart is 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 which would be assembled into the 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 here 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. 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. The radiation doses do not appear to represent a major risk for a craft designed for deep space operations, though a more detailed assessment of this factor must await further studies.

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 kick stage need not be developed to enable a manned mars mission.

MANNED MARS MISSION LOGISTICS SUPPORT



OTV APPLICATION TO BUILDUP & BOOST OF MANNED MARS SPACECRAFT

5xSYNCH ELLIPTICAL STAGING ORBIT TO MAXIMIZE ENERGY OF ASSEMBLED MMV

1) CHECKOUT OF MODULES IN LOW ORBIT
2) OTV BOOST OF MODULES TO 5xSYNCH
3) ASSEMBLE MODULES IN 5xSYNCH
4) EXPENDABLE OTV GIVES ESCAPE KICK

OTV PERFORMANCE (74K SPACE BASED OTV)

STATION TO 5xSYNCH: 60600 LB

EXPENDABLE OTV
- 5xSYNCH TO C3= 5: 499400 LB
- 5xSYNCH TO C3=10: 354300 LB
- 5xSYNCH TO C3=20: 218900 LB
- 5xSYNCH TO C3=50: 92800 LB

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ACC OTV SAFETY

ACC OTV SAFETY ASSESSMENT

The purpose of this task was to examine the key safety issues associated with the ACC OTV concept and evaluate the TECHNICAL risk in meeting the latest safety requirements.

The approach was to identify the major hazards apparent in the concept and assess the difficulty in controlling the hazard based on the current hazard control approaches used by the STS and payloads. For the purpose of this assessment, it was assumed that 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 requirements. The latest payload requirements were used as well as the draft "return to flight" payload requirements in development by NASA.

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.

The limitations on the chart were outside the scope of this assessment and must be evaluated to fully assess the acceptability of the OTV concept.

ACC OTV SAFETY ASSESSMENT

TASK:

- REVIEW ACC OTV CONCEPT TO DETERMINE IF THERE ARE ANY SAFETY "SHOW-STOPPERS THAT WOULD PROHIBIT THIS APPROACH. COMPARE TO CARGO-BAY APPROACH

APPROACH:

- IDENTIFY MAJOR SYSTEM HAZARDS AND MAKE ASSESSMENT OF THE TECHNICAL RISK INVOLVED IN CONTROLLING EACH HAZARD

  LOW: HAZARD SHOULD BE CONTROLLABLE USING STATE OF THE ART HAZARD CONTROL TECHNIQUES

  MED: HAZARD CONTROLS NOT PREVIOUSLY TRIED OR PROVEN ON STS. PROJECT THAT HAZARD CAN BE CONTROLLED IN A FASHION THAT MEETS STS REQUIREMENTS

  HIGH: CAN NOT SEE METHOD TO CONTROL HAZARDS IN A COMPLIANT MANNER. WOULD INVOLVE QUANTIFICATION AND ACCE[TANCE OF RISK. POTENTIAL SHOW-STOPPER

- USE LATEST REQUIREMENTS
  - LATEST NHB 1700.7A AND SEPTEMBER REV. B DRAFT
  - JSC DRAFT "RETURN TO FLIGHT" PAYLOAD REQUIREMENTS
  - DISCUSS WITH JSC PANEL MEMBERS

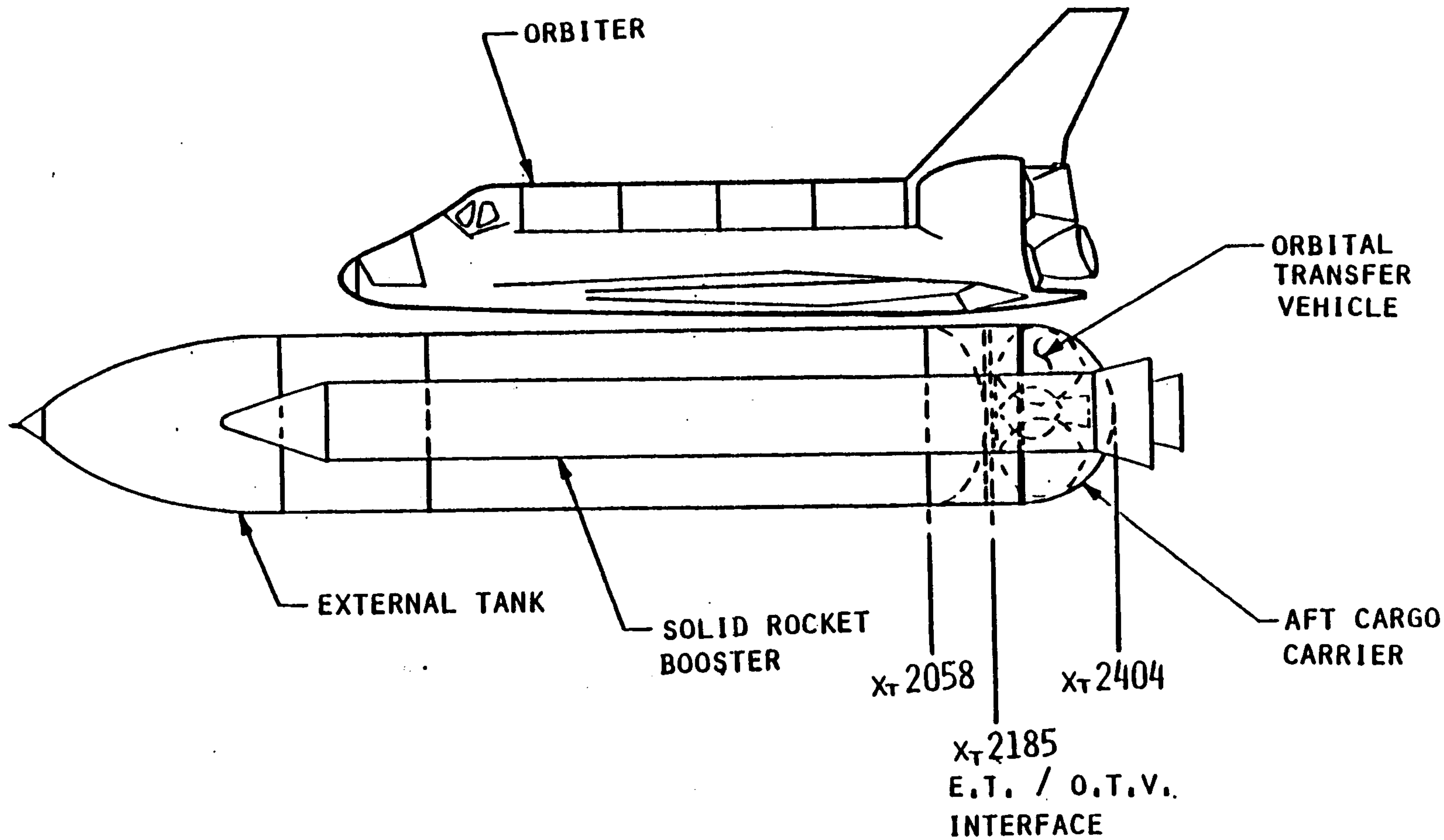
LIMITATIONS:

- COULD NOT CONSIDER:
  - ET FLIGHT PATH / FOOTPRINT IMPACTS
  - ET STRUCTURAL FAILURE POTENTIAL IMPACTS
  - STS FLIGHT DYNAMICS IMPACTS / NEW HAZARDS
  - ON-ORBIT THERMAL CONTRAINTS / IMPACTS IN MATED OPERATIONS

ACC OTV - OVERALL VEHICLE CONFIGURATION

This chart 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 some 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 by the Martin Marietta Manned Space Systems Division (formerly Michoud Division) under NASA contract NAS8-35564.

OVERALL VEHICLE CONFIGURATION



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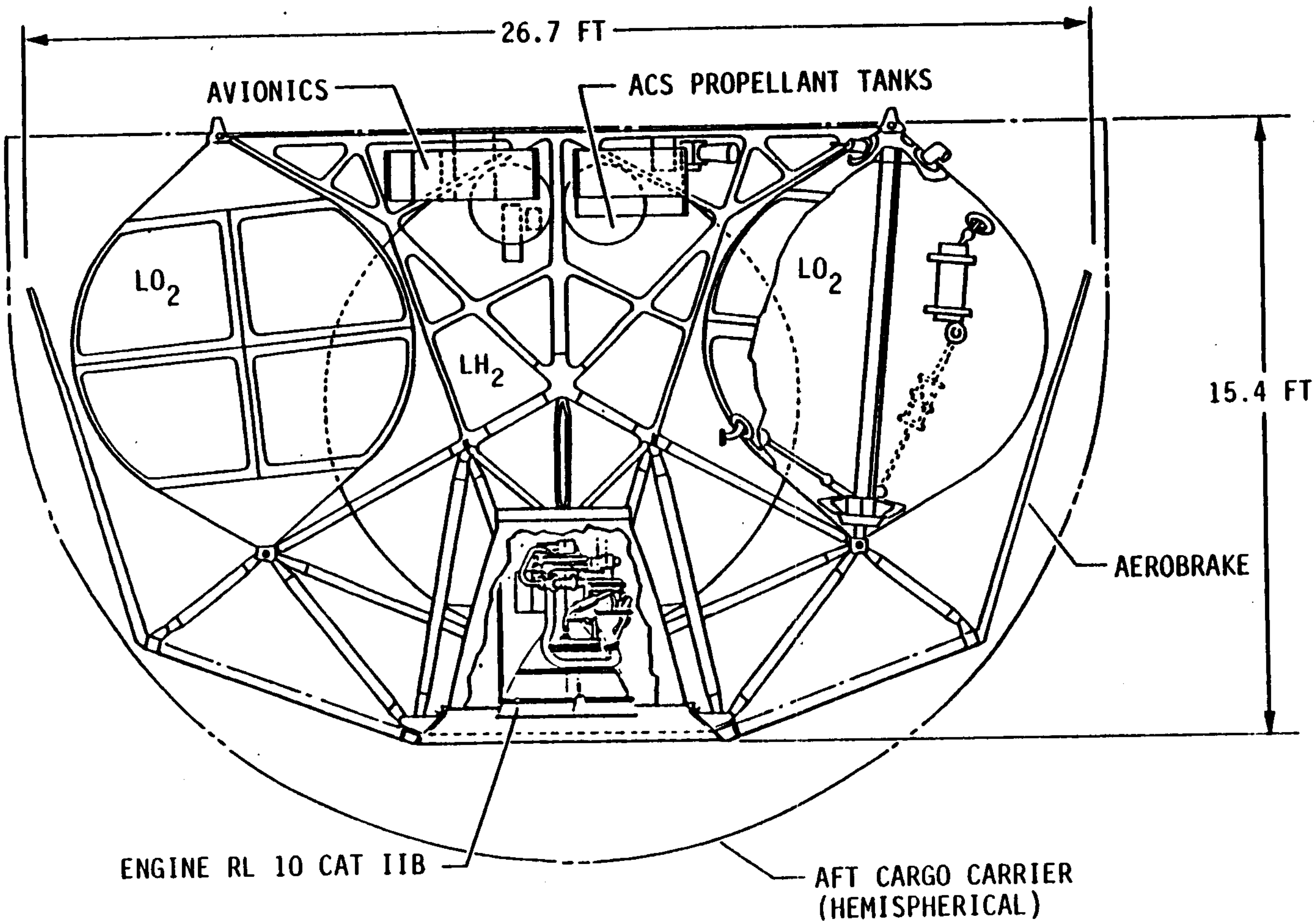
ACC OTV - BOOST CONFIGURATION

This chart 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.

AFT CARGO CARRIER OTV



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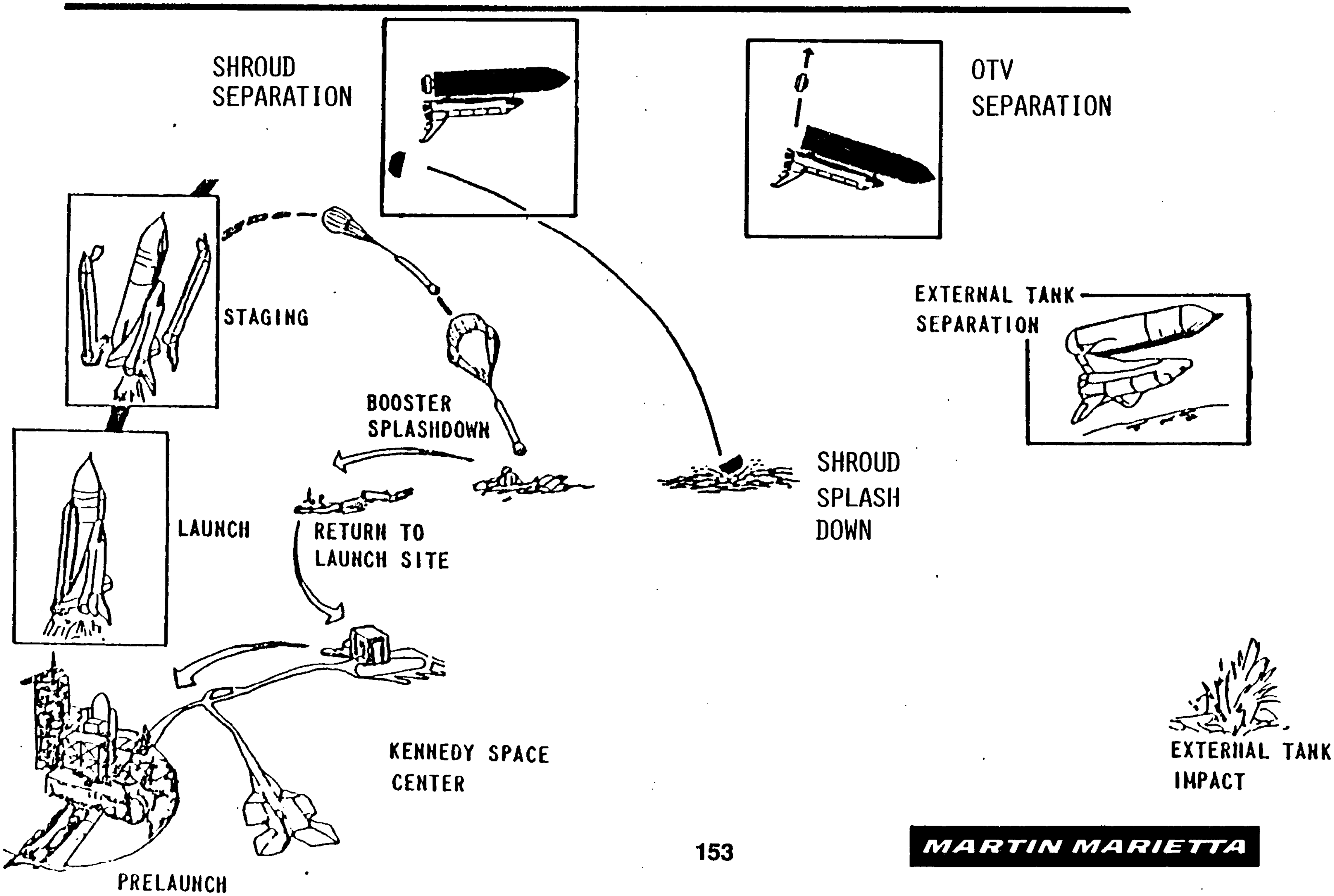
ASCENT PROFILE

This chart 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 fashion as now. ACC shroud separation occurs at T+156 sec, 24 sec after SRB separation. ET disposal targeting is identical to today's requirements. Shortly after STS main engine cutoff (MECO) the OTV is separated via springs and, after the shuttle has performed OMS-1 and departed the area, the OTV propells itself into a low park orbit. In this orbit, it awaits a renedezvous by the shuttle which then attaches its mission payload for subsequent boost.

ASCENT PROFILE



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MAJOR HAZARDS / ASSESSMENT

This chart shows the first of the major hazards that were assessed. Under a hazard group title, the individual hazards are listed with a technical risk assessment for both the ACC and cargo bay approaches. This risk is the TECHNICAL risk based on the ability of the concept to implement the typical control approaches listed (if they could be identified). Comments explaining the risk assessment are also provided.

The fire hazards generally involve release of the propellant into the cargo bay, the ACC carrier, or inadvertant 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 mechanism for these systems in the cargo bay. These must assure that no two failures will result in a partially released payload. 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 inadvertant commanding. This was rated as medium because discussions with the ACC OTV program indicated there may be other approaches to tank separation that would not involve the use of these valves. Otherwise, this would have been rated as high (a show-stopper).

The fact that the OTV is not dependent on pressure for structural integrity is a positive safety feature of both OTV concepts.

MAJOR HAZARDS / ASSESSMENT

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 - FORESEE 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.

MAJOR HAZARDS / ASSESSMENT - CONTINUED

This chart 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.

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 or positively deactivated prior to rendezvous with the Orbiter (could be mounted on the ET).

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 the safety requirements (mechanism failure will not result in Orbiter loss).

The highest risk collision hazard is associated with the failure to dump with the cargo bay configuration should dump be deemed necessaary. 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 addressed on the next figure and is likely to be required.

MAJOR HAZARDS / ASSESSMENT

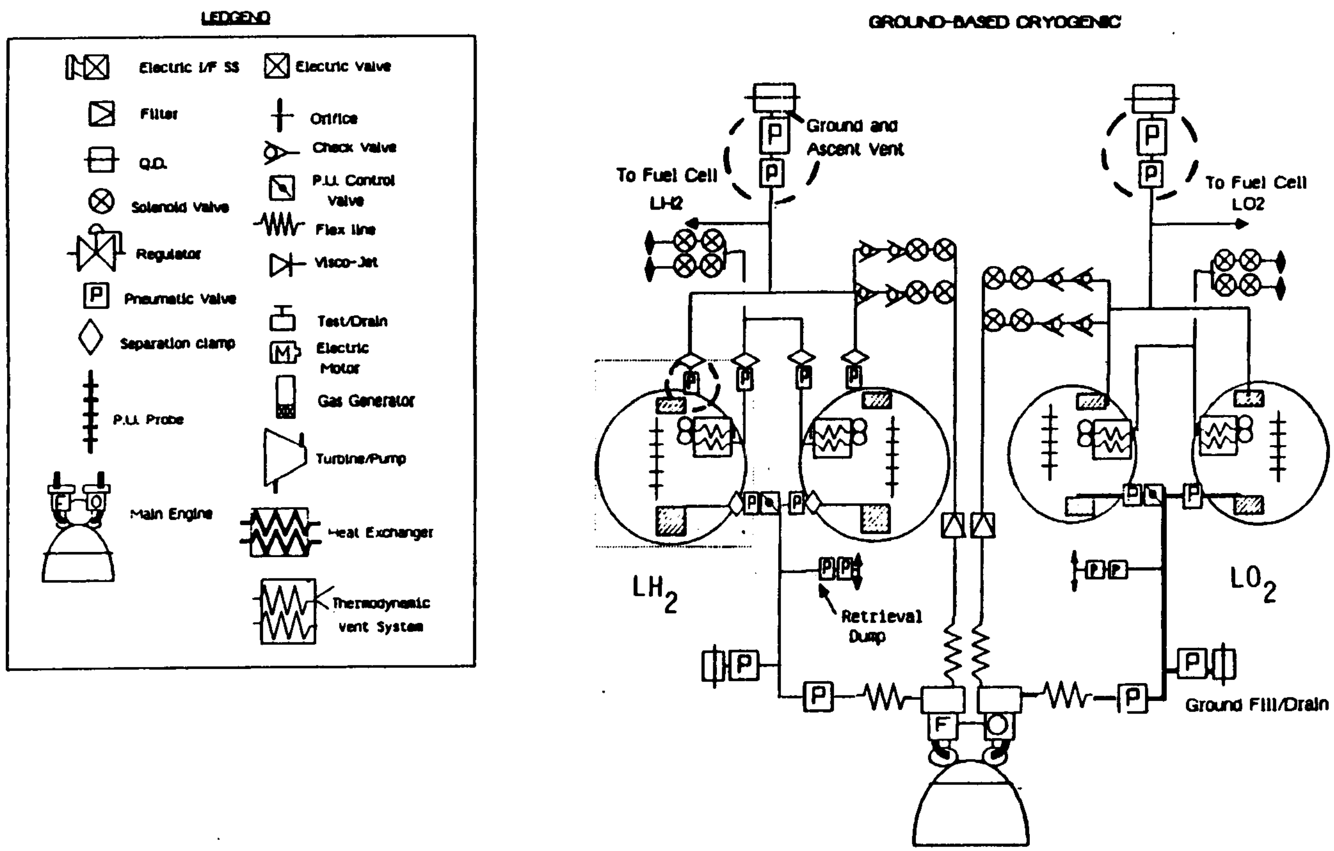
HAZARD GROUP / GENERIC HAZARD	ACC RISK	CARGO BAY RISK	TYPICAL CONTROL APPROACH	COMMENTS / CONCLUSIONS
EXPLOSION (CONT):				
B. PRESSURIZATION SYSTEM OVERPRESSURE	LOW	LOW	DO NOT OPERATE SYSTEM WITHIN SAFE DISTANCE	RESTRICTING ENGINE FIRING TO BE OUTSIDE OF SAFE DISTANCE ELIMINATES CONCERN. OTHERWISE, 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 NEED FOR SYSTEM TBD
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)
• FAILURE TO DUMP	N/A	N/A OR HIGH	UNKNOWN	IF VEHICLE MUST BE DUMPED FOR STS ABORT RETURN, DESIGN MUST BE 2 FT AGAINST PREMATURE DUMP. EXTREME CHALLENGE.
• 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.

GROUND-BASED LH_2/LO_2 SCHEMATIC

The cryogenic AOTV schematic is shown on the opposite page. Feedline and engine valves provide for three containments of propellants while the AOTV is near the orbiter during payload ops onorbit. A thermodynamic vent system (TVS) is used to nonpropulsively vent without settling. A propellant utilization system controls tank to tank dispersions and engine mixture ratio. Autogenous pressurization is provided after pump head idle is reached. Start traps are used to minimize the chilldown and settling time in tank head idle. The TVS line could be routed about the trap to minimize heat leak to the screens. Valves and self-sealing separation clamps isolate the LH_2 tank for removal and storage in the cargo bay. Helium and vent lines would be connected to pressurize the tank for return. Retrieval dump valves are used to expel the liquid remaining in the tank prior to rendezvous with the orbiter. The RCS is used to settle and dump while also completing the final apogee raising maneuver.

Thrust is provided by a single RL-10 III, 7500 lbf at 470 sec Isp and a 400:1 area ratio. The installed length is 55 in. with a two position nozzle, extending to 110 in. The exit diameter is 70 in., the envelope is the same as the RL10-IIB.

GROUND BASED LH_2/LO_2 SCHEMATIC



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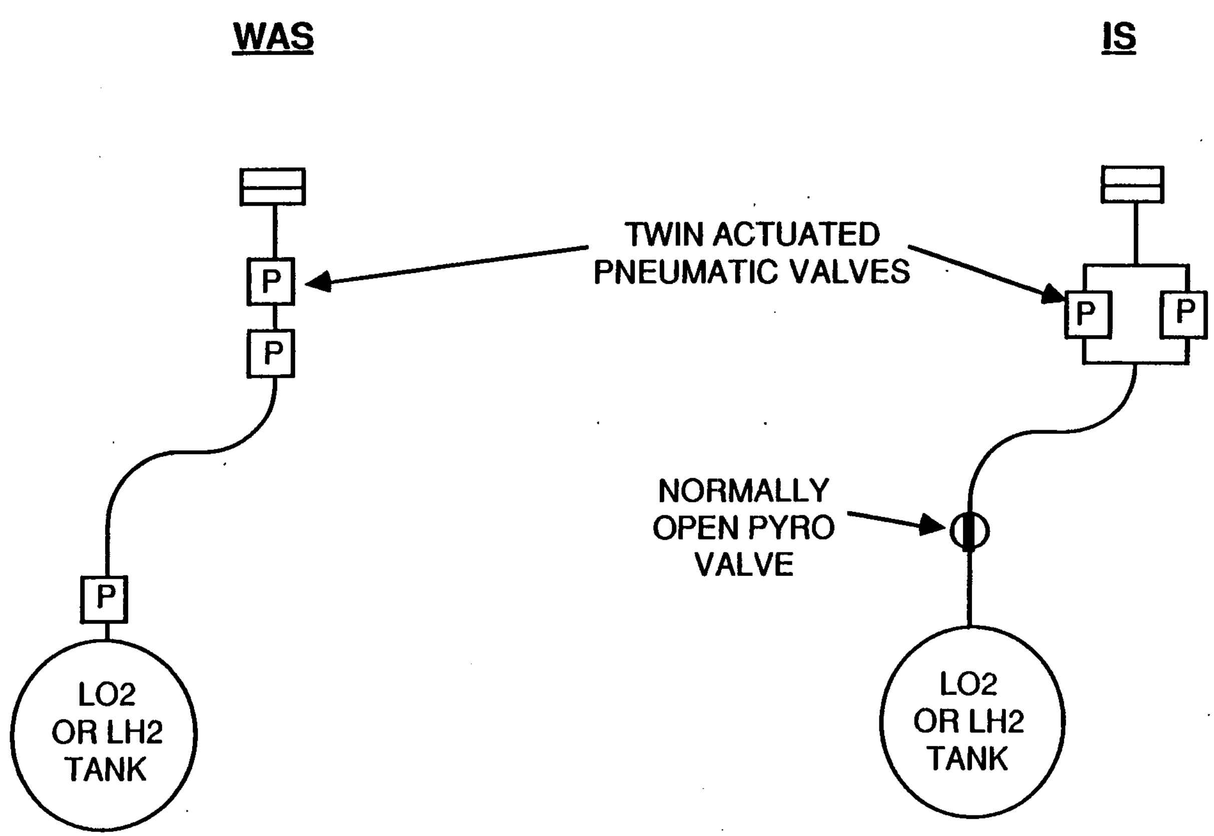
ASCENT VENT REDUNDANCY

The previous design of the ascent vent system consisted of three pneumatically actuated valves located in series in order to provide three inhibits downstream of the propellant tank. The pneumatic valves are intended to have twin actuators so that each valve can have one failure and continue to operate. The problem with the previous system design, however, is that if a valve was to have two failures, the vent system would not function, therefore creating a catastrophic condition of not relieving tank pressure.

The updated design cures this problem with parallel pneumatic valves to provide for venting control and a single pyro actuated valve with twin initiators. This system provides for two fault tolerance in the venting system as well as three inhibits for preventing loss of propellant from the tanks.

ASCENT VENT REDUNDANCY



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LRR

CONCLUSIONS

Within the constraints of this study, two major conclusions were reached.

The first is that there were no potential show stoppers identified for the ACC OTV concept and there is a potential show stopper with the cargo bay configuration (the need to dump would be directly opposing the controls to prevent premature dumping).

Secondly, it was concluded that the ACC OTV has definite safety advantages over the cargo bay configuration:

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).

The two medium risk items associated with the ACC configuration are not show-stoppers. The tank separation valve concern could be eliminated completely with other concepts. This leaves only the potential new destruct system as both a technical and additional safety risk. The safety risk associated with this system should be made to be acceptable since history in designing these systems exist.

JSC was 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 this is not prohibited but "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 JSC safety panel members contacted said they havew not seen a design that meets all of the requirements but would not project that it could not ever be done. A system requiring pressure for structural integrity would be the biggest challange and is probably not do-able without major safety compromises.

CONCLUSIONS

- THE ACC OTV CONCEPT CAN MOST LIKELY BE MADE TO MEET THE CURRENT REQUIREMENTS
  - NO SHOW STOPPERS SEEN
  - SAFETY ADVANTAGES OVER IN-BAY APPROACH
  - NEED TO DELETE TANK SEPARATION VALVES OR FIND DIFFERENT APPROACH

- THE CARGO BAY CONFIGURATION HAS POTENTIAL SHOWSTOPPERS

- NEED TO DUMP PRIOR TO RETURN WOULD REQUIRE A SYSTEM THAT IS BOTH TWO-FAILURE TOLERANT TO PREMATURE DUMPING AND AGAINST FAILURE TO

  - NO DESIGN HAS ACCOMPLISHED THIS

  - NEED TO DUMP IS TBD - WOULD BE DRIVEN BY:

    - NEED TO CHANGE CG; OR,
    - NEED TO DECREASE WEIGHT; OR,
    - NEED TO ELIMINATE CRYOS IF ALL LANDING SAFETY ISSUES ARE NOT SOLVED (FURTHER ANALYSIS REQUIRED)

  - EXTENSIVE MODIFICATIONS TO THE ORBITER REQUIRED PER JSC (VENTING)

  - "ALL THE CENTAUR ISSUES MUST BE RESOLVED" PER JSC

MARTIN MARIETTA

ACC PRESURE STABILIZATION

Currently the dedicated ACC (DACC) uses internal presure for stabilization during the STS SRB ignition overpresure pulse. A review of STS/Centaur Lessons Learned highlights that one of the main problems with the Centaur was its presure stabilized skin. In this case internal presure was required throughout the flight to maintain structural integrity. Hence one of the major prohibitions that has resulted from the Centaur cancellation is against presure 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 presurization is not required for flight, but only for the extremely brief period of time that the SRB ignition overpresure exists. An adequate presure in the ACC would then be one of the launch commit criteria to be honored before the SRB ignition command. 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 count could be halted before any ignition-critical presurization levels were reached. This represents a complication for the shuttle but not a flight-critical one.

An alternate approach was investigated, however, that assessed the design impact of making the ACC totally unpresurized for all phases of flight. This approach beefed up the ACC dome structure so that the SRB ignition pulse could be resisted solely with structural stiffness. In order to keep the flight weight manageable, a filament wound approach was neccesary. This approach results in significant manufacturing complication and an increase in weight of 210 lb. Further details may be found in the Structural Issues section.

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

ACC PRESURE STABILIZATION

- DEDICATED ACC CURRENTLY USES INTERNAL PRESURE

  STABILIZES STRUCTURE AT SRB IGNITION - TRANSIENT REQUIREMENT

  NOT REQUIRED FOR FLIGHT

- "NO PRESURE STABILIZED STRUCTURES" - CENTAUR LESSONS LEARNED

- TWO PATHS POSSIBLE:

  1) CURRENT APPROACH OK

     PRESURIZATION ONLY REQUIRED FOR DURATION OF IGNITION OVERPRESURE

     CRITICAL PRESURE DECAY WOULD BE DETECTED BEFORE SRB IGNITION COMMIT

  2) REDESIGN DEDICATED ACC

     ELIMINATE PRESURIZATION REQUIREMENT

     COMPOSITE DESIGN TO REDUCE WEIGHT GROWTH

     WEIGHT IMPACT = 210 LB

MARTIN MARIETTA

DACC COMPOSITE SHROUD

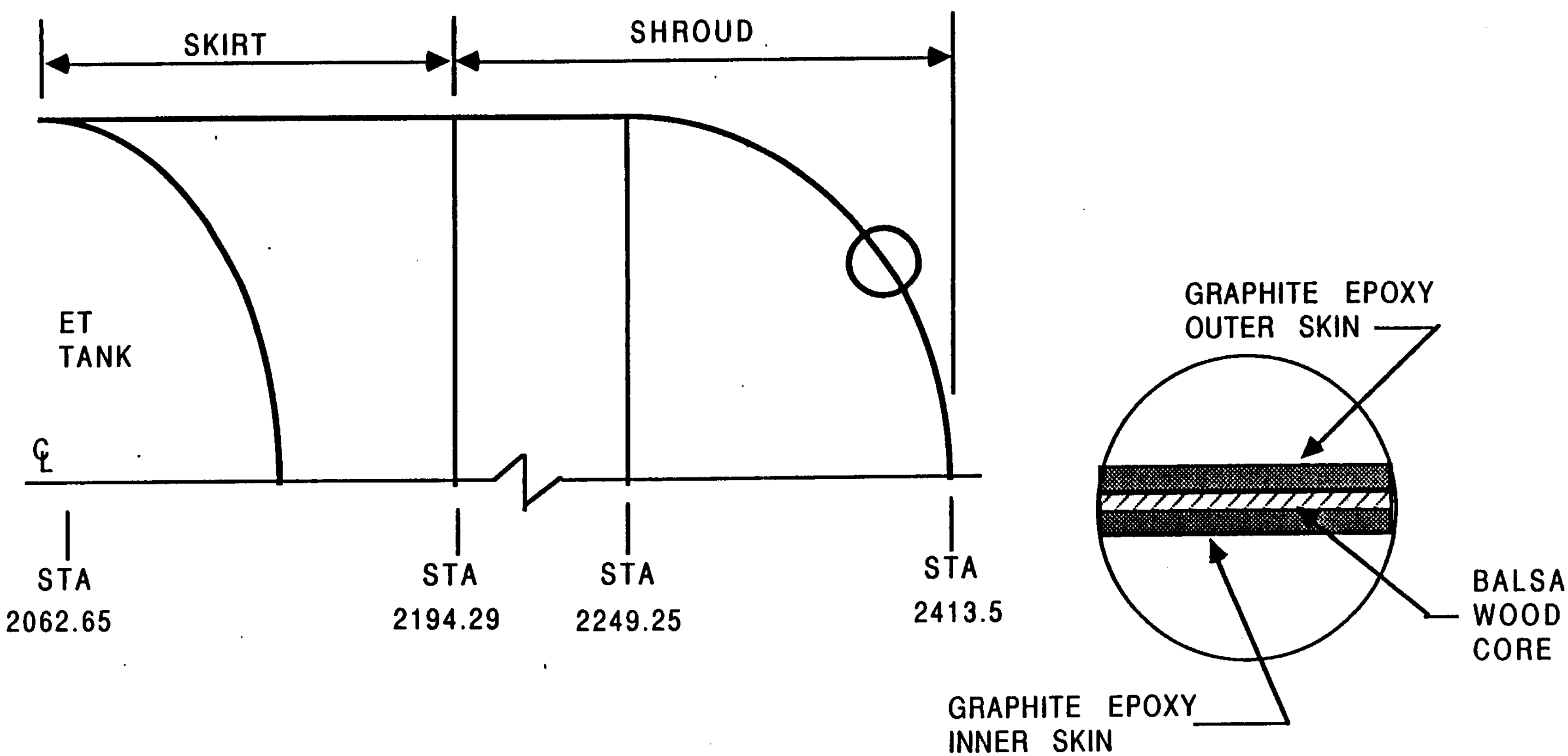
In the baseline design, the skins will be a sandwich structure. The inner and outer skins will be filament wound AS4W-12K graphite fiber using HBRF 55A epoxy resin. This composite will have 50% fiber by volume. The lamina properties for this composite are: the modulus in the fiber direction is 17.21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

The baseline 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 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.

DACC COMPOSITE SHROUD



DACC SHROUD WEIGHT COMPARISON

This chart shows the weight breakdown and comparison of the unpressurized 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.

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.

Translating the two different design concepts into a weight difference produces a net weight increase of 203 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.

DACC SHROUD WEIGHT COMPARISON

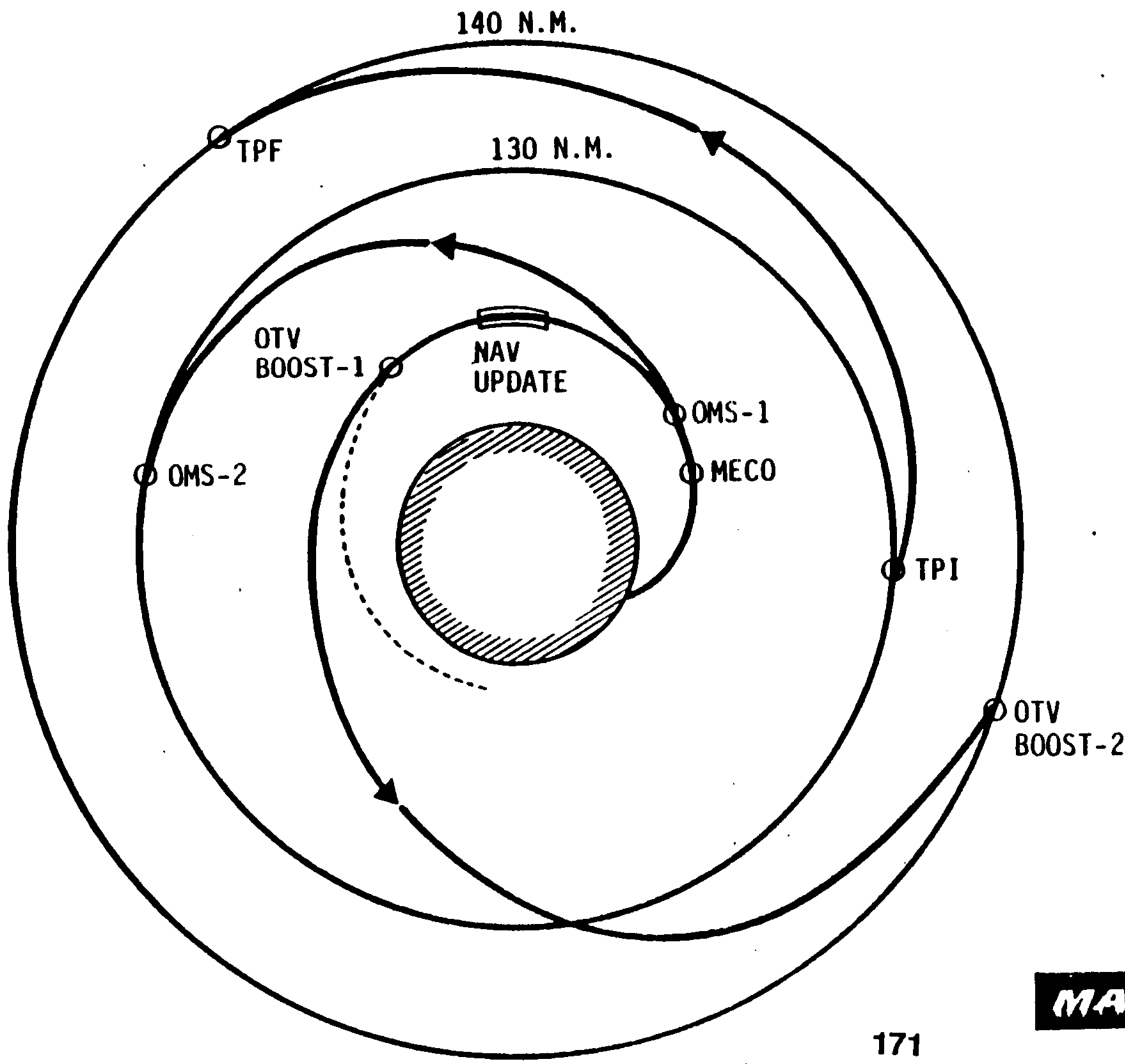
	METAL PRESSURIZED WEIGHT (LB)	COMPOSITE UNPRESSURIZED WEIGHT (LB)	DETLAS WEIGHT (LB)
SKIRT			
STRUCTURE	2556	2556	O
THERMAL PROTECTION	173	173	
AVIONICS/ELECTRICAL	152	152	
PROP/MECH	125	125	
ORDNANCE	23	23	
CONTINGENCY	454	454	
SUBTOTAL	3483	3483	O
SHROUD			
DOME	781	1248	+467
ATTACH FLANGE	62	62	0
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

OTV / ORBITER TRAJECTORY PLOT

This figure shows an overview of the ACC OTV ascent profile along with the associated Shuttle profile. The ACC OTV is deployed just after Shuttle MECO and flies itself independently into a 140 nmi park orbit. The Shuttle meanwhile flies into an initial 130 nmi orbit after which it performs rendezvous with the OTV at 140 nmi. OTV burns are timed to occur when the Shuttle is safely out of range according to STS safe separation criteria. The other major safety issue is how to safe the OTV after it has completed its orbital insertion sequence. This will be adressed in the following chart.

OTV/ORBITER TRAJECTORY PLOT

ASCENT THROUGH RENDEZVOUS NO. 1



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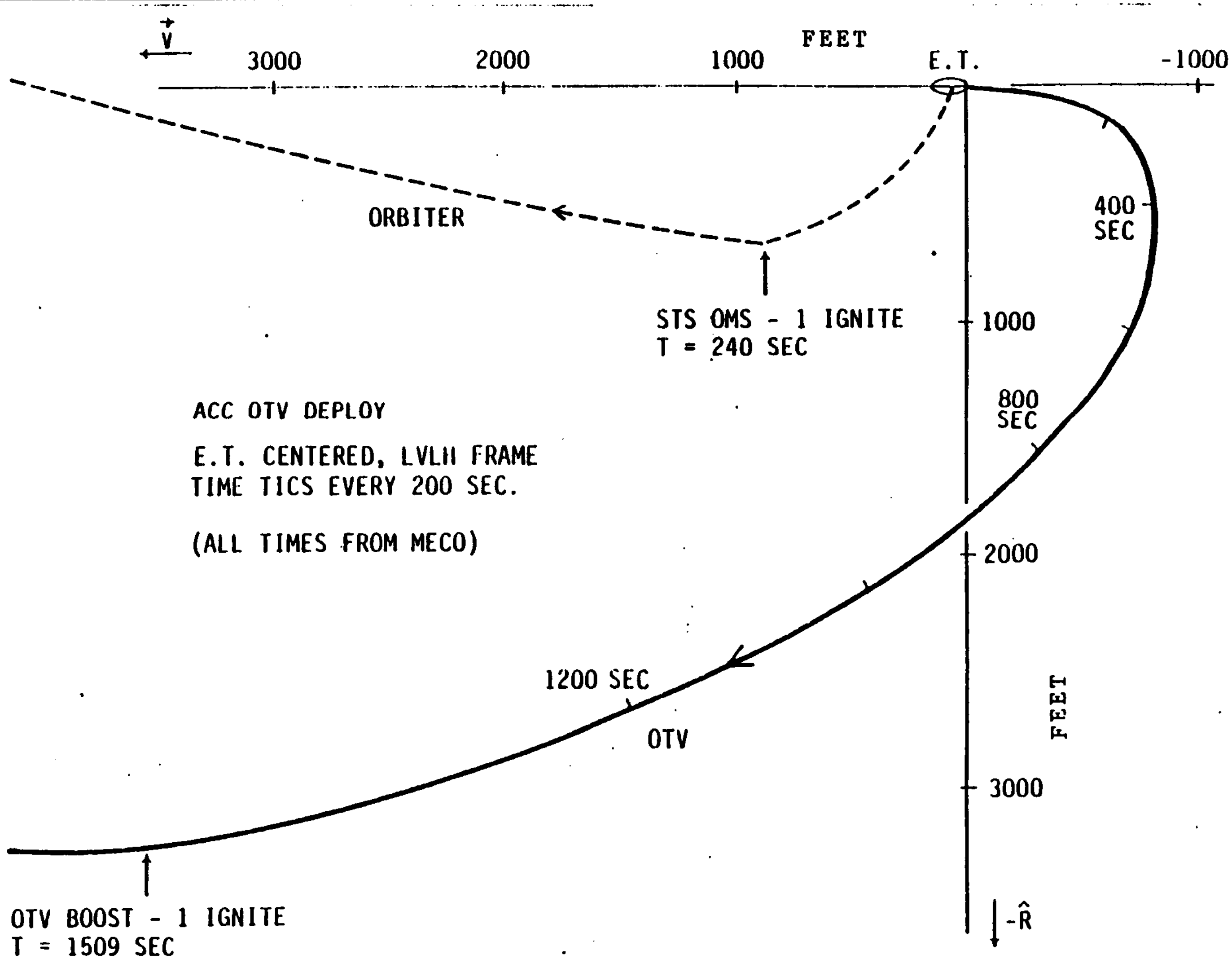
ORBITER-ET-OTV RELATIVE MOTION

Because the ACC OTV must fly independantly to low Earth orbit it must not interfere with the normal Shuttle operations. STS safe separation criteria have been used throughout in designing the ACC OTV flight sequence. This diagram 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 ACS system is turned on at a distance of 200 ft, consistent with STS safe separation criteria. When the Shuttle performs its OMS-1 burn the two are about 1800 ft apart which should be adequate from a plume impact standpoint. The first OTV MPS burn occurs at about 1500 sec 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 at T=4610 sec at an Orbiter separation of 228 nm. The Orbiter rendezvous sequence commences a few hours after this final OTV main engine burn.

