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Advanced Missions Safety

Volume II: Technical Discussion

Part I - Space Shuttle Rescue Capability

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Prepared by
SYSTEMS PLANNING DIVISION

1 September 1972

Prepared for
OFFICE OF MANNED SPACE FLIGHT
NATIONAL AERONAUTICS AND SPACE ADMINISTRATION
Washington, D. C.

Contract No. NASw-2301

Systems Engineering Operations
THE AEROSPACE CORPORATION

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VOLUME II - TECHNICAL DISCUSSION

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PREFACE

This study on Advanced Mission Safety has been performed as Task 2.6 of Contract NASw-2301, entitled Advanced Space Program Analysis and Planning. The task consists of three subtasks:

Subtask 1 - Space Shuttle Rescue Capability

Subtask 2 - Experiment Safety

Subtask 3 - Emergency Crew Transfer

Each subtask is an entity not related to or dependent upon any activity under either of the other two subtasks.

The results of Task 2.6 are presented in three volumes.

Volume I: Executive Summary Report presents a brief, concise review of results and summarizes the principal conclusions and recommendations for all three subtasks.

Volume II: Technical Discussion is in three parts, each providing a comprehensive discussion of a single subtask.

Part 1 provides an assessment of Earth Orbit Shuttle (EOS) capability to perform a rescue mission. It treats several concepts for augmenting this capability and increasing EOS rescue mission utility.

Part 2 presents an analysis of potential hazards introduced when experimental equipment is carried aboard the EOS. It identifies safety guidelines and requirements for eliminating or reducing these hazards.

Part 3 discusses the applicability and utility of various means of emergency crew transfer between a disabled and a rescuing vehicle.

Volume III: Appendices is in two parts, each devoted to an individual subtask. Part 1 contains detailed supporting analysis and backup material for Subtask 1, and Part 2 contains similar material for Subtask 2. Volume III is of interest primarily to the technical specialist.

Since the reader is not necessarily interested in all three subtasks, each part of Volumes II and III is a separate document.

All calculations were made using the customary system of units, and the data are presented on that basis. Values in the International System of Units (SI) are also indicated.

Subtask 1 was completed prior to the interest in a parallel-burn Space Shuttle configuration with a solid motor Booster and an expendable Orbiter propellant tank. Moreover, the reports were completed before the Space Shuttle RFP was issued and the Shuttle development contract was awarded. Publication of the Subtask 1 reports was, however, delayed until appropriate information on the parallel-burn Space Shuttle configuration could be developed and added to the Subtask 1 reports.

The Advanced Missions Safety Task was sponsored by NASA Headquarters and managed by the Advanced Missions Office of the Office of Manned Space Flight. Mr. Herbert Schaefer, the study monitor, provided guidance and counsel that significantly aided the effort.

ACKNOWLEDGEMENT

The principal participants in Part 1 of the Advanced Missions Safety Task were:

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1. INTRODUCTION

1.1 BACKGROUND

The need and feasibility of aiding distressed space crews have been of concern since the initiation of manned space flight. Numerous studies of hazards, the resulting emergencies, and escape/rescue concepts have been made (see Reference 1). The preferred solution to space emergencies is self-help. If this is not feasible, however, external help is required. Additional studies, therefore, have also been made concerning in-space operations that may be required to rescue a crew from a disabled spacecraft (see Reference 2).

Present plans place complete dependence on the Earth Orbit Shuttle (EOS) for putting men into space and returning them from earth orbit. Consequently, the EOS can be expected to play a major role in any space rescue operation, and its rescue mission utility is of obvious interest.

The Aerospace Corporation has been actively involved in studying both manned space flight safety and EOS performance. Because of that experience, the Aerospace Corporation was selected to perform this study.

1.2 STUDY OBJECTIVES

The objectives of the study were:

(1) Assess EOS Orbiter rescue capability and utility.
(2) Explore the feasibility of extending this rescue capability.

1.3 STUDY SCOPE

Low earth orbit (LEO), geosynchronous orbit, and lunar rescue missions were examined. A range of EOS design parameters such as payload capability, cargo bay volume, staging velocity, Orbiter propellant storage, etc., which influence its rescue mission utility, were considered. Characteristics of available EOS designs were employed to provide the desired parameter spectrum.

Three EOS performance augmentation techniques were examined:

(1) Increased propellant loading (cargo bay tank)
(2) On-orbit refueling
(3) EOS-launched tug

2. REFERENCE MISSIONS

2.1 GENERAL

Basic rescue mission requirements using the EOS as a space rescue vehicle can be established from current planning. When required, a rescue mission would probably involve flight to a region of planned manned space operation. Any accidental diversion of a distressed vehicle from a planned manned flight would be limited to the basic capability of the space vehicle used.

A current NASA mission model (see Reference 3) provided the basis for the rescue mission objectives used in this study. Additional mission details are discussed in Appendix A, Volume III Part 1.

2.2 MANNED MISSION DESCRIPTIONS

2.2.1 Low Earth Orbit (LEO)

LEO payloads fall into four general categories. These categories, their orbital altitudes (circular), inclination, launch site, and round-trip ΔV, referenced to 100 nmi (185 km)* and inclination of interest, are tabulated below.

Mission	Launch Site	Altitude		ΔV	
		nmi/55°	km/55°	kft/s	m/s
Space Station	ETR	270	500	1.5	460
Laboratories	ETR	350	650	2.0	610
Earth Physics Satellites	WTR	400	740	2.4	730
Earth Observation Module	WTR	100	185	0.25	76

The laboratories and satellites are unmanned but involve manned servicing or retrieval operations. The space station and the module are manned.

*SI unit values in parentheses are approximate

2.2.2 Geosynchronous Orbit

A synchronous earth orbit space station represents the only high energy earth orbit mission included in advanced manned space planning. Its altitude and round-trip ΔV characteristics are as follows:

Synchronous Space Station	19,323 nmi/0° (35,802 km)	ETR	28.5 kft/s (8.7 km/s)

The round-trip ΔV is referenced to 100 nmi (185 km) and 28.4° inclination.

2.2.3 Lunar Orbit

Manned lunar operations will be based on an orbiting station having these altitude and round-trip ΔV characteristics:

Orbiting Lunar Station	60 nmi/90° (110 km)	28.8 kft/s (8.8 km/s)

In this case, the round-trip ΔV is referenced to 100 nmi (185 km) and 31.5° inclination.

2.3 MISSION ΔV AND DURATION

A summary of the mission round-trip ΔVs and the duration of round-trip transportation is tabulated below. All values are referenced to a 100-nmi (185-km) earth orbit as the initial and final condition.

Low Earth Orbit	2.4 kft/s (0.73 km/s)	~0.5 days
Geosynchronous Orbit	29 kft/s (8.85 km/s)	1 - 3 days
Lunar Orbit	29 kft/s (8.85 km/s)	3 - 7 days

It can generally be assumed that a one-way trip involves half the required ΔV and takes half of the time indicated, except for multipass lunar returns.

In general, the lunar rescue mission will represent a more difficult requirement than the geosynchronous orbit rescue mission. The ΔV needed for lunar orbit departure is less than that required for transearth injection from geosynchronous orbit. The earth orbit injection ΔV, however, is greater. The net result is that approximately the same total ΔV is required for either mission. Both mission duration and earth return velocity are greater for the lunar mission. If a lunar capability is available, then a geosynchronous mission should also be possible. Shuttle capability, therefore, was compared only to the lunar rescue mission requirement.

3. SHUTTLE CONFIGURATIONS

3.1 GENERAL

Four EOS configurations were selected for analysis. Designs were utilized for which the basic characteristics and performance were readily available in late 1971. With NASA concurrence, specific selection was made to provide a spectrum of EOS Orbiter design parameters which might influence EOS rescue mission capability. The parallel-burn solid-motor-boosted Space Shuttle was not included as one of the four reference configurations. Interest in this configuration developed after this study was completed, and its characteristics are given separately in Appendix J of Volume III, Part 1.

Characteristics of the four reference Shuttles examined are listed in Table 3-1 and corresponding line drawings are given in Figure 3-1. The emphasis in Table 3-1 is on the Orbiter only, because the booster which is also reusable is obviously sized to accommodate an Orbiter with the tabulated characteristics.

3.2 DESCRIPTION

3.2.1 Configuration A

Configuration A is an integral-tank Orbiter design characteristic of Phase B Shuttle design concepts (see Reference 4). The payload bay is 60 ft (18.3 m) long and 15 ft (4.6 m) in diameter, and the system is designed to deliver a 40 klb (18.2 t) payload to a 50 × 100 nmi (90 × 185 km) polar orbit. The Orbiter is staged at ~11 kft/s (3.4 km/s) actual velocity relative to earth, and burns H_2/O_2 in both the main and OMS engines. The OMS system capacity is 2 kft/s (610 m/s).

3.2.2 Configuration B

Configuration B is a drop-tank Orbiter design characteristic of the Mark II Shuttle design concept (see Reference 5). The payload bay is 60 ft (18.3 m) long and 15 ft (4.6 m) in diameter, and the system is designed to deliver a 40 klb (18.2 t) payload to a 50 × 100 nmi (90 × 185 km) polar orbit. The

Table 3-1. Reference Shuttles

Design Specifications	A Fully Reusable	B Drop Tank	C Drop Tank	D Drop Tank
*Payload, klb (t)	40 (18.2)	40 (18.2)	25 (11)	10 (4.5)
Orbit, nmi (km)	polar 50 × 100 (90 × 185) once around abort	polar 50 × 100 (90 × 185)	polar 50 × 100 (90 × 185)	polar 100 × 100 (185 × 185)
ΔV, ft/s (m/s)	1000 (305)	650 (200)	650 (200)	850 (260)
Cargo Bay, ft (m) D × L	15 × 60 (4.6 × 18.3)	15 × 60 (4.6 × 18.3)	12 × 40 (3.7 × 12)	10 × 20 (3 × 6)
Staging Velocity (actual), kft/s (km/s)	11 (3.3)	7 (2.1)	7 (2.1)	7 (2.1)
Orbiter Dry Weight, klb (t)	228 (104)	130 (59)	108 (49)	46 (21)
Drop Tank Weight	--	35 (16)	31 (14)	23.4 (11)
Propellant Weight	518 (235)	704 (320)	603 (274)	237 (108)
Orbiter Propulsion - Main				
Propellants	H_2/O_2 Hi Pc	H_2/O_2 Hi Pc	H_2/O_2 Hi Pc	H_2/O_2 Hi Pc
Thrust, klb (10^3 N)	2 @ 632 (2820)	4 @ 306 (1360)	3 @ 306 (1360)	2 @ 242 (1080)
Orbiter Propulsion - OMS				
Propellants	H_2/O_2	Storable	Storable	H_2/O_2
Thrust, klb (10^3 N)	2 @ 15 (67)	2 @ 3.5 (16)	2 @ 3.5 (16)	1 @ 12 (53)
ΔV Capacity kft/s (m/s)	2 (610)	1 (305)	1 (305)	0.85 (260)

*no airbreather

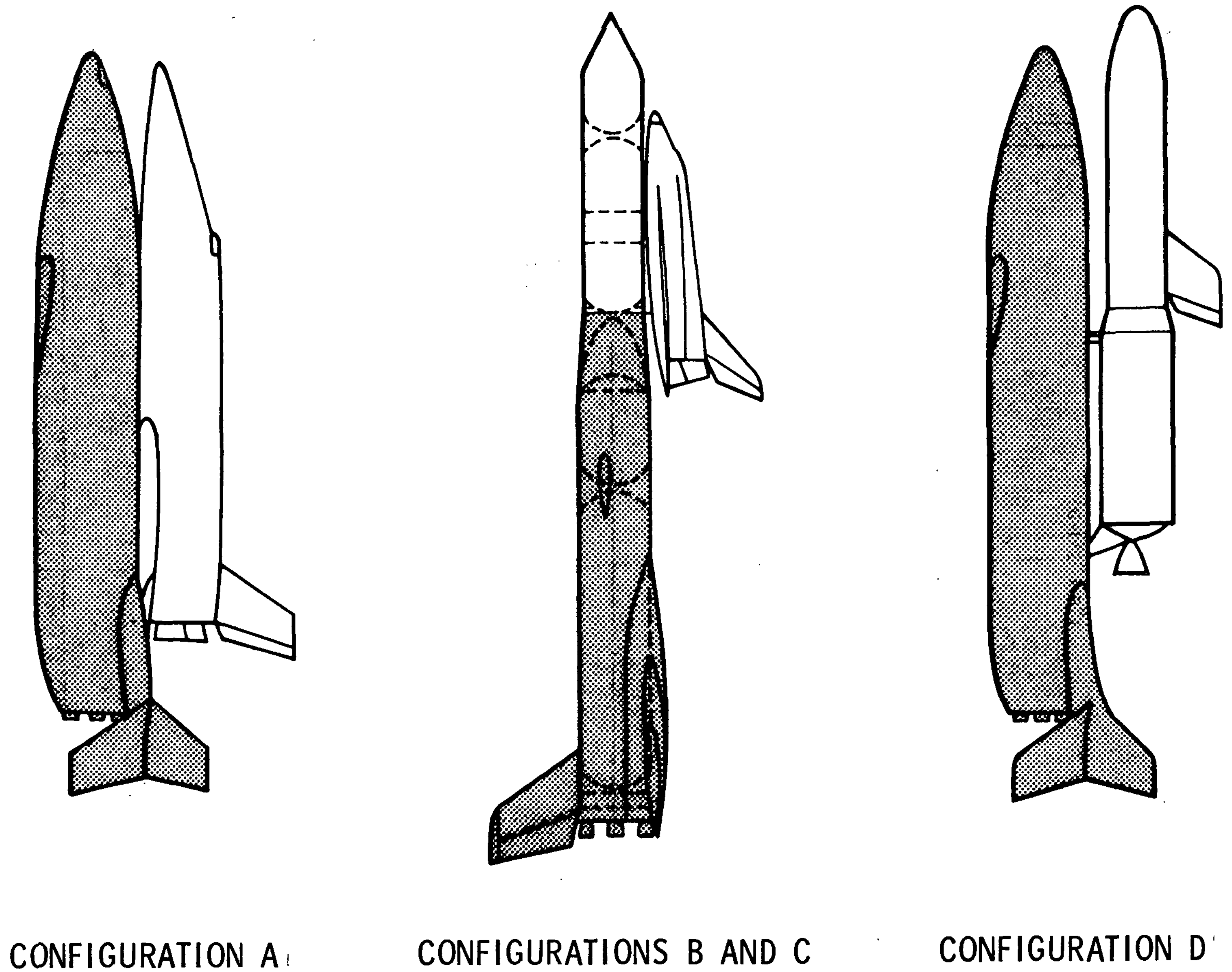

Figure 3-1. Shuttle Configurations

Orbiter is staged at ~7 kft/s (2.1 km/s) actual velocity relative to earth, and burns H_2/O_2 in the main engines only. The OMS system uses storable propellants and is sized to deliver 1 kft/s (305 m/s). (The Orbiter size, weight, and design characteristics for the parallel-burn Shuttle configuration are generally similar to this configuration.)

3.2.3 Configuration C

Configuration C is also a drop-tank Orbiter design similar in concept to Configuration B. The payload bay, however, is only 40 ft (12 m) long and 12 ft (3.7 m) in diameter. Also, the system is designed to deliver only a 25 klb (11 t) payload to a 50 × 100 nmi (90 × 185 km) polar orbit. In this case, also, the Orbiter is staged at ~7 kft/s (2.1 km/s), burns H_2/O_2 in the main engines and storable propellants in the OMS engines, and has an OMS capacity of 1 kft/s (305 m/s).

3.2.4 Configuration D

Configuration D is a drop-tank Orbiter based on an IDA design of a mini-Shuttle (see Reference 6). The payload bay is 20 ft (6 m) long and only 10 ft (3 m) in diameter. The system is designed to deliver a 10 klb (4.5 t) payload into a 100 × 100 nmi (185 × 185 km) polar orbit and is also staged at ~7 kft/s (2.1 km/s). Cryogenic propellants (H_2/O_2) are burned in both the main and OMS engines. The OMS system capacity is only 0.85 kft/s (260 m/s).

3.3 PARAMETERS EXAMINED

The four configurations described in section 3.2 provided a range in the following Orbiter design and operating parameters:

(1) Payload weight
(2) Payload bay dimension
(3) Orbiter staging velocity
(4) Main tank configuration (integral or drop-tank)
(5) OMS ΔV capacity
(6) OMS propellant (storable or cryogenic)

4. BASIC PERFORMANCE CAPABILITY

4.1 GENERAL

A summary plot of the 100 nmi (185 km) circular altitude basic payload capability for all four configurations is given in Figure 4-1. In all cases the Orbiter is placed in a 50 x 100 nmi (90 x 185 km) orbit by the main engines. The OMS system carries only sufficient fuel to circularize the orbit at 100 nmi (185 km) and then to deorbit. Excess OMS tank capacity is not utilized but, instead, is replaced with payload.

It should be noted that under current Orbiter specifications, main engine restart is not required. Thus, although the current engine design is capable of restart,* its installation in the Orbiter will not be provided with this capability. Once the main Orbiter engines are shut down, therefore, the OMS system represents the only ΔV source for on-orbit, intermittent application (all configurations).

Detailed performance curves are included in Appendix B of Volume III Part 1. The effects of inclination, circular orbital altitude, and OMS tank capacity limit on payload are illustrated there for all four reference EOS configurations. Similar data for the parallel-burn Shuttle configuration are included in Appendix J, Volume III, Part 1.

4.2 RESCUE MISSION ΔV

A rescue mission does not necessarily require the maximum EOS payload capability. The weight of a rescue module carried into orbit by a space rescue vehicle (SRV) or, alternatively, the weight of special rescue equipment for such a vehicle is in the order of 10 klb (4.5 t) (see Section 8).

On this basis two payload weights were considered in all rescue mission performance estimates, 10 klb (4.5 t) and 0. The latter value represents the maximum possible EOS capability.

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*according to the engine contractor

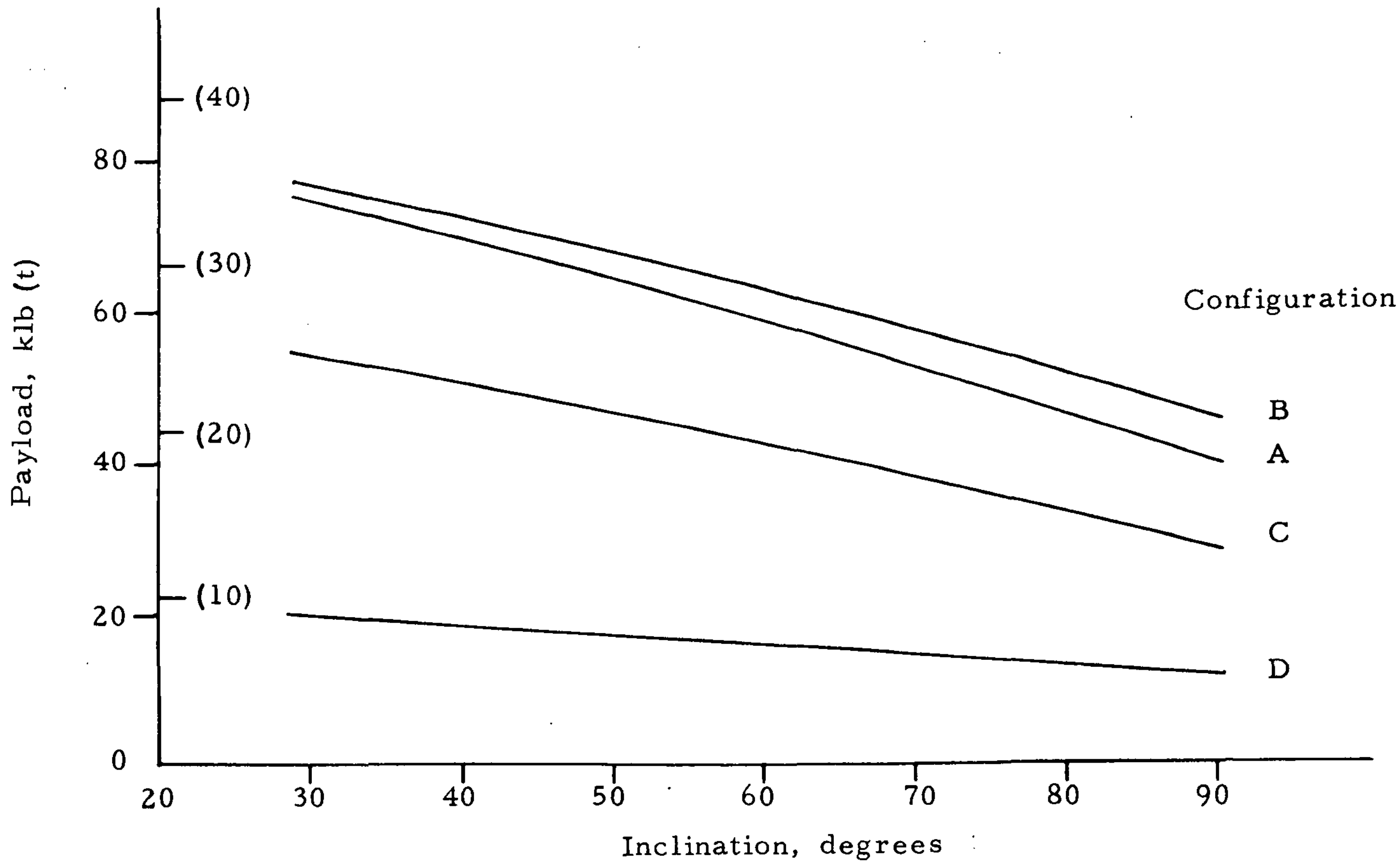


Figure 4-1. Shuttle Payload Capability, 100 nmi (185 km) Circular Altitude

A tabulation of the on-orbit ΔVs provided by each of the four Shuttles with 0 and 10 klb (4. 5 t) payloads is given in Table 4-1. Values are listed for two versions of Configuration B - the standard version described in paragraph 3. 2. 2, and an augmented OMS version. The latter configuration is equipped with a 2 kft/s (610 m/s) OMS system and is double the standard configuration.

The available OMS ΔVs in excess of circularization and deboost requirements given in Table 4-1 are generally established by the OMS tank capacity. At the higher altitudes and inclinations, however, a limit on the Shuttle weight-lifting capability will be encountered. In such cases, OMS propellant is off-loaded in order to carry the full rescue payload weight, thus causing a lower available ΔV than that established by the OMS tank capacity.

Detailed rescue mission ΔV curves for the four basic Shuttle configurations are presented in Appendix B, Volume III, Part 1. Similar data for the parallel-burn configuration are included in Appendix J. Volume III, Part 1.

4. 3 SUMMARY

Configurations A, B, and C are able to reach all four low earth orbit destinations listed in paragraph 2. 2. 1 with rescue mission payloads (see Appendix B, Volume III Part 1). Whether sufficient ΔV remains for on-orbit maneuvering, should any be required after the mission destination is reached, depends upon the specific destination and EOS configuration.

The rescue mission capability of the basic parallel-burn Space Shuttle configuration is very similar to that of Configuration B. (See Appendix J, Volume III, Part 1.)

