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October 1975

Mars Surface Sample Return Tradeoff Studies

MARTIN MARIETTA

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Contract 954205

MARS SURFACE SAMPLE RETURN

TRADEOFF STUDIES

Final Report

October 1975

Approved

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ABSTRACT

This report documents the results of four specific study tasks defined by the JPL study manager as critical to the evolving understanding of the Mars Surface Sample Return (MSSR) mission.

Task 1 compares the Mars mission opportunities in 1981, 1983/84, 1986, 1988, and 1990 to determine which mission modes are possible and appropriate for each.

Task 2 examines the design features of the hardware systems used to return the sample, in the Mars orbit rendezvous mode, to identify ways in which the probability of back contamination can be minimized.

Task 3 looks into the hardware and performance trade offs between the options of direct entry of the returning sample capsule at Earth and the orbital capture of that capsule for recovery by the Shuttle.

Task 4 explores the possibilities for increasing the landed weight at Mars, beyond that possible with minimally modified Viking lander, to support MSSR mission modes involving heavier systems.

FOREWORD

These technical memoranda were prepared in accordance with the Contract Schedule, Article 1, Paragraph a.(3), of JPL Contract No. 954205, Mars Surface Sample Return (MSSR) Tradeoff Studies. A technical memorandum is submitted for each of the four tasks in the Statement of Work as follows:

Task 1: Compile mission performance data for the leading mission mode candidates for the 1981, 1984, 1986, 1988 and 1990 opportunities. Single and dual shuttle launches, in-orbit weights, landed weights, Earth return weights, and ΔV budgets shall be considered. Titan IIIE/Centaur launches are included for comparison purposes only, but not emphasized.

Task 2: Identify and describe back contamination control options that could be incorporated into the Mars Ascent Vehicle (MAV) rendezvous and docking, docking and sample transfer scheme, orbiter, Earth Return Vehicle (ERV) and Earth recovery hardware and operational sequences.

Task 3: Perform tradeoffs for preliminary performance and hardware definitions to compare the direct entry and orbital capture modes for Earth recovery.

Task 4: Define and quantify available options for increasing landed weight.

Appendices A through E are technical notes completed during the course of this work that provide additional detail and substantiation for assumptions made in performing the contract trades.

ACKNOWLEDGEMENT

Martin Marietta wishes to recognize the contributions of the following individuals to this study:

Louis D. Friedman of the Jet Propulsion Laboratory, Technical Representative of the Contracting Officer, for management and direction.

Arthur L. Satin - Mission Analysis

Norm Phillips - Back Contamination Design Considerations and Illustrations

Scott K. Asnin - Mission Analysis

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GLOSSARY

ACS	Attitude Control System
EEC	Earth Entry Capsule
ERV	Earth Return Vehicle
IUS	Interim Upper Stage
L/D	Lift-to-Drag Ratio
LD/ED	Launch Date/Encounter Date
L/V	Launch Vehicle
M	Mach Number
MAV	Mars Ascent Vehicle
M/C	Midcourse
MOI	Mars Orbital Injection
MOR	Mars Orbital Rendezvous
MSSR	Mars Surface Sample Return
P/I	Propellant Inerts
P/L	Payload
q	Dynamic Pressure
RCS	Reaction Control System
VIS	Video Imaging System
*MSNS	Mission Number

LIST OF SYMBOLS

C_3	Earth Departure Energy
DLA	Declination of Launch Azimuth
hp	Periapsis Altitude
H_T	Terrain Height
Isp	Specific Impulse
Vhp	Hyperbolic Excess Velocity
V_E	Entry Velocity
W_E	Entry Weight
W_{ERV}	ERV Weight
W_{INS}	Insertion Weight
W_{LIFT}	Liftoff Weight
W_{LND}	Landing Weight
W_{LD}	Landed Weight
W_P	Propellant Weight
γ	Flight Path Angle
γ_E	Entry Flight Path Angle
ΔV	Delta-V
ΔV_{EOI}	Earth Orbital Entry ΔV
ΔV_{MOI}	Mars Orbital Injection ΔV
ΔV_{STAT}	Statistical ΔV
ΔV_{TEI}	Trans-Earth Injection ΔV
$\Delta\gamma$	Entry Corridor

I. INTRODUCTION

The Mars Surface Sample Return (MSSR) mission has been recognized by NASA mission strategists and planetary scientists alike as the most attractive and potentially valuable unmanned Mars exploration mission. The mission has been studied by a number of groups in recent years (References 1 and 2 are examples of previous work).

Several problems and challenges emerged from previous examinations of the mission that prompted Dr. Louis Friedman, JPL study manager for this contract, to request the four study tasks reported here.

The first of these is that the projected cost of an MSSR mission makes it very difficult to fit the program into the NASA new start plan. Therefore, it appeared advisable to examine all the mission opportunities in the 1980-1990 time period to see if there were launch years offering more potential than others. This information could then be factored into the mission planning process. Task 1 of this study is aimed at this objective. Task 1 also defines and examines the mission modes that are made possible by and are compatible with the Shuttle-IUS and Shuttle-Tug launch systems.

Another problem encountered in planning the MSSR mission is minimizing the probability of bringing Mars organisms back into the Earth biosphere in an uncontrolled condition. Task 2 of this study addresses the hardware design and operation features that could be incorporated into Mars Ascent Vehicle (MAV), the Earth Return Vehicle (ERV) and the hardware and sequences used in rendezvous and docking and recovery at Earth, to minimize back-contamination probabilities.

Task 3 derives also from the concerns over back contamination. It compares the hardware implementation requirements for two methods of recovering the sample capsule at Earth. One approach is to allow the capsule to enter the Earth's atmosphere directly from the Mars to Earth trajectory as was done with the Apollo returns from the Moon. The other option is to retro thrust the capsule into Earth orbit for subsequent retrieval by a Shuttle-based system. The choice between these options will ultimately be made on the basis of cost, back-contamination control and reliability.

Because of the greater potential MSSR performance capabilities made possible by the Space Transportation System, mission modes can be considered that require larger landed weights on Mars. Task 4 of this study examines possibilities for increasing landed weights beyond the levels allowed by Viking-derived systems.

This report, as prescribed by the contract statement of work, is comprised of four technical memoranda, one for each of the study tasks, and an assessment of the technology and program implications to a MSSR mission. Several other technical notes are appended to provide supporting data. Three of these notes, Appendices C, D and E, contain entry and descent data and are included to support the increased landed weight study (Task 4). The mission performance data contained in these particular notes are not in context with the performance data generated for the current study in Task 1.

II. TASK 1 MISSION PERFORMANCE DATA FOR VARIOUS MARS SAMPLE RETURN MODES

A. INTRODUCTION

This task was designed to explore the possibility of using the Space Tug and/or the Interim Upper Stage (IUS) to enhance MSSR mission reliability. Increased reliability would presumably result if the system mass margins are increased or the total mission time is shortened. The latter is accomplished by allocating the additional payload margin to the Earth return vehicle propellant load. The additional load allows non-optimum (non-conjunction) Mars departures and hence shorter stopover times and mission times. Previous MSSR studies with the Titan IIIE/Centaur launch system (Refs. 1 and 2) show that total mission times of the order of 1000 days are required primarily due to the long wait at Mars (stopover 400 days) while awaiting the minimum energy Mars-Earth geometry. If near minimum energy transfers are not required, only 30 days need be spent at Mars performing the baseline Mars orbital rendezvous (MOR) mission mode.

This study considered 5 different mission modes, 5 launch years (1981, 1983/4, 1986, 1988 and 1990), and 3 launch vehicles (Titan IIIE/Centaur, Shuttle/Tug and Shuttle/IUS). The mission modes were:

1. Single launch, out-of-orbit landing, direct return (SL, OO, DR)
2. Single launch, direct entry landing, direct return (SL, DE, DR)
3. Single launch, out-of-orbit landing, MOR (SL, OO, MOR)
4. Single launch, direct entry landing, MOR (SL, DE, MOR)
5. Dual launch, out-of-orbit landing, MOR (DL, OO, MOR)

Tug and IUS performance characteristics were supplied by JPL as summarized in Figure II-1 below.

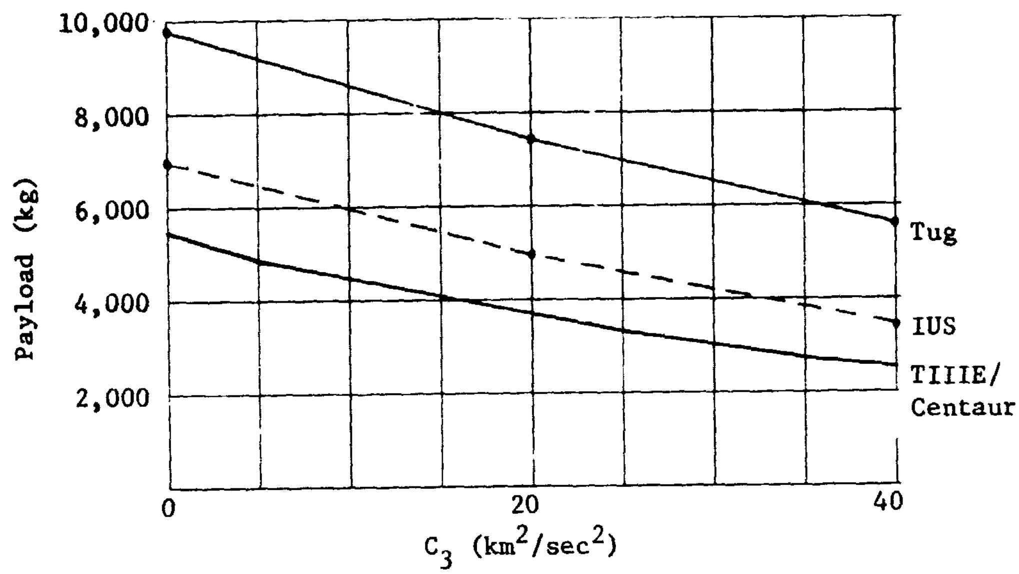

Figure II-1 IUS/Tug Performance

B. APPROACH

The basic approach taken was to match up the Earth Return Vehicle (ERV) weight available for Earth return with the weight required for Earth return. In other words, the approach arbitrarily accumulates all mission performance margins into the available ERV weight for that value and becomes a measure of the ease (or difficulty) of performing that particular mode with that launch vehicle in that mission opportunity. For the Mars-Earth leg, a basic dry weight of 20 kg was assumed for the Earth-orbital Capsule (EOC)*. This mass was "backed-up" to Mars using the reciprocal rocket equation to compute propellant weights for Earth orbit capture, Mars-to-Earth transfer and the insertion into the 1000 x 100,000 km Mars departure orbit. The non-propulsive mass assumed for the ERV was 137 kg**. The weight computations are as follows:

1. Earth orbit weight = W_{EOC} = 20 kg
2. Mars-Earth transit weight = $W_{ME} = W_{EOC}\, e^{\Delta V_{EOI}/gIsp} + 137$ kg
3. Mars orbit weight = $W_{ME}\, e^{V_{TEI}/gIsp}$ = Wet W_{ERV} required

* Based on the orbiting capsule design described in Task 3.

** See the ERV non-propulsive mass study of Appendix B.

The final orbit with a perigee of 500 km and a period of 24 hours can be reached with a Space Tug or IUS from a Shuttle-compatible parking orbit.

For each mission mode, launch year and launch vehicle (L/V), Earth-Mars launch/encounter windows were generated. ERV weights available for Earth return were computed for each launch encounter point. The final window consisted of the range of launch and encounter dates where available ERV weights were greater than 100 kg. (This was a somewhat arbitrary lower limit on weight. It turns out that a number more like 200 kg is closer to the minimum for return-to-earth*.) Other data generated included C_3, Vhp at Earth and Mars, transfer time, transfer angle, DLA, ΔV_{MOI}, injected weight to Mars and landed weight and entry weight when appropriate. The computer program which generates the Earth-Mars windows punches out for each point in the window a card with the launch date, encounter date and corresponding ERV weight available for Earth return. Likewise the program which generates the Mars-Earth return windows punches out similar cards--with the Mars launch date, the Earth arrival date and required Mars-Earth ERV weight. These sets of cards are input to program EMERGE (Earth-Mars-Earth Roundtrip GEnerator) whose function is to find all possible roundtrip missions.

The Mars-Earth launch window for a particular Earth launch year was the union of all Mars encounter dates for all mission modes and launch vehicles. This resulted from an initial search dimension of 400 days in Earth launch and 600 days in Mars encounter. (Types I and II transfers for the Earth launch windows were treated separately.) Final Earth-Mars windows (for Tug) were typically 6 months long in launch and 1 year in arrival. The range of Mars-Earth launch dates were extended by about 400 days to allow for various Mars stopover times. Initial Mars launch-Earth encounter searches were carried out over a launch range of 500 days and an encounter range of 700 days. (Points were computed every 10 days for all windows.) Punched output was obtained only for those LD/ED pairs whose corresponding ERV-weight-required was less than the maximum available ERV weight computed for the Earth-Mars leg.

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* See the ERV non-propulsive mass study of Appendix B.

The EMERGE program, mentioned earlier, accepts as input the Earth-Mars data and the Mars-Earth data and computes for output two types of tables. For each Earth launch date for which viable missions can be found, the program sorts the missions according to Mars stopover time and presents in tabular form the mission which affords minimum total mission time and the mission which affords maximum ERV margin (the margin being defined as the difference between the available ERV weight and required ERV weight). The program optimizes the two quantities by searching through varying Mars arrival dates for the same Earth launch date and stopover time. The Mars arrival date for the optimum mission is therefore shown along with the ERV margin (when the mission time is minimized) and with the mission time (when the ERV margin is maximized).

The five mission modes analyzed and the assumptions associated with each are described in the next section (C). That section is followed by a summary of the Earth-Mars performance (D), the Mars-Earth performance (E), and finally the Round Trip Possibilities (F).

C. MISSION MODE DESCRIPTIONS

The five MSSR mission modes considered are described here. For the dual launch case, in addition to computing the available ERV weight from the ERV/orbiter combination, it was necessary to check the Lander/Orbiter spacecraft weight after Mars orbit insertion to make certain that at least a 1205 kg (Ref. 1) lander was available for subsequent Mars orbit rendezvous. The following steps are performed to determine the weight available for the ERV.

Single Launch, Out-of-Orbit Landing, Direct Return

1. Trans-Mars Injection Weight = $f(C_3)$
2. Midcourse ΔV = 35 m/s (Isp = 306 sec)
3. Mars orbit insertion into 1000 x 100,000 km orbit ($\Delta V = f(Vhp)$) 60 m/s finite burn loss (Isp = 306 sec)
4. Transfer to 1 day orbit, hp = 1500 km (ΔV = 167.5 m/s) (Isp = 306 sec)
5. Subtract dry orbiter weight = 549 kg and propellant inert weight = 59 + .127 x W_p (App. B) to establish the lander weight at entry = W_E
6. Landed weight = W_{LD} = .75 W_E - 180 kg (see below)
7. Liftoff weight = W_{LIFT} = .5 W_{LD} (see below)
8. Weight inserted into 100 x 2200 orbit = W_{INS} = .1 W_{LIFT} (see below)

9. Transfer to 1000 x 100,000 km with solid rocket motor (Isp = 290 sec) ΔV = 1.077 m/s
10. Compute weight available for ERV using mass fraction = .9

The landed weight expression, (step 6), above is based on the results of Task 4. Its applicability for entry weights exceeding 1500 kg has not been established. It is, however, consistent with the 2700 kg direct entry/direct return mission design of Ref. 2. The assumption in step 7 is appropriate to landed weights of the order of 1000 kg and is probably conservative for larger landed weights. The 10 to 1 ratio identified in step 8 should apply through a relatively wide range of liftoff weights.

<u>Single Launch, Direct Entry Landing, Direct Return</u>

1. Trans-Mars injection weight = $f(C_3)$
2. Midcourse ΔV = 35 m/s
3. Compute entry velocity, V_E (= f (Vhp,γ)) for $\gamma = -24^o$ at 800,000 ft
4. Subtract dry cruise bus weight = 227 kg + propellant inerts = .127 x W_p to establish the lander weight at entry = W_E
5. Compute landed weight, W_{LD}, for entry velocity of 18822 fps, i.e., $W_{LD} = .75\ W_E - 142$ kg
6. Compute entry velocity correction to landed weight, i.e., $W'_{LD} = W_{LD} - .011\ (V_E \times 3280.8 - 18822.0)$. (This amounts to a loss of 11 kg of useful landed weight for each 1000 fps over 18822.)
7. $W_{LIFT} = .5\ W_{LD}$
8. $W_{INS} = .1\ W_{LIFT}$
9. Transfer to 1000 x 100,000 km with solid rocket motor (Isp = 290 sec) ΔV = 1.077 km/s
10. Compute weight available for ERV using mass fraction - .9

The relationship in Step 5 is based on results of Task 4. The entry velocity correction was ascertained from data in Ref. 3.

<u>Single Launch, Out-of-Orbit Landing, MOR</u>

1. Trans-Mars injection weight = $f(C_3)$
2. Midcourse ΔV = 35 m/s
3. Insert into 1000 x 100,000 km orbit (+ 60 m/s finite burn loss)
4. Transfer to a 1-day landing orbit (ΔV = 167.5 m/s)
5. Subtract off the lander mass = 1205 kg (Ref. 1)

6. Transfer orbiter to 2200 km circular (ΔV = .971 km/s)
7. Allow 69 m/s for orbiter active rendezvous derived from data from Reference 1.
8. ERV weight available in 2200 km orbit = total weight in 2200 km orbit less 59 kg + .127 x W_p less 735 for dry orbiter
9. Transfer ERV to 1000 x 100,000 orbit (ΔV = 1.044 km/s) using solid rocket motor (Isp = 290 sec)
10. Compute ERV weight available for return assuming .9 mass fraction for solid motor.

<u>Single Launch, Direct Entry Landing, MOR</u>

1. Trans-Mars injection weight = $f(C_3)$
2. Midcourse ΔV = 35 m/s
3. Compute entry velocity, V_E, for $\gamma = -24^{\circ}$ at 800,000 ft
4. Calculate W_E required to achieve landed weight of 776.4 kg (need min. of 776.4 kg for MOR type mission)
   a. Compute entry velocity correction to desired landed weight W'_{LD} = 776.4 kg + .011 (V_E x 3280.8 - 18822) kg
   b. Compute correspondingly larger entry weight = $W_E = (W'_{LD} + 142)/.75$
5. Subtract W_E from injected weight less midcourse propellant
6. Insert into 1000 x 100,000 km orbit (+ 60 m/s finite burn loss)
7. Transfer to 2200 km circular orbit (ΔV = 1.13 km/s)
8. Allow 69 m/s for orbiter rendezvous
9. ERV weight available in 2200 km orbit = total weight in orbit less propellant inerts = 59 kg + .127 x W_p less 735 kg for dry orbiter
10. Transfer ERV to 1000 x 100,000 km orbit with solid rocket motor, Isp = 290 sec, ΔV = 1.044 km/s
11. Compute ERV weight available for return assuming solid rocket mass fraction of .9

Steps 4a and 4b above are used to compute the increased entry weight needed to land 776.4 kg on the surface of Mars when the entry velocity is greater than 1822 fps. 776.4 kg is the baseline landed weight for the MOR mode (Ref. 1).

Dual Launch, Out-of-Orbit Landing, MOR

1. Trans-Mars injection weight = $f(C_3)$
2. Midcourse ΔV = 35 m/s
3. Insert into 1000 x 100,000 km orbit (60 m/s for finite burn loss)
4. Circularize at 2200 km (ΔV = 1.13 km/s)
5. Allow 69 m/s for orbiter active rendezvous
6. ERV weight available in 2200 km orbit = total weight less propellant inerts = 59 kg + .127 W_p, less 735 the weight of the dry orbiter
7. Transfer ERV to 1000 x 100,000 km (ΔV = 1.044 km/s) with solid rocket motor, Isp = 290 sec
8. Compute available ERV weight for Earth return assuming solid propellant mass fraction of .9

D. EARTH-MARS PERFORMANCE SUMMARY

A summary of available ERV weight in Mars orbit is presented in Tables II-1 through II-15 for each launch vehicle (L/V), launch year and mission mode. Type I and II trajectories to Mars were treated separately in the study. (Note the I and II headings in the Tables.) For each mission mode the launch and arrival window dates are presented along with the maximum ERV weight available for the window (MAX P/L) and the number of missions (#MSNS) in the window. The latter is the number of launch/encounter points with available ERV weight greater than 100 kg. These windows were determined by scanning the launch/encounter space until every launch/encounter point which afforded at least 100 kg of available ERV weight was found. This meant that very large windows had to be considered for possible roundtrip missions. The trajectory data for these windows cannot be presented in this document but for the sake of completeness the following parameters are summarized and discussed in Appendix A for the key missions of interest:

1. C_{3E}, Earth-to-Mars launch energy
2. Vhp_M, hyperbolic excess velocity at Mars
3. DLA, declination of outgoing asymptote from Earth (Missions considered here are not constrained by DLA. This effect is discussed in the Appendix.)
4. ΔV_{MOI}, velocity change for Mars orbit insertion
5. ΔV_{TEI}, velocity change for trans-Earth injection

6. C_{3M}, Mars-to-Earth launch energy

7. Vhp_E, hyperbolic excess velocity at Earth

8. ΔV_{EOI}, velocity change for Earth orbit insertion

9. θ_{ME}, Mars-to-Earth transfer angle

For the 1981/2 Single Launch, Direct Entry, Direct Return Mission flown with the Titan IIIE/Centaur (Table I-1) there are no missions since the largest ERV weights are 87 kg and 88 kg for the Type I and II trajectories respectively. Table II-16 further summarizes the maximum available ERV weights for the mission modes, launch years and launch vehicles. The numbers in parentheses are available ERV weights assuming liftoff weight = 75% of landed weight (as opposed to the 50% assumed for the rest of the calculations). As will be shown in Section F the liftoff weight fraction and the required ERV weight are critical factors in establishing the viability of direct return missions. The ERV weight available is greatest for the Dual Launch, MOR mission mode flown in 1988 with the Tug. The least ERV weight available is associated with the Single Launch, Out-of-Orbit, Direct Return Missions.

Table II-1 Earth-Mars Performance Summary:
1981/2 Launch Year; L/V = Titan IIIE/Centaur

Mission Mode	# MSNS		Launches		Arrivals		MAX W_{ERV} Avail.	
	I	II	I	II	I	II	I	II
Single Launch; Out-of-Orbit, Direct Return	0	0	-	-	-	-	-	-
Single Launch; Out-of-Orbit, MOR	0	1	-	11/27/81	-	09/23/82	-	104
Dual Launch - MOR	15	80	12/25/81 01/24/82	09/28/81 01/06/82	07/28/82 09/26/82	07/25/82 12/02/82	273	586
Single Launch; Direct Entry, Direct Return	0	0	-	-	-	-	87	88
Single Launch; Direct Entry, MOR	0	35	-	10/28/81 12/17/81	-	08/14/82 11/12/82	41	295

Table II-2 Earth-Mars Performance Summary:
1983/4 Launch Year; L/V = Titan IIIE/Centaur

Mission Mode	# MSNS		Launches		Arrivals		MAX W_{ERV} Avail.	
	I	II	I	II	I	II	I	II
Single Launch; Out-of-Orbit Direct Return	0	0	-	-	-	-	-	-
Single Launch; Out-of-Orbit, MOR	0	0	-	-	-	-	-	-
Dual Launch - MOR			02/19/84 03/20/84	11/08/83 02/16/84	08/24/84 10/23/84	08/22/84 12/10/84	260	408
Single Launch, Direct Entry, Direct Return	0	0	-	-	-	-	83	89
Single Launch, Direct Entry, MOR	0	11	-	12/18/83 01/17/84	-	09/21/84 10/31/84	27	148

Table II-3 Earth-Mars Performance Summary:
1986 Launch Year; L/V = Titan IIIE/Centaur

Mission Mode	# MSNS		Launches		Arrivals		MAX W_{ERV} Avail.	
	I	II	I	II	I	III	I	II
Single Launch; Out-of-Orbit Direct Return	0	0	-	-	-	-	-	29
Single Launch; Out-of-Orbit, MOR	0	0	-	-	-	-		-
Dual Launch - MOR	53	62	04/20/86 06/19/86	01/11/86 05/21/86	10/01/86 02/28/87	09/29/86 04/07/87	413	338
Single Launch, Direct Entry Direct Return	0	0	-	-	-	-	89	89
Single Launch, Direct Entry, MOR	12	3	05/10/86 05/20/86	04/21/86 05/01/86	11/20/86 01/29/87	01/17/87 01/27/87	154	111

Table II-4 Earth-Mars Performance Summary:
1988 Launch Year; L/V = Titan IIIE/Centaur

Mission Mode	# MSNS		Launches		Arrivals		MAX W_{ERV} Avail.	
	I	II	I	II	I	II	I	II
Single Launch; Out-of-Orbit, Direct Return	0	0	-	-	-	-	-	-
Single Launch; Out-of-Orbit, MOR	11	0	06/28/88 07/29/88	-	01/16/89 02/25/89	-	165	-
Dual Launch - MOR	86	94	05/10/88 08/28/88	04/10/88 08/28/88	11/27/88 05/16/89	11/27/88 09/23/89	653	412
Single Launch, Direct Entry, Direct Return	0	0	-	-	-	-	84	75
Single Launch, Direct Entry, MOR	46	0	05/30/88 08/08/88	-	12/17/88 03/27/89	-	332	84

Table II-5 Earth-Mars Performance Summary:
1990 Launch Year; L/V = Titan IIIE/Centaur

Mission Mode	# MSNS		Launches		Arrivals		MAX W_{ERV} Avail.	
	I	II	I	II	I	II	I	II
Single Launch; Out-of-Orbit, Direct Return	0	0	-	-	-	-	-	-
Single Launch; Out-of-Orbit, MOR	0	0	-	-	-	-	-	65
Dual Launch - MOR	19	95	08/30/90 10/09/90	06/25/90 10/13/90	03/08/91 0[illegible]/26/91	03/31/91 11/16/91	341	556
Single Launch, Direct Entry, Direct Return	0	0	-	-	-	-	65	79
Single Launch, Direct Entry, MOR	0	40	-	07/25/90 09/23/90	-	06/09/91 10/17/91	7	236

Table II-6 Earth-Mars Performance Summary:
1981/2 Launch Year; L/V = Shuttle/IUS

Mission Mode	# MSNS		Launches		Arrivals		MAX W_{ERV} Avail.	
	I	II	I	II	I	II	I	II
Single Launch; Out-of-Orbit, Direct Return	0	0	-	-	-	-	44	67
Single Launch; Out-of-Orbit, MOR	4	56	01/04/82 01/14/82	10/28/81 12/27/81	08/17/82 09/06/82	08/04/82 11/22/82	153	591
Dual Launch - MOR	23	103	12/05/81 01/24/82	10/08/81 01/26/82	07/18/82 09/26/82	07/15/82 01/01/83	607	1037
Single Launch, Direct Entry, Direct Return	41	143	11/25/81 01/14/82	10/28/81 03/27/82	06/08/82 09/06/82	07/05/82 03/12/83	128	130
Single Launch, Direct Entry, MOR	14	79	12/15/81 01/14/82	10/28/81 01/16/82	07/28/82 09/06/82	07/15/82 12/22/82	387	775

