(NASA-CR-183551) ORBITAL TRANSFER VEHICLE: CONCEPT DEFINITION AND SYSTEM ANALYSIS STUDY Midterm Review (Martin Marietta Corp.) 112 p CSCL 22B

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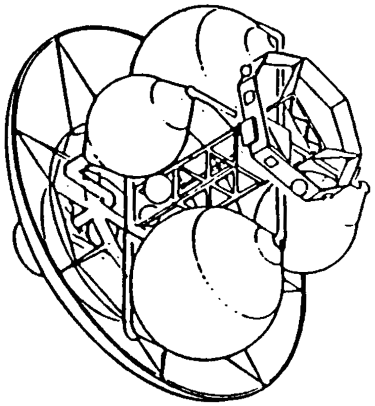
ORBITAL TRANSFER VEHICLE

CONCEPT DEFINITION
AND
SYSTEM ANALYSIS STUDY

CONTRACT EXTENSION II
MIDTERM REVIEW



NASA - MSFC
22 JULY 1987

MARTIN MARIETTA

AGENDA

EXECUTIVE SUMMARY	BILL WILLCOCKSON
RESULTS TO DATE INITIAL OTV PROGRAM LUNAR MISSIONS ACC OTV SAFETY ISSUES AEROASSIST SUMMARY	
PROGRAM / MISSION ISSUES	BILL WILLCOCKSON
MISSION OVERVIEWS ACC OTV SAFETY	
DESIGN ISSUES	LARRY REDD
NEAR TERM EXPENDABLE OTV ENGINE CONFIGURATION TRADES LUNAR LANDING IMPACTS PROGRAM EVOLUTION	
AEROASSIST	BILL WILLCOCKSON
AEROASSIST FOR MANNED MARS MISSION HIGH SPEED ENTRIES	

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EXECUTIVE SUMMARY

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RESULTS TO DATE

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 OTV PROGRAM THAT HAS A STRONG JUSTIFICATION FOR STARTUP
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

MAJOR 1984/85 STUDY FINDINGS

The major conclusions reached during the portion of the study conducted prior to this study extension are summarized in the viewgraph. These conclusions reflect the critical groundrules: 1) Selections are to be justified by the low Revision 8 mission model; 2) The cost of developing a new large capability propellant tanker is to be considered an OTV program cost; 3) a 72,000 pound STS lift capability to an east launch low orbit is available; 3) A space station capable of supporting OTV accommodations will be available in 2000; and 4) The cost of developing an ACC is to be considered an OTV program cost.

MAJOR 1984 / 85 STUDY FINDINGS

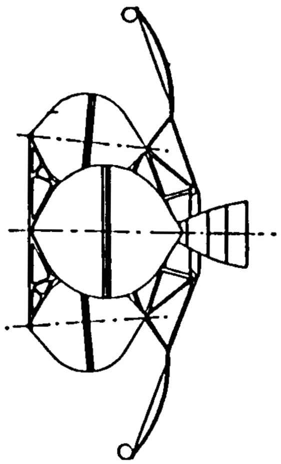
- OTV SHOULD BE REUSABLE, AEROASSISTED AND CRYOGENIC
- A 72 Klb ORBITER, A LOW USE SCENARIO AND SCAVENGING JUSTIFY DEVELOPMENT OF THE ACC AND THE ACC OTV
- AN INITIAL UNMANNED, GROUND BASED OTV SHOULD EVOLVE DIRECTLY TO A MANNED SPACE BASED OTV
- A LOW-USE MISSION MODEL CANNOT JUSTIFY DEVELOPMENT OF A LARGE PROPELLANT TANKER
- STS SCAVENGING CAN PROVIDE MOST OTV PROPELLANT (AND IS THE COST JUSTIFICATION FOR SPACE BASING)
- SPACE BASE ACCOMMODATIONS SHOULD BE INCORPORATED INTO THE SPACE STATION DESIGN
- SPACE BASE OTV SERVICING OPERATIONS SHOULD NOT USE EVA AS A PRIME OPERATIONS MODE

PROGRAM RECOMMENDATIONS (1984/85 STUDY)

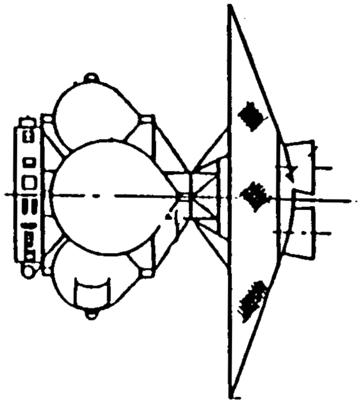
The selected Option I program uses the ground based Aft Cargo Carrier vehicle and the space based vehicle pictured. Both configurations use a four-propellant-tank concept. The ground based configuration is not man-rated and uses one main engine. This engine incorporates new technology, but presses it 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

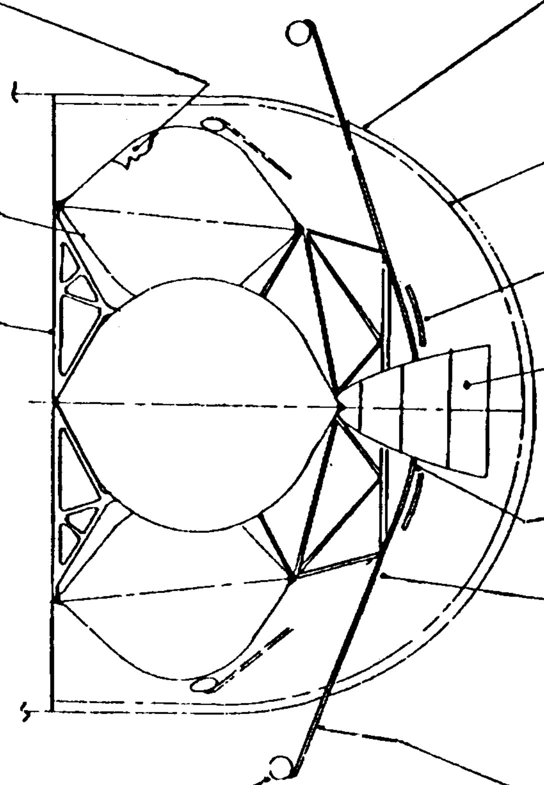
OPTION I

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

GROUND BASED CRYOGENIC OTV - ACC

The general arrangement and weight breakdown of our selected ground based cryogenic OTV is shown in the facing viewgraph. The four tank, single advanced technology engine configuration uses the volume and weight efficient principle suggested by Larry Edwards (NASA Headquarters) to fit easily into the Aft Cargo Carrier (ACC). The 38 foot diameter aerobrake folds forward while stowed in the ACC. It is discarded after flight and not stowed in the orbiter bay for retrieval. The aluminum/lithium propellant tanks are designed by engine inlet pressure requirements. Their thinnest gauges are .018 in. for the LO_2 tank and .014 in. for the LH_2 tank. The tanks are insulated with multi-layer insulation and spray-on foam insulation (SOFI). The hydrogen tanks are removed on-orbit and, with the core system (LO_2 tanks, structure, avionics, and propulsion), are stowed in the orbiter bay for retrieval after mission completion. The structure is of lightweight graphite epoxy. The propellant load was selected to enable full utilization of projected STS lift capability on GEO delivery missions.

GROUND BASED CRYOGENIC OTV



WEIGHT	
AEROBRAKE	1234
TANKS	524
STRUCTURE	774
ENVIRONMENTAL CTRL	424
MAIN PROPULSION	904
ORIENTATION CONTROL	187
ELECTRIC SYSTEMS	613
G, N & C	86
CONTINGENCY (15%)	712
DRY WEIGHT	5459
PROPELLANTS, ETC	50434
LOADED WEIGHT	55893

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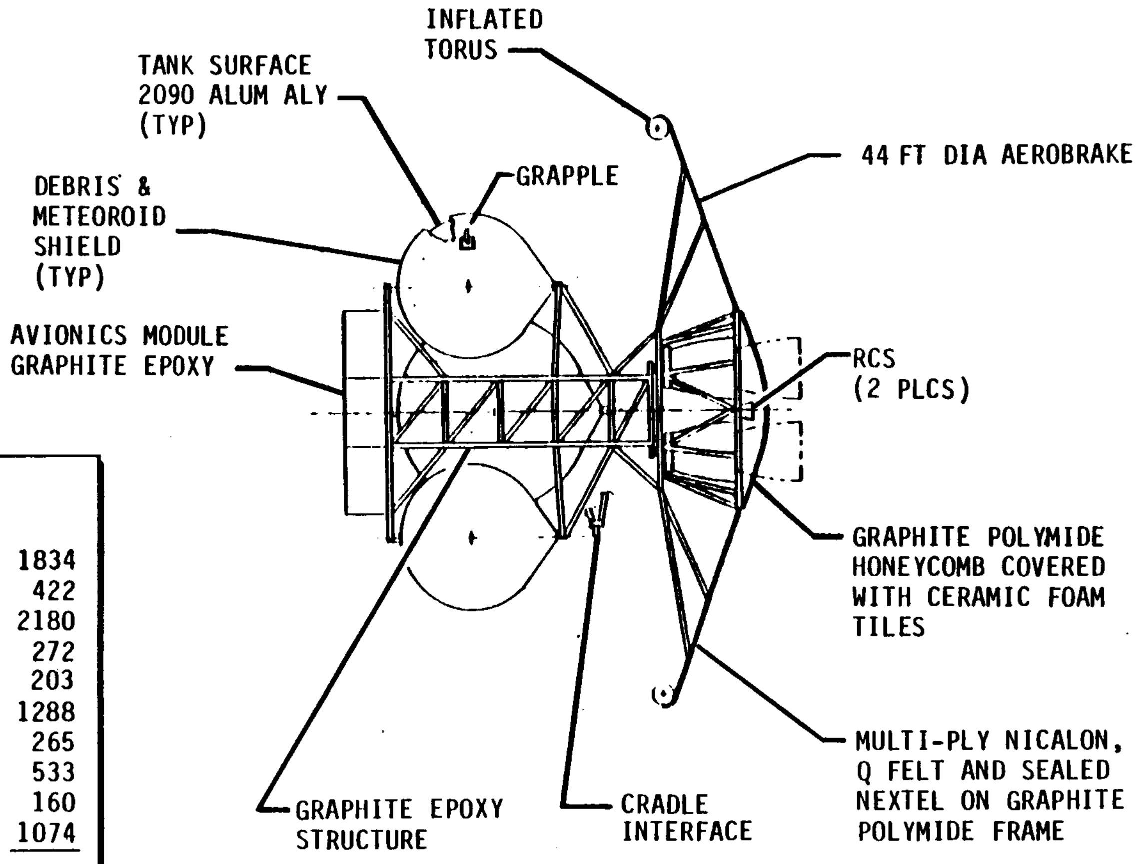
74 K SPACE BASED CRYO OTV

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.

74K SPACE BASED CRYO OTV



WEIGHTS

AEROBRAKE	1834
TANKS	422
STRUCTURE	2180
SUPPORT (ASE)	272
ENVIRONMENTAL CONTROL	203
MAIN PROPULSION	1288
ORIENTATION CONTROL	265
ELECTRIC SYSTEMS	533
G, N&C	160
CONTINGENCY (15%)	1074
DRY WEIGHT	8231
PROPELLANTS, ETC	74015
LOADED WEIGHT	82246

OTV EXTENSION #1 RESULTS

This figure summarizes the key results of the first extension to the OTV Phase A contract. We recommend development of the space base for OTV primarily due to the benefits of hitchiked propellant and launch logistics. Our prefered launch vehicle is the Large Cargo Vehicle (LCV) because of reduced launch costs and increased capacity to orbit. The OTV characteristics that correspond with LCV-launch are a three engine cluster to maintain engine-out capability while minimizing vehicle length, and a common vehicle design for ground based and space based applications. The ground based conclusions reached are very sensitive to the availability of empty Shuttles for return of OTV hardware.

OTV EXTENSION #1 RESULTS

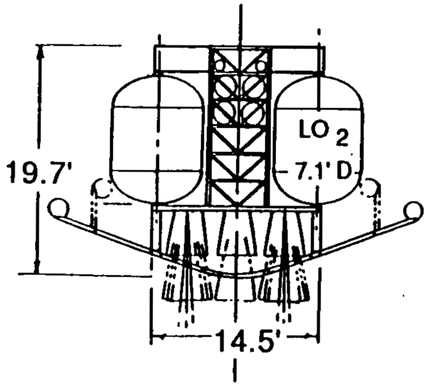
- RECOMMEND DEVELOPING SPACE BASED OTV CAPABILITY
  - ENHANCES OPERATION OF ADVANCED MISSIONS
  - REDUCED BOOSTER LAUNCHES
  - ECONOMIC VIABILITY DEPENDS ON PROPELLANT HITCHIKING AND COST EFFICIENT ACCOMODATIONS
- PREFER A LARGE CARGO VEHICLE BASED PROGRAM
  - 3-ENGINE CONCEPT REDUCES LENGTH FOR MANIFESTING
  - TWO VEHICLE SIZES (52K & 74K PROP LOAD)
  - GROUND & SPACE BASED VEHICLES COMMON
  - CONCLUSIONS DEPENDENT ON STS DOWNLEG AVAILABILITY
- HIGH TRAFFIC OPTIONS JUSTIFY SPECIALIZED OTV
- GROWTH MISSIONS DEFINED DO NOT JUSTIFY SUPER OTV
- SPACE BASING MAKES OTV OPERATIONS COSTS LESS SENSITIVE TO LAUNCH COSTS
- CONFIRM VIABILITY & EFFICIENCY OF LOW L/D CONCEPT AND RIGID/FLEX DESIGN FOR OTV AEROASSIST

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.

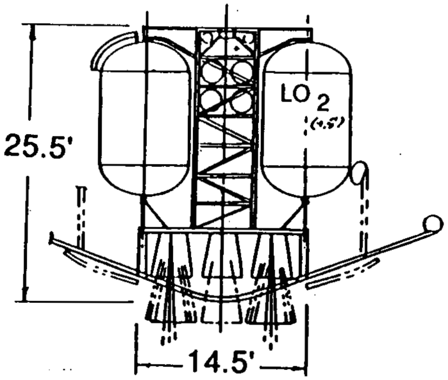
NOMINAL C/V OTV PROGRAM

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

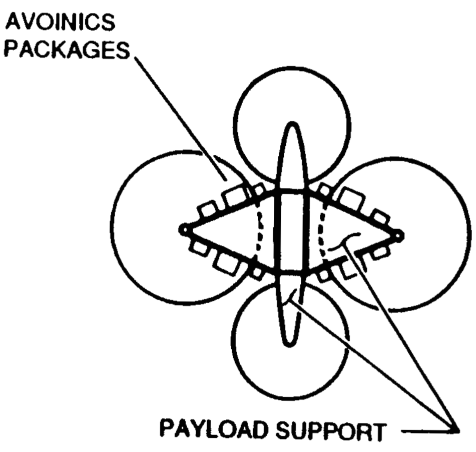
PROGRAM
- DECISIONS BASED ON REV 9, 2/2 MISSION MODEL
- KEY GROUNDRULE: AVAILABLE SHUTTLES TO RECOVER OTV'S
- ONLY TWO OTV CONFIGURATIONS REQUIRED
- 1995 IOC FOR GROUND BASED SYSTEM, 1996 FOR SPACE BASED
- MAN RATED VEHICLE CAN OPERATE FROM GROUND AS WELL
  AS SPACE WITH MINIMUM DELTAS

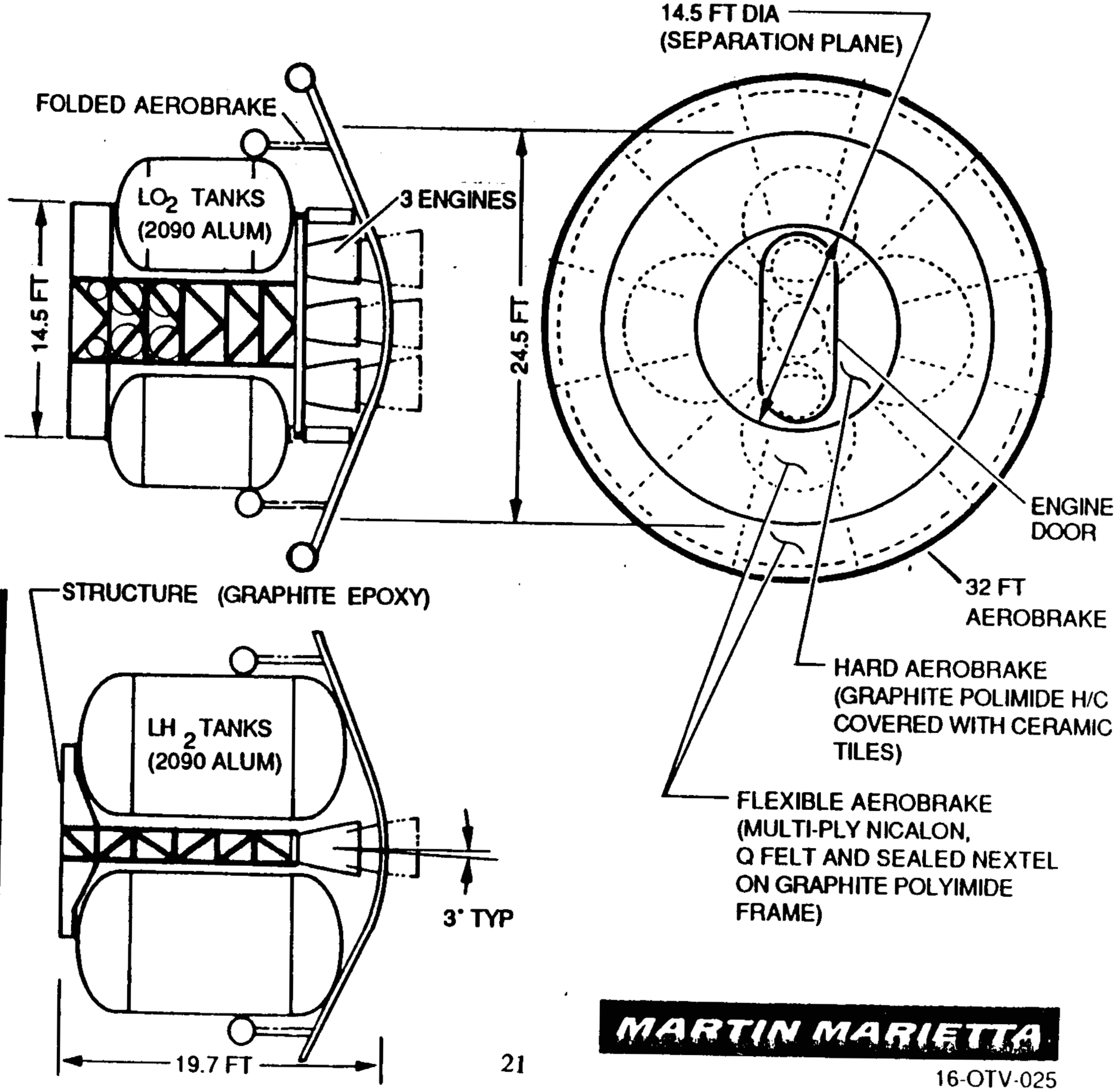
52K GROUND BASED OTV

The vehicle design concept shown in the figure was developed to fit within a 25 ft diameter large cargo vehicle. The tankage diameters were chosen such that the liquid oxygen tanks and the retracted engine lengths would total up to the same length as the liquid hydrogen tanks. This results in a vehicle length that is not driven by one single parameter.

The structure consists of a central core that ties the tankage, aerobrake, and payload adapter together from between the tanks. This assembly remains as a unit after the mission when the aerobrake is jettisoned and the tankage are removed and the structural core is returned to earth with the high cost unit items such as main engines, power system, avionics, RCS, etc.

52K GROUND BASED OTV - WIDE BODY TRANSPORT





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

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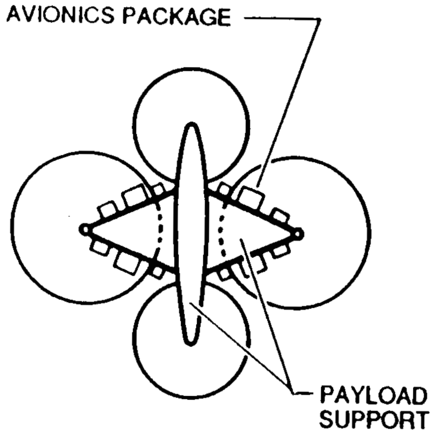
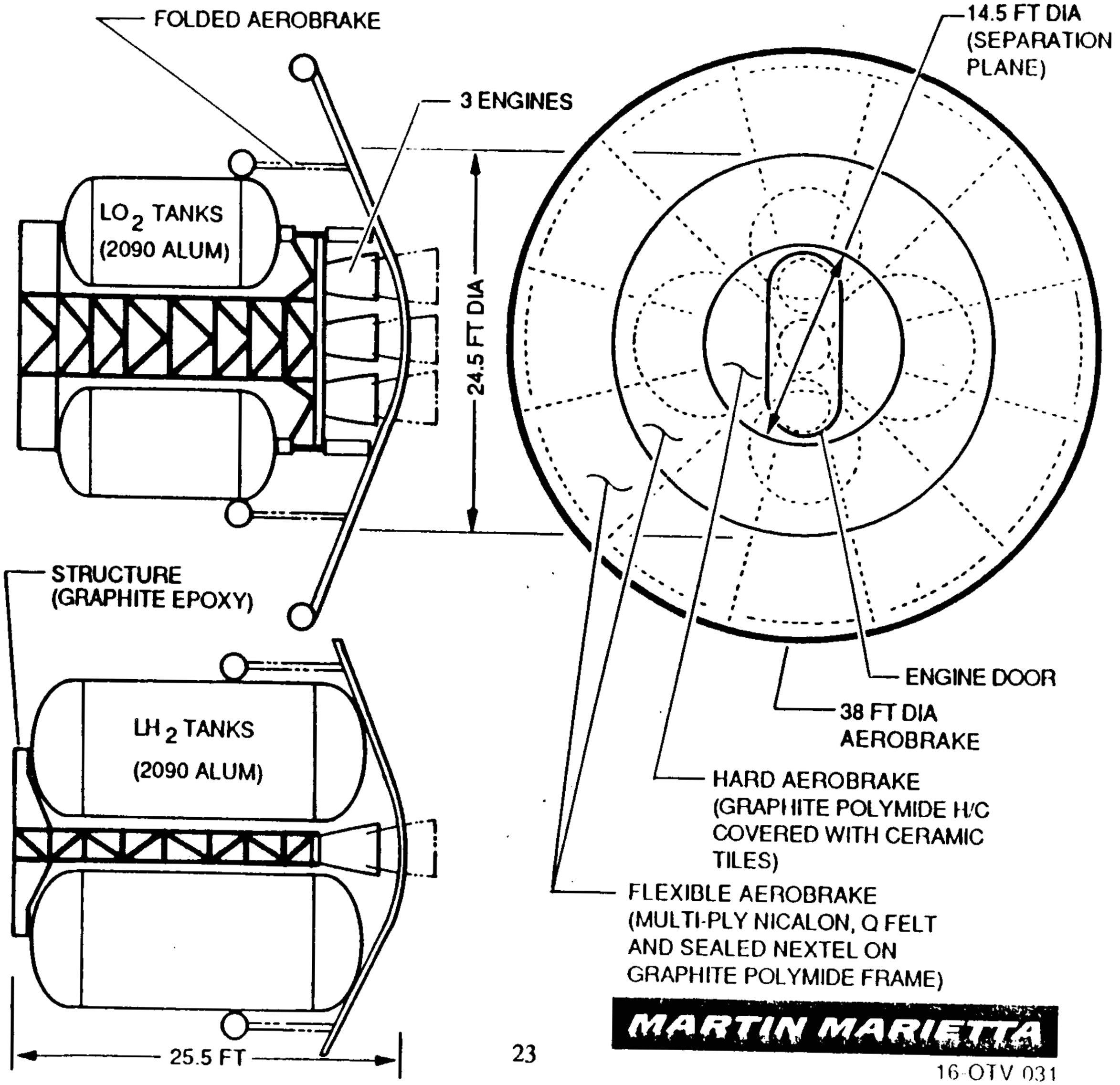
16-OTV-025

74K GROUND BASED OTV

The vehicle concept depicted in the figure is a "stretched" version of the 52K vehicle concept shown earlier. The major modifications are lengthened structure and added length in the propellant tank barrel sections. The aerobrake, of course, grew in diameter to protect the longer stage and corresponding payloads. The core arrangement of the vehicle remained essentially the same with regard to vehicle diameter, engine configuration, avionics location, aerobrake hard shell design, etc.

74K GROUND BASED OTV - WIDE BODY TRANSPORT





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

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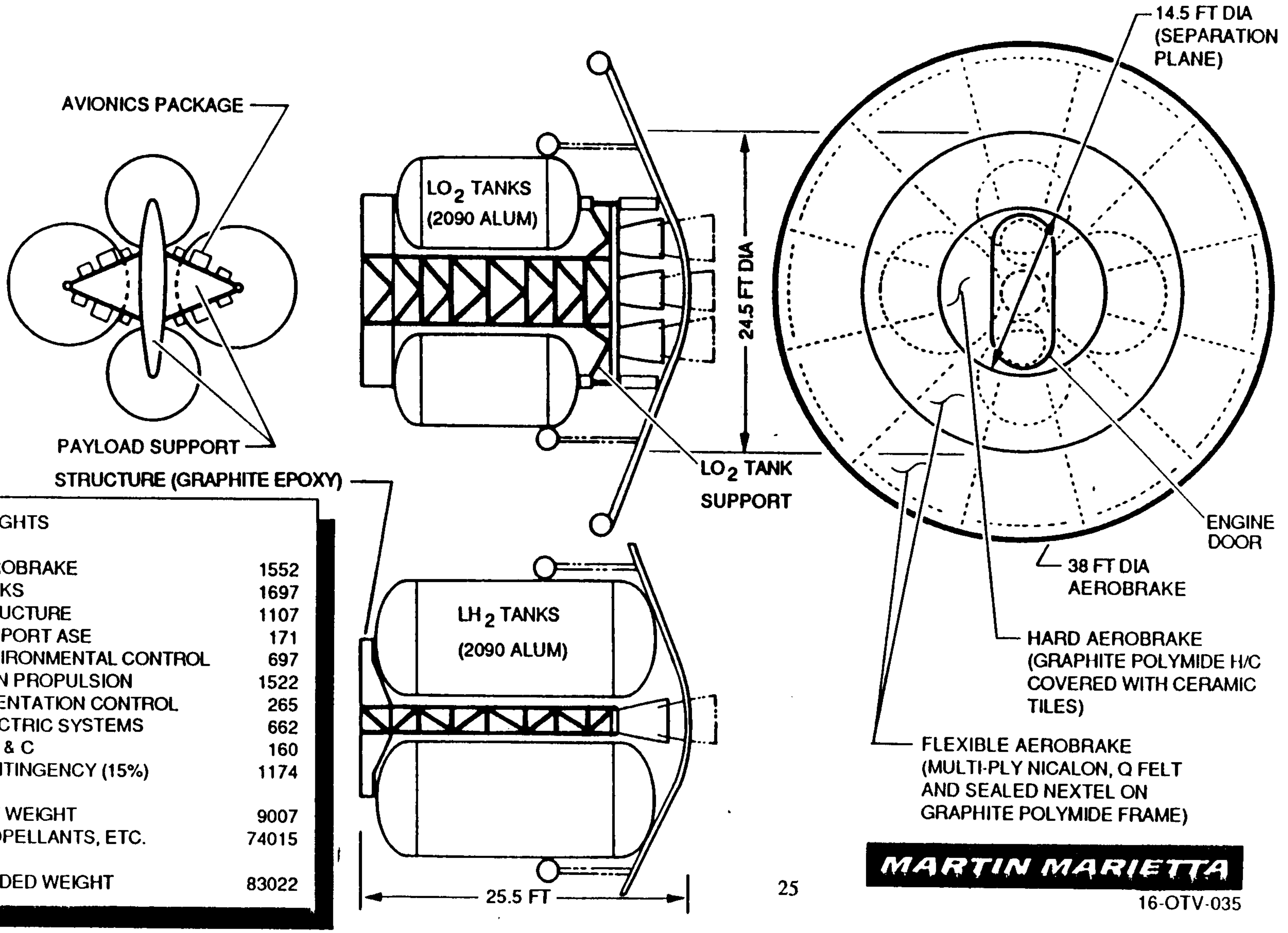
74K SPACE BASED OTV - WIDE BODY TRANSPORT

The figure shows the 74 Klbm propellant capacity OTV (ground based) modified for use as a space based vehicle. The weights reflect these modifications mentioned earlier. The vehicle is intended to be delivered to its space base in one piece by the large cargo vehicle, and then accommodated and operated out of this space base for its useful life as an OTV. The reason for only one size of space based OTV is that the propellant could be saved by having a smaller OTV (in addition to the large one) is small compared with the development cost for the extra size stage.

74K SPACE BASED OTV - WIDE BODY TRANSPORT



WEIGHTS	
AEROBRAKE	1552
TANKS	1697
STRUCTURE	1107
SUPPORT ASE	171
ENVIRONMENTAL CONTROL	697
MAIN PROPULSION	1522
ORIENTATION CONTROL	265
ELECTRIC SYSTEMS	662
G, N & C	160
CONTINGENCY (15%)	1174
DRY WEIGHT	9007
PROPELLANTS, ETC.	74015
LOADED WEIGHT	83022

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16-OTV-035

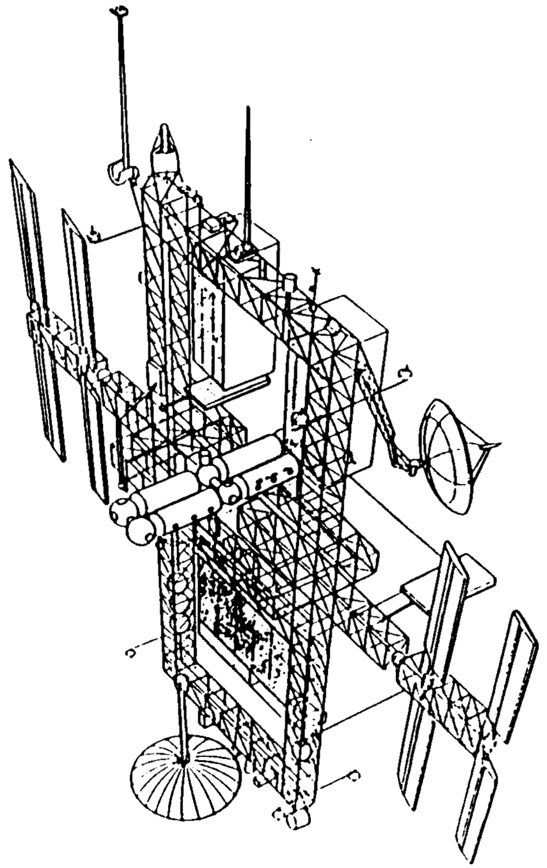
SPACE BASED OTV ACCOMMODATIONS ISSUES

The space base OTV support operations concept we have evolved over the past two years is summarized in the viewgraph. All OTV launch preparation and maintenance operations will nominally be performed without EVA. Initial operations will be performed using remote manipulators directed by an IVA crewman with limited high payoff automation supporting his activities. The growth trend will be towards more automation as the program evolves. Maintenance will be at a major orbital replacement unit level, e.g. tanks, engines, aeroshield, integrated avionics boxes, et. al.

Development build and transportation cost associated with space base accommodations must be minimized to improve the LCC picture as viewed in discounted dollars. Software cost is a major issue. Accommodations should be shared with OMV to the maximum extent possible. Hangar size and propellant farm size should be kept as small as possible to reduce the investment.