ORBITER-ET-OTV RELATIVE MOTION



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ACC OTV PROX OPS SAFETY SEQUENCE

A unique concern to an ACC OTV is vehicle safing for Shuttle rendezvous and payload mate. This figure 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 desireable 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. Analysis shows a capability for 6 hours of no-vent if the OTV tanks are first reduced to 16 psi. This will eliminate undesireable gaseous venting during the time the two vehicles are in collision range.

The final system to be safed will be the OTV Attitude Control System (ACS). The range at which this must be done is uncertain at present, it would be desireable to wait until as late as possible to reduce residual attitude rate disturbances.

ACC OTV PROX OPS SAFETY SEQUENCE

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

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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. The figure shows a payload adapter with one end being standardized to the OTV and the other being payload peculiar. The OTV end contains guide pins and electric latches to enable on-orbit docking with the propulsive stage. 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.

ACC OTV ON-ORBIT PAYLOAD INTEGRATION

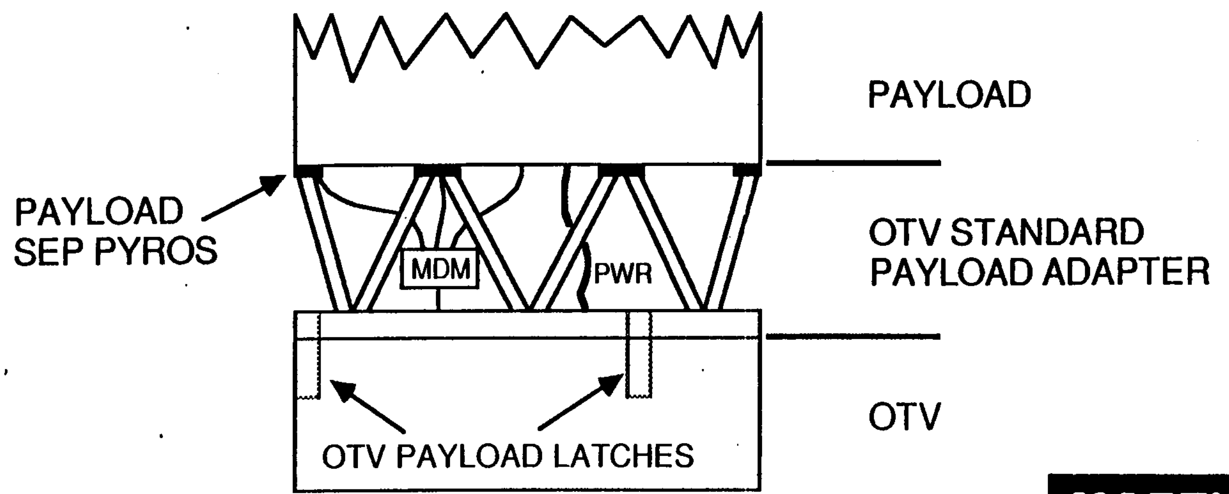
ON-ORBIT INTEGRATION OF TWO VEHICLES IS NOT A NEW ISSUE

- 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)



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ACC OTV HARDWARE JETTISON

This figure shows the sequence of events required to safely dispose of an OTV's aerobrake and tankage which reduces the amount of volume required to return the vehicle to Earth. Upon exiting the atmosphere after an aeroassist the aerobrake is jettisoned via springs. Because the trajectory is suborbital, the orbital life of the aerobrake is less than 1 revolution. It is felt that the aerobrake will disintegrate because of the very much higher heat pulse (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.

After the OTV coasts to its first apogee 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.

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 lowest performance ACS system, the ground based hydrazine system, is only 35 lb.

Thus this tankage and aerobrake disposal sequence shows promise as a method of reducing the downleg volume requirements which will be critical if Shuttle is the only available means of return.

ACC OTV HARDWARE JETTISON

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 BOTH MPS PROPELLANTS (LO2 & LH2)
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

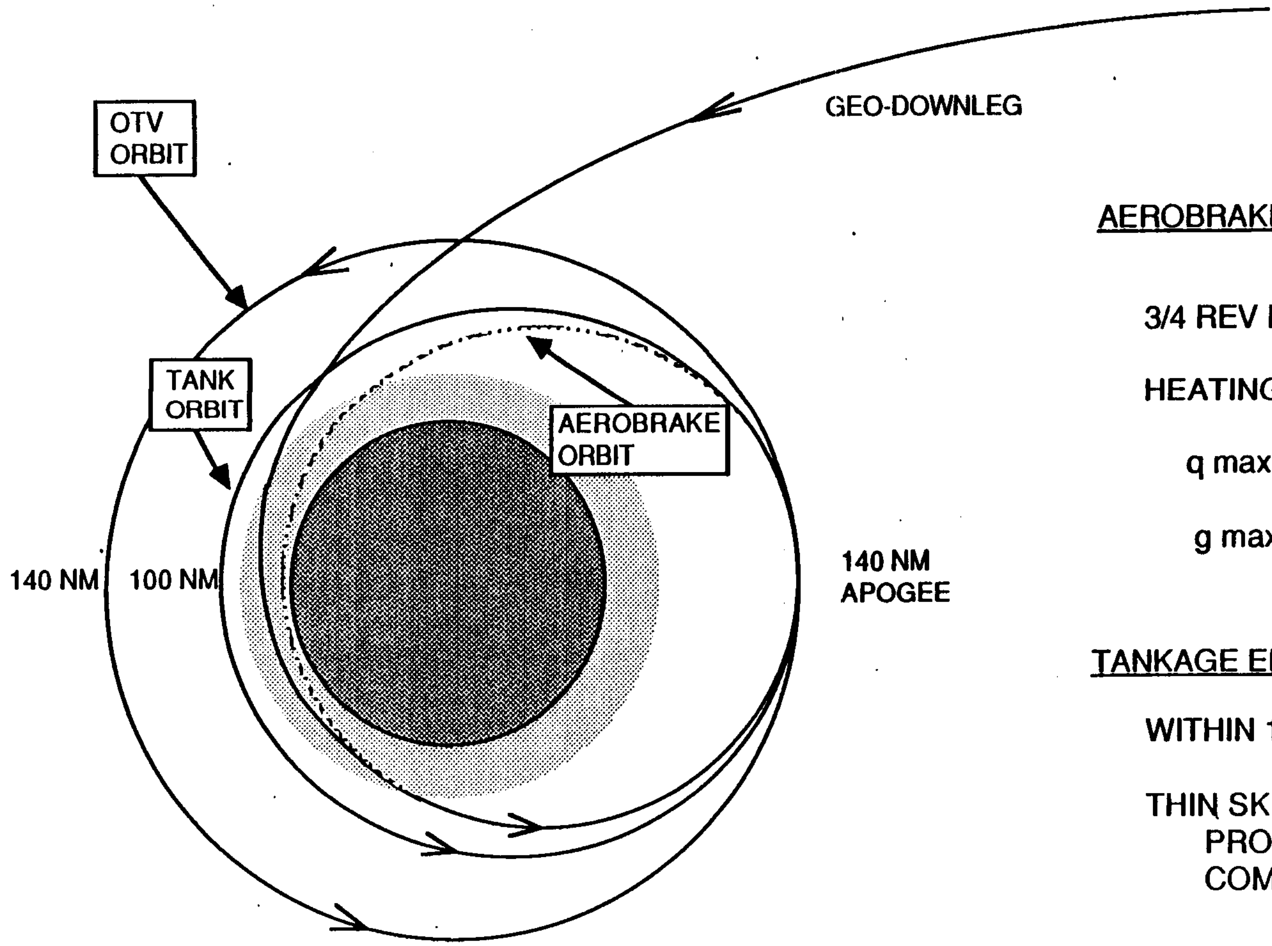
LH2 TANKS - LESS THAN 1 DAY ORBITAL LIFE

ACS VELOCITY REQUIREMENTS = 71 FPS (35 LB PROPELLANT)

ACC HARDWARE DISPOSAL

This figure illustrates the operation of tankage and aerobrake disposal discussed on the previous chart.

ACC HARDWARE DISPOSAL



AEROBRAKE ENTRY

3/4 REV FROM JETTISON

HEATING EXCEEDS TPS

q_{max} = 453 BTU/FT2 SEC

g_{max} = 40 g's

TANKAGE ENTRY

WITHIN 1 DAY OF JETTISON

THIN SKIN (0.025 INCH) PROBABLY BURNS UP COMPLETELY.

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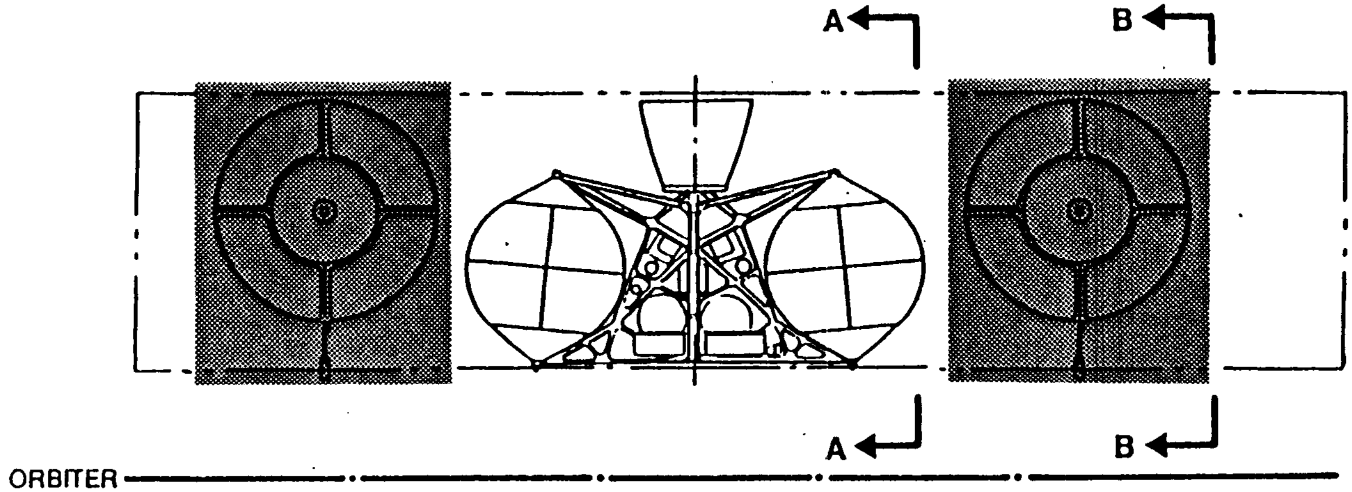
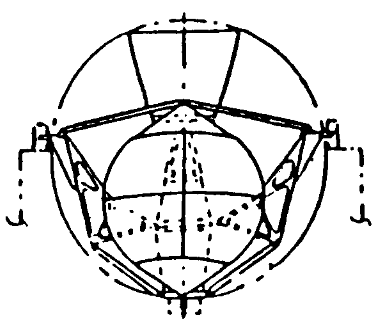
PAYLOAD BAY VOLUME SAVINGS BY JETTISONING LH2 TANKS

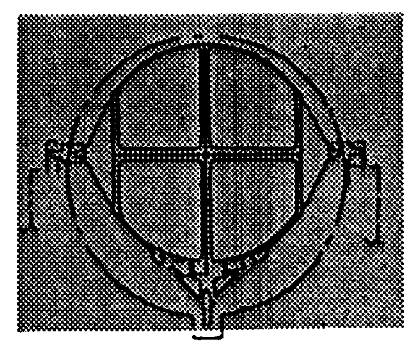
This figure shows the savings for the Shuttle payload bay volume if the large OTV LH2 tanks are jettisoned rather than being returned to Earth intact. Because of the increasing value of STS down capability, as heightened by Space Station operations, this approach appears to be an attractive one. It also 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.

PAYLOAD BAY VOLUME SAVINGS - LH2 TANK JETTISON





SECT A-A



SECT B-B

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ACC OTV RETURN RECOMMENDATIONS

RECOMMEND JETTISON OF LH2 TANKS

- REDUCES CARGO BAY VOLUME REQUIREMENTS (FROM 85% TO 40%)
- REDUCES STS CARGO BAY SUPPORTING ASE (FROM 2659 TO 920 LB)
- REDUCES OPERATIONAL COMPLEXITY (NO STS RE-CONFIGURATION)

RETURNED HARDWARE INCLUDES

- STRUCTURAL CORE
- ALL AVIONICS
- ALL ENGINE HARDWARE & PLUMBING DOWNSTREAM OF LH2 TANKS
- LO2 TANKS
- ALL ACS HARDWARE

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ACC OTV EVALUATION - PRO'S

- MINIMIZE STS SAFETY PROBLEMS
  - NO IN-FLIGHT DUMPS REQUIRED
  - NO POST-LANDING INERTING (ACUTE PROBLEM FOR TAL)
  - OTV CAN BE JETTISONED AFTER SRB SEP
  - OTV NOT IN CARGO BAY (SAFE VENTING, ETC)
- NOT PENALIZED BY STS LANDING LIMITS
- GOOD GROWTH TO LCV AND/OR SPACE BASING
- NO CANTILEVERED PAYLOADS (TRUNNION PIN MOUNTING)
- VEHICLE FLIGHT CHECK BEFORE PAYLOAD COMMIT
- STS PERFORMANCE ENHANCEMENT (OTV ORBIT INSERT WITH CRYO ENGINE)
- ACC CROSS BENEFIT TO OTHER PROGRAMS:
  - DELIVERY OF LARGE STRUCTURES, LOW DENSITY P/L'S, CRYO FLUIDS

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ACC OTV EVALUATION - CON'S

- UP FRONT COST / PARALLEL PROGRAM FOR ACC
- STS BOOST AERODYNAMICS RE-CERTIFICATION
- MORE COMPLEX PRE-DEPLOY OPERATIONS
- MORE TIME SPENT IN LOW EARTH ORBIT
- PAYLOAD MATE IN ORBIT (NOT SHOW-STOPPER FOR GEMINI / APOLLO)
- MORE COMPLEX RETRIEVAL IF LH2 TANKS ARE RECOVERED

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DESIGN ISSUES

The final review design issues begin with the initial expendable vehicle definition and comparisons of performance with the ground based reusable concept developed in earlier Phase A effort. The intent is to provide a vehicle concept that represents a program start in a time frame earlier than for the ground based reusable concept. The issues that pertain to the near term expendable include the choice of performance enhancements that fit the proper time frame.

The Lunar mission optimization, transfer vehicle and lander definitions, and cryogenic engine implications of Lunar landing are a major portion of this review. Final subjects include OTV concept definitions that correspond to Shuttle "C" and Large Cargo Vehicle (LCV) launch vehicle concepts.

DESIGN ISSUES

ACC EXPENDABLE VEHICLE DEFINITION
- PERFORMANCE ENHANCEMENT OPTIONS
- BASELINE SELECTION

GROUND BASED REUSE VS. EXPENDABLE
- PERFORMANCE TO GEO
- PAYBACK FOR REUSE

LUNAR MISSION ACCOMMODATION
- MISSION OPTIMIZATION
- LUNAR TRANSFER VEHICLE DEFINITION
- IMPLICATIONS OF LUNAR LANDINGS FOR CRYO ENGINES

SHUTTLE "C" OTV DEFINITION
- VEHICLE CHARACTERIZATION
- PERFORMANCE SUMMARY

DELTAS BETWEEN NEAR TERM EXPENDABLE AND G.B. REUSABLE

The table 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. The brake can be added or removed as a unit without impacting the remaining stage structure. Aluminum structure could perhaps be used in a near term expendable vehicle for cost and schedule benefits but would have performance impacts. Also, an existing RL10A could be used rather than a newly developed engine; once again to provide cost and schedule benefits but with dry weight and Isp impacts.

Other dry weight benefits for an expendable stage include less avionics and meteoroid protection requirements. This is primarily due to less time on orbit.

DELTAS BETWEEN NEAR TERM EXPENDABLE AND G.B. 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

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IOC DATE AND VEHICLE OPTIONS

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. These will be shown later in this presentation. The facing chart shows two examples of OTV character depending on IOC date.

IOC DATE AND VEHICLE OPTIONS

1993 - EARLIEST AVAILABILITY EXPENDABLE - RL10A ENGINE,
2219 ALUMINUM TANKS, COMPOSITE STRUCTURE.
ENHANCED STS GEO CAPABILITY (12.2 K WITH 55 K STS)

1995 - ALLOWS ADVANCED CAPABILITIES -
IOC ENGINE (475 SEC),
ALUMINUM LITHIUM ALLOY TANKS.
(14 K TO GEO WITH 55 K STS, 18.2 K WITH 65 K STS)

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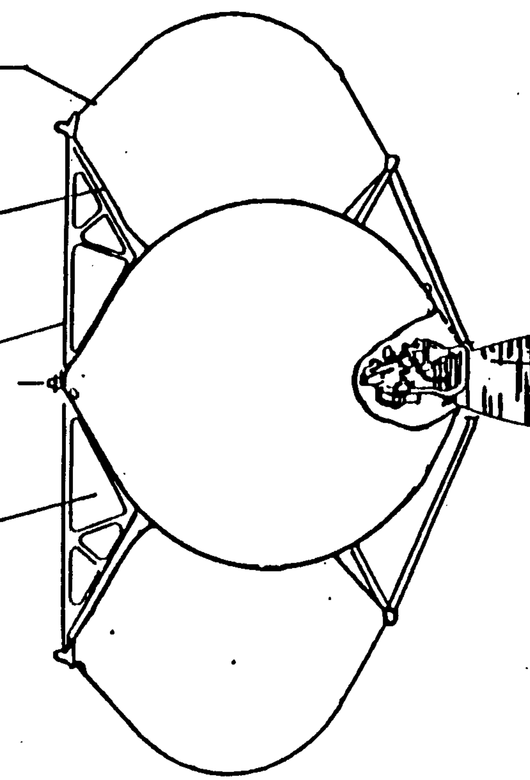
ACC EXPENDABLE OTV BASELINE

The general arrangement and weight breakdown for our selected expendable OTV transported in the ACC are shown on the facing viewgraph. The expendable OTV is based on the same arrangement as the groundbased 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.

The major differences are: aerobrake removal, Al 2219 tanks instead of Al-Li 2090 tanks, a RL10-A engine, and Ag-Zn batteries in place of the fuel cell system. Some GN&C equipment has been removed, or will be, replaced by a smaller system.

The total dry weight of the ACC expendable OTV is 4189 lb.

ACC EXPENDABLE OTV BASELINE



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

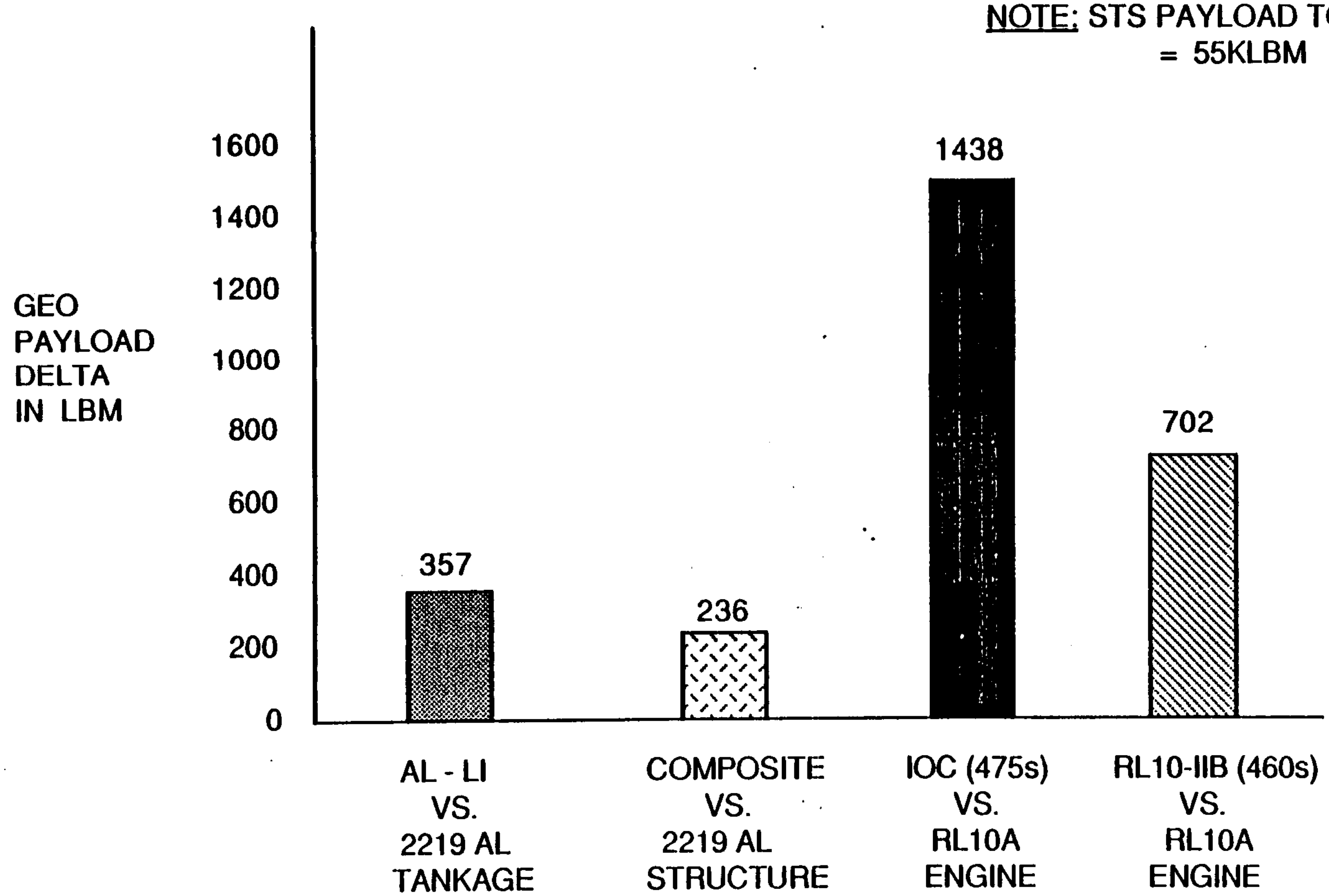
PERFORMANCE ENHANCEMENTS DELTAS

The performance improvement has been calculated for each of the vehicle enhancements under consideration for the near term expendable vehicle concept. These performance enhancement deltas are the benefits in GEO payload capability with a launch weight constraint of 55klbm. The tankage and structure performance deltas are nearly the same as the dry weight differences between the 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. These combined effects result in the differences shown in the figure.

PERFORMANCE ENHANCEMENT DELTAS - INITIAL EXPENDABLE

NOTE: STS PAYLOAD TO 110 NMI = 55KLBM



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LRR

COST GROUNDRULES: EXPENDABLE OTV TRADE STUDIES

The chart opposite 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.

The NSTS cost per flight used for purposes of transportation costs to Low Earth orbit (LEO) was baselined at $73M. This is 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-10 3A 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 candiates. 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 vehicles this is treated as a cost per mision item. The delta unit cost is combined with the third cost element, namely 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.

Cost Groundrules: Expendable OTV Trade Studies

- 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

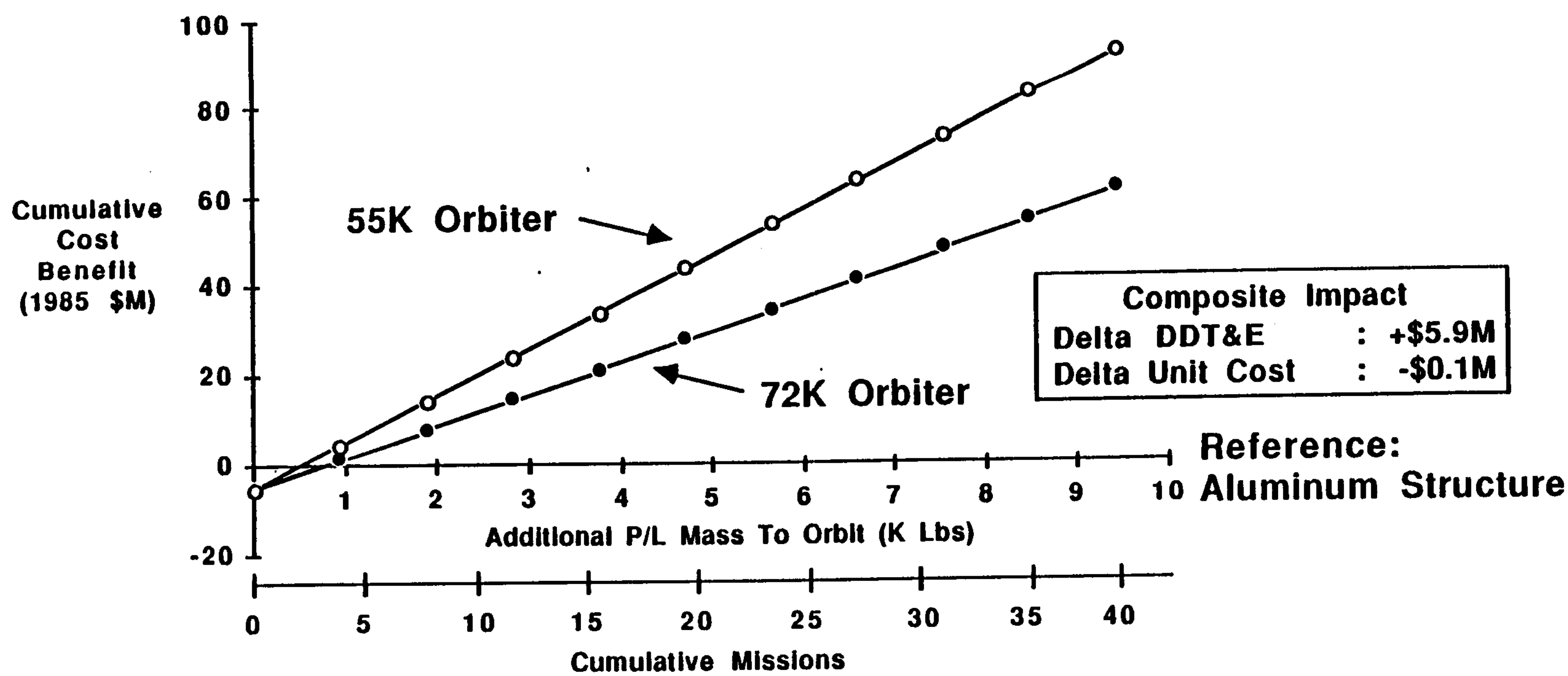
ALUMINUM VERSUS COMPOSITE STRUCTURES TRADE

The cost results of replacing the aluminum airframe with composite structures are shown on the page opposite. 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.

The two plots represent the cumulative cost benefit (on a per mission basis) 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.

Aluminum Versus Composite Structures Trade

	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



Composite Impact

Delta DDT&E : +$5.9M

Delta Unit Cost : -$0.1M

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MANNED SPACE SYSTEMS

M_OTV-STR-$/LB

ALUMINUM VERSUS ALUMINUM LITHIUM TANK TRADE

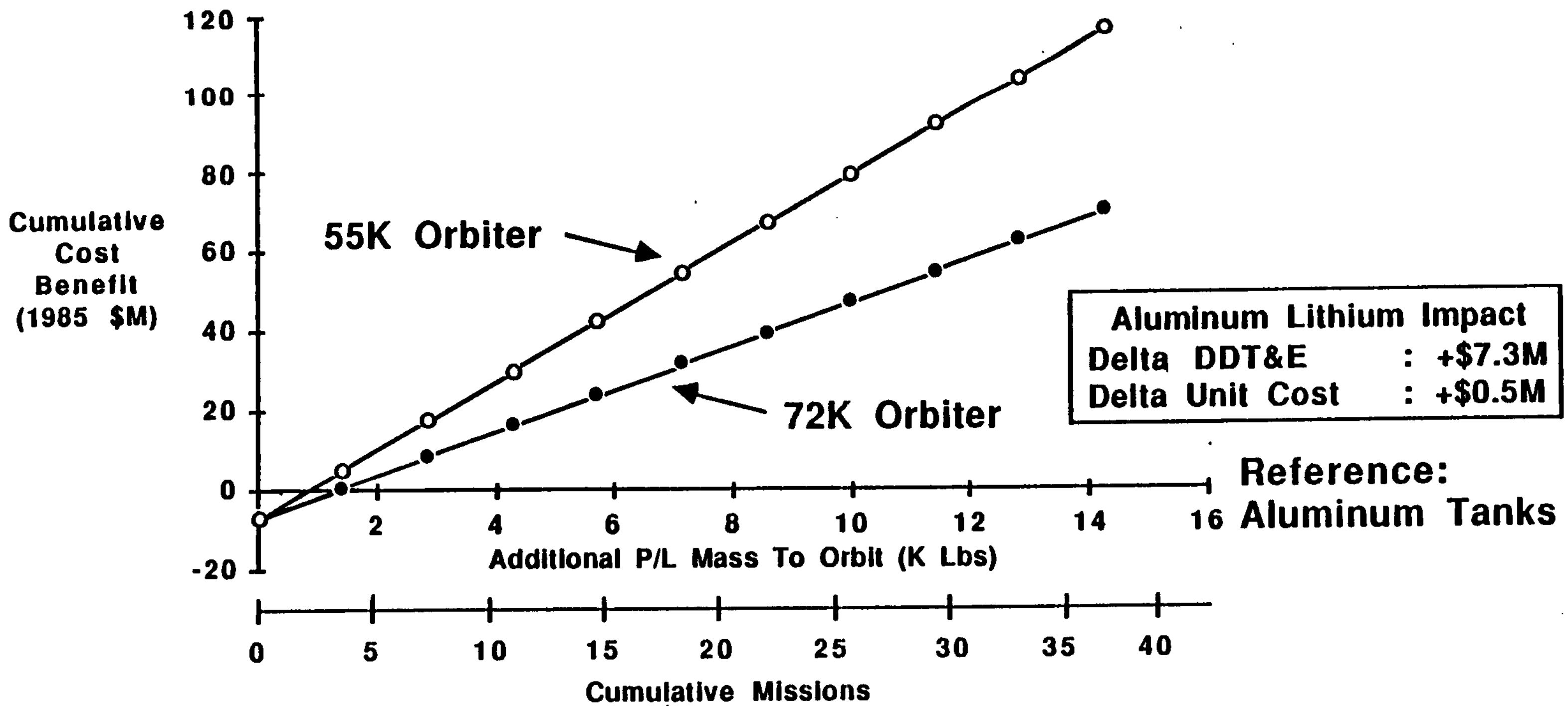
The cost results of replacing the aluminum tanks with aluminum lithium tanks are shown on the page opposite. The DDT&E and unit cost for the aluminum tanks are $14.6M and $2.4M respectively. The aluminum lithium tanks exhibit higher DDT&E ($27.5M) and unit costs ($2.9M). The higher DDT&E cost is driven 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.

The delta DDT&E cost estimate is represented by the offset on the Y-axis. The additional DDT&E investment required for the aluminum lithium tanks is $7.3M. The unit cost delta is approximetely $0.5M per set of tanks.

As in the previous trade study results, the two plots represent the cumulative cost benefit (on a per mission basis) 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.

Aluminum Versus Aluminum Lithium Tanks Trade

	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



Aluminum Lithium Impact	
Delta DDT&E	: +$7.3M
Delta Unit Cost	: +$0.5M

M_OTV-TNK-$/LB

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MANNED SPACE SYSTEMS

RL-10 VERSUS IOC ENGINE TRADE

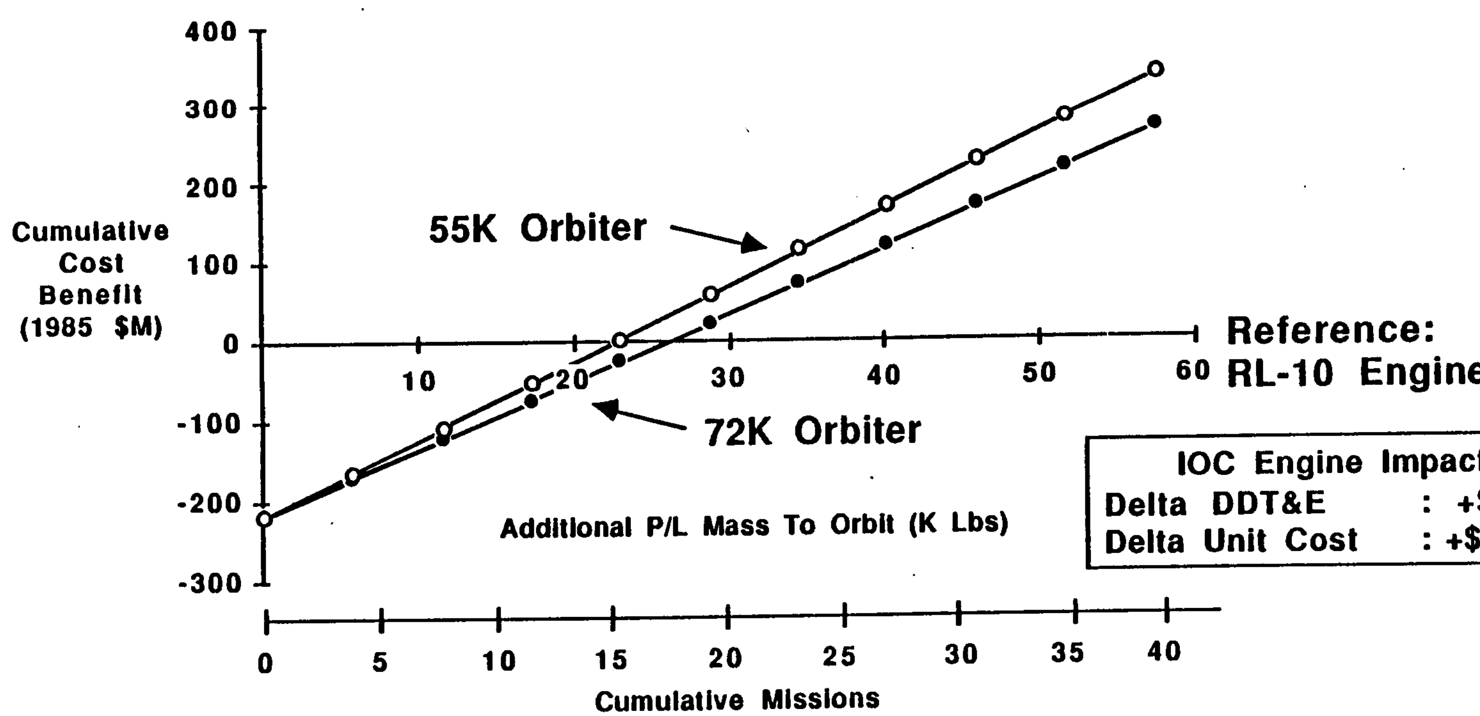
The cost impact for developing the IOC engine are shown on the page opposite. The DDT&E and unit cost for the RL-10 are $14.8M and $1.7M respectively. The DDT&E includes primarily ground test hardware and test operations requirements due to integration of the RL-10 to the new expendable stage. The DDT&E 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.

The delta DDT&E cost estimate is represented by the offset on the Y-axis. The additional DDT&E investment required for the IOC Engine is $220.0M. The unit cost delta is approximetely $0.5M engine.

As in the previous two trade studies, the two plots represent the cumulative cost benefit (on a per mission basis) 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 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 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 mission range. The overall benefit after 40 missions is much more significant than in the previous trades.

RL-10 Versus IOC Engine Trade

	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

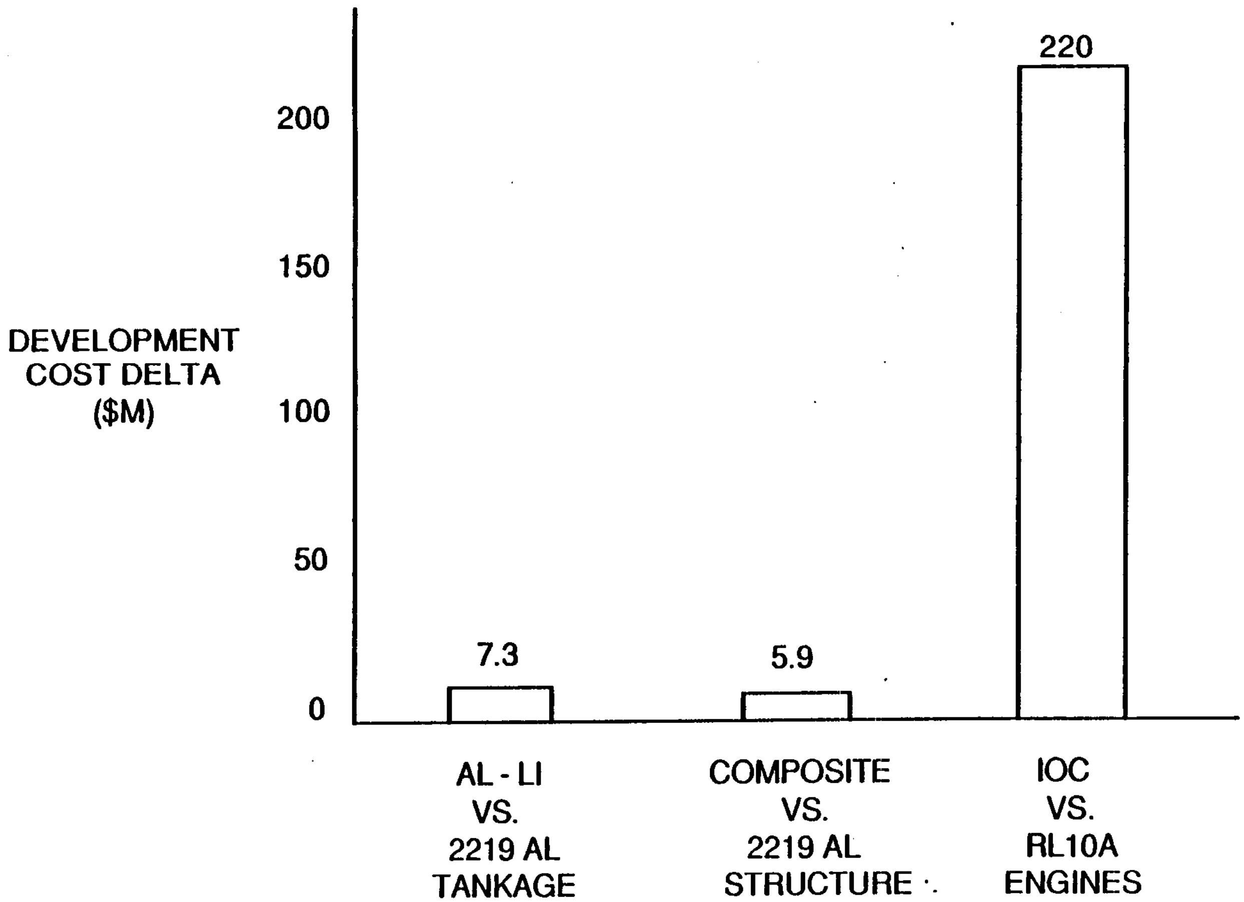


IOC Engine Impact	
Delta DDT&E	: +$220.0M
Delta Unit Cost	: +$ 0.5M

ENHANCEMENT DEVELOPMENT COST DELTAS

The figure 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.