Table II-7 Earth-Mars Performance Summary:
1983/4 Launch Year; L/V = Shuttle/IUS

Mission Mode	# MSNS		Launches		Arrivals		MAX W_{ERV} Avail.	
	I	II	I	II	I	II	I	II
Single Launch; Out-of-Orbit, Direct Return	0	0	-	-	-	-	47	54
Single Launch; Out-of-Orbit, MOR	9	33	02/19/84 03/19/84	11/[illegible]8/83 01/27/84	09/03/84 10/13/84	09/01/84 11/20/84	209	343
Dual Launch - MOR	31	84	01/30/84 03/20/84	11/08/83 03/17/84	08/14/84 11/02/84	08/12/84 01/19/85	662	798
Single Launch, Direct Entry, Direct Return	41	148	01/10/84 03/20/84	11/28/83 05/16/84	07/25/84 11/02/84	08/12/84 04/29/85	128	130
Single Launch, Direct Entry, MOR	22	64	01/30/84 03/20/84	11/28/83 02/26/84	08/24/84 11/02/84	08/12/84 12/30/84	433	552

Table II-8 Earth-Mars Performance Summary:
1986 Launch Year; L/V = Shuttle/IUS

Mission Mode	# MSNS		Launches		Arrivals		MAX W_{ERV} Avail.	
	I	II	I	II	I	II	I	II
Single Launch; Out-of-Orbit, Direct Return	0	0	-	-	-	-	67	49
Single Launch; Out-of-Orbit, MOR	60	27	03/31/84 06/09/86	02/20/86 05/11/36	10/01/86 02/08/87	10/29/86 02/26/87	580	286
Dual Launch - MOR	90	107	03/21/86 06/19/86	01/21/86 07/10/86	09/11/86 03/10/87	09/29/86 08/05/87	1037	733
Single Launch; Direct Entry, Direct Return	104	101	03/11/86 06/19/86	02/10/86 07/10/86	08/12/86 03/10/87	10/19/86 08/05/87	131	131
Single Launch; Direct Entry, MOR	79	80	03/21/86 06/19/86	02/10/86 06/20/86	09/21/86 03/10/87	10/19/86 06/16/87	768	510

Table II-9 Earth-Mars Performance Summary:
1988 Launch Year; L/V = Shuttle/IUS

Mission Mode	# MSNS		Launches		Arrivals		MAX W_{ERV} Avail.	
	I	II	I	II	I	II	I	II
Single Launch; Out-of-Orbit, Direct Return	0	0	-	-	-	-	72	49
Single Launch; Out-of-Orbit, MOR	51	17	05/30/88 07/29/88	07/09/88 08/18/88	12/07/88 03/17/89	06/05/89 09/13/89	680	245
Dual Launch - MOR	94	84	05/10/88 08/18/88	04/20/88 08/28/88	11/17/88 05/06/89	12/07/88 09/23/89	1149	715
Single Launch; Direct Entry, Direct Return	64	19	05/30/88 07/29/88	07/09/88 08/28/88	11/17/88 03/27/89	06/05/89 09/23/89	124	110
Single Launch; Direct Entry, MOR	64	19	05/30/88 08/08/88	07/09/88 08/28/88	11/27/88 03/27/89	06/05/89 09/23/89	843	443

Table II-10 Earth-Mars Performance Summary:

1990 Launch Year; L/V = Shuttle/IUS

Mission Mode	# MSNS		Launches		Arrivals		MAX W_{ERV} Avail.	
	I	II	I	II	I	II	I	II
Single Launch; Out-of-Orbit, Direct Return	0	0	-	-	-	-	57	64
Single Launch; Out-of-Orbit, MOR	8	39	08/20/90 08/30/90	08/14/90 10/03/90	03/18/91 04/17/91	06/29/91 11/06/91	354	534
Dual Launch - MOR	39	84	08/10/90 09/29/90	07/05/90 10/13/90	02/26/91 07/06/91	03/31/91 11/16/91	880	1007
Single Launch, Direct Entry, Direct Return	13	39	08/20/90 09/09/90	08/14/90 10/03/90	02/26/91 04/17/91	06/29/91 11/06/91	109	117
Single Launch, Direct Entry, MOR	15	45	08/20/90 09/09/90	08/04/90 10/03/90	02/26/91 05/07/91	06/09/91 11/06/91	576	702

Table II-11 Earth-Mars Performance Summary:

1981/2 Launch Year; L/V = Shuttle/Tug

Mission Mode	# MSNS		Launches		Arrivals		MAX W_{ERV} Avail.	
	I	II	I	II	I	II	I	II
Single Launch; Out-of-Orbit, Direct Return	0	10	-	11/17/81 12/07/81	-	09/03/82 10/13/82	75	106
Single Launch; Out-of-Orbit, MOR	22	100	12/05/81 01/24/82	10/08/81 01/16/82	07/18/82 09/16/82	07/15/82 12/22/82	727	1314
Dual Launch - MOR	49	151	11/25/81 02/13/82	09/28/81 02/05/82	07/08/82 10/26/82	07/05/82 01/21/83	1150	1740
Single Launch; Direct Entry, Direct Return	59	186	11/15/81 01/24/82	10/08/81 03/27/82	06/08/82 09/26/82	05/25/82 03/12/83	189	191
Single Launch; Direct Entry, MOR	37	134	11/25/81 02/03/82	09/28/81 02/05/82	07/08/82 10/06/82	07/05/82 01/21/83	934	1487

Table II-12 Earth-Mars Performance Summary:
1983/4 Launch Year; L/V = Shuttle/Tug

Mission Mode	# MSNS		Launches		Arrivals		MAX W_{ERV} Avail.	
	I	II	I	II	I	II	I	II
Single Launch; Out-of-Orbit, Direct Return	0	0	-	-	-	-	78	88
Single Launch; Out-of-Orbit, MOR	30	75	01/30/84 03/20/84	11/18/84 03/07/84	08/14/84 11/02/85	08/12/84 01/09/85	791	975
Dual Launch - MOR	65	160	01/20/84 04/09/84	10/19/83 04/26/84	08/04/84 12/12/85	07/23/84 03/30/85	1208	1395
Single Launch; Direct Entry, Direct Return	77	194	01/10/84 05/19/84	11/18/83 05/16/84	07/25/84 03/12/85	07/23/84 04/29/85	189	191
Single Launch; Direct Entry, MOR	49	123	01/20/84 03/30/84	10/29/83 04/16/84	08/04/84 11/22/84	08/12/84 03/10/85	998	1169

Table II-13 Earth-Mars Performance Summary:
1985/6 Launch Year; L/V = Shuttle/Tug

Mission Mode	# MSNS		Launches		Arrivals		MAX W_{ERV} Avail.	
	I	II	I	II	I	II	I	II
Single Launch; Out-of-Orbit, Direct Return	11	0	04/20/86 05/10/86	-	10/31/86 12/10/86	-	105	83
Single Launch; Out-of-Orbit, MOR	85	105	03/21/86 06/19/86	01/21/86 07/10/86	09/21/86 03/10/87	10/09/86 08/05/87	1291	875
Dual Launch - MOR	133	189	03/11/86 07/09/86	12/12/85 07/30/86	09/01/86 04/09/87	08/30/86 08/25/87	1714	1283
Single Launch; Direct Entry Direct Return	140	172	03/01/86 07/09/86	01/11/86 07/30/86	08/12/86 04/09/87	09/09/86 08/25/87	193	192
Single Launch; Direct Entry, MOR	113	136	03/11/86 06/29/86	12/22/85 07/20/86	09/11/86 03/20/87	09/09/86 08/15/87	1468	1080

Table II-14 Earth-Mars Performance Summary:
1988 Launch Year; L/V = Shuttle/Tug

Mission Mode	# MSNS		Launches		Arrivals		MAX W_{ERV} Avail.	
	I	II	I	II	I	II	I	II
Single Launch; Out-of-Orbit, Direct Return	26	0	06/19/88 07/29/88	-	12/27/88 03/17/89	-	114	84
Single Launch; Out-of-Orbit, MOR	95	96	05/10/88 08/18/88	04/20/88 08/28/88	11/17/88 05/06/89	12/07/88 09/23/89	1462	896
Dual Launch - MOR	129	151	04/30/88 08/28/88	03/31/88 09/17/88	11/07/88 06/05/89	11/07/88 10/13/89	1902	1334
Single Launch; Direct Entry, Direct Return	111	91	05/10/88 08/18/88	04/30/88 09/07/88	10/28/88 05/06/89	12/07/88 10/03/89	184	169
Single Launch; Direct Entry, MOR	117	135	05/10/88 08/28/88	03/31/88 09/07/88	11/17/88 06/05/89	11/17/88 10/03/89	1621	1082

Table II-15 Earth-Mars Performance Summary:
1990 Launch Year; L/V = Shuttle/Tug

Mission Mode	# MSNS		Launches		Arrivals		MAX W_{ERV} Avail.	
	I	II	I	II	I	II	I	II
Single Launch; Out-of-Orbit, Direct Return	0	13	-	08/14/90 09/03/90	-	06/29/91 09/07/91	96	105
Single Launch; Out-of-Orbit, MOR	42	95	08/20/90 09/29/90	07/05/90 10/13/90	02/26/91 07/06/91	03/31/91 11/16/91	1131	1288
Dual Launch - MOR	71	116	07/31/90 10/09/90	06/25/90 11/02/90	02/06/91 07/26/91	03/21/91 12/06/91	1578	1732
Single Launch; Direct Entry, Direct Return	47	77	08/10/90 09/19/90	07/25/90 10/23/90	01/27/91 06/06/91	05/10/91 11/26/91	167	177
Single Launch; Direct Entry, MOR	60	110	08/10/90 10/09/90	06/25/90 10/23/90	02/16/91 07/26/91	03/21/91 11/26/91	1297	1450

Table II-16 Maximum Available ERV Weight for Return to Earth

Year	Launch Vehicle	Single Launch; Out-of-Orbit, Direct Return	Single Launch; Direct Entry, Direct Return	Single Launch; Out-of-Orbit MOR	Single Launch; Direct Entry, MOR	Dual Launch; Out-of-Orbit, MOR
1981	Shuttle/Tug	-	-	-	-	-
	Shuttle/IUS	(101) 67	(195) 130	591	775	1037
	Titan IIIE/Centaur	0	(131) 87	104	295	586
1983/ 1984	Shuttle/Tug	-	-	-	-	-
	Shuttle/IUS	(81) 54	(195) 130	343	552	798
	Titan IIIE/Centaur	0	(134) 89	0	(222) 148	408
1986	Shuttle/Tug	(158) 105	(290) 193	1291	1468	1714
	Shuttle/IUS	(101) 67	(197) 131	580	768	1037
	Titan IIIE/Centaur	(44) 29	(134) 89	0	(231) 154	413
1988	Shuttle/Tug	(171) 114	(276) 184	1462	1621	1902
	Shuttle/IUS	(108) 72	(186) 124	680	843	1149
	Titan IIIE/Centaur	(47) 31	(126) 84	(248) 165	332	653
1990	Shuttle/Tug	(158) 105	(266) 177	1288	1450	1732
	Shuttle/IUS	(96) 64	(176) 117	534	702	1007
	Titan IIIE/Centaur	0	(119) 79	65	236	556

( ) = With 75% of landed weight available for liftoff

E. MARS-EARTH PERFORMANCE SUMMARY

The range of ERV weights (in the 1000 x 100,000 km Mars departure orbit) required to return to Earth for the MSSR launch years are shown in Table II-17. The window is constrained by W_{ERV} MAX. This quantity is the maximum ERV weight available in the given year. It is provided by the Dual Launch MOR mission mode flown with the Tug. Any launch/encounter pair with a required ERV weight less than or equal to W_{ERV} MAX is a candidate point in the window. Note that all mission years except 1990 have two window segments, denoted as a) and b). The a) segment consists of "fast" Type IIs of 240 day trip times and 265° transfer angles. These are return trajectories that dip inside the Earth's orbital radius. The b) segment consists of Type I returns. Note that the fast Type IIs require heavier ERVs to execute them. The 1990 returns consist of both Type Is and IIs, without the fast, large transfer angle, Type IIs. Trajectory related parameters associated with the Earth return windows for key missions of interest are summarized in Appendix A.

Table II-17 Potential Earth Return Windows

Earth Launch	#MSNS		Mars Launch Dates	Earth Arrival Dates	Max Avail. W_{ERV}	Min Req. W_{ERV}
1981	250	(a)	07/05/82-08/14/82	01/31/84-03/22/83	1750	1222
		(b)	03/12/83-11/07/83	04/05/84-09/02/84	1750	230
1983/1984	243	(a)	07/23/84-10/21/84	01/29/85-06/08/85	1400	687
		(b)	03/30/85-11/25/85	05/04/86-09/21/86	1400	264
1986	288	(a)	08/30/86-12/18/86	02/26/87-08/15/87	1725	529
		(b)	04/27/87-01/02/88	05/31/88-10/28/88	1725	334
1988	288	(a)	11/07/88-01/16/89	05/26/89-09/23/89	1925	688
		(b)	06/05/89-03/12/90	07/10/90-01/06/91	1925	484
1990	247		07/26/91-06/10/92	08/29/92-04/06/93	1750	293

The ERV weights required for a minimum energy return to Earth are shown in Table II-18. Reading across from left to right, the dry Earth orbital capsule (EOS) is backed up to Mars orbit. ΔV_{EOI} and ΔV_{TEI} are the Earth orbit insertion and trans-Earth injection ΔVs respectively. A dry ERV weight of either 85 kg or 137 kg is added to the wet EOC. The total ERV weight in Mars orbit is the wet W_{ERV} + the wet EOC. Again the numbers in parentheses are for a 137 kg dry ERV. These results will be used in Section F to evaluate the direct return mission opportunities.

Table II-18 Minimum Energy Return to Earth

Year	Dry EOC (kg)	w [redacted]$_{EOI}$ (km/s)	ΔV_{EOC} (kg)	Dry W_{ERV} + Wet EOC	ΔV_{TEI} (km/s)	W_{ERV} + EOC
1981	20	1.22	30.7	(167.7) 115.7	.87	(227.8) 157.1
1983	20	.98	28.2	(165.2) 113.2	.72	(212.8) 145.8
1986	20	.89	27.4	(164.4) 112.4	1.11	(243.0) 166.1
1988	20	.92	27.7	(164.7) 112.7	1.33	(263.0) 180.0
1990	20	.81	26.6	(163.6) 111.6	1.04	(235.9) 160.9
Dry W_{ERV} = 85 kg				Dry W_{ERV} = (137 kg)		

F. ROUNDTRIP MISSION OPPORTUNITIES

The Earth-Mars opportunities and the Mars-Earth return opportunities are used in the EMERGE program to generate all possible MSSR missions with stopover times from 0 to 400 days. This program was run for each mission mode, launch year, trajectory type, and launch vehicle. The program has two functional loops:

1. For each Earth launch date and stopover time, the program builds a library of feasible MSSR missions with varying Mars arrival dates, total mission times and ERV weight margins (available W_{ERV}, required W_{ERV}).

2. From this library of missions it then finds the mission which has the largest ERV weight margin and also the mission which has the shortest total time. For each of these missions it stores the Mars arrival date along with the ERV margin and the mission time for later output.

The routine performs functions 1) and 2) for each Earth launch date and stopover time and then prints out tables as shown (Tables II-19 and II-20 below). The stopover times are indicated across the top with Earth launch dates on the left. In the first output table (Minimum Time Missions) the minimum mission time is shown first, then ERV margin and finally the Mars arrival date for each Earth launch date and stopover time. In the second output table (Maximum ERV Margin Missions) the ERV margin appears first, followed by the corresponding mission time and Mars arrival date. The sample output shows that there are 20-day Earth launch windows for stopover times from 0 to 50 days with mission times in the 731 to 751 day range. A 20-day window affording maximum ERV weight margin should be flown in the interval 7/19/88 to 8/8/88 with a 50 day stopover. The maximum margin would be 664 kg.

An EMERGE run was made for each of the five launch years, three launch vehicles, two trajectory types and five mission modes--a total of 150 runs. These output sheets are available at MMC upon request. For more compact presentation, the output was further summarized as follows. From the minimum mission time tables only, the smallest mission time for a 20-day launch window (with the shortest stopover greater than 30 days) was noted. The ERV margin results were summarized by extracting the largest ERV weight margin available over a 20 day launch window (regardless of stopover time). These results are presented in Tables II-21. II-22 and II-23 for the three MOR mission modes. Only minimum energy conjunction or opposition missions are possible with the direct return mode. This will be discussed later.

Table II-19 Sample EMERGE Output: Dual Launch, MOR

Type II, 1988, L/V = Shuttle/Tug

Minimum Time Missions

Earth Launch Date \ Stopover Time (Days)	0	10	20	30	40	50
6/19/88	**** 0 0/ 0	**** 0 0/ 0	**** 0 0/ 0	**** 0 0/ 0	**** 0 0/ 0	**** 0 0/ 0
6/29,	**** 0 0/ 0	**** 0 0/ 0	**** 0 0/ 0	**** 0 0/ 0	**** 0 0/ 0	**** 0 0/ 0
7/ 9/88	741 44 6/15	741 69 6/ 5	751 226 6/ 5	751 193 6/ 5	751 124 6/ 5	751 28 6/ 5
7/19/88	741 138 6/25	741 154 6/25	741 86 6/25	751 250 6/25	751 196 6/25	751 117 6/25
7/29/88	731 28 7/15	741 192 7/15	741 138 7/15	741 59 7/15	751 224 7/15	751 157 7/15
8/ 8/88	731 60 8/ 4	741 193 8/ 4	741 146 8/ 4	741 79 8/ 4	751 215 8/ 4	751 156 8/ 4
8/18/88	731 48 8/24	741 157 9/ 3	741 117 8/24	741 58 8/24	751 172 8/24	751 120 8/24
8/28/88	741 134 9/23	741 103 9/23	741 51 9/23	751 150 9/23	751 102 9/23	751 39 9/23

Mission Time (Days)
ERV Margin (kg)
Mars Arrival Date

Table II-20 Sample EMERGE Output: Dual Launch, MOR
Type II, 1988, L/V = Shuttle/Tug

Maximum ERV Weight Margin Missions

Earth Launch Date \ Stopover Time (Days)	0	10	20	30	40	50	
6/19/88	0	0	0	0	0	0	ERV Margin (kg)
	****	****	****	****	****	****	Mission Time (Days)
	0/ 0	0/ 0	0/ 0	0/ 0	0/ 0	0/ 0	Mars Arrival Date
6/29/88	0	0	0	0	0	0	
	****	****	****	****	****	****	
	0/ 0	0/ 0	0/ 0	0/ 0	0/ 0	0/ 0	
7/ 9/88	44	232	227	283	307	347	
	741	751	761	761	761	771	
	6/15	6/15	6/15	6/15	6/ 5	6/15	
7/19/88	234	265	345	595	658	706	
	751	751	791	801	811	811	
	7/15	7/ 5	7/25	7/25	7/25	7/15	
7/29/88	536	599	648	682	706	721	
	791	801	801	801	811	811	
	8/24	8/24	8/14	8/ 4	8/ 4	7/25	
8/ 8/88	611	635	657	665	666	664	
	791	801	801	811	811	821	
	9/ 3	9/ 3	8/24	8/24	8/14	8/14	
8/18/88	581	589	587	585	582	581	
	791	801	811	811	811	821	
	9/13	9/13	9/13	9/13	9/13	9/13	
8/28/88	469	467	466	463	462	461	
	791	801	801	801	811	811	
	9/23	9/23	9/23	9/23	9/23	9/23	

MOR Summary

Referring to Tables II-21, II-22 and II-23, it should be noted that IUS data is presented for only 1981 and 1983/4. After that it is assumed the Tug will be available. The Titan IIIE/Centaur results are shown for comparison purposes only for all years. A dashed line entry means that the mission is not feasible with the specified launch vehicle. Also note that the left side of the tables present the minimum mission time summaries while the right side is for the maximum ERV weight margin summaries. Minimum stopover missions (30 days stopover) are possible only with the Tug. They can be flown in 1988 and 1990 with any of the three MOR modes but in 1986 only with the Dual Launch or Single Launch, Direct Entry modes (when DLA is not constrained, see Appendix A). The 1986 minimum stopover missions have a very short total mission time of 462 days. (These are comprised of Type I outgoing trajectories and the fast Type II returns.) The healthiest ERV weight margin going along with these minimum stopover missions is 102 kg for the year 1990. Margins range from 102 down to 0 kg for this class of mission. As expected, large maximum ERV weight margins occur for very long stopover times of 300 to 480 days. These are the minimum energy Mars-Earth returns which produce long total mission times of 902 to 1020 days.

The IUS allows the Single Launch, out-of-orbit, MOR to be flown in 1981 and 1983/4. There is little capacity, however, to reduce total mission time with this launch vehicle.

Direct Return Summary

The Earth-Mars results for direct return missions, Table I-16, Section D, were matched against the Mars-Earth minimum energy results of Table I-20, Section E, to determine what direct return missions are possible under what set of assumptions concerning 1) proportion of landed weight available for liftoff (50% or 75%) and 2) dry ERV weight (85 or 137 kg). Table I-24 shows the outcome. The single launch, out-of-orbit landing, direct return mode is practically impossible to fly even with the Tug and a minimum energy trajectory. The direct entry direct return mode, on the other hand, can be flown in every launch year except when the liftoff weight is assumed to be only 50% of the landed weight and the larger dry ERV weight (137 kg) is assumed. Then it can't be flown at all.

Table II-21 Dual Launch, Out-of-Orbit, MOR

Year	Type	L/V	Minimum Mission Time (Days)	ERV Margin (kg)	Stopover Time (Days)	Earth Launch Date	Mars Arrival Date	Max ERV Margin (kg)	Mission Time (Days)	Stopover Time (Days)	Earth Launch Date	Mars Arrival Date
1981	I	IUS	852	31	240	12/25/81 01/14/82	08/17/82 09/06/82	279	982	420	12/25/81 01/14/82	08/07/82 08/27/82
		T3E	-	-	-	-	-	-	-	-	-	-
	II	IUS	870	36	150	12/17/81 01/06/82	11/02/82 11/22/82	783	1020	380	11/17/81 12/07/81	09/13/82 09/23/82
		T3E	900	20	190	11/27/81 12/17/81	10/03/82 10/23/82	314	1010	360	11/17/81 12/07/81	09/13/82 10/03/82
1983/1984	I	IUS	865	31	230	02/29/84 03/20/84	10/03/84 11/02/84	348	945	380	02/19/84 03/10/84	09/13/84 10/03/84
		T3E	-	-	-	-	-	-	-	-	-	-
	II	IUS	898	22	190	01/27/84 02/16/84	11/30/84 12/10/84	496	1008	380	12/18/83 01/07/84	09/21/84 10/11/84
		T3E	928	20	250	12/28/83 01/17/84	10/11/84 10/21/84	123	1004	480	12/28/83 01/17/84	10/21/84
1986	I	TUG	462	59	30	03/31/86 04/20/86	10/01/86 10/11/86	1332	902	380	04/20/86 05/10/86	11/10/86 11/20/86
		T3E	882	0	320	05/10/86 05/30/86	11/10/86 01/09/87	14	938	480	05/10/86 05/30/86	01/19/87
	II	TUG	761	8	50	06/20/86 07/10/86	06/06/87 07/26/87	872	921	310	04/21/86 05/11/86	01/17/87 01/27/87
		T3E	947	13	480	04/11/86 05/01/86	01/17/87 01/27/87	48	957	480	04/21/86 05/11/86	01/17/87 01/27/87
1988	I	TUG	731	63	100	07/09/88 07/29/88	02/25/89	1351	921	410	06/29/88 07/19/88	01/16/89 01/26/89
		T3E	811	27	200	07/09/88 07/29/88	02/05/89 02/15/89	313	976	480	07/19/88 08/08/88	02/25/89
	II	TUG	751	59	30	07/09/88 07/29/88	06/05/89 07/15/89	890	986	330	07/19/88 08/08/88	07/25/89 08/04/89
		T3E	-	-	-	-	-	-	-	-	-	-
1990	I	TUG	780	0	140	09/09/90 09/29/90	04/17/91 05/27/91	884	9[illegible]0	450	08/20/90 09/09/90	03/18/91
		T3E	-	-	-	-	-	-	-	-	-	-
	II	TUG	756	102	[illegible]0	08/24/90 09/13/90	07/19/91 08/28/91	1311	966	300	08/14/90 09/03/90	07/29/91 08/08/91
		T3E	906	2	200	09/03/90 09/23/90	09/07/91 10/17/91	198	976	320	08/04/90 08/24/90	07/09/91 07/19/91

Table II-22 Single Launch, Direct Entry, MOR

Year	Type	L/V	Minimum Mission Time (Days)	ERV Margin (kg)	Stopover Time (Days)	Earth Launch Date	Mars Arrival Date	Max ERV Margin (kg)	Mission Time (Days)	Stopover Time (Days)	Earth Launch Date	Mars Arrival Date
1981	I	IUS	9[illegible]2	21	300	12/25/81 1/14/82	08/07/82 08/27/82	91	982	410	12/25/81 01/14/82	08/07/82 08/27/82
		T3E	-	-	-	-	-	-	-	-	-	-
	II	IUS	890	18	170	11/27/81 12/17/81	10/13/82 10/23/82	507	1020	380	11/17/81 12/07/81	09/13/82 10/03/82
		T3E	980	1	290	11/17/81 12/07/81	09/13/82 10/03/82	28	1010	360	11/17/81 12/07/81	09/13/82 10/03/82
1983/ 1984	I	IUS	885	25	270	02/19/84 03/10/84	10/13/84 10/03/84	119	945	360	02/19/84 03/10/84	09/13/84 10/03/84
		T3E	-	-	-	-	-	-	-	-	-	-
	II	IUS	908	3	220	01/17/84 02/06/84	10/31/84 11/20/84	248	1008	380	12/18/83 01/07/84	09/21/84 10/11/84
		T3E	-	-	-	-	-	-	-	-	-	-
1986	I	TUG	462	10	30	04/10/86 04/30/86	10/11/86 10/21/86	1086	922	380	04/20/86 05/10/86	11/10/86 11/20/86
		T3E	-	-	-	-	-	-	-	-	-	-
	II	TUG	801	0	30	06/20/86 07/10/86	06/26/87 07/26/87	682	921	330	04/21/86 05/11/86	01/17/87 01/27/87
		T3E	-	-	-	-	-	-	-	-	-	-
1988	I	TUG	741	61	110	07/09/88 07/29/88	02/15/89 02/22/89	1076	921	410	06/29/88 07/19/88	01/16/89 01/26/89
		T3E	986	6	480	07/19/88	02/25/89	6	986	480	07/19/88	02/25/89
	II	TUG	761	25	30	07/29/88 08/18/88	07/15/89 09/13/89	664	986	330	07/19/88 08/08/88	07/25/89 08/04/89
		T3E	-	-	-	-	-	-	-	-	-	-
1990	I	TUG	810	44	200	08/20/90 09/09/90	03/18/91 04/14/91	654	960	450	08/20/90 09/09/90	03/18/91
		T3E	-	-	-	-	-	-	-	-	-	-
	II	TUG	776	68	30	09/03/90 09/23/90	09/07/91 09/17/91	1076	966	280	08/14/90 09/03/90	07/29/91 08/28/91
			-	-	-	-	-	-	-	-	-	-