Propellant farm efficiency is also an issue. Our studies to date indicate that a cryogenic propellant farm can reflect a basically passive approach as long as its boiloff can be used by the Space Station. We understand that the Space Station program is likely to baseline hydrogen/oxygen for its reboost system. This provides a user for boiled off farm propellants. The remaining problem is to keep the quantities involved at the right level and at a suitable mixture ratio.

SPACE BASED OTV ACCOMMODATIONS ISSUES



OPERATIONS CONCEPT
- EVA RESERVED FOR CONTINGENCY
- MAINTAIN AT MAJOR ELEMENT LEVEL
- INITIAL CAPABILITY USES REMOTE MANIPULATION

DEVELOPMENT AND BUILD COST
- FRONT END COST MUST BE MINIMIZED
- SOFTWARE IS KEY ISSUE
- SHARING WITH OMV PROGRAM FEASIBLE AND NECESSARY

PROPELLANT FARM EFFICIENCY
- GH2/GO2 REBOOST ANTICIPATED
- GOAL: NEAR PASSIVE PROPELLANT FARM
- GOAL: BOILOFF MATCHED TO REBOOST NEED

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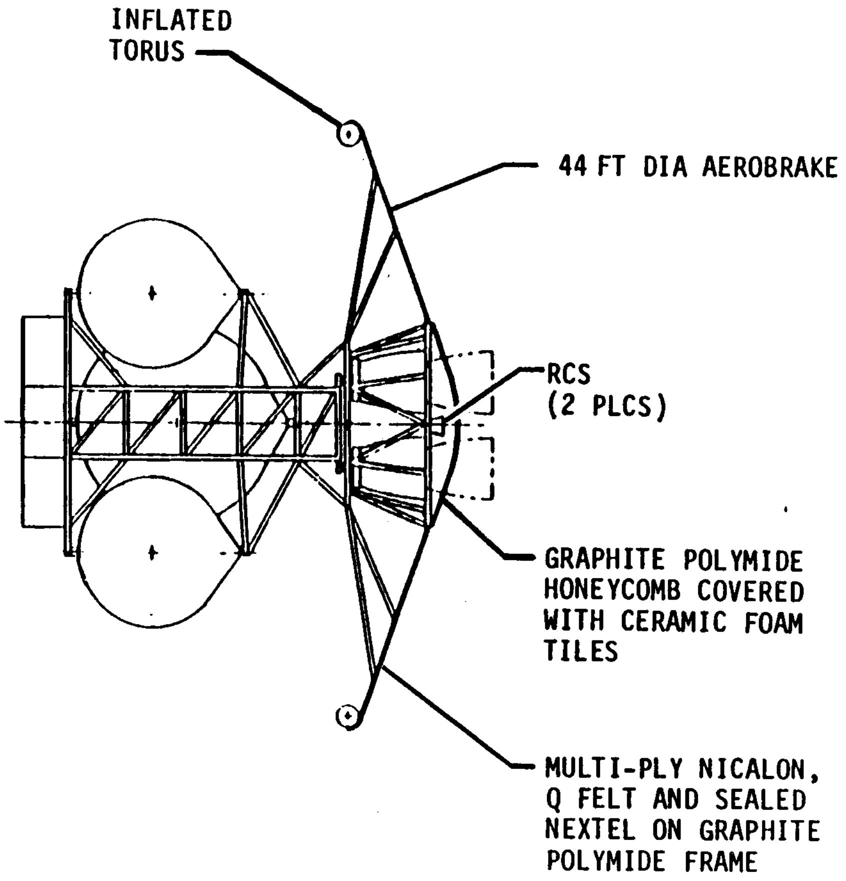
RECOMMENDED AEROASSIST CONCEPT

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

INITIAL OTV PROGRAM

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OTV EXTENSION #2 TASKS

DEFINE AN INITIAL OTV PROGRAM THAT HAS STRONG RATIONAL FOR INITIATION

DEVELOP EVOLUTION TO LONG TERM ADVANCED MISSIONS

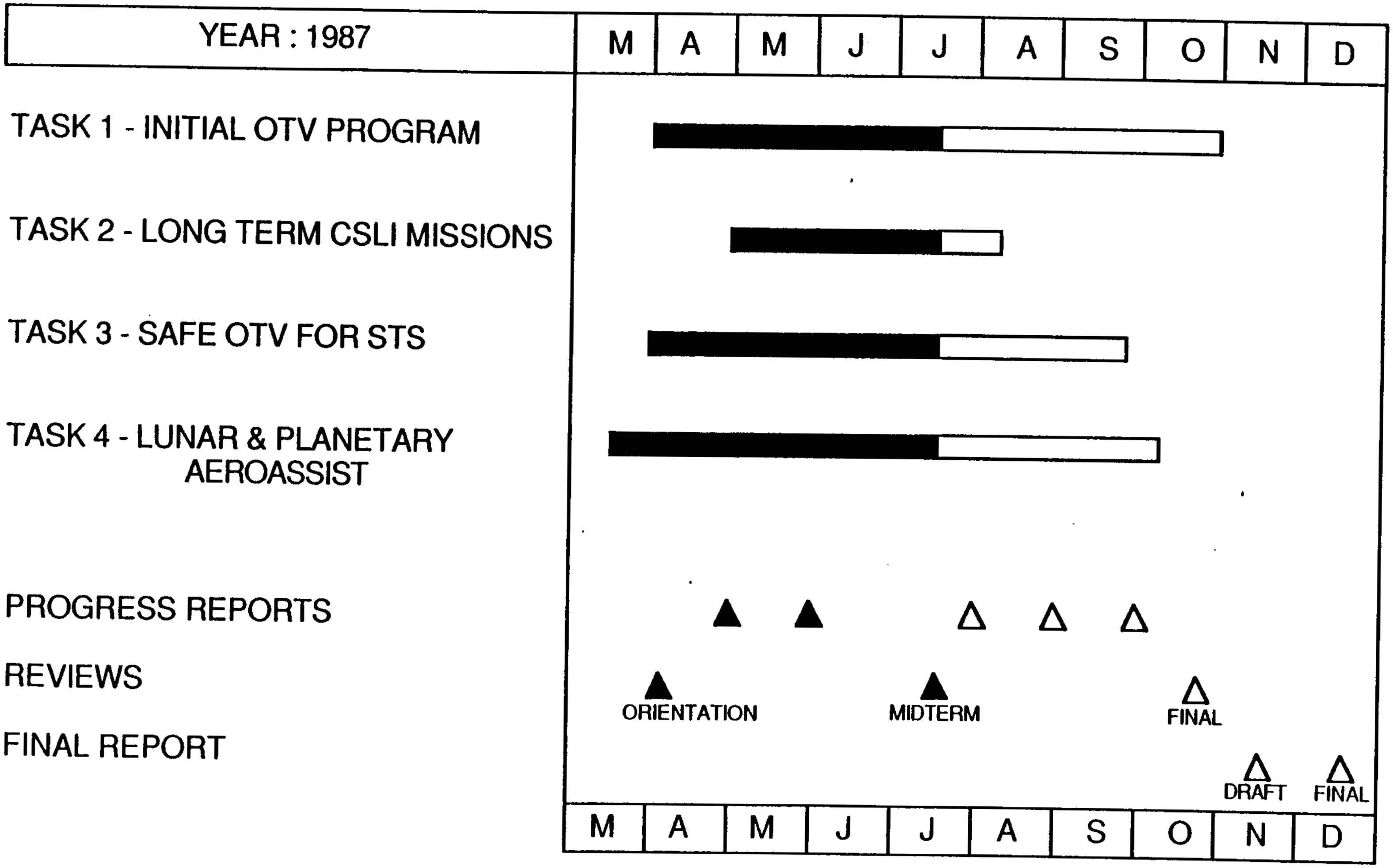
INVESTIGATE SAFETY IMPLICATIONS OF AN ACC OTV

EXPAND ANALYSIS OF HIGH SPEED AEROASSIST

OTV EXTENSION #2 SCHEDULE

This figure shows the operating schedule for the OTV Phase A, Extension #2. As shown, work is proceeding in all four study areas at the time of this midterm. The Long Term CSLI Mission study area will probably be extended somewhat. Work is proceeding satisfactorily toward a conclusion of study activities in December.

OTV EXTENSION #2 SCHEDULE



DRIVER MISSIONS

This figure summarizes the current set of driver missions supplied by MSFC. The baseline scenario 1 is combined with any one of three alternate scenarios to build a driver mission set. This results in either a Earth, Unmanned Planetary, or Lunar Initiative Scenario. The payload and time-phasing requirements are as indicated.

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 + EARTH INITIATIVE

OPTION # 2: BASELINE SCENARIO 1 + UNMANNED PLANETARY

OPTION # 3: BASELINE SCENARIO 1 + LUNAR INITIATIVE

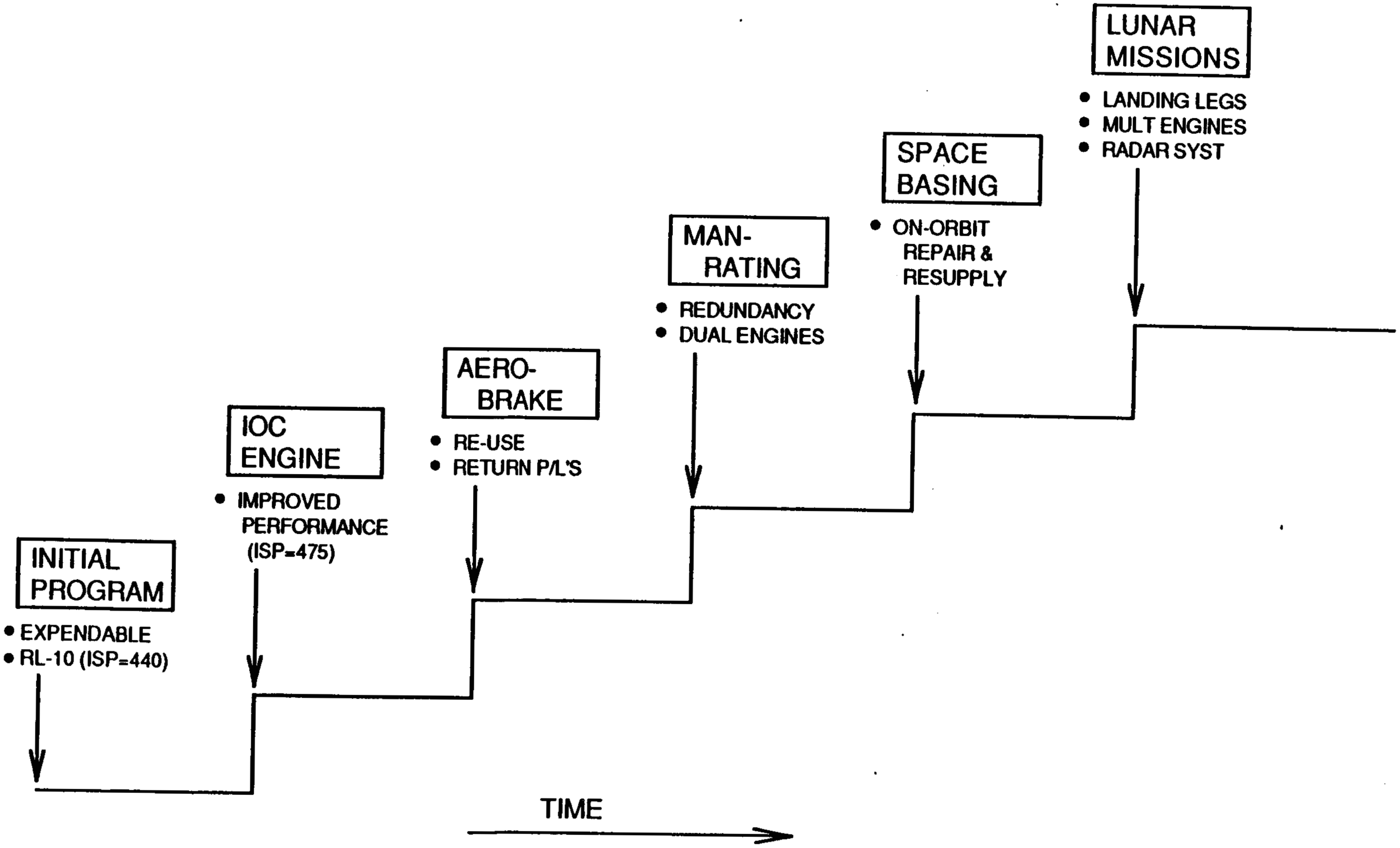
OTV PHASED GROWTH

This figure shows how the OTV program could grow in phases from fairly modest beginnings as an expendable vehicle through reusability ultimately to Lunar capability. The phases, taken in steps, give a growth-oriented program that is affordable.

The initial program step is an expendable vehicle which utilizes the proven RL-10 engine. By beginning expendable, the startup cost of the program can be minimized. The first upgrade to the vehicle is incorporation of the advanced IOC engine which raises ISP from 440 sec to 475 sec. This step increases performance with fairly well-understood technology. The next step is the incorporation of the more uncertain technology of aeroassist. Although the development unknowns are larger, the payoff is great as it allows efficient vehicle recovery as well as payload retrieval missions. With round-trip capability, the next logical step is to provide capability for manned missions. Man-rating the vehicle requires multiple engines as well as increased redundancy in the avionics. Space basing is brought on line when mission logistics become too cumbersome for the ground based system. Finally the ultimate capability currently envisioned is reached when round-trip Lunar missions. This last step requires landing legs, a new engine configuration, and avionics modifications.

By growing the vehicle in a step-by-step fashion, capabilities are achieved as they are needed which will reduce the front-end cost of the program.

OTV PHASED GROWTH



INITIAL PROGRAM
- EXPENDABLE
- RL-10 (ISP=440)

IOC ENGINE
- IMPROVED PERFORMANCE (ISP=475)

AERO-BRAKE
- RE-USE
- RETURN P/L'S

MAN-RATING
- REDUNDANCY
- DUAL ENGINES

SPACE BASING
- ON-ORBIT REPAIR & RESUPPLY

LUNAR MISSIONS
- LANDING LEGS
- MULT ENGINES
- RADAR SYST

TIME

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RESULTS FROM MISSION CAPTURE ANALYSIS

CORE MODEL - REUSABLE, AEROBRAKED VEHICLE (IOC ENGINE OR RL10-IIB) REQUIRED IN 1998 FOR MULTIPLE PAYLOAD DELIVERY

EARTH INITIATIVE - IOC ENGINE REQUIRED IN 1996 FOR 25K GEO LOW THRUST MISSION

- LARGE SPACE BASED VEHICLE REQUIRED FOR GEO SERVICING

LUNAR INITIATIVE - LARGE SPACE BASED VEHICLE REQUIRED FOR MANNED AND UNMANNED MISSIONS

- 4 ENGINE MODULE, 21:1 THROTTLING RATIO
- LANDING LEGS, RADAR

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

DESIGN ISSUES FOR NEAR TERM EXPENDABLE

The table shows the items that would differ between a ground based reusable vehicle and an expendable predecessor. The obvious difference between a reusable aeroassisted vehicle and an expendable version is the aerobrake. The brake can be 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 also 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.

DESIGN ISSUES FOR NEAR TERM EXPENDABLE

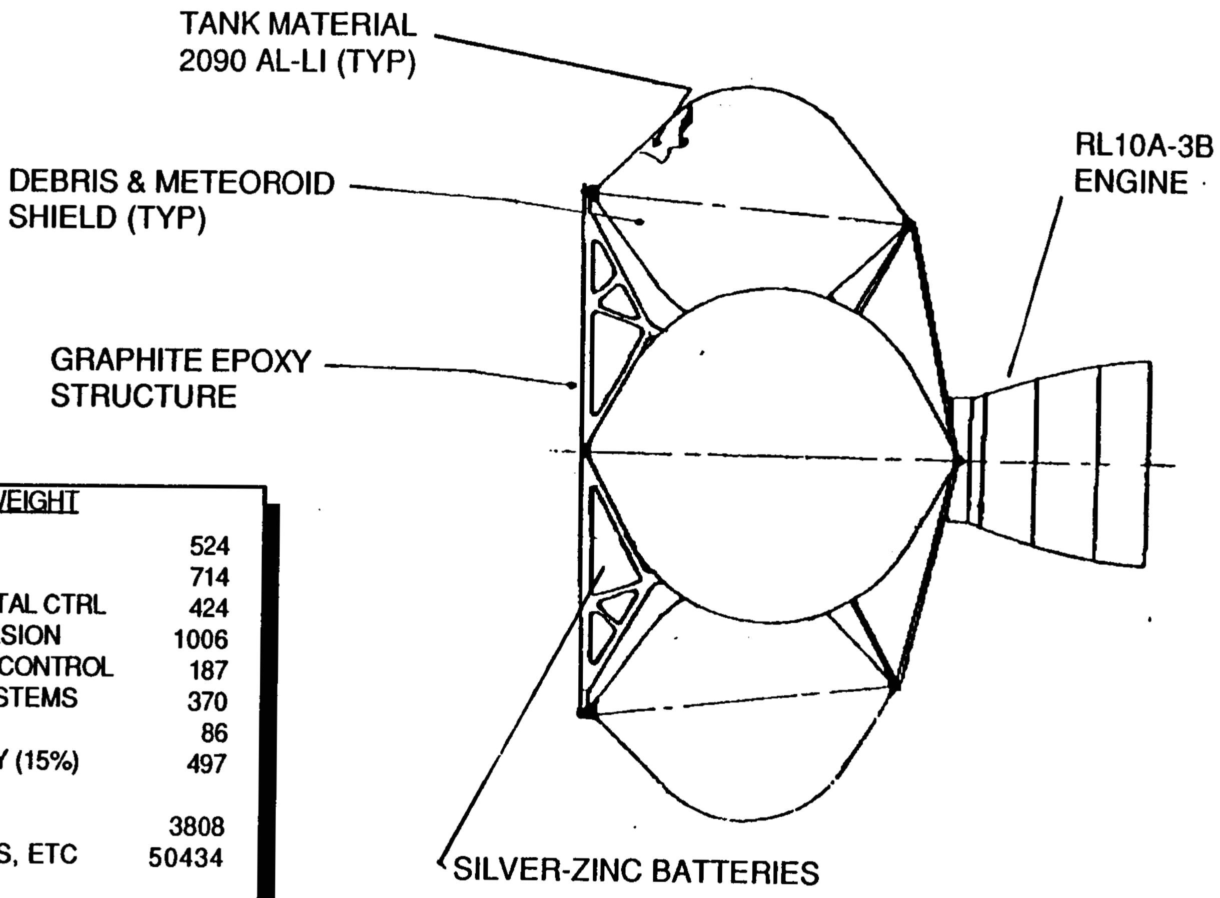
<u>ITEM</u>

- REMOVE AEROBRAKE
- ALUMINUM VS. COMPOSITE STRUCTURE
- RL10A-3 VS. IOC ENGINE
- LESS METEOROID PROTECTION (THINNER BUMPER)
- BATTERIES INSTEAD OF FUEL CELLS
- GPS VS. GROUND UPDATE FOR STATE VECTOR
- 2219 AL VS. 2090 AL-LI FOR TANKS
- REDUCE WEIGHT IMPACT OF COMM. ANTENNA

NEAR TERM EXPENDABLE (PRELIMINARY)

The concept on the facing page illustrates a version of OTV that is possible to develop in the near term with relatively low risk. For example, the concept incorporates the RL10A-3 which is an existing engine now in production. This vehicle concept is also intended to be expendable and not manratable. Additional features include lightweight silver-zinc batteries rather than fuel cells or heavy rechargable batteries.

NEAR TERM EXPENDABLE OTV (PRELIMINARY)



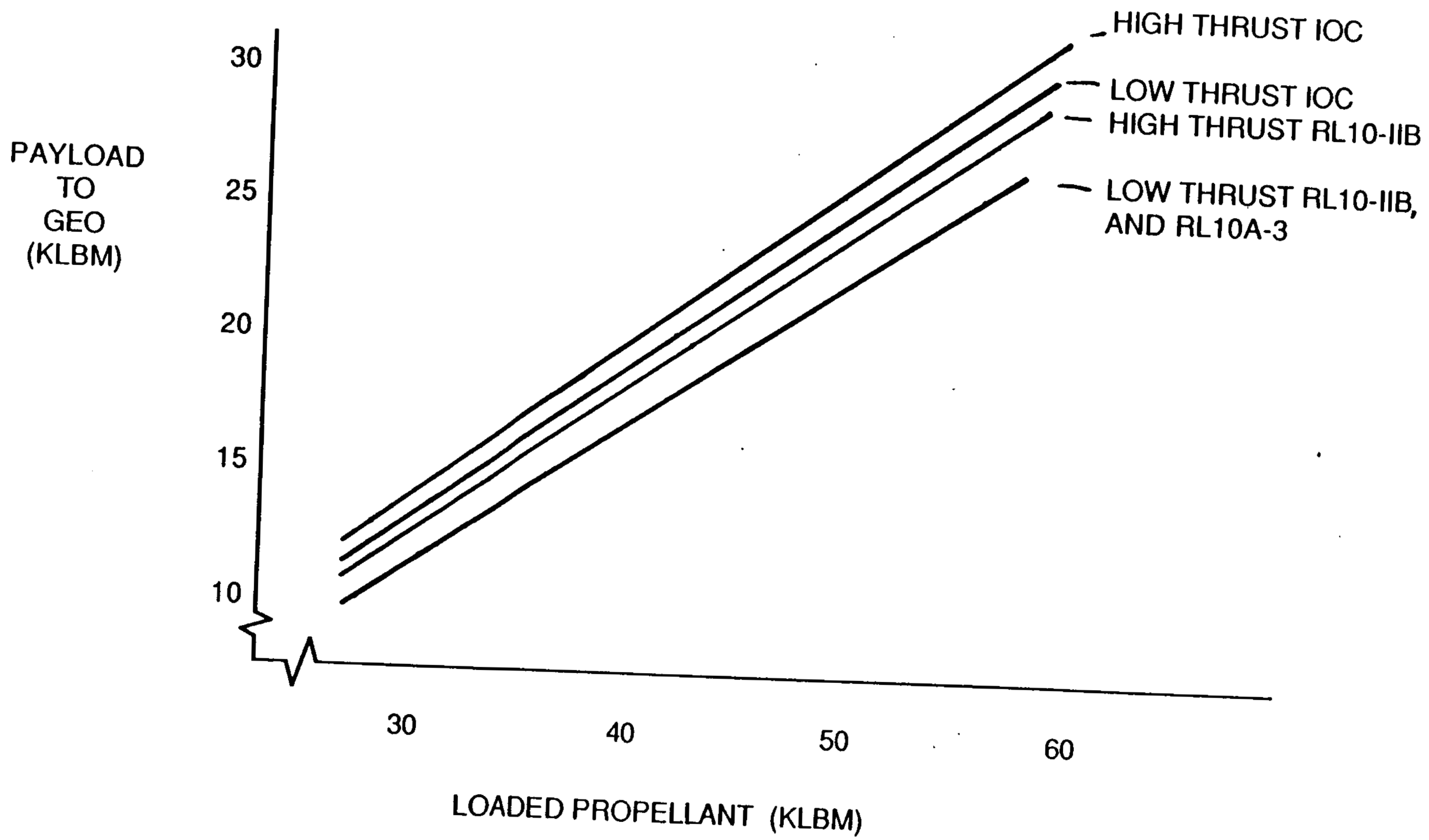
WEIGHT	
TANKS	524
STRUCTURE	714
ENVIRONMENTAL CTRL	424
MAIN PROPULSION	1006
ORIENTATION CONTROL	187
ELECTRIC SYSTEMS	370
G, N & C	86
CONTINGENCY (15%)	497
DRY WEIGHT	3808
PROPELLANTS, ETC	50434
LOADED WEIGHT	54242

EXPENDABLE OTV GEO DELIVERY CAPABILITY

The figure shows payload capability as a function of loaded propellant for an expendable ACC OTV. The cases presented are for GEO delivery. The importance of the figure is to illustrate the vehicle performance capability as the size of the OTV varies. Depending upon the STS lift capability, the ACC OTV propellant capacity may vary and thus result in a subsequent range of payload capability. This is important in choosing the proper size of the ground based vehicle.

Another important use of this parametric information is in deciding which main engine(s) may be appropriate for the OTV program. Performance data are shown here for the Advanced (IOC) engine (Isp of 475 sec), RL10-IIB (Isp of 460 sec), and the RL10A-3(Isp of 440 sec). The higher technology engines may enable certain missions to be flown from the ground and with STS because of limited STS capability. For instance, if the OTV propellant capacity was limited to 50,000 lbm, the IOC engine would be the only engine candidate that could enable an OTV to perform the 25,000 lbm to GEO (0.1g) mission. This type of data will be used in making recommendations for the OTV program.

EXPENDABLE OTV GEO DELIVERY CAPABILITY



INITIAL OTV DRIVER MISSIONS

This chart summarizes the performance and launch requirements for a near term expendable OTV. One mission each in two mission scenarios presents demanding requirements in the year 1996. The missions are: 1) the initial GEO platform in the Earth Initiative scenario and 2) the Mars Rover / Sample Return mission in the Unmanned Planetary scenario.

For the initial GEO platform mission, a propellant load of 59K lb is required to perform the low-g boost with the existing technology RL-10 main engine. If an advanced IOC engine is used instead, the required propellant can be reduced to 50k lb. These two vehicle options represent a cargo load of 94K and 85K lb respectively for a single-launch of a Large Cargo Vehicle (LCV), assuming an ASE weight of 6000 lb. For the Shuttle, the mission must be flown in two flights and must use the IOC engine to get within Shuttle lift constraints.

For the Mars Rover / Sample Return mission, an expendable OTV with an RL-10 engine requires 40K lb of fuel while an IOC engine requires 37K lb. These options yield LCV cargo loads of 71K and 68K lb respectively for a single launch. For the Shuttle, the mission must still be flown in two flights but the option of using either the RL-10 or IOC engine exists.

Thus these two driver missions indicate large, though not insurmountable, problems for the Shuttle system because of the multiple launches involved. An LCV system with a capacity of 94K lb to 140 nmi park orbit would be desireable to enable initial use of the RL-10.

INITIAL OTV DRIVER MISSIONS

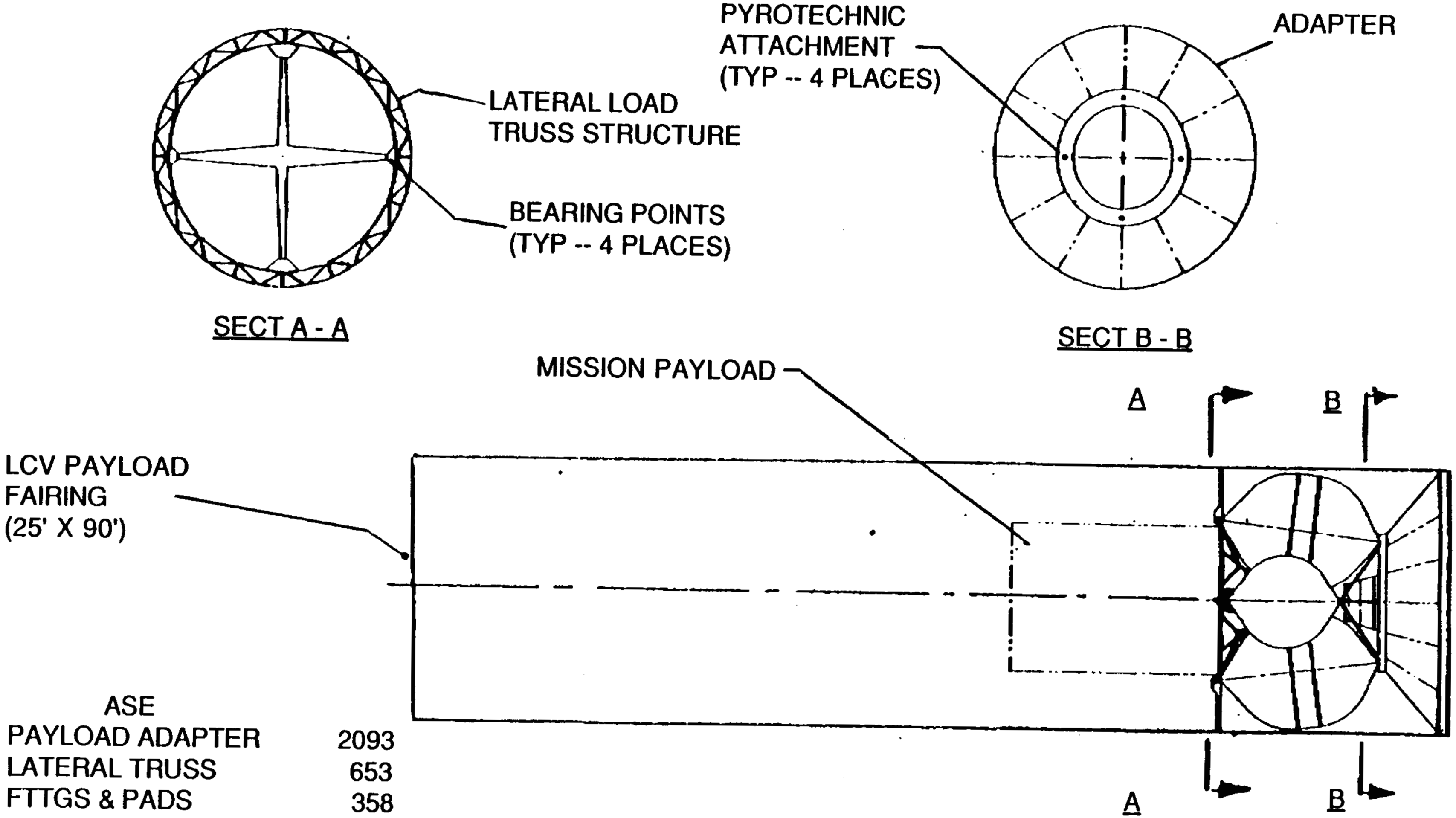
	EARTH INITIATIVE	UNMANNED PLANETARY
	INITIAL GEO PLATFORM 25 K TO GEO (0.1 G)	MARS ROVER SAMPLE RETURN 21 K, C3 = 10
MISSION SUMMARY FOR RL10A-3	OTV PROPELLANT = 59 KLBM (50 KLBM WITH IOC)	OTV PROPELLANT = 40 KLBM (37 KLBM WITH IOC)
# FLIGHTS - LARGE CARGO VEHICLE	1 -- TOTAL LOAD = 94K (TOTAL LOAD = 85K WITH IOC)	1 -- (TOTAL LOAD = 71K) (TOTAL LOAD = 68K WITH IOC)
# FLIGHTS - STS	2 -- TOTAL LOAD > 55K	2 -- TOTAL LOAD > 55K

LCV ADAPTER CONCEPTS FOR ACC OTV

It would be highly desirable to have the capability to launch OTV on more than one booster, i.e. the STS and a large cargo vehicle (LCV) of some sort. This dual compatibility would improve the access to high orbits, etc. considerably in addition to providing a growth path for the OTV.

The figure shows a possible way of mounting the ACC OTV concept into a LCV. The launch loads are carried into the vehicle structure via an aft thrust cone adapter. The forward lateral loads are carried by the truss structure shown at the four bearing points.

