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Final Report

A Study of System Requirements for Phobos/ Deimos Missions

Volume IV
Phase III Results
Combined Missions
to Mars & Its Satellites

June 1972

Contract NAS1-10873

A STUDY OF SYSTEMS REQUIREMENTS FOR PHOBOS/DEIMOS MISSIONS

FINAL REPORT

Volume IV

Phase III Results - Combined Missions to Mars and Its Satellites

Approved

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FOREWORD

This is Volume IV of the Final Report on A Study of Systems Requirements for Phobos/Deimos Missions, conducted by the Martin Marietta Corporation.

This study was performed for the Langley Research Center, NASA, under Contract NAS1-10873, and was conducted during the period 4 June 1971 to 4 June 1972. Mr. Edwin F. Harrison of Langley Research Center, NASA, was the Technical Representative of the Contracting Officer. The study was jointly sponsored by the Advanced Concepts and Mission Division of the Office of Aeronautics and Space Technology (OAST) and the Planetary Programs Division of the Office of Space Sciences (OSS) in NASA Headquarters.

This Final Report, which summarizes the results and conclusions of the three-phase study, consists of four volumes as follows:

- Volume I - Summary
- Volume II - Phase I Results - Satellite Rendezvous and Landing Missions
- Volume III - Phase II Results - Satellite Sample Return Missions and Satellite Mobility Concepts
- Volume IV - Phase III Results - Combined Missions to Mars and Its Satellites

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ABBREVIATIONS AND SYMBOLS

a	orbit semi-major axis
ACS	attitude control system
ARU	attitude reference system
Ax, Ay, Az	body acceleration
Az	azimuth angle
bps	bits per second
CC&S	control computer and sequencer
cg	center of gravity
db	decibel
DLA	declination of launch asymptote
DSN	Deep Space Net
DSS	Deep Space System
EL	elevation angle
ETC	engineering test capsule
$F1_c$, $F2_c$, $F3_c$, FN_c	lander engine thrust command
FOV	field of view
g	acceleration due to gravity, Earth
G&C	guidance and control
GCSC	guidance, control and sequencing computer
grms	gravity (rms)
HZ	hertz
i	orbit inclination
IRU	inertial reference unit
JPL	Jet Propulsion Laboratory
kbps	kilobits per second
km	kilometers
L/E	launch/encounter

LOS	line-of-sight
LPCA	lander pyrotechnic control assembly
LRC	Langley Research Center
mbps	megabits per second
MCC	midcourse correction
MLI	multilayer insulation
MMC	Martin Marietta Corporation
MOI	Mars orbit insertion
NASA	National Aeronautics and Space Administration
NW	net load factor times weight
OSR	optical solar reflector
p, q, r	body attitude rates
PTC	proof test capsule
PTO	proof test orbiter
R	range
$\dot{R}$	range rate
RCS	reaction control system
RF	radio frequency
RSS	root-sum-of-squares
RTG	radioisotope thermoelectric generator
R99	99 percentile closest approach radius
S/C	spacecraft
TA	orbit true anomaly
TEI	trans-Earth injection
T/M	thrust-to-mass
TMI	trans-Mars injection
TWTA	traveling wave tube amplifier
UHF	ultra-high frequency
UV	ultraviolet
VHE	hyperbolic excess velocity

VM	velocity meter
VO	Viking Orbiter
VRU	velocity reference unit
W	weight
α	solar absorptivity
ΔV	delta velocity
ΔV_{STAT}	navigation uncertainty delta velocity
ε	orbit eccentricity
θ	pitch attitude angle
π	3.1416
ρ	density
σ	standard deviation
ϕ	roll attitude angle
ψ	yaw attitude angle
μ_{MARS}	Mars central gravity potential constant
Ω	Mars longitude of ascending node
ω	Mars argument of periapsis
$\sim$	approximately

I. Objectives and Study Results

I. OBJECTIVES AND STUDY RESULTS

A. INTRODUCTION

This report, in four volumes, contains the results of a nine-month, three-phase study conducted for the Langley Research Center to evaluate the systems requirements to accomplish Phobos/Deimos missions in the 1977-1983 time period.

The study was initiated in June 1971 under NASA contract NAS1-10873. The study milestones are summarized in Table I-1. The study was based on a succession of three phases that allowed a logical progression from a straight-forward rendezvous and landing satellite mission conducted during Phase I, to a more meaningful sample return mission performed during Phase II, and finally culminating in a highly cost effective combined Mars landing and Phobos/Deimos mission studied during Phase III. Each succeeding phase effort built upon the results of the previous phase to a large degree. For example, the original concept of missions to the Martian satellites was developed by Messrs. Pritchard and Harrison of the NASA Langley Research Center. They demonstrated the technical feasibility of such space missions in a preliminary mission design that became the basis for the system study performed during Phase I. Using this basic knowledge, then, we generated basic data on mission analysis and spacecraft system requirements during Phase I which we applied to alternate mission concepts during Phases II and III in search for the most cost effective Phobos/Deimos exploration approach.

Table I-1 Study Milestones

Preliminary Mission Design by NASA/LRC-MAAB	January 1971
Systems Definition Study Contract to MMC	June 4, 1971
Phase I - Landing Roving Mission	June 4, 1971 thru September 9, 1971
First Presentation	September 9 and 10, 1971
Phase II - Sample Return Mission	September 13, 1971 thru December 9, 1971
Second Presentation	December 9 and 10, 1971
Phase III - Combined Mars and Phobos/Deimos Mission	December 13, 1971 thru April 6, 1972
Third Presentation	April 6 and 7, 1972
Final Report	May 5, 1972

Throughout the study phases, numerous trade studies and analyses were performed to progress through the many mission and system options available. These studies and analyses are documented in the appropriate study phases in which they were performed.

Each of the study phases are treated in separate volumes of this report. A brief summary of the study ground rules and guidelines applicable to that particular study effort are presented at the beginning of each of the study phases.

Overall program schedules and cost estimates were derived for each of the study phases. Detailed equipment lists were prepared and formed the basis for the cost estimates that were generated during the study.

B. PHASE III STUDY OBJECTIVES AND GUIDELINES

The overall objective of this phase of the study effort was directed at defining combined missions to Mars and Phobos/Deimos that could provide relatively large science returns at a low total cost.

In order to achieve this objective, it was necessary to perform several major categories of study effort. First, a matrix of all feasible combined Mars and Phobos/Deimos mission concepts was compiled. From this compilation, the most promising approach was selected from which a nominal mission profile was developed. Once the baseline mission profile was identified, a systems analysis study was performed to trade-off performance, configuration and cost characteristics of candidate mission concepts. Configuration and subsystem optimization analysis were then conducted from which a baseline and an alternate program were selected.

Program schedules and cost estimates were then prepared for the recommended and alternate program.

At the beginning of the Phase III study, a series of ground rules were mutually agreed upon by the MMC study team and by the Langley Reserach Center. These ground rules are summarized in Table I-2. Also, as preliminary results of the study began to develop, a series of study generated ground rules evolved. These ground rules are shown in Table I-3.

Two changes were introduced just after the conclusion of the Phase II study effort and just prior to the initiation of the Phase III effort:

1) A change was made in the launch vehicle nomeclature. The NASA versions of the Titan IIID series vehicles became Titan IIIE. This change was made to differentiate between the military vehicle and the NASA vehicles.

2) Allocated Viking spacecraft weights were updated as a result of formal approval received from NASA/LRC's Viking Project Office in early December 1971.

Table I-2 LRC Study Directed Ground Rules

- Launch vehicles considered: Titan IIIE/Centaur, Titan IIIE/Centaur GT, Titan IIIE7/Centaur, Shuttle/Centaur
- Launch opportunities shall be from 1977 to 1988
- Consider both Type I and II trajectories
- Consideration to be given to direct as well as out-of-orbit entry
- Consider: observation orbits, rendezvous orbits, landing, and sample return
- Consider use of space storable propellants
- Use revised (Viking Spacecraft Mass Properties Status Report, Issue 24) allocated weights
- Apply proven hardware and technology
- Minimize program costs

Table I-3 MMC Derived Study Ground Rules

- Titan IIIE/Centaur launch vehicle
- 1979 opportunity
- Type II trans-Mars trajectory
- Orbital operations to consist of: Mars capture orbit (97 hour period), phasing orbit (30 to 60 hour period), observation orbit (15.1 hour period)
- Stretched Viking Orbiter (26% propellant increase)
- Out-of-orbit Mars lander
- Phobos landing

C. STUDY RESULTS

Studies conducted during Phase III followed essentially the same study methodology used during Phase I and II. The study effort was concentrated primarily in three general categories; mission/science oriented analysis, system analyses and trade studies, and conceptual design studies.

The mission-oriented studies were conducted by developing a mission mode evaluation study in order to define the spectrum of potential mission approaches to be considered. This evaluation procedure provided a comprehensive screening and analysis of a large number of alternatives which in turn allowed us to select the baseline and leading alternative concept for in-depth conceptual design study. This mission mode evaluation was supported as necessary by preliminary system analysis data and science requirements as inputs.

System analysis of the candidate baseline mission system concepts developed during the mission mode evaluation were conducted. Major system and subsystem level trade studies were conducted for each candidate concept. These trades established relative cost, performance and development risk estimates for all mission concepts evaluated during the mission mode evaluation and allowed the selection of a baseline and leading alternate concept for further definition.

Conceptual design studies were then conducted which allowed us to select a recommended combined mission baseline concept and leading alternative.

This section summarizes and describes the baseline mission/ system as developed during the Phase III studies. A discussion of the mission and system options and trade studies that were considered are described in detail in Chapters II and III of this report.

1. Baseline Mission/System Description

A mission analysis was performed to establish the basic characteristics and tradeoffs of the mission profile and performance parameters necessary for definition of systems requirements, development of the baseline vehicle design concept and for the evaluation of its technical feasibility. Some of the more significant mission decisions and trade studies that were performed are delinated in Table I-4. The general goal in these studies was to achieve good science mission characteristics at both Mars and the satellites while still holding to minimum cost, minimum modification to proven hardware concept.

The mission mode evaluation analyzed combined mission concepts which considered all of the mission elements shown in Table I-5. In addition, four different spacecraft configurations were considered: two growth orbiters; a staged orbiter; and a space storable propellant orbiter. In total, performance data for 324 possible combined missions was determined and is presented in Chapter II. Cost comparisons were made for 19 of the most attractive combinations. This data is shown in Chapter III.

The selected baseline launch opportunity is a 1979 launch from Earth with arrival at the vicinity of Mars approximately 11 months later. The launch vehicle is the Titan IIIE/Centaur. The Earth-to-Mars portion of the mission is identical to the 1979 launch of Phase I. Type II trajectories were again selected.

Table I-4 Mission Analysis and Design Trade Studies

Selection of Capture Orbit

Mars Landing Profile

Mars Landing Latitude Accessibility

Selection of Observation Orbit

Spacecraft Configuration

Table I-5 Basic Mission Profile Options

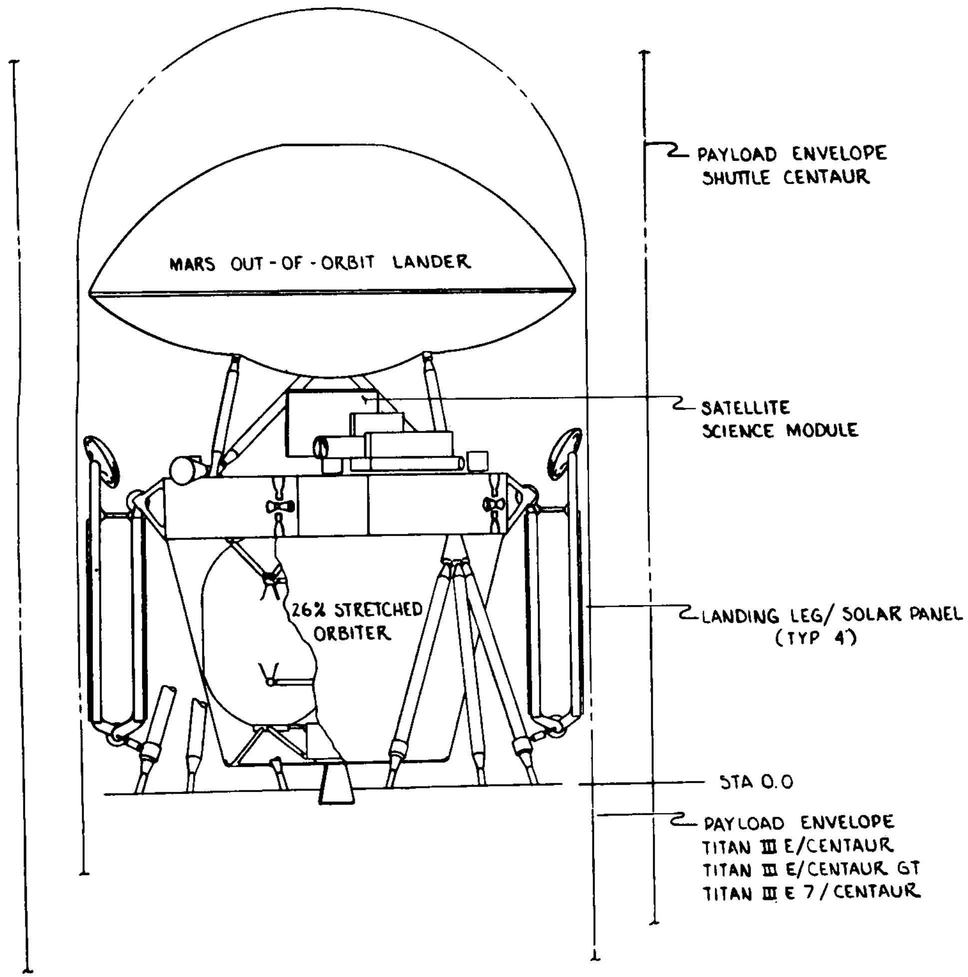
Mars Landings	**Mission Opportunities**
Direct Entry	1979
Out of 1-Day Orbit	1981
Out of 4-Day Orbit	1983/84
Phobos/Deimos Missions	**Launch Systems**
Observation Orbit	Titan IIIE/Centaur
Rendezvous and Station-keeping	Titan IIIE/Centaur GT
	Titan IIIE 7/Centaur
Landing	Shuttle/Centaur

The combined missions spacecraft as shown in Figure I-1 consists of two major elements; a minimally modified Viking Mars Lander and a modified landed Viking Orbiter with a 26% propulsion system stretch, basically the same orbiter configuration as our alternate Phase I orbiter. The total injected weight is approximately 4154 kg (9159 pounds). This compares with the Viking 75 injected weight of 3664 kg (8080 pounds). The science payload in this configuration is 67.5 kg in the orbiter and 62.6 kg in the Mars lander.

To accomplish the baseline combined mission, the Viking Orbiter is modified to incorporate landing legs, rendezvous radar, solar panels integrated with the landing legs, stretched propulsion system, (26% increase), addition of thermal control flip covers, addition of a terminal descent propulsion system, and the incorporation of the Phobos/Deimos payload.

Modifications to the existing Viking Mars Lander are minimal in nature and consist primarily of: an increase in heat shield ablator thickness; increase in propellant loading; and the incorporation of a geoscience payload.

The spacecraft will arrive at Mars about September 1980 at which time the Orbiter propulsion system will insert the spacecraft into a 97 hour capture orbit about Mars. The Mars orbit insertion (MOI) is performed at periapsis at an altitude of 1500 km (same as Viking '75) and the applied ΔV (870 mps) leaves the spacecraft in the 97 hour orbit with an apoapsis of 95,000 km and a periapsis in Deimos orbit plane. The Mars lander deorbits from this orbit approximately 5.5 hours prior to its entry interface the orbiter with its Phobos/Deimos science module continues in the 97 hour orbit. At the next apoapsis, a maneuver is per-



SCIENCE

MARS LANDER

IMAGERY
SEISMOMETER
METEOROLOGY
INTEGRATED GEOLOGY
ALPHA BACKSCATTER
X-RAY FLUORESCENCE
X-RAY DIFFRACTION

SATELLITE SCIENCE MODULE

X-RAY FLUORESCENCE
X-RAY DIFFRACTION

ORBITER

IMAGERY
ADVANCED IR SPECTROMETER
GAMMA RAY SPECTROMETER

SPACECRAFT CG IS 79.4" FWD OF STA O
VIKING '75 CG IS 84.2" FWD OF STA O

Figure I-1 Baseline Combined Missions Spacecraft formed to change the orbit plane to coincide with Deimos orbit plane and also to lift periapsis to 2660 km. A phasing orbit is then established by a retromaneuver at periapsis and its purpose is to allow the relative geometry between the spacecraft and Deimos to be adjusted so that after the observation orbit is established, the spacecraft will be at apoapsis when Deimos is near that same position. An observation orbit is established by another retro maneuver at periapsis reducing the period to 15.1 hours. From the observation orbit, the spacecraft is navigated to the vicinity of the target satellite (Phobos in the case of our baseline mission). Using the orbiter TV cameras, tracking data, and satellite ephemeris information, this maneuver leaves the spacecraft within 22 km of the deired separation distance between Phobos and the spacecraft of 50 km and with the same orbital period. At this point, the rendezvous radar acquires Phobos and the rendezvous and landing sequence is initiated. A small closing velocity (50 mps) is applied by the terminal descent thrusters. This velocity is then removed prior to impact to allow a touchdown of Phobos at a velocity of 1.5 mps $\pm$ 1.0 mps. Total delta velocity required from Earth injection to Phobos landing is 2235 mps.

Following touchdown on the satellites' surface as shown in Figure I-2, Earth communications are established and the science mission is begun. Total landed mission duration is approximately 90 days.

