JACK MULQUEEN

NASA

National Aeronautics and
Space Administration

December 1988

George C. Marshall Space Flight Center
Marshall Space Flight Center, Alabama 35812

Mars Exploration

Mission and System Analysis of the Split Sprint Mission

This is an historical document from NASA's study archives. It does not reflect NASA's current plans and should be viewed as simply one study exploring one potential approach to a human mission to Mars.

By: Program Development

MSFC - Form 454-1 (Rev. October 1976)

MARS EXPLORATION
MISSION AND SYSTEM ANALYSIS OF THE SPLIT SPRINT MISSION
TABLE OF CONTENTS

MARS EXPLORATION
MISSION AND SYSTEM ANALYSIS OF THE SPLIT SPRINT MISSION
TABLE OF CONTENTS (CONTINUED)

SECTION PAGE

MARS EXPLORATION
MISSION AND SYSTEM ANALYSIS OF THE SPLIT SPRINT MISSION
TABLE OF CONTENTS (CONTINUED)

SECTION | PAGE

MARS EXPLORATION
MISSION AND SYSTEM ANALYSIS OF THE SPLIT SPRINT MISSION
TABLE OF CONTENTS (CONTINUED)

MARS EXPLORATION
MISSION AND SYSTEM ANALYSIS OF THE SPLIT SPRINT MISSION
KEY CONTRIBUTORS TO THIS REPORT

PD32/A. YOUNG	LEAD ENGINEER AND MISSION ANALYSIS
PT31/F. HUFFAKER	EXPLORATION SCENARIO OPTIONS
PD12/J. FIKES	NAVIGATION AND CONTROL
PD13/R. CHAMPION	PROPULSION
PD14/R. GIUDICI	POWER AND ELECTRICAL SYSTEMS
PD14/R. WHITE	COMMUNICATION
PD22/N. BROWN	LONG TERM CRYOGENIC STORAGE
PD22/S. TUCKER	LONG TERM CRYOGENIC STORAGE
PD23/G. HAJOS	CONFIGURATION DESIGN
PD24/M. HERRMANN	SCALING EQUATIONS AND SYSTEM WEIGHT
PD24/S. HALL	SOLAR FLARES PREDICTIONS
PD31/T. FRENCH	STUDY CONSULTANT
PD31/L. RABY	DOCUMENTATION/TYPING
PD32/T. DICKERSON	STUDY TASKS OUTLINE AND DESCRIPTION
PD32/J. RUSSELL	MISSION ANALYSIS
PD32/J. MCCARTER	ORBITAL ANALYSIS
PD32/J. MULQUEEN	VEHICLE PERFORMANCE AND INTEGRATION/DOCUMENTATION COORDINATOR
PD32/M. SARGENT*	EARTH RETURN CAPSULE SYSTEM/INTEGRATION ANALYSIS
PD32/H. HARMON	ORBITAL LIFETIME
PD32/J. SKINNER	ORBITAL ANALYSIS
PD32/B. WALDEN	DOCUMENTATION/TYPING
PD33/J. LOWERY	AERODYNAMICS
PD33/J. HAYS	MARS ASCENT STAGE PERFORMANCE
PD34/M. VARNER	OPERATIONAL ANALYSIS
PD34/V. DAURO	AEROCAPTURE DYNAMICS

* UNIVERSITY SPACE RESEARCH ASSOCIATION VISITING SUMMER STUDENT; NOW EMPLOYED AT JPL

INTRODUCTION

The information presented in this report provides detailed mission and system analysis of the Split Sprint Mission to Mars. These data were generated from September 1987 through July 1988 and involved numerous individuals within Program Development. This mission derives its name from the fact that the total mission payload is split between two vehicles, a cargo vehicle, and a piloted vehicle. The cargo vehicle carries the Mars lander and propellant for the piloted vehicle return to Earth. Then approximately 1 1/2 years later, the piloted vehicle carries the crew on a higher energy "sprint" trajectory to Mars. The advantages of the Split Sprint Mission is that the payload for the piloted vehicle is minimized and the crew would be in space for only 14 months.

Mission and system data are presented for a Sprint Mission in the year 2004. Many important aspects of the mission were studied in detail. These include orbital lighting and lifetime analysis, Mars parking orbit selection, Earth departure window analysis, lander descent and ascent performance, vehicle aerobraking requirements, mission orbital maneuvers, and many other analyses listed in the Table of Contents. Emphasis was placed on vehicle performance analysis which ranged from recovery of the Earth departure stages to aerocapture of the Earth Return Capsule. There is also a section describing the computer programs that were used for this study. These programs provide many useful tools for future studies of manned planetary missions.

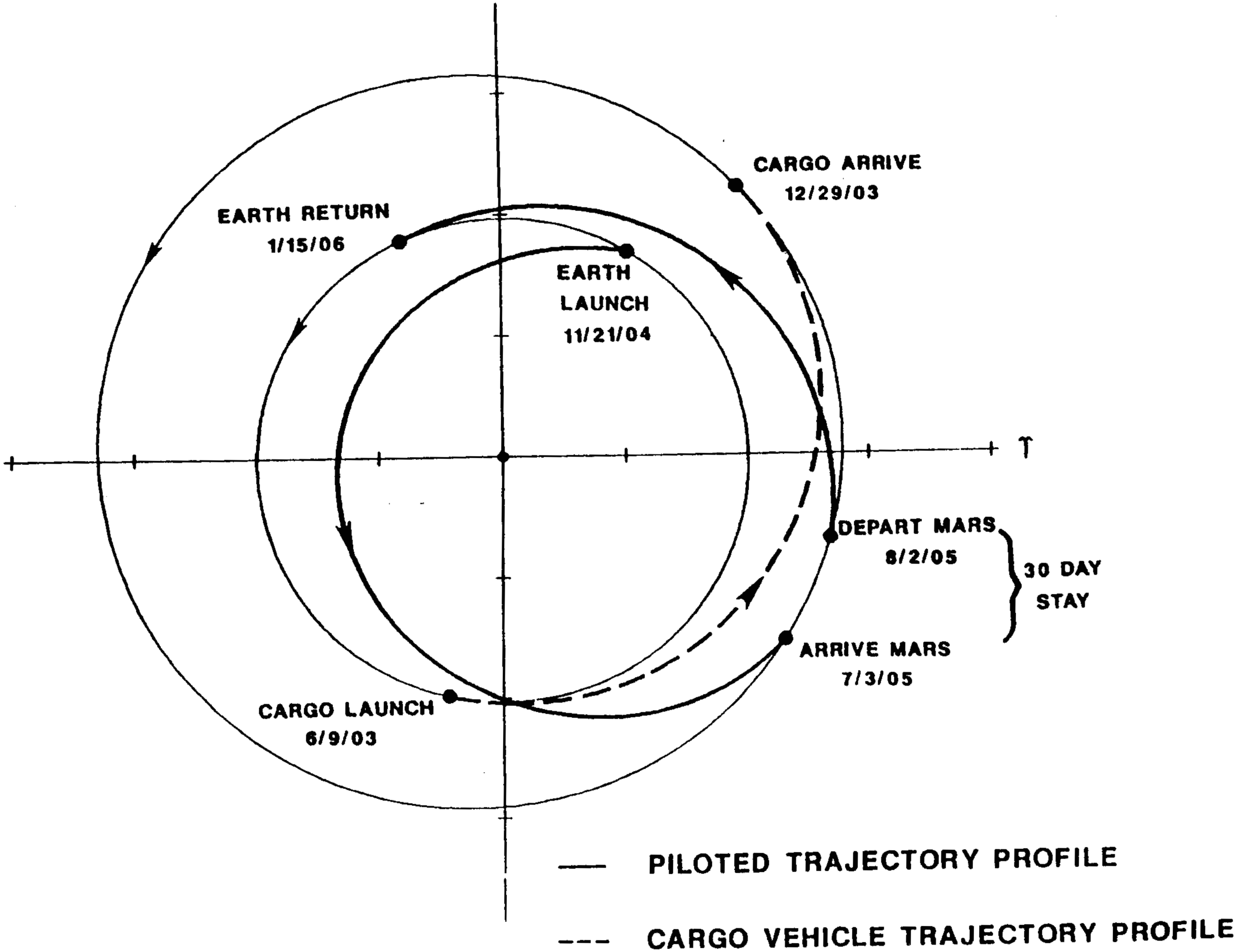
This introductory section will give an overall view of the Split Sprint Mission and describe the orbital operations and vehicles that were studied.

SPLIT SPRINT MISSION TRAJECTORIES

This chart shows the trajectory profiles for the cargo and piloted vehicles for the 2004 Split Sprint Mission. The cargo vehicle departs Earth in June 2003 on a low energy trajectory to Mars. When it arrives at Mars in December of 2003, it uses aerobraking to enter a 300 km X 4381 km altitude parking orbit. It remains in this orbit until the piloted vehicle arrives. The piloted vehicle departs Earth in November 2004 on a sprint trajectory to Mars. It arrives at Mars in July 2005 and enters a 1000 km altitude parking orbit. The cargo vehicle transfers to the piloted vehicle's orbit, where the piloted vehicle rendezvous with the cargo vehicle, then several crew members will descend to the Martian surface. Approximately 30 days after the piloted vehicle arrives, it will depart Mars using the propellant that was carried on the cargo vehicle. The mission will end in January 2006 when the small Earth return capsule separates from the piloted vehicle and returns to low Earth orbit via an aerobraking maneuver.

1-5447-7

SPLIT OPTION TRAJECTORY PROFILE



SPLIT SPRINT MISSION VEHICLES

This chart shows the two vehicles used in the Split Sprint Mission. The cargo vehicle is shown with the Mars Excursion Module (MEM), and the propellant storage tanks. It requires one stage for Earth departure. The piloted vehicle is shown with its two Earth departure stages. Both the cargo and piloted vehicles have aerobrakes which are used for aerocapture at Mars. The Earth departure stages also have aerobrakes. They are all brought back to Earth for reuse via a braking burn and aerobraking. The first stage for the piloted vehicle is the same stage that is used for the cargo vehicle Earth departure.

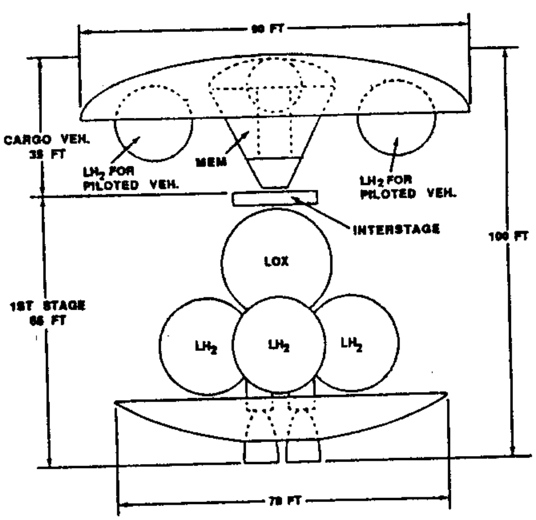
SPLIT SPRINT MISSION VEHICLES

CARGO MISSION VEHICLE CONFIGURATION

VEHICLE WEIGHTS (LBS)

CARGO VEHICLE	
ENGINES AND TANKS	46173
PROPELLANT	11016
TOTAL PROPULSION SYSTEM	57189
VEH. STRUCTURE AND SUBSYSTEMS	44500
AEROBRAKE	77276
PILOTED VEH. PROPELLANT	188253
TANKS FOR PILOTED VEH. PROP.	18908
MARS EXCURSION MODULE	133000
TOTAL GROSS WGT	519126
1ST STAGE	
ENGINES	15060
PROPELLANT TANKS	35611
PROPELLANT	790935
AEROBRAKE	9173
STRUCTURE AND SUBSYSTEMS	580
INTERSTAGE	20000
TOTAL GROSS WGT	871559
TOTAL EARTH DEPARTURE WGT	1,390,685

CARGO VEHICLE STACK AT LAUNCH



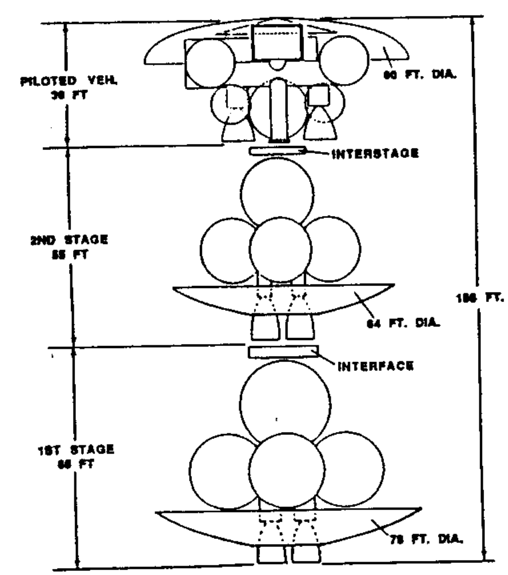
PILOTED MISSION VEHICLE CONFIGURATION

VEHICLE WEIGHTS (LBS)

PILOTED VEHICLE		
MAIN ENGINES (6)	=	2946
PROPELLANT TANKS	=	6870
PROPELLANT	=	188253
TOT. MAIN PROP. SYSTEM	=	198069
FLIGHT CONTROL SYSTEM	=	22325
AEROBRAKE	=	35500
CREW MODULES	=	131090
STRUCTURES AND SUBSYSTEMS	=	10320
EARTH RETURN CAPSULE	=	22032
TOTAL GROSS WGT.	=	419336
2ND STAGE		
ENGINES (4)	=	10040
PROPELLANT TANKS	=	17760
PROPELLANT	=	347220
AEROBRAKE	=	5884
SUBSYSTEMS	=	400
INTERSTAGE	=	20000
TOTAL GROSS WGT	=	401304
1ST STAGE		
ENGINES (6)	=	15060
PROPELLANT TANKS	=	35611
PROPELLANT	=	658938
AEROBRAKE	=	9173
SUBSYSTEMS	=	580
INTERSTAGE	=	20000
TOTAL GROSS WGT	=	739562

TOTAL GROSS DEPARTURE WGT = 1,560,202 LBS

PILOTED VEHICLE STACK AT LAUNCH



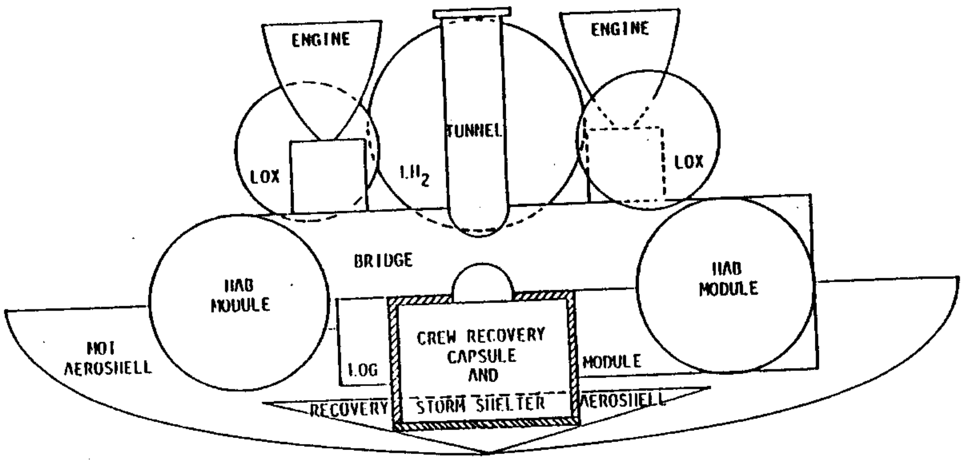
CREW VEHICLES

This chart shows the vehicles that will be used by the crew for the Split Sprint Mission. The schematic of the piloted vehicle shows two HAB modules similar to Space Station modules, and a logistics module. These modules are connected by a large bridge that also provides access to the Earth return capsule. The Earth return capsule is embedded into the piloted vehicle and surrounded by extra radiation shielding so that it can serve as a storm shelter if there is a solar particle eruption during the mission. The tunnel shown is used to transfer crew to the Mars excursion module when it is docked with the cargo vehicle.

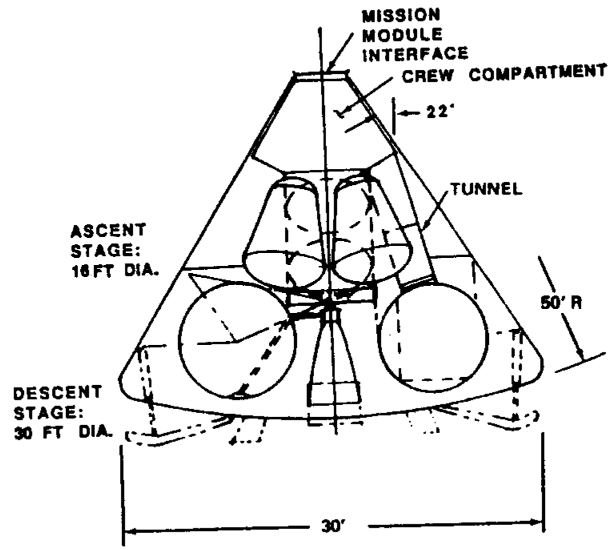
The Mars excursion module is the vehicle that descends to the Martian surface. It consists of a descent stage with a 30 ft heat shield and an ascent stage. It uses a combination of aerobraking, parachutes, and retro rockets for the descent. There are 3 crew compartments; one on top of the ascent stage, and two more in the descent stage.

CREW VEHICLES

PILOTED VEHICLE



MARS EXCURSION MODULE



FROM REFERENCE 22

ORBITAL OPERATIONS

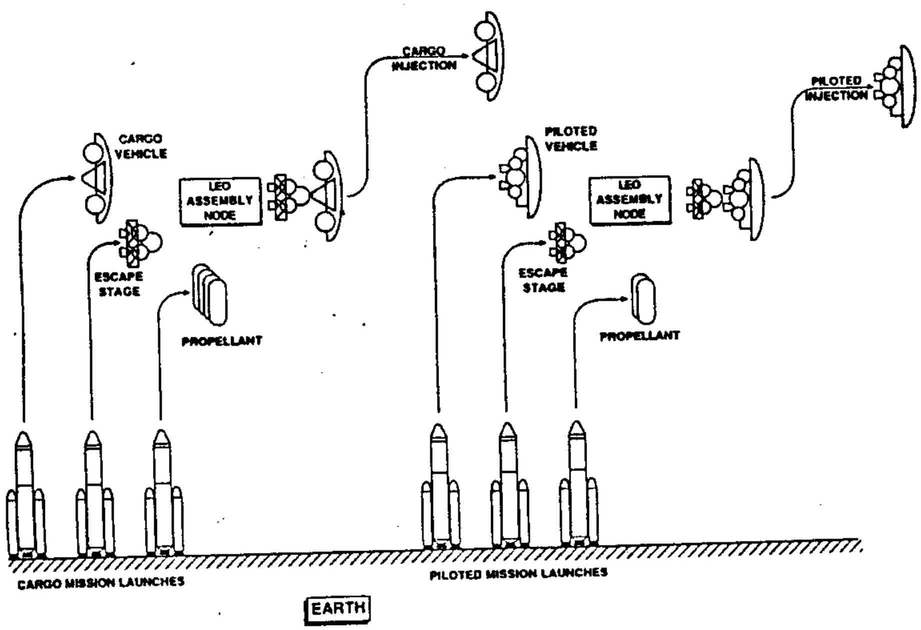
This chart shows a schematic of the orbital operations associated with a Split Sprint Mission. The first orbital operation begins with delivery of the cargo vehicle elements to low Earth orbit by some type of heavy lift launch vehicle. Assembly of the cargo vehicle and propellant loading would occur at a LEO assembly node. This node may or may not be associated with the Space Station. The cargo vehicle leaves Earth in June 2003, arrives at Mars in December 2003, and enters Mars orbit via aerocapture.

Approximately 1 1/2 years later, the piloted vehicle is assembled at the LEO assembly node. It leaves Earth in November 2004, and arrives at Mars in July 2005, where it will rendezvous with the cargo vehicle. After docking with the cargo vehicle, the crew transfers to the Mars excursion vehicle and descends to the surface for 20 days.

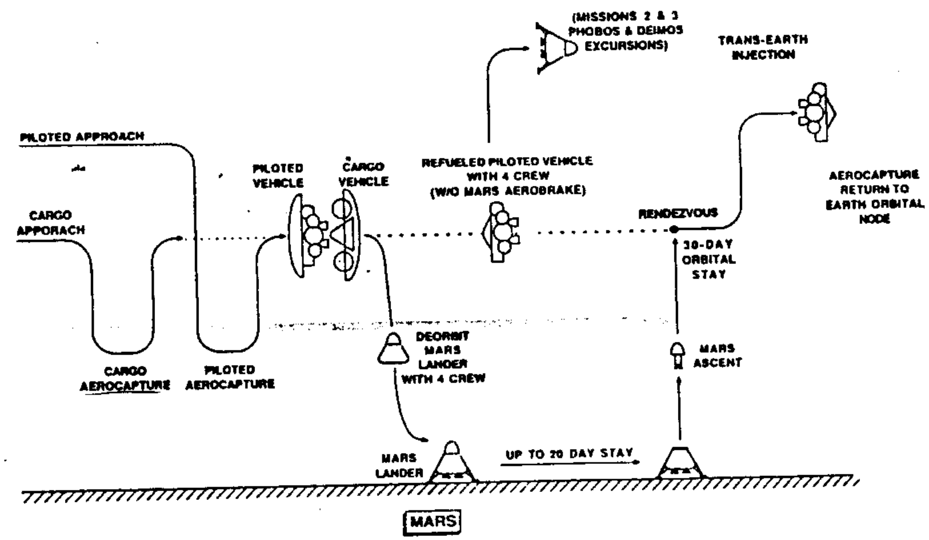
The piloted vehicle leaves Mars in August 2005, and arrives at Earth in January 2006. The small Earth return capsule is the only part of the piloted vehicle that is aerobraked into Earth orbit.

ORBITAL OPERATIONS

EARTH ORBITAL OPERATIONS



MARS ORBITAL OPERATIONS



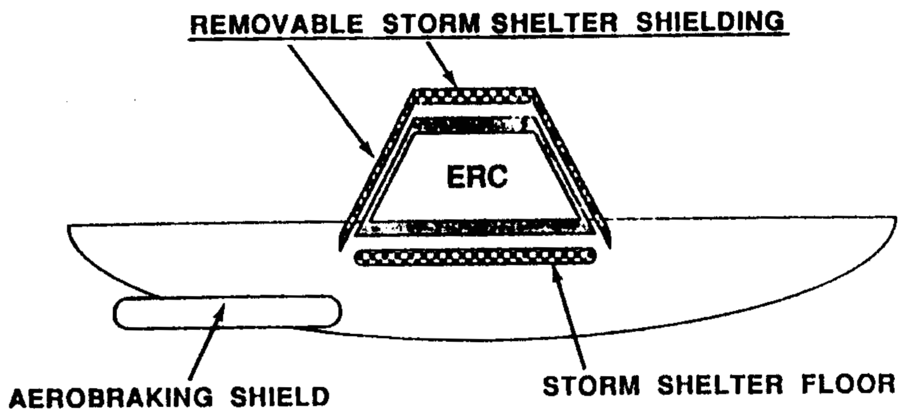
AEROCAPTURE OF THE EARTH RETURN CAPSULE

This chart shows how the Split Sprint Mission will end. The crew will enter the Earth return capsule and separate from the piloted vehicle just prior to arrival at Earth. The Earth return capsule will change its course so that it will enter the Earth's atmosphere to perform an aerocapture maneuver which will put it into low Earth orbit. The crew could then be recovered by an OMV and brought to the Space Station prior to returning to Earth, or they could be recovered by the Shuttle and returned directly to Earth. The rest of the piloted vehicle would remain in heliocentric orbit.

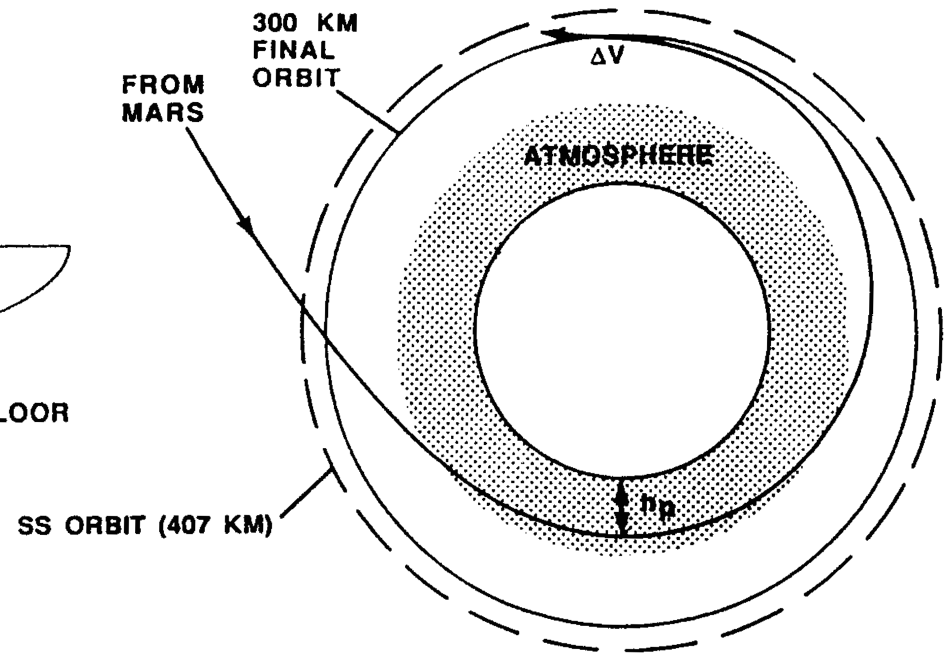
1-4171-8

AEROCAPTURE OF EARTH RETURN CAPSULE

EARTH RETURN CAPSULE



AEROCAPTURE TRAJECTORY



EARTH RETURN CAPSULE WGT.	=	19336 LBS
AEROBRAKE WGT.	=	2273 LBS
TOTAL BURN OUT WGT.	=	21609 LBS
TOTAL PROPELLANT WGT.	=	318 LBS
TOTAL RCS PROPELLANT	=	105 LBS
TOTAL GROSS WGT.	=	22032
AEROBRAKE DIAMETER	=	37.5 FT.
ISP	=	460 SEC.

INCOMING C_3	=	11.74 KM^2/SEC^2
hp	=	81.59 KM
TIME IN ATMOSPHERE	=	796.5 SEC
MAX. ACCELERATION	=	2.81 g's
AVG. ACCELERATION	=	1.28 g's
MAX. DYN. PRESSURE	=	30.37 LBS/FT^2
TOTAL DELTA VELOCITY	=	65.353 M/S

MARS EXPLORATION
SPLIT SPRINT MISSION
GROUNDRULES

The split sprint mission profile is used in order to minimize the Mars mission crew time required in space and to minimize the total mass required in low earth orbit (LEO) to perform the mission. The split sprint mission concept utilizes a cargo vehicle and a piloted vehicle; the cargo vehicle preceeds the piloted vehicle by about a year and a half. LEO departure is June 2003 for the cargo vehicle and November 2004 for the piloted vehicle with a crew of six. Aerodynamic braking both into Mars capture orbit and into Earth return orbit is used. Liquid hydrogen and liquid oxygen cryogenic propellants are used by the main propulsion system for the Earth-to-Mars transfer injection burn and for Mars-to-Earth transfer injection burn.

MARS EXPLORATION
SPLIT SPRINT MISSION
GROUNDRULES

- CONSIDER SPLIT SPRINT MISSION SCENARIO
  - CARGO VEHICLE PRECEEDS PILOTED VEHICLE
- USE EXISTING DEFINITION OF MARS CARGO VEHICLE ELEMENTS AND MARS SURFACE ACTIVITIES
- 2003 OPPOSITION FOR CARGO VEHICLE (JUNE 2003 LAUNCH)
- 2005 OPPOSITION FOR PILOTED VEHICLE (NOVEMBER 2004 LAUNCH)
- OPPOSITION SPRINT TRAJECTORY FOR PILOTED MISSION
- CREW SIZE OF 6
- AERODYNAMIC BRAKING INTO MARS ORBIT AND INTO EARTH RETURN ORBIT
- ACCELERATION LIMITS
  - NOT TO EXCEED 3.5G'S FOR MORE THAN 1 MINUTE WITH 7G'S PEAK DURING MEM DESCENT
- CRYO PROPELLANT FOR EARTH-MARS & MARS-EARTH TRANSFER
- PROVIDE SAFE HAVEN FOR CREW SAFETY

MARS EXPLORATION
SPLIT SPRINT MISSION
GROUNDRULES (CONTINUED)

Provision of safe haven and storm protection from solar flares with a 15 minute warning is specified for crew safety. Planned extra vehicular activity (EVA) is required in Mars orbit to prepare for surface operations, refuel the piloted vehicle, etc., and contingency EVA is assumed to be required for the Earth-Mars and Mars-Earth transit phases. Use is made of available space and launch elements for operations and derivatives of these elements for Mars vehicle components with a simple, safe, and low-cost concept desired.

MARS EXPLORATION
SPLIT SPRINT MISSION
GROUNDRULES (CONTINUED)

- PROVIDE STORM PROTECTION FOR SOLAR FLARES WITH 30 MIN WARNING
- UTILIZE PLANNED EVA IN MARS ORBIT
- CONTINGENCY EVA DURING EARTH-MARS & MARS-EARTH TRANSIT
- CONSIDER AVAILABLE SPACE AND LAUNCH ELEMENTS FOR OPERATIONS AND DERIVATIVES OF THESE ELEMENTS FOR MARS VEHICLE COMPONENTS
    - SPACE STATION
    - OTV
    - OMV
    - SDV OR HLLV
    - STS
- SIMPLE, SAFE, LOW COST CONCEPT DESIRED

2.0 MISSION AND OPERATIONS ANALYSIS

The following section presents information developed on mission and operations analysis for the Split Sprint Mission.

2.0 Mission and Operations Analysis

2.1 Mission Opportunities, Earth/Mars Oppositions for 1997-2031

2.2 Split Option Trajectory Profile

2.3 Earth Departure Analysis

2.3.1 Space Station Orbital Geometry
2.3.2 Space Station/Planetary Injection Strategies
2.3.3 Illustrative Energy Requirements for a Planetary Launch Window
2.3.4 In-Plane Earth Departure Delta-Velocities for a 50 Day Launch Window
2.3.5 Use of Intermediate Orbit for Departure Node Corrections (Plane Changes)
2.3.6 Comparison of Earth Orbit Departure Window Options
2.3.7 Optimizing Space Station Orbit to Allow In-Plane Departure/Arrival Conditions

2.4 Mars Parking Orbit Analysis

2.4.1 Mars Orbit Characteristics, Period, Regression Rate, Sun Synchronous Orbits
2.4.2 Mars Orbit Lifetimes (Atmospheric Decay)
2.4.3 Cargo Vehicle Parking Orbit Selection
2.4.4 Piloted Vehicle Parking Orbit Selection
2.4.5 Piloted Vehicle Percent Time In Sun During Mars Orbit

2.5 Transfers Between Mars Parking Orbit and Mars Surface

2.5.1 Orbit and Landing Site Geometry
2.5.2 Orbit to Surface Rendezvous Compatible Trajectory Characteristics
2.5.3 Rendezvous Compatible Repeating Mars Parking Orbits
2.5.4 Orbital Phasing Requirements to Deboost from a Non-Repeating Parking Orbit (Position Error and Phasing Orbit Altitude)

2.6 Mars Mission Communications

2.6.1 Piloted Vehicle Communications Distances
2.6.2 Light Travel Times Between Piloted Vehicle, Earth, and Mars
2.6.3 Sun Interference Angles Affecting Communication
2.6.4 Key Communications Issues

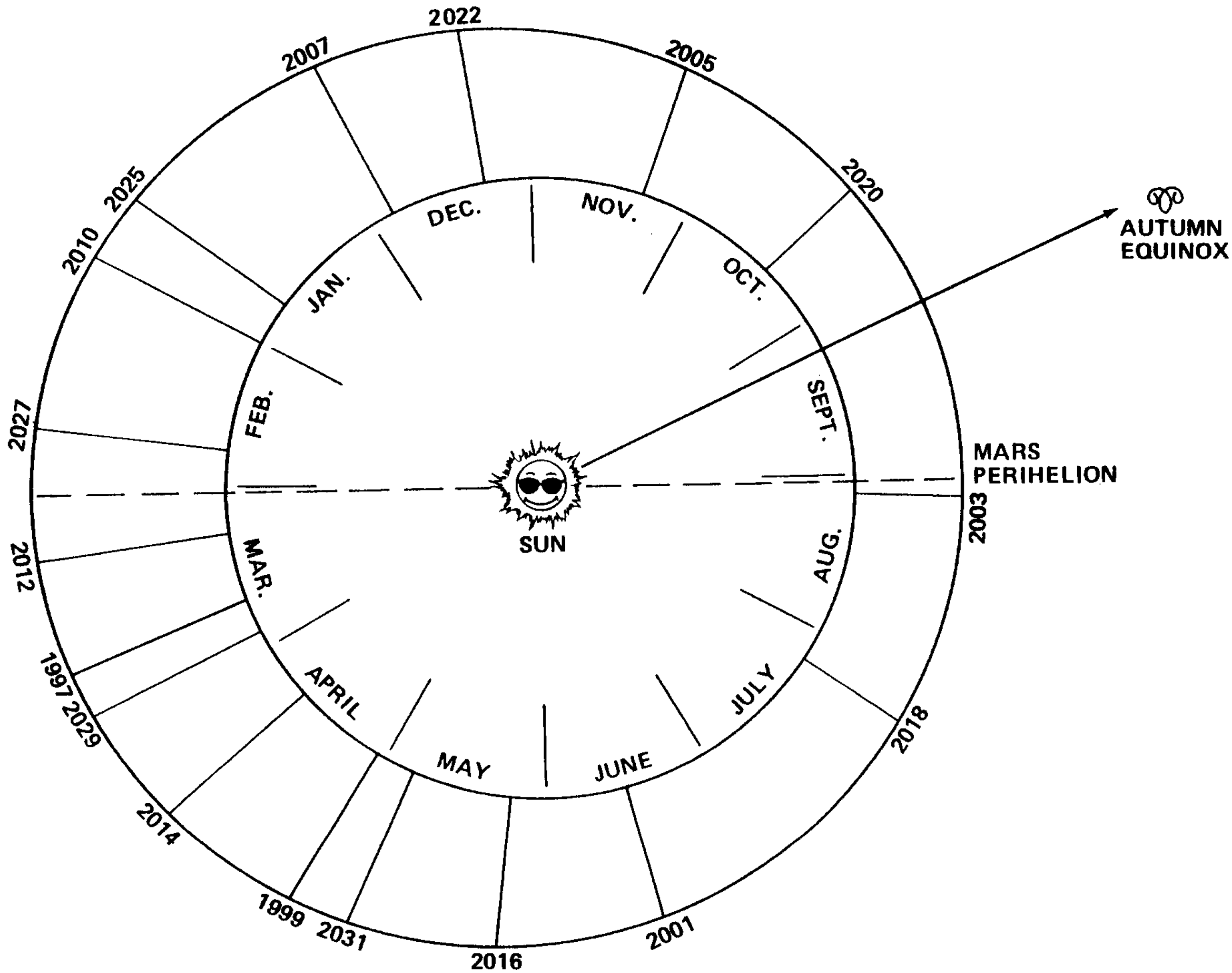
2.7 Mission Operations

2.7.1 Vehicle Assembly Operations
2.7.2 Piloted Mission Orbital Operations, Timeline
2.7.3 Mission Abort Scenario
2.7.4 Typical In-Flight Daily Schedule

EARTH-MARS OPPOSITION FOR YEARS 1997-2031

This chart shows the relative positions of Earth-Mars oppositions for the years 1997 to 2031. Each opposition is shown by a line joining the positions of Earth and Mars. Launch opportunities for standard direct flights to Mars occur 90 to 180 days before each opposition. The oppositions occur about every 26 months and follow an approximate 15 year cyclic pattern. Since Mars' orbit is slightly eccentric, some mission opportunities require more energy than others. The closer Mars is to the Earth's orbit, the less energy is required for the transfer to Mars. Thus, the 2003 opposition is a very good mission opportunity, while the 2012 opposition would be poor since it would require much more energy. An important characteristic of opposition class missions is the fact that either the inbound or outbound transfer leg must dip inside the Earth's orbit and travel more than 180 degrees around the Sun (see page 21). This requires a great deal of energy and is required to compensate for relative velocity of Mars with respect to the Earth. This high energy requirement can be diminished somewhat by using a Venus swingby for a gravity assist, or by performing a propulsive maneuver at the vehicle's perihelion to alter the trajectory.

3546–84



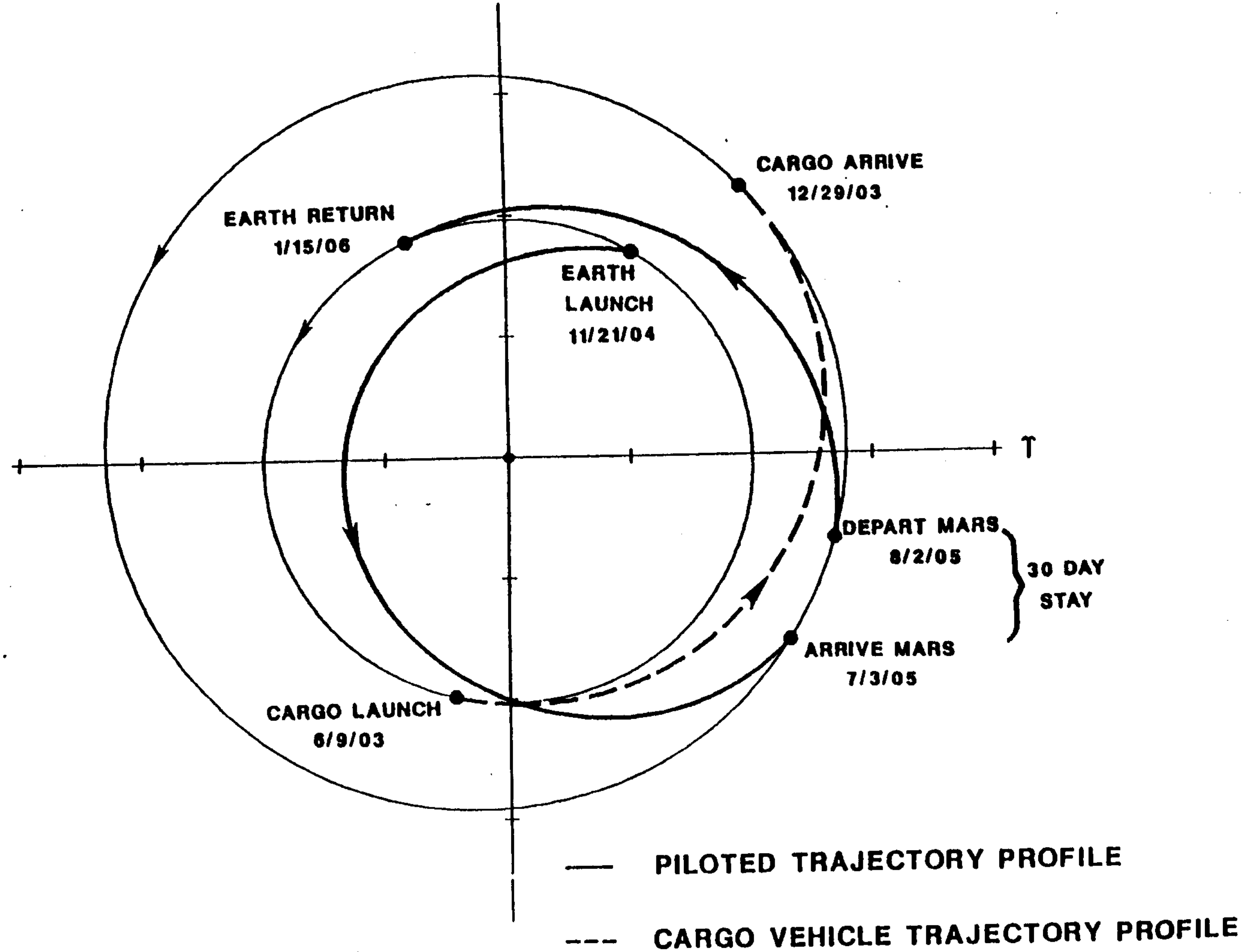
EARTH–MARS OPPOSITION FOR YEARS 1997–2031

SPLIT OPTION TRAJECTORY PROFILE

The split option trajectory profile consists of a low energy trajectory for the cargo vehicle and a high energy trajectory for the piloted vehicle. All cargo not required by the crew for the outbound leg of the mission to Mars is sent on a robotic vehicle using the low energy trajectory. The cargo on this unmanned vehicle contains the Mars lander and ascent vehicle, all deployed science packages, and the propellant for the return leg of the piloted vehicle mission. The trajectory used for the cargo vehicle departs Earth in June 2003 and arrives at Mars seven months later. The piloted vehicle would be launched on the sprint trajectory approximately 1 1/2 years later. The cargo and piloted vehicle will rendezvous and dock in a Mars 1000 Km altitude circular orbit. The total mission time for the sprint mission is 14 months. The trajectory allows for an aerobraked Mars flyby if the landing had to be aborted prior to Mars orbit insertion. This maneuver would adjust the trajectory energy level for a free return trajectory to Earth.

1-5447-7

SPLIT OPTION TRAJECTORY PROFILE



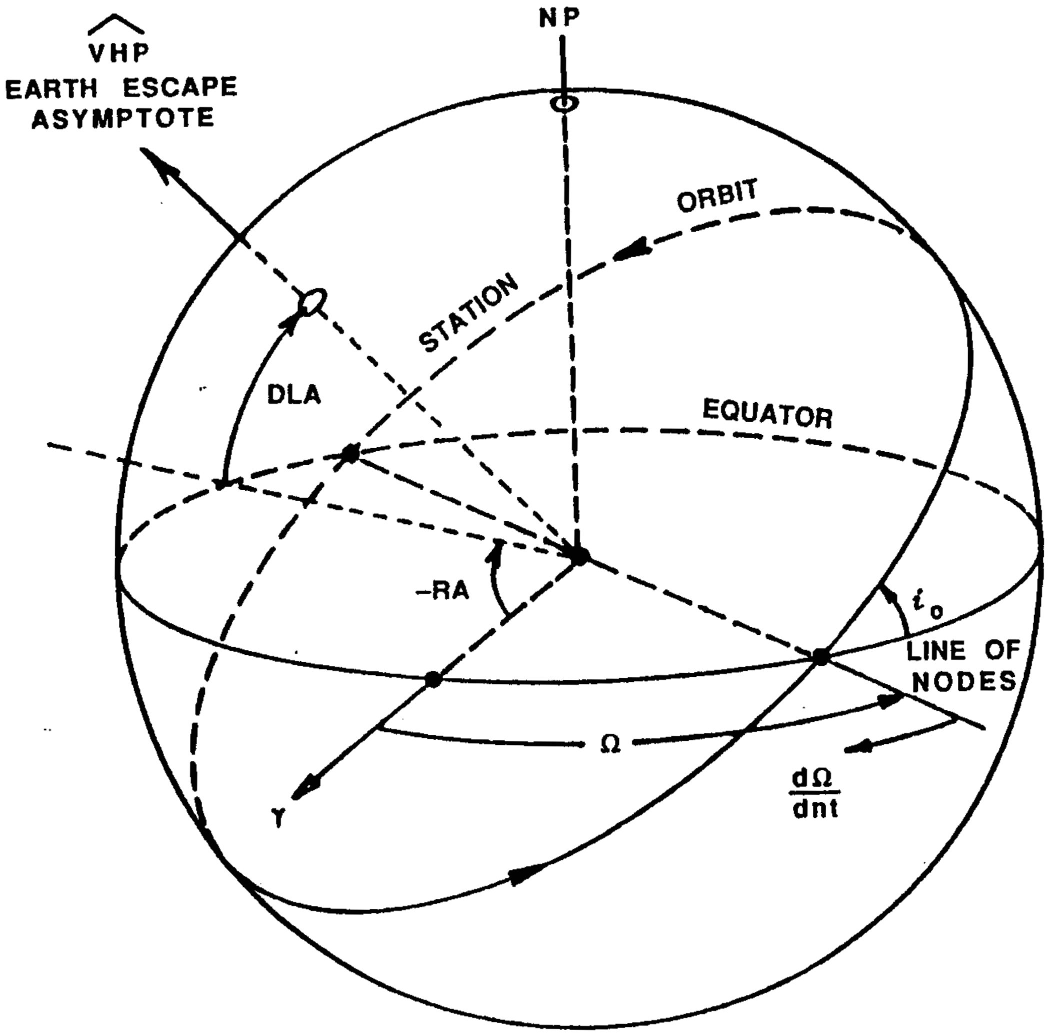
MARS EXPLORATION
SPACE STATION ORBITAL GEOMETRY

The ideal situation for launching a planetary mission from the Space Station orbit would be for in-plane launch conditions to exist from the Space Station orbit on the day of minimum Earth escape C_3. Since these conditions for optimum Space Station orbit departure will not, in general, coincide with the time of minimum C_3 (C_3 = VHP^2), a trade-off exists between the amount of plane change required and the C_3 of off-optimal launch dates. When a planetary mission opportunity occurs, the direction of the Earth escape (VHP) is relatively constant across the launch window, which typically lasts 10-50 days.

Conversely, the station orbit plane is constantly precessing due to the oblateness of the Earth; for the assumed orbit (220 NM circular at 28.5^{o} inclination) the nodal regression is 7.14^{o} deg./day westward. Assuming the angle (declination, DLA) from VHP to the Earth's equator is less than the orbit inclination, there will be only two times every 50 days when VHP lies in the station orbit plane, which is the condition for optimum coplanar escape. At all other times, a plane change (and performance loss) is required to achieve the correct escape conditions.

1-4223-8

SPACE STATION ORBITAL GEOMETRY



ORBIT PARAMETERS:

ALTITUDE, h=220nm (CIRCULAR)

INCLINATION, i_o =28.5°

ASCENDING NODE, Ω =VARIABLE

NODAL REGRESSION, $\frac{d\Omega}{dt}$ = -0.45938°/REV

= -7.13529°/DAY

$\Delta\frac{d\Omega}{dt}$ = 0.0067°/DAY PER. nm IN Δh

MARS EXPLORATION
PLANETARY MISSION DEPARTING FROM SPACE STATION ORBIT

There are several different options available for planetary missions departing from the Space Station orbit. These options may be characterized under the heading of active or passive launch strategies. In the active strategy, a propulsive maneuver by the main propulsion system is required to achieve the necessary plane change. The plane change can be made either in Earth orbit or on the interplanetary transfer trajectory. Plane change in Earth orbit can be made in different ways; listed here are three different ways: split maneuver, combined maneuver, and three-impulse maneuver. The split maneuver being the most expensive and the three-impulse being the less expensive in terms of extra delta velocity required to obtain desired Earth escape conditions. Broken plane transfer can be made on the interplanetary transfer trajectory in order for the Earth escape conditions to be compatible with the Space Station orbit orientation.

1-3815-8-14T

SPACE STATION/PLANETARY INJECTION STRATEGIES

	UTILIZATION PRIORITIES
● ACTIVE (PROPULSIVE PLANE CHANGES)	
● EARTH-ORBITAL	AS NEEDED FOR DLA TARGETING & LAUNCH DELAYS
– SPLIT MANEUVER	– STATION ORBIT REALIGNMENT: EXPENSIVE
– COMBINED MANEUVER	– NON-PLANAR ESCAPE: LESS EXPENSIVE
– THREE-IMPULSE MANEUVER	– APOAPSE PLANE CHANGE: LEAST EXPENSIVE (BUT REQUIRES $\geq 24^h$ INTERMEDIATE ORBIT)
● INTERPLANETARY	
– BROKEN PLANE TRANSFERS	VERY EFFECTIVE ON SOME MISSIONS IN REDUCING DLA PENALTIES AND IN IMPROVING OFF-OPTIMAL ESCAPE REQUIREMENTS FOR PASSIVE STRATEGY (SEE BELOW)
● PASSIVE (LAUNCH DATE TIMING)	BASELINE SOLUTION
– STATION ORBIT PRECESSION	WAITS FOR ORBIT REALIGNMENT, ACCEPTING SOME PERFORMANCE LOSS FROM RESULTING OFF-OPTIMAL LAUNCH DATE

ILLUSTRATIVE PLANETARY LAUNCH WINDOW

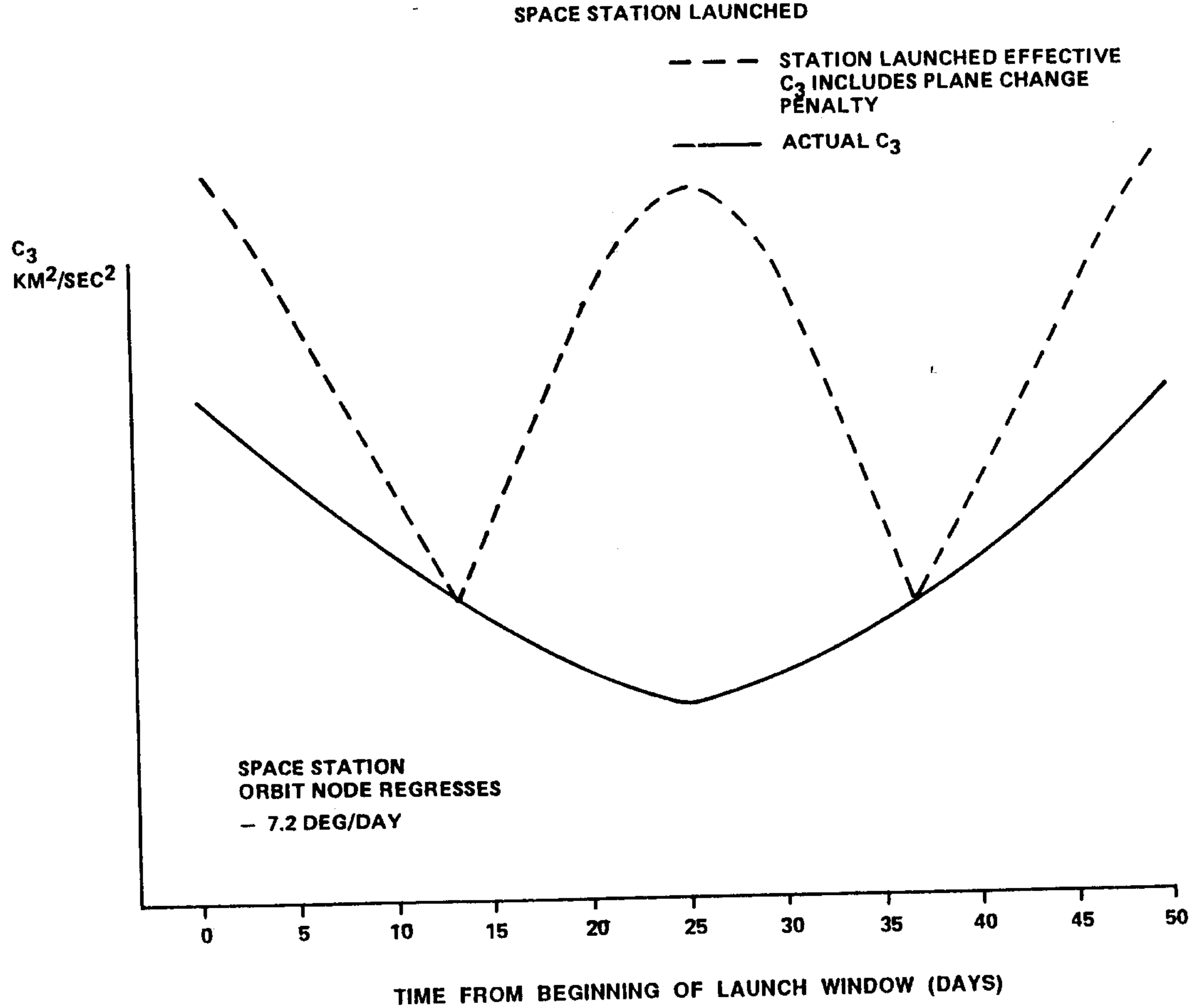
This chart illustrates how the departure C_3 (energy) requirements vary during a 50 day launch window for a Mars mission. The solid line is the in-plane C_3 requirement. This represents the departure C_3 required for the Earth-Mars trajectory if the Earth parking orbit plane contains the departure vector (asymptote). The variation in the C_3 requirements is due to the motion of Earth and Mars. The launch window is centered around the most favorable alignment of the two planets; therefore, the lowest in-plane departure condition occurs at the mid-point of the launch window. The vertical scale has been greatly magnified. The actual variation in C_3 requirements during the launch window would be much less than it appears.

Since the Mars vehicles would probably be departing from the Space Station orbit, which is constantly precessing, it would be impossible to always depart from the ideal orbital plane. In fact, during the 52 day Space Station orbital precession period, the orbit plane would pass through departure asymptote only twice. At all other times, a plane change would be required to leave the parking orbit in the direction of the departure asymptote. The dashed line represents the effective C_3 requirements due to the addition of the required plane change to the in-plane C_3 requirements. The two points where the dashed line meets the solid line represent the times when the Space Station orbit plane contains the departure asymptote. In this case, the best time to depart would be at 14 or 37 days into the window.

In reality, the dashed line could be shifted to the left or right so that it would intersect the solid line at different times. The most fortuitous alignment would be if the Space Station orbit passed through the departure asymptote on the day with the minimum in-plane C_3 requirement.

1-6906-8

ILLUSTRATIVE PLANETARY LAUNCH WINDOW



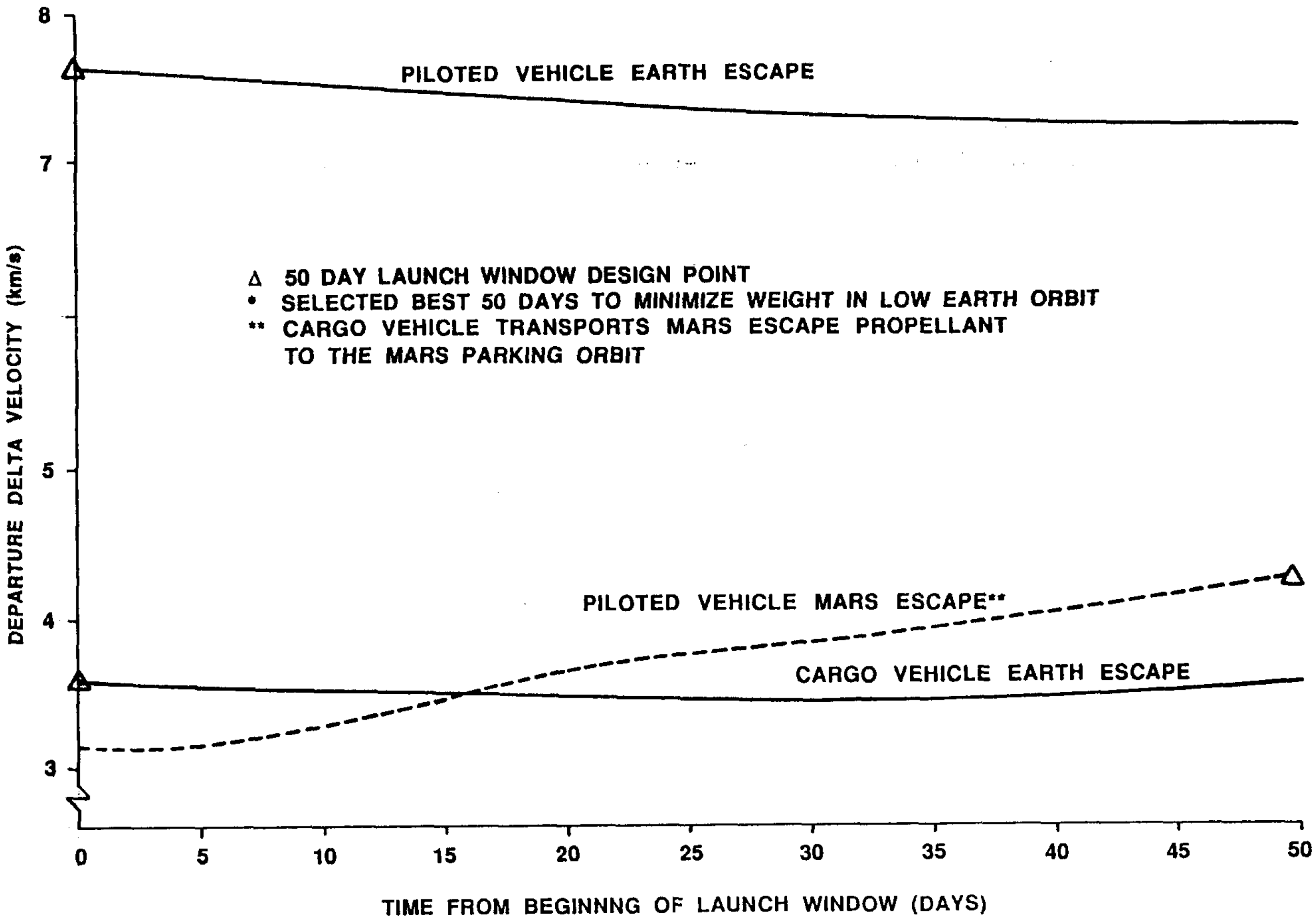
EARTH AND MARS DEPARTURE DELTA VELOCITIES FOR A 50 DAY EARTH LAUNCH WINDOW

This chart shows the required inplane Earth escape delta velocity for the cargo and piloted Mars missions for a 50 day launch window, and also the effect of a 50 day Earth launch window on the Mars escape delta velocity. The Earth escape delta velocities are shown with the solid lines. The 50 day launch window that was chosen represents the best 50 days, which result in the lowest total weight in low Earth orbit. This is slightly different than choosing the launch window, which results in the lowest Earth departure energy. These curves represent the in-plane departure delta velocities. When the Space Station orbit does not coincide with the departure vector, an additional plane change would be necessary. The dashed line shows how the Mars departure delta velocity is affected by the day of the Earth departure. The maximum values for each curve are identified by triangles. They represent the design values that would be used in determining the propulsion requirements of the vehicles.

1-5959-7

EARTH AND MARS DEPARTURE DELTA VELOCITIES FOR A 50 DAY LAUNCH WINDOW

SPLIT SRINT MISSION CARGO VEHICLE LAUNCH WINDOW STARTS JUNE 2003
PILOTED VEHICLE LAUNCH WINDOW STARTS NOVEMBER 2004

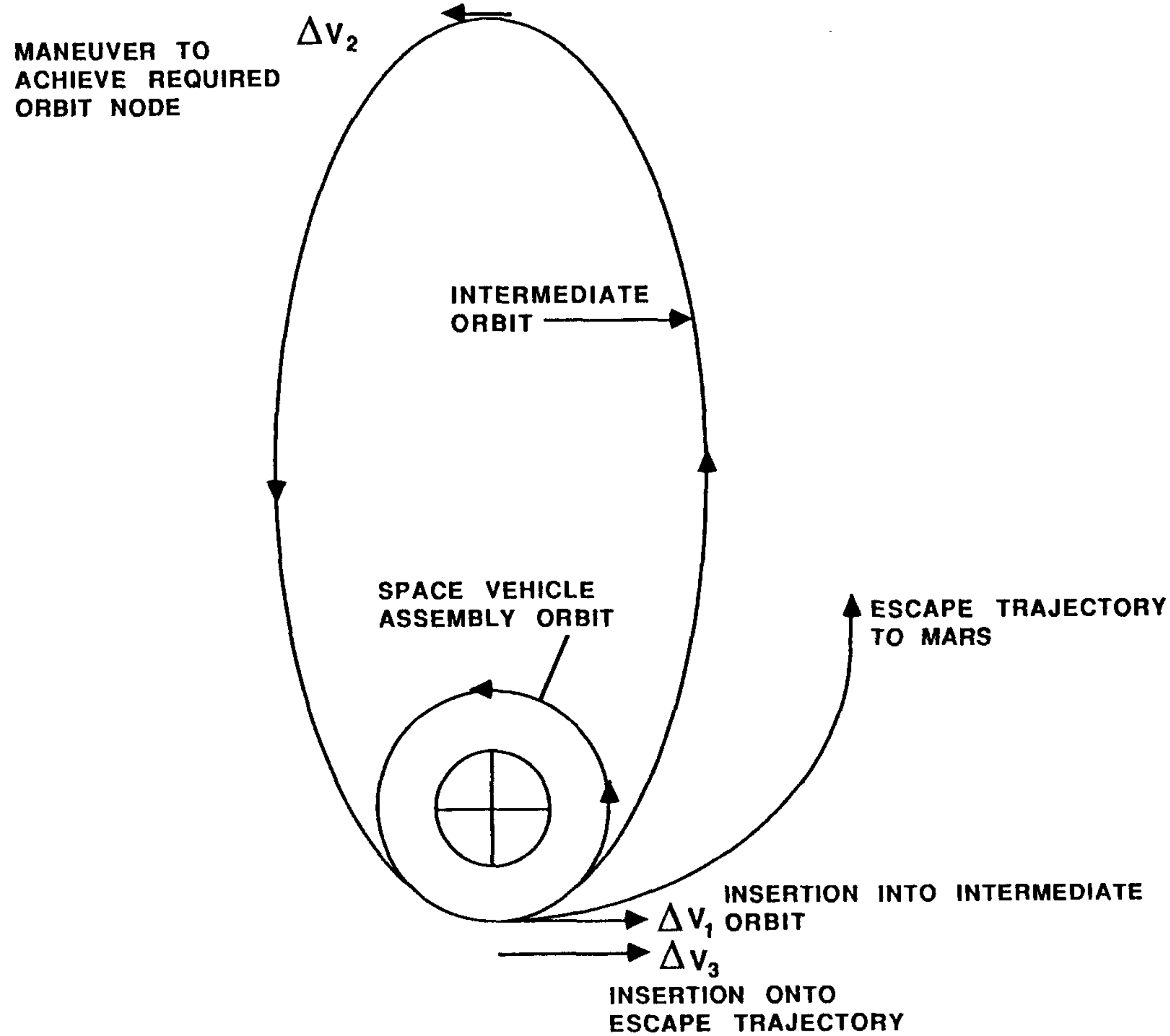


MARS EXPLORATION
EARTH ESCAPE VIA THREE-IMPULSE MANEUVER

In order to minimize the extra ΔV required to achieve Earth escape conditions for the Mars mission when non-co-planar situations exist, a three-impulse maneuver is employed. The first impulse (ΔV_1) places the space vehicle into an intermediate orbit where it coasts to apogee; at apogee, a second impulse (ΔV_2) is made to achieve the required orbit node which will provide co-planar conditions with the Earth escape vector VHP. At perigee, a third impulse (ΔV_3) is made to insert the space vehicle onto the desired escape trajectory. See Page 33 for ΔV penalty to achieve required orbit node (align orbit plane).

1-4225-8-18D

EARTH ESCAPE THROUGH INTERMEDIATE ORBIT TO CORRECT ORBIT NODE



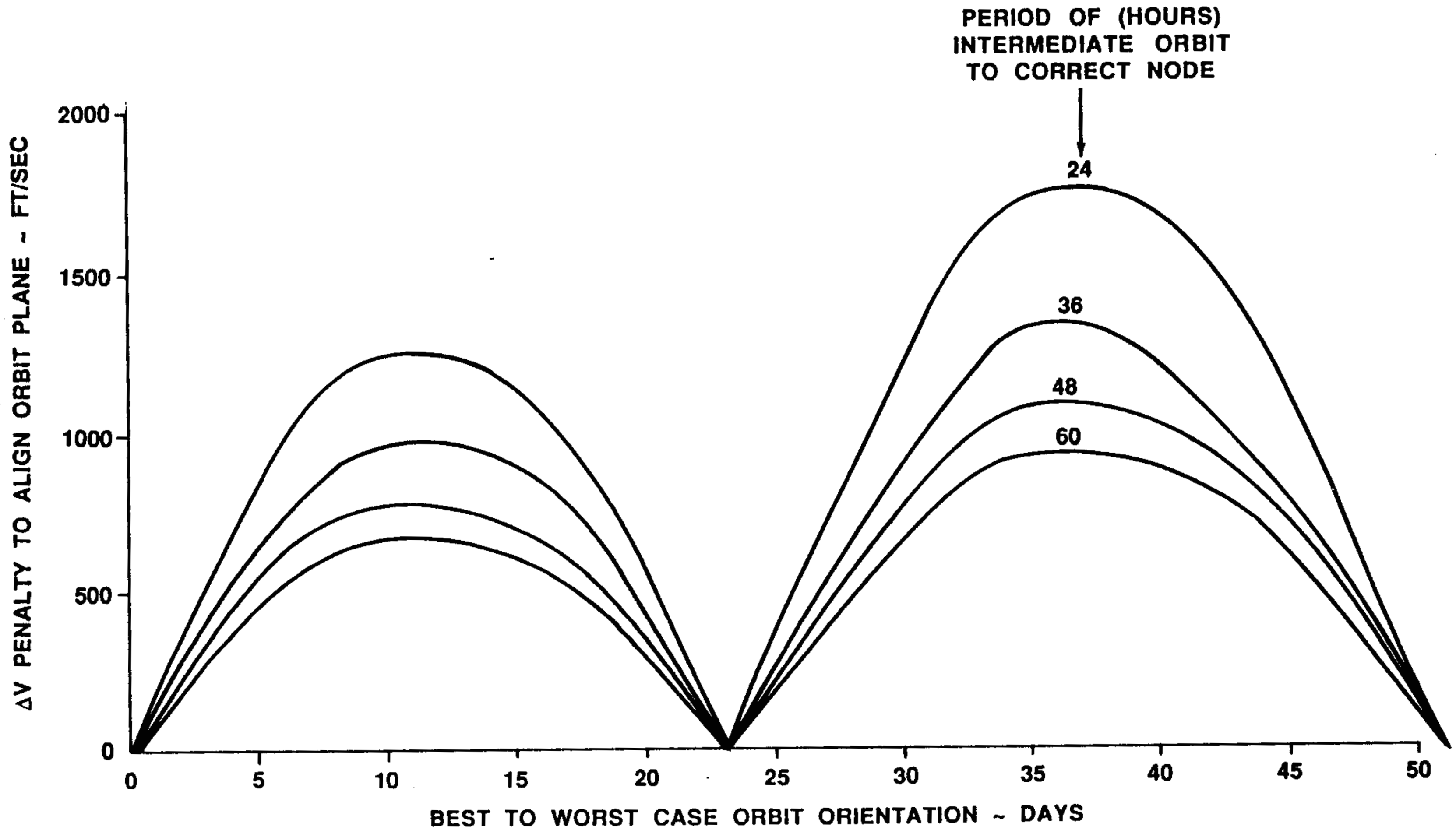
DELTA VELOCITY PENALTY ABOVE INPLANE LAUNCH CONDITION

This chart shows the delta velocity penalty that must be added to the inplane Earth departure delta velocities for Mars missions. The delta velocity penalty represents the additional delta velocity required to perform the 3 impulse departure described in the previous chart. This chart shows the delta velocity penalty as a function of time during a 50 day launch window. This corresponds to one orbital precession period of the Space Station orbit. There are two inplane launch opportunities during this time at which there is no plane change penalty. The delta velocity penalty at all other times depends on the time within the launch window and the period of the intermediate orbit (or time spent in the intermediate orbit). The longer the period of the intermediate orbit, the higher the altitude at which the plane change is made, and the lower the delta velocity penalties associated with four different intermediate orbits. The highest penalty occurs on the 37th day where the penalty is 930 to 1760 ft/sec.