LCV ADAPTER CONCEPTS FOR ACC OTV



ASE	
PAYLOAD ADAPTER	2093
LATERAL TRUSS	653
FTTGS & PADS	358
TOTAL	3104

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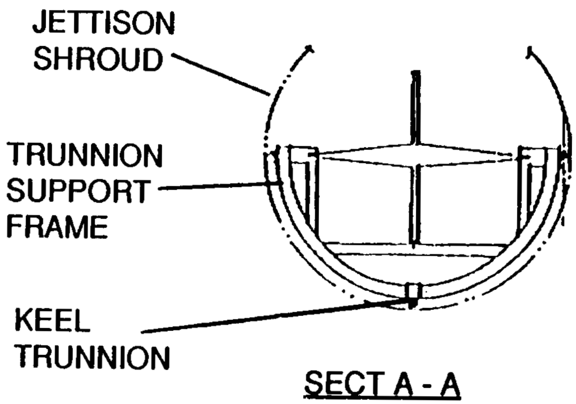
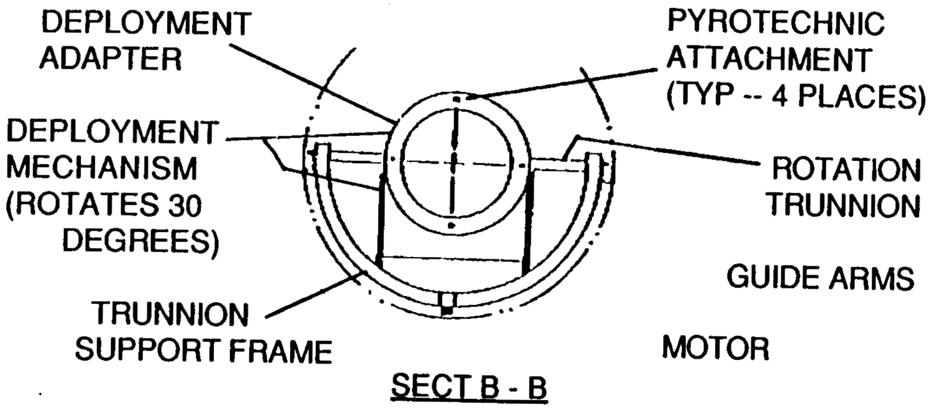
SDV ADAPTER CONCEPT FOR ACC OTV

The figure shows a concept for deployment of an OTV from a Shuttle Derived Vehicle (SDV). The sequence is to rotate the stage and payload from the SDV payload bay much the same way a tilt table would deploy a spacecraft (e.g. IUS/TDRSS) from STS. The deployment adapter attaches to the aft end of the OTV and provides the interface for the OTV deployment. It is assumed that the top of the SDV cargo shroud is jettisoned to allow this type of deployment.

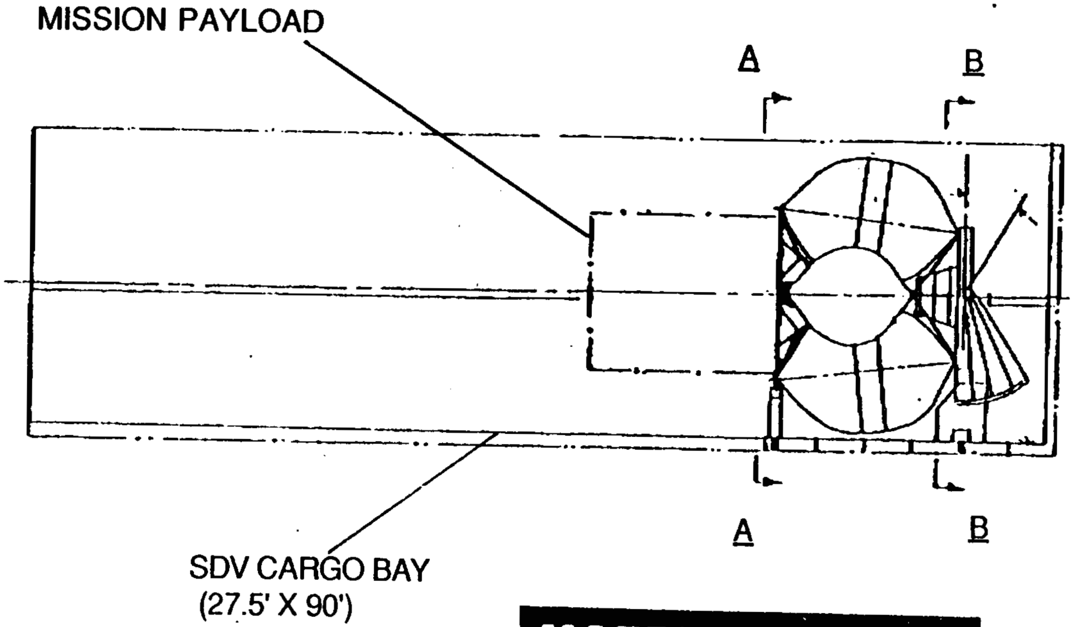
SDV ADAPTER CONCEPT FOR ACC OTV



SECT A - A



SECT B - B



ASE	
FWD FRAME	578
AFT FRAME	527
DEPLOY. ADAPTER	301
ROTATION TRUNNION	553
MOTOR & ARMS	315
SUBSYSTEMS	100
TOTAL	2374

LARGE CARGO VEHICLE ISSUES

This figure highlights a series of issues resulting from the emergence of a low cost heavy-lift booster. The reduction in cost and risk will shift many payloads from Shuttle to the LCV, with Shuttle performing only those missions that require manned presence. This will result in a drastic reduction in returning Shuttle flights with cargo bay space available which in turn will make it more difficult for a ground based OTV to be retrieved.

The net result is a shift in OTV operating mode to expendable or space based.

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

EXPENDABLE OR SPACE BASED OTV

SPACE BASE OPTIONS

This figure shows some of the options for OTV space basing. With the budgetary problems that Space Station has encountered it appears less likely that OTV can find a home on that facility. As was detailed in the Phase A and Extension #1 studies this first option is the most desireable from a program standpoint. The Space Station provides a stable base and power supply along with ready access by a servicing crew, if neccesary.

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

The third option, that of a decoupled free-flyer, has all the disadvantages of option 2 and few benefits. It does relieve the Station of any support role but manned access and propellant resupply become much more difficult.

Where such a de-coupled free-flyer might be attractive is in a minimum capability space base that is tended by the Shuttle, delineated in option #4. This would represent a low-cost approach to space basing. Only limited servicing would be possible, probably avionics changeout only. Propellant re-supply and tanking operations could be performed automaticly, removing any safety concerns.

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

SPACE BASE OPTIONS

1) SPACE STATION BASE

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

2) FREE FLYER BASE - COORBITAL WITH SPACE STATION

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

3) FREE FLYER DECOUPLED FROM SPACE STATION

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

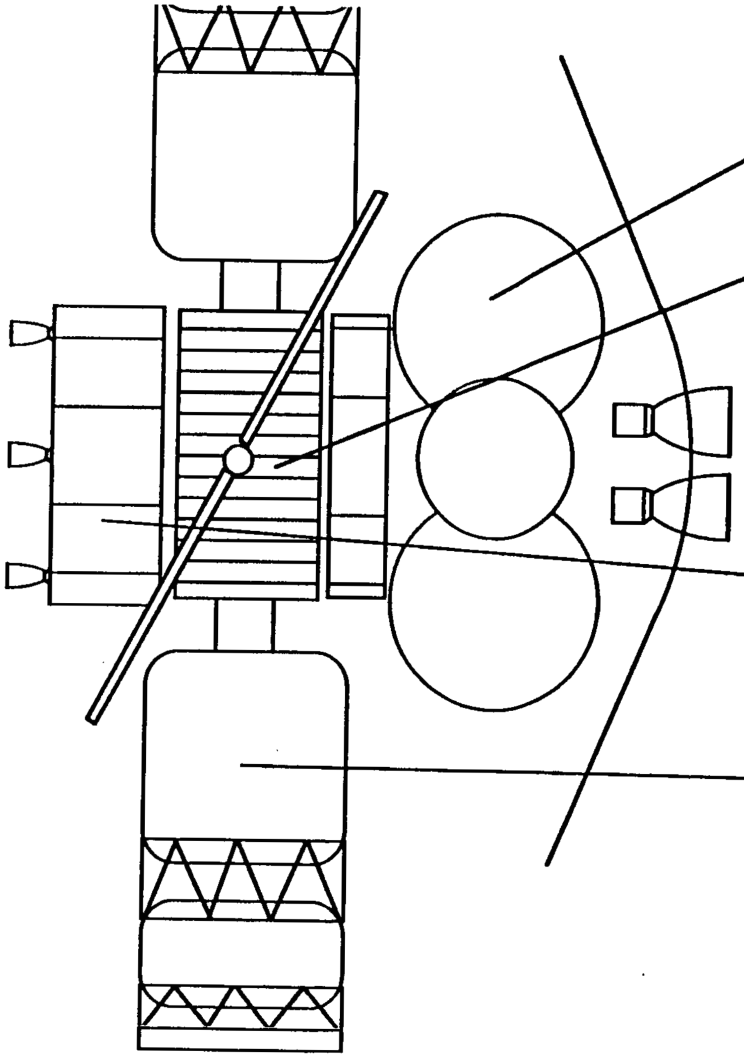
4) SPACE TENDED MINIMAL BASE

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

MINIMUM SPACE BASE

This figure shows a possible concept for a minimum space base for the OTV, as mentioned on the previous page. The only servicing functions performed would be avionics changeout by the Shuttle. The attached OMV would supply rendezvous and docking capability for joining the major modules together. Propellant resupply tanks injected by the LCV would be retrieved by the OMV and docked to the central stable core where automated refueling of the OTV would be controlled. The stable core also provides power (via solar arrays) and long-term attitude control via control moment gyro's (CMG). Because of the limited amount of servicing available, OTV's could not be re-used as often which could seriously limit the economic viability of the approach. However it may provide an attractive path for initial space based capability which grows to full-capability Space Station basing.

MINIMUM SPACE BASE



OTV

MINIMAL REPAIR CAPABILITY

STABLE BASE

PROVIDES LONG-TERM
ATTITUDE CONTROL (CMG'S)

DOCKING PORTS FOR OMV,
OTV & RESUPPLY TANKS

SOLAR ARRAYS FOR POWER

AUTOMATED PROPELLANT XFER

OMV

PROVIDES TRANSLATION
& DOCKING CAPABILITY

PROPELLANT RESUPPLY TANKS

ORBITAL INSERT BY LCV

RETRIEVAL BY OMV

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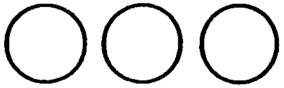
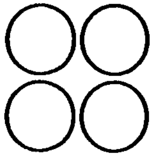
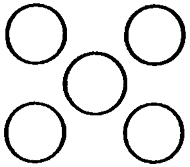
LUNAR MISSIONS

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 - 33.2KLBF	30:1	- HIGH THRUST REQUIRED - LARGE THROTTLING RATIO - WIDE PATTERN
	.9864	0.8K - 16.6 KLBF	21:1	- SMALLEST PATTERN - GOOD RELIABILITY - GROWTH FROM TWO ENGINES
	.9797	0.64K - 11.1KLBF	17:1	- LOWEST RELIABILITY - LARGEST PATTERN - COMPLEX DESIGN AND CONTROL

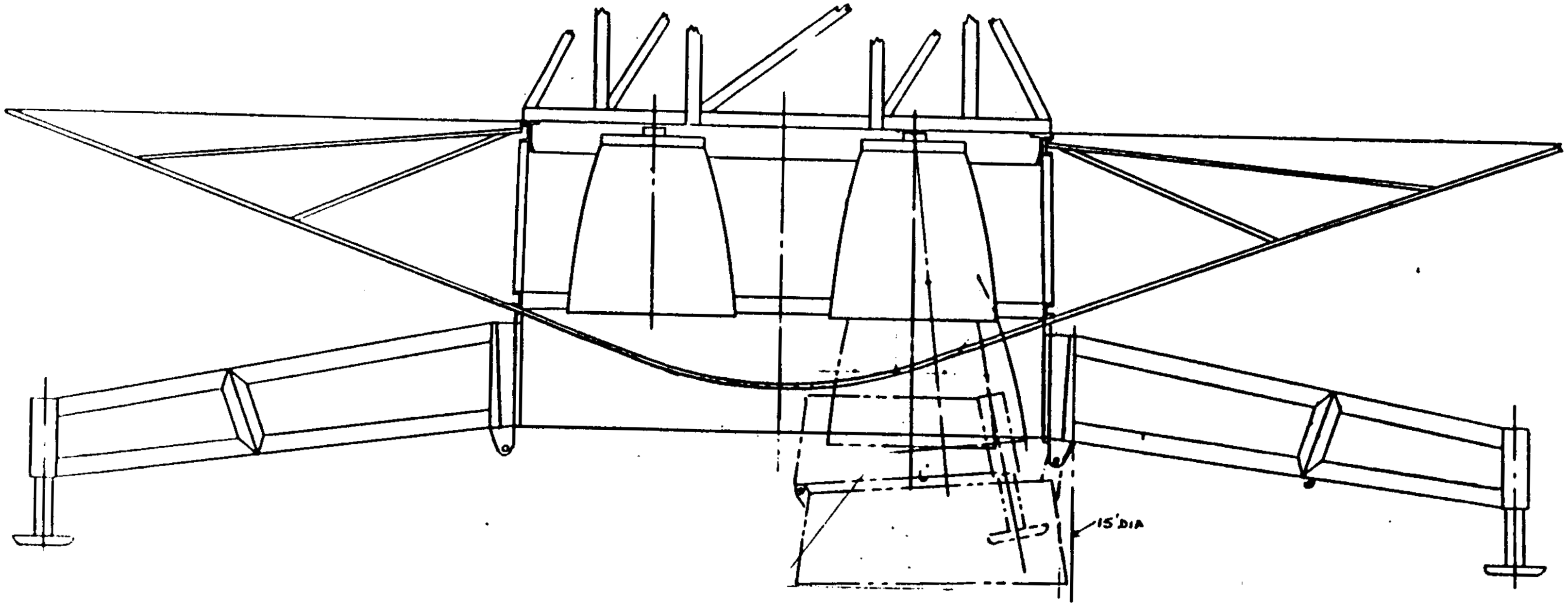
LUNAR LANDING LEGS

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



LUNAR TRANSFER OPTIONS

This figure shows a series of options for logistics to the surface of the Moon. Each mission option is shows an up and down scenario (designated by direction of arrows) along with payload capability (at the top of the page). Each mission is performed with the same 74K propellant capacity OTV to make the results comparable.

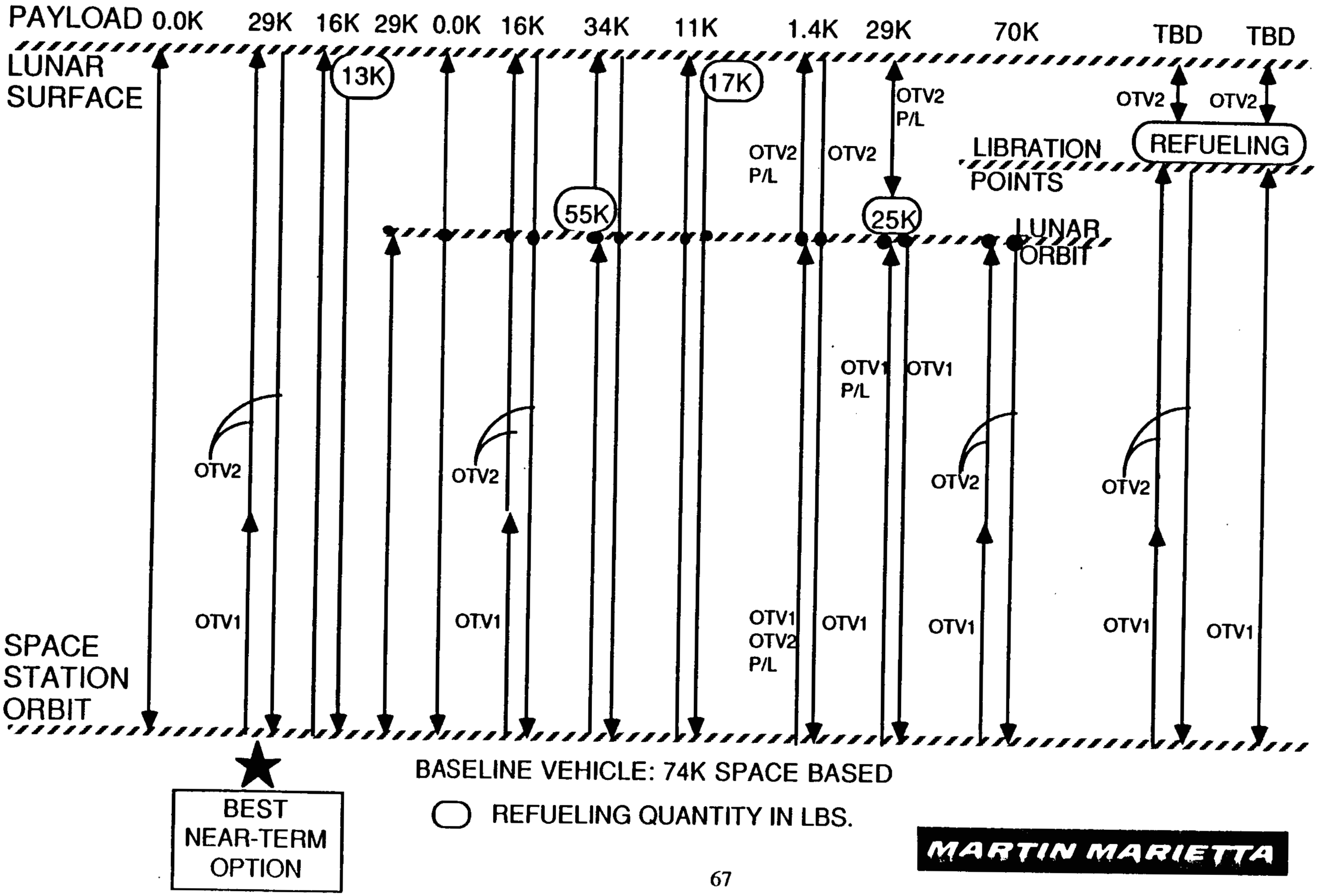
The three mission options shown at the left of the figure represent direct ascent options. In order they are: a single stage delivery mission (no payload capability), a two-stage delivery (29k capability), and a single stage lunar-surface refueling option (16k payload).

In the center of the figure are shown a series of lunar orbit missions (60 nm circular). For these mission modes a Lunar orbit support station is generally required. From the left they are: a single stage orbit delivery (29K to lunar orbit), a single stage surface delivery mission which first stops in lunar orbit (no payload capability, no return), a two stage delivery via orbit (19K to surface), single stage to the surface with orbital refueling (34K payload to surface), single stage with surface refueling via orbit (11K to surface), delivery of a specialized OTV lander to lunar orbit (1.4K bonus payload capability to surface), delivery of payload to lunar orbit by Earth-based OTV1 with lunar surface delivery by a Lunar-based OTV2 (29K to surface), and a two stage Lunar orbit delivery mission (70K to orbit).

Also shown are two Libration point missions which have not yet been assessed.

Of the direct ascent missions, the most attractive is the two stage delivery option as it has good performance without requiring Lunar support facilities. Of the Lunar orbit options the most attractive is the specialized lander (29K to Lunar surface) as it maximizes payload delivery for a minimum of orbital refueling.

LUNAR TRANSFER OPTIONS



BASELINE VEHICLE: 74K SPACE BASED

○ REFUELING QUANTITY IN LBS.

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ACC OTV SAFETY ISSUES

CENTAUR - STS SAFETY APPLICATION AREAS

This figure shows a listing of the most important STS/Centaur safety problem areas. This list was derived from NASA and General Dynamics data on fix-before-fly problems, safety reviews, Lewis/MMC's independent Centaur FMEA analysis and JSC's Centaur Lessons Learned.

The first four areas deal with in-flight dump capability. Problems were encountered with propellant dump dynamics disturbing the Shuttle's control stability and dump gasses being reingested into the Orbiter boat-tail. Inadequate dump redundancy (especially in valves) meant an inevitable growth in ASE weight. Another direct weight hit would be encountered if inadequate ASE helium reserves were augmented, thus adding more helium bottles. Propellant surge and waterhammer was encountered when high flow rate dump fluids encountered dead-ends in fill and drain lines.

Post-landing inerting was a severe problem for trans-Atlantic aborts since residual lox and hydrogen in the Centaur tanks must somehow be safed without a normal purge caravan.

Single point failures were identified in the presurization of the Centaur's lox/hydrogen common dome which could lead to dome failure and catastrophic propellant mixing. This concern is a Centaur-peculiar one which would not arise in vehicles with well-separated propellant tanks.

Other areas of concern include the following. Inadequate crew monitor and override capability of critical vehicle functions, inadequate control and/or containment of vented gasses, fire concerns with a vehicle insulation blanket which surrounds the Centaur tankage to prevent the formation of liquid air and/or the accumulation of hydrogen in the Shuttle cargo bay. The factors of safety used for qualifying the Centaur tankage was appropriate for primary structure and not for presure vessels. Deployment slosh uncertainty was a concern as it could cause re-contact between the Shuttle and Centaur.

The basic safety concerns with Centaur have been assessed for their impact on each of the two primary STS/OTV delivery modes: cargo bay and ACC. As may be seen, most of the safety problems for the cargo bay based Centaur would also be true for a cargo bay OTV.

The ACC boost location, however, would alleviate many of these concerns. By being located on the aft end of the External Tank, the OTV can be jettisoned in an abort and thus does not require propellant dump capability or post-landing inerting. Vent management is critical during boost phase only since the OTV flies away from the Shuttle at MECO. Vehicle insulation blankets are not neccesary since the ACC can be filled with a helium purge pre-launch.

CENTAUR - STS SAFETY APPLICATION AREAS

	CARGO BAY CONCERN	ACC OTV CONCERN
PROPELLANT DUMP DYNAMICS	YES	NO
PROPELLANT DUMP REDUNDANCY ISSUES	YES	NO
MAINTAINING SUFFICIENT HELIUM QUANITY	YES	NO
PROPELLANT LINE SURGE PROTECTION	YES	NO
POST-LANDING INERTING	YES	NO
VEHICLE COMMON DOME	*	*
CREW MONITOR & OVERRIDE CAPABILITY	YES	YES
VENT CONTROL AND CONTAINMENT	YES	NO (BOOST ONLY)
VEHICLE INSULATION BLANKET	YES	NO
TANKAGE FACTOR OF SAFETY	YES	YES
DEPLOYMENT SLOSH UNCERTAINTIES	YES	YES

* CENTAUR-SPECIFIC DESIGN CONCERN

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ACC OTV RETURN OPTIONS

1) RETURN ALL HARDWARE POSSIBLE

    JETTISON FABRIC PORTION OF AEROBRAKE (SUB-ORBITALLY)

    REMOVE LH2 TANKS AND HARDSHELL CORE AT SHUTTLE

    OPERATIONALLY COSTLY

2) EXPEND COMPLETE AEROBRAKE

    JETTISON ENTIRE AEROBRAKE (SUB-ORBITALLY)

    REMOVE LH2 TANKS AT SHUTTLE

3) EXPEND AEROBRAKE & LH2 TANKS

    JETTISON AEROBRAKE (SUB-ORBITAL)

    JETTISON LH2 TANKS INTO LOW ORBIT (1 DAY DECAY)

    STOW STRUCTURAL RACK (LO2 TANKS, AVIONICS & ENGINES) IN SHUTTLE

    NO DISASSEMBLY AT SHUTTLE

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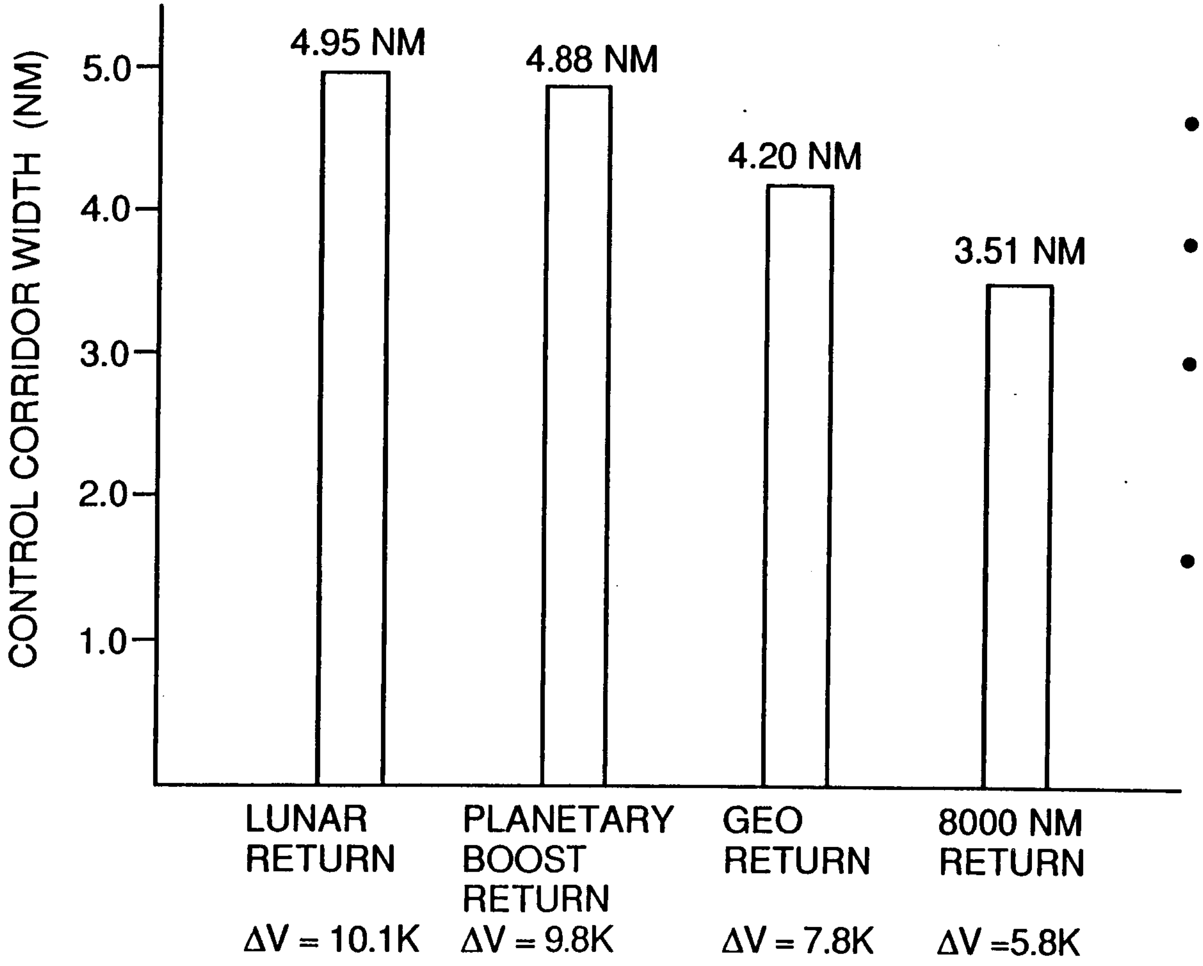
AEROASSIST

L/D EFFECTIVENESS

This figure shows the relative levels of control resulting from a fixed L/D for various entry velocities. As may be seen, the control corridor is a strong function of entry velocity and decays as velocity is reduced. These results impact the selection of L/D for new mission applications.

L/D EFFECTIVENESS

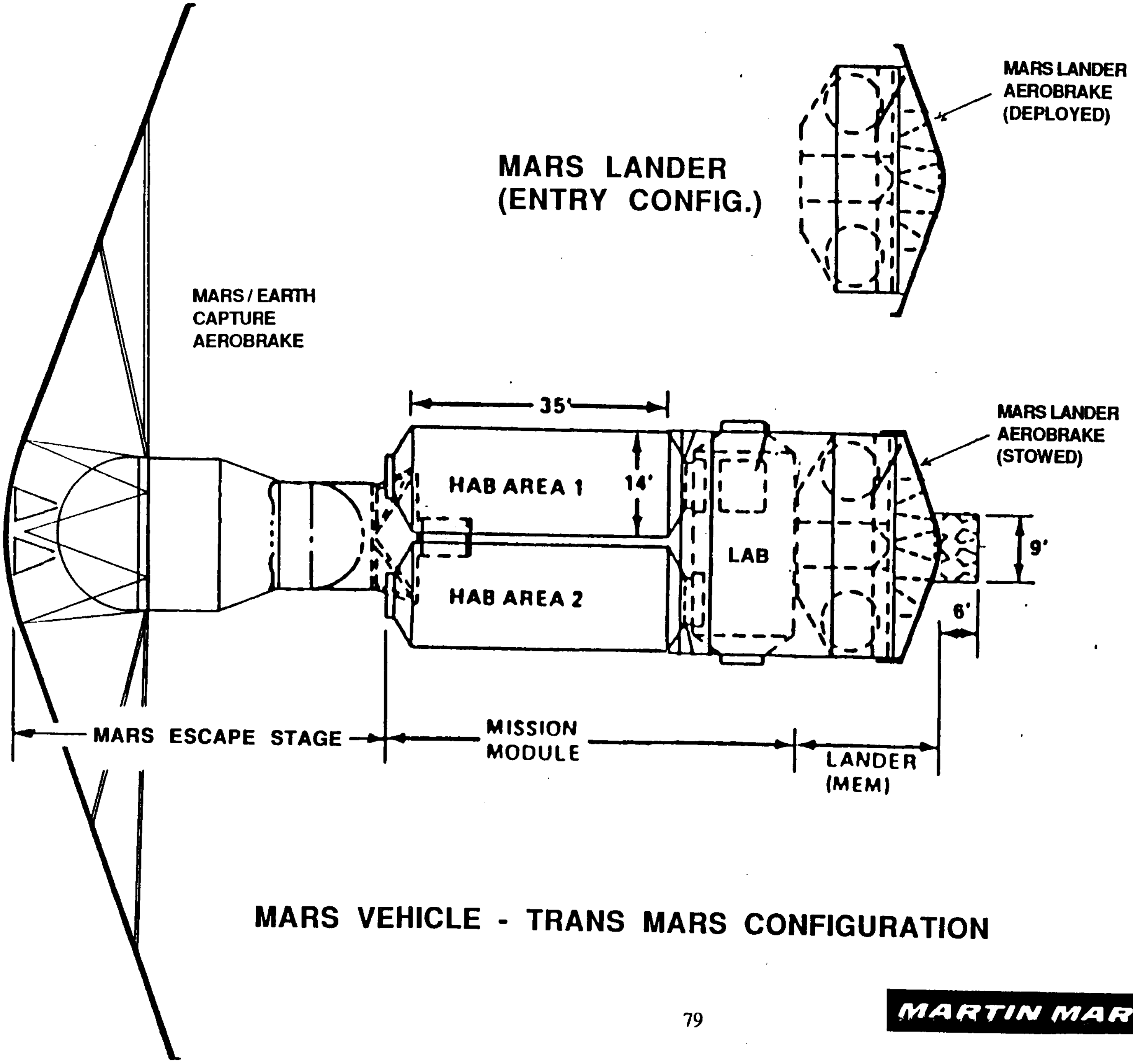


- AEROASSIST TO 245 NM
- L/D = 0.10
- CONTROL CORRIDOR WIDTH IS CREATED BY L/D
- L/D EFFECTIVENESS IS A FUNCTION OF THE AERO ΔV

MARTIN MARIETTA

MARS VEHICLE - TRANS MARS CONFIGURATION

The concept of a low L/D, ceramic fabric aerobrake developed in the OTV Phase A has been applied to a manned Mars mission. The resulting vehicle configuration is shown in this figure. The large Mars/Earth capture brake, which is permanently deployed, is shown on the left. The Mars excursion module is shown on the right in its trans-Mars configuration. The MEM's all-fabric brake is folded up in transit to prevent aerodynamic impingement during Mars aerocapture. Once in Mars orbit this brake is deployed in preparation for landing. Upon return to the Earth the entire remaining stack, consisting of the Mars escape stage, Lab/Hab modules, and the MEM stage 2 is aerocaptured by reusing the large brake.