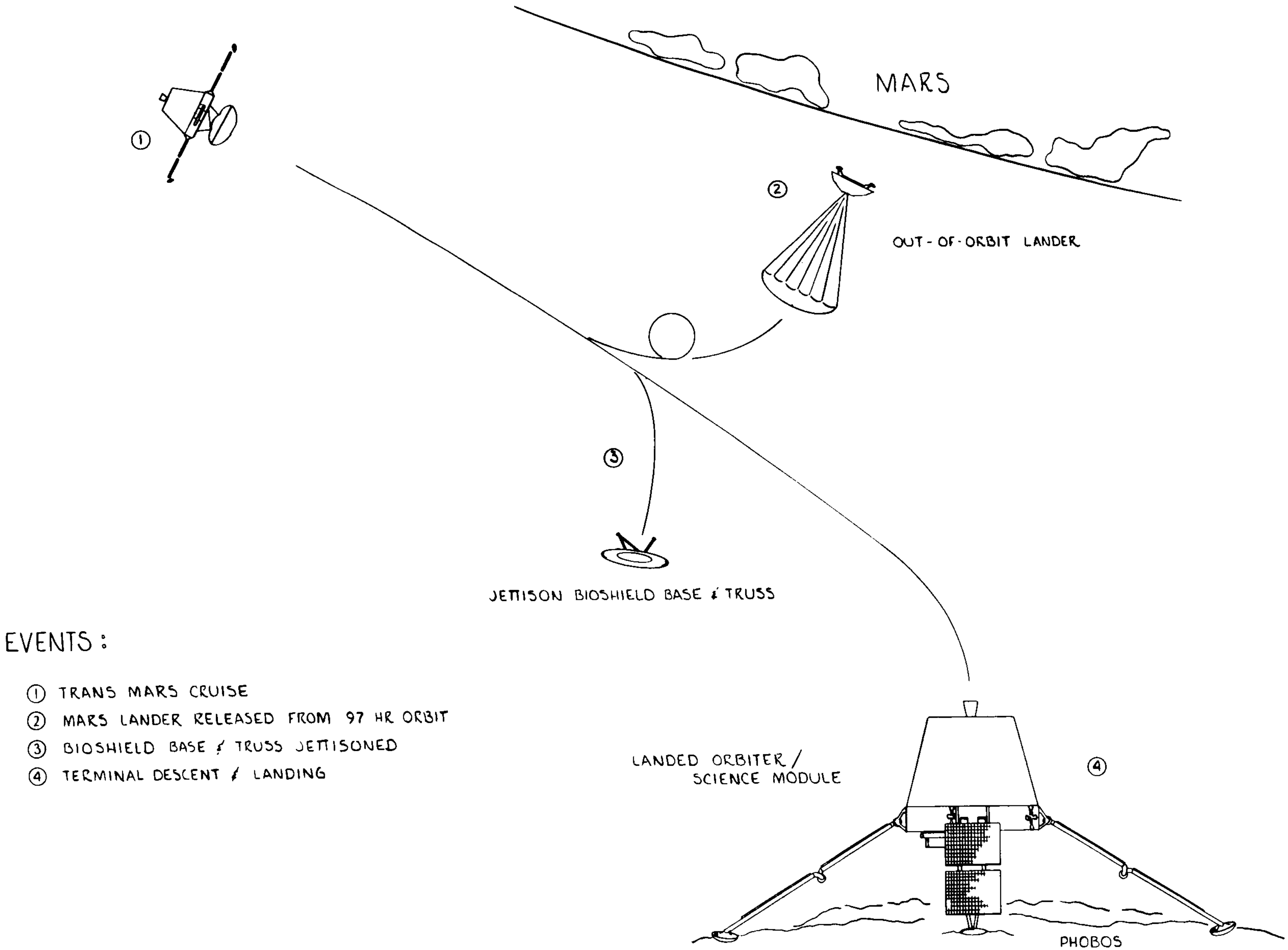


Figure I-2 Combined Missions Baseline Configuration Sequence

2. Alternate Mission/System Description

An alternate concept was also studied that met the increased performance requirements of the 1981 launch opportunity and used a direct entry lander weighing 1239 kgs combined with a landed modified Viking Orbiter weighing 1499 kgs fully loaded. The modified orbiter in this case required a 7% growth in the propulsion system to accomplish the mission. The launch vehicle was a Titan IIIE/Centaur. Total injected payload weight was 3973 kgs (8760 pounds). A system weight summary for the alternate configuration is given in Table I-6.

Table I-6 1981 Alternate Spacecraft Weight Summary, kgs

• Lander Capsule Loaded Weight	1238.6
• Orbiter Dry Weight	1027.1
• Orbiter Expendable Propellant Weight	1498.8
• Spacecraft Loaded Weight	3764.5 (8300 lb)
• Project Reserve	43.0
• LVMP	104.3
• V-S/C Adapter	61.2
•Injected Payload Weight	3973.0 (8760 lb)

II. Mission Analysis and Design

II. MISSION ANALYSIS AND DESIGN

A. PERFORMANCE ANALYSIS TRADE STUDIES

In selecting a mission which combines a Mars landing and a mission to investigate the two moons of Mars, there are many considerations and options which must be studied. Table II-1 is a list of the major events of a combined mission and some of the choices available for these events. In most cases, a particular choice has both advantages and disadvantages. For example, the direct entry mode for the Mars lander requires almost no modifications to the orbiter (which is used to carry out the Phobos/Deimos mission) but the more severe entry conditions require extensive modifications to the lander. Likewise, if the Mars lander de-orbits from a capture orbit, modifications to the orbiter are necessary to increase the propellant load since the additional lander weight has to be brought into the capture orbit along with the Phobos/Deimos spacecraft. This, however, requires less modifications to the lander. The capture orbit period involves similar compromises. The higher the period the less propulsion system changes to the orbiter and the more the required changes to the lander because of the higher entry velocities. The type of the Phobos/Deimos mission selected (observation orbit, rendezvous, or landing) is a trade between complexity of mission, modification to the orbiter, and scientific data return.

The spacecraft configuration used as a basis for the mission determines the relation between mission cost and potential scientific return. The launch opportunity chosen determines the relative ease of doing the mission (less energy required for earlier opportunities). The launch system selected for the baseline is a function of the

Table II-1 Basic Mission Profile Options

- Mars Landing
  - Direct Entry
  - Out-of-Orbit
- Capture Orbit
  - 1 to 11 Day Period
  - Periapsis Altitude
- Phobos/Deimos Mission
  - Observation Orbit
  - Rendezvous
  - Landing
  - Sample Return

- Spacecraft Configuration
  - Planetary Explorer/Pioneer
  - Modified Viking Orbiter
- Launch Opportunities (1977-88)
  - 1979
  - 1981
  - 1983/84
- Launch System
  - Titan IIIE/Centaur
  - Titan IIIE/Centaur GT
  - Titan IIIE 7/Centaur
  - Shuttle/Centaur

launch opportunity, the Phobos/Deimos mission, the type of Mars landing and period of capture orbit.

In order to develop the baseline mission, several trade studies were required, many involving the previously mentioned options. Table II-2 indicates the major trade studies and these will be discussed in more detail. There are additional trade studies involved which were primarily in the area of Mars entry dynamics, Mars mapping capability and the effects of periapsis altitude on ΔV requirements.

Table II-3 shows the pertinent design information as a function of orbital period for a range of capture orbits with period between 1 and 11 days. The Viking lander is designed for an entry from a 1 Martian day orbit. It is necessary, therefore, to investigate the effects of entry from the higher period orbits. The cause for concern is the increase in entry velocity and the resultant g load and various heat and heat rates. This results in an increase in lander weight as indicated. The increased weights for the 2, 3, and 4 day orbits are primarily due to additional ablator material. De-orbits from higher period orbits require additional structural changes since the "max g-loads" exceed the qualification level. These structural modifications would be relatively expensive as compared to the cost of additional ablator material. The savings in orbiter propellant as the orbital period increases indicates that the optimum choice is the highest orbital period not requiring the more expensive lander structural changes. This is the 97 hour class of orbit. Slight variations of several hours would have no significant effect. For example, the period could be changed to 98.4 hours (4 Martian days) in order to allow communication support to a Mars lander every 4th day and this would still not require a change to the structure other than the ablative material thickness. Before this 97 hour orbit was completely accepted, the

Table II-2 Mission Analysis and Design Trade Studies

Selection of Capture Orbit

Mars Lander Targeting Strategy

Plane Change Strategy

Observation Orbit Selection

Orbiter Propulsion Alternatives

Small Spin Stabilized Spacecraft Alternatives

Table II-3 Capture Orbit Selection Rationale

Orbit Period (Hr)	Apoapsis Altitude (km)	Entry* Velocity (MPS)	Maximum G Load (Gs)	Maximum Dynamic Pressure (lb/ft^2)	Plane Change ΔV (MPS)	Propellant** (kg)	Lander Weight (kg)
24.6	32,600	4628	13.0	163	150	2020	1115.6
49.2	56,000	4740	13.3	170	95	1890	1117.9
97.0	92,000	4816	13.5	178	65	1760	1124.2
123.0	112,000	4845	14.0	182	55	1720	1129.2
270.6	190,000	4895	14.6	185	40	1690	1165.5

* Assumes 30° Inclination

** For the Baseline Payload in 1979

degradation of the mapping capability was checked. Figure II-1 indicates the decrease in the mapping resolution element size of the 97 hour orbit from the 24-hour orbit. This parametric analysis assumes that the periapsis is over the equator. The indicated resolution element size is at the sub-spacecraft point and is a function of altitude and TV camera characteristics only. The resolution element size at the maximum latitude (a function on inclination) is degraded from 142 meters to 160 meters. At the 1500 km periapsis there is no degradation since the altitudes are the same.

The effects of the various entry modes on the Mars landing latitude capability are shown in Table II-4. This is a parametric study and assumes that the argument of periapsis is located over the equator and that these particular inclinations are possible (actually only limited inclinations are possible for each launch period). The latitude limits indicated for each inclination are the result of using the maximum yaw steering capability of the lander during its deorbit maneuver. This effect on latitude is greatest for zero inclination orbits and has basically no effect on latitude changes from a polar orbit. The basic differences in the maximum latitude capabilities is a result of the angle between periapsis and the landing point. As the entry velocities increase this angle (PER angle) increases which increases the effectiveness of the yaw steering portion of the deorbit maneuver. The direct entry landing latitude extremes are significantly greater than the two out-of-orbit cases because of the significantly larger PER angle (16° vs 10°). Also indicated in this table is the lander weight changes and orbiter propellant weight changes for each of these three general entry modes.

The Mars landing latitude capability using lander yaw steering vs the approach declination (DLA) is shown in Figure II-2. The

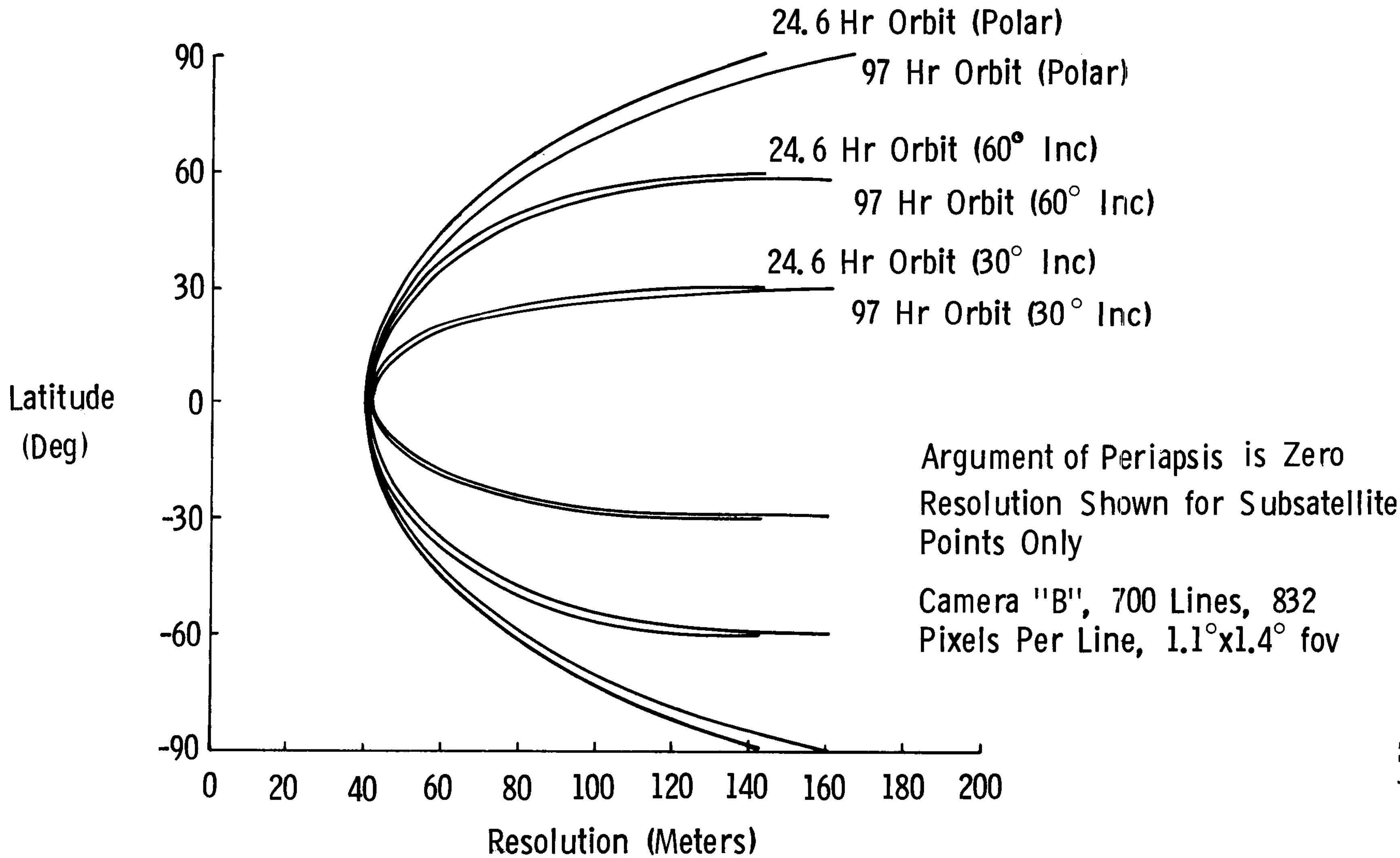


Figure II-1 Mars Orbital Mapping Capability

Table II-4 Mars Lander Targeting Strategy

Lander Mode	Inclination (Deg)	Entry Velocity (MPS)	Maximum* Landing Latitudes (Deg)	Lander Weight Change (kg)	Orbiter Propellant Change** (kg)
Direct Entry	0° 30° 90°	5913 6096 6218	-12° to +12° -18° to + 2° -16°	111	-576
97 Hour Orbit	0° 30° 90°	4694 4816 4938	-8.5° to +8.5° -13° to + 1° -12°	0 (Reference)	0 (Reference)
24 Hour Orbit	0° 30° 90°	4506 4628 4750	- 8° to + 8° -12° to + 1° -10°	-8.6	+263

* With argument of periapsis of zero and maximum Lander yaw steering

** Δ propellant to accomplish a Phobos landing mission in 1979

NOTE: Lander separation is 5.5 hours prior to entry in all cases.

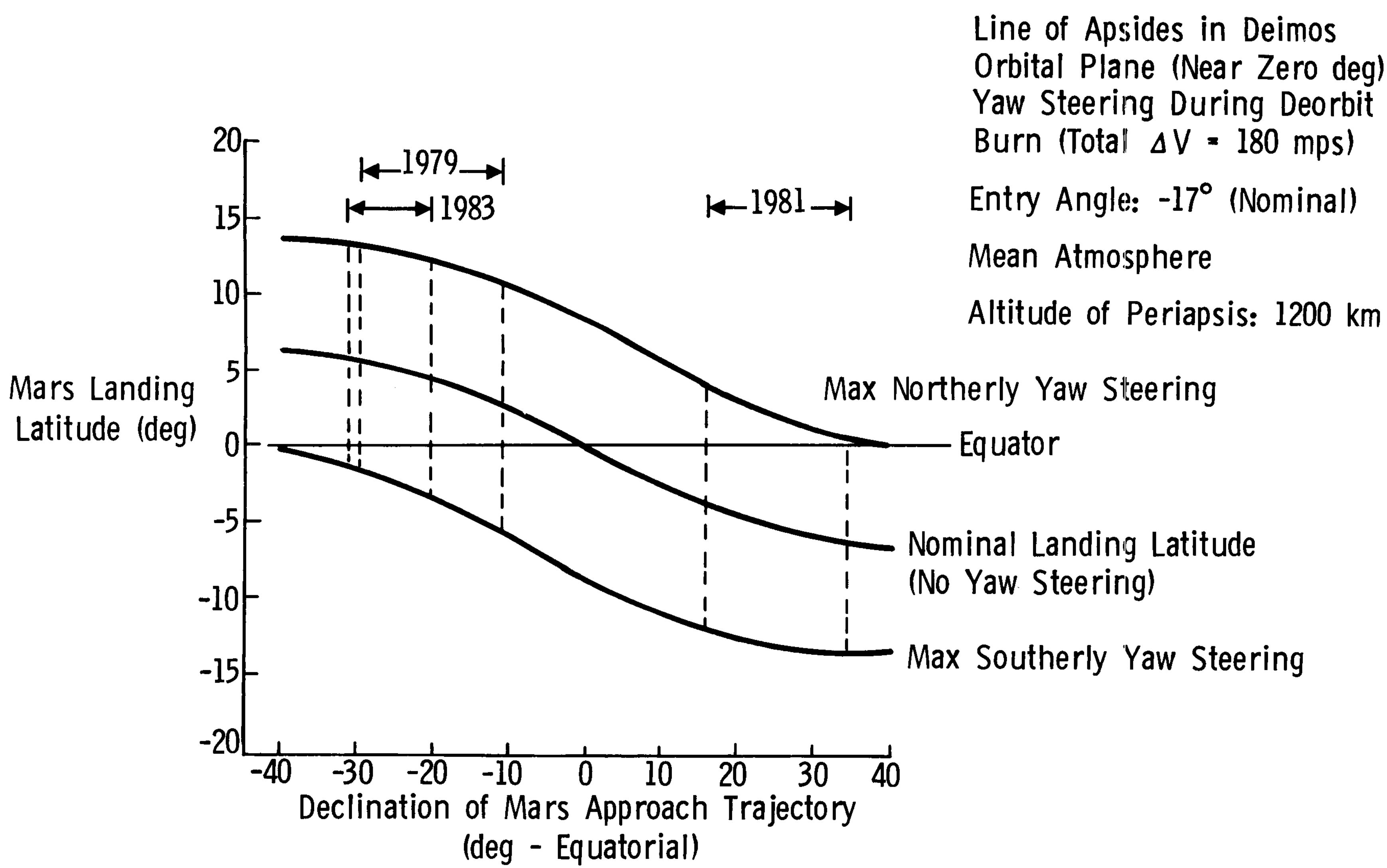


Figure II-2 Mars Latitude Accessibility Using Lander Yaw Steering

DLA range for a launch opportunity dictates the minimum inclinations that are available during that opportunity. Again, this assumes the periapsis is located over the equator. If the location of periapsis is allowed to more away from the equator (non-zero argument of periapsis) additional landing latitude capability is obtained as indicated in Figure II-3. If the argument of periapsis is not in Deimos' orbit plane ($\backsim$ equatorial) additional ΔV is required to match Deimos' orbit plane for the Phobos/Deimos portion of the mission. This function is shown on the right hand portion of the figure. The additional landing latitude capability is shown on the left hand figure as a function of the argument of periapsis and also indicated is the effect of reducing the Phobos/ Deimos payload and therefore, increasing the propellant to accommodate the additional ΔV requirements. As the curves indicate, the latitude capability is increased by over 50% by reducing the payload to 50 kg and over double by reducing the payload to zero. For missions which do not require the full capacity of the launch vehicle, significant additional latitude could be achieved by increasing the orbiter propellant. This would not reduce the payload but would increase the required modifications to the orbiter propulsion system.