1-5958-7

ΔV PENALTY ABOVE INPLANE LAUNCH CONDITIONS

CARGO VEHICLE TRAJECTORY DECLINATION IS –0.6 DEGREES
PILOTED VEHICLE TRAJECTORY DECLINATION IS 4.8 DEGREES

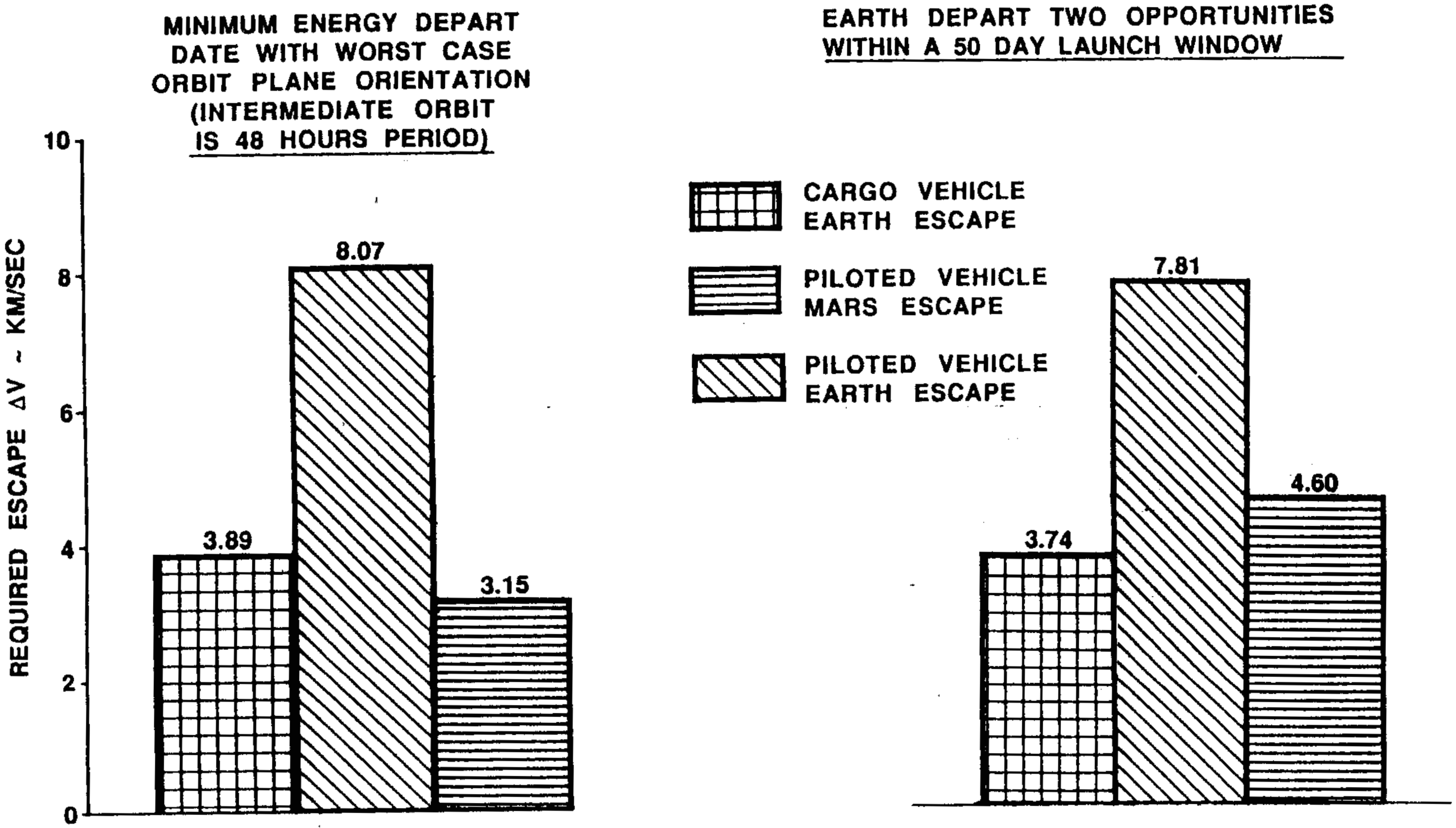


COMPARISON OF EARTH ORBIT DEPARTURE WINDOWS
MISSION DELTA VELOCITIES

This chart shows a comparison of the Earth and Mars departure delta velocities that correspond to two Earth departure window options. The data on the left side shows the delta velocities required if the departure occured on the day of the best alignment of Earth and Mars (minimum energy departure date), but with the orbit plane at its worst orientation with respect to the departure vector. A plane change would be required to leave the orbit in the direction of the departure vector. This plane change would be made using a 3 impulse maneuver in which the vehicle would enter a 48 hour intermediate orbit before leaving Earth. The data on the right shows the delta velocities required if departure occurs within a 50 day launch window at one of two in-plane departure opportunities (orbit plane contains the departure vector). The launch window would be centered around the minimum energy departure data, but the in-plane departure would not occur on this date. In order to determine which window option is better, it is necessary to consider the total weight in low Earth orbit required for each option.

1-5960-7

COMPARISON OF EARTH ORBIT DEPARTURE WINDOW OPTIONS MISSION DELTA VELOCITIES

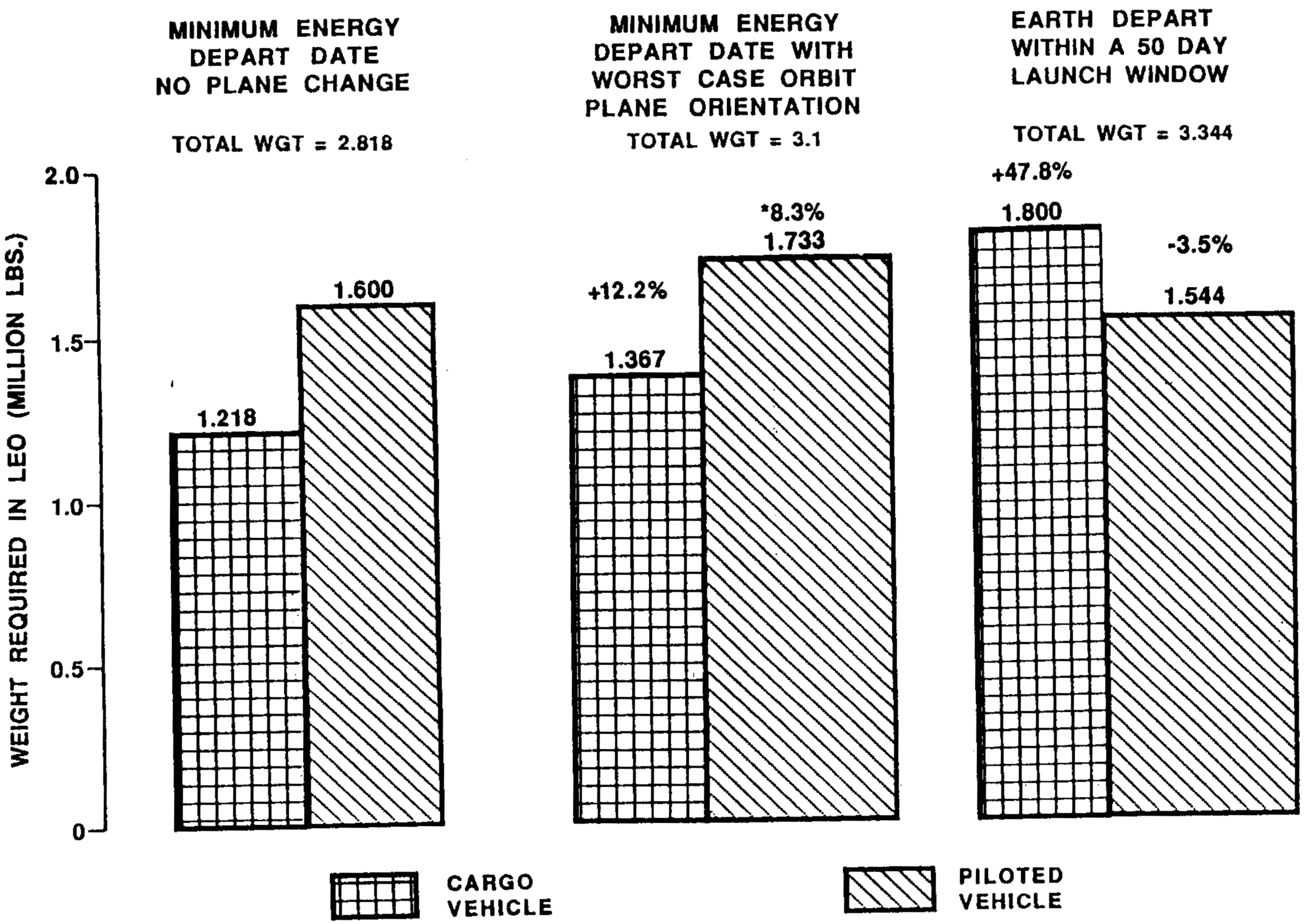


COMPARISON OF EARTH ORBIT DEPARTURE WINDOW OPTIONS
WEIGHT REQUIRED IN LOW EARTH ORBIT

This chart shows a comparison of the required weight in Low Earth Orbit (LEO) for the cargo and piloted vehicles for three launch window options. This roughly corresponds to the differences in propellant weight required. The first set of data shows the weight required in LEO if the missions were to depart on the minimum energy departure day with no plane change necessary. This would only be possible if the plane of the Earth parking orbit happened to contain the daparture vector on that day. The second set of data shows the weight required in LEO if the departure occurred on the day of minimum departure energy, but with the Earth parking orbit at the worst possible orientation with respect to the departure vector. A fairly large plane change would be required to leave from the parking orbit in the direction of the departure vector. The third set of data shows the weight in Earth orbit required if the departure occurred at one of two in-plane opportunities within a 50 day launch window. These two opportunities would not occur on the day of minimum energy departure conditions. Judging by the total weight required in LEO for both the cargo and piloted missions, it appears that the middle options would be best.

1-5961-7

COMPARISON OF EARTH ORBIT DEPARTURE WINDOW OPTIONS WEIGHT REQUIRED IN LOW EARTH ORBIT



SPACE STATION/SPRINT NODE ALIGNMENT
CARGO VEHICLE

The Space Station is used in support of the assembly of both the cargo and piloted Mars vehicles and as the return port for the piloted vehicle. The orbit plane of the Space Station must therefore be aligned with the departure asymptotes of both the cargo and piloted vehicles, as well as the Earth return asymptote of the piloted vehicle.

The trans-Mars injection of the cargo vehicle can be initiated from either an ascending or descending leg of the Earth departure orbit, which constrains the right ascension of the ascending Cargo Departure Node (CDN) to be either 63.9 deg or 246.1 deg.

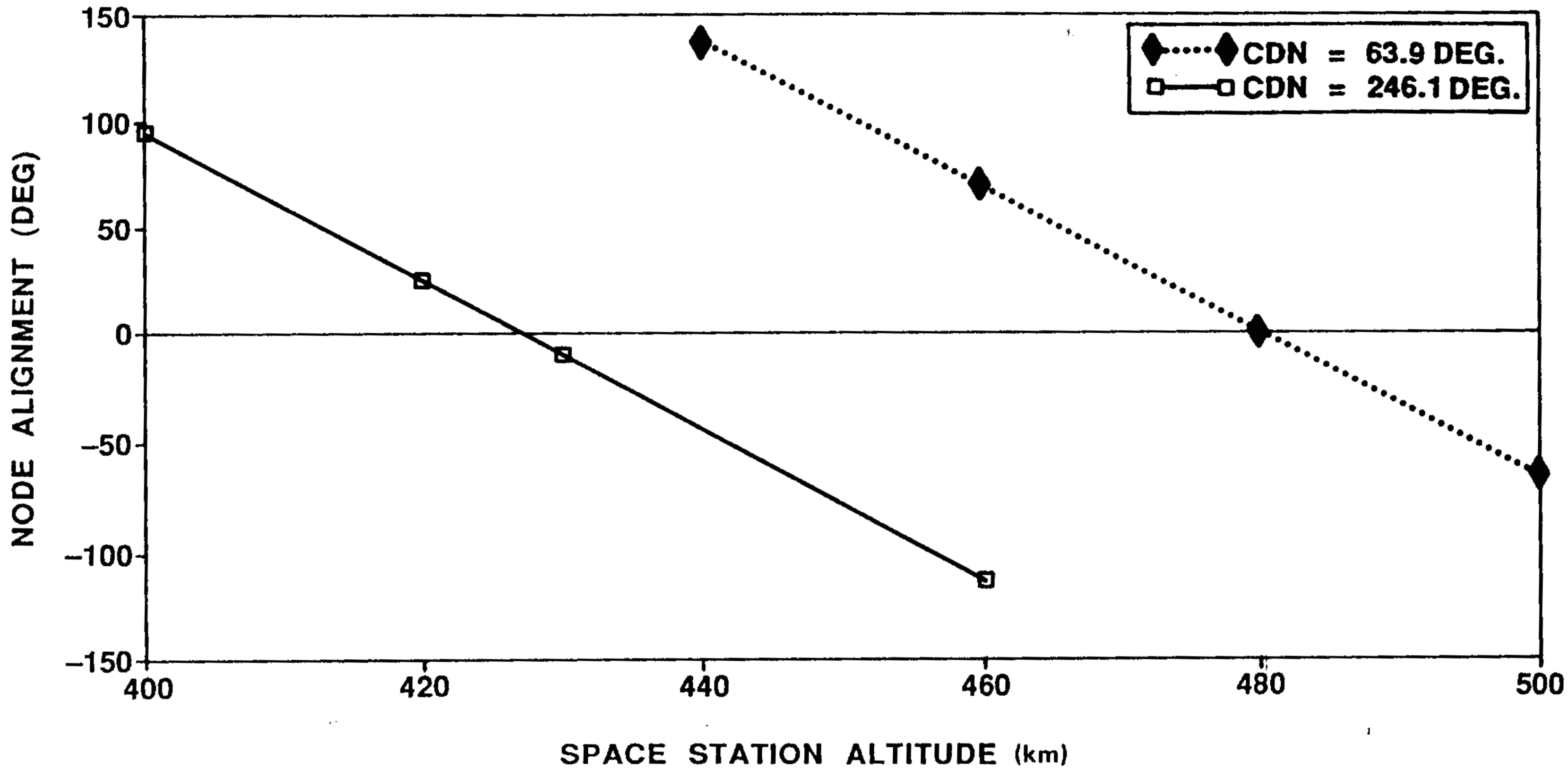
The right ascension of the Piloted Arrival Node (PAN) at Earth return can be either 12.5 deg or 102.5 deg depending on whether the vehicle is targeted for the northern or southern hemisphere.

The facing chart shows the nodal alignment of the Space Station orbit relative to the 12.5 degree ascending node of the Earth return orbit of the piloted vehicle. The curve indicates that either a Space Station orbit altitude of approximately 428 km or 480 km is required for the Space Station to be co-planar with the cargo vehicle Earth departure orbit and the piloted vehicle Earth departure and return orbits.

SPACE STATION/SPRINT NODE ALIGNMENT

CARGO VEHICLE

PAN = 12.5 DEG.



KEY

PAN - RIGHT ASCENSION OF SPACE STATION ASCENDING NODE AT PILOTED VEHICLE ARRIVAL

CDN - RIGHT ASCENSION OF SPACE STATION ASCENDING NODE AT CARGO VEHICLE DEPARTURE

SPACE STATION/SPRINT NODE ALIGNMENT
CARGO VEHICLE

The Space Station is used in support of the assembly of both the cargo and piloted Mars vehicles and as the return port for the piloted vehicle. The orbit plane of the Space Station must therefore be aligned with the departure asymptotes of both the cargo and piloted vehicles, as well as the Earth return asymptote of the piloted vehicle.

The trans-Mars injection of the cargo vehicle can be initiated from either an ascending or descending leg of the Earth departure orbit, which constrains the right ascension of the ascending Cargo Departure Node (CDN) to be either 63.9 deg or 246.1 deg.

The right ascension of the Piloted Arrival Node (PAN) as Earth return can be either 12.5 deg or 102.5 deg depending on whether the vehicle is targeted for the northern or southern hemisphere.

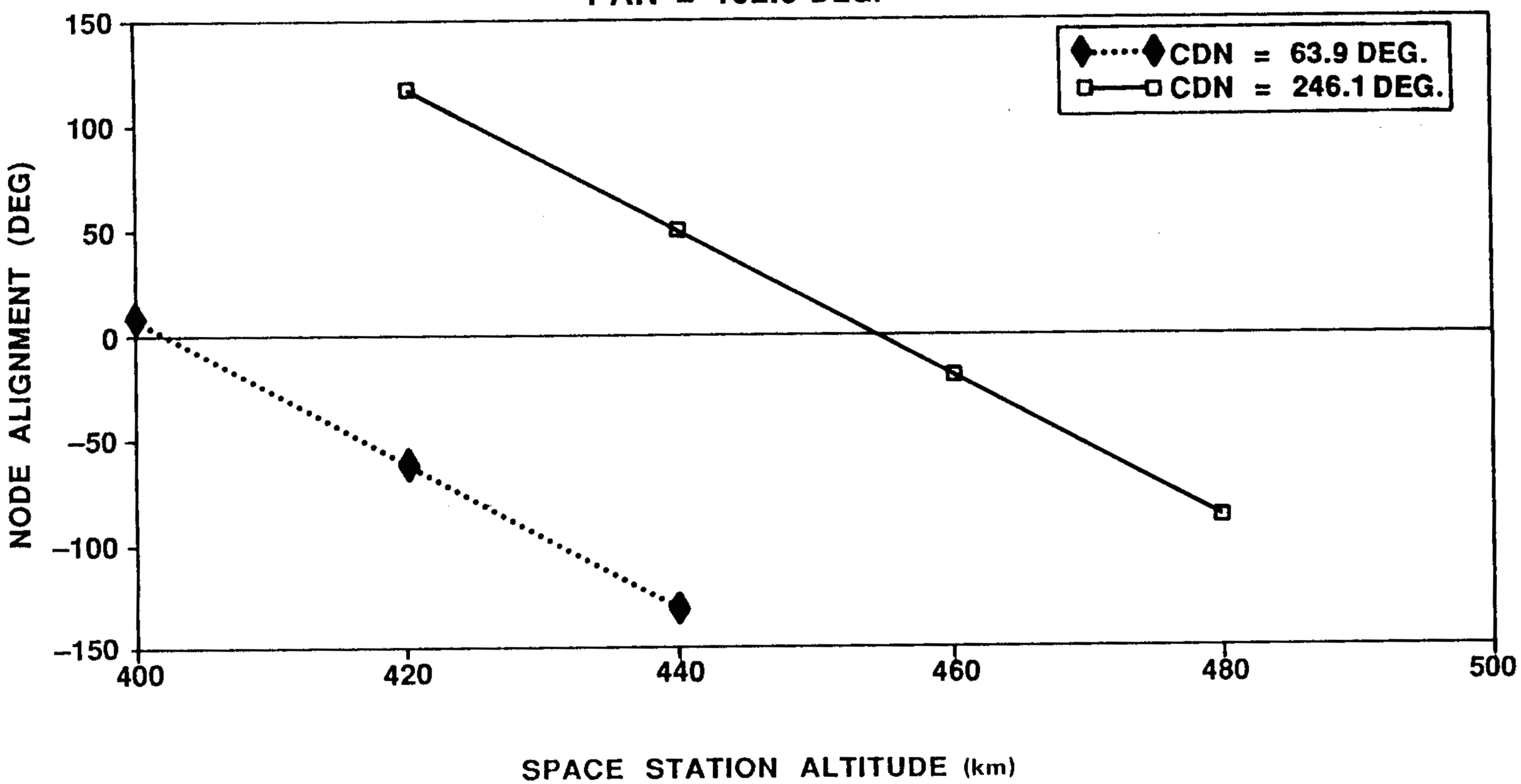
The facing chart shows the nodal alignment of the Space Station orbit relative to the 102.5 degree ascending node of the Earth return orbit of the piloted vehicle. The curve indicates that either a Space Station orbit altitude of approximately 402 km or 455 km is required for the Space Station to be co-planar with the cargo vehicle Earth departure orbit and the piloted vehicle Earth departure and return orbits.

1-4378-8

SPACE STATION/SPRINT NODE ALIGNMENT

CARGO VEHICLE

PAN = 102.5 DEG.



KEY

PAN - RIGHT ASCENSION OF SPACE STATION ASCENDING NODE AT PILOTED VEHICLE ARRIVAL

CDN - RIGHT ASCENSION OF SPACE STATION ASCENDING NODE AT CARGO VEHICLE DEPARTURE

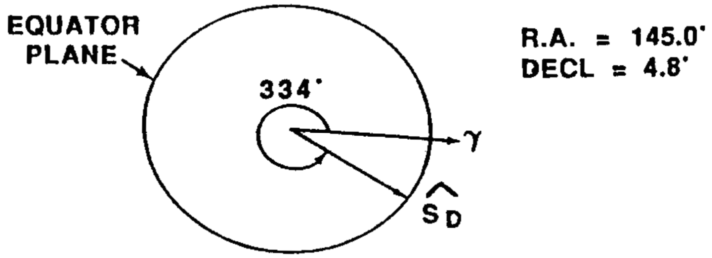
SPACE STATION ORBIT AVAILABILITY
TO SUPPORT 2005 MARS OPPOSITION SPLIT SPRINT MISSION

The piloted vehicle escape vector has a right ascension (R.A.) of 145.0° and a declination of 4.8°; there are two Space Station orbit orientations which would allow in-plane escape conditions for the piloted vehicle. One orientation would have a node of 334° from the vernal equinox (γ) direction and the second orientation would be 136° from (γ). Assuming the piloted vehicle departed from the Space Station orbit with a 334° node, 420 days later for Earth return, the actual existing node of the Space Station orbit would be in the direction of $\vec{\Omega}$ on the left side of the chart under piloted Earth return; however, the required node direction is $\hat{S}_A$. There is a 155.4° or 114.6° node angle misalignment which is shown. Assuming the piloted vehicle departed from the Space Station orbit with a 136° node, the node misalignment is only 6.8° for the piloted Earth return capsule. The lower right hand figures show how much the Space Station orbit will be misaligned for the cargo vehicle launch. This misalignment could be corrected by a 3-impulse escape maneuver or possibly through a deep space or broken plane maneuver on the Earth-to-Mars transfer trajectory. Another way the Space Station orbit node can be corrected is by adjusting the launch or Earth return date since the Space Station node is regressing at the rate of -7.14 degrees per day.

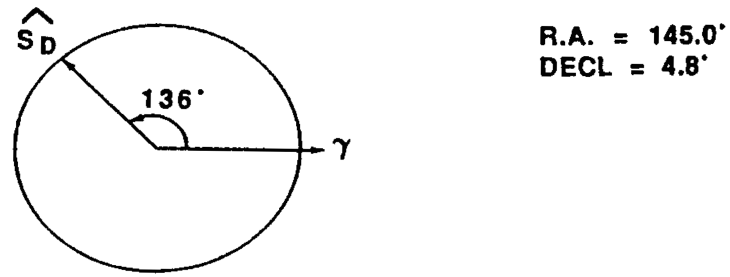
1-4229-8

SPACE STATION ORBIT AVAILABILITY TO SUPPORT 2005 MARS OPPOSITION SPLIT SPRINT MISSION

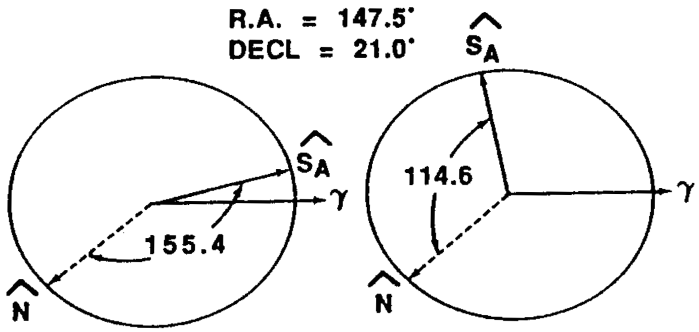
PILOTED VEHICLE LAUNCH



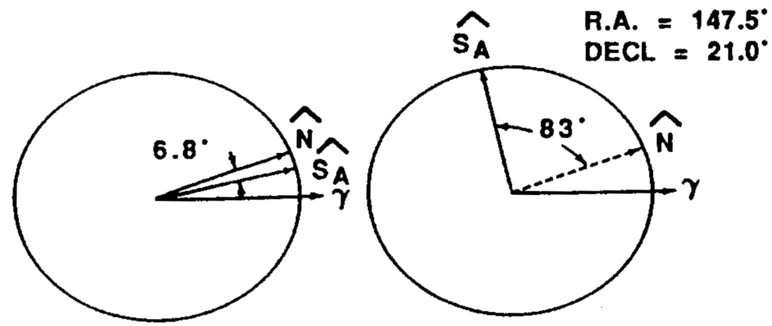
PILOTED VEHICLE LAUNCH



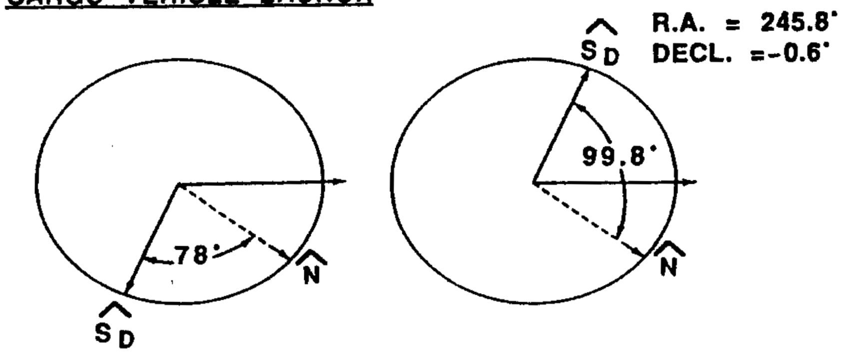
PILOTED EARTH RETURN CAPSULE



PILOTED EARTH RETURN CAPSULE



CARGO VEHICLE LAUNCH



$\hat{S}_D$ SPACE STATION ORBIT NODE REQUIRED FOR EARTH ESCAPE

$\hat{S}_A$ SPACE STATION ORBIT NODE REQUIRED FOR EARTH RETURN

$\hat{N}$ ACTUAL EXISTING NODE RESULTING FROM PILOTED VEHICLE IN-PLANE LAUNCH (TOP FIGURE)

R.A. ESCAPE OR RETURN VECTOR RIGHT ASCENSION

DECL ESCAPE OR RETURN VECTOR DECLINATION

ORBITAL PERIOD VS. ALTITUDE

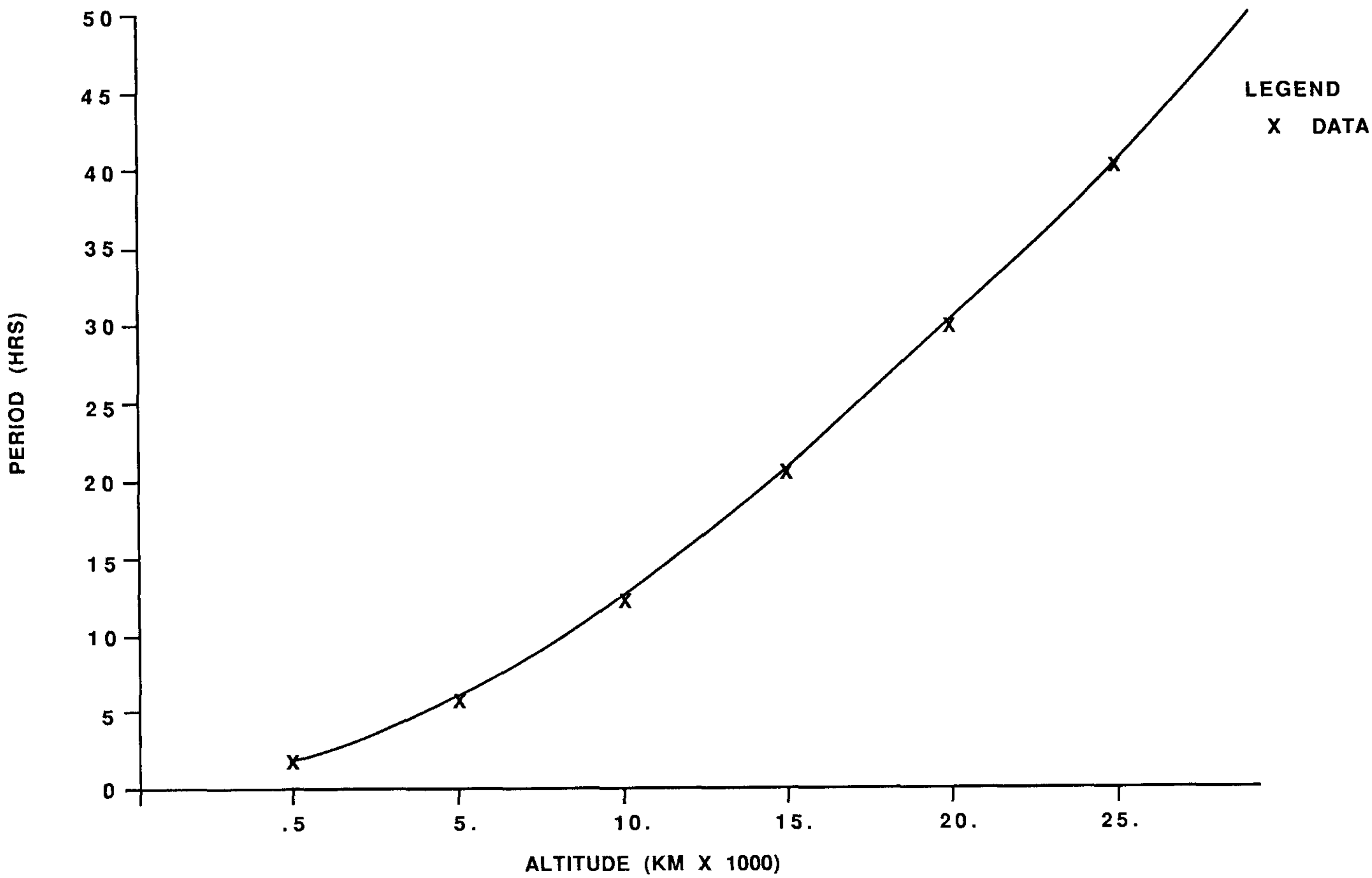
This chart shows the orbital period for circular orbits about Mars. Note that the altitude is in thousands of Km.

A Martian day is 24 hrs 39 min and 35.2 sec; hence, a synchronous orbit is approximately 17,000 km altitude. A 1000 km altitude parking orbit was chosen for this study resulting in a period of approximately 2.5 hours.

1-4066-8-16D

ORBITAL PERIOD VS ALTITUDE

CIRCULAR ORBIT



ORBITAL PERIOD VS. ALTITUDE

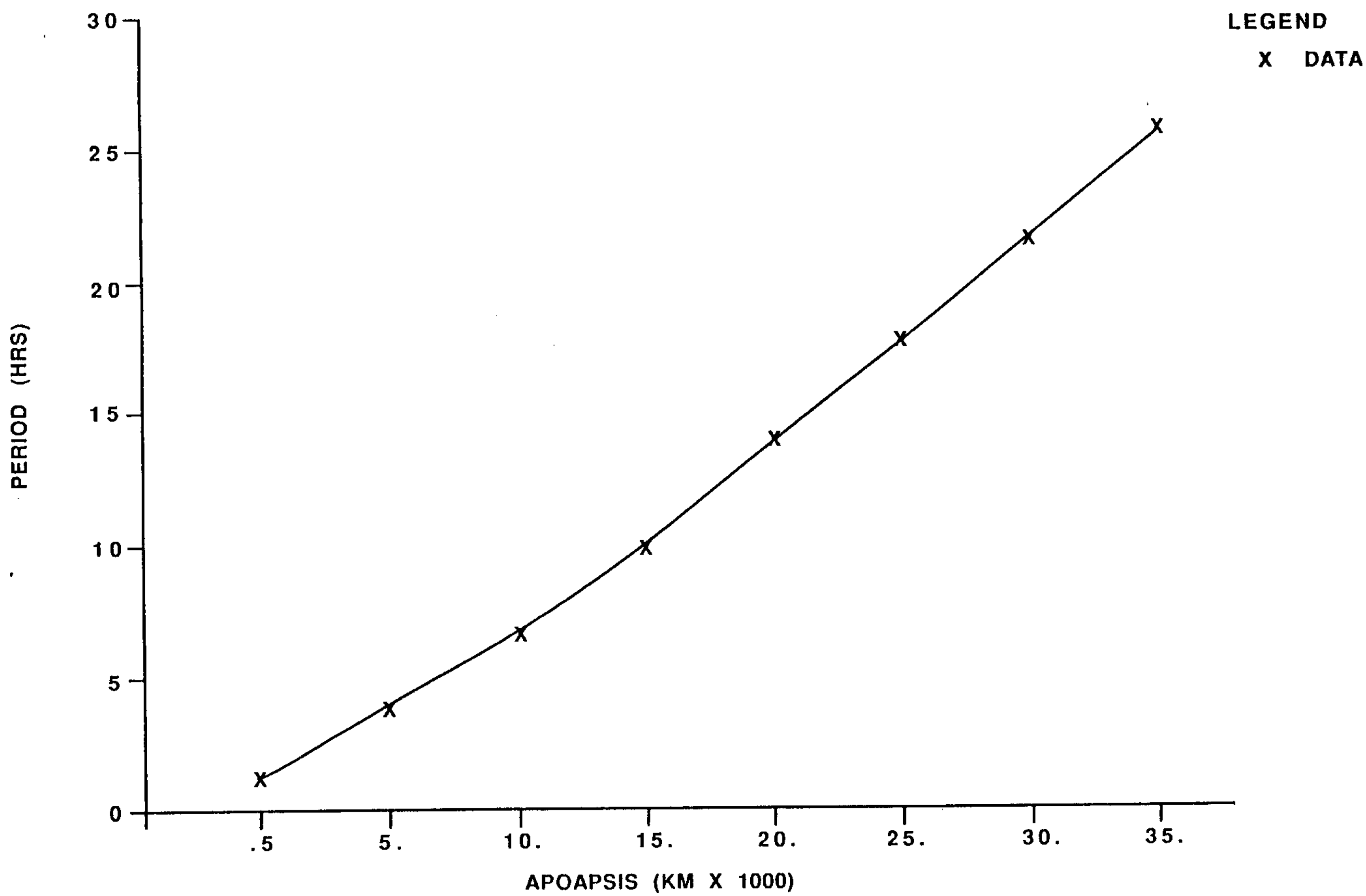
This chart shows the orbital period for elliptic orbits about Mars. The periapsis altitude is constant at 500 Km. The altitude of apoapsis varies from 500 Km to 35,000 Km.

An apoapsis altitude of approximately 33,500 km gives a Mars synchronous orbit. Elliptical parking orbits were considered, but a 10,000 km altitude circular orbit was selected for this study. This simplifies phasing considerations for landing site overflights and vehicle rendezvous.

1-4067-8-16D

ORBITAL PERIOD VS ALTITUDE

PERIAPSIS = 500 KM

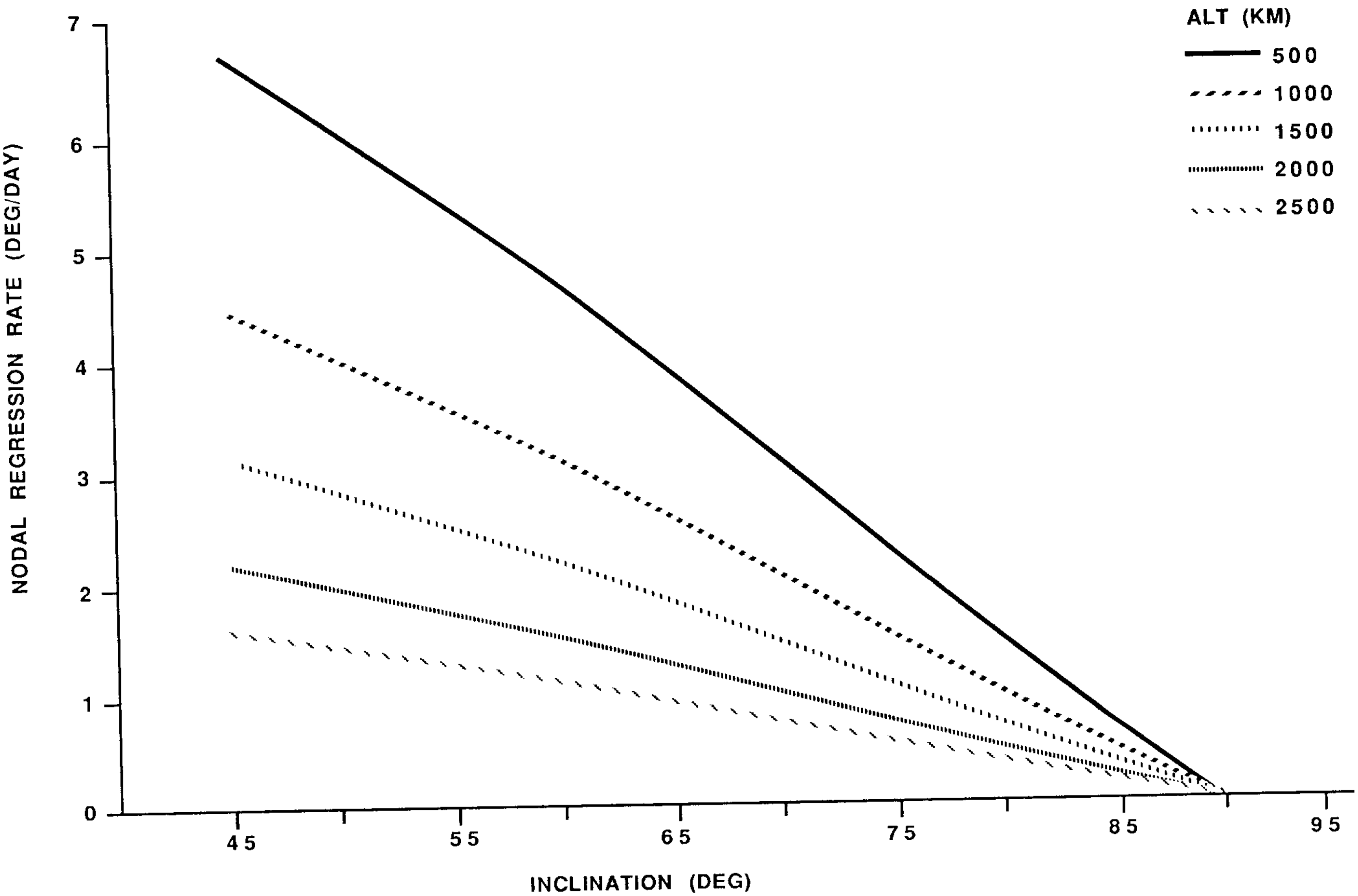


REGRESSION RATE VS. INCLINATION

Like Earth, Mars is an oblate spheroid which causes the nodes of an orbital plane to regress. This chart depicts the nodal regression about Mars as a function of inclination for various orbital altitudes. In comparison with Earth, this chart shows that a 500 kilometer, 45 degree inclination orbit has a regression rate of about 6.8 degrees/day, compared to a 5.4 degrees/day regression rate for an identical Earth orbit.

1-4068-8-16D

REGRESSION RATE VS INCLINATION



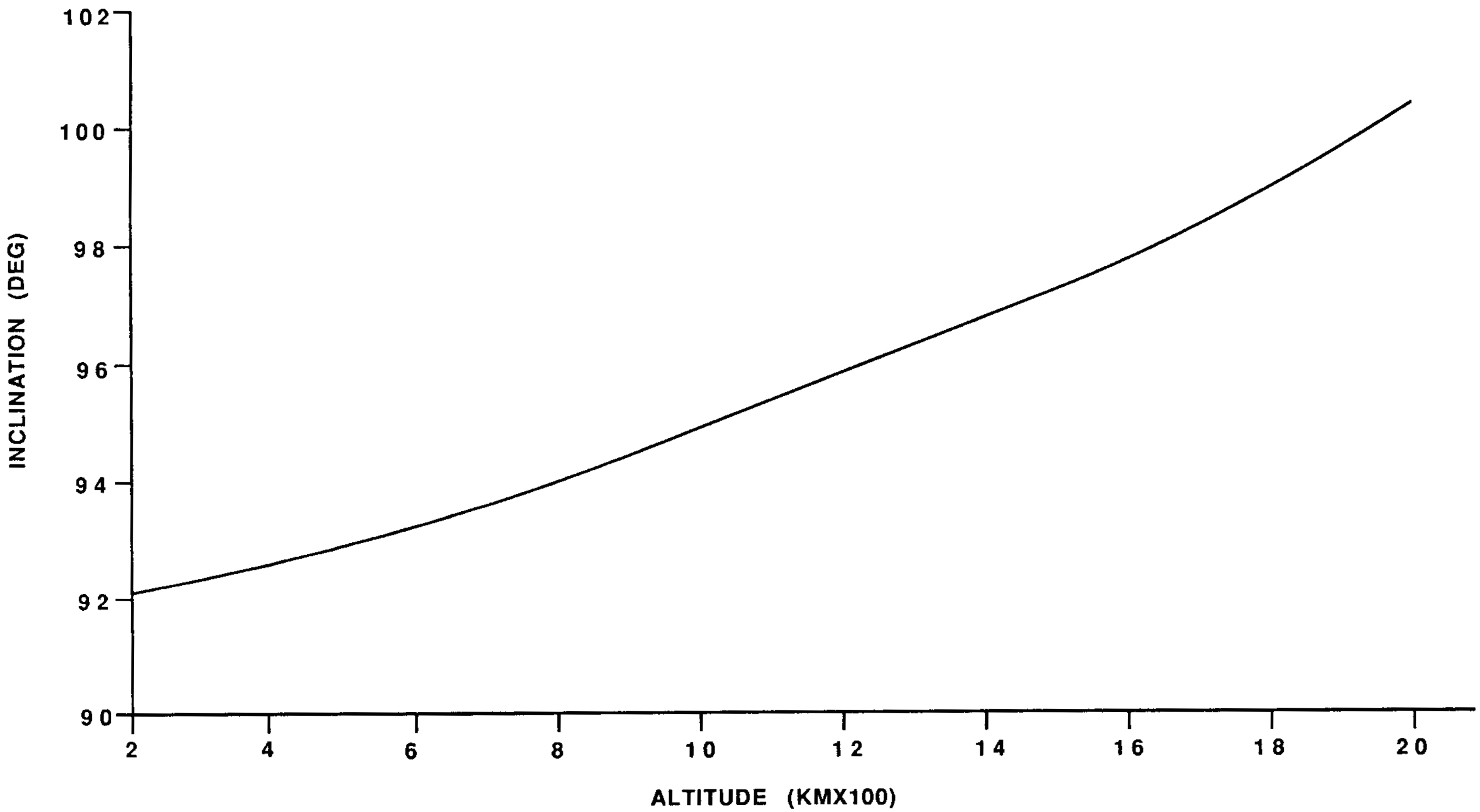
MARS SUN SYNCHRONOUS ORBIT

The orbital inclination is given vs. altitude for a sun synchronous orbit. A sun synchronous orbit does not necessarily provide continuous time in the sun for the spacecraft. The orbit does maintain a constant relationaship with the sun. As Mars orbits about the sun, the sun synchronous orbit must of neccessity precess Eastwardly 360 degrees every 686.979 days (0.524 degrees/day).

There is no requirement at this time for any of the manned Mars vehicles to maintain a sun synchronous orbit. This chart points out that for any Mars orbiter to continuously monitor the sun, it must be placed in a high inclination orbit, and the orientation of the orbit, with respect to the sun, chosen to provide continuous sun viewing.

1-4069-8-16D

MARS SUN SYCHRONOUS ORBIT



MARS ORBITAL LIFETIMES

An analysis was performed to determine the orbital lifetimes for the cargo and piloted vehicles in orbit about Mars. Maximum drag (worst case) orientation was used in making the orbit lifetime calculations. With an initial orbit altitude of 200 km, the orbit altitude decays to 185 km in 22 months for the cargo vehicle and in 24 months for the piloted vehicle. The actual time the cargo vehicle is in orbit around Mars is 20 months, and the piloted vehicle is in orbit around Mars for one month. From this analysis, it was determined that a 200 KM altitude orbit represents the minimum safe parking orbit for the cargo vehicle. Since an altitude of 1000 KM was chosen for this study, there is no orbital decay problem for the cargo or piloted vehicles.

MARS ORBITAL LIFETIMES

CARGO VEHICLE :

AEROBRAKE SHIELD SURFACE PERPENDICULAR TO VELOCITY VECTOR

BALLISTIC COEFF. = $WGT/C_D A$ = 170 LBS/FT^2

200 KM ORBIT DECAYS TO 185 KM IN 22 MONTHS

PILOTED VEHICLE :

AEROBRAKE SHIELD SURFACE PERPENDICULAR TO VELOCITY VECTOR

BALLISTIC COEFF. = $WGT/C_D A$ = 185 LBS/FT^2

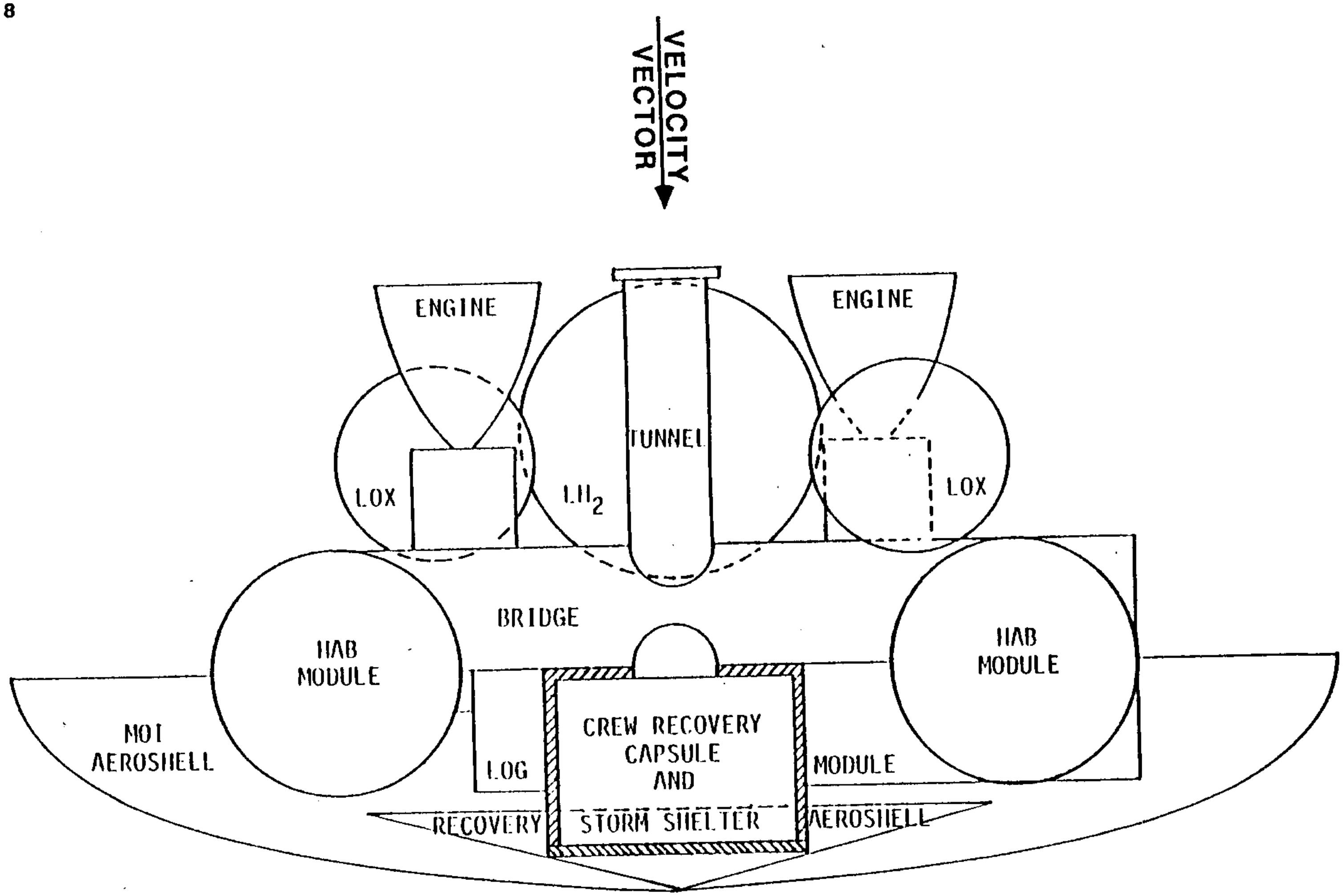
200 KM ORBIT DECAYS TO 185 KM IN 24 MONTHS

- HIGHER ORBITS (500 - 1000 KM) HAVE SIGNIFICANTLY LONGER ORBITAL LIFETIMES.

PILOTED VEHICLE-SIDE VIEW

This schematic depicts the piloted vehicle and shows the orientation of the aeroshell with respect to the velocity vector. This orientation (maximum drag) was used in making the orbital lifetime calculations.

1-4388-8



FROM REFERENCE 22

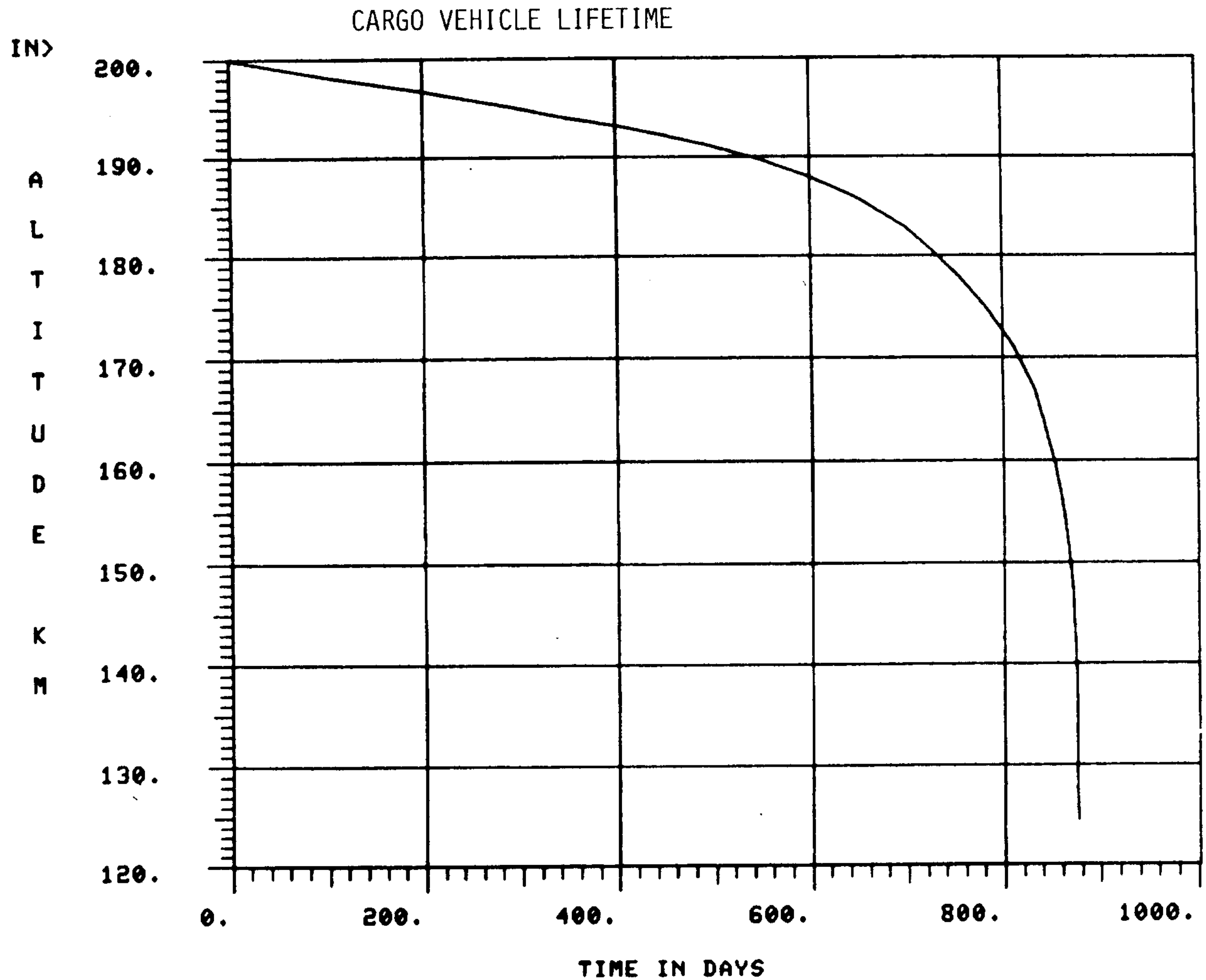
PILOTED VEHICLE SIDE VIEW

CARGO VEHICLE LIFETIME

This figure presents the predicted orbital decay profile, based on the maximum drag orientation of the cargo vehicle.

A nominal Mars atmospheric model, with no solar variations, was used to predict the orbital lifetimes for missions in orbit around Mars.

Initial orbital altitudes around Mars lower than 200 KM would compromise the cargo vehicle planned orbital stay time of 19 months for the maximum drag orientation.


CARGO VEHICLE LIFETIME
IN>
ALTITUDE KM
200.
190.
180.
170.
160.
150.
140.
130.
120.
0.
200.
400.
600.
800.
1000.
TIME IN DAYS

CARGO VEHICLE PROPELLANT USED AT MARS

A system trade was conducted to determine the most efficient parking orbit for the cargo vehicle so that it would be in the correct orbit plane upon arrival of the piloted vehicle at Mars. Shown on the facing chart are orbits that satisfy this requirement, together with the propellant required to inject into this orbit, and upon arrival of the piloted vehicle to transfer to a 980 km altitude circular orbit, for rendezvous with the piloted vehicle. An elliptical parking orbit with a 4381 km apoapsis altitude and a 300 km periapsis altitude, used in conjunction with aerobraking, provides the most efficient orbit of those studied.

1-3734-8-12T

MARS PARKING ORBIT TRADE STUDY
CARGO VEHICLE

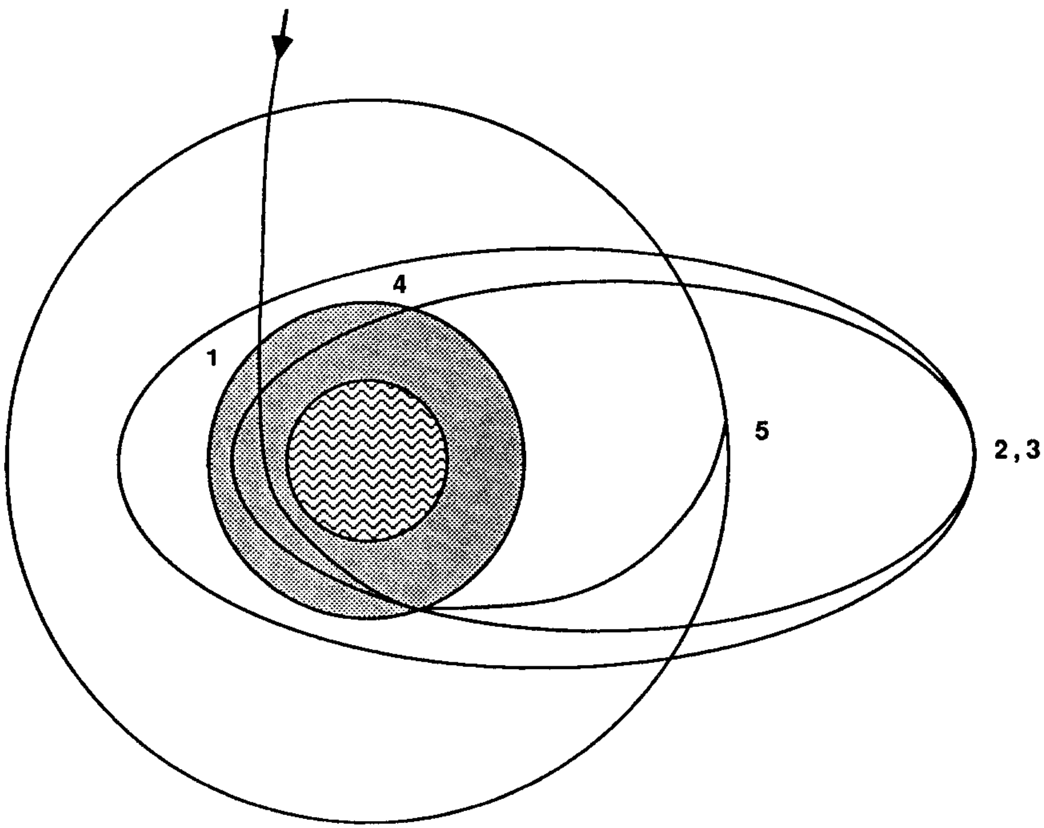
CASE NO.	PARKING ORBIT ALTITUDE (KM)	ORBIT TRANSFER MODE	TOTAL PROPELLANT REQUIRED AT MARS (LB.)
1.	2062 X 2062	ALL PROPULSIVE	74,000
2.	2062 X 2062	AEROBRAKE + PROPULSIVE	94,000
3.	3406 X 980	ALL PROPULSIVE	52,000
4.	4841 X 300	AEROBRAKE + PROPULSIVE	31,000

CARGO VEHICLE MISSION PROFILE

The orbital profile of the cargo vehicle is shown in the facing chart. The Mars arrival trajectory is guided to a virtual periapsis altitude of 47.4 km. Aerobraking is then used to lower the energy to a 4381.3 km apoapsis orbit. The space vehicle then coasts to the apoapsis and does a propulsive burn to raise the periapsis altitude to 300 km, which is above the sensible atmosphere. After a stay time of 522 days, the cargo vehicle uses a propulsive maneuver to lower the periapsis altitude to 56.2 km, and aerobraking is used to lower the apoapsis altitude to 980 km. The vehicle coasts to apoapsis and uses an additional propulsive maneuver to circularize at 980 km altitude. At this point, the vehicle is in position for terminal rendezvous with the piloted vehicle.

1-4081-8-16D

MARS MISSION CARGO VEHICLE



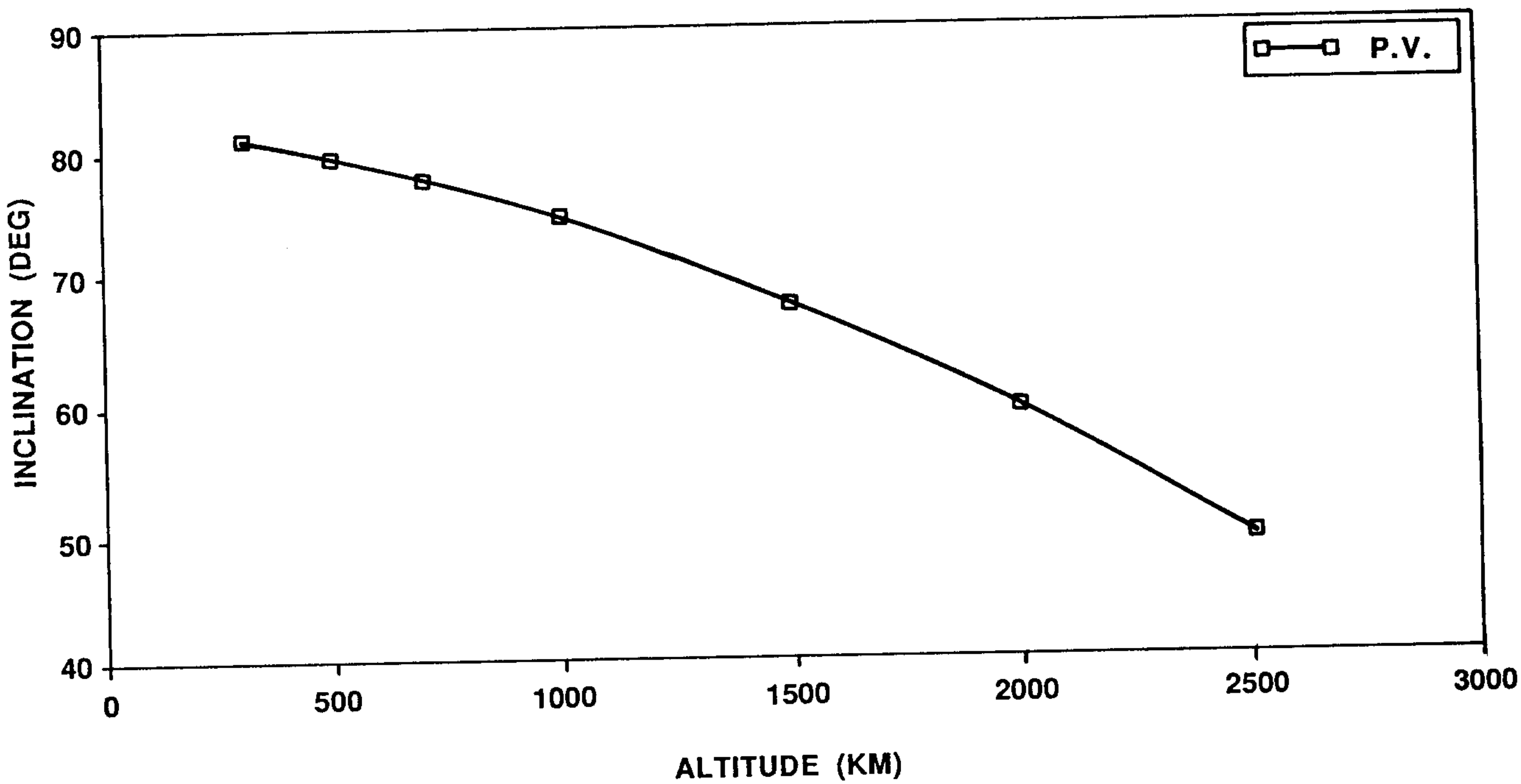
EVENT	TIME (DAYS/HRS)	EVENT DESCRIPTION	WEIGHT
1	0/0	AEROBRAKE TO 4381.3 X 47.4 KM	511,742
2	0/1.952	RAISE PERIAPSIS TO 300 KM	501,332
3	522/1.95	LOWER PERIAPSIS TO 56.2 KM	494,868
4	522/3.76	AEROBRAKE TO 980 X 56.2 KM	494,868
5	522/4.85	CIRCULARIZE AT 980 KM	467,283

SPRINT PARKING ORBIT ANALYSIS

The curve on the facing page shows altitude/inclination combination for the parking orbit of the piloted vehicle that is aligned with both the arrival and departure asymptotes, assuming a 30 day staytime for the piloted spacecraft. An altitude of 1000 km, with a corresponding inclination of 74.7 deg, was selected as the reference orbit for further analyses.

MARS MISSION ANALYSIS

SPRINT PARKING ORBIT ANALYSIS

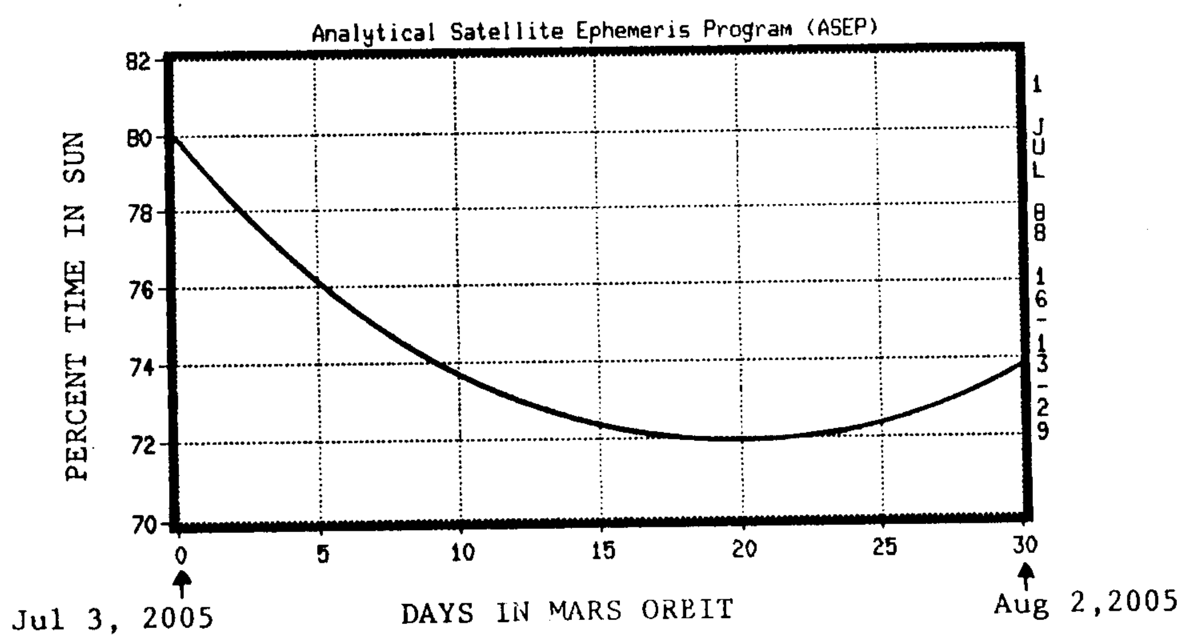


PERCENT TIME IN THE SUN DURING MARS ORBIT

This chart shows the precent of an orbital revolution that the vehicle spends in sunlight. The circular orbit has a 1000 km altitude, 74.73° inclination, and 292.4° right ascension of the ascending node. After approximately 20 days in Mars orbit, the orbit plane has precessed to an orientation of zero BETA angle and experiences minimum sunlight time per revolution. The percent time in sunlight would influence the design of the power and thermal subsystems.

PERCENT TIME IN THE SUN DURING MARS ORBIT

Altitude: 1000 Km   Incl: 74.73°

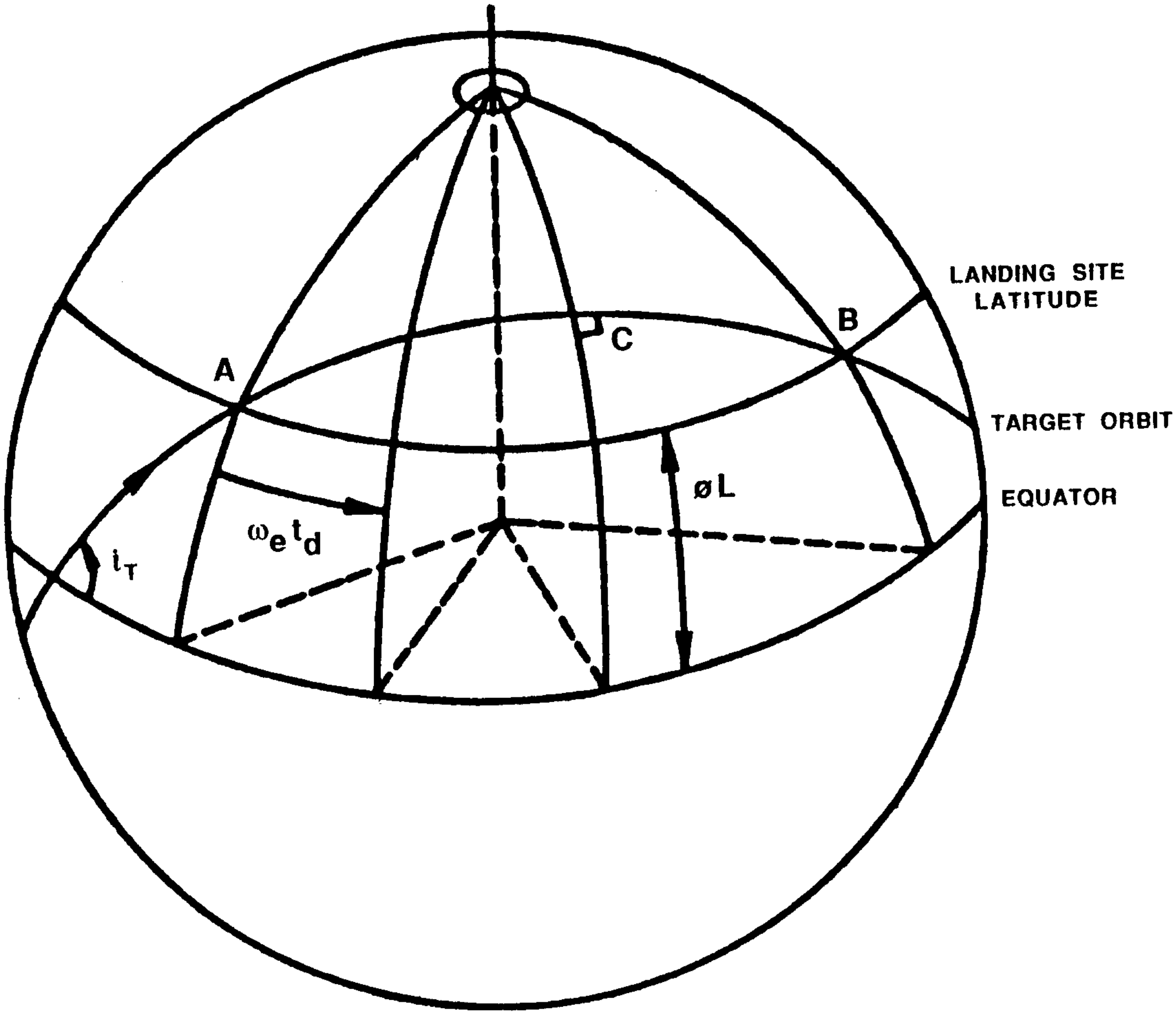


ORBIT AND LANDING SITE GEOMETRY

In-plane deboost and landing opportunities from the Mars orbit occur twice every Martian day (24.6 hours), where the landing latitude is less than the orbit inclination. One landing opportunity occurs from an ascending direction, South to North, as illustrated by position "A" on the figure. The second landing opportunity occurs from a descending direction, North to South, as illustrated by position "B" on the figure. If the landing spacecraft had cross range maneuvering capability, there could be more than two opportunities per day since the orbit would not have to pass directly over the landing site. If the landing site and orbiting spacecraft are not phased appropriately, an intermediate phasing orbit is required to achieve the desired landing site.