Table 4-1. Baseline Performance Summary

	Available ΔV,* ft/s (m/s)									
Configuration	A		B (std)		B (aux OMS)		C		D	
Payload, klb (t)	0	10 (4.5)	0	10 (4.5)	0	10 (4.5)	0	10 (4.5)	0	10 (4.5)
100 nmi (185 km)										
(28.4, 55, 90°)	1850 (565)	1760 (536)	1060 (323)	970 (296)	2180 (664)	2030 (619)	850 (259)	760 (232)	890 (271)	690 (210)
300 nmi (555 km)										
28.4, 55°	1340 (409)	1260 (384)	550 (168)	460 (140)	1670 (509)	1530 (466)	340 (104)	240 (73)	370 (113)	170 (52)
90°	1340 (409)	1220 (372)	550 (168)	460 (140)	1670 (509)	1230 (375)	340 (104)	240 (73)	370 (113)	170 (52)
500 nmi (925 km)										
28.4, 55°	730 (222)	650 (198)	-- --	-- --	1070 (326)	920 (280)	-- --	-- --	-- --	-- --
90°	710 (216)	140 (43)	-- --	-- --	930 (284)	260 (79)	-- --	-- --	-- --	-- --
700 nmi (1295 km)										
28.4°	150 (46)	60 (18)	-- --	-- --	490 (149)	340 (104)	-- --	-- --	-- --	-- --
55°	150 (46)	60 (18)	-- --	-- --	490 (149)	300 (91)	-- --	-- --	-- --	-- --
90°	-- --	-- --	-- --	-- --	10 (3)	-- --	-- --	-- --	-- --	-- --

*in excess of circularization and deboost requirements

5. INCREASED PROPELLANT LOADING

5.1 DESCRIPTION

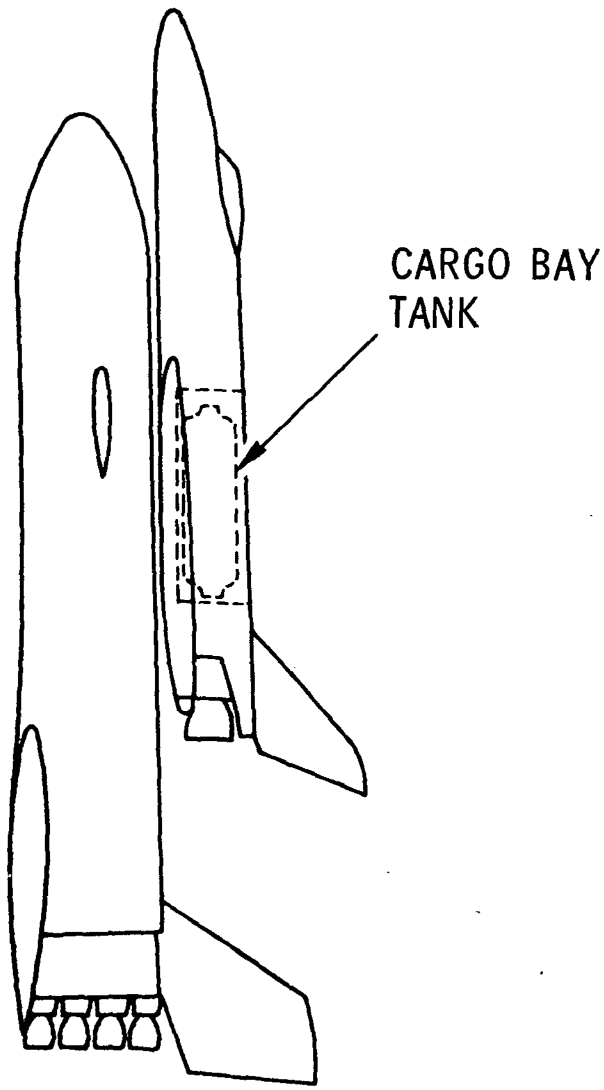
Increased Orbiter propellant loading (Figure 5-1) is an obvious method for improving the performance of a specific EOS. This is accomplished simply by placing an additional propellant tank in the cargo bay. The tank is sized to fill all available space in the cargo bay. Propellant is loaded on the launch pad, the quantity of propellant being adjusted to maintain the total weight in the cargo bay within EOS performance limits. Since the rescue mission payload (see section 4.2) is nominally 10 klb (4.5 t), considerable Orbiter capability and cargo bay space generally remain available for the added tank, plumbing, and propellant.

The approach described below for enhancing the Space Shuttle performance is applicable to all Shuttle configurations. Performance results for the four reference Shuttle configurations are given in section 5.3. Results for the parallel-burn Space Shuttle configuration are included in Appendix J, Volume III, Part 1.

5.2 METHOD OF APPROACH

The tank and propellant are considered as Orbiter payload. Removable tanks are provided, and they are mounted to cargo bay hard points. Tank weights include allowance for superinsulation designed for seven-day cryogenic propellant storage as well as tank pressurization and propellant transfer equipment. Mating with the Orbiter propellant system is via the payload fluid interface panel.

It was assumed that, except for a 20 ft (6 m) length reserved for the 10 klb (4.5 t) rescue-mission payload, the tank fills all available cargo bay volume. However, propellant is loaded only to the limit of EOS cargo capability.



- ADDITIONAL PROPELLANT LOADED ON PAD
- REMOVABLE CARGO BAY TANK
  - — PROVIDES OWN PRESSURIZATION
  - — CRYOGENIC STORAGE, 7 days

Figure 5-1. Increased Propellant Loading Schematic

The large tank size and superinsulation were assumed so that the same tank could also be used for orbital refueling (see Section 6).

All cargo bay propellant is burned through the Orbiter OMS engines. The propellant utilization sequence assumed is:

(1) Empty main tanks (main engines)
(2) Jettison main tanks (if appropriate)
(3) Burn OMS tank fuel (OMS engines)
(4) Burn cargo bay fuel (OMS engines)

Detailed tank weights are given in Appendix C, Volume III Part 1.

5.3 PERFORMANCE

The on-orbit ΔVs available with increased propellant loading on the launch pad are given in Figure 5-2 as a function of circular orbit altitude. Data are presented for each EOS configuration at inclinations of 28.4°, 55°, and 90° and for payloads of 0 and 10 klb (4.5 t). The curves were determined by assuming that a direct ascent is made to an elliptic orbit whose apogee is the altitude of interest. Except for the rescue mission payload indicated and the propellant needed to circularize and later deorbit directly, all remaining payload capability is devoted to propellant for on-orbit ΔV.

With increased propellant loading, OMS tank capacity no longer limits the available on-orbit ΔV. Instead, the available ΔV is now established by the Orbiter weight-lifting capability limit or the cargo bay tank capacity limit, whichever occurs first. For the conditions plotted in Figure 5-2, only the Configuration C case for 28.4° inclination and 10 klb (4.5 t) payload is partially limited by the volume of the cargo bay tank. All other cases are entirely weight-limited.

When data from Figure 5-2 and Figure B-5 of Appendix B, Volume III Part 1 are cross-plotted, available on-orbit ΔV comparisons can be made between the reference and augmented EOS configurations. Such comparisons are shown in Figure 5-3 for a 100 nmi (185 km) circular orbit. In

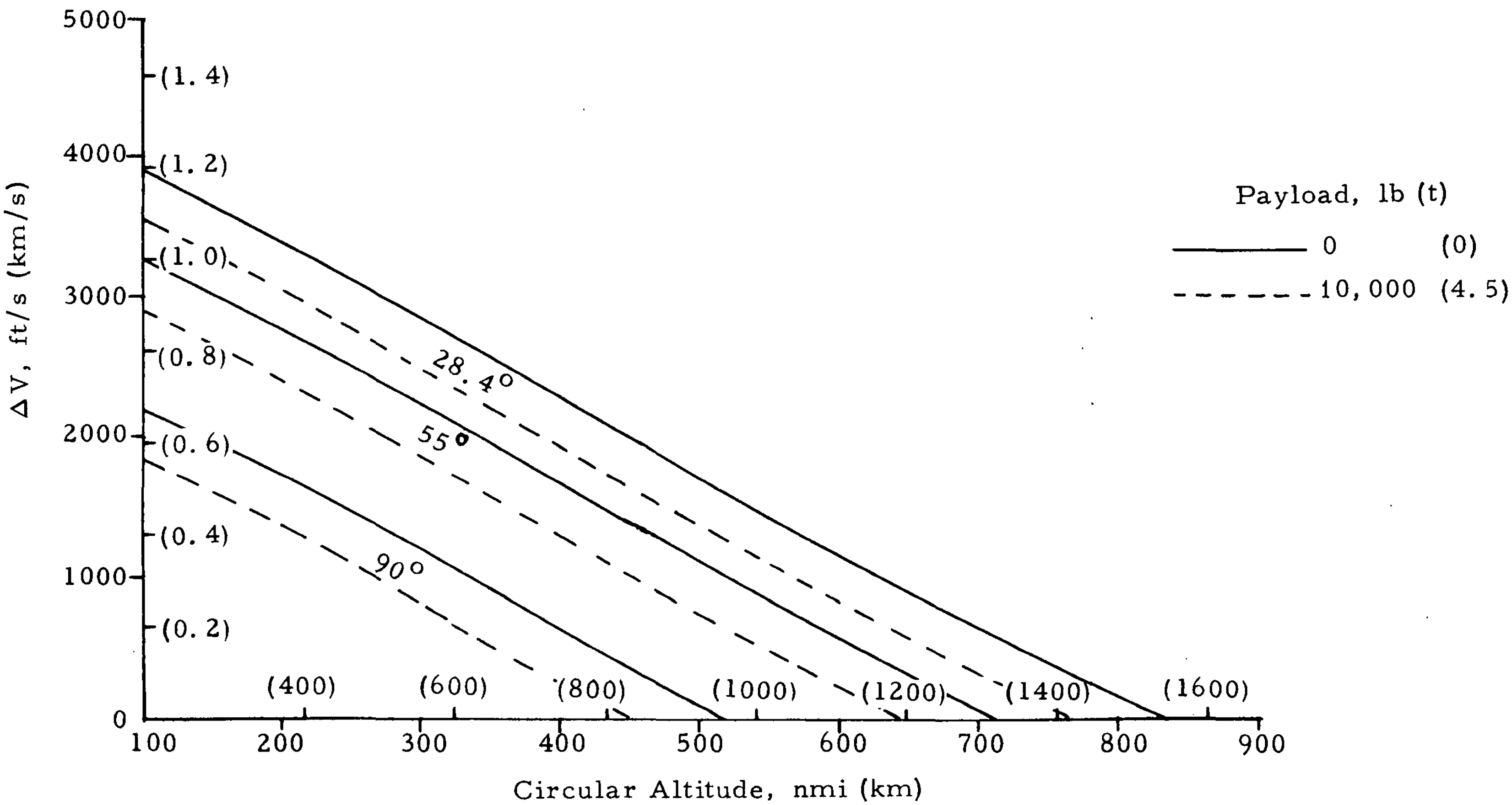


Figure 5-2a. Configuration A, On-Orbit ΔV with Increased Propellant Loading (Direct Reentry)

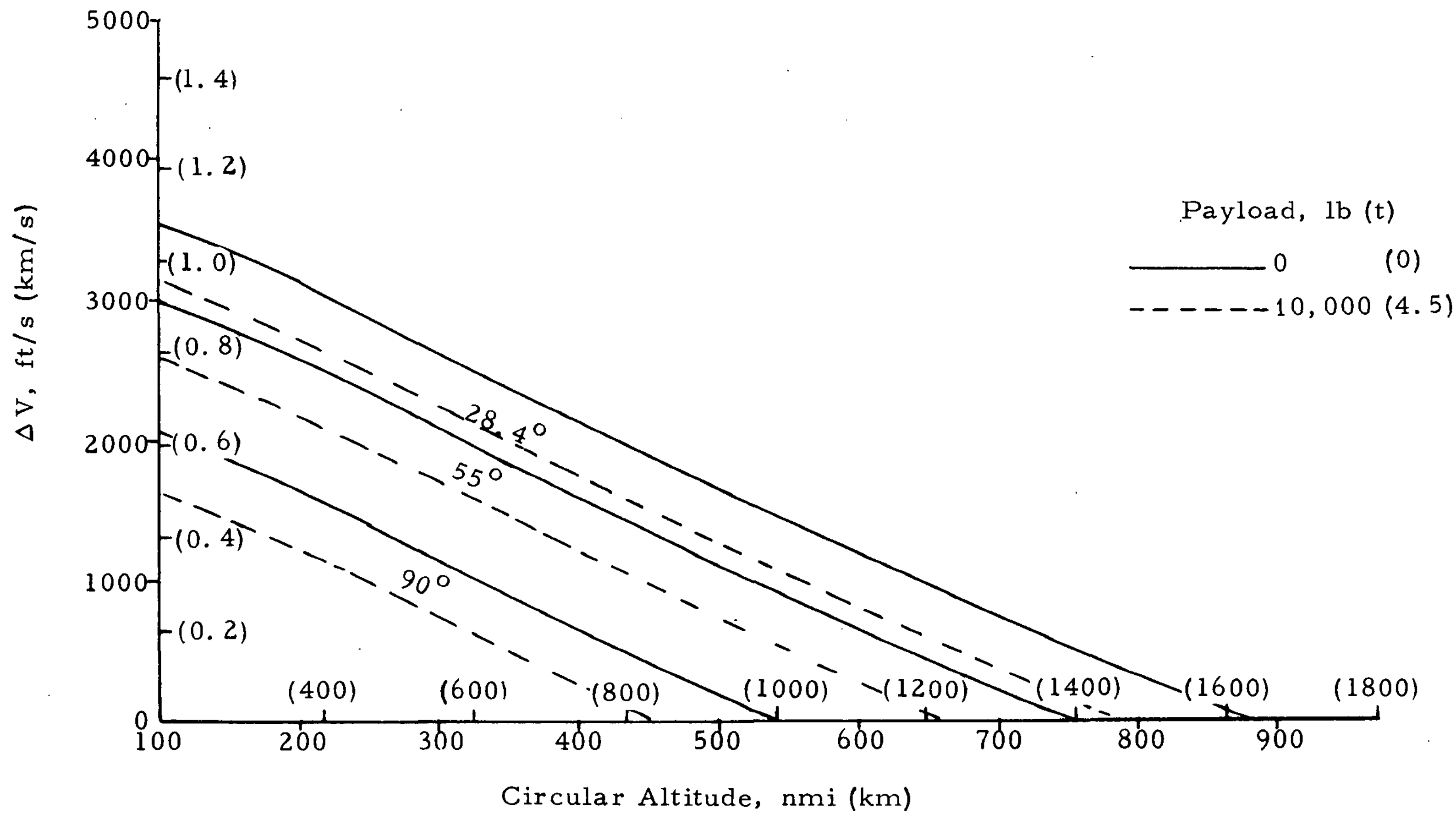


Figure 5-2b. Configuration B, On-Orbit ΔV with Increased Propellant Loading (Direct Reentry)

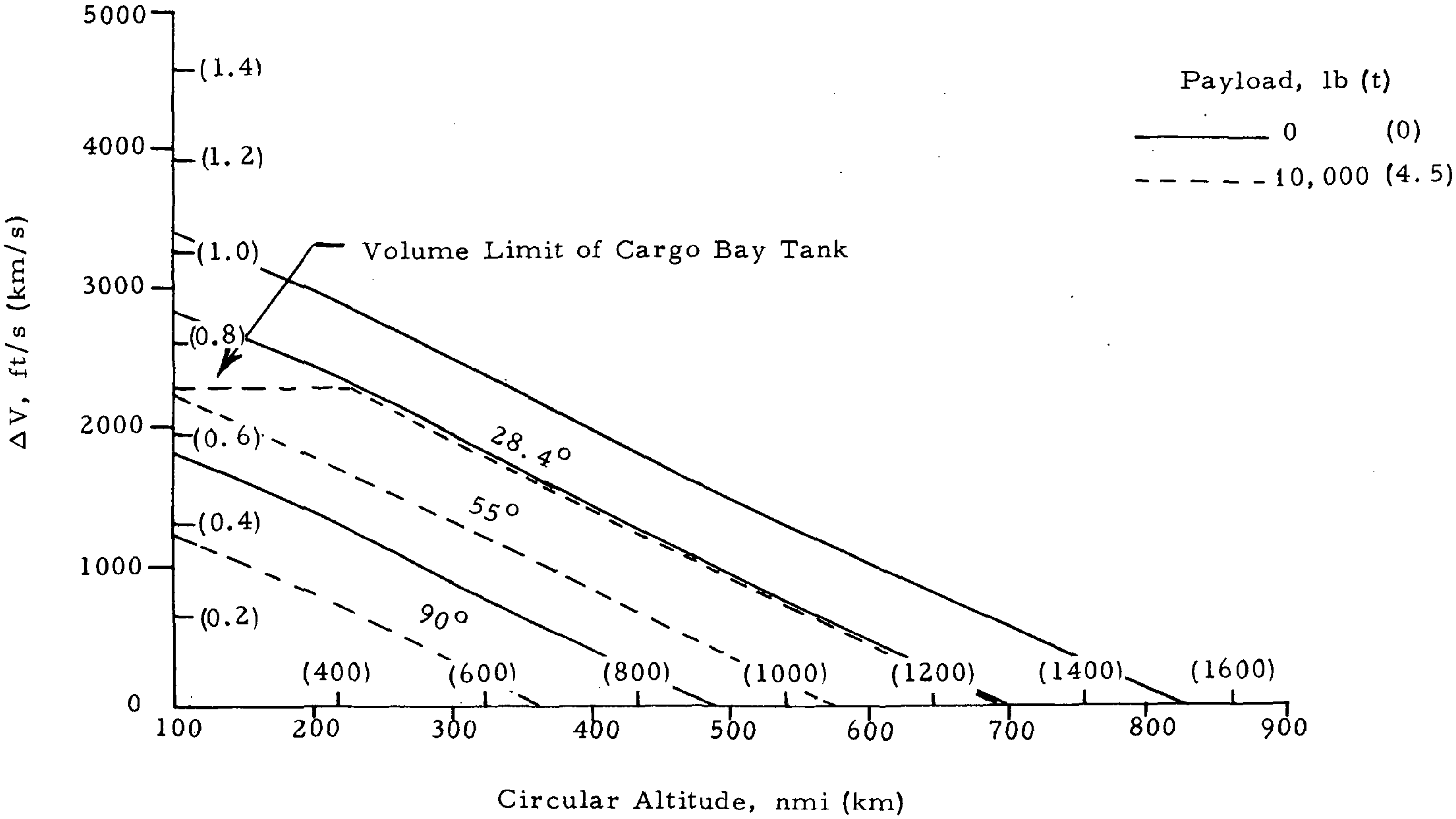


Figure 5-2c. Configuration C, On-Orbit ΔV with Increased Propellant Loading (Direct Reentry)

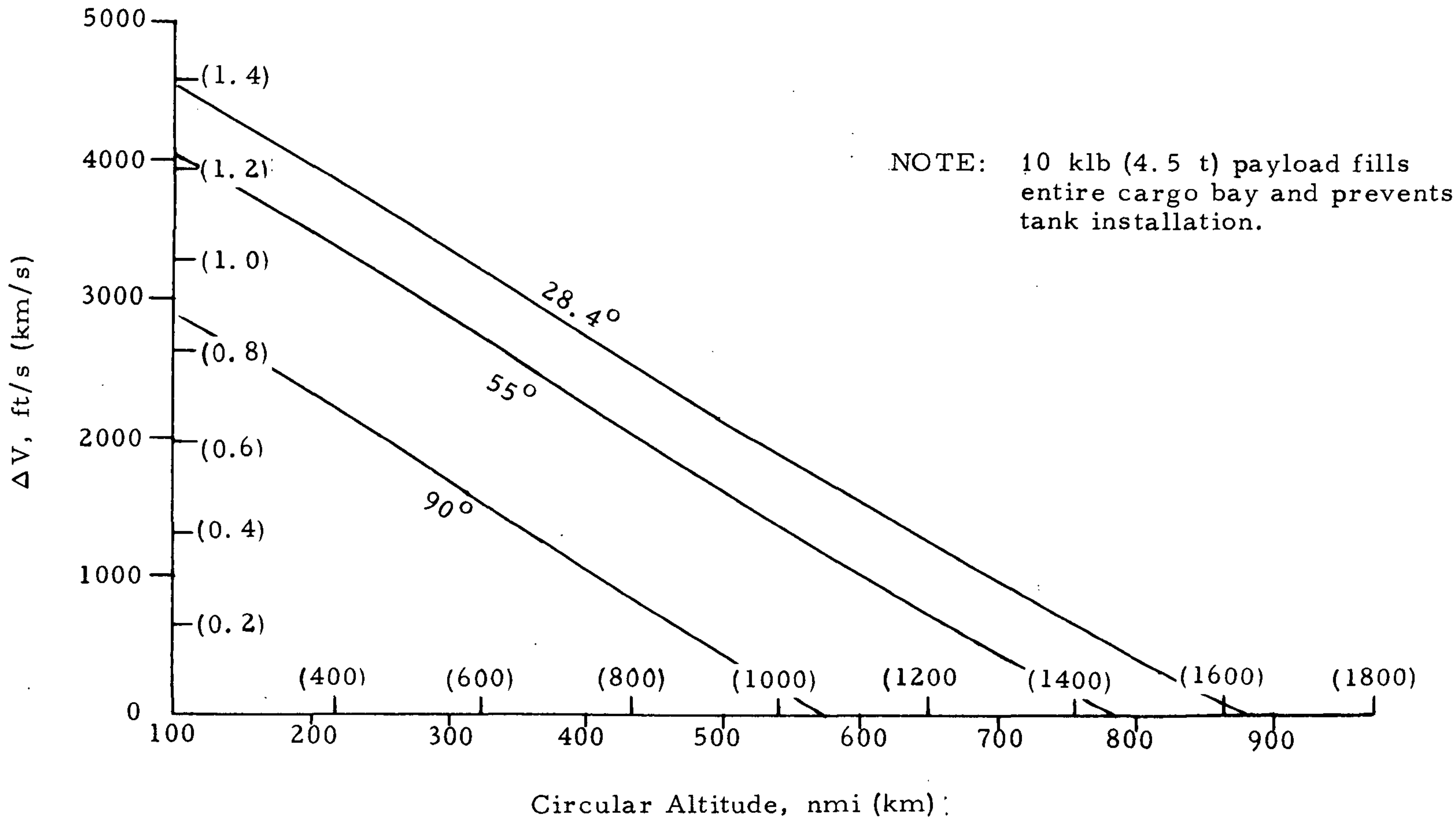


Figure 5-2d. Configuration D, "0" Payload Case - On-Orbit ΔV with Increased Propellant Loading (Direct Reentry)

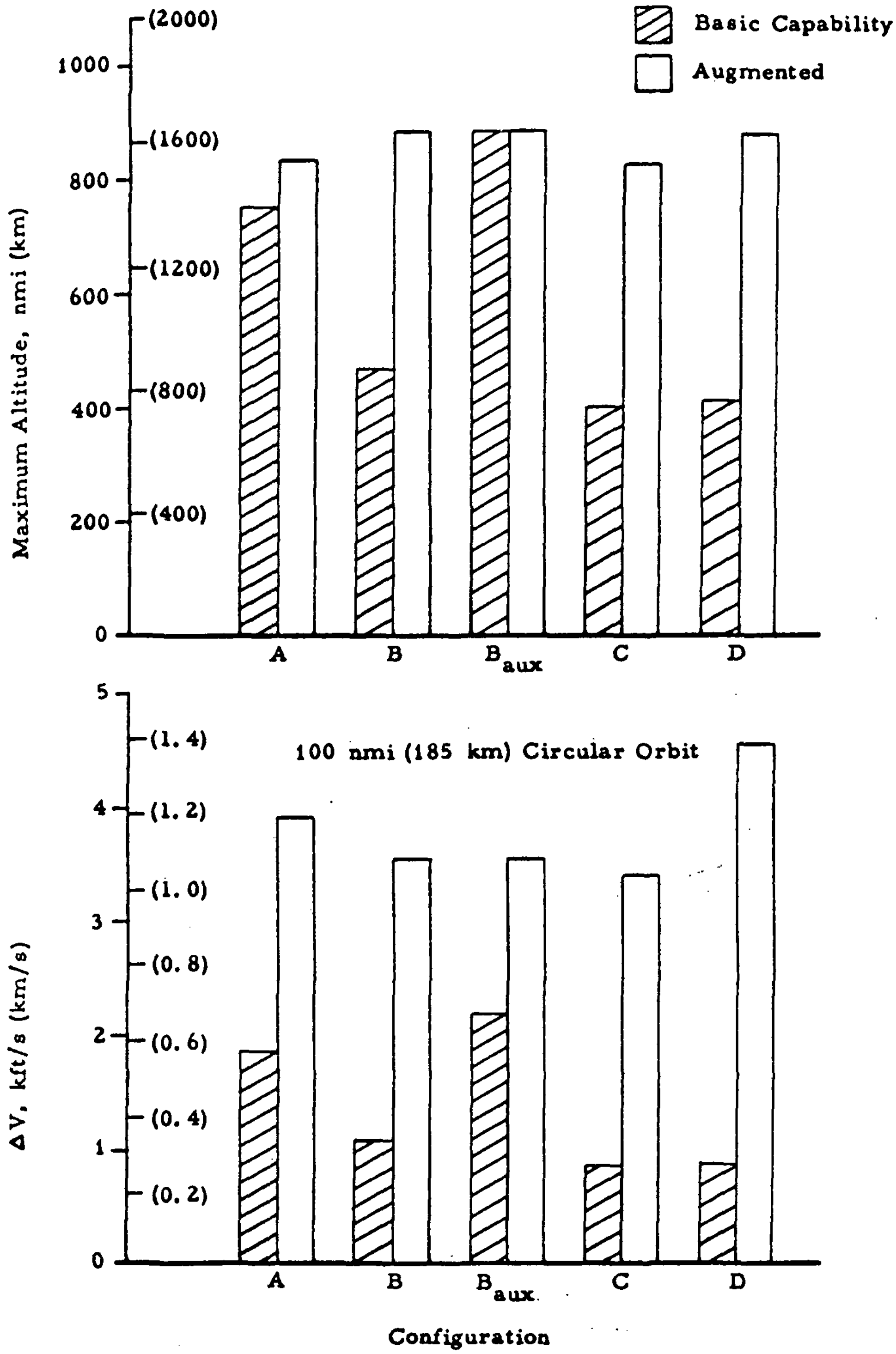


Figure 5-3a. "0" Payload, 28.4° Inclination - Shuttle Performance Comparison with Increased Propellant Loading (Direct Reentry)