Table II-23 Single Launch, Out-of-Orbit, MOR

Year	Type	Launch Vehicle	Minimum Mission Time	ERV Margin	Stopover Time	Earth Launch Date	Mars Arrival Date	Max ERV Margin	Mission Time	Stopover Time	Earth Launch Date	Mars Arrival Date
1981	I	IUS	-	-	-	-	-	-	-	-	-	-
		T3E	-	-	-	-	-	-	-	-	-	-
	II	IUS	900	16	190	11/27/81 12/17/81	10/03/82 10/23/82	321	1020	370	11/17/81 12/07/81	09/13/82 10/3/82
		T3E	-	-	-	-	-	-	-	-	-	-
1983/1984	I	IUS	-	-	-	-	-	-	-	-	-	-
		T3E	-	-	-	-	-	-	-	-	-	-
	II	IUS	978	3	290	12/18/83 01/07/84	10/1/84 10/11/84	38	1008	380	12/18/83 01/07/84	09/21/84 10/11/84
		T3E	-	-	-	-	-	-	-	-	-	-
1986	I	TUG	-	-	-	-	-	-	-	-	-	-
		T3E	-	-	-	-	-	-	-	-	-	-
	II	TUG	831	12	120	05/11/86 05/31/86	01/27/87 04/07/87	555	957	480	04/21/86 05/11/86	01/17/87 01/27/87
		T3E	-	-	-	-	-	-	-	-	-	-
1988	I	TUG	741	101	120	07/09/88 07/29/88	02/15/89	913	921	410	06/29/88 07/19/88	01/16/89 01/26/89
		T3E	-	-	-	-	-	-	-	-	-	-
	II	TUG	791	1	30	08/08/88 08/28/88	08/04/89 09/23/89	461	986	330	07/19/88 08/08/88	07/25/89 08/04/89
		T3E	-	-	-	-	-	-	-	-	-	-
1990	I	TUG	890	17	310	08/30/90 09/19/90	03/28/91 05/17/91	453	960	450	08/20/90 09/09/90	03/18/91
		T3E	-	-	-	-	-	-	-	-	-	-
	II	TUG	786	42	30	09/03/90 09/23/90	08/18/91 09/17/91	917	966	280	08/14/90 09/03/90	07/29/91 08/28/91
		T3E	-	-	-	-	-	-	-	-	-	-

Table II-24 Summary of Direct Return Mission ERV Weight Margins (kg)*
with Minimum Energy Mars-Earth Transfers

		Mission Mode							
		Single Launch, Out-of-Orbit Direct Return				Single Launch, Direct Entry Direct Return			
		$W_{LIFT} = .5\ W_{LND}$		$W_{LIFT} = .75\ W_{LND}$		$W_{LIFT} = .5\ W_{LND}$		$W_{LIFT} = .75\ W_{LND}$	
Year	L/V	ERV=85	ERV=137	ERV=85	ERV=137	ERV=85	ERV=137	ERV=85	ERV=137
1981	Tug	-	-	-	-	-	-	-	-
	IUS	0	0	0	0	0	0	+ 38.0	0
	T3E	0	0	0	0	0	0	0	0
1983/ 1984	Tug	-	-	-	-	-	-	-	-
	IUS	0	0	0	0	0	0	+ 49.2	0
	T3E	0	0	0	0	0	0	- 11.8	0
1986	Tug	0	0	-8.1	0	+ 26.9	0	+123.9	+ 47.0
	IUS	0	0	0	0	0	0	+ 30.9	0
	T3E	0	0	0	0	0	0	0	0
1988	Tug	0	0	-9.0	0	+ 4.0	0	+ 96.0	+ 13.0
	IUS	0	0	0	0	0	0	+ 6.0	0
	T3E	0	0	0	0	0	0	0	0
1990	Tug	0	0	-2.9	0	+ 16.1	0	+105.1	+ 30.1
	IUS	0	0	0	0	0	0	+ 15.1	0
	T3E	0	0	0	0	0	0	0	0

* Available ERV weight minus required ERV weight.

0 = Not Feasible - = Not Applicable

G. TASK 1 CONCLUSIONS

Of the five mission modes studied, only the three Mars Orbital Rendezvous (MOR) types could be flown in times shorter than the minimum-energy-out/minimum-energy-back mission time by utilizing the increased Tug and IUS launch performance. Missions as short as 462 days, with 30 day stopovers, are possible with the Tug in 1986. The direct return mission times could not be decreased from the 1000 day range because of the need for minimum energy Mars-Earth returns even with the Shuttle/Tug as the launch vehicle. The Single Launch, Out-of-Orbit, Direct Return mission is practically impossible even with the Tug. The Direct Entry, Direct Return mission can be flown with the Tug in 1986, 1988 and 1990 if the dry ERV weight is closer to 85 kg than 137 kg. The heavier dry ERVs can be flown every year from 1981 to 1990 if 75% of the landed weight can be used for liftoff.

Overall implications of the results of this mission survey task are discussed in Chapter VI.

III. TASK 2 BACK CONTAMINATION CONTROL OPTIONS

During this task, the problem of back contamination was studied as it pertains to the control of microorganisms external to the sample canister. No consideration has been given to the handling of the sample once it has been retrieved at Earth. Each phase of the sample return mission has been reviewed to determine what types of barriers could be incorporated to prevent the transporting of organisms from Mars to Earth. Three events in the sample return mission sequence provide opportunities for transferring the organisms if indeed they do exist. They are: (1) events during the time spent on the surface of Mars; (2) the transfer of the sample from the MAV to the ERV; and, (3) the retrieval of the sample at Earth. The intent of this study was to identify preventative schemes that could be integrated into the mission without jeopardizing the basic purpose of the mission, i.e., to obtain a sample of the surface of Mars.

Figure III-1 illustrates the sources of contamination acting upon each of these events in addition to the safeguards that are a part of the baseline mission design and measures that were investigated in the Task 2 study. Each of these events was approached as though there were no other back-contamination barriers in the entire mission, as opposed to the baseline MSSR concept which was an integrated plan with each phase of the mission further reducing the possibility of transporting Martian organisms back to Earth.

The baseline MSSR mission sequence was used as a reference configuration for this study. Modifications to this configuration as well as their impact on the total spacecraft will be discussed as we consider each of the events.

A. EVENT #1 (DURING TIME SPENT ON THE SURFACE OF MARS)

The back contamination concern for this phase of the mission is to keep the MAV, and particularly its third stage, free of any organisms in order to insure a sterile transfer of the sample in Mars orbit. One of the most common ways of doing this would be to enclose the MAV in a bioshield; however, the nature of this mission makes this a very difficult option to implement. For example, a bioshield would have to be capable of withstanding entry, landing, Martian surface environment, launch, aeroheating, and separation

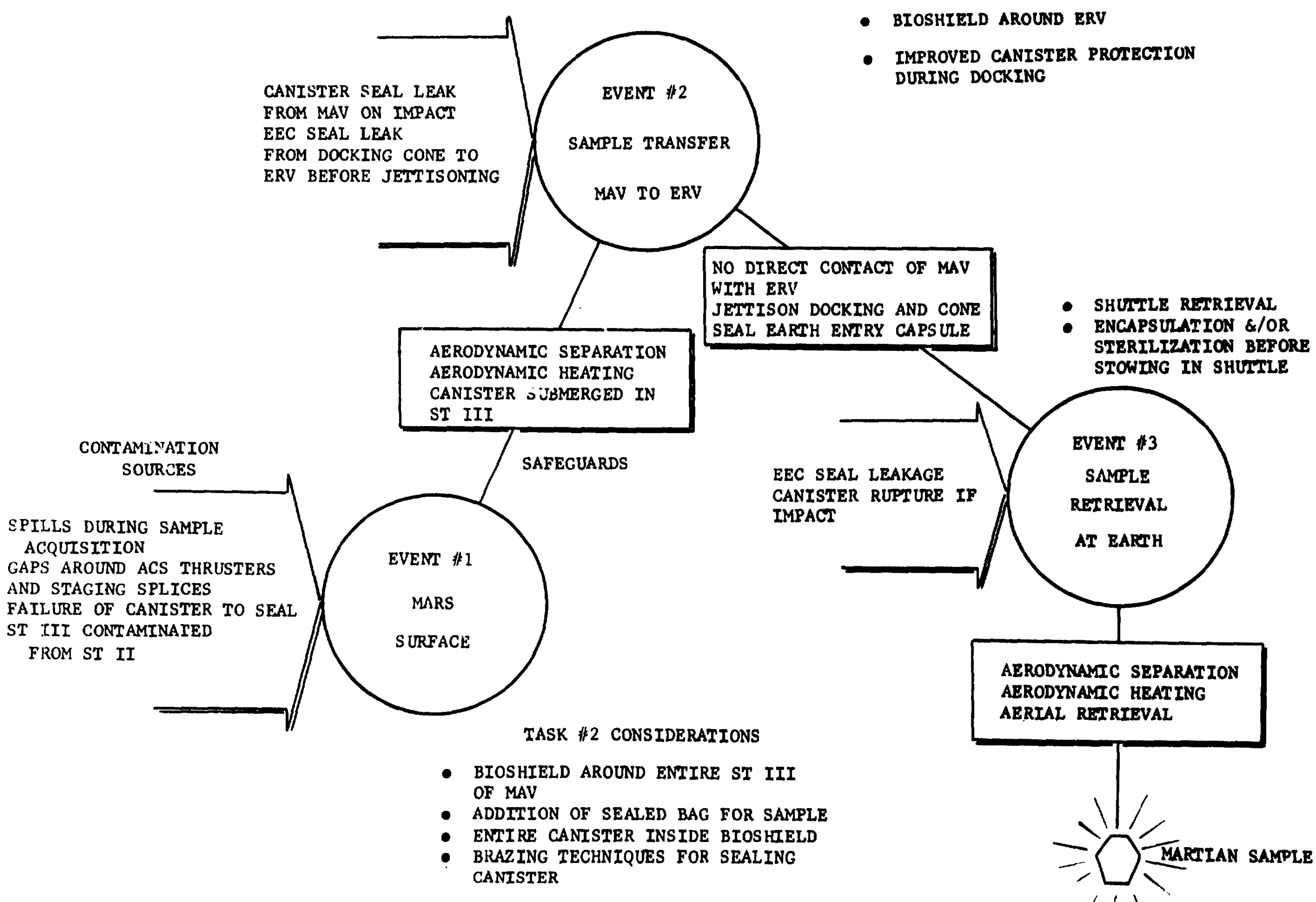


Figure III-1 Potential Contamination Sources and Preventive Measures

from the MAV. In addition, a bioshield must be capable of: (1) accommodating the transfer of the sample from the collector to the sample canister; (2) interface with the ACS thrusters so as to allow thrusting prior to bioshield separation; and, (3) not interfere with the operation of the sun sensors which are mounted on the ACS motor assemblies. A sample transfer and accommodation scheme is discussed in subsequent paragraphs. The bioshield interface with the ACS thrusters presents the most severe problems in that the thrusters point forward, aft, and laterally and all sets need to be fired prior to bioshield separation. This is an area that would require additional study should a decision be made to include the bioshield in the program; however, the scheme would probably include a sleeve from the bioshield to the thruster with breakaway seals at the thruster for both the forward and lateral thruster assemblies. The larger aft thrusters would seal directly to the bioshield (Figure III-2a). Blowout plugs could be used in the thrusters to prevent contamination through the motor assemblies while on the Martian surface. The sun sensor problem could probably best be worked by adding a sensor assembly to the bioshield for pre-launch use. This would then be jettisoned with the bioshield at the same time exposing the sensor assemblies on the ACS motors. To close off the bioshield, a seal would be installed between Stage II and Stage III of the MAV thus preventing the contamination of Stage III from the lower stages. Use of the bioshield would require modifying the baseline thermal control design and make necessary a separate thermal system for the Stage III components.

The bioshield would be a two-piece, rigid structure with the forward cap being ejected and the aft portion remaining with Stage II. Rather than have a dual forward cone, the canister and antenna would be totally exposed after bioshield separation which could eliminate the need for an extendable boom to expose the canister for docking. The bioshield being larger than the baseline MAV configuration would require a larger "bubble" on the assumed baseline Viking '75 lander capsule basecover to accommodate it.

In order to get the sample through the bioshield and into the canister without contaminating any surface inside the shield, we would propose to use a one-piece plastic sleeve that would line the sample canister and interface the outside of the bioshield. Once the sample transfer is complete,

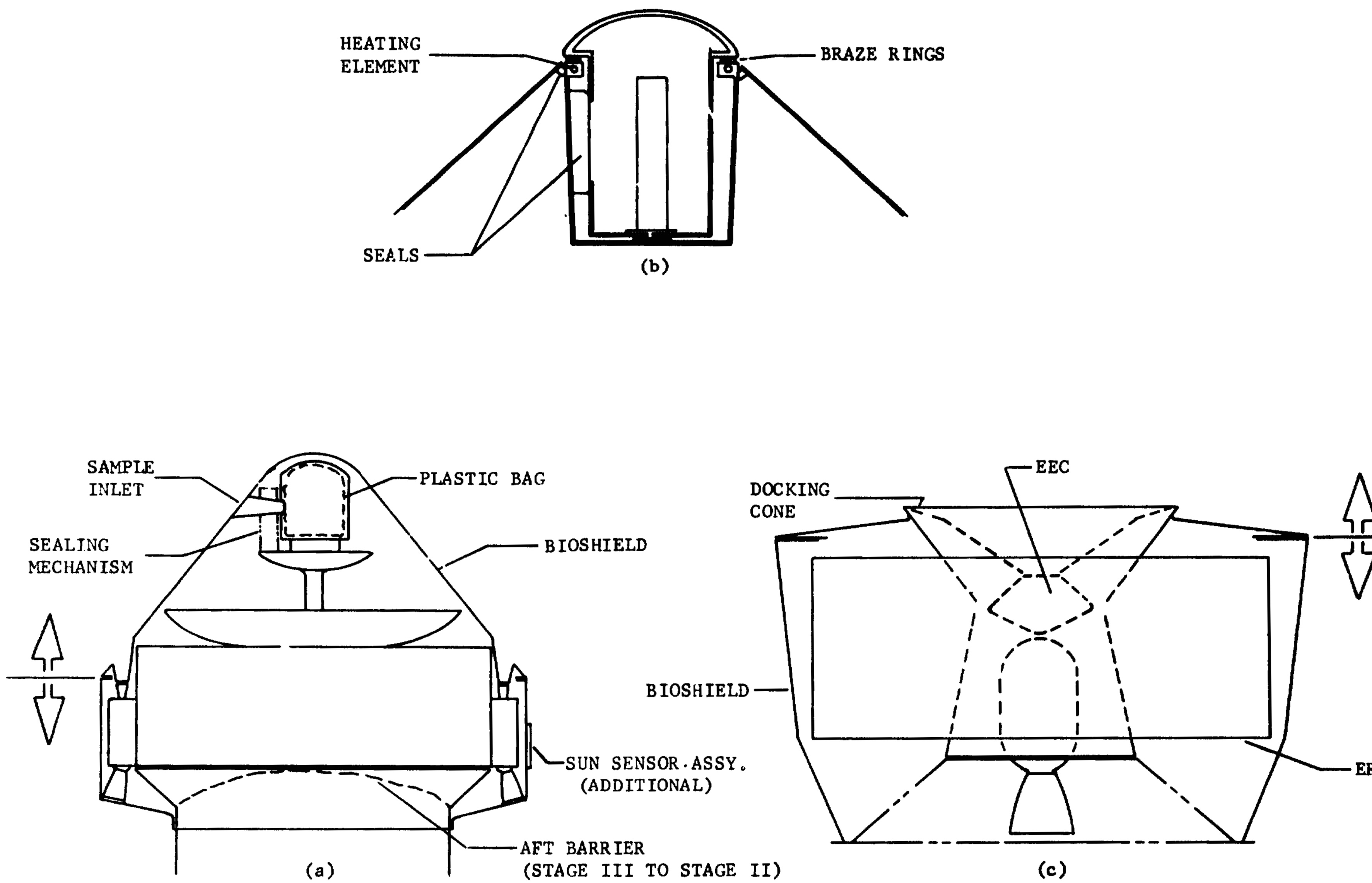


Figure III-2 Back Contamination Prevention Concepts

the sleeve would be heat sealed and then cut through the sealed portion thus providing a sealed sample, an uncontaminated canister which can then be closed off and sealed and also an uncontaminated inner surface of the bioshield. This scheme is illustrated in Figure III-3. A number of studies were conducted by Martin Marietta (Ref. 4) in the middle and lat 60s on the use of heat sealing plastics for sterile insertion techniques as applied to spacecraft maintenance. A number of materials (polyimide composites, polyesters, and fluorocarbons) were found to be capable of withstanding sterilization temperatures of 125°C and remain flexible enough to accomplish a heat sealing. This technique was adequate for sealing openings the approximate size of a manhole cover and, of course, our application would be on a much smaller scale making the task easier. Temperatures in the range of 355-360°C were required to insure a positive seal. Development work needs to be performed to insure that any gases or melted plastic impurities produced in the sealing process do not produce reactions with the soil sample and that their presence can be differentiated from the sample during subsequent chemical analysis. For MSSR application, the seal would be made while the MAV is attached to the lander and lander power could be used for the sealing. This scheme uses a top-loading canister rather than the end opening as proposed in the baseline design. This could also make the canister seal less susceptible to damage during the docking maneuver in that the MAV would not be "leading" with the canister cover.

An alternative means of enhancing back contamination control that presents fewer problems than a bioshield would be the inclusion of a pop-off outer cover for the forward end of the canister. This would prevent having to depend on the ablation process or the high surface temperature induced by aerodynamic heating (about 600°C) to effect decontamination as was the case with the Reference 1 baseline. In addition, a braze seal as shown in Figure III-2b could be used in place of the gold-deforming seal of Reference 1 to further reduce the possibility of seal leakage. These steps do not, however, afford the degree of control provided by the complete-MAV bioshield approach.

B. EVENT #2 (SAMPLE TRANSFER FROM MAV TO ERV)

This phase of the mission is more amenable to the use of a bioshield (on the ERV) in that the adverse conditions to which a MAV bioshield would

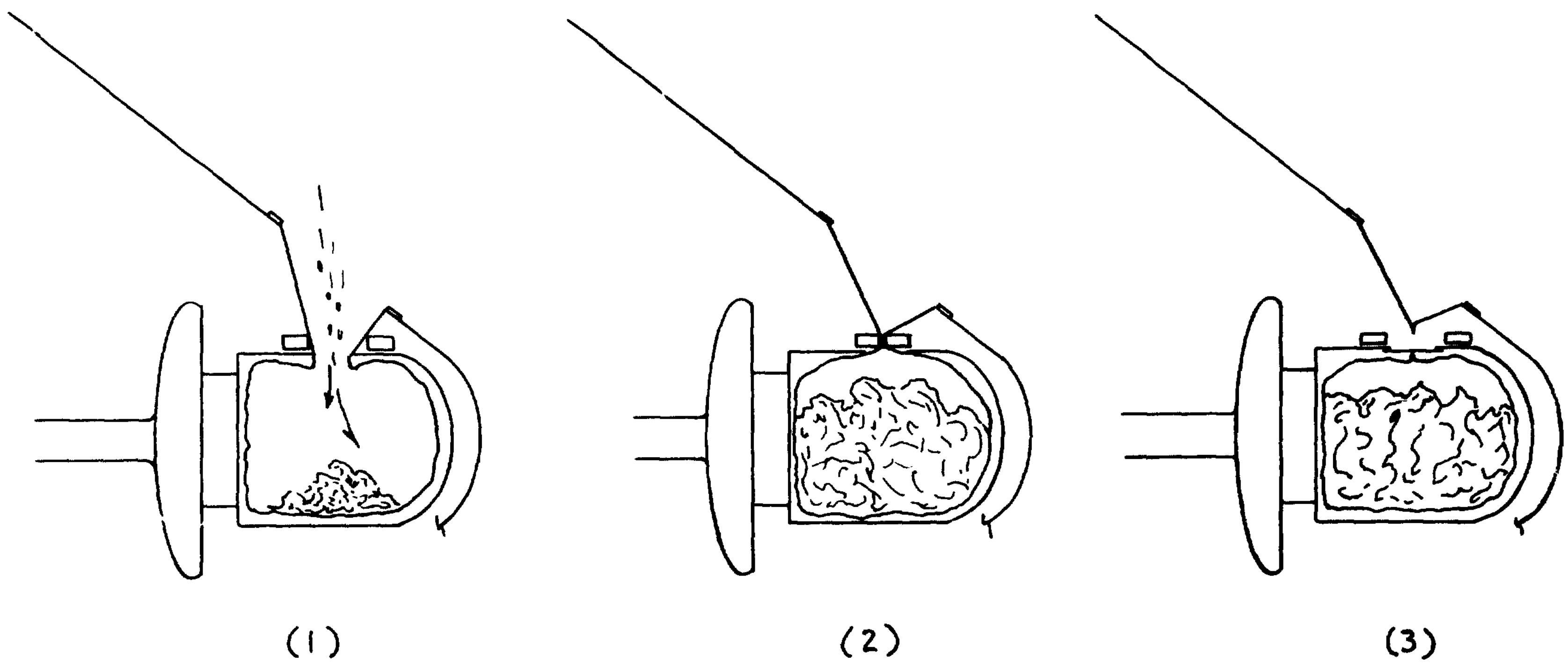


Figure III-3 Canister Sealing Sequence Using Plastic Liner

be subjected are not present. Figure III-2c shows the outline of a bioshield enclosure of the docking cone and the ERV. The MAV uses this docking cone to dock before it transfers the surface sample (see Ref. 1). This bioshield would also be a two-piece enclosure with the cap being attached to and jettisoned with the docking cone. The remainder would be attached to and remain with the orbiter. One feature of this design is that it does not affect the docking interfaces. The heat sealing plastic sleeve concept could also be applied once the MAV has completed transfer and been jettisoned (Figure III-4). This would provide a secondary sealing of the canister inside the EEC and also eliminate the penetration in the bioshield. The canister receptacle in the EEC contains latches to secure the canister once insertion is complete. To avoid interfering with the operation of these latches, the plastic sleeve would be terminated forward of the latches and sealed inside the receptacle. The mechanism for the heat sealing and cutting of the sleeve would be attached to the docking cone in order to jettison it so it would not interfere with separation of the EEC at Earth. The power required would be provided by the orbiter.

C. EVENT #3 (SAMPLE RETRIEVAL AT EARTH)

The use of the Space Shuttle to retrieve the sample in Earth orbit would open up a number of possibilities. With man in the loop, the sample could be retrieved with a clam shell device on the end of a manipulator arm (Figure III-5). Once the capsule had been enclosed, heat and/or chemical spray could be applied and the capsule monitored for days, if desired, prior to bringing it on board the spacecraft. This would necessitate replacing the Earth entry capsule of Reference 1 with an Earth Orbiting Capsule such as the one described in Task 3.