1-4228-8

ORBIT AND LANDING SITE GEOMETRY



MARS EXPLORATION
RENDEZVOUS COMPATIBLE ORBIT

One of the factors that will influence the Mars landing site and parking orbit selection will be the ability to perform an efficient rendezvous between the Mars spacecraft and the ascent vehicle when it returns from the surface. The parking orbit must be compatible with the landing site so as to enable repeated opportunities for the return of the crew. Figure 1 on the chart shows the three phases of the rendezvous sequence. The timing of the launch requires that the launch site be in the plane of the parking orbit. This is possible only if the latitude of the landing site is less than the inclination of the parking orbit. In addition to being launched into the proper orbital plane, the ascent vehicle must eventually end up in the same location as the piloted vehicle in its orbit. Figure 2 shows two ways in which this could be accomplished. The first is to launch the ascent vehicle when the landing site is in the orbit plane and the piloted vehicle is at a particular location in the orbit so that the two vehicles will arrive at the same location at the same time. The second option for the rendezvous would be to launch into an intermediate orbit that is co-planar with the parking orbit, remain in the intermediate orbit until the two spacecraft are phased properly, then transfer to the parking orbit for the rendezvous. The first rendezvous profile is the most efficient, but requires careful synchronization of the motion of the parking orbit with the rotation of Mars.

1-4224-8

MARS EXPLORATION
RENDEZVOUS COMPATIBLE ORBIT

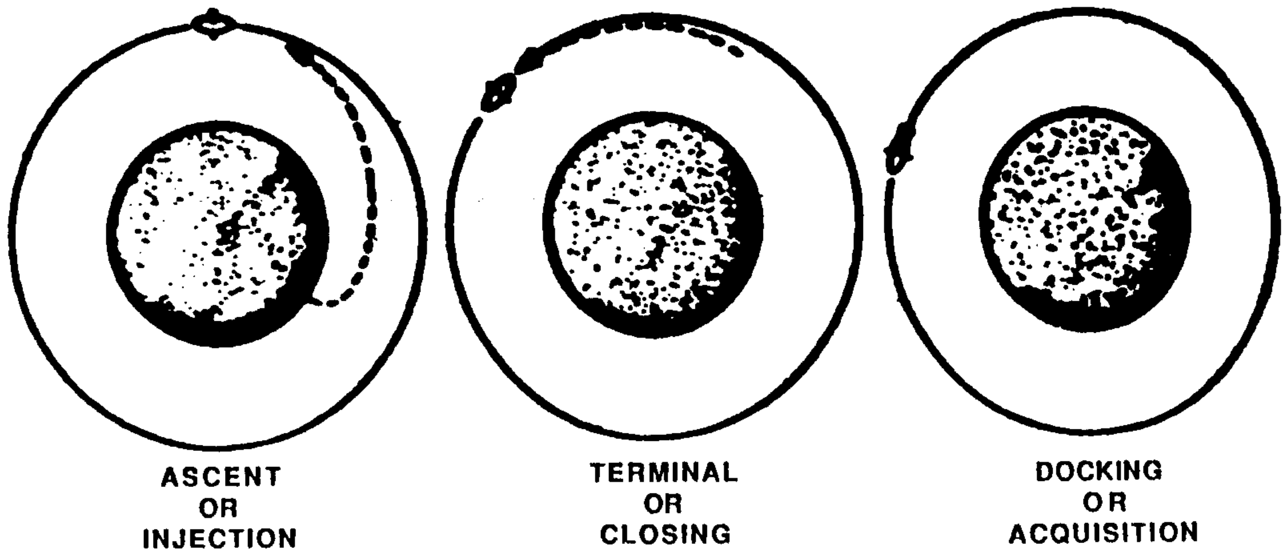


FIG. 1. PHASES OF RENDEZVOUS

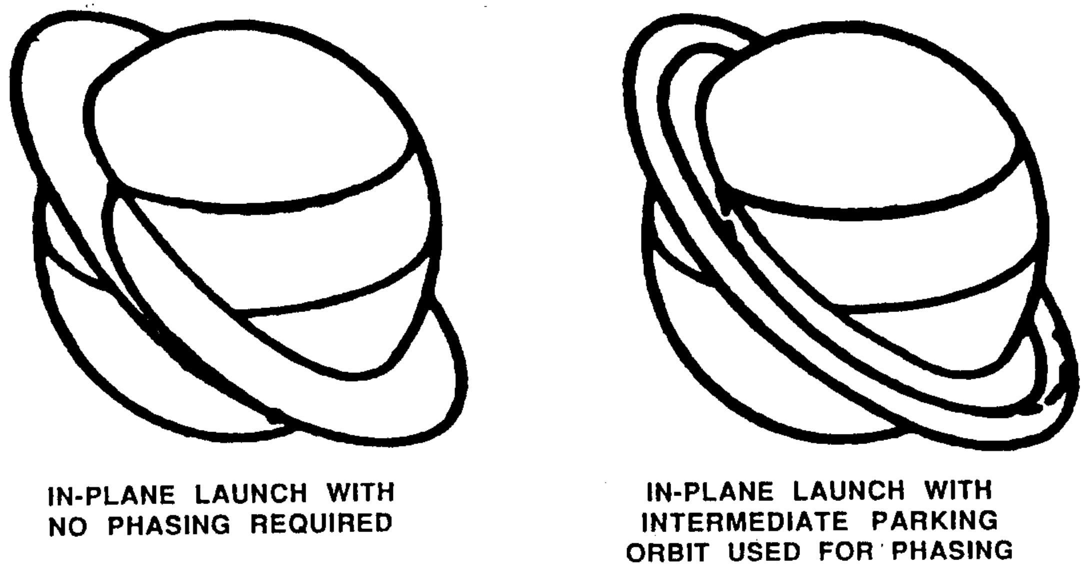


FIG 2. ASCENT/RENDEZVOUS TRAJECTORIES

RENDEZVOUS COMPATIBLE MARS ORBITS
MARTIAN DAILY REPEATING ORBITS

A rendezvous compatible orbit allows a spacecraft to repeatedly pass over a particular landing site so that descents and ascents between the spacecraft and the landing site are possible. Rendezvous conditions between a landing site and an orbiting spacecraft exist only when the landing site is in the orbit plane and the spacecraft is at a specific point in its orbit. To ensure that these conditions repeat at regular intervals, it is necessary to choose an orbit in which the orbital motion of the spacecraft is synchronized with the rotation of the planet. The altitude and inclination of the orbit determine the orbital velocity of the spacecraft, orbit period, and the precession rate of the orbit plane. By selecting the proper altitude and inclination, it is possible to achieve the synchronization that allows periodic rendezvous opportunities.

The next eleven pages show the proper combinations of altitude and inclination that will assure descent or ascent opportunities at the same time every day or every other day (Martian day, 24.6 hours). In these charts "M" is equal to the number of Mars rotations and represents the desired frequency of rendezvous opportunities. "N" is equal to the number of orbits completed by the spacecraft during "M" rotations of Mars. The synchronization is achieved by selecting the proper combination of altitude and inclination such that the time required for "N" orbital periods is equal to the time required for "M" rotations of Mars.

MARS EXPLORATION

```
*********************************************************
*                                                       *
*              RENDEZVOUS COMPATIBLE ORBIT              *
*            MARTIAN DAILY REPEATING ORBITS             *
*********************************************************
```

INPUT DATA

MU = 42828.3 J2 = 0.29400E-02 PP = 88643.000 R = 3397.200

CALCULATIONS

M	N	INCL (DEG)	ALTITUDE (KM)
1	1	0.0000	17032.7422
1	1	10.0000	17032.6758
1	1	20.0000	17032.4844
1	1	30.0000	17032.1992
1	1	40.0000	17031.8574
1	1	50.0000	17031.5137
1	1	60.0000	17031.2188
1	1	70.0000	17031.0254
1	1	80.0000	17030.9727
1	1	90.0000	17031.0820
1	1	100.0000	17031.3574
1	1	110.0000	17031.7832
1	1	120.0000	17032.3262
1	1	130.0000	17032.9355
1	1	140.0000	17033.5527
1	1	150.0000	17034.1152
1	1	160.0000	17034.5645
1	1	170.0000	17034.8555
1	1	180.0000	17034.9551
1	2	0.0000	9473.2207
1	2	10.0000	9473.1416
1	2	20.0000	9472.9180
1	2	30.0000	9472.5928
1	2	40.0000	9472.2266
1	2	50.0000	9471.8965
1	2	60.0000	9471.6816
1	2	70.0000	9471.6523
1	2	80.0000	9471.8633
1	2	90.0000	9472.3408
1	2	100.0000	9473.0840
1	2	110.0000	9474.0566
1	2	120.0000	9475.1953
1	2	130.0000	9476.4141
1	2	140.0000	9477.6084
1	2	150.0000	9478.6758
1	2	160.0000	9479.5176
1	2	170.0000	9480.0576
1	2	180.0000	9480.2432

M = NUMBER OF MARS' ROTATIONS

N = NUMBER OF SATELLITE REVOLUTIONS

MARS EXPLORATION

```
****************************************************
*                                                  *
*           RENDEZVOUS COMPATIBLE ORBIT            *
*          MARTIAN DAILY REPEATING ORBITS          *
****************************************************
```

INPUT DATA

MU = 42828.3 J2 = 0.29400E-02 PP = 88643.000 R = 3397.200

M	N	INCL	ALTITUDE
1	5	0.0000	3582.4817
1	5	10.0000	3582.4839
1	5	20.0000	3582.5127
1	5	30.0000	3582.6306
1	5	40.0000	3582.9319
1	5	50.0000	3583.5264
1	5	60.0000	3584.5227
1	5	70.0000	3586.0088
1	5	80.0000	3588.0356
1	5	90.0000	3590.6033
1	5	100.0000	3593.6536
1	5	110.0000	3597.0708
1	5	120.0000	3600.6868
1	5	130.0000	3604.2949
1	5	140.0000	3607.6682
1	5	150.0000	3610.5801
1	5	160.0000	3612.8257
1	5	170.0000	3614.2429
1	5	180.0000	3614.7273
1	6	0.0000	2778.3647
1	6	10.0000	2778.4238
1	6	20.0000	2778.6240
1	6	30.0000	2779.0315
1	6	40.0000	2779.7434
1	6	50.0000	2780.8728
1	6	60.0000	2782.5278
1	6	70.0000	2784.7905
1	6	80.0000	2787.6997
1	6	90.0000	2791.2351
1	6	100.0000	2795.3110
1	6	110.0000	2799.7764
1	6	120.0000	2804.4226
1	6	130.0000	2809.0005
1	6	140.0000	2813.2397
1	6	150.0000	2816.8728
1	6	160.0000	2819.6609
1	6	170.0000	2821.4148
1	6	180.0000	2822.0132

M = NUMBER OF MARS' ROTATIONS

N = NUMBER OF SATELLITE REVOLUTIONS

MARS EXPLORATION

****
*   *
*   RENDEZVOUS COMPATIBLE ORBIT   *
*   MARTIAN DAILY REPEATING ORBITS   *
****

INPUT DATA

MU = 42828.3 J2 = 0.29400E-02 PP = 88643.000 R = 3397.200

M	N	INCL	ALTITUDE
1	7	0.0000	2168.8359
1	7	10.0000	2168.9646
1	7	20.0000	2169.3752
1	7	30.0000	2170.1343
1	7	40.0000	2171.3406
1	7	50.0000	2173.1055
1	7	60.0000	2175.5322
1	7	70.0000	2178.6924
1	7	80.0000	2182.6069
1	7	90.0000	2187.2307
1	7	100.0000	2192.4456
1	7	110.0000	2198.0620
1	7	120.0000	2203.8289
1	7	130.0000	2209.4521
1	7	140.0000	2214.6177
1	7	150.0000	2219.0183
1	7	160.0000	2222.3809
1	7	170.0000	2224.4900
1	7	180.0000	2225.2090
1	8	0.0000	1687.2762
1	8	10.0000	1687.4878
1	8	20.0000	1688.1462
1	8	30.0000	1689.3181
1	8	40.0000	1691.1000
1	8	50.0000	1693.5983
1	8	60.0000	1696.9065
1	8	70.0000	1701.0807
1	8	80.0000	1706.1189
1	8	90.0000	1711.9459
1	8	100.0000	1718.4070
1	8	110.0000	1725.2708
1	8	120.0000	1732.2413
1	8	130.0000	1738.9789
1	8	140.0000	1745.1255
1	8	150.0000	1750.3345
1	8	160.0000	1754.2996
1	8	170.0000	1756.7806
1	8	180.0000	1757.6252

M = NUMBER OF MARS' ROTATIONS

N = NUMBER OF SATELLITE REVOLUTIONS

MARS EXPLORATION

RENDEZVOUS COMPATIBLE ORBIT
MARTIAN DAILY REPEATING ORBITS

INPUT DATA

MU = 42828.3 J2 = 0.29400E-02 PP = 88643.000 R = 3397.200

CALCULATIONS

M	N	INCL (DEG)	ALTITUDE (KM)
1	9	0.0000	1294.7775
1	9	10.0000	1295.0846
1	9	20.0000	1296.0291
1	9	30.0000	1297.6748
1	9	40.0000	1300.1134
1	9	50.0000	1303.4423
1	9	60.0000	1307.7404
1	9	70.0000	1313.0427
1	9	80.0000	1319.3192
1	9	90.0000	1326.4597
1	9	100.0000	1334.2689
1	9	110.0000	1342.4702
1	9	120.0000	1350.7212
1	9	130.0000	1358.6355
1	9	140.0000	1365.8116
1	9	150.0000	1371.8646
1	9	160.0000	1376.4563
1	9	170.0000	1379.3231
1	9	180.0000	1380.2979
1	10	0.0000	966.9578
1	10	10.0000	967.3736
1	10	20.0000	968.6427
1	10	30.0000	970.8248
1	10	40.0000	974.0026
1	10	50.0000	978.2603
1	10	60.0000	983.6569
1	10	70.0000	990.2039
1	10	80.0000	997.8294
1	10	90.0000	1006.3693
1	10	100.0000	1015.6434
1	10	110.0000	1025.2666
1	10	120.0000	1034.8679
1	10	130.0000	1044.0148
1	10	140.0000	1052.2629
1	10	150.0000	1059.1903
1	10	160.0000	1064.4286
1	10	170.0000	1067.6925
1	10	180.0000	1068.8010

M = NUMBER OF MARS' ROTATIONS

N = NUMBER OF SATELLITE REVOLUTIONS

MARS EXPLORATION

****

RENDEZVOUS COMPATIBLE ORBIT
MARTIAN DAILY REPEATING ORBITS

****

INPUT DATA

MU = 42828.3 J2 = 0.29400E-02 PP = 88643.000 R = 3397.200

M	N	INCL	ALTITUDE
1	11	0.0000	687.6889
1	11	10.0000	688.2272
1	11	20.0000	689.8617
1	11	30.0000	692.6450
1	11	40.0000	696.6476
1	11	50.0000	701.9350
1	11	60.0000	708.5406
1	11	70.0000	716.4401
1	11	80.0000	725.5300
1	11	90.0000	735.6153
1	11	100.0000	746.4069
1	11	110.0000	757.5305
1	11	120.0000	768.5461
1	11	130.0000	778.9748
1	11	140.0000	788.3312
1	11	150.0000	796.1580
1	11	160.0000	802.0591
1	11	170.0000	805.7288
1	11	180.0000	806.9740
1	12	0.0000	445.8335
1	12	10.0000	446.5090
1	12	20.0000	448.5520
1	12	30.0000	452.0053
1	12	40.0000	456.9226
1	12	50.0000	463.3448
1	12	60.0000	471.2733
1	12	70.0000	480.6440
1	12	80.0000	491.3076
1	12	90.0000	503.0190
1	12	100.0000	515.4364
1	12	110.0000	528.1337
1	12	120.0000	540.6211
1	12	130.0000	552.3745
1	12	140.0000	562.8693
1	12	150.0000	571.6155
1	12	160.0000	578.1915
1	12	170.0000	582.2734
1	12	180.0000	583.6572

M = NUMBER OF MARS' ROTATIONS

N = NUMBER OF SATELLITE REVOLUTIONS

MARS EXPLORATION

```
**************************************************
*                                                *
*           RENDEZVOUS COMPATIBLE ORBIT          *
*         MARTIAN DAILY REPEATING ORBITS         *
**************************************************
```

INPUT DATA

MU = 42828.3 J2 = 0.29400E-02 PP = 88643.000 R = 3397.200

CALCULATIONS

M	N	INCL (DEG)	ALTITUDE (KM)
1	13	0.0000	233.4234
1	13	10.0000	234.2523
1	13	20.0000	236.7506
1	13	30.0000	240.9481
1	13	40.0000	246.8761
1	13	50.0000	254.5443
1	13	60.0000	263.9142
1	13	70.0000	274.8746
1	13	80.0000	287.2239
1	13	90.0000	300.6612
1	13	100.0000	314.7892
1	13	110.0000	329.1280
1	13	120.0000	343.1386
1	13	130.0000	356.2534
1	13	140.0000	367.9109
1	13	150.0000	377.5914
1	13	160.0000	384.8506
1	13	170.0000	389.3487
1	13	180.0000	390.8723
1	14	0.0000	44.5863
1	14	10.0000	45.5860
1	14	20.0000	48.5912
1	14	30.0000	53.6141
1	14	40.0000	60.6569
1	14	50.0000	69.6900
1	14	60.0000	80.6262
1	14	70.0000	93.2990
1	14	80.0000	107.4472
1	14	90.0000	122.7091
1	14	100.0000	138.6287
1	14	110.0000	154.6717
1	14	120.0000	170.2508
1	14	130.0000	184.7573
1	14	140.0000	197.5961
1	14	150.0000	208.2211
1	14	160.0000	216.1683
1	14	170.0000	221.0843
1	14	180.0000	222.7481

M = NUMBER OF MARS' ROTATIONS

N = NUMBER OF SATELLITE REVOLUTIONS

MARS EXPLORATION

```
**************************************************
*                                                *
*          RENDEZVOUS COMPATIBLE ORBIT           *
*                                                *
**************************************************
```

INPUT DATA

MU = 42828.3 J2 = 0.29400E-02 PP = 88643.000 R = 3397.200

M	N	INCL	ALTITUDE
2	11	0.0000	3150.2185
2	11	10.0000	3150.2473
2	11	20.0000	3150.3569
2	11	30.0000	3150.6116
2	11	40.0000	3151.1077
2	11	50.0000	3151.9568
2	11	60.0000	3153.2678
2	11	70.0000	3155.1270
2	11	80.0000	3157.5791
2	11	90.0000	3160.6152
2	11	100.0000	3164.1638
2	11	110.0000	3168.0916
2	11	120.0000	3172.2109
2	11	130.0000	3176.2942
2	11	140.0000	3180.0925
2	11	150.0000	3183.3589
2	11	160.0000	3185.8716
2	11	170.0000	3187.4546
2	11	180.0000	3187.9951
2	13	0.0000	2454.3228
2	13	10.0000	2454.4150
2	13	20.0000	2454.7156
2	13	30.0000	2455.2910
2	13	40.0000	2456.2400
2	13	50.0000	2457.6748
2	13	60.0000	2459.7017
2	13	70.0000	2462.3987
2	13	80.0000	2465.7954
2	13	90.0000	2469.8604
2	13	100.0000	2474.4919
2	13	110.0000	2479.5203
2	13	120.0000	2484.7161
2	13	130.0000	2489.8076
2	13	140.0000	2494.5027
2	13	150.0000	2498.5142
2	13	160.0000	2501.5857
2	13	170.0000	2503.5151
2	13	180.0000	2504.1731

M = NUMBER OF MARS' ROTATIONS

N = NUMBER OF SATELLITE REVOLUTIONS

MARS EXPLORATION

RENDEZVOUS COMPATIBLE ORBIT

INPUT DATA

MU = 42828.3 J2 = 0.29400E-02 PP = 88643.000 R = 3397.200

M	N	INCL	ALTITUDE
2	15	0.0000	1914.9379
2	15	10.0000	1915.1064
2	15	20.0000	1915.6362
2	15	30.0000	1916.5941
2	15	40.0000	1918.0780
2	15	50.0000	1920.1975
2	15	60.0000	1923.0514
2	15	70.0000	1926.7042
2	15	80.0000	1931.1661
2	15	90.0000	1936.3774
2	15	100.0000	1942.2024
2	15	110.0000	1948.4308
2	15	120.0000	1954.7896
2	15	130.0000	1960.9618
2	15	140.0000	1966.6113
2	15	150.0000	1971.4114
2	15	160.0000	1975.0719
2	15	170.0000	1977.3652
2	15	180.0000	1978.1462
2	17	0.0000	1481.6647
2	17	10.0000	1481.9224
2	17	20.0000	1482.7190
2	17	30.0000	148[illegible].1201
2	17	40.0000	1485.2201
2	17	50.0000	1489.1216
2	17	60.0000	1492.9111
2	17	70.0000	1497.6353
2	17	80.0000	1503.2784
2	17	90.0000	1509.7487
2	17	100.0000	1516.8713
2	17	110.0000	1524.3931
2	17	120.0000	1531.9946
2	17	130.0000	1539.3132
2	17	140.0000	1545.9689
2	17	150.0000	1551.5958
2	17	160.0000	1555.8713
2	17	170.0000	1558.5437
2	17	180.0000	1559.4529

M = NUMBER OF MARS' ROTATIONS

N = NUMBER OF SATELLITE REVOLUTIONS

MARS EXPLORATION

```
**************************************************
*                                                *
*           RENDEZVOUS COMPATIBLE ORBIT          *
*                                                *
**************************************************
```

INPUT DATA

MU = 42828.3 J2 = 0.29400E-02 PP = 88643.000 R = 3397.200

M	N	INCL	ALTITUDE
2	19	0.0000	1123.9365
2	19	10.0000	1124.2963
2	19	20.0000	1125.3981
2	19	30.0000	1127.3041
2	19	40.0000	1130.1018
2	19	50.0000	1133.8827
2	19	60.0000	1138.7163
2	19	70.0000	1144.6254
2	19	80.0000	1151.5634
2	19	90.0000	1159.4010
2	19	100.0000	1167.9207
2	19	110.0000	1176.8228
2	19	120.0000	1185.7406
2	19	130.0000	1194.2645
2	19	140.0000	1201.9718
2	19	150.0000	1208.4585
2	19	160.0000	1213.3713
2	19	170.0000	1216.4354
2	19	180.0000	1217.4767
2	21	0.0000	822.0414
2	21	10.0000	822.5167
2	21	20.0000	823.9633
2	21	30.0000	826.4376
2	21	40.0000	830.0169
2	21	50.0000	834.7766
2	21	60.0000	840.7638
2	21	70.0000	847.9712
2	21	80.0000	856.3160
2	21	90.0000	865.6262
2	21	100.0000	875.6377
2	21	110.0000	886.0016
2	21	120.0000	896.3026
2	21	130.0000	906.0845
2	21	140.0000	914.8827
2	21	150.0000	922.2570
2	21	160.0000	927.8250
2	21	170.0000	931.2908
2	21	180.0000	932.4673

M = NUMBER OF MARS' ROTATIONS

N = NUMBER OF SATELLITE REVOLUTIONS

MARS EXPLORATION

```
**************************************************
*                                                *
*           RENDEZVOUS COMPATIBLE ORBIT          *
*                                                *
**************************************************
```

INPUT DATA

MU = 42828.3 J2 = 0.29400E-02 PP = 88643.000 R = 3397.200

M	N	INCL	ALTITUDE
2	23	0.0000	562.6417
2	23	10.0000	563.2466
2	23	20.0000	565.0798
2	23	30.0000	568.1891
2	23	40.0000	572.6375
2	23	50.0000	578.4789
2	23	60.0000	585.7314
2	23	70.0000	594.3519
2	23	80.0000	604.2148
2	23	90.0000	615.1005
2	23	100.0000	626.6942
2	23	110.0000	638.5958
2	23	120.0000	650.3405
2	23	130.0000	661.4266
2	23	140.0000	671.3488
2	23	150.0000	679.6331
2	23	160.0000	685.8704
2	23	170.0000	689.7455
2	23	180.0000	691.0599
2	25	0.0000	336.3546
2	25	10.0000	337.1048
2	25	20.0000	339.3693
2	25	30.0000	343.1851
2	25	40.0000	348.5952
2	25	50.0000	355.6261
2	25	60.0000	364.2601
2	25	70.0000	374.4106
2	25	80.0000	385.9030
2	25	90.0000	398.4649
2	25	100.0000	411.7273
2	25	110.0000	425.2372
2	25	120.0000	438.4802
2	25	130.0000	450.9101
2	25	140.0000	461.9836
2	25	150.0000	471.1953
2	25	160.0000	478.1122
2	25	170.0000	482.4018
2	25	180.0000	483.8554

M = NUMBER OF MARS' ROTATIONS

N = NUMBER OF SATELLITE REVOLUTIONS

MARS EXPLORATION

```
**************************************************
*                                                *
*          RENDEZVOUS COMPATIBLE ORBIT           *
*                                                *
**************************************************
```

INPUT DATA

MU = 42828.3 J2 = 0.29400E-02 PP = 88643.000 R = 3397.200

M	N	INCL	ALTITUDE
2	27	0.0000	136.3635
2	27	10.0000	137.2755
2	27	20.0000	140.0205
2	27	30.0000	144.6200
2	27	40.0000	151.0918
2	27	50.0000	159.4270
2	27	60.0000	169.5640
2	27	70.0000	181.3650
2	27	80.0000	194.5994
2	27	90.0000	208.9368
2	27	100.0000	223.9507
2	27	110.0000	239.1341
2	27	120.0000	253.9236
2	27	130.0000	267.7309
2	27	140.0000	279.9770
2	27	150.0000	290.1288
2	27	160.0000	297.7316
2	27	170.0000	302.4386
2	27	180.0000	304.0323

M = NUMBER OF MARS' ROTATIONS

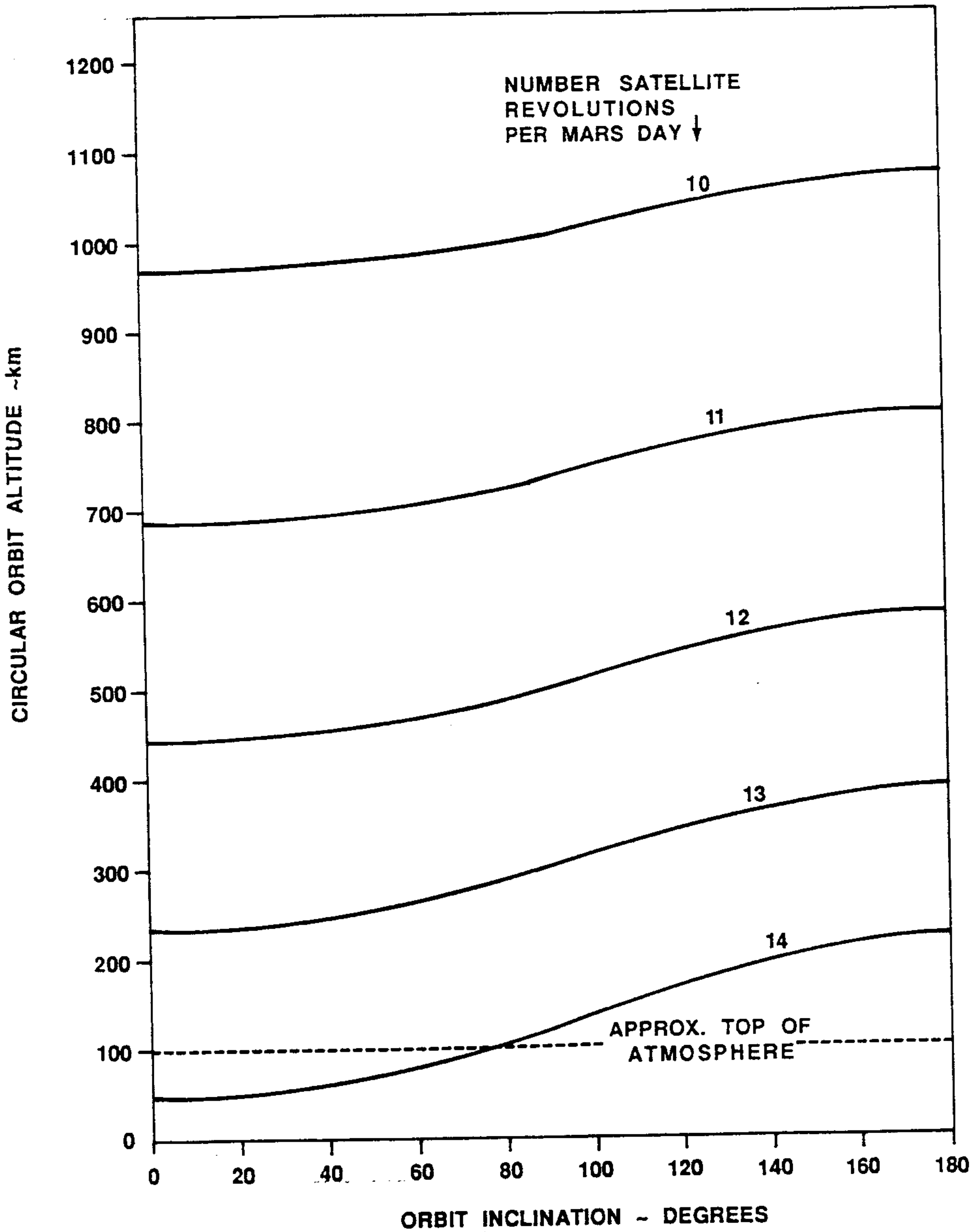
N = NUMBER OF SATELLITE REVOLUTIONS

RENDEZVOUS COMPATIBLE MARS ORBITS
ALTITUDE VERSUS INCLINATION FOR MARTIAN DAILY REPEATING ORBITS

This chart shows a graph of some of the preceeding data. A daily repeating orbit allows descent of ascent opportunities between the piloted vehicle and the landing site to occur at the same time each day. To synchronize the orbital motion of the piloted vehicle with the rotation of Mars, it is necessary to select a parking orbit with the proper combination of altitude and inclination. The orbit must complete an integer number of revolutions per day and the orbit plane must precess at a particular rate so that the landing site will be in the orbit plane at the same time each day. This chart shows the circular orbit altitude versus inclination for Mars daily repeating orbits with altitudes from 100 to 1200 kilometers.

1-5828-7

RENDEZVOUS COMPATIBLE MARS ORBITS ALTITUDE VERSUS INCLINATION FOR MARTIAN DAILY REPEATING ORBITS

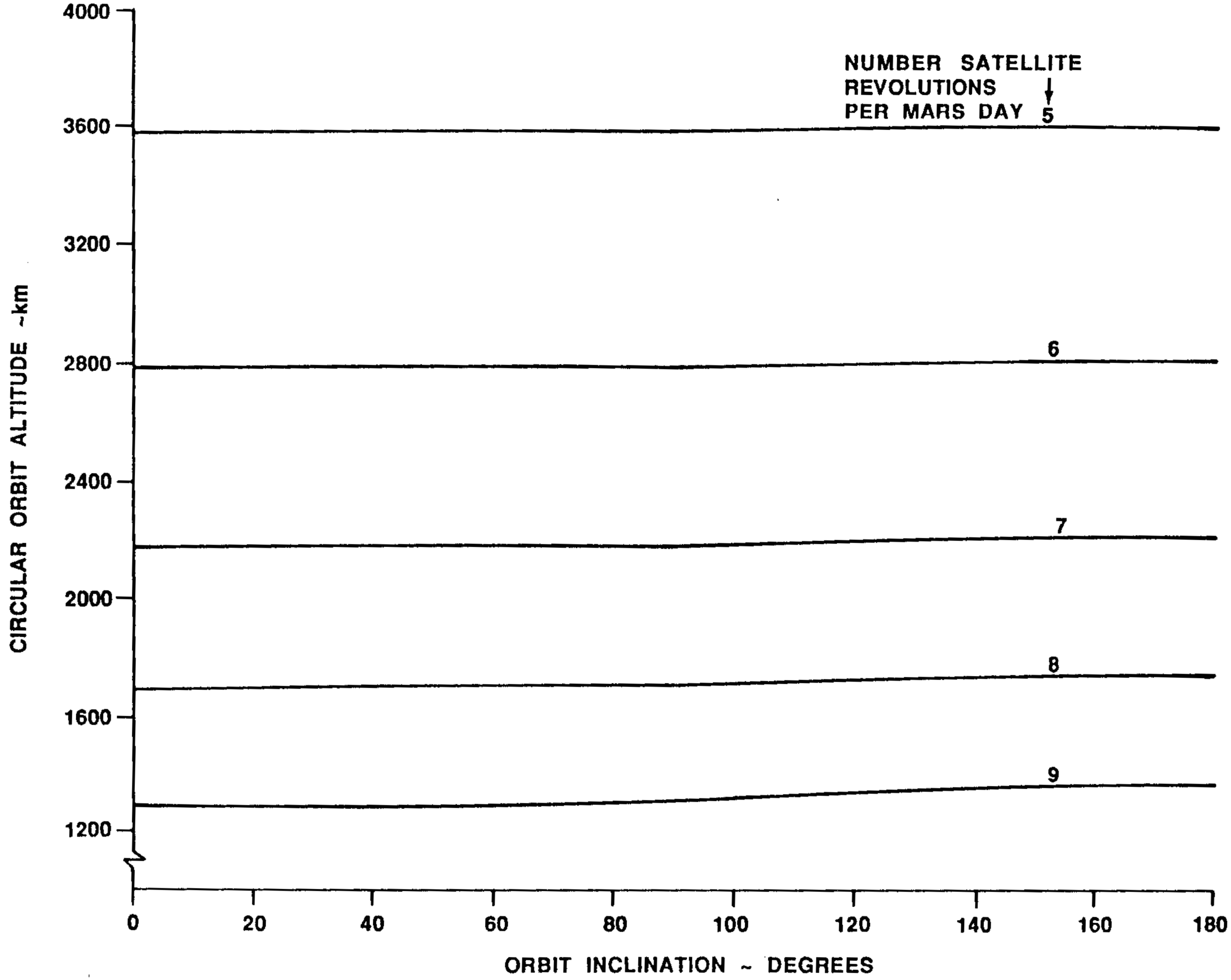


RENDEZVOUS COMPATIBLE MARS ORBITS
ALTITUDE VERSUS INCLINATION FOR MARTIAN DAILY REPEATING ORBITS

This chart shows a continuation of the data on the previous chart for circular orbit altitudes from 1200 to 3600 kilometers.

1-5827-7-

RENDEZVOUS COMPATIBLE MARS ORBITS
ALTITUDE VERSUS INCLINATION
FOR MARTIAN DAILY REPEATING ORBITS



IN-PLANE DEBOOST POSITION ERROR FOR DESCENT TO THE MARS SURFACE

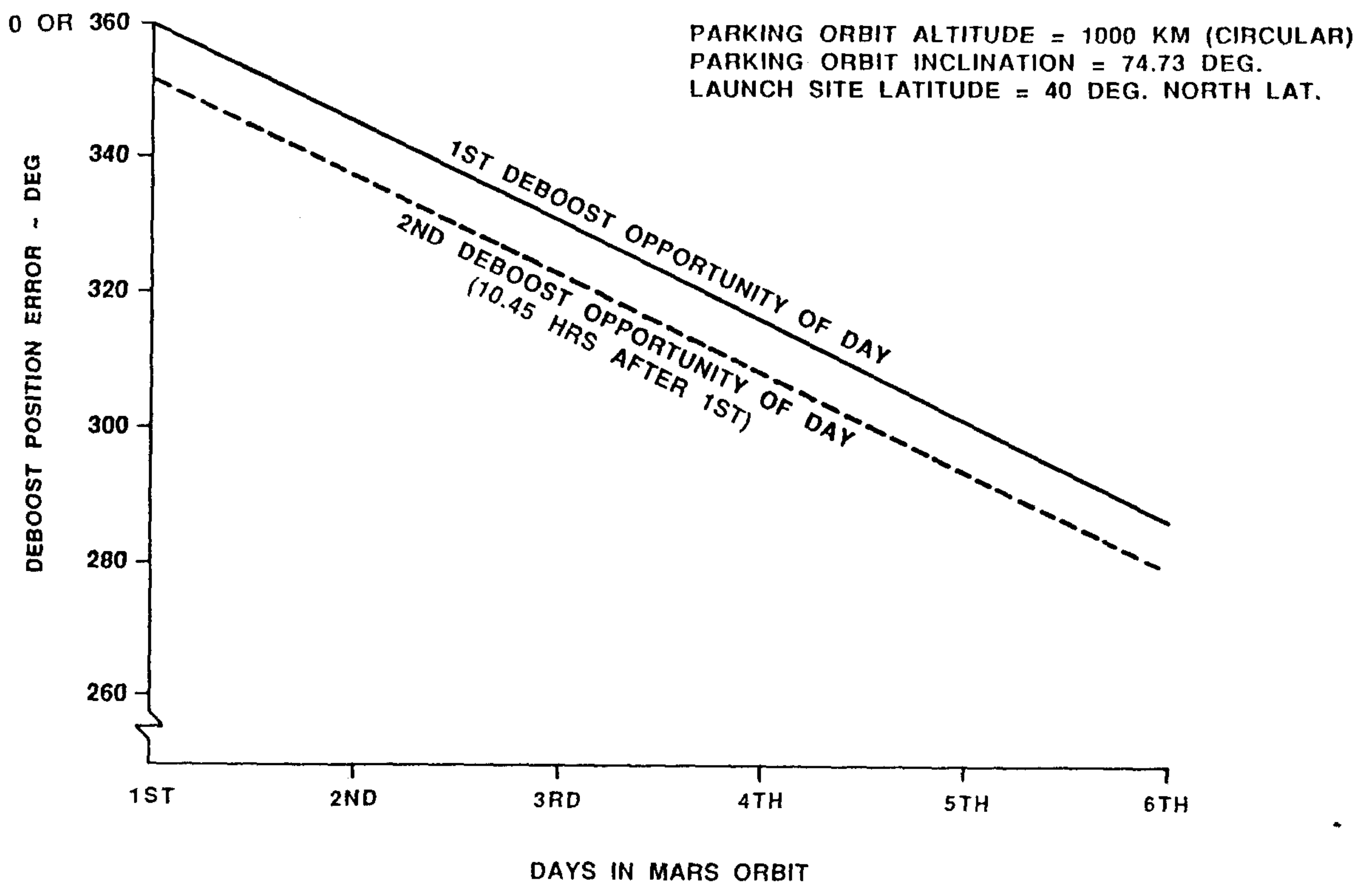
Since there are many factors that must be considered in selecting the Mars parking orbit, it may not be possible to select a repeating orbit in which the ideal descent and ascent conditions occur every day. As long as the landing site latitude is less than the orbit inclination, the landing site will pass through the orbit plane twice a day. Unfortunately, the piloted vehicle may not be at the proper location in the orbit from which the descent vehicle can descend to a specific longitude. In this case, the descent vehicle would have to enter an intermediate orbit to correct for the position error. The intermediate orbit would be selected so that when the landing site passes through the orbit plane, the descent vehicle would be at the proper deboost location. In this study, a 1000 kilometer altitude, 74.73 degree inclination circular parking orbit was chosen because of the Mars arrival and departure conditions, and the landing site was assumed to be at 40 degrees north latitude, 140 degrees west longitude.

This chart shows the in-plane deboost position error of the piloted vehicle of the first six days in Mars orbit. It was assumed that on the first day in orbit, the position error would be zero degrees. This means the descent vehicle could deboost directly from the parking orbit to the landing site. If this landing opportunity wasn't used, the position error would jump to 360 degrees, then decrease about 12 degrees per day. To correct for the positions error, the descent vehicle would have to enter an intermediate orbit before deboosting to the surface.

1-4227-8

IN-PLANE DEBOOST POSITION ERROR FOR DESCENT TO THE MARS SURFACE

IN-PLANE CENTRAL ANGLE BETWEEN PILOTED VEHICLE AND DEORBIT POINT AS A FUNCTION OF DAY IN MARS ORBIT.

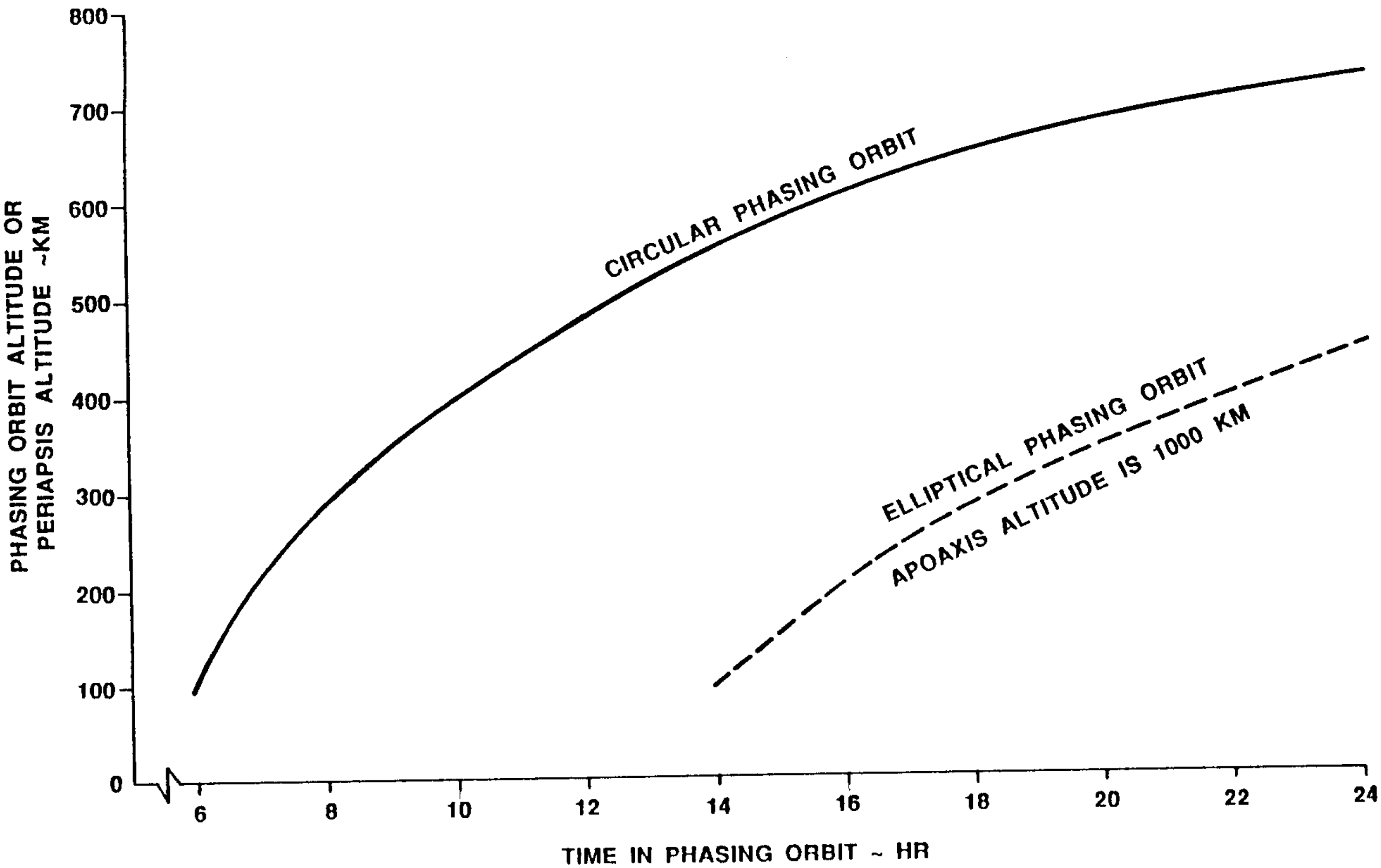


TIME REQUIRED IN PHASING ORBIT

If the landing site and orbiting spacecraft are not in phase, an intermediate phasing orbit is required to reach the desired landing site. The landing spacecraft is assumed to be in a 1000 km altitude circular orbit. Phasing orbit altitude as a function of phasing time is given for both circular and elliptical phasing orbits. It was assumed that the apoapsis altitude of the elliptical phasing orbit remains at 1000 kilometers. These curves were developed assuming worst case landing site and spacecraft phase angle misalignment of 360°. If a 100 km altitude circular phasing orbit were used, the time required for correct phase alignment would be six hours. Longer time is required in the phasing orbit as the phasing orbit altitude increases.

1-4226-8

TIME REQUIRED IN PHASING ORBIT
360° PHASE ANGLE MISALIGNMENT (WORST CASE)
PARKING ORBIT ALTITUDE IS 1000 KM



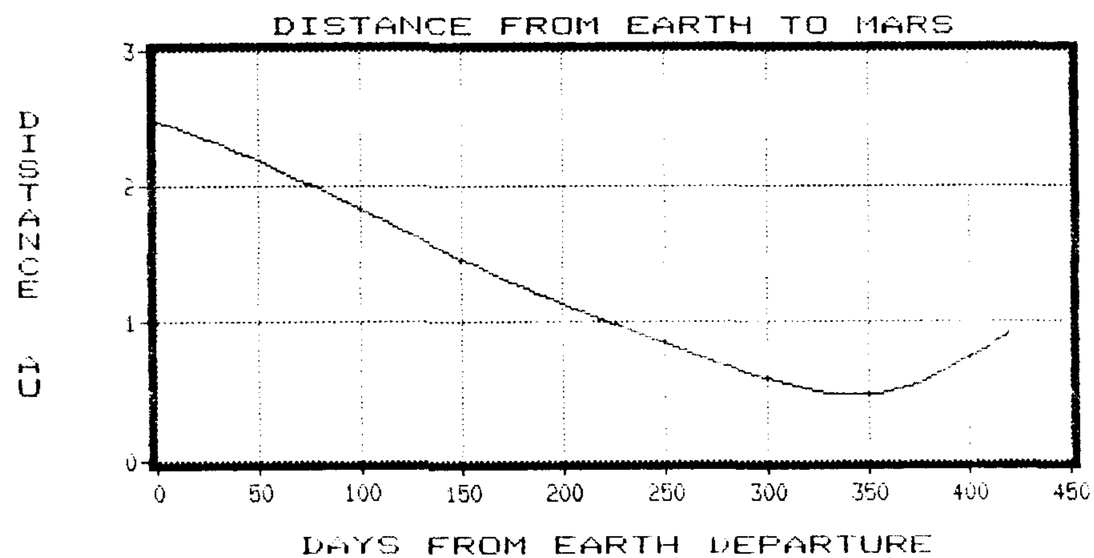
COMMUNICATIONS ANALYSIS

This section describes the communications analysis of the piloted Mars mission. It will show the communications distances, light travel times, and sun interference angles between the piloted vehicle, Earth, and Mars for the outbound and inbound legs of the mission. In the final part of this section, the key communications issues are described. These issues will determine how data is transmitted and stored, and what type of equipment is required for the mission.

2.6 Mars Mission Communications

EARTH-TO-MARS LINE-OF-SIGHT DISTANCE

This chart shows the line-of-sight distance between the Earth and Mars for the entire duration of the 420 day mission (Nov. 21, 2004 to Jan. 15, 2006). This represents the distance that communications would travel from Earth to the cargo vehicle which would be in Mars orbit.

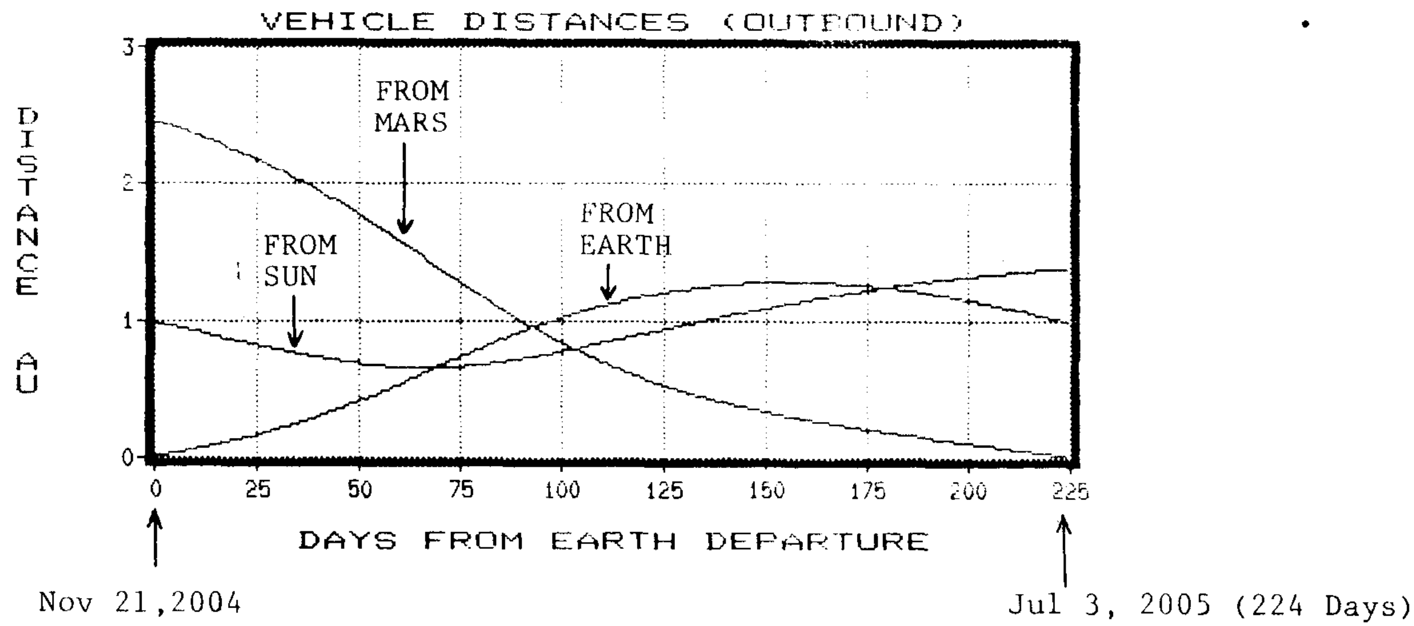

DISTANCE FROM EARTH TO MARS
DISTANCE AU
3
2
1
0
0
50
100
150
200
250
300
350
400
450
DAYS FROM EARTH DEPARTURE

TRANSFER VEHICLE DISTANCES FROM EARTH, MARS, SUN
(OUTBOUND LEG)

This chart shows the distances between the transfer vehicle and Earth, Mars, and the Sun during the time from Earth launch on Nov. 21, 2004 to Mars arrival on Jul. 3, 2005. The trip time for this outbound leg is 224 days.

PILOTED VEHICLE DISTANCE FROM EARTH, MARS, AND SUN

(Outbound Leg)

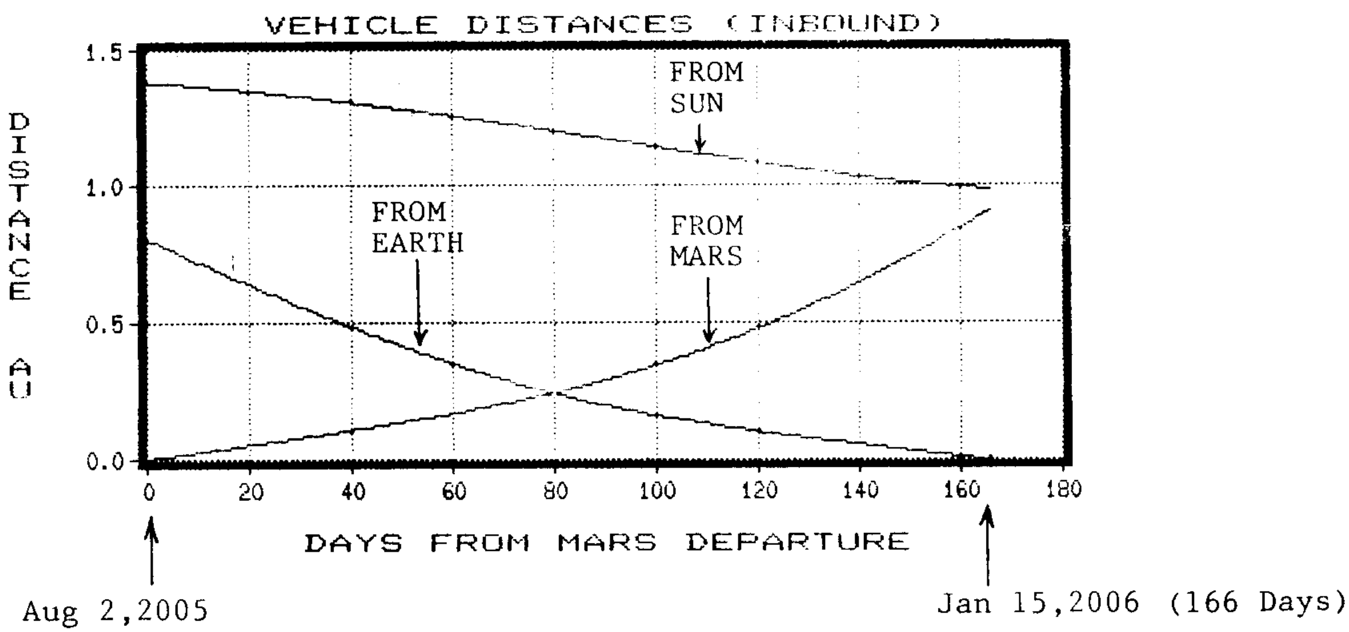


TRANSFER VEHICLE DISTANCES FROM EARTH, MARS, SUN
(INBOUND LEG)

This chart shows the distances between the transfer vehicle and Earth, Mars and the Sun during the time from Mars departure on Aug. 8, 2005 to Earth arrival on Jan. 15, 2006. The trip time for this inbound leg is 166 days.

PILOTED VEHICLE DISTANCE FROM EARTH, MARS, AND SUN

(Inbound Leg)

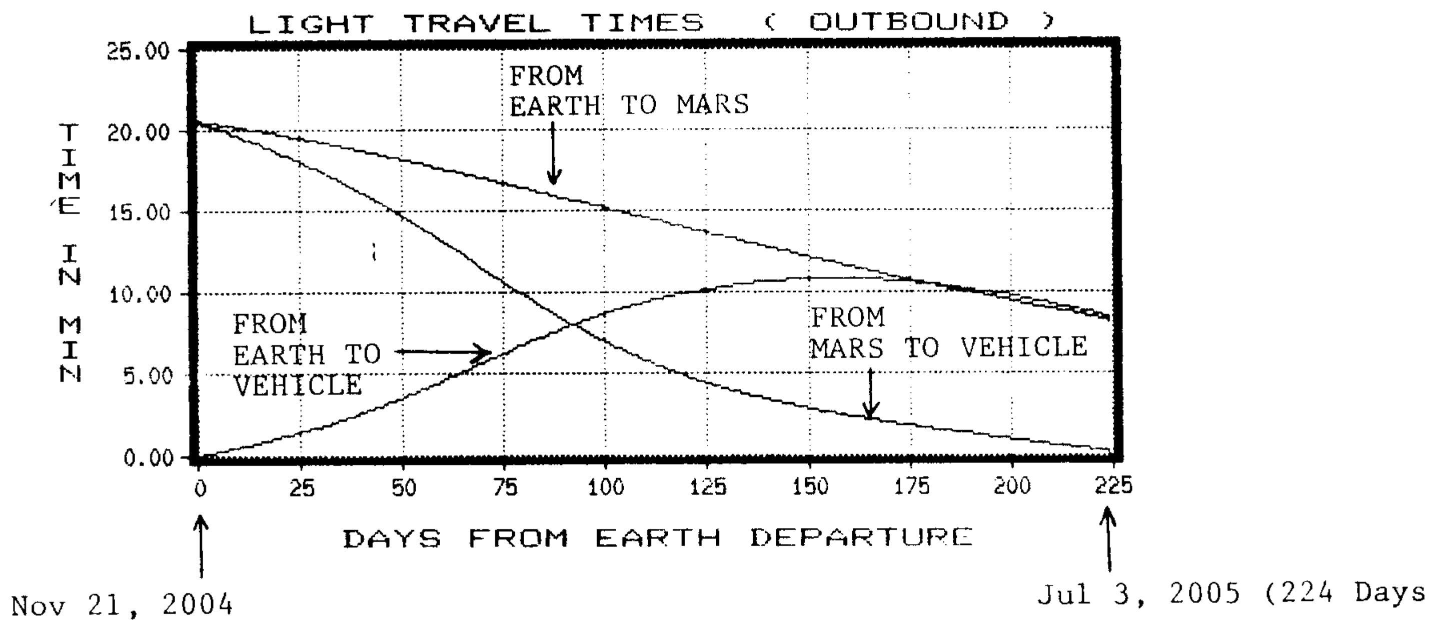


LIGHT TRAVEL TIMES (OUTBOUND LEG)

Communications between Earth and the transfer vehicle, or between Mars and the transfer vehicle, will have long travel times due to the large distances between the bodies. This chart shows the "light travel time" from Earth to the vehicle, from Mars to the vehicle, and from Earth to Mars. These data are for the outbound leg from Nov. 21, 2004 to Jul. 3, 2005.

LIGHT TRAVEL TIME FROM PILOTED VEHICLE, EARTH, AND MARS

(Outbound Leg)

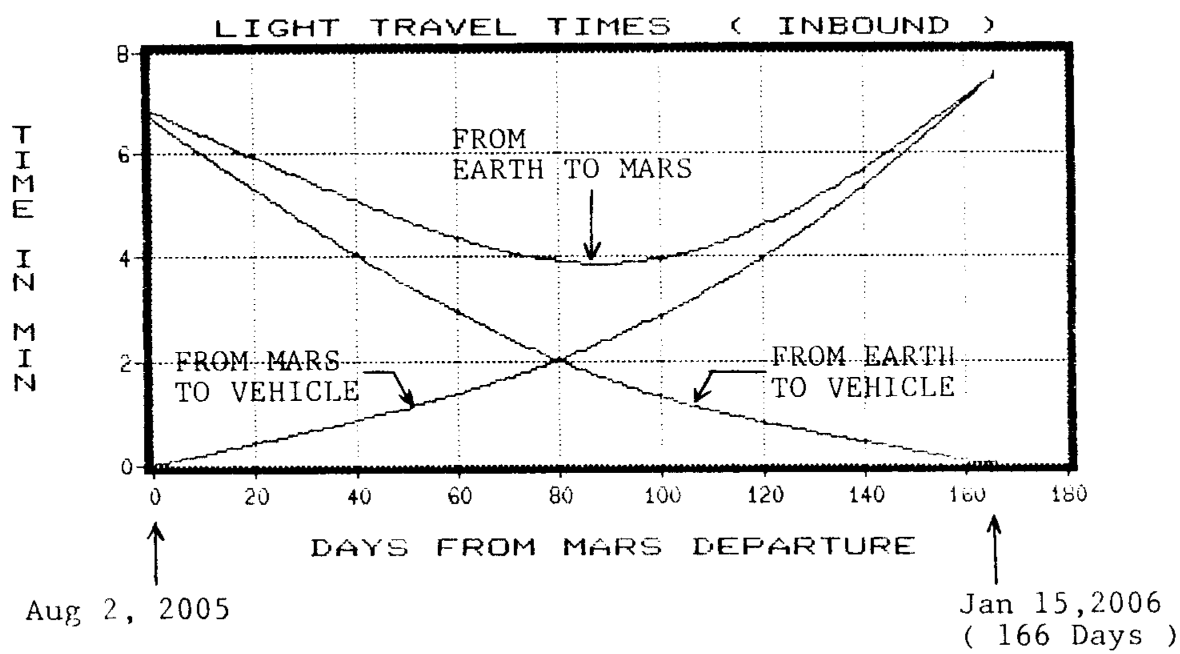


LIGHT TRAVEL TIMES (INBOUND LEG)

This chart shows the light travel time from Earth to the vehicle, from Mars to the vehicle, and from Earth to Mars for the inbound portion of the mission from Aug. 2, 2005 to Jan. 15, 2006.

LIGHT TRAVEL TIME FROM PILOTED VEHICLE, EARTH, AND MARS

(Inbound Leg)

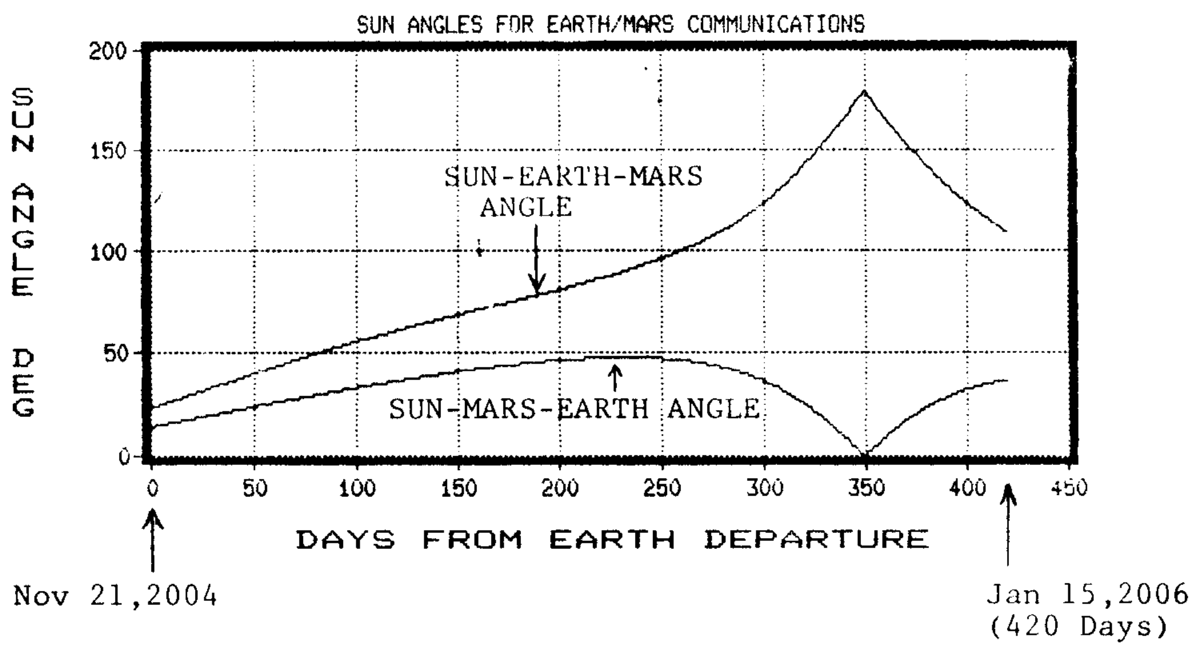


SOLAR INTERFERENCE ANGLE AFFECTING EARTH/MARS COMMUNICATIONS

This chart shows the angles between the solar vector and the Earth-to-Mars vector and the Mars-to-Earth vector. The zero sun angle at 350 days into the mission will not pose a communications problem since the vehicle will have left Mars approximately 95 days earlier.

SOLAR INTERFERENCE ANGLES AFFECTING COMMUNICATIONS
BETWEEN EARTH AND MARS

(Outbound and Inbound Legs)

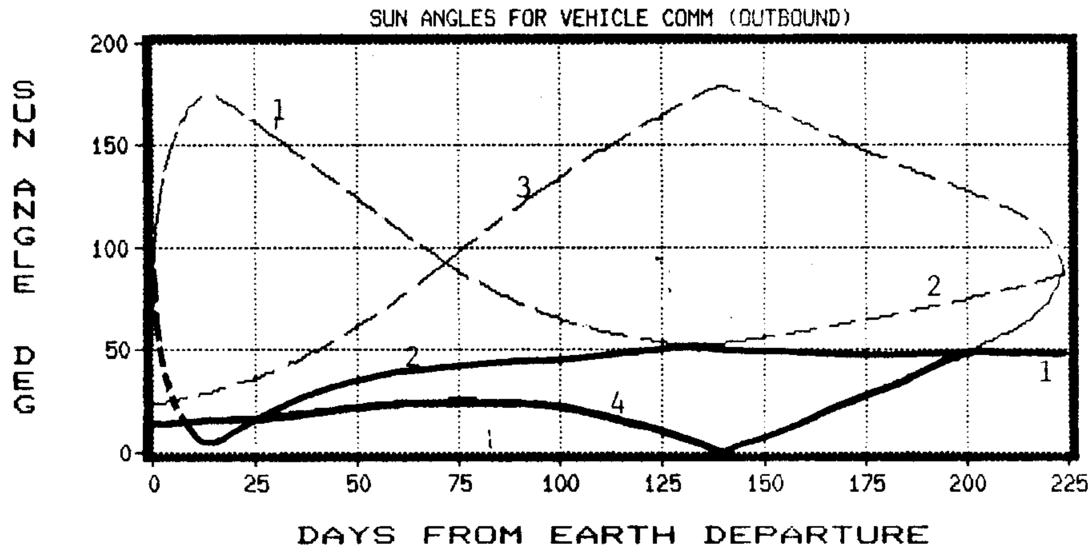


SUN INTERFERENCE ANGLES AFFECTING VEHICLE COMMUNICATIONS TO/FROM EARTH AND MARS (OUTBOUND LEG)

This chart shows the sun angles that are illustrated in the accompanying diagram for the outbound leg. The darkened curves represent the values that, potentially, could cause communication block-out periods. For example, communications between the piloted vehicle and the cargo vehicle at 130 days (from Earth departure of piloted vehicle) into the mission would be hampered for days due to angle #4 being small.