ENHANCEMENT DEVELOPMENT COST DELTAS



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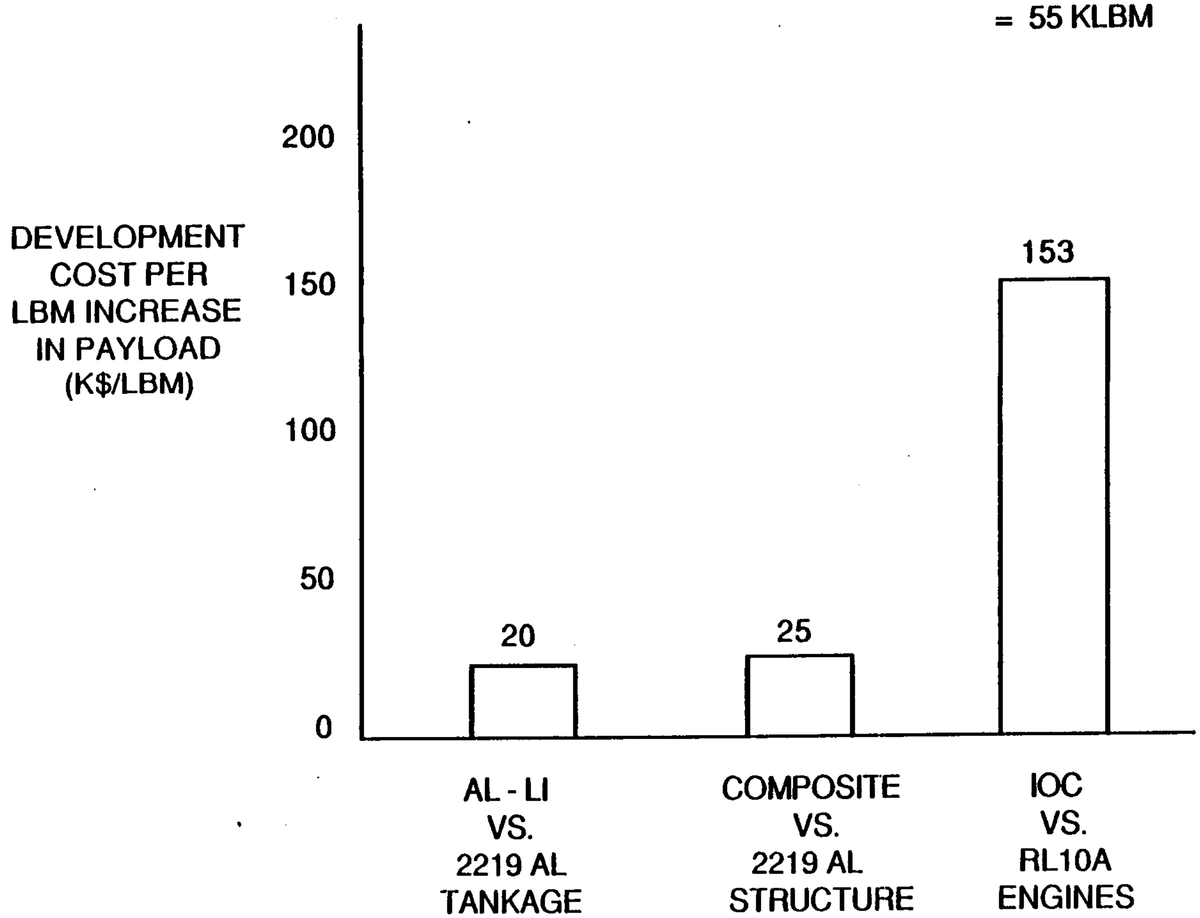
ENHANCEMENT DEVELOPMENT COST PER GEO LBM INCREASE

A good indication of the worth of each of the vehicle enhancements is the amount development dollars spent for the performance gained. The figure 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.

ENHANCEMENT COSTS PER LBM PAYLOAD IMPROVEMENT

NOTE: STS PAYLOAD TO 110 NMI
= 55 KLBM



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COST SUMMARY CHART

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 5 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.

COST SUMMARY CHART

EACH ENHANCEMENT TO THE EXPENDABLE APPEARS TO BE WORTHWHILE ECONOMICALLY, AND IS DEPENDANT UPON AVAILABILITY

ENGINE - REQUIRES 4 FLTS/YR TO PAYOFF IOC ENGINE IN 5 YEAR PERIOD

TANKAGE - FAIRLY QUICK (LESS THAN 5 FLTS) PAYBACK ONCE THE MATERIAL BECOMES AVAILABLE

STRUCTURE - ALSO QUICK PAYBACK (ABOUT 5 FLTS)

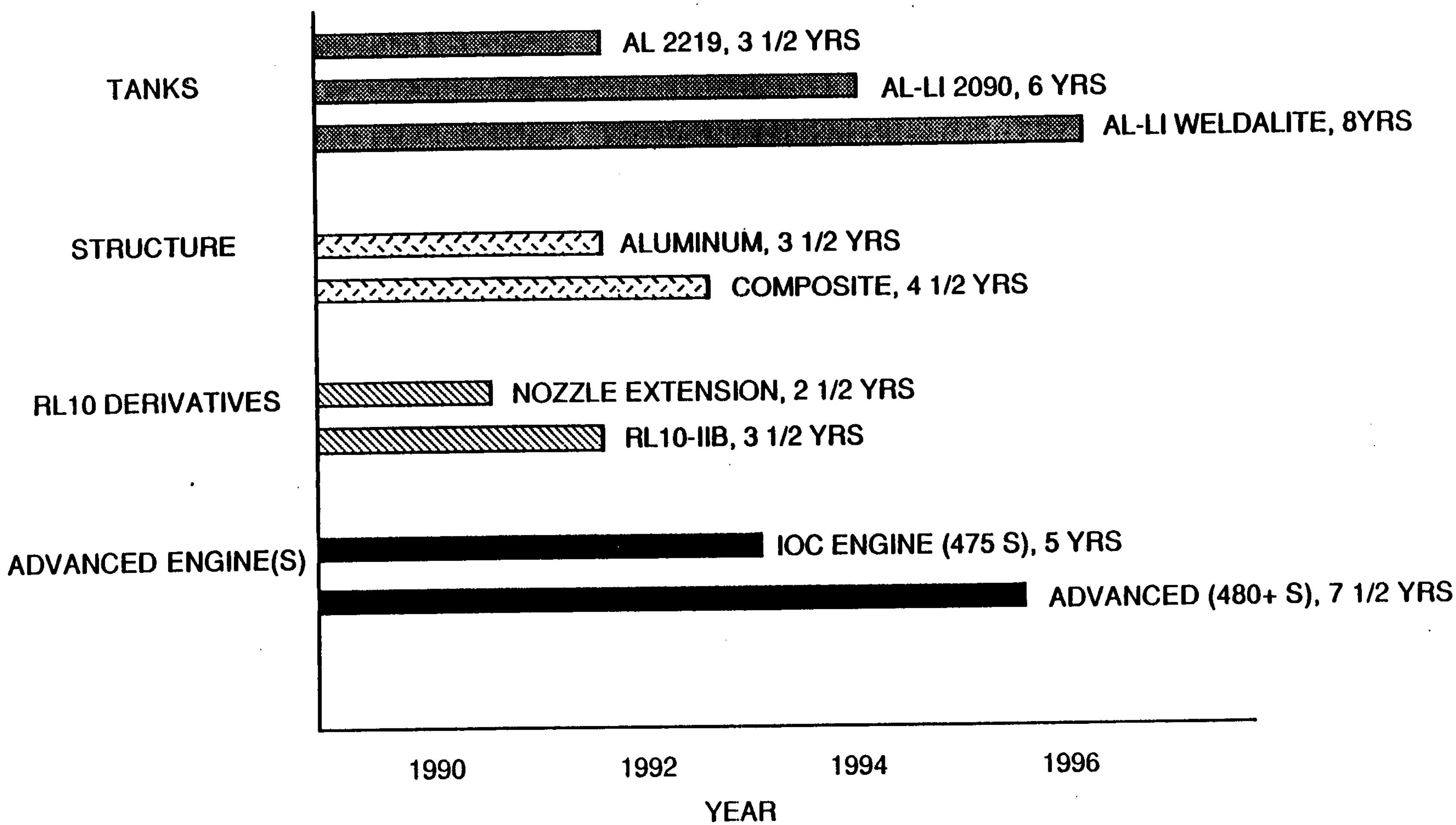
ENHANCEMENT DEVELOPMENT TIMES

The schedule of availability for each of the vehicle enhancements is shown in the figure 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; 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 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.

ENHANCEMENT DEVELOPMENT TIMES

NOTE: PROGRAMS BEGINNING IN 1988



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EXPENDABLE VEHICLE TRADE SUMMARY

The recommended characteristics of the initial expendable vehicle have been determined based upon cost trade studies. The recommendations 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.

EXPENDABLE VEHICLE TRADE SUMMARY

ALUMINUM VS ALUMINUM LITHIUM TANKAGE → RECOMMEND INCORPORATING ALUMINUM-LITHIUM TANKS AS SOON AS THE MATERIAL IS AVAILABLE

ALUMINUM VS COMPOSITE STRUCTURE → RECOMMEND USING COMPOSITE RATHER THAN ALUMINUM STRUCTURE

RL10A ENGINE VS IOC ENGINE → RECOMMEND USING IOC ENGINE AS SOON AS THE ENGINE CAN BE MADE AVAILABLE

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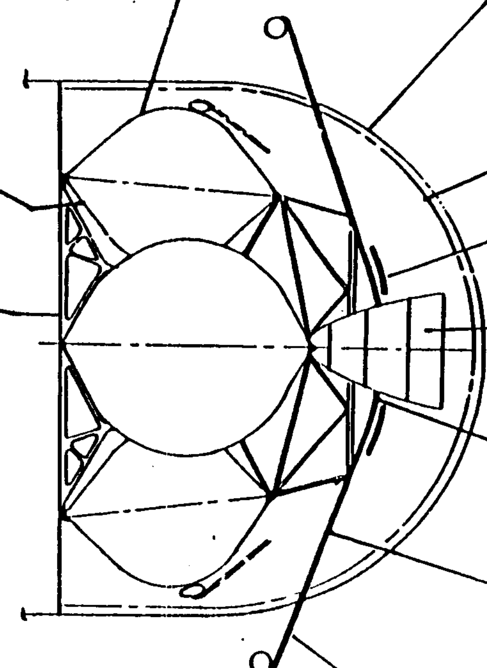
GROUNDBASED CRYOGENIC REUSABLE OTV

This viewgraph shows the general arrangement and weight breakdown of our selected groundbased cryogenic OTV transported in the ACC. 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).

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, along with the core system (LO2 tanks, structure, avionics, and propulsion) are stowed in the Orbiter cargo bay for retrieval after mission completion. The propulsion and avionics subsystems reflect the component count previously considered. The structure is of lightweight graphite/epoxy. The propellant load was selected to enable full use of the projected NSTS lift capability on GEO delivery missions.

GROUND BASED CRYOGENIC REUSABLE OTV



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

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STS ACC OTV PERFORMANCE BASELINE

Weight summaries are shown here for both the expendable vehicle baseline as well as the ground based reusable vehicle concept. The performance numbers are given for both a 55 k STS and a 65 k STS. The weights remain the same for each vehicle for the ACC, payload ASE, OTV ASE, and OTV dry for the two launch weight capabilities. The propellant, payload, and total liftoff weights differ for the two STS capacities.

STS ACC OTV 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)

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PAYLOAD TO GEO WITH STS

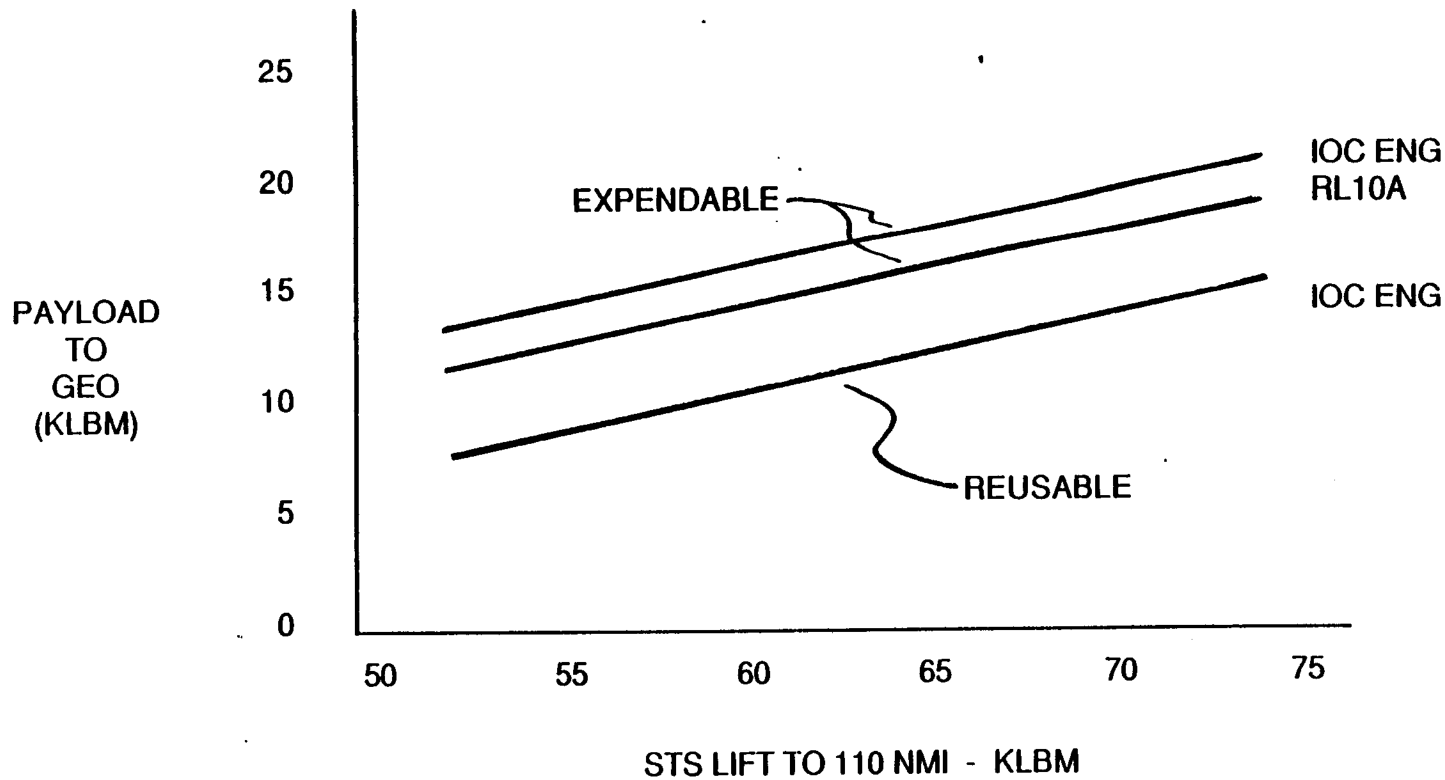
The figure shows OTV payload delivery capability to GEO as a function of STS delivery capability for the reusable and expendable vehicle concepts generated during this study. The STS lift capability shown corresponds to what the Shuttle can deliver to 110 nmi.

The conclusions to be drawn from the figure include the observation that the expendable vehicle concept is capable of delivering significantly greater payload to GEO than with the reusable concept. This may be a crucial realization 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 subsequent 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, could be higher than for the expendable for OTV class payloads.

PAYLOAD TO GEO WITH 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



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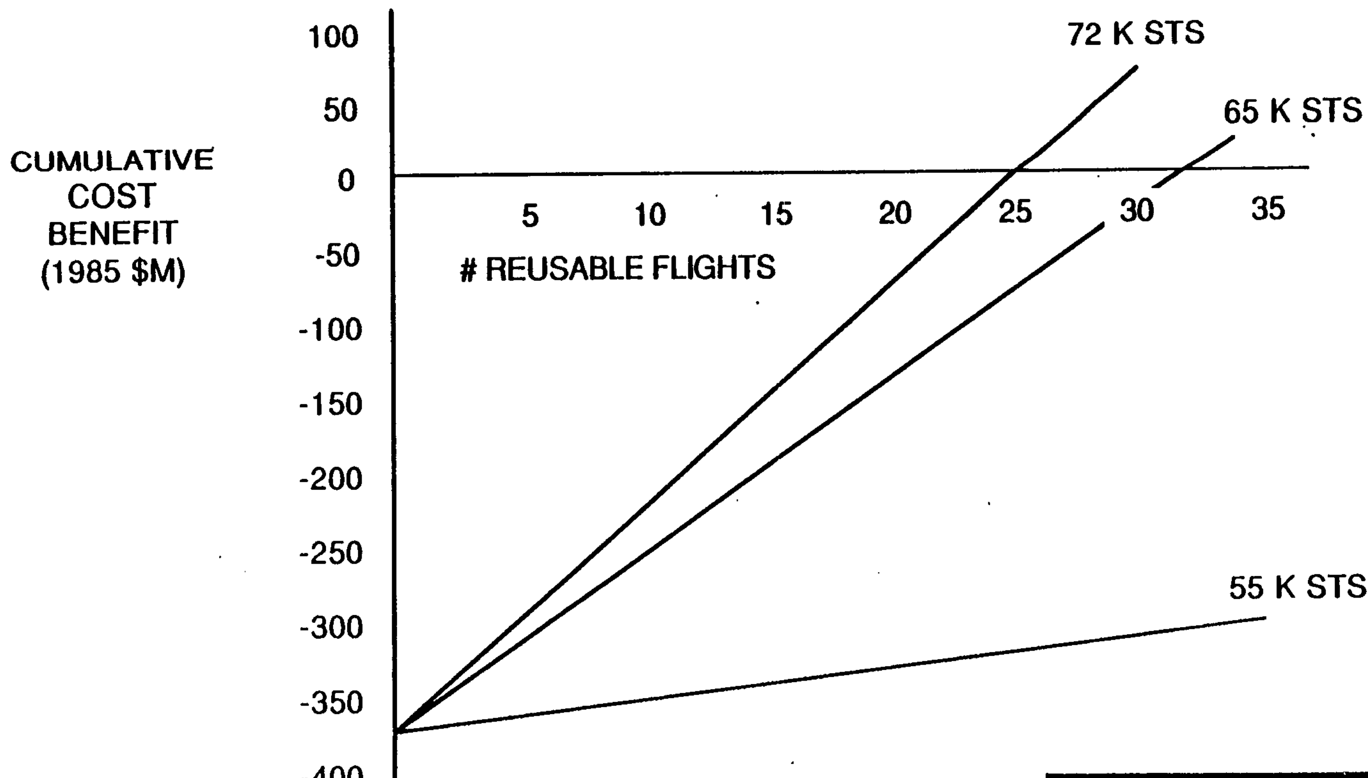
REUSABLE VEHICLE PAYBACK OVER EXPENDABLE

The ground based reusable vehicle concept has lower performance in terms of GEO payload than the expendable concept. However, the unit cost of the expendable vehicle can presumably be eliminated, or at least significantly reduced on a per-flight basis, by the reusable vehicle.

The figure shows the payback associated with the reusable concept after the investment is made to develop it. The crossover point with regard 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 larger payload relative to that of the expendable vehicle for higher STS capacities.

REUSABLE VEHICLE PAYBACK OVER EXPENDABLE

- EQUAL CUMULATIVE MASS TO GEO
- $10 M/FLT COST FOR USING REUSABLE VEHICLE
- DELTA DDT & E = $434M



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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 evaluation of the OTV for reuse, space basing, aeroassist, etc. These opportunities 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.

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

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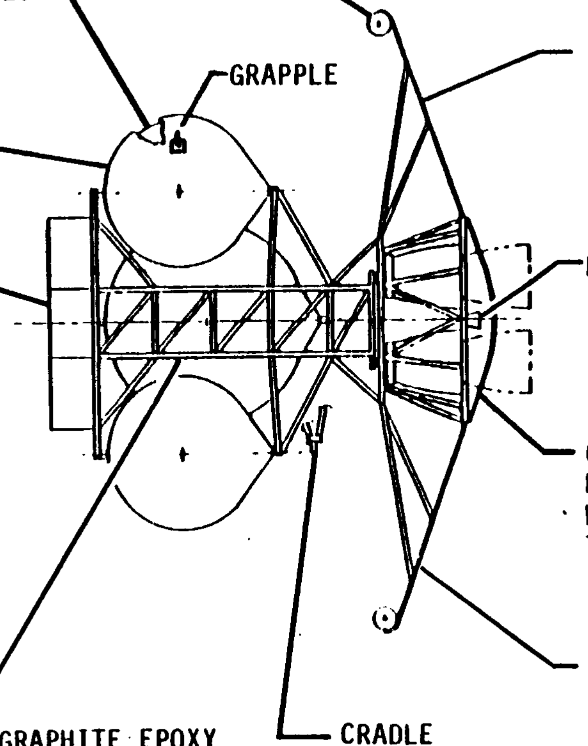
74 K SPACE BASED CRYO OTV

The flexible fabric brake OTV concept is shown in the figure. The brake/vehicle concept optimizes with a wide "squatty" tankage package. This usually suggest a central truss structure and subsequent side removable modular tankage.

The two main engines have extendable/retractable nozzles which protrude through openings in the nose of the aerobrake. These openings are closed during the aerocapture maneuver with actuated doors.

The vehicle and brake are intended to utilize a relatively low L/D (0.12) for control during the aerocapture maneuver and thus minimize the thermal loads on the fabric brake and therefore its weight. This results in a minimum weight OTV concept with adequate control capability during the aerotrajectory.

74 K SPACE BASED CRYO OTV



WEIGHT	
AEROBRAKE	1800
TANKS	1024
STRUCTURE	1897
ENVIRONMENTAL CTRL	203
MAIN PROPULSION	1403
ORIENTATION CONTROL	265
ELECTRIC SYSTEMS	535
G,N & C	160
CONTINGENCY (15%)	1093
DRY WEIGHT	8380
PROPELLANTS, ETC	74000
LOADED WEIGHT	82380

MARTIN MARIETTA

LRR

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.

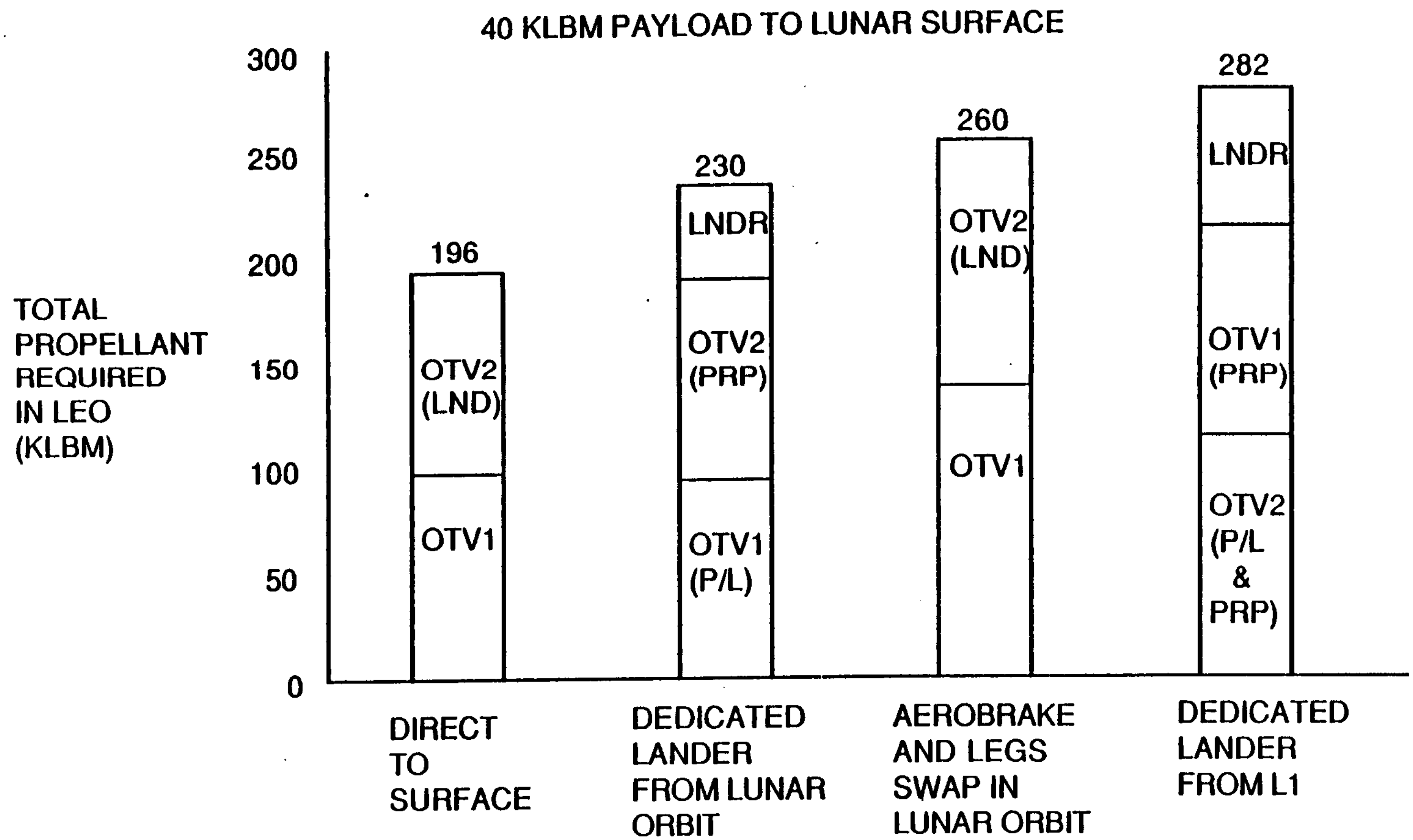
The dedicated lander approach uses two transfer vehicles to deliver the 40 Klbm payload and propellant for the lander to Lunar orbit. Then the propellant is transferred to the lander and the payload is delivered to the surface. The lander then returns to Lunar orbit.

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. Then on the return the landing stage would return to Lunar orbit to swap the landing legs back for its aerobrake and then 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 the figure. The most economical method of payload delivery to the Lunar surface appears to be the direct transfer to the surface. 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.

LUNAR TRANSFER COMPARISONS



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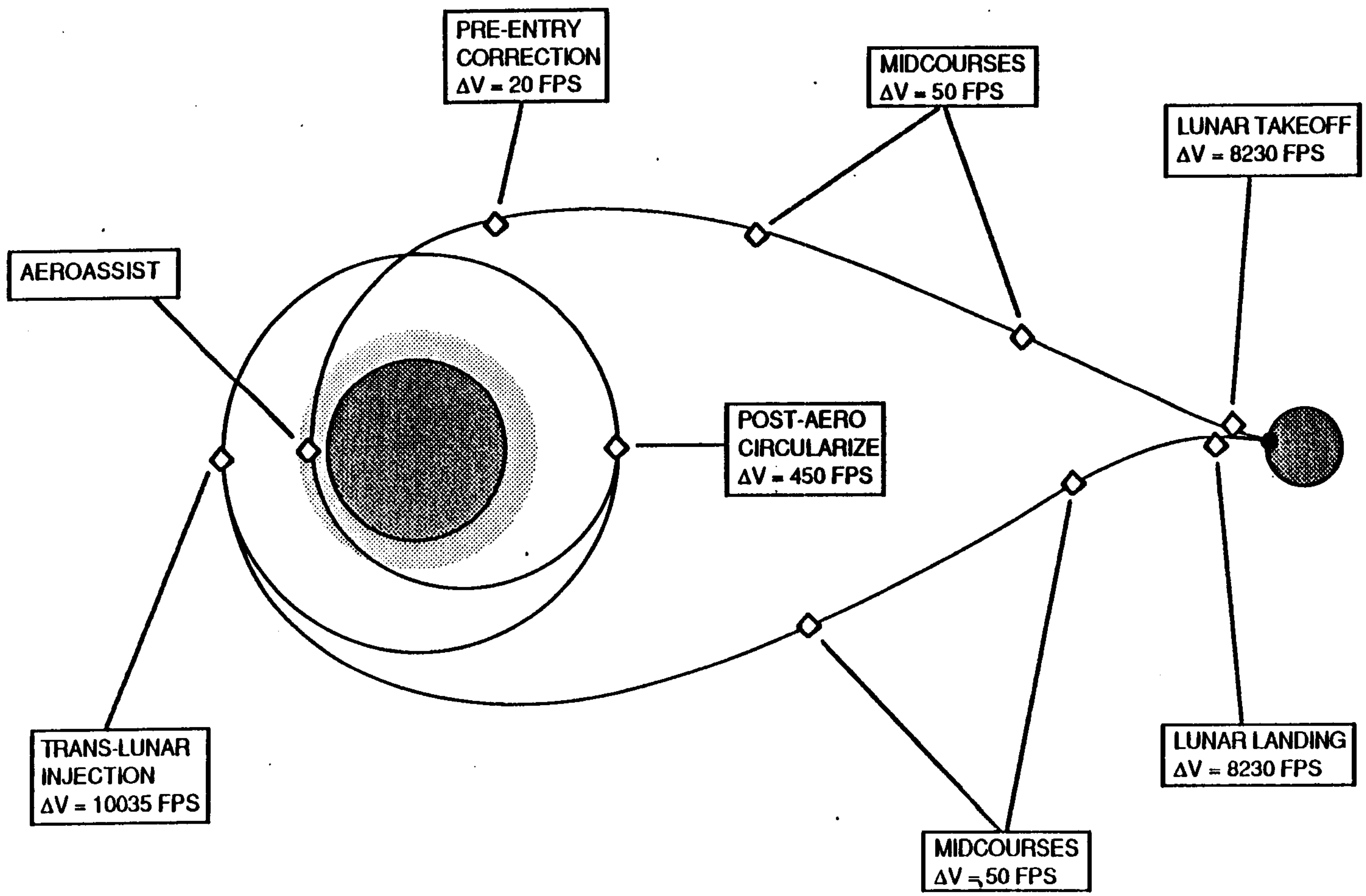
LUNAR PROFILE - DIRECT ASCENT

Various modes of lunar transfer were investigated for advanced missions. The first, shown here, is a direct transfer from low Earth orbit to the surface of the Moon followed by takeoff and direct injection into a trans-Earth trajectory. 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 Earth-moon flight. 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, with no assessment for gravity losses in descent.

LUNAR PROFILE - DIRECT ASCENT



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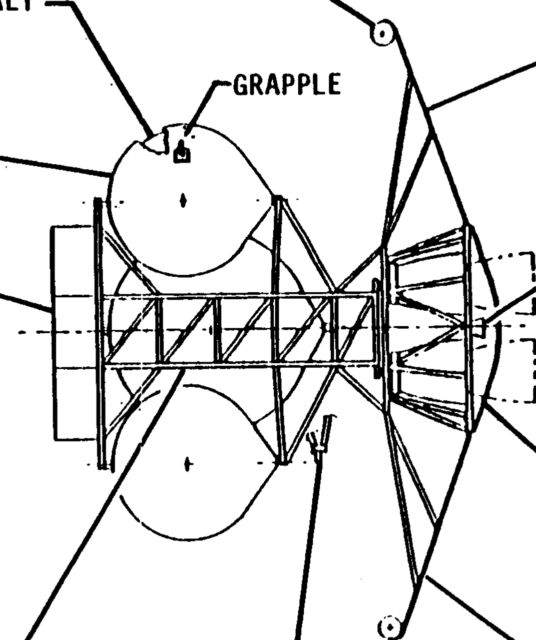
98 KLBM SPACE BASED LUNAR TRANSFER VEHICLE

The figure depicts the workhorse vehicle concept selected for delivering payloads, OTV's + payloads, etc. toward the Moon (the surface, Lunar orbit, or to a libration point). The vehicle was sized such that two stages of this concept (one containing the Lunar landing modifications) could deliver the 40Klbm 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.

98 KLBM LUNAR TRANSFER VEHICLE

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

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LUNAR LANDING GROUNDRULES

Several groundrules were assumed to apply to a Lunar landing scenario with an OTV. Some Lunar landings will most probably involve man, and due to the high cost of Lunar missions, engine out capability was imposed upon the configuration candidates. In addition, attitude misalignments were not allowed because of the desire 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.

The thrust level requirements associated with Apollo landings were adopted as ground rules 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.

LUNAR LANDING GROUNDRULES

- ENGINE OUT CAPABILITY

- NO ATTITUDE MISALIGNMENT

- CONTINUOUS THROTTLING CAPABILITY--BASED UPON APOLLO LANDING ACCELERATION REQUIREMENTS
  (FROM 0.31g AT DESCENT IGNITION TO 0.065g AT TOUCHDOWN)

- LEVEL SURFACE AND LANDING BEACONS FOR MAX CAPABILITY MISSIONS
  (REDUCES TIP-OVER RISK)

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THRUST LEVELS FOR LUNAR LANDING

The table 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.

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

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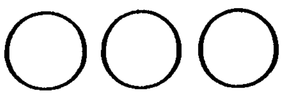
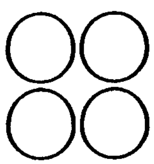
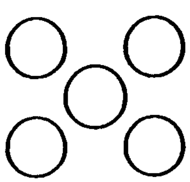
TP02

LUNAR LANDING ENGINE CONFIGURATIONS

Three (in-line), four, and five-engine configurations 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.

Four engines were chosen for Lunar landing applications. 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 (within a circular perimeter) and may offer the best growth path from a two engine system.

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
	.9797	0.66K - 11.7KLBF	18:1	- LOWEST RELIABILITY - LARGEST PATTERN - COMPLEX DESIGN AND CONTROL

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LRR

LUNAR LANDER DELTAS

Several modifications must be made in converting a space based OTV from GEO delivery capability, for instance, to delivering payloads to the Lunar surface. Four 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. Meteoroid protection requirements are not presently thought to differ much from those for LEO-GEO transfer.

LUNAR LANDER DELTAS FROM 98 K TRANSFER VEHICLE

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

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98 KLBM SPACE BASED LUNAR LANDING VEHICLE

The concept shown in the figure 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.

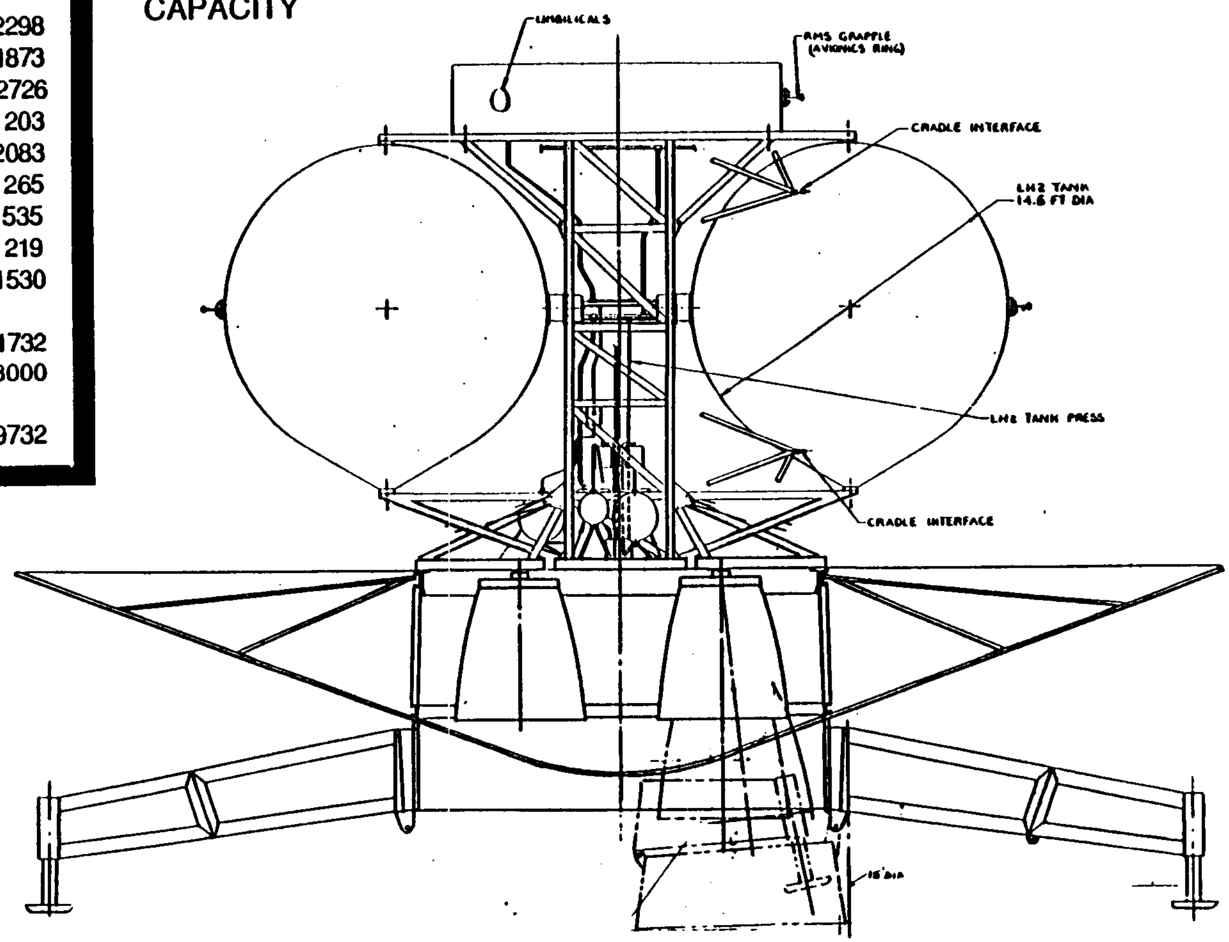
The figure shows a design concept for landing legs to accommodate the missions to the lunar surface. The legs fold under the aerobrake hard shell into a diameter compatible with delivery to LEO in the STS cargo bay. Therefore, the leg assembly could be attached to the vehicle after initial launch of 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.

98 KLBM LUNAR LANDER AND EARTH RETURN VEHICLE

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

98000 lbm PROPELLANT CAPACITY



45.2 FT DIA AEROBRAKE

4 - 17 KLBF ENGINES (THROTTLEABLE)

4 - LEG LANDING GEAR (ALUMINUM) LEFT ON SURFACE

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MARTIN MARIETTA

LRR

LUNAR LANDING ENGINE COMPARTMENT

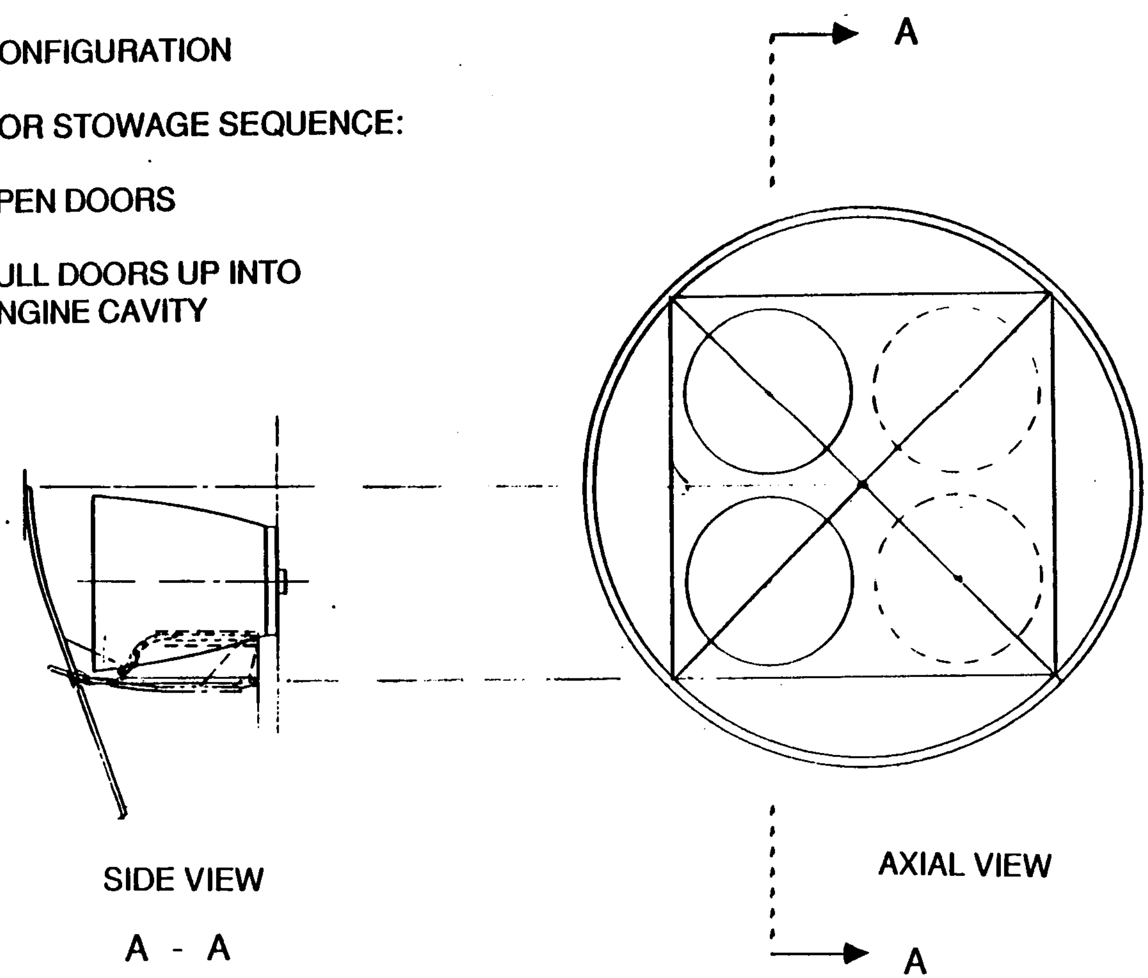
The figure shows the arrangement of the four engines recommended for lunar landings. The aerobrake doors are intended to rotate open to positions parallel to the engines' axes, and then withdrawn into the engine compartment alongside the engines during engine nozzle extension, engine operation, and nozzle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

LUNAR LANDING ENGINE COMPARTMENT

- 4 ENGINE CONFIGURATION
- ENGINE DOOR STOWAGE SEQUENCE:

    1) OPEN DOORS

    2) PULL DOORS UP INTO
    ENGINE CAVITY



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DEDICATED LUNAR LANDER

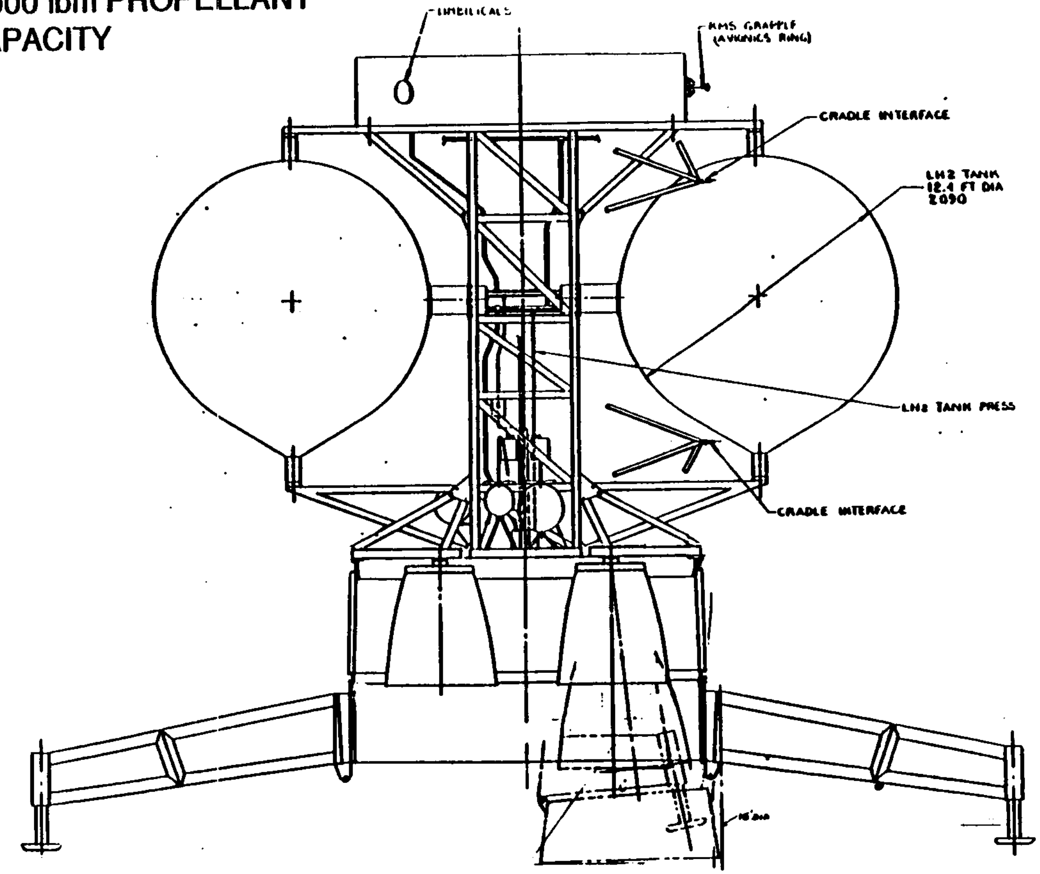
A Lunar lander concept was sized for the purpose of remaining in Lunar orbit and delivering to the surface the payload that the 98 Klbm vehicle could deliver to Lunar orbit. In other words the dedicated lander would be placed into Lunar orbit and serviced there (or perhaps on the surface) for use in 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.