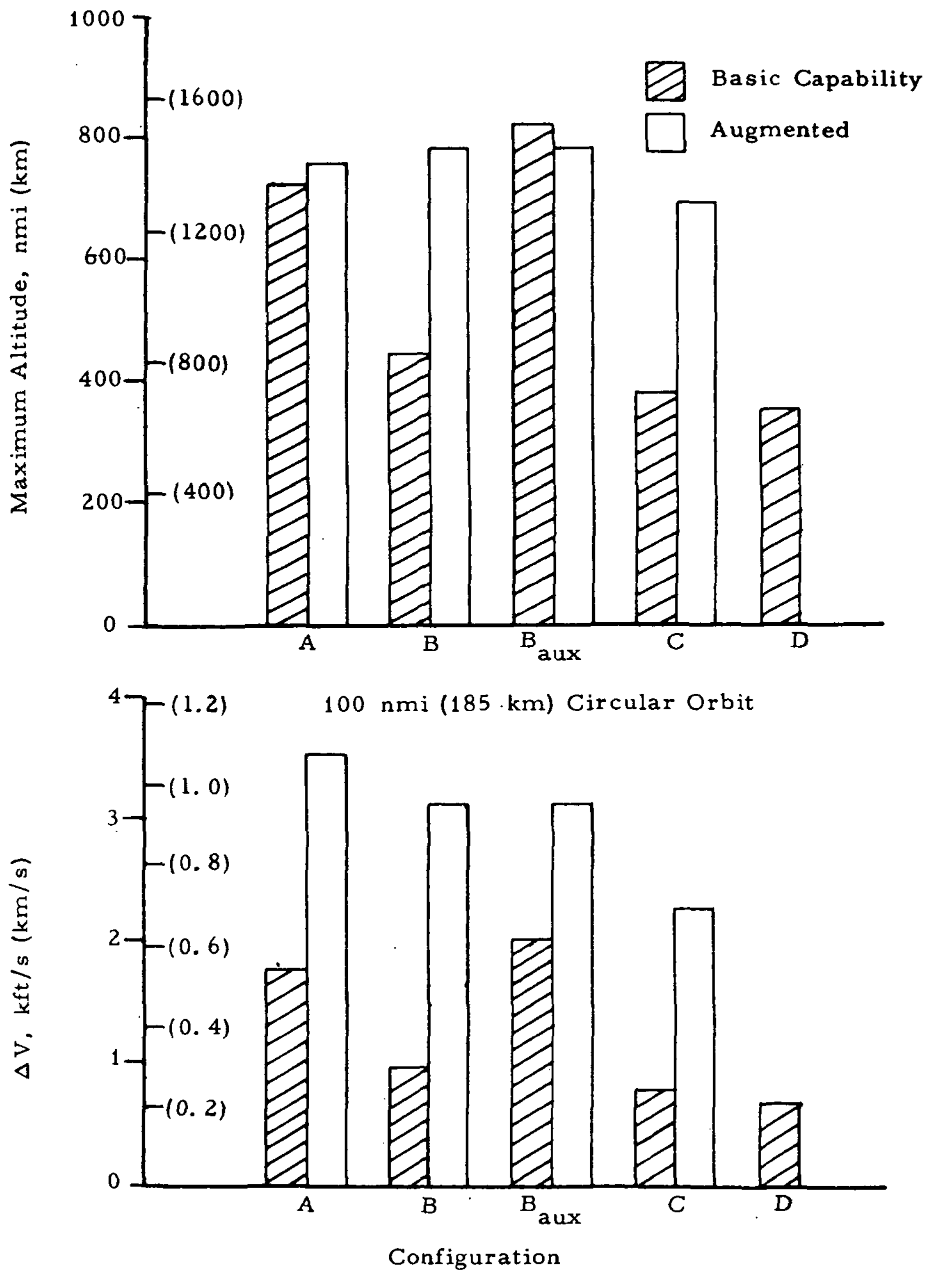


Figure 5-3b. 10 klb (4.5 t) Payload, 28.4° Inclination - Shuttle Performance Comparison with Increased Propellant Loading (Direct Reentry)

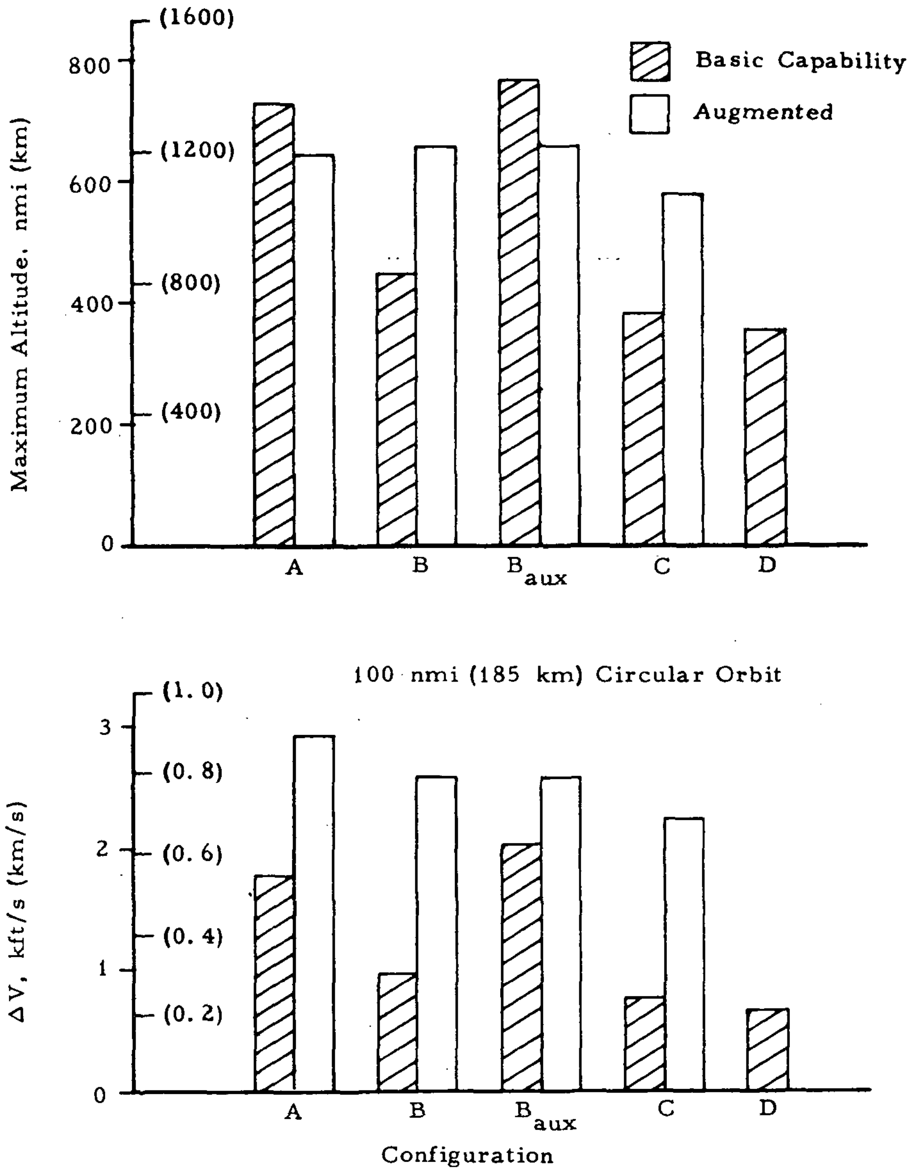


Figure 5-3c. 10 klb (4.5 t) Payload, 55° Inclination - Shuttle Performance Comparison with Increased Propellant Loading (Direct Reentry)

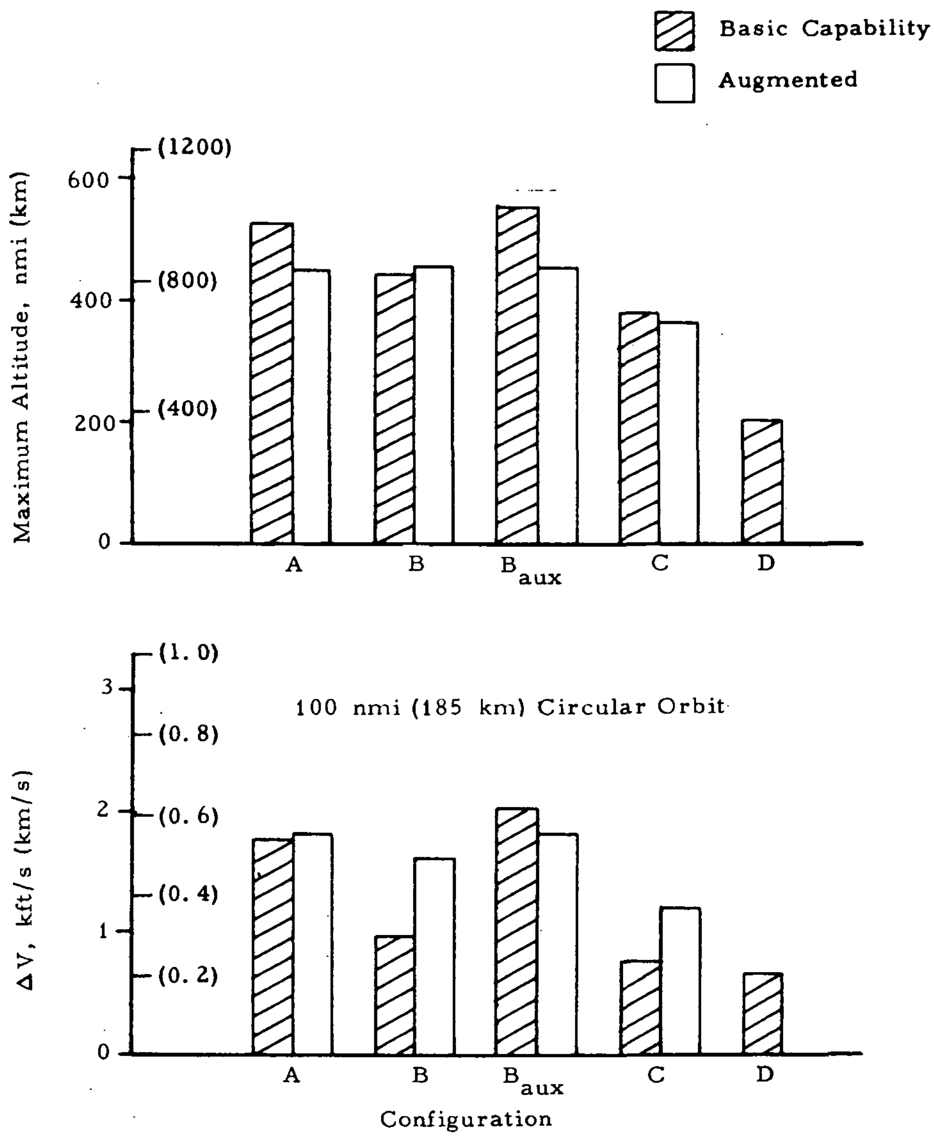


Figure 5-3d. 10 klb (4.5 t) Payload, 90° Inclination - Shuttle Performance Comparison with Increased Propellant Loading (Direct Reentry)

addition to the results for the standard Configuration B, results for its augmented OMS version are also shown.

The altitude at which the on-orbit ΔV becomes zero is the maximum altitude which can be attained carrying the rescue mission payload. A maximum altitude comparison between the reference and augmented EOS configurations is also included in Figure 5-3.

The effectiveness of increased propellant loading by means of a cargo bay tank depends not only on Shuttle configuration but on the specific rescue mission as well. If the mission imposes a requirement approaching the limit of the basic EOS capability, little can be gained with this approach. In fact, when allowance is made for the weight of the cargo bay tank and its related plumbing, the augmented Shuttle performance may be less than previously available with the basic EOS. The 10 klb (4.5 t), 90° case for Configuration B_{aux} falls into this category. It is noteworthy that even at lower inclinations increased propellant loading does little to improve the maximum altitude capability of this Shuttle configuration.

The primary benefit of this method occurs at low orbit altitudes and inclinations, where the available ΔV increase for Configurations A, B, B_{aux}, and C falls between 1 to 2 kft/s (0.3 to 0.6 km/s). Configuration D can accept a cargo bay tank for only the "0" payload case, since the 20 ft (6 m) length assumed for the 10 klb (4.5 t) payload fills the entire cargo bay. Thus, increasing the propellant loading of Configuration D with a cargo bay tank in order to augment its rescue mission capability is of doubtful utility.

5.4 TECHNICAL FEASIBILITY

Although some Orbiter modifications may be required (OMS plumbing, for example), no major Orbiter changes are anticipated.

Tanks designed for on-orbit cryogenic storage, self-pressurization, and fuel transfer are feasible and are considered to be current state of the art.

A removable tank installation for the Orbiter cargo bay is consistent with EOS payload handling and structural support philosophy. On-pad propellant loading uses conventional fueling procedures and precautions. Return of the empty tank to earth via the Orbiter is also a standard operation.

The payload fluid interface panel would be used to facilitate propellant transfer into the OMS system and would also provide basic fill, drain, vent, and electrical functions.

5.5 SUMMARY

Increased propellant loading generally offers a simple means for improving EOS rescue mission utility in LEO. The cargo bay tank installation and other required Orbiter modifications, as well as the operating mode involved, are considered technically feasible.

Maximum circular-orbit altitude varies little among augmented EOS configurations. At low inclination, the increase in ΔV at 100 nmi (185 km) falls between 1 to 2 kft/s (0.3 to 0.6 km/s) for all configurations. This ΔV increase falls as the inclination is raised and may even become negative.

Configuration B with the standard 1 kft/s (0.3 km/s) OMS capacity appears to be the best candidate for increased propellant loading. Configuration D can be disqualified, since it cannot simultaneously accommodate a 10 klb (4.5 t) rescue payload and a cargo bay propellant tank. Except for D, all augmented configurations can meet the LEO mission destinations referred to in paragraph 2.2.1 with some improvement in ΔV over that available with the basic configurations.

6. ORBITAL REFUELING

6.1 GENERAL DESCRIPTION

Orbital refueling (Figure 6-1) offers a means for obtaining a very large increase in ΔV. The gain with main tank refueling can be even further augmented by adding a cargo bay tank.

The results given in this section assume LEO refueling. Integral main tanks are refueled by propellant transfer from a donor vehicle. Drop tanks can also be refueled via propellant transfer from the donor (the tanks are obviously not jettisoned on the ascent to LEO). An alternate approach allows the drop tanks to be jettisoned during ascent to LEO and provides a fully fueled replacement tank on orbit, where it is mated to the Orbiter. If a cargo bay tank is carried it is assumed to be filled to capacity at refueling, and it is sized according to the rescue payload carried. The cargo bay tank is insulated for space storage. The replacement main tank (if used) can either be a duplicate of the original dropped main tank or a specially developed version for longer propellant storage.

All propellant acquired at the time of refueling, even cargo bay tank propellant, is fired through the Orbiter main engines. The propellant utilization sequence assumed after refueling is:

(1) Empty main tanks (main engines)
(2) Jettison main tanks (if appropriate)
(3) Empty cargo bay tank, when used (main engines)
(4) Burn OMS tank fuel (OMS engines)

Cargo bay tank sizes and weights were assumed to be identical with those used in Section 5 (see Appendix C, Volume III Part 1).

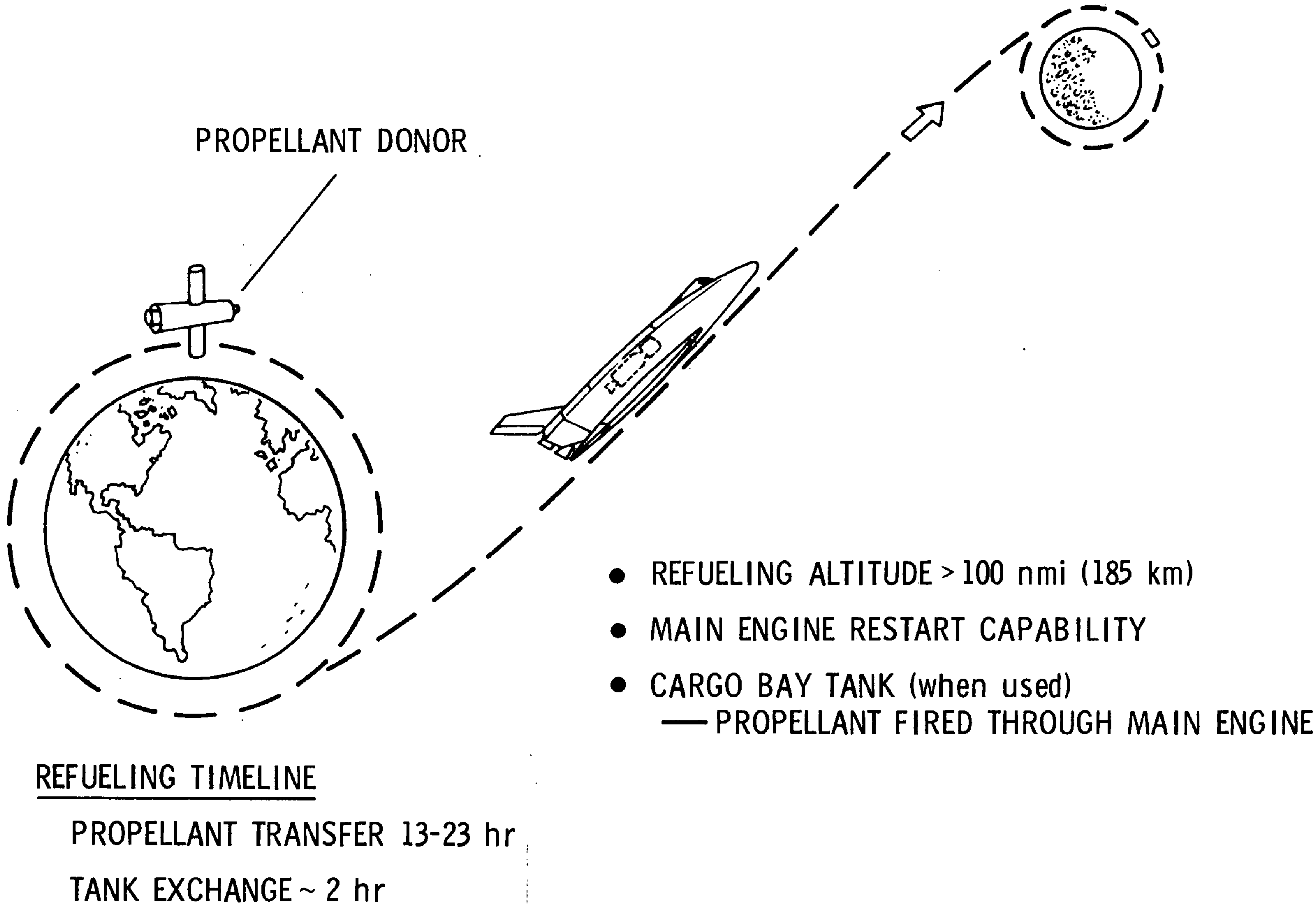


Figure 6-1. On-Orbit Refueling Schematic

6.2 REFUELING PROCEDURES

Means of providing large quantities of propellant in LEO have been examined for Reusable Nuclear Shuttle (RNS) and Chemical Interorbital Shuttle (CIS) applications (see References 7 and 8). For Orbiter application, three general techniques appear to be available. These are the Orbiting Propellant Depot (OPD), dedicated EOS flights, and an Expendable Second Stage (ESS). A discussion of each approach follows.

6.2.1 Orbiting Propellant Depot (OPD)

Typical depot sizes which have been examined for RNS and CIS application are:

	H_2		O_2	
	klb	t	klb	t
Reusable Nuclear Shuttle	430	195	150	68
Chemical Interorbital Shuttle	190	86	1060	481

Configuration B has the largest propellant capacity of the four configurations (see Table 3-1), approximately 60% of the size of the CIS depot. Clearly, large quantities of propellant are involved. The transfer of such large quantities from the OPD to the receiver can involve a significant delay, especially for a time-critical rescue mission.

Timelines were established (see Reference 7) for the procedures involved in transferring propellant from an OPD to receiver space vehicles. Typical timelines for an RNS and a Tug are reproduced in Appendix D, Volume III Part 1. Estimates based on these values were made for the total time required for propellant transfer to Configurations A, B, C, D, a Tug, and a cargo bay tank sized for A and B (see Table 6-1).

Table 6-1. Estimated Propellant Transfer Total Time (simultaneous propellant transfer)

Configuration	LH_2		LO_2		Total Time, hr
	klb	t	klb	t	
A	74	33.6	444	202	22
B	100	45.4	604	274	23
C	86	39.1	517	235	23
D	34	15.5	203	92.2	21
Tug Only	7.8	3.5	47	21.4	20
P/L Bay Tank Only	30.6	13.9	184	83.6	21

Docking, tank preparation, post transfer, and undocking activities occupy over half the time needed for propellant transfer. As a consequence, the estimated total time varies by just a few hours between Orbiter configurations, Tug, and cargo bay tank.

Even if the Tug timeline is only 12 hours, as estimated in Reference 8, instead of the 20 hours estimated in Reference 7, the on-orbit refueling operation will take between half a day and one full day.

A preferred alternate to orbital refueling via propellant transfer is tank exchange. The empty propellant tank is exchanged for a fully fueled tank previously placed in orbit as a part of or refueled from the OPD. This approach is applicable to Orbiter configurations with separable main tanks and/or a removable cargo-bay tank. Although a rendezvous and docking procedure is involved, the total time from hard docking to separation is estimated at less than two hours.

The tanks used in such a procedure will probably require better thermal insulation than the tanks of an Orbiter refueled by propellant transfer. In either case, if remaining main tank propellants are to be used at the mission destination, long-duration missions such as a lunar rescue mission require added tank insulation. The problem is less severe and may not even occur with the cargo bay tank since it is designed for a 7-day propellant storage period.

6.2.2 Dedicated EOS Flights

Without an OPD in which propellant is stored in anticipation of a need, orbital refueling as a means of augmenting Orbiter capability may be impractical. One approach would be to supply propellant from earth with a caravan of EOS flights. Numerous flights in rapid sequence would be required, owing to the large difference between Orbiter payload capacity and its propellant capacity. Assuming no loss due to boil-off, the number of EOS logistic flights needed to refuel the Orbiter in an orbit of 100 nmi (185 km) and 28.4° inclination is given in Table 6-2.

Table 6-2. Orbital Refueling Via EOS Logistic Flights

Config.	P/L Wt.		Tank Wt.		Net Propell. Wt.		Orbiter Propell. Wt.		No. of Flights
	klb	t	klb	t	klb	t	klb	t	
A	75	34	11	5	64	29	518	235	9
B	77	35	11	5	66	30	704	320	11
C	55	25	4.8	2.8	50.2	23	603	274	12
D	20	9.1	1.6	0.7	18.4	8.4	237	108	13

The Orbiter modifications and weights needed to accommodate propellant transfer from an Orbiter donor to an Orbiter receiver have not been considered, but they further reduce the propellant weight carried on each logistic flight. Assuming one logistic vehicle and a two-week vehicle turnaround between flights, 16 weeks are required for a single refueling of Configuration A. The other EOS configurations are proportionally longer. What with the large number of required flights, the small number of complete EOS systems expected to be available, and their required turnaround times, this method of orbital refueling appears unsatisfactory.

6.2.3 Expendable Second-Stage (ESS) Propellant Delivery

A possible method of reducing the large number of EOS logistic refueling flights involves replacing the Orbiter with an ESS (see Reference 9). This ESS is essentially a boosted propellant tank and serves as a propellant source for refilling the empty Orbiter tanks. For a design derived from the S-II stage and carrying main engine propellants in a 6:1 ratio, the propellant weight per flight delivered to a 100 nmi (185 km) orbit at 28.4° inclination is estimated at 177 klb (80 t). These values are based on Reference 9 data and are not applicable to Configuration D.

If no boil-off loss occurs, the number of flights needed to completely refuel Configurations A, B, and C are given in Table 6-3.

Table 6-3. Orbital Refueling Via Expendable Second Stage Logistic Flights

Configuration	No. of ESS Flights
A	3
B	4
C	4

The required number of propellant delivery flights is reduced on the average to one-third the flights with EOS refueling (see paragraph 6.2.2). For a two-week booster stage turnaround, the total refueling time for Configuration A is reduced to 4 weeks (6 weeks for Configurations B and C). These flight frequencies and total refueling times are considered unacceptable for a rescue mission.

6.3 PERFORMANCE

The Orbiter ΔVs available after LEO refueling are given in Figure 6-2 for each of the four EOS configurations as a function of the rescue payload weight carried. It is assumed that the refueling occurred at a 100 nmi (185 km) circular orbit and 28.4° inclination. Results are shown both with and without a cargo bay tank. Superimposed on the figure are the lunar one-way and round-trip ΔV requirements from LEO as well as the one-way plus lunar orbit departure (transearth injection only) case.