MARS VEHICLE - TRANS MARS CONFIGURATION

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SUMMARY

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WHY OTV?

- HIGH PERFORMANCE, GROWTH-ORIENTED CRYO SYSTEM
  - DUAL COMPATABILITY WITH STS & LCV
  - LOW-TECH EXPENDABLE CAN DELIVER AT LEAST 13.5K TO GEO
  - ALTERNATIVE TO CENTAUR GIVES ASSURED ACCESS
- RE-USABLE SYSTEM
  - LOWER COSTS THROUGH RE-USE
  - PAYLOAD RETRIEVAL CAPABILITY
- AEROASSIST
  - LARGE INCLINATION TURNS
  - AERO-TESTBED VEHICLE

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WHY OTV?

- TECHNOLOGY ENHANCEMENTS
  - AEROASSIST
  - NEW HIGH PERFORMANCE SPACE CRYO ENGINE
  - EFFICIENT TURNAROUND TECHNIQUES (DRIVING TO SPACE BASING)
    - VEHICLE SELF-CHECKOUT & DIAGNOSTICS
    - SUBSYSTEM HEALTH MONITORING (ENGINES, MECHANISMS, ETC)
    - ROBOTIC CHANGEOUT OF HARDWARE (APPLICABLE TO GROUND BASE)

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OTV TODAY

- CURRENTLY AT CRITICAL BRANCH IN HEAVY LIFT TO ORBIT (>50K)

  STS OR LCV

- TIMELY AVAILABILITY WOULD ASSURE LCV MODE
- LCV DELAYS COULD LEAVE STS AS BEST OPTION
- ACC OTV COVERS BOTH BRANCHES
  - INITIAL FLIGHTS ON STS, 13-14K CAPABILITY
  - TRANSITION TO LCV WHEN IT BECOMES OPERATIONAL
  - MAINTAIN DUAL PATH TO SPACE (ASSURED ACCESS)
- INITIALLY EXPENDABLE MODE CAN GET PROGRAM OFF THE GROUND
  - REPLACE LOST STS / CENTAUR CAPABILITY
  - INCREMENTAL GROWTH AS NEED AND BUDGET APPEAR

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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
  - HIGH PERFORMANCE VEHICLE (50K CRYO PROP, 0.92 M.F.)
- LAUNCH MODE
  - 4-TANK CONFIG OPTIMUM PACKAGE FOR STS OR LCV
  - TANK CHANGEOUT FOR GROWTH
  - 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
  - LUNAR LOGISTICS IS THE PRIMARY DRIVER
  - TWO 95KLB STAGES CAN DELIVER 40KLB TO LUNAR SURFACE

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PROGRAM / MISSION OPTIONS

DRIVER MISSIONS

This figure summarizes the current set of driver missions supplied by MSFC. The baseline scenario 1 is combined with any one of three alternate scenarios to build a driver mission set. This results in either a Earth, Unmanned Planetary, or Lunar Initiative Scenario. The payload and time-phasing requirements are as indicated.

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 + EARTH INITIATIVE

OPTION # 2: BASELINE SCENARIO 1 + UNMANNED PLANETARY

OPTION # 3: BASELINE SCENARIO 1 + LUNAR INITIATIVE

GEO SERVICING OPTIONS

This figure shows two different options for performing the GEO-servicing mission defined in the Earth Initiative Scenario. In the first option the OTV delivers and retrieves the OMV + servicer on the same flight. Because of the availability of the OTV to perform major orbital maneuvers, the OMV propellant load can be fairly small (500 lb is assumed to be adequate for proximity operations). This results in OTV delivery/retrieval requirements of 10.0klb up and 9.5klb down.

In the second option, the OMV is delivered by one OTV flight and retrieved by a different one. Because the OMV must perform all its own orbital maneuvers a full propellant load is required (7klb). This results in an OTV flight #1 delivery requirement of 16.5klb up and a flight #2 retrieval requirement of 9.5klb down.

GEO SERVICING OPTIONS

1) SINGLE MISSION

DELIVER & RETRIEVE OMV + SERVICER ON SAME OTV FLIGHT

ONE SATELLITE SERVICING LOCATION MINIMIZES OMV PROPELLANT

OTV DELIVERY REQMT: 2.5K LRU'S + 6.5K OMV + 1K SERVICER = 10K UP

OTV RETRIEVE REQMT: 2.5K LRU'S + 6.0K OMV (DRY)+ 1K SERVICER = 9.5K DOWN

2) SPLIT MISSION

DELIVER OMV ON OTV FLIGHT #1 AND RETRIEVE ON OTV FLIGHT #2

LONG DURATION & MULTIPLE SERVICING POSSIBLE

OTV FLIGHT #1 REQMT: 2.5K LRU'S + 13K OMV + 1K SERVICER = 16.5K UP

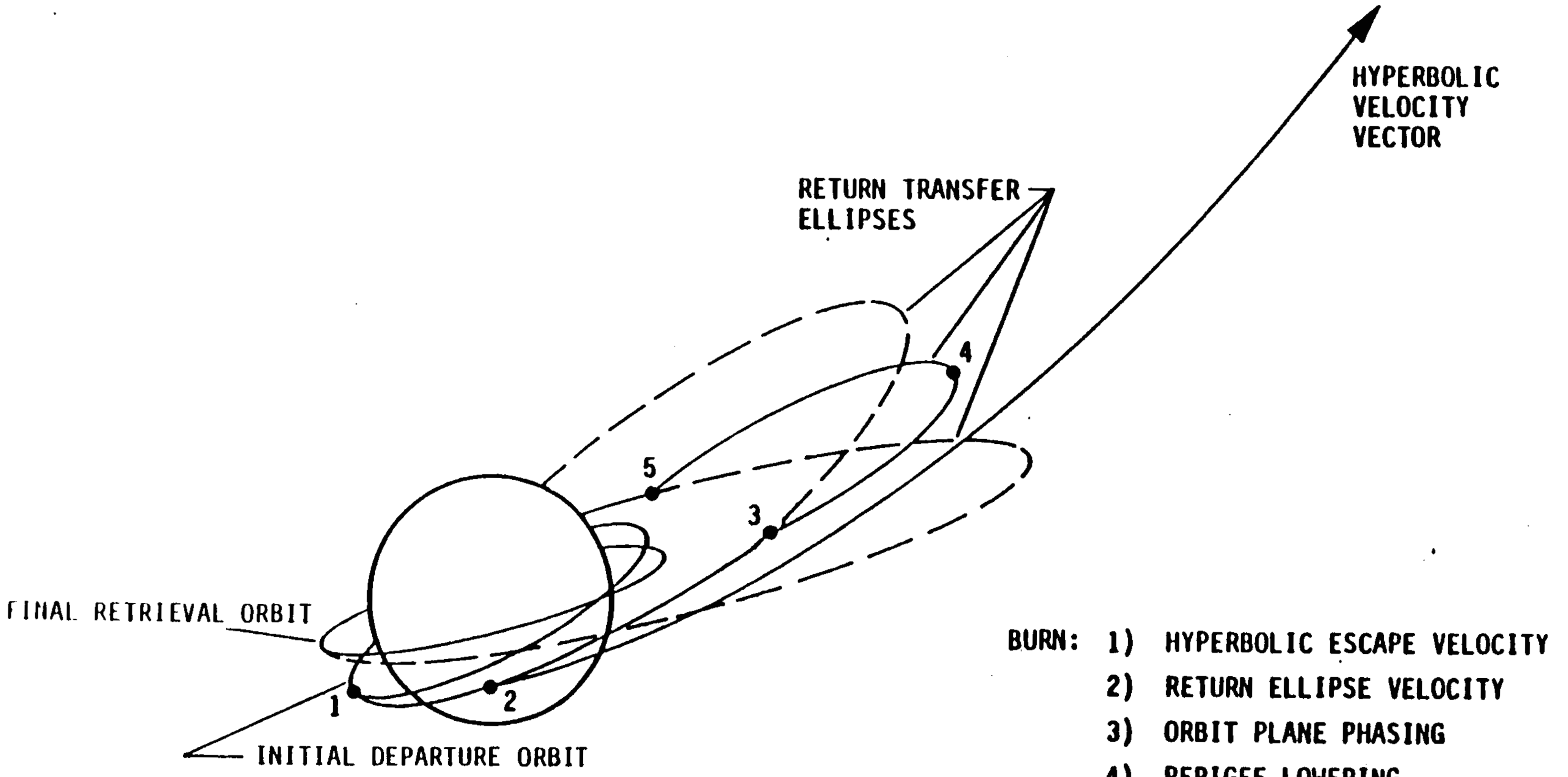
OTV FLIGHT #2 REQMT: 2.5K LRU'S + 6K OMV (DRY)+ 1K SERVICER = 9.5K DOWN

PLANETARY

Planetary missions flown by the OTV in a ground-based mode will begin from an orbit with an inclination and launch window optimized to achieve the desired hyperbolic velocity vector. After deployment and separation from the orbiter, the OTV main engine burn places the OTV and spacecraft on a hyperbolic escape trajectory. The OTV then separates from the spacecraft and increases the separation rate by performing a small RCS burn. About 15 minutes after separation, the OTV performs another burn to place itself in a highly elliptical orbit. Three additional burns are performed on this ellipse prior to the aerobraking maneuver. Two of these burns accomplish a plane change to align the OTV's trajectory with the orbiter rendezvous plane which is regressing relative to the OTV orbit. The third burn (actually the second burn in sequence) lowers the perigee of the ellipse to the appropriate altitude for the aerobraking maneuver. The plane change maneuvers are designed to minimize the energy required to transfer to the rendezvous plane.

The space-based planetary mission uses a strategy similar to the ground-based mission. The difference is in the launch window and orbital inclination of the OTV deployment orbit. Because the Space Station is in a 28.5 deg inclined orbit, the OTV launch window will be determined to minimize the total plane change required to achieve the desired velocity asymptote. As with the ground-based mission, the deorbit burns are designed to change planes and allow rendezvous in the Space Station orbit plane.

PLANETARY

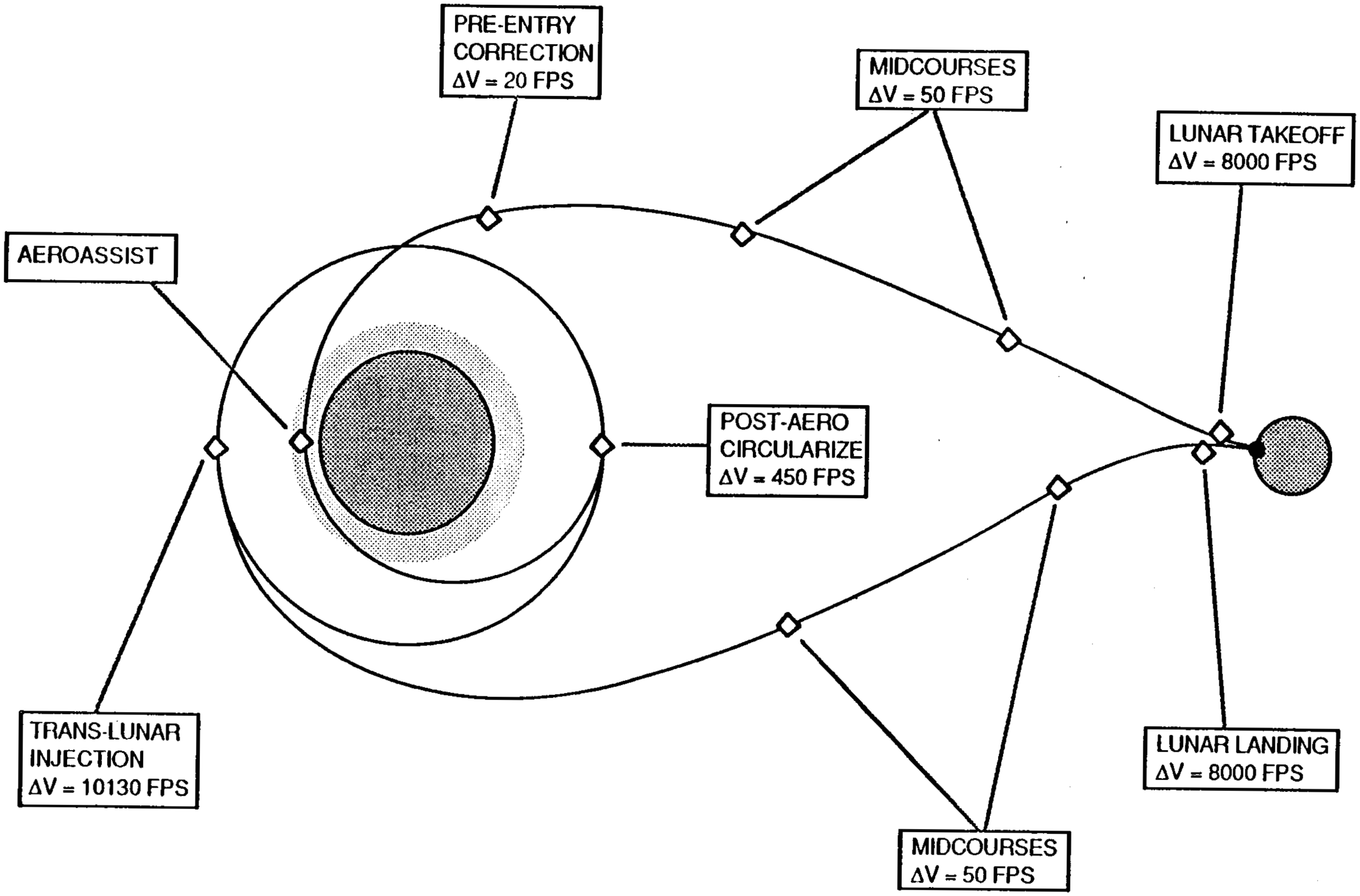


BURN:
1) HYPERBOLIC ESCAPE VELOCITY
2) RETURN ELLIPSE VELOCITY
3) ORBIT PLANE PHASING
4) PERIGEE LOWERING
5) ORBIT PLANE PHASING

LUNAR PROFILE - DIRECT ASCENT

Two 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. Velocities derived for this mission are shown and consist of Trans-Lunar Injection, Lunar Landing, Lunar Takeoff and several small midcourse burns. Velocities were derived from Surveyor data and two-body integrated trajectories.

LUNAR PROFILE - DIRECT ASCENT



PRE-ENTRY
CORRECTION
$\Delta V = 20$ FPS

MIDCOURSES
$\Delta V = 50$ FPS

LUNAR TAKEOFF
$\Delta V = 8000$ FPS

AEROASSIST

POST-AERO
CIRCULARIZE
$\Delta V = 450$ FPS

TRANS-LUNAR
INJECTION
$\Delta V = 10130$ FPS

LUNAR LANDING
$\Delta V = 8000$ FPS

MIDCOURSES
$\Delta V = 50$ FPS

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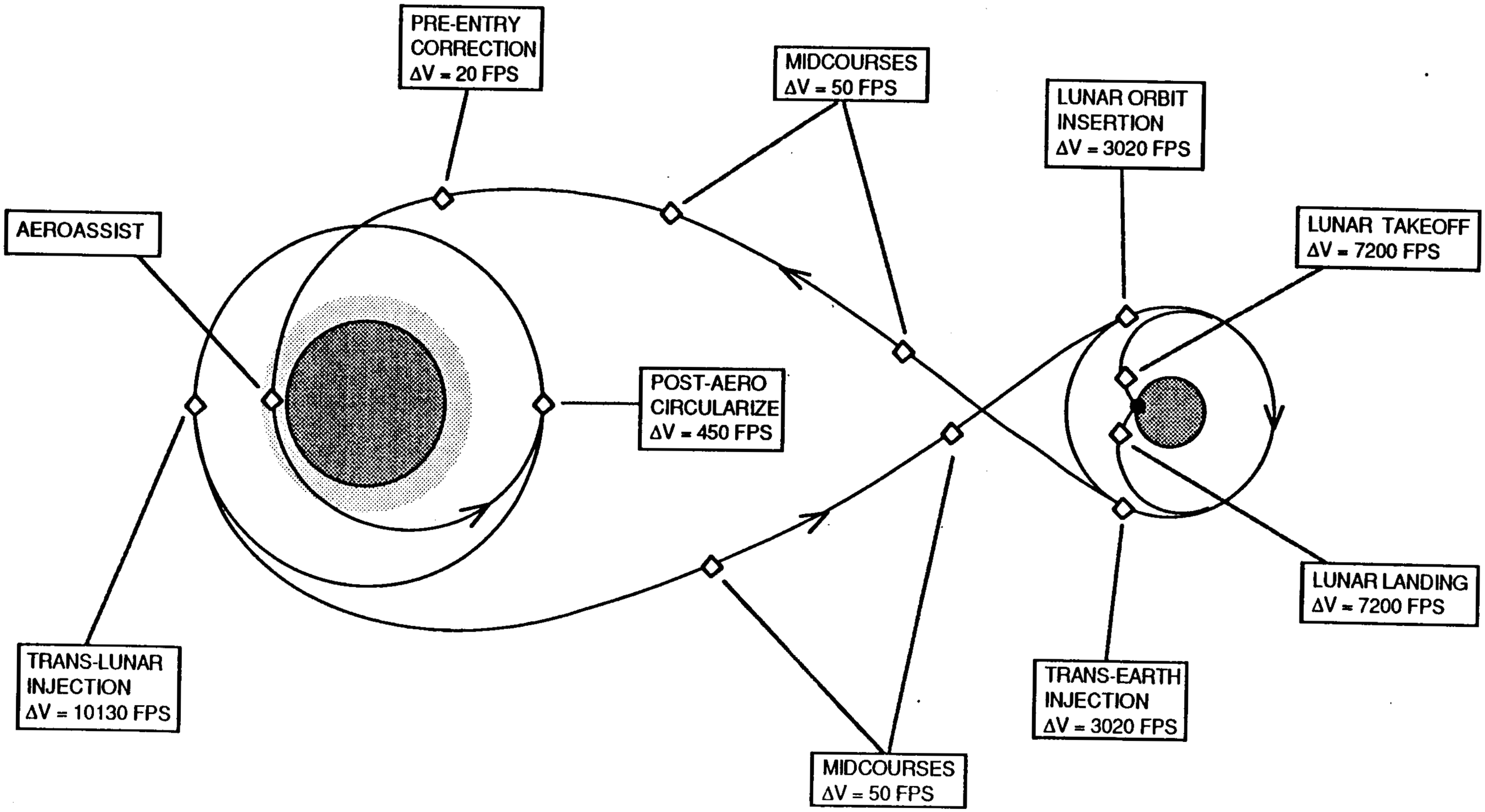
LUNAR PROFILE - LUNAR ORBIT

This Lunar profile uses an intermediate orbit about the Moon before descending to the surface. Velocities were derived from Apollo data and two-body integrated trajectories. These are Trans Lunar Injection, Lunar Orbit Insertion, Lunar Landing, Lunar Takeoff, and Trans Earth Injection. 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



PRE-ENTRY
CORRECTION
ΔV = 20 FPS

MIDCOURSES
ΔV = 50 FPS

LUNAR ORBIT
INSERTION
ΔV = 3020 FPS

AEROASSIST

LUNAR TAKEOFF
ΔV = 7200 FPS

POST-AERO
CIRCULARIZE
ΔV = 450 FPS

LUNAR LANDING
ΔV = 7200 FPS

TRANS-LUNAR
INJECTION
ΔV = 10130 FPS

TRANS-EARTH
INJECTION
ΔV = 3020 FPS

MIDCOURSES
ΔV = 50 FPS

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LUNAR TRANSFER OPTIONS

This figure shows a series of options for logistics to the surface of the Moon. Each mission option is shows an up and down scenario (designated by direction of arrows) along with payload capability (at the top of the page). Each mission is performed with the same 74K propellant capacity OTV to make the results comparable.

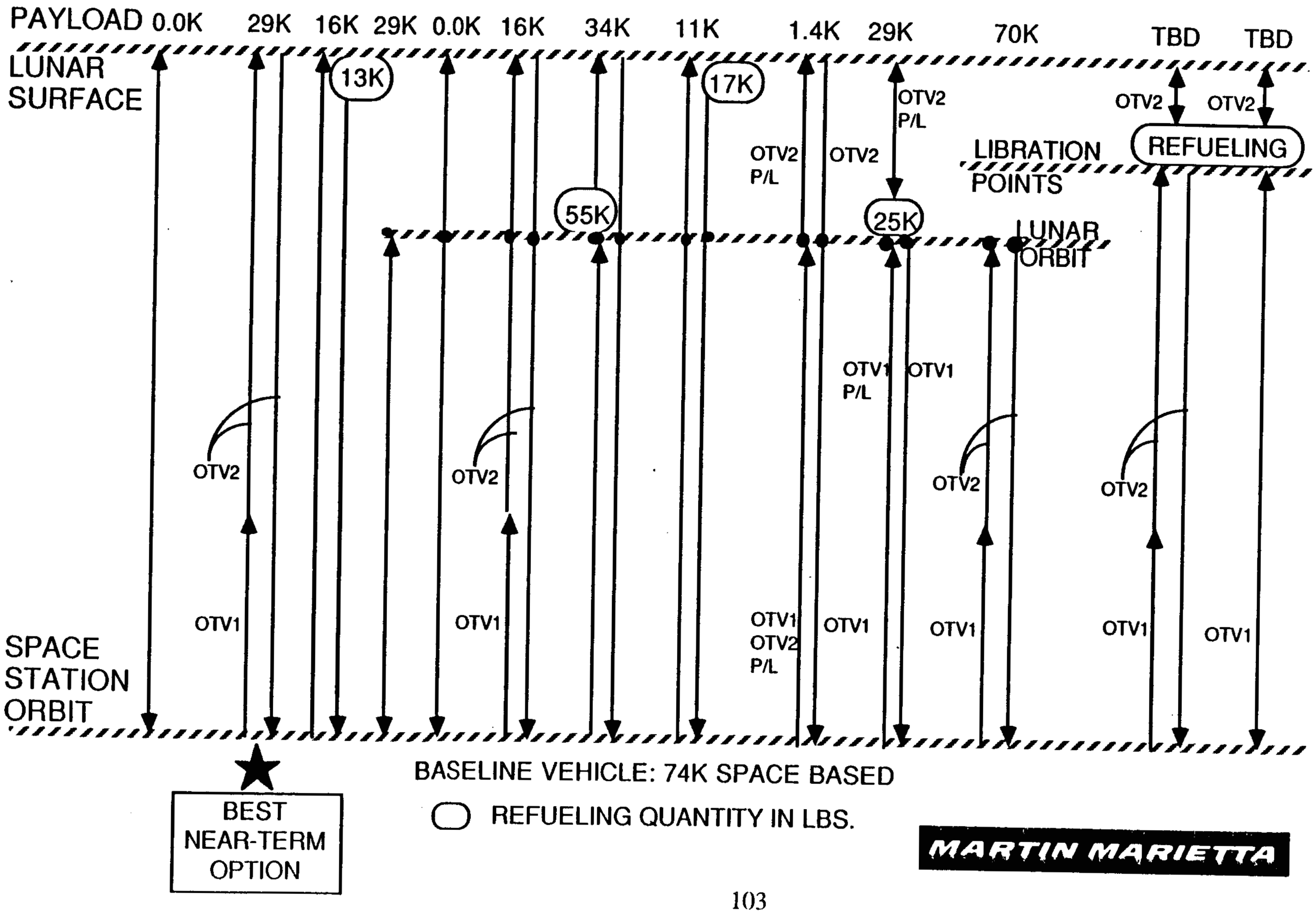
The three mission options shown at the left of the figure represent direct ascent options. In order they are: a single stage delivery mission (no payload capability), a two-stage delivery (29k capability), and a single stage lunar-surface refueling option (16k payload).

In the center of the figure are shown a series of lunar orbit missions (60 nm circular). For these mission modes a Lunar orbit support station is generally required. From the left they are: a single stage orbit delivery (29K to lunar orbit), a single stage surface delivery mission which first stops in lunar orbit (no payload capability, no return), a two stage delivery via orbit (19K to surface), single stage to the surface with orbital refueling (34K payload to surface), single stage with surface refueling via orbit (11K to surface), delivery of a specialized OTV lander to lunar orbit (1.4K bonus payload capability to surface), delivery of payload to lunar orbit by Earth-based OTV1 with lunar surface delivery by a Lunar-based OTV2 (29K to surface), and a two stage Lunar orbit delivery mission (70K to orbit).

Also shown are two Libration point missions which have not yet been assessed.

Of the direct ascent missions, the most attractive is the two stage delivery option as it has good performance without requiring Lunar support facilities. Of the Lunar orbit options the most attractive is the specialized lander (29K to Lunar surface) as it maximizes payload delivery for a minimum of orbital refueling.

LUNAR TRANSFER OPTIONS



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ACC OTV SAFETY

CENTAUR - STS SAFETY APPLICATION AREAS

This figure shows a listing of the most important STS/Centaur safety problem areas. This list was derived from NASA and General Dynamics data on fix-before-fly problems, safety reviews, Lewis/MMC's independent Centaur FMEA analysis and JSC's Centaur Lessons Learned.

The first four areas deal with in-flight dump capability. Problems were encountered with propellant dump dynamics disturbing the Shuttle's control stability and dump gasses being reingested into the Orbiter boat-tail. Inadequate dump redundancy (especially in valves) meant an inevitable growth in ASE weight. Another direct weight hit would be encountered if inadequate ASE helium reserves were augmented, thus adding more helium bottles. Propellant surge and waterhammer was encountered when high flow rate dump fluids encountered dead-ends in fill and drain lines.

Post-landing inerting was a severe problem for trans-Atlantic aborts since residual lox and hydrogen in the Centaur tanks must somehow be safed without a normal purge caravan.

Single point failures were identified in the presurization of the Centaur's lox/hydrogen common dome which could lead to dome failure and catastrophic propellant mixing. This concern is a Centaur-peculiar one which would not arise in vehicles with well-separated propellant tanks.

Other areas of concern include the following. Inadequate crew monitor and override capability of critical vehicle functions, inadequate control and/or containment of vented gasses, fire concerns with a vehicle insulation blanket which surrounds the Centaur tankage to prevent the formation of liquid air and/or the accumulation of hydrogen in the Shuttle cargo bay. The factors of safety used for qualifying the Centaur tankage was appropriate for primary structure and not for presure vessels. Deployment slosh uncertainty was a concern as it could cause re-contact between the Shuttle and Centaur.

CENTAUR - STS SAFETY APPLICATION AREAS

PROPELLANT DUMP DYNAMICS

PROPELLANT DUMP REDUNDANCY ISSUES

MAINTAINING SUFFICIENT HELIUM QUANITY

PROPELLANT LINE SURGE PROTECTION

POST-LANDING INERTING

VEHICLE COMMON DOME

CREW MONITOR & OVERRIDE CAPABILITY

VENT CONTROL AND CONTAINMENT

VEHICLE INSULATION BLANKET

TANKAGE FACTOR OF SAFETY

DEPLOYMENT SLOSH UNCERTAINTIES

* CENTAUR-SPECIFIC DESIGN CONCERN

CENTAUR - STS SAFETY APPLICATION AREAS (PART 2)

The basic safety concerns with Centaur identified in the previous chart have been assessed for their impact on each of the two primary STS/OTV delivery modes: cargo bay and ACC. As may be seen, most of the safety problems for the cargo bay based Centaur would also be true for a cargo bay OTV.

The ACC boost location, however, would alleviate many of these concerns. By being located on the aft end of the External Tank, the OTV can be jettisoned in an abort and thus does not require propellant dump capability or post-landing inerting. Vent management is critical during boost phase only since the OTV flies away from the Shuttle at MECO. Vehicle insulation blankets are not neccesary since the ACC can be filled with a helium purge pre-launch.

Overall, the ACC location for the OTV eliminates most of the safety concerns encountered by a cargo bay based vehicle.