DEDICATED LUNAR LANDER

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)

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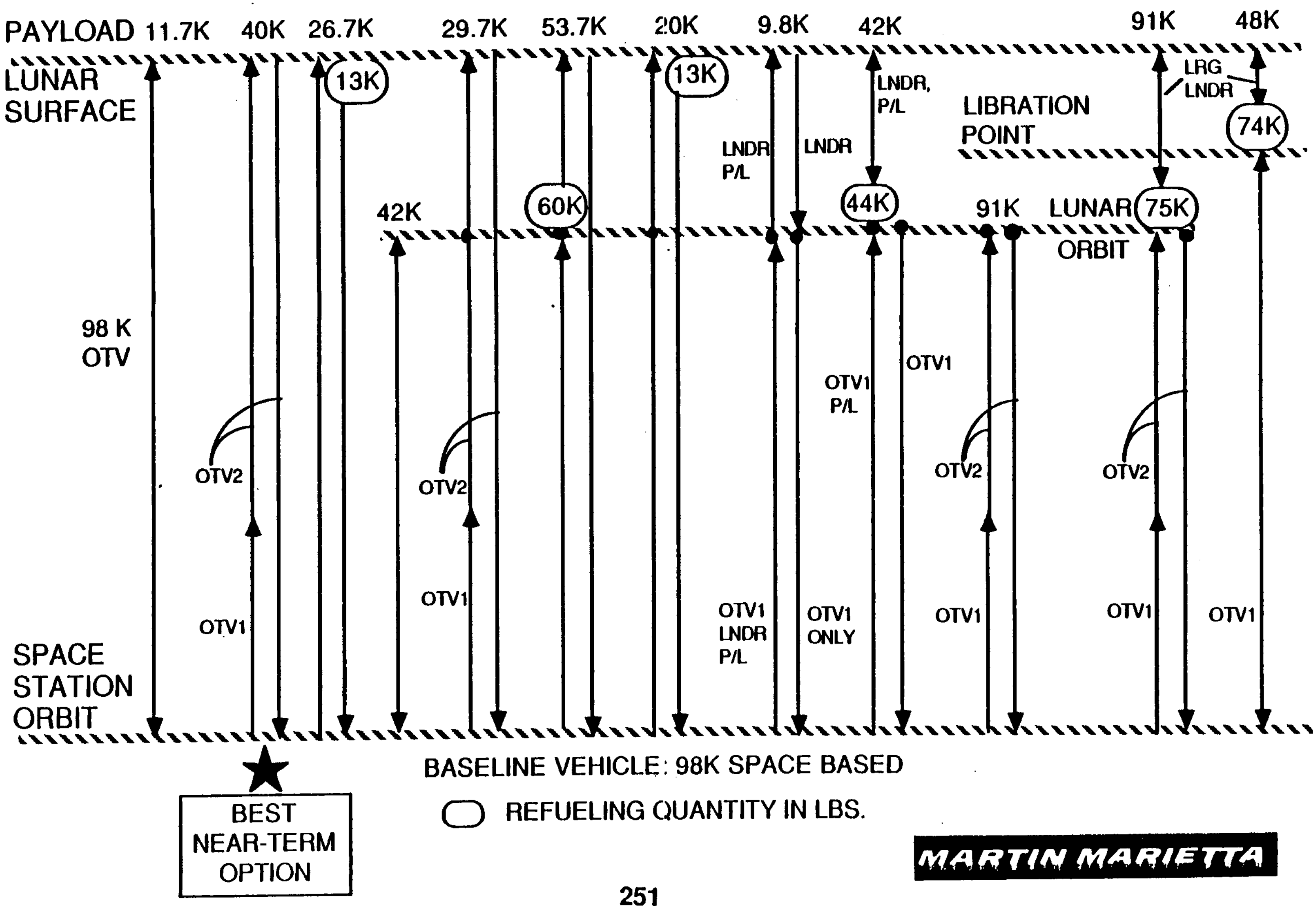
LUNAR DELIVERY OPTIONS

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 the figure along with the payload amounts to the surface that correspond to each of these options. 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 lander, a dedicated lander concept is shown with its function of delivering to the surface (from Lunar orbit or L1) a payload and then returning itself to its basing location.

LUNAR DELIVERY OPTIONS / PAYLOAD CAPABILITIES



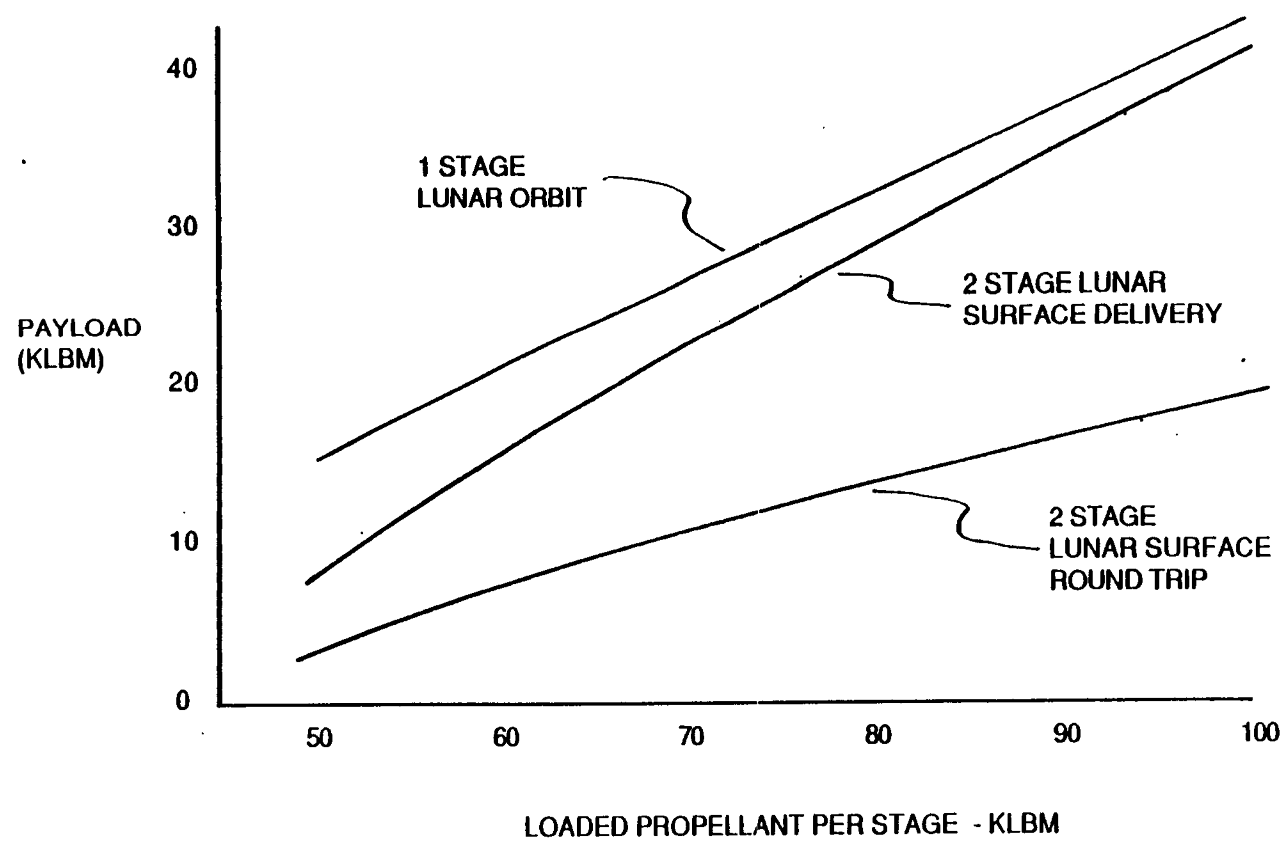
LUNAR OTV PERFORMANCE

Performance parametrics for the 98 klbm transfer vehicle and 98 klbm lander are shown in the figure. 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 using one transfer vehicle and one landing vehicle. 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. The third case is for delivery capability of one 98 klbm transfer vehicle from LEO to Lunar orbit.

LUNAR OTV PERFORMANCE

NOTE: SPECIFIC IMPULSE = 475 SEC



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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 three 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 over the full range of thrust.

CRYO ENGINE THROTTLING FOR LUNAR LANDING

- ABOUT 20:1 REQUIRED FOR LANDING

- 10:1 DEMONSTRATED AT PRATT & WHITNEY WITH RL10A-3-7
  (CONTINUOUS THROTTLING WITH NO MAJOR RL10 ENGINE MODS)

- GREATER THAN 10:1 WOULD REQUIRE MODS TO RL10-TYPE GAS/LIQUID ENGINE CYCLE
  (HEAT EXCHANGER UPSTREAM OF OXYGEN INJECTOR TO GASIFY LOX
  AT LOW THRUST LEVELS AND AVOID COMBUSTION INSTABILITIES)

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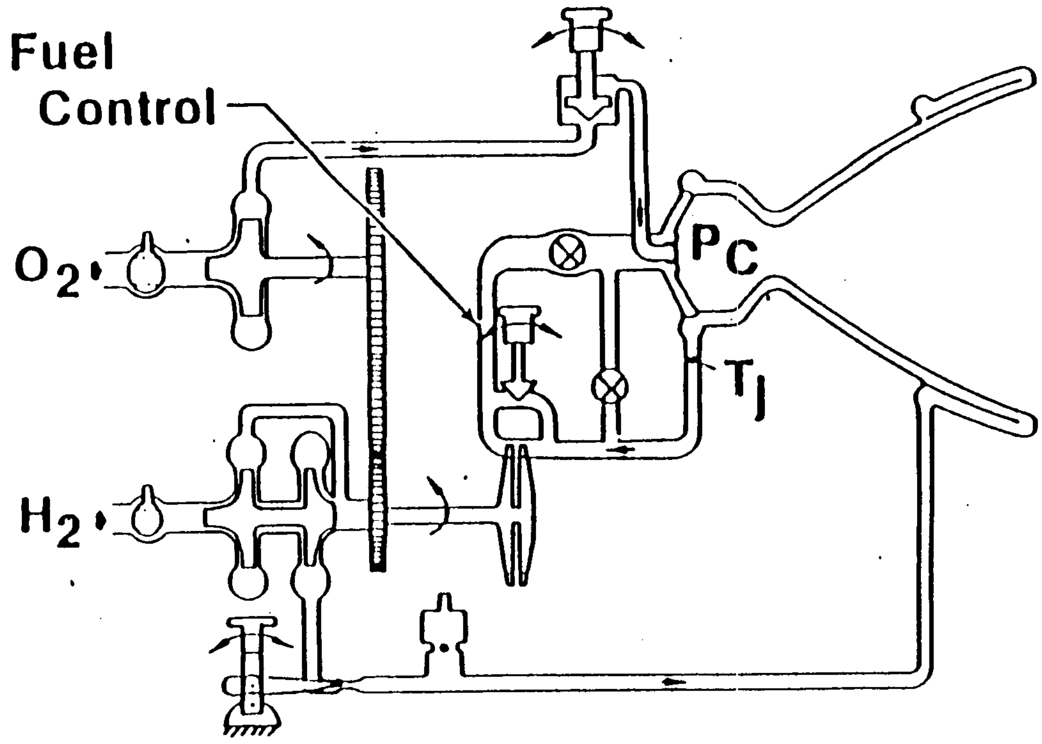
RL10A-3-7 PROPELLANT FLOW SCHEMATIC

The RL10A-3 engine has been successfully tested to demonstrate throttlability over a wide range of thrust. Throttling ratios of up to 10:1 have been demonstrated with no need for major engine modifications. For example, the three flow control devices shown in the figure can be electronically controlled to provide throttling. 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 in order to prevent instabilities in combustion. In other words, for low thrust operation of a large engine, the pump discharge pressure is relatively low. Thus, the delta P across the injector may be to 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.

RL10A-3-7 PROPELLANT FLOW SCHEMATIC



Oxidizer Control
- f (Throttle - Pumped Operation)
- f (P_C and T_j in THI)

Fuel Control
- f (Throttle - Pumped Operation)
- (Closed in THI)

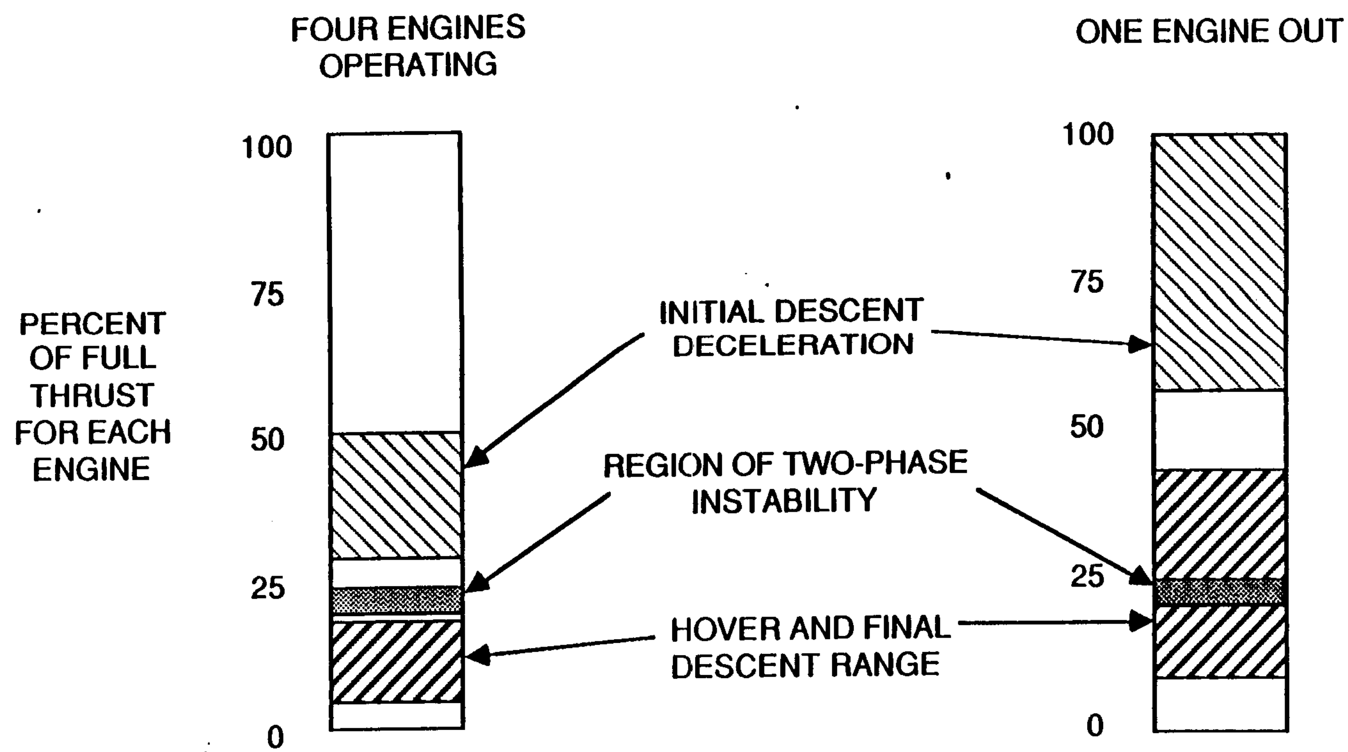
Cavitating Venturi
- f (Throttle - Pumped Operation)
- (Open in THI)

EXPANDER CYCLE THRUST RANGE DISCONTINUITY (4 ENG.)

The current RL10 engine cycle is capable of modification to perform 20:1 throttling but not of a purely continuous nature. In other words, due to a discontinuity caused by a required oxygen phase change (via a heat exchanger at low thrust levels in order to preclude combustion instabilities) 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 in this throttling range. 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 intended to provide gaseous oxygen to the injector and the combustion chamber. The region in between 20 and 25% of full thrust is where the 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 where four engines were selected, 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 thru the phase change discontinuity and into the gaseous oxygen operation range (the 5 to 20% range). The problem results when an engine-out condition occurs and the hover/final descent thrust range for the remaining engines exceeds or spans the thrust discontinuity. 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.

EXPANDER CYCLE THRUST RANGE DISCONTINUITY (4 ENG.)



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LRR

SOLUTIONS TO CRYO 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.

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).

c. Change the groundrules on engine-out so that when it occurs the contingency operation requires return to Earth 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).

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.

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.

SOLUTIONS TO CRYO THROTTLING DISCONTINUITY

A. HEAT EXCHANGER CIRCUIT AND ENGINE CONTROL SYSTEM MODIFICATIONS

B. MISSION PROFILE DESIGNED FOR CONTINGENCY (ENGINE-OUT) OPERATION, PROBABLY WITH PERFORMANCE DEGRADATION

C. CHANGE THE GROUNDRULES ON ENGINE-OUT (RETURN TO EARTH INSTEAD OF LANDING)

D. GAS/GAS ENGINE CYCLE (SUCH AS AEROJET TECHSYSTEMS HAS PROPOSED) THAT GASIFIES OXYGEN AT ALL THRUST LEVELS

E. USE SIX MAIN ENGINES (INSTEAD OF FOUR) IN ORDER TO PROVIDE FOR ENGINE-OUT

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SHUTTLE "C" OTV CHARACTERISTICS

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 the figure is 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 #STS flights, etc. from previous mission capture analyses. Therefore, the emphasis upon short length is necessary in this situation.

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

SHUTTLE "C" OTV CHARACTERISTICS

SHUTTLE "C": 100 KLBM LIFT AND 15 FT X 60 FT LONG PAYLOAD BAY

- SHORT: TANDEM TOROID CONFIGURATION SELECTED DUE TO LENGTH CRITICALITY FROM PREVIOUS MISSION CAPTURE EXERCISES
- SINGLE ENGINE: HIGHEST PERFORMING MAIN PROPULSION SYSTEM, UNMANNED APPLICATION, COST SAVINGS FROM SHORT LENGTH AND HIGH PERFORMANCE OUTWEIGH MISSION LOSS COSTS WITH NO ENGINE OUT CAPABILITY
- EXPENDABLE: NO RETURN CAPABILITY WITH SHUTTLE "C"
- 58.4 KLBM PROPELLANT CAPACITY: STACK WEIGHT EQUAL TO SHUTTLE "C" GROSS OF 100 KLBM
  → GEO PAYLOAD OF 26 K

PROPELLANT		DRY		PAYLOAD		ASE		GROSS
58.4	+	5.6	+	26	+	10	=	100 KLBM

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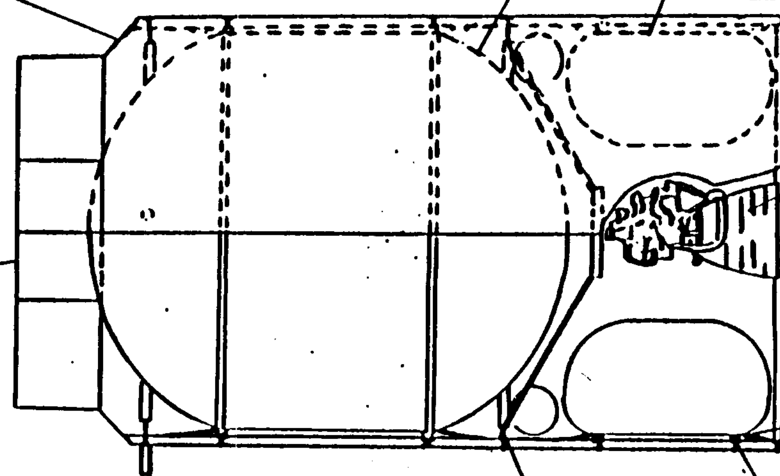
SHUTTLE-C EXPENDABLE OTV

This figure shows a cargo bay expendable OTV capable of derivering 15,000 lbm to GEO from Shuttle-C deployment in LEO. This concept is attractive because of its high performance and the vehicle's short length (compared to other cryogenic configurations).

The main contributor to the shortened length is incorporation of a toroidal LO2 tank in which the main engine is packaged. This concept was developed to emphasize short length while maintaining high performance, i.e., payload capability at minimum gross weight. According to the mission model assessment, the stage length plus ASE should not exceed 30 ft in order to minimize NSTS launch costs. In other words, the 30 ft payload capability and sufficient performance are the major desirable characteristics for a cargo bay OTV. This stage meets these criteria, i.e., 26.7 ft length, ASE length, and ASE packaging characteristics.

Minimum tank gages are 0.025 for the toroidal LO2 tank and 0.025 for the LH2 tank. The two tanks are protected by a cylindrical debris shield of graphite/epoxy, supported by longerons and ring frames of the same material. Each tank is attached to the longerons and frames by fiberglass/epoxy struts which accommodate the temperature differences. The avionics units have been mounted on an avionics ring that also serves as the payload interface. Ag-Zn batteries provide the power source, and the propulsion unit is a RL10-A engine.

SHUTTLE-C EXPENDABLE OTV (15' DIA)



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

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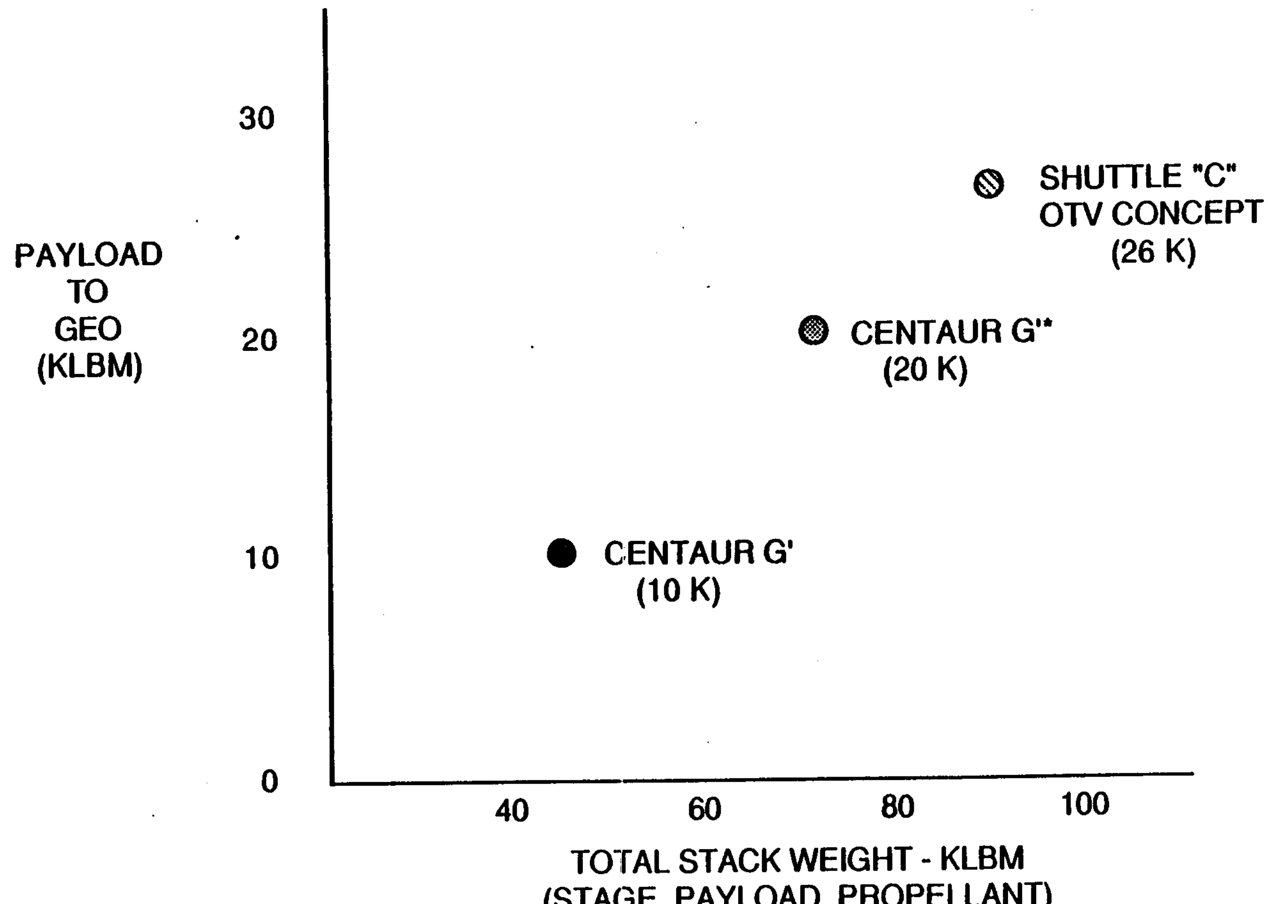
MANNED SPACE SYSTEMS

EXPENDABLE VEHICLE COMPARISON

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

EXPENDABLE VEHICLE COMPARISON



* CENTAUR REQUIRES STRUCTURAL MODS
(MAX CAPABILITY TODAY = 10 K P/L)

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Structural Subjects to be Covered

ACC Expendable OTV Definition

-- AL-LI TANKS -- OPTION
-- IOC ENGINE -- OPTION
-- METEOROID SHIELD
-- COMPOSITE ACC
-- BATTERY SELECTION

LCV Expendable OTV

-- ASE FOR SIDE MOUNT LCV
-- ASE FOR INLINE LCV
-- AIRFRAME ANALYSIS

Ground Base Cryogenic Reusable OTV

-- AEROBRAKE
-- TANKS
-- METEOROID SHIELD

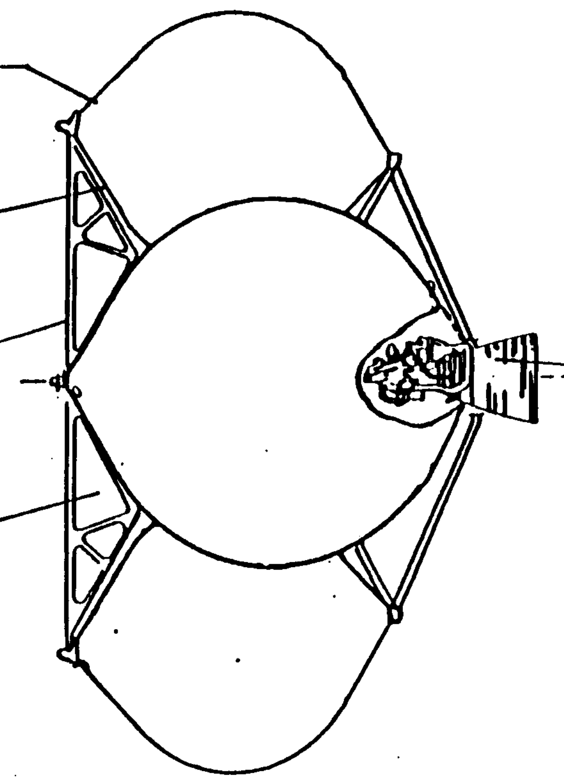
ACC EXPENDABLE OTV BASELINE

The general arrangement and weight breakdown for our selected expendable OTV transported in the ACC are shown on the facing viewgraph. The expendable OTV is based on the same arrangement as the groundbased 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.

The major differences are: aerobrake removal, Al 2219 tanks instead of Al-Li 2090 tanks, a RL10-A engine, and Ag-Zn batteries in place of the fuel cell system. Some GN&C equipment has been removed, or will be, replaced by a smaller system.

The total dry weight of the ACC expendable OTV is 4189 lb.

ACC EXPENDABLE OTV BASELINE



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

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MANNED SPACE SYSTEMS

ACC EXPENDABLE ENHANCEMENTS

This chart shows the enhancements and weight breakdown for the ACC expendable OTV.

The first modification to the vehicle is the replacement of the Al 2219 tanks with Al-Li 2090 tanks which results in a weight saving of 349 lb.

The second modification incorporates an IOC engine into the propulsion system which saves 424 lb from the baseline vehicle and 65 lb from the first enhanced vehicle.

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

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MANNED SPACE SYSTEMS

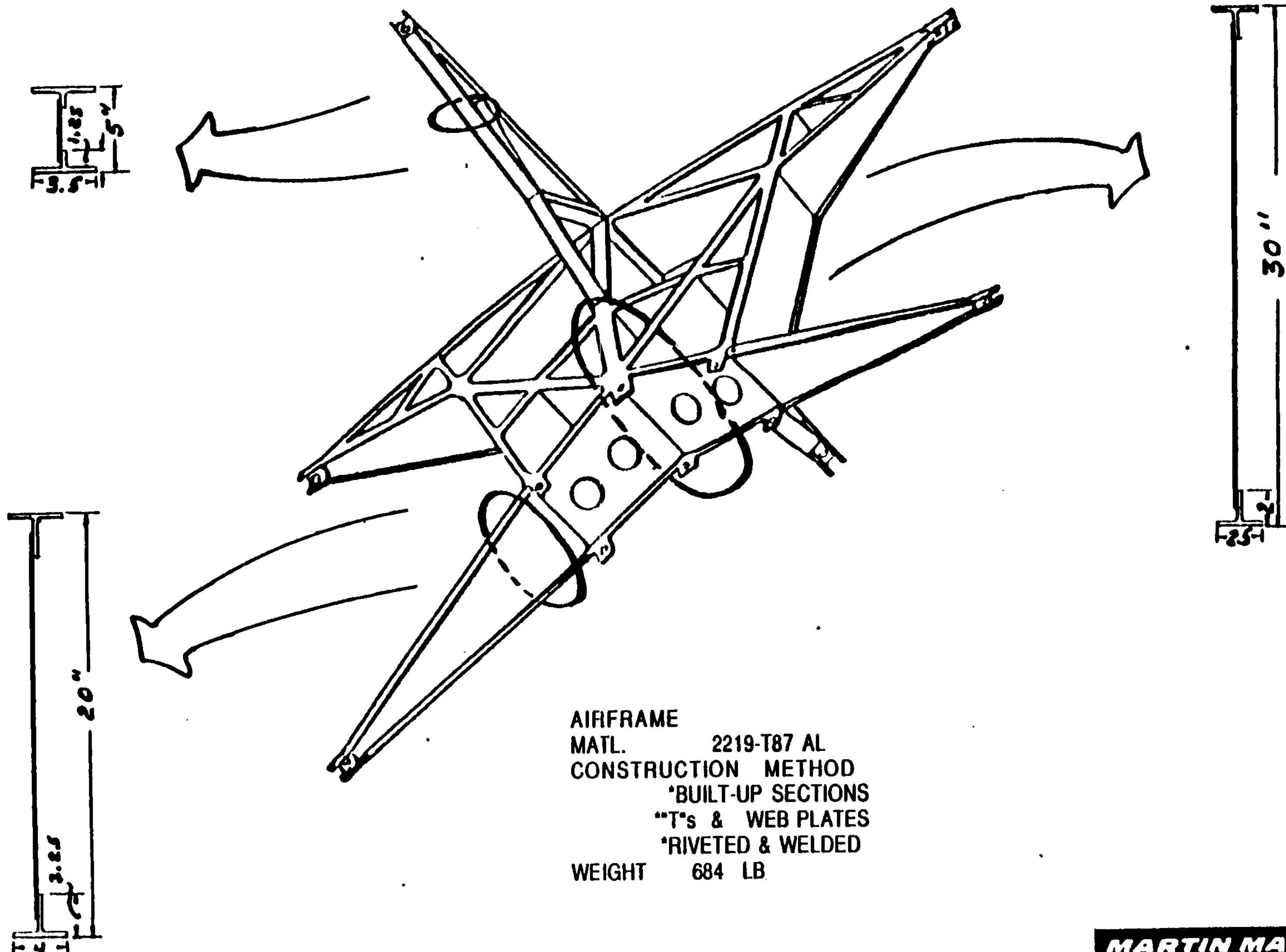
AIRFRAME ALUMINUM

The 2219 aluminum airframe is a multi-member truss work based on the volume and weight efficient principals suggested by Larry Edwards (NASA HQ). 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 strucuture.

This figure shows a view of the airframe and some typical cross sectional views of the builtup members. The airframe weighs 684 lb, including fittings and attachments.

AIRFRAME - ALUMINIUM



AIRFRAME
MATL. 2219-T87 AL
CONSTRUCTION METHOD
*BUILT-UP SECTIONS
*"T"s & WEB PLATES
*RIVETED & WELDED
WEIGHT 684 LB

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MANNED SPACE SYSTEMS

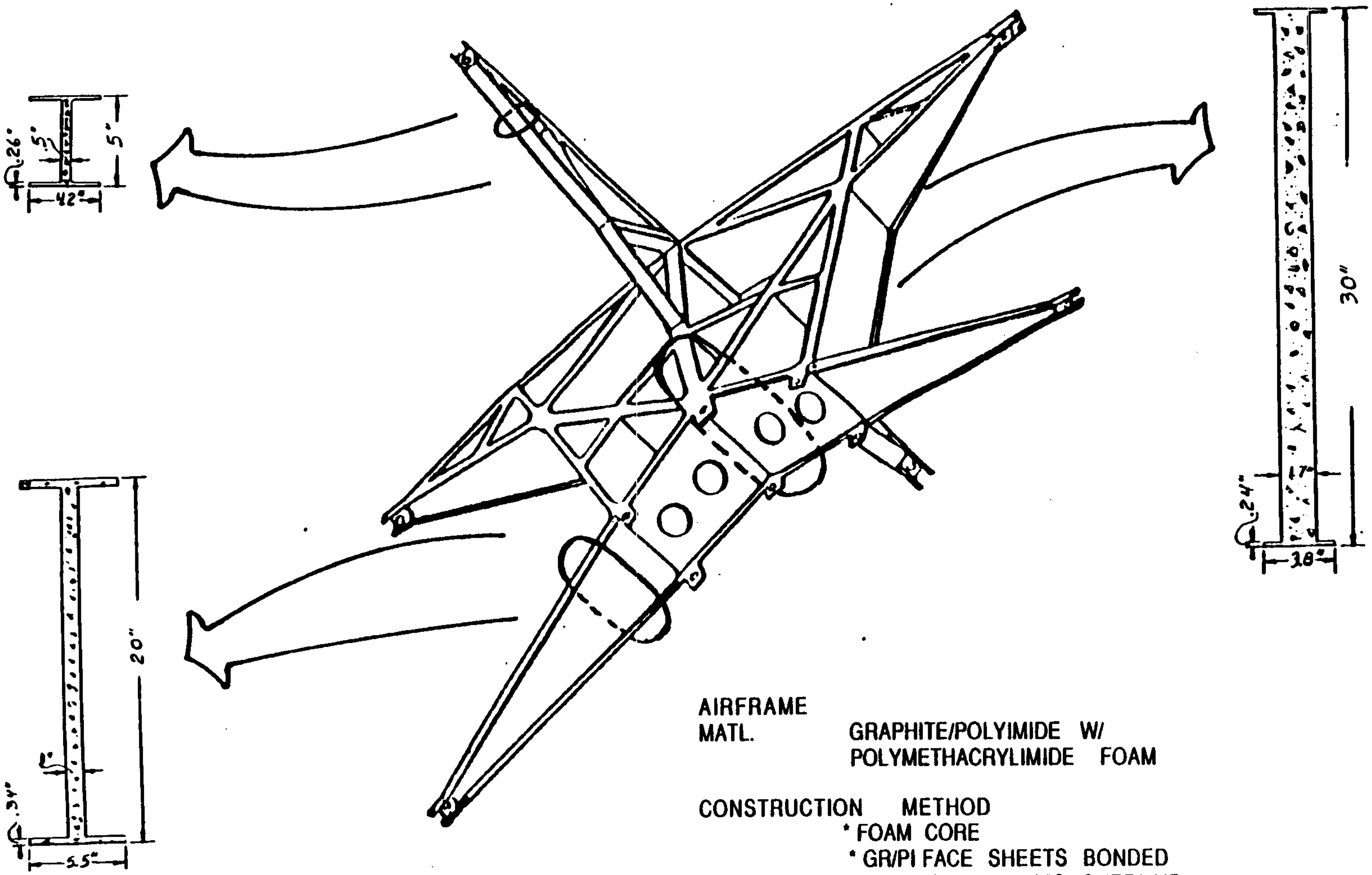
AIRFRAME COMPOSITE

As part of the weight optimization effort, the air frame was recalculated using Graphite/Polymide (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.

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.

This figure shows a view of the airframe and some typical cross sectional views of the builtup members. The airframe weights 454 lb, including fittings and attachment. 230 lb are saved by using a composite structure instead of a similar aluminum structure.

AIRFRAME - COMPOSITE



AIRFRAME
MATL. GRAPHITE/POLYIMIDE W/ POLYMETHACRYLIMIDE FOAM

CONSTRUCTION METHOD
- FOAM CORE
- GR/PI FACE SHEETS BONDED
- GR/PI JOINT PLATES OVERLAID and BONDED

WEIGHT 454 LB

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MANNED SPACE SYSTEMS

MAIN PROPELLANT TANKS

The procedure for determining propellant tank wall thickness is shown on the chart. 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.

This chart shows the calculation results for the required proof test pressures.

MAIN PROPELLANT TANKS

- Design Process

Pressure Requirements → Proof Test Requirements → Internal Loads Stress Resultants → Wall Thickness Sizing → Yield-Ult Check

Pressure Requirements	Proof Test Requirements	Internal Loads Stress Resultants	Wall Thickness Sizing	Yield-Ult Check
- Ullage Pressure - Inertia G's (Head Pressure)	- Proof Factor - Fracture Data	- Geometry - Membrane Analysis - Bosor V	- Tension - Compression	- Margins

- 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

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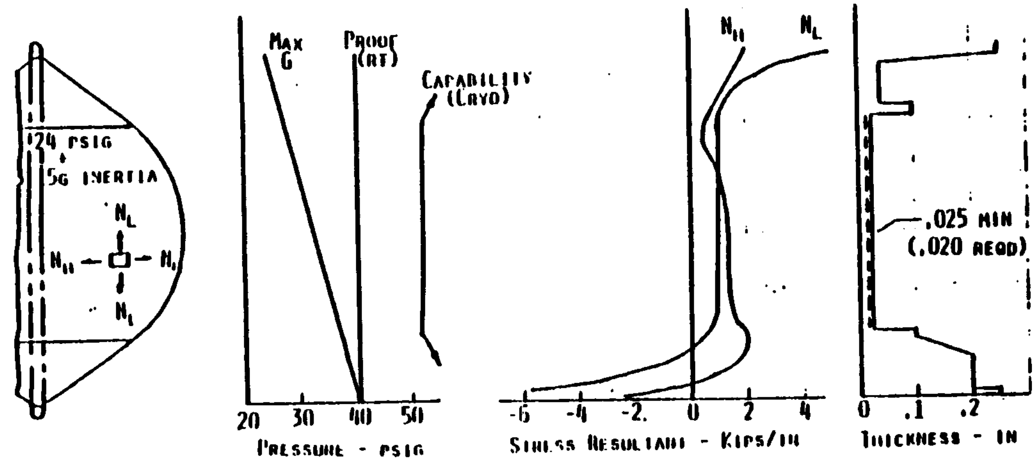
MANNED SPACE SYSTEMS

LO2 TANK DESIGN

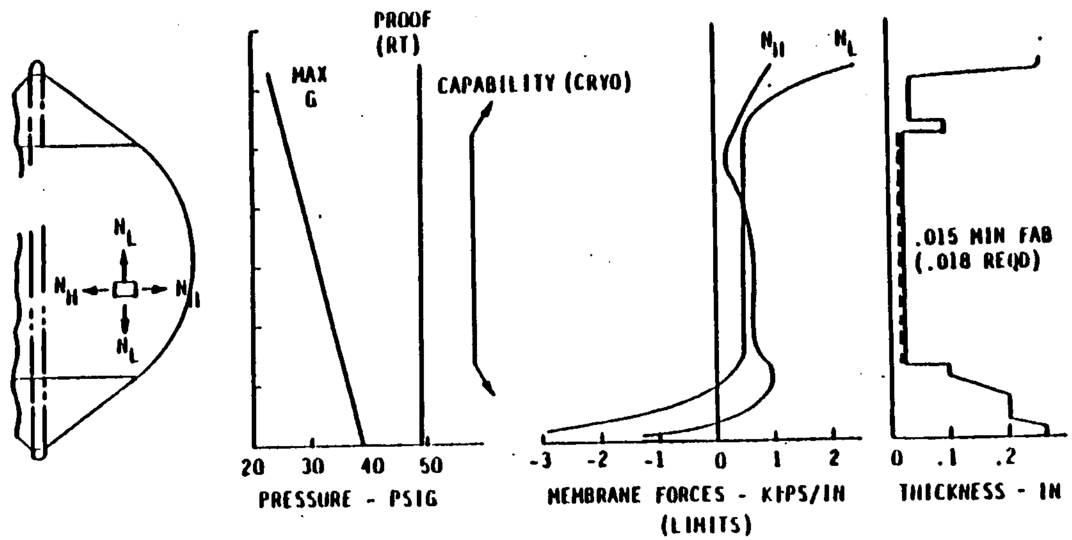
This chart shows the results of the LO2 tank stress analysis (using the BOSOR shell program), including capability margin, membrane force, and wall thickness. The tank was originally sized using AL 2219 and a 0.025-in. minimum gage was recommended. As a weight optimization alternative, Al-Li 2090 was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

LO2 TANK DESIGN

2219



2090



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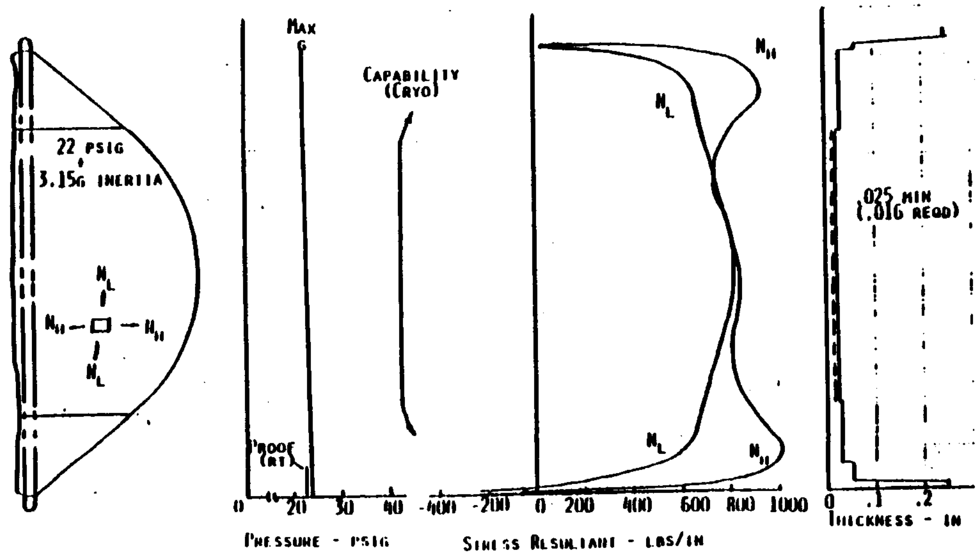
MANNED SPACE SYSTEMS

LH2 TANK DESIGN

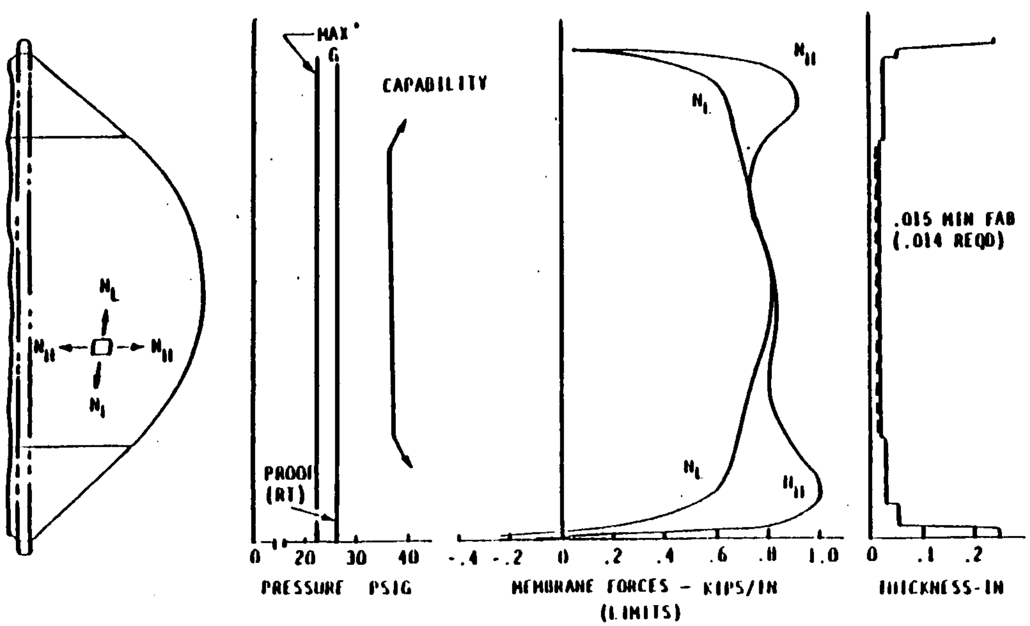
This chart shows the results of the LH2 tank stress analysis (using the BOSOR shell program), including capability margin, membrane force, and wall thickness. The tank was originally sized using AL 2219 and a 0.025-in. minimum gage was recommended. As a weight optimization alternative, Al-Li 2090 was considered and the minimum gage was reduced to 0.015.