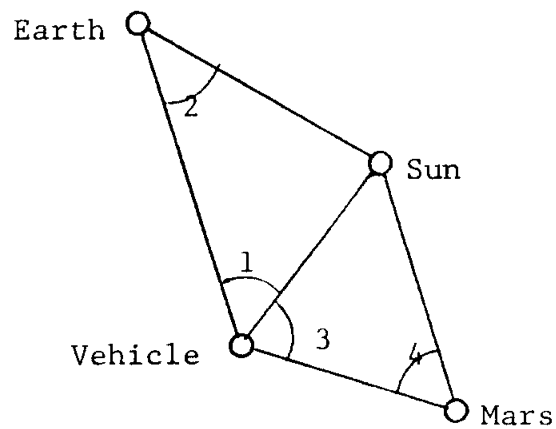
SUN INTERFERENCE ANGLES AFFECTING VEHICLE COMMUNICATIONS

(Outbound Leg)



Nov 21, 2004

Jul 3, 2005
(224 Days)

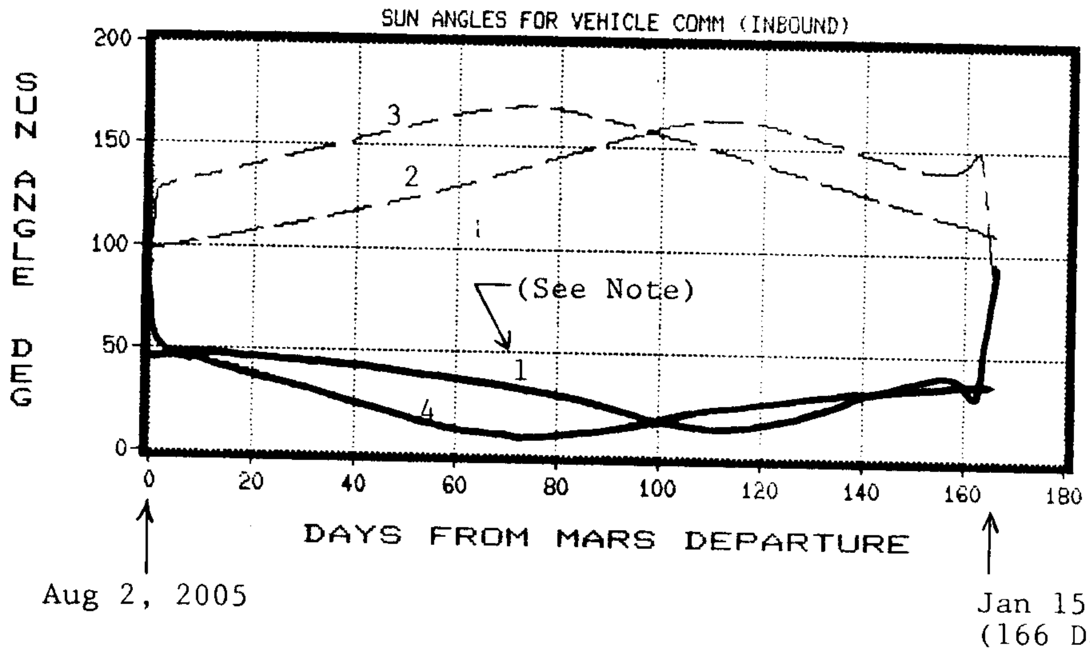


SUN INTERFERENCE ANGLES AFFECTING VEHICLE COMMUNICATIONS TO/FROM EARTH AND MARS (INBOUND LEG)

This chart shows the sun angle illustrated on the diagram in the previous chart for the inbound leg. The darkened curves represent the values that, potentially, could hamper communications - curve 1 for vehicle/Earth communications and curve 4 for vehicle/Mars communications.

SUN INTERFERENCE ANGLES AFFECTING VEHICLE COMMUNICATIONS

(Inbound Leg)



Note: See Diagram on Outbound chart.

KEY COMMUNICATIONS ISSUES

The data rates, the amount of communication coverage required and the level of subsystem redundancy for a manned Mars mission will have a major impact on the overall sizing, complexity, and architecture of the communications system. These parameters will also influence in an interactive fashion the choice of technologies such as communications frequencies and data processing methods used.

MANNED MARS

KEY COMMUNICATION ISSUES

- O DATA RATES
- O COMMUNICATION COVERAGE
- O FREQUENCY OF THE COMMUNICATIONS LINKS
- O REDUNDANCY/SPARES

DATA RATES

The data rate requirement for each of the various communication links used to support a manned mars mission is needed not only to size the system in terms of antenna size and power levels, but also to determine applicable technologies; such as microwave versus optical transmissions. The data rate requirements will likely be driven by the science requirements and the video requirements.

The use of data compression techniques can result in a very significant reduction in the amount of data that must be transmitted. As an example, the Japanese are transmitting color video at a rate of 44 Kbps as compared to a normal commercial rate of nearly 100 Mbps. The amount of reduction is dependent on a number of factors including desired quality.

MANNED MARS
DATA RATES

- O VIDEO REQUIREMENTS EXPECTED TO DRIVE DATA RATE
  - USED FOR PUBLIC RELATIONS, SCIENTIFIC OBSERVATIONS, SERVICING/OPERATIONS SUPPORT
- O VIDEO DATA RATES GREATLY DEPENDENT ON DESIRED QUALITY, SUBJECT MATTER AND AMOUNT OF DATA COMPRESSION
- O COMMERCIAL QUALITY WITHOUT DATA COMPRESSION IS APPROXIMATELY 100 MBPS
- O *THE JAPANESE ARE EXPERIMENTING WITH FULL-MOTION COLOR VIDEOPHONES* THAT OPERATE AT 44 KBPS AND 112 KBPS (MAX. FRAME RATE OF 10 FRAMES PER SECOND PLUS DATA COMPRESSION)

RETURN LINK ANALYSIS

The facing chart shows the data rates that can be supported by 3-foot and 8-foot diameter spacecraft antennas as a function of RF power level when communicating over the maximum spacecraft-to-Earth distance of 1.3 AU. The supporting link analysis assumes use of the planned Deep Space Network (DSN) 70-meter subnet operating at a frequency of 32 GHz. In order to support the data rates indicated the following conditions must be meet:

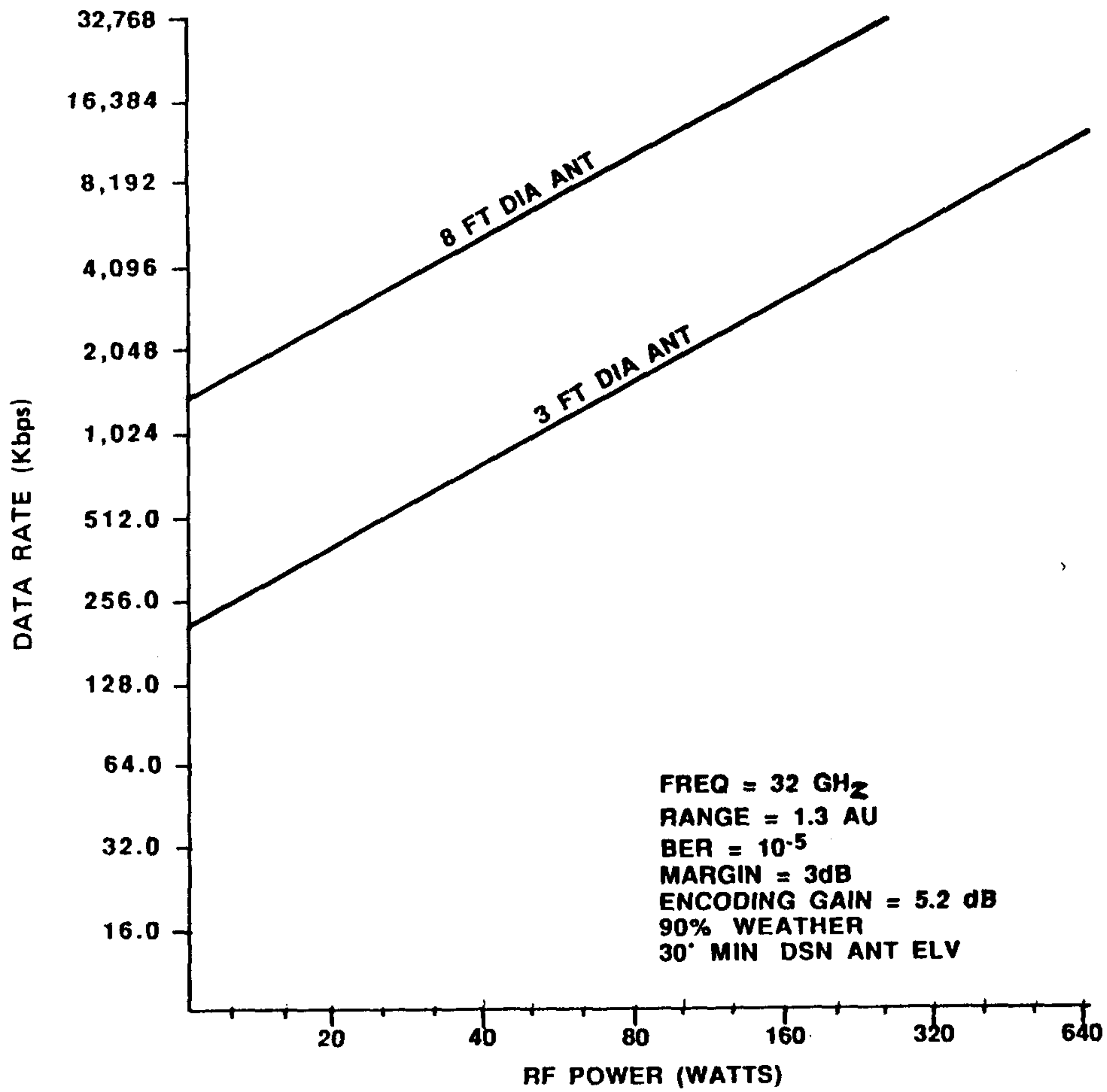
- GND antenna cannot be pointed closer than 30^{o} to the horizon.

- Current weather conditions relative to RF link degradation are no worst than conditions expected for 90 percent of the time.

- The data transmitted is rate 1/2 convolutionally encoded.

1-4022-7

MANNED MARS MISSION RETURN LINK*



* UTILIZES PLANNED DSN 70 METER ANT WITH IMPROVED SURFACE EFF.

COMMUNICATIONS COVERAGE

The facing chart indicates a number of options for providing communications coverage between the Earth and a Mars Lander and gives the amount of coverage available with each option. Without some type of communications relay in orbit around Mars, communications between the Earth and the Lander is limited to approximately 50 percent of the time. If the Orbiter for the Sprint Mission is utilized as a communications relay, it could support a communications contact once every 2.45 hours for a Lander near the Equator, but its capability to support a Lander is reduced for landing sites at higher latitudes. If continuous or nearly continuous coverage for the Lander is required, it can be provided by placing relay satellites in isosynchronous orbit around Mars.

MANNED MARS
COMMUNICATIONS COVERAGE

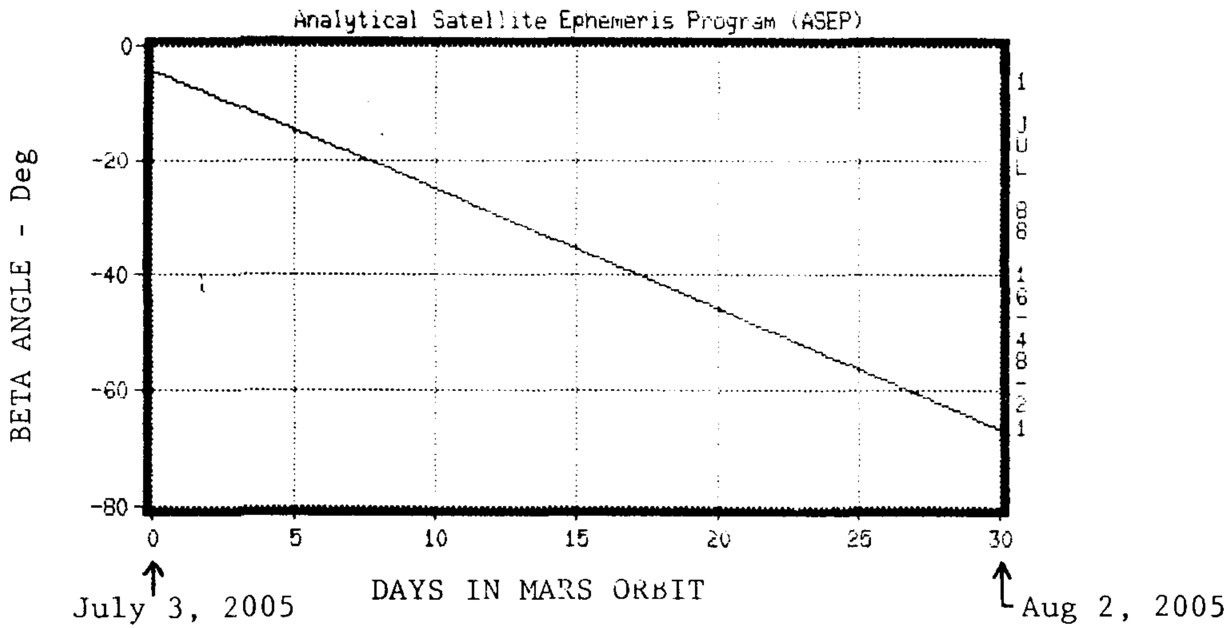
- ROUND TRIP DELAY TIME AT MAX. DISTANCE OF 1.3 AU IS APPROXIMATELY 21 MINUTES (APPROX. 16 MIN. WHEN AT MARS).
- WITHOUT AN ORBITING RELAY, COMMUNICATIONS WITH THE LANDER ON THE SURFACE WILL BE BLOCKED FOR APPROXIMATELY 12 HOURS EACH DAY.
- THE ORBITER (IN A 1000 KM ORBIT) COULD PROVIDE AN UPDATE ONCE EVERY 2.45 HOURS FOR EQUATORIAL OPERATIONS. REDUCED CONTACTS FOR HIGHER LATITUDES.
- ONE COMMUNICATIONS RELAY SATELLITE PLACED IN ISOSYNCHRONOUS ORBIT ABOVE THE LANDER WOULD PROVIDE CONTINUOUS RELAY CAPABILITY BETWEEN EARTH AND LANDER EXCEPT FOR APPROXIMATELY 77 MIN. PER DAY.

EARTH BETA ANGLE FROM MARS ORBIT

When the piloted vehicle arrives at Mars, it is aerobraked into a 1000 km altitude orbit at 74.73° inclination. The ascending node position is 292.4° from the Vernal Equinox. This chart shows the angle between this orbit plane and the Mars/Earth line-of-sight for the 30 day period in Mars orbit.

EARTH BETA ANGLE FROM MARS ORBIT

Altitude: 1000 Km   Incl: 74.73°

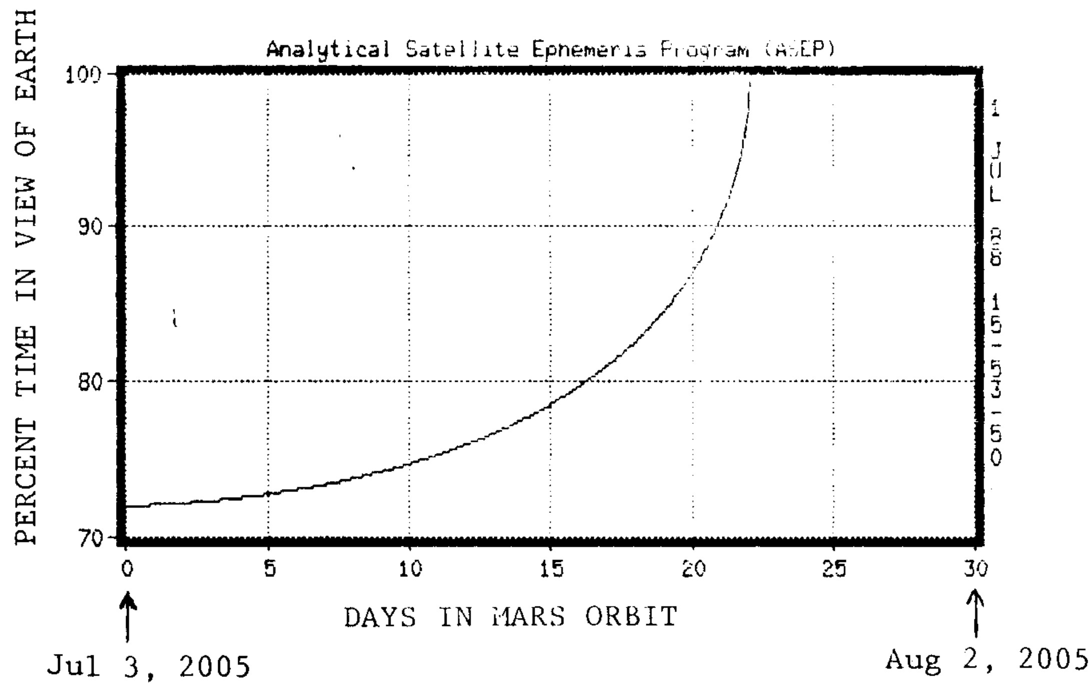


PERCENT TIME IN VIEW OF EARTH FROM MARS ORBIT

This chart shows the percentage of an orbital revolution that the Mars/Earth line-of-sight is not blocked by the planet Mars. The circular orbit about Mars has a 1000 km altitude, 74.73° inclination, and 292.4° ascending node right ascension. After 22 days in orbit, the orbit plane has precessed to an orientation which provides continuous view of the Earth and, hence, continuous communication opportunity.

PERCENT TIME IN VIEW OF EARTH FROM MARS ORBIT

Altitude: 1000 Km   Incl: 74.73°



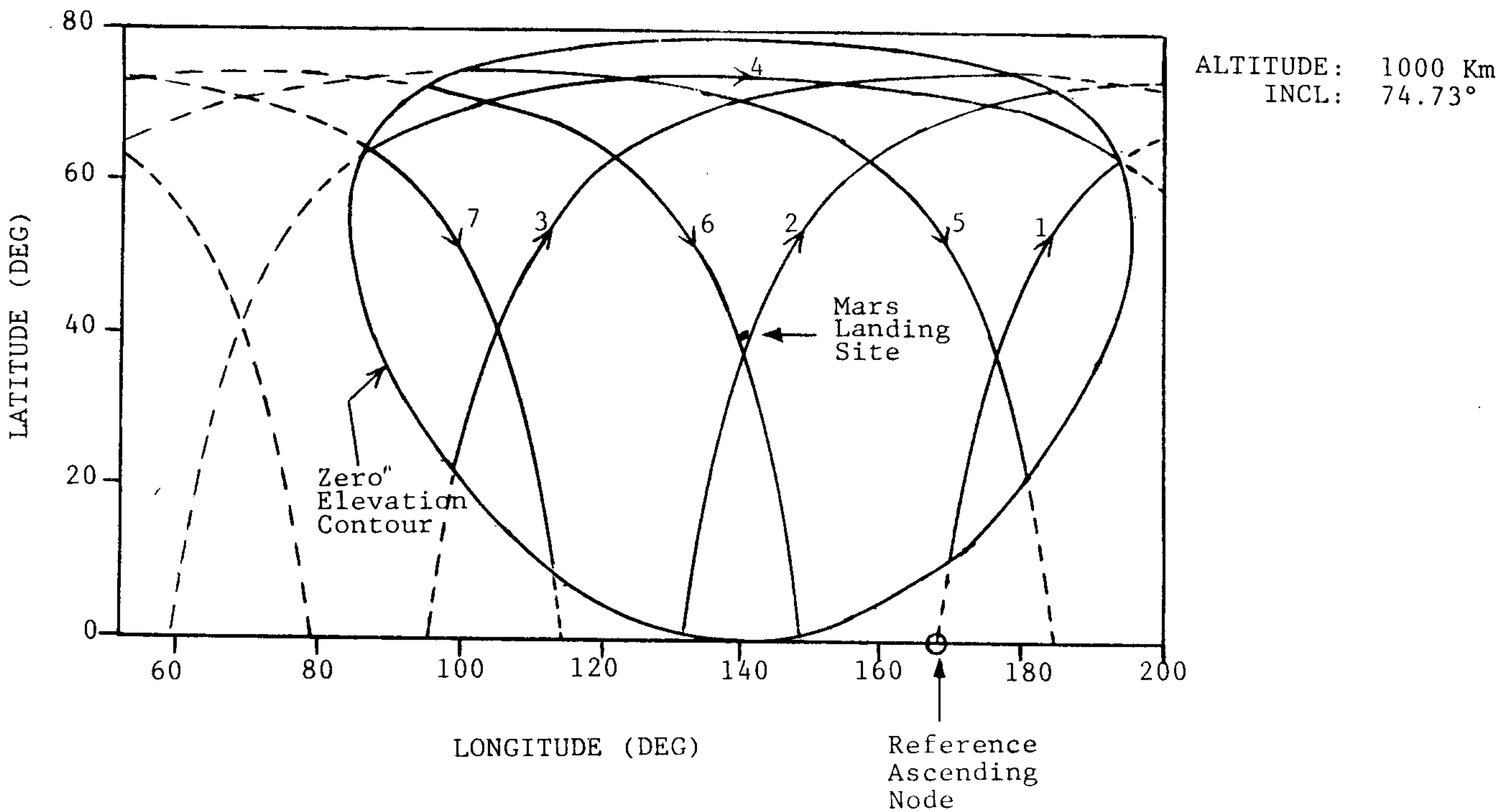
MARS LANDING SITE COMMUNICATIONS OPPORTUNITIES

Upon arrival of the piloted vehicle at Mars, it will insert into a 1000 Km orbit at 74.73^{o} inclination. From this orbit, the Mars Excursion Module (MEM) will deorbit to a landing site located at 40^{o} N latitude and 140^{o} E longitude. Communications between this site and the piloted vehicle in orbit will be required during the surface visit. The accompanying chart shows ground tracks for a typical series of passes over the ground site. The solid ground track lines indicate the position of the orbiting vehicle where its elevation angle from the site is 0^{o} or larger. Line-of-sight communication can occur during passes within the zero degree elevation contour shown on the chart.

The table shows the approximate times of acquisition and loss of communications from a reference node. These communications opportunities occur approximately every 23.5 hours as the landing site rotates through the orbit plane.

For communications between the landing site and Earth, there will be approximately a 9.6 hour communication opportunity each day. This is the period between Earth rise and Earth set as viewed from the Mars landing site.

LANDING SITE/ORBITER COMMUNICATIONS OPPORTUNITIES

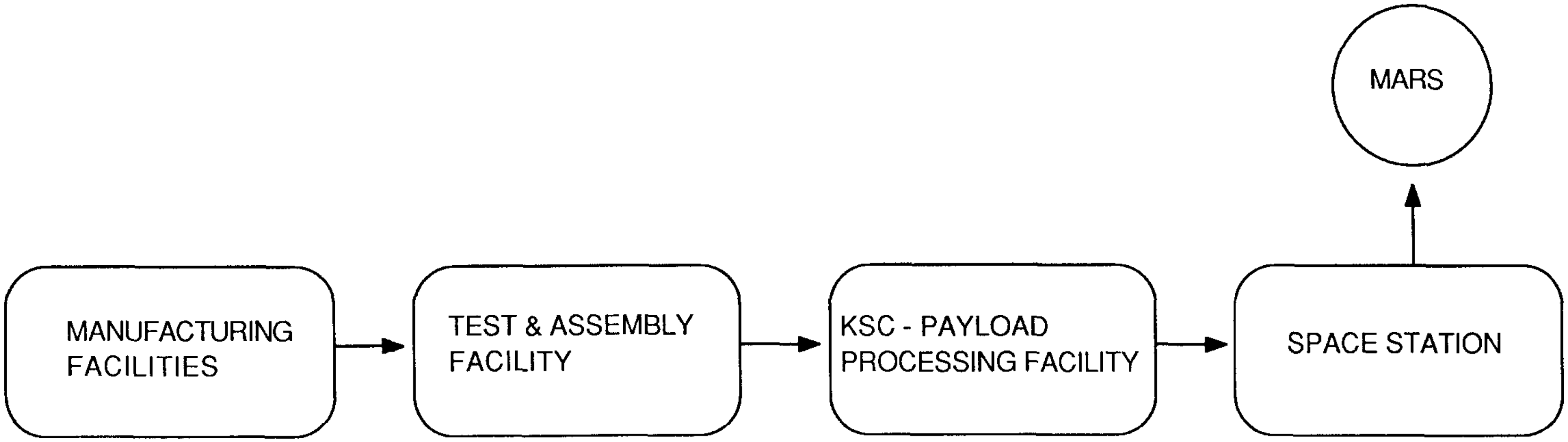


PASS	TIME FROM REFERENCE ASCENDING NODE (HRS) AOS	LOS	PASS DURATION (MIN)
1	.073	.47	23.67
2	2.47	3.03	33.11
3	5.09	5.54	26.87
4	7.87	8.17	17.87
5	10.45	10.96	30.14
6	13.01	13.57	33.66
7	15.56	15.98	25.67
		TOTAL :	190.99 Min
			3.18 Hrs

MARS VEHICLE ASSEMBLY OPERATIONS

This chart summarizes the assembly operations for the Mars vehicles. After each vehicle is assembled and checked at the manufacturing facility, it is disassembled into subsystem assemblies small enough to be transported to KSC and to be installed in the HLLV cargo bay. Assembled portions of the vehicle are as large as possible to reduce orbital assembly. This hardware is transported to Space Station or a similar orbiting platform for assembly, reverification, and launch.

MARS VEHICLE ASSEMBLY OPERATIONS



MANUFACTURING FACILITIES

- MANUFACTURE

TEST & ASSEMBLY FACILITY

- ASSEMBLE
- TEST, CHECK, AND VERIFY
- DISASSEMBLE
- TRANSPORT TO KSC

KSC - PAYLOAD PROCESSING FACILITY

- PROCESS AS PAYLOAD
- INSTALL IN LAUNCH VEHICLE
- LOAD HAZARDOUS MATERIALS
- LAUNCH TO LEO

SPACE STATION

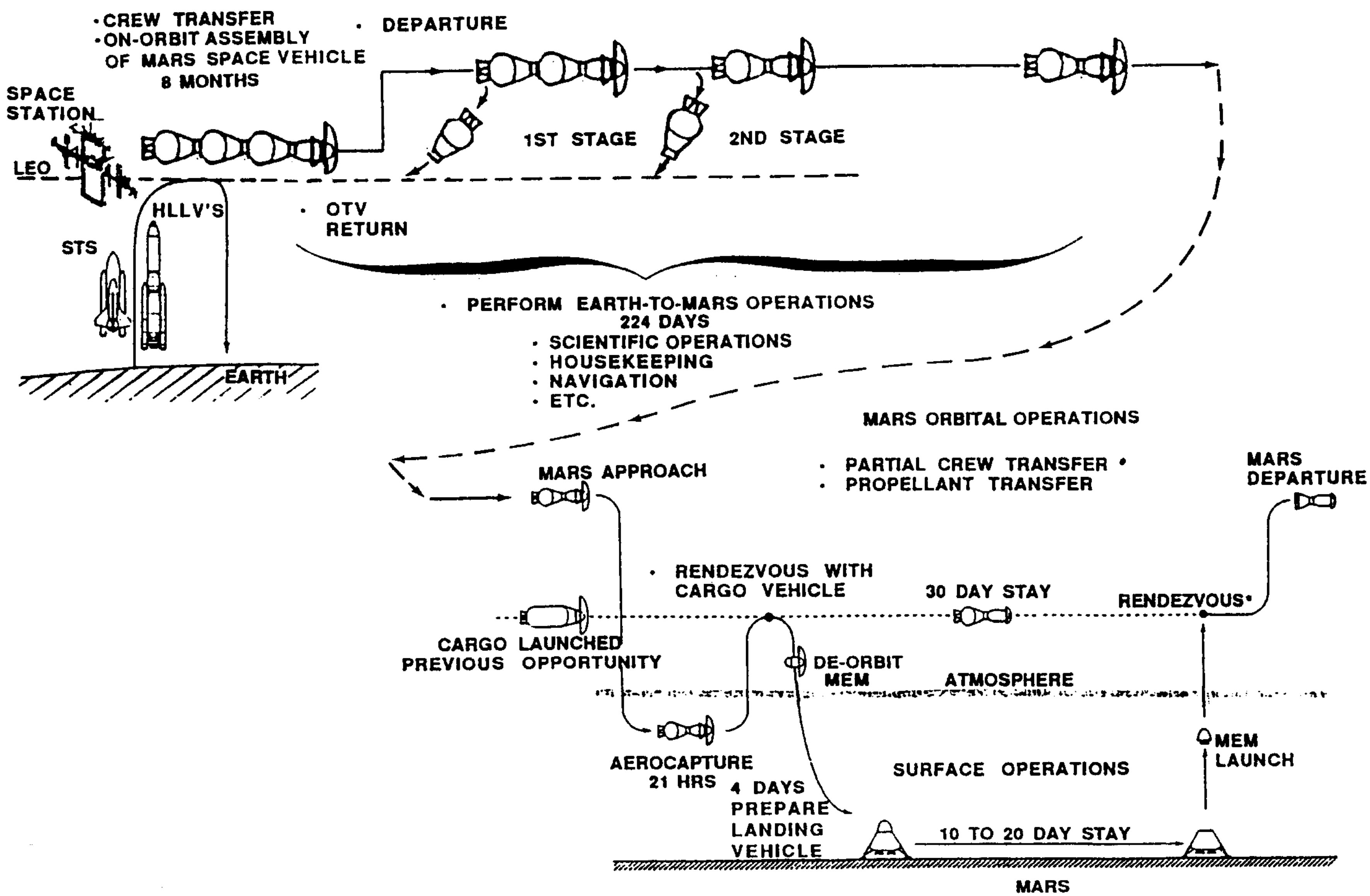
- ASSEMBLE
- RETEST, RECHECK, AND REVERIFY
- LOAD PAYLOADS, NECESSITIES, AND FUEL
- LAUNCH TO MARS

PILOTED MARS MISSION SCENARIO - SPLIT OPTION

The facing page gives an overview of the Mars vehicle operations for the piloted mission. The vehicle will require some assembly at LEO due to the payload weight and volume limitations of the launch vehicles. The cargo vehicle is launched into a Mars orbit before the piloted vehicle is launched from LEO. The piloted vehicle is launched using two Earth departure stages. When it arrives at Mars, it utilizes aerobraking to enter a parking orbit in which it rendezvous with the orbiting cargo vehicle. After the rendezvous and docking, the Mars departure stage is mated to the piloted vehicle and the Mars Excursion Module is prepared for use. Part of the crew remains on the piloted vehicle to prepare for the trip home while the other crew members visit Mars. Operations of the Mars to Earth voyage are continued on the next page.

1-5748-8

PILOTED MARS MISSION SCENARIO - SPLIT OPTION

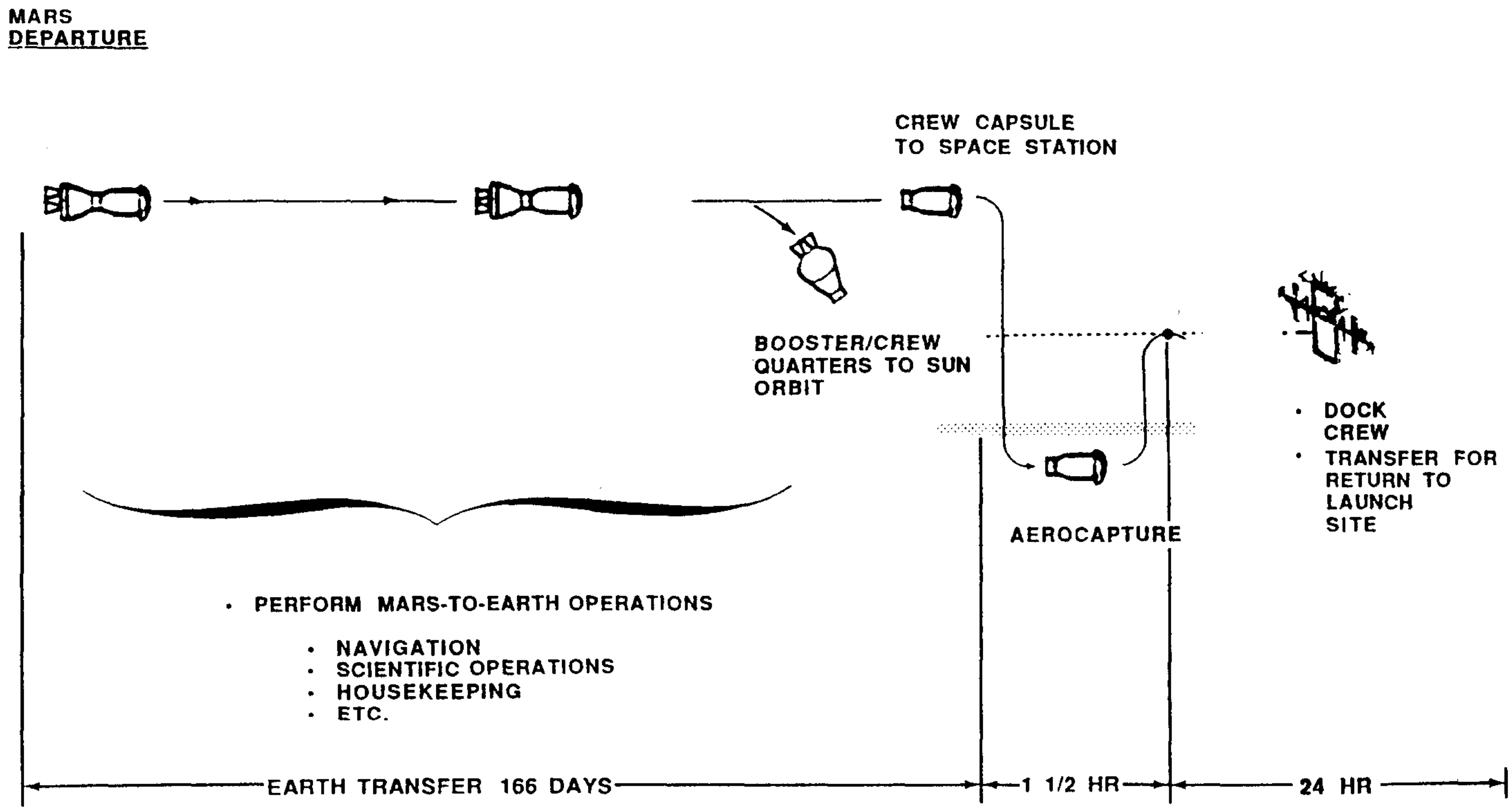


PILOTED MARS MISSION SCENARIO - SPLIT OPTION (CONTINUED)

The facing page shows the mission operations of the Mars to Earth voyage of the piloted vehicle. Only the Earth Return Capsule (ERC) and a small aerobrake are utilized for Earth return. The other vehicle parts are jettisoned and remain in heliocentric orbit. The OMV will retrieve the Earth Return Capsule and return the crew to the Space Station where they may be quarantined prior to returning to Earth.

1-5747-8

PILOTED MARS MISSION SCENARIO - SPLIT OPTION (CONTINUED)



MARS MANNED VEHICLE TRANSFER TIMELINE

The facing page assigns times to selected activities during the transfer to Mars and the return to Earth. Much of the vehicle processing can be done on Earth with some final assembly and checkout at LEO. The burn and travel times were obtained through mission analysis.

MARS MANNED VEHICLE TRANSFER TIMELINE

SPACE STATION		
VEHICLE ASSEMBLY	7	MONTHS
CHECKOUT & PREPARATION	1	MONTHS

mars transfer = 224 days

INITIAL BURN (3 STAGES)	50	MINS
DEADLINE FOR ABORT TO LEO	9.9	MINS INTO 3RD STAGE BURN
TRAVEL	112	DAYS
MID COURSE BURN	5	MINS
TRAVEL	111	DAYS
CORRECTION BURN & MARS SWINGBY DEADLINE	15	SEC
TRAVEL	1	DAY

MARS AEROBRAKE	6	MINS
ADJUST TO ORBIT	21	HOURS

30 days

RENDEVOUS WITH CARGO VEHICLE	1.5	DAYS
DOCK WITH CARGO VEHICLE	6	HOURS
PREPARE LANDING VEHICLE	4	DAYS
BURN FOR MARS SURFACE	30	SEC
TRAVEL	30	MINS
MARS SURFACE	10-24	DAYS
BURN FOR MARS ORBIT	10	MINS
TRAVEL	1	HOUR
RENDEVOUS WITH MISSION VEHICLE	6	HOURS
DOCK WITH MISSION VEHICLE	3	HOURS
PREPARE FOR EARTH RETURN	1	DAY

earth transfer = 166 days

INITIAL BURN	16	MINS
TRAVEL	83	DAYS
MID COURSE BURN	5	MINS
TRAVEL	82	DAYS
SEPARATE MISSION MODULE & EARTH RETURN CAPSULE (ERC)	2	HOURS
ORIENT ERC FOR EARTH CAPTURE	30	MINS
CORRECTION BURN	15	SEC
TRAVEL	1	DAY

EARTH AEROBRAKE	9	MINS
ADJUST TO ORBIT	1.5	HOURS

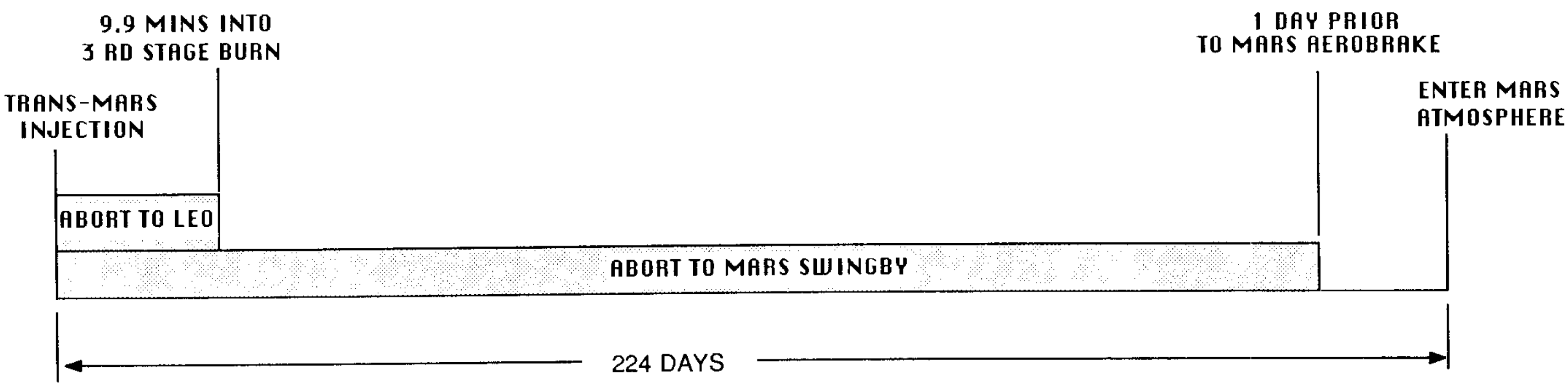
OMV RENDEVOUS	18	HOURS
OMV DOCK	3	HOURS
MANEUVER TO SPACE STATION	24	HOURS
SPACE STATION		

MARS MISSION ABORT SCENARIO

This chart shows the times at which a mission abort is possible. During the trans-Mars injection, it is possible to abort to low Earth orbit until 9.9 minutes into the mission. This is the point at which the third stage still has enough propellant to brake itself back into Earth orbit. Once the abort maneuver is made, it would take 1-2 days to recover the vehicle in low Earth orbit.

After 9.9 minutes into the 3rd stage burn, the vehicle is committed to at least a Mars swingby. It is possible to abort Mars orbit insertion up until 1 day prior to Mars arrival. If the Mars orbit insertion is aborted, it would be impossible to rendezvous with the cargo vehicle, therefore, the piloted vehicle must have the resources to completely sustain the return mission. If the Mars orbit insertion was aborted, the total mission duration would not be significantly reduced. The total trip time would be approximately 410 - 440 days.

MARS MISSION ABORT SCENARIO



A TYPICAL IN-FLIGHT SCHEDULE

This table shows the schedule of a typical day during the voyage from LEO to Mars for the 0-G baseline as well as 1-G and Skylab for comparison. Eating and personal hygiene/housekeeping require more time to complete in zero gravity due to the extra time required to secure all items that could float away. In zero gravity, more exercise is necessary to keep the body strong and healthy for one gravity standards. The time spent on scheduled work, education, and versatile time can be increased in one gravity due to less time needed to complete exercise, eating, and hygiene/housekeeping. A crew member would influence his/her versatile or passive/rest and relaxation (R and R) time. Examples of versatile time activites are family/Earth communications, private time, optional work, or optional exercise. Scheduled work/scientific applications include telescope observations, onboard experiments, health/human reaction monitors, radiation research study toward and educational degree, and mission task simulations. The recommended Skylab daily schedule is shown for reference. The planned Skylab schedule is 13.5 hours for personal activity, 2.5 hours for station operational/ housekeeping, and 8 hours for experiments. Personal activity consists of sleeping, eating, hygiene, training, and R and R. This schedule is for six days. Only necessary activites are scheduled for the seventh day. Any extra time on the seventh day can be used to catch up on work activities or for R and R.

A TYPICAL IN-FLIGHT DAILY SCHEDULE

	MARS		
SKYLAB (HRS)	1-G TIME (HRS)	0-G TIME (HRS)	TASK
7	6	4	SCHEDULED WORK/SCIENTIFIC APPLICATIONS
1	1	1	MONITOR VEHICLE SYSTEMS
1	2	1	EDUCATION/TRAINING
2	1.5	2	PERSONAL HYGINE/HOUSEKEEPING
0	1	2.5	EXERCISE
8	8	8	SLEEP
4.5	3.5	4.5	EATING
.5	1	1	VERSATILE OR PASSIVE TIME/R&R

3.0 VEHICLE PERFORMANCE AND INTEGRATION

The following section provides information on the Split Sprint Mission vehicle performance and integration.

3.0 Vehicle Performance and Integration

3.1 Aerobraking Dynamics

3.2 Mars Vehicle Aerodynamics and Aerobrake Sizing

3.3 Cargo Vehicle Mission Analysis

- 3.3.1 Vehicle Configuration
- 3.3.2 Propulsion System
- 3.3.3 Earth Departure and Recovery of 1st Stage
- 3.3.4 Cargo Vehicle Maneuvers at Mars
- 3.3.5 Vehicle Performance Summary

3.4 Piloted Vehicle Mission Analysis

- 3.4.1 Vehicle Configuration
- 3.4.2 Propulsion System
- 3.4.3 Earth Departure and Recovery of Departure Stages
- 3.4.4 Piloted Vehicle Maneuvers at Mars
- 3.4.5 Vehicle Performance Summary

3.5 Mars Excursion Module Analysis

- 3.5.1 Descent Analysis
- 3.5.2 Ascent Analysis

3.6 Earth Return Capsule Analysis

- 3.6.1 System/Integration Analysis
- 3.6.2 Aerocapture and Recovery of Earth Return Capsule

AERODYNAMIC FORCES

The facing page illustrates some of the symbols and nomenclature used in the calculation of the aerodynamic forces during aerobraking.

Basic Aerodynamics

Drag and lift are generated when the vehicle moves through an atmosphere. Drag is opposite to the velocity relative to the atmosphere and is the working force to dissipate energy. Lift is perpendicular to the relative velocity and to the wings (assumed if not present).

Aerodynamic Forces (CD, CL are functions of angle of attack, mach number)

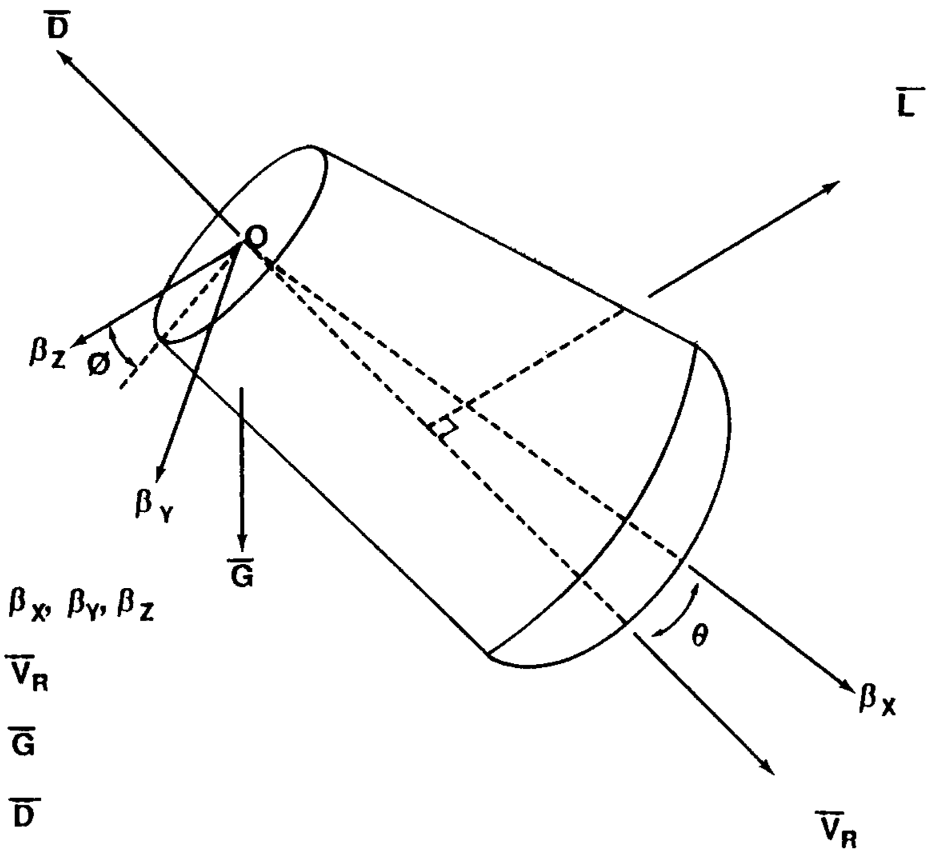
Drag = (Atmos Density/2) * CD * Area * VR ** 2

Lift = (Atmos Density/2) * CL * Area * VR ** 2

The direction of lift is a major parameter in all entries as lift influences braking by moving the vehicle in the atmosphere. Lift must work in harmony with drag, i.e., if more/less drag is needed, lift should pull the vehicle down/up in the atmosphere. When not essential to braking, lift can be used for terminal latitude/longitude or flight plane control.

1-4169-8

AERODYNAMIC FORCES



BODY AXES	–	$\beta_X, \beta_Y, \beta_Z$
VELOCITY	–	$\overline{V}_R$
GRAVITY	–	$\overline{G}$
DRAG	–	$\overline{D}$
LIFT	–	$\overline{L}$
ANGLE OF ATTACK	–	θ
ANGLE OF BANK	–	Ø

AEROBRAKING TO ORBIT

The facing page shows altitude, velocity, and acceleration profiles versus time for a typical aerobraking and capture on arriving at Mars.

What is aerobraking to orbit?

A dissipation of energy using aerodynamic drag to change from an incoming orbit to a second orbit. Lift influences braking by moving the vehicle in the atmosphere and may be used for orbit plane control.

Aerobraking to orbit has never been demonstrated, and is going to be very difficult. Initial attempts should plan for the use of chemical rockets to correct dispersions.

Aerobraking Time (Airtime) Through Atmosphere

About 30 degrees of orbit through periapsis. (5 to 12 minutes)
Cannot be less than incoming orbit time!
Cannot be more than outgoing orbit time!

Delta-Velocity and Airtime Determine G-Loads

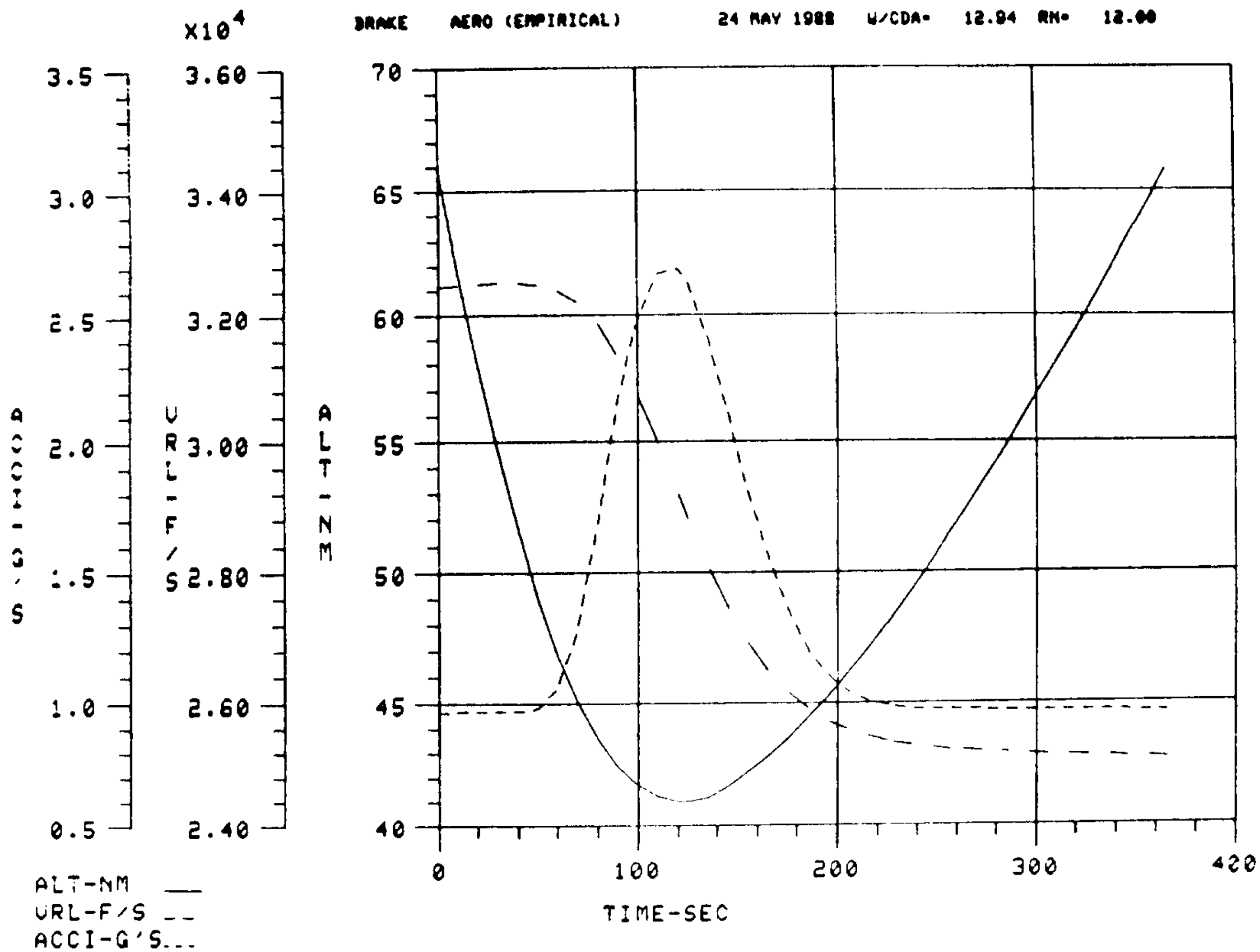
Delta V	~	Vin-Vout (brakes along V)
Average G Load	~	Delta-V/Airtime
Max G Load	~	About 3 * average

Delta-Energy and Airtime Determine Heat Loads

Delta Energy ~ Mass * (Vin ** 2 - Vout ** 2)/2

Recommendations

Use angle of attack to change CD to modulate braking. And/or use variable drag device such as ballute or flaps. Use angle of bank to modulate heading for plane control. Lift must work in harmony with drag, i.e., if more/less drag is needed, lift should pull the vehicle down/up in the atmosphere. Limits on bank angle variation ease guide logic decisions. Normal aerobody (positive CL), Nominal Bank = 180 degs. Typical brakes (Negative CL), Nominal Bank = 0 degs.



BRAKE AERO (EMPIRICAL) 24 MAY 1988 W/CDA= 12.94 RN= 12.00
X10^4
3.5
3.0
2.5
2.0
1.5
1.0
0.5
3.60
3.40
3.20
3.00
2.80
2.60
2.40
70
65
60
55
50
45
40
ACCI-G'S
VRL-F/S
ALT-NM
0
100
200
300
420
TIME-SEC
ALT-NM
VRL-F/S
ACCI-G'S

AEROBRAKING TO SURFACE
(TO A CAPTURE ORBIT IS VERY DIFFERENT)

A typical aerobraking to the surface schematic is shown on the facing page. Energy management commences when sufficient aerodynamic control is available and is used to control the maximum peak constraint values. Cross and down range controls are adjustable phases to ensure touchdown at the desired position.

What is aerobraking to surface?

A dissipation of energy using aerodynamic drag such that the vehicle will impact the surface. Lift may be used to guide to a specified recovery position (altitude/longitude/latitude).

Aerobraking Travel/Time in Atmosphere

From the top of the atmosphere to Earth site is about 2000 NM and 10 minutes of time. (varies greatly)

The algorithm used to command the aerodynamic angles of attack and bank determine time, G loads, heating and terminal position. Aerobodies have a positive lift at a positive angle of attack. Aerobrakes have a negative lift at a positive angle of attack. The guide algorithm must be tailored with this in mind.

Constraints G Loads, Peak Heating, Integrated Heat Load

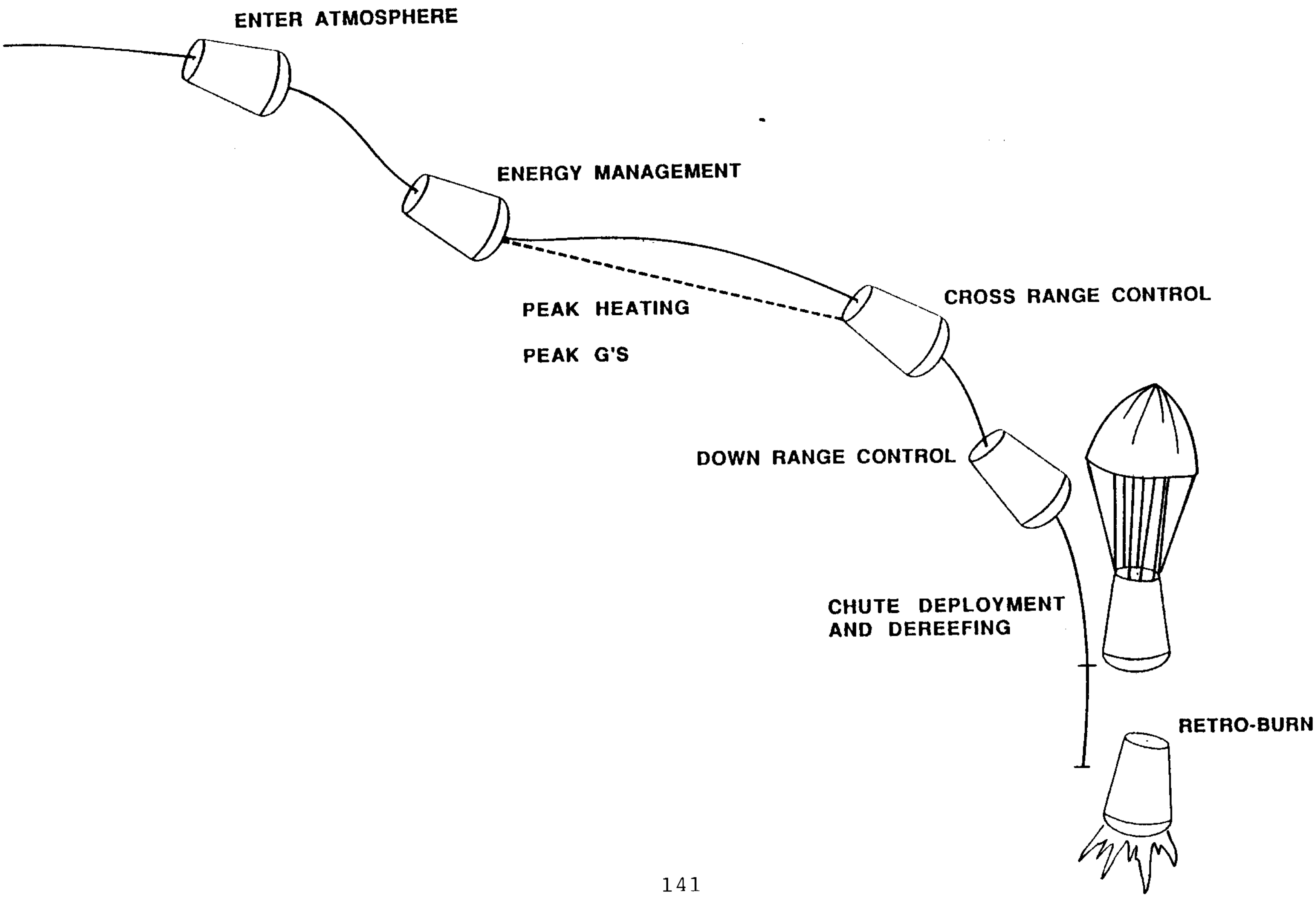
G loads set by velocity delta (DV). Heating by energy (M*V**2)/2. Peak g loads are normally controlled with angle of attack. Peak heating rates are a function of atmos density, shield curvature and initial entry state. Peak rates can be controlled by techniques that slow the vehicle high in the atmosphere. Integrated heat loads are a major factor in most entries.

Recommendations

Use high angle of attack for maximum drag and minimum airtime. Modulate angle of attack as necessary for g control. During peak lift regimes, direct lift as necessary to prevent skipout and work in concert with drag. Variable drag devices such as flaps may also be used. After slowing to reasonable velocity (mach 1 to 2) use angle of attack to control downrange and angle of bank to control crossrange.

1-4170-8

AEROBRAKING TO SURFACE



MARS MISSION VEHICLES
AERODYNAMIC CHARACTERISTICS

The hypersonic aerodynamic characteristics are shown for the Mars entry vehicle. The aero shell is assumed to be the same shape as the Apollo. The vehicle is trimmed at an angle of attack of 15° for the moment reference point shown. The data were calculated with Hypersonic Arbitrary Body Program.

MARS MISSION VEHICLES
AERODYNAMIC COEFFICIENTS

REFERENCE AREA = 6358.5 FT^2
REFERENCE LENGTH = 90 FT
MOMENT REFERENCE POINT (X, Y, Z) = 18., 0., -2.71586
BETA = 0

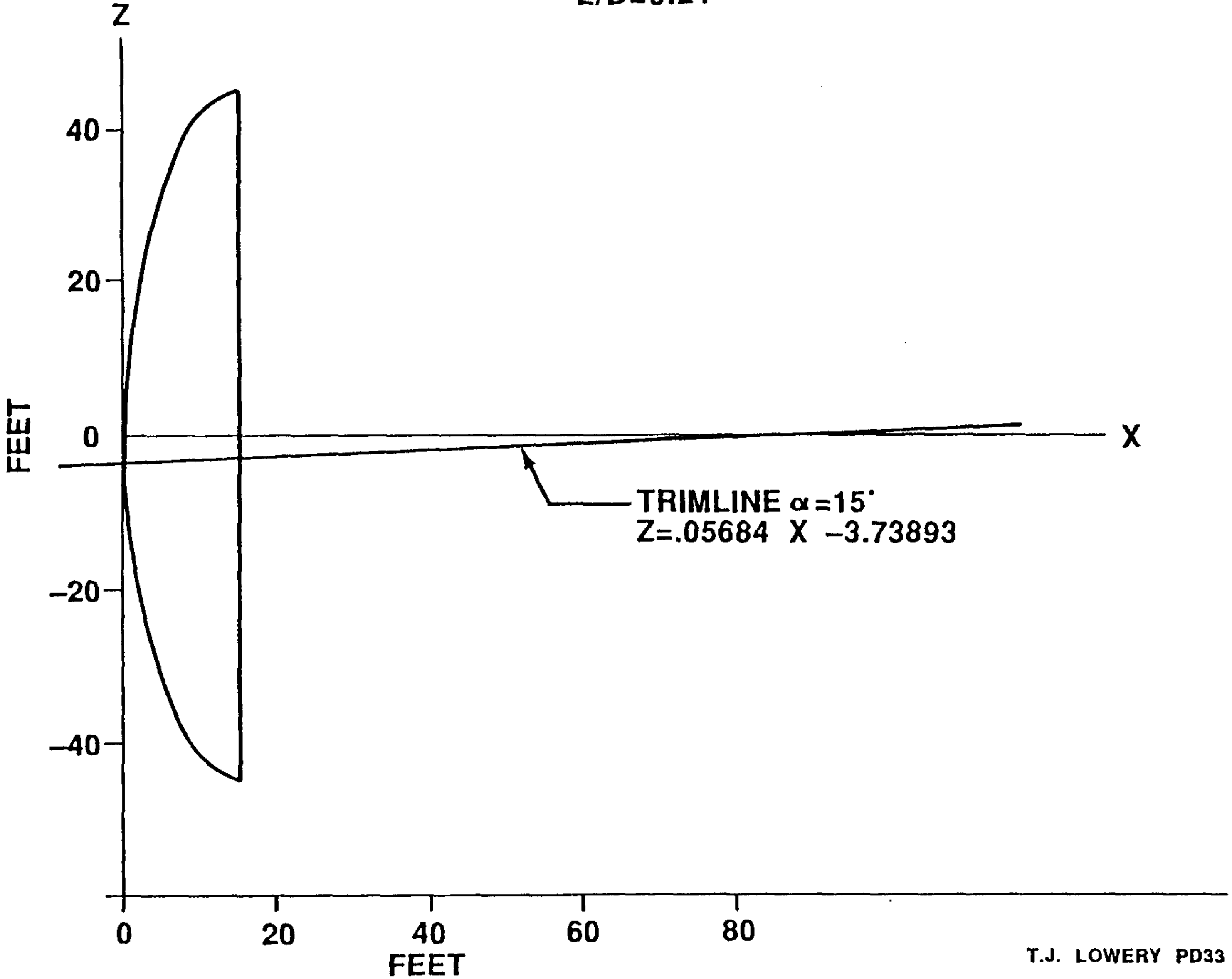
ALPHA	CD	CL	CA	CY	CN	L/D	CM
-20.00000	1.23047	0.34351	1.27376	0.00000	-0.09805	0.27917	0.09033
-15.00000	1.31145	0.27271	1.33734	0.00000	-0.07601	0.20795	0.08071
-10.00000	1.37216	0.18925	1.38417	0.00000	-0.05190	0.13792	0.06937
-6.00000	1.40419	0.11587	1.40861	0.00000	-0.03154	0.08252	0.05929
-4.00000	1.41433	0.07773	1.41631	0.00000	-0.02112	0.05496	0.05397
-2.00000	1.42045	0.03901	1.42095	0.00000	-0.01058	0.02747	0.04851
0.00000	1.42249	0.00000	1.42249	0.00000	0.00000	0.00000	0.04293
2.00000	1.42045	-0.03901	1.42095	0.00000	0.01058	-0.02747	0.03725
4.00000	1.41433	-0.07773	1.41631	0.00000	0.02112	-0.05496	0.03151
6.00000	1.40419	-0.11587	1.40861	0.00000	0.03155	-0.08252	0.02573
8.00000	1.39010	-0.15313	1.39788	0.00000	0.04182	-0.11016	0.01994
10.00000	1.37216	-0.18925	1.38417	0.00000	0.05190	-0.13792	0.01417
15.00000	1.31145	-0.27271	1.33734	0.00000	0.07601	-0.20795	0.00000
17.00000	1.28130	-0.30273	1.31382	0.00000	0.08511	-0.23627	-0.00549
20.00000	1.23047	-0.34351	1.27376	0.00000	0.09805	-0.27917	-0.01346
24.00000	1.15358	-0.38904	1.21208	0.00000	0.11380	-0.33724	-0.02344
28.00000	1.06829	-0.42348	1.14206	0.00000	0.12762	-0.39641	-0.3253
30.00000	1.02324	-0.43635	1.10433	0.00000	0.13373	-0.42644	-0.03669
35.00000	0.90608	-0.45564	1.00356	0.00000	0.14647	-0.50287	-0.04578
40.00000	0.78625	-0.45685	0.89596	0.00000	0.15543	-0.58105	-0.05285

MARS CARGO VEHICLE
CENTER OF GRAVITY TO TRIM REQUIREMENTS

The trim line is shown for the required L/D of 0.21 at an angle of attack of 15°. The trim line represents the CG locations in the X-Z plane that will yield zero pitching moment at the desired L/D. The trim line is a straight line with the equation given.

1-4251-8-18D

MARS CARGO VEHICLE AEROBRAKE
90 FT. DIAMETER
CENTER OF GRAVITY TO TRIM REQUIREMENTS
$\alpha = 15^\circ$
L/D=0.21



T.J. LOWERY PD33 9/15/87

AERODYNAMIC FLOW DURING AEROBRAKING

There are many factors that will influence the design of the aerobrakes used for Mars missions. Two of these factors that need to be considered are the shape of the aerobrake and the size of the payload or vehicle behind the aerobrake. The shape of the aerobrake determines the aerodynamic characteristics of the vehicle, and the size of the payload determines the size of the aerobrake. The aerobrake should be shaped such that it will have a lift-to-drag ratio that will allow adequate vehicle control during aerobraking, and it should be large enough so that flow past the aerobrake will not be adversely disturbed by the payload.

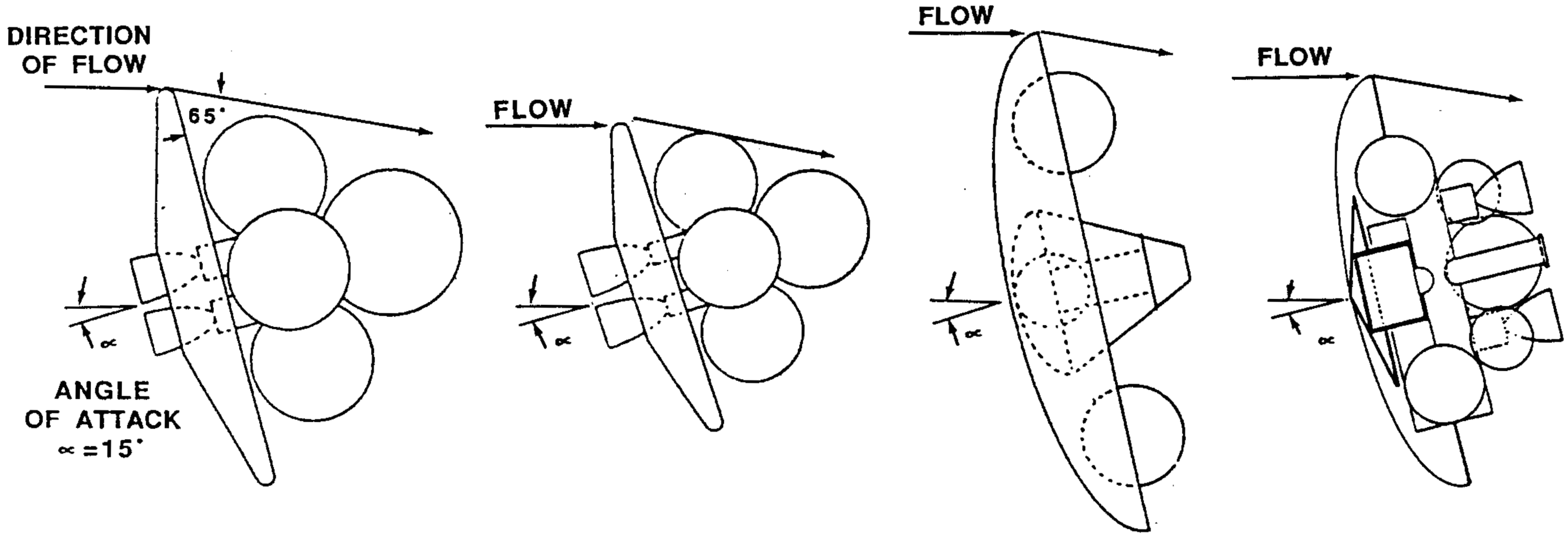
The aerobrakes selected in the SAIC Sprint Mission report were evaluated using data from a wind tunnel test conducted at Langley Research Center in 1983 (LaRC test 117), and the current design for the Aeroassist Flight Experiment. the shapes of the aerobrakes used in the SAIC report are basically round dish-shaped shells. These shapes match those tested in the wind tunnel test. It was assumed that a lift to drag ratio of .2 - .25 would allow adequate controllability during aerobraking. Based on the wind tunnel tests, the vehicle would have to be "trimmed" to fly at an angle of attack of 15 degrees to attain the proper lift-to-drag ratio. At this angle of attack, the flow bends 10 degrees as it goes past the aerobrake. This means that the flow will be at a 65 degree angle from the base of the aerobrake.

As shown on this chart, the flow would be adversely disturbed by the propellant tanks on the two Earth departure stages. This indicates that the aerobrakes should be made larger for these two stages.