CENTAUR - STS SAFETY APPLICATION AREAS

	CARGO BAY CONCERN	ACC OTV CONCERN
PROPELLANT DUMP DYNAMICS	YES	NO
PROPELLANT DUMP REDUNDANCY ISSUES	YES	NO
MAINTAINING SUFFICIENT HELIUM QUANITY	YES	NO
PROPELLANT LINE SURGE PROTECTION	YES	NO
POST-LANDING INERTING	YES	NO
VEHICLE COMMON DOME	*	*
CREW MONITOR & OVERRIDE CAPABILITY	YES	YES
VENT CONTROL AND CONTAINMENT	YES	NO (BOOST ONLY)
VEHICLE INSULATION BLANKET	YES	NO
TANKAGE FACTOR OF SAFETY	YES	YES
DEPLOYMENT SLOSH UNCERTAINTIES	YES	YES

* CENTAUR-SPECIFIC DESIGN CONCERN

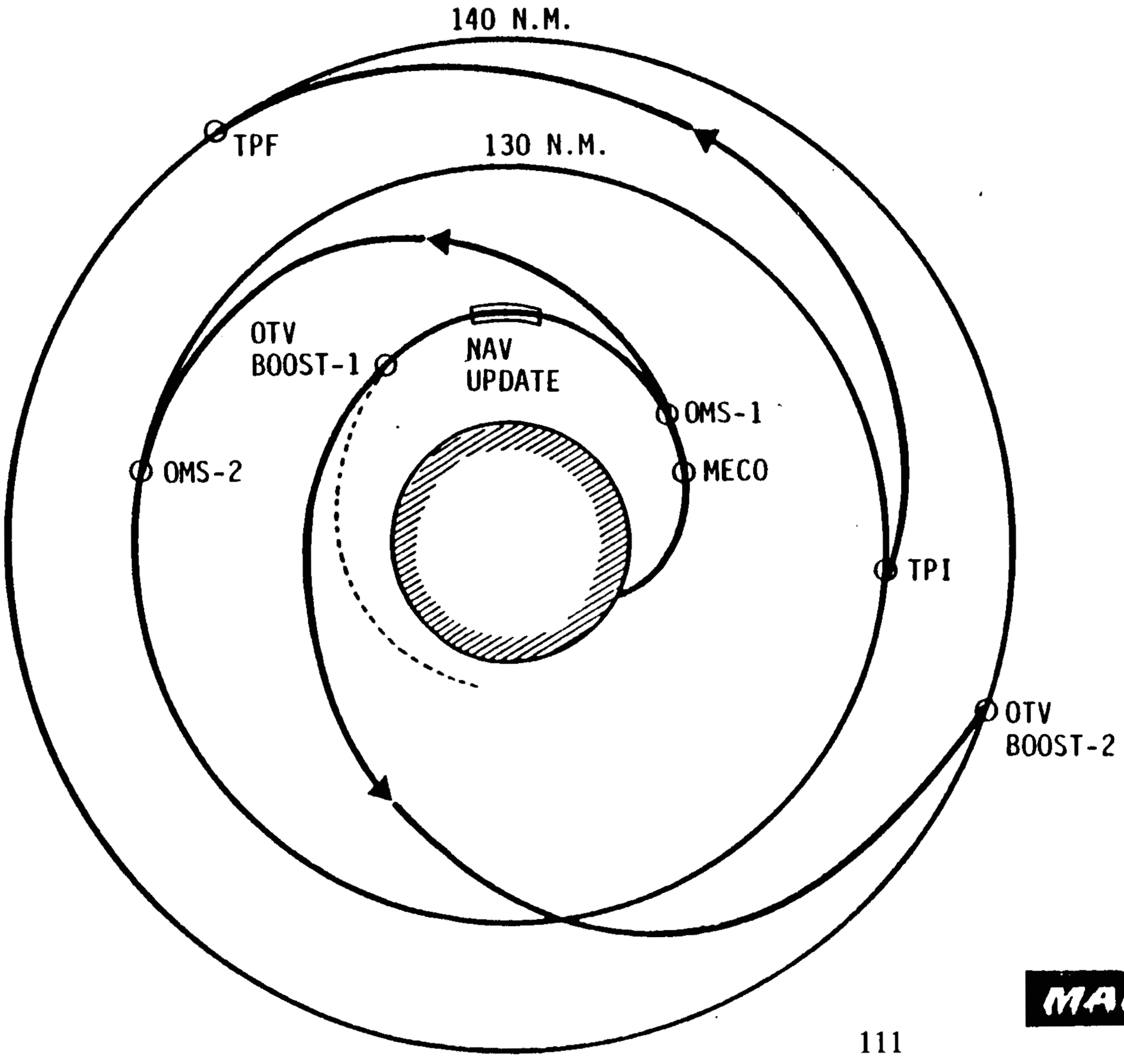
OTV / ORBITER TRAJECTORY PLOT

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

C-2



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

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ACC OTV RETURN OPTIONS

1) RETURN ALL HARDWARE POSSIBLE
    - JETTISON FABRIC PORTION OF AEROBRAKE (SUB-ORBITALLY)
    - REMOVE LH2 TANKS AND HARDSHELL CORE AT SHUTTLE
    - OPERATIONALLY COSTLY
2) EXPEND COMPLETE AEROBRAKE
    - JETTISON ENTIRE AEROBRAKE (SUB-ORBITALLY)
    - REMOVE LH2 TANKS AT SHUTTLE
3) EXPEND AEROBRAKE & LH2 TANKS
    - JETTISON AEROBRAKE (SUB-ORBITAL)
    - JETTISON LH2 TANKS INTO LOW ORBIT (1 DAY DECAY)
    - STOW STRUCTURAL RACK (LO2 TANKS, AVIONICS & ENGINES) IN SHUTTLE
    - NO DISASSEMBLY AT SHUTTLE

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 acting upon an unsupported structure with its engine doors open, though this cannot be verified without much more extensive analysis and test. 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 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 on 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)

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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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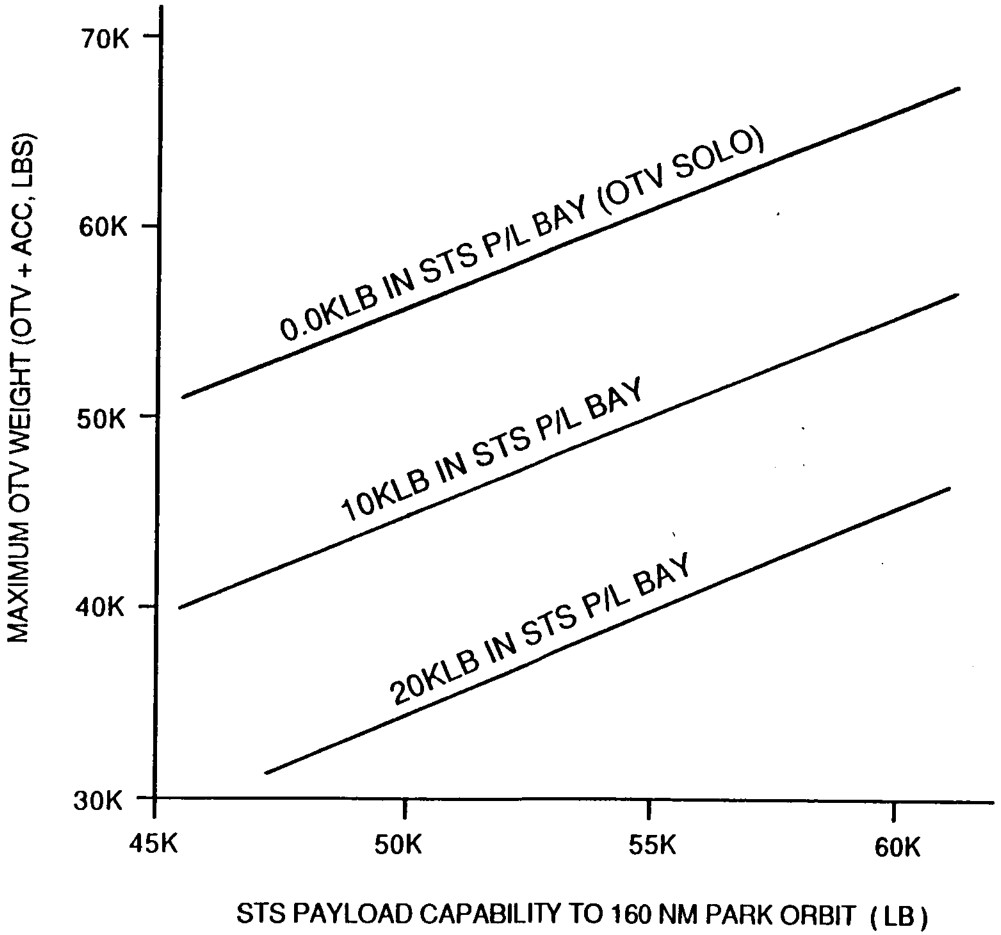
STS ACC PERFORMANCE

This figure summarizes the STS lift capability for an ACC OTV which performs its own orbital insertion starting at STS MECO. Because the OTV's engines are cryogenic they have an appreciably higher ISP than that of the Shuttle and so save boost mass over a standard Shuttle insertion. This translates to STS OMS propellant savings which increases the Shuttle lift capability. The savings is essentially doubled by the Shuttle groundrule that sufficient OMS propellant be carried to deorbit any payload bay carried payload in case of a failed deployment.

STS ACC PERFORMANCE



COMPARE STANDARD STS DELIVERY WITH ACC MODE

OTV SELF BOOST FROM MECO TO 140 NM PARK ORBIT

SHUTTLE ORBITER PERFORMS RENDEZVOUS AT 140 NM

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ACC PROGRAM COSTS

PROGRAM COSTS FOR THE DEDICATED ACC PAYLOAD CONTAINER

	NON-RECURRING	RECURRING
STANDARD DEDICATED ACC	$158M	$2.1M
42 INCH STRETCHED ACC	$165M	$2.2M

COSTS FROM THE 1985 MSFC/MMC AFT CARGO CARRIER STUDY

CONSTANT 1984 DOLLARS

64 FLIGHT BASIS

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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

DESIGN ISSUES

The design issues contained in this midterm review are listed in the facing chart. The first to be discussed is with regard to the differences that would result between a ground based expendable vehicle and a reusable vehicle. The next issues correspond to engine configuration tradeoffs for the ground based, space based, and lunar landing vehicle concepts. In addition, a lunar landing leg structural concept is shown as well as a concept for engine packaging and aerobrake door stowage.

The final issue is program evolution and recommendations for vehicle growth path and development.

DESIGN ISSUES

- GROUND BASED EXPENDABLE WEIGHTS
- TWO ENGINES IN THE ACC
- ENGINE CONFIGURATION RECOMMENDATIONS:
    - GROUND BASED EXPENDABLE AND REUSABLE
    - SPACE BASED REUSABLE
    - LUNAR LANDING
- LUNAR LANDING LEGS AND ENGINE PACKAGING
- PROGRAM EVOLUTION

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NEAR TERM EXPENDABLE OTV

DESIGN ISSUES FOR NEAR TERM EXPENDABLE

The table shows the items that would differ between a ground based reusable vehicle and an expendable predecessor. The obvious difference between a reusable aeroassisted vehicle and an expendable version is the aerobrake. The brake can be 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 also 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.

DESIGN ISSUES FOR NEAR TERM EXPENDABLE

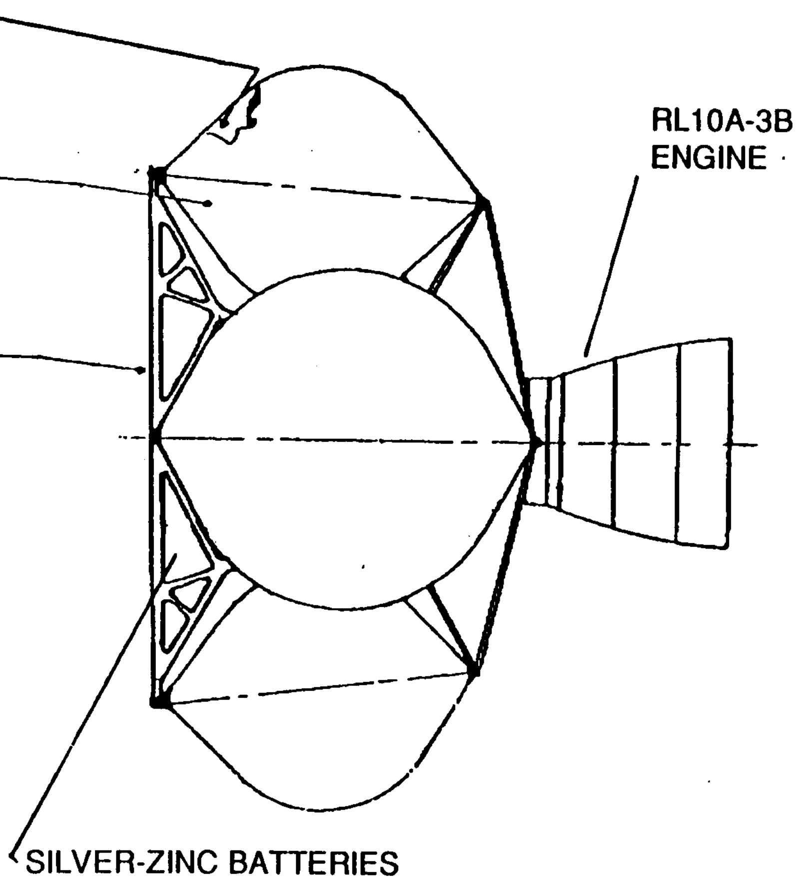
ITEM

- REMOVE AEROBRAKE
- ALUMINUM VS. COMPOSITE STRUCTURE
- RL10A-3 VS. IOC ENGINE
- LESS METEOROID PROTECTION (THINNER BUMPER)
- BATTERIES INSTEAD OF FUEL CELLS
- GPS VS. GROUND UPDATE FOR STATE VECTOR
- 2219 AL VS. 2090 AL-LI FOR TANKS
- REDUCE WEIGHT IMPACT OF COMM. ANTENNA

NEAR TERM EXPENDABLE (PRELIMINARY)

The concept on the facing page illustrates a version of OTV that is possible to develop in the near term with relatively low risk. For example, the concept incorporates the RL10A-3 which is an existing engine now in production. This vehicle concept is also intended to be expendable and not manratable. Additional features include lightweight silver-zinc batteries rather than fuel cells or heavy rechargable batteries.

NEAR TERM EXPENDABLE OTV (PRELIMINARY)



WEIGHT	
TANKS	524
STRUCTURE	714
ENVIRONMENTAL CTRL	424
MAIN PROPULSION	1006
ORIENTATION CONTROL	187
ELECTRIC SYSTEMS	370
G, N & C	86
CONTINGENCY (15%)	497
DRY WEIGHT	3808
PROPELLANTS, ETC	50434
LOADED WEIGHT	54242

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ENGINE CONFIGURATION TRADES

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TWO ENGINE ISSUE

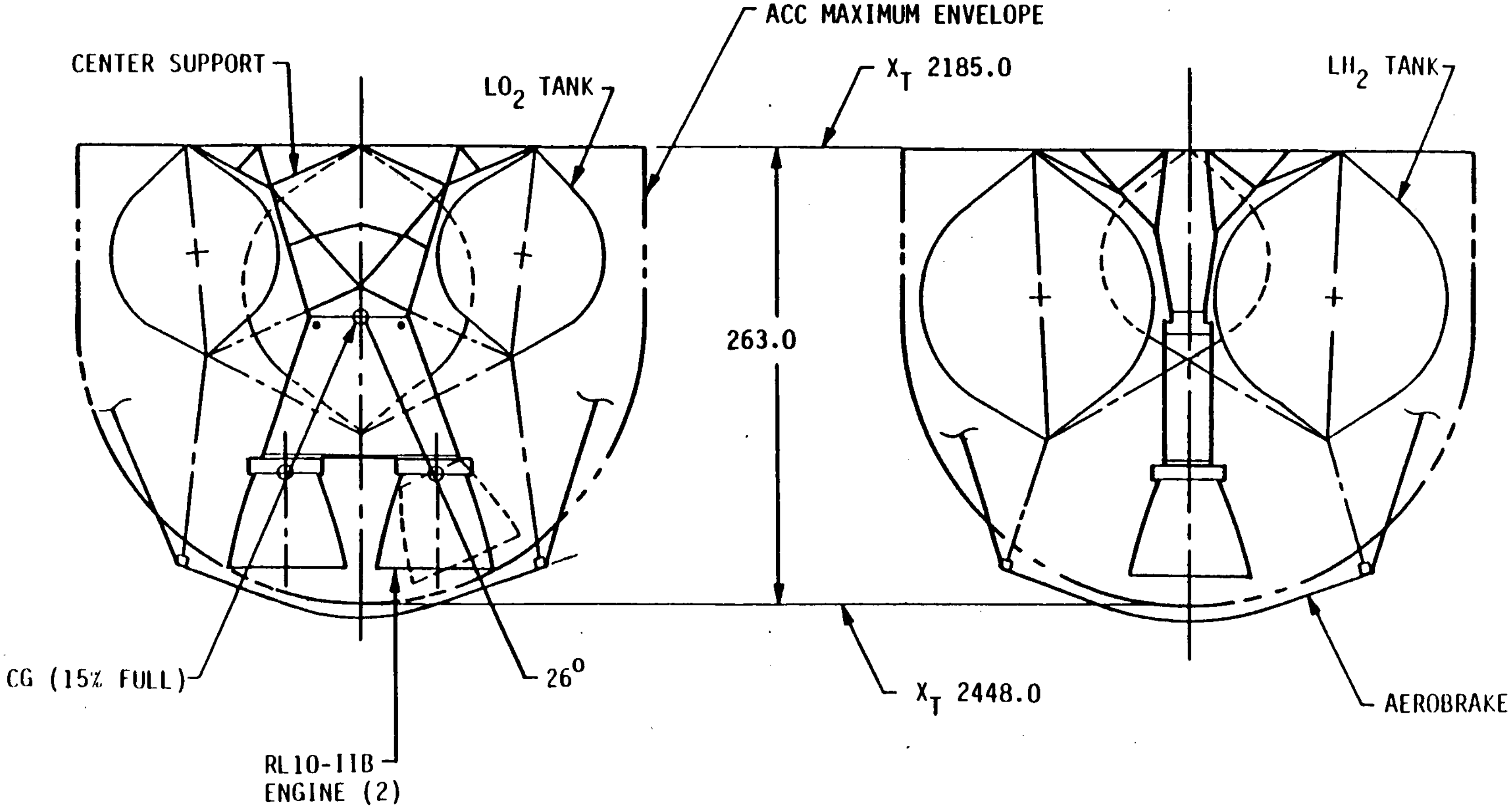
- PREVIOUS STUDIES SHOW LARGE IMPACT FOR 2-ENGINE REDUNDANCY
  - STRUCTURE IS LENGTH DRIVEN IN PROVIDING ENGINE-OUT CAPABILITY
  - TWO ENGINE OTV IS A PACKAGING PROBLEM IN ACC
- NEW CONCEPT OF A SHORT 2-ENGINE VEHICLE
  - WILL FIT INTO ACC
  - STILL PROVIDES ENGINE-OUT CAPABILITY WITH PAYLOAD ON FRONT END

2 ENGINE GB CRYO STUDY RESULTS

A study was performed to determine whether or not two RL10-IIB engines would fit with the OTV in the ACC. A constraint imposed upon the the study was that the vehicle be able to control itself with an engine out in a worst case cg condition; i.e. with no payload and with 15% fuel load. The other ground rules used during the study include 6" clearance between nozzles and 26 degree total outboard engine gimbal which includes a 10 degree outboard null setting.

The results of the two engine packaging study are shown on the facing figure. The two engine ACC OTV is shown with a spherical dedicated ACC and demonstrates that the vehicle with this particular configuration is not compatible with ACC launch. Therefore, additional insight is required in determining the applicability and/or proper configuration of two engines in a ground based vehicle.

2 ENGINE GB CRYO STUDY RESULTS



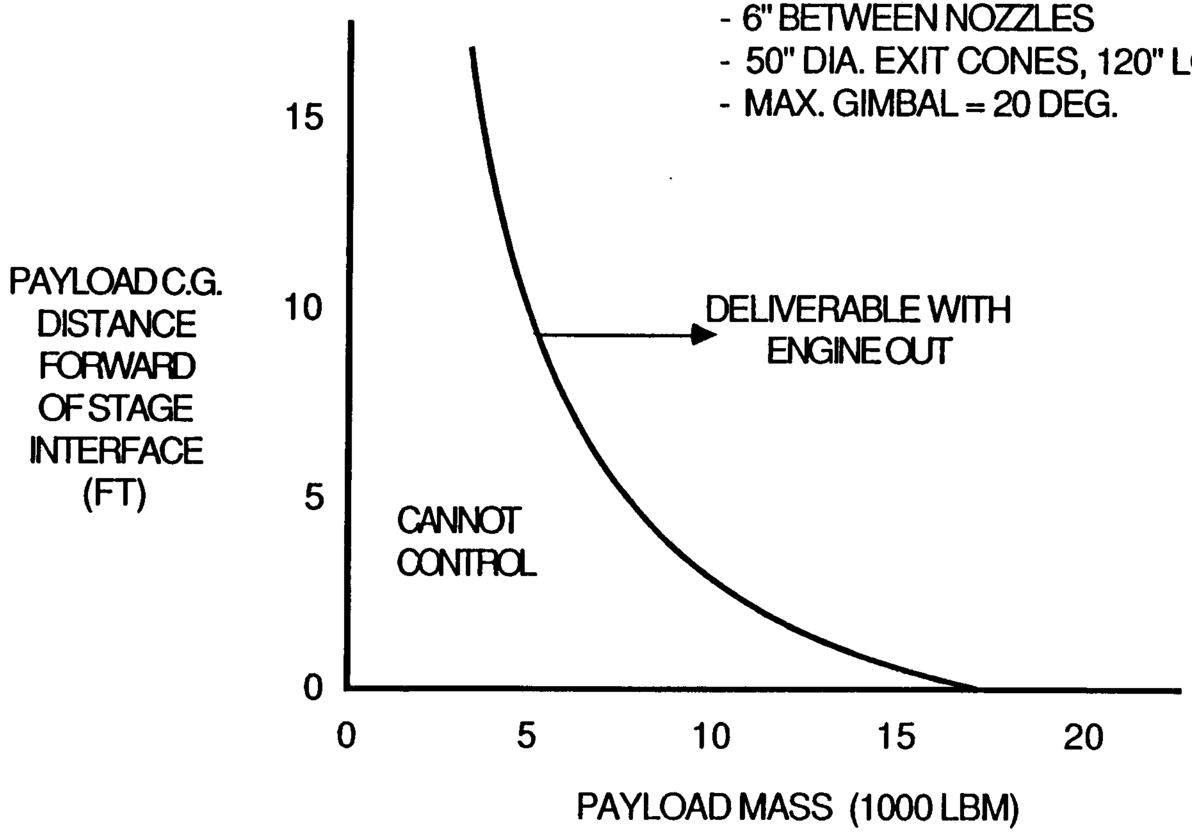
TWO-ENGINE ACC OTV

In order to package two engines into the ACC with the OTV, the vehicle must be made as short as possible. This may preclude the ability for the resulting vehicle to be able to control itself (without a payload on the front) with an engine out. However, in order to fully consider a two engine ground based configuration, it is important to understand the limitations of the squatty two engine concept.

The figure shows the payload size and weight combinations necessary for the short two-engine ACC configuration to be able to maintain control during an engine-out condition. For example, the longer or heavier a payload is, the more likely the stage will be able to maintain control. This is due to the cg being moved far enough forward of the engine gimbal points to allow vehicle thrust vector control within the bounds of engine gimbal angle. The short two-engine concept will not be able to maintain control in returning itself after a mission because of cg travel aft of the engines' gimbal control range.

TWO-ENGINE ACC OTV -- ENGINE OUT

- NO LENGTH IMPACT TO STAGE
- 6" BETWEEN NOZZLES
- 50" DIA. EXIT CONES, 120" LONG
- MAX. GIMBAL = 20 DEG.

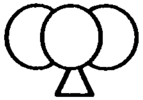
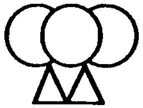
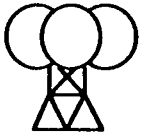
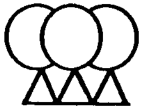


ENGINE CONFIGURATION TRADE DATA

The facing table shows comparison data which has been used in engine configuration trade studies for the engine configurations shown. The candidates include 1, 2, and 3 engine configurations. One of the 2 engine configurations (the stretched out version) has the ability to provide thrust vector control with engine out with no payload. The worst case cg for the 2 engine case (when the cg is furthest aft) is with about 15% propellant on-board. The best 3 engine configuration is the in-line rather than the cluster pattern because of the resulting shorter vehicle length.

Mission reliabilities are shown for the candidate configurations for both OTV return (four burns) and spacecraft delivery (three burns). Notice that the 2 engine candidates differ on OTV return reliability since the short version cannot return itself with one engine out and no payload. Also shown are OTV replacement cost estimates for both reusable and expendable OTV scenarios. For instance, the reusable OTV replacement costs are calculated by using the OTV mission reliability (four total burns) and the expendable OTV replacement costs are calculated by using the spacecraft delivery reliability (three total burns). $450M was assumed to be the cost of repeating a mission; e.g. STS launch plus spacecraft cost.

ENGINE CONFIGURATION TRADE DATA

CONFIGURATION	MISSION RELIABILITY*		100 MISSION LOSS COSTS		
	OTV	S/C	REUSABLE OTV'S ($40M EA)	MISSION REDO ($450M/LOSS)	EXPENDABLE OTV'S ($50M EA)
	0.9840	0.9880	$64M	$540M	$60M
	0.9683	0.9992	$127M	$36M	$4M
	0.9988	0.9992	$5M	$36M	$4M
	0.9978	0.9985	$9M	$67M	$8M

* BURN RELIABILITY = 0.996
NON-INDEPENDANT FAILURE RATE = 3%
4 BURNS-OTV RETURN, 3 BURNS-S/C DELIVERY

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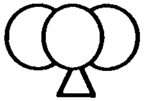
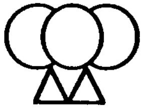
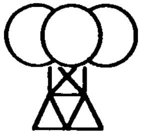
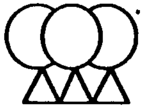
EXPENDABLE OTV COST COMPARISONS

Operational cost comparisons are shown in the facing table for expendable vehicle engine configurations. The first set of comparison totals is for equal launch costs; i.e. STS launch of stage and payload is assumed to be the same for all candidates. The second set of comparisons is for large cargo vehicle launch costs which are based upon both stage weight and length. The result is reduced costs by using a short 2 engine stage instead of a 1 engine stage. However, these cost comparisons are for 100 missions and must be weighed against the development cost differences between engine configuration candidates (not shown). The 3 engine case has reduced mission loss costs from the 1 engine case, however, the unit costs of the engines just about evens this effect out.

EXPENDABLE OTV COST COMPARISONS

CONFIGURATION	MISSION LOSSES: OTVS,REDO	ENGINE UNITS ($2M EA)	COMPARISON * TOTALS	LCV LAUNCH COST DELTAS	LCV COMPARISON DELTAS
	$600M	$204M	$804M	REF.	REF.
	$40M	$404M	$444M	+$64M	-$296M
	$40M	$404M	$444M	+$333M	-$27M
	$75M	$606M	$681M	+$128M	+$5M

* NO DIFFERENCES IN LAUNCH COSTS

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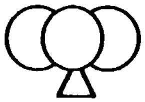
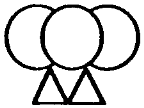
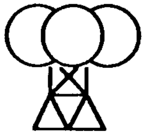
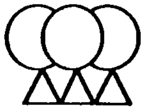
GROUND BASED REUSABLE OTV COST COMPARISONS

The table shows the operational cost comparisons of a reusable ground based OTV with the four engine configurations. The long 2 engine configuration will not package into the ACC and it is questionable that the 3 engine configuration would. The results show that the short 2 engine candidate has lower operational costs than the 1 engine configuration due to the reduced spacecraft replacement costs. It was assumed that engine life was 10 missions.

GROUND BASED REUSABLE OTV COST COMPARISONS

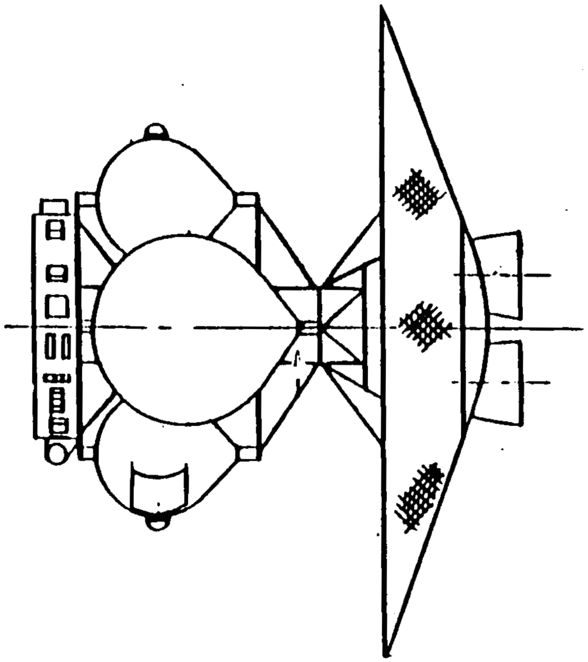
CONFIGURATION	ACC LAUNCH	MISSION LOSS OTV'S + REDO	ENG UNITS ($2M EA)	COMPARISON TOTALS	REMARKS
	OK	64 +540 =$604M	$24M	$628M	- SIMPLEST DESIGN - LOWEST NON-REC AND UNIT COSTS
	OK	127 + 36 =$163M	$56M	$219M	- REDUCED MISSION LOSSES - COMPACT DESIGN
	TOO LONG	5 + 36 =$41M	$44M	$85M	- CAN'T FLY IN ACC - N/A -- INTENDED FOR SPACE BASING
	?	9 + 67 =$76M	$66M	$142M	- COMPLEX CONTROL - LARGE AEROBRAKE DOORS AND ENGINE PACKAGING CONCERNS

OTV MAIN PROPULSION OPTIMIZATION

In optimizing a space based ("ultimate advanced") OTV, several aspects of the design have to be considered. A space based OTV can be relatively unconstrained by launch vehicle geometric envelope except for initial launch, perhaps. This is because the OTV does not necessarily have to be launched in one piece. Therefore, the OTV geometry and performance can be optimized somewhat separate from the traditional drivers (length and diameter) of a ground based vehicle.

The figure lists the key items to be considered in configuring a space based OTV. For instance, long engines (high expansion ratios) typically have improved Isp over short engines. However, a vehicle long enough to accommodate long engines will suffer penalties associated with additional structure and increased aerobrake diameter. Also, a single engine configuration usually provides the lightest weight propulsion system but cannot offer engine out capability. Therefore, with these things considered, an optimum configuration with engine out capability has been determined.

OTV MAIN PROPULSION OPTIMIZATION



OPTIMIZATION CONSIDERED:

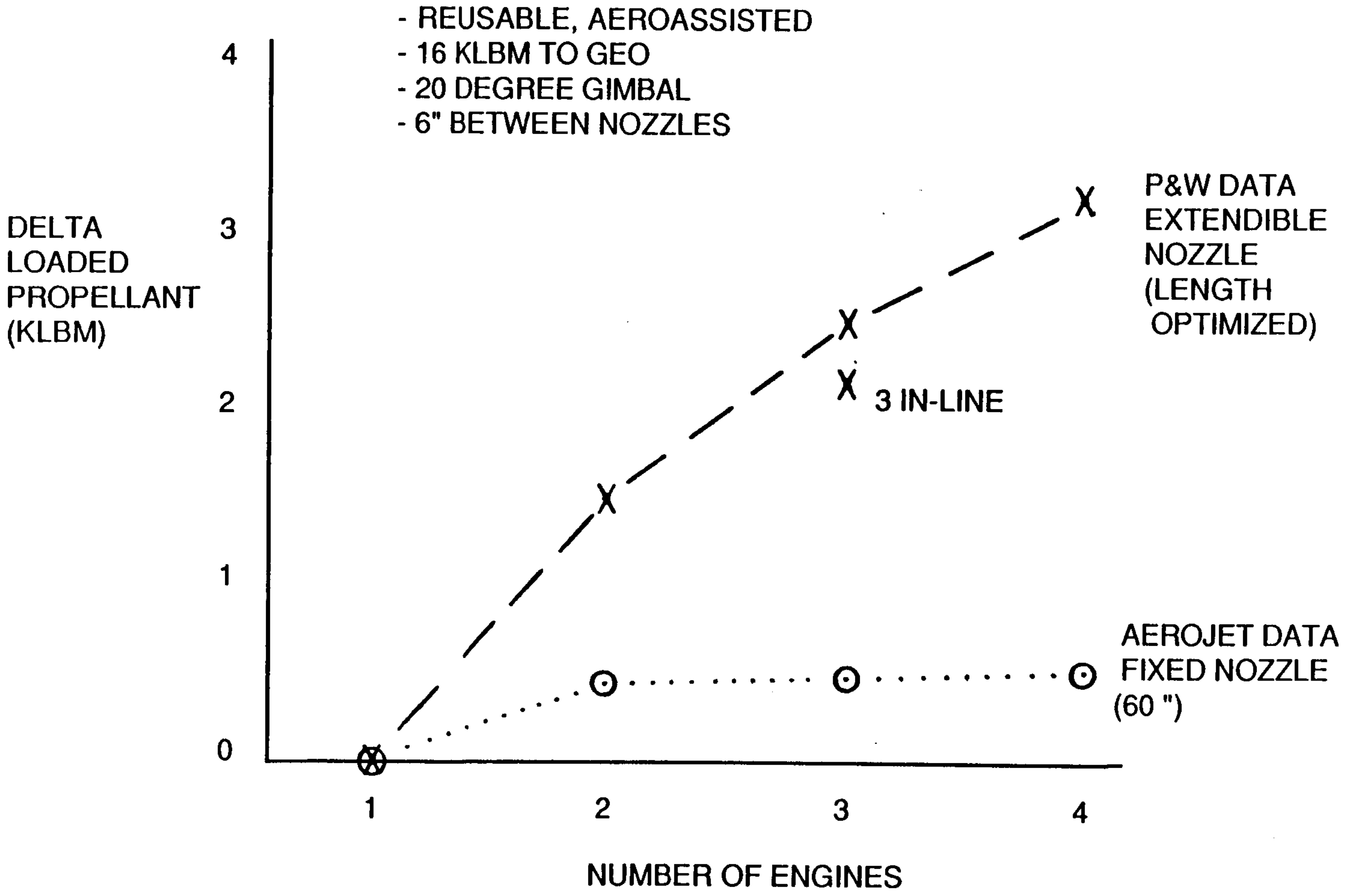
- AEROBRAKE LOCATION, SIZE, AND WEIGHT
- ENGINE PERFORMANCE AS F(THRUST, AREA RATIO, ETC.)
- STRUCTURAL AND PROPULSION SYSTEM WEIGHTS

OTV PERFORMANCE VS. #ENGINES

The figure shows the results of considering vehicle performance differences for various numbers of engines. Delta propellant load as a function of number of engines is given for delivery of 16000 lbm to GEO. The vehicle concept is reusable and aeroassisted.