Comparable performance results for the parallel-burn Space Shuttle configuration are presented in Appendix J, Volume III, Part 1.

Although adding a cargo bay tank improves the capability of a refueled Orbiter, none of the configurations can achieve lunar round trip with a return to LEO. The cargo bay propellant does in fact contribute about one-third of the total ΔV available to a refueled Orbiter. (See Table D-3, Appendix D, Volume III Part 1.) The added payload weight derates the total capability, however, and results in a net ΔV gain of only 2 to 3 kft/s (0.6 to 0.9 km/s) over the refueled Orbiter without a cargo bay tank.

All configurations have one-way lunar capability with a rescue payload. Except for Configuration A, the available ΔV is sufficient also to allow lunar orbit departure and transearth injection. With a cargo bay tank, Configuration A also acquires this capability.

As discussed in Section 2, approximately the same total ΔV is required to achieve either lunar or geosynchronous orbit from LEO. To depart geosynchronous orbit for return to earth, however, requires a ΔV approximately 2500 ft/s (760 m/s) greater than for lunar orbit departure. To identify the requirement for one-way plus departure from geosynchronous orbit requires that this increment be added to the one-way plus lunar orbit departure line on Figure 6-2.

Even with this added requirement, only Configuration A lacks the necessary capability. As was the case for the lunar mission, however, with a cargo bay tank even Configuration A offers one-way plus geosynchronous orbit departure capability.

Configuration B is the best candidate for orbital refueling. Transport of a 10 klb (4.5 t) rescue module as payload results in less than a 10% reduction in available ΔV. But even with no payload and a cargo bay tank, the Configuration B Orbiter is approximately 2000 ft/s (610 m/s) short of lunar round-trip capability from LEO. Configuration C is approximately 1 to 1.5 kft/s (305 to 460 m/s) less capable than Configuration B.

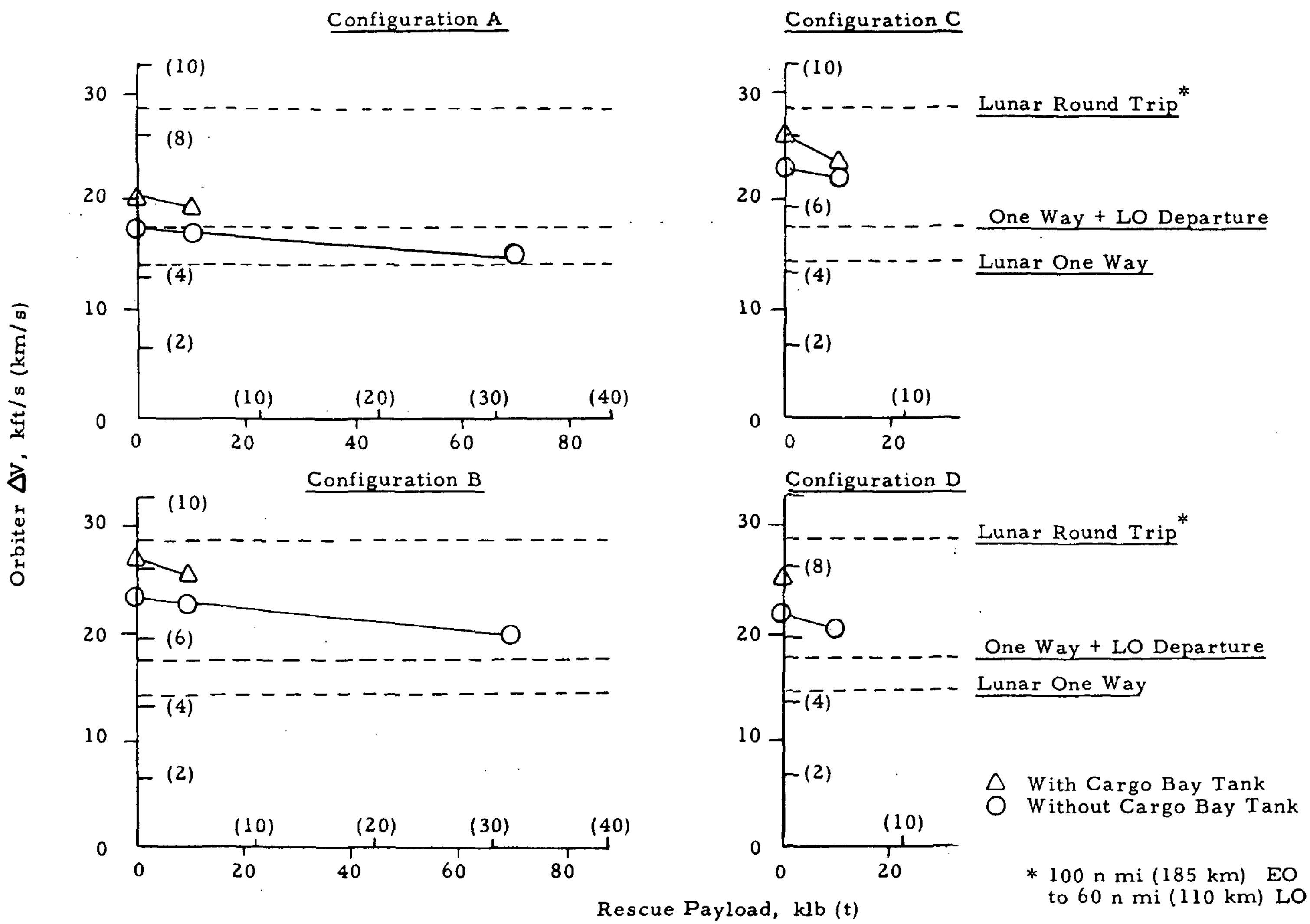


Figure 6-2. On-Orbit ΔV with Orbital Refueling

Although the ΔV which remains after transearth injection is insufficient for earth orbit insertion, significant amounts may be available for maneuvering and braking, depending on the EOS configuration. The feasibility of Orbiter earth reentry from a lunar mission using only the available ΔV was considered. The multiple-pass grazing mode of earth reentry (discussed in section 9.2) appears useful and, if employed, would result in Orbiter lunar round-trip capability.

6.4 TECHNICAL FEASIBILITY

Orbital refueling of Orbiter tanks with cryogenic propellants appears technically feasible. Both fluid transfer and tank exchange are considered practical. Both techniques require rendezvous and docking by the receiver and donor. Where feasible, tank exchange is the operationally preferred mode.

Use of an OPD as a propellant donor appears technically feasible. Direct propellant delivery to an Orbiter via dedicated EOS logistic flights, or via an ESS, also appear technically feasible. A significant number of earth launches, however, would be required with these latter two techniques. If, as may be the case with a rescue mission, rapid response is desired, the long durations involved would be unacceptable. Propellant delivery to an OPD by EOS logistics flights or an ESS is practical, but it must occur before the need for a rescue mission involving Orbiter refueling has developed.

The modifications required and the operating mode introduced by refueling the Orbiter in LEO are considered feasible. The Orbiter must be provided with a capability to dock to the OPD, and its propellant system must include refueling provisions. If tanks are exchanged, the interface between the main tanks and the reentry vehicle must be equipped for remating in space. Main engine restart capability is necessary. Also, to avoid exceeding the cargo bay structural limits, single engine operation is required if a refueled cargo bay tank is utilized.

As stated in section 5.4, a cryogenic cargo bay tank for space application is current state of the art. The Orbiter main tank is not designed for more than a few hours of propellant storage, however. Unless a cargo bay tank is included, a refueled Orbiter sent to the moon would therefore require additional main-tank insulation.

6.5 SUMMARY

Refueling the Orbiter in LEO may offer both lunar and geosynchronous rescue mission capability. Propellant transfer to the empty Orbiter tanks is an acceptable refueling mode. However, when feasible, tank exchange (full for empty) is preferred. If an OPD is not available, EOS logistics flights or an ESS might be used as the propellant donor.

All refueled configurations have one-way lunar and geosynchronous rescue-mission capability. Although the remaining ΔV is inadequate for return to earth orbit, it is generally sufficient for transearth injection. If aerodynamic braking via multipass-grazing reentry is employed, earth return appears feasible. If available, a small braking ΔV on the first perigee significantly reduces the time required for such reentry (see section 9.2).

Among the four initial Space Shuttle reference designs, Configuration B is the best candidate for orbital refueling and is closely followed by Configuration C. However, the parallel-burn Space Shuttle design, because of its substantially larger external propellant tank, offers greatly superior performance with on-orbit refueling (see Appendix J, Volume III, Part 1). With this latter Shuttle configuration, both lunar and geosynchronous orbit round trips appear possible from low earth orbit with a 10 klb (4.5 t) rescue payload and without the need for aerodynamic braking.

7. EOS-LAUNCHED TUG

7.1 DESCRIPTION

7.1.1 General

Adding a stage to an existing launch system is the most frequently employed technique for augmenting the system's capability. In the case of the EOS, the Orbiter cargo bay is sized to accept an upper stage for orbital delivery (see Figure 7-1). This stage, variously identified as a Tug or OOS (Orbit-to-Orbit Shuttle), converts the two-stage EOS into a three-stage system. It is the rescue mission capability of such a three-stage system that is treated in this section.

It was not the purpose of this study to design a Tug for specific application to a rescue mission. Instead, the analysis was based on a Tug configuration selected from available designs. It was assumed that the third stage is reusable and that neither the stage nor the rescue payload have reentry capability. Both the stage and payload are delivered to and returned from LEO via the Orbiter. Thus, both the Tug and the rescue payload must fit simultaneously into the Orbiter cargo bay.

A rescue mission payload of 10 klb (4.5 t) was assumed (see Section 8). A volume 15 ft (4.6 m) in diameter and 20 ft (6.1 m) long was assumed for the payload and the cargo erection and support structure.

Only Configurations A and B can accommodate the Tug. The cargo bay of Configuration D is too short even for an Agena stage with a zero rescue payload. The Configuration C cargo bay which could accommodate a Centaur stage is too short to simultaneously accommodate a rescue module (see paragraph 8.3).

It should be noted that the performance of the parallel-burn Space Shuttle with the Space Tug as the upper stage is the same as that given in Section 7.2 for Configuration B.

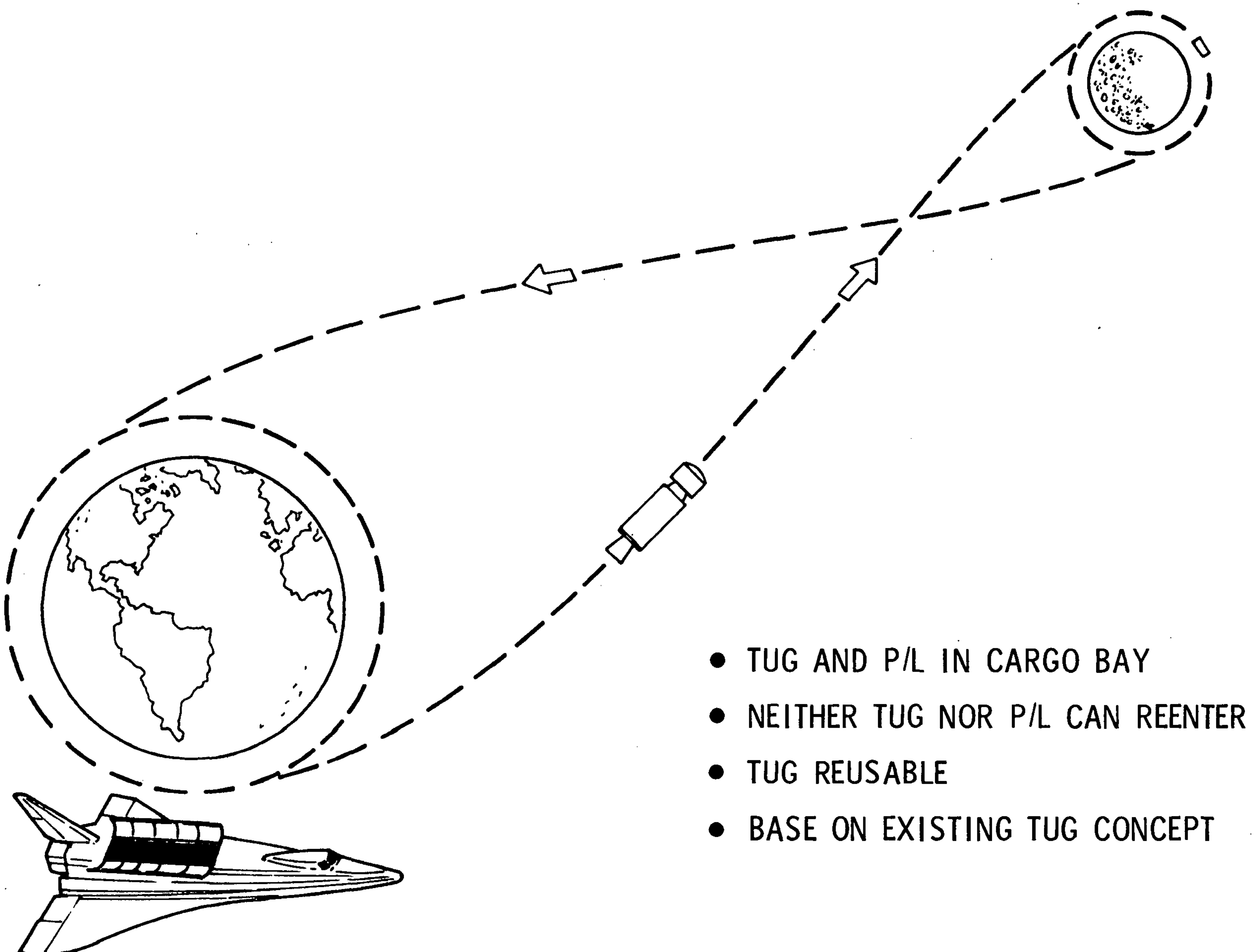


Figure 7-1. EOS-Launched Tug Schematic

7.1.2 Tug Characteristics

An integral Tug configuration based on a design presented in Reference 10 was selected as the third stage. The basis for this selection and a Tug inboard profile are given in Appendix E, Volume III Part 1. In addition to reuse and cargo bay installation requirements, the design also meets manned application requirements. Specific Tug characteristics are summarized in Table 7-1.

7.2 PERFORMANCE

7.2.1 Basic Three-Stage System

The Tug ΔV available at staging in a 100 nmi (185 km) circular orbit is given in Figure 7-2 as a function of orbit inclination. Curves are shown for both 0 and 10 klb (4.5 t) payloads with Configurations A and B. The break in each curve occurs when it becomes necessary to off-load Tug propellant to avoid exceeding the EOS payload capability.

Lunar orbit rescue capability of the basic three-stage system is shown in Figure 7-3. For a mission initiated from a 28.4° inclination, 100 nmi (185 km) circular orbit, Configurations A and B have identical capabilities. Available Tug ΔV is plotted as a function of payload weight, and dashed lines are superimposed to represent the ΔVs needed for a one-way trip and a round trip to a 60 nmi (110 km) polar lunar orbit.

A one-way Tug lunar trip appears feasible with the 10 klb (4.5 t) rescue mission payload. The round trip, however, would be limited to a payload of approximately 4 klb (1.8 t). If, as suggested in Reference 11, two of these Tugs in a tandem configuration are sequentially staged from LEO, the round-trip payload increases to ~ 10 klb (4.5 t), allowing for return of both stages to LEO. This latter approach introduces the added complexity of a second EOS flight plus rendezvous and assembly of the two Tugs.

Table 7-1. Tug Characteristics*

Gross wt, lb	61,893	Propellants	LH_2/LO_2
Propellant wt, lb	54,833	Thrust, lb	1 at 20,000
Inert wt, lb	5,786	Specific Impulse, sec	471
Burnout wt, lb	6,441	Stay Time, days	14
Length, ft	35.1	Restartable Engine	
Diameter, ft	15		

*Reference 10

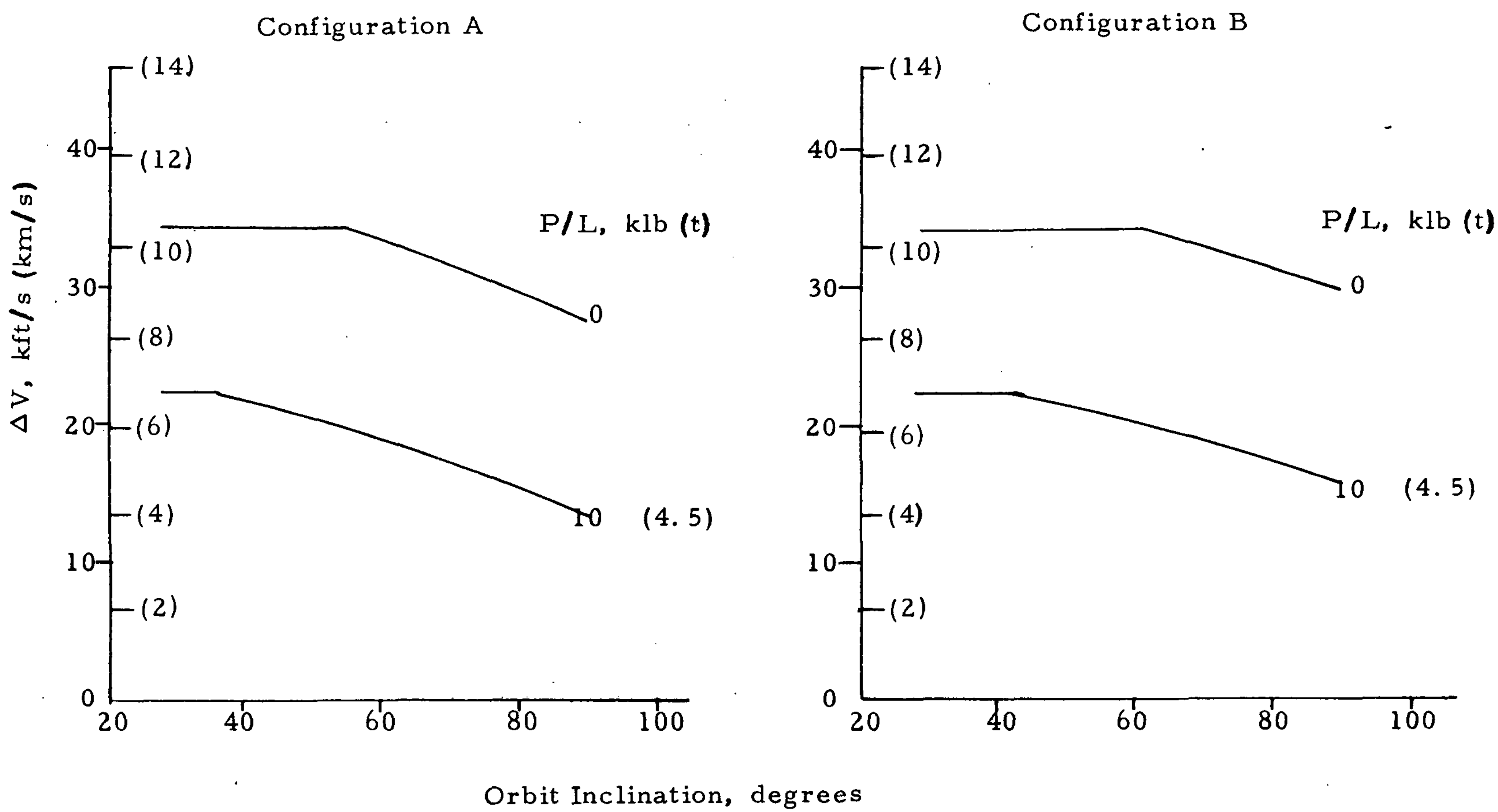


Figure 7-2. Tug ΔV at Staging in 100 nmi (185 km) Circular Orbit

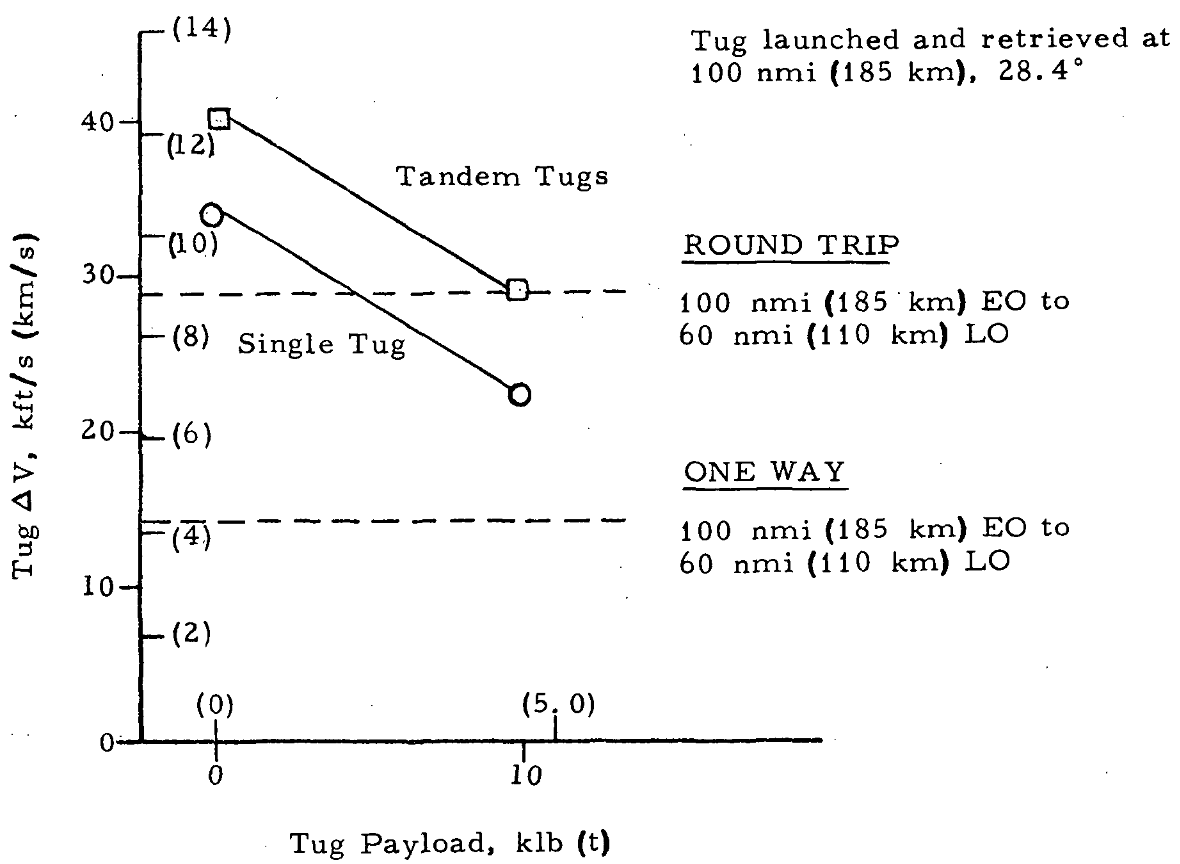


Figure 7-3. Lunar Mission Capability of Shuttle-Launched Tug

7.2.2 Refueled Three-Stage System

Performance of the three-stage system can be improved by refueling the Orbiter in LEO and also topping the Tug tanks, if the Tug is launched off-loaded. The total ΔV available in LEO is thereby approximately doubled. It is the ΔV distribution between Tug and Orbiter, however, that establishes the overall system performance capability. The ΔV available in LEO after refueling and its distribution between Tug and Orbiter are shown in Figure 7-4 as a function of rescue payload weight and EOS configuration. On the right side of Figure 7-4, this ΔV has been translated into a lunar round-trip capability.

The lunar round trip is made by the Tug only. Staging and Tug retrieval are performed at the highest circular altitude that the refueled Orbiter can achieve and from which it can return to LEO and reenter.

Both Configurations A and B could be used for placing a 10 klb (4.5 t) payload into lunar orbit with a staged Tug and would be able to return both Tug and payload to earth from the Tug staging/retrieval altitude. Owing to its greater ΔV capability, however, Configuration B stages the Tug at a higher altitude than Configuration A and thus provides a ΔV margin for lunar orbit activity. No such margin is available with Configuration A unless the rescue payload is reduced below 10 klb (4.5 t).

The Orbiter is capable of carrying a fully fueled Tug plus a 10 klb (4.5 t) rescue payload into lunar orbit from LEO. In the case of Configuration A, main tank propellant alone is inadequate, and OMS propellant must also contribute some ΔV. As a result, the remaining OMS capability is less than that required for Orbiter transearth injection from lunar orbit. Configuration B, on the other hand, reaches lunar orbit with a ΔV reserve of approximately 4 kft/s (1.2 km/s). The multigrazing earth reentry mode discussed in paragraph 9.2 would be useful here and would give Configuration B a lunar round-trip capability with a loaded Tug and a 10 klb (4.5 t) rescue payload.