Although it could be possible to incorporate all of the aforementioned safeguards into a single mission, this would not necessarily be the thing to do. The Shuttle retrieval described for Event #3 provides the most positive results partly because of the adaptability afforded by having man directly involved. The ERV bioshield described in Event #2 would be the easiest to implement and in light of the results of the biota transfer analysis performed by Dr. Vandrey (see Ref. 1, App. G) which indicated the probability of a single spore reaching the ERV is 1/200 000, this step alone might be adequate. The bioshield for the MAV as described in Event #1 would be the most difficult to implement because of the interface

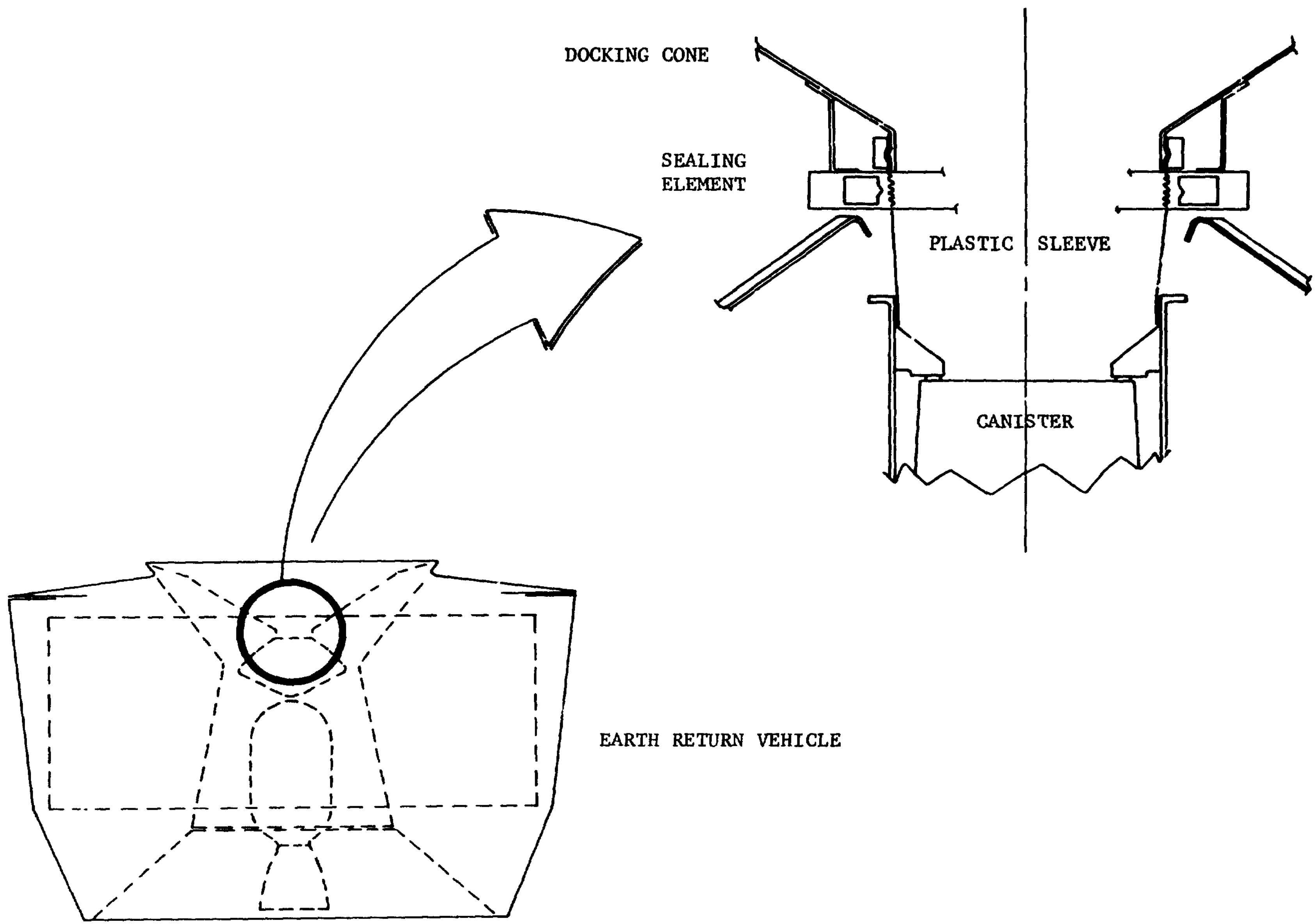


Figure III-4 Earth Entry Capsule Sealing Concept

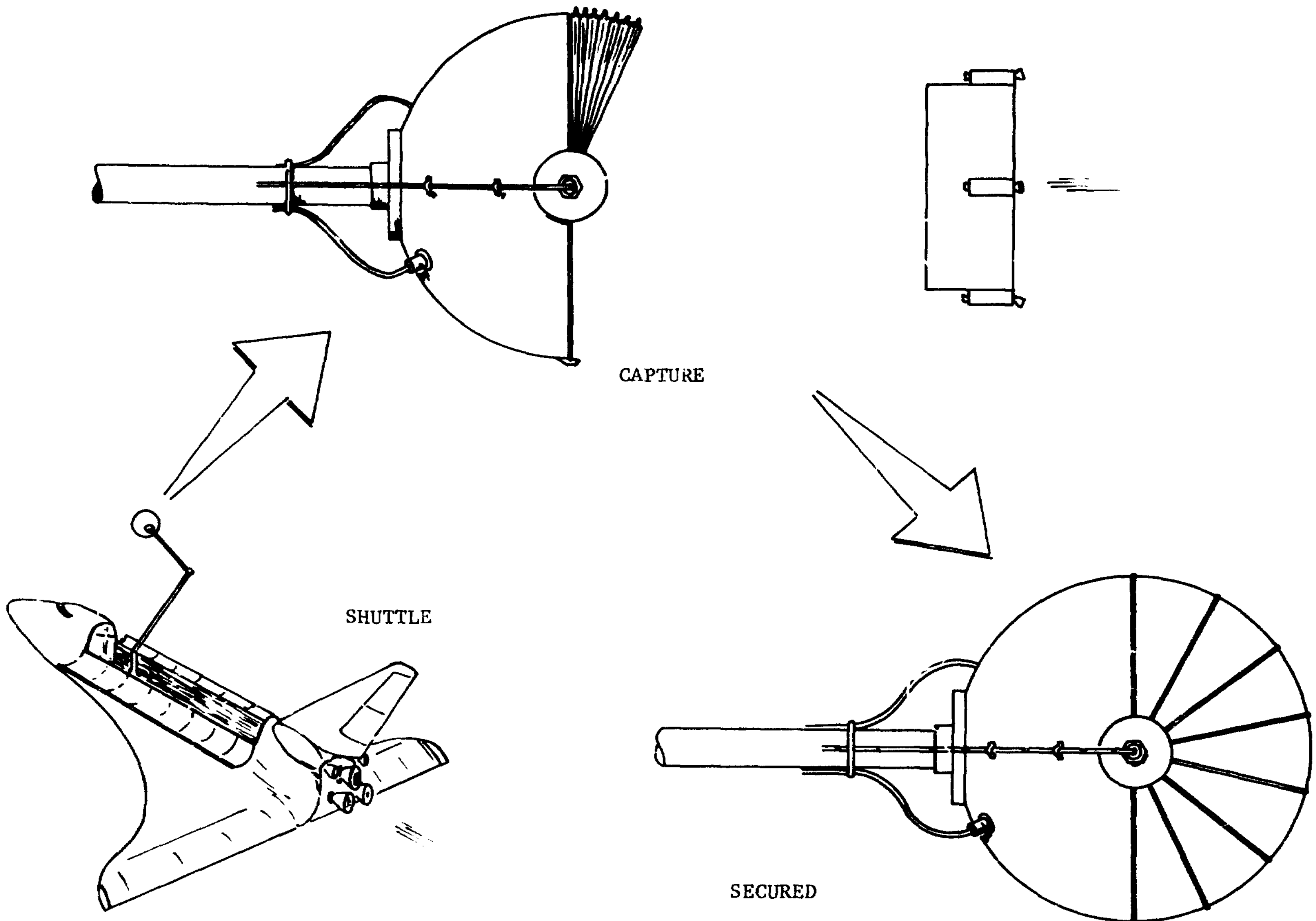


Figure III-5 Sample Recovery in Earth Orbit

problems previously discussed. While enhancing contamination control, this approach could degrade mission success probability.

D. ALTERNATIVE DOCKING CONFIGURATION

An alternative docking scheme was considered for the purpose of determining the adaptability of the previously discussed back contamination measures to configurations other than the MSSR baseline. The success of the Apollo probe and drogue system makes it a desirable candidate for sample return application. With this design, transfer of the sample from the MAV to the ERV becomes much more difficult and leads to consideration of returning the entire MAV third stage to Earth orbit. Such a mode would eliminate any unreliability associated with remote sample transfer, but would require heavier systems.

This configuration is shown in Figure III-6. Although the baseline configuration was also essentially a probe and drogue device, the mass limitations and the use of the sample canister as the probe made it impossible to duplicate the Apollo probe/drogue docking geometry. The alternative design would be a duplicate, though scaled down version, of the Apollo system.

With this configuration, emphasis is placed upon the safeguards associated with Event #1. Should the entire MAV third stage be returned, it would be essential that an effective MAV bioshield be used in addition to a positive sealing of the canister. The schemes considered for Event #2 (bioshield, EEC sealing, and docking cone removal) would not apply for this configuration. The added precaution of making a final external-surface decontamination in Earth orbit would be indicated with this approach. Also, the possibility of having an ERV in Earth orbit that could have been internally contaminated by a failure in the MAV bioshield must be considered. The possibility of a canister seal rupture, however, is greatly reduced with this design in that the canister is nested inside the MAV, sealed with the plastic bag and lid, and never disturbed.

Overall, this approach would appear to afford less-certain back contamination control features than the baseline (sample-transfer-in-Mars-orbit) approach. Early in the original study (Ref. 1), consideration was given to "tossing and catching" sample transfer techniques, to lessen back contamination potential, but these were dropped as being too unreliable from a mission success standpoint.

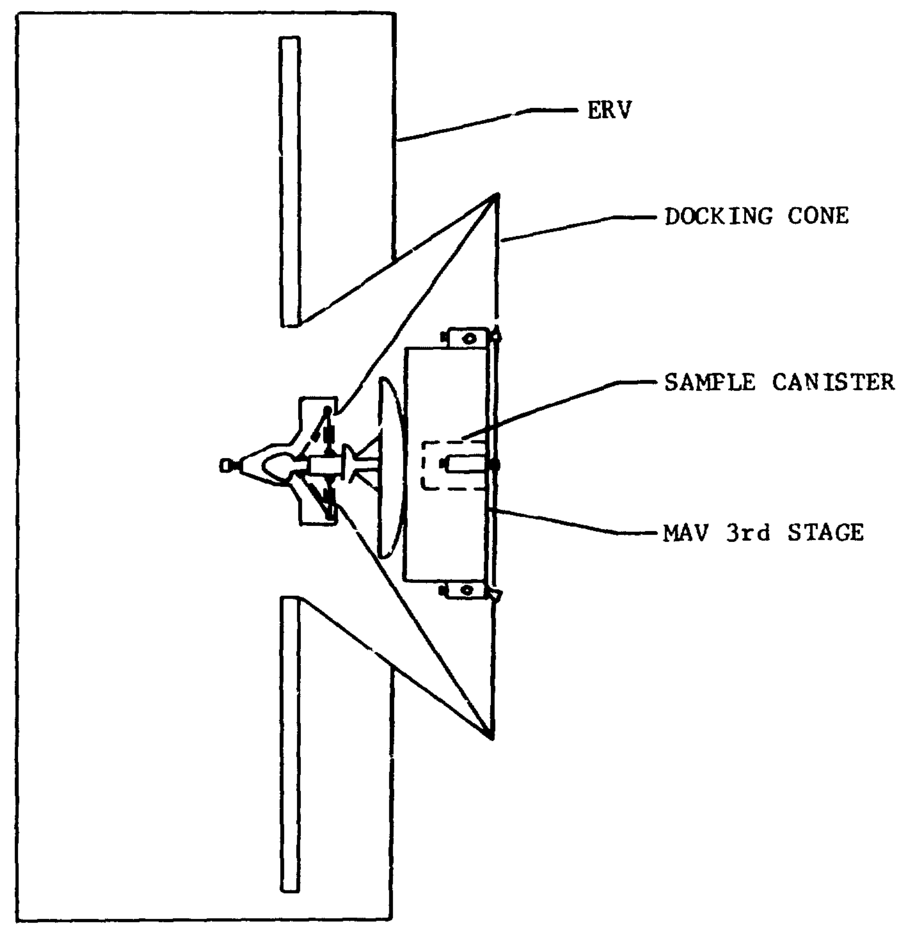


Figure III-6 Alternative Docking Concept Based on Apollo Type Probe/Drogue Design

IV. TASK 3 EARTH ORBITING CAPSULE

Capturing the Mars surface sample in Earth orbit instead of after Earth entry affords potential advantages in terms of reducing back contamination. The capture could be accomplished by the Shuttle orbiter, the Shuttle upper stage, or a satellite inspection/recovery vehicle such as the "Free Flyer" type of vehicle (see Ref. 5) that is proposed to operate in conjunction with the Shuttle orbiter. In the case of the upper stage it is conceivable that a hyperbolic rendezvous could be effected since the Tug will be capable of accelerating to escape velocity and then returning to the orbit of the Shuttle orbiter. Such a mode of operation would eliminate the need for retropropulsion on the returning sample-containing vehicle, but would represent a relatively high risk approach (high risk in terms of mission success but probably lower risk in terms of back contamination). Although this approach might be used as a backup mode, the primary capture mode will more likely involve restoring the sample carrying vehicle into an orbit compatible with that of the Shuttle orbiter and effecting rendezvous and sample transfer in that orbit.

Within this general mode of operation a number of suboptions exist. The entire Earth Return Vehicle can be injected into Earth orbit and captured intact. In this case the sample could have been previously transferred from the MAV to the ERV (in Mars orbit), or the entire MAV third stage could simply be transported back to Earth by the ERV and be injected along with the ERV into the Shuttle rendezvous orbit. Either of these options imposes significant retropropulsion requirements but the latter affords advantages in terms of reducing the sample transfer operations, see Task 2.

The most efficient approach (weight-wise) is to transfer the sample while in Mars orbit into a small capsule capable of injecting itself into Earth orbit after being returned to Earth by the ERV. This approach also permits a more direct comparison with the Earth Entry Capsule mode evaluated in Ref. 1. Therefore a preliminary system design for this type of Earth orbiting capsule has been performed. The subsystems mass breakdowns for this vehicle are presented in Table IV-1 and its configuration is shown in Figure IV-1. Table IV-1 and Figure IV-1 also show the Earth entry capsule of Ref. 1. A comparison reveals that the two systems are

simil-- in terms of total mass and size. This means that final choice of the mission mode would depend more on a detailed comparison of mission success probability and back contamination risk than on payload performance.

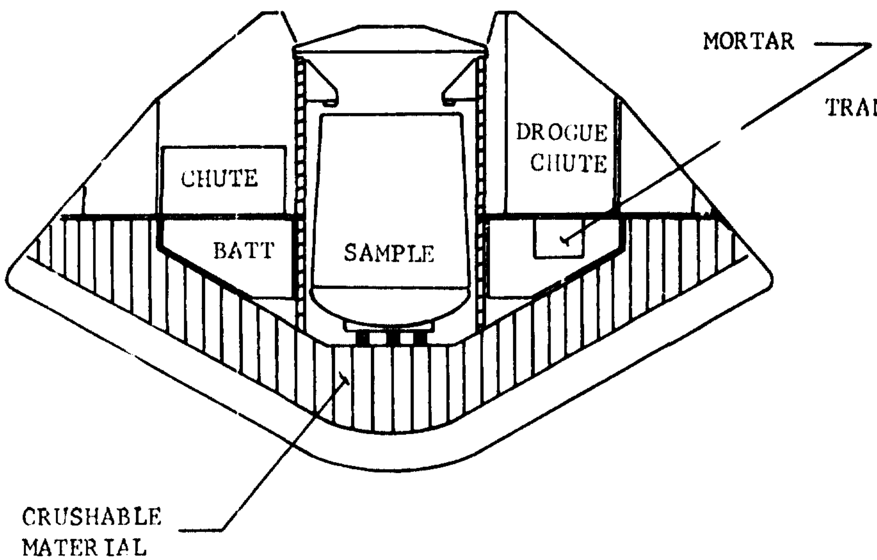
Table IV-1 Comparison of Mass Breakdown for Earth Orbiting and Entry Capsules

A. Mass Breakdown for Earth Orbiting Capsule

ITEM		MASS (kg)
Structure		5.0
Sample Container	.9	
Sample Container Receptacle	1.8	
Basic Structure	2.3	
Telecommunications		2.8
Power		
Battery, Power Control	2.3	
Solar Array (Earth Orbit)	1.3	
Guidance/Attitude Control		3.8
Propulsion Inerts		1.6
Science (Sample)		1.0
Sub Total		17.8
Contingency (10%)		1.8
Total Dry Mass		19.6
Orbit Insertion Propellant		10.2
(For 24 hour, 500 Km Periapsis Orbit)		
Total		29.8

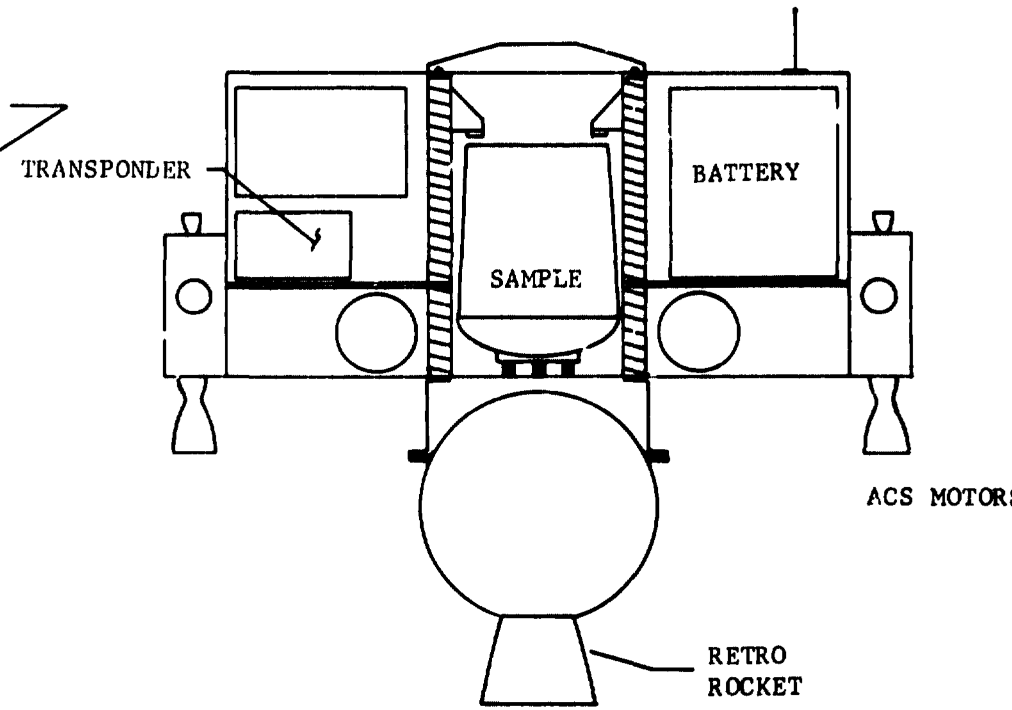
B. Mass Breakdown for Earth Entry Capsule (From Ref. 1)

ITEM		MASS (kg)
Structure		8.14
Sample Receptacle	1.86	
Aeroshell Structure (incl. 2.5# Crush Material)	3.91	
Inner Structure	.93	
Upper Frustum	.61	
Lower Frustum	.83	
Ablator		5.76
Parachute System		3.54
Flotation System		1.36
Power and Cabling System		1.81
Electronics		2.95
Pyrotechnics		.91
Contingency 5%		1.53
		26.00
Sample and Container		2.00
		28.00



EARTH ENTRY CAPSULE

28 Kg



EARTH ORBITING CAPSULE

29.8 Kg

Figure IV-1 Earth Orbiting Capsule and Earth Entry Capsule Configurations

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A. INFLUENCE OF ENTRY ANGLE OF LANDED MASS

The Viking '75 out-of-orbit Mars Entry Capsule and the direct entry capsule patterned after it that was baselined for the initial sample return study (Ref. 1) were designed to enter at angles as close to skip-out as possible in order to reduce entry environment severity. The skipout angle is somewhat steeper for direct entry (-17.5^{o} vs -15.0^{o}) but in either case, due to a dynamic pressure overshoot phenomenon that occurs near skipout*, selecting entry angles near the skipout boundary results in having to delay deploying the parachute until lower altitudes are reached than would be possible with slightly steeper entry angles. This situation is illustrated in Figure V-1. The figure shows that the optimum entry, from a parachute deployment standpoint, occurs at an angle a few degrees steeper than the skipout-bounded entry angle. Figure V-1 was constructed for out-of-orbit entries but similar curves exist for the direct entry case only they are shifted to the right. The out-of-orbit mode affords the advantages of landing site certification and the added mass of orbit insertion propellant required is available in most of the mission options under consideration.

Selecting the optimum entry angle allows a greater entry mass to be accommodated before reaching a point where insufficient altitude exists at chute deployment to allow the chute and retropropulsion system to effect a landing. The steeper angles do, however, result in a heavier aeroshell. The variation of propellant weight, aeroshell weight, and resultant landed weight for a fixed entry weight are shown in Figure V-2. Although this figure shows that the net effect, for a fixed entry weight, is a decrease in landed weight with increasing entry angle, the important thing is that greater landed weights can be achieved by selecting higher entry angles when additional entry weight is available. This is apparent from Figure V-3.

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* When altitude-velocity plots are constructed for a family of entry angles (γ_Es) just above the angle for skipout, it is observed for lifting entry that there is a region where these curves cross-over each other, i.e., at a given altitude the velocity is greater for a steep γ_E than for a shallower one. As larger and larger γ_Es are considered, this trend reverses and the velocity at a given altitude is greater for the steeper entry angles.

B. INFLUENCE OF ENTRY CORRIDOR WIDTH, L/D AND DESCENT PROPULSION THRUST LEVEL ON LANDED MASS

From Figure V-1 it is apparent that narrowing the entry angle corridor increases the allowable parachute deployment altitude and thus produces the same beneficial effect in terms of increasing landed mass described above. Narrowing the corridor width to 2 degrees has been found to be possible by going to optical guidance. Conceivably the 2^o entry corridor could be narrowed even further. The entry sensitivities were discussed in the URDMO final report (see Ref. 1).

A higher L/D (Lift/Drag ratio) also produces higher parachute deployment altitudes, See Figure V-4, but it causes a shift of the optimum entry angle to a still steeper value which means some of the added entry weight is absorbed in aeroshell structural weight. The increased angle of attack required by the higher L/D also causes some heat shield weight increase.

Finally, increasing the thrust level of the descent engines has been examined since this causes the deceleration impulse to be accomplished in a smaller altitude span. This in turn means that lower parachute deployment altitudes can be accepted, and that finite burn losses are reduced.

The total increase in landed weight of decreasing the corridor width to 2^o, increasing the L/D from 0.20 to 0.25 and increasing the thrust level 50% is shown in Figure V-5. It is apparent that for a fixed entry weight only relatively small landed weight increases result from the fairly large variations in entry corridor width, L/D and thrust level examined. The main advantage of modifying these parameters is to shift the point at which constraints are reached and thus to make it possible to use the added entry weight to increase landed weight. The resulting landed weight/entry weight relation is shown in Figure V-6. From this figure it is seen that the ratios of landed mass to entry mass stays relatively constant at about 70% over the range of entry mass evaluated.

C. EXTENSION OF STUDY DATA TO LARGE SYSTEMS

The preceding data were generated for relatively small increases in entry mass (10% to 25%) relative to the baseline system derived in Ref. 1, and as pointed out earlier, this provides adequate increased lander mass for rendezvous type missions; however, some of the Shuttle/Tug missions of

interest involve entry masses that are 3 or more times greater than those of the earlier baseline. Due to the diameter constraint of the Shuttle payload compartment, however, the Mars entry aeroshell diameter cannot easily be increased much more than about 16% (larger diameters would require deployable or assembly-in-space techniques). This means the entry ballistic coefficients may well increase by almost a factor of three if the full launch vehicle capability is utilized. Figure V-1 shows that a ballistic coefficient increase of a much smaller amount (33%) has a significant impact on allowable parachute deployment altitude. In fact, that amount of increase in ballistic coefficient was found to degrade the deployment altitude by the same degree that it would be improved by increasing the L/D ratio from 0.20 to 0.25. If even greater L/D values are used to offset the high ballistic coefficients the resulting entry angle of attack becomes very large (> 20°) and the aeroshell afterbody no longer stays in the mild environment of the shadowed flow region. Using a sharper entry cone angle would help solve this problem, or the parachute could simply be designed to withstand the higher dynamic pressure conditions incurred with the large ballistic coefficient vehicles. For such vehicles a new set of component weight relations should be developed to properly interpret the parametric mission analysis data of Task 1, i.e., the values of landed mass to entry mass for such vehicles could differ substantially from the values established in this task. Consequently it is recommended that system design studies be performed for high-ballistic coefficient classes of entry/lander vehicle.

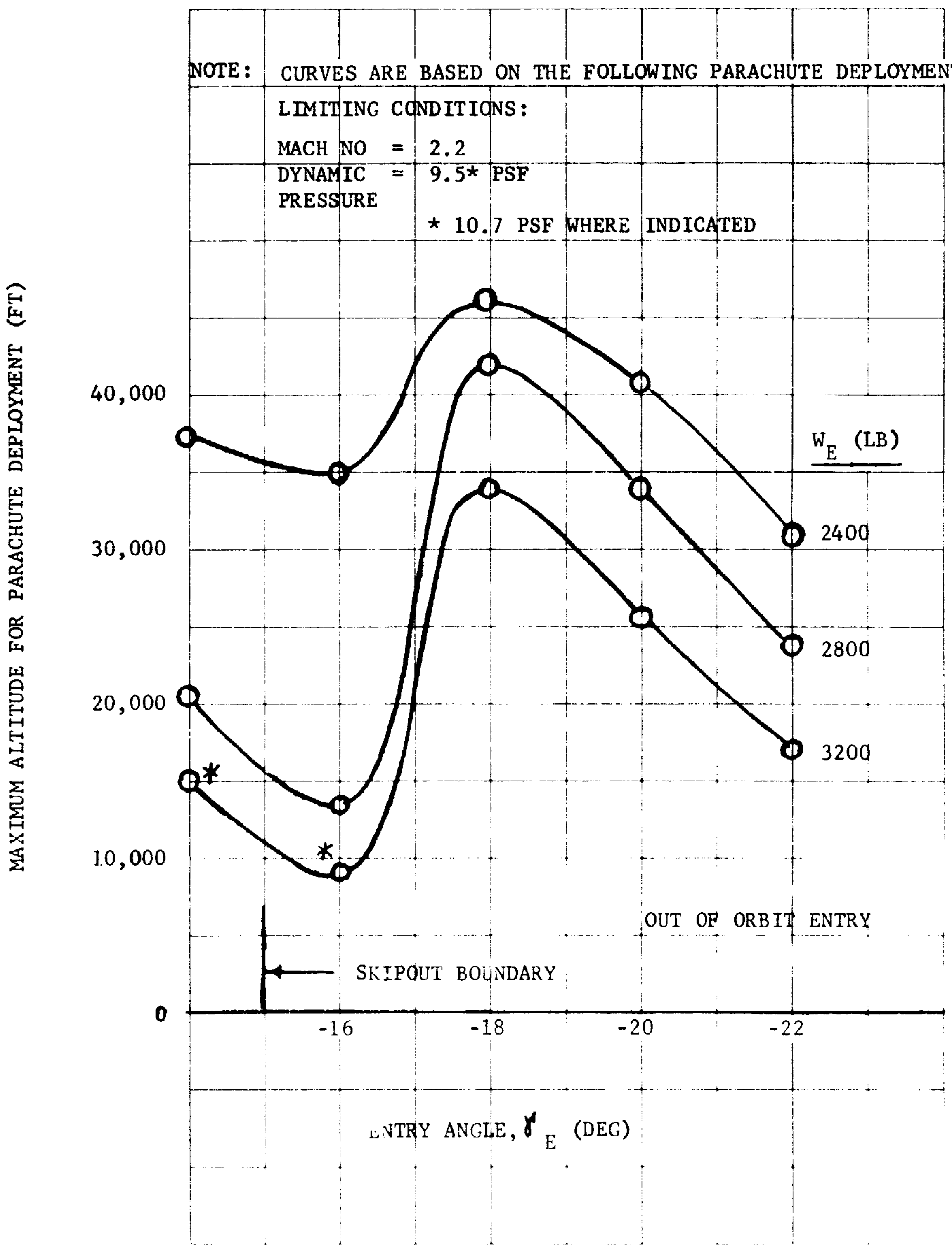

Figure V-1 Influence of Entry Angle on Allowable Parachute Deployment Altitude

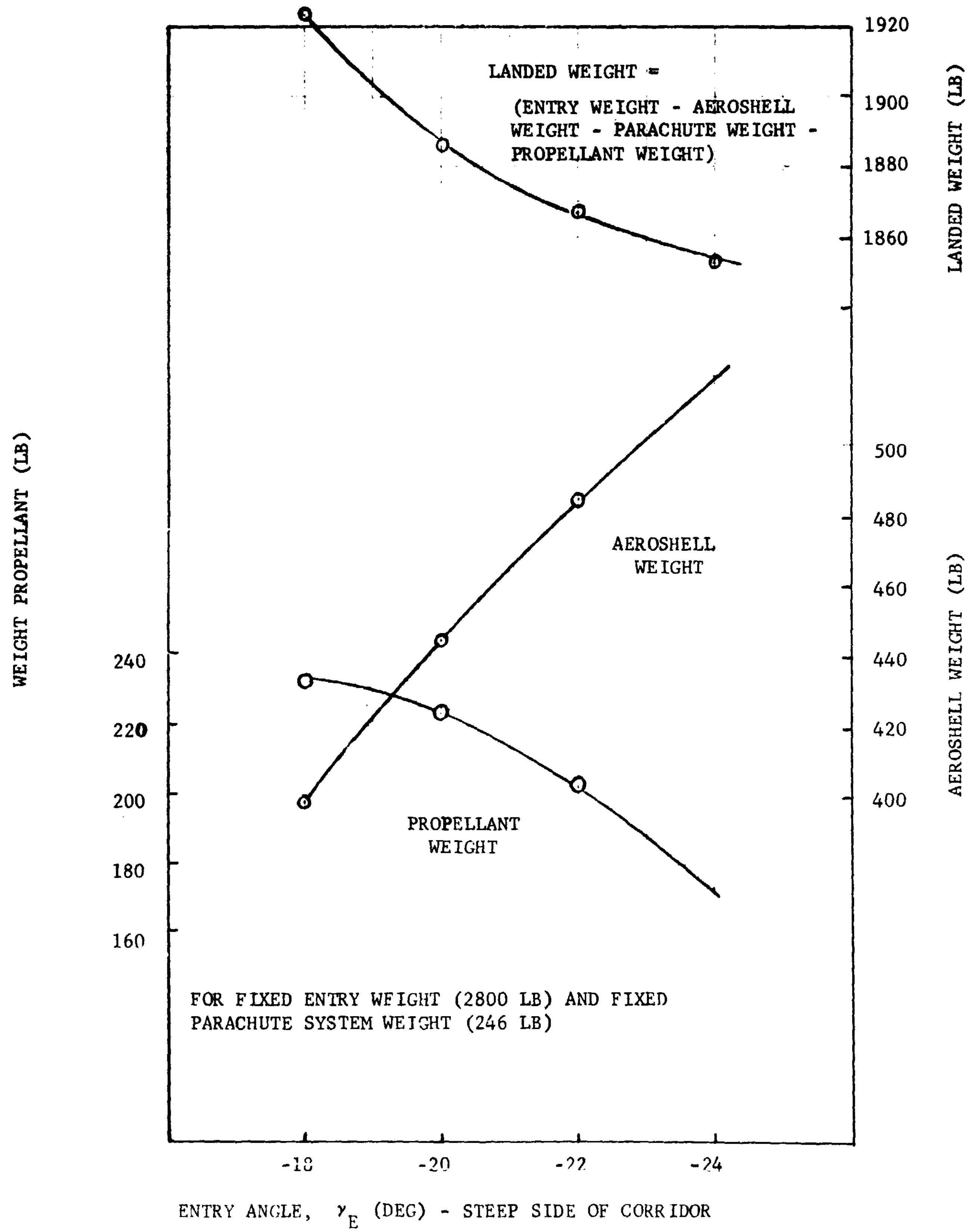

Figure V-2 Variation of Propellant Weight, Aeroshell Weight and Landed Weight with Entry Angle for Fixed Entry Weight