A comparison of the two possible observation orbits (30.6 hours and 15.15 hours), using the OBSERV program (Appendix A). yielded the encounter conditions as shown in Figure II-4. The 15.15 hour observation has repeated observations of Deimos every other orbit and Phobos moves relative to the spacecraft approximately 7.5° per spacecraft orbit. The relative position of Phobos and the spacecraft has a thirty day cycle and during that cycle, there are two orbit crossings encounters. Each crossing yields two passages at less than 1000 km closest approach. The minimum approach is apprximately 200 km due to the slight difference in

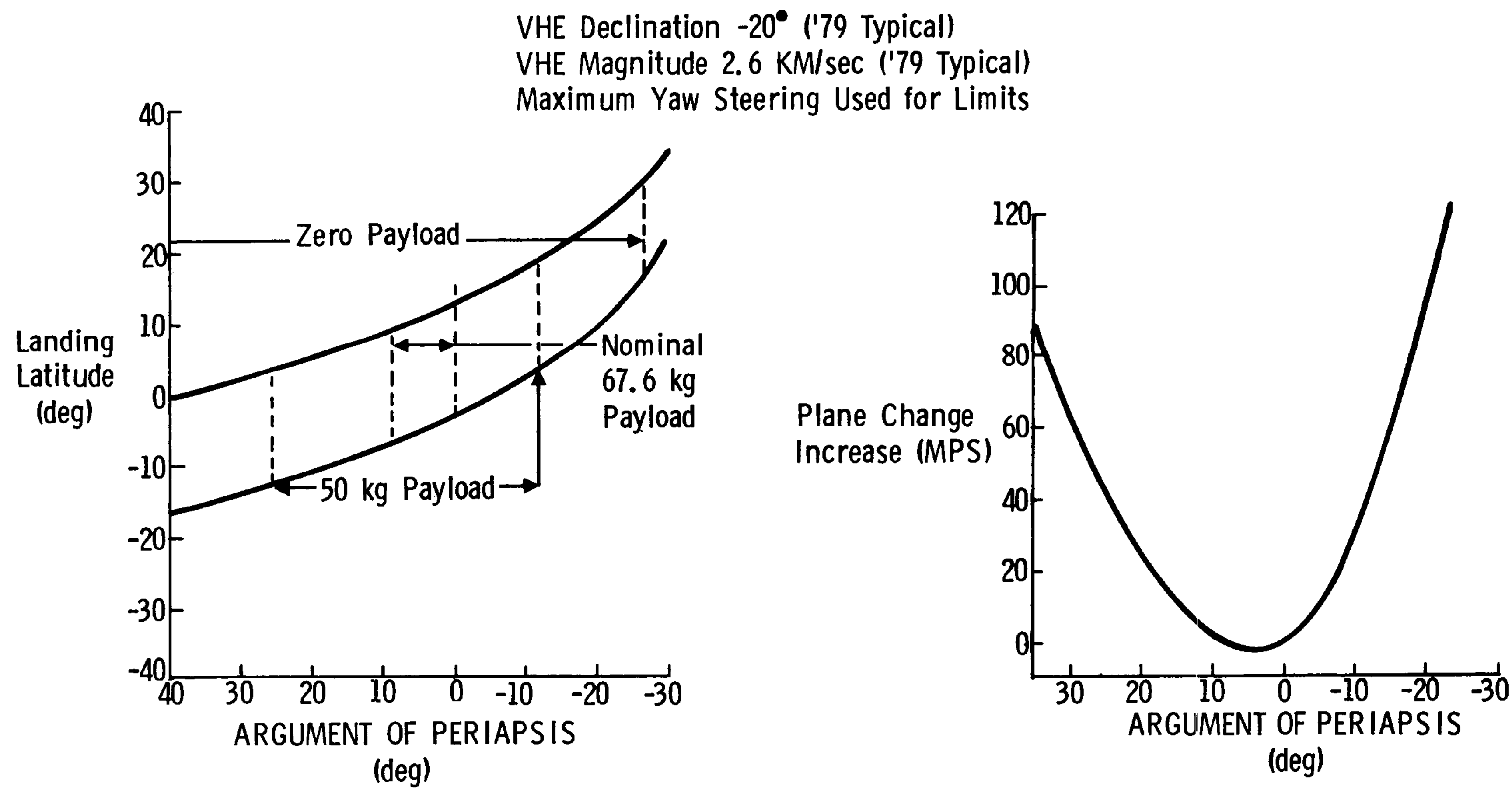


Figure II-3 Mars Latitude Accessibility Using Off-Optimal Plane Change Strategy

	15.15 hr Phobos	15.15 hr Deimos	30.6 hr Phobos	30.6 hr Deimos
Relative Range	1000 km	200 km	200 km	1000 km
Relative Velocity	1400 m/s	490 m/s	640 m/s	865 m/s
TV Max Slew Rates	.1°/sec	.1°/sec	.03°/sec	.1°/sec
Occurance	4/30 day	1/30.3 hr	1/30.6 hr	4/120 day

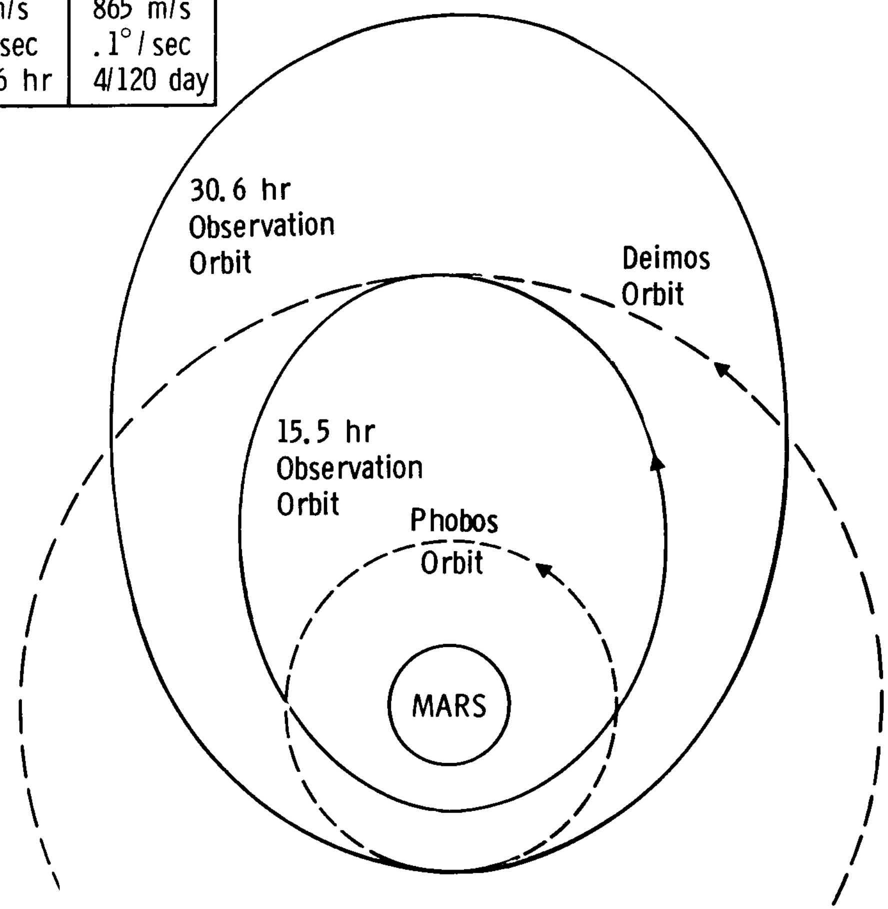


Figure II-4 Comparison of Observation Orbits inclination. The 30.6 hour observation orbit has a relative position cycle between Deimos and the spacecraft of 120 days and also contain four closest approaches of less than 1000 km. This observation orbit which is synchronous with Phobos and Deimos moves in relation to the spacecraft orbit. The encounter conditions between the two possible observation orbits are approximately the same, however, the shorter cycle of observations and the lower periapse altitude (lower ΔV) of the 15.15 hour orbit make it more desirable for the baseline mission. The 15.15 hour observation orbit has, in addition to the four viewings with relative ranges, less than 1000 km, 16 more viewings at less than 3500 km. This is a feature of the closeness of the relative orbits near periapsis.

In determining the baseline mission profile, it was necessary to select a propulsion system concept since this choice has effects throughout the mission design. Figure II-5 indicates the result of a parametric study comparing the growth orbiter (increased propellants), staged orbiter, and space storable propellants (specific impulse of 385 sec). The space storable propellants yield significantly higher payload capabilities, however, it is a new development item and therefore costly. The comparison of the growth orbiter and the staged orbiter indicates that the growth orbiter provides a slightly higher payload. This is because of the higher propulsion system weight for the staged configuration as compared to the growth orbiter. The propellant requirements for the growth orbiter did not exceed the 50% growth limit for any of the ΔV requirements with this initial weight of 300 kg. For heavier initial weights or larger ΔV requirements where the required propellants exceed this 50% growth limit, a four tank, two engine system is required and a decrease in payload capability of approximately 170 kg occurs because of the heavier propulsion system weight. For payloads of ΔV requirements in this region,

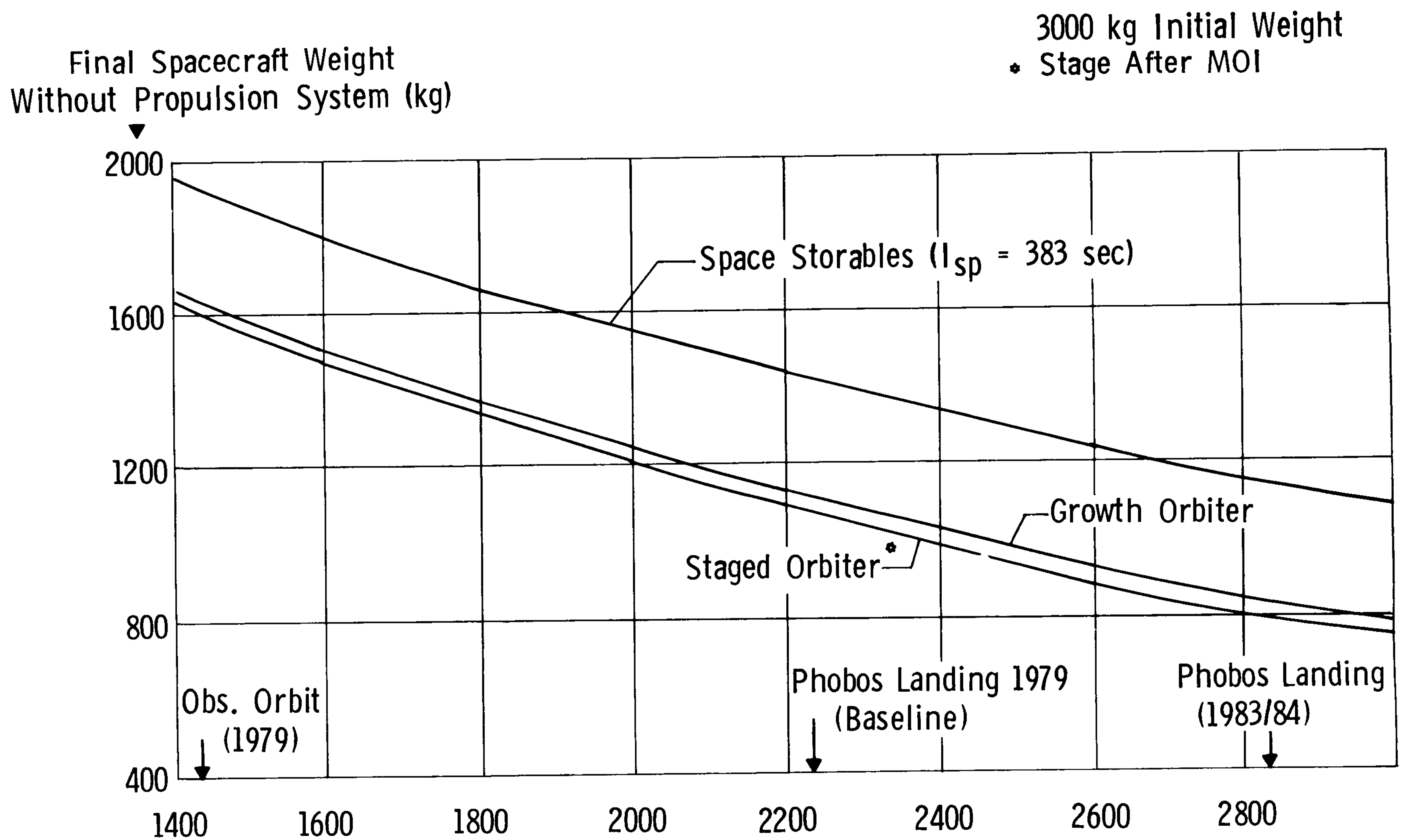


Figure II-5 Orbiter Propulsion Alternatives the staged orbiter would provide superior payload performance.

A final trade study involved the possibility of using a light-weight spacecraft for an observation orbit or rendezvous with Phobos or Deimos without a Mars Lander. This information is described in Table II-5. Mariner and Pioneer spacecraft technology were investigated to determine if either is adaptable for a science mission to Phobos or Deimos. The observation orbit mission required a 1430 meters per second (mps) ΔV budget and a Phobos rendezvous and stationkeeping orbit mission required 2060 mps ΔV budget. The Mariner '71 has a ΔV capability of 1540 mps. Therefore, it could achieve the observation orbit with no modifications. The Pioneer F and G configuration was used as the spacecraft concept for the spin stabilized candidate. It has less than 100 mps ΔV capability, therefore, a propulsion module similar to that on the Mariner vehicle was added. The propulsion systems on both Mariner and Pioneer vehicles were stretched for the station-keeping orbit missions. The Mariner vehicle carried its existing science package which could be modified to include x-ray fluorescence or other equipment. The Pioneer science was assumed to include spin scan imaging similar to Earth weather satellites along with instruments previously discussed for these mission modes. The Pioneer spacecraft appears to have the same data rate capability as the Mariner spacecraft, however, this is true only when the antenna is pointed at Earth. Since the science experiments and communication will have to share the vehicle pointing time, the total data transmission is reduced by the time spent in gathering the science data, i.e., non-Earth pointed. This pointing conflict can be reduced by use of mechanically or electrically despun antennas. However, these have lower gain than the Pioneer F and G systems which would again reduce the total data transmission.

Table II-5 Alternate Spacecraft Concepts - 1979 Opportunity (Phobos)

	Mariner '71	Pioneer/Planetary Explorer
Spacecraft Weight		
Observation Orbit	1013 kg	460 kg
Station-keeping	1200 kg	618 kg
Propulsion Subsystem		
Observation Orbit	553 kg	210 kg
Station-keeping	694 kg	260 kg
Science Payload		
Observation Orbit	68 kg	30 kg
Station-keeping	68 kg	40 kg
Data Rate	8-16 kilobit/sec	8-16 kilobits/sec
Operational Life	3-6 months	2-4 years
Relative Cost	100-125 mill	75-100 mill

B. MISSION DESCRIPTION

Figure II-6 is an overview of the baseline mission profile. This mission involves the use of a 97 hour capture orbit from which a Mars lander deorbits and a Phobos/Deimos mission proceeds. The Mars orbit insertion (MOI) is done at periapsis at an altitude of 1500 km (same as Viking '75) and the applied ΔV there leaves the spacecraft in a 97 hour orbit with an apoapsis at 95000 km and periapsis in Deimos' orbit plane. The lander de-orbits from this orbit 5.5 hours prior to its entry interface and the orbiter with its Phobos/Deimos science module continues in the 97 hour orbit. At the next apoapsis, a maneuver is performed to change the orbit plane to coincide with Deimos' orbit plane and also to lift periapsis to the periapsis altitude of the observation orbit (2660 km). This orbit is not shown in the figure. Also not shown, is the phasing orbit which has a variable period between 30 and 60 hours. This phasing orbit is established by a retromaneuver at periapsis and its purpose is to allow the relative geometry between the spacecraft and Deimos to be adjusted so that after the observation orbit is established, the spacecraft will be at apoapsis when Deimos is near that same position. The observation orbit is established by another retro maneuver at periapsis reducing the period to 15.149 hours and this orbit has an apoapsis about 100 km less than the orbital altitude of Deimos. This period is half that of Deimos so that Deimos will be near the spacecraft every other time the spacecraft is at apoapsis. As mentioned earlier, periodic close viewings of Phobos occur over a 30 day cycle. After an adequate time for viewing of both satellites and for an Earth-based decision to be made as to which satellite is to be more fully investigated, the spacecraft leaves the observation orbit for one of the two satellites. If Deimos