LH2 TANK DESIGN

2219



2090



MARTIN MARIETTA
MANNED SPACE SYSTEMS

OTV DEBRIS/METEOROID ASSUMPTIONS

To meet a proposed 0.999 probability of no damager 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, it is only necessary to vary the bumper thickness and location to achieve appropriate levels of penetration resistance. Additional thickness of the pressure wall or thermal blanket will not be analyzed.

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 $140 m^2$, 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.

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

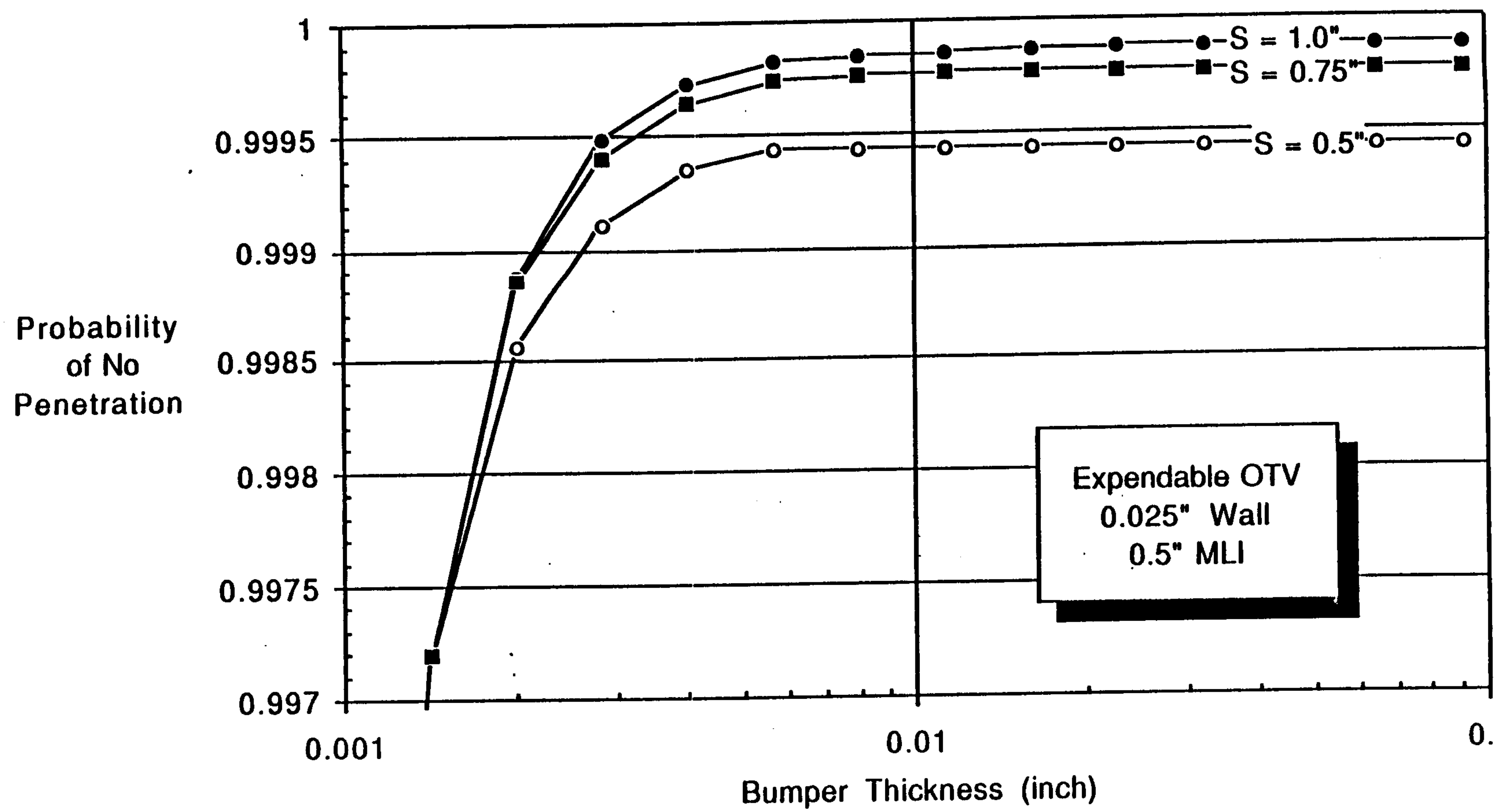
MARTIN MARIETTA
MANNED SPACE SYSTEMS

PROBABILITY CALCULATION

Bumper thickess has a strong influence on the probability of penetration for thin bumpers. If the incident particle is not broken up by the bumper, than 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.

The size of a meteoroid and the size of debris which can be stopped by each design is used to calculatge 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.

OTV DEBRIS/METEOROID EXPENDABLE



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MANNED SPACE SYSTEMS

OTV DEBRIS/METEOROID BUMPER SIZE .

For an expendable vehicle, a layer of Beta Cloth will suffice as a bumper with a 0.6-in. standoff.

Although expected increases in the space debris and meteoroid environment will affect these numbers, these are projections in the environment, 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 or 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.

OTV DEBRIS/METEOROID BUMPER SIZE

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

MARTIN MARIETTA
MANNED SPACE SYSTEMS

DACC COMPOSITE SHROUD

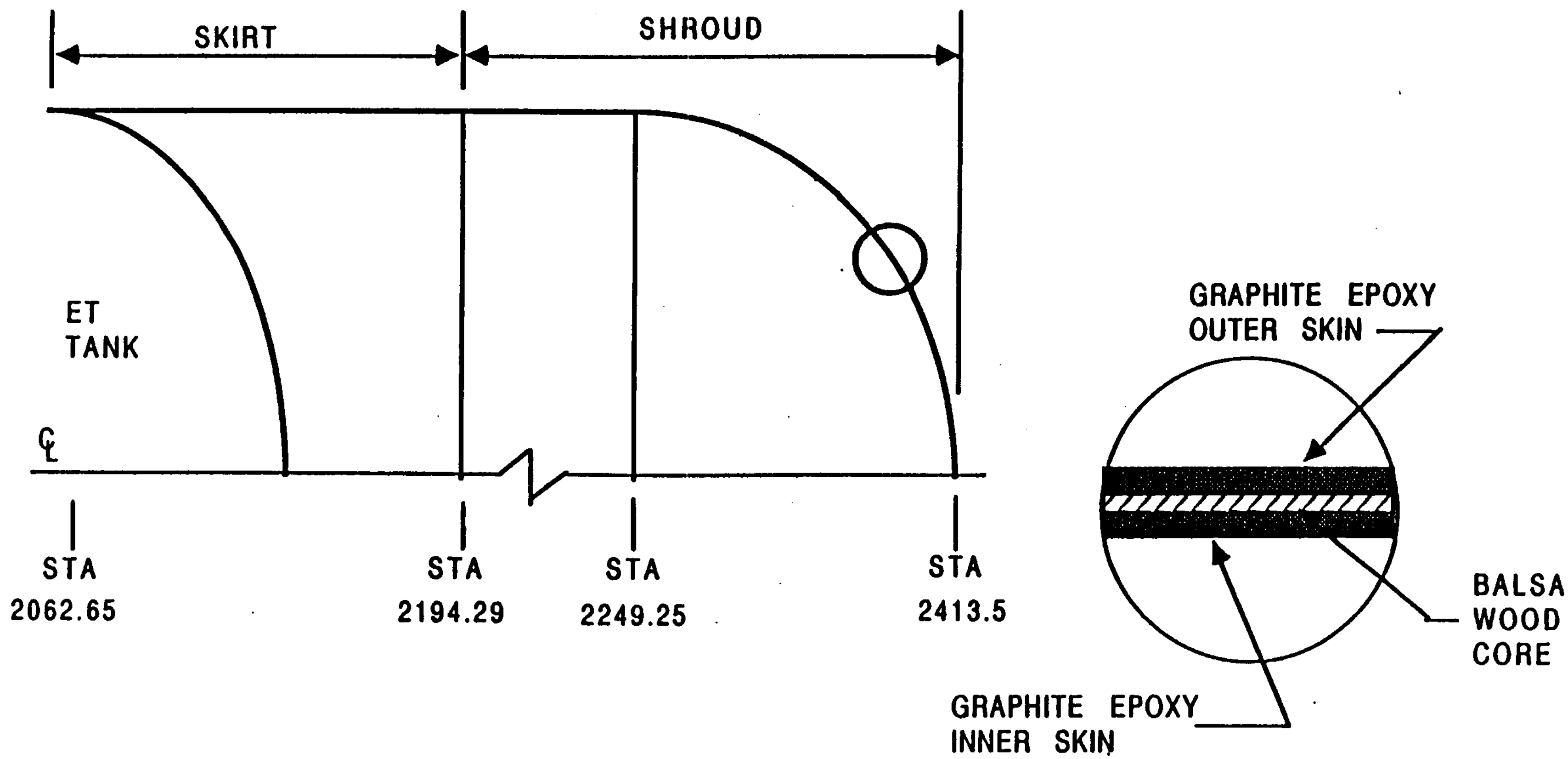
In the baseline design, the skins will be a sandwich structure. The inner and outer skins will be filament wound AS4W-12K graphite fiber using HBRF 55A epoxy resin. This composite will have 50% fiber by volume. The lamina properties for this composite are: the modulus in the fiber direction is 17.21×10^6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

The baseline 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 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.

DACC COMPOSITE SHROUD



MARTIN MARIETTA
MANNED SPACE SYSTEMS

DACC SHROUD WEIGHT COMPARISON

This chart shows the weight breakdown and comparison of the unpressurized 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.

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.

Translating the two different design concepts into a weight difference produces a net weight increase of 203 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.

DACC SHROUD WEIGHT COMPARISON

	METAL PRESSURIZED WEIGHT (LB)	COMPOSITE UNPRESSURIZED WEIGHT (LB)	DETLAS WEIGHT (LB)
SKIRT			
STRUCTURE	2556	2556	O
THERMAL PROTECTION	173	173	
AVIONICS/ELECTRICAL	152	152	
PROP/MECH	125	125	
ORDNANCE	23	23	
CONTINGENCY	454	454	
SUBTOTAL	3483	3483	O
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

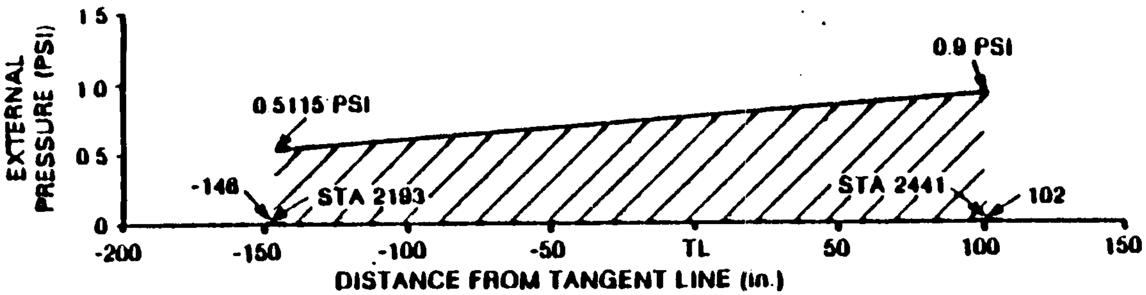
MARTIN MARIETTA
MANNED SPACE SYSTEMS

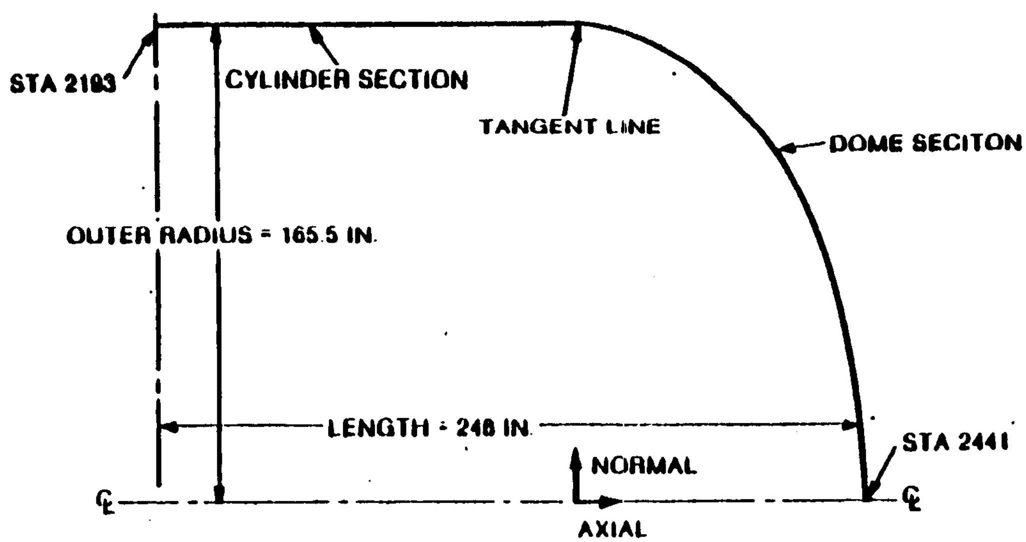
PRESSURE AT IGNITION AND SHROUD GEOMETRY

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.

The sketch on the opposite page shows the external pressure distribution on the shroud.

PRESSURE AT IGNITION AND SHROUD GEOMETRY





HERCULES

MARTIN MARIETTA
MANNED SPACE SYSTEMS

STS STRUCTURAL DESIGN REQUIREMENTS

Structural requirements are outlined on the opposite page. 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.

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

HERCULES

MARTIN MARIETTA
MANNED SPACE SYSTEMS

STRUCTURAL ANALYSIS SUMMARY

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. The minimum FS is 1.4 at the shroud-to-skirt joint under acceleration loading and in the shroud cylinder under external pressure.

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

HERCULES

MARTIN MARIETTA
MANNED SPACE SYSTEMS

BATTERY CANDIDATES

This table 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 capa-city 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.

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.

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 -60 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

MARTIN MARIETTA
MANNED SPACE SYSTEMS

BATTERY SELECTION

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 following five design criteria: (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.

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 ($\leq$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.

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)

MARTIN MARIETTA
MANNED SPACE SYSTEMS

LCV EXPENDABLE OTV

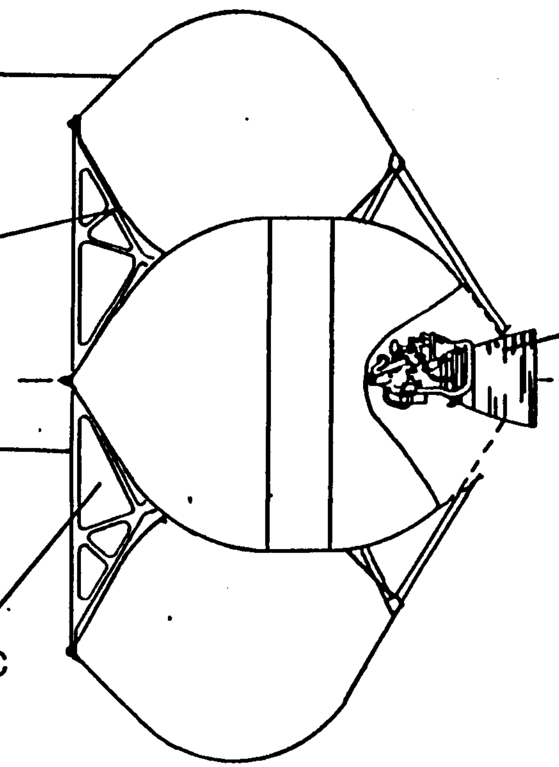
This viewgraph shows the general arrangement and breakdown of our selected expendable configuration which will be used in either a sidemount or inline LCV payload element.

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.

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.

LCV EXPENDABLE OTV



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

ASE FOR 50K OTV - SIDEMOUNT CONFIGURATION

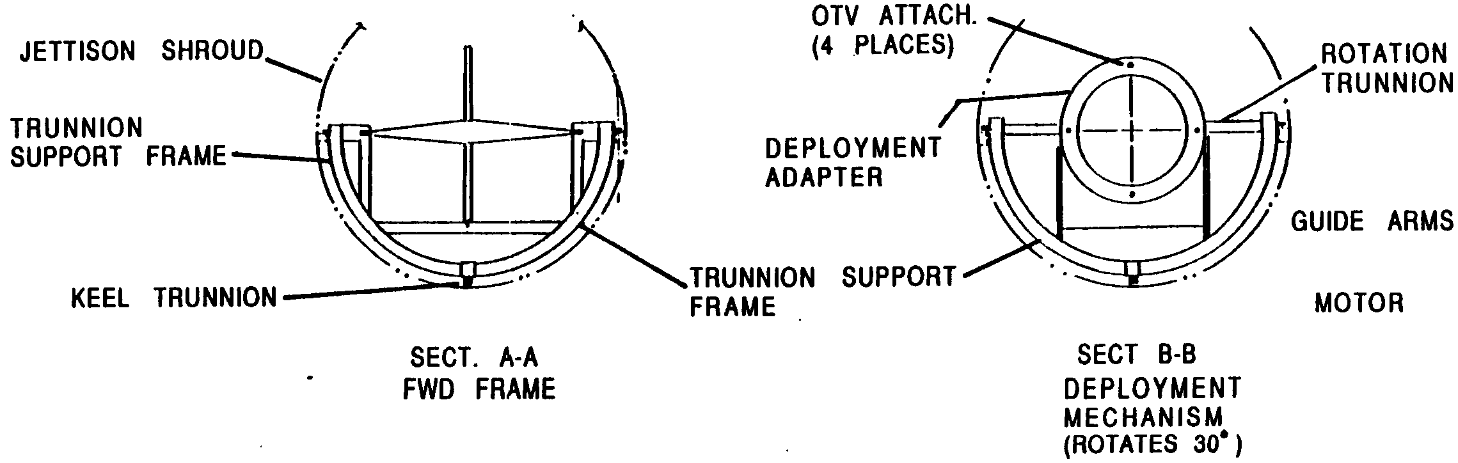
This viewgraph shows the ASE components and weight breakdown for the LCV expendable OTV Sidemount configuration.

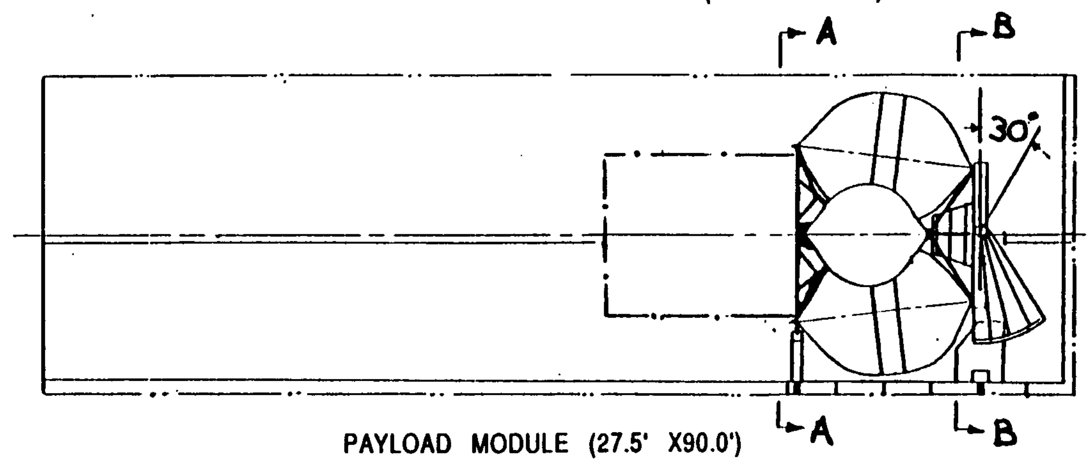
The ASE is designed to support and launch the OTV from a 27.5 ft x 90 ft unmanned Payload (P/L) Module.

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 for 50K OTV SDV SIDE-MOUNT CONFIGURATION





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

MARTIN MARIETTA
MANNED SPACE SYSTEMS

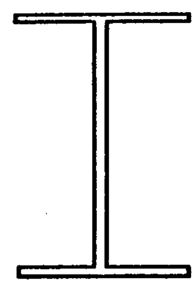
ASE FOR 50K OTV - INLINE CONFIGURATION

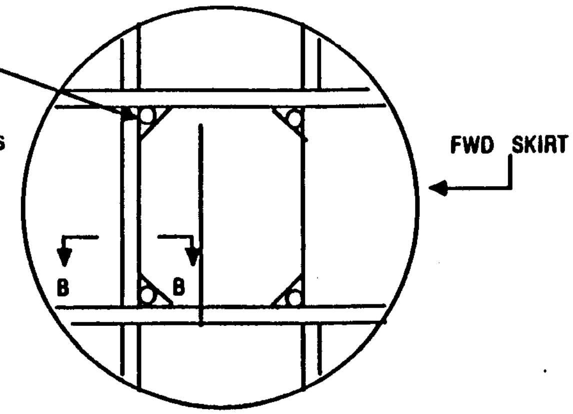
This figure shows the ASE components and weight breakdown for the LCV expendable OTV Inline configuration. The ASE equipment (skirt, support beams, and hardware) is the same structure as on the ACC.

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.

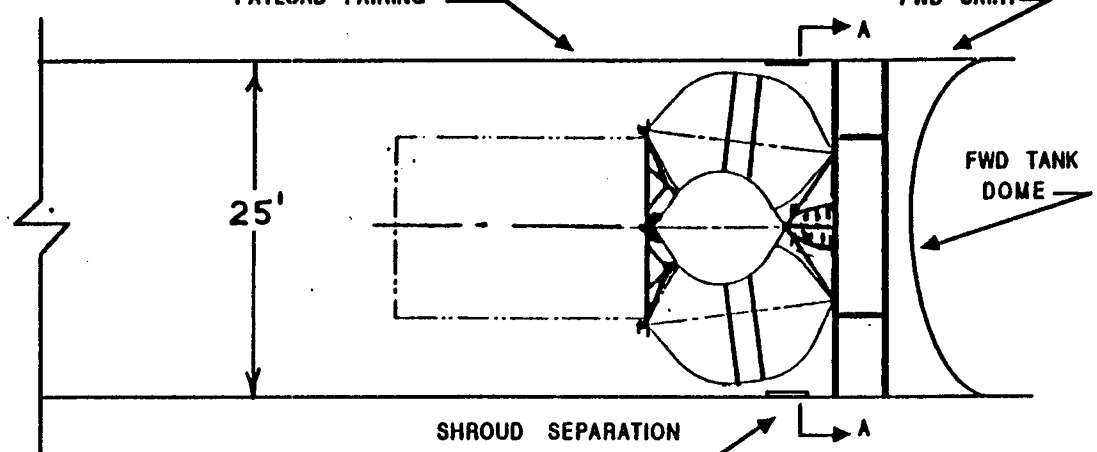
ASE for 50K OTV LCV IN-LINE CONFIGURATION


SECTION B-B
SUPPORT BEAM CROSS SECTION



SECTION A-A
OTV ATTACHMENT & SUPPORT BEAMS

ASE	WEIGHT (LB)
SKIRT	1746
FRAMES	810
ATTACH HRDW	108
PROP/MECH	125
AVIONIC/ELEC	152
ORDNANCE	23
CONTINGENCY	445
TOTAL	3409



MARTIN MARIETTA
MANNED SPACE SYSTEMS

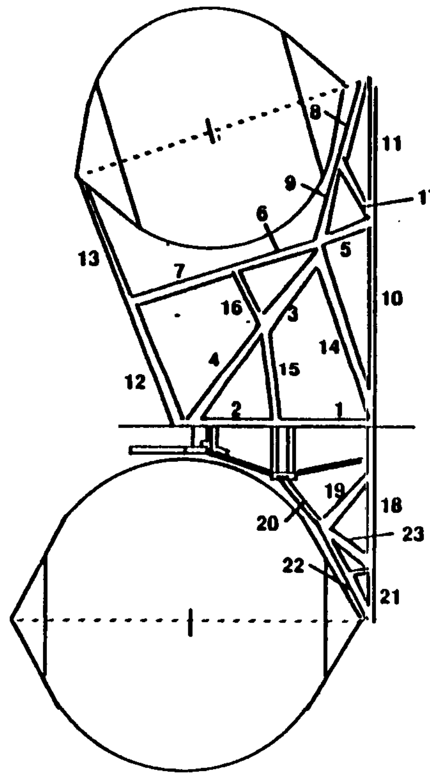
COMPARISON OF OTV DESIGN CAP LOADS

The viewgraph shows a tabulation of the old and new cap loads. To maintain the structural capability of the rack, new loads were designated to transfer the payload axial (X) and Y and Z moment loads directly into the rack support structure. This is accomplished byplacing a 6-in. diaameter tube along the axis of the fuel tanks (as shown in the figure).

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

COMPARISON of LCV vs ACC OTV CAP LOADS

LO2
TK AXLE
6" DIA



LH2
TK AXLE
6" DIA.

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

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

MARTIN MARIETTA
MANNED SPACE SYSTEMS

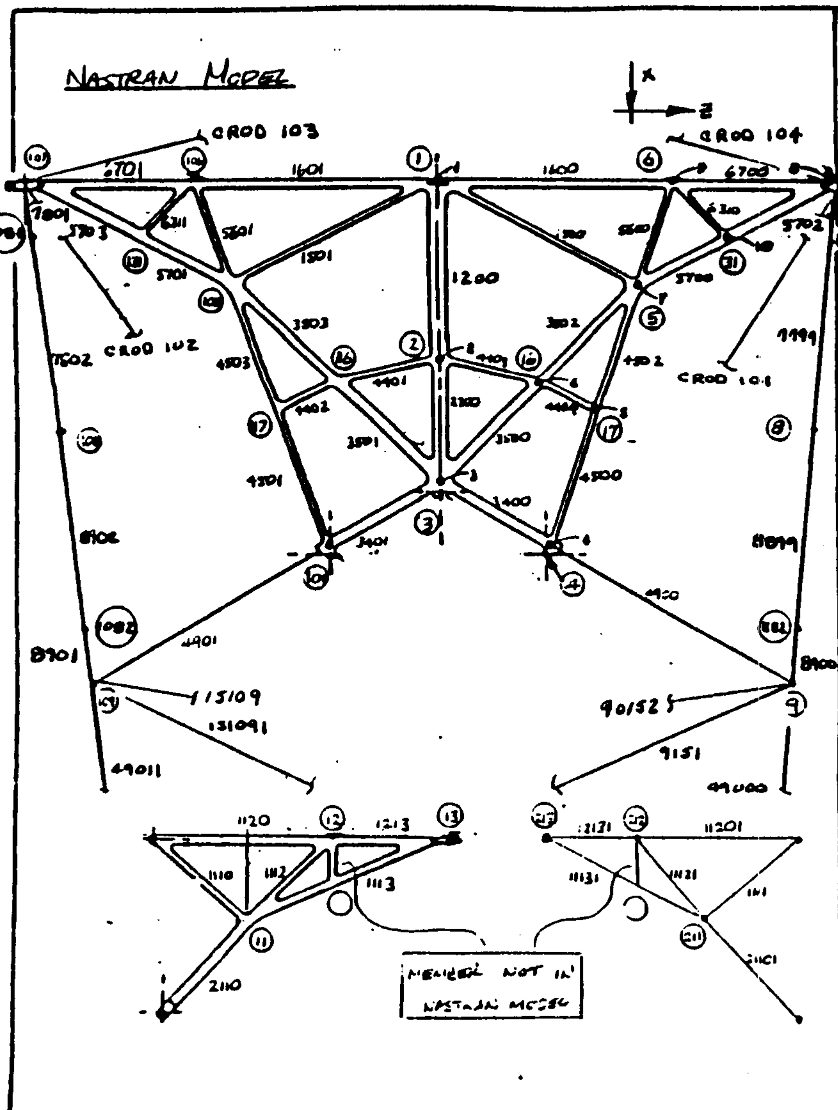
LCV OTV AIRFRAME STRUCTURAL ANALYSIS

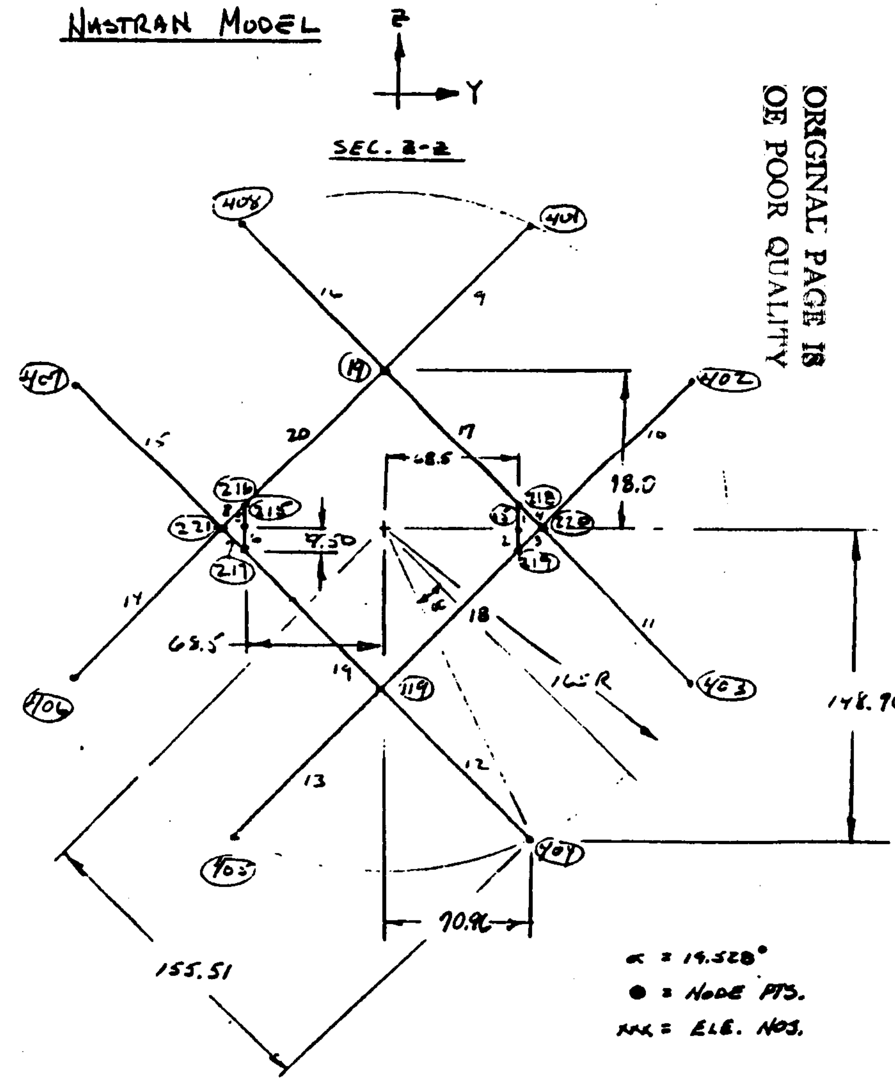
A stress analysis of the new/modified OTV rack and support structure was made to substantiate the integrity of the structure. The two principal requirements of the new design are:

(1) The new/modified rack must react the payload (14 klb) and fuel tank loads, whereas the current rack is designed to react any 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.

LCV OTV AIRFRAME STRUCTURAL ANALYSIS





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MARTIN MARIETTA
MANNED SPACE SYSTEMS

LCV OTV AIRFRAME STRUCTURAL ANALYSIS

Preliminary beam sizes were calculated by hand or 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 (see NASTRAN Model, CRODS 101 through 104.) Note that these tubes could be part of the payload support structure. The aft ends of the fuel tanks are also tied together with four struts (see NASTRAN Model, element numbers 15109, 151091, 90152, and 9451). These struts remain with the OTV to stabilize the fuel tanks.

LCV OTV AIRFRAME STRUCTURAL ANALYSIS

NASTRAN MODEL

ELEMENT	NODE 1	NODE 2	A	I_1	I_2
200	1	2	.956	107.8	1.055
2300	2	3	1.61	181.5	.871
2502	5	16	3.03	151.1	2.05
3500	16	3	3.03	151.1	2.05
3400	3	4	4.516	18.7	18.7
4500	4	17	3.49	28.2	2.09
4502	17	5	.668	27.9	.50
4900	4	9	4.516	18.7	18.7
1600	1	6	2.94	171.4	5.65
6700	6	7	2.94	171.4	5.65
1500	1	5	1.62	101.7	2.92
5700	5	31	3.49	85.4	1.21
5702	31	7	3.49	28.2	2.09
5600	6	5	.462	14.0	.18
[illegible]	6	31	1.85	4.80	1.62
1110	11	1	1.0	3.5	.2
1120	1	12	1.85	4.8	1.62
1212	12	13	1.85	4.8	1.62
2110	2	11	1.85	4.8	1.62
1113	11	12	1.85	4.8	1.62
1112	11	13	1.85	4.8	1.62
4403	2	16	.134	10.5	.034
4404	16	17	.3	15.9	.041
5151	215	9	4.516	18.7	18.7
49000	9	19	6.627	26.33	26.33
49011	109	119	6.627	26.33	26.33
151091	109	215	4.516	18.7	18.7
15109	15	109	4.516	18.7	18.7

NASTRAN MODEL

ELE	NODE 1	NODE 2	A	I_1	I_2
90152	15	9	4.516	18.7	18.7
7800	7	881			
7799	881	8			
8899	8	882			
8900	7	882			
7801	107	1081			
7802	1081	108			
8902	108	1082			
8901	1082	109	4.516	18.7	18.7
1	15	218	16.7	1170.	30.6
2	15	219			
3	219	220			
4	220	218			
5	215	216			
6	215	217			
7	217	221			
8	221	216			
9	19	401			
10	220	402			
11	220	403			
12	119	404			
13	119	405			
14	221	406			
15	221	407	16.7	1170.	30.6

NASTRAN MODEL

ELE	NODE 1	NODE 2	A	I_1	I_2
16	19	408	16.7	1170.0	30.6
17	19	218			
18	219	119			
19	119	217			
20	216	19	16.7	1170.0	30.6
3401	3	104	1.99	172.9	1.47
3601	3	116	3.03	216.	2.05
4501	104	117	.668	35.9	.44
4503	111	105	.668	27.9	.50
4901	104	109	4.516		
1601	1	106	2.94	171.4	5.65
6701	106	107	2.94	343	287
1501	1	105	1.62	101.7	2.92
5701	105	131	3.49	85.4	1.21
5601	105	106	.462	14.0	.18
6311	131	106	1.0	30	.2
1111	211	1	1.0	3.5	.2
11201	1	212	1.85	4.8	1.62
12131	212	213			
21101	211	2			
11131	213	211			
11121	211	212	1.85	4.8	1.62
4401	2	116	.134	10.5	.034
4402	116	117	.3	15.9	.041
3503	116	105	3.03	151.1	2.29
5703	131	107	3.49	28.2	2.09

CRODS			
NO.	NODE 1	NODE 2	AREA
101	213	7	1.82
102	213	107	1.82
103	107	13	1.82
104	13	7	1.82

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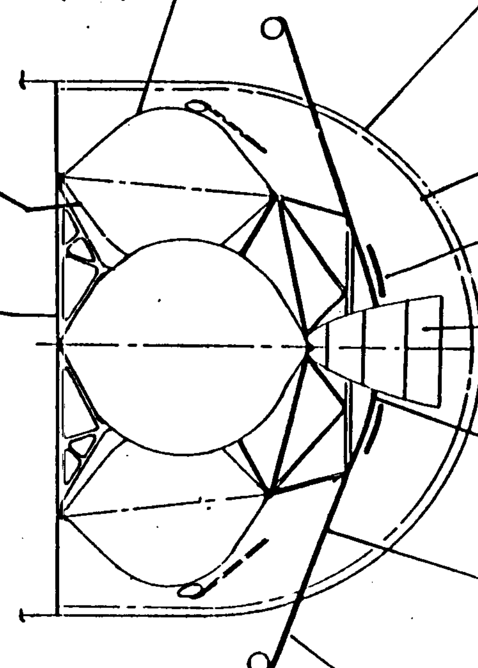
GROUNDBASED CRYOGENIC REUSABLE OTV

This viewgraph shows the general arrangement and weight breakdown of our selected groundbased cryogenic OTV transported in the ACC. 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).

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, along with the core system (LO2 tanks, structure, avionics, and propulsion) are stowed in the Orbiter cargo bay for retrieval after mission completion. The propulsion and avionics subsystems reflect the component count previously considered. The structure is of lightweight graphite/epoxy. The propellant load was selected to enable full use of the projected NSTS lift capability on GEO delivery missions.

GROUND BASED CRYOGENIC REUSABLE OTV



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

MARTIN MARIETTA
MANNED SPACE SYSTEMS

GROUNDBASED CRYOGENIC OTV WEIGHT CHANGE SUMMARY

This table shows the updated weight changes to the recommended groundbased OTV.

(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 reevaluation was a weight increas 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.

G.B. CRYOGENIC OTV WT CHANGE SUMMARY

REASONS FOR CHANGES

(1) AEROBRAKE	REDUCED HARD SHELL CORE FROM 25.0 FT TO 13.5 FT IN DIAMETER REMOVED 25.5 FT DIAMETER RIB SUPPORT FRAME
(2) TANKS	INCREASED WELD LANDS AND MEMBRANE THICKNESS PER LATEST PROPERTY INFORMATION FOR THE AL-LI 2090 MATERIAL
(3) ENVIRONMENTAL CONTROL	REDUCED THICKNESS OF BUMPER BASED ON REFINED DATA DEVELOPED DURING THE SPACE STATION STUDY PROGRAM
(4) ELECTRICAL SYSTEM	REPLACED S-BAND SYSTEM WITH A LIGHTER SYSTEM

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G.B. CRYOGENIC OTV WT CHANGE SUMMARY

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

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AEROBRAKE WEIGHT CHANGES

This table shows the weight changes in the 38 ft diameter aerobrake that occur due to a reduction in the diameter of the hard shell center. The center, with a t 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.

A secondary effect occurs from removing the rib supported at 25.5 ft out 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.