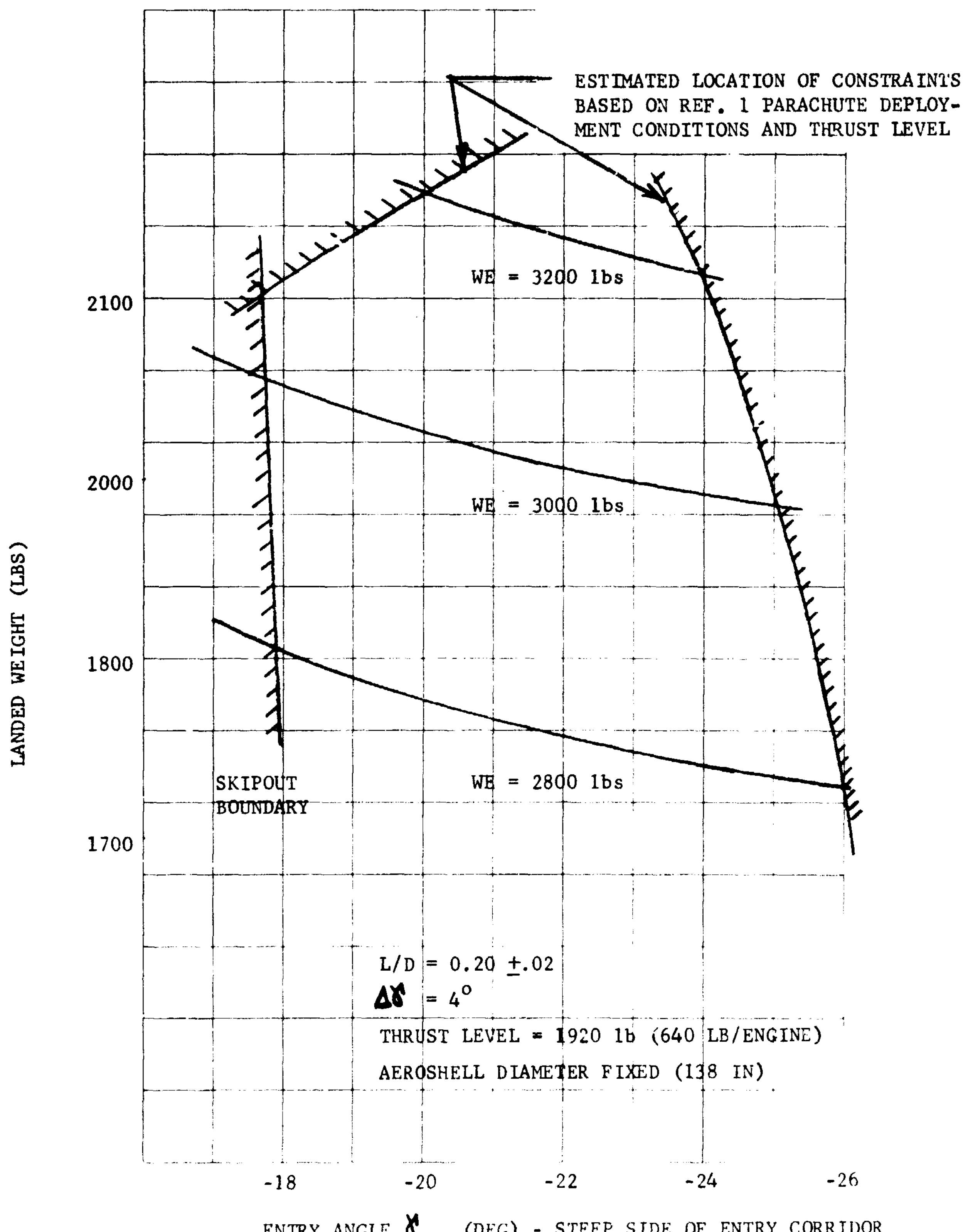

Figure V-3 Variation in Landed Weight and Entry Weight Constraints with Entry Angle

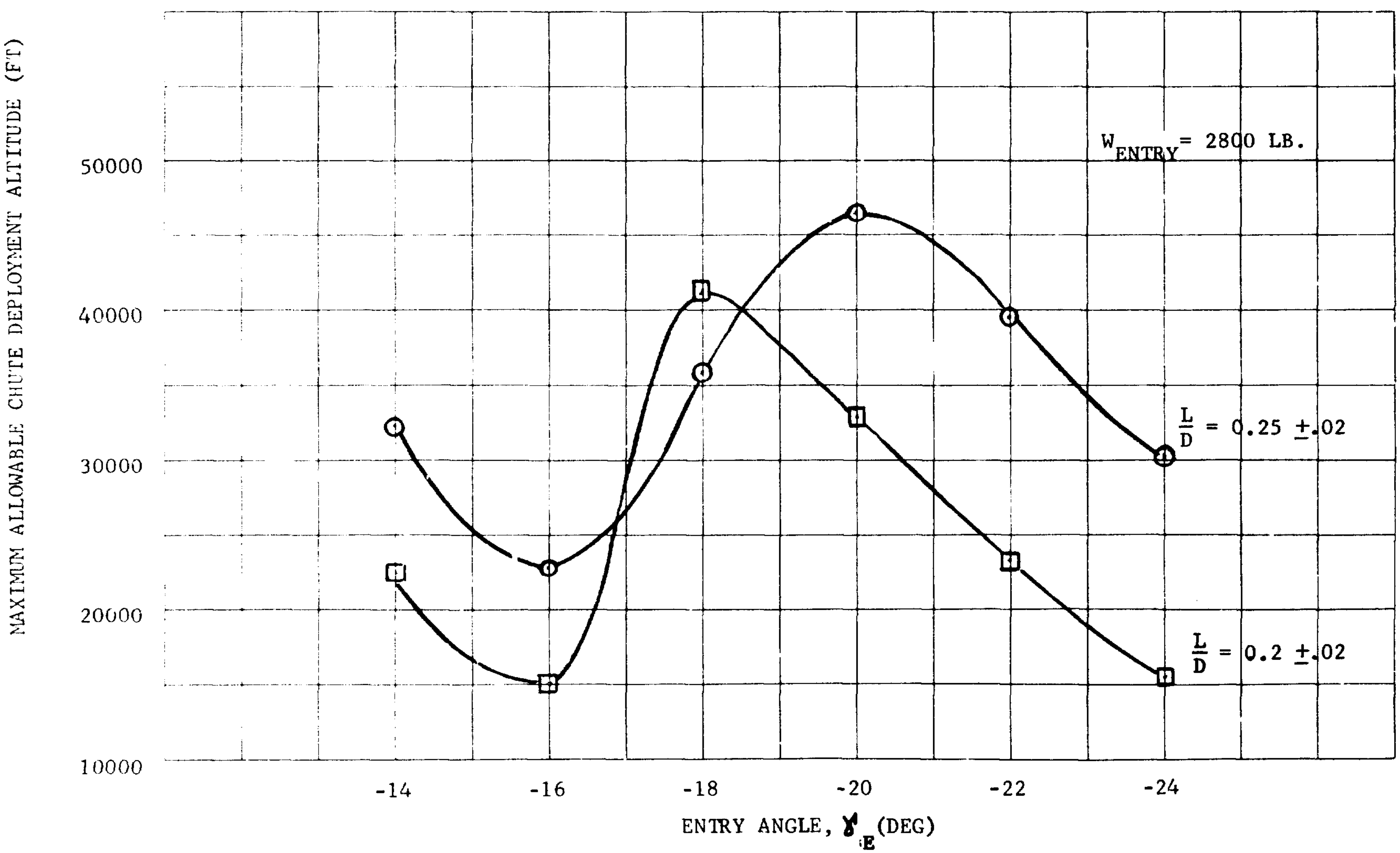

Figure V-4 Altitude for Parachute Deployment as Influenced by L/D

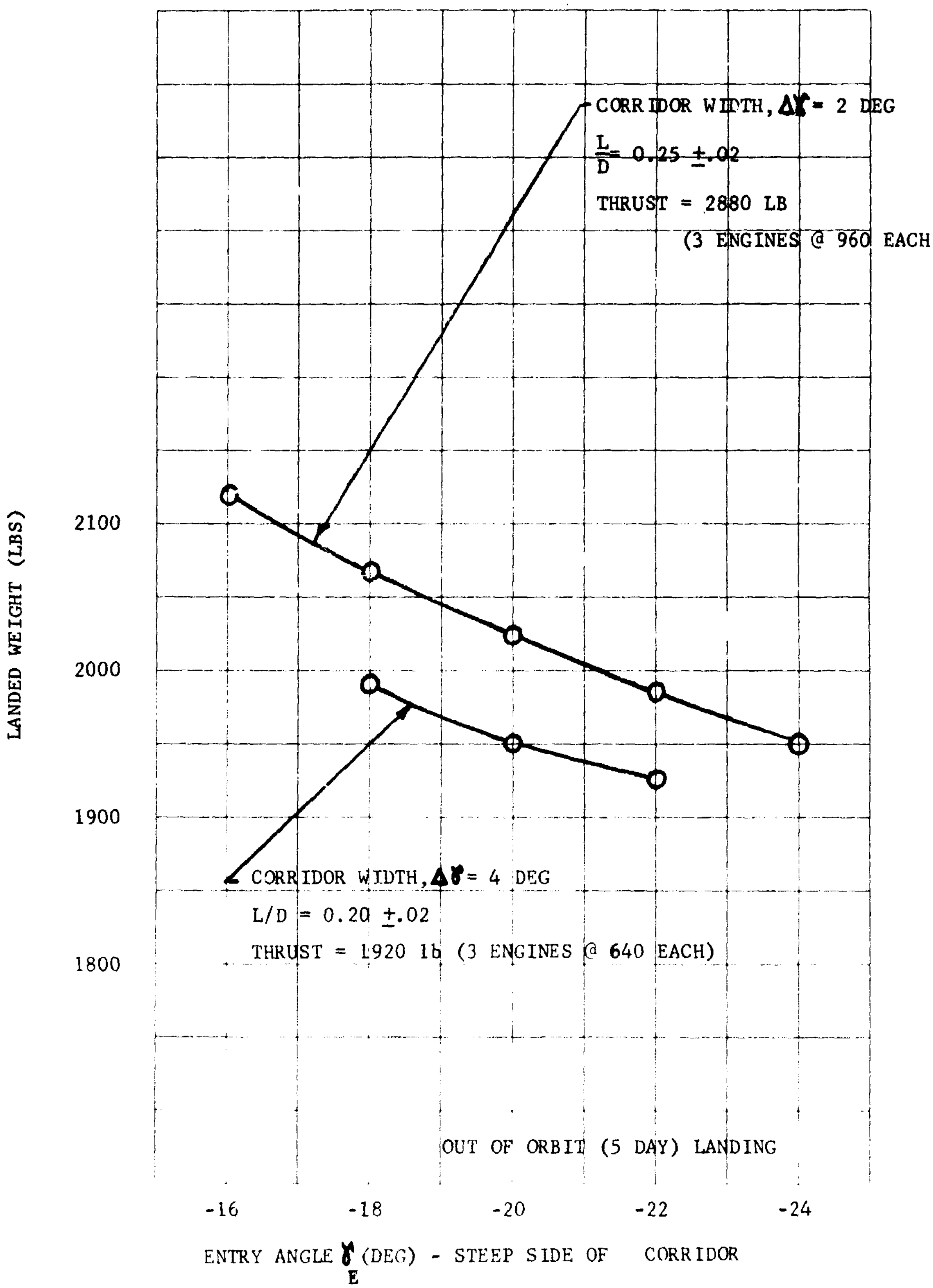

Figure V-5 Influence of Smaller Corridor, Higher L/D and Higher Thrust on Landed Weight

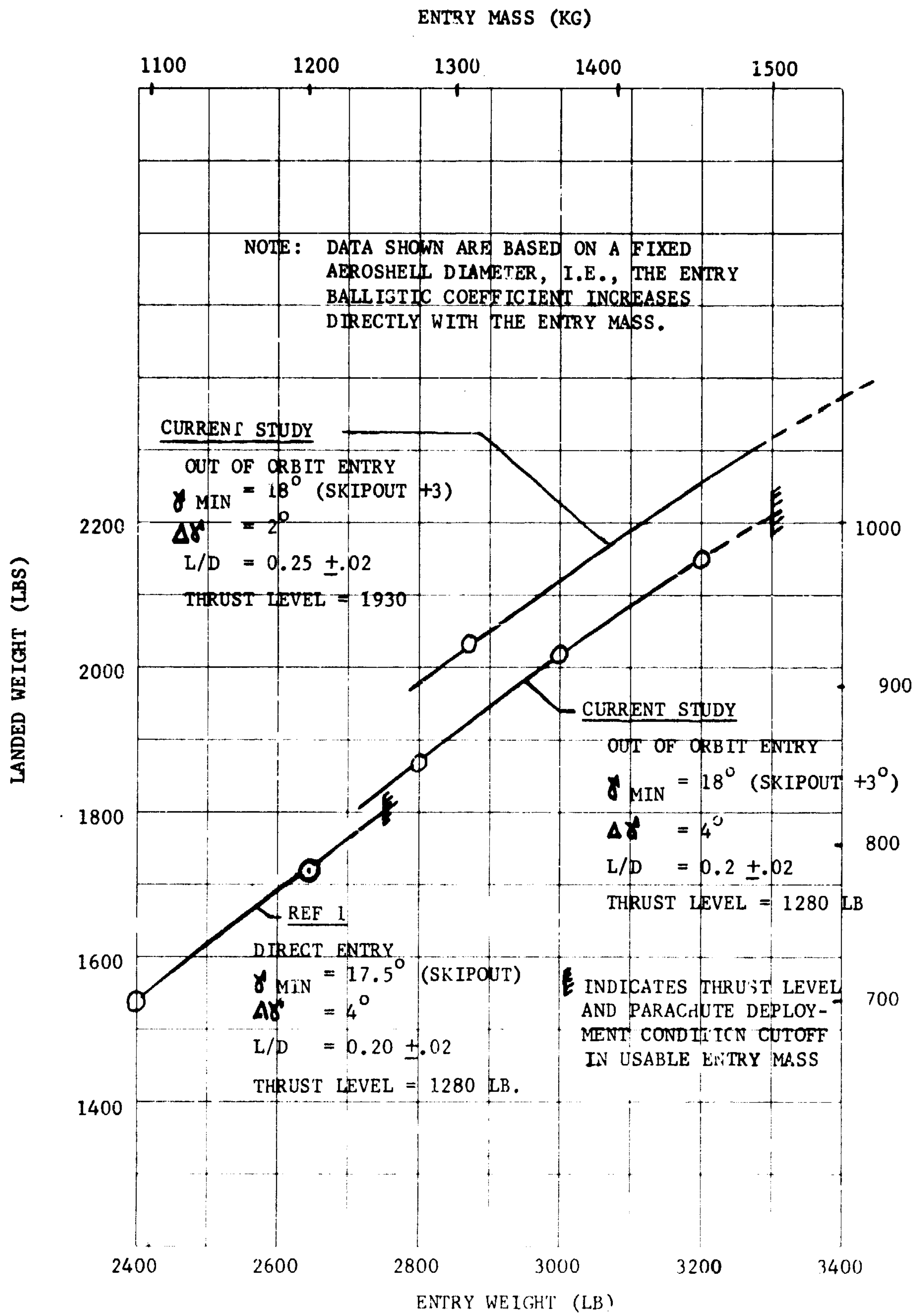

Figure V-6 Improvement in Landed Mass and Relaxation of Entry Mass Constraints is made possible by adjustment of Entry/Descent Parameters

VI. SUMMARY OF PROGRAM AND TECHNOLOGY IMPLICATIONS

After a number of studies and reviews of the MSSR mission by NASA, industry, and the science community, several key conclusions have emerged:

1. The Mars sample return is an exciting mission concept, one that promises to answer more important, first order science questions than any other unmanned venture to Mars, and one that should and will be performed.

2. The MSSR mission will be an expensive one that is and will be difficult to fit within the NASA budget for planetary programs.

3. The problem of adequately safeguarding against back contamination is a formidable and perhaps even insurmountable one, depending upon the allowable probabilities finally agreed to.

4. Most of the technology required to perform a minimum MSSR mission, neglecting the potential impact of stringent back contamination control regulations, is in hand now. In fact, much of this minimum mission technology can be derived directly from the Viking program.

At least two possible MSSR mission scenarios present themselves:

1. The conservative planning approach. Allow the results of the Viking'75 landings and of a possible follow-on Mars mission in 1981 to be evaluated. Then plan an MSSR mission for a 1988 launch to capitalize on the favorable Earth to Mars launch/arrival performance requirements of that opportunity. This approach would probably be most compatible with NASA budget projections and is in fact the mission timing currently in the NASA planetary mission model.

   The choice of mission mode for performing the MSSR in 1988 can proceed from two sets of logic. On the one hand the probable availability of the Shuttle/Tug launch system makes it possible to use heavier spacecraft elements. This would allow either short total mission times (e.g., 731 days) or the simpler but heavier mission modes such as the direct return without Mars orbit rendezvous.

On the other hand, the later launch date (1988) should allow technology development to support the lighter weight Mars orbital rendezvous mode with a high degree of confidence.

It would seem that the more energy-efficient MOR mode would be the choice over the brute force direct return technique, given the availability of reliable rendezvous and docking technology. Also, the MOR mode offers significantly better control of back contamination.

2. The Mars-emphasis planning approach. If the results of the Viking '75 landings were sufficiently spectacular to stimulate an intensive program of Mars exploration, an MSSR mission could be flown as early as the 1981 or the 1983/84 opportunities. These early missions would probably emphasize the use of proven hardware concepts. This again would probably favor the MOR mode because such a mission could be derived from Viking technology.

The results of this study, therefore, reaffirm that MOR-related technology developments, particularly those in the areas of remote rendezvous algorithm development, rendezvous sensor development, and multi-degree of freedom experimental docking simulation would greatly benefit the accomplishment of the sample return mission no matter when it is flown. Additional areas have been identified where new technology developments could further improve mission performance, reduce costs, or increase mission success. These areas which support back contamination control, increasing the landed weight, and retrieval of the sample from Earth orbit are summarized as follows:

1. Development of "plastic bag" sealing techniques to minimize contamination of sample container external surfaces.
2. Development of bioshielding techniques for the Mars ascent vehicle upper stage that are compatible with autonomous launch operation.
3. Optical guidance to decrease Mars entry angle dispersions (to improve landed weight).
4. Development of methods for extending Mars entry capsule aeroshell diameter after separation from Shuttle launch vehicle (to improve

landed weight capability where massive landers are involved, e.g. direct return missions).

5. Shuttle manipulator arm modification for sample capsule docking or capture.

Recommended areas for further sample return studies include:

1. Development of detailed requirements for Mars rendezvous sensors. (Study sensitivity of rendezvous performance to sensor accuracy and range capability.)
2. Earth return vehicle design.
3. Earth orbit capture--rendezvous and docking modes and algorithm development.

VII. REFERENCES

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APPENDIX A

SUMMARY OF MISSION PERFORMANCE PARAMETERS

The range of mission performance parameters for the key mission modes (dual launch, out-of-orbit, MOR; single launch, out-of-orbit, MOR; and single launch, direct entry, MOR) are presented in Tables A-1 through A-6. Tables A-1, A-3, and A-5 are for minimum time missions and A-2, A-4 and A-6 for missions which afford maximum ERV weight margin. The parameters are defined as follows:

C_{3E}	Earth launch energy (km^2/sec^2)
Vhp_M	hyperbolic excess velocity at Mars (km/sec)
DLA	declination of outgoing asymptote from Earth (deg)
ΔV_{MOI}	Mars orbit insertion velocity change (km/sec)
C_{3M}	Mars launch energy (km^2/sec^2)
Vhp_E	hyperbolic excess velocity at Earth return (km/sec)
ΔV_{EOI}	Earth orbit insertion velocity change (km/sec)
ΔV_{TEI}	trans-Earth injection velocity change (km/sec)
θ_{ME}	Mars-Earth transfer angle (deg)

The range of parameter values presented in the tables do not exactly correspond to the total range for the viable missions. The latter, however, are contained in the former. Complete launch/encounter grids for these parameters and others are available at MMC upon request.

Several interesting features of the data should be pointed out. As expected, the Earth-to-Mars legs for both the minimum time missions and the maximum ERV weight missions optimizes with the same Earth launch/ Mars encounter window. The primary difference is in the Mars-Earth leg and particularly in the time the ERV departs Mars for Earth. By comparing A-1, -3, -5 with A-2, -4, -6, respectively, it can be seen that the potential ERV weight margin is spent on ΔV_{TEI} and ΔV_{EOI} in order to shorten Earth-return trip time. Large ΔV_{EOI} maneuvers are possible because of the very light (20 kg) entry capsule. For example, the 5.03 km/sec ΔV_{EOI} for the 1986 Type I Tug mission (Table A-1) can be accomplished with ~47 kg of propellant.

Table A-1 Dual Launch, Out of Orbit, MOR: Minimum Mission Time

Year	Type	L/V	Minimum Mission Time (Days)	ERV Margin (kg)	Stopover Time (Days)	Earth Launch Date	Mars Arrival Date	C_{3E}	Vhp_M	DLA	ΔV_{MOI}	C_{3M}	Vhp_E	ΔV_{EOI}	ΔV_{TEI}	θ_{ME}
1981	I	IUS	**852	31	240	12/25/81 01/14/82	08/17/82 09/06/82	12.66 25.88	3.95 12.35	- 0.6 -61.1	1.46 8.79	33.17 74.65	7.48 12.60	2.79 6.26	2.93 5.38	150.4 190.7
		T3E	---	---	---	--	--	---	---	---	---	---	---	---	---	---
	II	IUS	**870	36	150	12/17/81 01/06/82	11/02/82 11/22/82	8.90 16.82	3.71 4.40	18.8 49.0	1.45 1.91	28.80 66.66	6.77 12.69	2.40 6.32	2.62 4.99	175.1 194.7
		T3E	**900	20	190	11/27/81 12/17/81	10/03/82 10/23/82	8.94 11.77	3.13 3.53	17.3 44.9	1.09 1.33	24.53 74.65	7.48 12.60	2.06 6.26	2.93 5.38	185.1 194.5
1983/ 1984	I	IUS	**865	31	230	02/29/84 03/20/84	10/03/84 11/02/84	14.78 70.26	3.75 7.09	-24.1 -65.4	1.47 4.03	21.84 36.08	5.76 7.98	1.89 3.08	2.10 3.13	188.6 198.1
		T3E	---	---	---	--	--	---	---	---	---	---	---	--	---	---
	II	IUS	898	22	190	01/27/84 02/16/84	11/30/84 12/10/84	8.78 10.52	4.55 4.88	- 3.1 23.1	2.01 2.25	21.84 26.08	5.76 7.98	1.89 3.08	2.10 3.13	188.6 198.1
		T3E	928	20	250	12/28/83 01/17/84	10/11/84 10/21/84	10.59 12.81	3.60 3.90	3.2 29.5	1.37 1.54	18.62 95.33	5.11 13.47	1.60 6.93	1.85 6.39	183.7 202.9
1986	I	Tug	* 462	59	30	03/31/86 04/20/86	10/01/86 10/11/86	8.30 11.72	3.79 4.26	-47.8 -60.7	1.50 1.81	41.69 48.30	10.74 12.34	5.03 6.06	3.50 3.91	273.5 305.1
		T3E	**882	0	320	05/10/86 05/30/86	11/10/86 01/09/87	8.52 18.92	3.18 3.76	-26.8 -42.7	1.12 1.47	28.65 40.69	6.78 8.88	3.07 14.85	2.90 11.93	190.6 235.5
	II	Tug	**761	8	50	06/20/86 07/10/86	06/06/87 07/26/87	17.49 51.13	4.74 5.33	- 8.6 -58.0	2.15 3.10	32.64 60.99	7.96 22.76	2.41 2.64	2.61 3.43	138.4 170.7
		T3E	947	13	480	04/11/86 05/01/86	01/17/87 01/27/87	8.37 12.46	4.00 4.31	2.2 -15.5	1.63 1.84	10.11 10.74	17.24 30.36	0.89 1.07	1.12 1.17	136.8 177.6
1988	I	Tug	731	63	100	07/09/88 07/29/88	 02/25/89	12.19 16.12	2.55 2.97	15.8 37.3	0.77 1.00	--- 65.91	--- 11.69	--- 5.58	--- 4.92	--- 151.7
		T3E	811	27	200	07/09/88 07/29/88	02/05/89 02/15/89	12.19 16.12	2.52 2.72	9.2 28.7	0.76 0.86	54.76 66.39	8.59 10.94	3.45 5.03	4.29 4.94	159.3 188.6
	II	Tug	**751	59	30	07/09/88 07/29/88	06/05/89 07/15/89	18.83 48.52	3.24 4.85	1.3 -48.1	1.15 2.23	55.79 71.72	8.96 28.18	3.69 19.86	4.35 15.95	141.0 164.6
		T3E	---	---	---	--	--	---	---	---	---	---	---	---	---	---
1990	I	Tug	**780	0	140	09/09/90 09/29/90	04/17/91 05/27/91	19.65 31.87	2.33 3.02	28.1 60.8	0.67 1.03	62.48 70.01	4.79 14.26	1.46 7.55	4.73 8.15	159.6 170.1
		T3E	---	---	---	--	--	---	---	---	---	---	---	---	---	---
	II	Tug	**756	102	30	08/24/90 09/13/90	07/19/91 08/28/91	14.96 44.32	2.68 8.47	8.4 -60.9	0.84 5.23	80.62 96.81	8.18 44.69	3.51 35.65	5.71 26.71	186.0 201.3
		T3E	906	2	200	09/03/90 09/23/90	09/07/91 10/17/91	14.38 20.95	2.92 3.39	- 6.5 17.2	0.97 1.24	28.97 45.66	3.12 6.86	0.89 7.01	2.46 3.75	190.6 235.5

* Constraint violated over the entire Earth launch window.
** Constraint violated over a portion of the window.