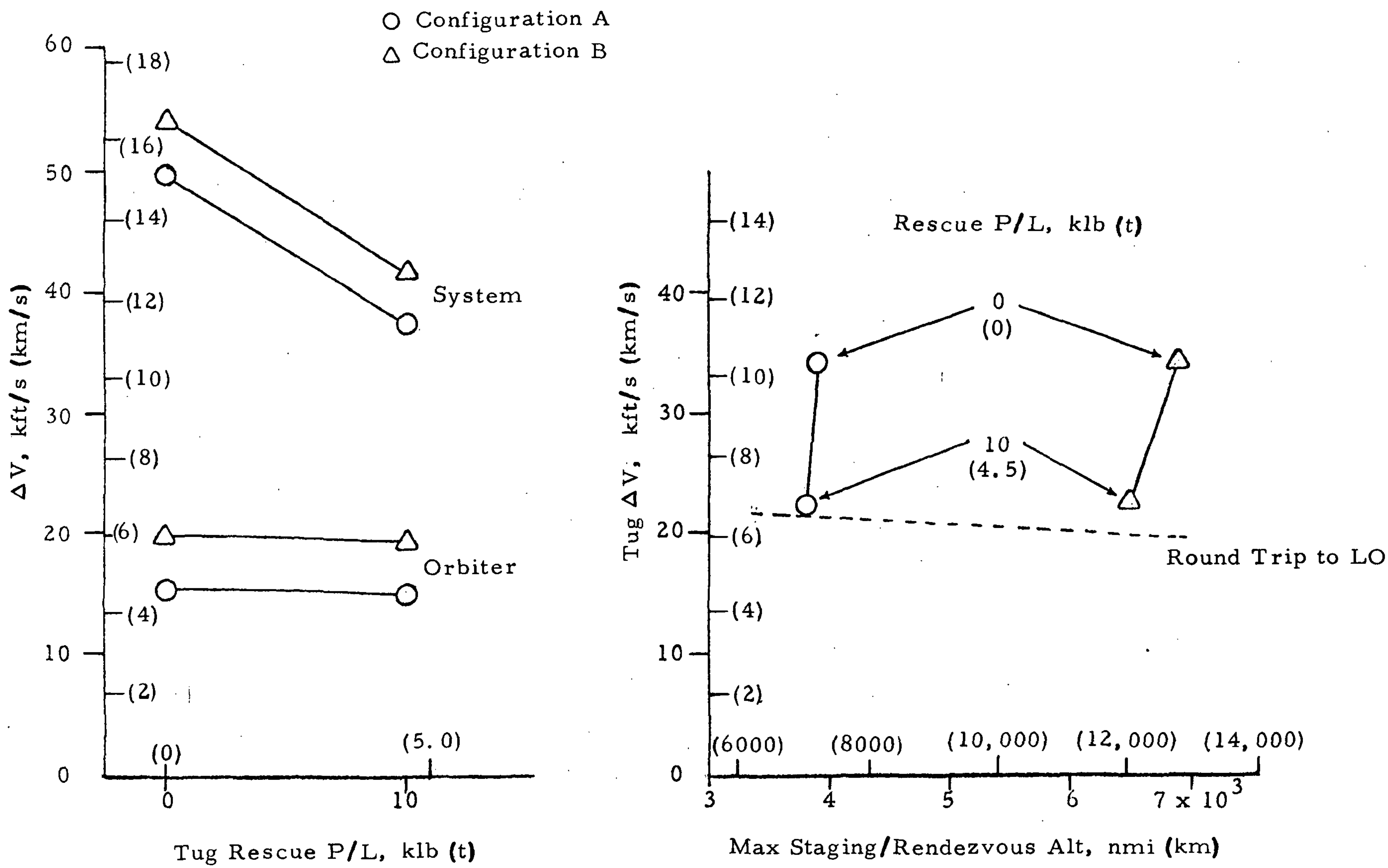


Figure 7-4. Lunar Mission Capability of Refueled Shuttle-Launched Tug
(Orbiter Refueled in LEO)

7.3 TECHNICAL FEASIBILITY

Design and development of a Tug compatible with rescue mission requirements is technically feasible. Extensive design effort has already been devoted to Tug configurations intended for manned application, EOS cargo bay installation, and extended in-space stay time. Moreover, current EOS specifications intend that a fully fueled Tug, 60 to 65 klb (27 to 30 t) gross weight, can be carried as a payload, launched, and retrieved.

Specific Tug designs for installation in a 15 ft (4.6 m) diameter by 60 ft (18.3 m) length cargo bay are available. They include appropriate weight and volume margins for a rescue module and for support and erection structure as well.

Combining the EOS-launched Tug with orbital refueling is also considered technically feasible. The Orbiter refueling discussion in section 6.4 remains applicable here. If Tug fueling is also included, the operating mode and the necessary Tug modifications depend upon whether the Tug is refueled inside of or outside the Orbiter.

7.4 SUMMARY

A three-stage system involving a Tug launched by a two-stage EOS is technically feasible and represents a simple method of achieving lunar and synchronous orbit rescue capability. The cargo bays of Configurations A and B are sized to accommodate current Tug designs and yet provide the necessary volume and weight capacity for a rescue payload plus support and erection structure as well. The cargo bay dimensions of Configurations C and D are undersized for these Tug dimensions, and smaller Tug configurations lack the desired capability.

Configurations A and B both give the staged Tug a 4 klb (1.8 t) payload lunar orbit round-trip capability from LEO. A 10 klb (4.5 t) payload lunar orbit round-trip capability would be available if the Orbiter were refueled in LEO and the Tug staging/retrieval altitude were raised accordingly.

Configuration B is the best candidate for orbital refueling. In an alternate mode, the Orbiter as well as a fully fueled Tug and a 10 klb (4.5 t) rescue payload could be placed in lunar orbit with an Orbiter ΔV reserve of approximately 4 kft/s (1.2 km/s). A large rescue vehicle ΔV capability is thus provided in lunar orbit. In addition, the Orbiter ΔV reserve appears sufficient for a multiple-pass grazing earth return carrying both the Tug and rescued personnel.

8. RESCUE PAYLOAD

8.1 GENERAL

Previous studies have identified payloads appropriate for rescue missions (see References 2 and 12). Such payloads fall into two general categories: the equipment to be carried aboard an SRV (an EOS, for example) and a space rescue module dedicated to rescue and powered by a propulsive stage/module such as a Tug. Usually one of these payload categories is carried on a rescue mission, the choice depending upon the specific situation.

No differentiation was made in this study between these two payload categories. The 10 klb (4.5 t) weight allowance is appropriate for either. Also, the 20 ft (6 m) cargo bay length allowance is more than adequate for the rescue module, which imposes the greater volume requirement.

8.2 SPACE RESCUE VEHICLE (SRV) EQUIPMENT LIST

An equipment list for a manned SRV was presented in Reference 2 and is included here as Table 8-1. Also listed is the corresponding estimated weight of each item. The total weight represented by this list is 8.3 klb (3.8 t). If more than one unit of an item (EVA suit, AMU backpack, one-man transfer capsule, tether, etc.), is carried, the total weight could be approximately 10 klb (4.5 t), the nominal value assumed for this study.

8.3 RESCUE MODULE

The space rescue module is reusable and manned. It is intended to house a rescue crew on missions beyond the capability of the Orbiter. Delivery to space and return to earth is, however, via the Orbiter. This rescue module would contain as standard equipment some of the items listed in Table 8-1.

Table 8-1. Recommended Equipment for Manned SRV*

Item	Unit Weight, lb
Communications and Survey Equipment	700
Despin Devices	250
Soft Docking Fixture	250
Attachable Docking Fixture	800
Portable Airlock	1,600
EVA Suits	70
AMU Backpack	150
Manipulator (Shirtsleeve)	2,000
Transfer Capsule	500
Sampling and Analysis Kit	50
Damage Control Equipment	150
Remote Manipulator	1,000
Medical Kit	60
Extended Survival Kit	500
Tethers (Umbilicals)	45
Personnel Carriers	10
Miscellaneous and Spares	200

*Reference 2

A representative rescue module design from Reference 12 is shown in Figure 8-1. It incorporates a docking port, side hatch and airlock, manipulator arm, and, of course, habitable space for crew and passengers. The specific design illustrated has a 50-day capability with a three-man crew at a total weight of 9.8 klb (4.5 t). More men can be accommodated (to 15) in an emergency for a shorter time. The module diameter is 14 ft (4.3 m) and its length 10 ft (3 m), which is well under the volume assumed in the EOS performance estimates.

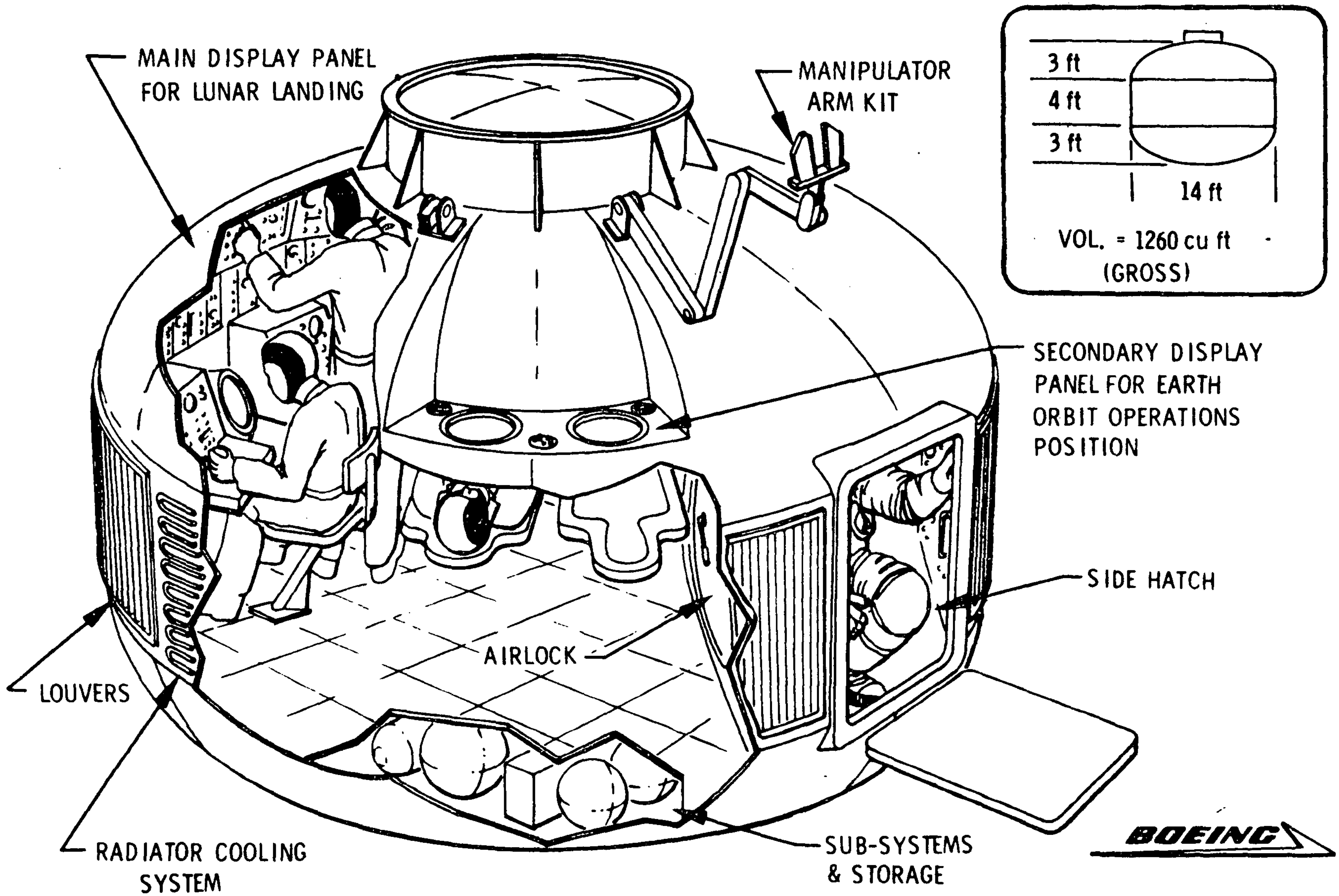


Figure 8-1. Representative Space Rescue Module Design (Reference 12)

9. SPECIAL SUBJECTS

9.1 ORBITER REENTRY CAPABILITY

9.1.1 General

The maximum orbital altitude from which Orbiter reentry is feasible is established by the available deorbit ΔV and the maximum allowable surface temperature of the thermal protection system (TPS). The peak TPS surface temperature is, in turn, dependent upon the Orbiter aerodynamic characteristics (L/D and W/S) and the crossrange desired.

The current Orbiter objective for all four configurations is reentry from 100 nmi (185 km) with an 1100 nmi (2040 km) crossrange. Also, the TPS temperature along the bottom surface centerline is limited to 2200°F (~1480°K). Reentry from orbital altitudes greater than 100 nmi (185 km) involves transferring to a 100 nmi orbit and then initiating reentry.

9.1.2 Direct Reentry from > 100 nmi (185 km)

The feasibility of direct Orbiter reentry from altitudes >100 nmi (185 km) is of interest in rescue missions. This subject was treated in Reference 13, where it was concluded that an Orbiter designed to the basic specifications indicated in paragraph 9.1.1 had a direct reentry capability from altitudes significantly greater than 100 nmi (185 km).

A comparison is made in Figure 9-1 for altitudes of 200 to 700 nmi (370 to 1300 km) between the ΔV required for a "standard" Orbiter reentry involving transfer to 100 nmi (185 km) and direct deorbit and reentry. The crossrange is held to zero and TPS limits are observed. Over the entire altitude range, direct reentry requires only about half the ΔV that is required for a "standard" reentry.

The variation in peak lower surface temperature for direct reentry and zero crossrange is given in Figure 9-2 as a function of the orbital altitude from

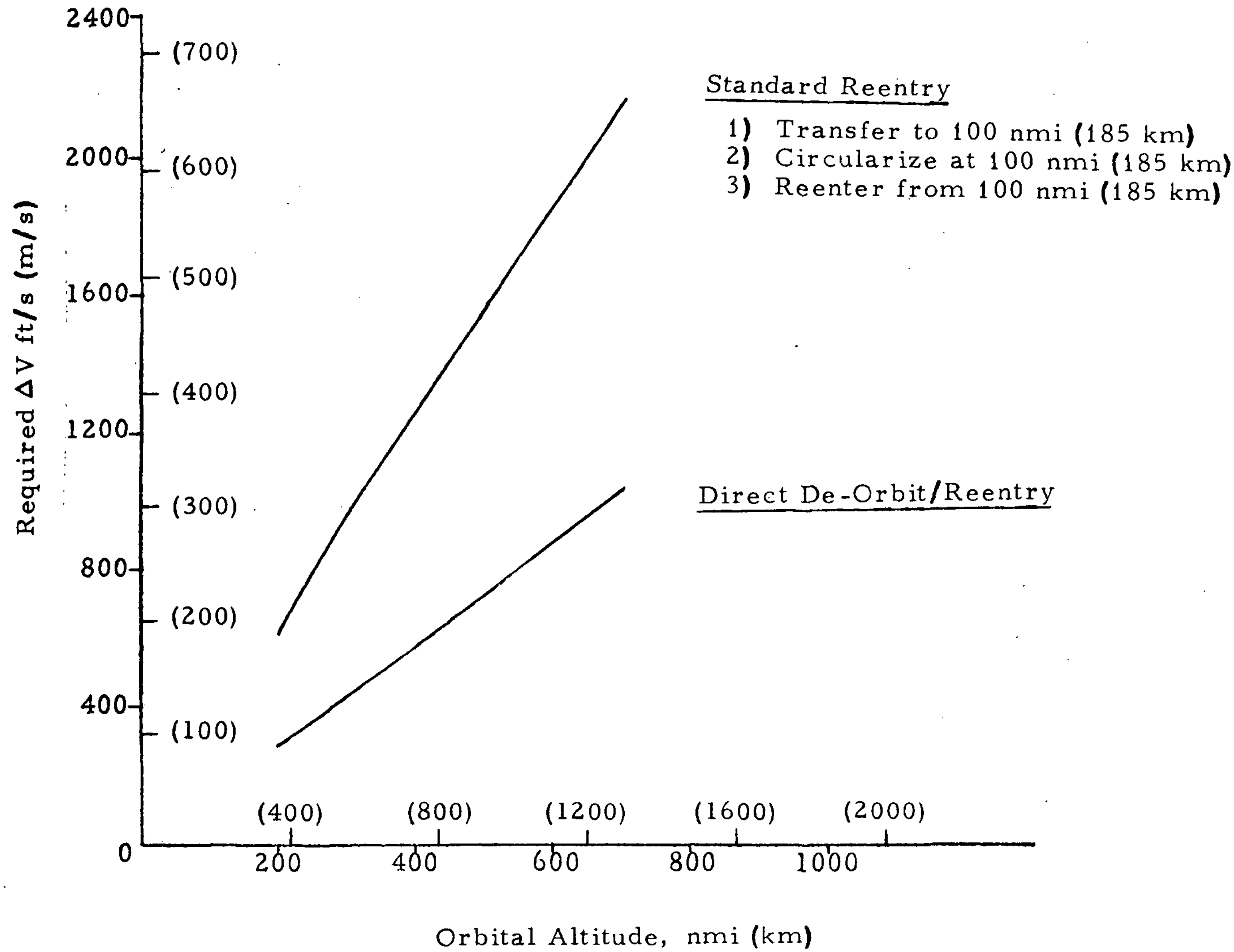


Figure 9-1. Orbiter Reentry ΔV Requirement

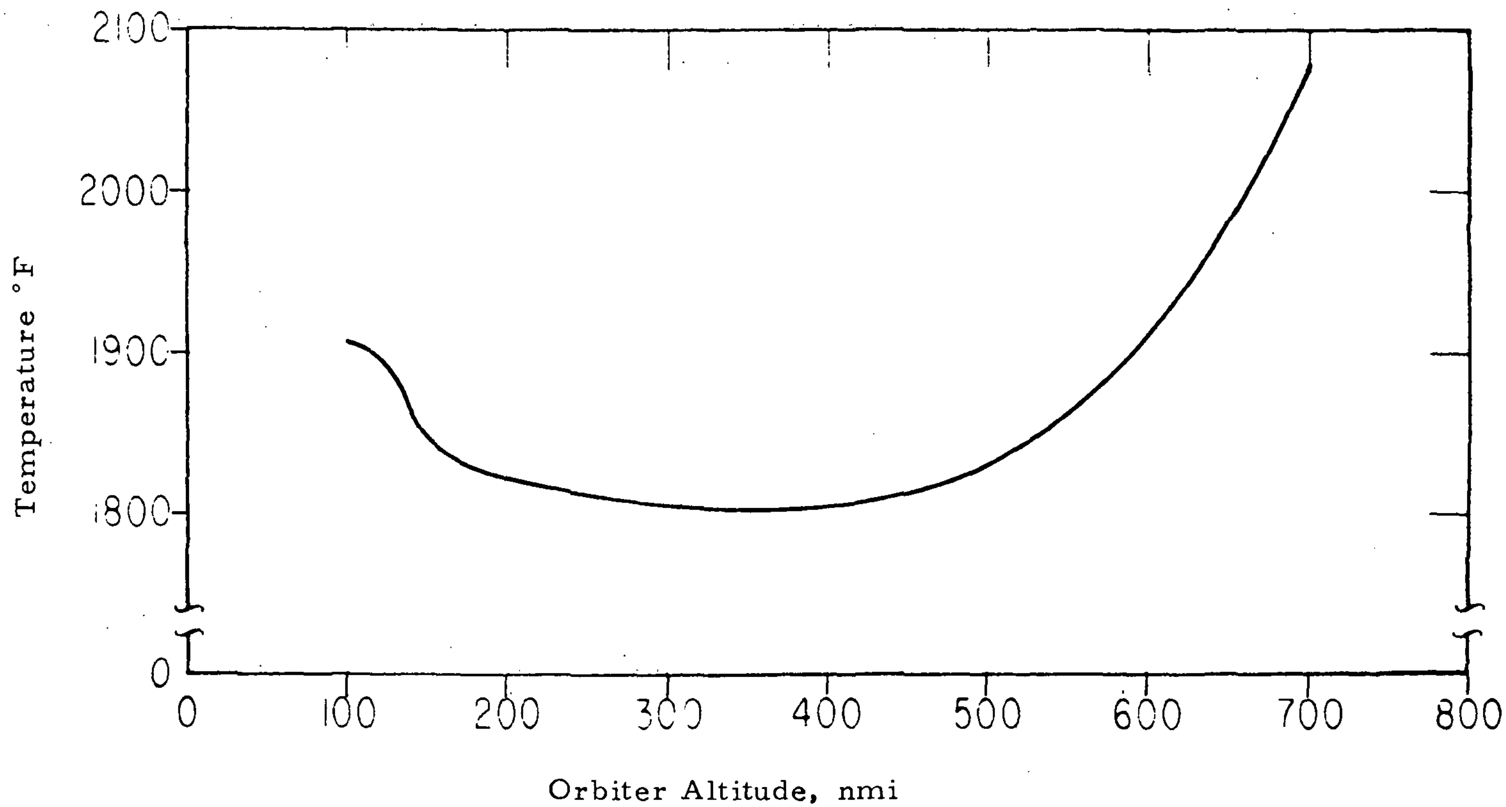


Figure 9-2. Orbiter Lower Surface Peak Temperature for Direct Reentry, Zero Crossrange (Reference 13)

which reentry is initiated. With a TPS surface temperature limit of 2200°F (~1480°K), direct reentry from a circular orbital altitude slightly in excess of 700 nmi (1300 km) appears feasible.

Additional discussion of direct Orbiter reentry is given in Appendix F, Volume III Part 1.

9.1.3 Summary

If the crossrange is held to zero, direct Orbiter reentry from altitudes to ~700 nmi (~1300 km) appears feasible for all four configurations while observing the 2200°F (1480°K) TPS surface temperature limit. The required ΔV is approximately half that required for "standard" reentry from those altitudes.

9.2 ORBITER LUNAR RETURN - MULTIPLE-PASS MODE

9.2.1 General

As already discussed, the Orbiter of all four configurations is capable, with certain performance augmentation techniques, of achieving geosynchronous and lunar orbits. However, the Orbiter is neither designed for earth reentry from such high-energy missions, nor is the remaining ΔV adequate for a return to LEO from which a standard reentry could be initiated. In this case, aerodynamic braking by multiple-pass grazing reentry may be useful.

A diagram of multiple-pass grazing reentry is shown in Figure 9-3. The trajectory is designed to successively graze the earth's atmosphere on each pass without exceeding TPS temperature limits. This repetitive aerodynamic braking gradually reduces the Orbiter velocity until earth capture and normal reentry occur. The only ΔV required is for transearth injection at the moon and midcourse correction. If desired, a small ΔV could be provided in order to raise the perigee altitude on the last orbit for delaying reentry and providing control over the landing point (in excess of the crossrange). Inherent disadvantages of this approach are the duration of the reentry procedure and the level of guidance accuracy required.