1-4230-88

AERODYNAMIC FLOW DURING AEROBRAKING

FLOW AROUND AEROBRAKE FOR MARS MISSION VEHICLES
– BASED ON NASA LaRC WIND TUNNEL TEST 117, APRIL 1983



FIRST STAGE	SECOND STAGE	CARGO VEHICLE	PILOTED VEHICLE
PROPELLANT TANKS DISTURB FLOW	PROPELLANT TANKS DISTURB FLOW	PAYLOAD PLACEMENT PROVIDES ADEQUATE CLEARANCE FOR FLOW	PAYLOAD PLACEMENT PROVIDES ADEQUATE CLEARANCE FOR FLOW
AEROBRAKE SHOULD BE LARGER	AEROBRAKE SHOULD BE LARGER		

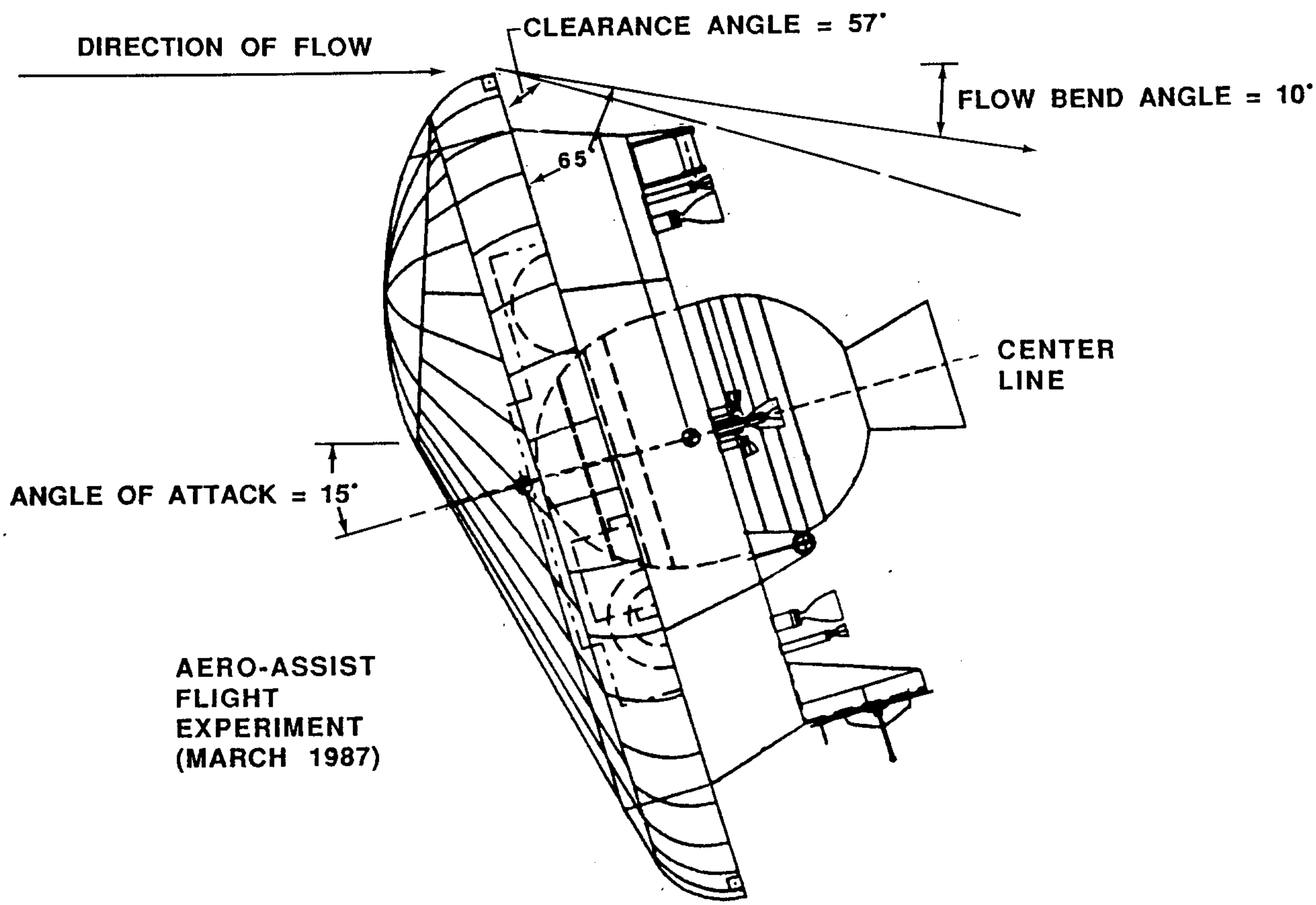
ALL CONFIGURATIONS ARE FROM SAIC/JOHN NIEHOFF REPORT

ASSUMED AERODYNAMIC FLOW FOR AEROASSIST FLIGHT EXPERIMENT

In order to determine the proper clearance angle between the edge of the aerobrake and the payload, it was assumed that the clearance angle of the Aeroassist Flight Experiment would be adequate. From the figure it can be seen that this angle is approximately 57 degrees. This was used as a guide to determine how much larger the aerobrakes for the two Earth departure stages should be.

ASSUMED AERODYNAMIC FLOW FOR AERO-ASSIST FLIGHT EXPERIMENT

– BASED ON NASA LaRC WIND TUNNEL TEST 117, APRIL 1983



MODIFICATIONS TO AEROBRAKE SIZES FOR EARTH ESCAPE STAGES

Based on data from the LaRC test 117 and the configuration of the Aeroassist Flight Experiment, the aerobrakes for the two Earth departure stages were modified. The shapes were chosen to be round spherical shells to match the shape used for the piloted and cargo vehicles. The aerobrake diameters were increased to provide a 57 degree clearance angle between the back of aerobrake and the propellant tanks. The aerobrake weights were increased by using the weight/diameter ratio of the original aerobrakes in the SAIC report.

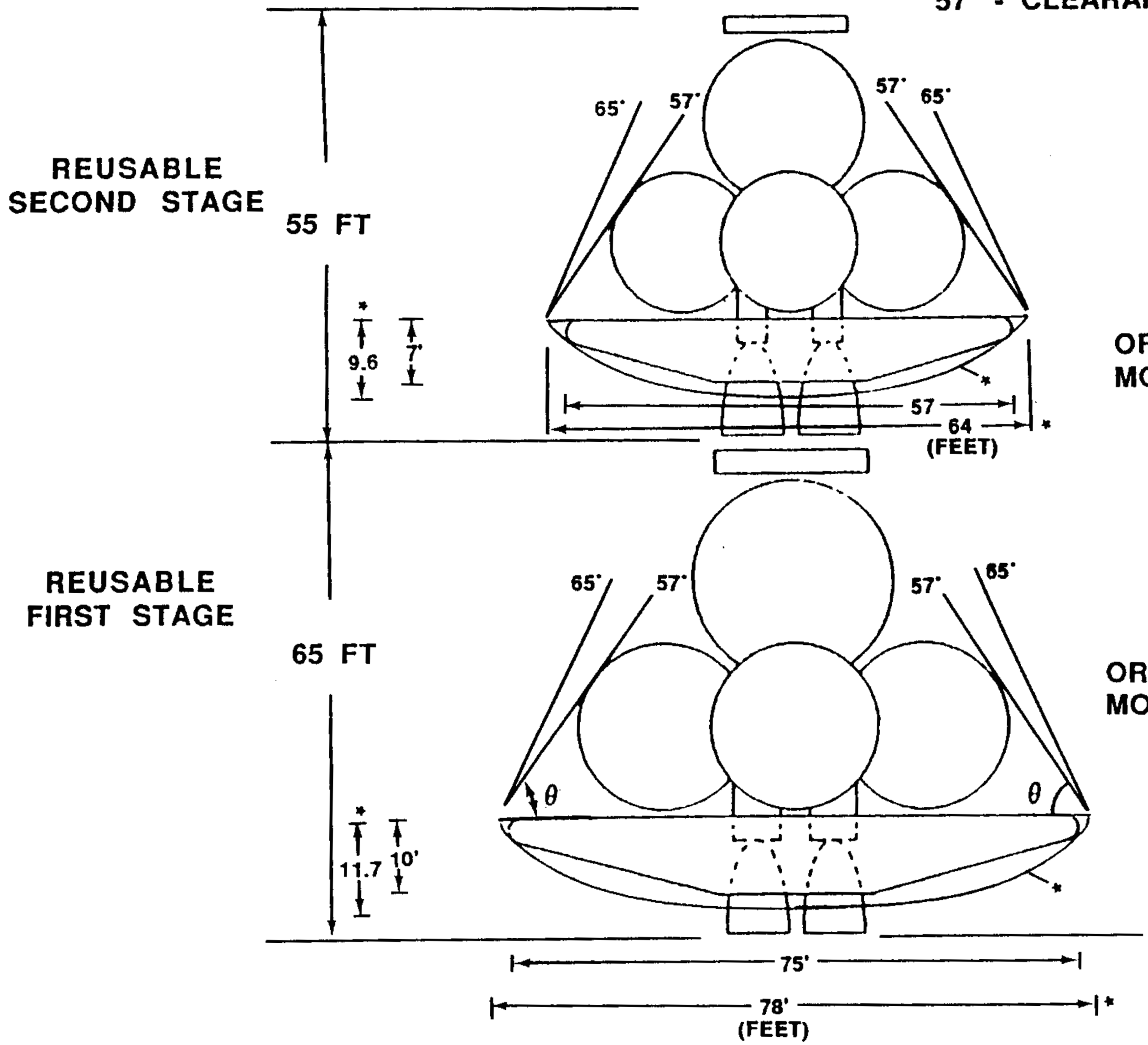
1-4233-8

MODIFICATIONS TO AEROBRAKE SIZES FOR EARTH ESCAPE STAGES

– BASED ON NASA LaRC WIND TUNNEL TEST 117 AND AERO-ASSIST FLIGHT EXPERIMENT (AFE)

65° - DIRECTION OF FLOW AT 15° ANGLE OF ATTACK
57° - CLEARANCE ANGLE BASED ON AFE

*MODIFICATION TO AEROBRAKE CONFIGURATION IN THE SAI/JOHN NIEHOFF REPORT



ORIGINAL AEROBRAKE WGT. = 5240 LBS
MODIFIED AEROBRAKE WGT. = 5884 LBS

ORIGINAL AEROBRAKE WGT. = 8820 LBS
MODIFIED AEROBRAKE WGT. = 9173 LBS

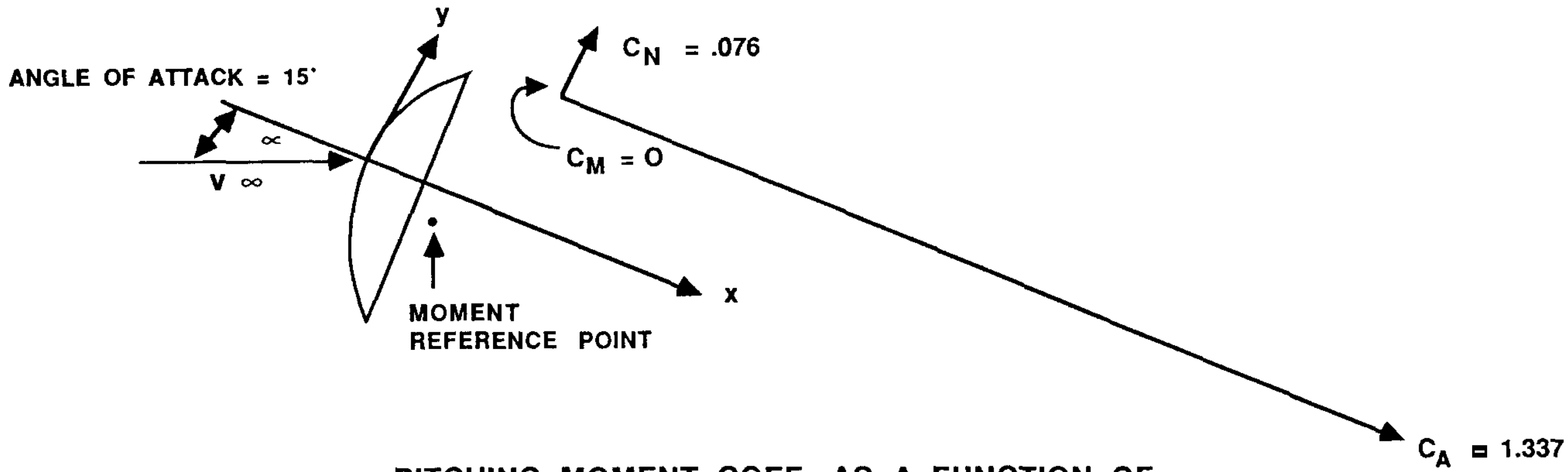
LONGITUDINAL STATIC STABILITY DURING AEROBRAKING

Using data from the Langley wind tunnel test, Joe Lowery (MSFC/PD33) calculated aerodynamic coefficients for the Mars vehicle aerobrakes. The pitching moment coefficient was used to evaluate the longitudinal static stability of the vehicles. The graph shows that if the vehicle is "trimmed" to be in equilibrium at an angle of attack of 15 degrees, the pitching moment coefficient is zero. For static stability, the vehicle should return to this position if it is disturbed. This is achieved if the slope of the curve is negative as it is here. The significance of a negative slope is that if the vehicle is disturbed such that it pitches up or down, the aerodynamic forces will act to restore the vehicle back towards its equilibrium position at 15 degrees angle of attack.

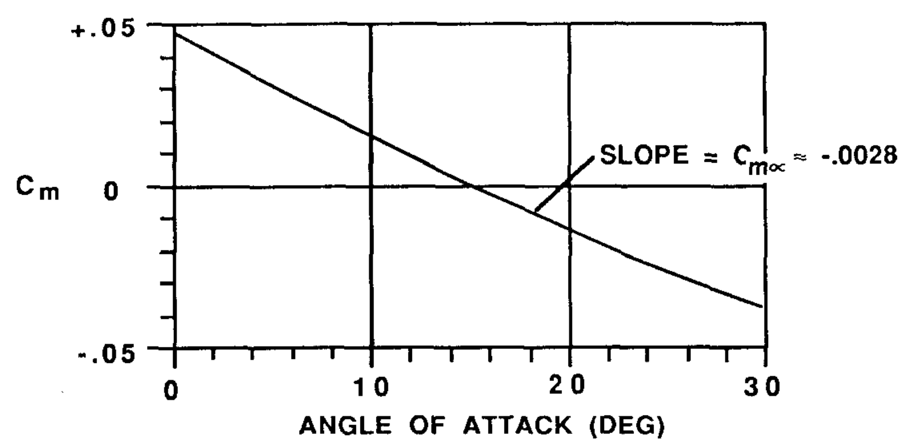
1-4235-8-18D

LONGITUDINAL STATIC STABILITY DURING AEROBRAKING

BASED ON NASA LaRC WIND TUNNEL TEST 117 APRIL 1983.



PITCHING MOMENT COEF. AS A FUNCTION OF ∝



CONDITION FOR STATIC STABILITY: $C_{m\alpha} < 0$

FOR ANY DISTURBANCE CAUSING AN INCREASE IN ∝ (CLOCKWISE ROTATION) THE AERODYNAMIC FORCES WILL RESULT IN AN INCREASE IN C_m IN THE NEGATIVE DIRECTION (COUNTER-CLOCKWISE ROTATION)

CARGO MISSION VEHICLE CONFIGURATION

This chart shows the approximate vehicle configuration and weight breakdown for the cargo mission. The configurations are taken from the SAIC/John Niehoff study report. The weights have been slightly refined as a result of this study. The 1st stage provides all of the propulsion for Earth departure and is reusable. It returns to Earth using aerobraking and can be used again for the Earth departure of the piloted mission.

1-4350-8

CARGO MISSION VEHICLE CONFIGURATION

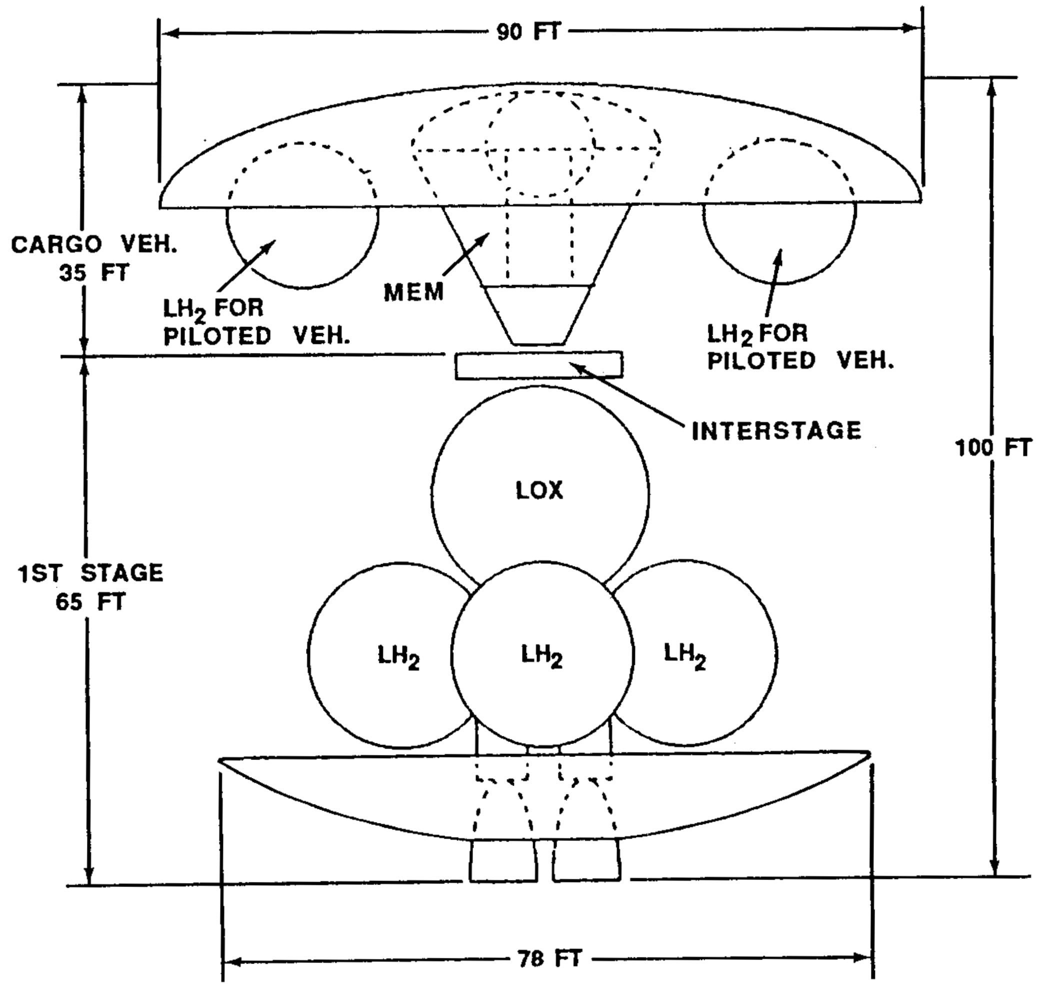
VEHICLE WEIGHTS (LBS)

CARGO VEHICLE	
ENGINES AND TANKS	11016
PROPELLANT	46173
TOTAL PROPULSION SYSTEM	57189
VEH. STRUCTURE AND SUBSYSTEMS	44500
AEROBRAKE	77276
PILOTED VEH. PROPELLANT	188253
TANKS FOR PILOTED VEH. PROP.	18908
MARS EXCURSION MODULE	133000
TOTAL GROSS WGT	519126

1ST STAGE	
ENGINES	15060
PROPELLANT TANKS	35811
PROPELLANT	790935
AEROBRAKE	9173
STRUCTURE AND SUBSYSTEMS	580
INTERSTAGE	20000
TOTAL GROSS WGT	871559

TOTAL EARTH DEPARTURE WGT 1,390,685

CARGO VEHICLE STACK AT LAUNCH



CARGO VEHICLE PROPULSION SYSTEM

This chart describes the flight control propulsion system for the cargo vehicle. The engines and tank weights were provided by Robert Champion (PD13/MSFC) based on the propellant requirements for each maneuver.

CARGO VEHICLE PROPULSION SYSTEMS

FLIGHT CONTROL SYSTEM

SEPERATE SYSTEMS ARE USED FOR EACH MANEUVER.
EACH SYSTEM IS JETTISONED AFTER MANEUVER.

MANEUVERS

- OUTBOUND MID-COURSE CORECTION
- CIRCULARIZATION AFTER MARS AEROCAPTURE
- RENDEZVOUS WITH CARGO VEHICLE
- RETURN MID-COURSE CORRECTION

ENGINE CHARACTERISTICS

ENGINE TYPE - XLR-132
PROPELLANT - NTO/MMH (STORABLE)

NUMBER OF ENGINES	=	6
TOTAL THRUST	=	125000 LBS
ISP	=	342 SEC
MIXTURE RATIO	=	2
CHAMBER PRESSURE	=	1500 PSIA
NOZZLE EXPANSION RATIO	=	400

SYSTEM WEIGHTS

NTO	=	30872
MMH	=	15391
TOTAL TANKS	=	678
ENGINES	=	9378
THRUST STRUCTURE	=	960
TOTAL SYSTEM WGT.	=	57189

CARGO VEHICLE EARTH DEPARTURE AND RECOVERY OF FIRST STAGE

This chart shows some of the performance data that was developed for the Earth departure of the cargo vehicle. The diagram shows the approximate mission profile for the Earth departure. The first stage provides all of the propulsion for Earth escape. It was assumed that after the stage separates from the cargo vehicle and the interstage is jettisoned, the first stage performs a propulsive braking maneuver to bring itself back into an Earth orbit with a period of about 24 hours. This orbit would have a perigee altitude of 407 Km and an apogee altitude of 71,319 Km. The first stage then returns to low Earth orbit by using aerobraking to transfer into a 300 Km altitude parking orbit. The stage would subsequently be retrieved from this orbit.

The Delta velocity and propellant used for each maneuver is shown, and a summary of the aerobraking analysis is listed.

1-4255-8-18D

CARGO VEHICLE EARTH DEPARTURE AND RECOVERY OF FIRST STAGE

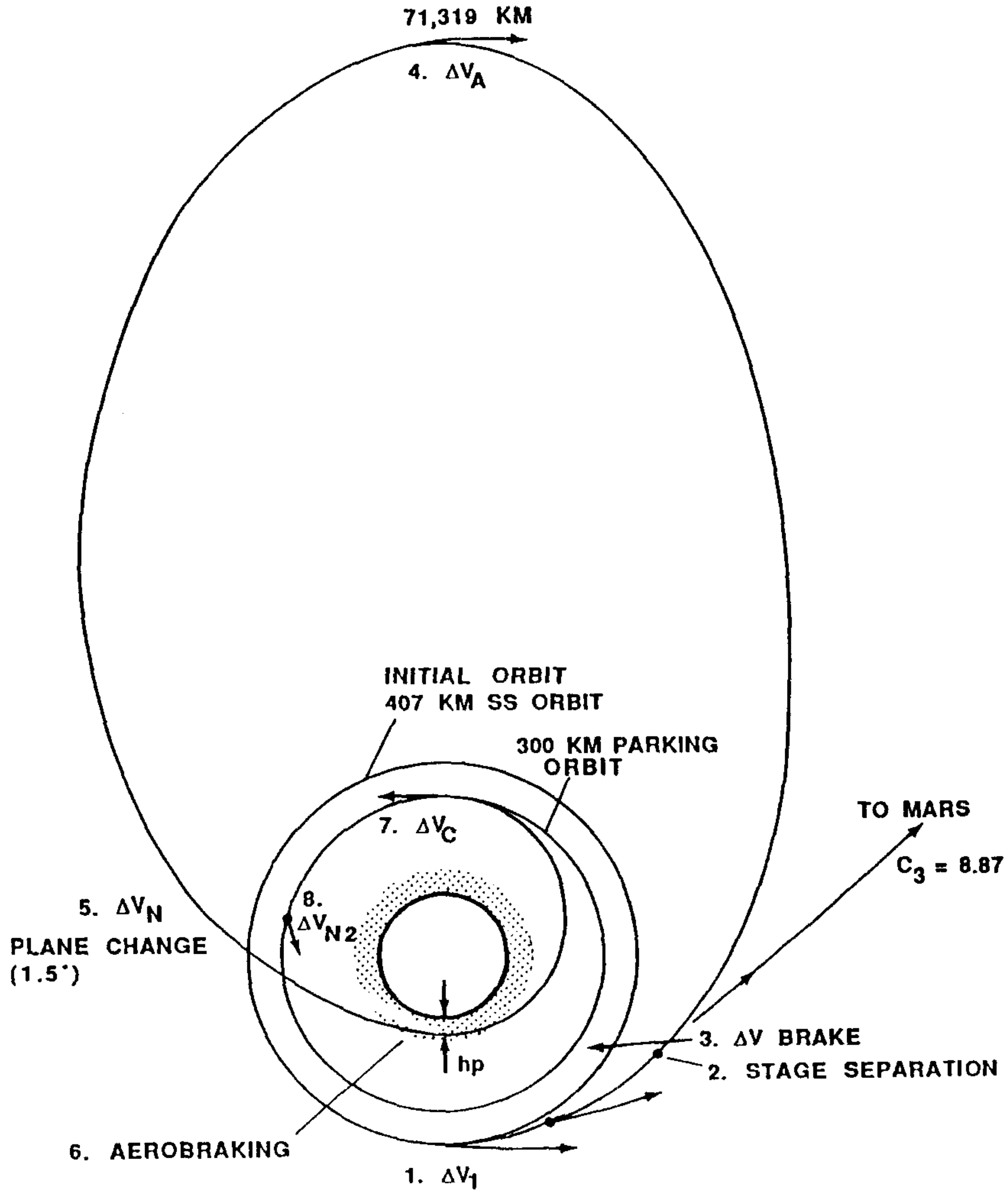
ALL PROPULSION PROVIDED BY FIRST STAGE

THRUST = 35000 LBS
ISP = 342 SEC.

1ST STAGE MANEUVERS

1. EARTH DEPARTURE, Δv_1 = 3759 M/S
   PROPELLANT USED = 766,625 LBS
2. STAGE SEPARATION
3. 1ST STAGE BRAKING BURN, Δv BRAKE = 1205 M/S
   PROPELLANT USED = 19238 LBS
4. LOWER PERIGEE INTO ATMOSPHERE, Δv_A = 23.5 M/S
   PROPELLANT USED = 330 LBS
5. 1ST PLANE CHANGE, Δv_{N2} = 187 M/S
   PROPELLANT USED = 2565 LBS
6. AEROBRAKING
   AEROBRAKE DIA. = 78 FT
   BALLISTIC COEFF. = 8.48 LBS/FT^2
   HP = 81.13 KM
   MAX. DYNAMIC PRES. = 13.77 LBS/FT^2
   MAX. ACCELERATION = 2.12 G'S
   AVG. ACCELERATION = 1.18 G'S
   TIME IN ATMOSPHERE = 769 SEC.
7. CIRCULARIZATION, Δv_C = 64 M/S
   PROPELLANT USED = 855 LBS
8. 2ND PLANE CHANGE, Δv_{N2} = 101 M/S
   PROPELLANT USED = 1322 LBS

TOTAL PROPELLANT USED = 790935
TIME FROM LAUNCH TO STAGE RETURN = 27.25 HRS



CARGO VEHICLE PROPULSION SYSTEM MANEUVERS

Except for the outbound mid-course correction, the cargo vehicle performs all of its maneuvers at Mars. These maneuvers are shown in this diagram. Maneuvers 2 and 3 are performed when the cargo vehicle first arrives at Mars. Aerobraking is used to reduce the vehicle's energy and insert it into a 300 X 4381 Km altitude parking orbit. This parking orbit was selected such that the cargo vehicle is in the same orbital plane as the piloted vehicle when it arrives. Maneuvers 4 through 7 are performed after the piloted vehicle arrives at Mars. Aerobraking is used a second time to transfer from the parking orbit to 980 Km altitude circular phasing orbit. This is followed by a transfer to the 1000 Km altitude piloted vehicle orbit.

The Delta velocity, and the propellant used for each maneuver, is listed, and summaries of the two aerobraking maneuvers are given.

1-4256-8-18D

CARGO VEHICLE PROPULSION SYSTEM MANEUVERS MARS ARRIVAL PROFILE

CARGO VEHICLE PROPULSION SYSTEM

THRUST = 125,000 LBS.
ISP = 342 SEC.

MANEUVERS

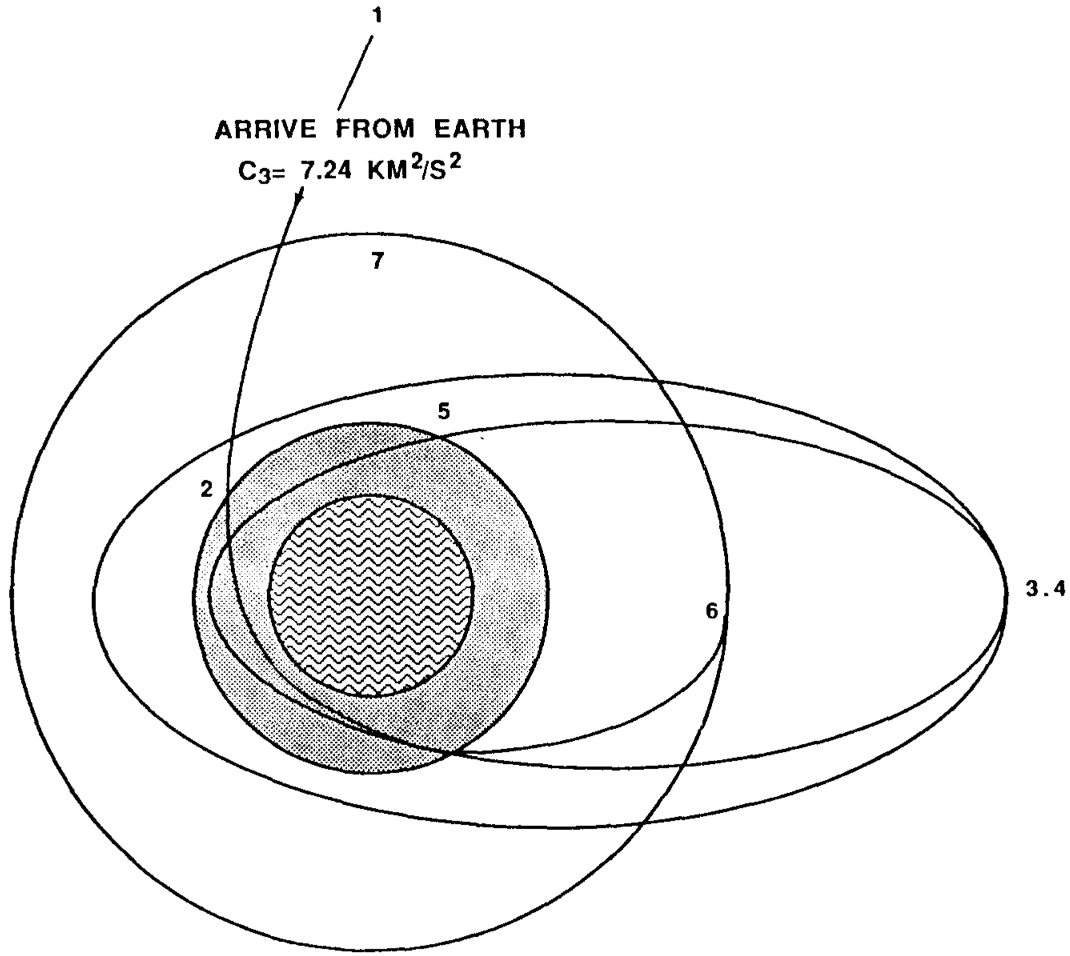
AEROCAPTURE INTO PARKING ORBIT

1. OUTBOUND MID COURSE CORRECTION
   ΔV = 25 M/S
   PROPELLANT USED = 3884
2. AEROCAPTURE AT MARS
   BALLISTIC COEF. = 51.91 LBS/FT^2
   hp = 47.68 KM
   MAX. DYN. PRESS. = 47.19 LBS/FT^2
   MAX ACCELERATION = 1.08 g's
   AVG. ACCELERATION = .57 g's
3. INSERT INTO 300 X 4381 KM PARKING ORBIT
   ΔV = 45 M/S
   PROPELLANT USED = 6877 LBS

RENDEZVOUS WITH PILOTED VEHICLE

4. LOWER PERIAPSIS INTO ATMOSPHERE (hp = 56.9 KM)
   ΔV = 44 M/S
   PROPELLANT USED = 6496 LBS
5. AEROBRAKE TO LOWER APOAPSIS
   BALLISTIC COEF. = 50.2 LBS/FT^2
   hp = 56.87 KM
   MAX. DYN. PRES.= 7.79 LBS/FT^2
   MAX ACCELERATION = .43 g's
   AVG. ACCELERATION = .38 g's
6. INSERT INTO 980 KM PHASING ORBIT
   ΔV = 191 M/S
   PROPELLANT USED = 27767 LBS
7. TRANSFER TO 1000 KM CIRCULAR ORBIT
   ΔV = 8.2 M/S
   PROPELLANT USED = 1149 LBS

TOTAL PROPELLANT USED = 46173 LBS



CARGO VEHICLE PERFORMANCE SUMMARY

This chart summarizes the performance analysis for each of the maneuvers performed by the cargo vehicle. The propellant weights for each maneuver include a flight performance reserve equivalent to .75% of the delta velocity (or Isp/1.0075). The tank, engine, and thrust structure weight for each maneuver is listed since they are jettisoned after each maneuver.

CARGO VEHICLE PREFORMANCE SUMMARY

1. OUTBOUND MID-COURSE CORRECTION

FLIGHT CONTROL SYSTEM

THRUST	=	125000 LBS
ISP	=	342 SEC
DELTA VELOCITY	=	25 M/S
PROPELLANT USED	=	3884 LBS
PROP. TANKS	=	30 LBS
ENGINES (2)	=	3126 LBS
THRUST STRUCTURE	=	320 LBS
FINAL WEIGHT	=	511742 LBS

2. PARKING ORBIT INSERTION AFTER MARS AEROCAPTURE (300 X 4381 KM ORBIT)

FLIGHT CONTROL SYSTEM

THRUST	=	125000 LBS
ISP	=	342 SEC
DELTA VELOCITY	=	45 M/S
PROPELLANT USED	=	6877 LBS
PROP. TANKS	=	97 LBS
ENGINES (2)	=	3126 LBS
THRUST STRUCTURE	=	320 LBS
FINAL WEIGHT	=	501322 LBS

3. RENDEZVOUS WITH PILOTED VEHICLE (AFTER 2ND AEROBRAKING)

FLIGHT CONTROL SYSTEM

THRUST	=	125000 LBS	
ISP	=	342 SEC	
DELTA VELOCITY	=	244 M/S	
PROPELLANT USED	=	31412 LBS	
PROP. TANKS	=	527 LBS	(NOT JETTISONED)
ENGINES (2)	=	3126 LBS	(NOT JETTISONED)
THRUST STRUCTURE	=	320 LBS	(NOT JETTISONED)
FINAL WEIGHT	=	465910 LBS	

PILOTED MISSION VEHICLE CONFIGURAITON

This chart shows the approximate vehicle configuration and weight breakdown for the piloted mission. The first two stages are used to attain Earth escape velocity and return to low Earth orbit after stage separation. The piloted vehicle's main propulsion system is used to achieve the final velocity required for the transfer to Mars.

1-4353-8-19D

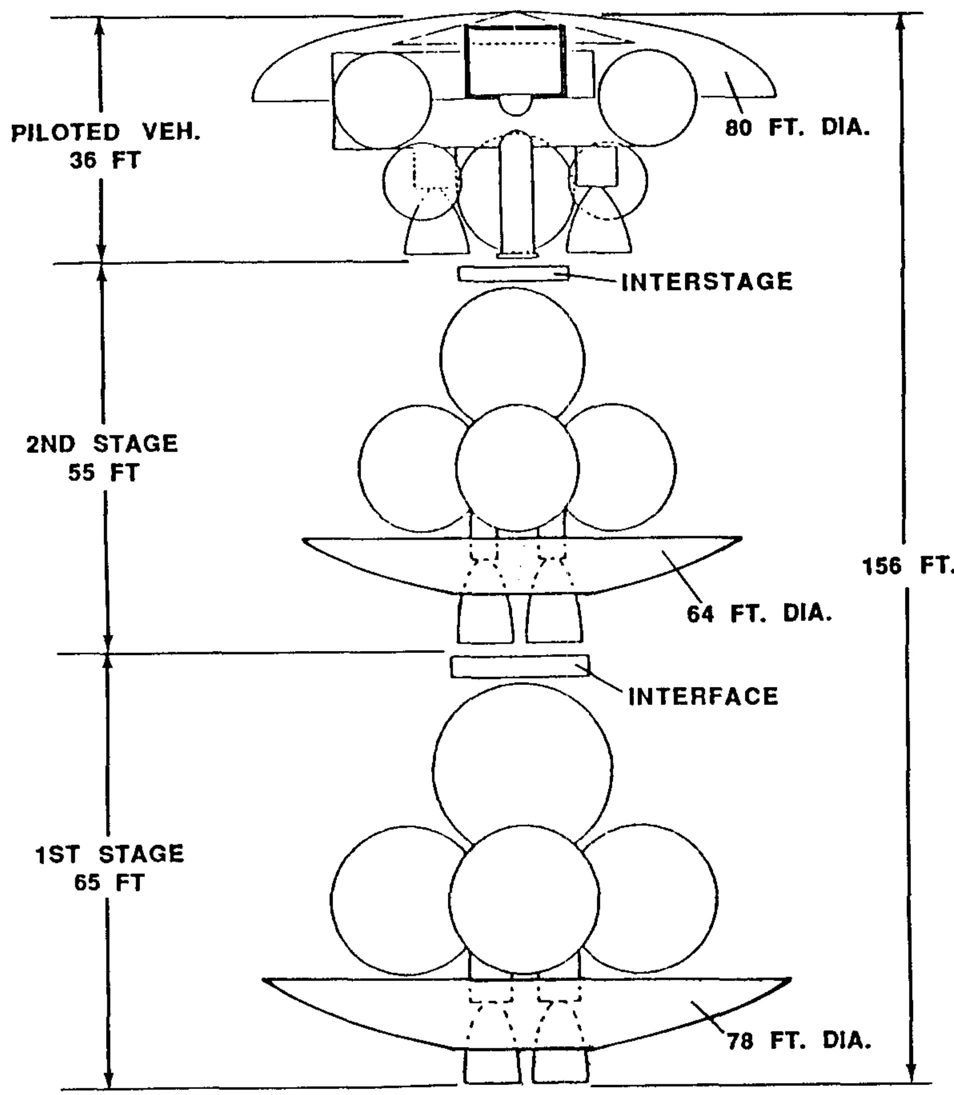
PILOTED MISSION VEHICLE CONFIGURATION

VEHICLE WEIGHTS (LBS)

PILOTED VEHICLE		
MAIN ENGINES (6)	=	2946
PROPELLANT TANKS	=	6870
PROPELLANT	=	188253
TOT. MAIN PROP. SYSTEM	=	198069
FLIGHT CONTROL SYSTEM	=	22325
AEROBRAKE	=	35500
CREW MODULES	=	131090
STRUCTURES AND SUBSYSTEMS	=	10320
EARTH RETURN CAPSULE	=	22032
TOTAL GROSS WGT.	=	419336
2ND STAGE		
ENGINES (4)	=	10040
PROPELLANT TANKS	=	17760
PROPELLANT	=	347220
AEROBRAKE	=	5884
SUBSYSTEMS	=	400
INTERSTAGE	=	20000
TOTAL GROSS WGT	=	401304
1ST STAGE		
ENGINES (6)	=	15060
PROPELLANT TANKS	=	35811
PROPELLANT	=	658938
AEROBRAKE	=	9173
SUBSYSTEMS	=	580
INTERSTAGE	=	20000
TOTAL GROSS WGT	=	739562

TOTAL GROSS DEPARTURE WGT = 1,560,202 LBS

PILOTED VEHICLE STACK AT LAUNCH



PILOTED VEHICLE PROPULSION SYSTEMS

This chart describes the two propulsion systems that are used by the piloted vehicle. The engines and tank weights were provided by Robert Champion (PD13/MSFC) based on the propellant requirements for each system.

PILOTED VEHICLE PROPULSION SYSTEMS

MAIN PROPULSION SYSTEM

MANEUVERS
- EARTH DEPARTURE (3RD STAGE)
- MARS DEPARTURE

ENGINE CHARACTERISTICS
- ENGINE TYPE - ADVANCED RL10 ENGINES
- PROPELLANT - LOX/LH2 (CRYOGENIC)

NUMBER OF ENGINES	=	6
TOTAL THRUST	=	90000 LBS
ISP	=	482 SEC
MIXTURE RATIO	=	6
CHAMBER PRESSURE	=	1500 PSIA
NOZZLE EXPANSION RATIO	=	642

SYSTEM WEIGHTS

LIQUID OXYGEN	=	161360
LIQUID HYDROGEN	=	26893
LOX TANKS	=	2063
LH2 TANK	=	4538
ENGINES	=	2946
THRUST STRUCTURE	=	269
TOTAL SYSTEM WGT.	=	198069

FLIGHT CONTROL SYSTEM

SEPERATE SYSTEMS ARE USED FOR EACH MANEUVER.
EACH SYSTEM IS JETTISONED AFTER MANEUVER.

MANEUVERS
- OUTBOUND MID-COURSE CORECTION
- CIRCULARIZATION AFTER MARS AEROCAPTURE
- RENDEZVOUS WITH CARGO VEHICLE
- RETURN MID-COURSE CORRECTION

ENGINE CHARACTERISTICS
- ENGINE TYPE - XLR-132
- PROPELLANT - NTO/MMH (STORABLE)

NUMBER OF ENGINES	=	8
TOTAL THRUST	=	50000 LBS
ISP	=	342 SEC
MIXTURE RATIO	=	2
CHAMBER PRESSURE	=	1500 PSIA
NOZZLE EXPANSION RATIO	=	400

SYSTEM WEIGHTS

NTO	=	11179
MMH	=	5590
TOTAL TANKS	=	236
ENGINES	=	5000
THRUST STRUCTURE	=	320
TOTAL SYSTEM WGT.	=	22325

AEROBRAKING OF THE TRANSFER STAGES FOR THE MARS PILOTED VEHICLE

This chart summarizes the performance analysis that was performed for the Earth departure of the piloted mission. This analysis included analyses of the recovery of the two transfer stages using aerobraking. The first stage provides the initial delta velocity increment for the piloted vehicle. After separation, the first stage enters a highly elliptical orbit. When it reaches the apogee of this orbit, the perigee is lowered into the atmosphere and aerobraking is used to bring the stage back to a 300 kilometer altitude orbit. Appropriate plane changes are made to put the stage into the Space Station orbit plane.

The second stage provides enough delta velocity for the piloted vehicle to escape Earth. After separation, the stage brakes itself into a 24 hour orbit, then uses aerobraking to return to the same orbit as the first stage.

The piloted vehicle's main propulsion system provides the final velocity increment needed to achieve the necessary Earth departure velocity.

1-5826-7

AEROBRAKING OF THE TRANSFER STAGES FOR THE MARS PILOTED VEHICLE

1ST STAGE

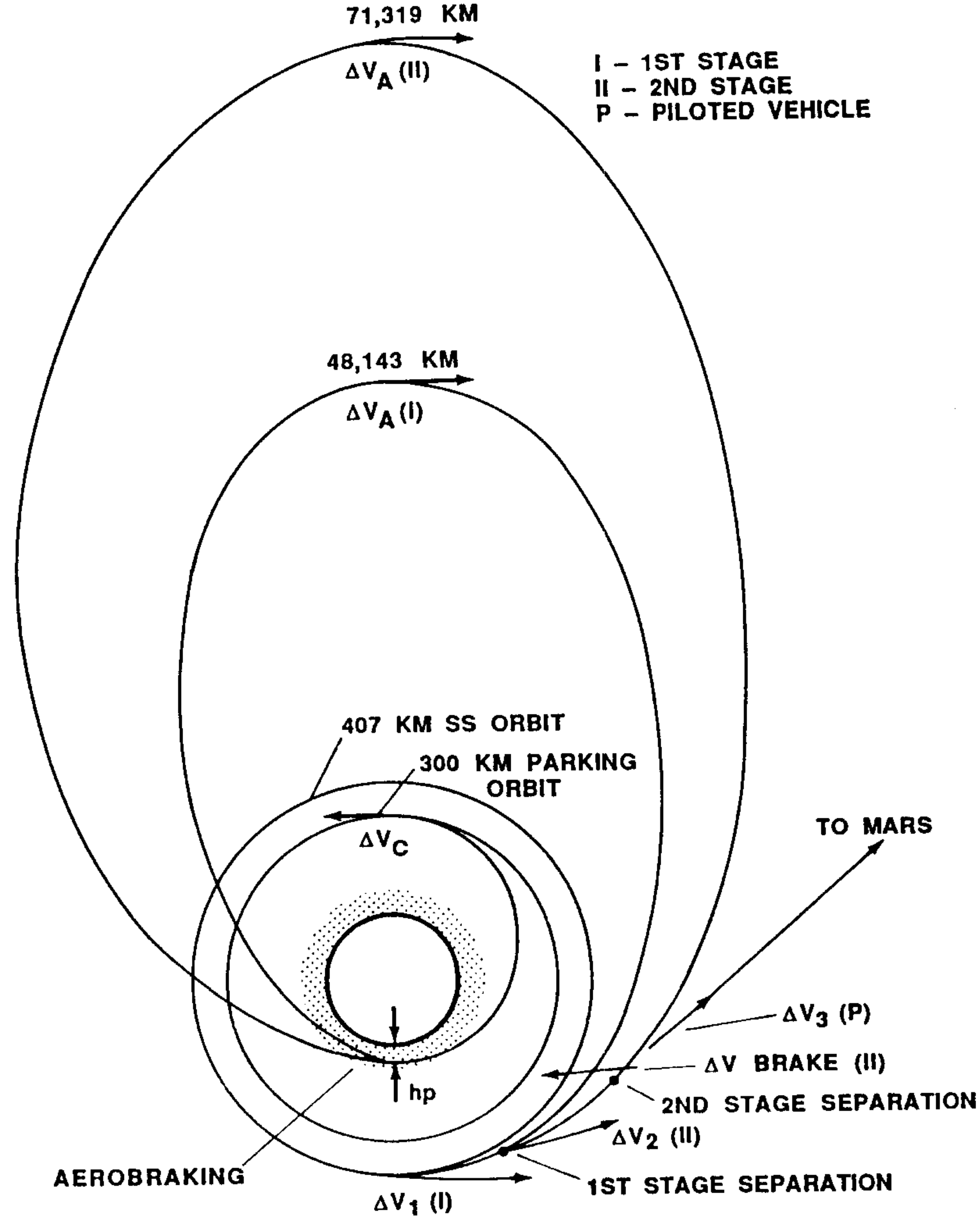
I_{SP}	= 482 SEC
ΔV_1	= 2568 M/S
ΔV_A	= 27 M/S
ΔV_C	= 64 M/S
PROPELLANT REQUIRED	= 6,589,291 LBS
AEROBRAKE DIA.	= 78 FT.
BALLISTIC COEF.	= 7.76 LBS/FT2
hp	= 85.82 KM
TIME IN ATMOSPHERE	= 763 SEC
MAX ACCELERATION	= 2.0 G's
AVG. ACCEL	= 1.16 g's
MAX DYN. PRES	= 11.61 LBS/FT^2
TIME TO RECOVER STAGE	= 18.7 HRS

2ND STAGE

I_{SP}	= 482 SEC
ΔV_2	= 2297 M/S
ΔV_{BRAKE}	= 2842 M/S
ΔV_A	= 20 M/S
ΔV_C	= 64 M/S
PROPELLANT REQUIRED	= 324700 LBS
AEROBRAKE DIA	= 64 FT.
BALLISTIC COEF.	= 6.94 LBS/FT^2
hp	= 86.19 KM
TIME IN ATMOSPHERE	= 757 SEC
MAX ACCELERATION	= 2.12 g's
AVG ACCELERATION	= 1.18 g's
MAX DYN. PRESSURE	= 11.25 LBS/FT^2
TIME TO RECOVER STAGE	= 24.7 HRS

PILOTED VEHICLE

I_{SP}	= 482 SEC
THRUST	= 90000 LBS
ΔV_3	= 2796 M/S
PROPELLANT REQUIRED	= 187316 LBS



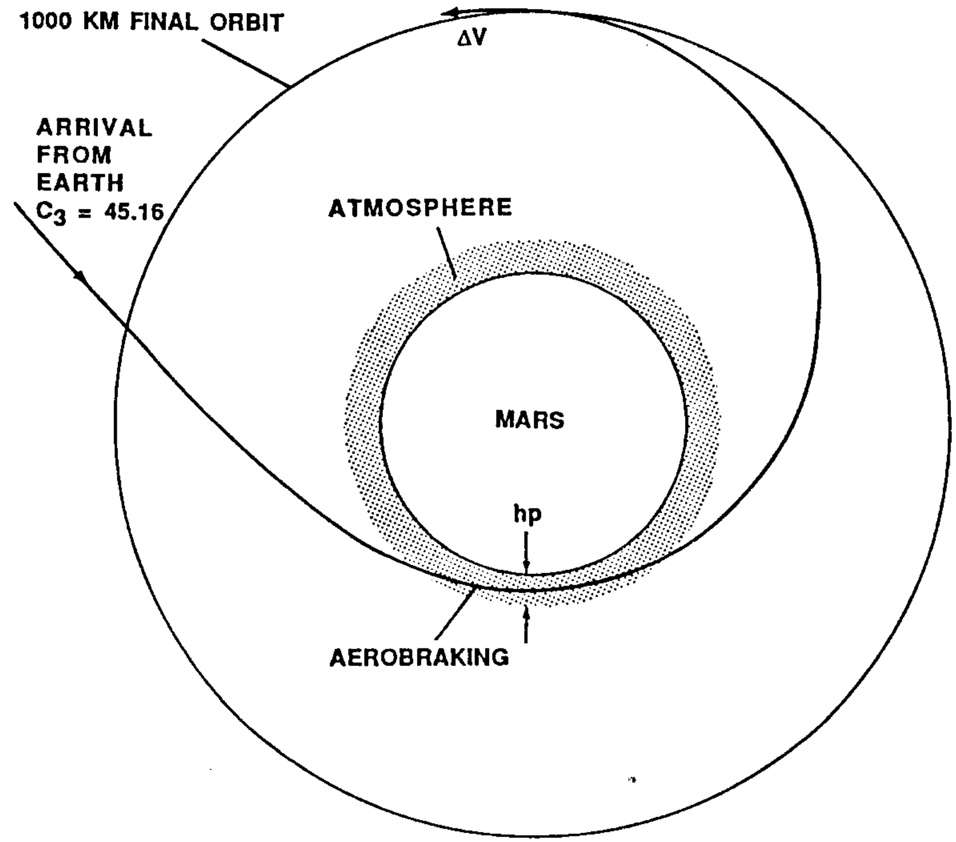
AEROCAPTURE OF THE PILOTED VEHICLE AT MARS

The integrated mission program was used to analyze the capture of the piloted vehicle into the Mars orbit. The piloted vehicle arrives at Mars with a C_3 of 45.16 Km^2/Sec^2, and uses aerobarking to reduce its energy enough to enter a transfer orbit with a 1000 Km altitude apoapsis. When the vehicle reaches apoapsis, the piloted vehicle uses its flight control propulsion to circularize and enter the final 1000 Km altitude parking orbit. A summary of the aerobraking analysis is shown on the chart.

1-4355-8-19D

AEROCAPTURE OF THE PILOTED VEHICLE AT MARS

PILOTED VEHICLE MARS AEROCAPTURE PROFILE



INCOMMING C_3		= 45.16 KM^2/SEC^2
PROPULSION SYSTEM:	FLIGHT CONTROL SYSTEM	
	ISP	= 342 SEC
	THRUST	= 50000 LBS
PILOTED VEHICLE WGT		= 228001 LBS

AEROBRAKING SUMMARY

AEROBRAKE DIA.	= 80 FT
BALLISTIC COEFF.	= 29.27 LBS/FT^2
hp	= 48.45 KM
MAX. DYN. PRESS.	= 93.35 LBS/FT^2
MAX. ACCELERATION	= 3.38 G'S
AVG. ACCELERATION	= .93 G'S
TIME IN ATMOSPHERE	= 711 SEC.

CIRCULARIZATION

ΔV	= 195.4 M/S
PROPELLANT USED	= 12904 LBS

PILOTED VEHICLE PERFORMANCE SUMMARY

This chart summarizes the performance analysis for each of the maneuvers performed by the piloted vehicle. The propellant used for each maneuver is listed. This includes a flight performance reserve equivalent to .75% of the delta velocity (or Isp/1.0075). The tank, engine, and thrust structure weights are listed for the maneuvers if they are jettisoned after the maneuver. The aerobrake is jettisoned after the third maneuver. Allowance was made for the piloted vehicle to rendezvous with the cargo vehicle (maneuver 4). This is a redundant capability since the cargo vehicle also has the capability to rendezvous with the piloted vehicle.

PILOTED VEHICLE PREFORMANCE SUMMARY

PILOTED VEHICLE GROSS WGT = 419336 LBS

1. EARTH DEPARTURE (3RD STAGE)

MAIN PROPULSION SYSTEM

THRUST	= 90000 LBS
ISP	= 482 LBS
DELTA VELOCITY	= 2796 M/S
PROPELLANT USED	= 187316 LBS
RESIDUALS	= 937 LBS
FINAL WEIGHT	= 231083 LBS

2. OUTBOUND MID-COURSE CORRECTION

FLIGHT CONTROL SYSTEM

THRUST	= 50000 LBS
ISP	= 342 SEC
DELTA VELOCITY	= 25 M/S
PROPELLANT USED	= 1729 LBS
PROP. TANKS	= 23 LBS
ENGINES (2)	= 1250 LBS
THRUST STRUCTURE	= 80 LBS
FINAL WEIGHT	= 228001 LBS

3. CICULARIZATION AFTER MARS AEROCAPTURE

FLIGHT CONTROL SYSTEM

THRUST	= 50000 LBS
ISP	= 342 SEC
DELTA VELOCITY	= 195 M/S
PROPELLANT USED	= 12998 LBS
PROP. TANKS	= 186 LBS
ENGINES (2)	= 1250 LBS
THRUST STRUCTURE	= 80 LBS
AEROBRAKE	= 35500 LBS
FINAL WEIGHT	= 177987 LBS

4. RENDEZVOUS WITH CARGO VEHICLE (ROUND TRIP FROM 1000 KM ORBIT TO 980 KM ORBIT)

FLIGHT CONTROL SYSTEM

THRUST	= 50000 LBS
ISP	= 342 SEC
DELTA VELOCITY	= 15 M/S
PROPELLANT USED	= 800 LBS
PROP. TANKS	= 11 LBS
ENGINES (2)	= 1250 LBS
THRUST STRUCTURE	= 80 LBS
FINAL WEIGHT	= 175846 LBS

MAIN PROPULSION SYSTEM REFUELING

PROPELLANT WGT.	= 187316 LBS
VEHICLE GROSS WGT.	= 363162 LBS

5. MARS DEPARTURE

MAIN PROPULSION SYSTEM

THRUST	= 90000 LBS
ISP	= 482 LBS
DELTA VELOCITY	= 3339 M/S
PROPELLANT USED	= 187316 LBS
ENGINES	= 2946 LBS
TANKS AND STRUCTURE	= 6870 LBS
FINAL WEIGHT	= 166030 LBS

6. RETURN MID-COURSE CORRECTION

FLIGHT CONTROL SYSTEM

THRUST	= 50000 LBS
ISP	= 342 SEC
DELTA VELOCITY	= 25 M/S
PROPELLANT USED	= 1242 LBS
PROP. TANKS	= 16 LBS
ENGINES (2)	= 1250 LBS
THRUST STRUCTURE	= 80 LBS
FINAL WEIGHT	= 163442 LBS

MARS EXCURSION MODULE

The Mars excursion module used in the SAIC study is based on a concept that was originally developed by Rockwell in 1968, then modified by MSFC in 1986. The propulsion and parachute requirements were found using the integrated mission program and the chutes program. The integrated mission program was used to simulate the trajectory from the deorbit burn to leave the parking orbit to the top of the atmosphere. The exact location of this burn was determined so the vehicle would land at a site 40 degrees north latitude, 140 degrees east longitude. The size of the parachutes and the drag factor were based on the past studies. The drag factor is a combination of the drag coefficient, porosity and shape of the parachutes. The thrust was found by iteration such that the touchdown velocity would be less than 1 m/s.

1-4247-8

MARS EXCURSION MODULE

VEHICLE ASSUMPTIONS

CREW OF 4

GROSS WGT. = 133000 LBS

PROPELLANT WGT. = 17680 LBS

AEROBRAKE DIA. = 30 FT

BALLISTIC COEFFICIENT = 106 LBS/FT^2

NUMBER OF ENGINES = 6

TOTAL THRUST = 90000 LBS

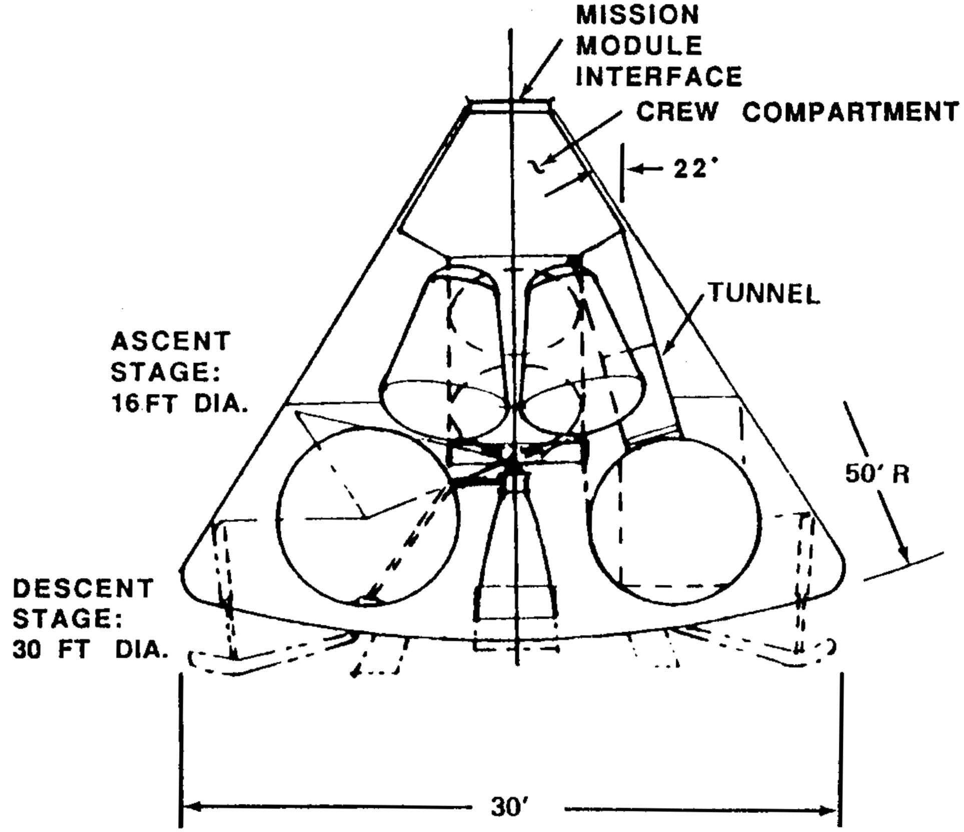
ISP = 342 SEC

NUMBER OF PARACHUTES = 3

PARACHUTE DIA. = 35 FT

DRAG FACTOR = .56

CONFIGURATION CONCEPT*



*TUCKER, MEREDITH AND BROTHERS (MSFC), SPACE VEHICLE CONCEPTS. MANNED MARS MISSION WORKING GROUP PAPERS, VOL. I., NASA M002, JUNE 1986

MARS EXCURSION MODULE DESCENT TRAJECTORY

This chart shows the descent trajectory of the Mars Excursion Module. The detailed trajectory was calculated using the chutes program. The descent profile is summarized below.

DESCENT PROFILE

1. Deboost from 1000 km altitude parking orbit to 46.6 x 1000 km altitude orbit

   Delta Velocity = 196 m/s
   Propellant Used = 7606 lbs

2. @ 100 km altitude: Enter atmosphere,

   Velocity = 3614 m/s
   Flight Path Angle = -3.25 deg

3. @ 20 km altitude: Deploy chutes to 25% open

   Velocity = 2055 m/s (Mach 9)

4. @ 16 km altitude: Dereef chutes to 50% open

   Velocity = 1076 m/s (Mach 4.6)

5. @ 14.3 km altitude: Dereef chutes to fully open

   Velocity = 712 m/s (Mach 3)

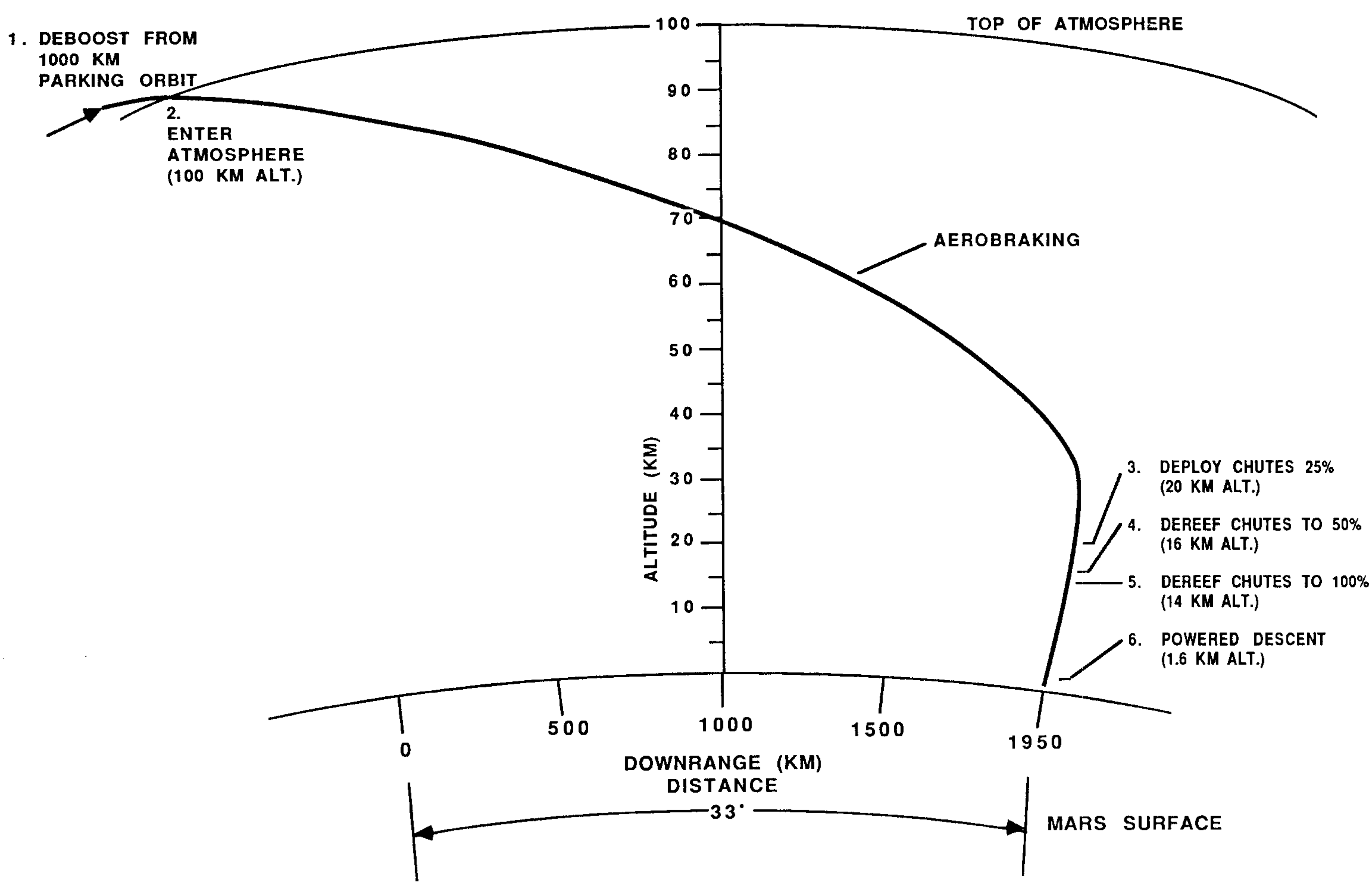
6. @ 1.57 km altitude: Begin powered descent

   Velocity = 132 m/s
   Thrust = 90000 lbs
   Burn Time = 38 sec
   Propellant Used = 10074 lbs

The entire landing takes 67 minutes from deboost to touchdown; it takes approximately 13 minutes to travel from the top of the atmosphere to touchdown.

1-4246-8-18D

MARS EXCURSION MODULE DESCENT TRAJECTORY

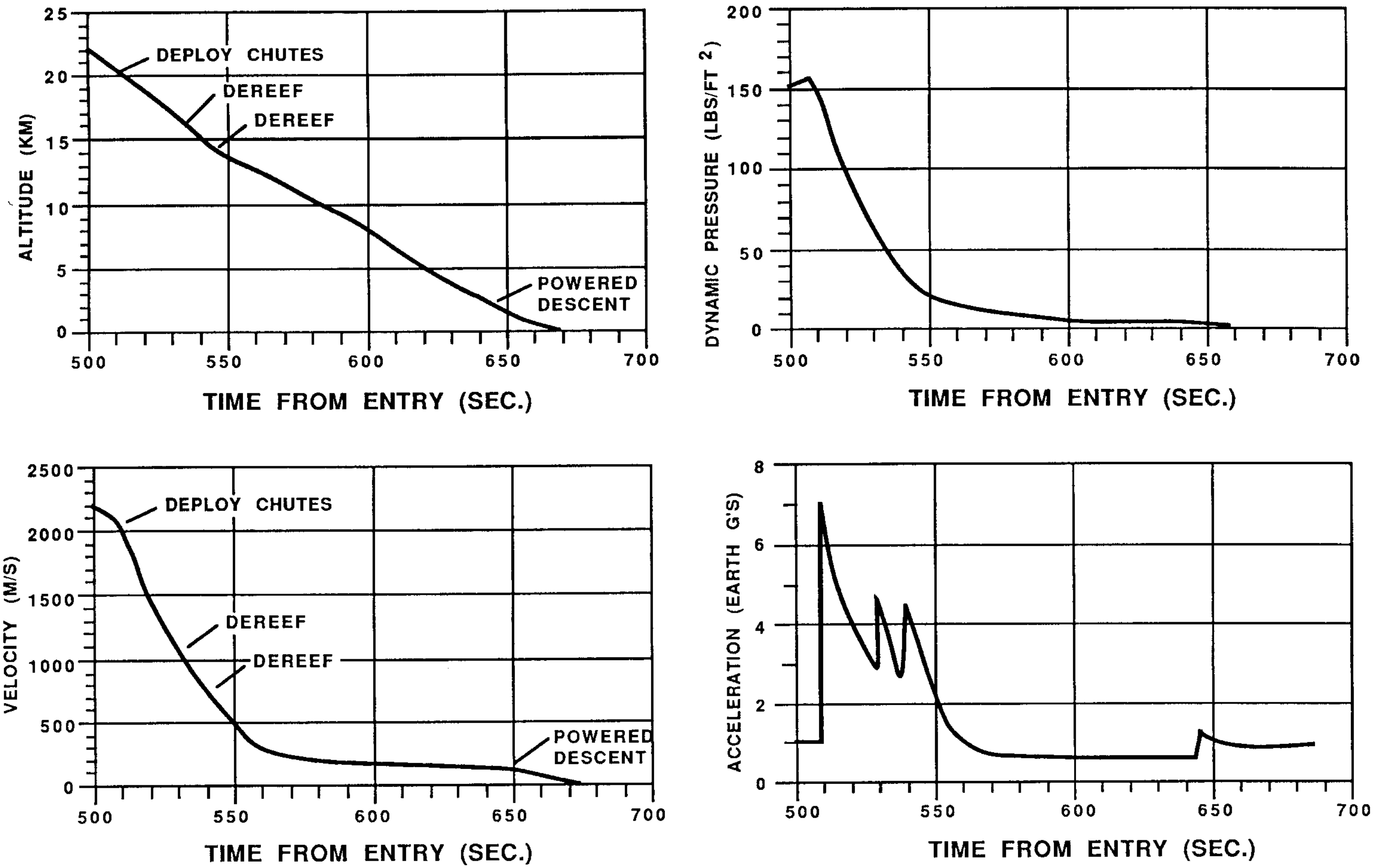


MARS EXCURSION MODULE DESCENT ANALYSIS

This chart shows some of the results for the Mars excursion module descent analysis. The chutes program provides over 80 output variables pertaining to the descent trajectory. Four are shown here. These graphs show altitude, velocity, dynamic pressure and acceleration as function of time from atmosphere entry, during the final 185 seconds of the descent.