Also included in the results is the consideration of fixed nozzles and nozzles with extendable/retractable segments. The results show that the highest performing vehicle with engine out capability results with two engines, independant of nozzle design or engine contractor data.

OTV PERFORMANCE VS. #ENGINES

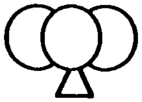
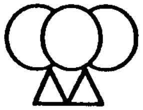
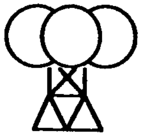
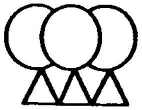


- REUSABLE, AEROASSISTED
- 16 KLBM TO GEO
- 20 DEGREE GIMBAL
- 6" BETWEEN NOZZLES

SPACE BASED REUSABLE OTV COST COMPARISONS

Operational cost comparisons are shown in the facing table for various space based OTV engine configurations. It is assumed in this comparison that the $450M mission repeat cost covers the launch of a second spacecraft and the cost of the spacecraft. An additional $100M is charged for the delivery of each replacement OTV. The results show that the long 2 engine version is the least costly to operate due to its low mission loss costs. The "stretched out" 2 engine configuration also offers advantages as a space based vehicle because of the easier access it offers to various components for on-orbit servicing.

SPACE BASED REUSABLE OTV COST COMPARISONS

CONFIGURATION	MISSION LOSS OTV'S + REDO	S.B. OTV DELIVERY ($100M/REPLACE)	ENGINES ($2M EA)	PROPELLANT COST($2K/LBM)	COMPARISON DELTAS
 *	64 + 540 =$604M	$160M	$24M	REF	REF
	127 + 36 =$163M	$317M	$56M	+$190M	-62M
	5 + 36 =$41M	$12M	$44M	+$300M	-391M
	9 + 67 =$76M	$22M	$66M	+$420M	-204M

* NOT MAN-RATABLE

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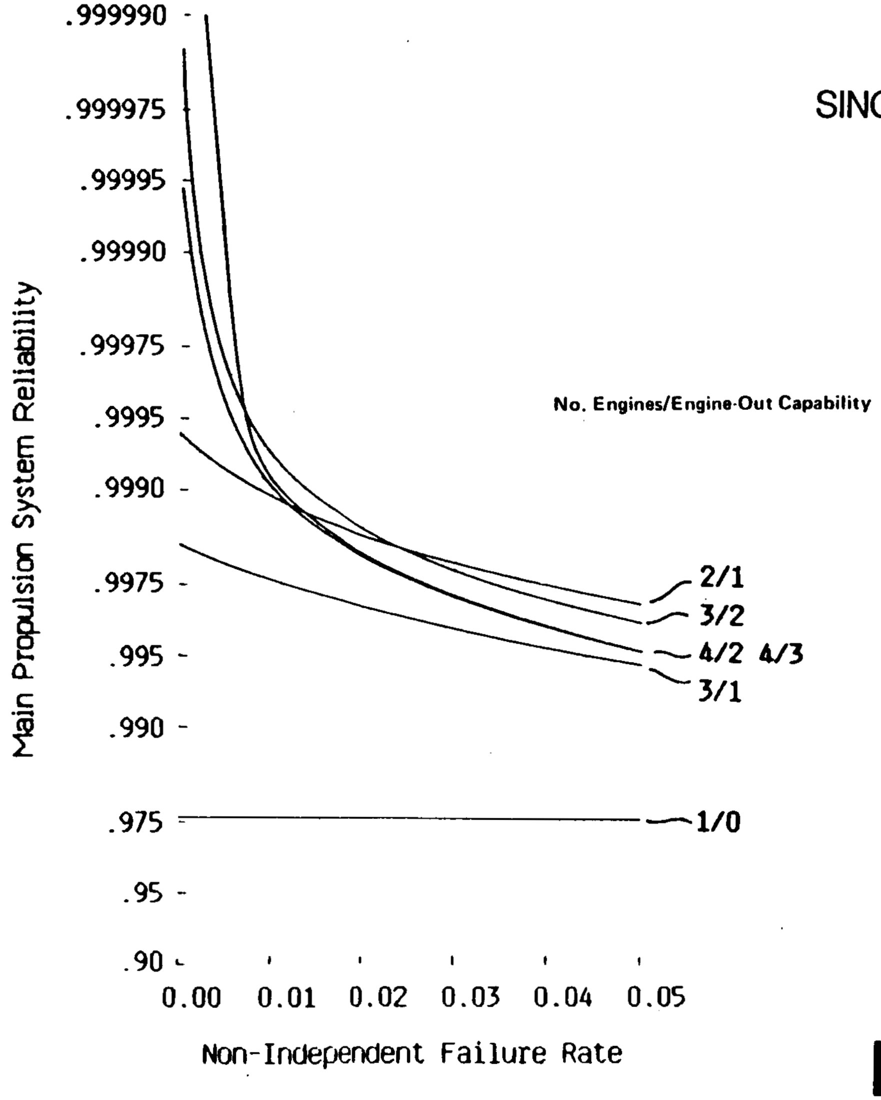
MAIN PROPULSION SYSTEM RELIABILITY COMPARISONS

Mission success probabilities have been computed for propulsion systems with various number of engines. The analysis assumed six burns per mission and .996 reliability per burn for an advanced space engine. Mission reliability is also a function of the non-independant failure rate (fraction of the time that an engine failure results in catastrophe). An accepted industry non-independant failure rate is somewhere around 3%.

The results show that two engines with one engine out capability provide the highest mission success probability for non-independant failure rates over 2 1/2%. Note that these reliability figures represent only the engine reliabilities and not the associated plumbing, controls, etc. Therefore, more than two engines would be penalized with lower system reliability due to the additional hardware (assuming the same level of redundancy).

MAIN PROPULSION SYSTEM RELIABILITY

SINGLE BURN RELIABILITY = .996



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ENGINE CONFIGURATION RECOMMENDATIONS

- SINGLE ENGINE RECOMMENDED FOR NEAR TERM EXPENDABLE AND GROUND BASED REUSABLE OTV'S
- TWO ENGINES WILL BE RECOMMENDED FOR GROUND BASED REUSABLE ONLY IF LARGE NUMBER OF MISSIONS ARE TO BE FLOWN
- TWO ENGINES ARE RECOMMENDED FOR SPACE BASED OTV BECAUSE OF:
    - COST (LOWEST)
    - RELIABILITY (HIGHEST)
    - MANNED PERFORMANCE (HIGHEST)
    - VEHICLE GEOMETRY (ACCESSABILITY)

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LUNAR LANDING ISSUES

LUNAR LANDING GROUNDRULES

Several groundrules were assumed to apply to a Lunar landing scenario with an OTV. Since 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 attitute 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.

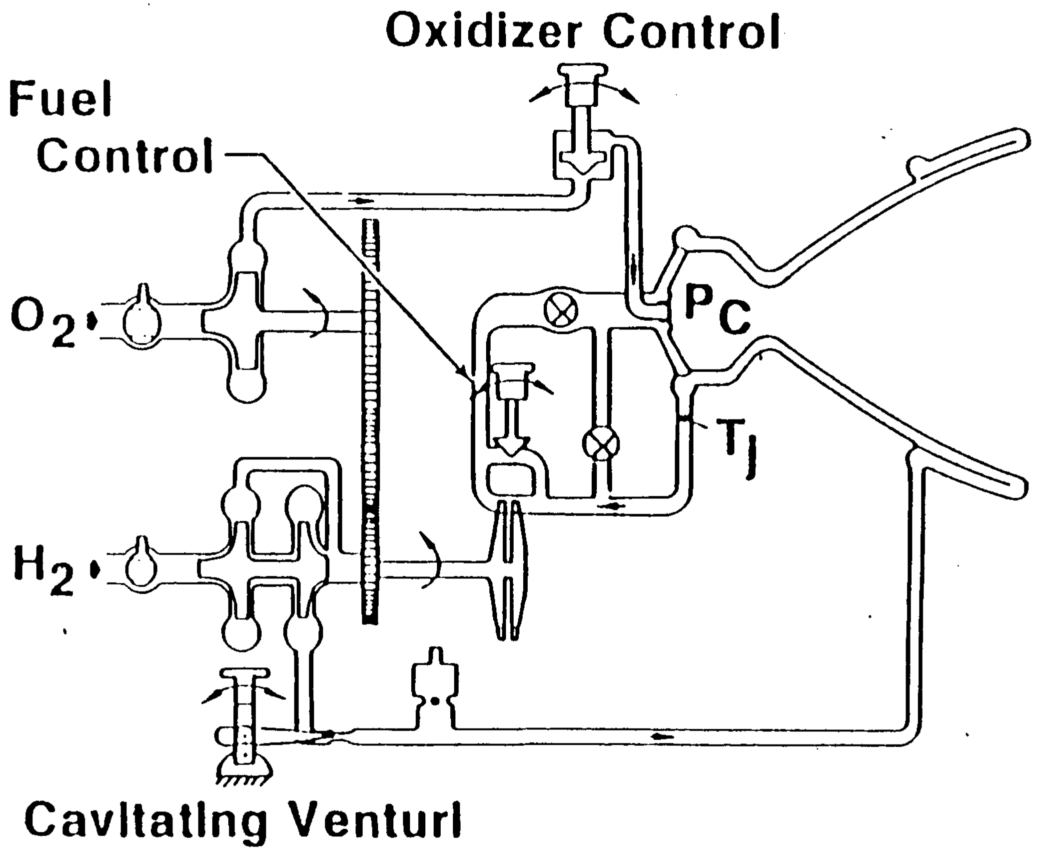
LUNAR LANDING GROUNDRULES

- 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)

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 atomize the liquid oxygen sufficiently for smooth combustion; 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)

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	10.4K + 23.4K = 33.8KLBM	10.4K + 12.1K = 22.5KLBM
TOTAL TOUCHDOWN WT.	15K + 33.8K = 48.8KLBM	62.5KLBM
MINIMUM THRUST	0.065g(48.8K) = 3.2KLBF	0.065g(62.5K) = 4.1KLBF
DESCENT IGNITION WT.	83.4KLBM	107.1KLBM
MAXIMUM THRUST	0.31g(83.4) = 25.9KLBF	0.31g(107.1) = 33.2KLBF

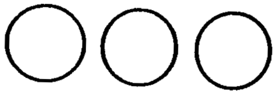
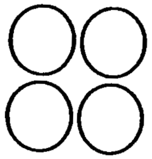
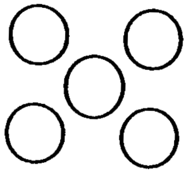
RESULTS: CONTINUOUS THRUST RANGE REQUIREMENTS = 3.2KLBF TO 33.2KLBF

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 - 33.2KLBF	30:1	- HIGH THRUST REQUIRED - LARGE THROTTLING RATIO - WIDE PATTERN
	.9864	0.8K - 16.6 KLBF	21:1	- SMALLEST PATTERN - GOOD RELIABILITY - GROWTH FROM TWO ENGINES
	.9797	0.64K - 11.1KLBF	17:1	- LOWEST RELIABILITY - LARGEST PATTERN - COMPLEX DESIGN AND CONTROL

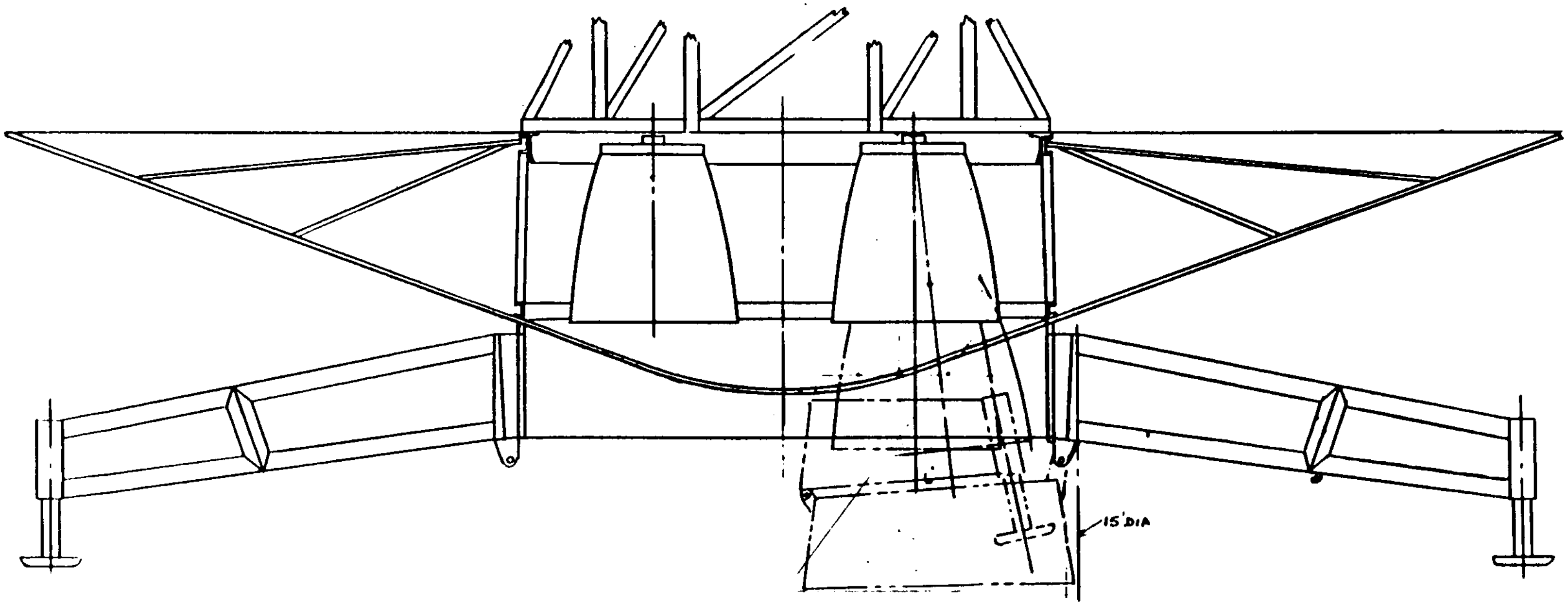
LUNAR LANDING LEGS

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

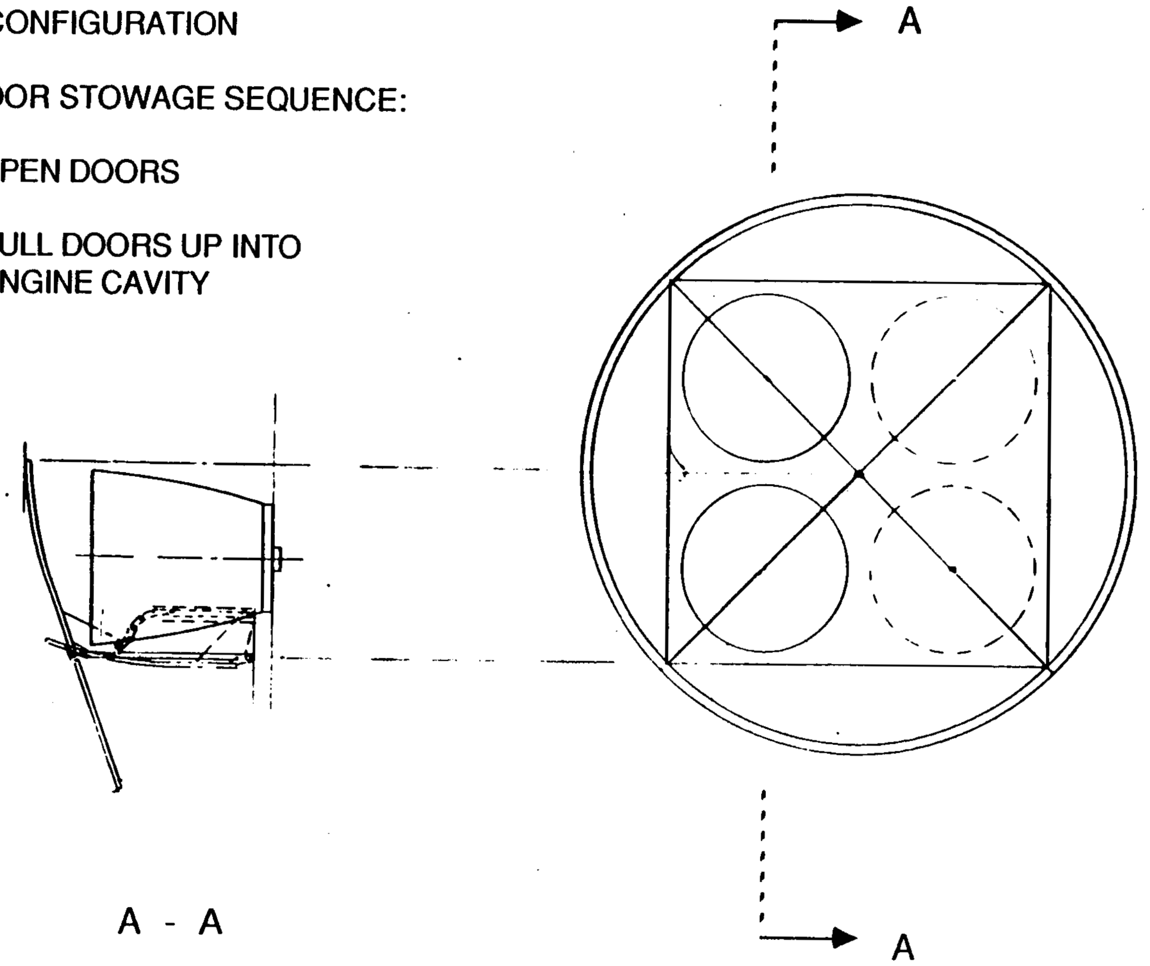


LUNAR LANDING ENGINE COMPARTMENT

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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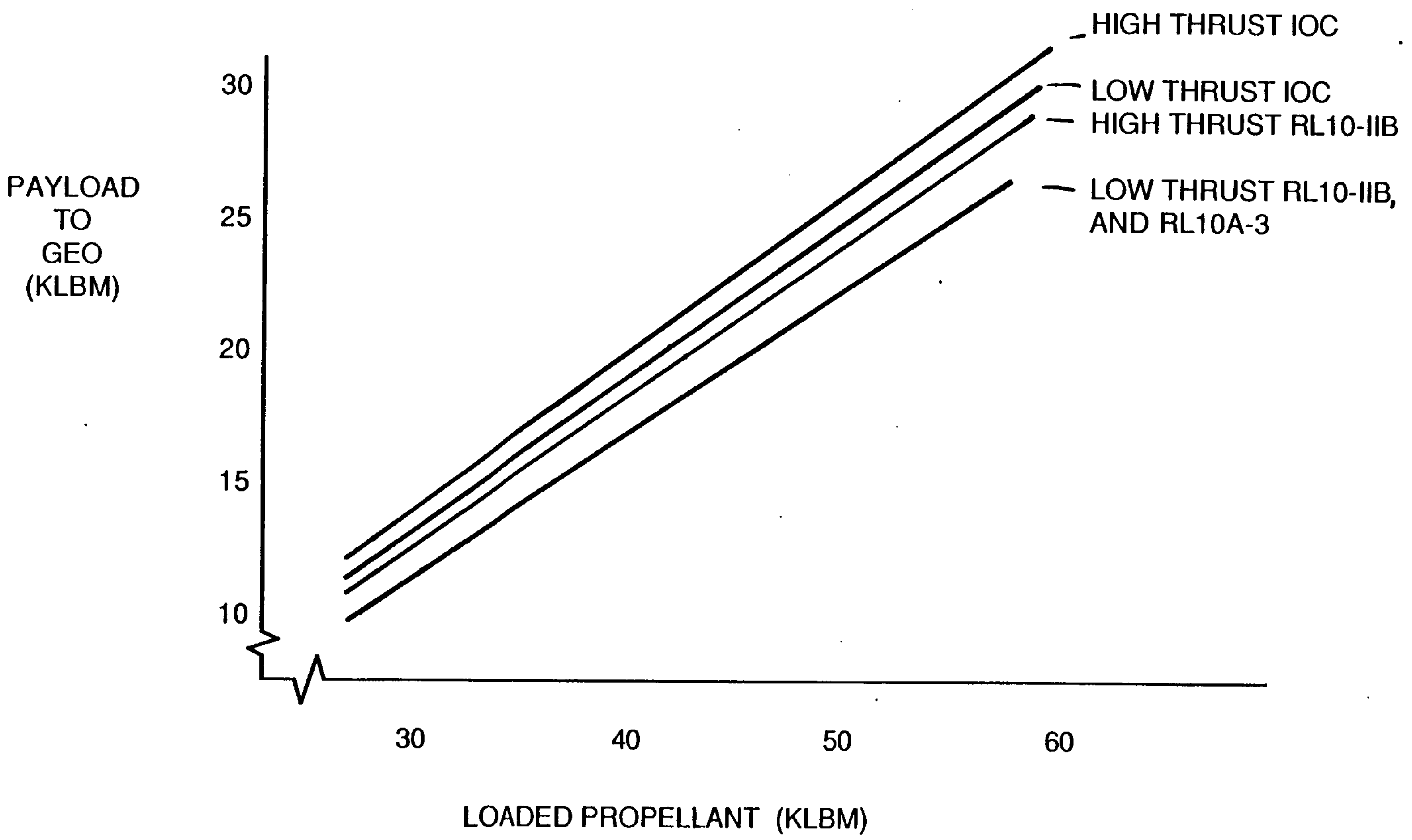
PROGRAM EVOLUTION

EXPENDABLE OTV GEO DELIVERY CAPABILITY

The figure shows payload capability as a function of loaded propellant for an expendable ACC OTV. The cases presented are for GEO delivery. The importance of the figure is to illustrate the vehicle performance capability as the size of the OTV varies. Depending upon the STS lift capability, the ACC OTV propellant capacity may vary and thus result in a subsequent range of payload capability. This is important in choosing the proper size of the ground based vehicle.

Another important use of this parametric information is in deciding which main engine(s) may be appropriate for the OTV program. Performance data are shown here for the Advanced (IOC) engine (Isp of 475 sec), RL10-IIB (Isp of 460 sec), and the RL10A-3(Isp of 440 sec). The higher technology engines may enable certain missions to be flown from the ground and with STS because of limited STS capability. For instance, if the OTV propellant capacity was limited to 50,000 lbm, the IOC engine would be the only engine candidate that could enable an OTV to perform the 25,000 lbm to GEO (0.1g) mission. This type of data will be used in making recommendations for the OTV program.

EXPENDABLE OTV GEO DELIVERY CAPABILITY



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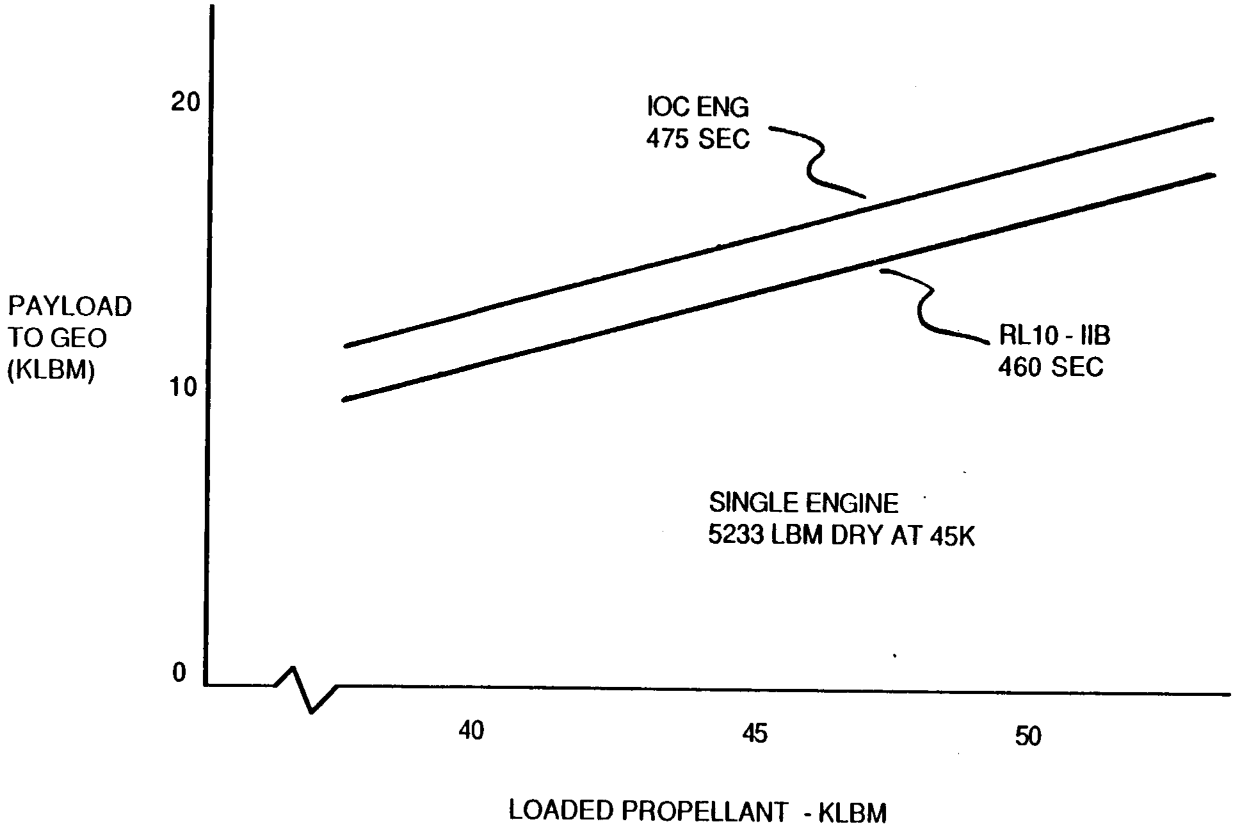
REUSABLE ACC OTV PERFORMANCE

The figure shows GEO payload delivery capability for a reusable ACC OTV. Performance data is shown for both Advanced (IOC) and RL10-IIB engines. As noted the dry vehicle weight is 5233 lbm at 45000 lbm loaded propellant. Loaded propellant values of 55000 lbm and greater are not shown since a vehicle of this size would not be deliverable in the STS and would require a large cargo vehicle. Therefore, it is unlikely that a reusable OTV would be of interest in this size range.

REUSABLE ACC OTV PERFORMANCE

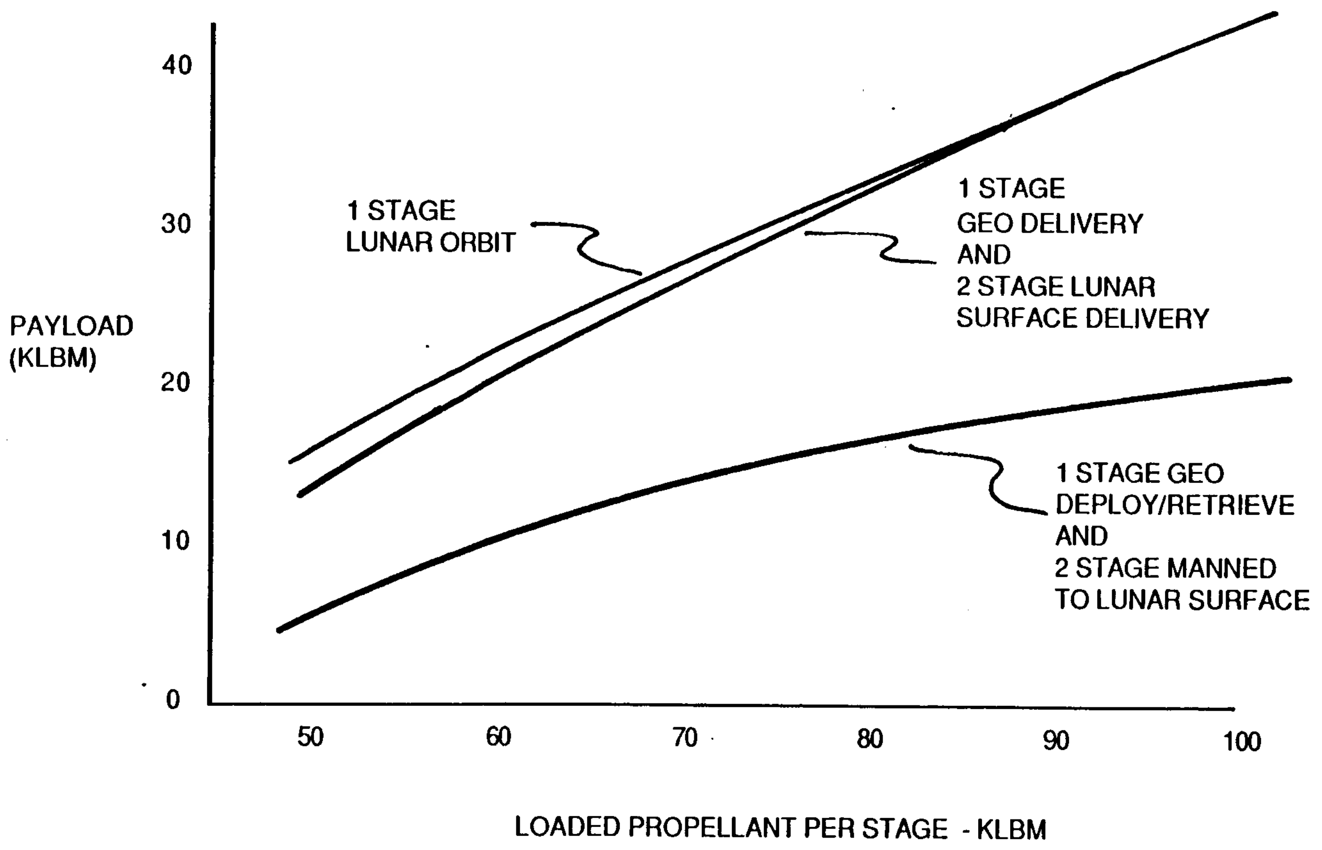


LARGE OTV PERFORMANCE

The figure shows performance capabilities for a large man-ratable OTV (that is, one that is capable of performing large or manned missions and will possibly be space based). The mission capabilities shown are for GEO delivery, GEO retrieval, and for GEO deploy/retrieve missions.

Performance data is also shown here for a variety of Lunar missions with use of one and two stages. The performance data for Lunar landing missions includes the extra dry weight associated with landing legs, two extra engines, and radar equipment required for landing. These data also include the specific impulse of 475 sec of the IOC engine concept.

LARGE OTV PERFORMANCE



MARTIN MARIETTA

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 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 2006	GEO PLATFORM 22 K 2010
STS LAUNCH, EXPENDABLE, RL10A-3 ENG.	27 KLBM	26 KLBM	65 KLBM* 55 KLBM (IOC)*	33 KLBM	49 KLBM
STS LAUNCH, AEROBRAKE, REUSABLE, IOC ENGINE	35 KLBM 38 KLBM (IIB)	44.3 KLBM	42 KLBM 44 KLBM (IIB)	40 KLBM 43 KLBM (IIB)	55 KLBM*

* EXCEEDS STS LIFT CAPABILITY -- REQUIRES LCV

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 KLBM 59 KLBM (RL10A)* 54 KLBM (IIB)*		10 K, C3 = 80 45 KLBM C3 = 110: 62 KLBM*		
AEROBRAKE/ REUSABILITY (STS LAUNCH)		9 K UP, 7.5 K DN 45 KLBM 49 KLBM (IIB)	21 K, C3 = 10 40 KLBM	8.8 K UNMANNED 33 KLBM 35 KLBM (IIB)	
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 - 80 KLBM 40 K UNMANNED 2 - 95 KLBM

* EXCEEDS STS CAPABILITY -- REQUIRES LCV

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RESULTS FROM MISSION CAPTURE ANALYSIS

CORE MODEL - REUSABLE, AEROBRAKED VEHICLE (IOC ENGINE OR RL10-IIB) REQUIRED IN 1998 FOR MULTIPLE PAYLOAD DELIVERY

EARTH INITIATIVE - IOC ENGINE REQUIRED IN 1996 FOR 25K GEO LOW THRUST MISSION

- LARGE SPACE BASED VEHICLE REQUIRED FOR GEO SERVICING

LUNAR INITIATIVE - LARGE SPACE BASED VEHICLE REQUIRED FOR MANNED AND UNMANNED MISSIONS

- 4 ENGINE MODULE, 21:1 THROTTLING RATIO
- LANDING LEGS, RADAR

GROWTH PATH DEVELOPMENT PROGRAMS

OTV evolution will consist of vehicle improvements over a period of time as the missions demand and as certain technologies become available. Program evolution will be decided by cost considerations of which vehicle improvements to make and when to make them.