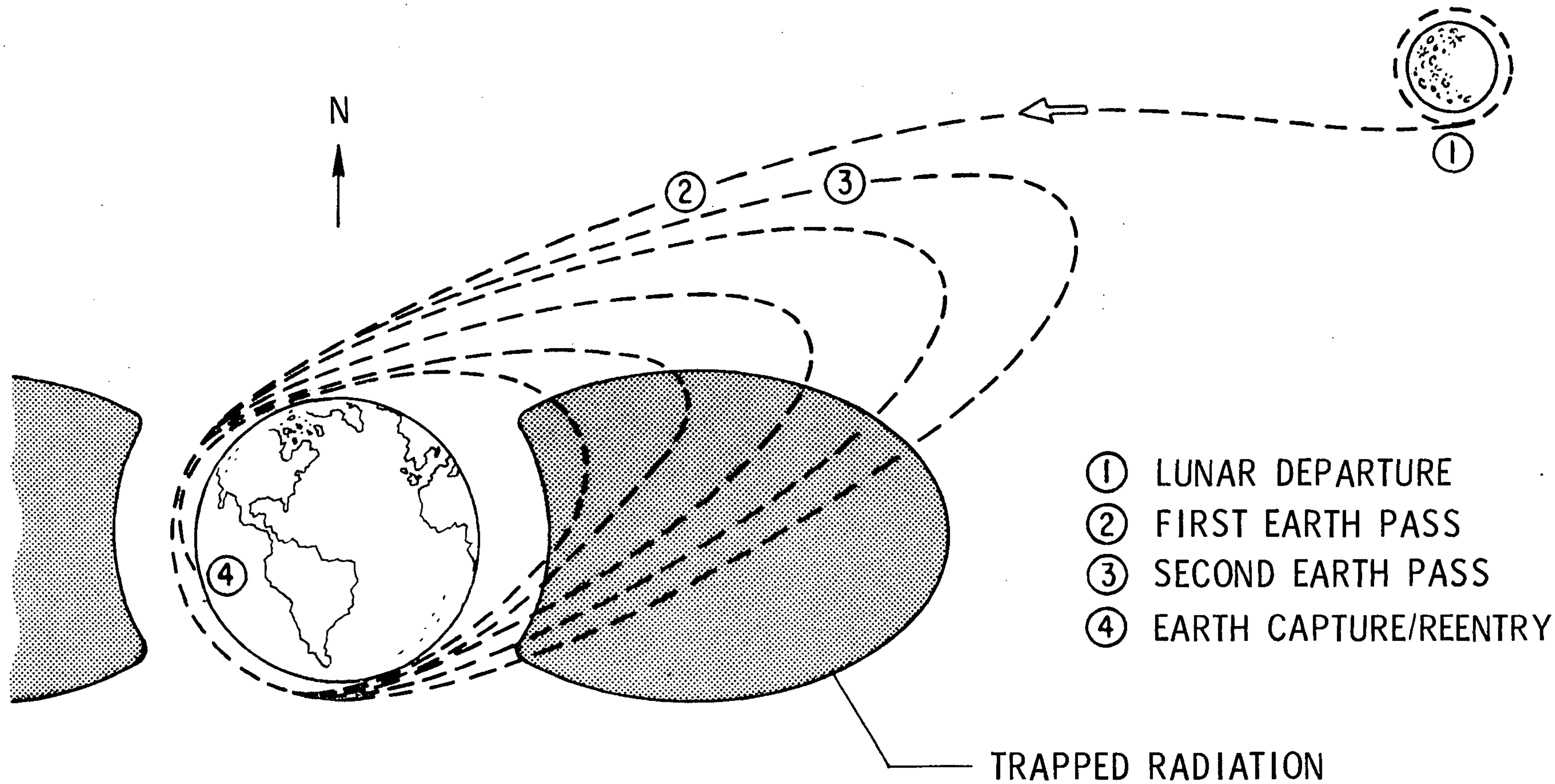


Figure 9-3. Schematic Illustration of Multiple-Pass Grazing Reentry from Lunar Orbit

The Orbiter characteristics assumed for these calculations were

$$L/D = 0.8$$

$$\frac{W}{C_L S} = 41.3 \text{ lb/ft}^2 \ (1980 \text{ N/m}^2)$$

$$\alpha = 50°$$

The results are applicable to any Orbiter configuration which matches these parameters, regardless of size or weight.

9.2.2 Return Duration

If the atmosphere were not present, the first perigee velocity would be 36.4 kft/s (11.1 km/s). Since the equivalent velocity for a multiple-pass grazing reentry from geosynchronous orbit is 33.8 kft/s (10.3 km/s), this latter case is less severe and involves fewer passes before capture and reentry. All further discussion will therefore address lunar return only, since it represents the more difficult case.

The variation in total transit time for pure grazing reentry, as a function of the maximum lower surface temperature (which occurs on every pass through perigee), is shown in Figure 9-4. Included in the total transit time is 60.2 hrs for the period from lunar orbit departure to first perigee and 0.5 hr for final reentry to touchdown.

Lunar return time is very sensitive to the allowable lower surface maximum temperature. The current design value of 2200°F (1480°K) results in ten grazing passes and a total lunar return time to touchdown of 323 hrs or 13.5 days, as compared to a direct reentry of 60.7 hrs or 2.5 days.

The perigee altitudes for 2200°F (1480°K) range from 47 nmi to 42 nmi (87 km to 78 km). The variation in temperature with displacement from these nominal targeted perigee altitudes is shown in Figure 9-5. Small errors in perigee altitude due to guidance or navigational imprecision or a departure of the

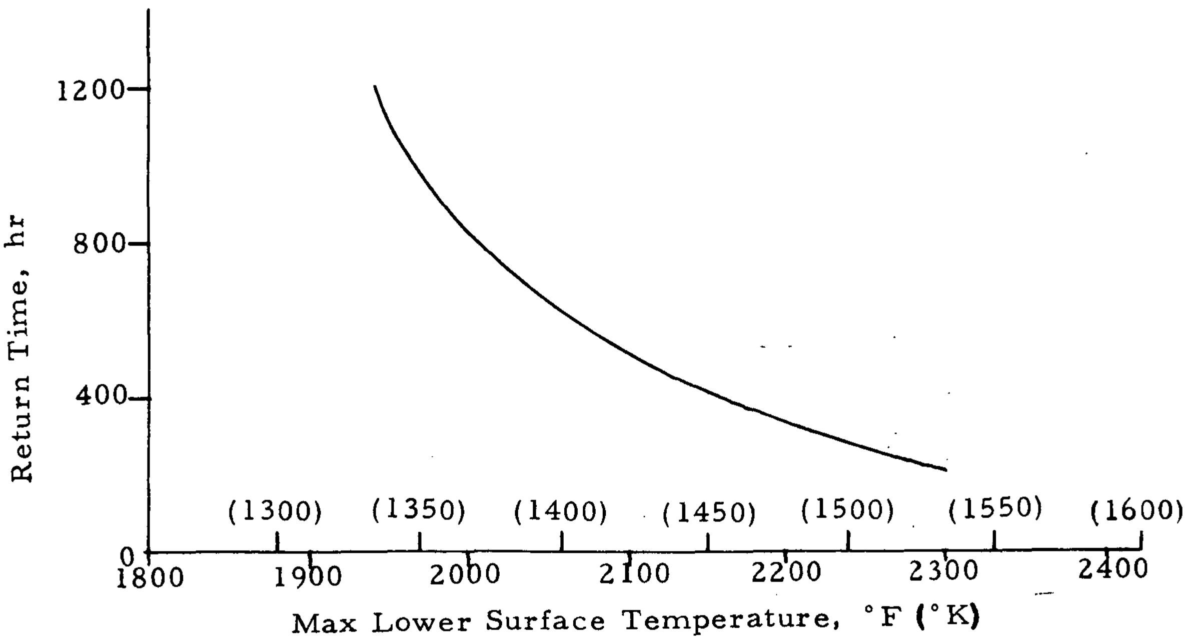


Figure 9-4. Total Lunar Return Time for Unassisted Grazing Reentry

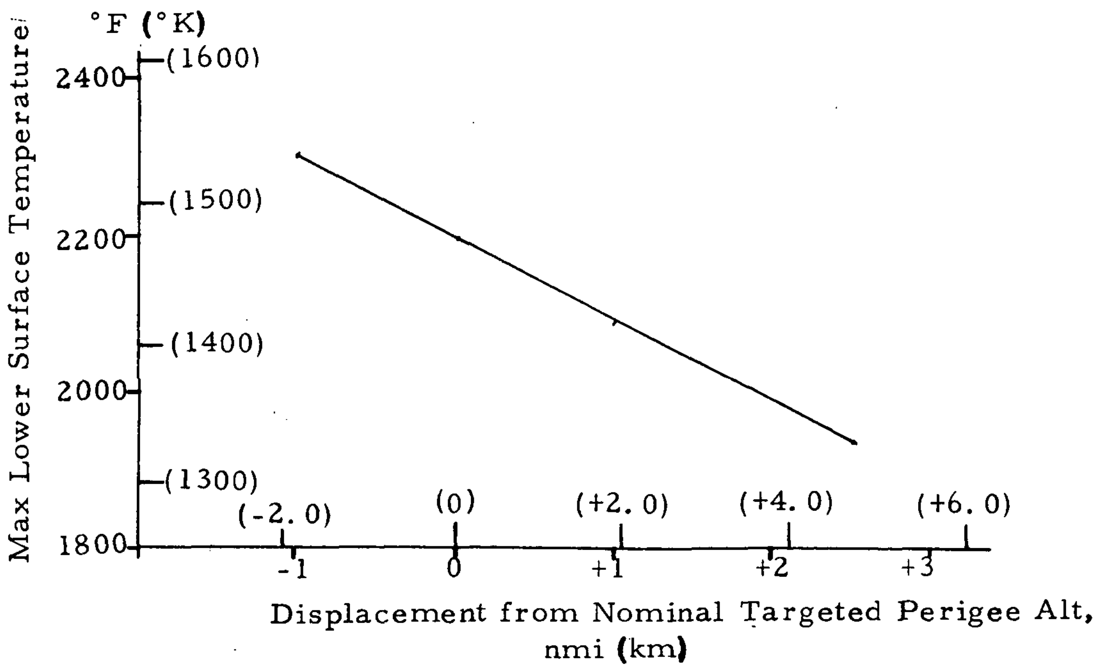


Figure 9-5. Sensitivity of Lower Surface Temperature to Perigee Altitude

perigee density from the model atmosphere value can significantly influence the surface temperature and cause large changes in the total lunar return time. If, for example, the perigee altitude on each pass is raised by 2.5 nmi (4.6 km), the equivalent density reduction is approximately 50%. The resulting maximum lower surface temperature is 1940°F (1330°K), and the corresponding total return time is 1195 hrs or 49.8 days. Conversely, if the perigee altitude is lowered by 1 nmi (1.8 km), the maximum temperature rises to 2300°F (1530°K), and the total return time is reduced to 8.4 days.

The total transit time for a lunar rescue mission is the sum of the ascent and return durations. The ascent duration, which is beyond the scope of this discussion, is not unique to the Orbiter but depends on the earth-moon geometry at any particular time.

9.2.3 Earth Return with Perigee Assist

The Orbiter ΔV budget for lunar return via multiple-pass grazing reentry typically includes 3.3 kft/s (1 km/s) for transearth injection and midcourse correction. In addition, it is necessary to apply a small ΔV--~1 ft/s (~0.3 m/s) or less--on each pass to maintain the lower surface maximum temperature at the selected design value.

Any excess ΔV available after these required expenditures are met is very effective in reducing the Orbiter lunar return time if applied as a retro-velocity to remove some of the energy from the orbit. The most efficient ΔV usage is a single burn at the first perigee, where the greatest change in orbit period can be achieved.

The total lunar return time with a ΔV assist at the first-pass perigee is plotted as a function of the applied retro-ΔV in Figure 9-6. The first perigee is targeted at 100 nmi (185 km), so that the Orbiter, flying tail first, can apply a retro-ΔV at that perigee. At the next apogee the Orbiter is reoriented, and the perigee is adjusted down to the temperature limit. A ΔV of only a few ft/s is required for this adjustment. The perigee-assist procedure offers no time

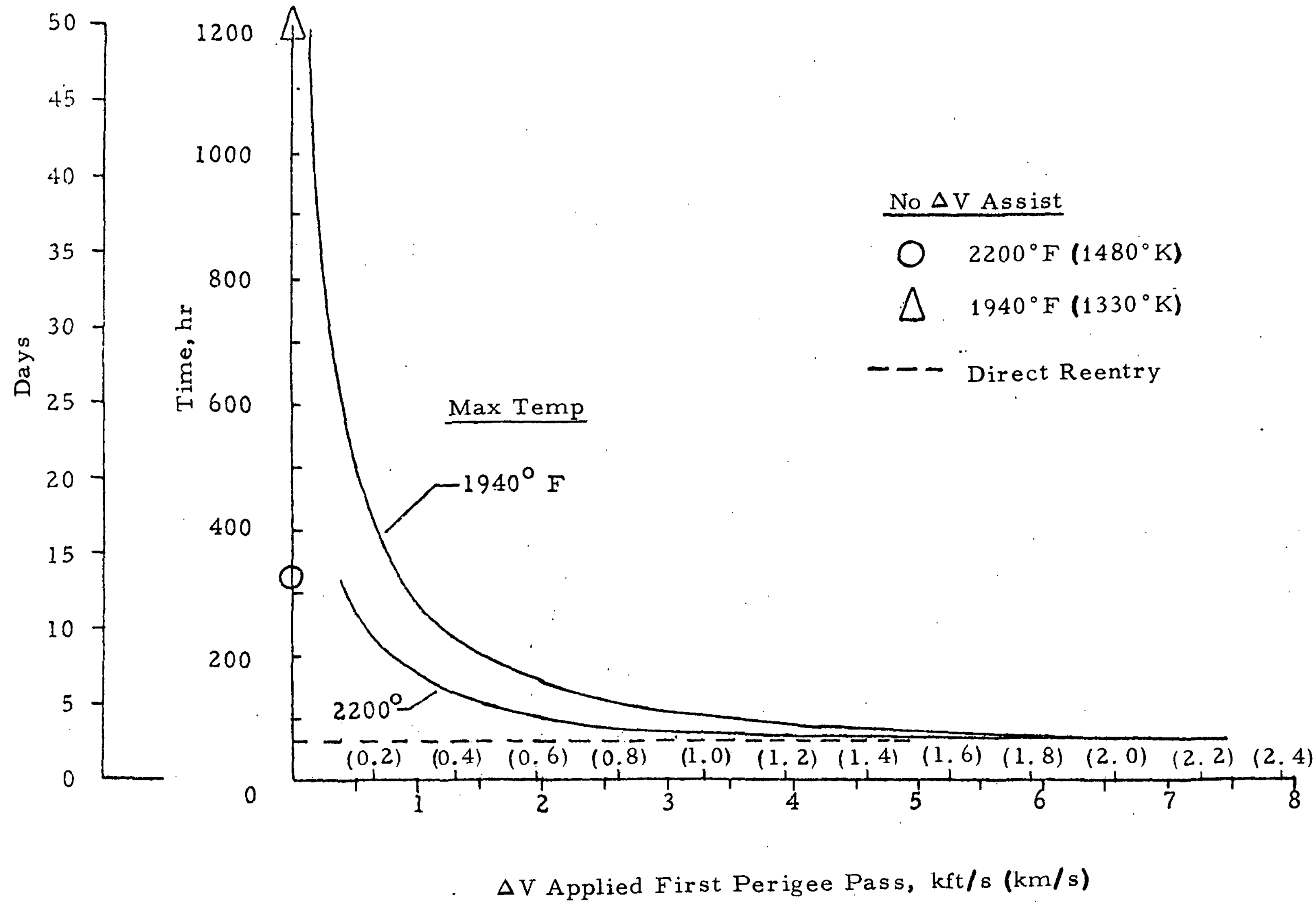


Figure 9-6. Lunar Return Time for Assisted Grazing Reentry, Lunar Orbit to Touchdown

saving until the retro-ΔV is larger than the velocity loss due to aerodynamic drag on a first pass targeted to the temperature limit.

Curves are plotted for two values of the maximum lower surface temperature, 2200°F and 1940°F (1480°K and 1330°K). Also shown are a line at 60.7 hours for the limiting case of direct lunar return and data points for no first-perigee pass ΔV assist.

Small retro-ΔVs cause large reductions in lunar return time, especially for the 1940°F (1330°K) case. Once beyond the knee of the curve, which occurs between 1 and 2 kft/s (305 and 610 m/s), the rate of reduction in return time slows markedly at either temperature. Nevertheless, with only a 3 kft/s (~1 km/s) retro assist on the first perigee, the total return time is only 84 hours for the 2200°F (1480°K) case and 113 hours for the 1940°F (1330°K) case. The number of orbits is correspondingly reduced to less than half the value for the no-ΔV assist case.

Additional discussion of the orbital mechanics aspects of Orbiter lunar orbit return via multiple-grazing reentry is given in Appendix G, Volume III Part 1.

9.2.4 Summary

A comparative tabulation of multiple-pass grazing Orbiter lunar return for two lower surface temperature limits with and without ΔV assist is given in Table 9-1. Based on only flight mechanics considerations, multiple-pass grazing reentry appears feasible, and lunar return can be achieved with about one-fourth the ΔV budget required for lunar return via LEO reentry. Similarly, geosynchronous orbit return can be achieved with about one-half the ΔV budget required for return via LEO reentry. The difference is in the value of the transearth injection ΔV which is about 3300 ft/s (1 km/s) for lunar orbit and about 6000 ft/s (1.8 km/s) for geosynchronous orbit.

Table 9-1. Comparison Summary of Orbiter Grazing Reentry

	2200°F	(1480°K)	1940°F	(1330°K)
Lunar Departure ΔV	3300 ft/s	(1 km/s)	3300 ft/s	(1 km/s)
Velocity (first earth pass)	36,405 ft/s	(11.1 km/s)	36,405 ft/s	(11.1 km/s)
First-Pass Altitude	284,000 ft	(86.6 km)	299,000 ft	(91.1 km)
No-Assist Case				
No. of passes	10		26	
Total time, hr	323		1195	
ΔV Assist Case Total Time, hr				
1000 ft/s (0.3 km/s)	164		285	
2000 ft/s (0.6 km/s)	104		157	
3000 ft/s (0.9 km/s)	84		113	
Direct Reentry, hr	60.7		60.7	

9.3 ORBITER TPS MULTIPLE-PASS CAPABILITY

9.3.1 Discussion

The feasibility of Orbiter multiple-pass return from lunar orbit is established not merely by flight mechanics considerations. Other factors, including adequacy of the Orbiter TPS and the crew's accumulated exposure to trapped radiation surrounding the earth, must also be considered.

An analysis of the adequacy for multipass reentry (see Figure 9-3) of a TPS using Reusable External Insulation (REI) to protect an aluminum primary structure and designed for normal Orbiter reentry is given in Appendix G, Volume III Part 1. The case treated was no perigee ΔV assist and a maximum lower surface temperature of 2200°F (1480°K).

Based on presently considered Orbiter insulation concepts, the temperature history of the aluminum structure under the TPS was computed for two values of TPS initial surface temperature (Figure 9-7). On the basis of available experience, 0°F (255°K) and 100°F (310°K) were selected. The actual value will depend on Orbiter orientation relative to space and the sun during the cooling period. Due to the relative durations of the heating and cooling periods, the Orbiter can be oriented to allow the TPS surface temperature to return to its initial value after each perigee heat pulse.

The effect on the structure temperature of the passes preceding reentry is small. Most of the temperature rise occurs during soakback after reentry, and only a small rise occurs during the last few grazing passes prior to reentry.

The estimated peak structure temperatures after reentry are summarized below:

	°F		°K	
TPS surface temp.	0	100	255	310
Peak structure temp.	271	350	406	449

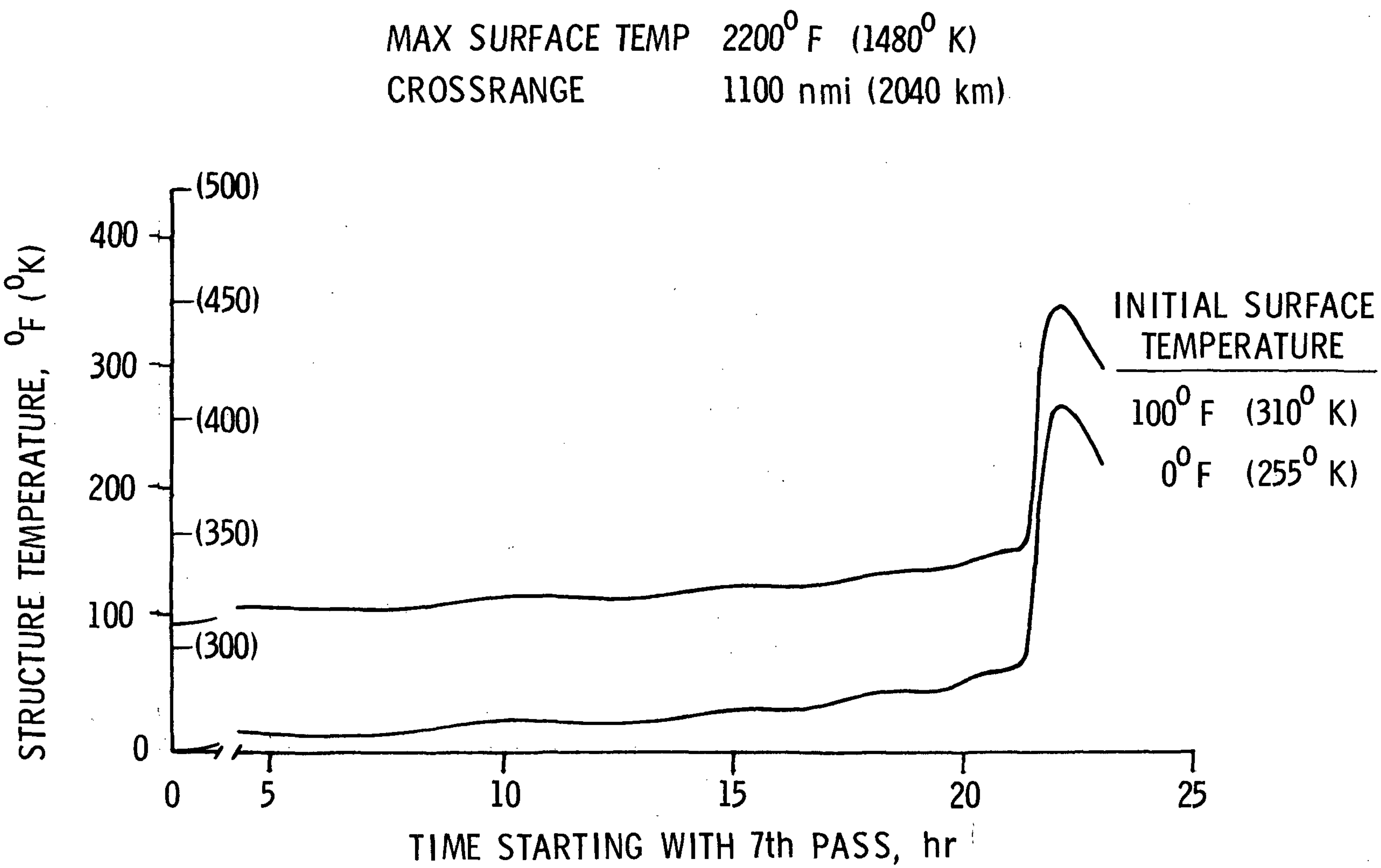


Figure 9-7. Structure Temperature History for Multiple-Pass Grazing Reentry with No Perigee Assist

Both cases include a 1100 nmi (~2000 km) crossrange. By reducing the crossrange to 730 nmi (1350 km), the peak value for the 100°F (310°K) case can be held to the design maximum of 300°F (420°K). The preferred way of holding the structure temperature below its design value, however, is to operate at an initial TPS surface temperature below 100°F (310°K).

9.3.2 Summary

An REI thermal protection system normally designed for a maximum TPS surface temperature of 2200°F (1480°K) and a maximum aluminum structure temperature of 300°F (420°K) provides adequate protection during unassisted multipass-grazing return from lunar orbit. Even the 1100 nmi (~2000 km) Orbiter crossrange capability can be maintained.

Both perigee ΔV assist and geosynchronous orbit multiple-pass return represent less severe heating conditions than the case analyzed.

9.4 EXPOSURE TO TRAPPED RADIATION

9.4.1 General

An analysis of Orbiter crew exposure to trapped radiation surrounding the earth is also given in Appendix G, Volume III Part 1. The calculations were based on an electron and proton environment model obtained from Reference 14. Based on available Orbiter designs, an inherent Orbiter shielding equivalent of approximately 2 g/cm^2 aluminum was assumed for all four EOS configurations.