1-4245-8-18D

MARS EXCURSION MODULE DESCENT ANALYSIS
PARACHUTE DESCENT ANALYSIS



INTRODUCTION

This section of the Split Sprint Mission Report deals with the performance of the Mars Ascent Vehicle. The main objective is to place a 6,000 lb. payload into a 1,000 km (540 NMI) circular Mars orbit, with an inclination of 74.73 degrees. Various trade studies and sensitivities will be addressed.

PD33/HAYS

MARS

ASCENT VEHICLE

PERFORMANCE

MARS ASCENT VEHICLE PERFORMANCE

The facing chart introduces the first main topic to be covered within the report on Mars Ascent Vehicle Performance. This topic, entitled "Initial Performance Data (Sep. 1987)," is the first of three iterations dealing with the main objective: to place a 6,000 lb. payload into a 1,000 KM circular orbit, with an inclination of 74.73 degrees. It should be noted that the final performance data is presented within the third main topic, entitled "2nd Performance Update (Jan. 1988)."

The fourth main topic, entitled "Performance to Mars Synchronous Orbit (Feb. 1988)," deals with performance to a highly elliptical, Mars synchronous orbit, and is not directly related to the main objective of this report.

MARS ASCENT VEHICLE PERFORMANCE

- O INITIAL PERFORMANCE DATA (SEP. 1987)
  - o PRELIMINARY PERFORMANCE TO 1,000 KM (540 NMI) / 74.73 DEG INC.
  - o SINGLE STAGE VS. DROP TANKS
  - o FIXED VS. OPTIMUM INJECTION

- O 1ST PERFORMANCE UPDATE (OCT. 1987)
  - o PRELIMINARY PERFORMANCE TO 1,000 KM (540 NMI) / 74.73 DEG INC.
  - o SENSITIVITY TO:
    - FLIGHT PERFORMANCE RESERVE
    - ACCELERATION LIMIT
    - ATMOSPHERE MODEL
  - o TRAJECTORY

- O 2ND PERFORMANCE UPDATE (JAN. 1988)
  - o FINAL PERFORMANCE TO 1,000 KM (540 NMI) / 74.73 DEG INC.
  - o LAUNCH WINDOW

- O PERFORMANCE TO MARS SYNCHRONOUS ORBIT (FEB. 1988)
  - o PERFORMANCE TO 500 X 33,562 KM (270 X 18,122 NMI) / 74.73 DEG INC.
  - o SENSITIVITY TO:
    - VACUUM THRUST
    - THRUST-TO-WEIGHT AT LIFT-OFF

PD33 / ETHERIDGE / LOWERY / HAYS

MARS ASCENT PERFORMANCE (INITIAL)

The facing chart lists the groundrules and assumptions used in generating the initial Mars ascent performance data. The ascent trajectory employs two burns: the first to inject the Mars Ascent Vehicle into an elliptical transfer orbit, and the second to circularize at 1,000 KM (540 n.mi.)/74.73 deg. inclination.

MARS ASCENT PERFORMANCE
(INITIAL)

GROUNDRULES AND ASSUMPTIONS:

- LAUNCH SITE: GDLAT = 40°, LONG = 140°
- INJECTION INTO ELLIPTICAL TRANSFER ORBIT
- CIRCULARIZATION AT 1,000 KM (540 N.MI.)/74.73° INCLINATION
- NO FLIGHT PERFORMANCE RESERVE (FPR)
- NO ACCELERATION LIMIT
- COOL ATMOSPHERE
- VACUUM THRUST = 40,000 LBS (1 ENGINE)
- ENGINE WEIGHT = 500 LBS
- VACUUM ISP = 326.8 SEC.
- ALL LAUNCHES OCCUR AT SECOND IN-PLANE OPPORTUNITY (SOUTHERLY AZIMUTH)

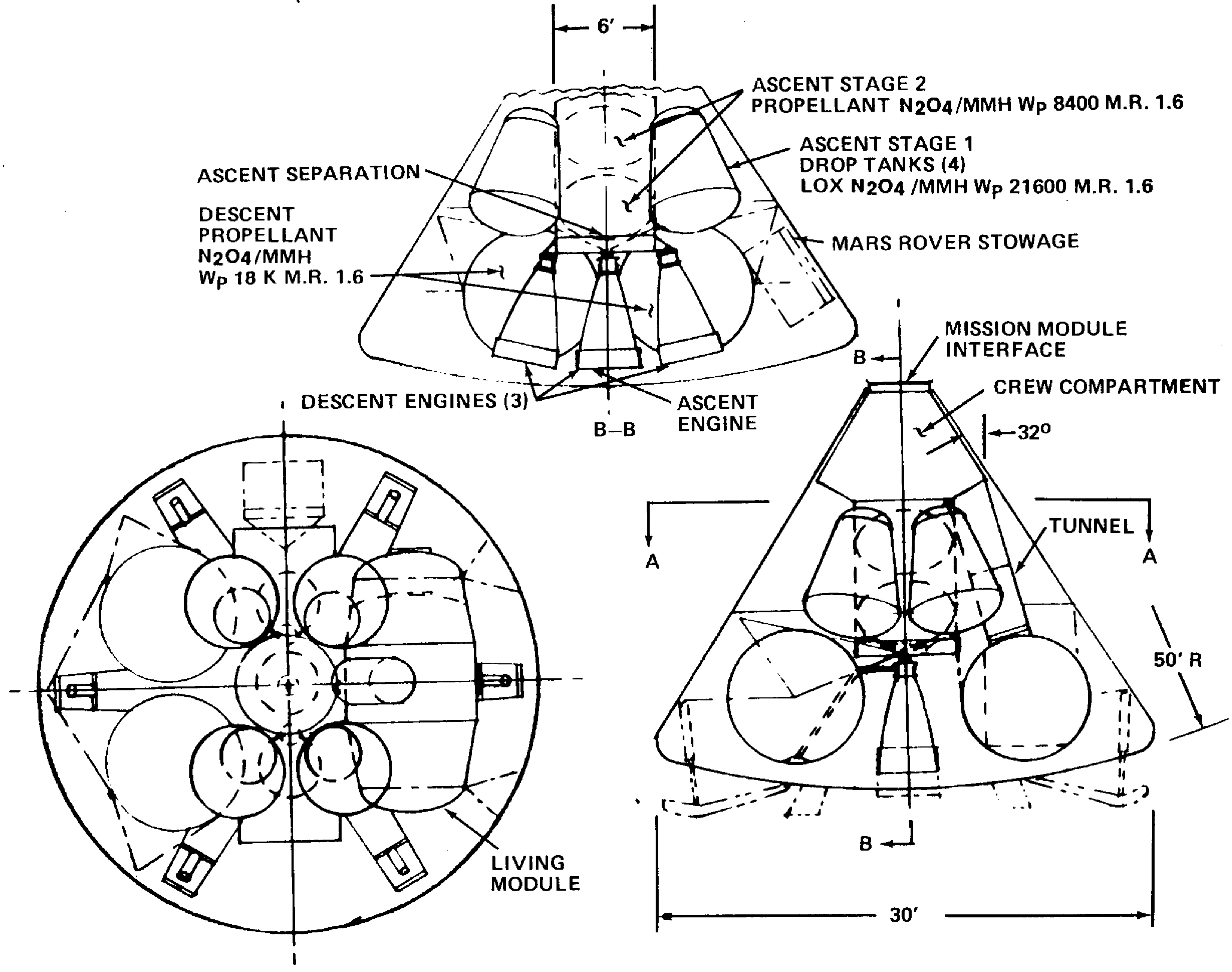
PD33
SEPT. 16, 1987

MARS EXCURSION MODULE CONCEPT

The Mars Excursion Module (MEM) shown consists of both an ascent vehicle and a descent vehicle. For ascent, the Mars Ascent Vehicle separates from the rest of the MEM and flies into Mars orbit using the center engine. The Mars Ascent Vehicle shown was used as a point of departure for determining ascent aerodynamics to be used in this study.

1-4186-8

FIGURE 6 MARS EXCURSION MODULE CONCEPT (APOLLO COMMAND MODULE DERIVATIVE)



PRELIMINARY MEM SUBSYSTEMS INCOMPLETE

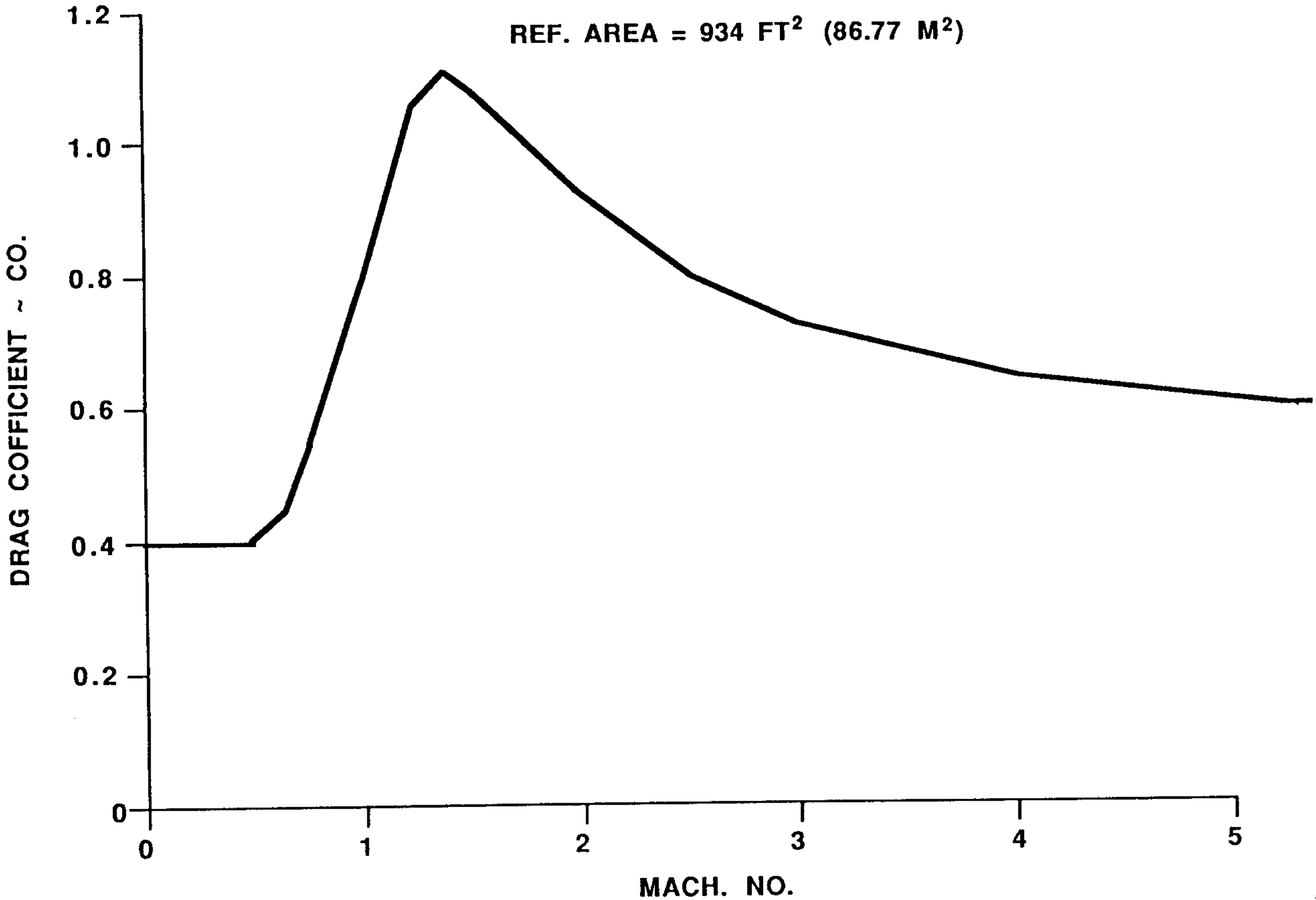
MARS EXCURSION MODULE ASCENT DRAG

This chart shows the zero angle of attack drag coefficient for the Mars ascent vehicle as a function of mach number. This curve was developed using cone drag tables. The drag coefficients shown on this graph must be used with a reference area equal to the base area of the ascent vehicle. These coefficients were used as input to analyze the performance for the ascent from the Martian surface.

[Ed. Note: As noted on the chart, the refernce area used as input to the ascent performance calculations was 934 ft^2. This corresponds to a 34.5 ft diameter ascent vehicle. Subsequent to the ascent vehicle analysis, it was determined that the reference area should have been 201 ft^2, corresponding to a 16 ft diameter ascent vehicle. The drag coefficients for this case remain the same, but the corresponding drag forces would be reduced by 78%. This decrease in drag would result in slightly better performance (less propellant) than shown in the ascent analysis.]

1-4387-8-19D

MARS EXCURSION MODULE ASCENT STAGE

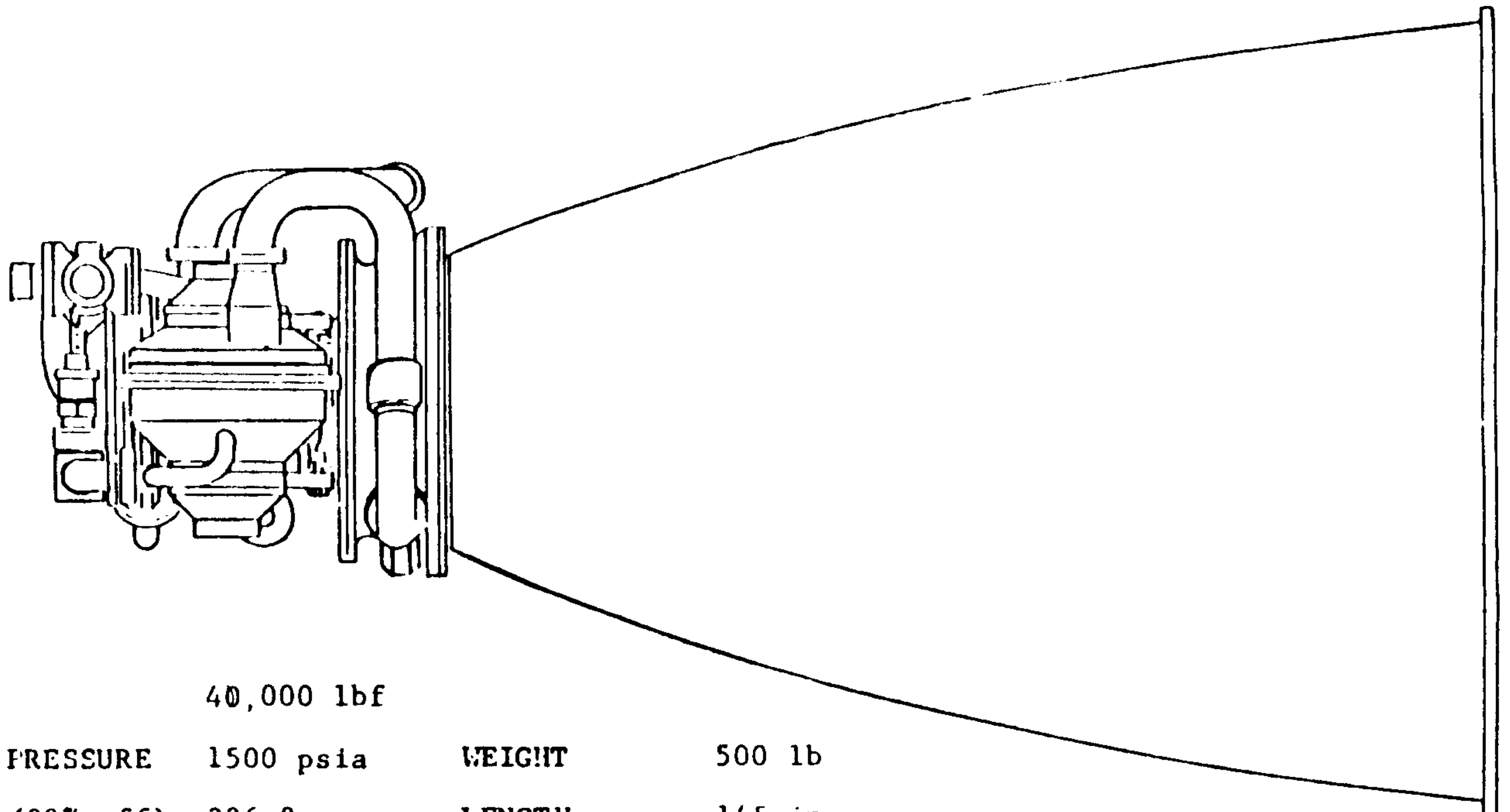


3/29/87
TJL

EARTH STORABLE PUMP-FED BIPROPELLANT ENGINE

This chart shows the engine used by the Mars Ascent Vehicle. The 326.8 sec. vacuum Isp shown was used for the initial performance data and the first performance update. The vacuum Isp was increased to 342 sec. for the second performance update. The engine weight was also increased from 500 lbs. to 1000 lbs. during the course of this study. However, because the change came late in the study, all the ascent performance data presented in this memo is based on an engine weight of 500 lbs.

EARTH STORABLE PUMP-FED BIPROPELLANT ENGINE



THRUST	40,000 lbf		
CHAMBER PRESSURE	1500 psia	WEIGHT	500 lb
Isp vac (92% eff)	326.8 s	LENGTH	145 in
PROPELLANTS	N_2O_4/MMH	EXIT DIAMETER	83 in
EXPANSION RATIO	400:1	MIXTURE RATIO	2

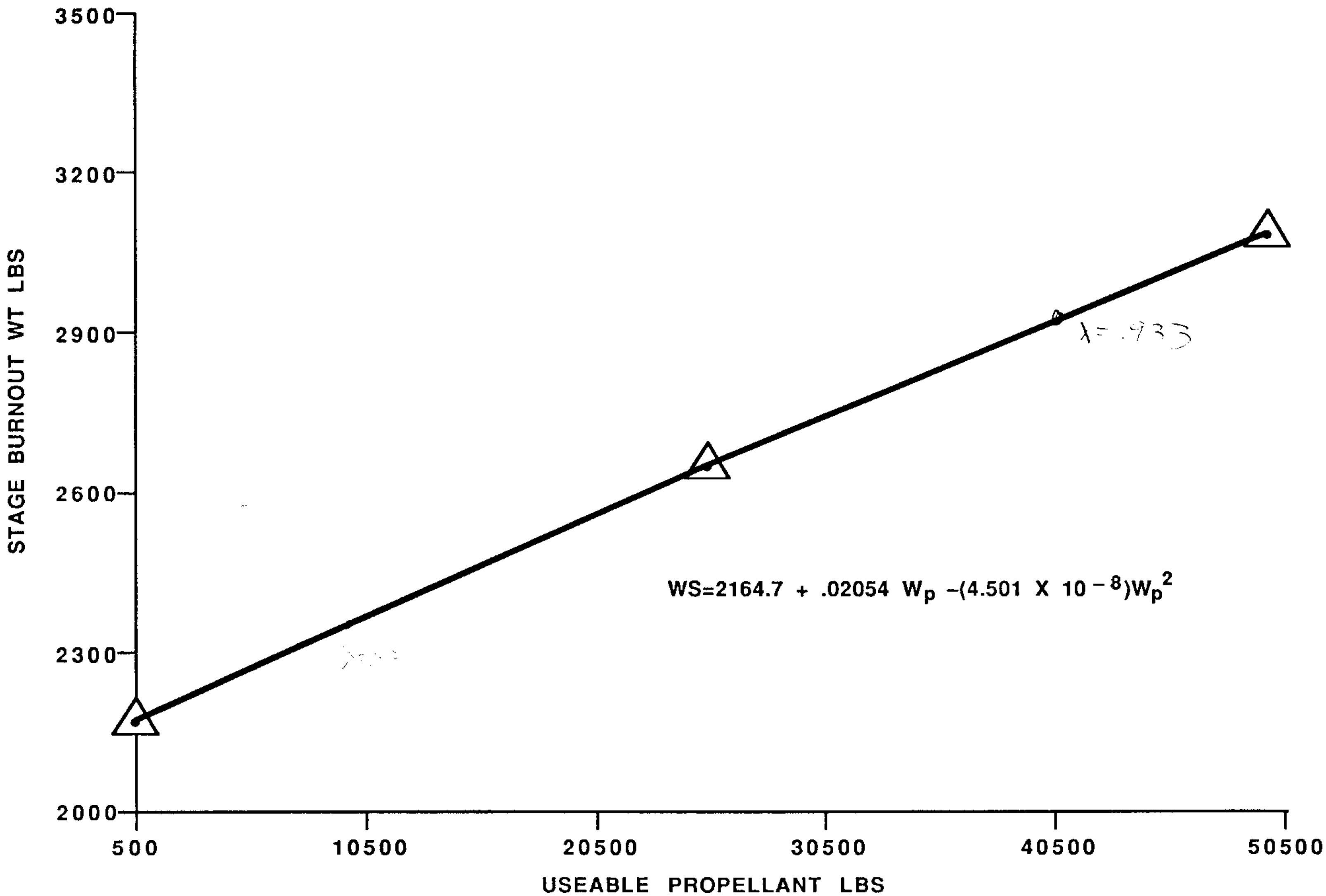
PD13
070687

MARS STUDY MEM ASCENT CORE

The scaling equation for the storable propellant (N_2O_4/MMH) MEM Ascent Core Stage is shown on this chart. The equation given at the bottom of the chart is good for a propellant range of 500 to 50,500 pounnds. The weights include one 40,000 pound thrust (vac) engine with complete TPS, propulsion, avionics, and structures. The propellant tank is sized for each propellant with spherical bulkheads up to eight feet (max) diameter, and then if more volume is required, a cylindrical section is added. A 15 percent contingency and residuals are included in the burnout weights given by this curve.

1-4183-8-17D

MARS STUDY MEM ASCENT CORE
N204/MMH 1-40K ENG/INLINE TANKS



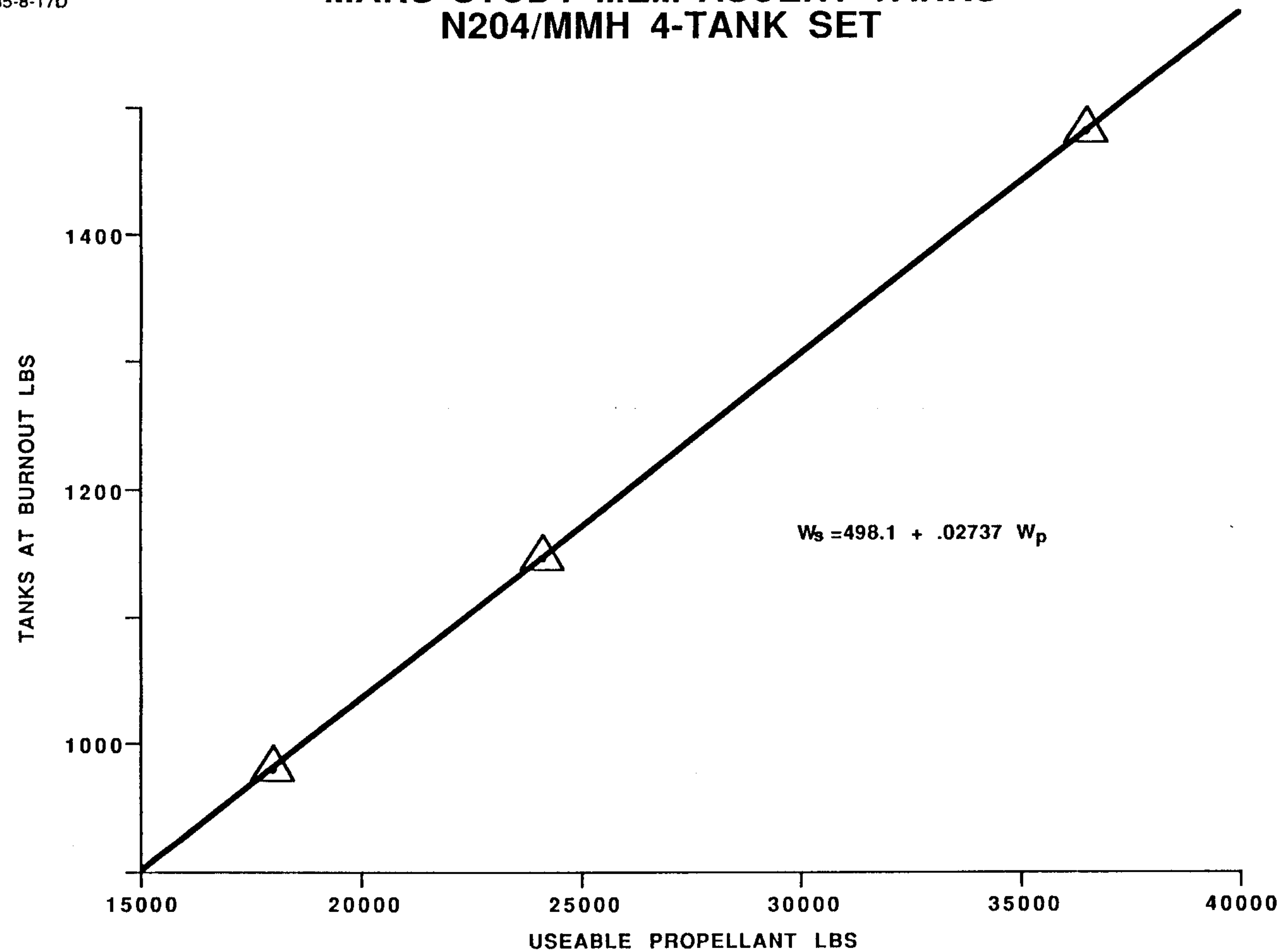
MARS STUDY MEM ASCENT TANKS

This is the scaling equation used for the drop tanks used on the Mars Ascent Vehicle. The tank set includes two N_2O_4 tanks and two MMH tanks which are cylindrical in shape with $\sqrt{2}$ bulkheads, and each is 65 inches in diameter.

The burnout weight includes residuals and complete propellant tank systems including TPS, structures, feed system, pressurization system and minimum avionics. A 15% weight contingency is also included. No engine, thrust structures, TVC, etc., are included with the tank set. The usable propellant range for this tank set is 15,000 pounds to 40,000 pounds.

1-4185-8-17D

MARS STUDY MEM ASCENT TANKS
N204/MMH 4-TANK SET



PD24/BROTHERS
03-22-85

MARS ASCENT PERFORMANCE - SINGLE STAGE

This chart shows the performance for a single stage Mars Ascent Vehicle. All cases shown are for a 106 x 1,000 km (57 x 540 n.mi.) transfer orbit, with circularization at 1,000 KM (540 n.mi.).

MARS ASCENT PERFORMANCE

SINGLE STAGE

ORBIT = 1,000 KM (540 N.MI.) CIRCULAR

INCL = 74.73 DEG

LAUNCH POINT

LONGITUDE 140. DEG

LATITUDE 40. DEG

CASE #	1	2	3
PAYLOAD =	7,408.	11,130.	12,231.
WP (LBS) =	34,714.	45,846.	49,694.
WI (LBS) =	2,878.	3,024.	3,075.
GLOW (LBS) =	45,000.	60,000.	65,000.

*ALL CASES WERE INJECTED INTO A 106 X 1,000 KM (57 X 540 N.MI.) ELLIPSE AND CIRCULARIZED AT 1,000 KM (540 N.MI.)

MARS ASCENT PERFORMANCE - DROP TANK CONFIGURATION

This chart shows the performance for a Mars Ascent Vehicle with drop tanks. All cases shown are for a 106 x 1,000 KM (57 x 540 n.mi.) transfer orbit, with circularization at 1,000 KM (540 n.mi.).

MARS ASCENT PERFORMANCE

DROP TANK CONFIGURATION

ORBIT = 1,000 KM (540 N.MI.) CIRCULAR
INCL = 74.73 DEG.

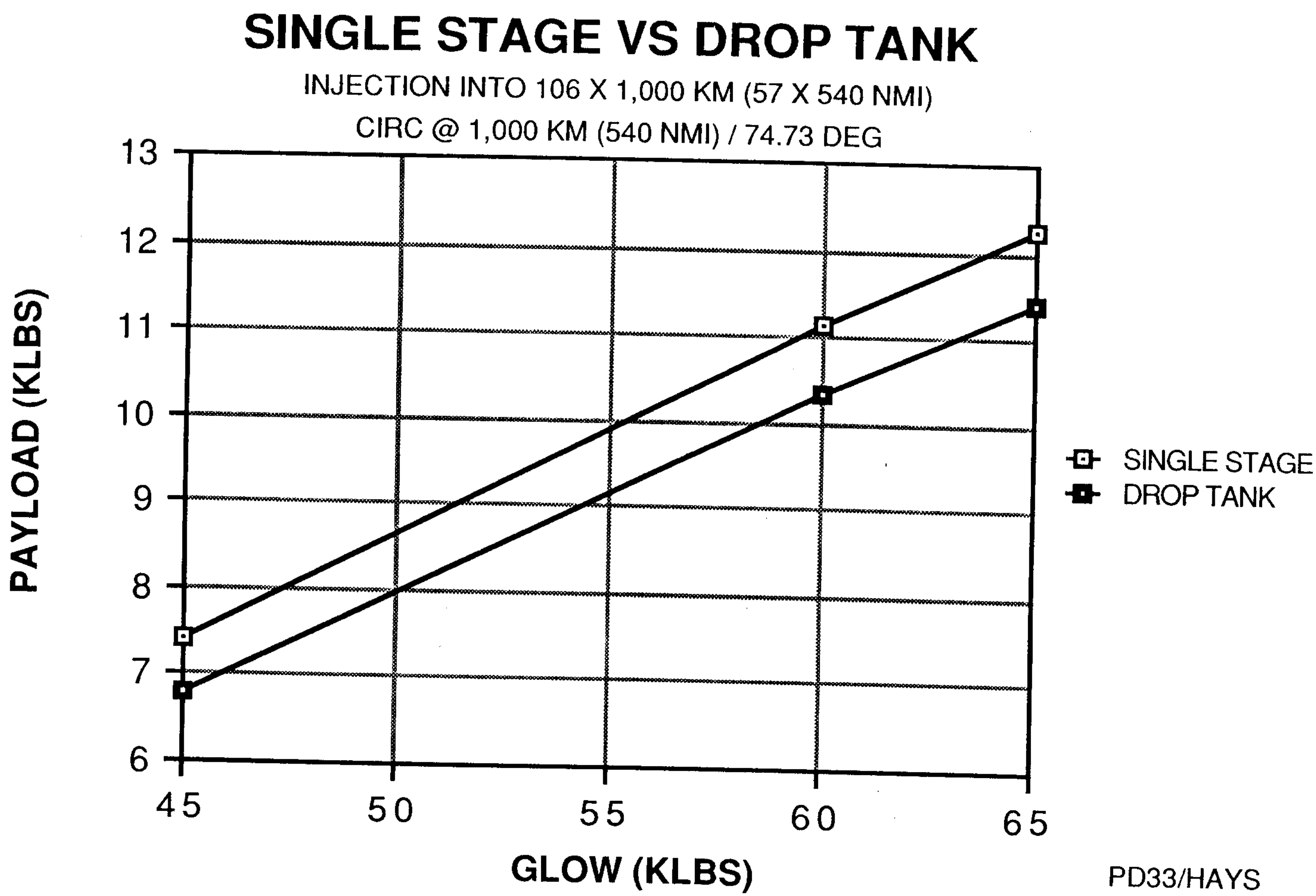
LAUNCH POINT
LONGITUDE = 140. (DEG)
LATITUDE = 40. (DEG)

*CASE #		1	2	3
PAYLOAD (LBS)	=	6,778.	10,339.	11,392.
WP (TANKS) (LBS)	=	34,095.	45,020.	48,800.
WI (TANKS) (LBS)	=	1,428.	1,729.	1,833.
WP (CORE) (LBS)	=	523.	732.	793.
WI (CORE (LBS)	=	2,176.	2,180.	2,181.
GLOW (LBS)	=	45,000.	60,000.	65,000.

*ALL CASES WERE INJECTED INTO A 106 X 1,000 KM (57 X 540 N.MI.) ELLIPSE AND CIRCULARIZED AT 1,000 KM (540 N.MI.)

SINGLE STAGE VS. DROP TANK

This chart shows the payload vs gross lift-off weight (GLOW) data shown on the previous two charts. The single stage vehicle out-performs the vehicle with drop tanks, and therefore will be assumed for the remainder of this study.



SINGLE STAGE VS DROP TANK
INJECTION INTO 106 X 1,000 KM (57 X 540 NMI)
CIRC @ 1,000 KM (540 NMI) / 74.73 DEG
PAYLOAD (KLBS)
13
12
11
10
9
8
7
6
45
50
55
60
65
GLOW (KLBS)
SINGLE STAGE
DROP TANK
PD33/HAYS

MARS ASCENT PERFORMANCE - SINGLE STAGE

This chart shows the ascent performance for a single stage vehicle with optimized injection into an elliptical transfer orbit. Once again, circularization occurs at 1,000 KM (540 n.mi.).

MARS ASCENT PERFORMANCE

SINGLE STAGE

ORBIT = 1,000 KM (540 N.MI.) CIRCULAR

INCL = 74.73 DEG

LAUNCH POINT

LONGITUDE 140. DEG

LATITUDE 40. DEG

CASE #	1	2	3	
PAYLOAD =	7,810.	9,038.	11,355.	
WP (LBS) =	34,317.	38,040.	45,639.	λ = .9382
WI (LBS) =	2,873. λ = .922	2,922.	3,006.	
GLOW (LBS) =	45,000.	50,000.	60,000.	
INJECTION POINT				
ALT (N.MI.) =	52.55	50.35	43.5	
γ_I (DEG) =	12.32	9.28	4.26	
V_I (FPS) =	11,702.	11,897.	11,934.	

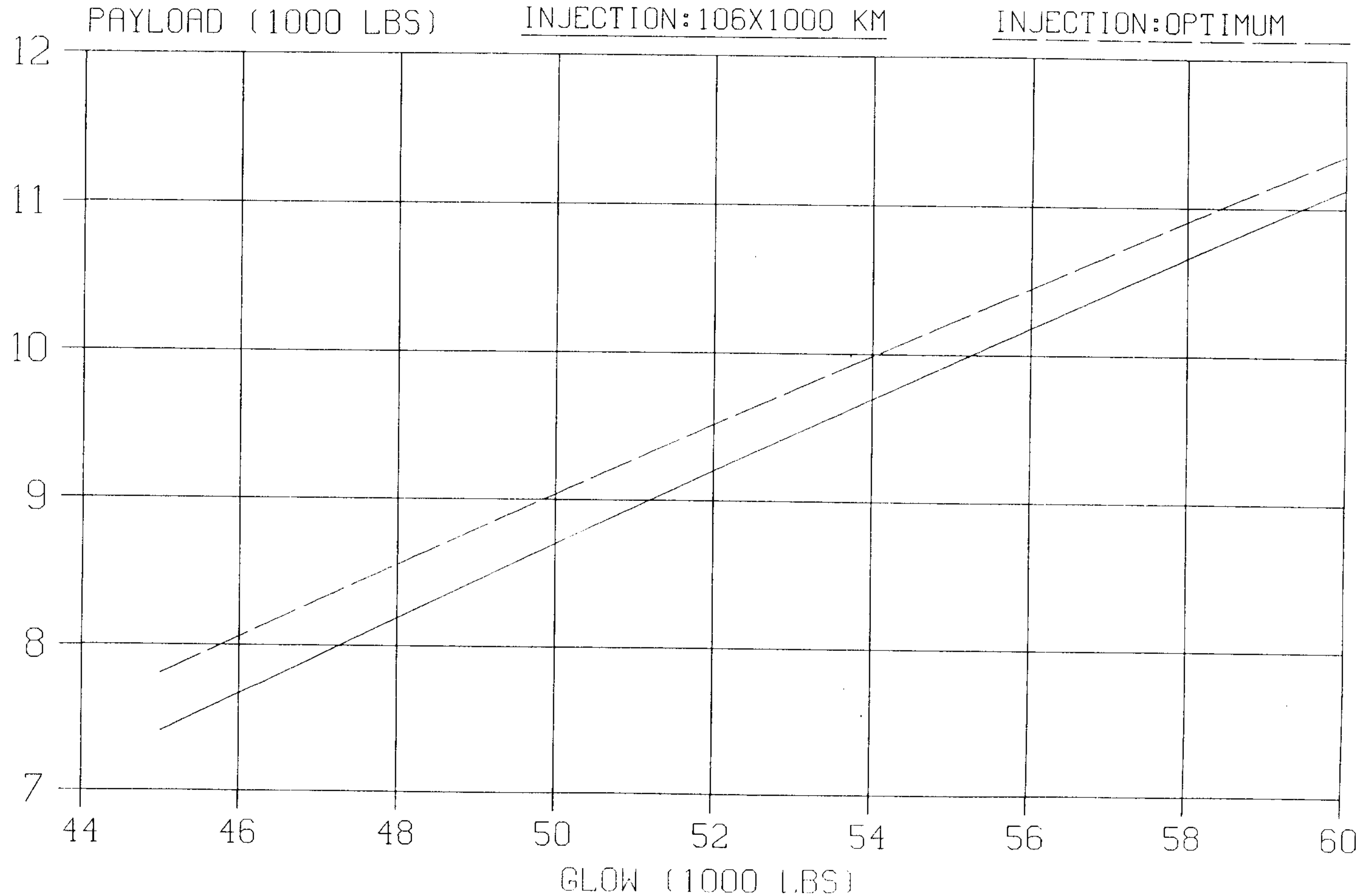
*ALL CASES WERE CIRCULARIZED @ 1,000 KM (540 N.MI.)

MARS ASCENT PERFORMANCE

The facing chart shows the performance data presented on the previous chart (optimum injection) compared with the performance obtained when injection into transfer orbit occurs at 106 KM (57 n.mi.). Optimum injection into transfer orbit results in better performance, and therefore will be used for the remainder of this study.

MARS ASCENT PERFORMANCE
SINGLE STAGE

1000 KM/74.73 DEG



THIS PAGE INTENTIONALLY LEFT BLANK

MARS ASCENT VEHICLE PERFORMANCE

- O INITIAL PERFORMANCE DATA (SEP. 1987)
  - o PRELIMINARY PERFORMANCE TO 1,000 KM (540 NMI) / 74.73 DEG INC.
  - o SINGLE STAGE VS. DROP TANKS
  - o FIXED VS. OPTIMUM INJECTION
- O 1ST PERFORMANCE UPDATE (OCT. 1987)
  - o PRELIMINARY PERFORMANCE TO 1,000 KM (540 NMI) / 74.73 DEG INC.
  - o SENSITIVITY TO:
    - FLIGHT PERFORMANCE RESERVE
    - ACCELERATION LIMIT
    - ATMOSPHERE MODEL
  - o TRAJECTORY
- O 2ND PERFORMANCE UPDATE (JAN. 1988)
  - o FINAL PERFORMANCE TO 1,000 KM (540 NMI) / 74.73 DEG INC.
  - o LAUNCH WINDOW
- O PERFORMANCE TO MARS SYNCHRONOUS ORBIT (FEB. 1988)
  - o PERFORMANCE TO 500 X 33,562 KM (270 X 18,122 NMI) / 74.73 DEG INC.
  - o SENSITIVITY TO:
    - VACUUM THRUST
    - THRUST-TO-WEIGHT AT LIFT-OFF

PD33 / HAYS

MARS ASCENT PERFORMANCE (FIRST UPDATE)

The facing chart lists the groundrules and assumptions used in generating the first update to the Mars ascent performance data. Changes to the initial set of groundrules and assumptions include single stage, optimized injection into elliptical transfer orbit, and a fixed payload weight of 6,000 lbs. Once again, the final orbit is at 1,000 KM (540 n.mi.)/74.73 deg. inclination.

MARS ASCENT PERFORMANCE
(1ST UPDATE)

GROUNDRULES AND ASSUMPTIONS:

- SINGLE STAGE (2 BURNS)
- LAUNCH SITE: GDLAT = 40°, LONG = 140°
- OPTIMIZED INJECTION INTO ELLIPTICAL TRANSFER ORBIT
- CIRCULARIZATION AT 1,000 KM (540 N.MI.)/74.73° INCLINATION
- PAYLOAD = 6,000 LBS
- VACUUM THRUST = 40,000 LBS (1 ENGINE)
- ENGINE WEIGHT = 500 LBS
- VACUUM ISP = 326.8 SEC.
- ALL LAUNCHES OCCUR AT SECOND IN-PLANE OPPORTUNITY (SOUTHERLY AZIMUTH)

PD33/HAYS
OCT. 28, 1987

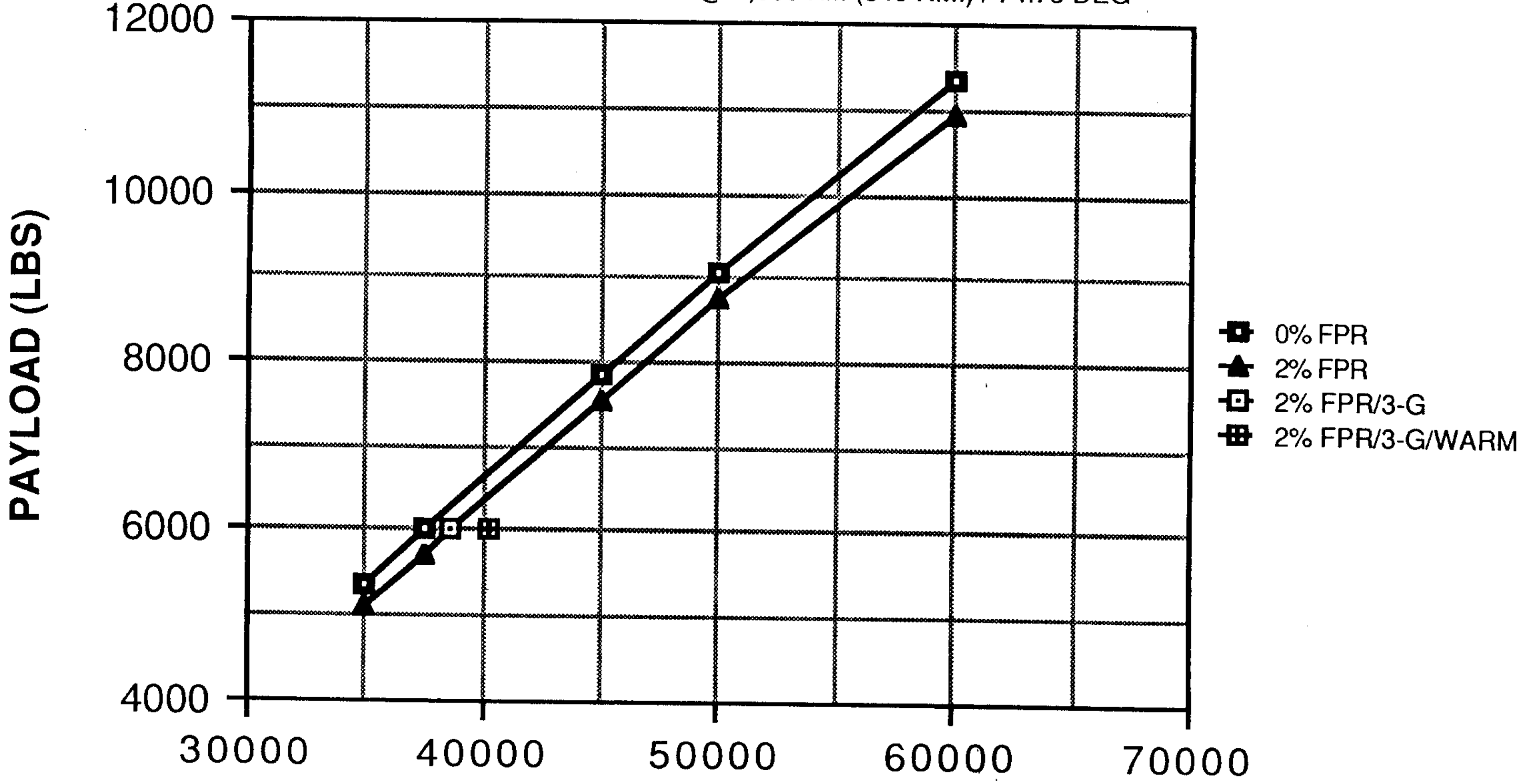
MARS ASCENT VEHICLE

This chart shows payload as a function of gross lift-off weight (GLOW). The effect of adding a 2% delta-V flight performance reserve (FPR), a 3-G acceleration limit, and a warm Mars atmosphere model are shown. The 3-G limit is measured in Earth G's. This limit and the 2% FPR add more realism to the performance data. The warm atmosphere model represents a worst case scenario (compared with the cool atmosphere model).

MARS ASCENT VEHICLE

SINGLE STAGE (2 BURNS)

OPTIMIZED INJECTION / CIRC @ 1,000 KM (540 NMI) / 74.73 DEG



PD33/HAYS
OCT. 28, 1987

MARS ASCENT PERFORMANCE

This chart shows performance data from the previous chart for a fixed payload of 6,000 lbs. The addition of a 2% FPR and warm atmosphere model both cause GLOW to increase. the addition of a 3-G acceleration limit has no effect on GLOW.

MARS ASCENT PERFORMANCE

- o PAYLOAD = 6K LBS
- o THRUST (VAC.) = 40K LBS
- o ISP (VAC.) = 326.8 SEC. (FOR ALL POWER SETTINGS)

FPR(% ΔV)	MAX G'S	ATMOSPHERE	MIN POWER SETTING (%)	GLOW (LBS)
0	4.6	COOL	100	37,534
2	4.4	COOL	100	38,578
2	3.0	COOL	67	38,578
2	3.0	WARM	68	40,314

ADDITIONAL PERFORMANCE CONSIDERATIONS

- o ENGINE THROTTLE EFFICIENCY
- o LOWER ENGINE THRUST LEVEL (?)
- o LAUNCH WINDOW

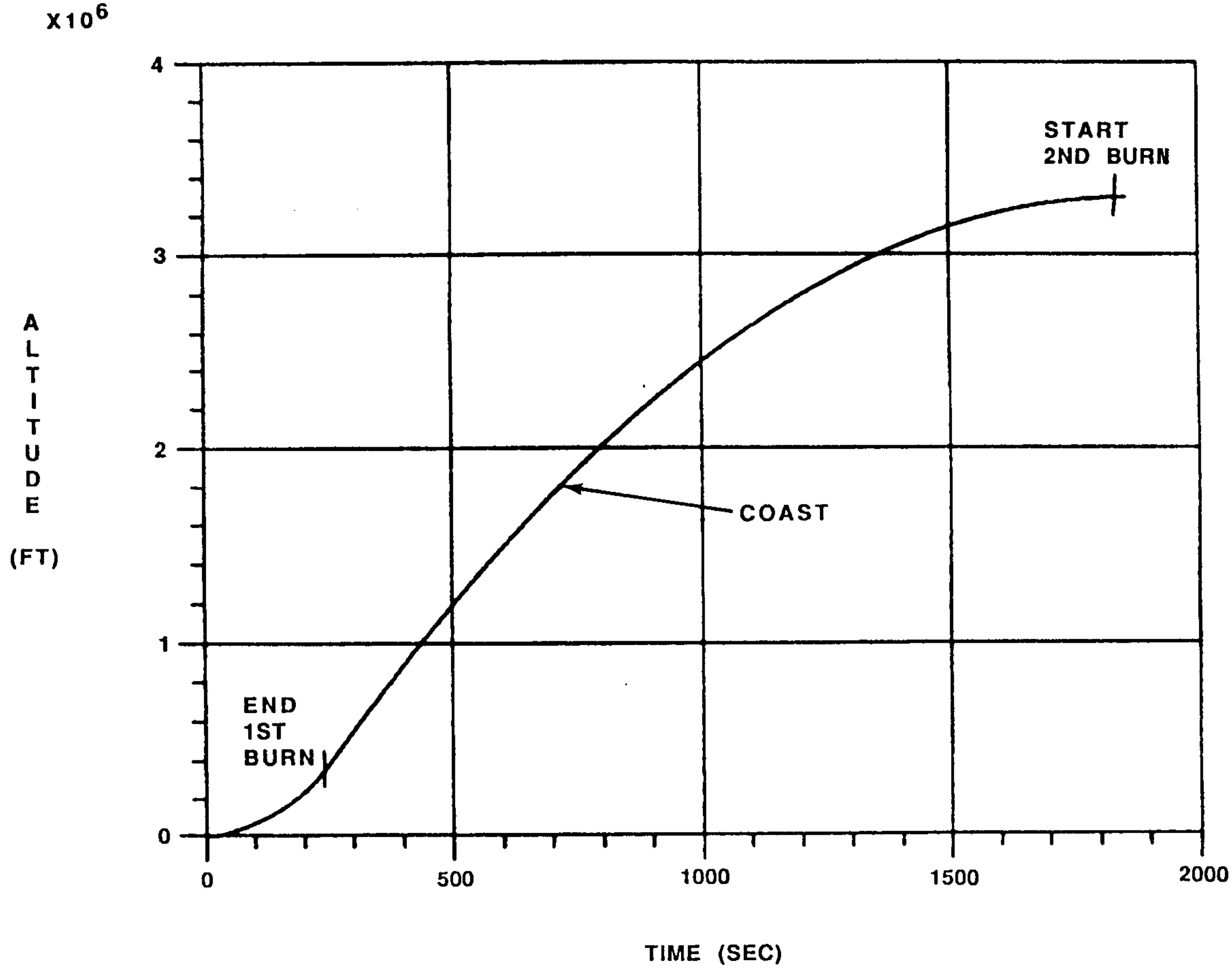
PD33/HAYS
OCT 28, 1987

ALTITUDE VS. TIME

This graph, and the following three graphs, are for the 2% FPR/3-G/warm atmosphere case shown on the previous two charts. This graph shows altitude vs. time, and indicates where the first burn, coast, and second burn occur.

1-4383-8

MARS ASCENT

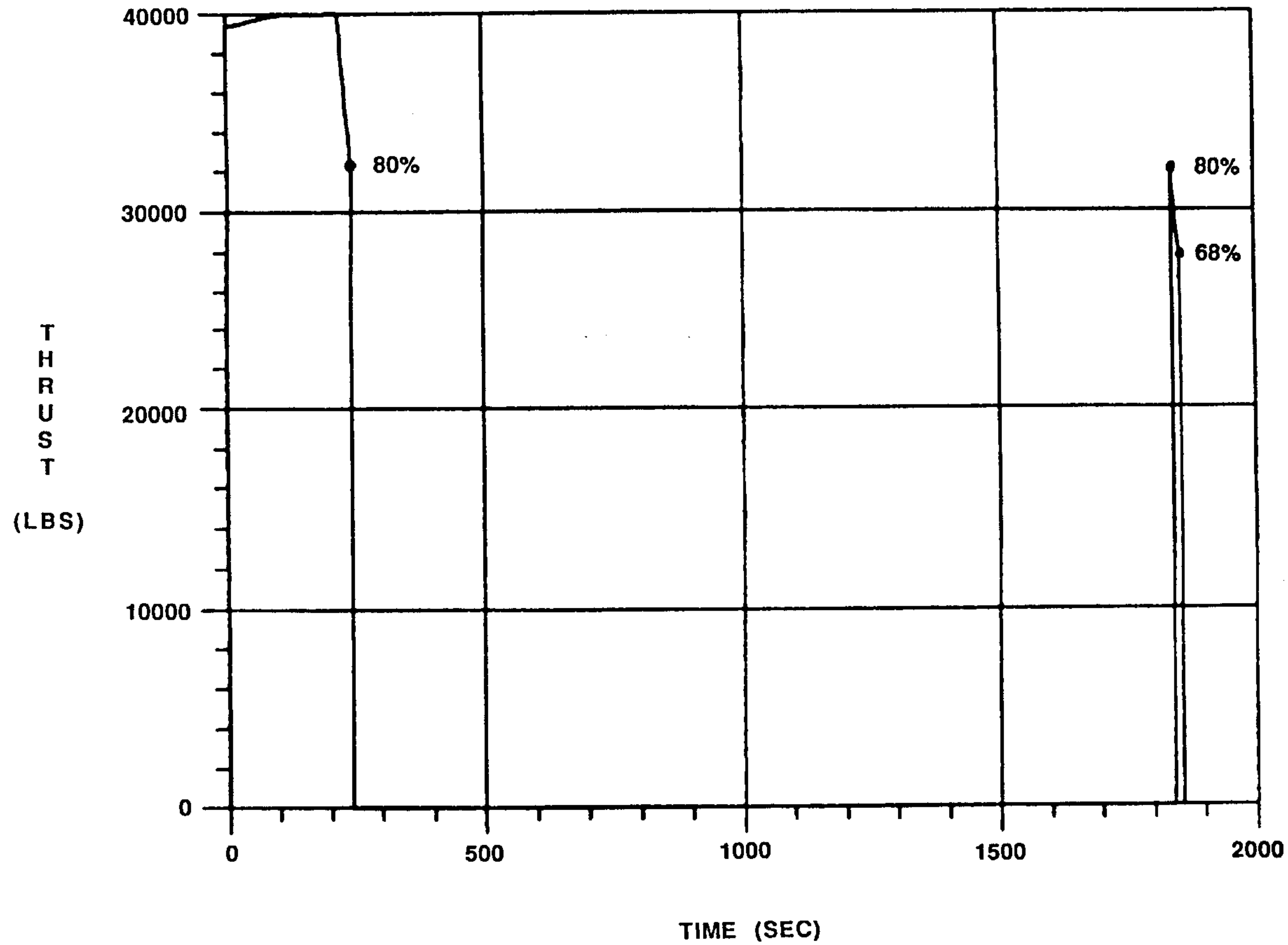


THRUST VS. TIME

This chart shows thrust vs time for both burns. Because of the 3-G acceleration limit, the engine is throttled to 80% power setting at the end of the first burn, and 68% power setting at the end of the second burn.

1-4384-8

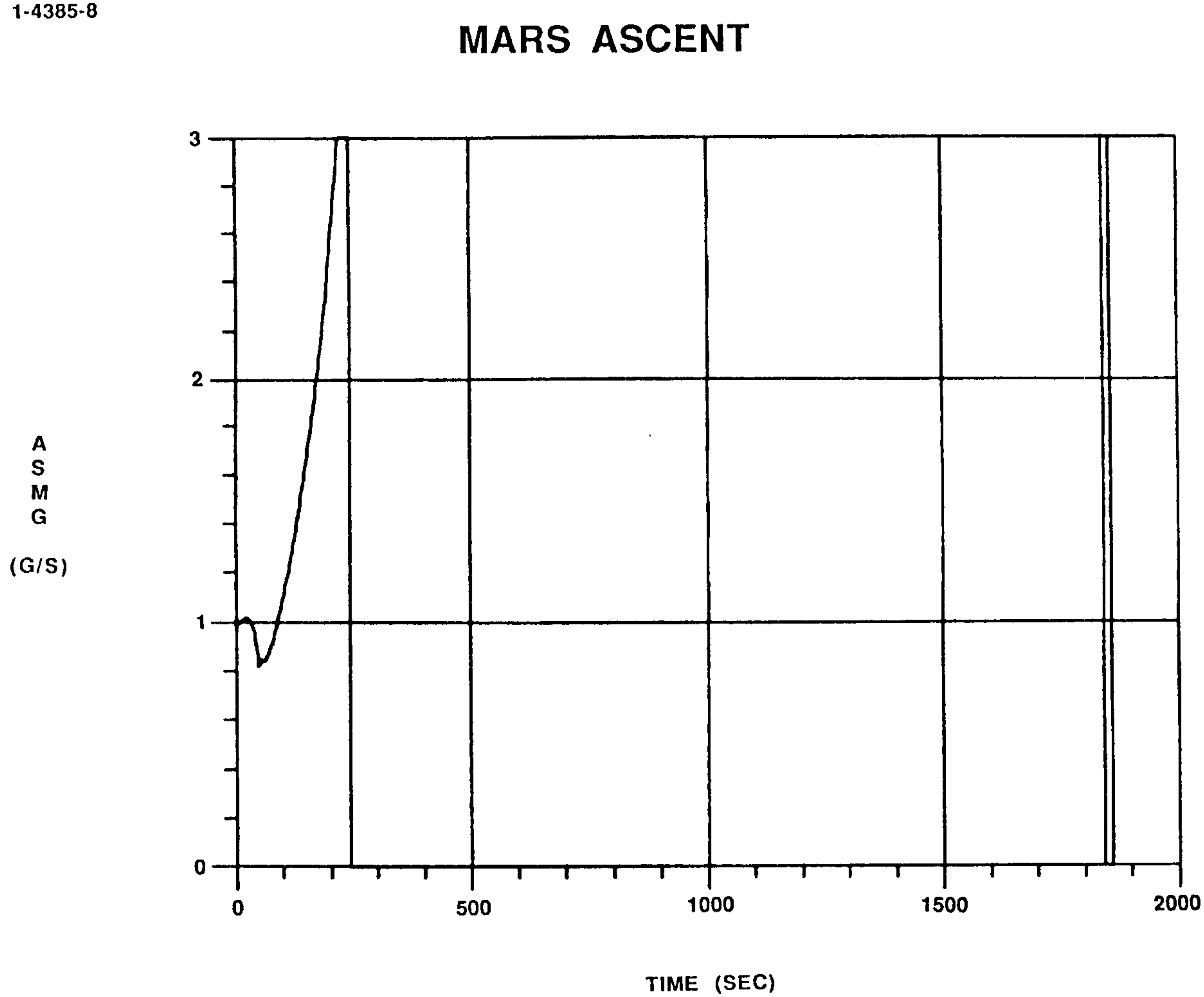
MARS ASCENT



SENSED ACCELERATION VS TIME

This chart shows the G-loading for both burns. The acceleration is limited to 3-G's for crew safety. Note that this is about eight times the G-force experienced by the crew on the surface of Mars.

1-4385-8

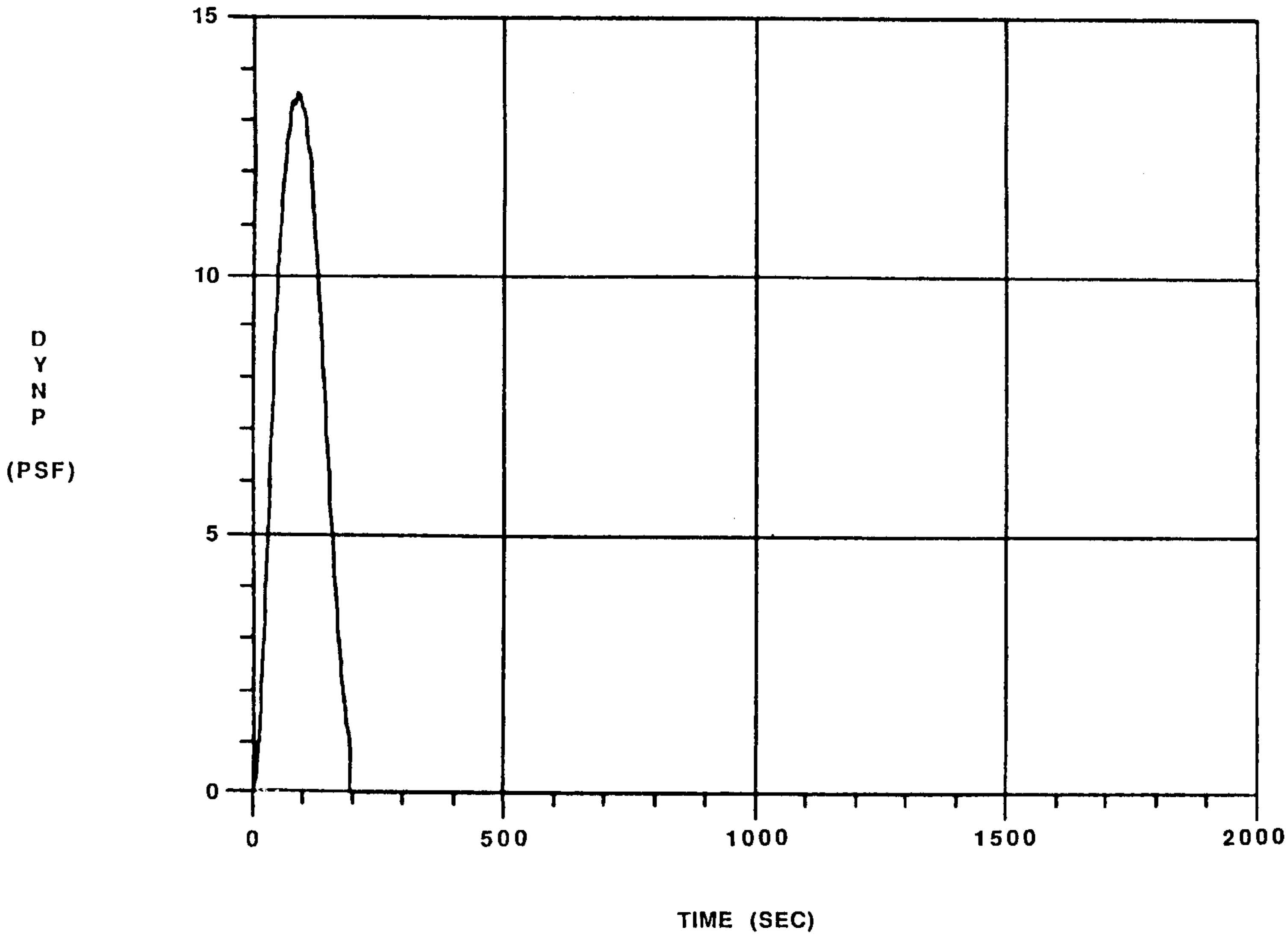


DYNAMIC PRESSURE VS. TIME

This chart shows the dynamic pressure during Mars ascent. Because Max-Q is only about 13.5 psf, aeroheating effects should be small.

1-4386-8

MARS ASCENT



THIS PAGE INTENTIONALLY LEFT BLANK

MARS ASCENT VEHICLE PERFORMANCE

- O INITIAL PERFORMANCE DATA (SEP. 1987)
  - o PRELIMINARY PERFORMANCE TO 1,000 KM (540 NMI) / 74.73 DEG INC.
  - o SINGLE STAGE VS. DROP TANKS
  - o FIXED VS. OPTIMUM INJECTION
- O 1ST PERFORMANCE UPDATE (OCT. 1987)
  - o PRELIMINARY PERFORMANCE TO 1,000 KM (540 NMI) / 74.73 DEG INC.
  - o SENSITIVITY TO:
    - FLIGHT PERFORMANCE RESERVE
    - ACCELERATION LIMIT
    - ATMOSPHERE MODEL
  - o TRAJECTORY
- O 2ND PERFORMANCE UPDATE (JAN. 1988)
  - o FINAL PERFORMANCE TO 1,000 KM (540 NMI) / 74.73 DEG INC.
  - o LAUNCH WINDOW
- O PERFORMANCE TO MARS SYNCHRONOUS ORBIT (FEB. 1988)
  - o PERFORMANCE TO 500 X 33,562 KM (270 X 18,122 NMI) / 74.73 DEG INC.
  - o SENSITIVITY TO:
    - VACUUM THRUST
    - THRUST-TO-WEIGHT AT LIFT-OFF

PD33 / HAYS

MARS ASCENT PERFORMANCE (SECOND UPDATE)

The facing chart lists the groundrules and assumptions used in generating the second update to the Mars ascent performance data. Changes to the groundrules and assumptions, as a result of trades done in the first update, include a 2% Delta-V FPR, 3-G acceleration limit, and a warm Mars atmosphere model. Other changes include a 342 sec. vacuum Isp and a northerly launch azimuth. Once again, the final orbit is at 1,000 KM (540 n.mi.)/74.73 deg. inclination. Also, a launch window is provided by an impulsive orbital yaw maneuver.

It should be noted that while this second update was being worked, the engine weight was increased to 1,000 lbs. However, all performance data shown in this memo is for a 500 lb. engine. Since engine weight and payload can be traded pound for pound, a 1,000 lb. engine would result in a 5,500 payload.

MARS ASCENT PERFORMANCE
(2ND UPDATE)

PURPOSE: DETERMINE THE GROSS LIFT-OFF WEIGHT (GLOW) REQUIRED TO PROVIDE A MARS ASCENT LAUNCH WINDOW

GROUNDRULES AND ASSUMPTIONS:

- SINGLE STAGE (2 BURNS)
- LAUNCH SITE: GDLAT = 40°, LONG = 140°
- OPTIMIZED INJECTION INTO ELLIPTICAL TRANSFER ORBIT
- CIRCULARIZATION AT 1,000 KM (540 N.MI.)/74.73° INCLINATION
- 2% DELTA-V FPR
- 3-G ACCELERATION LIMIT
- WARM ATMOSPHERE
- PAYLOAD = 6,000 LBS
- VACUUM THRUST = 40,000 LBS (1 ENGINE)
- ENGINE WEIGHT = 500 LBS
- VACUUM ISP = 342 SEC.
- ALL LAUNCHES OCCUR AT SECOND IN-PLANE OPPORTUNITY (NORTHERLY AZIMUTH)
- LAUNCH WINDOW PROVIDED BY IMPULSIVE ORBITAL MANEUVER

PD33/HAYS
JAN 11, 1988

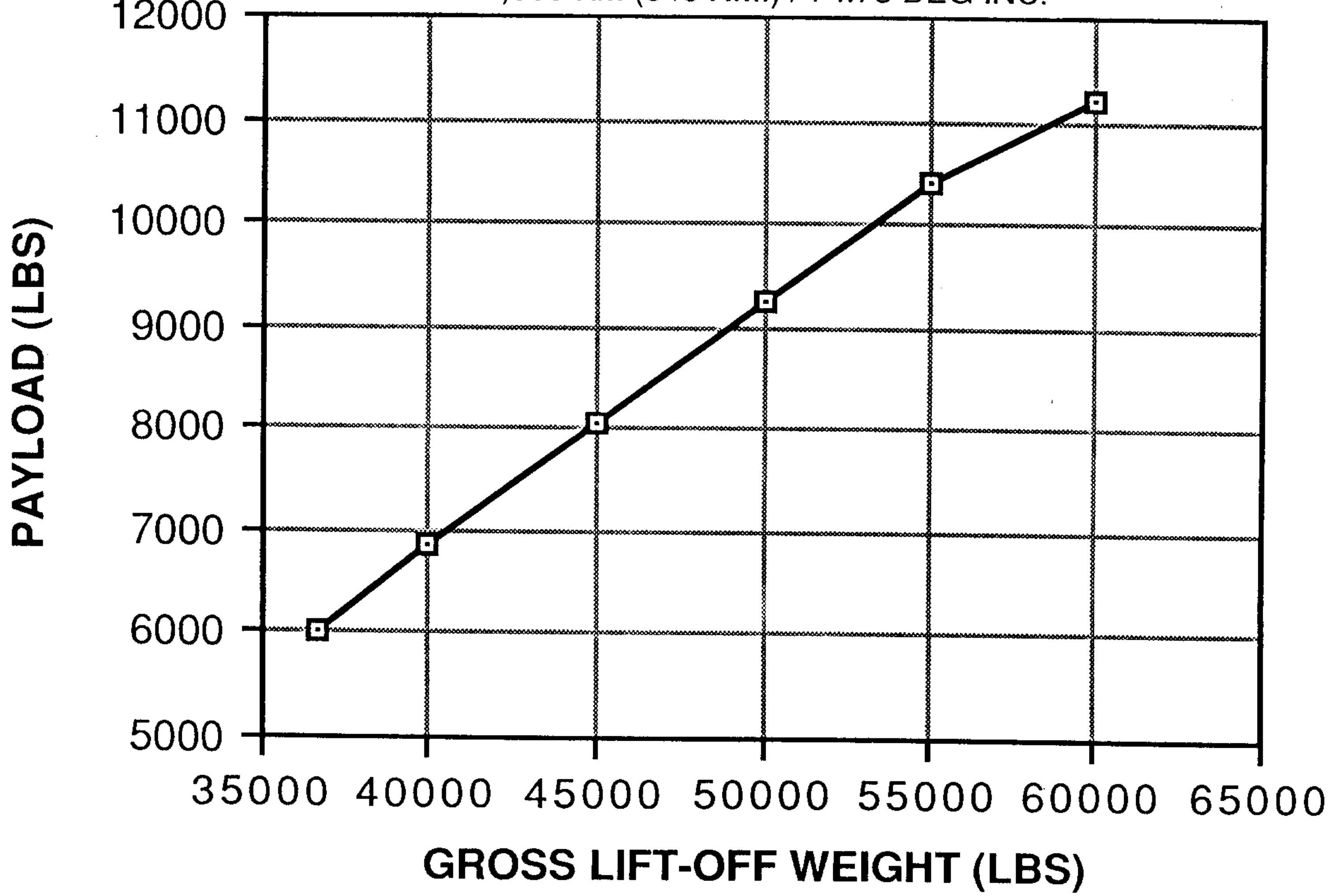
MARS ASCENT VEHICLE

The facing chart shows payload as a function of GLOW for a Mars Ascent Vehicle. As shown, a 6,000 lbs. payload requires a GLOW of 36,661 lbs. The performance data shown here does not include any launch window.