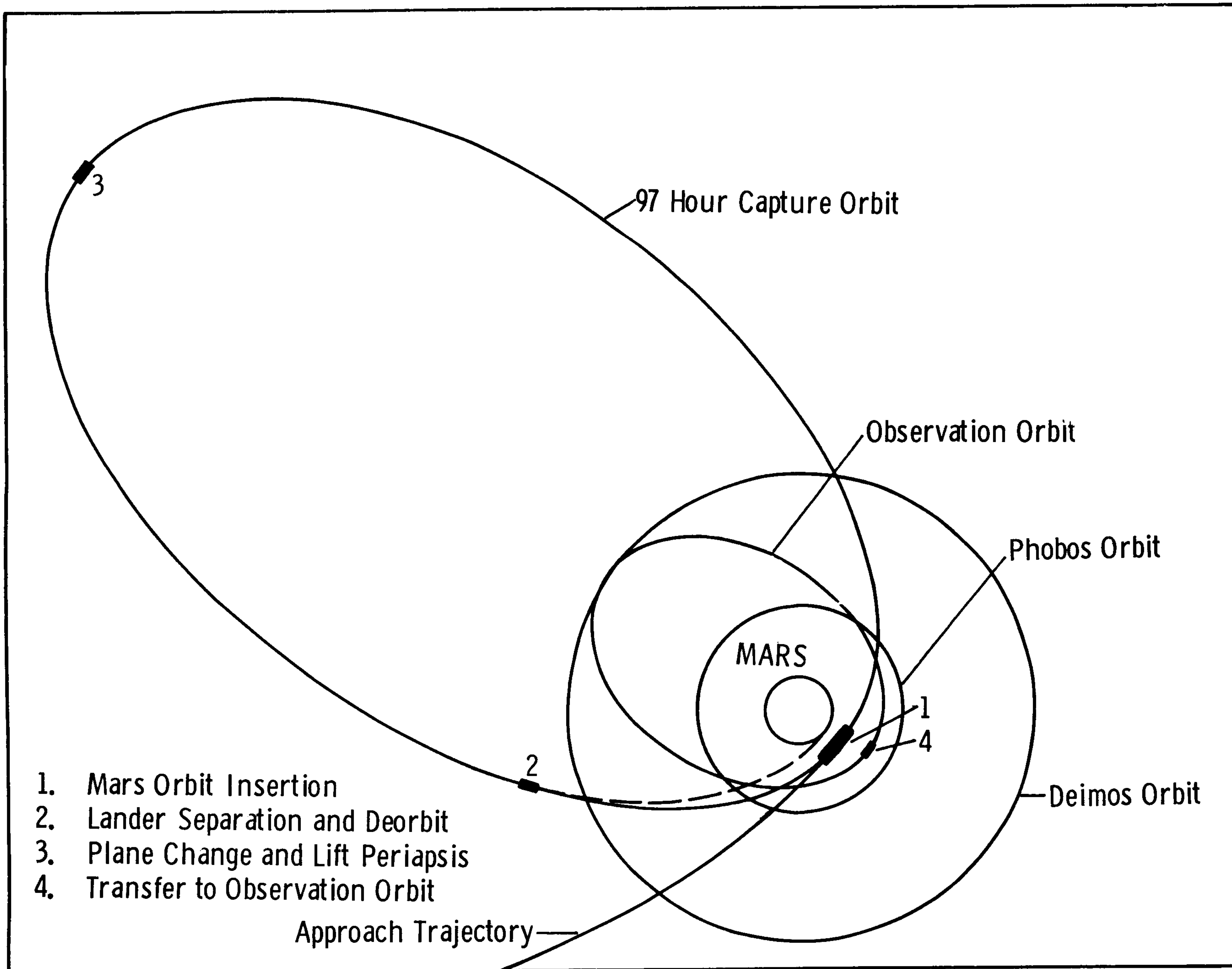


Figure II-6 Overview of Mission Profile

is the target satellite, a maneuver is performed at apoapsis, when Deimos is there, to match Deimos' orbit. Using the Orbiter's TV cameras, tracking data, and satellite ephemeris information, this maneuver leaves the spacecraft within 22 km of the desired separation distance of 50 km. At this point, the rendezvous radar acquires Deimos and the rendezvous and landing sequence is initiated. A small closing velocity is obtained using the Orbiter engines and is removed prior to impact to allow a touchdown on Deimos at a velocity of less than 2 meters per second. This portion of the mission is more fully described in Section III.

If the choice is to investigate Phobos rather than Deimos, a maneuver is performed at apoapsis to lift periapsis a portion of the way to the orbital altitude of Phobos. This establishes another phasing orbit to allow the geometry to change so that Phobos will be in position when the spacecraft completes its sequence of maneuvers. The next maneuver is to raise periapse to Phobos' altitude. At periapse, the spacecraft fires its engines to leave the spacecraft within 22 km of the desired separation distance between Phobos and the spacecraft of 50 km and with the same orbital period. From this point, the sequence of events is the same as with the Deimos landing.

Figure II-7 indicates the 1979 payload capability using the growth orbiter and 97 hr. capture orbit for the various Phobos/Deimos missions as a function of the initial weight at Earth injection. Included in this figure are the launch vehicles injection capability limits for the Titan IIIE/Centaur, Titan IIIE/Centaur GT (Growth Tank Centaur), and Titan IIIE 7/Centaur. The Titan IIIE nomenclature is the new name for the NASA version of the Titan IIID vehicles. This change was made recently to differentiate between the military vehicle and the NASA vehicle which has some small differences (not affecting performance). As can be seen,

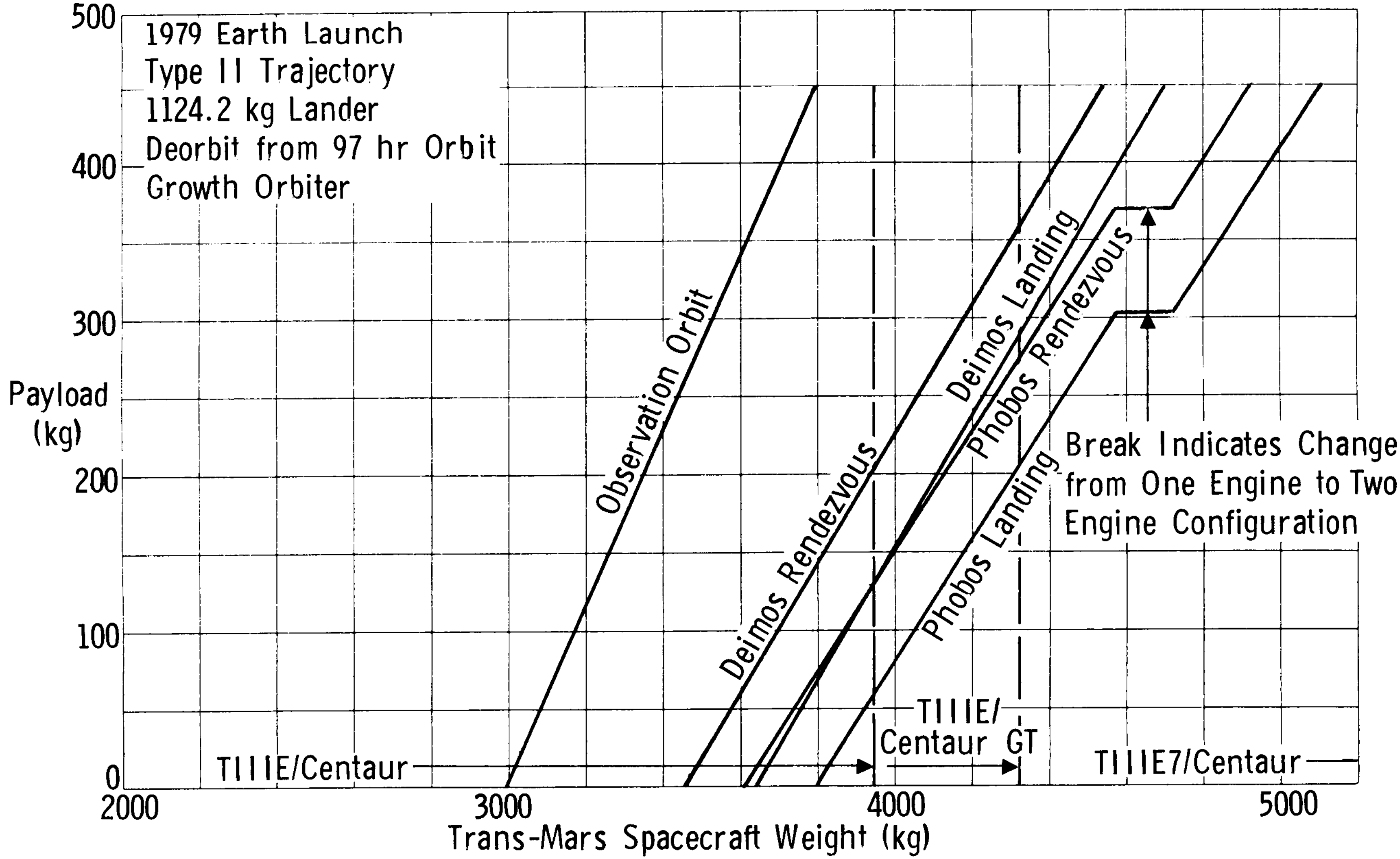


Figure II-7 Trans-Mars Spacecraft Weight vs Payload - 1979 - 97 Hour Orbit

the TIIIE/Centaur can accommodate a Phobos landing mission with 67 kg of science payload. Data for launch opportunities in 1979, 1981, and 1983/84; three propulsion systems (Growth, Staged, and Space Storable); and three Mars Lander modes (Direct Entry, 97 hour capture orbit, and 24.6 hour capture orbit) are shown in Figures II-8 through II-33. In each of these figures, the applicable launch vehicle injection limits are indicated. The baseline mission performance characteristics are indicated in Table II-6. The ΔV budget for this mission is the same as in Phase I except for a reduction in the allocation for navigation uncertainties and a reduction in the ΔV required for MOI caused by performing this maneuver at a lower altitude (1500 km instead of 2660 km) in Phase I. These two reductions yielded a combined savings of 140 meters per second. The post maneuver weights are significantly different from Phase I since the Titan IIIE/Centaur's capability is more fully utilized and also the Mars lander is included until after the MOI maneuver. The weight landed on Phobos is greater since a landed orbiter concept is utilized although the total orbiter/payload weight is less. The combined weight of orbiter and Phobos lander in Phase I was 1465 kg which included 482 kg of lander and 983 kg of orbiter. The sequence of events is a little longer since 30 days rather than 15 days is allowed in the observation orbit prior to the rendezvous and landing on Phobos.

The navigation analysis and requirements utilized in this phase of the study are briefly indicated in Table II-7 and are fully described in Phases I and II.

A side study was investigated during this final phase of the study concerning the applicability of Venus "swingbys" both from Earth to Mars and from Mars to Earth. Figure II-34 indicates the general geometry and results of this side study. Both Earth