AEROBRAKE WEIGHT CHANGES

THE AEROBRAKE HARDCORE CENTER WAS CHANGED FROM 25.5 FT TO 13.5 FT

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

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MANNED SPACE SYSTEMS

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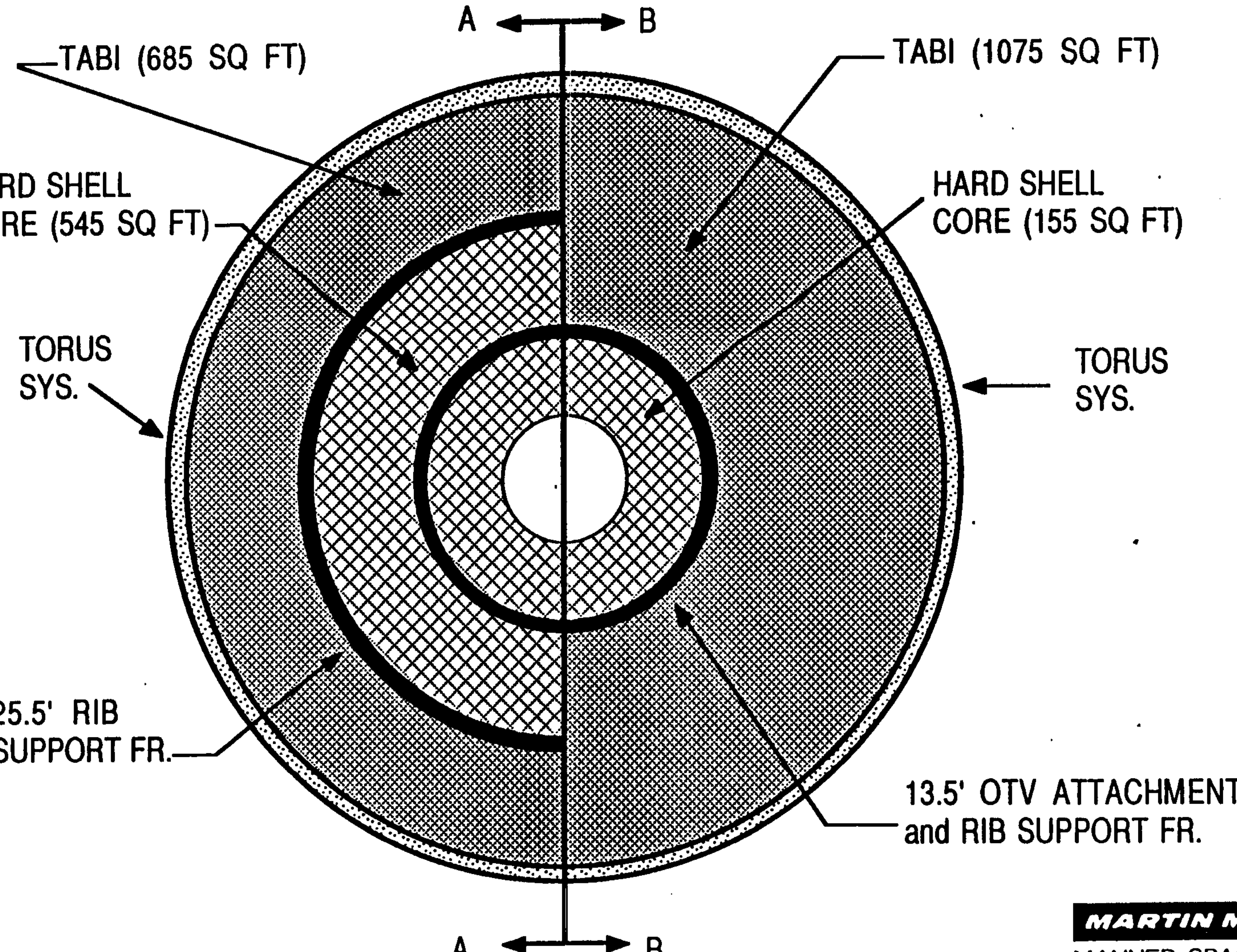
AEROBRAKE DESIGN CHANGES



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MANNED SPACE SYSTEMS

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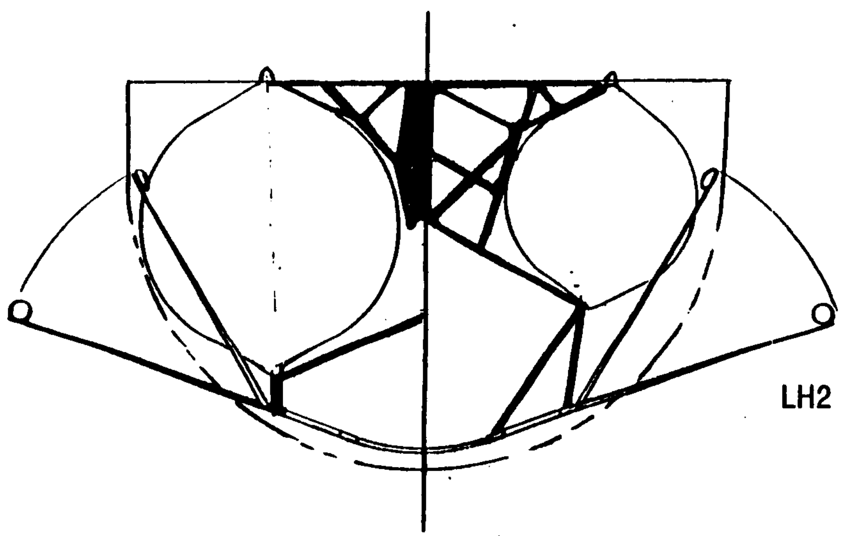
AEROBRAKE STOWAGE ARRANGEMENT

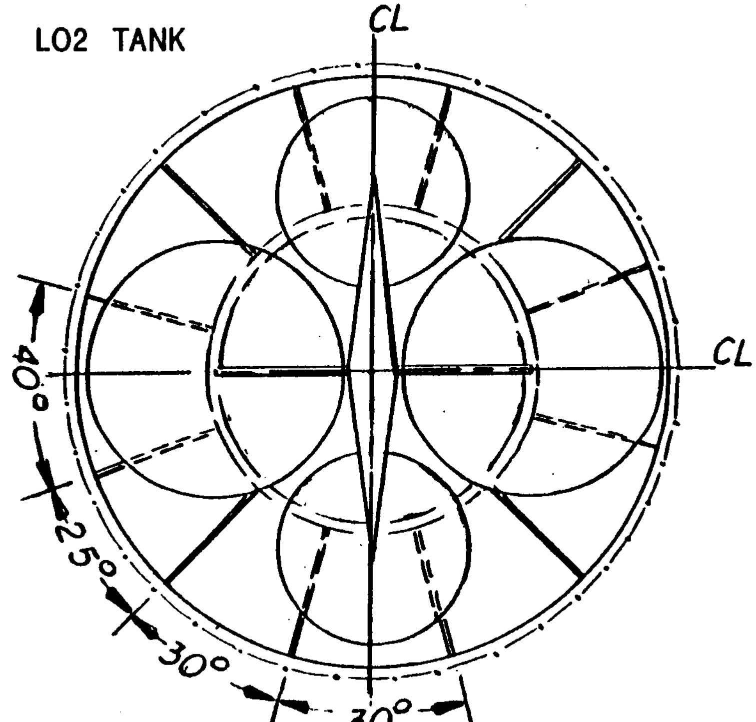
The facing view shows the stowage arrangement for the 38 ft aerobrake. To accommodate the aerobrake with a 13.5 ft diameter hard shell center, the ribs have been clocked off the centerline of the tanks by 15°, with a 30° typical spacing. However, the ribs on either side of the LH2 were clocked 20° off 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.

AEROBRAKE STOWAGE ARRANGEMENT



FIXED AEROBRAKE
(13.5 FEET DIAMETER)

DEPLOYED AEROBRAKE
(38 FEET DIAMETER)



RIB ARRANGEMENT TO ACCOMMODATE STOWAGE
btwn LH2 TANK - 40° APART
20° OFF CL OF TANK
btwn LO2 TANK - 30° APART
15° OFF CL OF TANK
TYP SPACING - 30° APART

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MANNED SPACE SYSTEMS

OTV DEBRIS/METEOROID ASSUMPTIONS

To meet a proposed 0.999 probability of no damager 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, it is only necessary to vary the bumper thickness and location to achieve appropriate levels of penetration resistance. Additional thickness of the pressure wall or thermal blanket will not be analyzed.

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 [illegible]12.

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.

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 $\geq$ 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

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PROBABILITY CALCULATION

Bumper thickess has a strong influence on the probability of penetration for thin bumpers. If the incident particle is not broken up by the bumper, than 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.

The size of a meteoroid and the size of debris which can be stopped by each design is used to calculatge 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.

OTV DEBRIS/METEOROID BUMPER SIZE

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

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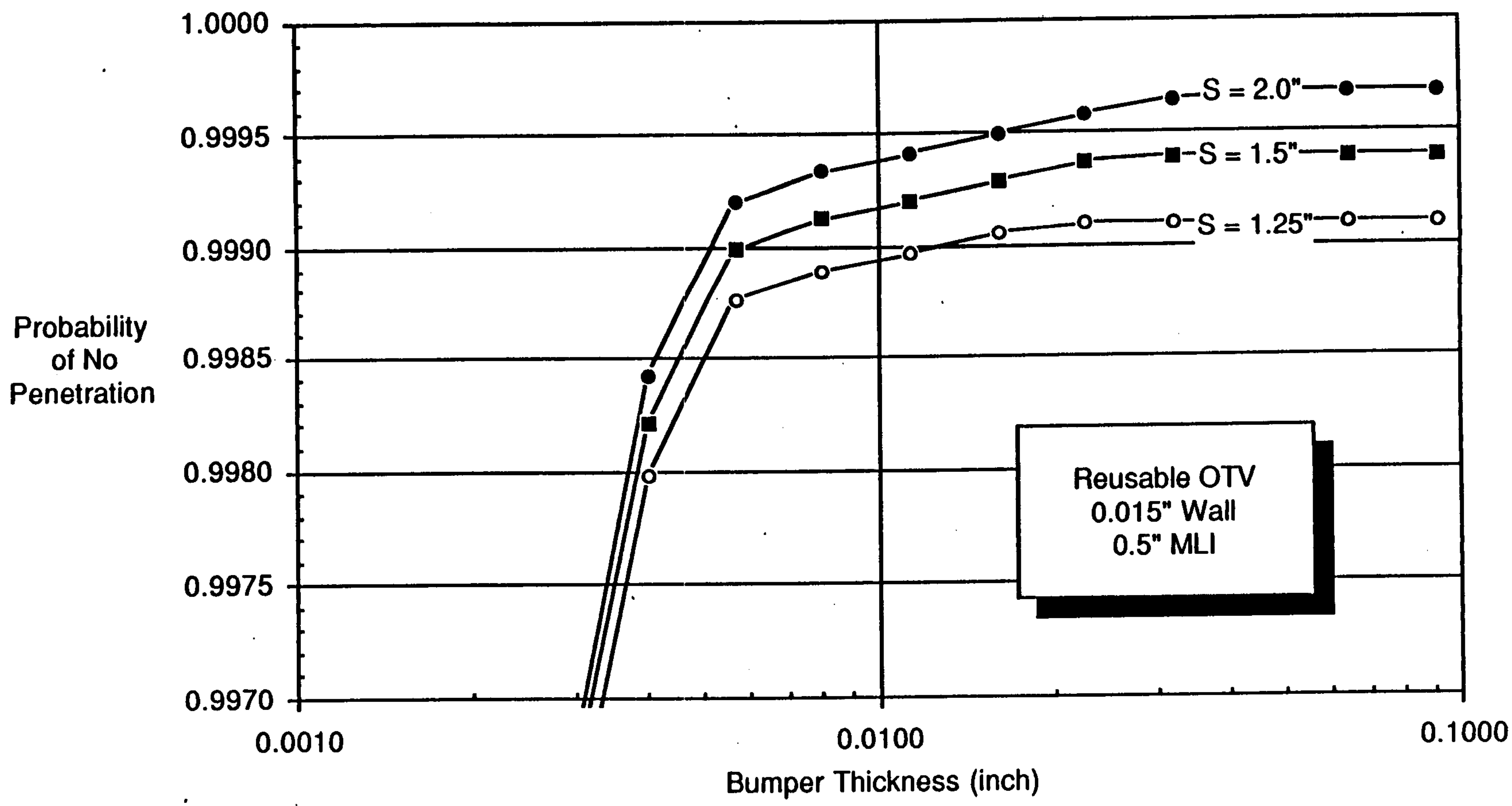
MANNED SPACE SYSTEMS

OTV DEBRIS/METEOROID BUMPER SIZE

For a reusable OTV, at least two layers of Beta Cloth should be used with at least a 1.5-in. standoff.

Although expected increases in the space debris and meteoroid environment will affect these numbers, these are projections in the environment, 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 or 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.

OTV DEBRIS/METEOROID G.B. REUSABLE



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MANNED SPACE SYSTEMS

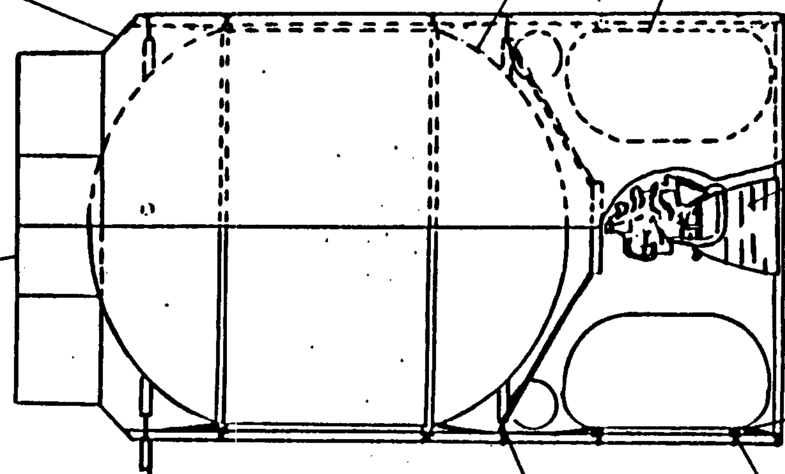
SHUTTLE-C EXPENDABLE OTV

This figure shows a cargo bay expendable OTV capable of derivering 15,000 lbm to GEO from Shuttle-C deployment in LEO. This concept is attractive because of its high performance and the vehicle's short length (compared to other cryogenic configurations).

The main contributor to the shortened length is incorporation of a toroidal LO2 tank in which the main engine is packaged. This concept was developed to emphasize short length while maintaining high performance, i.e., payload capability at minimum gross weight. According to the mission model assessment, the stage length plus ASE should not exceed 30 ft in order to minimize NSTS launch costs. In other words, the 30 ft payload capability and sufficient performance are the major desirable characteristics for a cargo bay OTV. This stage meets these criteria, i.e., 26.7 ft length, ASE length, and ASE packaging characteristics.

Minimum tank gages are 0.025 for the toroidal LO2 tank and 0.025 for the LH2 tank. The two tanks are protected by a cylindrical debris shield of graphite/epoxy, supported by longerons and ring frames of the same material. Each tank is attached to the longerons and frames by fiberglass/epoxy struts which accommodate the temperature differences. The avionics units have been mounted on an avionics ring that also serves as the payload interface. Ag-Zn batteries provide the power source, and the propulsion unit is a RL10-A engine.

SHUTTLE-C EXPENDABLE OTV (15' DIA)



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

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AEROASSIST RESULTS

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AEROASSIST

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- MISSION DESCRIPTIONS
- ANALYSIS ASSUMPTIONS
- EARTH RETURN RESULTS
- MARS CAPTURE RESULTS
- EARTH CAPTURE RESULTS
- HIGH SPEED AERO SUMMARY
- LUNAR RETURN AEROBRAKE

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AEROASSIST CLASSES

Several different classes of entries have been studied in the course of this contract as is summarized in this figure. Earth return missions utilize aeroassist to reduce the energy of an existing elliptical Earth orbit. The desired end condition is 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. Cases consistant with return from Mars have been investigated with encounter C_3's ranging from 8.0 to 68 km^2/sec^2. The third class of missions are those of Mars capture. 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).

AEROASSIST CLASSES

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

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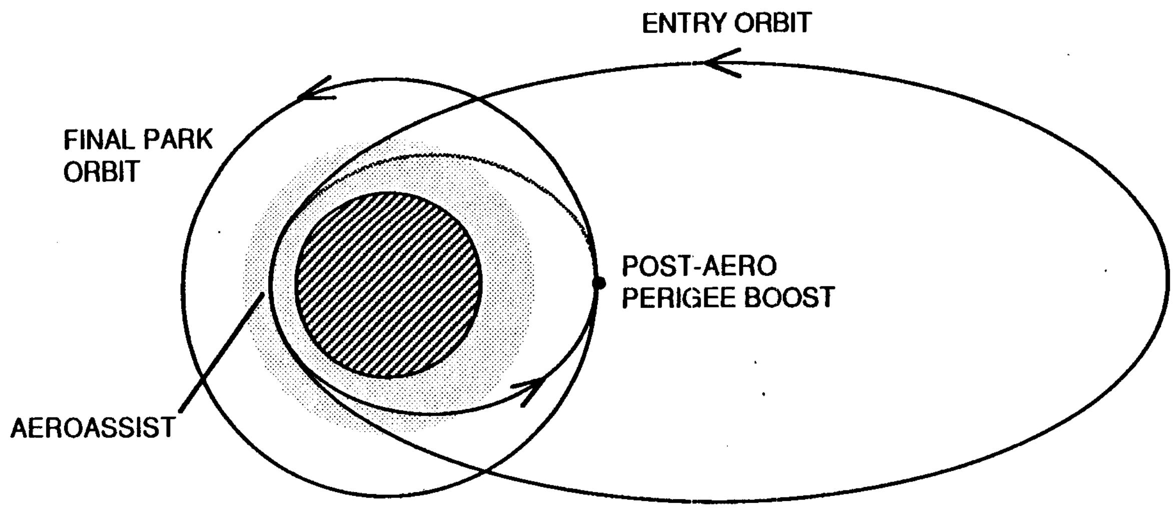
EARTH RETURN TO LOW ORBIT

This figure 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 (indicated by the shaded region of the figure). 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.

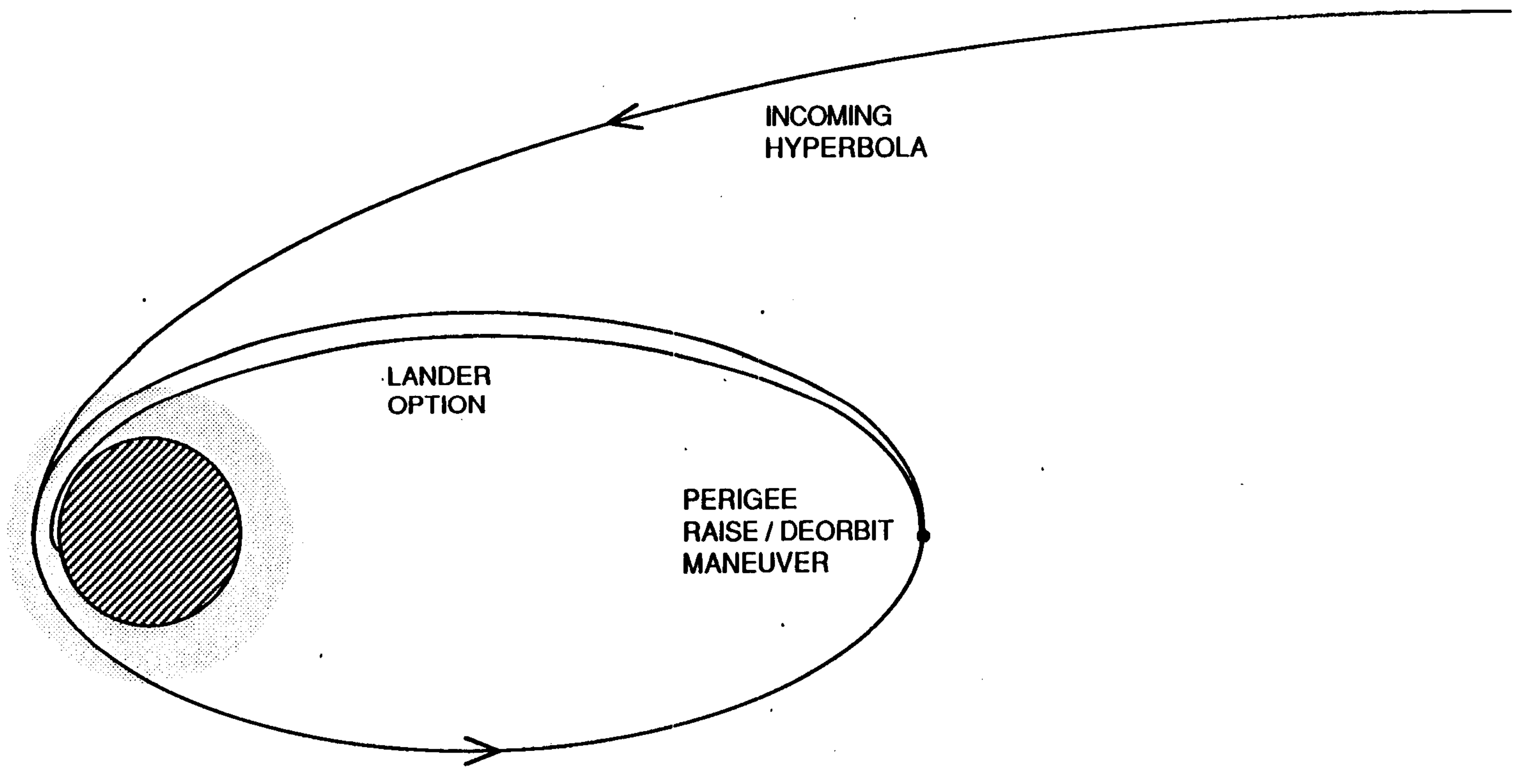
EARTH RETURN TO LOW ORBIT



PLANETARY AERO-CAPTURE

The process of aer-capture is very similar to that of aeroassist, illustrated previously, 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.

PLANETARY AERO-CAPTURE



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AEROASSIST CONDITIONS - EARTH ENTRIES

This chart summarizes important parameters associated with aeroassists at Earth. This includes both Earth return and capture missions discussed previously. The parameters of interest are as follows. 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.

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

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AEROASSIST CONDITIONS - MARS ENTRIES

This chart summarizes the same information as the previous one but 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 in the planetary community because of its good combination of site reconaisance as well as communication relay links.

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

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PLANETARY DATA

This figure 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.

PLANETARY DATA

	EARTH	MARS
EQUATORIAL RADIUS	2.09256627E7 FT	1.114567E7 FT
POLAR RADIUS	208555024E7 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.)

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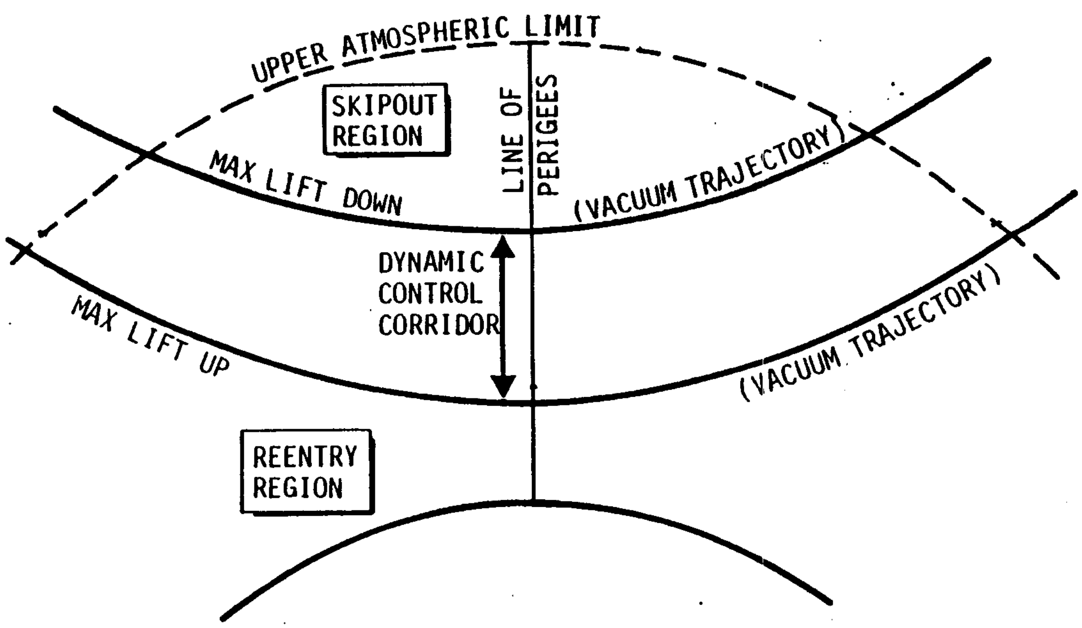
CONTROL CORRIDOR DEFINITION

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, 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 (subsequent charts).

CONTROL CORRIDOR DEFINITION



- 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

MARTIN MARIETTA

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. This chart summarizes error analysis assumptions that are common to all entries. These variables are discussed in greater detail in each error analysis page.

AERO ERROR ANALYSIS ASSUMPTIONS

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

FOLLOWING ASSUMPTIONS ARE COMMON

<u>NAVIGATION:</u>

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 SEP FROM MARS

   FINAL NAV UPDATE FOR MIDCOURSE AT 1.5 HR FROM MARS ENTRY

<u>MIDCOURSE:</u> FINAL MIDCOURSE CORRECTION AT ENTRY MINUS 1.0 HOUR

<u>ATMOSPHERE:</u>

1) EARTH DENSITY VARIABILITY = ± 30 %

2) MARS DENSITY VARIABILITY = ± 50 %

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

<u>BALLISTIC COEFFICIENT UNCERTAINTY:</u> ± 8% ON W/CDA

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

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EARTH RETURN RESULTS

GEO AERO-ENTRY ERROR ANALYSIS

The following chart 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.

Aerodynamic Variations - No two aero-entries will be quite the same. 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 increased by 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.

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% $W/C_D A$

• 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

MARTIN MARIETTA

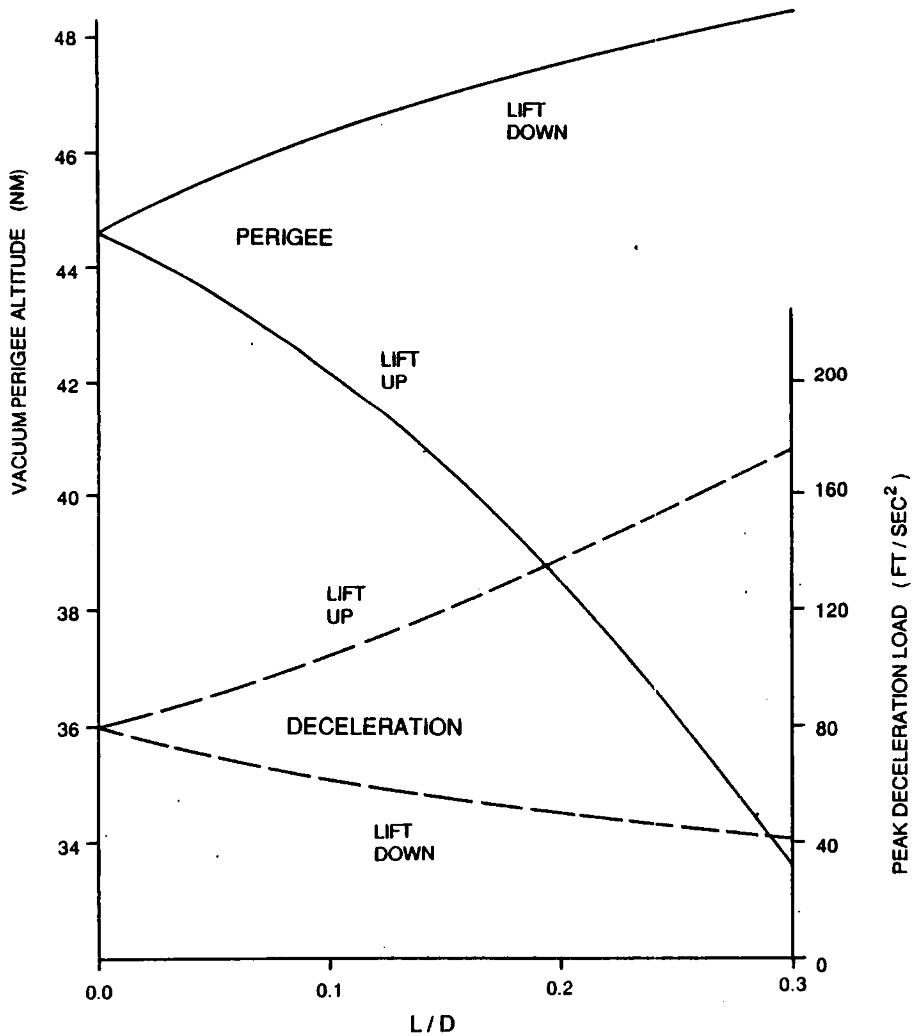
GEO RETURN CONTROL & LOADS

Various entry trajectories were generated utilizing a pre-entry ellipse with an apogee of 19323 nm. By targeting to an post-aero exit orbit with an apogee of 245 nm the return to a Space Station from an initial geosynchronous transfer is derived. 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 while the peak heating and integrated heating is shown as a function of ballistic coefficient.

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 controlled to the desired exit conditions with the available lift. Since the error analysis of the previous chart 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.

GEO RETURN CONTROL & LOADS



- RETURN FROM GEO TO S.S.
  - ENTRY APOGEE = 19323 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

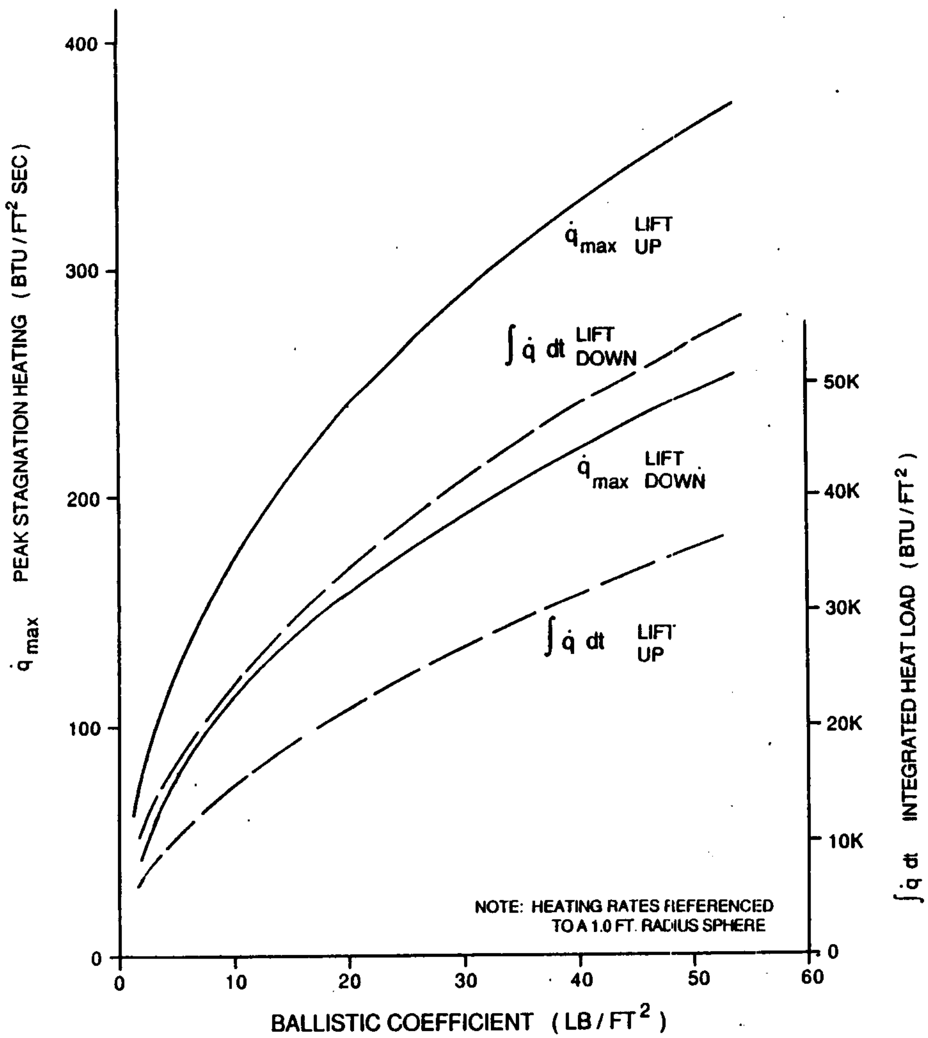
MARTIN MARIETTA

GEO RETURN HEATING

This chart shows heating information for the GEO return to 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.

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.

GEO RETURN HEATING



- RETURN FROM GEO TO S.S.

  ENTRY APOGEE = 19323 NM

  AERO EXIT APOGEE = 245 NM

  BASE L/D = 0.20

- PEAK STAGNATION HEATING

  SETS TPS MATERIAL REQMTS

- INTEGRATED HEATING

  SETS TPS THICKNESS

MARTIN MARIETTA

LUNAR AERO-ENTRY ERROR ANALYSIS

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 the dispersions identified. The lunar entry condition is faster because of the much higher apogee of the incoming orbit, consistent with a lunar free return. The actual dispersions are the same because of a common Earth environment for entry. The 5.53 nm net control corridor size sets a minimum L/D requirement of 0.11 for the entry vehicle. A further analysis of aerobrake sizing actually increased this L/D for load relief peculiar to the lunar vehicle application. This issue is discussed further on in the presentation.

LUNAR 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

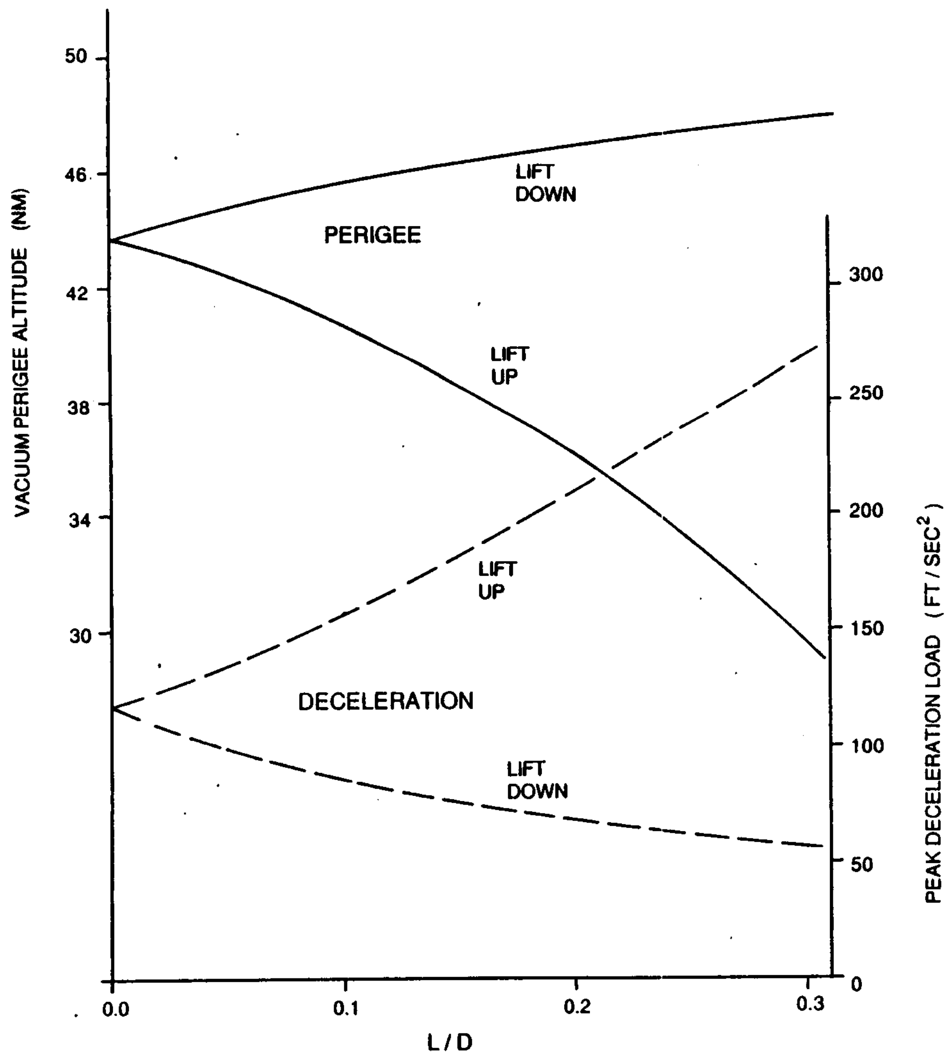
CONCLUSION: 5.53 N.M. CONTROL CORRIDOR REQUIRED TO COVER ERRORS WITH 33% MARGIN

MARTIN MARIETTA

LUNAR RETURN CONTROL & LOADS

This figure shows data for Lunar return. Initial entry orbit has an apogee of 287700 nmi which corresponds to a free-return Lunar trajectory. The exit apogee is 245 nmi which corresponds to return to the Space Station. 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.

LUNAR RETURN CONTROL & LOADS



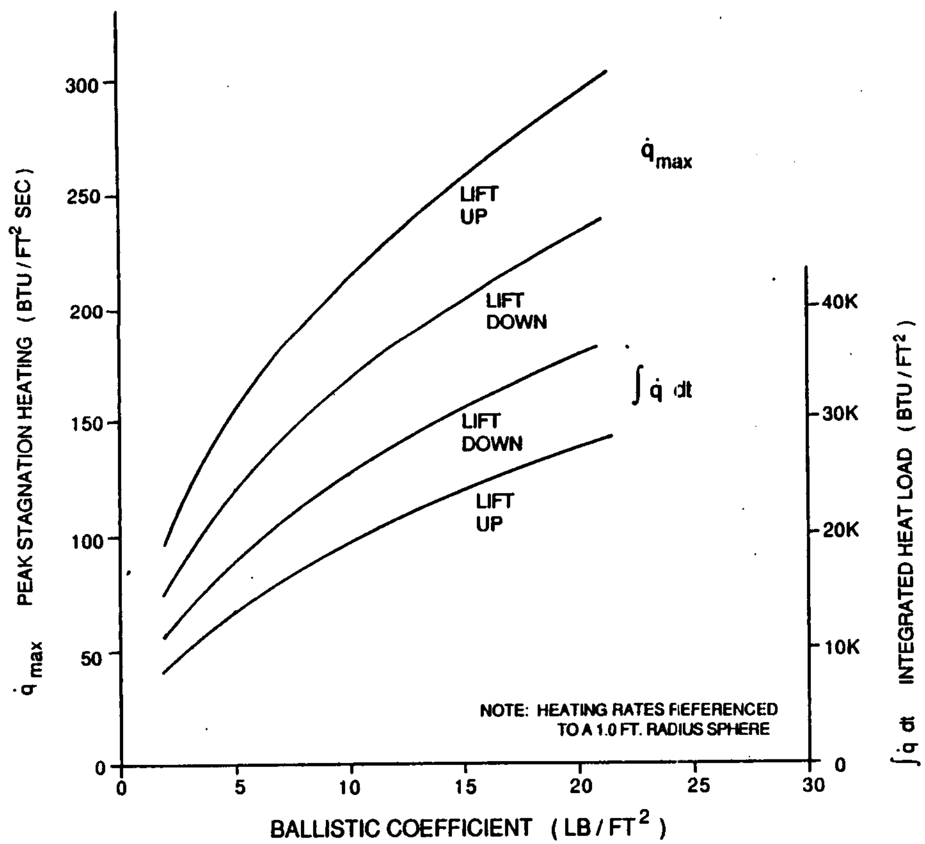
- RETURN FROM MOON TO S.S.
  - ENTRY APOGEE = 287700 NM
  - AERO EXIT APOGEE = 245 NM
  - BASE W/CdA = 5.0 LB/FT^2
- AEROASSIST CONTROL CORRIDOR
  - WIDTH = DELTA OF PERIGEES
  - ERROR ANALYSIS SETS REQMT
  - CONTROL CORRIDOR SETS L/D
- PEAK DECELERATION
  - SETS STRUCTURAL REQMTS

MARTIN MARIETTA

LUNAR RETURN HEATING

These curves show 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.

LUNAR RETURN HEATING



- 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

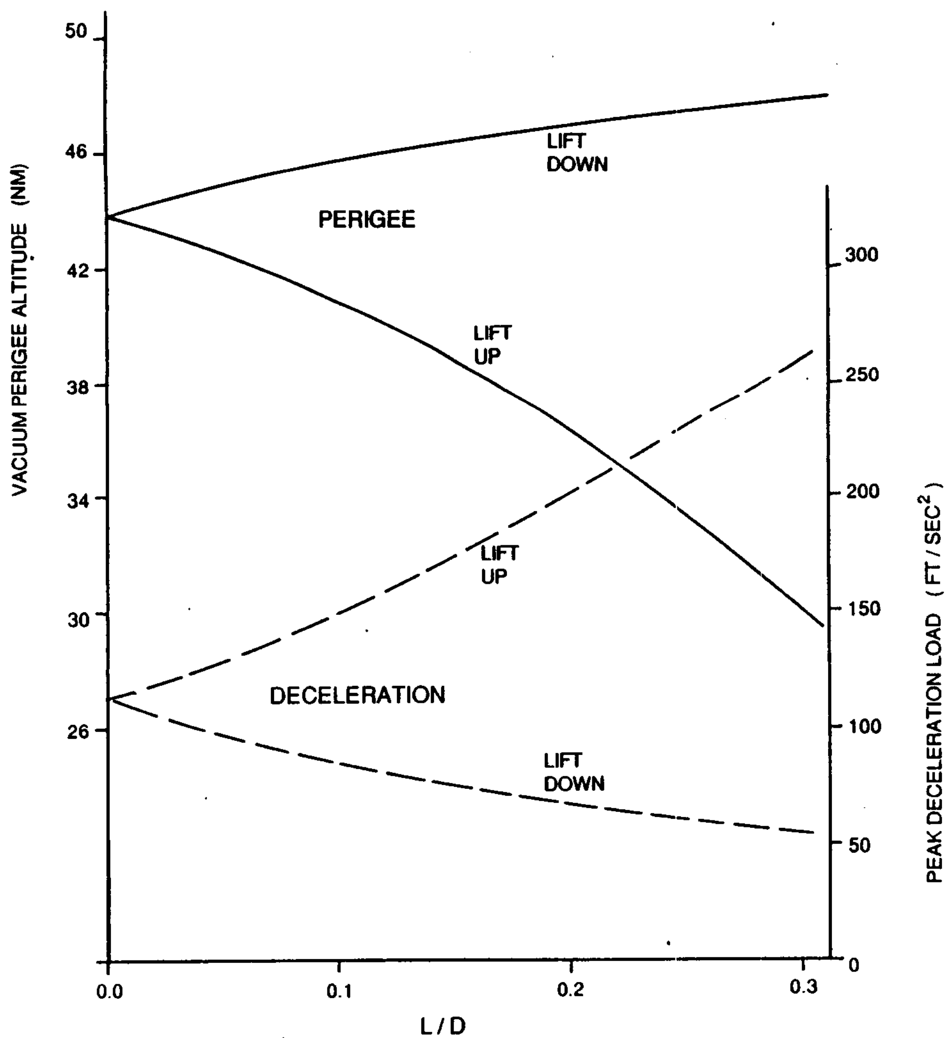
  SETS TPS THICKNESS

MARTIN MARIETTA

PLANETARY BOOST RETURN - CONTROL & LOADS

This figure 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, PlanetB & 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 the latter case.