The biological dose to the crew was estimated for two situations:

(1) Circular high-altitude earth orbits

(2) Multiple-pass grazing reentry from lunar orbit

9.4.2 Circular High-Altitude Earth Orbits

The biological dose in rem acquired by an Orbiter crew in one day is plotted in Figure 9-8 as a function of orbit altitude for an inclination of 30°. Circular

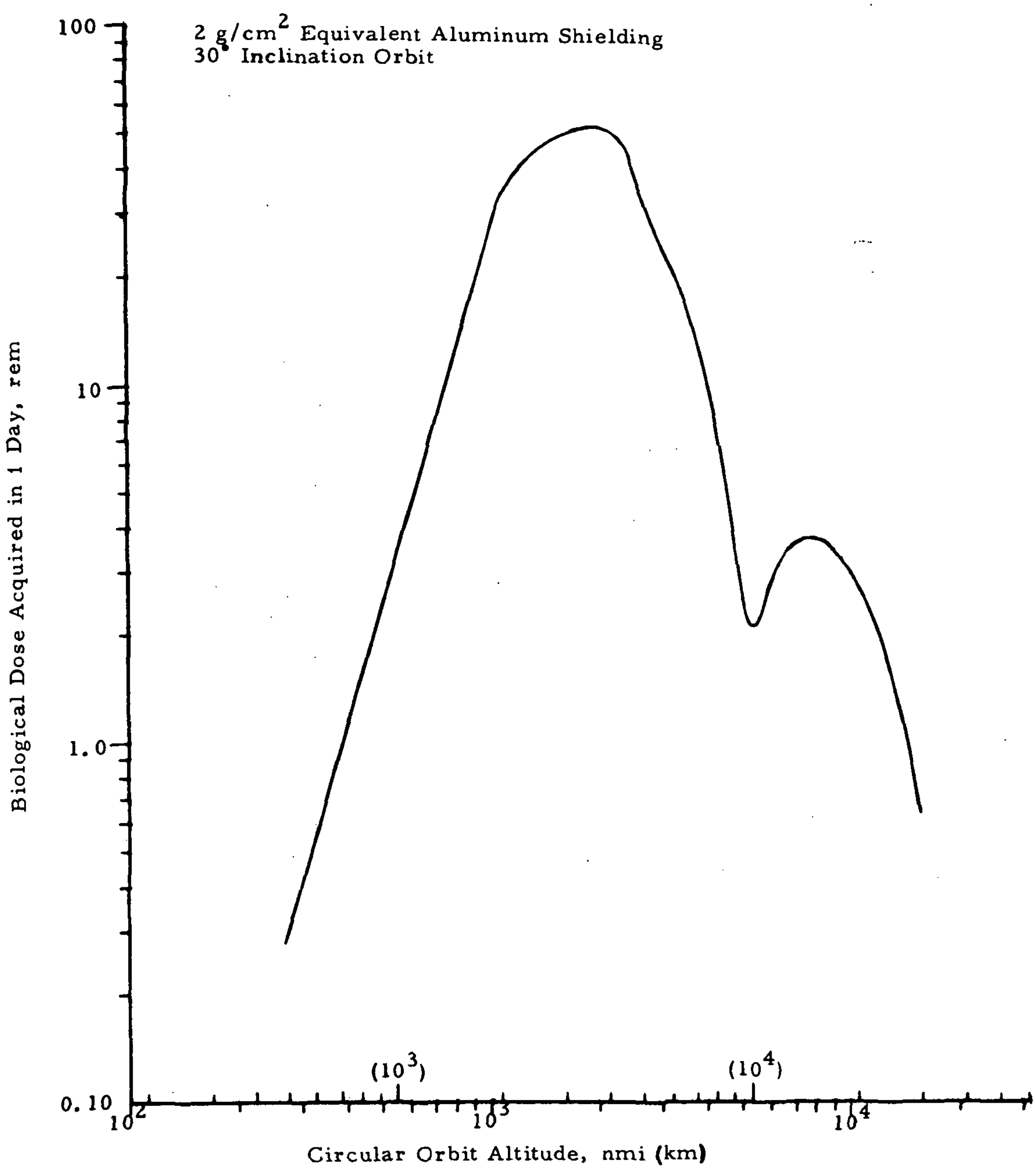


Figure 9-8. Biological Dose (Skin) Due to Earth's Naturally Trapped Radiation Environment

orbits above 270 nmi (500 km) are considered; orbits below this altitude have already been examined in previous EOS and space station studies and were found to have less severe environments.

Except for prolonged exposure in the South Atlantic Anomaly, orbit altitudes below 270 nmi (500 km) present no problem. Above this altitude the dose increases rapidly to a peak between 1500 and 2200 nmi (2800 and 4100 km).

The dose changes with either orbit plane inclination or with shield thickness. At 2000 nmi (3700 km), for example, the dose in an equatorial orbit is about twice that shown in Figure 9-8, whereas in a polar orbit it is only about half the value shown in Figure 9-8.

An increase in shielding equivalence above 2 g/cm^2 would reduce the dose only a few percent for orbits up to about 5000 nmi (~9000 km). Above that altitude increased shielding becomes more effective (see Figure G-12, Appendix G, Volume III Part 1).

Extravehicular activity (EVA) produces dramatically higher radiation doses than those indicated in Figure 9-8 because the shielding equivalence provided by an Apollo-type space suit is only ~0.5 g/cm^2. For altitudes up to about 4000 nmi (7500 km) the total radiation dose is between 3 to 10 times greater than the values as shown in Figure 9-8; above 5000 nmi (~9000 km) this value can rise to values 100 times greater (see Figure G-12, Appendix G, Volume III Part 1).

The EVA phase of a rescue mission could take 14 ± 7 hours in an unfavorable situation (see Reference 2). Although individual EVA excursions could be held to a few hours duration, the accumulated dose may limit rescue mission EVA at high altitudes.

The ordinate of Figure 9-8 is for a skin dose (0.1 mm depth). Corresponding exposure limits used by NASA are given in Table 9-2 (see Reference 15).

Table 9-2. Radiation Exposure Limits (Skin, rem at 0.1 mm depth)

1-year average daily rate	0.6
30-day maximum	75
Quarterly maximum*	105
Yearly maximum	225
Career limit	1200

*May be allowed for two consecutive quarters followed by six-months restriction from further exposure to avoid yearly limit.

By applying the skin dose limits from Table 9-2 to the curves of Figure 9-8, mission constraints can be readily established. Altitude-duration constraints for an Orbiter having 2 g/cm^2 shielding equivalence and in a 30° inclined orbit are illustrated in Figure 9-9 for a 75-rem 30-day skin dose limit. At a circular altitude of 800 nmi (1500 km) and 30° inclination, a typical peak altitude for the unrefueled EOS configurations (see Figure 5-3), the daily dose would be 12 rem. To avoid exceeding the 30-day limit, only 6.25 days could be spent on orbit.

Allowable Orbiter parking periods at the maximum staging/rendezvous altitude (see Figure 7-4) for the refueled, EOS-launched Tug case are given below:

Config.	Payload		Daily Dose	Max. Time*
	klb	t	rem	days
A	0	0	13	5.8
A	10	4.5	12	6.3
B	0	0	3.2	23.4
B	10	4.5	3.5	21.4

*based on 30-day limit

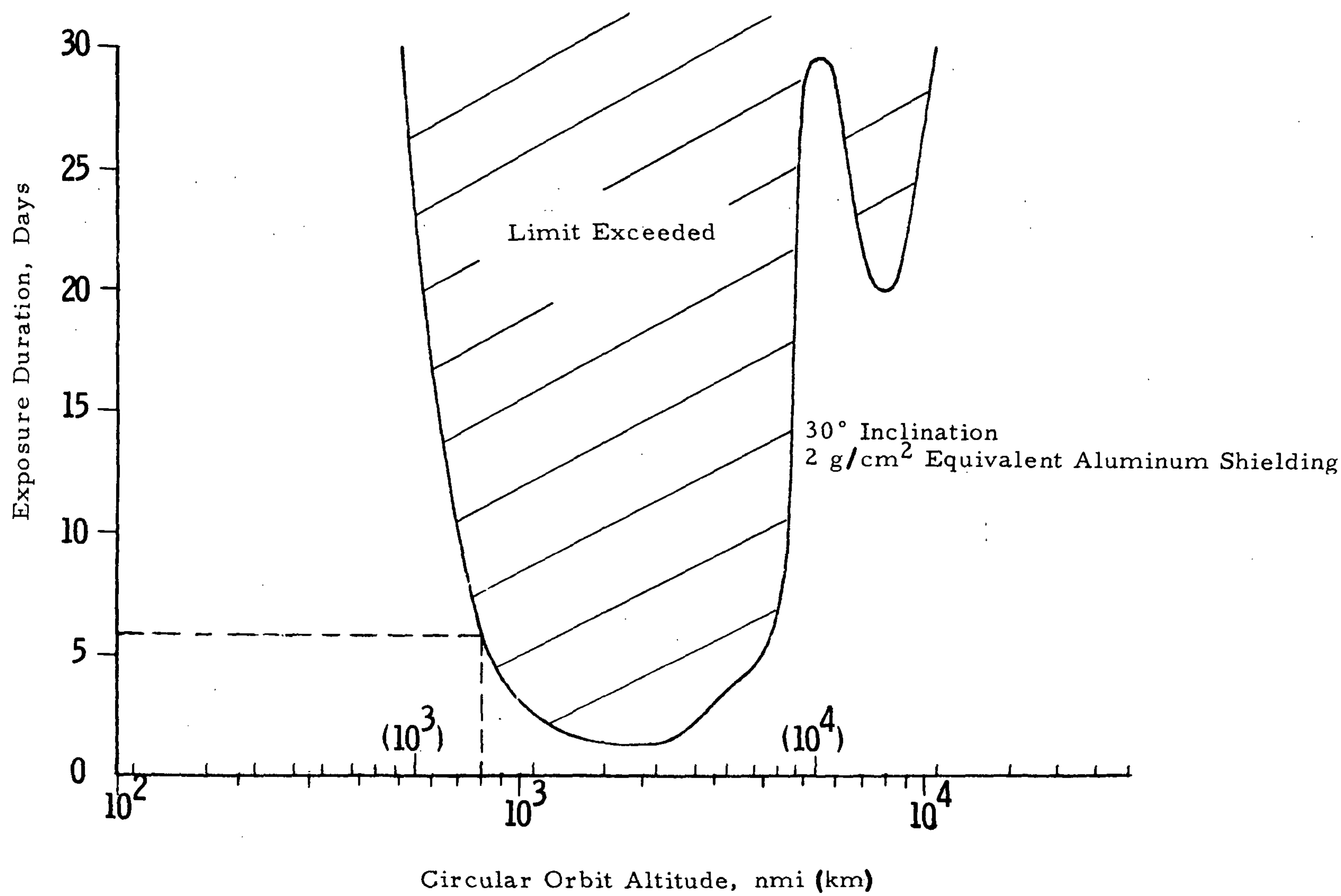


Figure 9-9. Allowable Crew Radiation Exposure for 75 rem, 30-Day Skin Dose Limit

With respect to the round-trip transit time duration for a lunar rescue mission, the time available to Configuration A appears marginal.

9.4.3 Multiple-Pass Grazing Reentry From Lunar Orbit

When multiple-pass grazing reentry from lunar orbit is employed, the Orbiter makes numerous passes through the trapped radiation field. The total dose acquired depends, of course, on the number of passes, the orbit orientation about the earth, and the orbital dimensions and periods.

Weakest field intensities are encountered in elliptic orbits with polar inclinations. Coincidentally, a polar return orbit can always be achieved with no ΔV increase over the transearth injection (TEI) value (see References 16 and 17). It was assumed, therefore, that all multiple-pass grazing reentry orbits would have 90° inclinations, and all radiation exposure is estimated on this basis.

The no-perigee assist case offers the maximum number of passes and thus the greatest accumulated exposure. Corresponding apogee and perigee altitudes, the approximate period, and the incremental radiation dose for each orbit are listed in Table 9-3. Total return time is about 13.5 days and the total dose approximately 4 rem. If no attempt were made to follow a minimum dose return, this value would be about four times higher. In any event, it appears that by itself multiple-pass grazing reentry introduces less crew exposure than the limits given in Table 9-2, and the return orbits may not necessarily have to be limited to a 90° inclination. Nevertheless, lower inclinations and alternate trajectories may combine to exceed acceptable dose limits, and any proposed multiple-pass grazing reentry path should be evaluated in detail when such a rescue mission is considered.

9.4.4 Summary

The shielding effectiveness of the assumed Orbiter structure against trapped radiation meets the needs of all augmented performance EOS configurations examined. Over six days can be spent in a circular orbit at about 800 nmi (1500 km), the maximum altitude generally attainable with an unrefueled

Table 9-3. Lunar Return Multiple-Pass Orbit Characteristics and Radiation Exposure Behind 2 g/cm^2 Aluminum Shielding (no perigee assist)

Orbit No.	Apogee Alt.		Perigee Alt.		Approx. Period, hr	Incremental Dose, rem
	n mi	km	n mi	km		
1	208,000	385,000	~ 45	~ 83	--	--
2	137,000	254,000	~ 45	~ 83	134	0.09
3	67,000	124,000	~ 45	~ 83	50	0.24
4	42,000	77,800	~ 45	~ 83	26.7	0.33
5	29,000	53,700	~ 45	~ 83	16.6	0.13
6	20,500	38,000	~ 45	~ 83	11.2	0.15
7	15,000	27,800	~ 45	~ 83	8.0	0.20
8	10,800	20,000	~ 45	~ 83	5.8	0.26
9	7,600	14,100	~ 45	~ 83	4.3	0.38
10	4,900	9,100	~ 45	~ 83	3.2	0.68
11	2,500	4,600	~ 45	~ 83	2.3	1.32
12	1,000	1,850	--	--	1.7	0.18

Total Dose $\cong$ 4 rem

Orbiter. For the refueled Orbiter EOS-launched Tug case, the Configuration B Orbiter can remain at the staging/retrieval altitude in excess of three weeks. Even multiple-pass grazing reentry from lunar orbit with as many as 11 passes through the earth's radiation belts falls below the acceptable crew dose limit.

9.5 GROUND-LAUNCHED ASCENT/RENDEZVOUS TIME

9.5.1 In-Plane Ascent

A general analysis was made of the time required for the worst-case, in-plane ascent of an EOS from ETR to a rendezvous with a distressed vehicle. This case represents the simultaneous occurrence of the worst in-plane delay and the worst parking orbit delay which combine to give the maximum time required for an in-plane ascent. Previous effort (see Reference 2) had addressed only the specific case of a 270 nmi (500 km), 55° target.

A normal EOS ascent was assumed; that is, launch into a 50 × 100 nmi (90 × 185 km) orbit which is circularized into a 100 nmi (185 km) parking orbit and, after an appropriate phasing delay, a Hohmann transfer to the final orbit and rendezvous with the distressed vehicle. Only inclinations >28.4° were considered. The results are shown in Figures 9-10a and b.

Large ΔVs are obviously required to attain altitudes beyond LEO. With any of the four configurations considered the maximum Orbiter capability is barely in excess of 800 nmi (1500 km), even with increased propellant loading (see Figure 5-3). The region of Figure 9-10a below 800 nmi (1500 km) has, therefore, been plotted to an expanded scale, Figure 9-10b. When information from Figures 5-3 and 9-10 a and b is combined, the maximum time for ascent and rendezvous at the Orbiter's highest altitude capability can be determined. With a 10 klb (4.5 t) rescue payload this duration varies from one to two days, depending upon EOS configuration. At lower altitudes even longer times are involved.

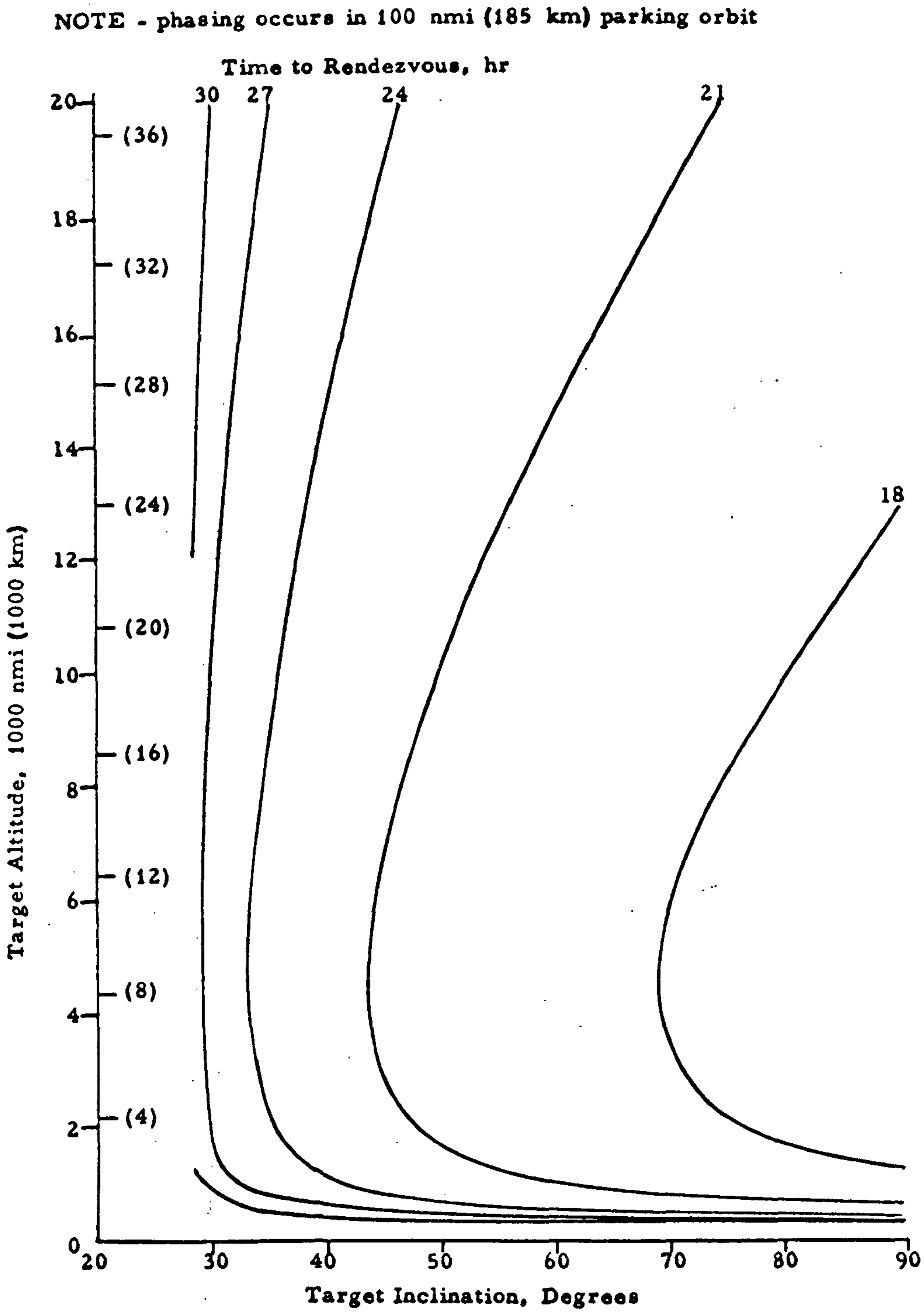


Figure 9-10a. Worst-Case Time to Rendezvous for In-Plane Ascent - Target Altitude 100-20,000 nmi (185-37,000 km)

NOTE - phasing occurs in 100 nmi (185 km) parking orbit

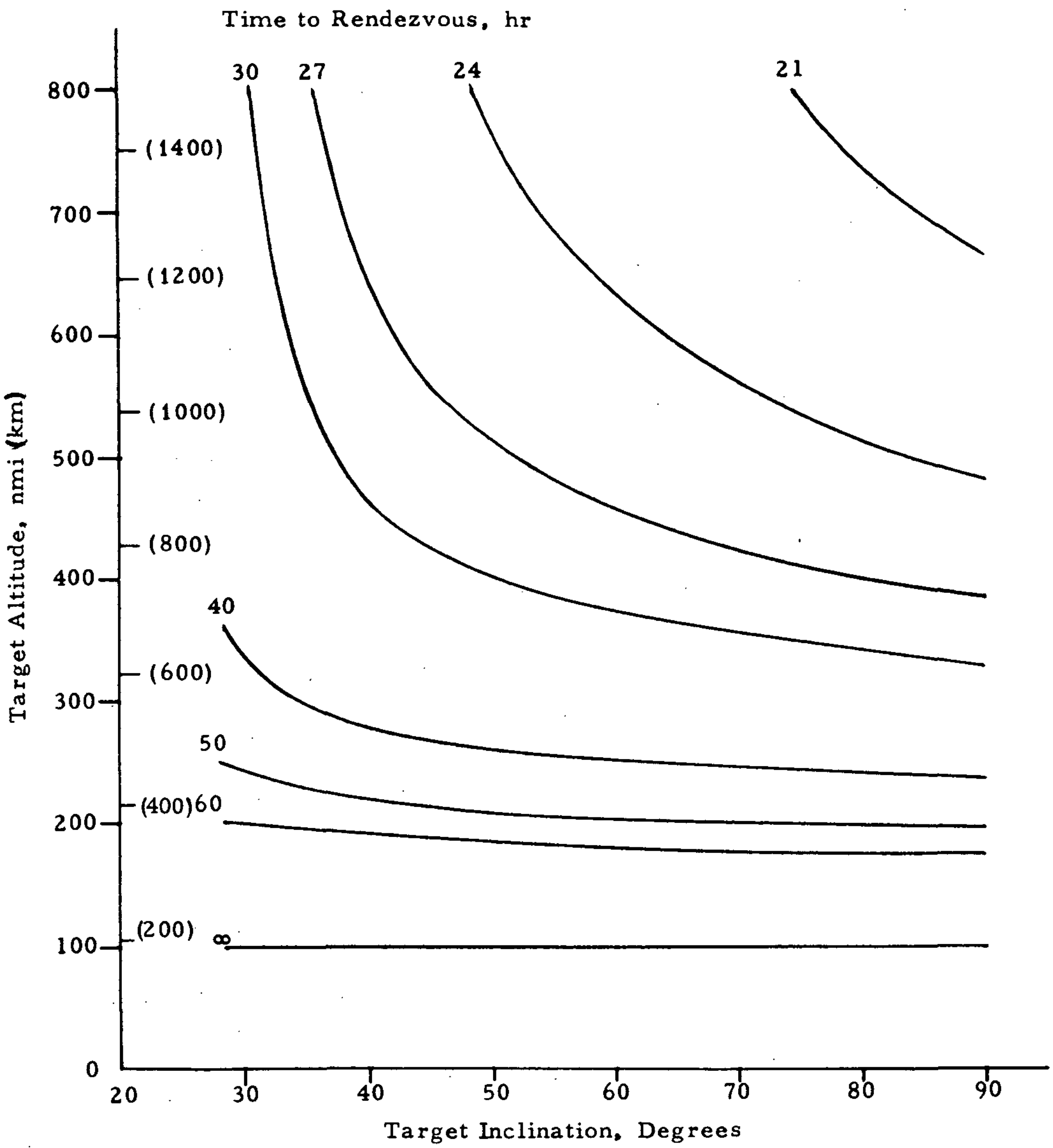


Figure 9-10b. Worst-Case Time to Rendezvous for In-Plane Ascent - Target Altitude 100-800 nmi (185-1480 km)

9.5.2 Ascent with Excess ΔV

Some reduction in the worst-case ascent and rendezvous time is possible if ΔV in excess of that needed for a standard in-plane ascent is available. One technique (as discussed in Reference 2) is to combine parking orbit phasing with a plane change. Another technique, which in some cases offers more effective use of available ΔV, is to enter a phasing orbit at an altitude above the target orbit. As a third alternate, direct ascent can be employed, but only if a sufficiently large ΔV margin exists.

Representative trends for the worst-case time with excess ΔV are plotted in Figure 9-11. At each value of excess ΔV the most effective mode of ΔV expenditure was selected. At a 55° inclination and over an altitude range of from 100 to 500 nmi (185 km to 925 km) even a large ΔV expenditure does not reduce the worst-case ascent and rendezvous time below ~18 hr.

9.5.3 Summary

The worst-case, in-plane ascent and rendezvous duration is independent of EOS configuration. Any excess ΔV available can be used to reduce this time period. Thus, Configuration A, which has the greatest OMS capacity and largest ΔV excess, would give the maximum time reduction.

In general, the worst-case ascent and rendezvous time to the maximum Orbiter altitude for all four EOS configurations is between one to two days. The longer times are for low altitudes. With increased propellant loading the low altitude time is approximately halved.

A detailed discussion is given in Appendix H, Volume III Part 1.

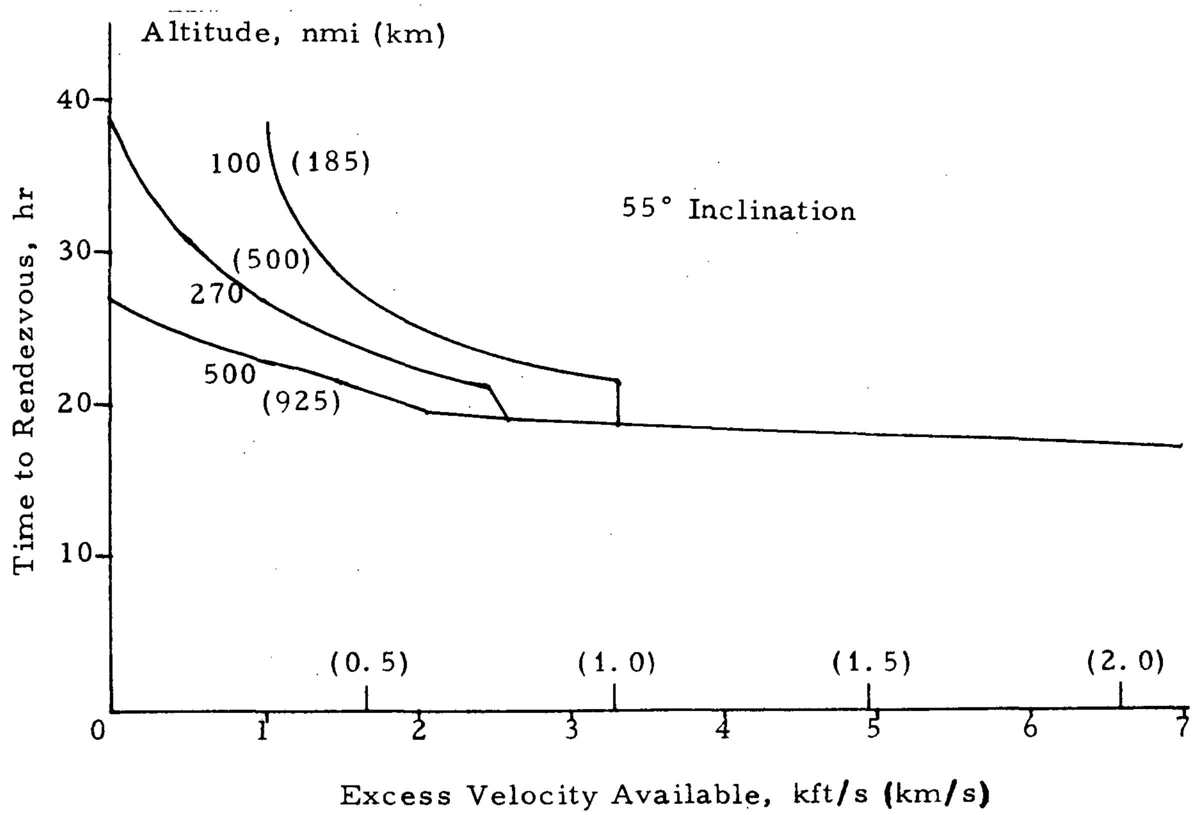


Figure 9-11. Worst-Case Time to Rendezvous with Excess Available ΔV

10. CAPABILITY SUMMARY

10.1 PERFORMANCE COMPARISON

10.1.1 General

A summary matrix of the basic and augmented performance for the initial four EOS configurations is given in Table 10-1. (Corresponding data for the parallel-burn Space Shuttle configuration are given in Appendix J, Volume III, Part 1.) In general, each augmentation mode represents a different level of capability. Increased propellant loading is the least effective method for increasing EOS rescue mission capability. Orbiter refueling in LEO is the next level of capability, and, when combined with an Orbiter-launched Tug, offers the greatest capability of all.