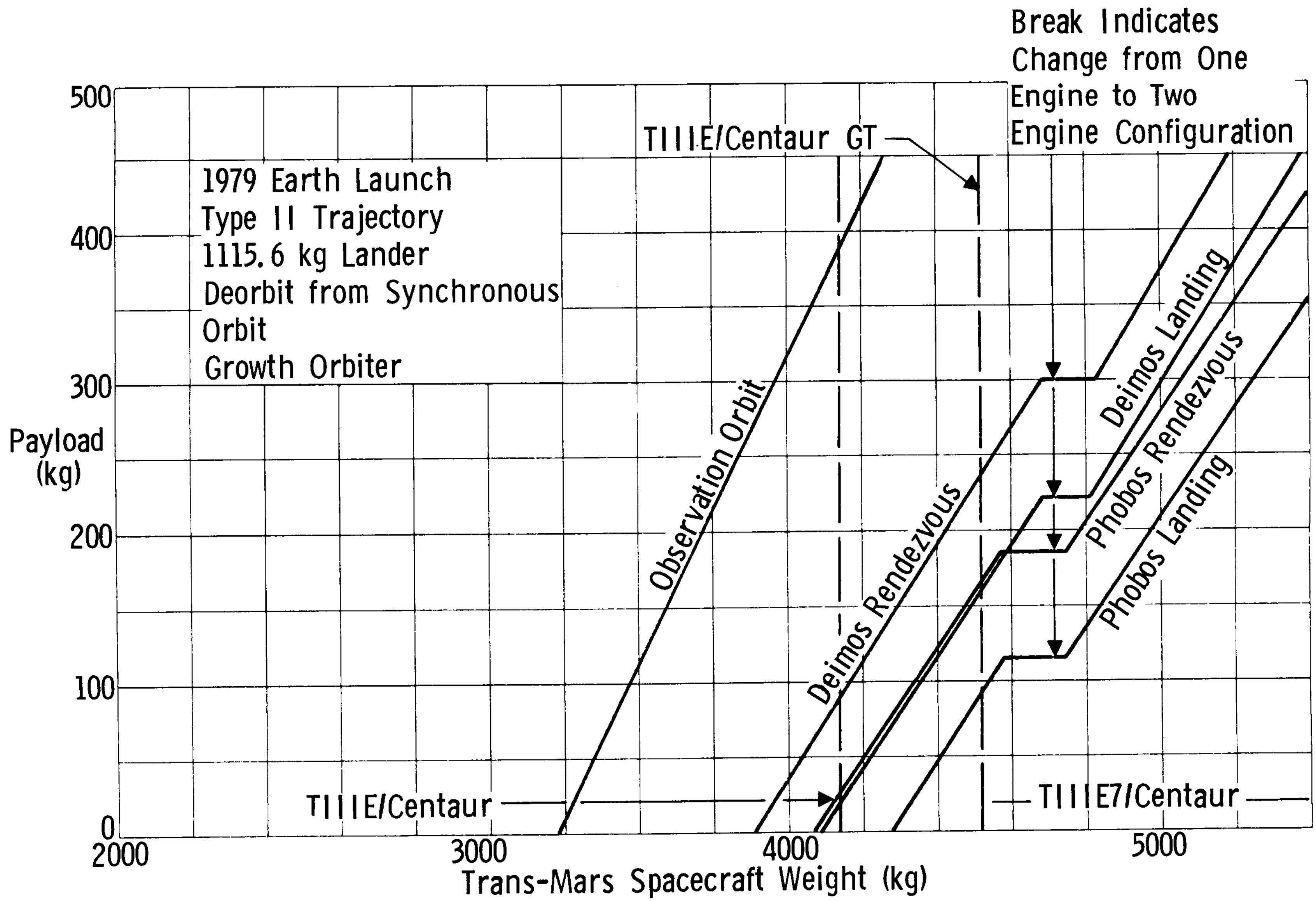


Figure II-8 Trans-Mars Spacecraft Weight vs Payload - 1979 - Synchronous Orbit

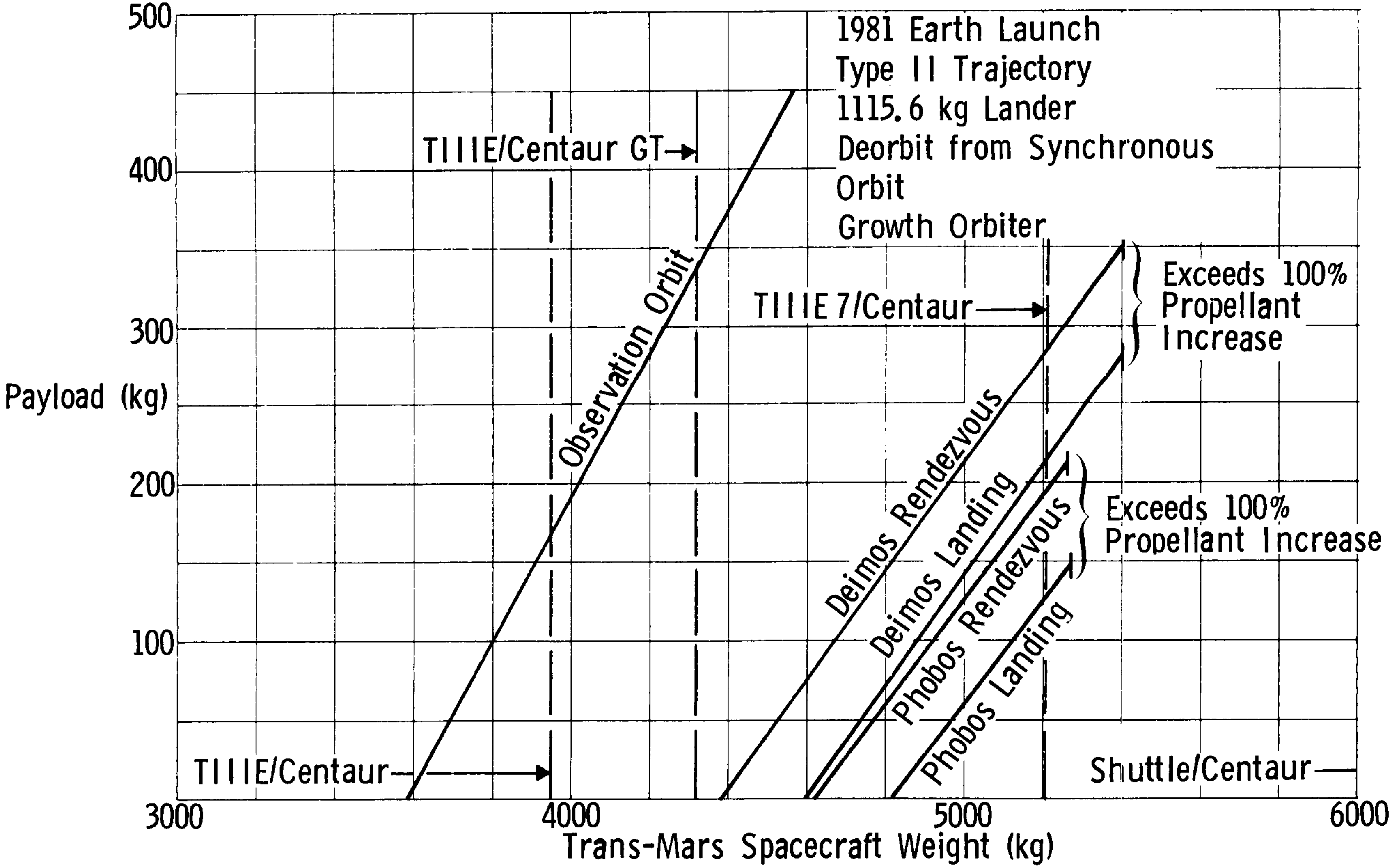


Figure II-9 Trans-Mars Spacecraft Weight vs Payload - 1981 - Synchronous Orbit

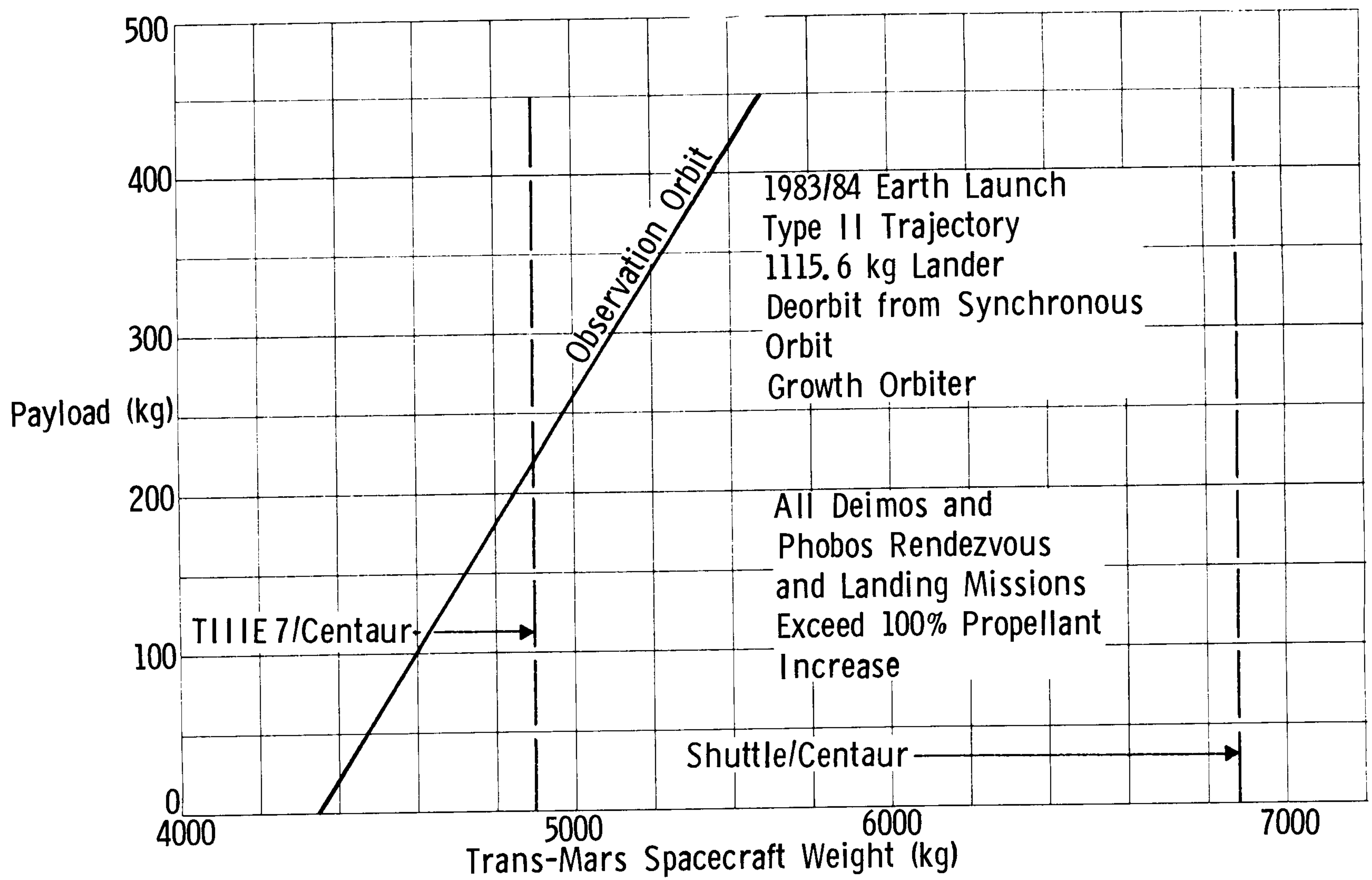


Figure II-10 Trans-Mars Spacecraft Weight vs Payload - 1983/84 - Synchronous Orbit

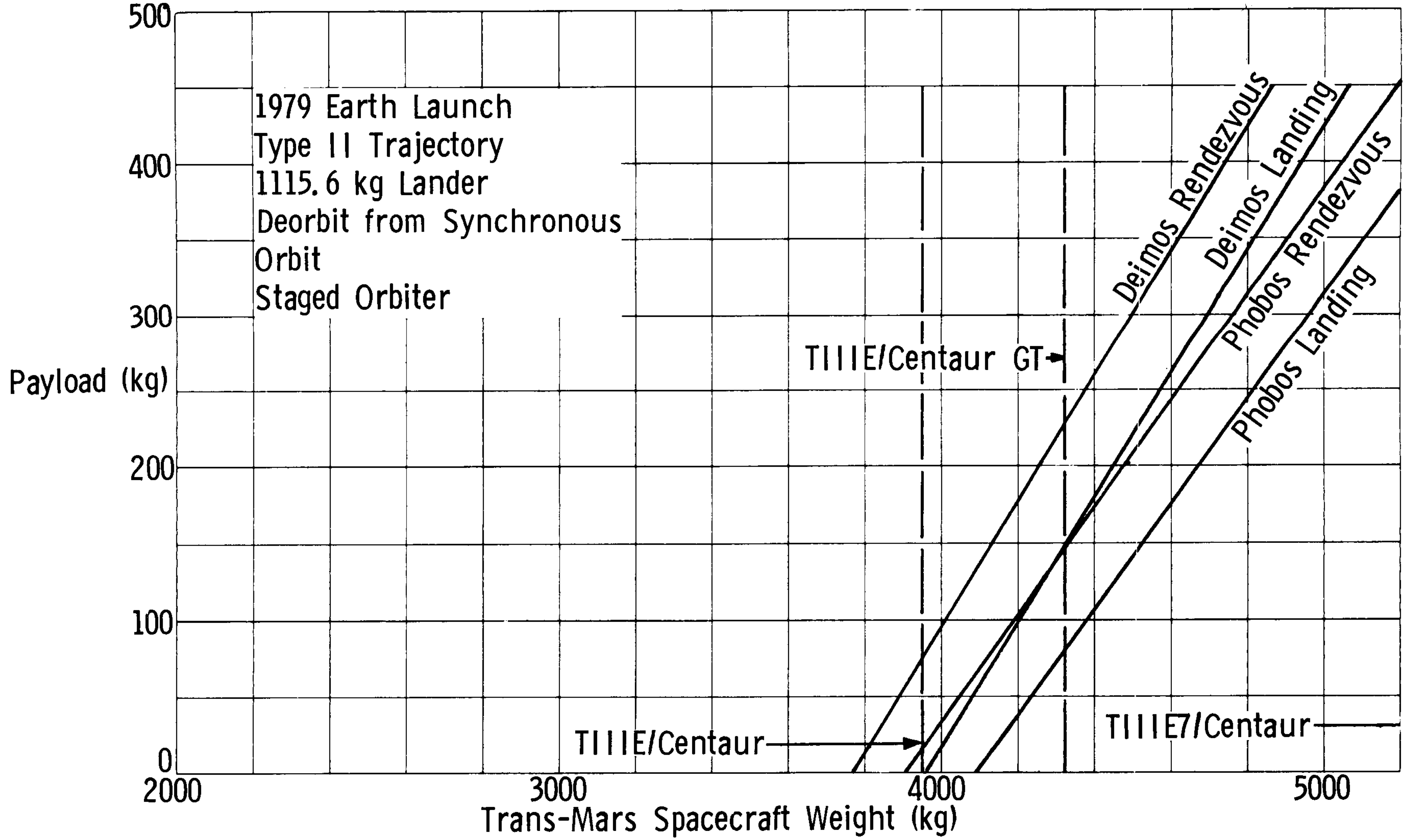


Figure II-11 Trans-Mars Spacecraft Weight vs Payload - 1979 - Synchronous Orbit

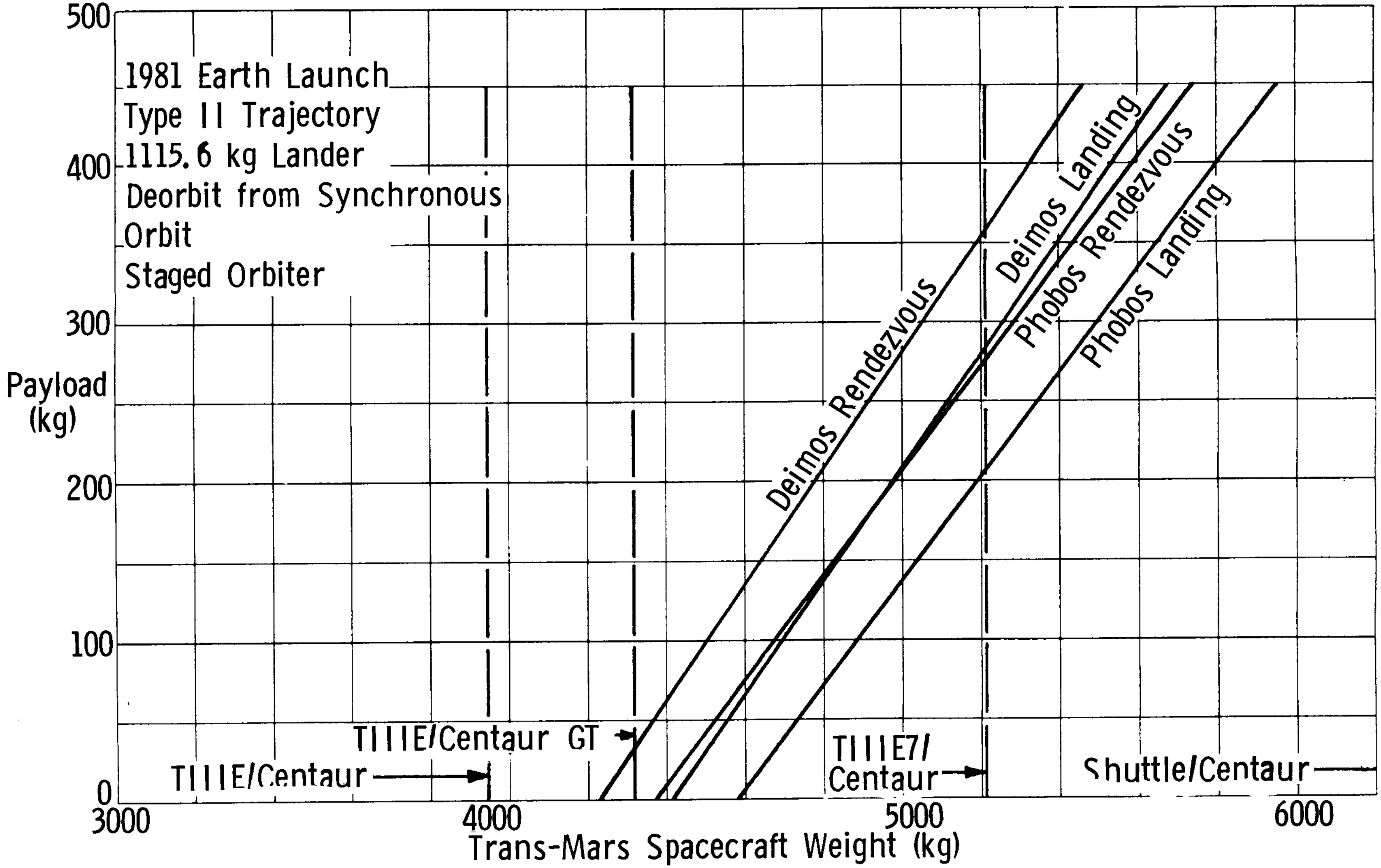


Figure II-12 Trans-Mars Spacecraft Weight vs Payload - 1981 - Synchronous Orbit

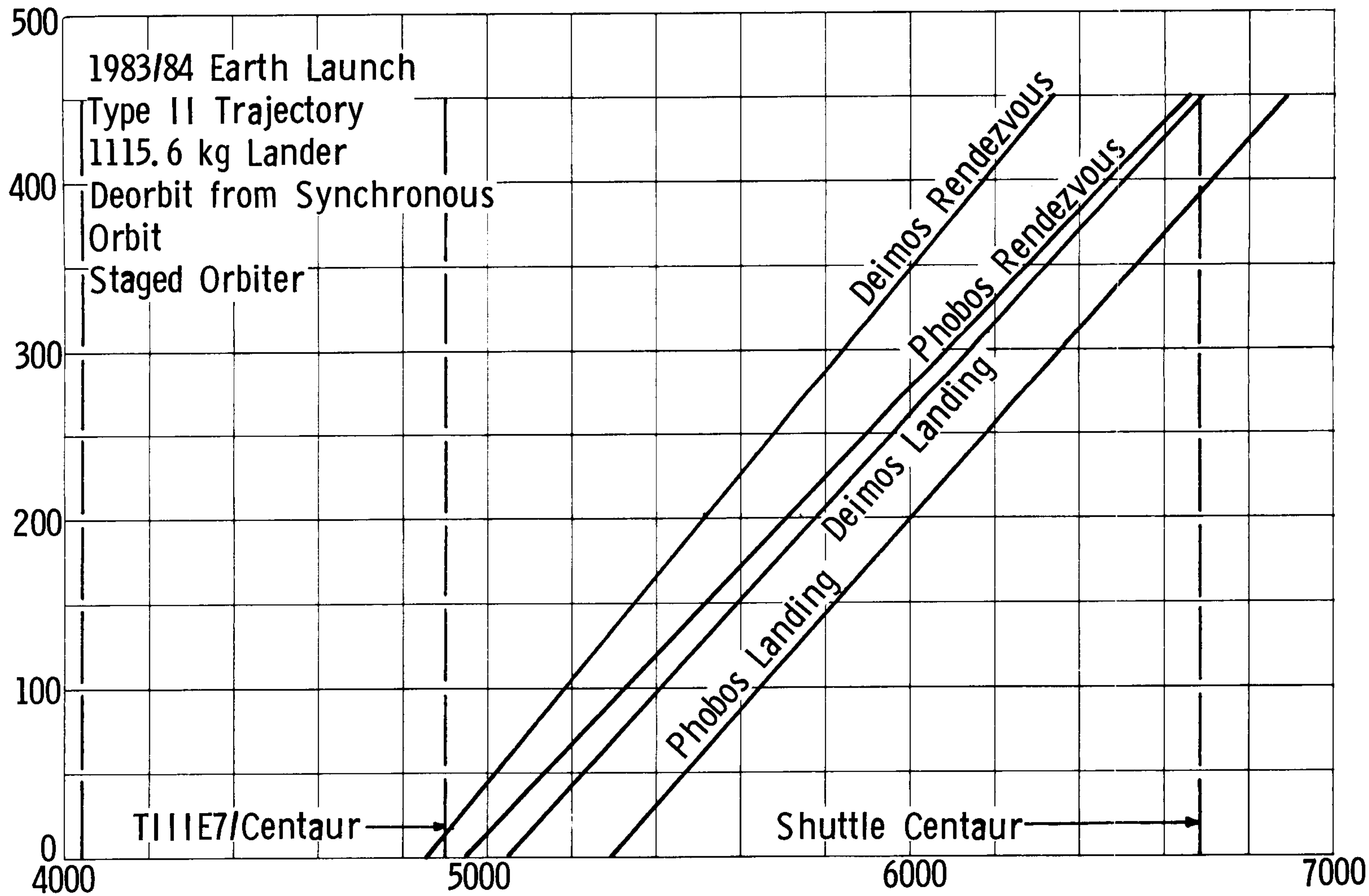


Figure II-13 Trans-Mars Spacecraft Weight vs Payload - 1983/84 - Synchronous Orbit