PLANETARY BOOST RETURN - CONTROL & LOADS



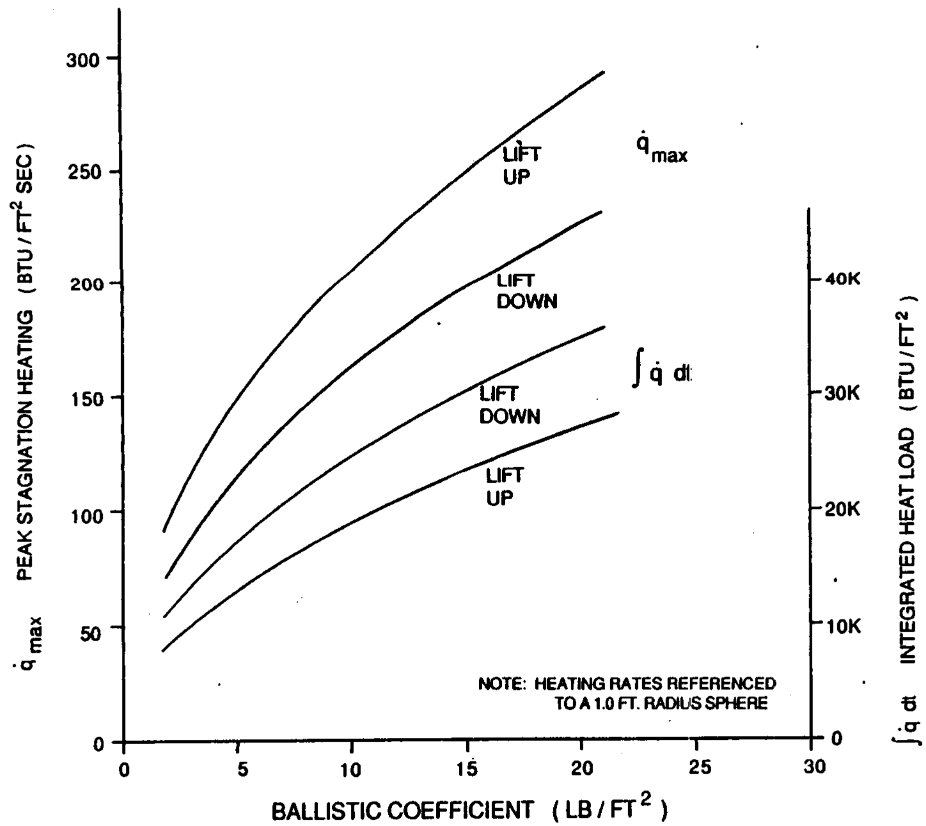
- RETURN FROM PL. BOOST TO S.S.
  - ENTRY APOGEE = 130900 NM
  - AERO EXIT APOGEE = 245 NM
  - BASE W/CdA = 5.0 LB/FT^2
- AEROASSIST CONTROL CORRIDOR
  - WIDTH = DELTA OF PERIGEES
  - ERROR ANALYSIS SETS REQMT
  - CONTROL CORRIDOR SETS L/D
- PEAK DECELERATION
  - SETS STRUCTURAL REQMTS

MARTIN MARIETTA

PLANETARY BOOST RETURN HEATING

These curves show the convection heating data for the planetary-boost return case.

PLANETARY BOOST RETURN HEATING



- 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

MARTIN MARIETTA

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MARS CAPTURE RESULTS

MARTIN MARIETTA

MARS CAPTURE ERROR ANALYSIS

This figure summarizes the error analysis conducted to derive Mars capture control requirements for an encounter orbit with a C_3 of 8.2 km^2/sec^2. 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.

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 for a final update 90 minutes before entry. 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 with ± 30% for Earth applications) which is derived from the cool versus warm density models contained in the Mars Reference Atmosphere. The L/D uncertainty results from a vehicle trim attitude variability of ± 2° in the continuum flow 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 aerodynamic 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 trajectory must be 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 nmi. which then sets the L/D of the Mars entry vehicle at 0.32 using the control sensitivity data contained in the next chart.

MARS CAPTURE AERO-ENTRY ERROR ANALYSIS: C3=8.2

	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

$\boxed{= \pm 17900 \text{ FT} = \pm 2.94 \text{ NM NET VARIATION}}$

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

MARTIN MARIETTA

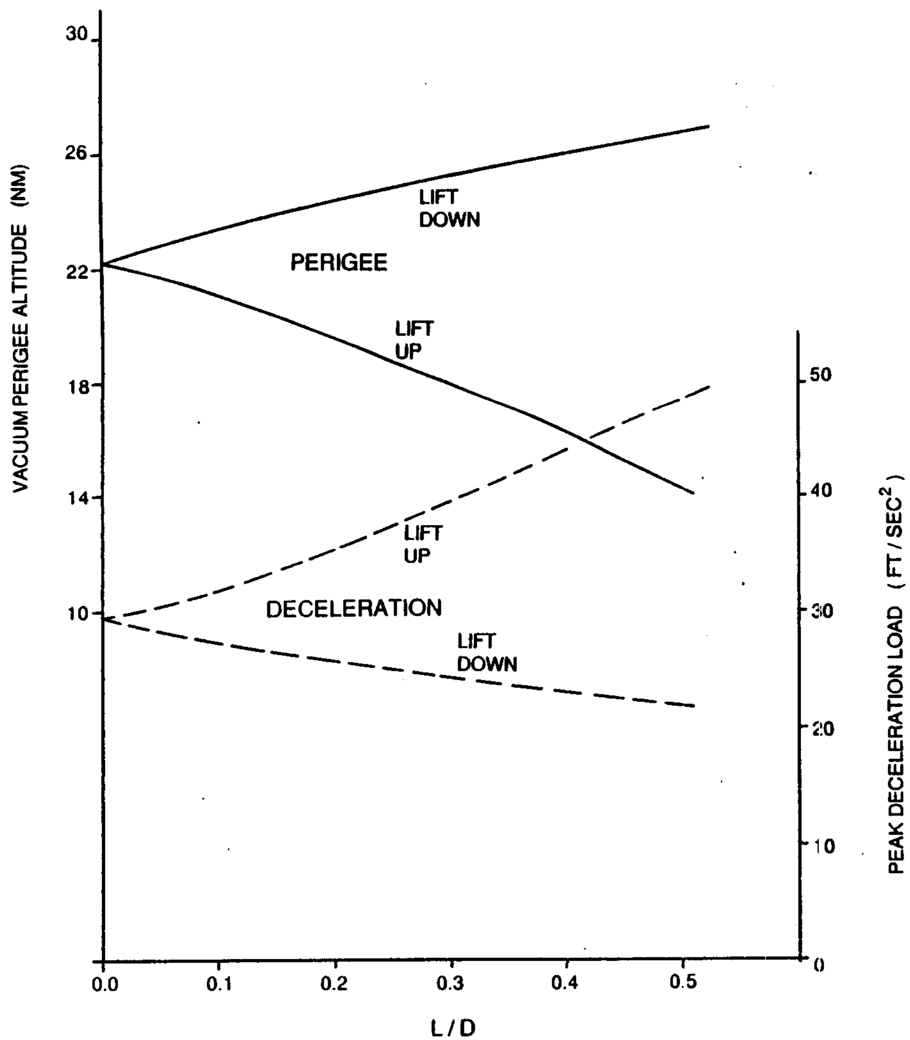
MARS CONTROL & LOADING PARAMETRICS

Various entry trajectories were generated utilizing a pre-entry hyperbols with a 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 while the peak heating and integrated heating is shown as a function of ballistic coefficient.

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 controlled to the desired exit conditions with the available lift. With error analysis haying 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.

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.

MARS CAPTURE, C3=8.2 - CONTROL & LOADS



- MARS CAPTURE
  - ENTRY C3=8.2 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

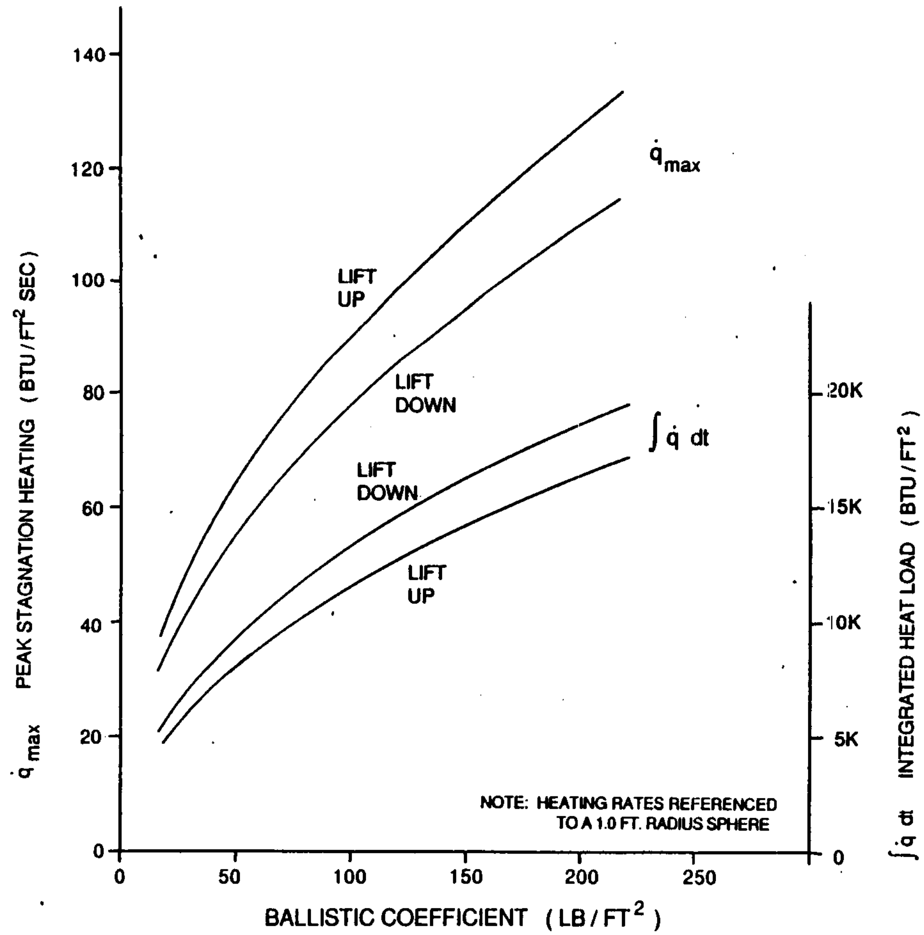
MARTIN MARIETTA

MARS CAPTURE HEATING

This chart 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.

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.

MARS CAPTURE, C3=8.2 - HEATING



- MARS CAPTURE

  ENTRY C3 = 8.2 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

MARTIN MARIETTA

MARS CAPTURE AERO-ENTRY ERROR ANALYSIS: C3=13

This figure 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 fix 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. The 8.12 nm net control corridor size sets a minimum L/D requirement of 0.26 for the entry vehicle when control parametrics (next chart) are analysed.

MARS CAPTURE AERO-ENTRY ERROR ANALYSIS: C3=13

	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

$\boxed{= \pm 18500 \text{ FT} = \pm 3.05 \text{ NM NET VARIATION}}$

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

MARTIN MARIETTA

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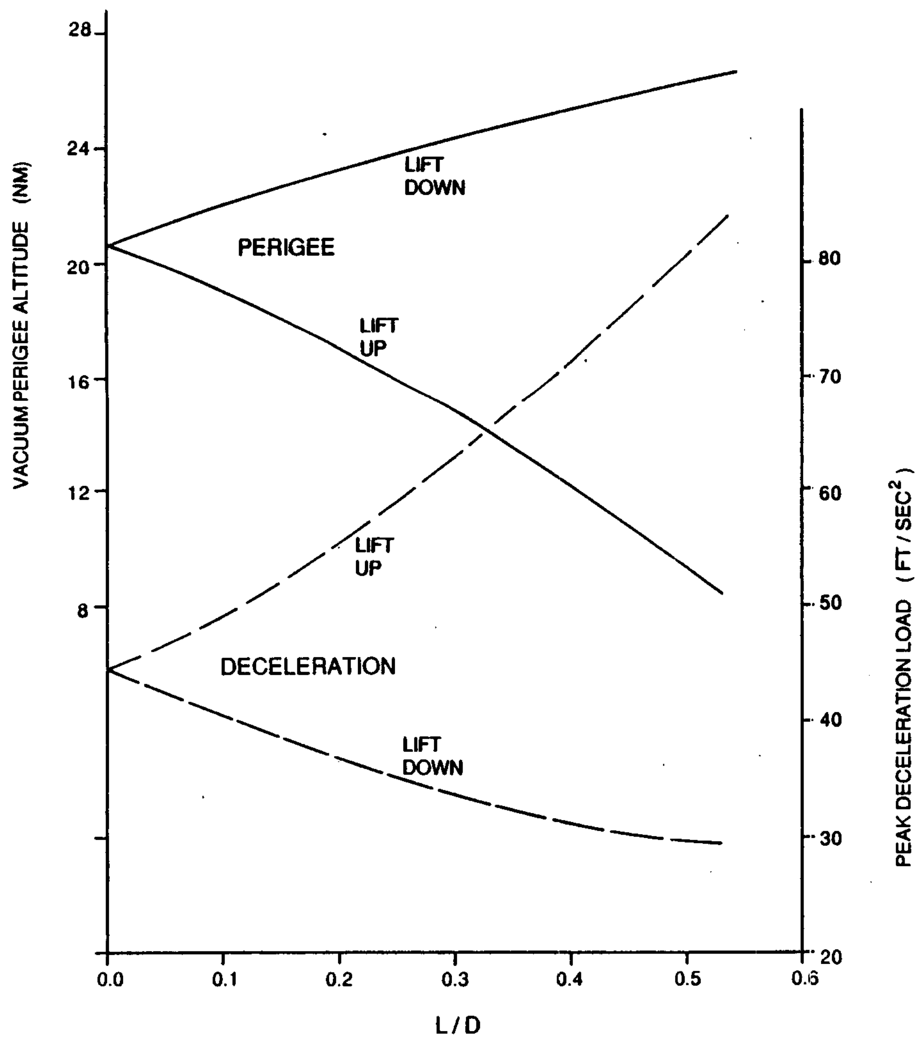
MARS CAPTURE, C3=13 - CONTROL & LOADS



- MARS CAPTURE
  - ENTRY C3=13 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

MARTIN MARIETTA

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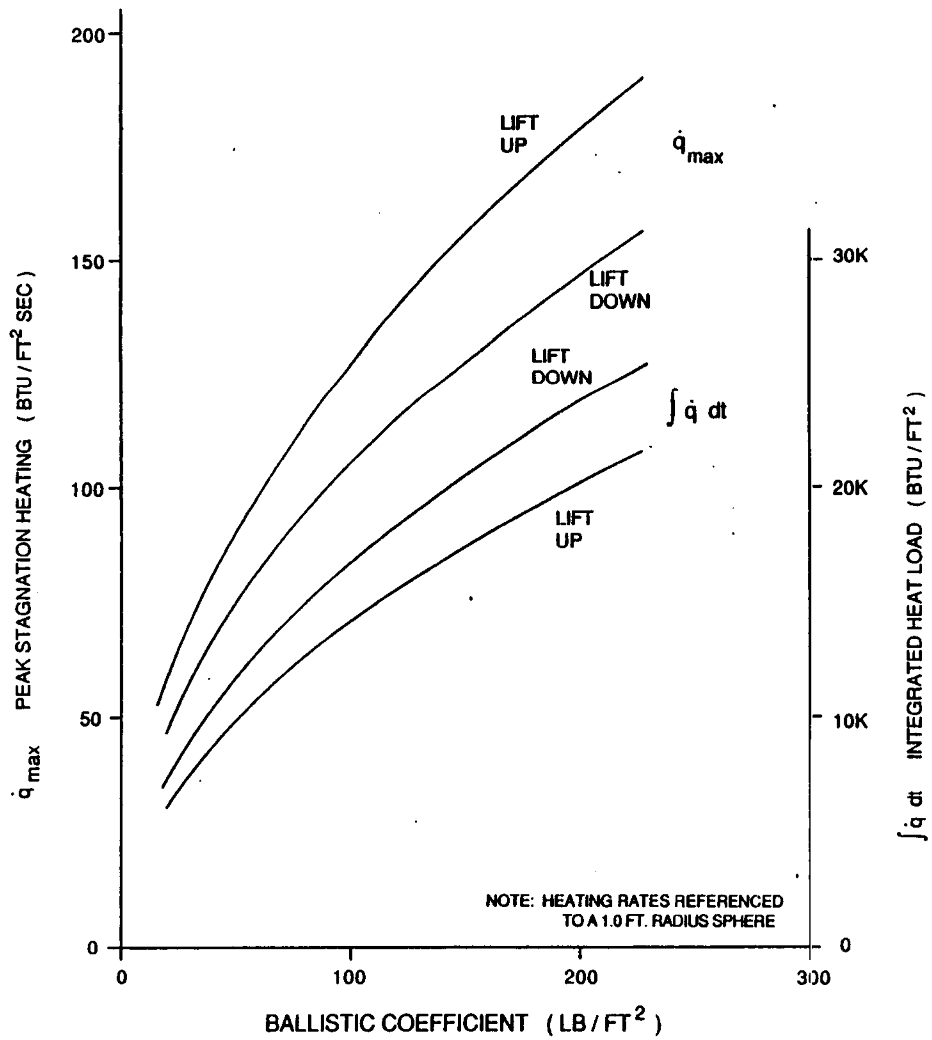
MARS CAPTURE, C3=13 - HEATING



- MARS CAPTURE
  - ENTRY C3 = 13 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

MARTIN MARIETTA

MARS CAPTURE AERO-ENTRY ERROR ANALYSIS: C3=31

This figure 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 fix 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. The 9.76 nm net control corridor size sets a minimum L/D requirement of 0.19 for the entry vehicle when control parametrics (next chart) are analysed.

MARS CAPTURE AERO-ENTRY ERROR ANALYSIS: C3=31

	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

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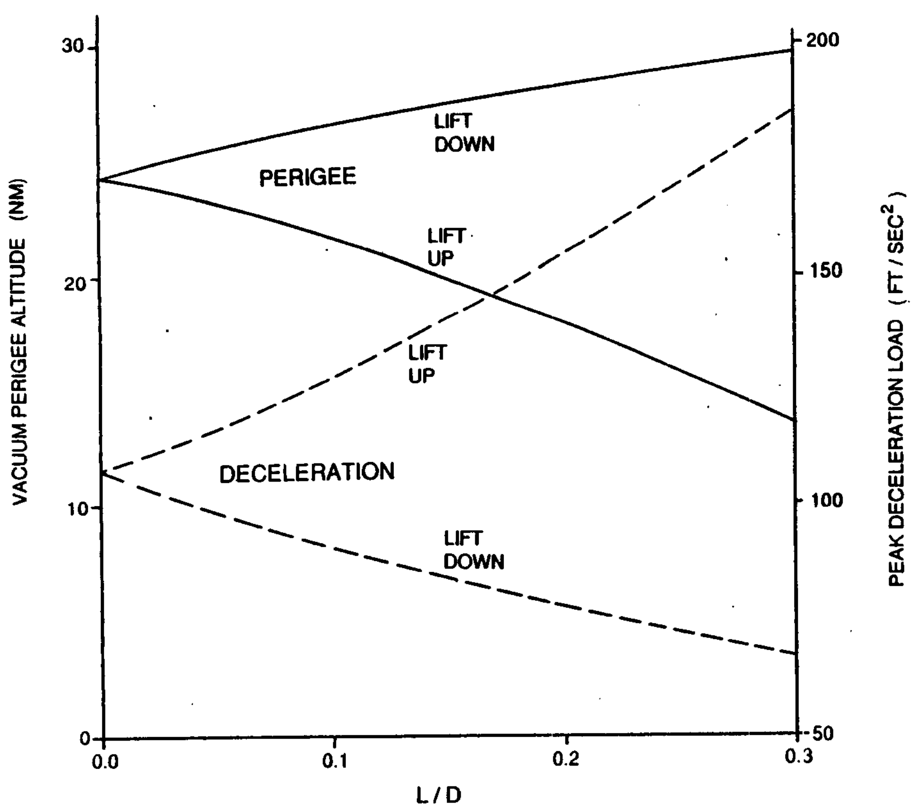
MARS CAPTURE, C3=31 - CONTROL & LOADS



- 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

MARTIN MARIETTA

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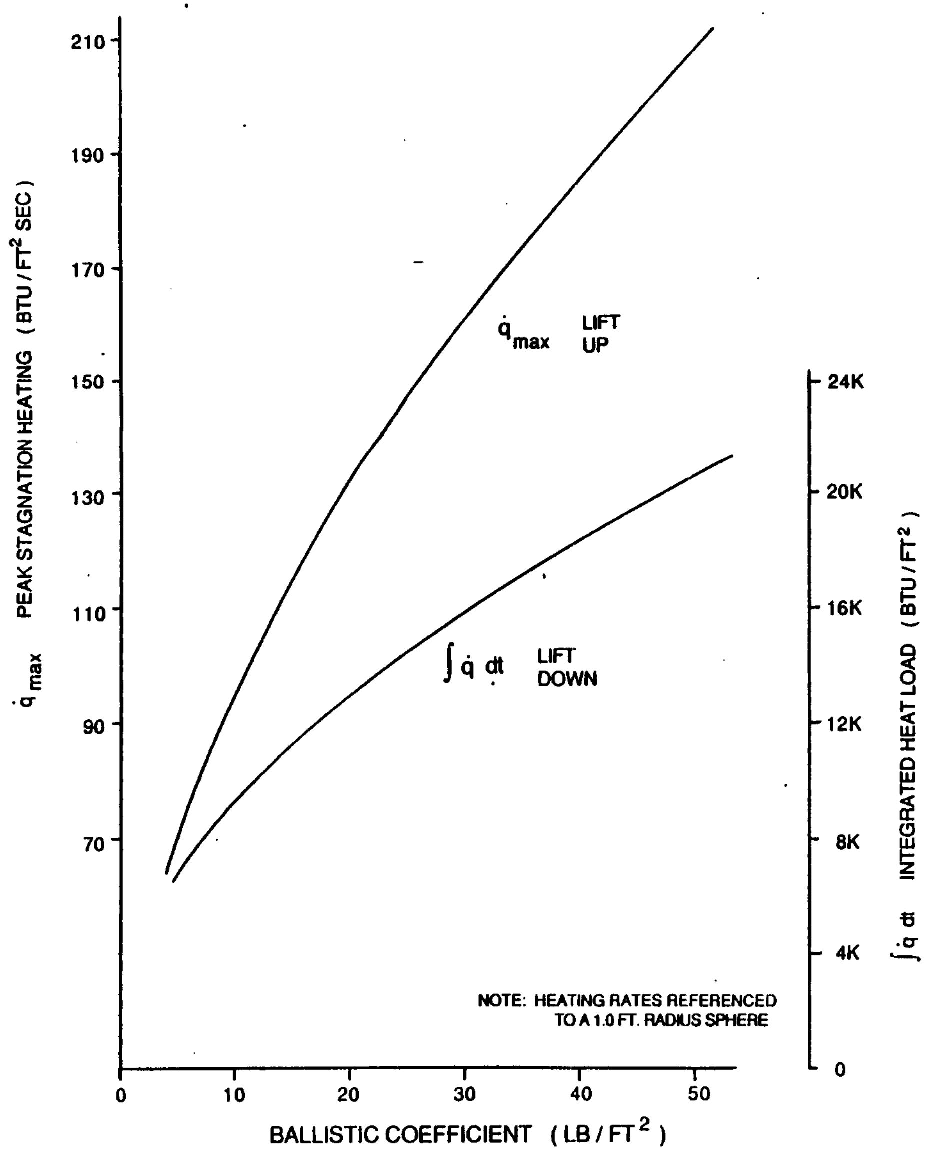
MARS CAPTURE, C3= 31 - HEATING



- 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

MARTIN MARIETTA

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MARS CAPTURE AERO-ENTRY ERROR ANALYSIS: C3=60

This figure 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 fix 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. The 12.18 nm net control corridor size sets a minimum L/D requirement of 0.16 for the entry vehicle when control parametrics (next chart) are analysed.

MARS CAPTURE AERO-ENTRY ERROR ANALYSIS: C3=60

	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% $W/C_D A$

• 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

MARTIN MARIETTA

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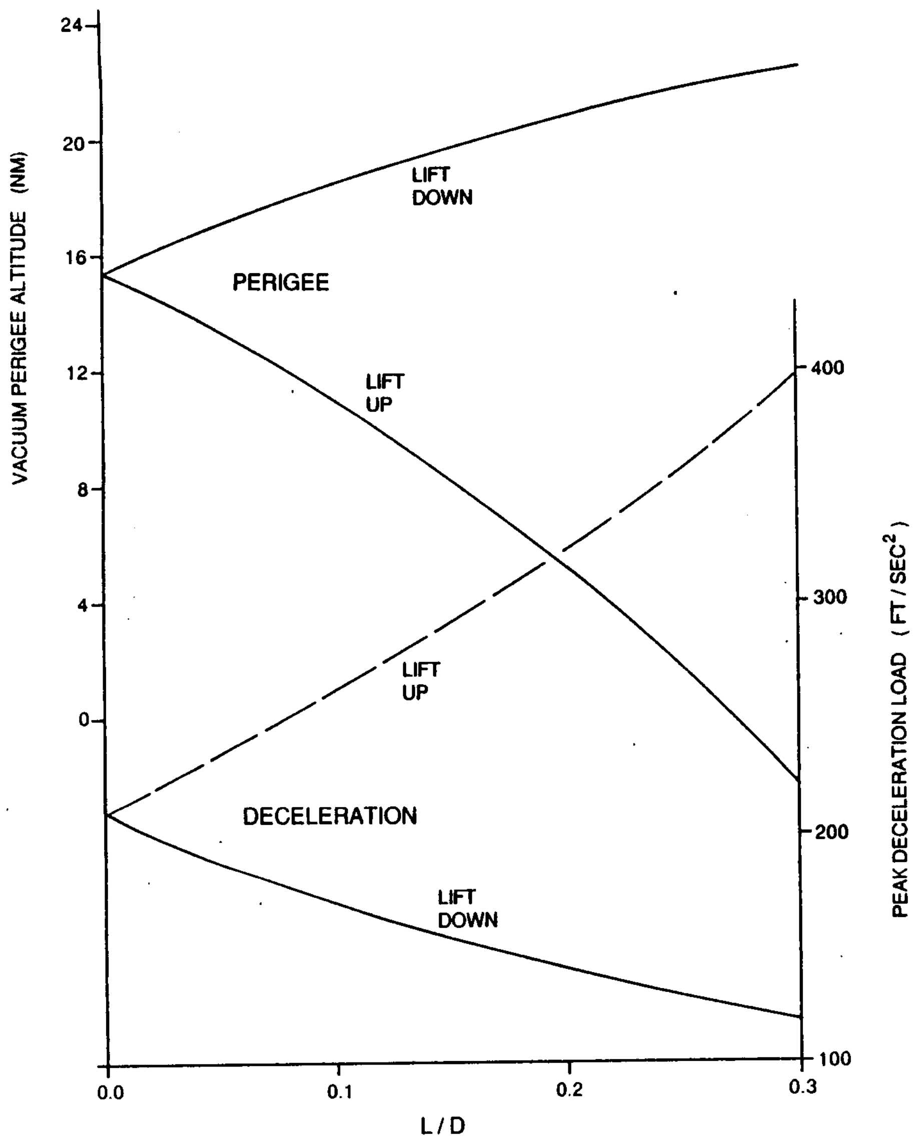
MARS CAPTURE, C3=60 - CONTROL & LOADS



- 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

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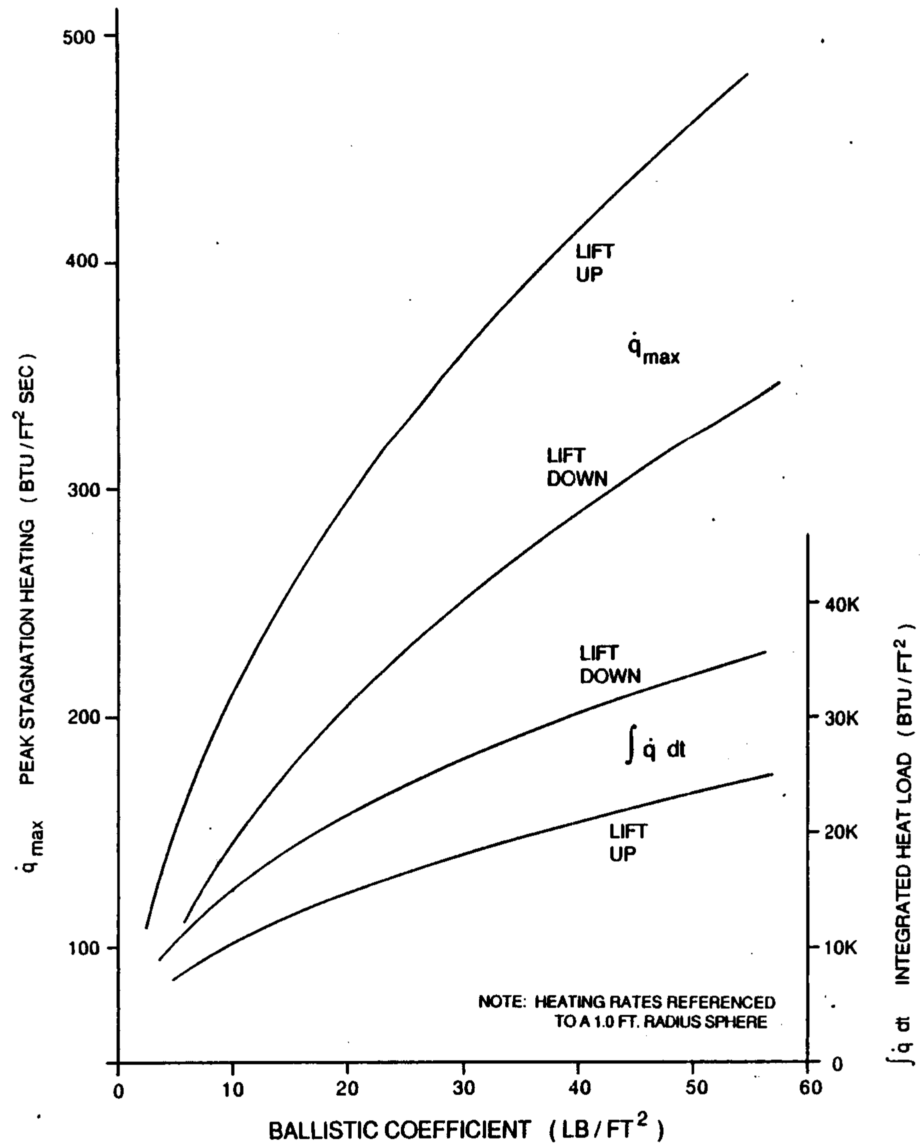
MARS CAPTURE, C3= 60 - HEATING



- 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

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EARTH CAPTURE RESULTS

EARTH CAPTURE ERROR ANALYSIS

This figure shows the results of entry error analysis conducted for the Earth capture mission phase. 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 cases. The net result of this error analysis for a entry C_3 of 8.0 km^2/sec^2 is a 2.83 nmi control corridor requirement. This control corridor requirement translates to a vehicle L/D of 0.25 using the control parametric chart.

EARTH CAPTURE AERO-ENTRY ERROR ANALYSIS: C3=8

	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

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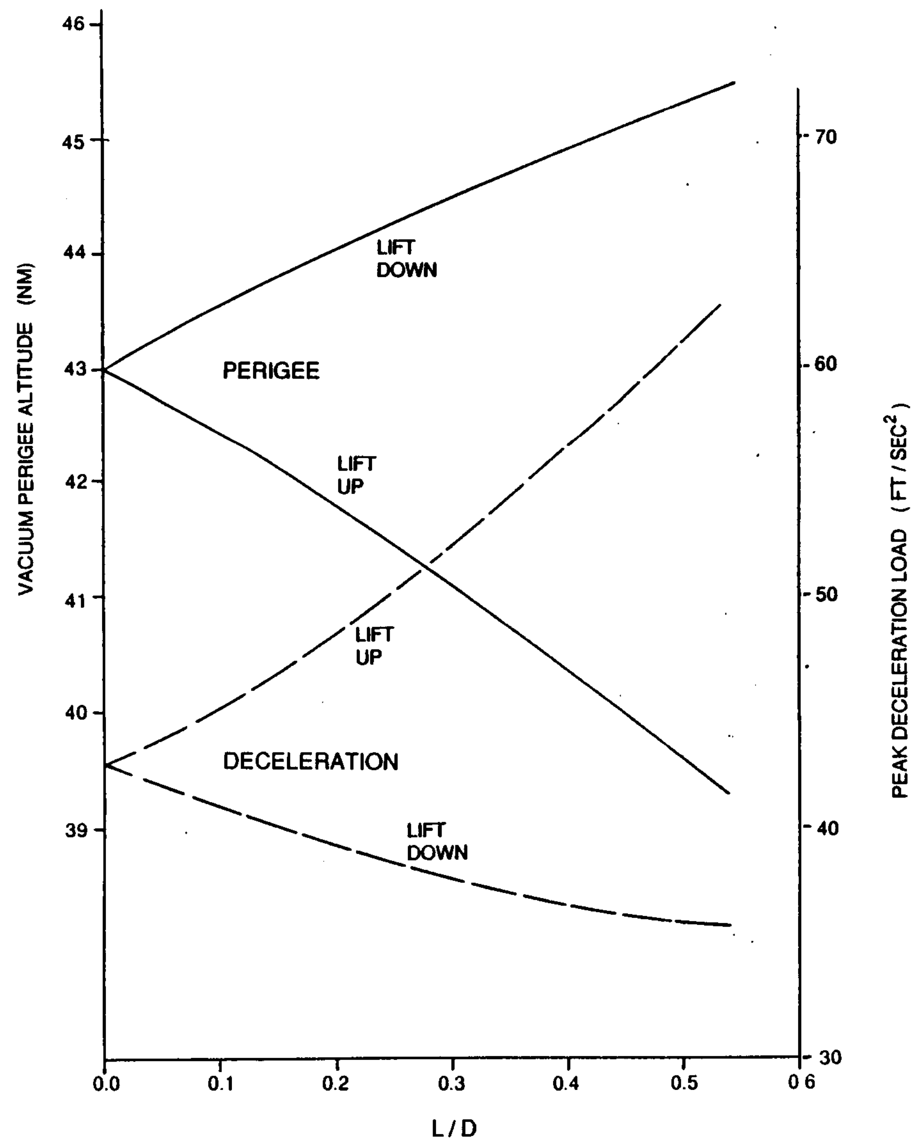
EARTH CAPTURE CONTROL & LOADS

This figure shows the aeroentry data base for the Earth capture phase. As in the Mars capture phase this includes data on lift up and lift down trajectories for vacuum perigees (whose difference yields control corridor), and deceleration loads.

EARTH CAPTURE, C3=8 - CONTROL & LOADS



- EARTH CAPTURE
  - ENTRY C3=8.0 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

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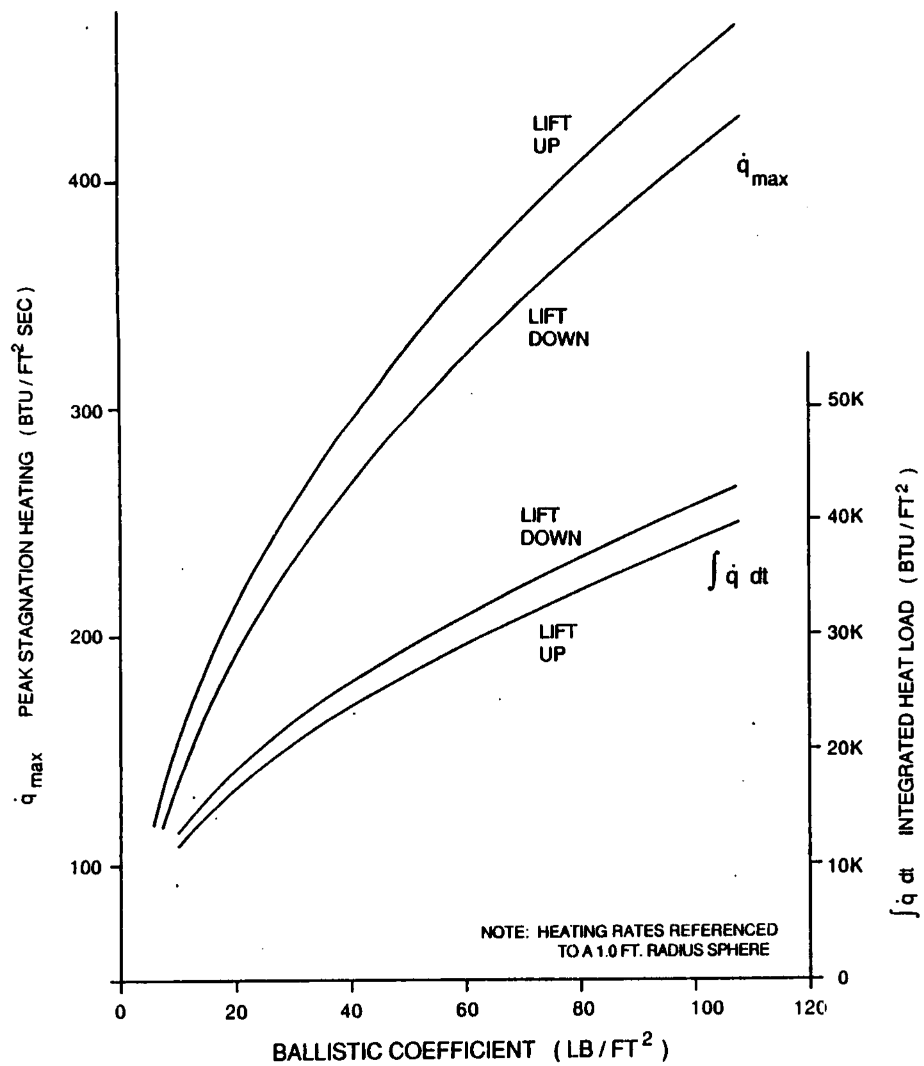
EARTH CAPTURE HEATING

This figure shows the aeroentry heating data base for the Earth capture phase. As with the Mars capture phase this includes convective heating data for lift up and lift down trajectories. Both the peak stagnation point heating as well as the time-integrated heat flux values are shown as a function of ballistic coefficient.

EARTH CAPTURE, C3= 8.0 - HEATING



- MARS CAPTURE

  ENTRY C3 = 8.0 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

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EARTH CAPTURE AERO-ENTRY ERROR ANALYSIS: C3=16

This figure 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. The 3.05 nm net control corridor size sets a minimum L/D requirement of 0.195 for the entry vehicle when control parametrics (next chart) are utilized.

EARTH CAPTURE AERO-ENTRY ERROR ANALYSIS: C3=16

	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

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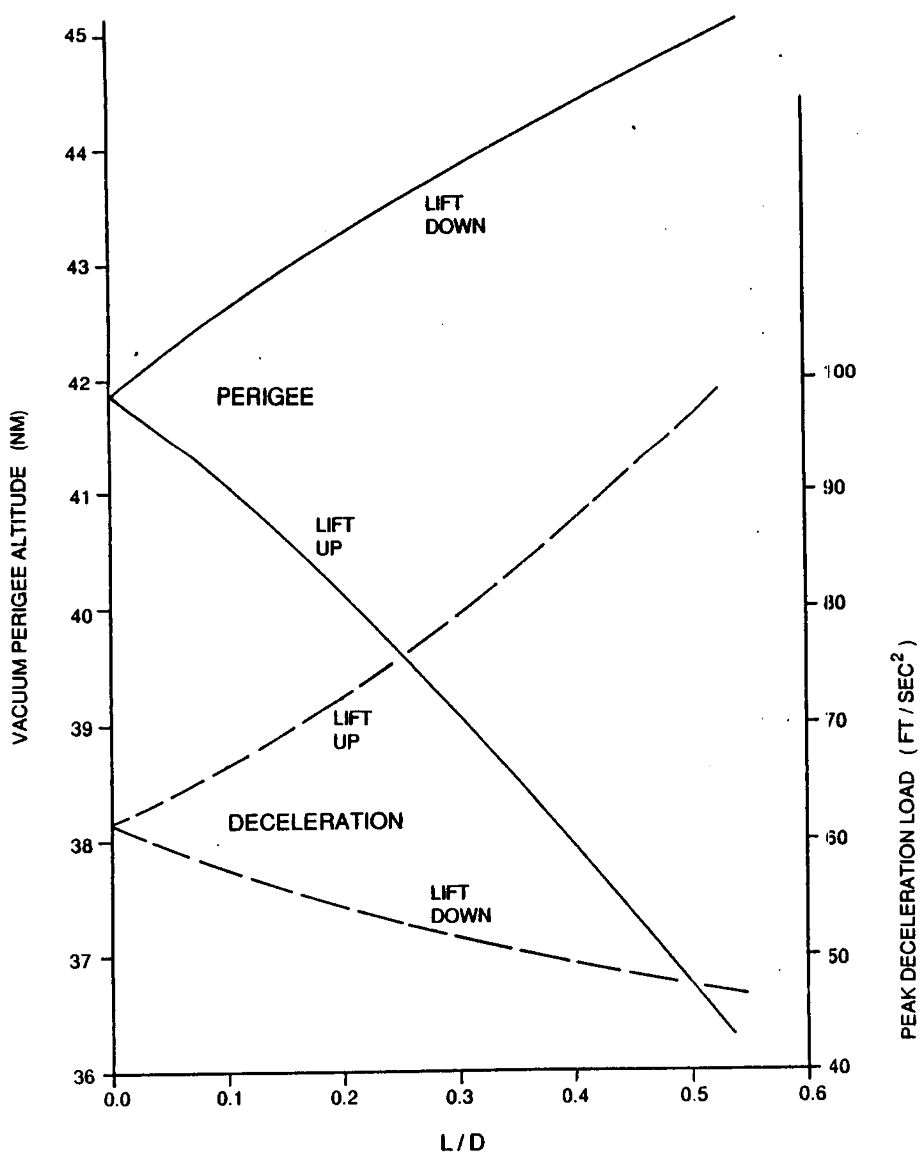
EARTH CAPTURE, C3=16 - CONTROL & LOADS



- 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

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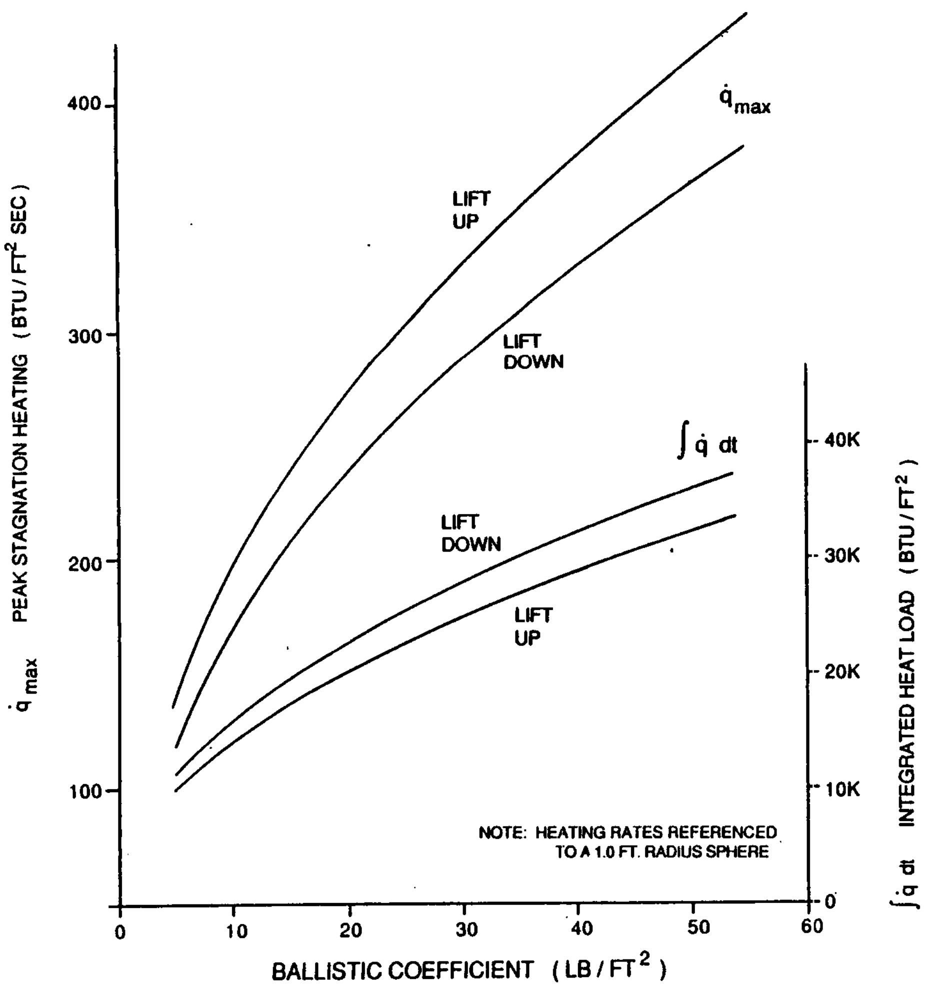
EARTH CAPTURE, C3= 16 - HEATING



- 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

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EARTH CAPTURE AERO-ENTRY ERROR ANALYSIS: C3=32

This figure 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. The 3.54 nm net control corridor size sets a minimum L/D requirement of 0.155 for the entry vehicle when control parametrics (next chart) are utilized.