MARS ASCENT VEHICLE

SINGLE STAGE (2 BURNS)

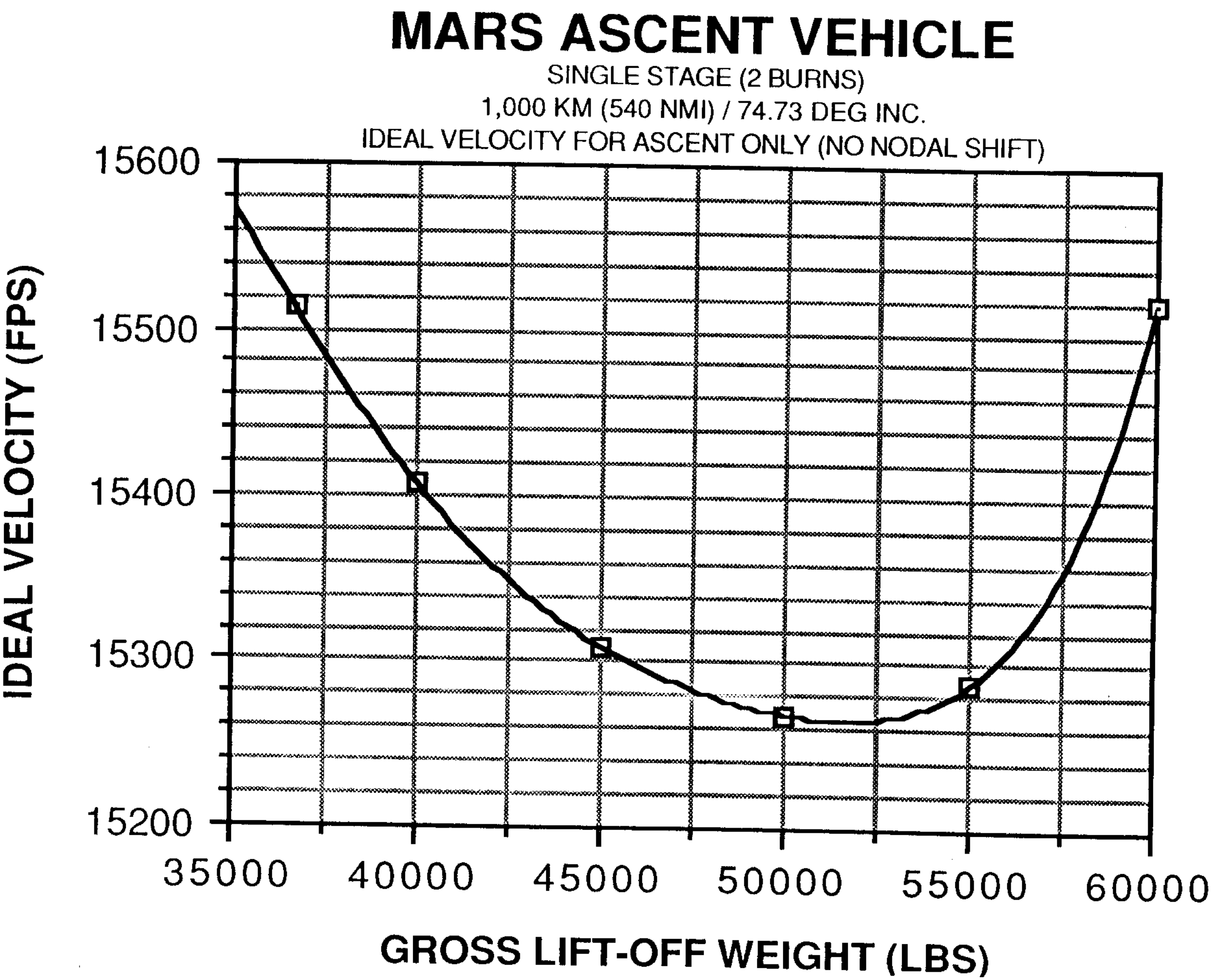
1,000 KM (540 NMI) / 74.73 DEG INC.



PD33/HAYS
DEC. 16, 1987

MARS ASCENT VEHICLE

This chart shows ideal velocity vs. GLOW for a Mars Ascent Vehicle. This data is for ascent only, and is required when using the rocket equation to calculate launch window performance. The "bucket" in ideal velocity, at 52K-lbs. GLOW, suggests that the vehicle may be over-thrusted on the left side of the curve and under-thrusted on the right side.



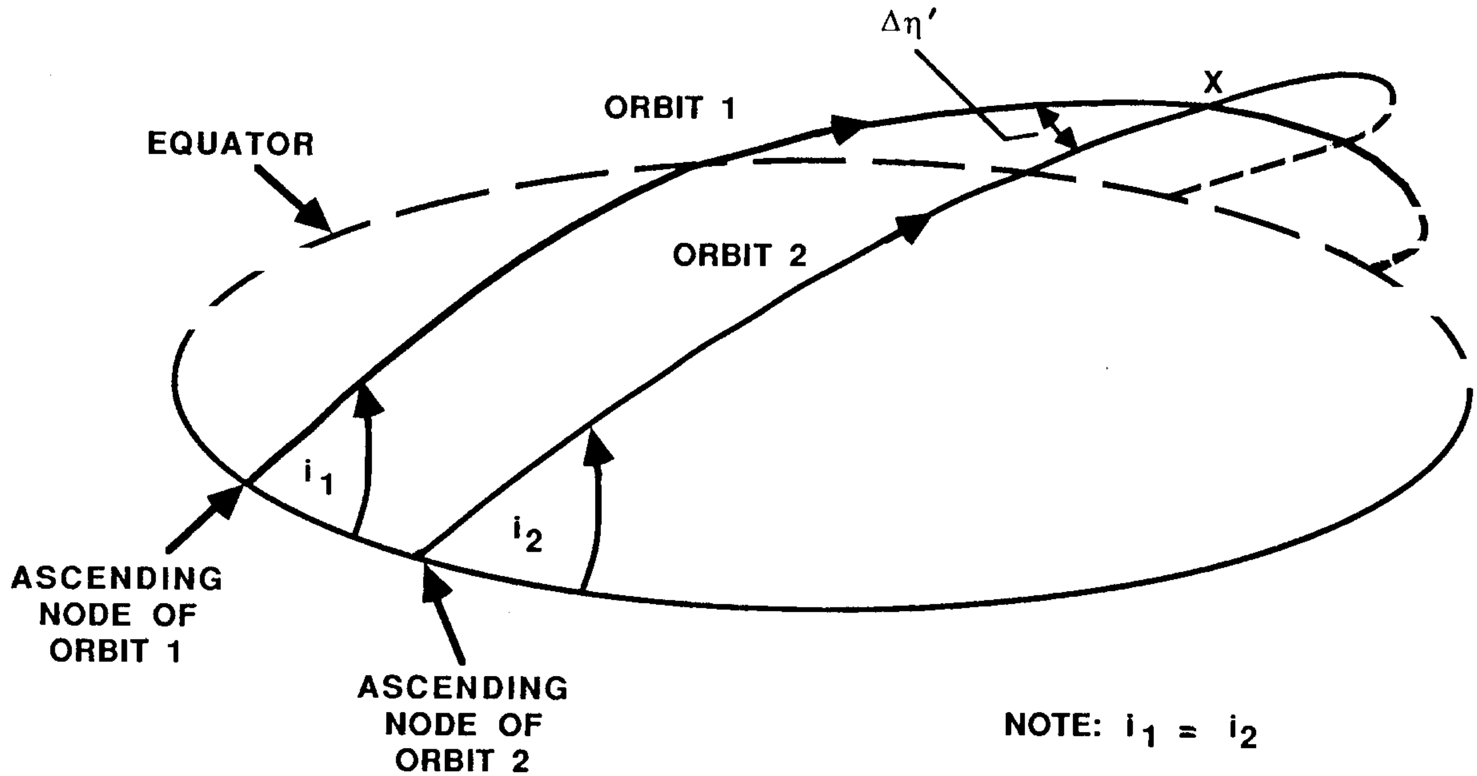
PD33/HAYS
DEC. 16, 1987

LAUNCH WINDOW PROVIDED BY ORBITAL MANEUVER TO SHIFT NODE

This diagram shows how an orbital maneuver is used to provide a launch window. For example, if orbit "2" is the desired orbit, then an early launch will put the spacecraft into orbit "1." To place the spacecraft into the desired final orbit, a yaw maneuver of angle $\Delta\eta'$ must be accomplished when the two orbits cross at point "X."

1-5795-8-26D

LAUNCH WINDOW PROVIDED BY ORBITAL MANEUVER TO SHIFT NODE



NOTE: $i_1 = i_2$

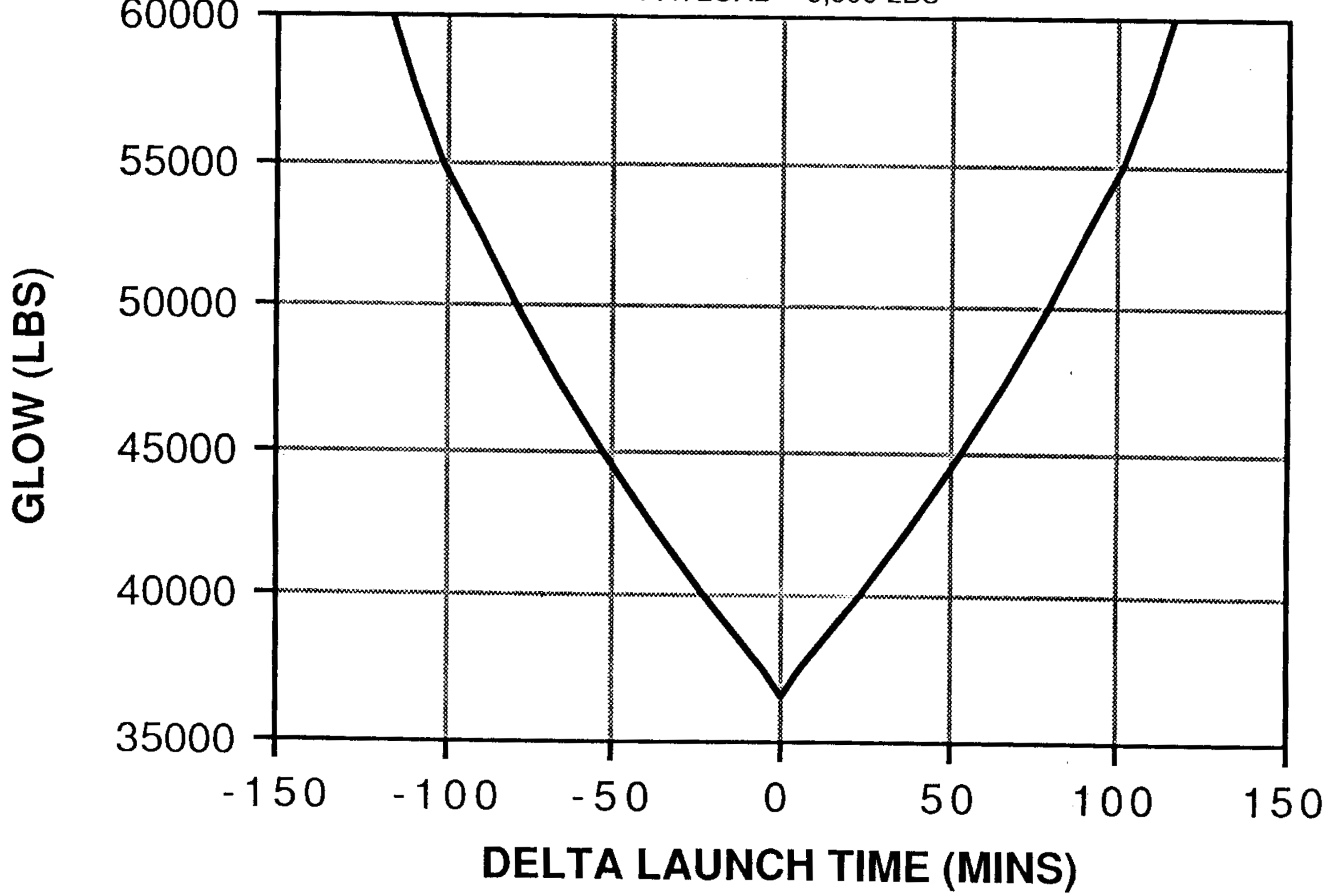
PD33/HAYS
12/17/87

MARS ASCENT VEHICLE

This chart shows the effect on GLOW of an early or late launch time. Negative delta launch time refers to an early launch, and positive a late launch. The Gross Lift-Off Weight (GLOW) increases as the launch time deviates from the in-plane condition (delta launch time = 0). This data is for a payload of 6,000 lbs.

MARS ASCENT VEHICLE

IMPULSIVE ORBITAL MANUEVER FOR NODAL SHIFT
1,000 KM (540 NMI) / 74.73 DEG INC.
PAYLOAD = 6,000 LBS



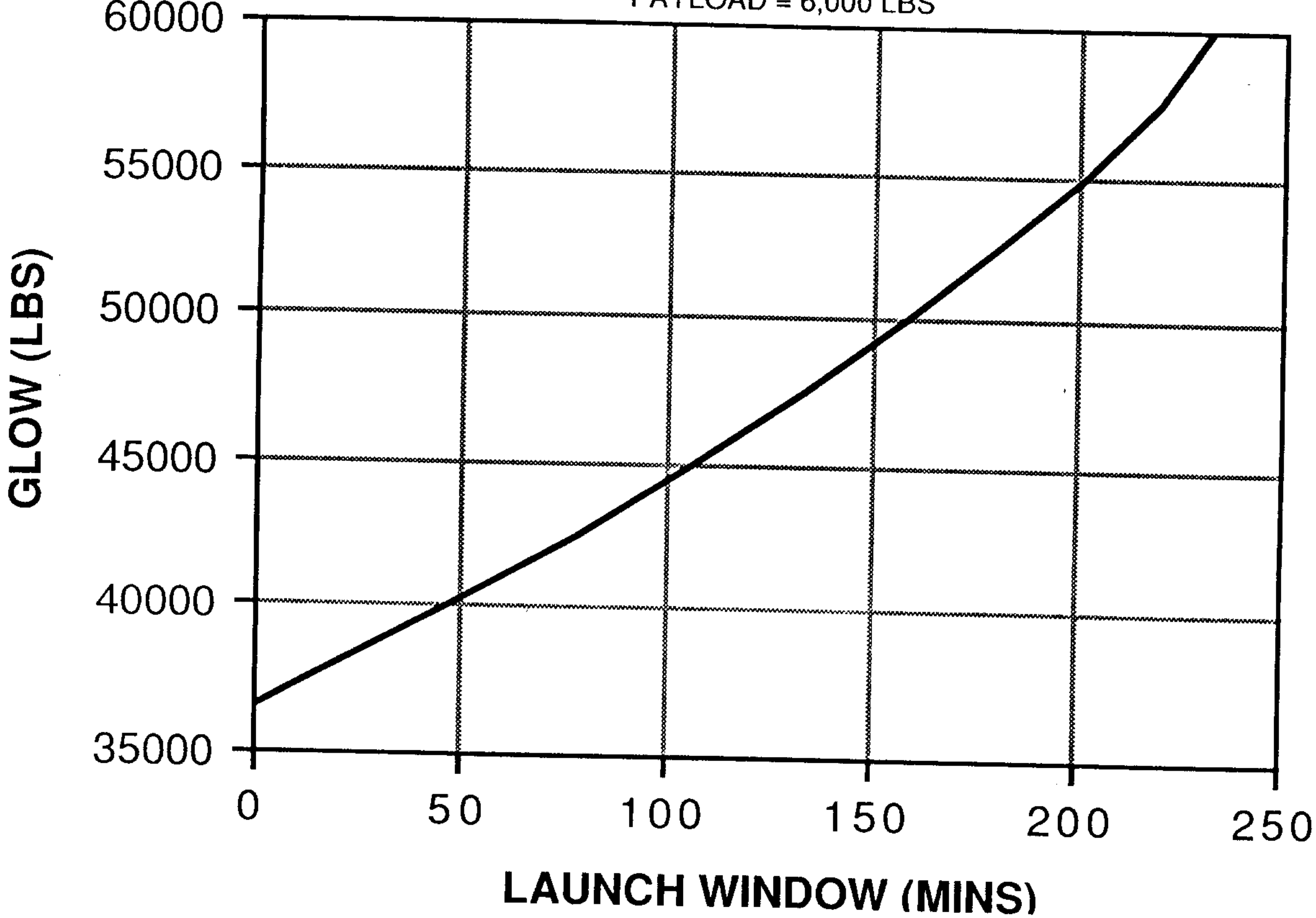
PD33/HAYS
DEC. 16, 1987

MARS ASCENT VEHICLE

The facing chart depicts GLOW vs. Launch Window for a 6,000 lb. payload. As expected, a wider launch window requires a larger GLOW.

MARS ASCENT VEHICLE

IMPULSIVE ORBITAL MANUEVER FOR NODAL SHIFT
1,000 KM (540 NMI) / 74.73 DEG INC.
PAYLOAD = 6,000 LBS



PD33/HAYS
DEC. 16, 1987

MARS ASCENT PERFORMANCE

This chart summarizes the results of the second performance update. A GLOW of 36,661 lbs. is required to place a 6,000 lb. payload into a 1,000 KM (540 n.mi.)/74.73 deg. inclination orbit with no launch window. As shown on the previous chart, a GLOW of 45,000 lbs. will provide a 106 minute launch window.

MARS ASCENT PERFORMANCE

1,000 KM (540 N.MI.) / 74.73 DEG. INCLINATION

RESULTS:

0 GLOW = 36,661 LBS IS REQUIRED FOR THE NOMINAL TRAJECTORY (NO LAUNCH WINDOW)

0 GLOW = 45,000 LBS WILL PROVIDE A 106 MINUTE LAUNCH WINDOW

PD33/HAYS
JAN. 28, 1988

THIS PAGE INTENTIONALLY LEFT BLANK

MARS ASCENT VEHICLE PERFORMANCE

- O INITIAL PERFORMANCE DATA (SEP. 1987)
  - o PRELIMINARY PERFORMANCE TO 1,000 KM (540 NMI) / 74.73 DEG INC.
  - o SINGLE STAGE VS. DROP TANKS
  - o FIXED VS. OPTIMUM INJECTION
- O 1ST PERFORMANCE UPDATE (OCT. 1987)
  - o PRELIMINARY PERFORMANCE TO 1,000 KM (540 NMI) / 74.73 DEG INC.
  - o SENSITIVITY TO:
    - FLIGHT PERFORMANCE RESERVE
    - ACCELERATION LIMIT
    - ATMOSPHERE MODEL
  - o TRAJECTORY
- O 2ND PERFORMANCE UPDATE (JAN. 1988)
  - o FINAL PERFORMANCE TO 1,000 KM (540 NMI) / 74.73 DEG INC.
  - o LAUNCH WINDOW
- O PERFORMANCE TO MARS SYNCHRONOUS ORBIT (FEB. 1988)
  - o PERFORMANCE TO 500 X 33,562 KM (270 X 18,122 NMI) / 74.73 DEG INC.
  - o SENSITIVITY TO:
    - VACUUM THRUST
    - THRUST-TO-WEIGHT AT LIFT-OFF

PD33 / HAYS

MARS ASCENT PERFORMANCE

This chart deals with performance to a highly elliptical, Mars synchronous orbit. Changes to the groundrules and assumptions used in the second update include a final orbit of 500 KM x 33,562 KM (74.73 deg. inclination), and no launch window. A GLOW of 49,062 lbs. is required to place a 6,000 lb. payload into this orbit.

Note: Because of the Mars synchronous orbit, this data does not apply directly to the Split Sprint Mission.

MARS ASCENT PERFORMANCE

PURPOSE: DETERMINE THE GROSS LIFT-OFF WEIGHT (GLOW) AND IDEAL VELOCITY REQUIRED FOR LAUNCH INTO A HIGHLY ELLIPTICAL, MARS SYNCHRONOUS ORBIT.

GROUNDULES AND ASSUMPTIONS SAME AS "2ND UPDATE" EXCEPT:

0 FINAL ORBIT : 500 X 33,562 KM (270 X 18,122 N.MI.) / 74.73 DEG. INCLINATION

0 NO LAUNCH WINDOW

RESULTS:

0 GLOW = 49,062 LBS

0 IDEAL VELOCITY = 18,395 FPS = 5.606 x

PD33/HAYS
JAN. 28, 1988

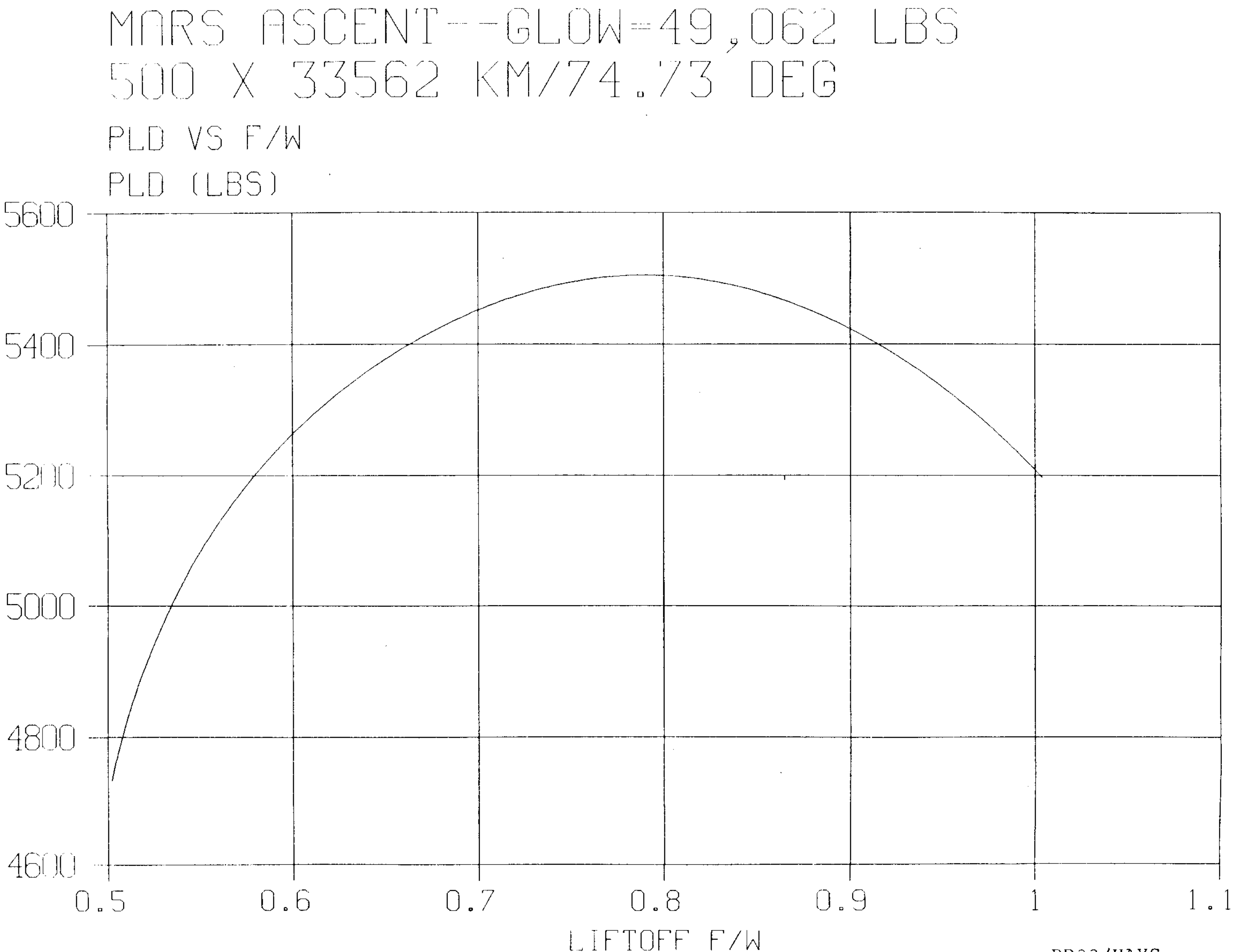
PAYLOAD VS. VACUUM THRUST

This chart shows the effect of varying thrust level for the Mars synchronous mission discussed on the previous chart. Payload is maximized when the vacuum thrust is approximately 39,000 lbs. This indicates that the 40,000 lb. thrust which was assumed on the previous chart is about optimum (assuming no launch window is required).

PD33/HAYS
2/29/88

PAYLOAD VS. THRUST-TO-WEIGHT AT LIFT-OFF

This chart shows payload as a function of F/W at lift-off for the Mars synchronous mission. Once again, GLOW is held constant at 49,062 lbs. (optimum for a 40,000 lb. thrust engine). Payload is maximized when F/W is approximately 0.8 Earth G's at lift-off.



PD33/HAYS
2/29/88

CONCLUSION

The facing chart lists the pertinent results in regard to Mars Ascent Vehicle Performance.

CONCLUSION

- o SINGLE STAGE OFFERS BETTER PERFORMANCE THAN DROP TANKS
- o OPTIMUM INJECTION OFFERS BETTER PERFORMANCE THAN FIXED INJECTION
- o WARM ATMOSPHERE MODEL IS WORSE CASE
- o LAUNCH WINDOW PROVIDED BY ORBITAL MANEUVER TO SHIFT NODE
- o PERFORMANCE TO 1,000 KM (540 N.MI.)/74.73 DEG INC./6 K-LBS PAYLOAD:
  - GLOW = 36,661 LBS. FOR NOMINAL TRAJECTORY (NO LAUNCH WINDOW)
  - GLOW = 45,000 LBS. WILL PROVIDE 106 MINUTE LAUNCH WINDOW
- o PERFORMANCE TO 500 X 33,562 KM (270 X 18,122 N.MI.)/74.73 DEG INC/6 K-LBS PAYLOAD*:
  - GLOW = 49,062 LBS. (NO LAUNCH WINDOW)
  - 40 K-LBS VAC. THRUST PROVIDES OPTIMUM PERFORMANCE

*THIS DATA IS FOR COMPARATIVE USE ONLY, AND IS NOT RELATED TO THE MAIN OBJECTIVE OF THIS REPORT (AS STATED IN INTRODUCTION).

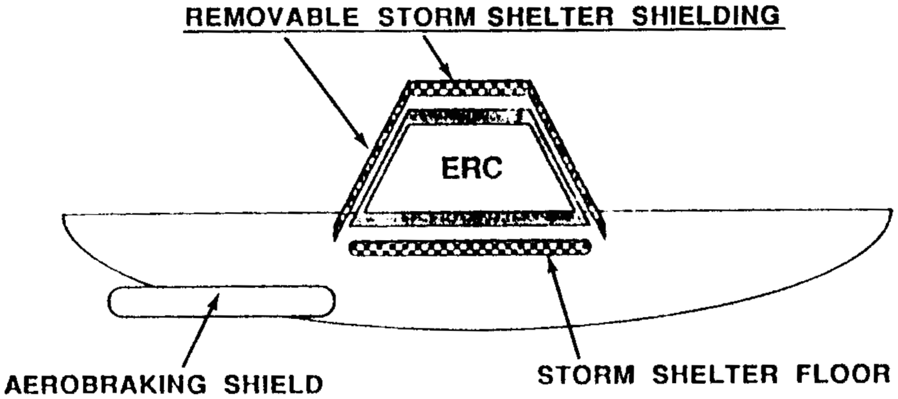
PD33/HAYS

AEROCAPTURE OF THE EARTH RETURN CAPSULE

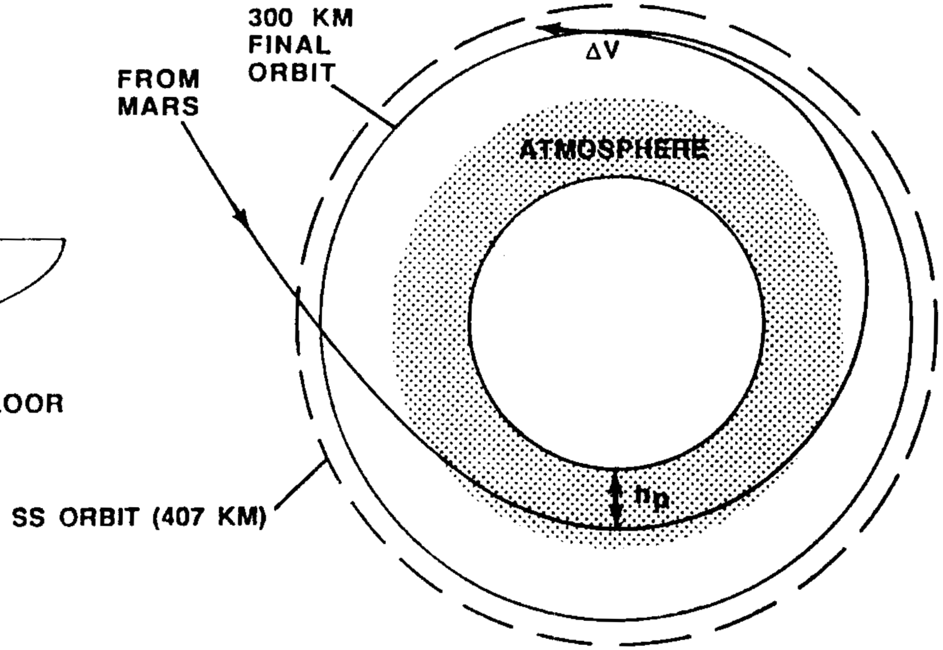
Rather than return the entire piloted vehicle to Earth orbit, it would be much more efficient to return only a smaller crew vehicle that would be subsequently retrieved from low Earth orbit. A detailed trajectory simulation was performed to analyze the propulsion requirements of the Earth return capsule. This analysis used the results of a vehicle weight analysis performed by Mark Sargent (MSFC). The trajectory simulation was run using the integrated mission program created by Vince Dauro (MSFC/PD34). The program was used to simulate the return of the vehicle from its heliocentric orbit to a 300 kilometer altitude Earth parking orbit. The Earth return vehicle uses aerobraking to achieve the necessary reduction in energy to enter Earth orbit. By aiming for the proper perigee altitude prior to aerobraking, the Earth return capsule can be placed into an orbit with an apogee of 300 km. When the vehicle reaches apogee, the orbit can be circularized using its propulsion system. The Earth return capsule and/or its crew could be retrieved by another vehicle based at the Space Station or on the ground.

AEROCAPTURE OF EARTH RETURN CAPSULE

EARTH RETURN CAPSULE



AEROCAPTURE TRAJECTORY



EARTH RETURN CAPSULE WGT.	=	19336 LBS
AEROBRAKE WGT.	=	2273 LBS
TOTAL BURN OUT WGT.	=	21609 LBS
TOTAL PROPELLANT WGT.	=	318 LBS
TOTAL RCS PROPELLANT	=	105 LBS
TOTAL GROSS WGT.	=	22032
AEROBRAKE DIAMETER	=	37.5 FT.
ISP	=	460 SEC.

INCOMING C_3	=	11.74 KM^2/SEC^2
hp	=	81.59 KM
TIME IN ATMOSPHERE	=	796.5 SEC
MAX. ACCELERATION	=	2.81 g's
AVG. ACCELERATION	=	1.28 g's
MAX. DYN. PRESSURE	=	30.37 LBS/FT^2
TOTAL DELTA VELOCITY	=	65.353 M/S

EARTH RETURN CAPSULE WEIGHT STATEMENT VALIDATION STUDY

The next section of this document presents a detailed analysis of the Earth return capsule. This study was performed by Mark Sargent of the University of Illinois at Champaign-Urbana as part of the Universities Space Research Association Summer Fellowship Program. The report that follows represents a seven week study in which detailed weight statements for all of the subsystems of the Earth return capsule were developed.

USRA/NASA UNIVERSITIES ADVANCED ENGINEERING DESIGN PROGRAM

UNIVERSITIES SPACE RESEARCH ASSOCIATION/NATIONAL AERONAUTICS AND SPACE ADMINISTRATION

EARTH RETURN CAPSULE WEIGHT STATEMENT VALIDATION STUDY

WEIGHT STATEMENT RESEARCH FOR AN EARTH-AEROBRAKING CREW RECOVERY VEHICLE FOR A MANNED MARS MISSION

AUGUST 1987

COMPILED BY MARK SARGENT, USRA VISITING SUMMER FELLOW

EARTH RETURN CAPSULE

INTRODUCTION

This section summarizes the results of a study of the weight breakdown of an Earth Return Capsule (ERC). This breakdown is used to validate the ERC weight suggested by John Niehoff of Science Applications International Corporation (SAIC) in Humans to Mars, a Space Leadership Initiative. A secondary purpose of this study is to provide a roadmap to personnel expertise and reference material in order to facilitate further research at both Marshall Space Flight Center and the author's school, the University of Illinois at Champaign-Urbana.

This study was performed under the direction of the Preliminary Design office at Marshall Space Flight Center. The research was funded by the University Space Research Association (USRA) under the USRA/NASA Universities Advanced Engineering Design Program. The objective of this annual summer program is to determine how NASA and the university community environment can best work together to produce a meaningful product in engineering design.

What is an Earth Return Capsule?

The Earth Return Capsule (ERC) is a life support capsule to house the crew of a manned Mars mission. The ERC has two primary functions: 1) it serves as a life support capsule within the radiation-shielding "Storm Shelter" in the event of a Solar Particle Event, and 2) it is the command module for the Earth Return Vehicle which houses the crew during aerobraking and recovery at Earth..

Why study the weight breakdown of an Earth Return Capsule?

It is important to have a valid estimate of the weight of the ERC because of the significant mass leverage which it possesses. An additional pound of spacecraft returned to Earth would very likely mean thousands of pounds of additional initial Low Earth Orbit (LEO) mass.

EARTH RETURN CAPSULE

INTRODUCTION

- **What is an Earth Return Capsule?**
  - **life support capsule within the radiation-shielding "Storm Shelter"**
  - **command module for the Earth Return Vehicle**

- **Why study the weight breakdown of an Earth Return Capsule?**
  - **significant mission-mass leverage demands a valid estimate**

STUDY APPROACH

First, the ERC requirements were determined from the Manned Mars Mission model. Next, the ERC was broken down into specific subsystems for analysis by subsystem engineers within the preliminary design office. Subsystem weight were determined by the subsystem engineers and assembled into a final weight statement for comparison to the Niehoff baseline.

STUDY APPROACH

- ERC requirements determined from Mission Model
- ERC subsystem breakdown determined
- Subsystem Engineers consulted, subsystem weights determined
- Subsystems integrated into final weight statement

AGENDA

The outline of the report is presented on this chart. The report consists of three main portions. The first is an introduction to the Manned Mars Mission and ERC. This is followed by the specific subsystem analyses. Finally, the ERC weight statement is given.

AGENDA

I. Introduction
- Manned Mars Mission Description
- Earth Return Capsule Mission Description
- Earth Aerocapture
- Earth Return Vehicle Configuration
- General Requirements and Assumptions

II. Subsystem Analyses
- Guidance, Navigation and Control
- Communications
- Life Support
- Electrical Power
- Primary Propulsion and RCS
- Storm Shelter Shielding
- Thermal Control
- Crew and Seating
- Structures

III. Summary
- Earth Return Capsule Weight Statement
- Recommended Further Study

IV. Attachments
- Acronyms and Abbreviations
- References

Manned Mars Mission

The following is a brief description of the piloted portion of the Manned Mars Mission, only those details which drive the Earth Return Capsule design are included. This summary is included in order to establish a framework from which the missions of the Earth Return Capsule can be better understood.

Six astronauts are launched from low earth orbit (LEO) into an Earth-Mars transfer orbit. They spend the 224-day trip in a command module called the Mission Module (MM). Upon arrival at Mars, three of the crew descend to the surface for a 30-day stay, the other three remain in the MM orbiting the planet. After 30 days, the surface crew leaves Mars and transfers to the Mission Module. The Mission Module carries the crew for the return Mars-Earth transfer which takes 165 days. One day out from Earth, the crew climbs into the Earth Return Vehicle (ERV), which then breaks away from the Mission Module. The astronauts pilot the ERV through an aerobraking maneuver in the Earth's atmosphere to dissipate their transfer orbit energy, and end up in a highly elliptical Earth orbit. An apogee burn places the ERV in a circular orbit suitable for pickup by an Orbital Maneuvering Vehicle (OMV) and transfer to the Space Station.

For our purposes the overall mission can be broken into three pieces: The initial leg, the 30 day stay at Mars, and the return leg. During all three segments of the journey, solar flare radiation could easily kill an unprotected crew. Protection from these Solar Particle Events (SPE) while in space is provided by a "Storm Shelter". As currently conceived, the Storm Shelter is a small life support capsule surrounded by suitable radiation shielding. The Earth Return Capsule surrounded by detachable aluminum radiation shielding panels fulfills all Storm Shelter requirements: life support for up to five days, external communications, and all other neccessities for a possible five day SPE. This establishes the first mission of the Earth Return Capsule - to function as the life support capsule for a Storm Shelter.

The Earth Return Vehicle is a completely autonomous spacecraft composed of two major pieces: First, the Earth Return Capsule, which serves as the control module for the ERV. Second, the aerobraking shield, which enables the ERV to decelerate in Earth's atmosphere. This Earth Return Vehicle configuration establishes the second mission of the Earth Return Capsule - to function as the control module for the Earth Return Vehicle.

Manned Mars Mission

Space Station Assembly of Mars Vehicle Checkout and Preparation	240 days
Earth-Mars Transfer and Mars Aerobrake	224 days
Rendezvous with cargo vehicle Prepare landing vehicle Mars surface exploration (3 crew) Prepare for Earth return	30 days
Mars-Earth Transfer	165 days
Mission Module and ERC separation ERC completes Earth-aerobraking maneuver	1 day
OMV rendezvous and dock Maneuver to Space Station	2 days

Dual Function of the Earth Return Capsule

I. Life Support Capsule for the Storm Shelter

"Space missions to other planets, including Mars, where most of the mission is outside the protection of the Earth's magnetosphere, will subject the mission crew to radiation hazards from solar particle events (SPE) produced by solar flares. Radiation from this source may reach levels of several hundred rads in periods of a few hours." (Ref 2)

The Storm Shelter will provide this protection, maintaining full life support as well as restricted Mission Module control and communication for up to to five days. The life support consummables can be recharged so that the Storm Shelter can be used as needed for the fourteen month mission.

II. Command Module for the Earth Return Vehicle

As the Mission Module nears Earth on the return leg from Mars, the crew climbs into the Earth Return Vehicle (ERV), which then breaks away from the Mission Module. It is a fully operational spacecraft which takes the crew through the aerobraking maneuver. The ERC is the command module portion of the ERV, in the same way that the Mission Module served as the command module for the entire Earth-Mars piloted spacecraft.

Dual Function of the Earth Return Capsule

- **Life Support Capsule for the Storm Shelter**
  - provides full life support within the radiation-shielded Storm Shelter during a potentially lethal Solar Particle Event
- **Command Module for the Earth Return Vehicle**
  - serves as the command module for the ERV during the aerobraking maneuver at Earth

Earth Aerocapture Mission Profile

The following page illustrates the mission profile of the aerobraking maneuver used at the end of the return leg to decelerate the Earth Return Vehicle (ERV). (Ref 3)

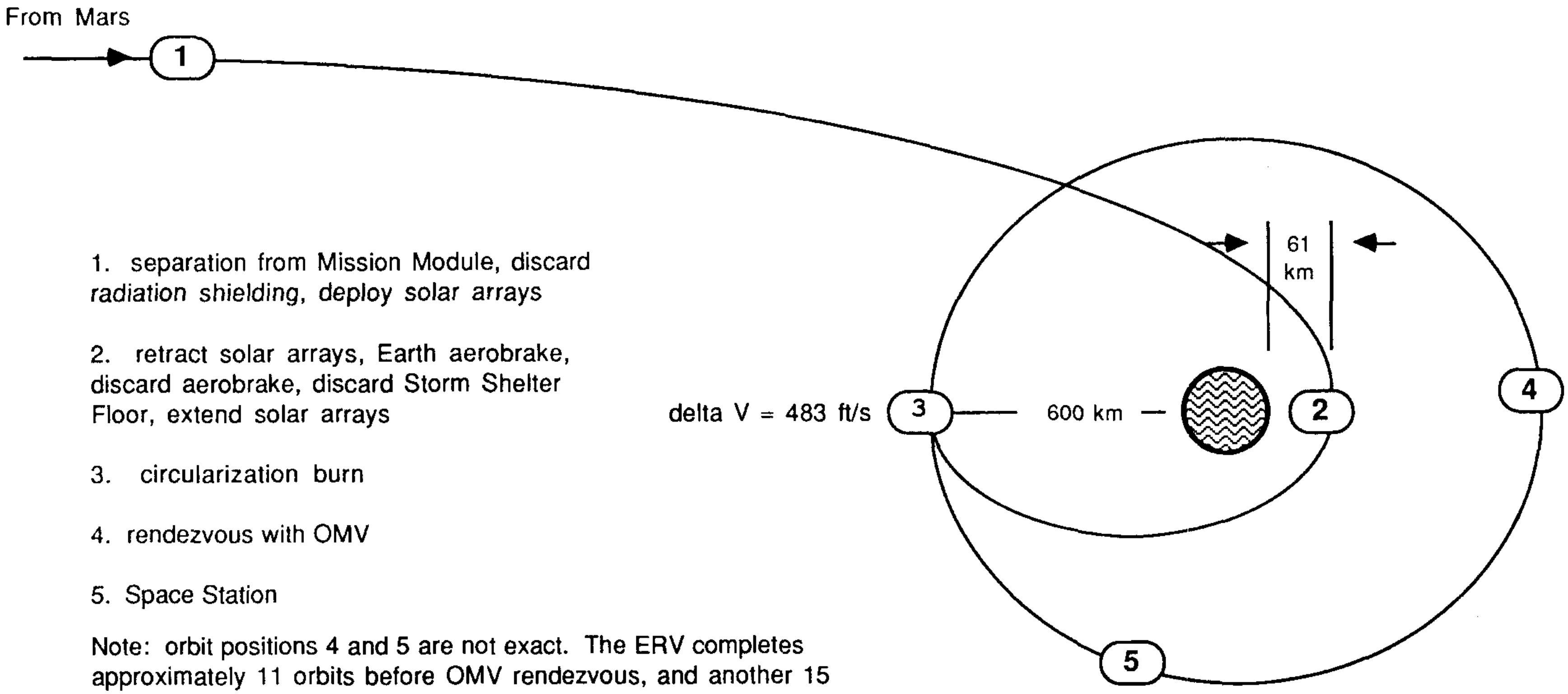
Approximately one day out from Earth the crew will abandon the Mission Module and board the ERV. The ERV separates from the Mission Module, which enters a heliocentric orbit. In the baseline mission for this study, the Mission Module is discarded. Next, the aluminum radiation shielding covering the outside walls of the ERC is jettisonned to lighten the spacecraft and allow the body mounted thermal radiators to function. The portion of shielding which is mounted between the ERC and the aerobrake cannot be jettisonned until the ERC separates from the aerobrake. Finally, the solar panel arrays are deployed on two extendible booms.

The ERV coasts in this configuration for approximately 24 hours; the reaction control system (RCS) lines up the spacecraft precisely for the aerobraking maneuver. Approximately one hour before the aerobraking maneuver, the solar panels are retracted and the ERV switches to battery power.

The duration of the aerobraking maneuver is between five and twelve minutes. After this aerobraking maneuver, the aerobrake and Storm Shelter floor are jettisonned, and the solar arrays are redeployed. When the spacecraft reaches the apogee of its' elliptical orbit, an impulsive burn from the 500 lbF main engine circularizes the orbit. An OMV requires approximately 18 hours to match positions with the ERC, and three hours to dock.

The OMV transports the ERC to the Space Station in approximately 24 hours.

Earth Aerocapture Mission Profile



1. separation from Mission Module, discard radiation shielding, deploy solar arrays

2. retract solar arrays, Earth aerobrake, discard aerobrake, discard Storm Shelter Floor, extend solar arrays

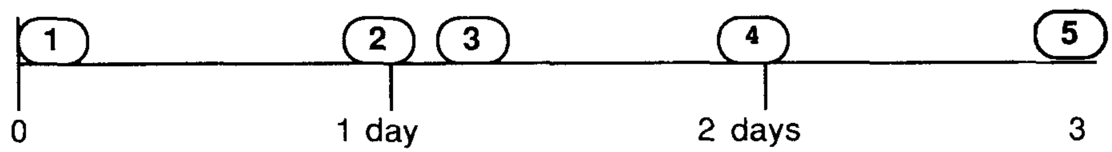
3. circularization burn

4. rendezvous with OMV

5. Space Station

Note: orbit positions 4 and 5 are not exact. The ERV completes approximately 11 orbits before OMV rendezvous, and another 15 orbits before docking with the Space Station.

Aerocapture Timeline



Earth Return Vehicle Configuration

The Earth Return Capsule (ERC) is the life support capsule containing the crew.

The Earth Return Vehicle (ERV) is composed of the ERC, Aerobraking shield, and the Storm Shelter Floor.

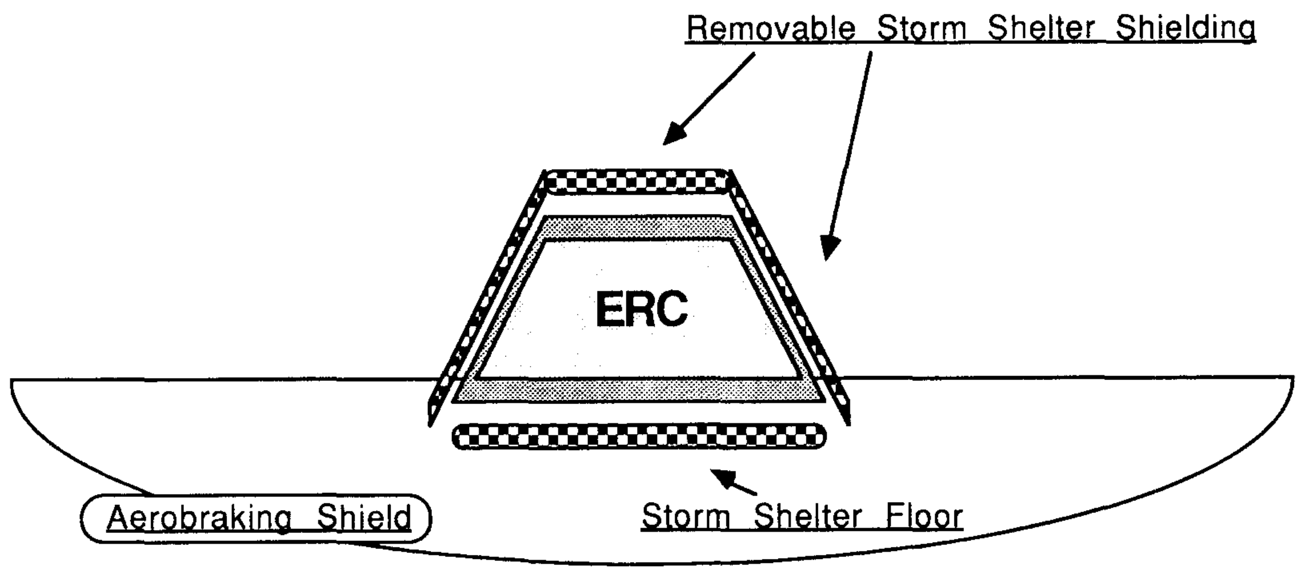
The Aerobraking shield is the detachable shield used at the Earth aerobraking maneuver.

A Solar Particle Event (SPE) is a solar flare releasing very high levels of radiation.

The Storm Shelter is composed of the ERC, enclosed by Aluminum radiation shielding. The astronauts remain within the ERC during Solar Particle Events.

The Storm Shelter Floor is that portion of the Storm Shelter that cannot be jettisonned until the aerobrake is detached.

Earth Return Vehicle Configuration



Fundamental ERC Requirements

Given the dual function description and Manned Mars Mission description, it is possible to determine the fundamental requirements of the ERC.

FUNDAMENTAL REQUIREMENTS: STORM SHELTER FUNCTION

The astronauts will retreat into the Storm Shelter in the event of a Solar Particle Event. The ERC must provide up to 5 days of full life support in the event of this emergency. In addition, the ERC must provide some capability of Mission Module control, and communications during this retreat.

FUNDAMENTAL REQUIREMENTS: EARTH RETURN VEHICLE FUNCTION

The ERC is the command module for the Earth Return Vehicle. Following separation from the mission module, the ERV is a completely functional spacecraft composed of the ERC, the aerobraking shield, and little else. The ERC must provide all of the functions of a command module during aerobraking and orbital adjustment using chemical propulsion.

Fundamental ERC Requirements

STORM SHELTER

- provide life support: six men for five days
- provide Mission Module control, communications

EARTH RETURN VEHICLE

- Fully functioning command module during Earth aerobrake
- provide life support: six men for three days

GUIDANCE, NAVIGATION AND CONTROL

The Earth Return vehicle has to contain sufficient equipment to accurately accomplish the aerobrake maneuver. A list of components that can provide the needed knowledge and accuracy is shown. To determine the state vector of the ERV, the Global Positioning System (GPS) is used. The GPS data can also be used to determine the altitude at any given time. Use of the GPS becomes effective at geosync altitudes and lower. To accurately determine the attitude and attitude rate during the aerobrake maneuver, the ring laser gyro IMU is used. Updates from the star trackers and the GPS can be used prior to atmospheric entry to optimize the aerobrake maneuver. A computer is needed to process the information and control the thruster system. Also communications is provided between the ERV and ground to allow updates to various system parameters to be uplinked from the ground.

Guidance, Navigation and Control

Component	Quantity	Wt (lbs) each	Total weight (lbs)
Ring Laser Gyro IMU	2	25	50
Control Electronics	2	40	80
Computer/Communications	2	25	50
Power Amplifiers	2	15	30
Transponder	2	15	30
Antenna	1	2	2
Cabling	1	20	20
GPS	1	20	20
Star Tracker	2	10	20
Growth and Contingency (5% - Ref 5)			15
Total Weight			317

Life Support

ANALYSIS: All subsystems are sized for 6 men for 5 days. Important subsystems are typically given an extra day for contingency, critical subsystems even more.

Temperature and Humidity Control is provided by a condensing heat exchanger with a temperature and humidity control package (Ref. 13). Carbon Dioxide and Contamination Control is provided by a CO2 Control Assembly with replaceable LiOH cannisters (Ref. 13)

Pressure Control is provided by the O2/N2 Control Assembly (Ref. 13) The Nitrogen supply system is modeled after that of the Orbiter (Ref. 14). Oxygen supply was estimated for 60 man-days at 1.59 lbs per man-day.

Food management is provided by a Galley which contains all required dispensers, heaters, rehydration ports, water chiller, and stowage (Ref 13).

Water management: Grey water provided by the condensing heat exchanger fulfills hygiene requirements. Food reconstruction and drinking is estimated to be 6.9 lbs/man-day (Ref 17), storage tanks and plumbing (Ref. 13)

Waste Management consists of a commode, waste storage tank, and an estimate of liquid waste produced: urine, respiration/perspiration, and waste water.

Medical supplies include a medical life support module, physicians equipment, pharmacy, and medical supplies kit (Ref. 6).

Fire Protection is modeled after the Orbiter's systems (Ref. 14). One fixed Freon 1301 extinguisher, two portable extinguishers, and three detectors.

Miscellaneous equipment includes 28 items. For example: a workbench, clean wipes, radiation monitors, clothing, spacesuits, camera equipment and film, etc.

Life Support

Component	weight (lbs)
Temperature and Humidity Control	43
Carbon Dioxide and Contamination Control	194
Pressure Control	457
Oxygen Supply	249
Food Management	134
Water Management	319
Medical	103
Fire Protection	28
Miscellaneous Equipment	1562
growth and contingency (5% - Ref 5)	178
Total Weight	3267

COMMUNICATIONS AND DATA MANAGEMENT

The Communications and Data Management (C&DM) subsystem for the Earth Return Vehicle (ERV) is capable of supporting autonomous ERV operations during final return and capture in Low Earth Orbit (LEO) when it is separated from the Mission Module. The C&DM subsystem also provides sufficient controls and interfaces with the Mission Module to permit the crew to control the Mission Module and to communicate with the ground from the ERV when it is being used as a storm shelter. A preliminary equipment list for the ERV communications and data management system is provided on the facing chart. This set of equipment supports a dual redundant system.

COMMUNICATIONS AND DATA MANAGEMENT

Component		N	Wt (lbs) each	Total Weight
S-BAND:	Transponder	2	20	40
	parametric amplifier assembly	2	10	20
	power amplifier (30 watts)	2	16	32
	signal processor	2	30	60
	encrypt/decode	2	10	20
	low gain antennae	2	2	4
	communications and tracking processor	2	35	70
AUDIO:	audio controller	2	30	60
	audio storage unit	2	30	60
	voice recognition	2	35	70
	crew audio unit	2	2	4
	wall plugs	2	1	2
	remote speakers	2	3	6
K-BAND:	transponder	2	20	40
	power amplifier	2	10	20
	signal processor	2	15	30
	encrypt/decode	2	10	20
	low gain antennae	2	1	2
DATA:	central processor	2	20	40
	module processor	2	20	40
	database processor	2	20	40
	operations processor	2	20	40
	time/freq unit	2	10	20
	mass storage	10	14	140
	remote interface units	6	10	60
	multi-function control station	2	80	160
	caution and warning electronics	2	22	44
	annunciator assembly	2	6	12
	caution and warning sensors	12	0.17	2.04
	wiring and cables			125
	growth and contingency (5% - ref 5)			64
Total Weight				1347

Electrical Power

ERV Requirements	watts	
ECLSS: 6 men at 1000 w/each	6000	
Guidance, Navigation, and Control	400	
Communications and Data Handling	1000	
Propulsion and Thermal Control	1500	
Total	8900	W

Analysis: (Reference 11) This power requirement must be met for three days: one day after separation from the Mission Module, and for two days after the aerobrake maneuver. Because weight is critical, solar arrays were chosen. During the aerobrake, the arrays must retract and AgZn batteries are used until the arrays can be extended again after aerobrake. The requirement is rounded for 9 kW for calculations.

9 kW x (2.2 safety factor) = 20 kW array --> two 10 kW arrays
20 kW / 75 w/kg --> 267 kg --> 600 lbs of solar array

during aerobrake: assume a 3 hour retracted period, 3 hr x 9 kW = 27 kW-hr.
- 27 kW-hr / 38 W-hr/lb --> 710 lb + 90 lb packaging
- each battery can provide approximately 3500 W-hr --> 8 batteries + redundancy. Ten batteries at 100 lbs/ea = 1000 lbs
- Charger/Regulator 2 @ 80 lbs/ea = 160 lbs, Distribution and Wiring = 140 lbs

The ERV is deeply embedded within the Mission Module and the solar arrays can't be extended before separation. Therefore, the Mission Module must supply the required power during Storm Shelter use.

Electrical Power

Component	N	weight each	total weight (lbs)
10 kW Solar Array	2	300	600
AgZn Batteries	10	100	1000
Charger/Regulator	2	80	160
Wiring and Distributors	1	140	140
growth and contingency (5% - Ref 5)			95
Total Weight			1995

EARTH RETURN CAPSULE ALTERNATIVE ELECTRICAL POWER SYSTEM

During the review process for this document, Bob Giudici (MSFC/PD14) (Ref 25) determined that the power requirements of the Earth Return Capsule could be decreased significantly since the crew would only be using it for two or three days. He also determined that lithium batteries should be used rather than silver-zinc batteries. This chart summarizes Mr. Giudici's calculations for the electrical power system using these two modifications. It is shown that the weight of the power system is decreased significantly, this would allow weight reductions in the thermal control system as well as reducing total vehicle weight. This weight reduction in the Earth Return Capsule would filter back through all of the propellant requirements to reduce the total Earth departure weight of the piloted vehicle by about 9600 lbs. The impacts of this weight reduction on the other subsystems was not investigated.

EARTH RETURN CAPSULE ALTERNATIVE ELECTICAL POWER SYSTEM

ERC REQUIREMENTS	WATTS
ECLSS : 6 PEOPLE AT 250 WATTS/ PERSON	1500
GUIDANCE,NAVIGATION,AND CONTROL	100
COMMUNICATIONS AND DAT HANDLING	200
PROPULSION AND THERMAL CONTROL	200
TOTAL	2000

ANALYSIS

LITHIUM BATTERIES

SOLAR ARRAYS

TWO 1 KW ARRAYS NEEDED	
2 KW @ 75 W/KG = 27 KG =	60 LBS

DURING AEROBRAKING

3 HOUR ON BATTERIES = 6 KW-HR	
6000 W/ (67 W/LB) =	90 LBS
6000 W/ (600 W-HR/BATTERY)= 10 BATTERIES	
10 BATTERIES @ 9 LBS EACH =	90 LBS
2 CHARGERS/REGULATORS =	20 LBS
DISTRIBUTION AND WIRING =	50 LBS
5% CONTINGENCY	11 LBS
TOTAL WEIGHT	321 LBS

Primary Propulsion and RCS

ANALYSIS: Assumptions - 1) the Primary Propulsion burns are performed after jettisonning the Storm Shelter floor and the aerobrake shield. 2) all RCS burns are performed while shield, brake, and capsule are still attached. 3) both systems are cryogenic, and use the same tanks.

The mission profile calls for aerobraking at an altitude of 81 km, resulting in an elliptical orbit with an apogee and perigee radius of 600 km and 81 km respectively. At apogee, the main 500 lbF engine will produce a delta-V of 483 ft/s to circularize the orbit (Ref. 3).

The main engine is an XLR-134 (Ref. 10), producing 500 lbF at an Isp of 460 lbf-s/lbm. The circularization burn requires 497 lbs of LO2/LH2.

The RCS engines (Ref. 8 and 9) produce 70 lbF at 420 lbf-s/lbm. Three RCS delta-V's are assumed: from separation to aerobrake: 200 ft/s, during aerobraking: 500 ft/s, and from aerobrake to Space Station: 50 ft/s. These maneuvers require 1168 lbs of LO2/LH2.

Boiloff is 83 lbs. Giving a total propellant weight of 1748 lbs.

Cryogenic tankage, boiloff, and contingency were sized by Robert Champion (Ref. 8). The insulation and distribution systems are sized as shown (Ref. 8, 10). Twelve RCS thrusters provide full redundancy. Growth and contingency includes fuel lines, metering, bracketing, etc.

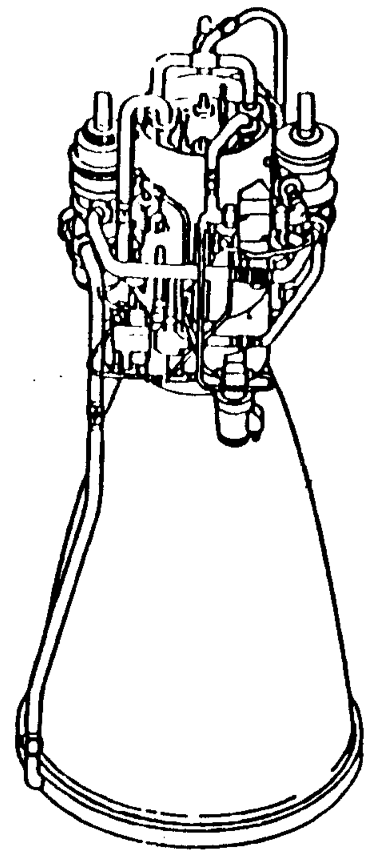
Primary Propulsion and RCS

Component	N	weight each	total weight (lbs)
Primary Propulsion Subsystem:			
XLR-134 Engine (500 lbF)	1	125	125
Liquid Hydrogen Tanks	2	21	42
Insulation	2	16	32
Vapor Cooled Shield (VCS)	2	30	60
Liquid Oxygen Tanks	1	14	14
Insulation	1	12	12
VCS	1	23	23
Reaction Control System:			
70 lbF Thrusters	12	15	180
Pressurization System for RCS and Primary Propulsion	1	129	129
Growth and Contingency (15% - Ref. 5)			92
Total Dry Weight			709
Cryogenic Fuels:			
Liquid Hydrogen			270
Liquid Oxygen			1478
Total Wet Weight			2457

XLR-134

LISTED ON THE FACING PAGE ARE THE CHARACTERISTICS OF THE XLR-134. THE XLR-134 WAS USED FOR THE MAIN PR0PULSION SYSTEM ON THE ERC.

XLR-134



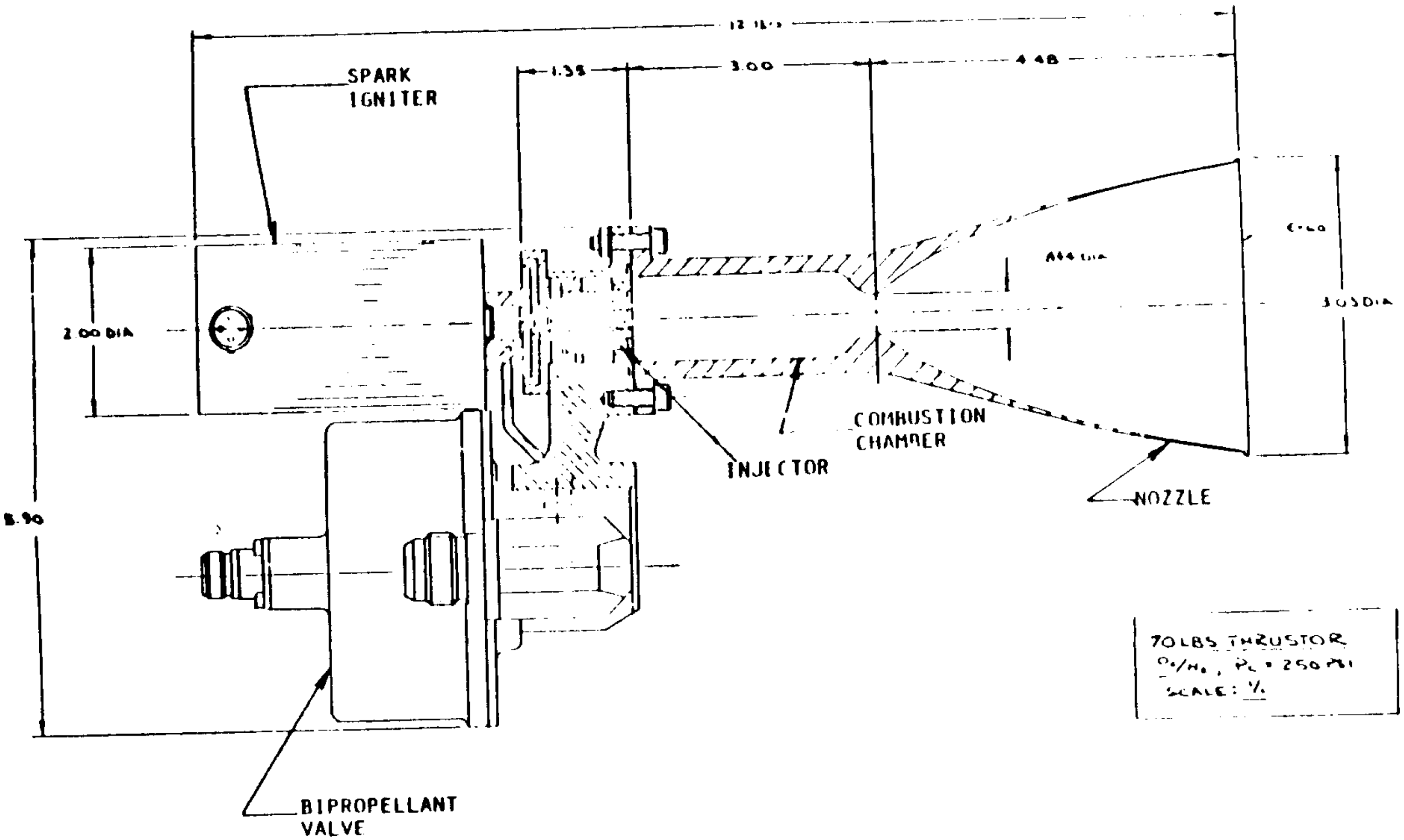
- Thrust 500 lbf
- Propellants LO_2/LH_2
- Cycle Expander
- ISP 460 + sec
- Length 48 In.

CHAMBER PRESSURE 500 PSIA
EXPANSION RATIO 770
WEIGHT 125 LB
EXIT DIAMETER 22 IN

AEROJET TechSystems COMPANY

70 LBF THRUSTER

LISTED ON THE FACING PAGE ARE THE CHARACTERISTICS OF THE RCS THRUSTER. THE THRUSTER WAS USED FOR ATTITUDE CONTROL ON THE ERC.



10-2

THRUST	70 LBF	WEIGHT	15 LB
PROPELLANTS	LOX/LH	LENGTH	12.1 IN
ISP VAC	420.5 S	EXIT DIAMETER	3.0 IN
EXPANSION RATIO	60:1	MIXTURE RATIO	4
CHAMBER PRESSURE	250 PSIA		

PD13
063087

Thermal Control

Analysis: (Reference 12) The total power requirement of 9000 wattts is also the heat rejection requirement, and drives the design of this subsystem. The heating that occurs during the aerobraking maneuver is brief enought so as not to be a major concern. Body mounted radiators are chosen because their design is not dramatically affected by the aerobraking maneuver, they don't have to be retracted or extensively shielded. The Replaceable TAG 54 Heat Pipe Radiator was selected in the Space Station study, and is scaled from 14.2 kW to 9.0 kW for the table of weights below. Growth and contingency includes the fluid lines, capsule insulation, etc.

Assumed Thermal Load: 9 kW
Scale Factor = Thermal Load / 14.2 kW = 0.633802

Before separation from the Mission Module, the radiators will be covered by Storm Shelter Shielding. Therefore, the Mission Module must provide heat rejection capability through an interface with the Earth Return Capsule during Storm Shelter use.

Thermal Control

Component	weight (lbs)
Main Panel	973
Heat Exchanger	98
Panel Attachments	64
Panel Fluid	17
Pumped Loop Dry Weight	160
Pumped Loop Fluid Weight	126
growth and contingency (20% - Ref 5)	287
Total Weight	1725

Crew and Seating

<u>ANALYSIS:</u> The seats must withstand high-G loading (approximately 3 g's - Ref. 3) during aerobrake. The shuttle seats are designed to survice a high-G crash landing (Ref 6). The aluminum seats in Reference 4 are designed for up to 30 g's.

Crew and Seating

Component	Wt (lbs)
Man	170
Personal	5
Seat	59
Clothes	5
Miscellaneous	2
Total x 6 men	1446

Structure

ANALYSIS: This subsystem includes all supporting structure, the physical walls and structure of the Earth Return Capsule. Lacking the resources to create a detailed design, a first-order estimate of the structural weight can be estimated as a percentage of the total spacecraft weight (References 5 and 19). Previous studies on similar spacecraft indicated a structural weight of approximately 25% of total spacecraft dry weight (Ref. 4)

Structure

Assumed ERC Dry Weight: 15500 lbs

Structure = 25% of ERC Dry Weight

= 3875 lbs

Earth Return Capsule Weight Statement

This table summarizes the subsystem weight analyses. A dry weight of 15546 lbs is indicated for the ERC. This is compared to a weight of 12877 lbs for the ERC baselined in the SAIC study. The propellant required for this ERC mission is 1748 lbs as shown.