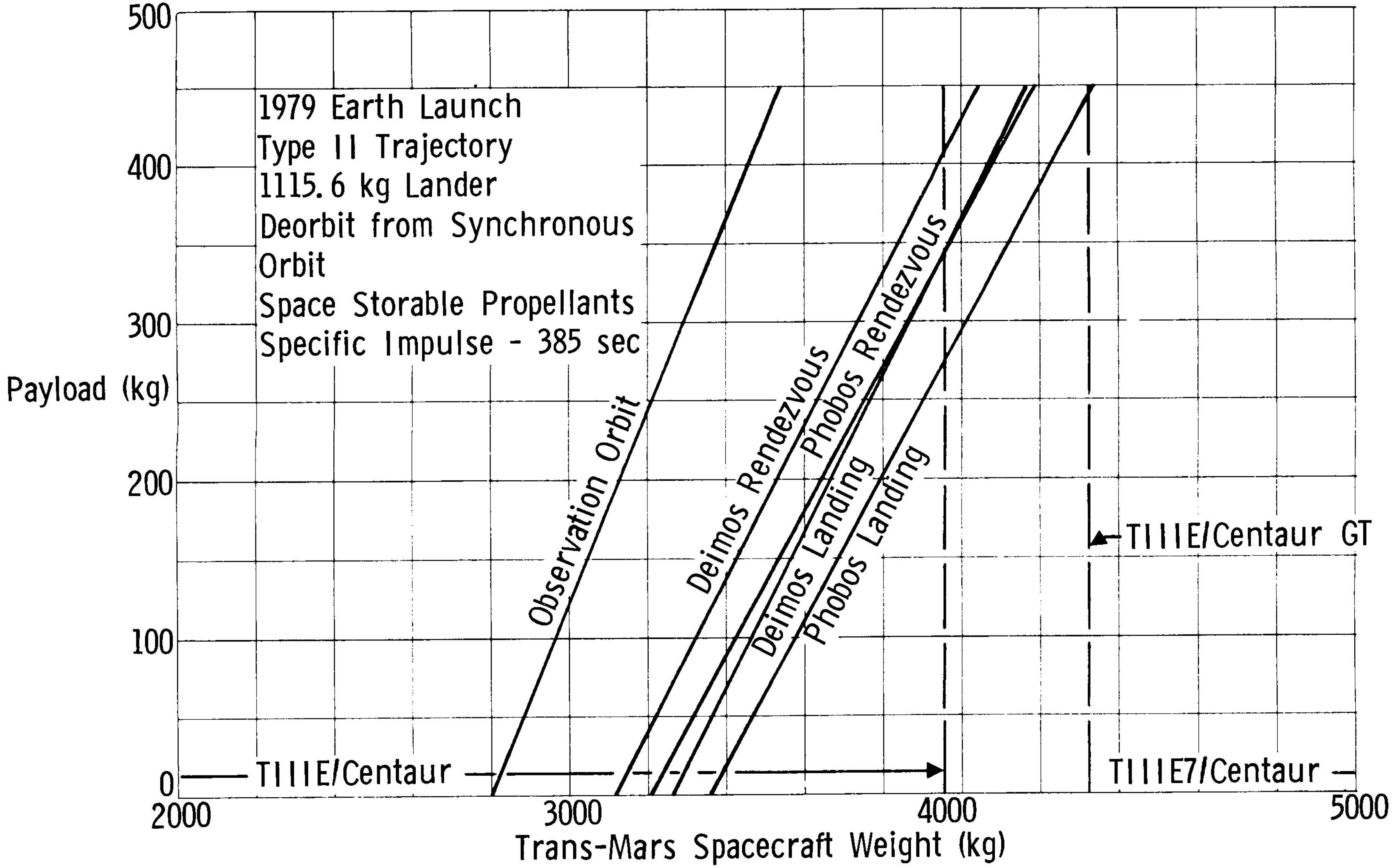


Figure II-14 Trans-Mars Spacecraft Weight vs Payload - 1979 - Synchronous Orbit

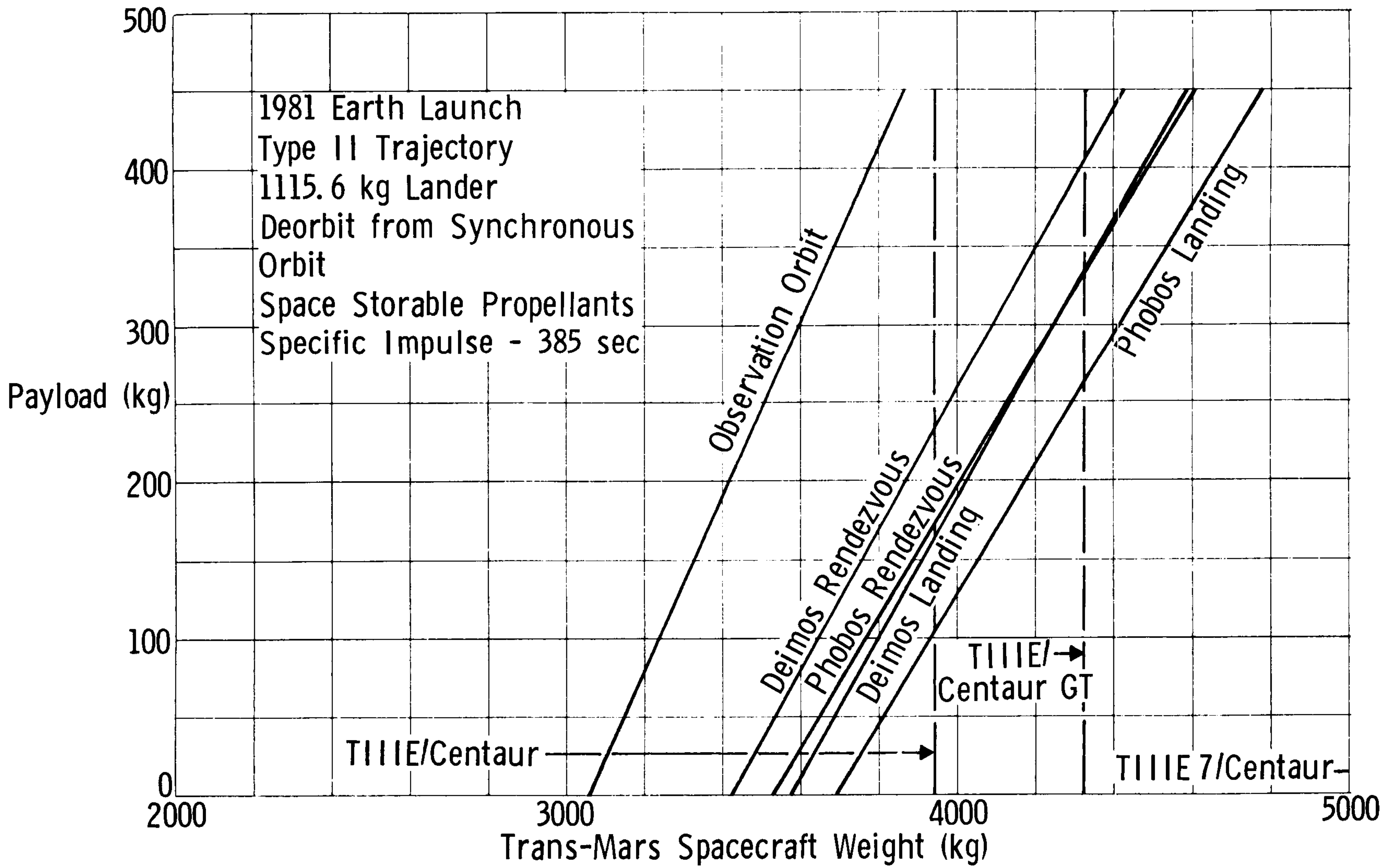


Figure II-15 Trans-Mars Spacecraft Weight vs Payload - 1981 - Synchronous Orbit

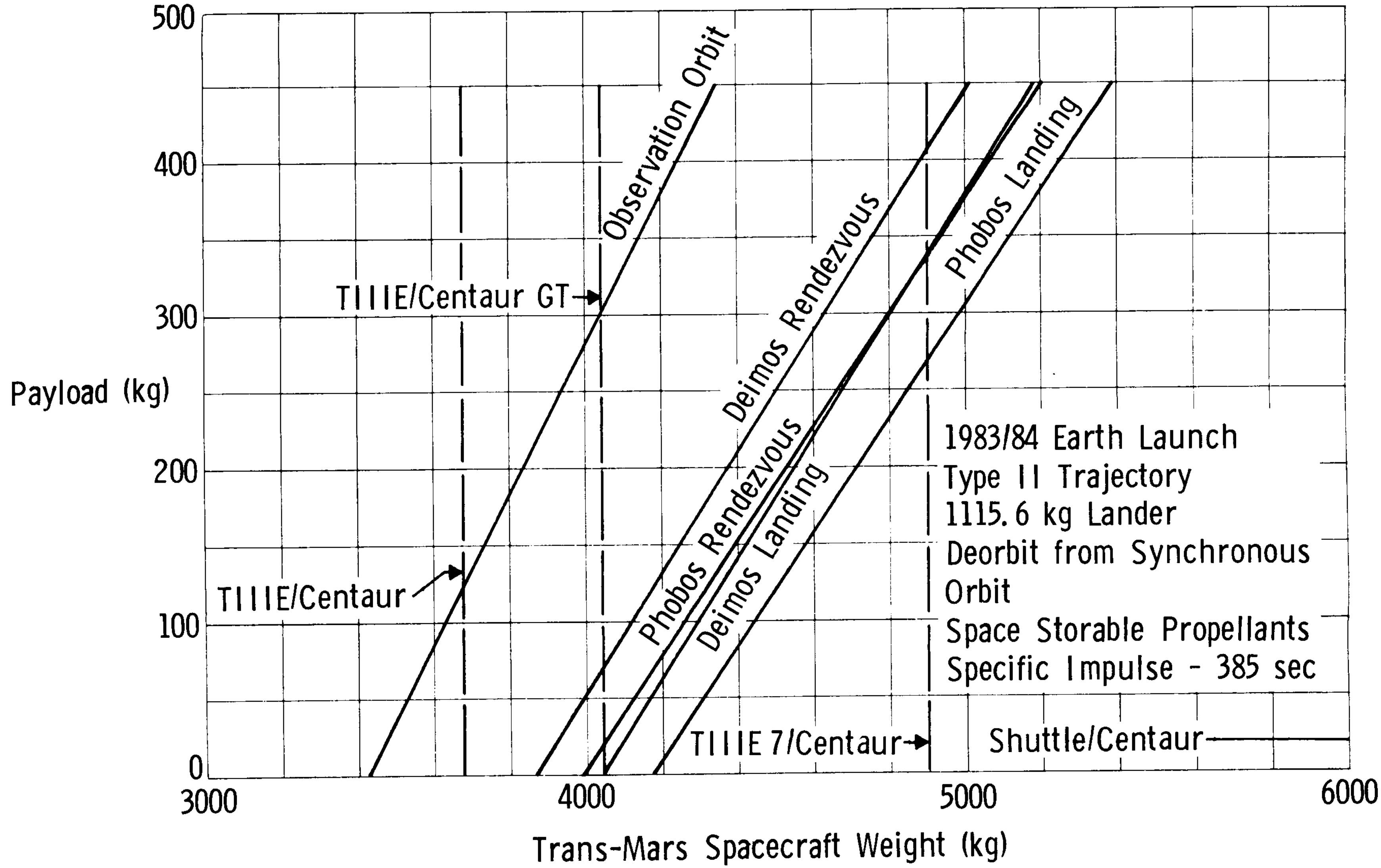


Figure II-16 Trans-Mars Spacecraft Weight vs Payload - 1983/84 - Synchronous Orbit