EARTH CAPTURE AERO-ENTRY ERROR ANALYSIS: C3=32

	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
	$\boxed{= \pm 8100 \text{ FT}}$	$\boxed{= \pm 1.33 \text{ NM NET VARIATION}}$

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

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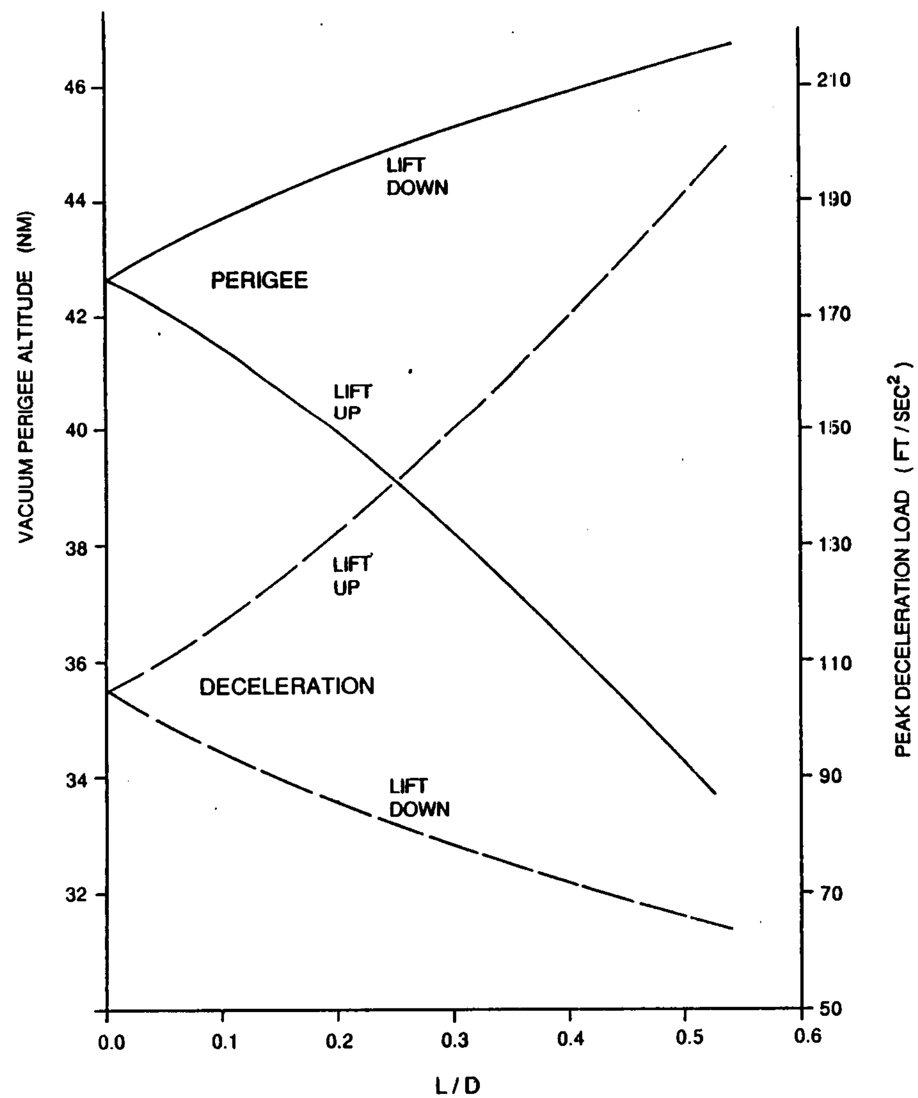
EARTH CAPTURE, C3=32 - CONTROL & LOADS



- 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

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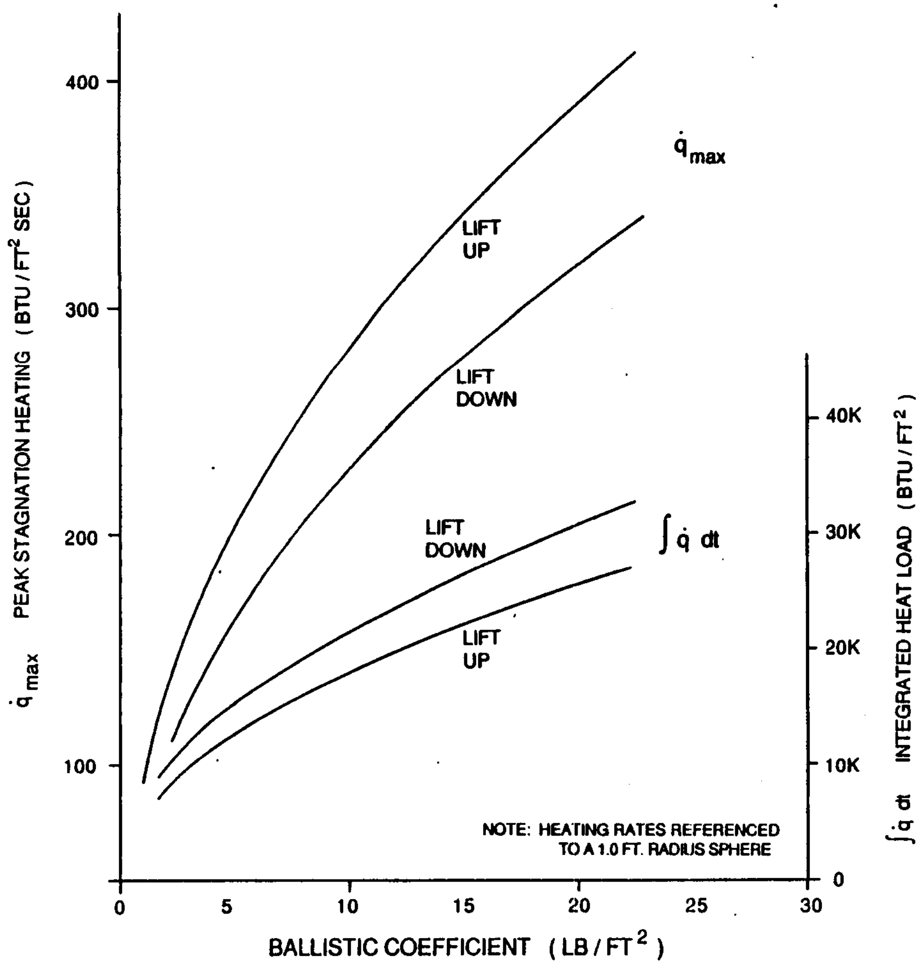
EARTH CAPTURE, C3= 32 - HEATING



- 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

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EARTH CAPTURE AERO-ENTRY ERROR ANALYSIS: C3=68

This figure 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. The 4.35 nm net control corridor size sets a minimum L/D requirement of 0.13 for the entry vehicle when control parametrics (next chart) are utilized.

EARTH CAPTURE AERO-ENTRY ERROR ANALYSIS: C3=68

	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

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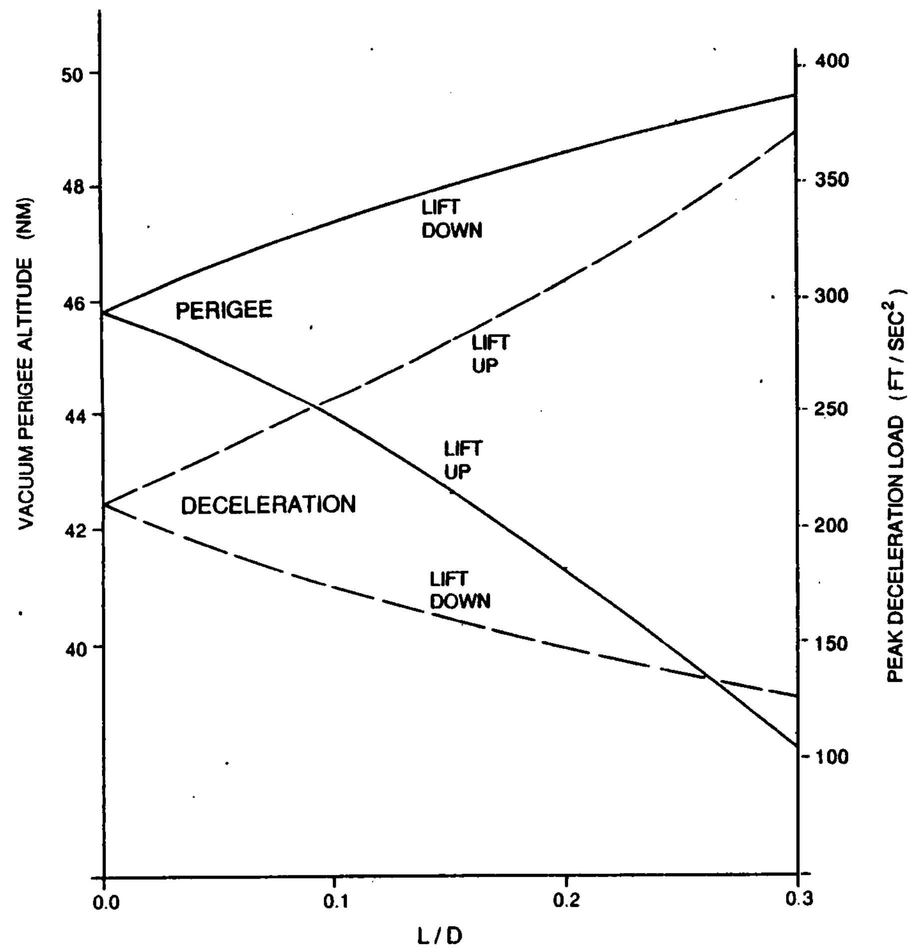
EARTH CAPTURE, C3=68 - CONTROL & LOADS



- 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

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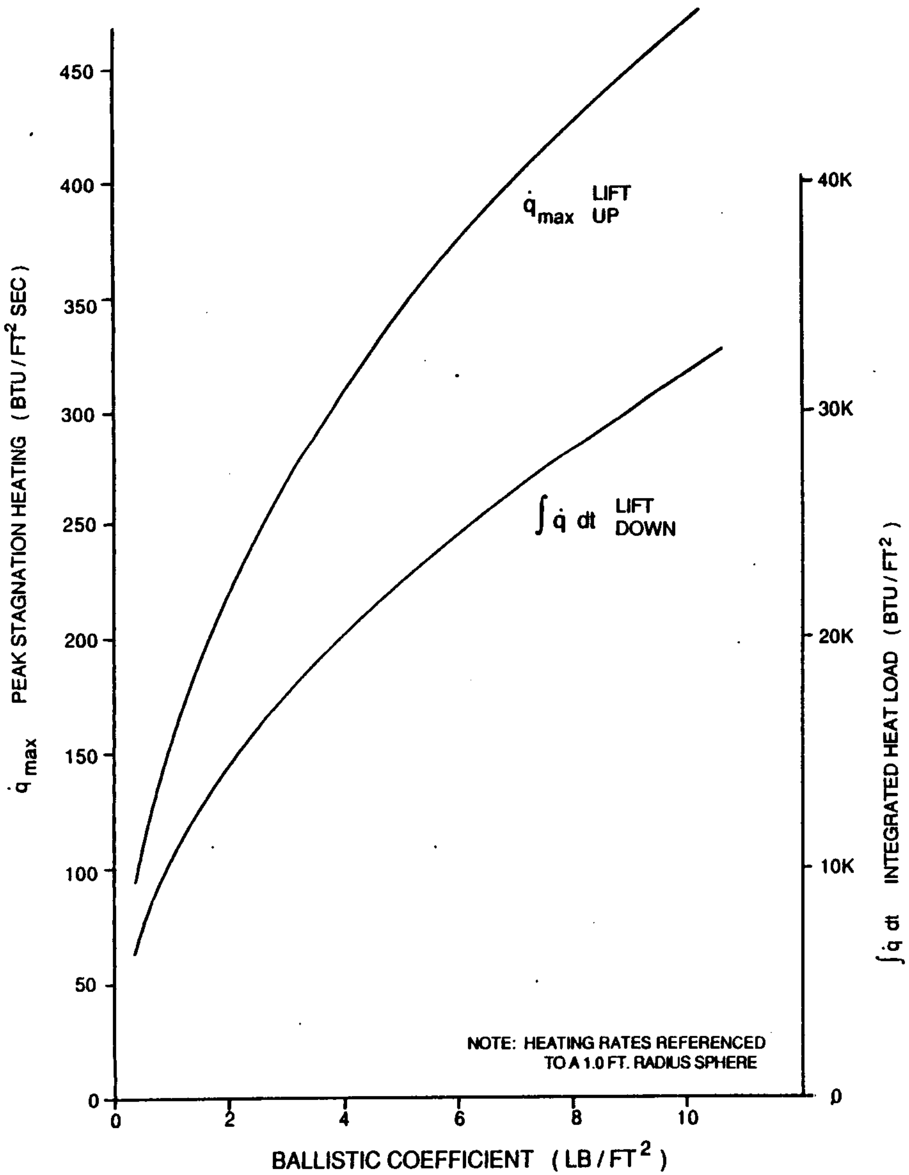
EARTH CAPTURE, C3= 68 - HEATING



- 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

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HIGH SPEED AEROASSIST

DATA SUMMARY

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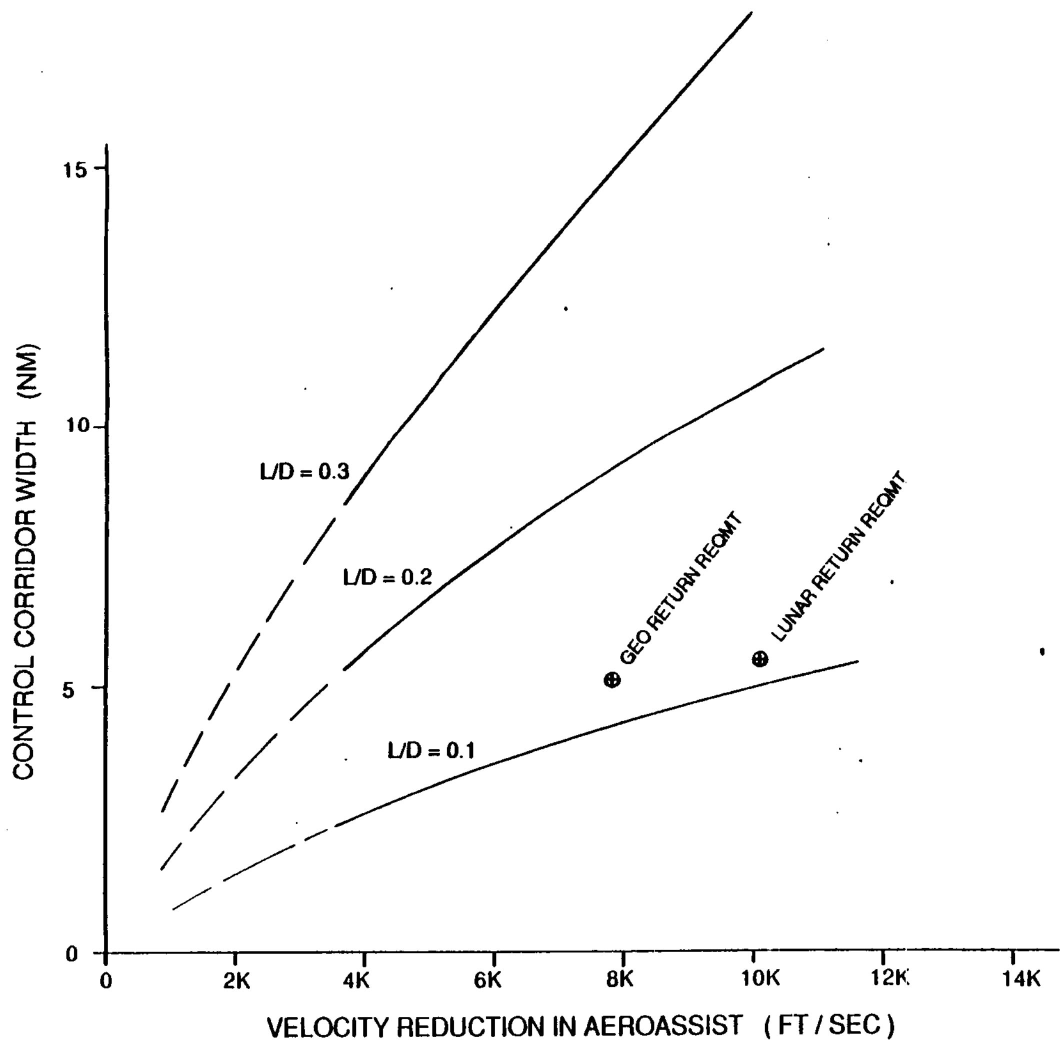
CONTROL VS AERO DELTA-V: EARTH RETURN

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 the following chart which plots control corridor magnitudes for given L/D values vs aeroassist velocity reduction.

The higher the aero ΔV the larger the control corridor (i.e. the larger the amount of trajectory control available). These trends are 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. It may be seen that although the control corridor requirements grow 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.

CONTROL VS AERO DELTA-V : EARTH RETURN



- L/D EFFECT VS AERO DELTA-V FOR GEO & LUNAR RETURN
- EXIT APOGEE = 245 NM
- CONTROL REQUIREMENTS FROM ERROR ANALYSIS

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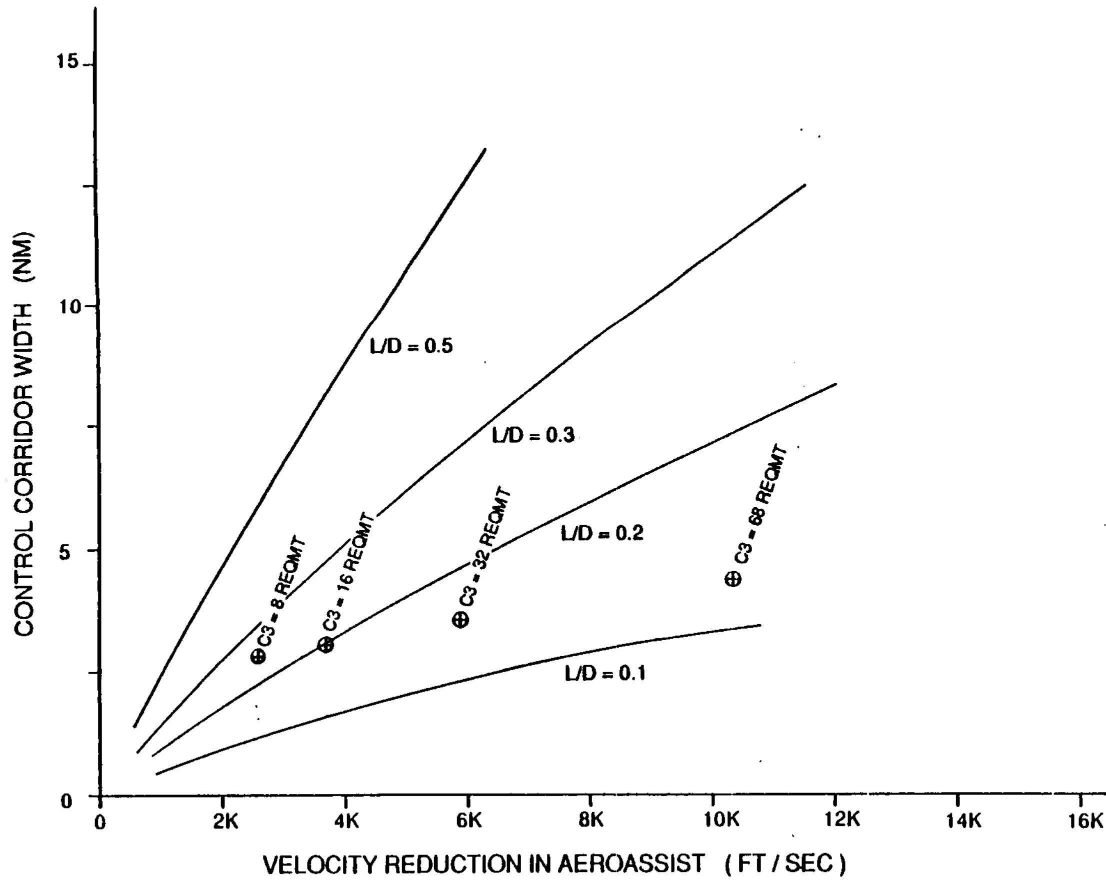
CONTROL VS AERO DELTA-V: EARTH CAPTURE

This figure 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.

CONTROL VS AERO DELTA-V : EARTH CAPTURE



- L/D EFFECT VS AERO DELTA-V FOR EARTH CAPTURE
- ENTRY C3 = 8, 16, 32, 68
- EXIT APOGEE = 38485 NM
- CONTROL REQUIREMENTS FROM ERROR ANALYSIS

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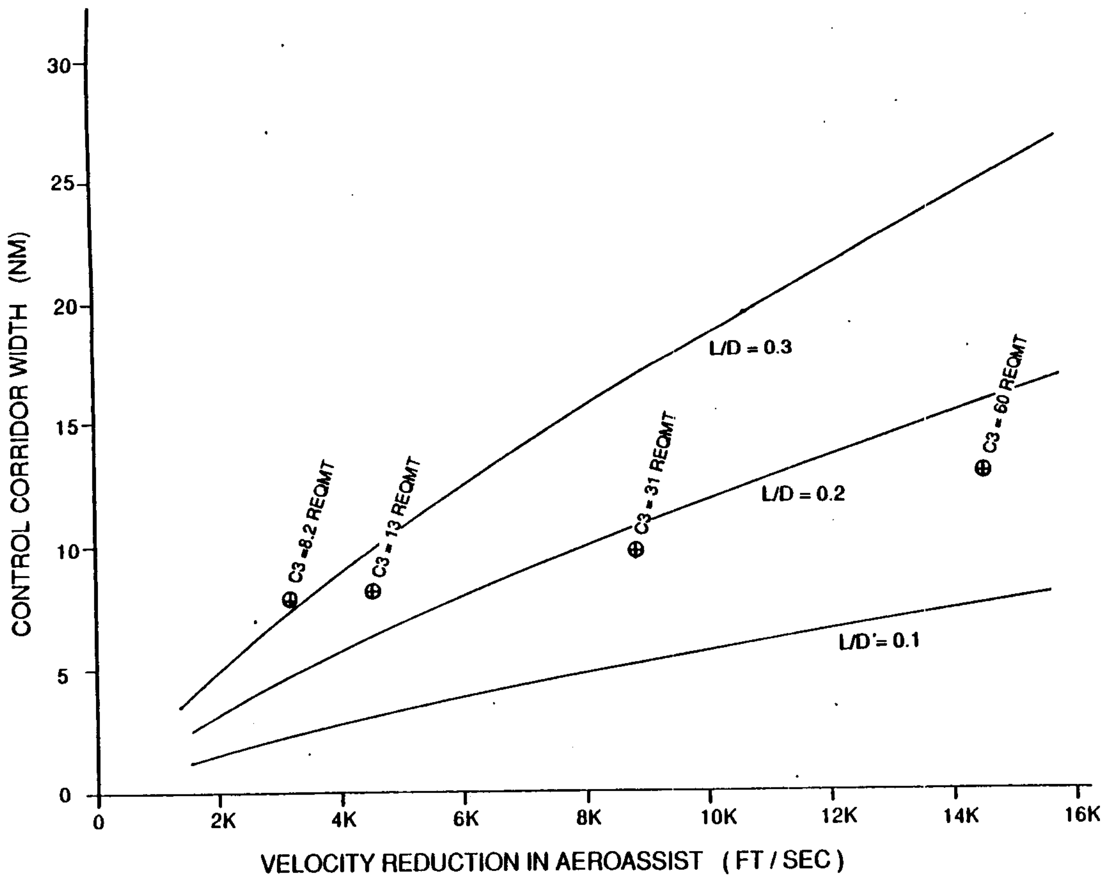
CONTROL VS AERO DELTA-V: MARS CAPTURE

This figure 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 with the previous graphs for the Earth aeroassist, 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). Again, as with the Earth cases 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.

CONTROL VS AERO DELTA-V : MARS CAPTURE



- L/D EFFECT VS AERO DELTA-V FOR MARS CAPTURE
- ENTRY C3 = 8.2, 13, 31, 60
- EXIT APOGEE = 18108 NM
- CONTROL REQUIREMENTS FROM ERROR ANALYSIS

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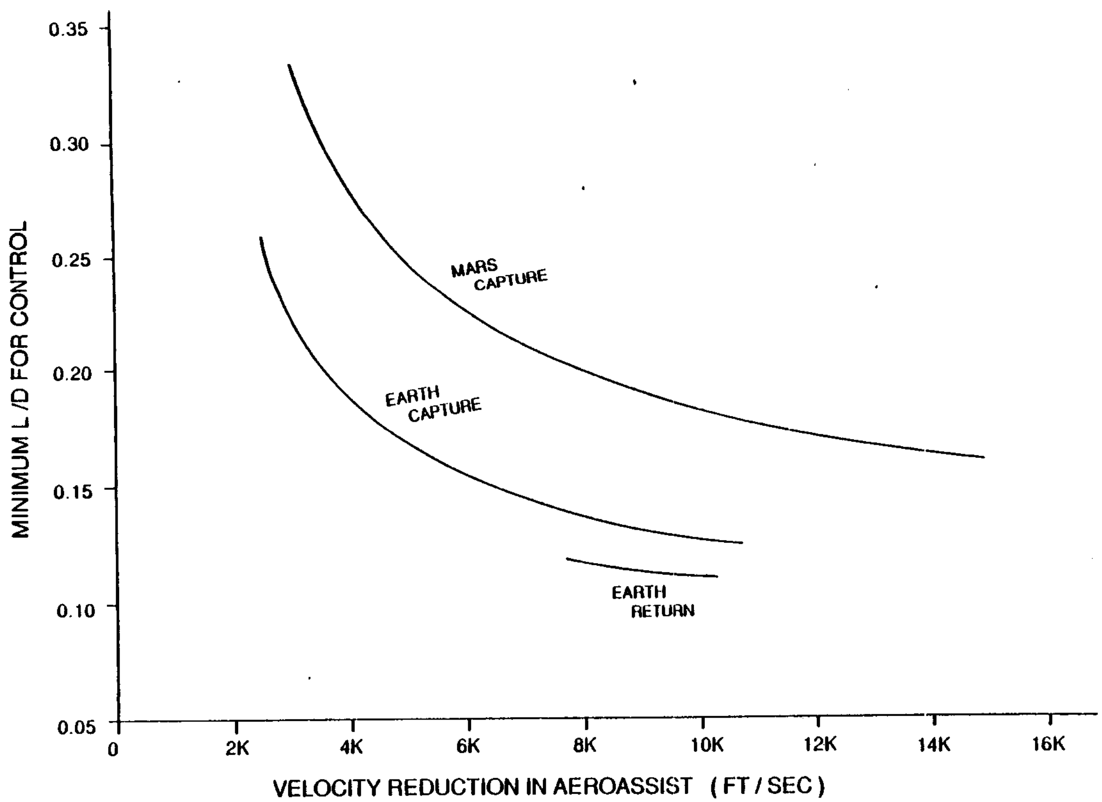
C-5

MINIMUM L/D REQUIREMENTS FOR AEROASSIST

This figure shows the decreasing L/D requirements for increasingly energetic aeroassist maneuvers. As the three previous charts 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 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.

MINIMUM L/D REQUIREMENTS FOR AEROASSIST



- L/D REQMTS VS AERO DELTA-V
- EARTH RETURN
  EXIT APOGEE = 245 NM
- EARTH CAPTURE
  EXIT APOGEE = 38485 NM
- MARS CAPTURE
  EXIT APOGEE = 18108 NM
- CONTROL REQUIREMENTS
  FROM ERROR ANALYSIS

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LUNAR RETURN AEROBRAKE

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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 figure), 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 neccesary to expand the control capability such that when 25% of it is eliminated, the remaining peice 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 two figures down 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.

LUNAR LOAD RELIEF

LUNAR RETURN REQUIRES 5.53 NM AERO CONTROL CORRIDOR

- MINIMUM L/D = 0.11
- RESULTING PEAK g-LEVELS = 4.8
- THIS WOULD REQUIRE LARGE STRUCTURAL MODS TO OTV

INCREASE L/D TO 0.14 (CONTROL CORRIDOR CAPABILITY = 7.3 NM)

- FLY VEHICLE IN UPPER PART OF CONTROL CORRIDOR
- RESULTING PEAK g-LEVELS = 4.0
- INCREASED AEROBRAKE WEIGHT (LARGER DIAMETER, THICKER TPS)
- LESSER OTV CORE WEIGHT INCREASE (LOWER PEAK g-LEVELS)

LOAD RELIEF MAXIMIZES LUNAR & GEO OTV CORE COMMONALITY

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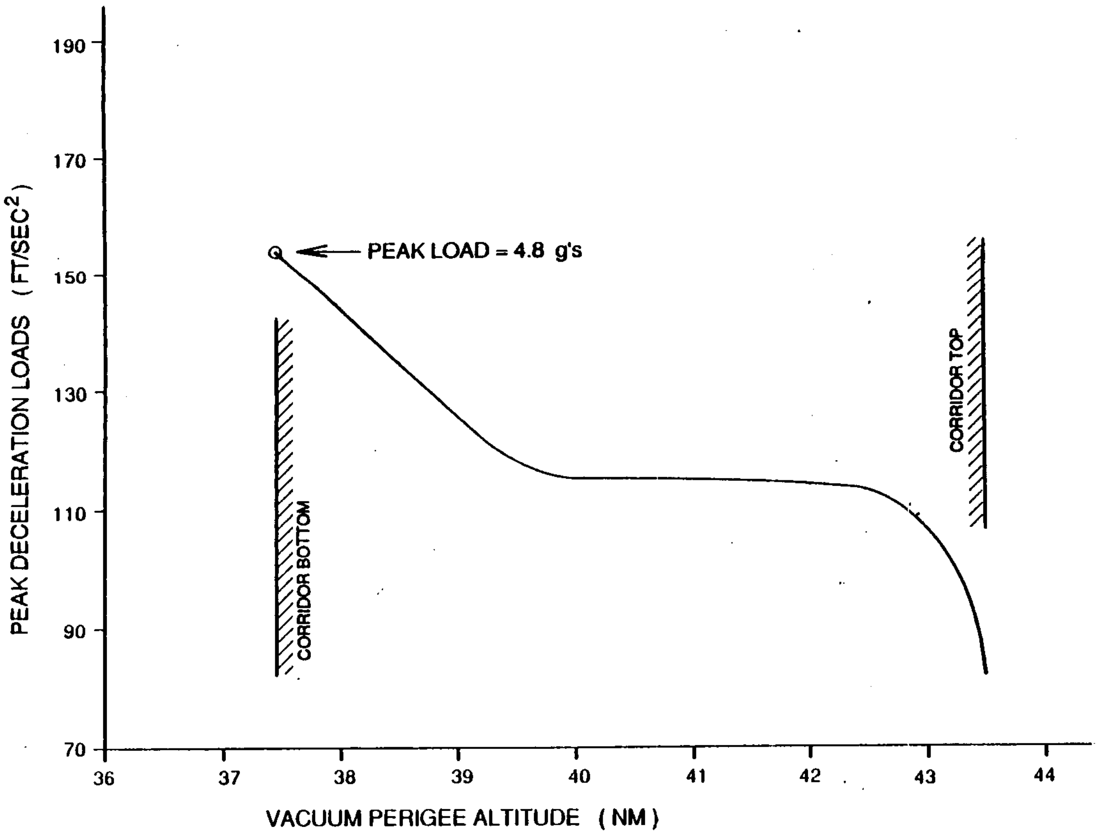
LUNAR LOADS, L/D = 0.12

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

LUNAR LOADS, L/D = 0.12



- LUNAR RETURN
  (PRE-ENTRY APOGEE
  = 287700 NM)
- SPACE STATION PICK-UP
  (ALTITUDE = 245 NM)
- L/D = 0.12
- CONTROL CORRIDOR = 5.5 NM
- PEAK g-LEVEL = 4.8

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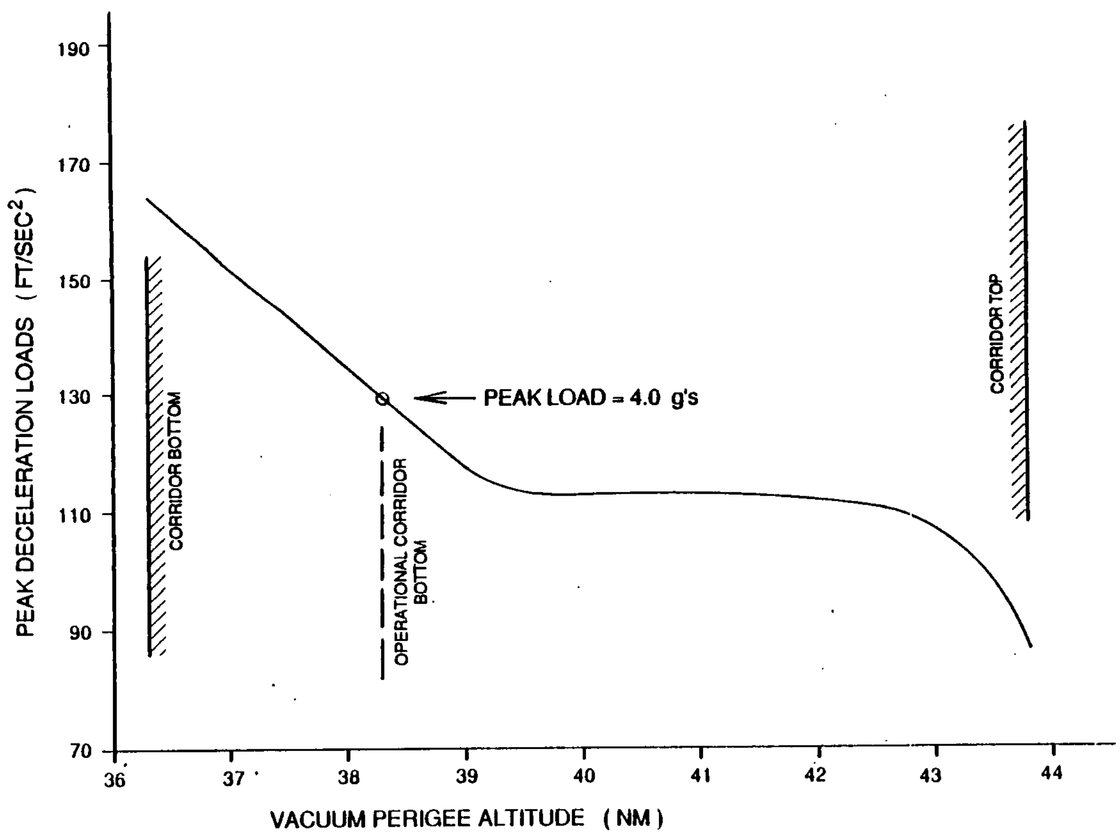
LUNAR LOADs, L/D = 0.14

This chart 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 utilizing the upper 5.5 nm for flight, peak loads are reduced to 4.0 g's.

LUNAR LOADS, g-RELIEF: L/D = 0.14



- 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

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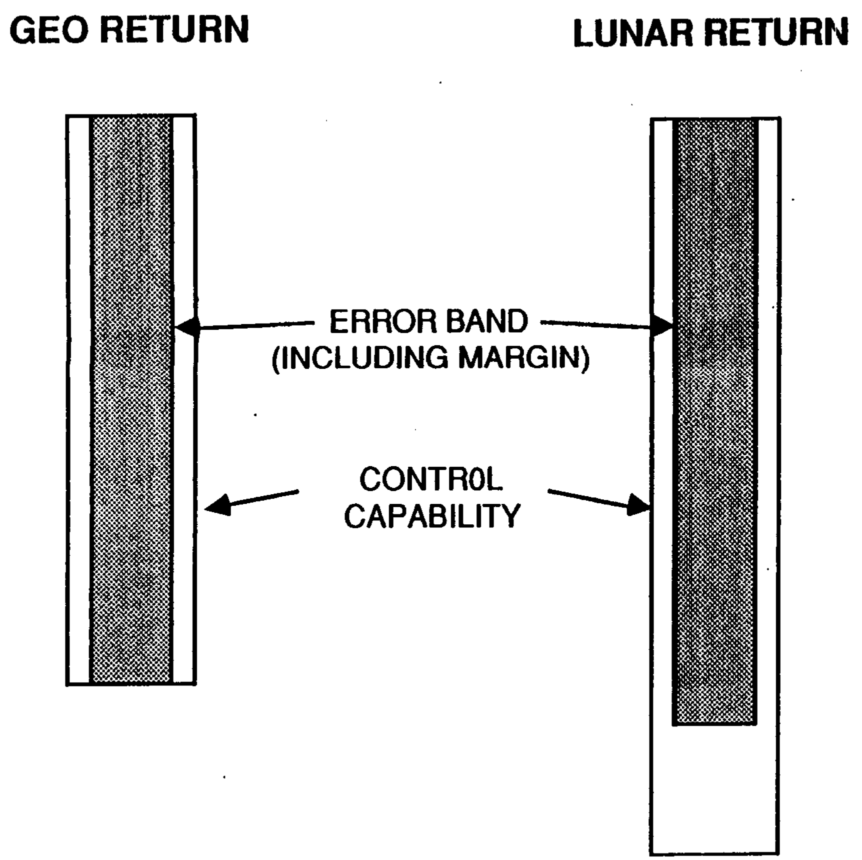
LUNAR AERO CONTROL

This chart illustrates the principal of using excess control for load relief. The basic 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 AERO LOAD RELIEF



<u>GEO RETURN</u>

ACCEPTABLE AERO LOADS

CENTERING ERROR BAND IN CONTROL CAPABILITY GIVES WEIGHT OPTIMUM AEROBRAKE

<u>LUNAR RETURN</u>

OVERSIZE CONTROL CAPABILITY

BIAS ERROR BAND TO TOP OF CONTROL BAND FOR LOAD RELIEF

GIVES ACCEPTABLE LOADS

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LUNAR AEROBRAKE CHARACTERISTICS

In order to assess lunar logistics missions, a design for the lunar return aerobrake had to be undertaken. This chart summarizes the salient features of this device. The heaviest return payload was used which is the 15000 lb manned cab. Load relief, discussed previously, was used to reduce the peak deceleration loads to 4.0 g's.

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.

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

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LUNAR AEROBRAKE WEIGHTS

This chart 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.

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 2298 lb for an increase of 458 lb over the GEO return brake weight.

LUNAR AEROBRAKE WEIGHTS

	LUNAR BRAKE	GEO BRAKE
OTV CORE - 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	1046	799
TOTAL AEROBRAKE WEIGHT	2298	1840

ALL WEIGHTS IN POUNDS

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AEROASSIST

AEROASSIST IS CRUCIAL TO RE-USEABILITY OF OTV

CAPABILITY EXPANDS OTV MISSIONS

- EFFICIENT RETURN / RETRIEVAL FROM HIGH ENERGY MISSIONS
  - GEOSYNCHRONOUS
  - LUNAR & PLANETARY-BOOST
  - MOLNIYA
- ENHANCES HIGH-INCLINATION MISSION PERFORMANCE

CAPABILITY APPLICABLE TO FUTURE MISSIONS

- EXTEND TECHNOLOGY TO PLANETARY CAPTURE
- AERO-TESTBED VEHICLE

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