The purpose of the table is to list the possible vehicle improvements in the logical order of evolution and mission need to assess the subsystem updates/redesigns/developments. Visibility of these is essential in order to group them in an attempt to minimize program evolution costs and schedule impacts. For example, when evolving to a large size OTV the developments required to make the vehicle man-ratable and perhaps space based should just as well be done all at the same time. This type of "grouping" is intended to minimize test hardware/operations, tooling, demonstration missions, design duplicity, development time, qualification paperwork, etc. As the vehicle improvements progress, the overlap in subsystem development groups help tie the program together into a smooth evolution of continuing enhancements in OTV capabilities.

GROWTH PATH DEVELOPMENT PROGRAMS

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

GOALS FOR OTV EVOLUTION

During the OTV program maximum vehicle performance and mission flexibility will be required for all phases of the program evolution. At the same time, impacts to program schedule and cost will need to be minimized in order to enable a cost effective program. These items are listed on the facing chart and have been considered in this study in order to recommend a cost-effective approach to incorporating program improvements as they are needed.

GOALS FOR OTV PROGRAM EVOLUTION

MAXIMIZE:

- VEHICLE PERFORMANCE, MISSION FLEXIBILITY, ETC.

MINIMIZE:

- TEST HARDWARE/TEST OPERATIONS
- DEMONSTRATION MISSIONS
- DRY WEIGHT IMPACTS OF BLOCK CHANGES
- TEST ARTICLE AND PRODUCTION TOOLING
- DEMONSTRATION AND QUALIFICATION PAPERWORK
- DEVELOPMENT SCHEDULE IMPACTS
- DESIGN DUPLICITY

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

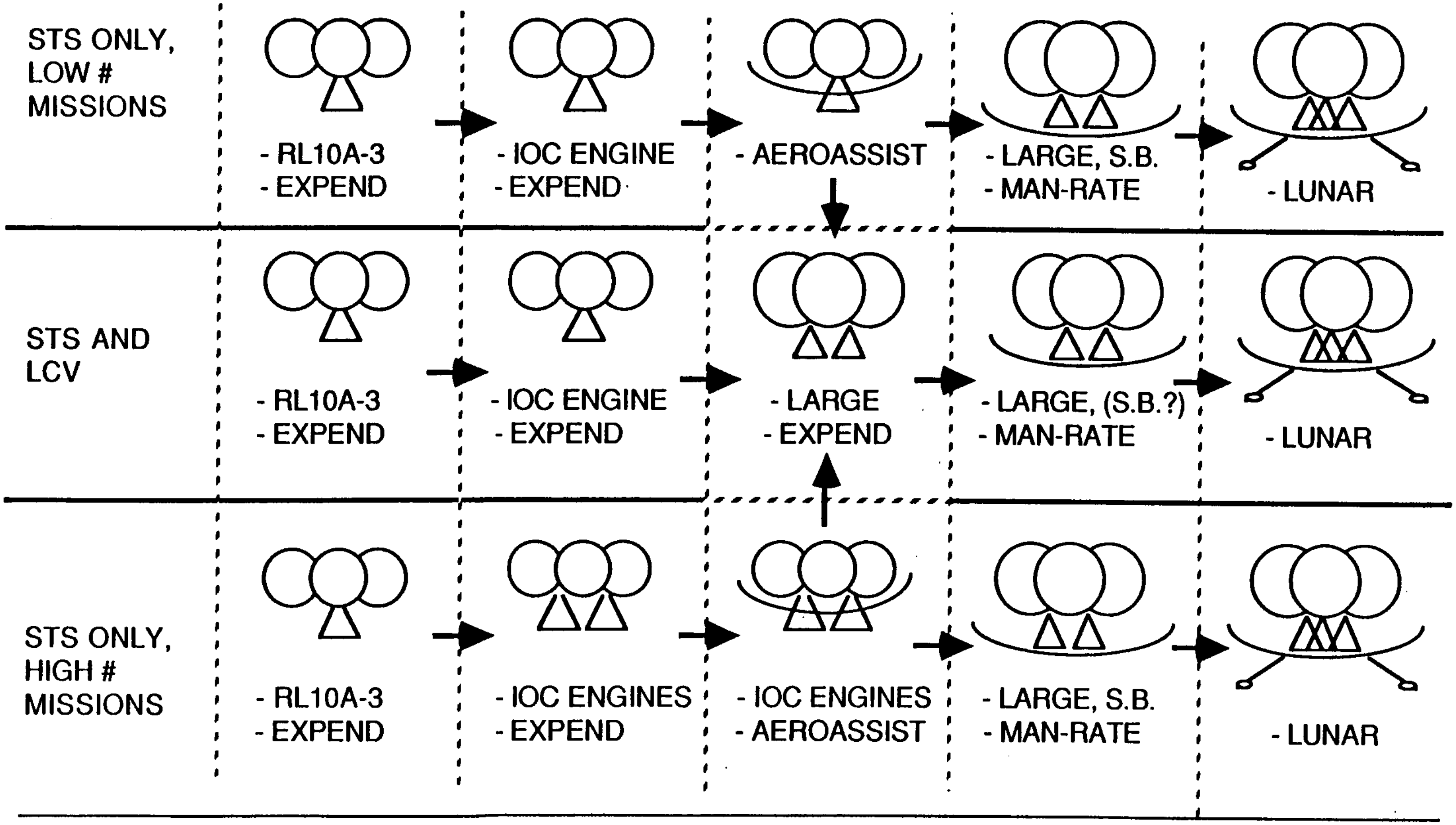
RECOMMENDED OTV EVOLUTION PROGRAMS

The figure shows the recommended evolution paths that apply to OTV in a variety of conditions. These include the cases of STS-only launch capability with a low number of missions, STS followed by a large cargo vehicle as a means of launch to LEO, and STS-only launch with a high number of missions to be flown.

The STS-only scenarios begin with a near term expendable vehicle, introduce an advanced engine, and then evolve into a reusable, aeroassisted vehicle. The next step is to develop a large vehicle to accommodate the larger missions. This requires a decision of whether to use a large cargo vehicle (if one exists) for deployment of a large expendable OTV or whether to begin space basing a large vehicle since the STS will not be able to lift a loaded large OTV.

The above scenario combined with a high number of missions to be performed would differ slightly in evolution. The increased spacecraft delivery reliability of 2 engines may be justified unless the non-recurring costs are significantly higher than for equipping an OTV with a single engine.

RECOMMENDED OTV EVOLUTION PROGRAMS



STS ONLY, LOW # MISSIONS	- RL10A-3 - EXPEND	- IOC ENGINE - EXPEND	- AEROASSIST	- LARGE, S.B. - MAN-RATE	- LUNAR
STS AND LCV	- RL10A-3 - EXPEND	- IOC ENGINE - EXPEND	- LARGE - EXPEND	- LARGE, (S.B.?) - MAN-RATE	- LUNAR
STS ONLY, HIGH # MISSIONS	- RL10A-3 - EXPEND	- IOC ENGINES - EXPEND	- IOC ENGINES - AEROASSIST	- LARGE, S.B. - MAN-RATE	- LUNAR

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AEROASSIST

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AEROASSIST FOR MANNED MARS MISSION

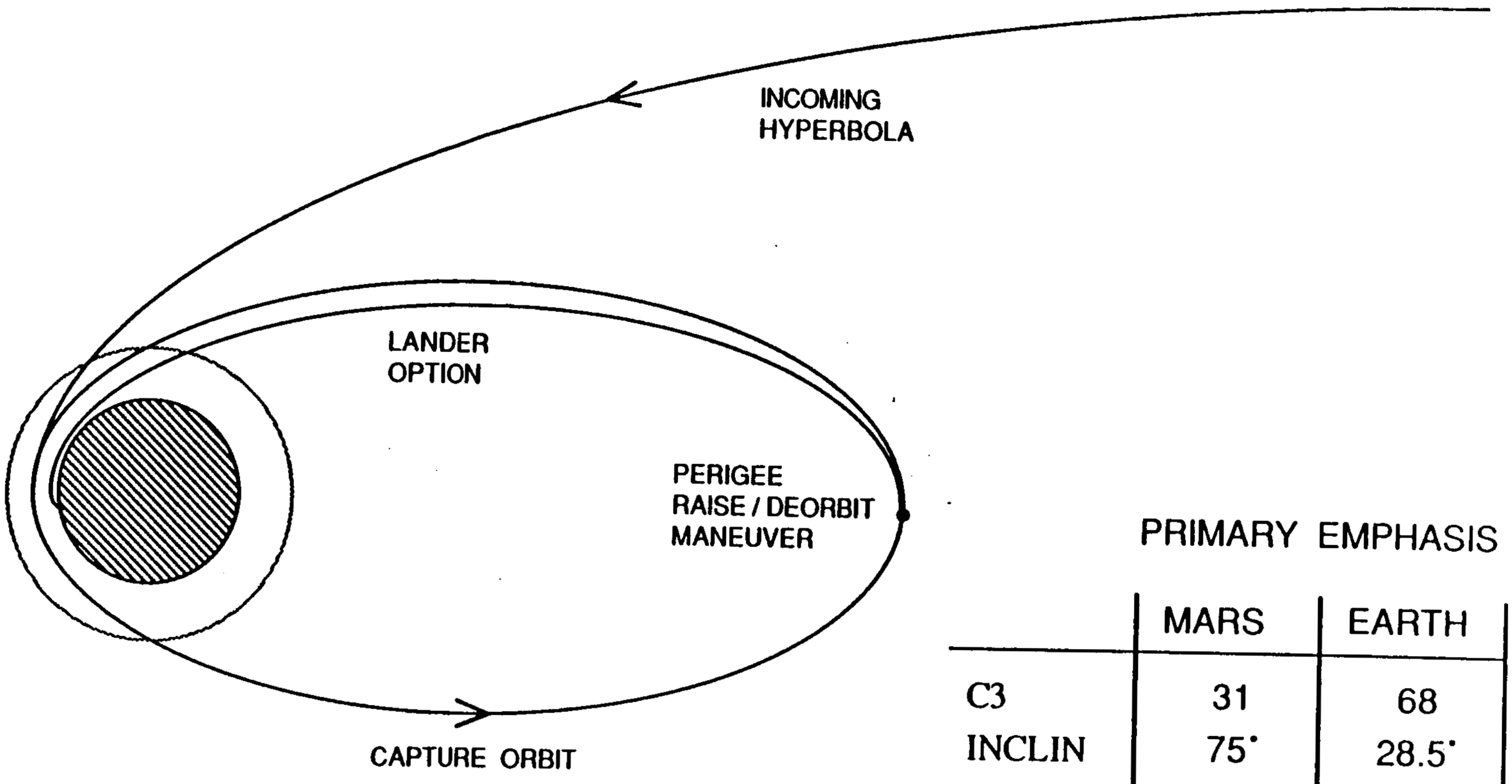
MARTIN MARIETTA

PLANETARY AERO-CAPTURE

The principal of aeroassist can also be applied to hyperbolic planetary encounters where it can be used to capture a vehicle into a closed orbit, hence the term "aerocapture". In order to keep heating and g-loads manageable it is preferable to brake into a highly elliptic initial orbit. This can be modified by subsequent aeropasses or rocket burns. Additionally, a lander can be dispatched directly from this orbit as was done in the Viking project.

The primary concentration of our Mars study will be the encounter/landing conditions shown in the enclosed table.

PLANETARY AERO-CAPTURE



PRIMARY EMPHASIS

	MARS	EARTH
C3	31	68
INCLIN	75°	28.5°
APOGEE	18100 nm	38500 nm
LANDER LAT.	40°	---

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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.)

INTERPLANETARY NAVIGATION

This chart presents an overview of the near-term status of interplanetary navigation. This has a significant impact on the amount of control required in a Mars aeroassist.

Very long base interferonmetry joins electrically the capabilities of a number of widely separated earth tracking stations to achieve high state accuracies very far from earth. The technique has been used in the Voyager project and will be used on Galileo.

Once a spacecraft can optically detect a target planet, terminal navigation using onboard sensors can very accurately locate its position. Two techniques are indicated.

The first uses an onboard video camera (assumed to be part of the science payload) which photographs the planet as the encounter proceeds. Ultimately, an hour before entry, this technique can result in a 1.5 nmi position accuracy.

The second technique uses an autonomous stellar sextant and has higher accuracies because of a wider effective field of view. Based on data for the ANARS Space Sextant Program one could expect positional accuracies on the order of 1/2 nmi at the entry minus one hour point.

In the entry error analysis a middle ground approach was chosen and the accuracies for the video navigation technique used (1.5 nmi error at final correction).

INTERPLANETARY NAVIGATION

- VLBI (VERY LONG BASE INTERFEROMETRY) TECHNIQUES WILL YIELD 5 NM PER AU POSITION ACCURACY IN THE NEAR FUTURE

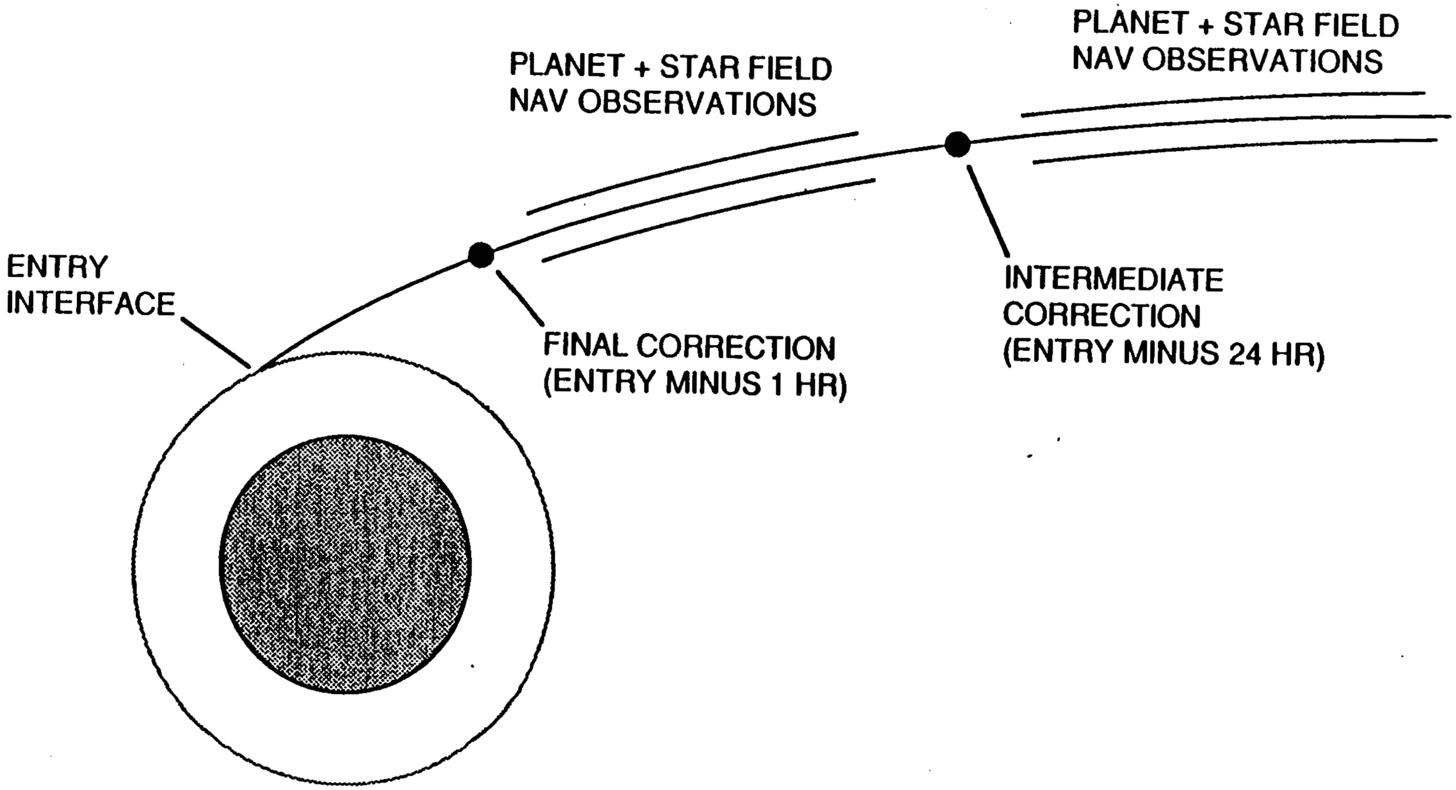
- CURRENT VIDEO NAV TECHNIQUES (VIKING, VOYAGER, ETC) YIELD 1 NM AND 0.15 FPS STATE ACCURACY PER 10000 NM SEPARATION FROM PLANET

- ANARS (SPACE SEXTANT) SHOULD BE ABLE TO YIELD 0.5 NM AND 0.1 FPS ACCURACY IN FINAL 24 HRS. OF ENCOUNTER

TERMINAL NAVIGATION & CORRECTIONS

This figure shows an overview of the planetary encounter process. Terminal navigation updates are coupled with trajectory correction maneuvers to achieve accurate entry conditions. An intermediate correction at about 24 hours out will achieve the general target while a final correction at entry minus one hour will lock in the entry point.

TERMINAL NAVIGATION & CORRECTIONS

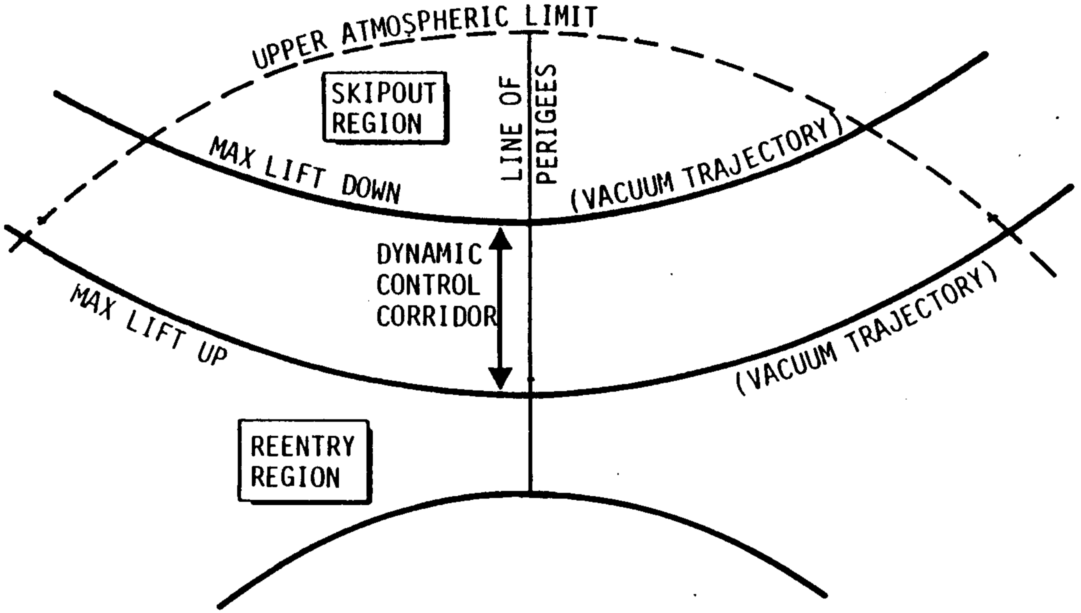


CONTROL CORRIDOR DEFINITION

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

MARS CAPTURE ERROR ANALYSIS

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

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 naviation 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 as discussed in Section 6.1. These independent error contributions are RSS'ed together to yield a net perigee variation due to targeting errors of ± 1.52 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 arodynamic variations is ± 3.59 nmi. in nominal perigee altitude.

When the targeting and aerodynamic errors are combined a net perigee variation of ± 3.89 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 10.2 nmi. which then sets the L/D of the Mars entry vehicle at 0.2 using the parametric data contained in the next chart.

MARS AERO-ENTRY ERROR ANALYSIS

	EQUIVALENT PERIGEE ERROR		
• TARGETING ERRORS (FINAL CORRECTION BURN AT ENTRY MINUS 1 HR)			
- POINTING ERROR	= 130 FT	± .1 DEG	
- CUTOFF ERROR	= 1200 FT	.33 FPS ACCELEROMETER	
- NAV ERROR	= 9100 FT	FROM 1.5 NM POSITION UNCERTAINTY	
	750 FT	FROM 0.2 FPS VELOCITY UNCERTAINTY	
• AERODYNAMIC VARIATION			
- ATMOSPHERIC UNCERTAINTY	= 18800 FT	± 50% DENSITY	
- L/D UNCERTAINTY	= 10900 FT	± 2° AT 13° ANGLE OF ATTACK (± 30% L/D)	
- BALLISTIC UNCERTAINTY	= 1600 FT	WT = ± 150 LB (RESIDUALS) C_D = ± 5% (STS/VIKING DATA) A = ± 5%	± 8% $W/C_D A$

• RSS

= ± 9210 FT = ± 1.52 NM FROM TARGETING

= ± 21800 FT = ± 3.59 NM FROM AERODYNAMICS

= ± 23700 FT = ± 3.89 NM NET VARIATION

CONCLUSION: 10.3 N.M. CONTROL CORRIDOR REQUIRED TO COVER ERRORS WITH 33% MARGIN

MARS CAPTURE PARAMETRICS

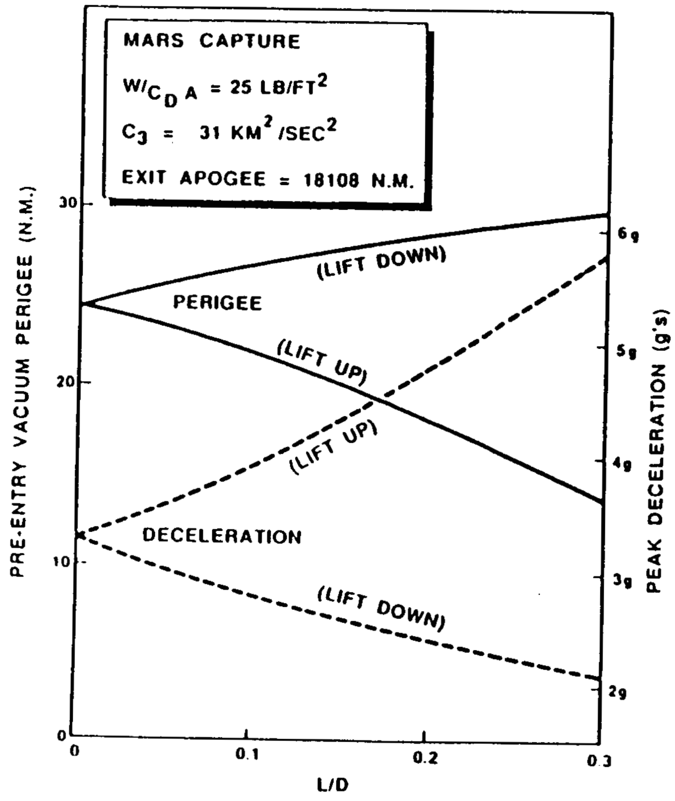
Various entry trajectories were generated utilizing a pre-entry hyperbols with a C_3 of 32 km^2/sec^2 and a Mars capture agogee of 18108 nmi. (post-aero). Aerodynamic L/D and ballistic coefficient were varied for continuous lift up and lift down trajectories to generate the parametric data base. Because of nature 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.

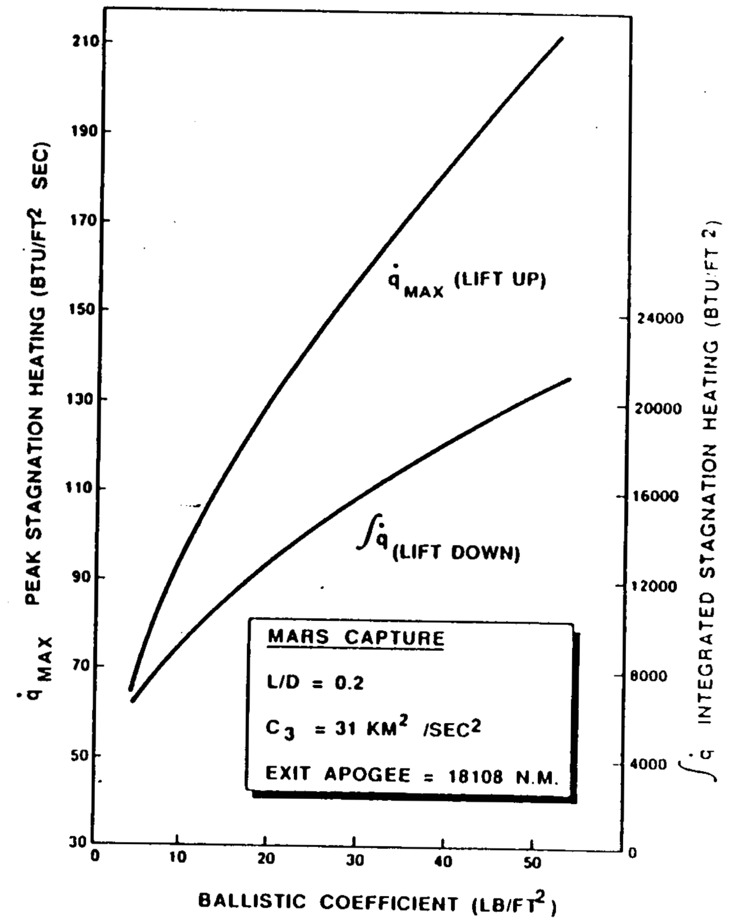
The difference between the pre-entry vacuum perigees for lift up and lift down aerotrajectories defines a control corridor width which represents the region in which the vehicle can be controlled to the desired exit conditions with the available lift. Once error analysis has defined the magnitude of this control corridor, the vehicle's required L/D is set. For a control corridor width of 10.3 nm an L/D of 0.2 is required for Mars capture.

Peak entry deceleration is shown in g's 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.

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 arobrake's insulating TPS.

MARS CAPTURE PARAMETRICS

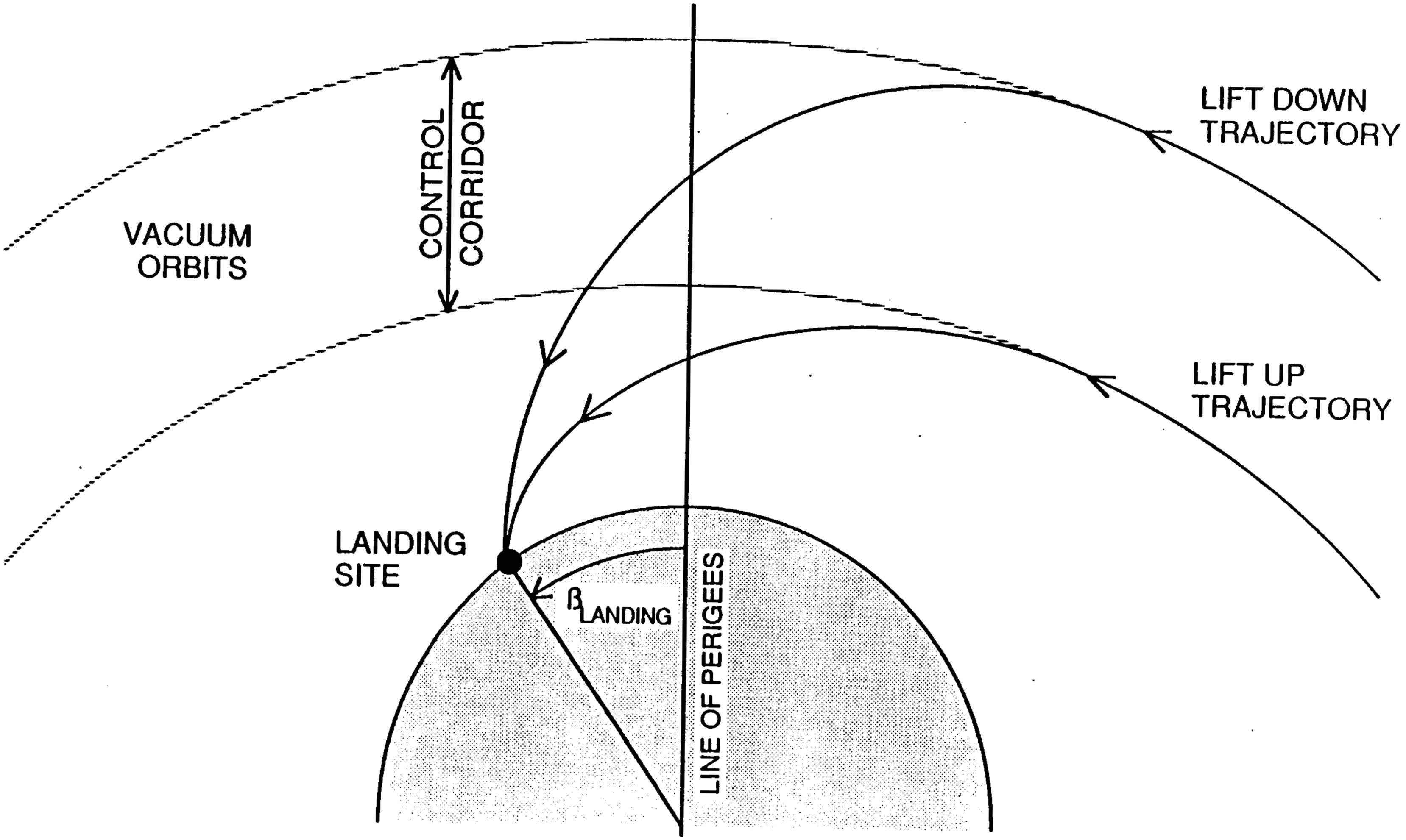




MARS LANDING

This figure shows an overview of the Mars landing process. A touchdown point is defined with respect to the vacuum perigee of the downleg orbit. This parameter, $\beta_{LANDING}$, determines the vehicle loading conditions: the further downrange the landing site is the lower the peak heating and airloads. For the Mars application, avalue of 8.0 degrees was chosen which minimizes loads without getting too close to the skipout boundary. As with the capture phase, lift up and lift down trajectories are used to define an aero control corridor which is used with error analysis results to set the required L/D.

MARS LANDING



MARS AERO-LANDING ERROR ANALYSIS

This figure shows the results of entry error analysis conducted for the Mars landing mission phase. The primary differences from the capture phase is in the sensitivity of aerodynamic variations due to the lower energy entry condition. The net result of this error analysis is a 5.22 nmi control corridor requirement.