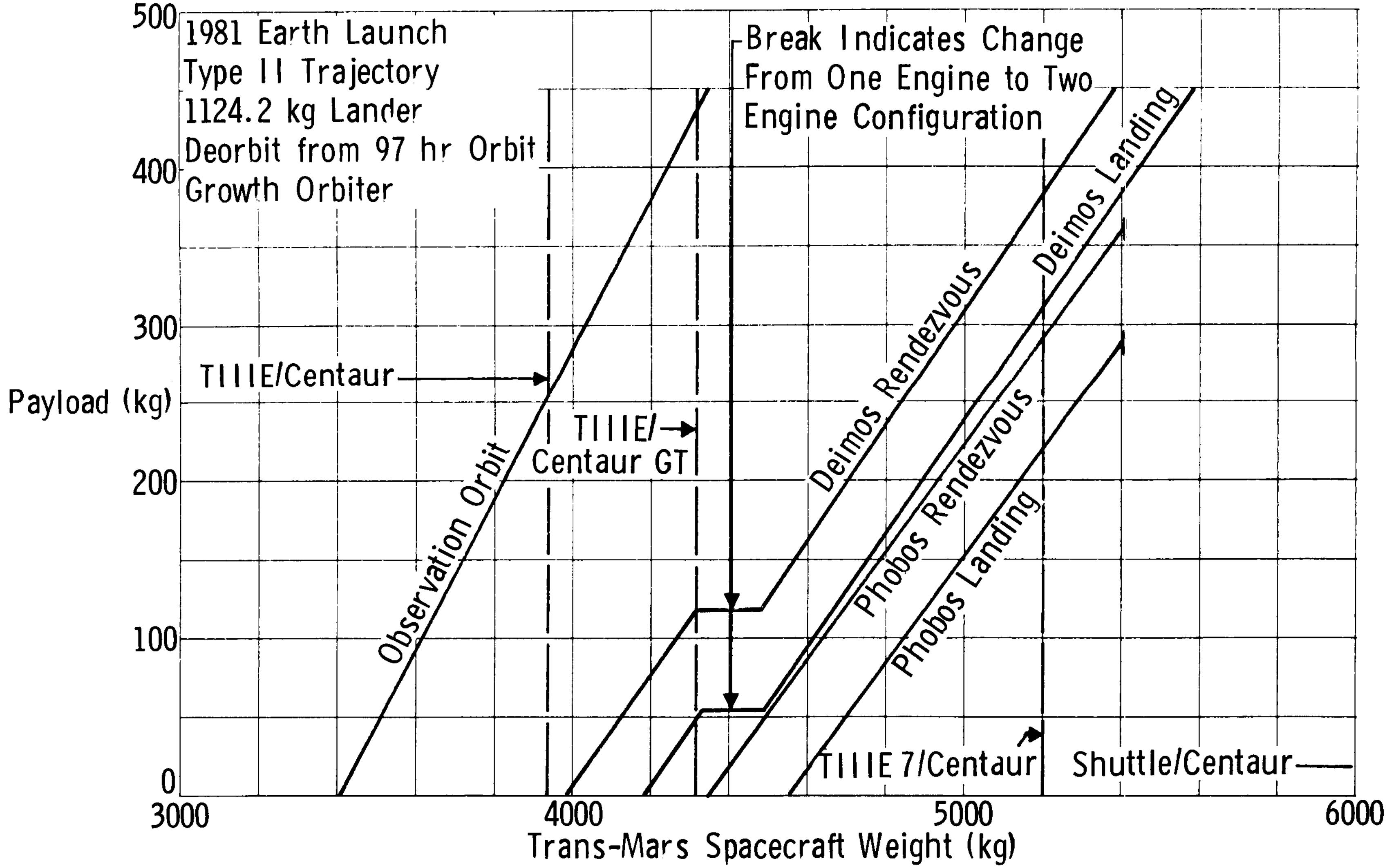


Figure II-17 Trans-Mars Spacecraft Weight vs Payload - 1981 - 97 Hour Orbit

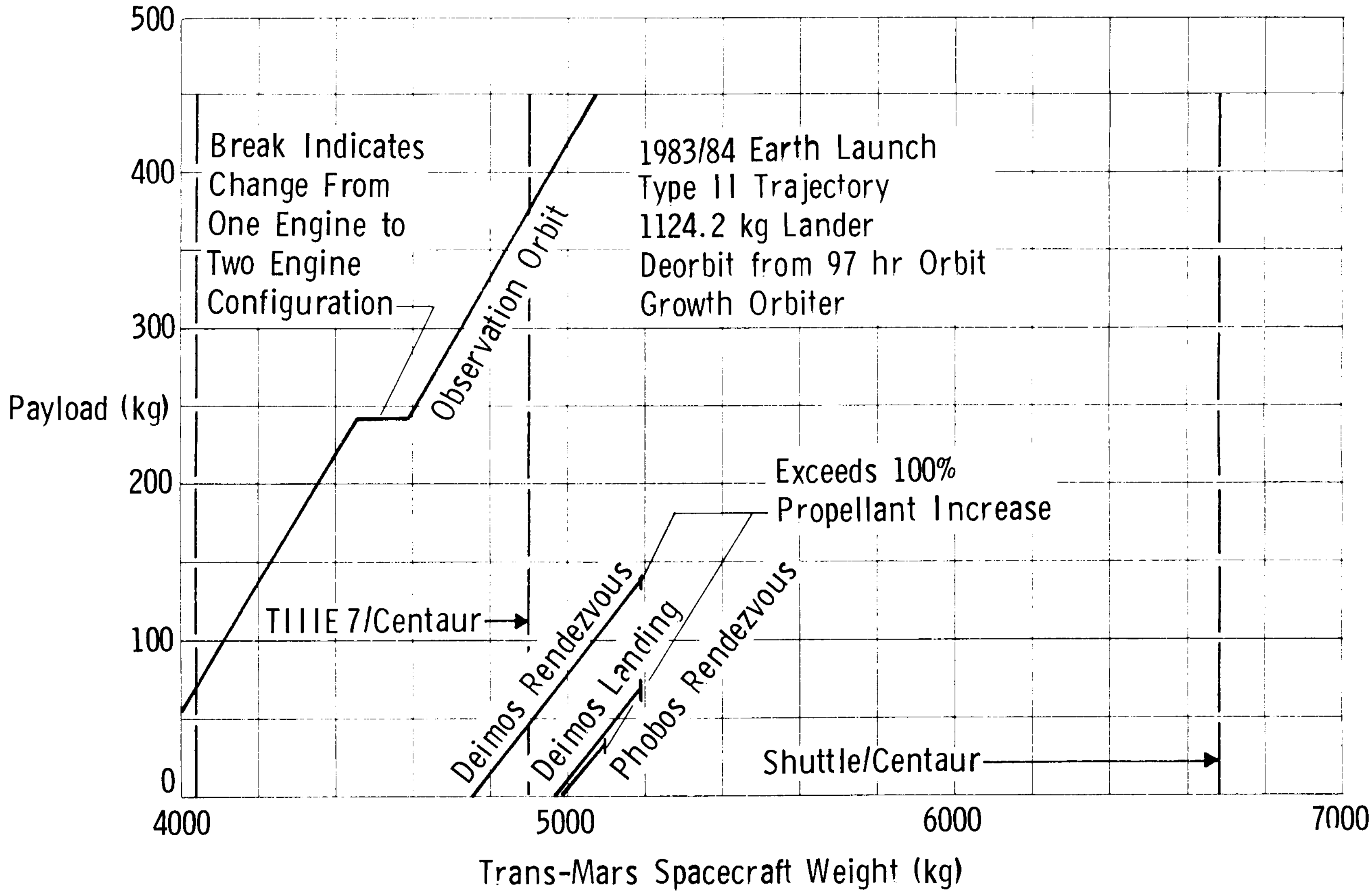


Figure II-18 Trans-Mars Spacecraft Weight vs Payload - 1983/84 - 97 Hour Orbit

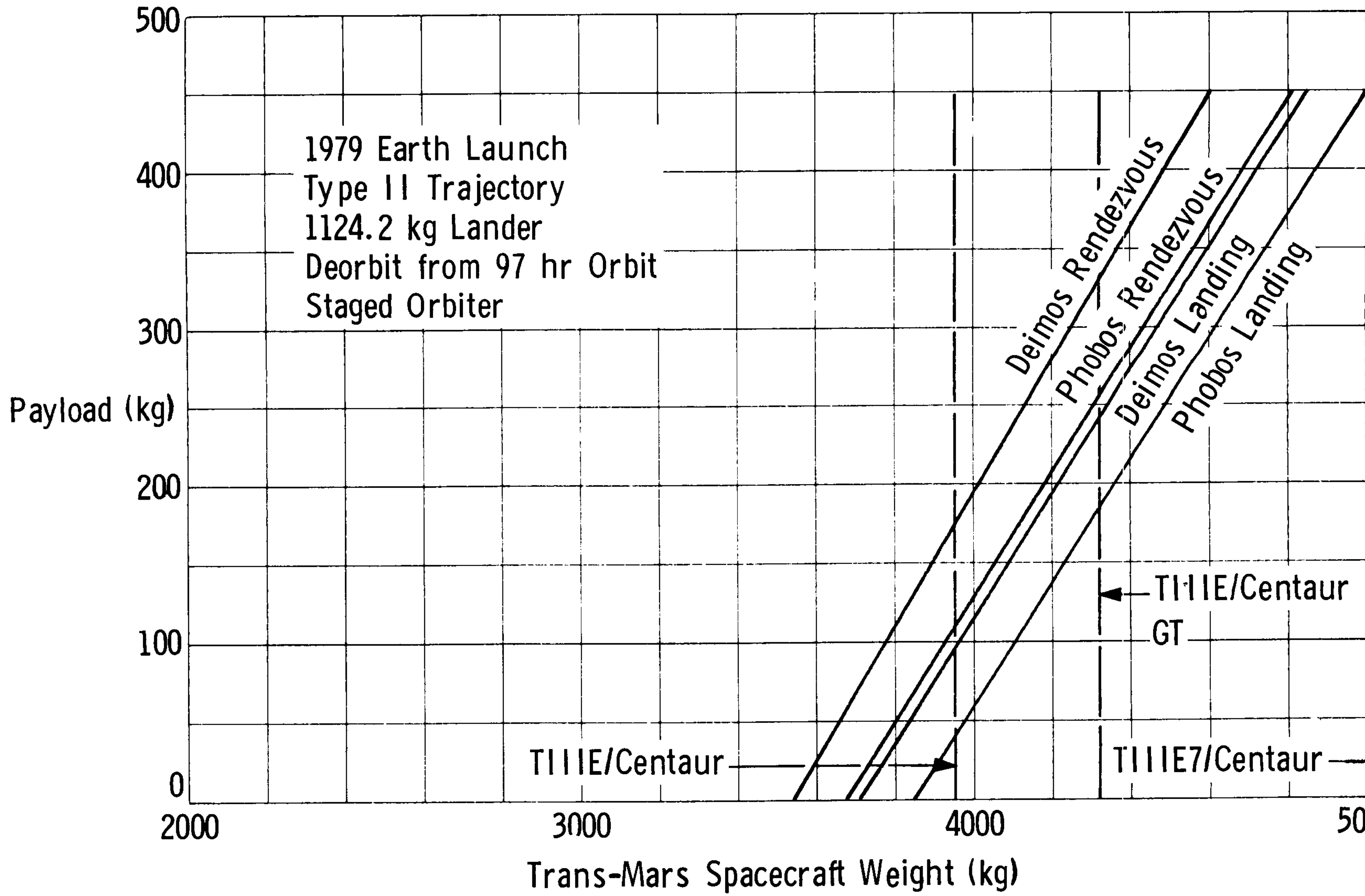


Figure II-19 Trans-Mars Spacecraft Weight vs Payload - 1979 - 97 Hour Orbit

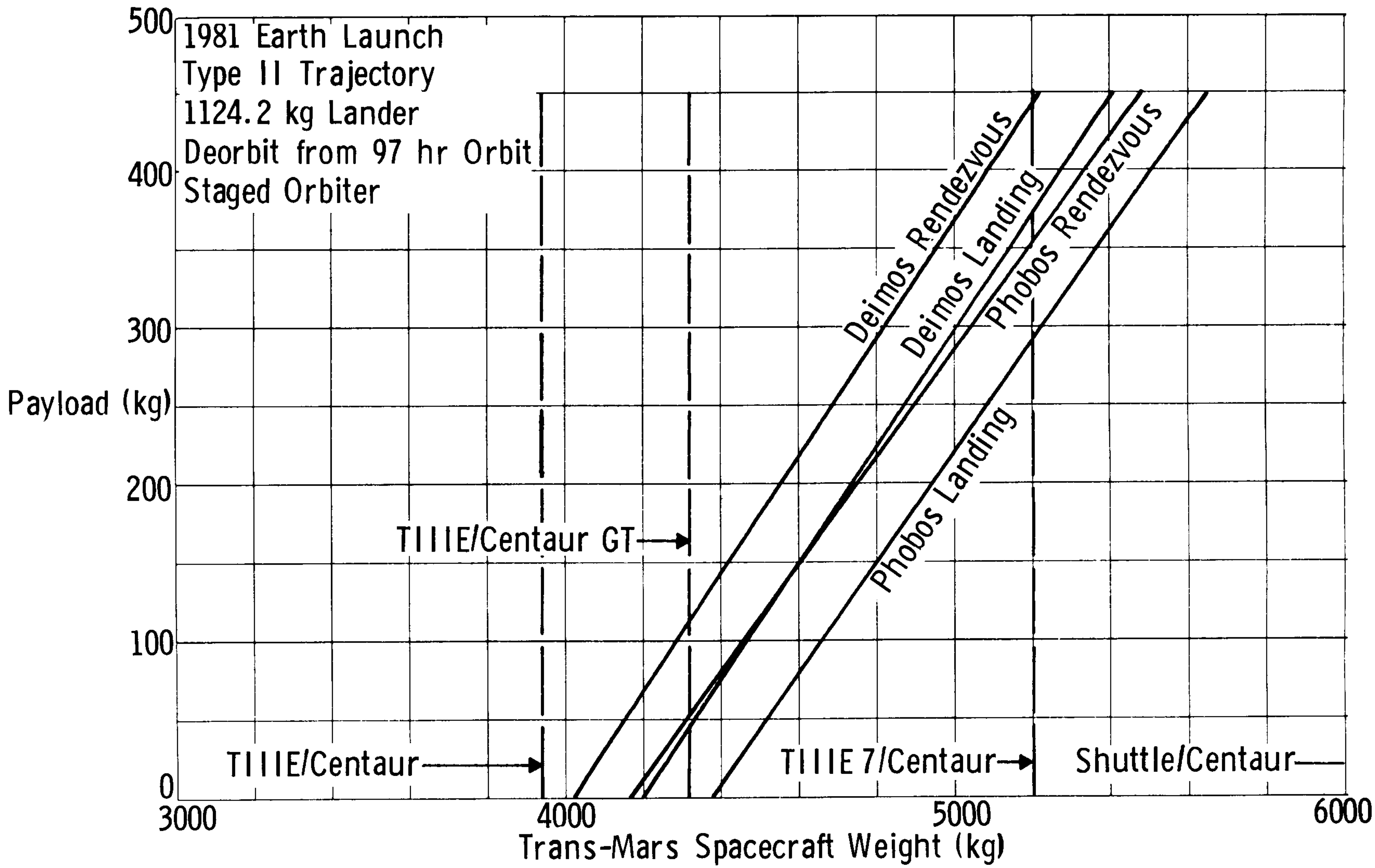


Figure II-20 Trans-Mars Spacecraft Weight vs Payload - 1981 - 97 Hour Orbit

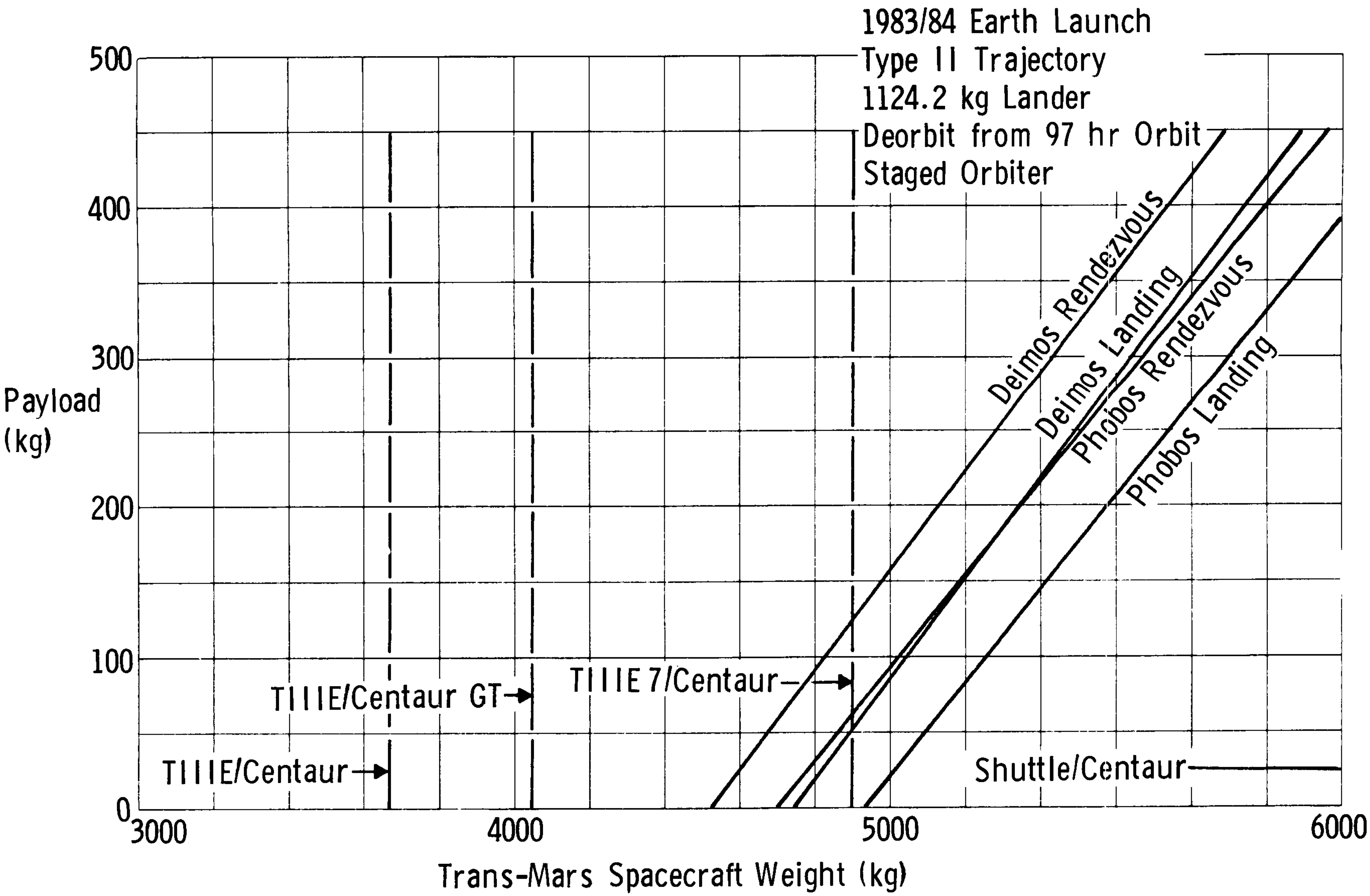


Figure II-21 Trans-Mars Spacecraft Weight vs Payload - 1983/84 - 97 Hour Orbit

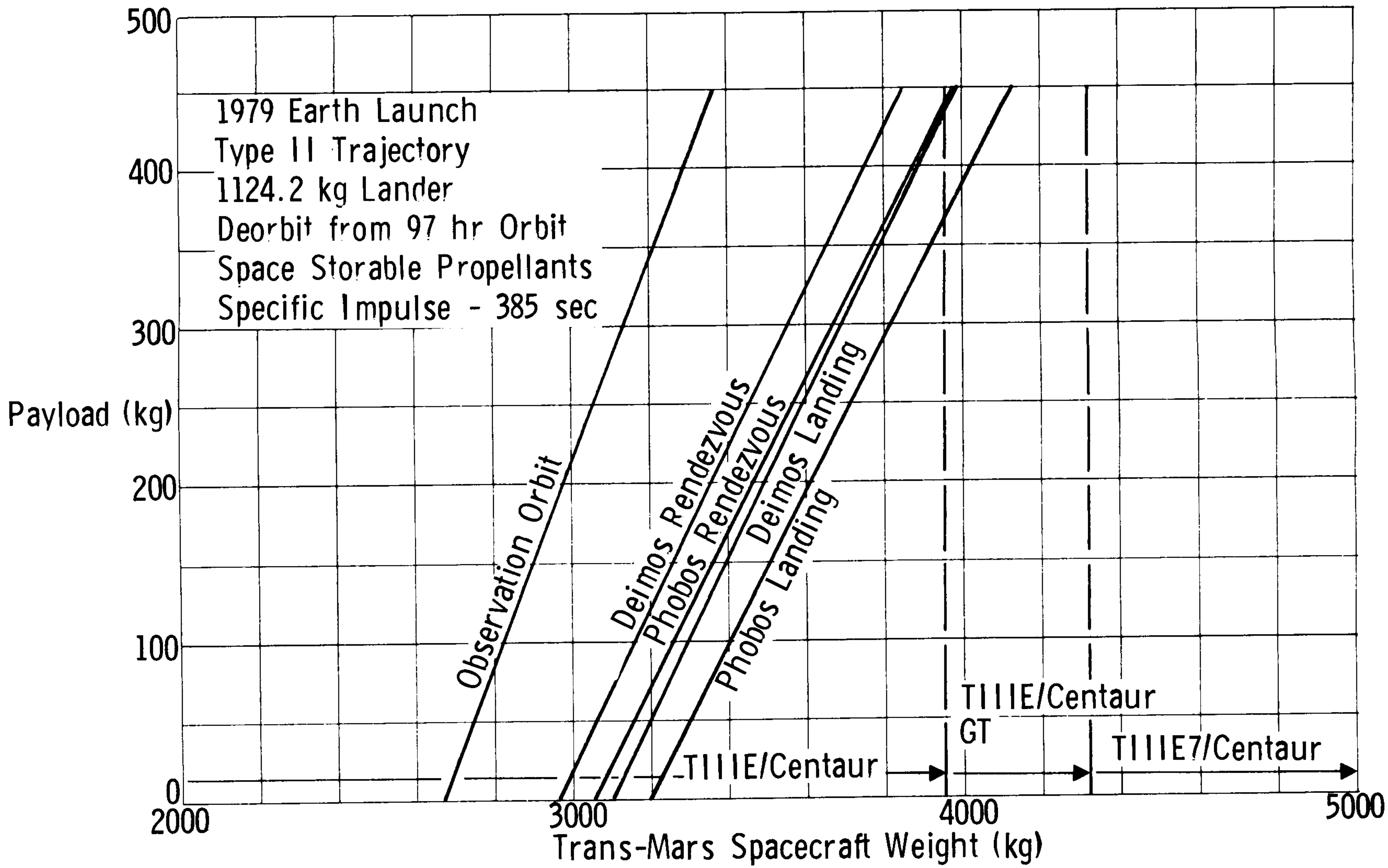


Figure II-22 Trans-Mars Spacecraft Weight vs Payload - 1979 - 97 Hour Orbit