Earth Return Capsule Weight Statement

Subsystem	Weight (lbs)
Crew and Seats	1446
Guidance, Navigation and Control	317
Environmental Control and Life Support	3742
Communications	1347
Primary Propulsion and Reaction Control	709
Electrical Power	1995
Thermal Control	1725
Science Return	300
Structure	3875
Dry Weight	**15456 lbs**

Propellant Weight: 1748 lbs

Niehoff's Return Capsule Dry Weight: **12877 lbs**

(Reference: page 12, 11 June 87 DATAFAX from John Niehoff of Science Applications International Corporation to Tom French, MSFC)

Conclusion

The purpose of this study was to provide a preliminary weight breakdown of an Earth Return Capsule and compare it to that proposed by SAIC. The breakdown is shown above, comparing the numbers is more difficult because of the lack of information available concerning the Niehoff study. More data is needed from SAIC before these figures can be meaningfully compared. First and foremost, a more detailed breakdown of Niehoff's Capsule Dry Weight would indicate differences in analysis. For example: Is Niehoff's weight a "dry" weight or a "wet" weight? In the SAIC proposal, it is termed a "damp" weight, but no propellant is manifested. What assumptions were made, if any, of weight-saving advanced technology? Are the aerobrake and Storm Shelter Floor discarded before the apogee burn, or saved for future missions? These are just a few of the questions raised during the comparative analysis. Given the unknowns of the Niehoff study, and the first-order analyses in this study, the Niehoff capsule weight is within acceptable limits.

Conclusion

- Estimated ERC Dry Weight ≈ 15,846 lbm
- Niehoff/SAIC ERC Weight 12,877 lbm
- To first-order, the Niehoff/SAIC ERC Weight is validated
- More information is required from the Niehoff study for a detailed comparison:
    - detailed subsystem breakdown
    - is it a "dry weight" or a "wet weight"?
    - what are the level-of-technology assumptions?
    - spacecraft configuration during key engine firings

Recommended Further Study

- Complete a true preliminary design of the vehicle concept as opposed to a mass analysis.

- Investigate configuration of external components for best fit with the Mission Module.

- Investigate use of disposable shielding for the Storm Shelter Floor that could be discarded before the aerobraking maneuver: liquid, propellant, pellets, etc.

- Life Support interface: What if the Mission Module Life Support System can be used during Storm Shelter operation? The ERC ECLSS can be sized for the three day ERV mission instead of the five day Storm Shelter mission.

- Equipment commonality and portability: Significant advantage could be gained by using modular equipment in the Mission Module that can be transferred to, and used in the ERV following separation from the Mission Module.

Recommended Further Study

- True preliminary design of the Earth Return Vehicle Concept
- Investigate external component configuration for best fit with Mission Module
- Investigate possible use of disposable Storm Shelter shielding
- ERC Life Support Requirements may be reduced
- Overall equipment commonality and portability could reduce duplication

4.0 SPECIAL ANALYSES

The following section contains the results of several special analyses that were performed during this study. These include a brief investigation into alternative mission profiles to the split sprint mission, solar flare prediction, and long term cryogenic propellant storage in space.

4.0 Special Analyses

4.1 Visits to Phobos or Deimos
4.2 Alternate Mission Profile Options
4.3 Solar Flare Analysis
4.4 Long Term Cryogenic Storage

TRANSFERS TO PHOBOS OR DEIMOS

Several overall mission and vehicle requirements were evaluated in order to determine the impact of adding visits to Phobos and/or Deimos to the mission. It was assumed that an extra Phobos/Deimos transfer vehicle would be added to the cargo vehicle payload for this purpose. This vehicle would be required to transfer from the 1000 kilometer altitude/74 degree inclination parking orbit to the near equatorial orbits of Phobos and Deimos. The mission options that were considered were visits to Phobos, Deimos, or both. The trajectory options that were considered were two impulse transfers and three transfers with and without aerobraking.

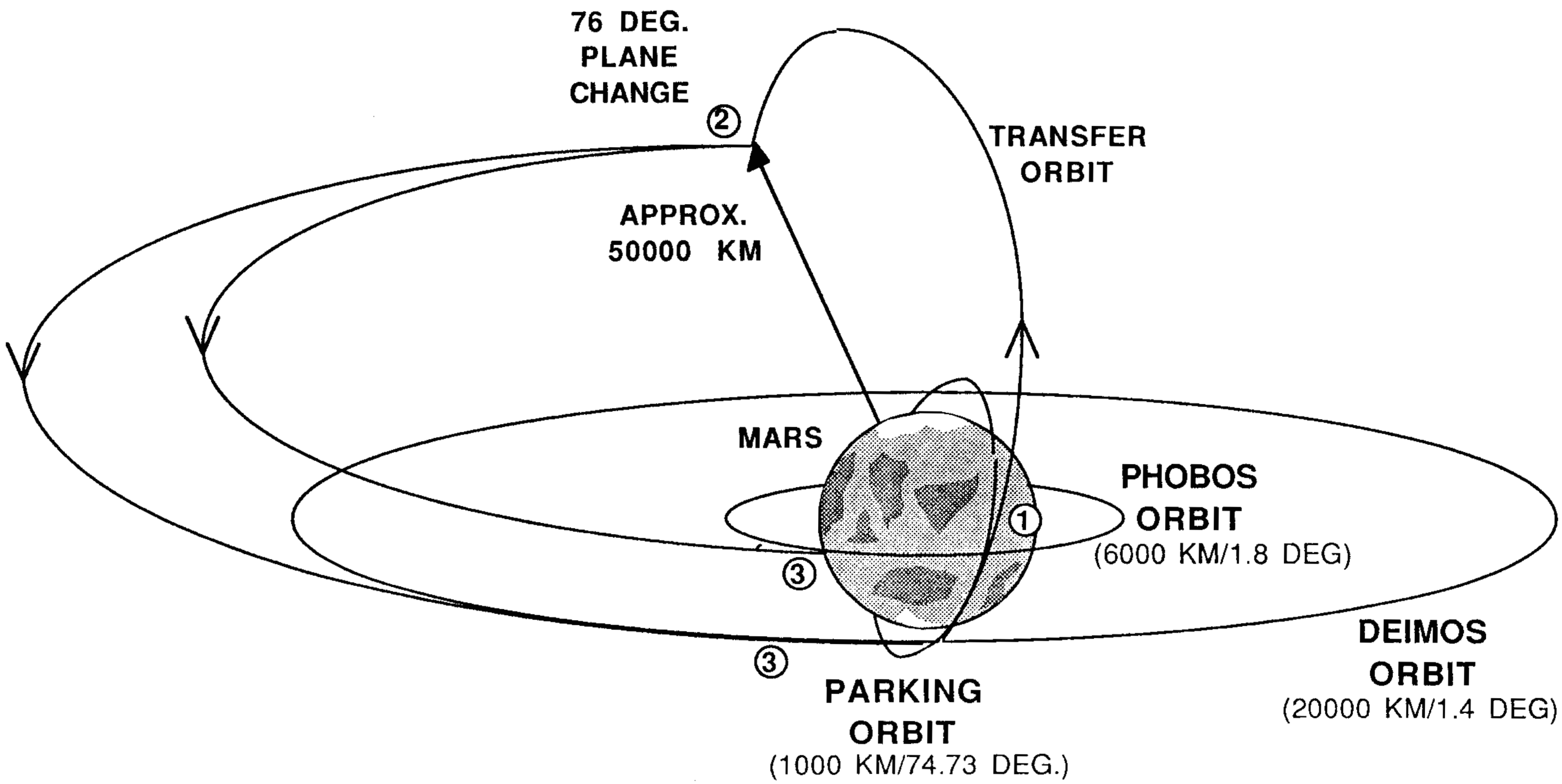
The two impulse transfers were direct Hohmann type transfers with the large plane change divided optimally between the two burns. This allowed short transfer times, but requires large velocity increments. The three impulse transfer is shown on the chart. The large plane change is made at a very high altitude where the orbital velocity is lowest. This is more efficient than the two impulse transfer in terms of velocity increments, but requires a much longer transfer time of two days.

It is possible to further reduce the delta velocity requirements of the three impulse transfer if aerobraking is used to decrease the orbit apoapsis prior to the final orbit insertion. After making the large plane change, the vehicle would transfer into the Mars atmosphere. Aerobraking would decrease the vehicle apoapsis altitude from 50,000 kilometers to the moon's orbit altitude. After leaving the atmosphere, the vehicle would use the third impulse to complete the insertion into the moon's orbit.

TRANSFERS TO PHOBOS OR DIEMOS

3 IMPULSE TRANSFER FROM PARKING ORBIT TO MARS MOONS

1. LEAVE PARKING ORBIT (RAISE APOAPSIS)
2. CHANGE ORBIT INCLINATION AND CHANGE PERIAPSIS ALTITUDE
3. INSERT INTO MOON'S ORBIT



DELTA VELOCITY REQUIREMENTS FOR TRANSFERS TO MARS MOONS

This chart summarizes the velocity increments required for the various mission and trajectory options that were considered. The delta velocities shown are for transfers from the parking orbit to the Mars moons and back. The maximum transfer apoapsis is given for each 3 impulse transfer.

DELTA VELOCITY REQUIREMENTS FOR TRANSFERS TO MARS MOONS

ORBITS PARKING ORBIT : 1000 KM /74.73 DEG.
PHOBOS : 5823 X 6141 KM/1.8 DEG.
DEIMOS : 20014 X 20146 KM/1.4 DEG

ASSUMPTIONS: 2 DAY TRANSFERS FOR 3 IMPULSE TRAJECTORIES
PERIAPSIS = 50 KM FOR AEROBRAKING TRAJECTORIES
WORST CASE PLANE CHANGES USED

MISSION OPTIONS

I VISIT PHOBOS ONLY

2 IMPULSE (2.4 HR. TRANSFER)

	DELTA V (M/S)
OUTBOUND	= 2844.2
RETURN	= 2844.2
TOTAL	= 5688.4

3 IMPULSE (2 DAY TRANSFER)

	DELTA V (M/S)	MAX. ALT. (KM)
OUTBOUND	= 2288.5	53536
RETURN	= 2288.5	53536
TOTAL	= 4567	

3 IMPULSE W/AEROBRAKING

	DELTA V (M/S)	MAX. ALT. (KM)
OUTBOUND	= 2066.7	54457
RETURN	= 1327.7	53065
TOTAL	= 3394.4	

II VISIT DEIMOS ONLY

2 IMPULSE (7 HR. TRANSFER)

	DELTA V (M/S)
OUTBOUND	= 2289
RETURN	= 2289
TOTAL	= 4578

3 IMPULSE (2 DAY TRANSFER)

	DELTA V (M/S)	MAX. ALT. (KM)
OUTBOUND	= 2081	46107
RETURN	= 2081	46107
TOTAL	= 4162	

3 IMPULSE W/AEROBRAKING

	DELTA V (M/S)	MAX. ALT. (KM)
OUTBOUND	= 2192	50446
RETURN	= 1154	45600
TOTAL	= 3346	

III VISIT BOTH

2 IMPULSE

	DELTA V (M/S)
OUTBOUND =	2844
PHOBOS TO DEIMOS =	755
RETURN =	2289
TOTAL =	5888

3 IMPULSE (2 DAY TRANSFER)

	DELTA V (M/S)	MAX. ALT. (KM)
OUTBOUND =	2289	53536
PHOBOS TO DEIMOS =	755	
RETURN =	2081	46107
TOTAL =	5125	

3 IMPULSE W/AEROBRAKING

	DELTA V (M/S)	MAX. ALT (KM)
OUTBOUND =	2067	54457
PHOBOS TO DEIMOS =	755	
RETURN =	1154	45600
TOTAL =	3976	

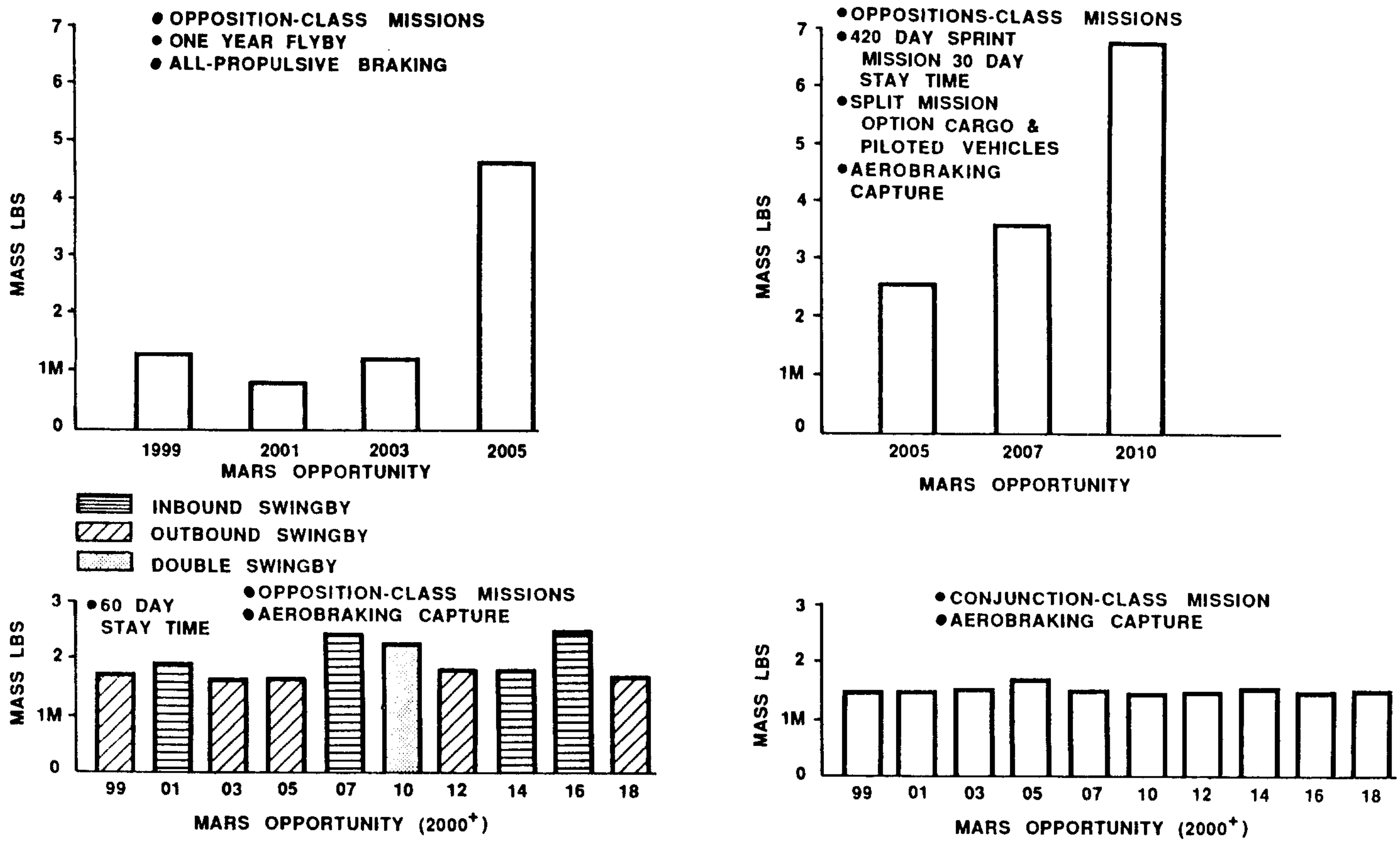
MARS EXPLORATION
MASS IN EARTH ORBIT REQUIREMENTS
CHEMICAL PROPULSION (LOX/LH_2)

The initial mass required in low Earth orbit (LEO) for four different Mars mission profile options is given on this figure. The initial mass required in LEO ranges from 850,000 lbs, lowest value for a one year Mars flyby, to 6,800,000 lbs, highest value for the 2010 split sprint mission opportunity. Shown in this figure is a great variation in initial mass required in LEO of the interplanetary space vehicle over a number of mission opportunities. This variation is due to the eccentricity of Mars orbit, which has a perihelion distance of 1.38 A.U. and an apohelion distance of 1.66 A.U. The wide variation in mass required in LEO may be reduced in the Mars flyby and split sprint mission profiles by adjusting the total mission time of more optimum value and by optimum deep space propulsive maneuvers. The wide variation in initial mass is reduced by aerocapture at Mars arrival and Earth return for the Venus swingby and conjunction class mission mode. The variation in initial mass for conjunction class mission mode over a number of mission opportunities is relatively small because there is more freedom to optimize the outbound trajectory transfer to Mars and the return trajectory transfer to Earth. Range of mass required in LEO for missions and opportunities considered are:

Mars Flyby	0.85 to 4.70M lbs
Split Sprint	2.65 to 6.80M lbs
Venus Swingby	1.60 to 2.50M lbs
Conjunction	1.50 to 1.70M lbs
Nuclear Electric Propulsion	~ 1.1M lbs

1-3955-7

MARS EXPLORATION
MASS IN EARTH ORBIT REQUIREMENTS
CHEMICAL PROPULSION (LOX/LH_2)

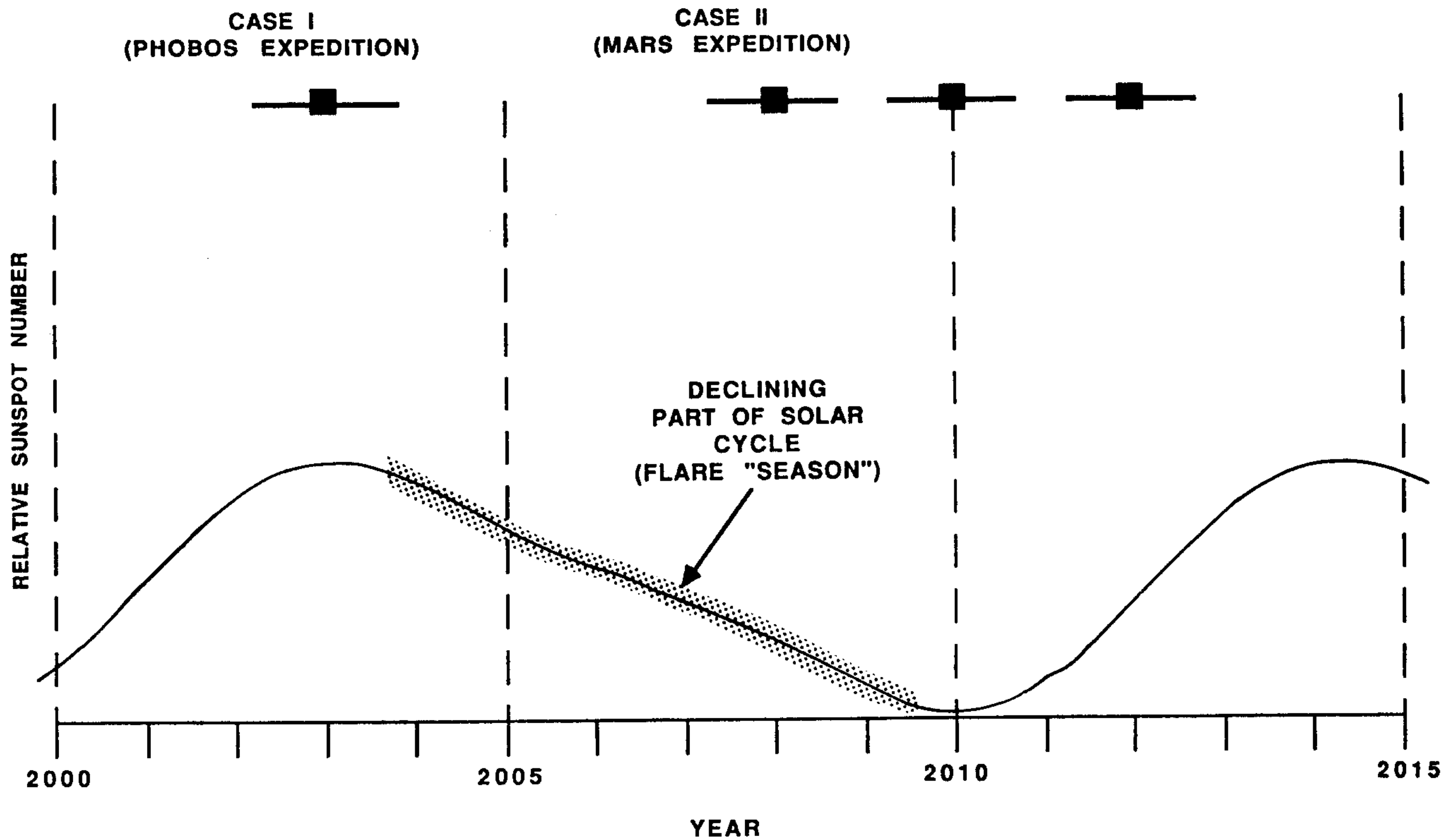


IDEALIZED SUNSPOT NUMBER VS. TIME FRAME FOR SELECTED MISSIONS

The chart on the facing page shows the relationship between mission times and projected fluctuations in solar activity. The vertical scale depicts the relative number of sunspots. The occurance of solar flares is more likely in the period when the number of sunspots is declining, for example the period from 2004 to 2009. The first manned Mars mission is nearly centered on this period, suggesting that of all the missions shown that it is the most likely to experience radiation from a major solar flare. The least likely mission to experience solar flare radiation is the last manned Mars expedition. The Phobos expedition and the second manned Mars expedition fall between these two extremes. If the series of three manned Mars missions were slipped by two years, the likelihood of experiencing a major solar flare during flight would be reduced significantly.

1-4180-8

IDEALIZED SUNSPOT NUMBER VS. TIME FRAME FOR SELECTED MISSIONS



THIS PAGE INTENTIONALLY LEFT BLANK

CRYOGENIC PROPELLANT STORAGE ANALYSIS

PROPELLANT STORAGE FACILITY THERMAL ANALYSIS

The facing page chart outlines the objectives, guidelines, and assumptions used to perform the thermal analysis of a Mars propellant storage facility. The facility thermal model consists of a cylinder with flat end caps, sized to accommodate propellant loading, having nominal surface coating properties for absorptivety, and emissivity of 0.4 and 0.9, respectively.

MANNED MARS MISSION

PROPELLENT STORAGE FACILITY THERMAL ANALYSIS

OBJECTIVES:

- Determine On-Orbit Propellant Boil-Off Rates For Storage Facility in Three Phases of Flight

  1. Low Earth Orbit (250 NMI, 28.5°)
  2. Heliocentric Mars Transfer Orbit
  3. Mars Orbit (550 NMI, 0.°)

GUIDELINES AND ASSUMPTIONS:

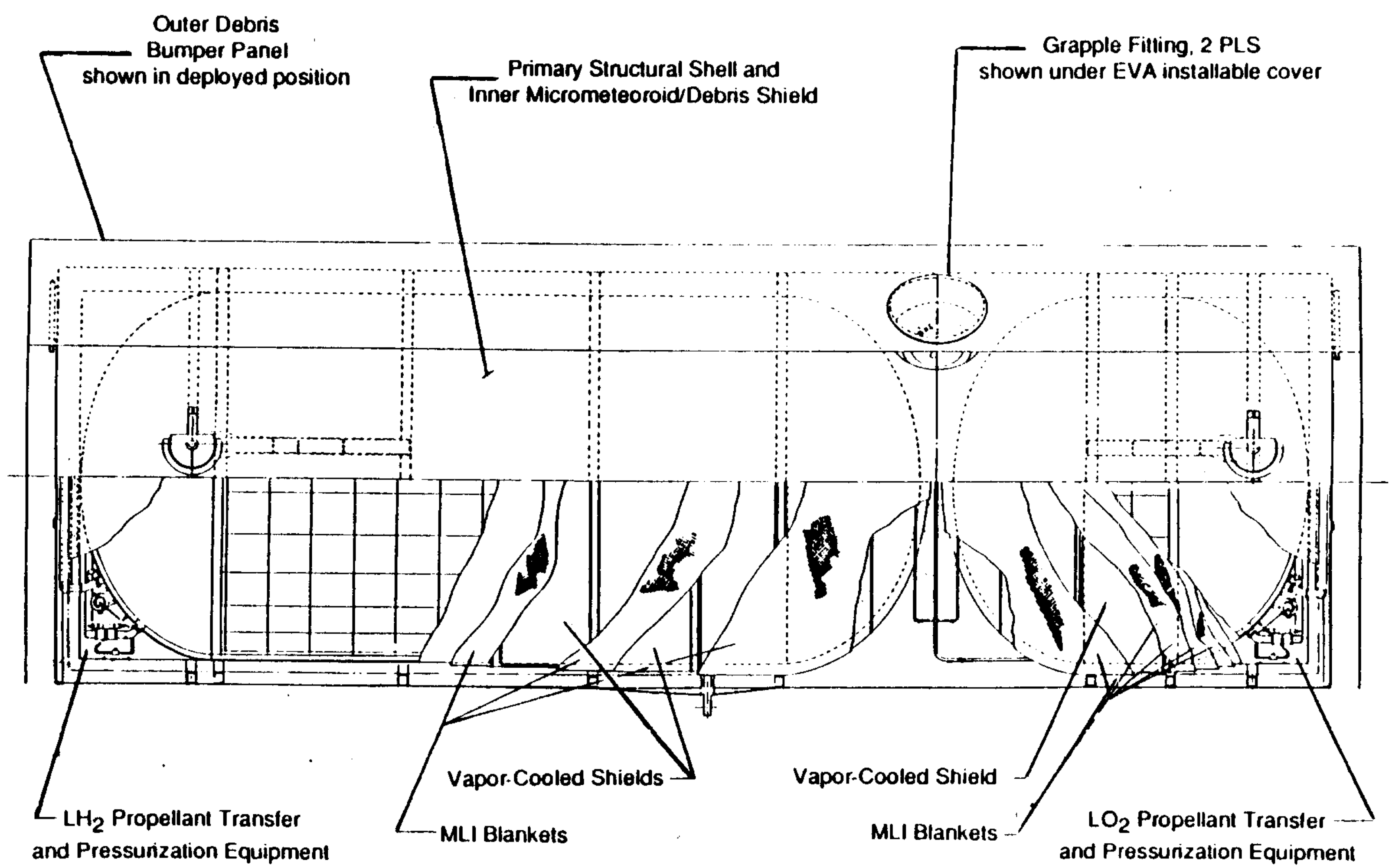
- 20.4 ft. Diameter, 35 ft. Long Cylinder
- Exterior Surface Properties: $\alpha = 0.4$
  $\varepsilon = 0.9$
- Storage Facility is Analyzed in Two Orientations in LEO and MARS Orbits

  1. Planet Oriented
  2. Sun Oriented

Cryogenic Storage Systems

Long term, orbital storage of cryogens in the Mars cargo and transit vehicles will require advanced propellant storage, management, and transfer hardware. The Mars cargo vehicle may resemble the conceptual design of a storage facility for liquid hydrogen and liquid oxygen. The facing page displays the characteristics of a cryogen storage tankset, combining multilayer insulation (MLI), vapor cooled shields (VCS), a thermodynamic vent system (TVS), and low conductance struts to form an efficient storage system. Additionally, a micro-gravity liquid acquisition device (LAD), fluid slosh baffles, and propellant disconnects enable the management and transfer of the cryogens on orbit. The insulation system provides a thermal barrier from the space environment and the micro-meteroid and debris shield adequately protects the tankset from penetrations. Analytically the passive thermal protection scheme has relatively low boiloff rates; however, there are still unproven flight systems (thick MLI, MLI/VCS supports, micro-g fluid transfer) in the represented conceptual design.

ALL-PASSIVE FACILITY

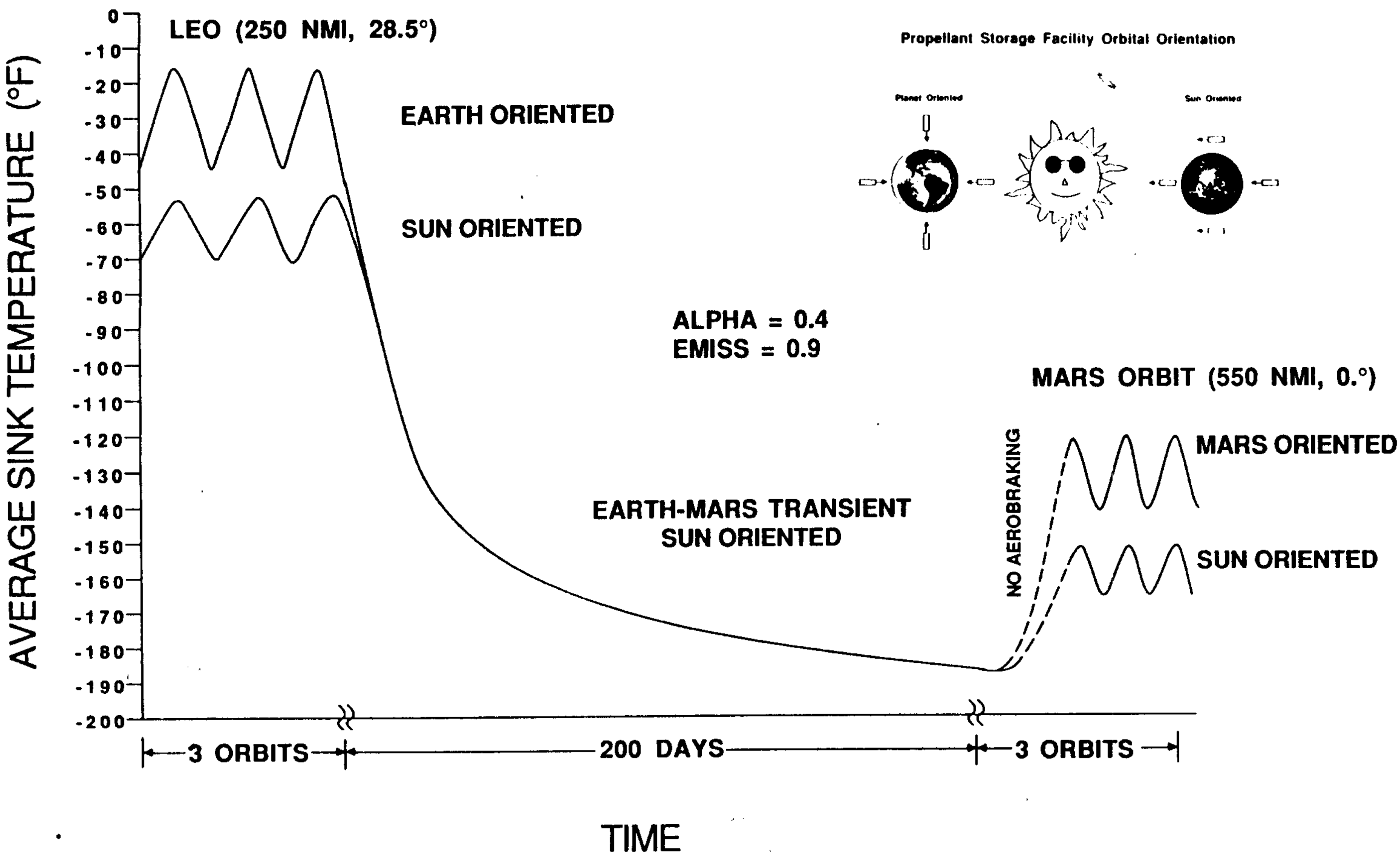


MARS PROPELLANT CARGO STORAGE ANALYSIS

The thermal analysis of the Mars propellant storage configuration involved determining the environmental heat rates on the external micrometeoroid shield using the computer code TRASYS. Circumferential and end cap heat rates were then spatially overaged to obtain a single, equivalent, environmental sink temperature.

The accompanying chart shows the average sink temperature for the three phases of flight and two orbital orientations. In low earth orbit, the orbital average sink temperature is approximately -30°F for earth oriented and -65°F for sun oriented. In Mars orbit, the planet and sun oriented sink temperatures are -130°F and -160°F, respectively. The dashed line indicates that aerobraking at Mars was not considered.

MARS PROPELLANT CARGO STORAGE THERMAL ANALYSIS



Propellant Boiloff Analysis

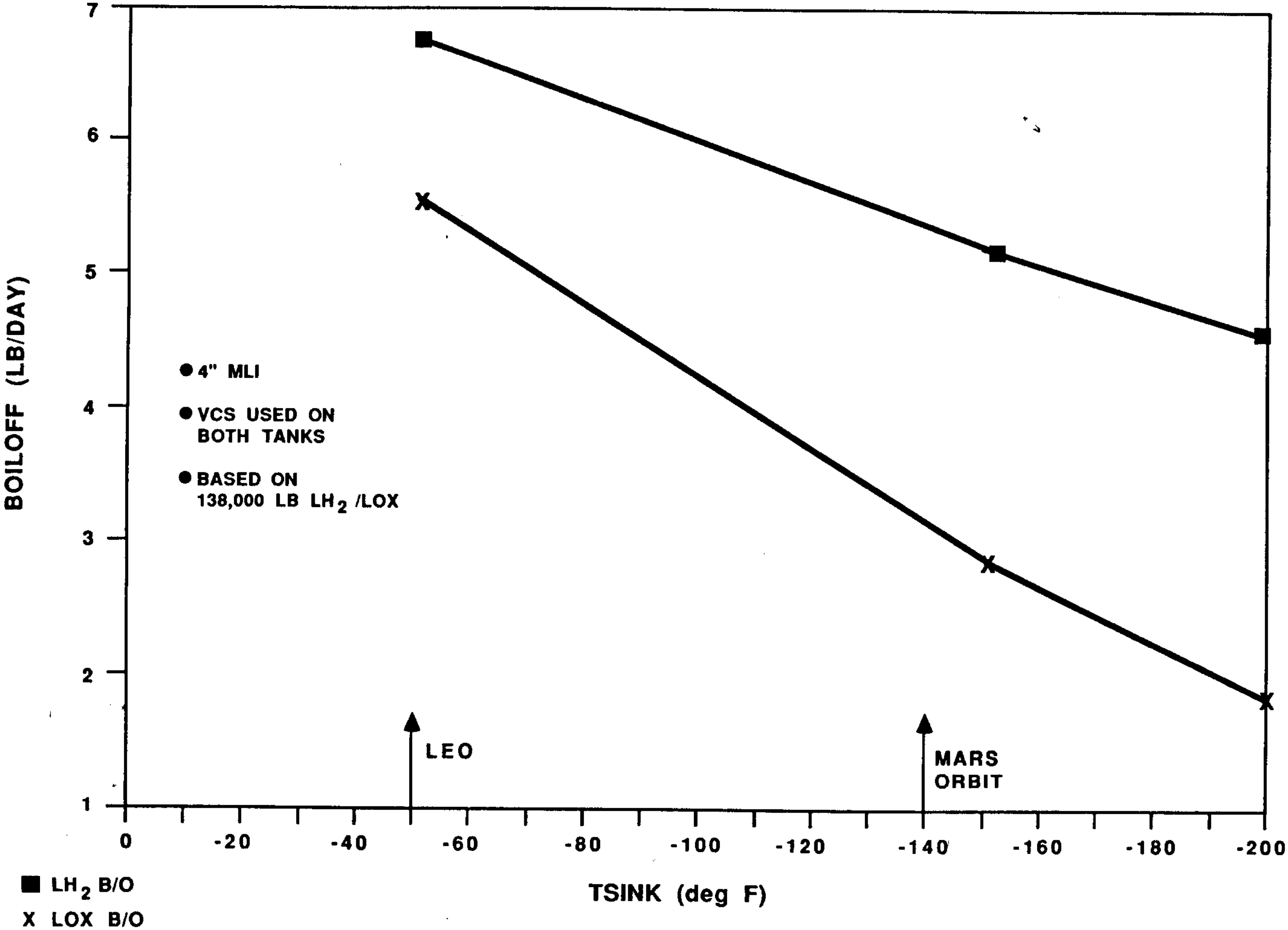
The analysis of the cargo vehicle thermal performance utilized a sink temperature calculation to determine tank heat leak (resulting in cryogen boil-off). Based on the conceptual design and 138,000 lb. propellant loading with 4 in. of MLI, the curves on the facing page were generated for the hydrogen and oxygen. The plots show the boil-off variation against the environmental sink temperature. The lower Earth and Mars orbits sink temperatures were calculated from environmental data obtained from TRASYS. The conclusions that can be drawn from the curves are as follow:

- With the system setup for efficient storage, the boil-off rates are low, even in LEO (hydrogen-1% per month, and oxygen-.15% per month).

- The system can certainly, based on the analysis, remain in Mars orbit for extended periods with minimum cryogen loss.

1-4250-8-18D

CARGO VEHICLE LH_2/LOX BOILOFF AS F(TSINK)



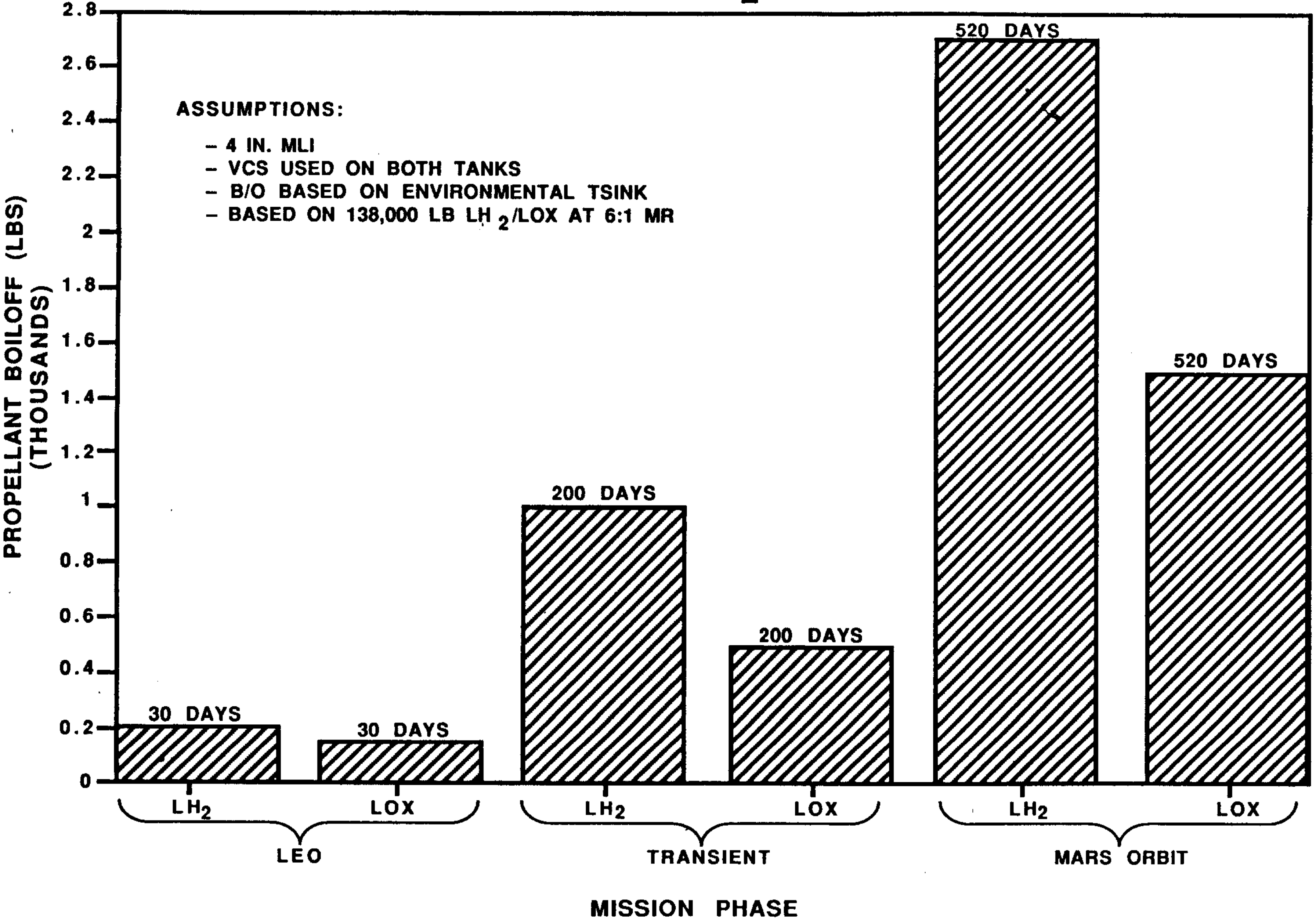
Cryogen Boil-Off Summary

Given the previously shown sink temperature and boil-off plots the following bar graph was constructed. For the estimated stay times, LEO-30 days, transit-200 days, and Mars orbit-520 days the boil-off for each mission phase is plotted. In the LEO vehicle buildup phase the storage system will vent approximately 370 lbs. of propellant (200 lb of hydrogen and 170 lb of oxygen). The Mars transit phase will lose 1500 lbs (1000 lb hydrogen and 500 lb oxygen). The most demanding phase, from a time standpoint, is the Mars orbit phase. Neglecting coating degradation, the boil-off loss is approximately 4200 lbs. (2700 lb hydrogen and 1500 lb oxygen). Therefore, the total mission losses are 6070 lbs. or 4% of the 138,000 lbs. (propellant transfer losses are neglected). The boil-off losses represent a 20% hydrogen loss and 2% oxygen loss based on a mixture ratio of 6 parts oxygen to 1 part hydrogen.

In conclusion, the mission appears to be feasible from a propellant management point of view. There are additional trades that could be performed to optimize the insulation system and VCS, trading boil-off versus insulation system weight. In general, for a long duration mission such as this the cargo vehicle performance is not as sensitive to the insulation system weight as a manned earth return stage might be.

1-4249-8-18D

CARGO VEHICLE LH_2/LOX BOILOFF



SUMMARY

This report describes the work completed at MSFC over the past few months in developing detailed and sensitivity data for the Mars Split Sprint Mission, which is one of the mission profile options being considered in the Space Exploration New Initiatives. The facing page provides a summary of the key findings from this study. A significant long-term commitment to developing several critical technologies and to establishing the substantial transportation capabilities and orbital facilities needed is essential to the success of a Manned Mars Mission. The initial Manned Mars Exploration will require the development of a number of technologies including aerocapture (which significantly reduces the amount of mass which must be delivered to low Earth orbit), efficient interplanetary propulsion, automation and robotics, and storage and transfer of cryogenic propellant in space. Also required is the successful demonstration of the functional abilities of human beings to cope with space flights of more than one year. For this successful demonstration, the Space Station would be critical. It must support the life science research, medical technique development, and crew rehabilitation so essential to a piloted Mars mission.

Considerable effort must also be given to the advanced planning of the transportation elements needed for this mission profile option, some of which are of unique design. The vehicles which must be developed include a piloted vehicle, a Mars lander, a cargo vehicle, and an Earth-recovery capsule. Development of the refuelable third stage and the large recoverable first/second stages is also necessary.

5.0 SUMMARY

- ANALYTICAL COMPUTER PROGRAMS HAVE BEEN MODIFIED AND DEVELOPED TO PERFORM MISSION DESIGN AND TRANSPORTATION REQUIREMENTS ANALYSIS FOR LUNAR AND PLANETARY MISSIONS.

- MARS ORBIT SELECTION CAN BE ACHIEVED TO MINIMIZE THE ENERGY REQUIRED FOR MARS CAPTURE AND ESCAPE MANEUVERS.

- ORBITS ABOUT MARS WITH ALTITUDES GREATER THAN 200 KM WILL HAVE LIFETIMES OF OVER 600 DAYS.

- FOR THE CARGO AND PILOTED VEHICLES, A 50 DAY LAUNCH WINDOW CAN BE ACHIEVED WITH ABOUT 12% INCREASE IN TOTAL WEIGHT REQUIRED IN LOW EARTH ORBIT ABOVE LAUNCHING ON THE MINIMUM ENERGY DATE.

- RENDEZVOUS COMPATIBLE MARS PARKING ORBITS HAVE BEEN DETERMINED.

- THE PILOTED VEHICLE HAS THREE MAIN PROPULSIVE STAGES FOR EARTH ORBIT ESCAPE, THE FIRST PROPULSIVE STAGE IS ALSO USED BY THE CARGO VEHICLE FOR EARTH ORBIT ESCAPE, THE THIRD STAGE IS REFUELED IN MARS ORBIT AND IS USED FOR MARS ORBIT ESCAPE.

- THE PILOTED AND CARGO VEHICLE PROPULSION SYSTEM CHARACTERISTICS AND PROPELLANT REQUIREMENTS HAVE BEEN DEVELOPED.

- VEHICLE AEROBRAKE DESIGN WAS INVESTIGATED; AEROBRAKE SHAPE AND SIZE, FLIGHT ATTITUDE, AERODYNAMIC FLOW, AND LONGITUDINAL STATIC STABILITY WERE ADDRESSED.

- ALL OF THE AEROBRAKING MANEUVERS WERE ANALYZED IN DETAIL WITH INTEGRATED TRAJECTORY SIMULATIONS.

- A DETAILED PERFORMANCE ANALYSIS OF THE MARS EXCURSION MODULE WAS PERFORMED, WHICH INCLUDED ANALYSIS OF THE DESCENT TO MARS SURFACE, AND ASCENT BACK TO MARS ORBIT.

- A DETAILED WEIGHT ANALYSIS AND AEROBRAKING TRAJECTORY SIMULATION WERE PERFORMED FOR THE EARTH RETURN CAPSULE.

THIS PAGE INTENTIONALLY LEFT BLANK

5.0 SUMMARY (CONTINUED)

- CRYOGENIC PROPELLANT STORAGE AND MANAGEMENT FOR THE CARGO VEHICLE WAS INVESTIGATED. LOSSES CAN BE MINIMIZED BY ADVANCED THERMAL PROTECTION SYSTEMS.
- SOLAR FLARE HAZARDS CAN BE REDUCED BY SHIFTING LAUNCH TIMES TO PERIODS OF LOW SOLAR FLARE OCCURANCES.
- BIPROPELLANT PROPULSION SYSTEMS HAVE BEEN DESIGNED FOR THE MARS DESCENT AND ASCENT VEHICLE, AND THE FLIGHT CONTROL SYSTEM FOR BOTH THE CARGO AND PILOTED VEHICLES.
- CRYOGENIC SYSTEMS WERE DESIGNED FOR THE MAIN PROPULSION SYSTEMS OF BOTH THE CARGO AND PILOTED VEHICLES, AND ALSO THE MAIN PROPULSION AND RCS OF THE EARTH RETURN CAPSULE.
- SUN ANGLES ENCOUNTERED DURING THE SPRINT MISSION DO NOT BLOCK COMMUNICATIONS BETWEEN THE PILOTED VEHICLE AND EARTH AT ANY TIME.
- THE WINDOW FOR EARTH/MARS LANDING SITE COMMUNICATIONS DURING THE 30 DAY STAY TIME IS ABOUT 9.5 HOURS PER DAY.
- COMMUNICATION OPPORTUNITIES BETWEEN THE MARS LANDING SITE AND THE PILOTED VEHICLE (IN MARS ORBIT) IS CONFINED TO 7 CONSECUTIVE ORBITAL REVOLUTIONS EACH 24.6 HOURS, WITH AN AVERAGE COMMUNICATION WINDOW OF 27.3 MINUTES PER ORBIT.
- ALTERNATE MISSION PROFILE OPTIONS HAVE BEEN IDENTIFIED FOR THE SPLIT SPRINT PROFILE.

6.0 COMPUTER PROGRAM DEVELOPMENT AND MODIFICATION

The following section describes the computer programs that were developed or modified during this study. They provide a significant capability and will be of great value in future analyses related to the Lunar and Mars New Space Exploration Initiatives.

6.0 Computer Program Development and Modification

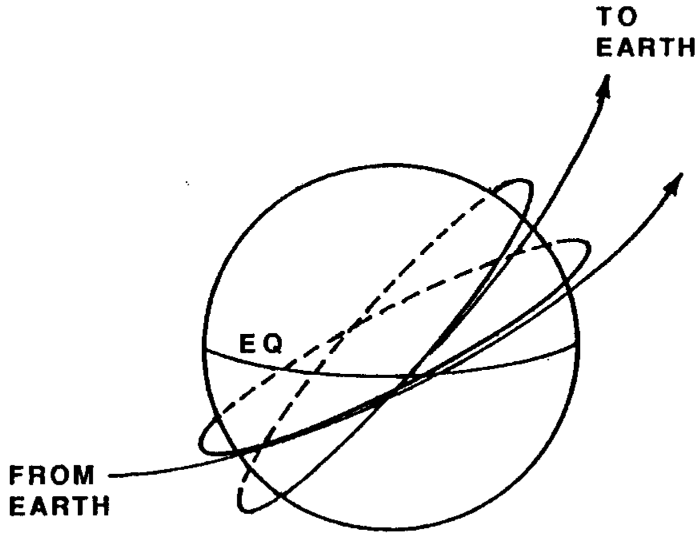
6.1 Mars Orbital Launch Window Program
6.2 Orbit Lifetime Program
6.3 Planetary Ephemeris
6.4 Integrated Mission Program
6.5 Analytical Satellite Ephemeris Program
6.6 Orbit Transfer Program
6.7 Helio
6.8 Mars

MARS ORBITAL LAUNCH WINDOW PROGRAM

A computer program was developed to evaluate proposed Mars parking orbits for matching the arrival and departure asymptotes of the heliocentric transfer trajectories. The right ascension of the ascending node, and the right ascension of the periapsis of the parking orbit, are computed from the heliocentric arrival asymptote. The nodal precession and apsidal rotation of the parking orbits are then computed as a function of altitude, inclination orbit eccentricity, and Mars stay time. The resulting parking orbit is then compared to the orbit required to achieve the desired departure asymptote and the nodal and apsidal misalignments displayed.

1-4082-8-16D

MARS ORBITAL LAUNCH WINDOW PROGRAM



PROGRAM INPUT	PROGRAM OPERATIONS	PROGRAM OUTPUT
ARRIVAL ASYMPTOTE ● RIGHT ASCENSION ● DECLINATION ● C_3 (ENERGY) DEPARTURE ASYMPTOTE ● RIGHT ASCENSION ● DECLINATION ● C_3 STOP-OVER TIME AT MARS	PARKING ORBIT VARIATION ● ALTITUDE ● ECCENTRICITY ● INCLINATION ● NODAL PRECESSION ● APSIDAL ROTATION PROPULSIVE MANEUVERS ● LOCATION ● ATTITUDE ● MAGNITUDE	OPTIMAL PARKING ORBIT CHARACTERISTICS ● ALTITUDE ● INCLINATION PROPULSIVE REQUIREMENTS ● MINIMUM PLANE CHANGE ● MINIMUM ENERGY

PROGRAM NAME:	Mars Orbit Lifetime Program
DESCRIPTION:	A modification of the MSFC LTIME Program incorporating the Mars gravitational and oblateness constants. Also incorporated is the Viking "High Density" atmospheric model.
CAPABILITIES:	Provides a time history of the orbital elements of a spacecraft subjected to the perturbative forces of atmospheric drag and Mars gravitational harmonics. The atmospheric model incorporated is a mean model and does not model the effects of solar heating as a function of predicted solar activity.
FEATURES:	The general method of averaging is used to define the mean orbital elements and their time rates of change. These elements change slowly until the satellite is in its final stages of decay. This allows the average rates of change of the elements to be integrated over relatively long time spans with negligible error.
APPLICATION:	The program was originally developed in the 1960's by LMSC under contract to MSFC. The program has been used to predict Earth satellite orbital decay for all U.S. space programs. The validity of the results when used for Mars satellites will depend primarily on the accuracy of the atmospheric model used.

PROGRAM NAME: PLANETS

DESCRIPTION: Planetary Ephemeris Generation

CAPABILITIES: User inputs month, day, year and GMT. Displayed on the terminal screen are the names of the eight planets and the sun along with the respective Right Ascension, Declination, and unit vector in the Earth's Equitorial/Vernal Equinox coordinate system.

FEATURES: At the end of each display, the program control options are:

1. Press <Ret> to increment time by 1 day.
2. Enter a new time increment.
3. Enter "R" to restart with a new date.
4. Enter "F" to toggle on file output.
5. Enter "X" to exit the program.

APPLICATION: General Purpose Mission Analysis

INTEGRATED MISSION PROGRAM
(IMP)

DEVELOPED AND PROGRAMMED
BY
V.A. DAURO, SR.
(205) 544-0546
FTS 824-0546

ABSTRACT

This is a simulation lauguage that can be used to model most present or future missions about Earth, Mars, or the Moon. Mission profiles are user controlled through selection from a large event/maneuver menu.

PROGRAM CODE AND USER'S MANUAL

This program is a dynamic growing code, and is always being improved and expanded. The latest version and its user's manual may be obtained by request through Program Development, MSFC.

The following six pages describe and give the capabilities of the Integrated Mission Program (IMP).

SLIDE #
MSFC/PD34/DAURO
6 JULY 1988

IMP CHARACTERISTICS

- A FEHLBERG 7/13 RUNGE-KUTTA INTEGRATOR WITH ERROR AND STEP SIZE CONTROL IS USED TO NUMERICALLY INTEGRATE THE EQUATIONS OF MOTION.

- INPUT/OUTPUT HAS BEEN SIMPLIFIED AND IS IN METRIC UNITS, WITH THE EXCEPTION OF THRUST AND WEIGHTS WHICH ARE IN ENGLISH UNITS.

- INPUT IS READ FROM THE VDT KEYBOARD AND THE USER'S EDITED INPUT FILE. REAL TIME KEYBOARD INPUT HAS BEEN MINIMIZED.

* THE MAIN OUTPUT IS STORED IN THE USER'S PRINT FILE. SOME DATA IS DISPLAYED ON THE VDT DURING EXECUTION.

* THE OUTPUT INCLUDES VEHICLE STATE, ORBITAL AND GUIDE PARAMETERS. IT ALSO SHOWS EVENT AND TOTAL VELOCITY CHANGES, WEIGHT CHANGES AND PROPELLANT USAGE.

* THE CODE IS IN DOUBLE PRECISION FORTRAN, AND CONSISTS OF ABOUT 13000 FORTRAN V STATEMENTS.

SLIDE #
MSFC/PD34/DAURO
6 JULY 1988

IMP CAPABILITIES

* ONE TO THREE SPACECRAFTS MAY BE SIMULATED. (MAIN, TARGET, AND AN OBSERVER)
* PROFILES MAY BE GENERATED ABOUT THE EARTH, MOON OR MARS.
* OBLATE OR SPHERICAL GRAVITY AT EARTH OR MARS IS AVAILABLE.
* SPHERICAL GRAVITY IS USED FOR THE SUN AND THE MOON.
* ATMOSPHERES AT EARTH OR MARS ARE INCLUDED WHEN DESIRED.
* THE PERTURBATIVE EFFECTS OF SOLAR PRESSURE, SOLAR GRAVITY AND MOON GRAVITY MAY BE SELECTED WHEN IN EARTH ORBIT.
* THE PERTURBATIVE EFFECTS OF SOLAR PRESSURE AND GRAVITY MAY BE INCLUDED WHEN IN MARS ORBIT.
* THE EFFECTS OF EARTH GRAVITY MAY BE USED WHEN IN MOON ORBIT.
* VELOCITY CHANGES MAY BE IMPULSIVE FOR PRELIMINARY PLANNING OR OF FINITE DURATION CONTROLLED BY INTERNAL ALGORITHMS.
* NON IMPULSIVE VELOCITY CHANGES MAY BE SEMI-OPTIMIZED, AND BURN DURATIONS UP TO 3600 SECONDS ARE FEASIBLE.
* OVER 40 EVENTS SUCH AS LIFTOFF, POST MECO, RENDEZVOUS, COAST, TRANSFER, AEROCAPTURE, DEBOOST, REENTRY AND MANY OTHER USEFUL MANUEVERS ARE PREPROGRAMMED.
* SOME EVENTS CONTAIN A SEQUENCE OF FIXED MANUEVERS, HOWEVER IN GENERAL THE USER CAN SELECT THE MANUEVERS AND CHAIN THEM TOGETHER IN THE ORDER DESIRED.

SLIDE #
MSFC/PD34/DAURO
6 JULY 1988

LATEST IMP ADDITIONS

* TRANSFERS TO EARTH-SUN OR EARTH-MOON LIBRATION POINTS AND/OR HALO ORBITS.
* TRANSFERS BETWEEN EARTH AND MOON.
* MANUEVERS ABOUT THE MOON.
* EPHEMERIDES FOR THE INNER AND OUTER PLANETS.
* EARTH TO MOON OR MOON TO EARTH COORDINATE SYSTEM TRANSFORMATION.
* PROVISIONS FOR A USER SUPPLIED ATMOSPHERE.

SLIDE #
MSFC/PD34/DAURO
6 JULY 1988

• * • EQUATION OF MOTION FEATURES • • •

THE FOLLOWING TABLE SHOWS THE EQUATION OF MOTION ACCELERATIONS PRESENTLY PROGRAMMED. THRUST IS AUTOMATICALLY INCLUDED WHEN USED. THE BASE GRAVITY FIELD IS ALWAYS PRESENT; THE OTHERS SELECTABLE.

ORBITING	-------GRAVITY			-----------		
	EARTH	SUN	MOON	MARS	ATMOS	SOLPRES
EARTH	BASE	YES	YES	NO	YES	YES
MARS	NO	YES	NO	BASE	YES	YES
MOON	YES	NO	BASE	NO	NO	YES

SLIDE #
MSFC/PD34/DAURO
6 JULY 1988

* * * LINE OF SIGHT FROM THE S/C * * *

IN ADDITION TO SIGHT TO THE OBSERVER AND/OR TARGET.

	----LINE OF SIGHT TO-----			
	EARTH	SUN	MOON	MARS
EARTH		YES	YES	NO
MARS	NO	YES	NO	
MOON	YES	YES		NO

IMP SAMPLE INPUT FOR LUNAR RETURN

```
COMMENT
      TRANSFER FROM MOON TO 200 NM CIRCULAR AT EARTH
      INITIAL PARKING ORBIT AT 1000 KM
      GRAVITY; OBLATE EARTH, MOON
END
DATE                       7    29 1992.     6.
PROP                             1 446.4     360.
RKMAX                              3600.     10.
SUNMOON                   10
INSERT                    22     0 120.                15790.96  5000.     360.0
 -9.78257     152.18352    1000.         1338.04257    0.0             111.96885
COASTP                              3.58
TRANSFER                   2     4  400000.
COAST                               24.
SWITCH                    00
COAST                               60.
ORBIT                    220     4  370.4     340000.
COAST                               15.2
TARGET                     0     2  223.5
 -22.70434    77.51425     407.44        0.            0.              28.5
RKMAX                               360.     1.
PLANE                     12     4
COAST                      4
CIRCLE                           4
PROP                             2
STOP
```

PROGRAM NAME:	ASEP (Analytical Satellite Ephemeris Program)
DESCRIPTION:	Simulates the motion of a Satellite in Earth or Mars orbit. The solution to the satellite motion is completely analytic and includes secular, short period, and long period terms. The gravity model includes J_2 through J_4 spherical harmonic terms.
CAPABILITIES:	Computes satellite ephemeris, ground tracks, day/night cycles, times of terminator crossings, Sun/Earth/Mars positional geometry, and other geometrical quantities associated with the satellite orbit.
FEATURES:	Interactive and menu-driven. Custom designed for the DEC VT200 series terminals and the VAX 11/780 computer.
APPLICATION:	General purpose tool for both real-time and pre-mission analysis and mission planning.

MSFC ANALYTICAL TOOLS

PROGRAM:	ORBIT TRANSFER PROGRAM
DESCRIPTION:	CALCULATES IMPULSIVE DELTA VELOCITY REQUIREMENTS FOR ORBIT TRANSFERS AT EARTH AND MARS, AND PERFORMANCE CHARACTERISTICS, INCLUDING PAYLOAD CAPABILITIES AND PROPELLANT REQUIREMENTS.
PURPOSE:	PROVIDES QUICK ASSESSMENTS OF SPACECRAFT PERFORMANCE REQUIREMENTS FOR ORBIT TRANSFERS.
FEATURES:	DELTA VELOCITY REQUIREMENTS CAN BE CALCULATED FOR HOHMANN (2-IMPULSE) AND BI-ELLIPTIC (3-IMPULSE) ORBIT TRANSFERS. PLANE CHANGES ARE OPTIMALLY DIVIDED BETWEEN EACH BURN. THE THREE IMPULSE TRANSFERS HAVE AN AEROBRAKING OPTION.
HISTORY:	WRITTEN FOR USE IN PRELIMINARY DESIGN ORBIT TRANSFER AND PERFORMANCE ANALYSIS.

MSFC ANALYTICAL TOOLS

PROGRAM NAME:	HELIO
DESCRIPTION:	Earth and Mars planetary ephemeris generator combined with an Earth to Mars transfer trajectory propogation.
CAPABILITIES:	Computes vehicle distances from Earth and Mars, communication delay times (light travel times), and sun angles.
FEATURES:	User inputs date and time of Earth departure, semi-major axis, eccentricity, and inclination of the transfer orbit. Initial true anomaly in the transfer orbit, time increment for output, and number of days to run.
APPLICATIONS:	Mars Mission Analysis

PROGRAM NAME:	MARS
DESCRIPTION:	Mars and Earth ephemeris generation and geometry calculator.
CAPABILITIES:	User inputs data and time, and on the terminal screen is displayed 29 quantities describing various geometrical aspects of the Earth and Mars orbits.
FEATURES:	Computes the angles in the Sun-Earth-Mars triangle, Right Ascension and Declination of the Sun and Mars in the Earth Equitorial/Vernal Equinox system. Right Ascension and Declination of the Sun and Earth in the Mars Equitorial/Vernal Equinox system. The program runs on the VAX 11/780 computer with a VT200 (or compatible) terminal.
APPLICATION:	General Purpose Mars Mission Analysis

ACRONYMS AND ABBREVIATIONS

AFE	Aeroassist Flight Experiment
AgZn	Silver-Zinc
ANT	Antenna
AREF	Aerodynamic Reference Area
ASEP	Analytical Satellite Ephemeris Program
ASMG	Acceleration Sensed
ave	Average
C_3	Hyperbolic Orbit Energy
CA	Aerodynamic Axis Force Coefficient
CD	Drag Coefficient
CDN	Cargo Descending Node
CG	Center of Gravity
CIRC	Circularization Burn, OMS2
CL	Lift Coefficient
CLL, Cl, or CMR	Aerodynamic Roll Moment Coefficient
CLN, Cn, or EMY	Aerodynamic Yaw Moment Coefficient
CM, Cm, or CMP	Aerodynamic Pitch Moment Coefficient
CN	Change Noticed
CO_2	Carbon Dioxide
CV	Cargo Vehicle
dB	Decibel
DECL	Direct Energy Conversion Laboratory
delta-V or DV	Delta Velocity, Velocity Change
DIA	Diameter
diref	Reference Dia
DLA	Escape Orbit Declination
DSN	Deep Space Network
DYNP	Dynamic Pressure
ECLSS	Environmental Control and Life Support System
EFF	Efficiency
ELV	Expendable Launch Vehicle
ERC	Earth Return Capsule
ERV	Earth Return Vehicle
EVA	Extravehicular Activity
FPR	Flight Performance Reserve
ft/s	Feet Per Second

g	1 Gravity
GDLAT	Geodetic Latitude
GH	Gaseous Hydrogen
GH2	Gaseous Hydrogen Molecule
GLOW	Gross Loft Off Weight
GND	Ground
GPS	Global Positioning System
HLLV	Heavy Lift Launch Vehicle
Δh	Delta h
hp	Horsepower
in	Inches
Isp	Specific Impulse
J2	Legendre Polynominal Coefficient Used in Gravity Potential Function
Kbps	Kilobits Per Second
km	Kilometers
KSC	Kennedy Space Center
kW	Kilowatts
kW-hr	Kilowatt-Hour
LAD	Liquid Acquisition Device
LaRC	Langely Research Center
lbF	Pounds of Force
lbs	Pounds
L/D	Lift/Drag
LEO	Low Earth Orbit
LH	Liquid Hydrogen
LH2	Liquid Hydrogen Molecule
liOH	Lithium Hydroxide
LO2	Liquid Oxygen
LREF	Reference Moment Arm (Aerodynamic)
Mbps	Megabits Per Second
M	Number of Mars Rotations
MEM	Mars Excursion Module
MLI	Mulit-Layer Insulation
MM	Mission Module
MMH	Monomethyl Hydrazine
MR	Mixture Ratio
MRP	Moment Reference Point
MV	Mars Vehicles
N	Number of Items

N_2	Molecular Nitrogen
NM or NMI	Nautical Miles
NTO/MMH	Nitrogen Heteroxide/Monomethylhydrazine
O_2	Molecular Oxygen
OTV	Orbital Transfer Vehicle
OMV	Orbital Maneuvering Vehicle
PAN	Piloted Ascending Node
Pc	Combustion Chamber Pressure
PD or PDD	Program Development Directorate
PDO	Preliminary Design Office
PLD	Payload
PLS	Propellant Load System
Psi	Pounds Per Square Inch
Psia	Pounds Per Square Inch Absolute
psf	Pounds Per Square Foot
Q	Spacecraft Pitch Rate
R	Spacecraft Yaw Rate
RA	Right Ascension
RCS	Reaction Control System
Ref	Reference
RF	Radio Frequency
RKMAX	Maximum Recharge
R&R	Rest & Relaxation
SAI or SAIC	Science Applications International Corporation
SDV	Shuttle Derived Vehicle
sec	Seconds
SPE	Solar Particle Event
ss	Space Station
STS	Space Transportation System
TRASYS	Thermal Radiation Analysis System
TVS	Thermodynamic Vent System
USRA	Universities Space Research Association
Vin	Velocity Entering the Atmosphere
VCS	Vapor Cooled Shield
VHP	Hyperbolic Orbit Excess Velocity
Vout	Velocity Leaving the Atmosphere
VR or VRL	Relative Velocity
W	Watt
WAGSPOT	Weight and Geometry Sizing Program of Tanks
We	Earth's Angular Rotation Rate

W-hr	Watt-Hour
WI	Inert Weight
Wp	Propellant Weight
wt	Weight

REFERENCES

1. Margaret Varner, PD34 - Operations Analysis Branch, MSFC
2. Manned Mars Mission - Working Group Papers, Volume II, NASA M002 1986, "Solar Particle Event Predictions for Manned Mars Missions"
3. Jack Mulqueen, PD32 - Mission Analysis Branch, MSFC
4. Biconic Earth Entry Module Configuration: Catalog # 2-113544 MSFC Microfiche
5. Shuttle Systems Weight and Performance Status Report, 31 Mar 1987, Walt Johanson, PD24 - Systems Integration Branch, MSFC
6. Steve Hall, PD24 - Systems Integration Branch, MSFC
7. Ron White, PD14 - Electrical Systems Branch, MSFC
8. Robert Champion, PD13 - Propulsion Systems Branch, MSFC
9. John Fikes and Larry Brandon, PD12 - Navigation and Control Systems Branch, MSFC
10. Norman Brown, PD22 - Structures and Thermal Analysis Branch, MSFC
11. Larry Hill, PD14 - Electrical Systems Branch, MSFC
12. Space Station Advanced Development Program Review - Thermal, October/November 1986, Clay Horan, PD22 - Structures and Thermal Analysis Branch, MSFC
13. Component Manual for Potential MSP ECLS Hardware, Steve Hall, PD24 - Systems Integration Branch, MSFC
14. Dictionary of Space Technology, Dave Lanier, PD34 - Operations Analysis Branch, MSFC

15. Orbiter Detail Weight Statement, 2 Feb 1984, Bobby Brothers, PD24 - Systems Integration Branch, MSFC

16. Final Report: Definition of Experimental Test for a Manned Excursion Module [Rockwell Study], Steve Hall, PD24 - Systems Integration Branch, MSFC

17. Space Systems Technology, Steve Hall, PD24 - Systems Integration Branch, MSFC

18. Archie Young, PD32 - Mission Analysis Branch, MSFC

19. Bart Graham and Alan Philips, PD22 - Structures and Thermal Analysis Branch, MSFC

20. Sally Ride, "Leadership and America's Future in Space", a Report to the NASA Administrator, 1987

21. Manned Mars Missions, Working Group Papers, Volumes I and II, NASA M002 June, 1986

22. Piloted Sprint Missions to Mars, Science Application International Corporation, 1987

23. Scenario Requirements Document, Document No. Z-MAS-SRD-001, June 2, 1988

24. Exploration Requirements Document, Document No. Z-MO-ERD-001, May 13, 1988

25. Bob Giudici, PD14 - Electrical Systems Branch, MSFC

NOTES

NOTES