MARS AERO-LANDING ERROR ANALYSIS

	EQUIVALENT PERIGEE ERROR		
• TARGETING ERRORS (FINAL CORRECTION BURN AT ENTRY MINUS 1 HR)			
- POINTING ERROR	= 130 FT	± .1 DEG	
- CUTOFF ERROR	= 1200 FT	.33 FPS ACCELEROMETER	
- NAV ERROR	= 9100 FT	FROM 1.5 NM POSITION UNCERTAINTY	
	750 FT	FROM 0.2 FPS VELOCITY UNCERTAINTY	
• AERODYNAMIC VARIATION			
- ATMOSPHERIC UNCERTAINTY	= 14400 FT	± 50% DENSITY	
- L/D UNCERTAINTY	= 15800 FT	± 2° AT 9° ANGLE OF ATTACK (± 30% L/D)	
- BALLISTIC UNCERTAINTY	= 3500 FT	WT = ± 150 LB (RESIDUALS) C_D = ± 5% (STS/VIKING DATA) A = ± 5%	± 8% W/C_D A

• RSS

= ± 9210 FT = ± 1.52 NM FROM TARGETING

= ± 21700 FT = ± 3.57 NM FROM AERODYNAMICS

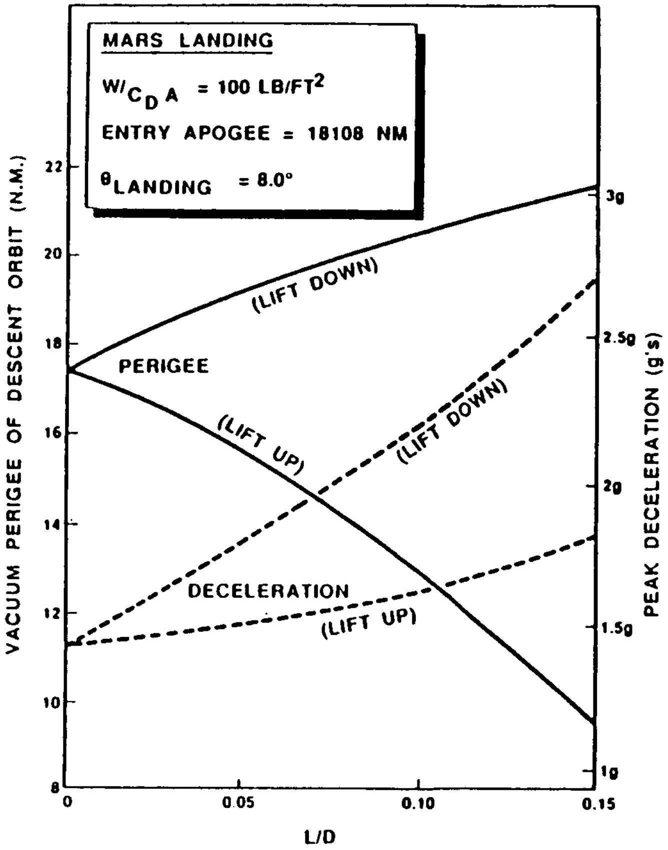
= ± 23500 FT = ± 3.87 NM NET VARIATION

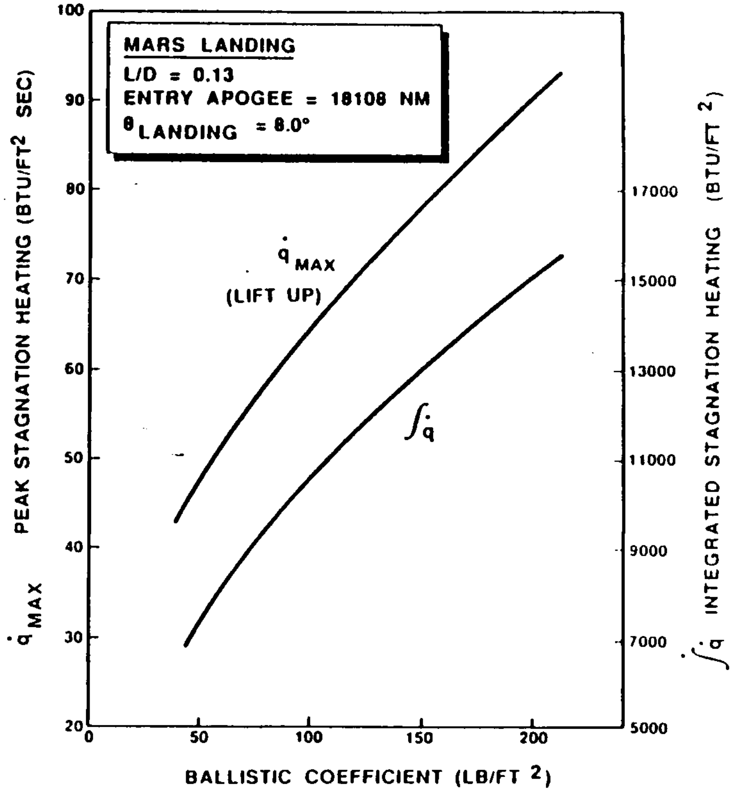
CONCLUSION: 5.22 N.M. CONTROL CORRIDOR REQUIRED TO COVER ERRORS WITH 33% MARGIN

MARS LANDING PARAMETRICS

This figure shows the aeroentry data base for the Mars landing phase. As in the capture phase this includes data on lift up and lift down trajectories for vacuum perigees (whose difference yields control corridor), deceleration loads, peak stagnation heating, and integrated heating (both for a 1 ft. sphere).

MARS LANDING PARAMETRICS





EARTH CAPTURE ERROR ANALYSIS

This figure shows the results of entry error analysis conducted for the Earth capture mission phase. The primary differences from the Mars capture phase is in the accuracy of the terminal navigation (use of GPS is assumed here) and in the sensitivity of aerodynamic variations (due to the lower energy entry condition). The net result of this error analysis is a 4.68 nmi control corridor requirement.

EARTH CAPTURE ERROR ANALYSIS

	EQUIVALENT PERIGEE ERROR	
• TARGETING ERRORS (FINAL CORRECTION BURN AT ENTRY MINUS 1 HR)		
- POINTING ERROR	= 1095 FT	± .1 DEG
- CUTOFF ERROR	= 2217 FT	.33 FPS ACCELEROMETER
- NAV ERROR	= 899 FT	FROM 1020 FT POSITION UNCERTAINTY
	1342 FT	FROM 0.1 FPS VELOCITY UNCERTAINTY
• AERODYNAMIC VARIATION		
- ATMOSPHERIC UNCERTAINTY	= 5100 FT	± 30% DENSITY
- L/D UNCERTAINTY	= 7300 FT	± 2° AT 9° ANGLE OF ATTACK (± 30% L/D)
- BALLISTIC UNCERTAINTY	= 2000 FT	WT = ± 150 LB (RESIDUALS) C_D = ± 5% (STS/VIKING DATA) A = ± 5% ± 8% $W/C_D A$
• RSS	= ± 3000 FT = ± 0.49 NM FROM TARGETING	
	= ± 10300 FT = ± 1.69 NM FROM AERODYNAMICS	
	= ± 10700 FT = ± 1.76 NM NET VARIATION	

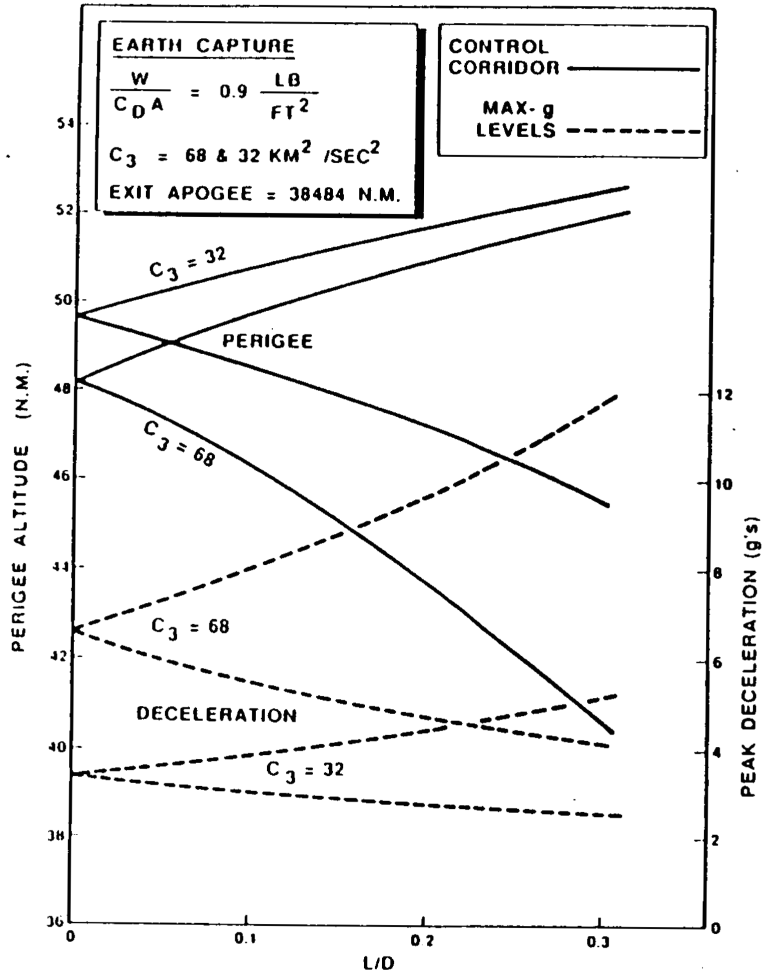
CONCLUSION: 4.68 N.M. CONTROL CORRIDOR REQUIRED TO COVER ERRORS WITH 33% MARGIN

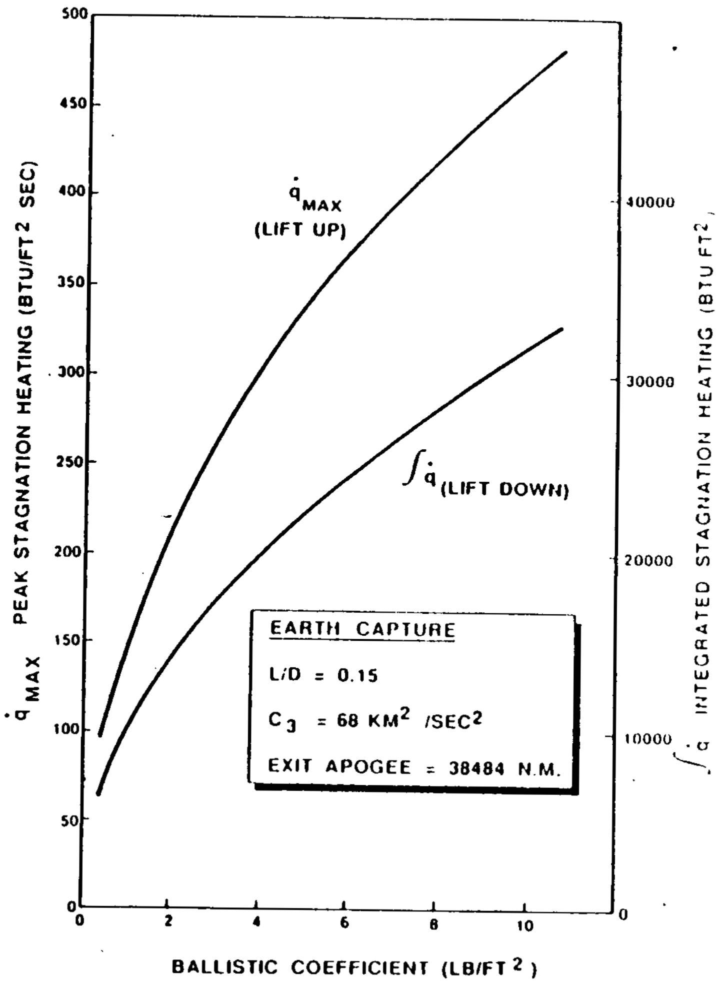
EARTH CAPTURE PARAMETRICS

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), deceleration loads, peak stagnation heating, and integrated heating (both for a 1 ft. sphere).

C-3

EARTH CAPTURE PARAMETRICS





AERO CONFIGURATION

This figure summarizes the aeroassist data for the Mars vehicle by mission phase (Mars capture, Mars landing, Earth capture). The Mars capture brake is also used as the Earth capture brake to save weight by avoiding duplication. This brake is used at an L/D of 0.20 for Mars capture and an L/D of 0.15 for Earth capture. The brake's structural characteristics are primarily set by the very stressful Earth encounter condition. It is 142 ft in diameter and weighs 23371 lb of which 401 lb is Rigid Surface Insulation (RSI) and 8890 lb is Flexible Surface Insulation (FSI). The Mars landing brake is used at an L/D of 0.133 and is constructed entirely of FSI due to the lower heating of entry. It is 36 ft in diameter and weighs 1392 lb of which 786 lb is FSI.

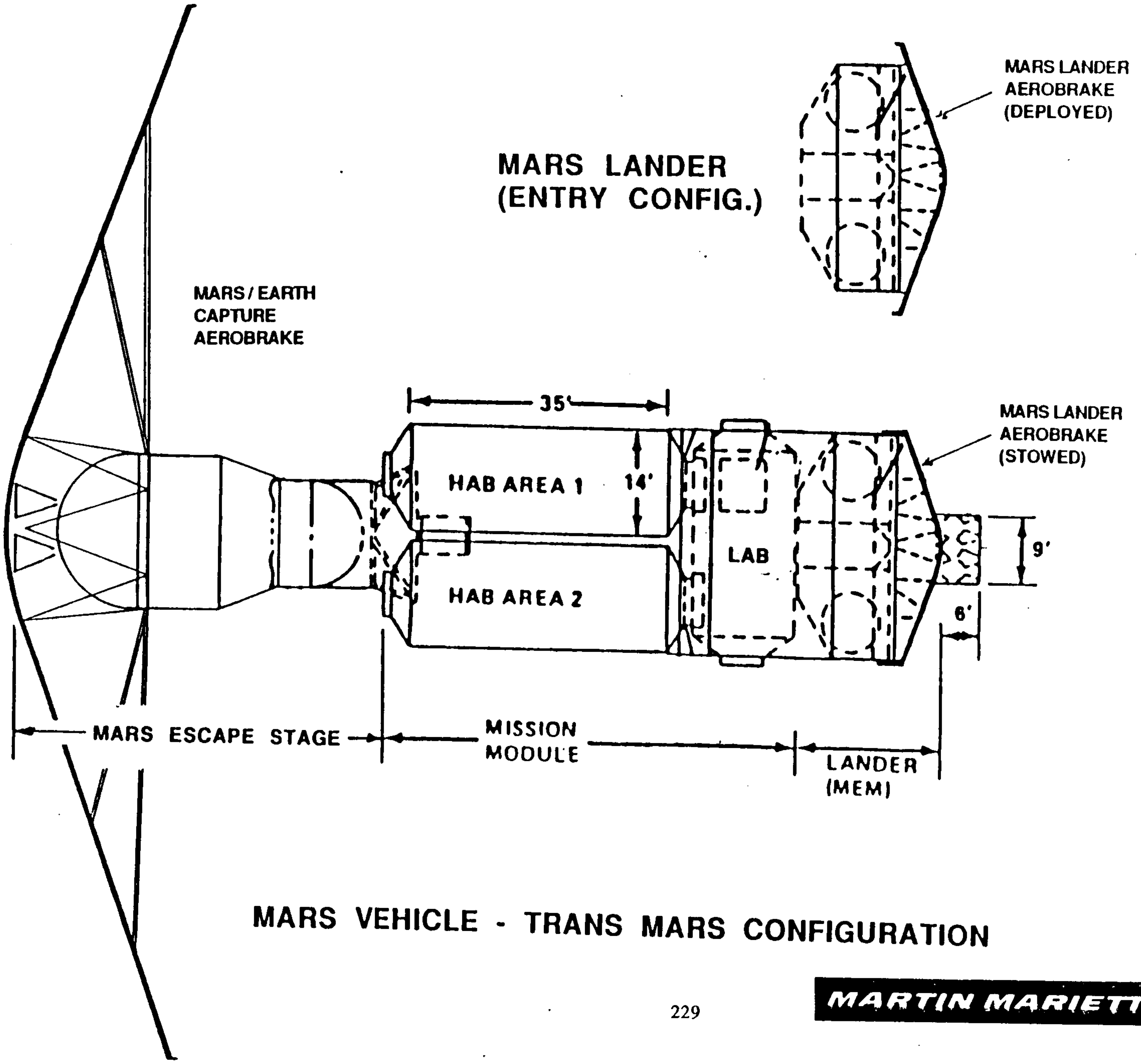
AERO CONFIGURATION DATA

	MARS CAPTURE	MARS LANDING	EARTH CAPTURE
VEHICLE WEIGHT	465300 LB	157500 LB	163000 LB
L/D	0.2	0.133	0.15
ANGLE OF ATTACK	12.42 DEG	8.28 DEG	9.33 DEG
AEROBRAKE DIAMETER	142 FT	36 FT	142 FT
BALLISTIC COEFFICIENT	19.3 LB/FT2	100 LB/FT2	6.6 LB/FT2
PEAK LOADING	4.9 g's	2.5 g's	8.3 g's
NUMBER OF RIBS	38	12	38
AEROBRAKE TOTAL WEIGHT (*)	23371 LB (*)	1392 LB (*)	23371 LB (*)
RSI WEIGHT	401 LB	0	401 LB
FSI WEIGHT	8890 LB	786 LB	8890 LB
STRUCTURAL WEIGHT	11032 LB	424 LB	11032 LB

(* NOTE: AEROBRAKE TOTAL WEIGHT INCLUDES A 15% CONTINGENCY)

MARS VEHICLE - TRANS MARS CONFIGURATION

The concept of a low L/D, ceramic fabric aerobrake developed in the OTV Phase A has been applied to a manned Mars mission. The resulting vehicle configuration is shown in this figure. The large Mars/Earth capture brake, which is permanently deployed, is shown on the left. The Mars excursion module is shown on the right in its trans-Mars configuration. The MEM's all-fabric brake is folded up in transit to prevent aerodynamic impingement during Mars aerocapture. Once in Mars orbit this brake is deployed in preparation for landing. Upon return to the Earth the entire remaining stack, consisting of the Mars escape stage, Lab/Hab modules, and the MEM stage 2 is aerocaptured by reusing the large brake.



MARS VEHICLE - TRANS MARS CONFIGURATION

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REMAINING ISSUES

- G-LOAD REDUCTION FOR HIGH SPEED EARTH ENTRY ($C_3 = 68\ KM^2/SEC^2$)
  - INCREASED L/D
    OR
  - HYBRID AEROCAPTURE (PARTIAL PROPULSIVE)
- LOWER ENERGY EARTH ENCOUNTER ($C_3 < 40\ KM^2/SEC^2$)
  - G-LOAD REDUCTION
  - AEROBRAKE STRUCTURAL & TPS REDUCTION
- BETTER LANDING PHASE CHARACTERIZATION
  - PARACHUTE / LANDING ROCKET CONSTRAINTS
  - SENSOR POINTING DURING AEROPHASE

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HIGH ENTRY SPEED
AEROASSIST

MARTIN MARIETTA

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HIGH SPEED ENTRY AEROASSIST

FOUR DIFFERENT CLASSES OF ENTRIES HAVE BEEN DEFINED:

1) LUNAR RETURN

2) PLANETARY BOOST RETURN

3) EARTH CAPTURE C3 = 8.0 16.0 32.0 $\boxed{68.0}$ KM2 / SEC2

4) MARS CAPTURE C3 = 8.23 13.0 $\boxed{31.0}$ 60.0 KM2 / SEC2

▭ INDICATES CONDITIONS COVERED IN 1986 STUDY

STATUS: PARAMETRIC DATA GENERATION HAS BEEN COMPLETED

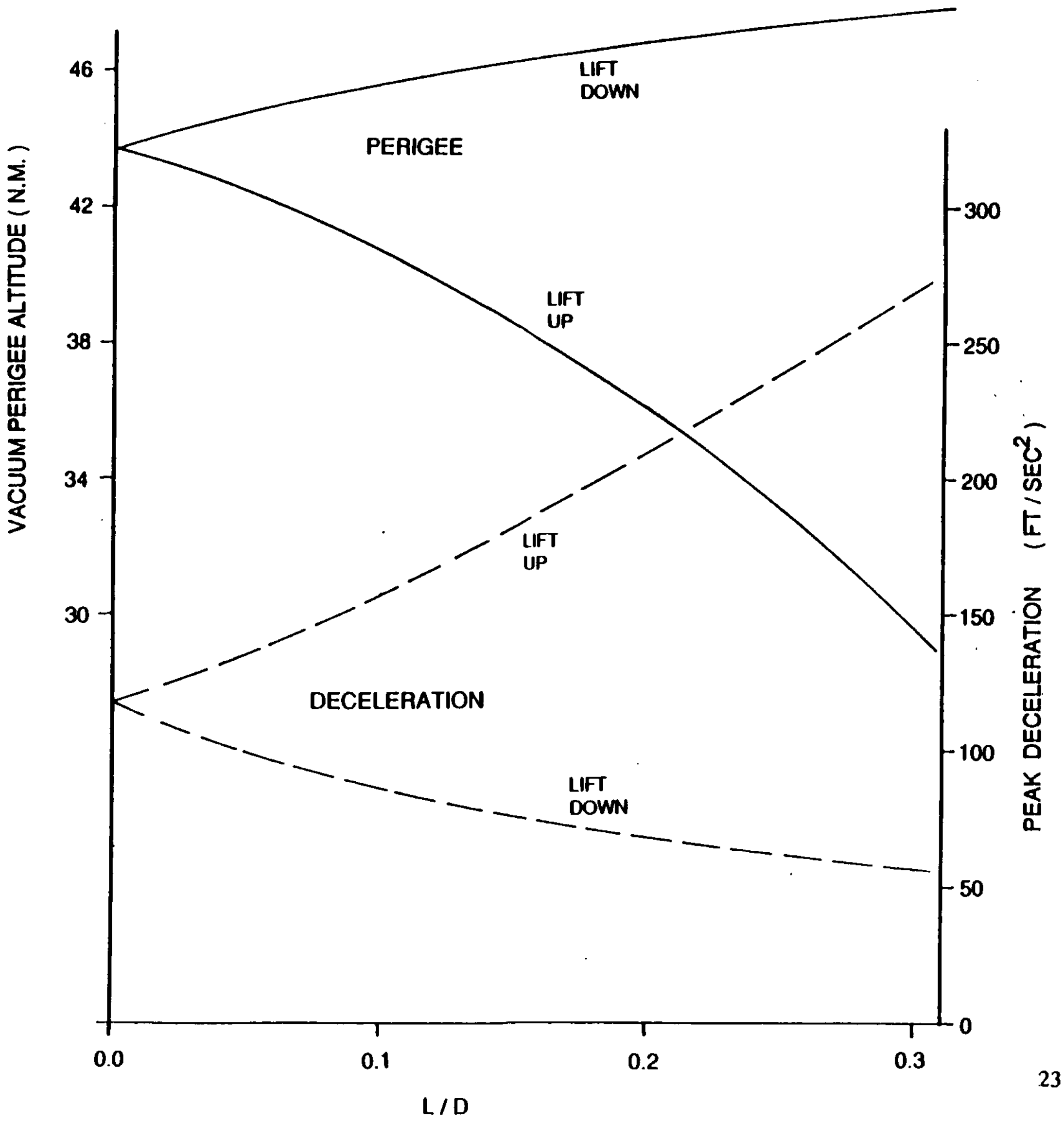
REDUCTION & ANALYSIS IS UNDERWAY

MARTIN MARIETTA

LUNAR RETURN CONTROL & LOADS

This figure shows an example of the data base being developed for the high velocity aeroassist conditions. This shows data for the Lunar return case. 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. This parameter is important in establishing the L/D requirements for a particular entry case. Also shown is the peak deceleration which is used to size structural elements.

LUNAR RETURN CONTROL & LOADS



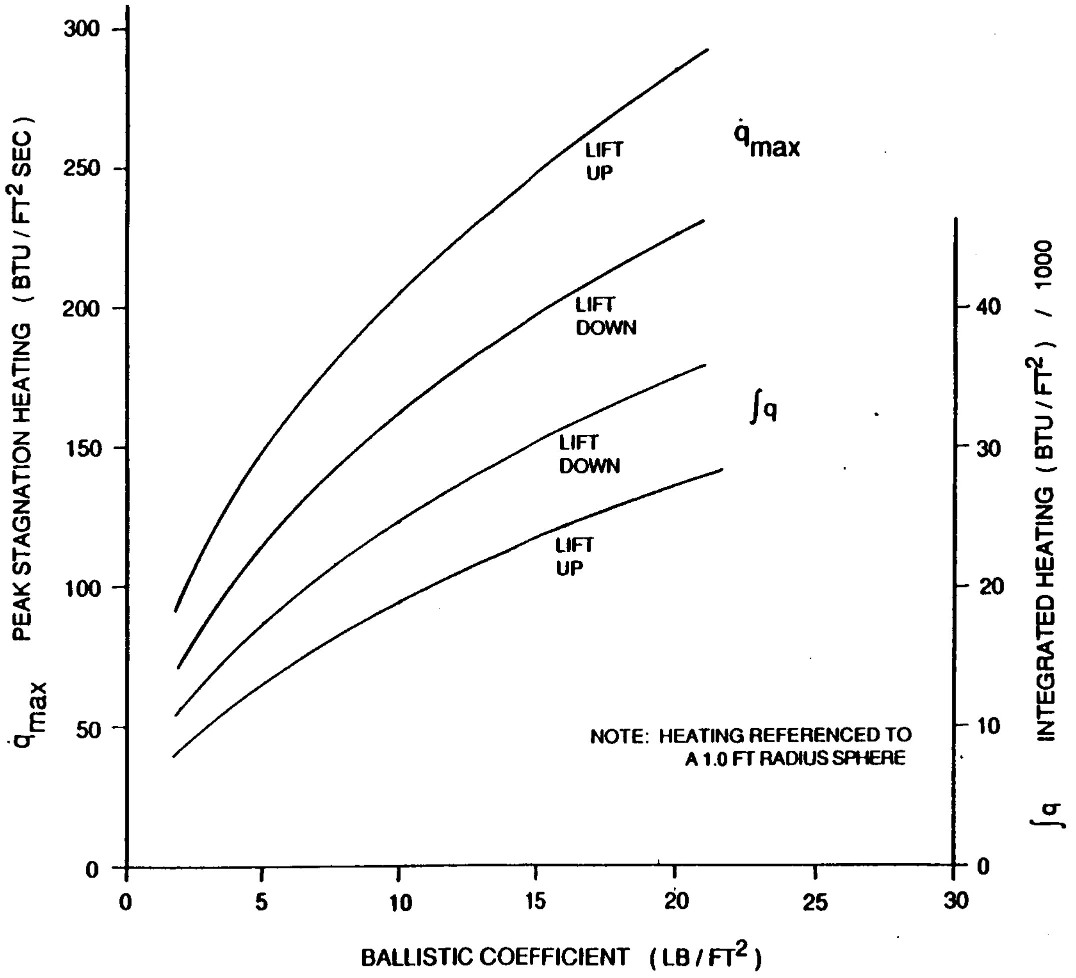
- RETURN FROM MOON TO S.S.
- AEROASSIST CONTROL CORRIDOR
  - WIDTH = DELTA OF PERIGEES
  - ERROR ANALYSIS SETS REQMT
  - CONTROL REQMT SETS L/D
- PEAK DECELERATION
  - SETS STRUCTURAL REQUIREMENTS

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. In the coming months, analysis will attempt to derive meaningful design points for some of the baseline entries.

LUNAR RETURN HEATING



- RETURN FROM MOON TO S.S.
- PEAK STAGNATION HEATING

  SETS TPS MATERIAL REQMTS
- INTEGRATED HEATING

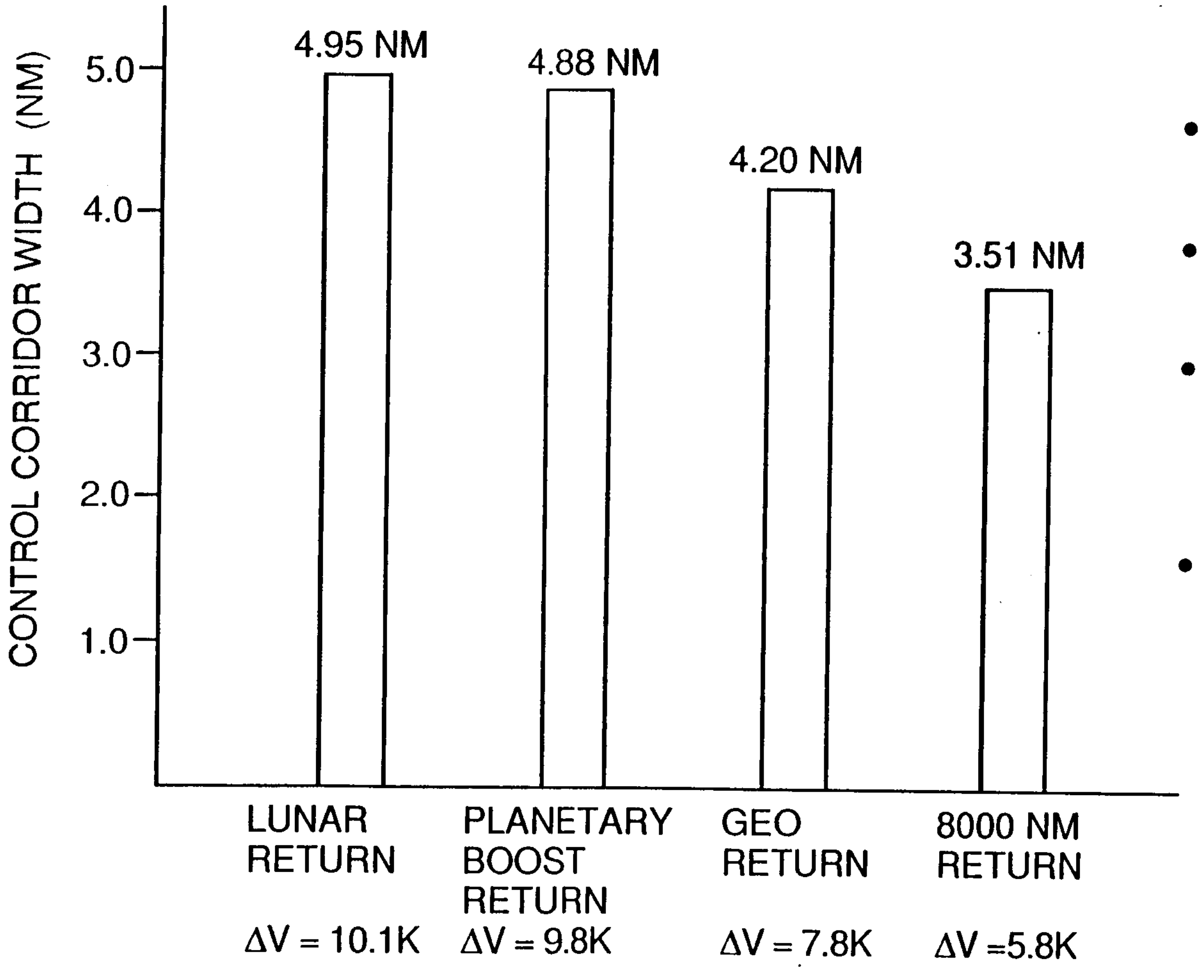
  SETS TPS THICKNESS

MARTIN MARIETTA

L/D EFFECTIVENESS

This figure shows the relative levels of control resulting from a fixed L/D for various entry velocities. As may be seen, the control corridor is a strong function of entry velocity and decays as velocity is reduced. These results impact the selection of L/D for new mission applications.

L/D EFFECTIVENESS



- AEROASSIST TO 245 NM
- L/D = 0.10
- CONTROL CORRIDOR WIDTH IS CREATED BY L/D
- L/D EFFECTIVENESS IS A FUNCTION OF THE AERO ΔV

MARTIN MARIETTA