Table A-2 Dual Launch, Out-of-Orbit, MOR: Maximum ERV Weight Margin

Year	Type	L/V	Max. ERV Margin	Mission Time	Stopover Time	Earth Launch Date	Mars Arrival Date	C_{3E}	Vhp_M	DLA	ΔV_{MOI}	C_{3M}	Vhp_E	ΔV_{EOI}	ΔV_{TEI}	θ_{ME}
1981	I	IUS	**279	982	420	12/25/81 01/14/82	08/07/82 08/27/82	10.21 25.88	3.95 7.71	- 0.6 -44.2	1.48 2.38	5.85 7.19	4.48 5.39	1.34 1.72	0.71 0.84	202.5 230.8
		T3E	---	--	---	--	--	---	---	---	---	---	---	---	---	---
	II	IUS	783	1020	380	11/17/81 12/07/81	09/13/82 09/23/82	9.33 10.92	3.06 3.20	17.5 40.7	1.05 1.13	6.04 7.19	4.48 5.39	1.34 1.72	0.73 0.84	206.9 230.8
		T3E	314	1010	360	11/17/81 12/07/81	09/13/82 10/03/82	9.18 11.21	3.06 3.20	12.0 40.7	1.05 1.13	6.41 7.80	4.35 4.88	1.29 1.50	0.77 0.90	201.7 230.0
1983/ 1984	I	IUS	**348	945	380	02/19/84 03/10/84	09/13/84 10/03/84	11.65 20.25	3.75 4.43	-23.6 -58.1	1.47 1.93	10.32 12.85	4.38 5.13	1.30 1.60	1.14 1.36	202.0 230.4
		T3E	---	--	---	--	--	---	---	---	---	---	---	---	---	---
	II	IUS	496	1008	380	12/18/83 01/07/84	09/21/84 10/11/84	11.85 17.69	3.55 3.92	1.4 29.2	1.34 1.58	9.89 11.95	4.96 4.43	1.32 1.54	1.10 1.28	191.7 226.0
		T3E	123	1004	480	12/28/83 01/17/84	10/21/84	10.59 12.30	3.72 3.90	3.2 21.3	1.45 1.50	5.92 8.03	3.47 3.79	0.99 1.09	0.72 0.92	148.1 167.7
1986	I	Tug	*1332	902	380	04/20/86 05/10/86	11/10/86 11/20/86	8.37 9.89	3.15 3.25	- 4.24 -60.0	1.10 1.16	62.86 21.11	4.50 21.40	1.35 13.64	2.05 11.80	183.5 193.2
		T3E	** 14	938	480	05/10/86 05/30/86	01/19/87	8.65 11.04	3.77 3.87	-22.8 -42.1	1.48 1.55	11.53 21.87	3.44 13.86	0.98 7.23	1.25 7.76	162.3 193.0
	II	Tug	872	921	310	04/21/86 05/11/86	01/17/87 01/27/87	8.37 9.91	3.87 4.05	2.2 -37.7	1.55 1.72	19.77 22.76	3.84 4.41	1.11 1.32	1.94 2.18	198.6 223.2
		T3E	48	957	480	04/21/86 05/11/86	01/17/87 01/27/87	8.37 9.91	3.87 4.13	2.2 -37.7	1.55 1.72	10.14 22.07	3.11 12.43	0.89 6.89	1.12 6.61	150.6 187.0
1988	I	Tug	1351	921	410	06/29/88 07/19/88	01/16/89 01/26/89	11.67 13.12	2.61 2.75	8.5 23.8	0.81 0.86	33.01 36.05	3.01 3.72	0.86 1.07	2.92 3.13	196.4 222.4
		T3E	313	976	480	07/19/88 08/08/88	02/25/89	13.94 21.13	2.55 2.64	9.8 24.6	0.77 0.82	13.64 14.40	3.12 3.28	0.89 0.94	1.43 1.50	238.7 264.5
	II	Tug	890	986	330	07/19/88 08/08/88	07/25/89 08/04/89	17.92 21.16	3.50 3.70	1.5 -20.3	1.31 1.43	18.50 28.00	3.59 5.00	1.03 1.55	1.84 2.57	206.3 235.8
		T3E	---	--	---	--	--	---	---	---	---	---	---	---	---	---
1990	I	Tug	**884	960	450	08/20/90 09/09/90	03/18/91	17.82 19.56	3.54 3.58	36.2 50.6	1.34 1.36	13.96 15.13	2.88 3.47	0.83 0.99	1.46 1.56	204.9 244.5
		T3E	---	--	---	--	--	---	---	---	---	---	---	---	---	---
	II	Tug	1311	966	300	08/14/90 09/03/90	07/29/91 08/08/91	15.91 19.77	2.61 3.04	7.0 -15.8	0.81 1.03	13.96 19.22	2.88 3.47	0.83 0.89	1.46 1.90	204.9 237.2
		T3E	198	976	320	08/04/90 08/24/90	07/09/91 07/19/91	17.85 20.99	2.51 2.72	-17.0 6.6	0.75 0.97	13.96 19.22	2.88 3.47	0.83 0.89	1.46 1.90	204.9 237.2

* Constraint violated over the entire Earth launch window.
** Constraint violated over a portion of the window

Table A-3 Single Launch, Out-of-Orbit, MOR: Minimum Mission Time

Year	Type	L/V	Minimum Mission Time (Days)	ERV Margin (kg)	Stopover Time (Days)	Earth Launch Date	Mars Arrival Date	C_{3E}	Vhp_M	DLA	ΔV_{MOI}	C_{3M}	Vhp_E	ΔV_{EOI}	ΔV_{TEI}	θ_{ME}
1981	I	IUS	---	---	---	--	--	---	---	---	---	---	---	---	---	---
		T3E	---	---	---	--	--	---	---	---	---	---	---	---	---	---
	II	IUS	**900	16	190	11/27/81 12/17/81	10/03/82 10/23/82	8.94 11.77	3.13 3.53	17.3 44.9	1.09 1.33	24.53 74.65	7.48 12.60	2.06 6.26	2.93 5.38	185.1 194.5
		T3E	---	---	---	--	--	---	---	---	---	---	---	---	---	---
1983/ 1984	I	IUS	---	---	---	--	--	---	---	---	---	---	---	---	---	---
		T3E	---	---	---	--	--	---	---	---	---	---	---	---	---	---
	II	IUS	978	3	290	12/18/83 01/0[illegible]/84	10/01/84 10/11/84	10.92 14.20	3.63 4.34	11.4 22.2	1.34 1.42	13.97 14.91	4.25 4.49	1.26 1.34	1.42 1.54	208.3 217.9
		T3E	---	---	---	--	--	---	---	---	---	---	---	---	---	---
1986	I	Tug	---	---	---	--	--	---	---	---	---	---	---	---	---	---
		T3E	---	---	---	--	--	---	---	---	---	---	---	---	---	---
	II	Tug	**831	12	120	05/11/86 05/31/86	01/27/87 04/07/87	9.22 52.88	3.87 5.92	+15.6 -70.5	1.55 3.06	24.51 50.35	5.13 17.61	1.61 10.32	2.31 8.69	187.7 202.5
		T3E	---	---	---	--	--	---	---	---	---	---	---	---	---	---
1988	I	Tug	741	101	120	07/09/88 07/29/88	02/15/89	12.19 16.12	2.55 2.97	15.8 37.3	0.77 1.00	65.91	11.69	5.58	4.92	151.7
		T3E	---	---	---	--	--	---	---	---	---	---	---	---	---	---
	II	Tug	**791	1	30	08/08/88 08/28/88	08/04/89 09/23/89	17.31 76.12	3.64 6.24	- 0.3 -49.9	1.40 3.32	37.81 44.40	4.24 6.35	1.25 2.18	3.25 3.67	190.5 210.3
		T3E	---	---	---	--	--	---	---	---	---	---	---	---	---	---
1990	I	Tug	**890	17	310	08/30/90 09/19/90	03/28/91 05/17/91	18.01 45.28	2.40 4.19	31.4 74.6	0.70 1.76	39.88 51.30	3.22 5.59	0.92 1.81	3.38 4.09	195.4 239.5
		T3E	---	---	---	--	--	---	---	---	---	---	---	---	---	---
	II	Tug	786	42	30	09/03/90 09/23/90	08/18/91 09/17/91	14.65 45.37	2.83 4.74	10.8 -34.7	0.92 2.15	62.48 70.01	4.79 14.26	1.46 7.55	4.73 8.15	186.0 201.3
		T3E	---	---	---	--	--	---	---	---	---	---	---	---	---	---

* Constraint violated over the entire Earth launch window.
** Constraint violated over a portion of the window.

Table A-4 Single Launch, Out-of-Orbit, MOR: Maximum ERV Weight Margin

Year	Type	L/V	Max. ERV Margin	Mission Time	Stopover Time	Earth Launch Date	Mars Arrival Date	C_{3E}	Vhp_M	DLA	ΔV_{MOI}	C_{3M}	Vhp_E	ΔV_{EOI}	ΔV_{TEI}	θ_{ME}
1981	I	IUS	---	--	---	--	--	---	---	---	---	---	---	---	---	---
		T3E	---	--	---	--	--	---	---	---	---	---	---	---	---	---
	II	IUS	321	1020	370	11/17/81 12/07/81	09/13/82 10/03/82	9.18 11.21	3.06 3.20	16.1 40.7	1.05 1.13	6.04 8.00	4.45 5.51	1.33 1.77	0.73 0.93	206.9 235.3
		T3E	---	--	---	--	--	---	---	---	---	---	---	---	---	---
1983/ 1984	I	IUS	---	--	---	--	--	---	---	---	---	---	---	---	---	---
		T3E	---	--	---	--	--	---	---	---	---	---	---	---	---	---
	II	IUS	38	1008	380	12/18/83 01/07/84	09/21/84 10/11/84	10.92 14.96	3.56 4.34	1.4 29.4	1.34 1.58	9.89 11.95	4.43 4.96	1.32 1.54	1.10 1.28	197.7 230.4
		T3E	---	--	---	--	--	---	---	---	---	---	---	---	---	---
1986	I	Tug	---	--	---	--	--	---	---	---	---	---	---	---	---	---
		T3E	---	--	---	--	--	---	---	---	---	---	---	---	---	---
	II	Tug	555	957	480	04/21/86 05/11/86	01/17/87 01/27/87	9.22 9.91	3.87 4.13	2.2 -37.7	1.55 1.72	10.14 22.07	3.11 13.43	0.89 6.89	1.12 6.61	150.6 170.9
		T3E	---	--	---	--	--	---	---	---	---	---	---	---	---	---
1988	I	Tug	913	921	410	06/29/88 07/19/88	01/16/89 01/26/89	11.67 13.12	2.61 2.75	4.6 17.1	0.81 0.88	33.01 36.05	3.01 3.72	0.86 1.07	2.92 3.13	196.4 222.4
		T3E	---	--	---	--	--	---	---	---	---	---	---	---	---	---
	II	Tug	461	986	330	07/19/88 08/08/88	07/25/89 08/04/89	17.92 21.16	3.50 3.70	1.5 -20.3	1.31 1.43	12.64 16.23	3.23 4.05	0.92 1.18	1.34 1.65	219.7 245.6
		T3E	---	--	---	--	--	---	---	---	---	---	---	---	---	---
1990	I	Tug	**453	960	450	08/20/90 09/09/90	03/18/91	17.82 19.56	3.54 3.58	36.2 50.6	1.34 1.36	13.96 15.13	2.88 3.47	0.83 0.99	1.46 1.56	204.9 244.5
		T3E	---	--	---	--	--	---	---	---	---	---	---	---	---	---
	II	Tug	917	966	280	08/14/90 09/03/90	07/29/91 08/28/91	14.96 19.77	2.61 3.04	7.3 -15.8	0.81 1.03	16.21 26.58	2.94 3.42	0.34 0.98	1.65 2.46	211.3 249.8
		T3E	---	--	---	--	--	---	---	---	---	---	---	---	---	---

* Constraint violated over the entire Earth launch window.
** Constraint violated over a portion of the window.

Table A-5 Single Launch, Direct Entry, MOR: Minimum Mission Time

Year	Type	L/V	Minimum Mission Time (Days)	ERV Margin (kg)	Stopover Time (Days)	Earth Launch Date	Mars Arrival Date	C_{3E}	Vhp_M	DLA	ΔV_{MOI}	C_{3M}	Vhp_E	ΔV_{EOI}	ΔV_{TEI}	θ_{ME}
1981	I	IUS	**922	21	300	12/25/81 01/14/82	08/07/82 08/27/82	10.21 25.88	4.31 5.04	- 0.6 -44.2	1.48 2.38	12.32 15.64	3.92 4.89	1.14 1.51	1.32 1.60	201.0 220.1
		T3E	---	---	---	--	--	---	---	---	---	---	---	---	---	---
	II	IUS	890	18	170	11/27/81 12/17/81	10/13/82 10/23/82	8.94 10.24	3.27 3.53	17.3 39.2	5.85 6.00	33.94 53.30	7.48 10.33	2.79 4.60	2.98 4.21	181.1 190.7
		T3E	980	1	290	11/17/81 12/07/81	09/13/82 10/03/82	9.18 11.21	3.06 3.20	12.0 40.7	1.05 1.13	10.47 11.58	3.89 4.10	1.13 1.20	1.15 1.25	210.1 224.6
1983/ 1984	I	IUS	**885	25	270	02/19/84 03/10/84	09/13/84 10/03/84	11.65 20.25	3.75 4.43	-23.6 -58.1	1.47 1.93	18.62 95.33	5.11 13.47	1.60 6.93	1.85 6.39	188.6 202.9
		T3E	---	---	---	--	--	---	---	---	---	---	---	---	---	---
	II	IUS	**908	3	220	01/17/84 02/06/84	10/31/84 11/20/84	9.14 19.67	3.89 4.71	1.7 43.2	1.56 2.13	14.75 17.71	4.35 5.28	1.29 1.67	1.54 1.69	203.2 212.8
		T3E	---	---	---	--	--	---	---	---	---	---	---	---	---	---
1986	I	Tug	**462	10	30	04/10/86 04/30/86	10/11/86 10/21/86	8.21 9.42	3.47 3.84	-45.5 -58.2	1.29 1.53	37.13 66.95	9.47 14.55	4.02 7.78	3.20 4.97	105.9 129.5
		T3E	---	---	---	--	--	---	---	---	---	---	---	---	---	---
	II	Tug	801	0	30	06/20/86 07/10/86	06/26/87 07/26/87	18.05 25.32	4.79 5.02	- 8.6 -37.9	2.19 2.36	22.83 42.60	4.64 14.29	1.40 3.39	2.18 6.61	188.5 207.8
		T3E	---	---	---	--	--	---	---	---	---	---	---	---	---	---
1988	I	Tug	741	61	110	07/09/88 07/29/86	02/15/89 02/22/89	13.20 16.03	2.52 2.97	37.3 12.2	0.76 1.00	--- 61.14	--- 10.82	--- 4.95	--- 4.65	--- 151.7
		T3E	986	6	480	07/19/88	02/25/89	13.94	2.64	24.6	0.82	13.91	3.28	0.94	1.45	226.0
	II	Tug	761	25	30	07/29/88 08/18/88	07/15/89 09/13/89	17.31 25.03	3.45 3.86	4.3 -27.9	1.28 1.54	53.72 80.84	8.07 28.18	3.14 19.86	4.23 15.95	142.3 174.1
		T3E	---	---	---	--	--	---	---	---	---	---	---	---	---	---
1990	I	Tug	**810	44	200	08/20/90 09/09/90	03/18/91 04/14/91	17.82 24.07	2.96 3.58	36.2 62.8	0.99 1.36	61.96 68.03	4.37 6.11	1.30 2.06	4.70 [illegible].03	161.3 196.0
		T3E	---	---	---	--	--	---	---	---	---	---	---	---	---	---
	II	Tug	776	68	30	09/03/90 09/23/90	09/07/91 09/17/91	14.60 20.95	2.92 3.32	10.8 - 6.5	0.97 1.20	63.38 89.76	5.35 14.26	1.70 7.55	5.14 8.15	156.5 186.0
		T3E	---	---	---	--	--	---	---	---	---	---	---	---	---	---

* Constraint violated over the entire Earth launch window.
** Constraint violated over a portion of the window.

Table A-6 Single Launch, Direct Entry MOR: Maximum ERV Weight Margin

Year	Type	L/V	Max. ERV Margin	Mission Time	Stopover Time	Earth Launch Date	Mars Arrival Date	C_{3E}	Vhp_M	DLA	ΔV_{MOI}	C_{3M}	Vhp_E	ΔV_{EOI}	ΔV_{TEI}	θ_{ME}
1981	I	IUS	** 91	982	410	12/25/81 01/14/82	08/07/82 08/27/82	10.21 25.98	3.77 5.04	2.4 -44.2	1.48 2.38	5.85 8.00	4.48 5.38	1.34 1.72	0.71 0.80	202.5 230.8
		T3E	---	--	---	--	--	---	---	---	---	---	---	---	---	---
	II	IUS	507	1020	380	11/17/81 12/07/81	09/13/82 10/03/82	9.18 11.21	3.06 3.20	16.1 40.7	1.05 1.13	6.04 7.19	4.48 5.39	1.34 1.72	0.73 0.84	206.9 230.8
		T3E	** 28	1010	360	11/17/81 12/07/81	09/13/82 10/03/82	9.18 11.21	3.77 5.04	2.4 -44.2	1.48 2.38	6.41 7.80	5.35 4.88	1.29 1.50	0.77 0.90	201.7 230.8
1983/ 1984	I	IUS	**119	945	360	02/19/84 03/10/84	09/13/84 10/03/84	11.65 20.25	3.75 4.43	-23.6 -58.1	1.47 1.93	11.10 14.94	4.38 5.51	1.30 1.77	1.21 1.54	210.7 239.1
		T3E	---	--	---	--	--	---	---	---	---	---	---	---	---	---
	II	IUS	248	1008	380	12/18/83 01/07/84	09/21/84 10/11/84	10.92 14.96	3.55 3.92	1.4 29.4	1.34 1.58	10.32 12.85	4.38 5.13	1.30 1.60	1.14 1.36	202.0 230.4
		T3E	---	--	---	--	--	---	---	---	---	---	---	---	---	---
1986	I	Tug	**1086	922	380	04/20/86 05/10/86	11/10/86 11/20/86	8.37 9.89	3.15 3.25	-42.4 -60.0	1.10 1.16	19.77 22.76	3.84 4.41	1.11 1.32	1.94 2.18	198.1 223.2
		T3E	---	--	---	--	--	---	---	---	---	---	---	---	---	---
	II	Tug	682	921	330	04/21/86 05/11/86	01/17/87 01/27/87	8.37 9.91	3.87 4.13	-37.7 2.2	1.55 1.72	19.77 20.99	3.61 4.25	1.04 1.25	1.89 2.04	199.4 214.1
		T3E	---	--	---	--	--	---	---	---	---	---	---	---	---	---
1988	I	Tug	1076	921	410	06/29/88 07/19/88	01/16/89 01/26/89	11.67 13.12	2.61 2.75	8.5 23.8	0.81 0.88	33.01 36.05	3.01 3.72	0.86 1.07	0.92 1.18	196.4 222.4
		T3E	6	986	480	07/19/88	02/25/89	13.94	2.64	24.6	0.82	40.61	10.05	4.41	3.43	94.79
	II	Tug	664	986	330	07/19/88 08/08/88	07/25/89 08/04/89	17.92 21.16	3.50 3.70	1.5 -20.3	1.31 1.43	12.64 16.23	3.23 4.05	2.92 3.13	1.34 1.65	219.7 245.6
		T3E	---	--	---	--	--	---	---	---	---	---	---	---	---	---
1990	I	Tug	**654	960	450	08/20/90 09/09/90	03/18/91	17.82 19.56	3.54 3.58	50.6 36.2	1.36 1.34	13.96 15.13	2.88 3.47	0.83 0.99	1.46 1.56	204.9 224.5
		T3E	---	--	---	--	--	---	---	---	---	---	---	---	---	---
	II	Tug	1076	966	280	08/14/90 09/03/90	07/29/91 08/28/91	14.96 19.77	2.61 3.04	12.1 -15.8	0.81 1.03	16.21 26.58	2.94 3.42	0.84 0.98	1.65 2.46	211.3 249.8
		T3E	---	--	---	--	--	---	---	---	---	---	---	---	---	---

* Constraint violated over the entire Earth launch window.
** Constraint violated over a portion of the window.

The asterisks in the tables (column 4) call attention to DLA constraint violations. A single asterisk means that the constraint is violated over the entire Earth launch window. Double asterisks denote a violation over a portion of the window. The DLA is limited to the interval $-40.3 < \mathrm{DLA} < 40.3$ because of the 60° to 120° launch azimuth requirement at Cape Canaveral. Only the 1986/Type I/Tug mission is eliminated due to this constraint.

The θ_{ME}s for the minimum time missions are generally smaller than the θ_{ME}s for the corresponding maximum ERV weight margin missions. The smaller transfer angles mean shorter missions but again the price is paid in terms of trajectory mismatch at Mars launch and Earth arrival. Some θ_{ME}s for minimum time missions are larger however (e.g. 1983/4/Type II/IUS, 1986/Type I/Tug). These are the fast Type II Earth returns where the transfer trajectory dips below the Earth's orbital radius.

APPENDIX B

ERV NON-PROPULSIVE MASS

In Task 1 of Contract 9540205, Mars Surface Sample Return Tradeoff Studies, we are to examine the performance requirements for the launch years 1981, 1983/84, 1986, 1988 and 1990, to determine which mission modes are most feasible and appropriate for each opportunity. This information will provide inputs to the NASA mission planning process.

Our approach will be to examine opposition and conjunction class missions, Mars orbital rendezvous and direct return modes, and direct entry and out of orbit landing techniques to determine which choices or combinations are most compatible with the performance requirements and the available launch systems.

One assumption required in initiating a search for such compatible mission profiles is an estimate of the non-propulsive mass of the Earth Return Vehicle (ERV) including the Earth entry capsule or Earth orbit insertion capsule. The definition of an ERV configuration has not been an official part of our previous MSSR contract studies although some preliminary sizing exercises have been performed.

This technical note will summarize the various approaches which have been applied in establishing the size of the ERV and will recommend a non-propulsive weight to be used in Task 1.

A. ERV FOR BASELINE MSSR STUDY (PV/THOR DELTA)

The mission baseline in our previous study assumed an ERV with a dry mass of 105 kg, carrying 130 kg of propellant and a 28 kg EEC. This configuration was based on an early version of the Pioneer Venus spacecraft that was to have been compatible with the Thor Delta launch vehicle. To meet the 263 kg total wet mass, PV components are used for the most part but a new more efficient spaceframe structure would be required. Removing the propulsion inerts (31 kg) brings the non-propulsive (with the EEC) mass to 102 kg.

B. MODIFIED PV/ATLAS CENTAUR CONFIGURATION

Summary mass statements for two versions of modified Pioneer Venus orbiters of the later Atlas Centaur class are shown in Table B-1 below. The minimally modified system (Version A) uses the basic PV configuration from which some non-essential structural elements have been removed. The maximum modification Version B incorporates a reduced spacecraft diameter from 2.54m to 1.8m to improve structural efficiency. In both the A and B versions, the subsystems and components not required for the MSSR mission were removed. The non-propulsive masses of Versions A and B are 177 kg and 164 kg respectively including 28 kg for the EEC.

C. JPL/LRC STUDY CONFIGURATION

The study of the direct return MSSR mission performed by a JPL/LRC team in 1974 assumed an ERV mass of 169 kg broken down as shown in Table B-2. Removal of propellants and propulsion dry weight yields a non-propulsive mass of 58 kg. Adding the 28 kg for the EEC brings the comparative mass to 86 kg.

D. PIONEER 10/11

The Pioneer 10 and 11 vehicles weigh 270 kg after elimination of science (30 kg), propellants (27 kg) and propulsion system dry weight (approximately 13 kg). The non-propulsive mass becomes approximately 200 kg. Adding 28 kg for the EEC brings the total to 228 kg.

E. HELIOS

The Helios spacecraft mass is 364 kg including 55 kg of science instruments. The non-propulsive mass would be approximately 265 kg. Adding the EEC (28 kg) brings the total to 293 kg.

F. PIONEER 6/9

These smaller interplanetary vehicles weighed 62 kg including 25 kg of science. Non-propulsive mass would be about 30 kg. Attaching a 28 kg EEC would bring the total non-propulsive mass to 58 kg.

Table B-1 Pioneer Venus Based ERV Mass Breakdown

Element	Mass - kg Version A (Min Mod)	Version B (Max Mod)
Communications	10.50	10.50
Data Handling	7.44	7.44
Control	10.42	10.42
Structure	87.40	74.60
Power	25.20	25.20
Propulsion	19.60	19.60
Contingency 5%	8.04	7.44
Total ERV Dry	168.60	155.20
Earth Entry Capsule	28.00	28.00
Total ERV Dry + EEC	196.60	183.20
RCS Propellant	6.00	6.00
Liquid Propellant	50.30	46.90
Solid Rocket Motor	138.80	129.60
ERV Gross	391.70	365.70

Table B-2 JPL/LRC ERV Mass Breakdown

SYSTEM	MASS (kg)
Structure	14
Propulsion and Attitude Control	111
Telecommunications	7
Power	17
Data Handling and Command	4
Pyrotechnics	5
Cabling	3
Temperature Control	5
Mechanisms	3
TOTAL	169

G. RECOMMENDATION

The masses of all the vehicles discussed are plotted in Figure B-1. Several Mariner class vehicles are also shown. A figure of 165 kg is recommended as the reference ERV mass since it is close to the central value for the designs which have had the benefit of most detailed study and which are compatible with the selected Earth orbit capture module.