Best augmentation possibilities occur with the parallel-burn Space Shuttle and with Configuration B. Configuration A imposes a long refueling timeline, whereas the size of the cargo bay of Configurations C and D forces a reduction in Tug size and capability.

10.1.2 Increased Propellant Loading

Increased propellant loading is useful for LEO rescue missions. Depending on the EOS configuration, a ΔV increase between 1 to 2 kft/s (0.3 to 0.6 km/s) can be obtained.

10.1.3 Orbital Refueling

Orbital refueling in LEO offers a rescue mission capability to geosynchronous and lunar orbits. The refueled Orbiter can be placed in lunar orbit with both a fully fueled Tug and a rescue payload weighing 10 klb (4.5 t) in the cargo bay.

Such missions require multiple-pass grazing reentry because the remaining ΔV is insufficient for returning the Orbiter to LEO. The procedure appears

Table 10-1. Summary of Basic and Augmented Shuttle Performance

Payload klb(t)	Basic Performance				Increased Propellant Loading				Orbital Refueling		Three-Stage EOS		
	28.4°		90°		28.4°		90°		ΔV @ OPD, kft/s (km/s)		Tug ΔV @ 100 nmi, kft/s (185 km, km/s)		Refueled ΔV @ OPD, kft/s (km/s)
	Max Alt nmi (km)	ΔV @ 100 nmi, ft/s (185 km, m/s)	Max Alt nmi (km)	ΔV @ 100 nmi, ft/s (185 km, m/s)	Max Alt nmi (km)	ΔV @ 100 nmi, ft/s (185 km, m/s)	Max Alt nmi (km)	ΔV @ 100 nmi, ft/s (185 km, m/s)	No Cargo Tank	Cargo Tank	28.4°	90°	
Config A													
0 (0)	755 (1400)	1850 (565)	640 (1185)	1850 (565)	835 (1545)	3900 (1190)	515 (955)	2220 (678)	17.6 (5.4)	20.6 (6.3)	34.3 (10.5)	27.7 (8.5)	49.7 (15.2)
10 (4.5)	725 (1340)	1760 (537)	525 (970)	1760 (537)	760 (1405)	3540 (1080)	450 (833)	1840 (560)	17.2 (5.3)	19.4 (5.9)	22.4 (6.8)	13.5 (4.1)	37.4 (11.4)
70 (31.8)	--	--	--	--	--	--	--	--	15.1 (4.6)	--	--	--	--
Config B (std)													
0 (0)	470 (870)	1060 (323)	470 (870)	1060 (323)	885 (1640)	3540 (1080)	540 (1000)	2040 (622)	23.1 (7.1)	26.9 (8.2)	34.3 (10.5)	29.9 (9.1)	54.1 (16.5)
10 (4.5)	445 (825)	970 (296)	445 (825)	970 (296)	785 (1450)	3130 (955)	455 (840)	1640 (500)	22.5 (6.9)	25.3 (7.7)	22.4 (6.8)	15.7 (4.8)	41.7 (12.7)
70 (31.8)	--	--	--	--	--	--	--	--	19.5 (6.0)	--	--	--	--
Config B (aux)													
0 (0)	885 (1635)	2180 (665)	705 (1305)	2180 (665)									
10 (4.5)	825 (1525)	2030 (620)	555 (1020)	2030 (620)									
Config C													
0 (0)	405 (750)	850 (259)	405 (750)	850 (259)	830 (1535)	3400 (1040)	485 (900)	1820 (555)	22.9 (7.0)	26.1 (8.0)			
10 (4.5)	380 (704)	760 (232)	380 (704)	760 (232)	695 (1285)	2280 (695)	365 (675)	1230 (375)	22.2 (6.8)	23.6 (7.2)			
Config D													
0 (0)	415 (764)	890 (272)	415 (764)	890 (272)	880 (1625)	4550 (1385)	575 (1065)	2880 (880)	21.6 (6.6)	24.8 (7.6)			
10 (4.5)	355 (657)	690 (210)	355 (657)	690 (210)	N/A (N/A)	N/A (N/A)	N/A (N/A)	N/A (N/A)	20.1 (6.1)	--			

feasible, and the total return time can be shortened by a small retro-ΔV application at the first perigee to reduce the number of passes before reentry.

The duration of such a lunar mission, excluding the refueling time, would be in the order of 7 to 17 days. For the geosynchronous case the mission duration would be in the order of 4 to 14 days.

It should be noted that the parallel-burn Space Shuttle design offers the possibility of achieving a lunar or a geosynchronous round trip with a 10 klb (4.5 t) payload and without the need for aerodynamic braking before return to low earth orbit.

10.1.4 EOS-Launched Tug

A Tug carried to and launched from LEO by the Orbiter also has geosynchronous and lunar orbit capability (Configurations A, B, and parallel-burn Space Shuttle only). The Tug is capable of a round trip from LEO with a small rescue payload not exceeding about 4 klb (1.8 t). With a four-stage configuration (tandem Tugs) this payload can be raised to 10 klb (4.5 t). The mission duration, excluding the time for rescue operations, is in the order of 7 days for the lunar mission and 3 days for the geosynchronous mission.

10.1.5 Orbital Refueling + EOS-Launched Tug

If orbital refueling in LEO is combined with an EOS-launched Tug, the rescue mission payload in lunar orbit can be increased to about 10 klb (4.5 t). The Tug staging/retrieval altitude is between 3800 and 6900 nmi (7000 and 12,800 km) and normal Orbiter reentry is employed.

If, instead, the Orbiter is placed in lunar orbit (see paragraph 10.1.3), it can deliver a fully fueled Tug plus the 10 klb (4.5 t) payload.

10.2 TECHNICAL FEASIBILITY

10.2.1 General

Certain EOS design features influence the utility and adaptability of the Orbiter to the augmentation modes considered. Items which fall into this category are listed in Figure 10-1.

- ORBITER MAIN ENGINE SYSTEM
  - — INSTALLATION NOT DESIGNED FOR RESTART
  - — NOT INTENDED TO BE OPERATED SINGLY
- OMS TANKS/ENGINES
  - — SEPARATE PROPELLANT SYSTEM
  - — ISOLATED FROM MAIN PROPELLANT SYSTEM
  - — ONLY SOURCE OF ON-ORBIT ΔV
- ORBITER MAIN TANKS NOT INSULATED FOR EXTENDED MISSIONS
- LIMITED ORBITER REENTRY CAPABILITY (LEO)
- RESCUE MODULE ACCESSIBLE VIA CARGO BAY TUNNEL

Figure 10-1. EOS Design Features Impacting Adaptability to Rescue Operations

Heading this general list is the Orbiter main engine. These engines are intended for long life and multiuse, and the planned engine design will allow both restart and single engine operation. The currently planned Orbiter engine installation, however, does not permit such operation. System installation changes needed to acquire these capabilities appear feasible.

OMS tanks and engines represent the only source of sizable on-orbit ΔV. Even if identical OMS and main engine propellants are used (Configurations B and C use different propellants), no provision is currently planned for feeding unused main tank propellants to the OMS engines. With identical propellants, an increase in on-orbit ΔV, now limited by OMS tank capacity, could be obtained. Such an increase would be beneficial for rescue missions and would also improve general Orbiter utility. Feeding unused main tank propellants to the OMS engines is feasible, but propellant boil-off may be excessive.

The Orbiter (including its main propellant tanks) is not designed for extended space missions. Consequently, any performance augmentation mode which depends on main tank storage beyond the basic ascent phase can suffer large propellant storage losses.

Since the Orbiter is basically designed for LEO activity, any augmentation scheme that raises its operating altitude range imposes many new requirements. One prominent consideration is Orbiter reentry capability. Direct reentry with little or no crossrange allows some increase in operating altitude. For return from lunar or geosynchronous missions, multiple-pass grazing reentry appears feasible.

A manned rescue module is among the payloads that may be carried in the cargo bay. In-flight accessibility of the manned rescue module from the crew compartment would be via the cargo bay tunnel.

10.2.2 Augmentation Mode

Each augmentation mode imposes a unique situation. In addition, complementary hardware items are required. The following paragraphs identify some of the considerations that fall into these two categories.

10.2.2.1 Increased Propellant Loading

The cargo bay tank required with this approach appears technically feasible and is currently included among possible Orbiter payloads.

An appropriate fluid interface for fill, vent, and transfer of the propellants to the OMS engines is also required and could be incorporated in the payload interface panel.

10.2.2.2 Orbital Refueling

Augmentation through orbital refueling obviously requires a large propellant source. Possibilities include an OPD or a caravan of propellant logistics flights of the EOS or an ESS configuration. Any of these approaches appears technically feasible.

Because the refueling period may be long, main tank exchange is preferred to direct propellant transfer from the donor to the Orbiter. (Tank exchange is not feasible with designs represented by Configuration A.) In either case, the appropriate modifications to the Orbiter are considered technically feasible.

Refueling both the main engine and OMS propellant tanks has an added complication if each uses different propellants. Configurations B and C fall into this category. Nevertheless, if required, multiple-propellant refueling would be possible.

10.2.2.3 EOS-Launched Tug

The man-rated Tug required for this augmentation mode has been extensively examined in other studies and is considered technically feasible.

Current EOS design criteria include such a reusable Tug as a payload. It is anticipated, therefore, that the necessary capability for handling, launching, and retrieving a Tug in space will be included in the Orbiter design.

11. COST

11.1 GENERAL ASSUMPTIONS

In estimating the costs associated with each EOS performance augmentation technique, it was assumed that certain necessary hardware elements were already available. For example, orbital refueling from an OPD would not be undertaken on a rescue mission unless an OPD had already been developed and was operational. Similarly, an EOS-launched Tug mission would not be undertaken unless an operational Tug, compatible with the Orbiter, was already in the inventory. Also, an ESS for direct in-space propellant transfer to a spent Orbiter would not be developed merely for this application, but, if available, its use would be considered.

It was also assumed that if such hardware is in the inventory, then the necessary compatibility modifications have already been incorporated in the EOS. Only the incremental costs for rescue mission-peculiar needs are considered here. In addition, neither the EOS propellant cost in performing the mission nor the actual cost of the rescue payload was considered.

On this basis, only the costs for the increased propellant loading case needed to be examined. However, because Orbiter refueling represents interesting possibilities and only OPD costs are available in the existing literature, the additional EOS costs to allow orbital refueling via propellant transfer as well as via tank exchange were also examined.

11.2 ESTIMATED COSTS

Detailed estimating ground rules and cost values are discussed in Appendix I, Volume III Part 1.

11.2.1 Increased Propellant Loading

A summary of the costs associated with acquiring increased propellant loading capability for each EOS configuration is given in Table 11-1. The major

Table 11-1. Estimated Costs for Increased Propellant Loading
(Millions of 1971 Dollars)

Configuration	Tank Size, ft (m)	RDT&E	Unit Total*
A	15 x 60 (4.6 x 18.3)	114	3.2
	15 x 40 (4.6 x 12.2)	93	2.4
B	15 x 60 (4.6 x 18.3)	114	3.2
	15 x 40 (4.6 x 12.2)	93	2.4
C	12 x 40 (3.7 x 12.2)	81	1.7
	12 x 20 (3.7 x 6.1)	60	0.9
D	10 x 20 (3.0 x 6.1)	52	0.7

*includes manufacturing, spares, engineering, tooling support, and program management costs for added modifications portion of the expense is for RDT&E, which includes ground and flight testing. The unit recurring cost is relatively small.

The major effect on both non-recurring and recurring cost is tank size as established by available cargo bay dimensions. Thus, if allowance is made for a rescue payload, or if smaller Orbiter vehicles are used, the cost is reduced.

11.2.2 Orbital Refueling

The estimated costs for modifying the EOS second stage to accommodate both LEO propellant transfer into the main tank from a donor and resumed operation after refueling are given in Table 11-2a.

Similar costs for orbital refueling via the main tank exchange mode are given in Table 11-2b. It should be noted that this latter mode is not applicable to Configuration A, which has integral Orbiter tanks.

Modifications are necessary to both the Orbiter and the separable main tank to allow on-orbit tank exchange and continued second-stage operation. As a result, this mode of refueling, although much less time-consuming than propellant transfer (see section 6.2), is much more expensive to acquire.

The cost per flight of propellant delivery into LEO by EOS and by ESS is given in Table 11-3. By application of these values and the flight frequencies given in Tables 6-2 and 6-3, the propellant delivery cost for a single refueling operation was determined, per Table 11-3. It was assumed that the same number of propellant delivery logistics flights would be required with either direct propellant transfer from donor to Orbiter or via an OPD.

The cost of a single Orbiter refueling can approach and may even exceed $100 million, regardless of whether the EOS or an ESS is used for propellant delivery. This is especially so if propellant cost and hardware amortization are considered, and if dedicated vehicles are acquired to ensure propellant

Table 11-2. Estimated Orbital Refueling Costs
(Millions of 1971 Dollars)

(a) Propellant Transfer Mode

	RDT&E	Added Unit Cost*
All Configurations	45	1.5

(b) Main Tank Exchange Mode

Configuration	RDT&E	Added Unit Cost*
B	318	13.8
C	302	12.5
D	270	9.9

* includes manufacturing, spares, engineering, tooling support, and program management costs for added modifications

Table 11-3. Estimated Propellant Delivery Cost*
(Millions of 1971 Dollars)

(a) Earth Orbit Shuttle Delivery

Configuration	No. of Flights	Cost/Flight**	Delivery Cost/Refueling
A	9	4.5	40
B	11	7.5	83
C	12	7.0	84
D	13	4.0	52

(b) Expendable Second Stage Delivery

Configuration	No. of Flights	Cost/Flight***	Delivery Cost/Refueling
A	3	33	99
B	4	33	132
C	4	33	132

* to LEO and 28.4°

** ~800-flight program; does not include hardware amortization

*** Reference 9; adjusted to 1971 dollars delivery without delay. With such high propellant delivery costs, the expense of a single Orbiter refueling exceeds the RDT&E cost for the propellant transfer mode and represents over one third the cost of RDT&E for the tank exchange mode.

11.2.3 EOS-Launched Tug

The basic EOS design is capable of delivering, launching, and retrieving a Tug in LEO. Thus, if a Tug is already developed, no additional expenditures are anticipated in applying this system to the rescue mission.

If the Orbiter is refueled before Tug staging, the costs given in section 11.2.2 also apply here.

11.3 COST SUMMARY

It is unlikely that any performance augmentation mode will be specifically acquired for the sole purpose of improving only the EOS rescue mission capability. The utility and capability represented by a performance-augmented Orbiter will be considered for additional applications as well and will be balanced against acquisition cost.

It was assumed that a Tug or an OPD would be used to augment basic rescue mission capability only if already available. On this basis, the EOS-launched Tug represents the lowest additional-cost approach in augmenting Orbiter performance. Orbital refueling via propellant transfer is the least costly new capability, with an acquisition cost somewhat lower than increased propellant loading. Orbital refueling via main tank exchange is the most costly performance augmentation mode to acquire.

The highest operational cost also occurs with orbital refueling. Propellant delivery cost to LEO is less with the Shuttle than with an ESS configuration; but even with direct propellant transfer from logistics vehicles to the Orbiter, propellant delivery cost will probably exceed $100 million for a single refueling.

12. SUMMARY AND CONCLUSIONS

The results of this study task indicate the feasibility of using the Orbiter, with some modification, as a space rescue vehicle not only in the vicinity of the earth but also for rescue from lunar orbit. Obviously, any Orbiter capability improvement would be of use to other missions as well.

This section provides a summary overview of the more significant results presented in detail throughout the report.

12.1 GENERAL

Three augmentation modes for increasing Shuttle rescue mission capability were examined. They are:

(1) Increased propellant loading (cargo bay tank)
(2) On-orbit refueling
(3) EOS-launched Tug

The effectiveness of each mode is influenced by certain Shuttle design and operating characteristics. They include:

(1) Orbiter staging velocity
(2) Payload bay dimensions
(3) Allowable payload weight
(4) Integral or droppable Orbiter main tanks
(5) OMS ΔV capacity

Large payload bay dimensions and allowable payload weight are preferred. Also, a Mark II type of design with droppable Orbiter tanks and a lower staging velocity is preferred over a fully reusable integral tank design which has identical payload bay dimensions and payload weight specifications. Raising the OMS ΔV capacity from a value of 1 kft/s (0.3 km/s) to a value of 2 kft/s (0.6 km/s) is also desirable.

All three augmentation modes are considered technically feasible. The required Orbiter design changes vary with each augmentation technique. Each mode has its unique capability and resulting region of utility. Thus, one approach is not necessarily a competitive alternate to the other two modes. Instead, selection should be made on the basis of whether the capability represented by the specific mode is desired.

12.2 INCREASED PROPELLANT LOADING

Adding a cargo bay tank to augment Orbiter capability is limited to LEO applications. A ΔV increase between 1 to 2 kft/s (0.3 to 0.6 km/s) is obtained at low-orbit altitudes and inclinations while also carrying a rescue payload weighing 10 klb (4.5 t). This ΔV increase falls rapidly as the mission altitude and inclination are raised.

12.3 ON-ORBIT REFUELING

Refueling the Orbiter in LEO is useful for synchronous and lunar rescue mission applications. An OPD is the most practical means of refueling the Orbiter. Although propellant transfer is an acceptable refueling mode, tank exchange (full for empty) is preferred, when feasible, in order to reduce the time involved.

All refueled Orbiters have one-way lunar rescue mission capability with a 10 klb (4.5 t) rescue payload. Once the Orbiter is in lunar orbit, the remaining Orbiter ΔV is sufficient for transearth injection but insufficient (except for the parallel-burn Space Shuttle configuration) for earth-orbit insertion as well. Multiple-pass grazing reentry offers a means for earth return in this situation.

Current Orbiter TPS designs and radiation shielding appear adequate for multiple-pass grazing reentry and even allow retention of crossrange capability. The long reentry duration can be markedly reduced by applying a small retro-ΔV on the first perigee pass.

12.4 EOS-LAUNCHED TUG

A three-stage system involving a Tug launched and retrieved by the EOS is also technically feasible and represents a method of achieving round-trip lunar mission rescue capability. An Orbiter with Mark II characteristics can accommodate an appropriately-sized Tug plus a rescue payload.

If the Tug is staged and retrieved in LEO, a Tug lunar round-trip rescue payload of ~4 klb (1. 8 t) could be achieved. By refueling the Orbiter in LEO and then raising the Tug staging/retrieval altitude, the round-trip rescue payload weight could be increased to 10 klb (4. 5 t). After the Tug has been retrieved, the Orbiter altitude is reduced to 100 nmi (185 km) for a normal reentry.

A further step in using the Space Tug to augment Shuttle capability is to join two Tugs in tandem and launch them from low-earth orbit. A round trip, lunar-orbit rescue mission with a 10 klb (4. 5 t) payload can be accomplished with this four-stage configuration, without requiring refueling of the Orbiter in low-earth orbit or Orbiter operation at altitudes above 100 nmi (185 km).

In an alternate mode of operation with a refueled Orbiter, Configuration B (see paragraph 3. 2. 2) has the capability of not only placing a 10 klb (4. 5 t) payload in lunar orbit (see section 12. 3) but of simultaneously placing a fully loaded Tug in lunar orbit as well. This added gain in rescue utility requires multiple-pass grazing earth reentry, and the Orbiter ΔV reserve of ~4 kft/s (1. 2 km/s) appears adequate to accommodate the Tug and rescued crew during such a return. The parallel-burn Space Shuttle configuration offers a greater payload and/or mission ΔV capability for this same refueled mode of operation.

13. RECOMMENDATIONS

In view of the conclusions reached, a number of recommendations are appropriate. They are offered from the perspective of EOS rescue mission utility and are grouped into three general categories.

13.1 SHUTTLE DESIGN CONSIDERATIONS

In this general area, it is recommended that:

(1) A drop-tank Orbiter design (similar to Mark II) be encouraged.

(2) The Mark II cargo capability not be reduced.

(3) The 15×60 ft (4.6×18.3 m) cargo bay size not be reduced.

(4) An OMS capacity of 2 kft/s (0.6 km/s) be adopted.

(5) Consideration be given to increasing on-orbit stay time of the Orbiter.

(6) Consideration be given to including Orbiter main engine system restart provisions. (The basic engine design will accommodate restart.)

(7) Consideration be given to including Orbiter single main engine operation capability.

(8) On-orbit refueling provisions be included in the Orbiter design.

(9) Compatibility of the payload fluid interface panel be ensured with a cargo bay propellant tank installation and propellant transfer to either the OMS or main engines.

13.2 OPERATIONAL PREFERENCE

Orbital refueling offers the greatest improvement in Shuttle utility. For this augmentation mode, it is recommended that:

(1) An OPD is preferred to direct refueling from logistics vehicles.

(2) If feasible, Orbiter tank exchange (full for empty) is preferred to propellant transfer.

(3) The same propellants be used in the OMS as in the main engine system and alternate feed from the main tanks be provided.

(4) Multiple-pass grazing reentry be considered to reduce the lunar and geosynchronous mission total ΔV requirement.

Using the EOS to launch/retrieve a Tug offers a unique capability, whether or not the Orbiter is simultaneously refueled. It is recommended that:

(1) The Tug be sized to allow simultaneous Orbiter delivery to LEO of both the Tug and a rescue module.

(2) The Tug be man-rated and designed for use with a 10 klb (4.5 t) rescue module plus specialized rescue equipment.

13.3 STUDY AREAS

It is recommended that future study efforts regarding the use of the EOS for rescue missions include the following considerations:

(1) Extending Orbiter main tank propellant storage duration

(2) Orbiter refueling by on-orbit main tank exchange (full for empty)

(3) Feasibility of burning excess main tank propellants in OMS engines

(4) Design of a cargo bay tank installation for extended storage duration

(5) Lunar and geosynchronous mission return trajectories for multiple-pass grazing reentry (ΔV requirements, transit time, departure and arrival timing, radiation exposure)

(6) Guidance and control requirements for multiple-pass grazing reentry

(7) TPS requirements for multiple-pass grazing reentry

(8) Control over landing site with multiple-pass grazing reentry

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15. SYMBOLS AND ABBREVIATIONS

AMU	Astronaut Maneuvering Unit
CIS	Chemical Interorbital Shuttle
C_L	Lift Coefficient
EO	Earth Orbit
EOS	Earth Orbit Shuttle
ESS	Expendable Second Stage
ETR	Eastern Test Range
EVA	Extra-Vehicular Activity
IDA	Institute for Defense Analyses
L/D	Lift-Drag Ratio
LEO	Low Earth Orbit
LO	Lunar Orbit
OMS	Orbital Maneuvering System
OOS	Orbit-to-Orbit Shuttle
OPD	Orbiting Propellant Depot
P/L	Payload
REI	Reusable External Insulation
RNS	Reusable Nuclear Shuttle
SRV	Space Rescue Vehicle
TEI	Transearth Injection
TPS	Thermal Protection System
W/S	Wing Loading
WTR	Western Test Range
α	Angle of Attack

16. DIMENSIONS

ft	foot
ft/s	foot per second
kft/s	kilofoot per second
°F	degree Fahrenheit
hr	hour
lb	pound
klb	kilopound
lb/ft^2	pound per square foot
nmi	nautical mile
g/cm^2	gram per square centimeter
kg	kilogram
°K	degree Kelvin
m	meter
m/s	meter per second
km	kilometer
km/s	kilometer per second
N	Newton
N/m^2	Newton per square meter
t	metric ton = 1000 kg