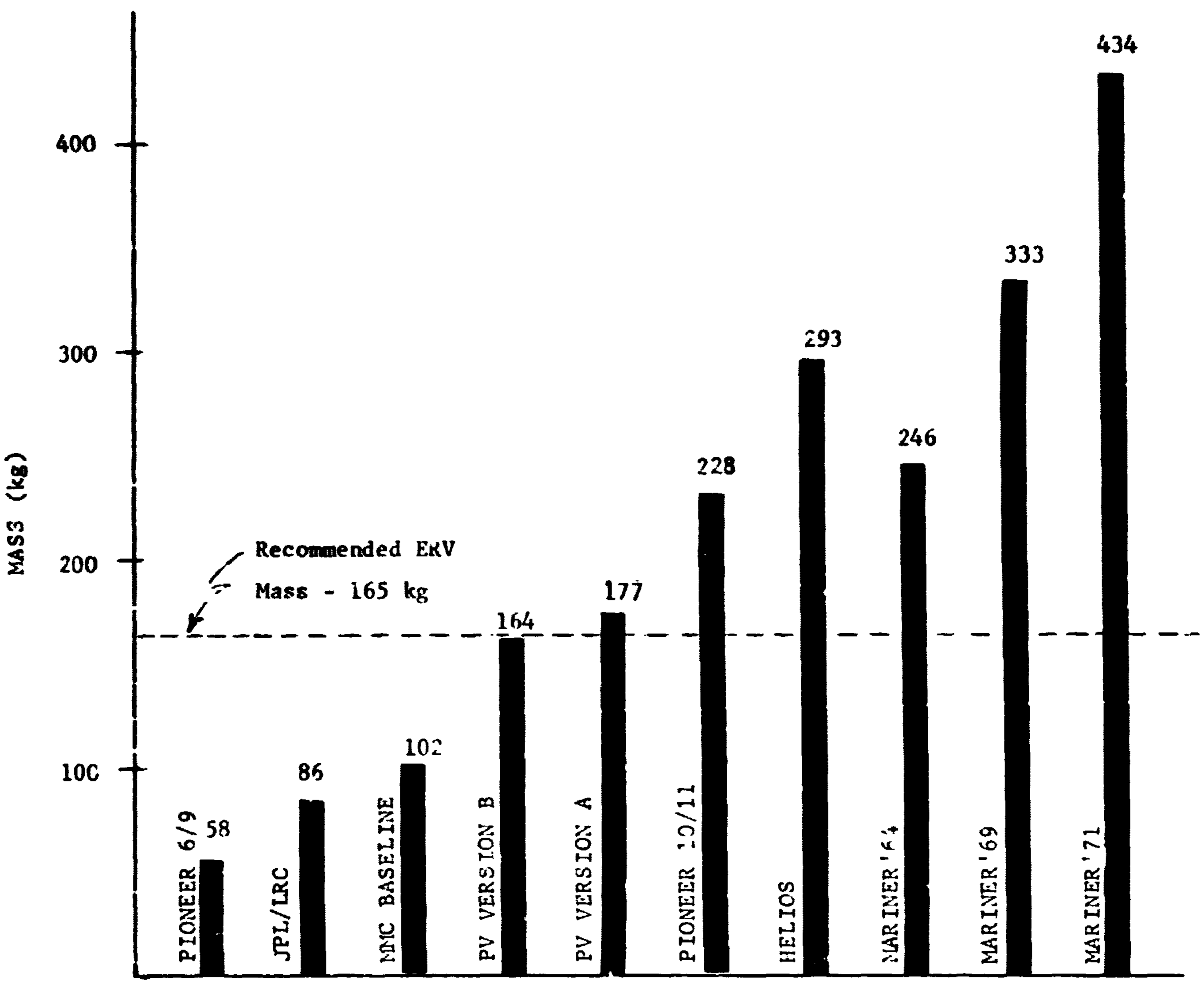

Figure B-1 Non-propulsive Mass of Interplanetary Cruise Vehicles (Includes 28 kg for Earth Entry Capsule)

APPENDIX C

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PERFORMANCE AND WEIGHT ASSESSMENTS FOR DUAL LAUNCH MSSR IN 1981 AND 1983/84

A. INTRODUCTION

Our earlier study of MSSR for JPL was directed to consider a mission characterized by relatively constrained performance--a single Titan IIIE/Centaur launch applied to the weight requirements of a multiple-module spacecraft designed to accomplish a variety of complex phases in orbit at Mars. The baseline which was developed to achieve mission feasibility required the selection of certain features which would have preferably been avoided in a less restricted weight environment, notably direct entry landing, small weight margins assigned to the MAV, and an optimistically low weight allocation to the ERV limiting its flexibility to return-to-Earth, hence limiting landing site accessibility.

When the more attractive performance offered by a dual Titan launch can be considered, options are opened which lead to a more realistic mission baseline. By dividing the lander/MAV and ERV modules between separately launched spacecraft, the weight critical nature of the single launch 1981 opportunity is eliminated, and the 1983/84 Mars opportunity can be reconsidered for MSSR. Most importantly, dual launch allows the design of out-of-orbit landing, permitting the lander to wait out any troublesome Martian weather while the landing site is certified. For 1981 launches, sufficient performance exists to baseline a Viking-type landing orbit of 24.623 hours, planet synchronous, while in 1983/84 a larger 5-day synchronous landing orbit would be required for ample weight allocation to the lander/MAV at entry. Both designs account for much larger and heavier ERV definitions, and incorporate reasonable growth margins on all vehicles.

B. DISCUSSION

<u>ERV Sizing</u> - Current designs defined in the MSSR study of this vehicle call for a 100% increase in non-propulsive weight returned to Earth over that considered in the earlier baseline. Payload for return now includes 172.9 kg for useful EF[illegible] ..t (with 20% contingency), 21.2 kg for the full survival EEC, and 6.0 kg of ACS propellant, for a total return payload of

210.1 kg. Two propulsion systems accomplish trans-Earth insertion (TEI) as defined and baselined in the MSSR study. A solid rocket motor does the initial ΔV from circular rendezvous orbit to the large ellipse (2200 by 100,000 km nominally). After its jettison, a liquid system does all remaining burns.

In developing ERV propulsion system size for 1981 and 1983/84, the following ΔV values were assumed. Starting from a 2200 km circular rendezvous orbit, the solid motor must generate 1044 m/sec to achieve the 100,000 km apoapsis altitude. The liquid system then trims periapsis to 1000 km with 22 m/sec, and finally transfers to the Earth return trajectory with 667 m/sec in 1981, 715 m/sec for 1983/84. An additional 100 m/sec is allocated to this system for other trims, plane change maneuvers, and burn loss. Total ERV ΔV requirements are then 1833 m/sec for the 1981 mission, and 1881 m/sec for 1983/84.

For the solid rocket motor, Isp is 285 sec and the mass fraction is 0.88. For the liquid system, Isp is 306 sec, with inerts = 8.9 kg + 0.17 (propellant), and 6 kg RCS propellant included in the liquid tanks. Given this background, the resulting ERV sizing is presented in Table C-1.

Table C-1 ERV Sizing for 1981 and 1983/84

	1981	1983/84
Reference C_3 (km^2/sec^2)	5.42	5.90
Total TEI ΔV (km/sec)	1.833	1.881
ERV useful, 20% margin (kg)	172.9	172.9
EEC (full survival)	31.2	31.2
Total non-propulsive	204.1	204.1
ACS propellant	6.0	6.0
Total P/L for return	210.1	210.1
Solid propulsion	165.4	168.7
Liquid propulsion	91.5	97.5
Total ERV allocation	467.0	476.3

Orbiter Sizing - Changes to the weight definition of the orbiter bus involve the deletion of much science hardware and the splitting of functional requirements between the orbiter/lander and orbiter/ERV configurations. Beginning with a nominal bus of 690.5 kg nonpropulsive mass for the orbiter/lander, mass deletions subtract 87.0 kg of science, 32.8 kg for scan platform, 31.4 kg data storage, and 46.2 kg for the cold gas RCS. An addition of 21 kg for the auxiliary RCS propulsion system, and 35 kg of margin brings the orbiter total mass, less main propulsion, to 549 kg. The implication is that this orbiter serves essentially as only a propulsive bus to carry the lander/MAV into orbit. All of the complex functions associated with site certification, rendezvous, and relay communication are assigned to the orbiter/ERV spacecraft. That bus, from a base mass of 690.5 kg, deletes 87.0 kg science and 46.2 kg cold gas system, but adds 43.7 kg for the VIS (TV), 9.8 kg for UHF relay radio, 15.3 kg docking cone and rendezvous radar. Scan platform and data storage are retained. With 74 kg allocated to the auxiliary propulsion system, which performs terminal rendezvous and RCS maneuvers, and a 35 kg margin, the total orbiter mass comes to 735 kg, less main propulsion.

To size propulsion systems for these two orbiters, the orbiter/ERV was assumed to reach the familiar 2200 km circular orbit, while the orbiter/lander was to achieve the standard Viking orbit of 1500 km periapsis and 24.623 hour period. MOI conditions were selected to reflect the worst case, highest Vhp, situation for each opportunity, assuming 20-day launch windows, where each individual launch was subjected to worst case conditions. Additional ΔV budget was defined to include 35 m/sec for midcourse correction, 60 m/sec finite burn loss, 40 m/sec ΔV_{STAT}, and 69 m/sec rendezvous and trims for the orbiter/ERV (50 m/sec trims for the orbiter/lander).

C. MISSION PERFORMANCE FOR 1981 AND 1983/84

With the ERV and orbiter sized more realistically to accomplish MSSR functions, given the premise of dual Titan IIIE/Centaur launch, both Mars opportunities were evaluated to determine weight allocations which could be assigned to the lander/MAV. The intent was hopefully to provide 100-200 kg additional weight to that configuration at entry, thereby opening the possibilities of greater MAV weight margins and/or lander mobility.

The reference entry weight for our direct entry baseline was 1205 kg. Additionally it was felt that keeping to landing out of a Viking-type orbit (1500 km periapsis, 24.623 hour period) would be a desirable feature for MSSR, since significant experience would be gained through a successful Viking '75 flight. Table C-2 presents the specific designs which were treated for 1981 and 1983/84, utilizing all available launch weight in all cases.

Table C-2 MSSR Performance for Out-of-Orbit, Dual Launch

	1981		1983/84			
S/C	ORB/ERV	ORB/LND	ORB/ERV	ORB/LND	ORB/LND	ORB/LND
Orbit	2200	1-day	2200	1-day	2-day	5-day
Ref Vhp (km/sec)	3.12	3.12	3.66	3.66	3.66	3.66
Main ΔV^* (km/sec)	2.317	1.444	2.645	1.784	1.686	1.612
Throw Wt. (kg)	4407	4407	4341	4341	4341	4341
Adapt/Mp	165	165	165	165	165	165
Bio. Cap	0	54	0	54	54	54
Cruise Wt.	4242	4188	4176	4122	4122	4122
M/C Wp	51	51	51	51	51	51
Pre-MOI Wt.	4191	4137	4125	4071	4071	4071
Bus + Marg.	661	528	661	528	528	528
Aux. Prop.	74	21	74	21	21	21
ERV + Marg.	467	0	476	0	0	0
Main Prop. Wp*	2330	1643	2492	1892	1816	1757
Net P/I	361	274	382	306	296	289
Remaining Orb. Wt.	298	1671	40	1324	1410	1476
Adapt + Bio Base		87		87	87	87
Sep. Lander		1584		1237	1323	1389
Deorb. Prop.		80		80	80	80
Entry Wt.		1504		1157	1243	1309

* Excludes M/C, ΔV = 35 m/sec.

The obvious result of this analysis is that the 1981 opportunity, with dual launch, offers an exceptional performance picture for a Viking-derived MSSR, with ample weight margins all around. Even after allowance is made for a much heavier ERV and orbiter bus, and propulsion system stretch, an orbited weight margin of 298 kg exists for the rendezvous spacecraft. The orbiter/lander spacecraft, from a 1-day orbit, can provide an entry weight of 1504 kg (3316 lb), or 300 kg excess over our direct entry baseline.

For 1983/84, the Viking-derived MSSR becomes feasible with dual launch, although the ample margins of 1981 do not exist. The rendezvous spacecraft has an orbited weight margin of 40 kg, which is still reasonable. But the orbiter/lander provides only 1157 kg entry weight from a 1-day orbit, less than the direct entry baseline weight of 1205 kg. From a 2-day synchronous orbit, entry weight can be increased to 1243 kg. To reach a 100 kg excess over our previous design, landing must be out of a 5-day orbit. This last case has been tentatively accepted as our reference 1983/84 design, upon which landed weight studies have been based. The 5-day orbit provides an entry weight of 1309 kg (2886 lbs), and still appears to allow entry characteristics not too dissimilar from the nominal Viking entry.

D. SUMMARY

Opportunities in 1981 and 1983/84 for MSSR have been re-evaluated given the acceptance of a dual launch and out-of-orbit landing. Assigning the rendezvous and landing functions to two separately launched spacecraft produces a very attractive performance scenario for 1981, with ample weight margins for all systems and landing from a 1-day Viking-type orbit. For 1983/84, much smaller performance margins exist; yet by specifying landing from a 5-day orbit, an entry weight of 1309 kg can be provided, exceeding the 1981 direct entry baseline by over 100 kg.

This study on the 1981 and 1983/84 missions was conducted before the contract start and was company-funded. The 1990 and 1986 missions have larger performance margins than the 1983/84 launch opportunities, but have smaller performance margins than in 1981. Figure II-16 shows the maximum ERV weight for Earth return vehicles for the dual launch, out-of-orbit, and Mars orbital rendezvous, and indicates the relative performance needed for each mission.

APPENDIX D

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ENTRY AND LANDING PARAMETRICS FOR THE 5-DAY ORBIT

A. INTRODUCTION

With the intent of providing increased entry weight for the lander/MAV in 1983/84, a 5-day (123-115 hour) planet synchronous landing orbit has been tentatively accepted as our reference for the opportunity. Adequate landed weight can not be achieved by using a one-day orbit similar to Viking '75, so a 5-day orbit was used to get enough landed weight. This orbit has been examined to determine its compatibility with the considered Viking-class entry systems and constraints. Deflection ΔV requirements are defined as a function of entry flight path angle (γ_e) and coast time. Entry parametrics are then developed for the characteristic entry velocities, considering variations in entry weight (W_e) and γ_e. A corridor width of 4^o is assumed, with $L/D = .2 \pm .02$. Pressure regulated terminal descent propulsion is chosen for the mission, with thrust equal to 640 lb, no blowdown, and nominal Viking parachute dimensions are accepted. Both the aeroshell and ablator are allowed to grow heavier in response to variations in the maximum dynamic pressure (q max.) for the various entry conditions. Mean Mars atmosphere is used throughout.

The analysis defines the relationship between entry weight, terminal descent propellant, and dry landed weight for the 5-day orbit, and relates these results to the considered mission designs for 1981 and 1983/84 (see Appendix C).

B. DISCUSSION

Deflection Conditions for the 5-Day Orbit - An early concern with a landing orbit of this size was the feasibility of achieving coast times (deflection to entry) of 4 to 6 hours with reasonable ΔV requirements. This coast time constraint derives from lander battery limits. Using the program DEORBIT, the deflected trajectory was targeted to various entry conditions, optimizing the burn to minimize ΔV. Constraining coast time to 4, 5, and 6 hours resulted in the deflection ΔV characteristics shown in Figure D-1, varying with γ_e.

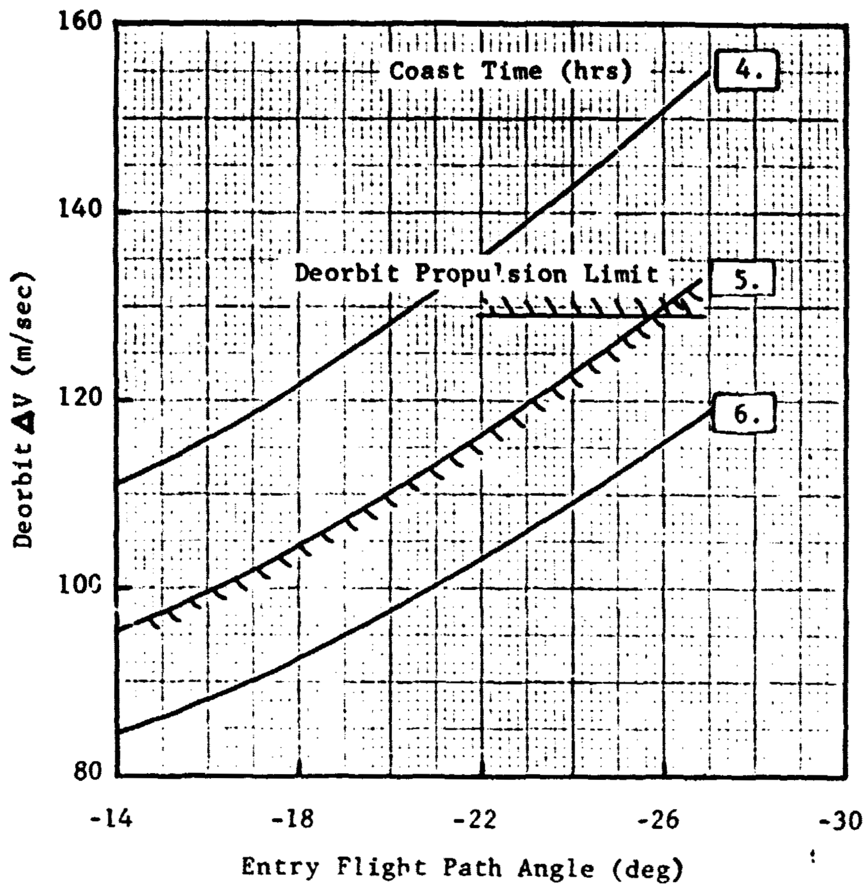

Figure D-1 Deorbit ΔV for the 5-Day Orbit

The limits noted in the figure represent 1) a maximum available ΔV of 129 m/sec for deflection given nominal lander deorbit performance (78.9 kg usable propellant and 225.4 sec Isp) and 2) a maximum coast time of 5 hours from battery power limits. These bounds establish the steepest entry angle, -25.5°, that is achievable by the lander systems without a deorbit (RCS) propulsion stretch or relaxation of the coast time constraint. Entry velocity was found to be insensitive to entry angle in the range -14° to -26°, and to coast time in the range of 4 to 6 hours. The reference entry velocity is 4.770 km/sec (15650 fps), much nearer the VO'75 value of 15010 fps than the direct entry velocities of 19-20,000 fps. Other than the obvious effect on deflection ΔV, varying flight path angle produces significant changes in the location of entry with respect to periapsis, and will therefore have important implications for landing site accessibility.

Landed Weight Analysis - The technique used to generate this analysis corresponds to that used by D. Howard in his previous studies for MSSR and Viking '79 (see Ref. 3). The study requires two trajectory simulation programs, UD288 and TEPOT, with supporting graphical and analytical work to patch together the results of each, converting the final results to data relevant for our mission design.

A matrix of parameters was defined for the 5-day landing orbit that included variation in entry weight from 2400 to 3400 lb (1091 to 1541 kg) and in flight path angle from near skipout at -14^{o} to -22^{o}. To control the study, assumptions were accepted for L/D (.2 $\pm$ .02), corridor width (4^{o} $\Delta\gamma_e$), terminal engine thrust (640 lb - 2816 N - with no blowdown), parachute size and weight (16 m and 112 kg), and entry velocity (constant at 7114 mps). The desired result was a definition of landed weight for varying W_e and γ_e.

Entry trajectories for the various conditions were propagated from the entry interface altitude of 800,000 ft (242.8 km) to the mean Mars surface, considering a mean Mars atmospheric model (Ref. 6). For each entry condition (W_e, γ_e) a trajectory must be generated for three L/D values representing the expected range--here, for L/D equal to .18, .20, and .22. From that trajectory set the lowest altitude where chute deployment conditions are satisfied (nominally $q \leq 9.5$ psf and $M \leq 2.2$) is selected. The state associated with that minimum altitude is then input to TEPOT, which simulates parachute and terminal engines phases of the landing. After mechanical iteration on terminal engine phase initiation, the point where velocity is reduced to 8 fps (2.4 m/sec) is graphically determined for a specific terrain height (zero for our mission). This establishes the vernier propellant requirement for each (W_e, γ_e) and allows definition of landed weight.

Rather than treat individual γ_e cases, each 4^{o} corridor must be considered as a set, or range, of γ_e. Here the corridors are denoted by their steepest γ_e, so that the "-20^{o} corridor" extends from -16^{o} to -20^{o}. Within each 4^{o} corridor the severest case must be accounted for. Maximum q is taken from the steepest γ_e trajectory, with the smallest L/D, and sizes the aeroshell. Minimum deployment altitude is taken from whichever γ_e (and L/D) yields the lowest value, but state conditions for TEPOT are

then selected from the shallow γ_e trajectory at that minimum altitude, and always from the L/D = .2 case.

Illustrated in Figure D-2 are the resulting chute deployment altitudes for the various sets of (W_e, γ_e). The circled points indicate minimum altitudes considering all three L/Ds, and represent specific trajectories and the standard deployment conditions ($q \leq 9.5$, $M \sim 2.2$). It was found that for the heavier entry weights, low γ_e trajectories (-14° and -16°) represented a critical area not satisfying the 9.5 psf condition at positive altitudes, so could therefore not be considered. However, the consensus of opinion concerning real chute capabilities, based on test results, indicates that a 10.7 psf q limit is more realistic, which if is indeed the case would reopen a region of (W_e, γ_e) space at the shallow γ_e end. For this analysis, then, the higher limit is used whenever the standard limit fails to produce a possible trajectory. These points are noted by asterisks in Figure D-2.

The deployment altitude curves, with 4° corridors fitted into their contours, yield minimum altitude conditions which, with the corresponding q max, fold into the landing simulation done by the program TEPOT for various entry weights. Table D-1 summarizes the results of TEPOT, which generates terminal engine propellant requirements for various altitudes. Certain combinations of (W_e, γ_e) produce no landing solution at a positive altitude, and are so marked. These cases represent real situations arising from the inability of chute and/or vernier engines to slow the lander in the available time before zero altitude is reached. This condition corresponds to the "Lander Systems Limits" discussed in our direct entry study.

The results give some indication of maximum landed weight for this orbit. Up to and including a 3200 lb (1455 kg) entry weight, a 4° corridor exists which produces a valid solution, hence a usable landing trajectory. As entry weight increases, however, providing solutions does become more difficult. At W_e = 3000 lb (1364 kg), the q limit is raised to 10.7 psf for the -18° and -20° corridors. At W_e = 3200 lb, even raising the q limit is not sufficient to avoid the loss of those corridors. Only the -22° corridor is available for landing. For W_e = 3400 lb (1545 kg) no corridors were found to provide landing solutions, therefore setting

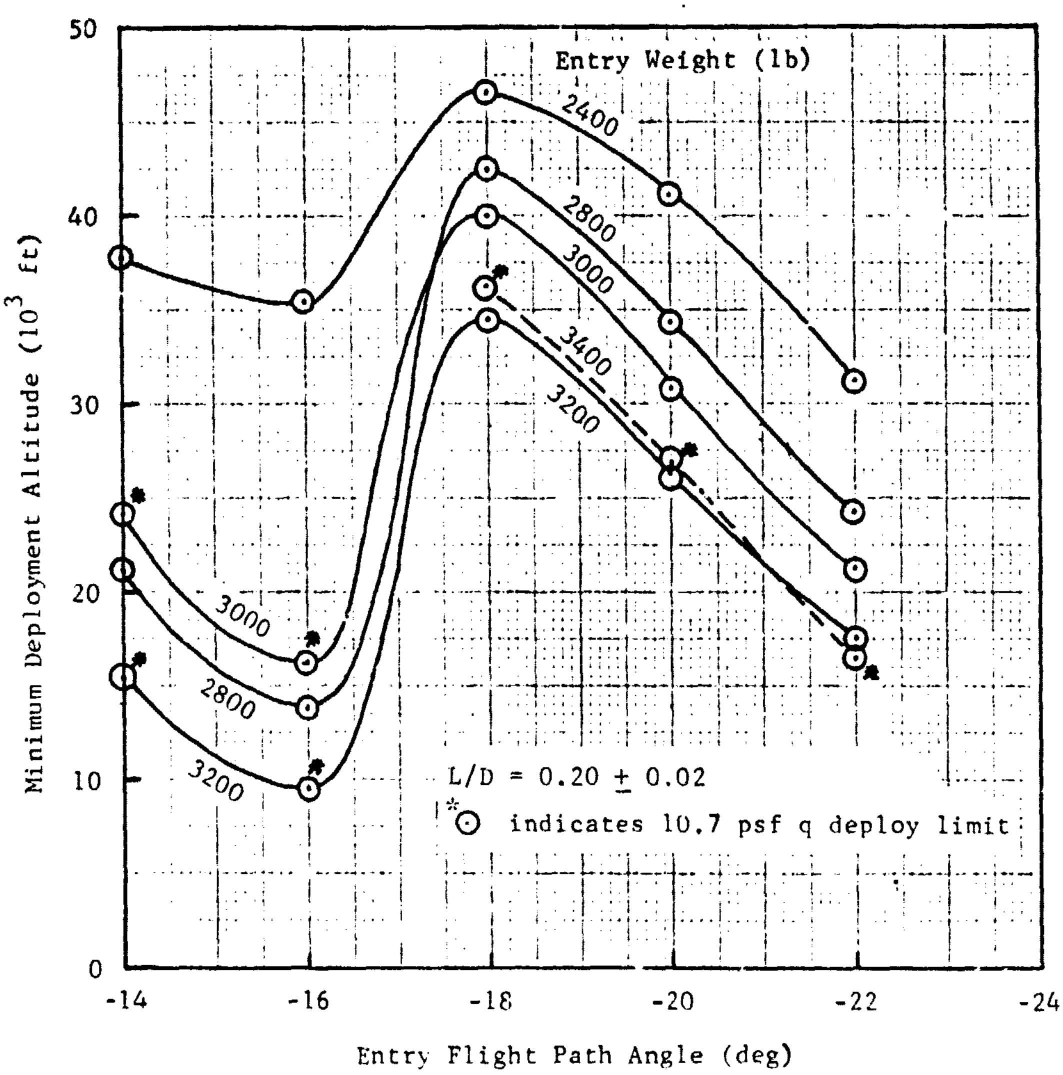

Figure D-2 Chute Deployment Altitudes for the 5-Day Orbit

Table D-1 Performance Summary for 5-day Landing Orbit, With V_e = 15650 fps and $\Delta\gamma_e$ = 4^o, H_T = 0

We (lb)	Steep γ_e (deg)	q max (psf)	Ablator (lb)	Aero-Shell (lb)	Wt. on Chute (lb)	Wt. on Vernier (lb)	Wt. Pro-pellant (lb)	Wt. Dry Landed** (lb)
2400	-18	150	40	381	2019	1773	151	1622
	-20	188	64	425	1975	1729	146	1583
	-22	221	77	463	1937	1691	140	1551
2800	-18	171	50	398	2402	2156	233	1923
	-20	214	64	444	2356	2110	224	1886
	-22	252	77	485	2315	2069	202	1867
3000	-18*	181	50	406	2594	2348	274	2074
	-20*	227	64	454	2546	2300	264	2036
	-22	267	77	496	2504	2258	243	2015
3200	-18*	191	50	414	2786	2540	(No Pos. H_T soln)	
	-20*	240	64	463	2737	2491	(No Pos. H_T soln)	
	-22	283	77	506	2694	2448	305	2143
3400	-22*	298	77	516	2884	2533	(No Pos. H_T soln)	

* q deploy at 10.7 psf required

** does not account for Lander propellant tank stretch on strut beefup.

an upper limit to landed weight potential. Further fine tuning would be required if this bound is to be better identified.

Propellant requirements are plotted in Figure D-3 as a function of entry weight and corridor. Solutions requiring a q deployment of 10.7 psf are denoted by asterisks. Figure D-4 presents the final product--the variation in landed weight as a function of entry weight and corridor, again noting the cases requiring a higher deployment q. Here landed weight does not yet adjust for propellant tank stretch or landing strut beefup. Still, comparisons with the old '81 direct entry baseline can be made by applying the above results to the dual launch weight definitions for 1981 and 1983/84.

Our reference entry weight for 1981 direct was 2657 lbs (1205 kg) with a landed weight of 1711 lb (776 kg). For 1981 dual launch a 1-day landing orbit has been selected, so the results reported here can be applied only approximately. (Using 5-day landing parametrics for a 1-day orbit should be conservative.) This design yields an ample entry weight of 3316 lb (1504 kg) which by extrapolating Figure D-4 indicates a possible landed weight of 2200 lb (998 kg), probably near the systems limit. For 1983/84 with dual launch the 5-day landing orbit provides 2886 lb (1309 kg) of entry weight and between 1930 and 1990 lb (875 and 902 kg) landed, varying with corridor selection. Skipout for these cases occurs just "above" -14^{o} (-13.7^{o} more exactly), so the more conservative corridor of -20^{o} (steep end) should be used. That corridor gives 1950 lb (885 kg) for the nominal landed weight. If the shorter period landing orbits of TN 47 are considered, decreased entry weight translates into decreased landed weight. The 2-day orbit provides 2740 lb (1243 kg) entry weight, therefore, 1840 lb (835 kg) landed. The 1-day orbit provides 2551 lb (1157 kg) entry. 1700 lb (771 kg) landed.

This analysis allows us to choose the type of landing orbit best suited to the needs of lander/MAV weight increases for 1983/84. The 5-day orbit produces a 109 kg increase over the '81 direct entry landed configuration. Designing the 2-day orbit would yield a 59 kg increase, and the 1-day orbit would yield a 5 kg decrease. For 1981, the dual launch mission produces for a 1-day landing orbit a relative landed weight increase of 222 kg.

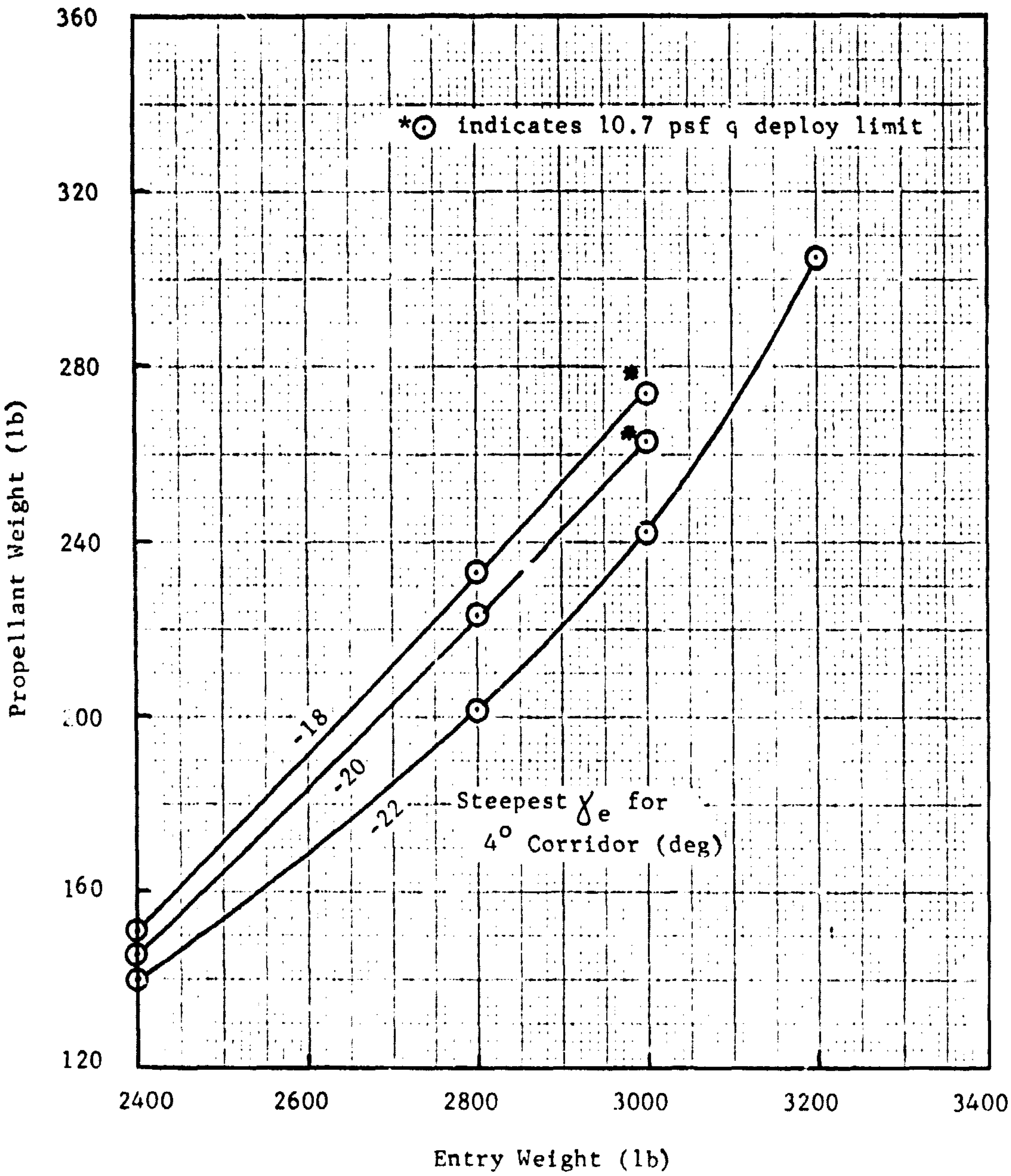

Figure D-3 Lander Descent Engine Propellant Requirements

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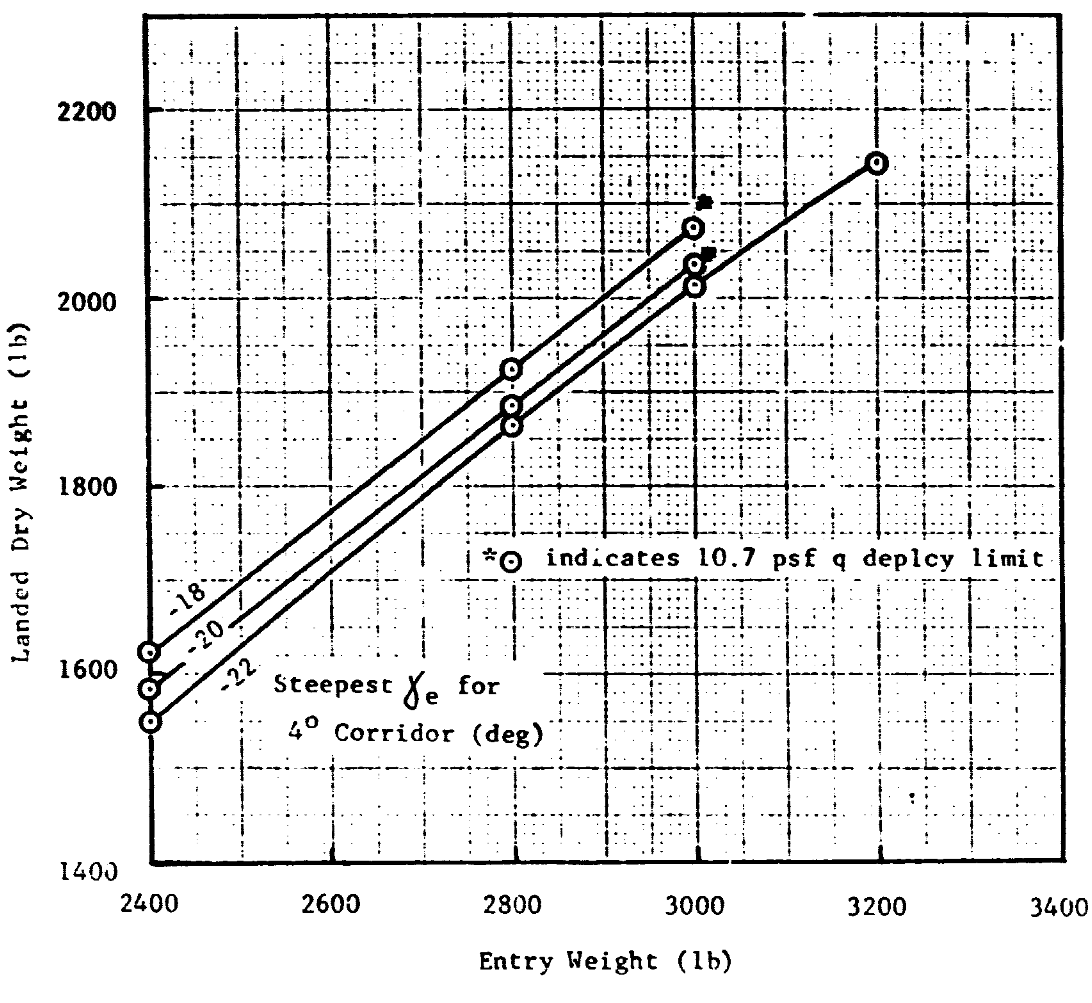

Figure D-4 Landed Weight for the 5-Day Orbit

C. CONCLUSIONS

Landed weight studies have been completed for the trajectory characteristics of a 5-day landing orbit. With nominal deorbit performance, a 5-hour coast time can be achieved for entry flight path angles shallower than -25.5°, while skipout occurs near -14°. Corridor selection for a 4° width indicates that entry weight is limited to an upper bound between 3200 lb and 3400 lb (1455 kg and 1545 kg). The parametrics can be related to the proposed mission designs (landing orbit designs) as follows. For 1983/84 landed weight with respect to the direct entry baseline of 775 kg is +109 kg with the 5-day orbit, +59 kg with the 2-day orbit, and -5 kg with the 1-day Viking orbit. For 1981 the 1-day orbit provides +222 kg. Some minor adjustment of these figures must be made to account for lander propellant tank stretch and strut beef-up.

APPENDIX E

TN 52

HIGH PERFORMANCE ENTRY TO LANDING FOR THE 1983/84 MSSR

A. INTRODUCTION

Landed weight potential has earlier been defined parametrically for the 5-day landing orbit tentatively referenced for the 1983/84 MSSR. In TN 48 (see Appendix D) landed weight was developed as a function of entry weight and entry angle for a set of basically nominal lander characteristics--4° $\Delta\gamma_E$ entry corridor, L/D = .20 $\pm$ .02, and terminal engine thrust of 640 lb (2816 N), pressure regulated. This technical note defines landed weight for two specific entry weights provided by the 1-day and 5-day orbits in 1983/84 (2551 lb and 2886 lb--1160 kg and 1312 kg--respectively), considering a higher performance entry-to-landing with a 2° $\Delta\gamma_E$ entry corridor, L/D = .25 $\pm$ .02, and thrust = 960 lb (4224 N). Landed weight variations with flight path angle are developed.

B. DISCUSSION

From the reference weights defined in TN 47 (see Appendix C), a 5-day landing orbit provides 2886 lb (1309 kg) entry weight, and the 1-day orbit provides 2551 lb (1157 kg). With nominal lander characteristics, landed weight for these conditions was found to be 1950 lb (885 kg) and 1700 lb (771 kg) respectively, considering -16° as the shallowest design γ_E for a 4° corridor. For comparison, the '81 direct entry baseline landed weight was 1711 lb (776 kg). In the present study, entry-to-landing has been simulated for the enhanced performance available with a 2° entry corridor, high L/D (.25 $\pm$ .02), and 50% increase in engine thrust (960 lb). The analysis elaborates and expands the earlier parametrics, and while some modification of the entry digital computer program UD288 tables was required, entry trajectory design techniques will not be addressed.

Entry flight path angle was selected as the primary control variable. For both landing orbits, γ_E was varied from -14 to -24°. Skipout occurs just under -14°, so a shallow γ_E of -16° is chosen as a conservative bound on entry corridor. Corridor width for these cases is 2°. Parachute deployment conditions are assumed satisfied during the entry trajectory when

$M \leq 2.2$ and dynamic pressure (q) $\leq$ 10.7 psf, as has been suggested in previous testing. A nominal 53 foot (16.2 m) diameter chute is chosen.

Minimum deployment altitude for both landing orbits is presented in Figure E-1, showing the variation with γ_E and considering the L/D range of .25 $\pm$ .02. The 5-day orbit, with higher entry velocity (15650 fps--4.8 km/sec) and entry weight (2886 lb--1312 kg) produces lower deployment altitudes than the 1-day orbit (1510 fps--4575 m/sec and 2551 lb--1160 kg). Lower deployment means the chute has less time to decelerate the system, and leads to high ΔV and propellant requirements for the terminal descent engines. Maximum dynamic pressure determines aeroshell size and so influences landed weight. Max q for both orbits is shown by Figure E-2. These factors show up first in the variation of lander propellant load, illustrated in Figure E-3 varying with entry corridor. Here the points represent the steep γ_E end of each corridor. Propellant weight is about 25% greater in the 5-day orbit.

Finally, these conditions translate into dry landed weight for both orbits, presented in Figure E-4. It is immediately apparent that the shallower corridors provide the maximum landed weight. The upper two curves for the 5-day orbit compare the different entry-to-landing performance characteristics, with the upper curve showing the enhanced landed weight capability with a narrow corridor, high L/D, and high thrust. The nominal 4° parametrics are extracted from Appendix D. Ignoring corridor width, the effect of L/D and thrust can be approximately isolated by comparing landed weights at the same steep γ_E. For instance, the 2° corridor at the steep γ_E of -20° (from -18° to -20°) and the 4° corridor at a steep γ_E of -20° (-16° to -20°) represent nearly the same max q condition. At this point the higher performance adds 75 lb (34 kg) to landed weight, increasing it from 1950 lb (885 kg) to 2025 lb (919 kg). The effect of the narrow corridor can be seen by moving the γ_E range of the 2° case to the shallow limit of -16°, (-16° to -18°) thereby truncating the steeper angles which must be included in the 4° corridor. This alone adds 40 lb (18 kg) to landed weight. Considering all the enhancements, landed weight for the 5-day orbit can be increased to 2065 lb (937 kg) from 1950 lb (885 kg), or about 6%. Compared with the direct entry baseline for 1981, this 937 kg represents a 20% increase over the early

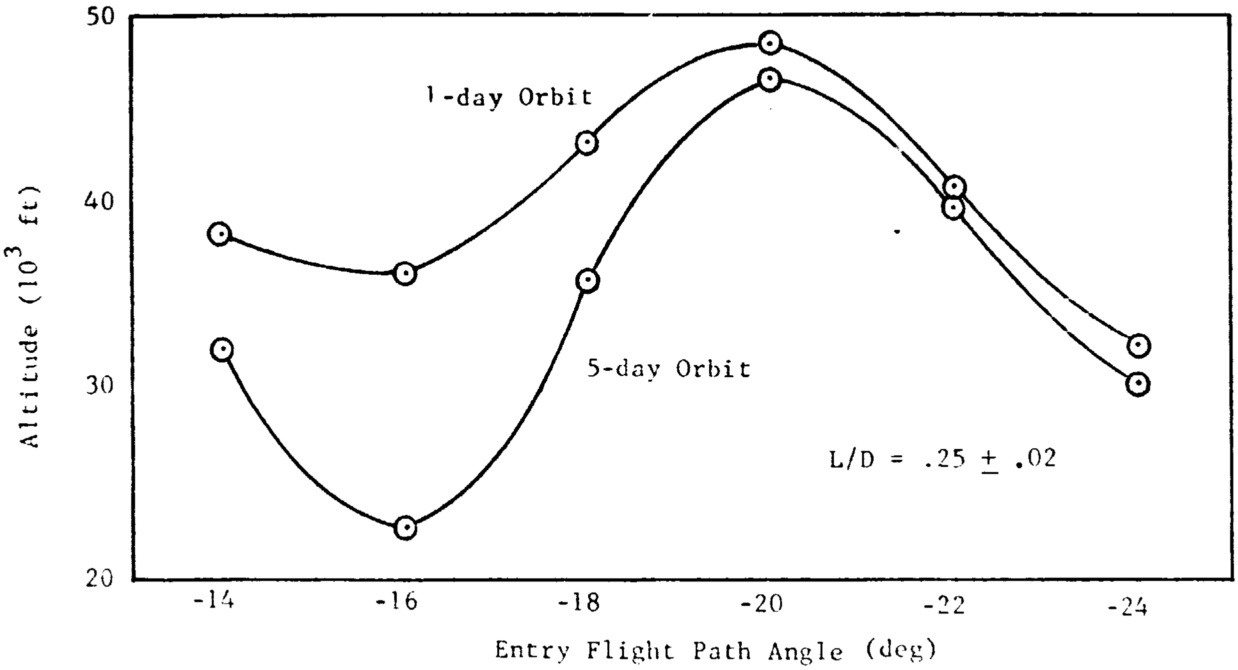

Figure E-1 Minimum Deployment Altitude

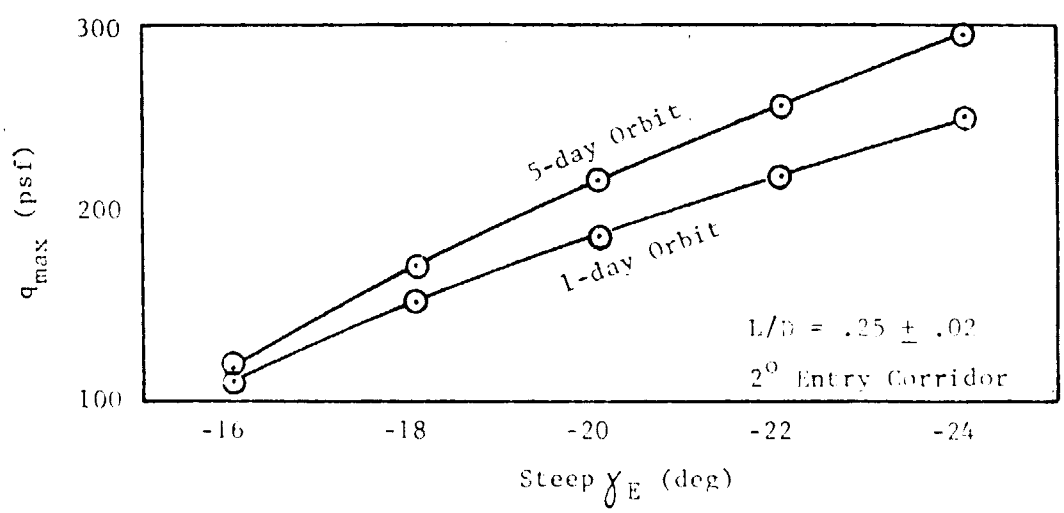

Figure E-2 Maximum Dynamic Pressure

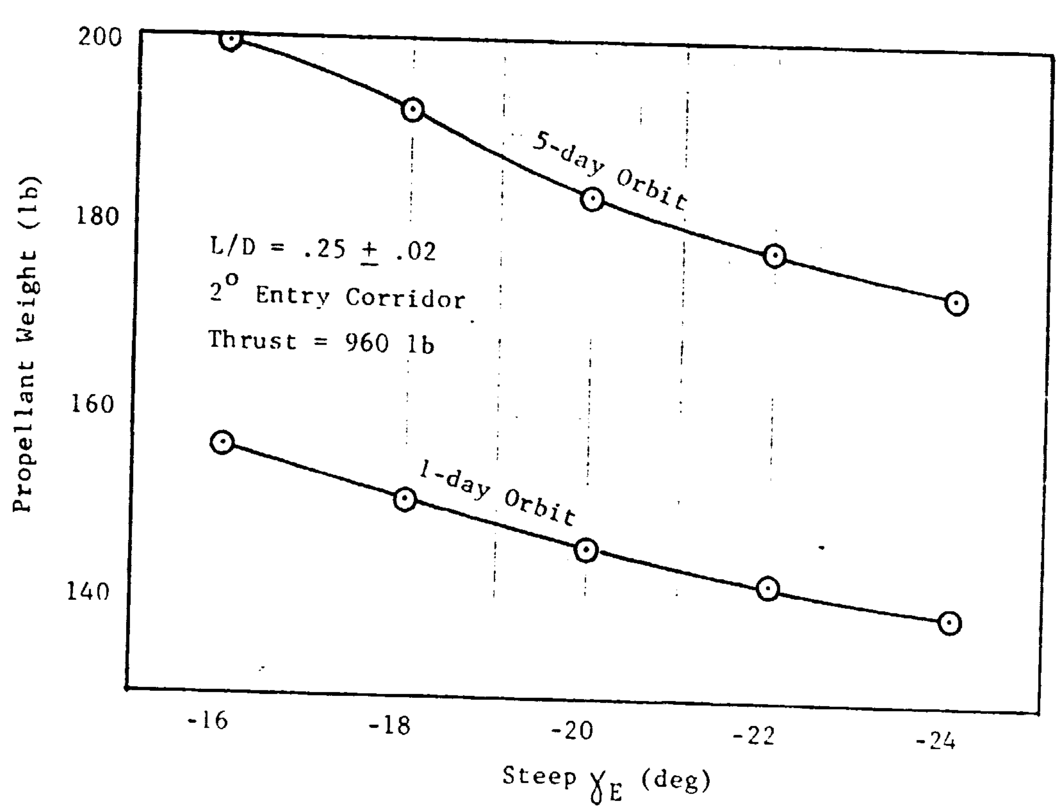

Figure E-3 Propellant Requirements

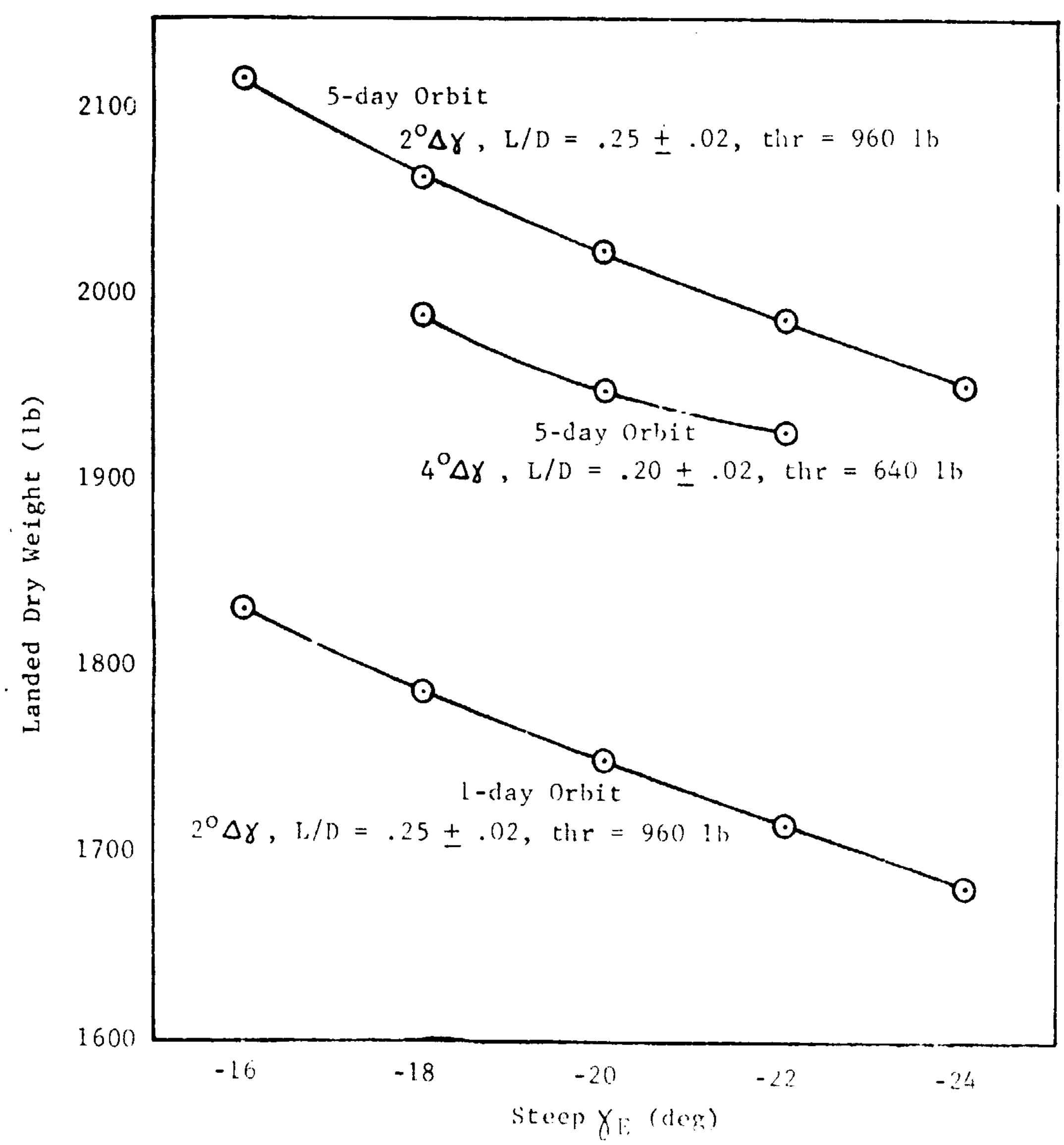

Figure E-4 Landed Dry Weight

value of 776 kg, which will be very useful in application to a larger MAV. The primary gain is, however, derived from the dual launch concept for the mission and the larger landing orbit.

Landed weight for the 1-day Viking landing orbit is shown by the lower curve of Figure E-4, and is considerably less than the 5-day design. The decreased entry weight of 2551 lb (1157 kg) leads to a large decrease in landed weight, to 1787 lb (811 kg), illustrating the importance of providing as much entry weight as possible.

C. CONCLUSIONS

For the current 1983/84 MSSR baseline landing from a 5-day orbit, greatly increasing entry-to-landing performance with a narrow corridor, high L/D, and high thrust, produces only a moderate increase in landed weight, from 885 kg to 937 kg. The primary gains for our design derive from providing increased entry weight by accepting dual launch and selecting the 5-day landing orbit. For the cases studied here, landed weight is maximized at the shallowest acceptable entry corridor, from -16° to -18° with a 2° entry angle spread.