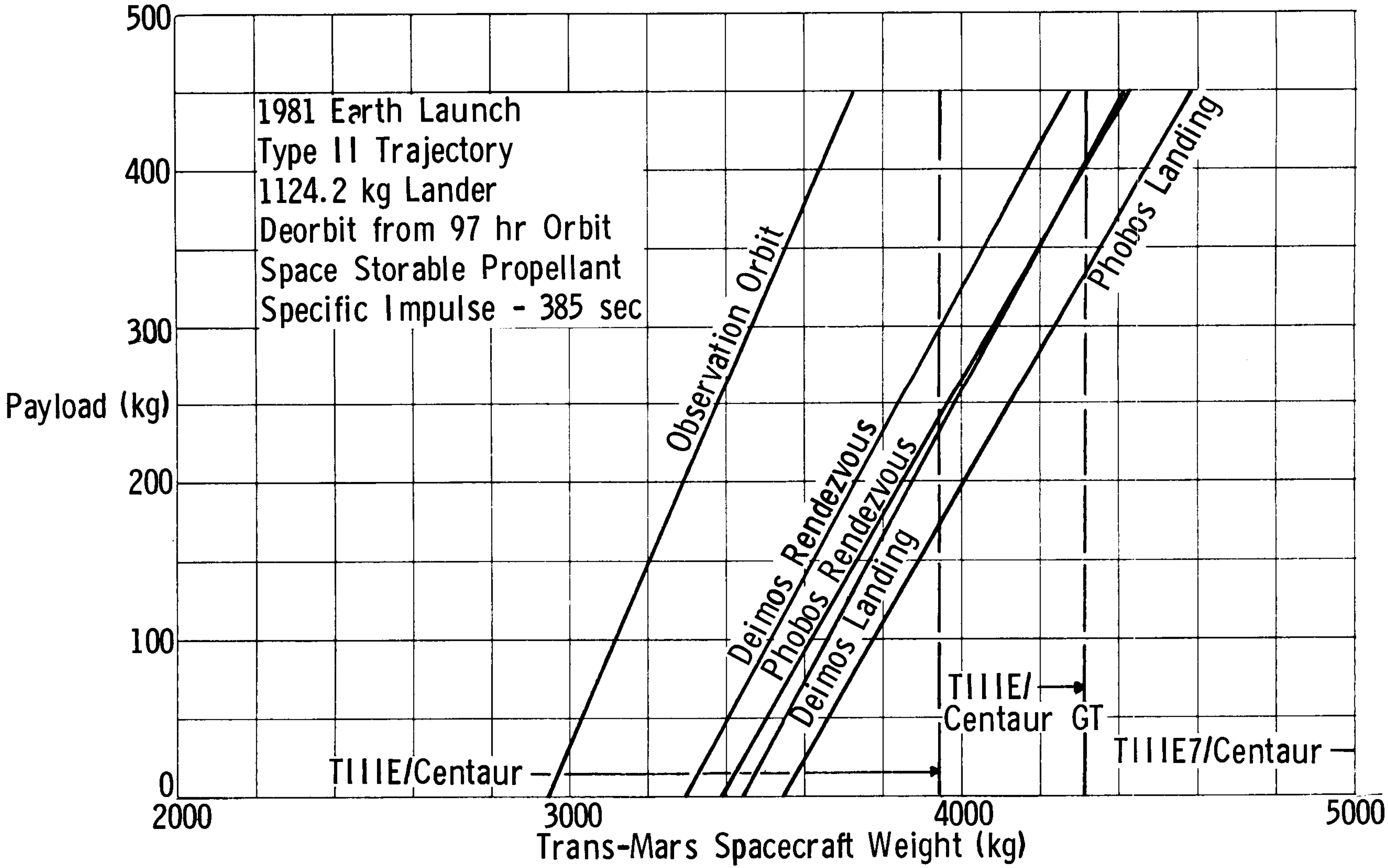


Figure II-23 Trans-Mars Spacecraft Weight vs Payload - 1981 - 97 Hour Orbit

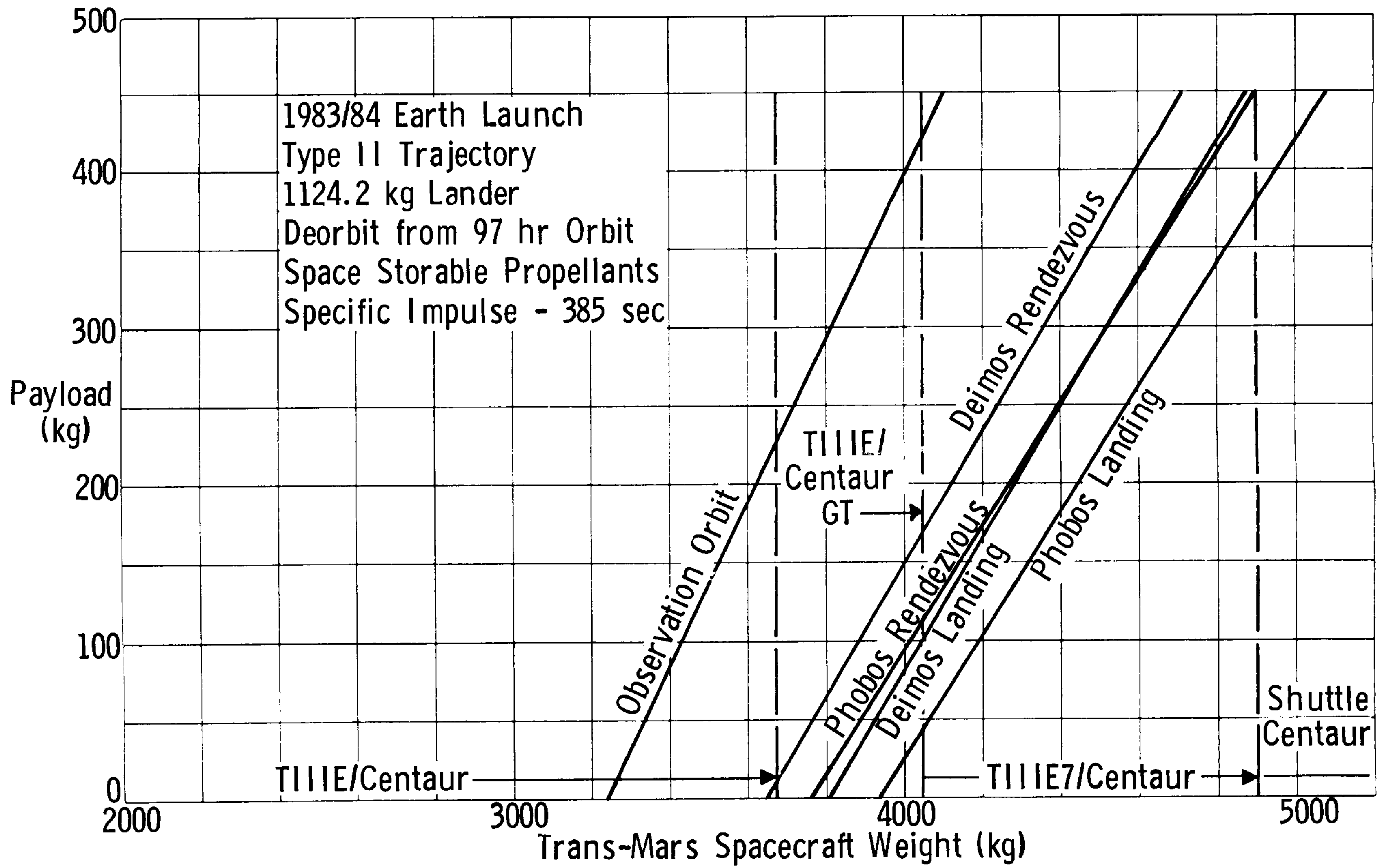


Figure II-24 Trans-Mars Spacecraft Weight vs Payload - 1983/84 - 97 Hour Orbit

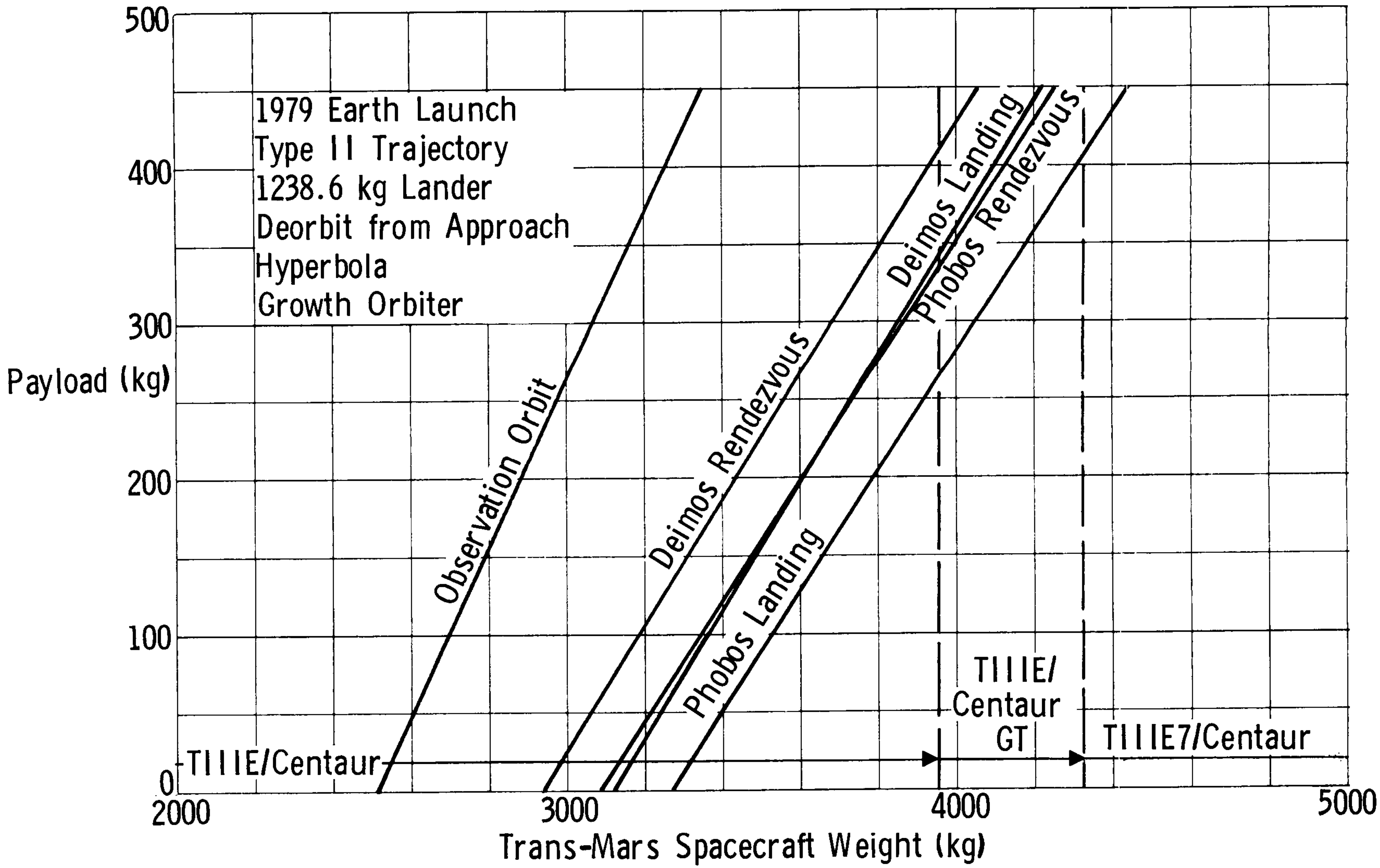


Figure II-25 Trans-Mars Spacecraft Weight vs Payload - 1979 - Direct Entry

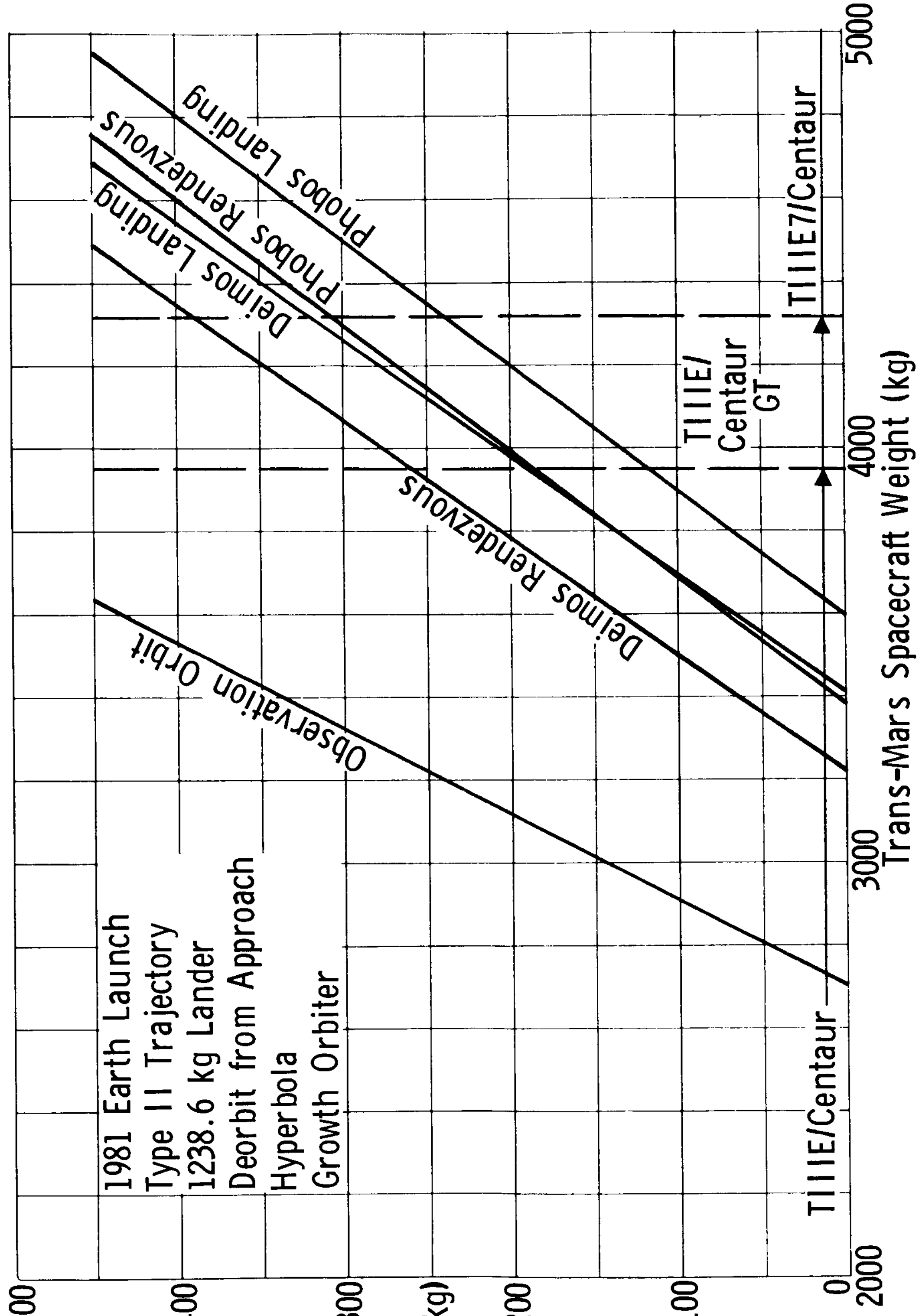


Figure II-26 Trans-Mars Spacecraft Weight vs Payload - 1981 - Direct Entry

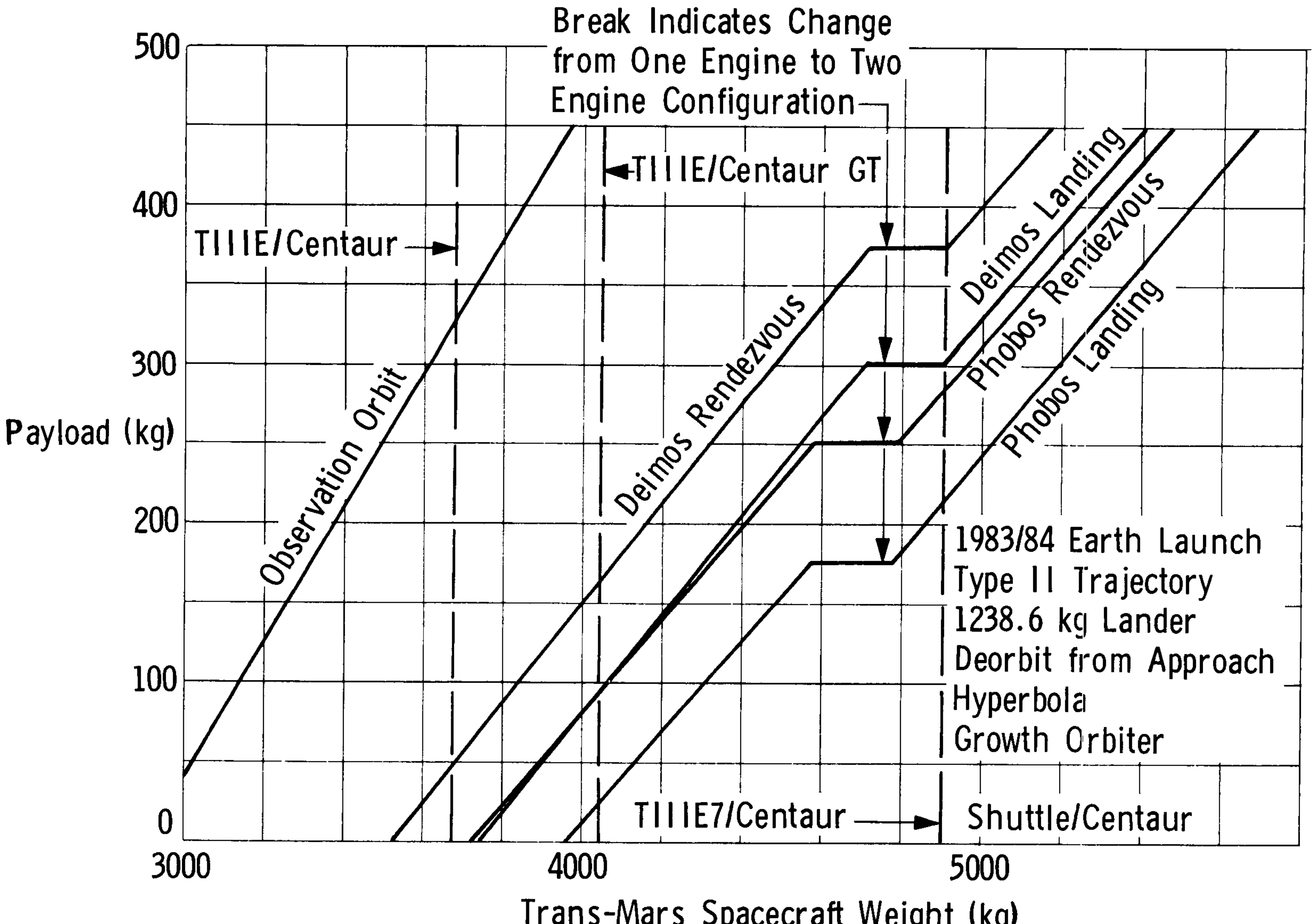


Figure II-27 Trans-Mars Spacecraft Weight vs Payload - 1983/84 - Direct Entry

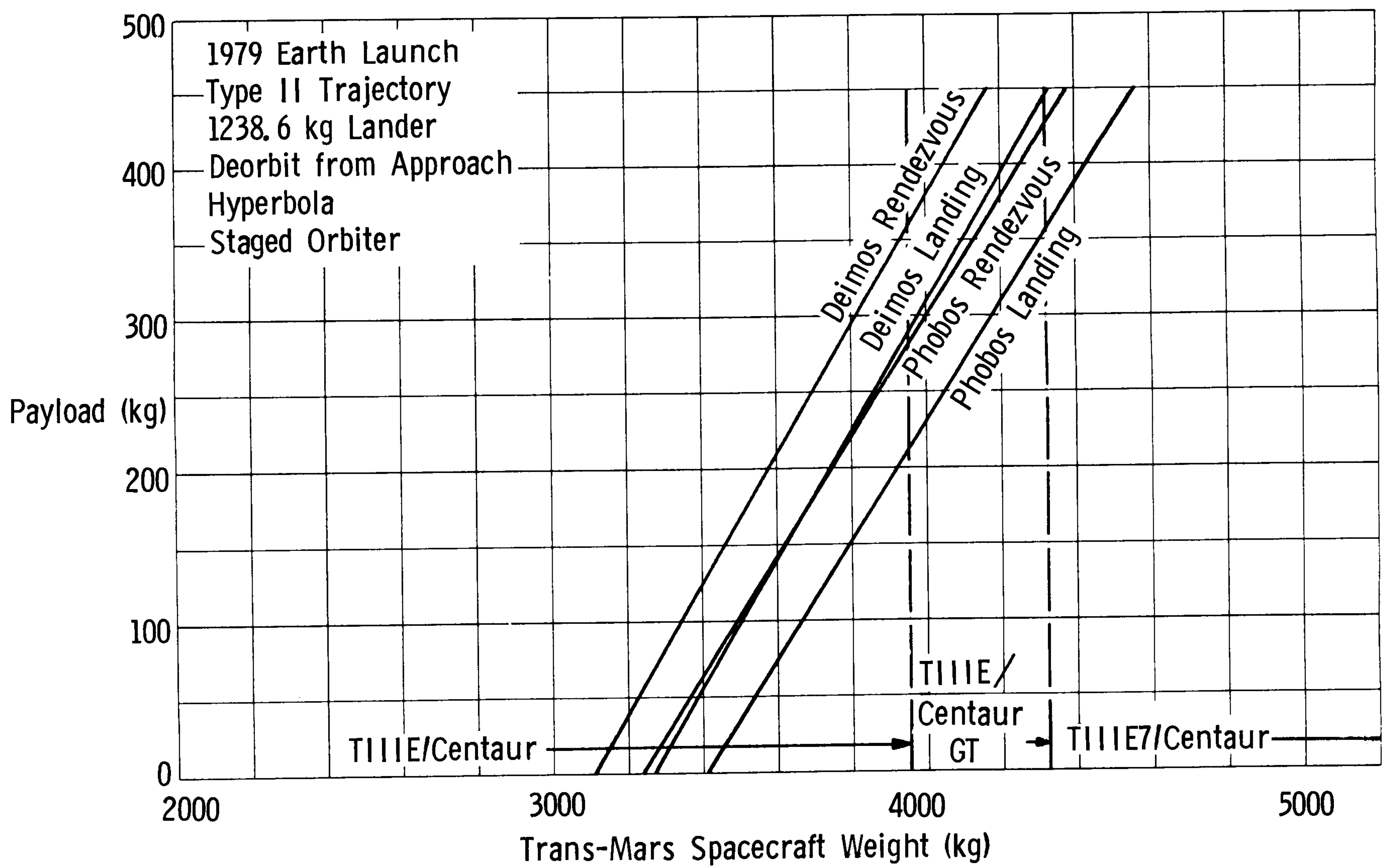


Figure II-28 Trans-Mars Spacecraft Weight vs Payload - 1979 - Direct Entry

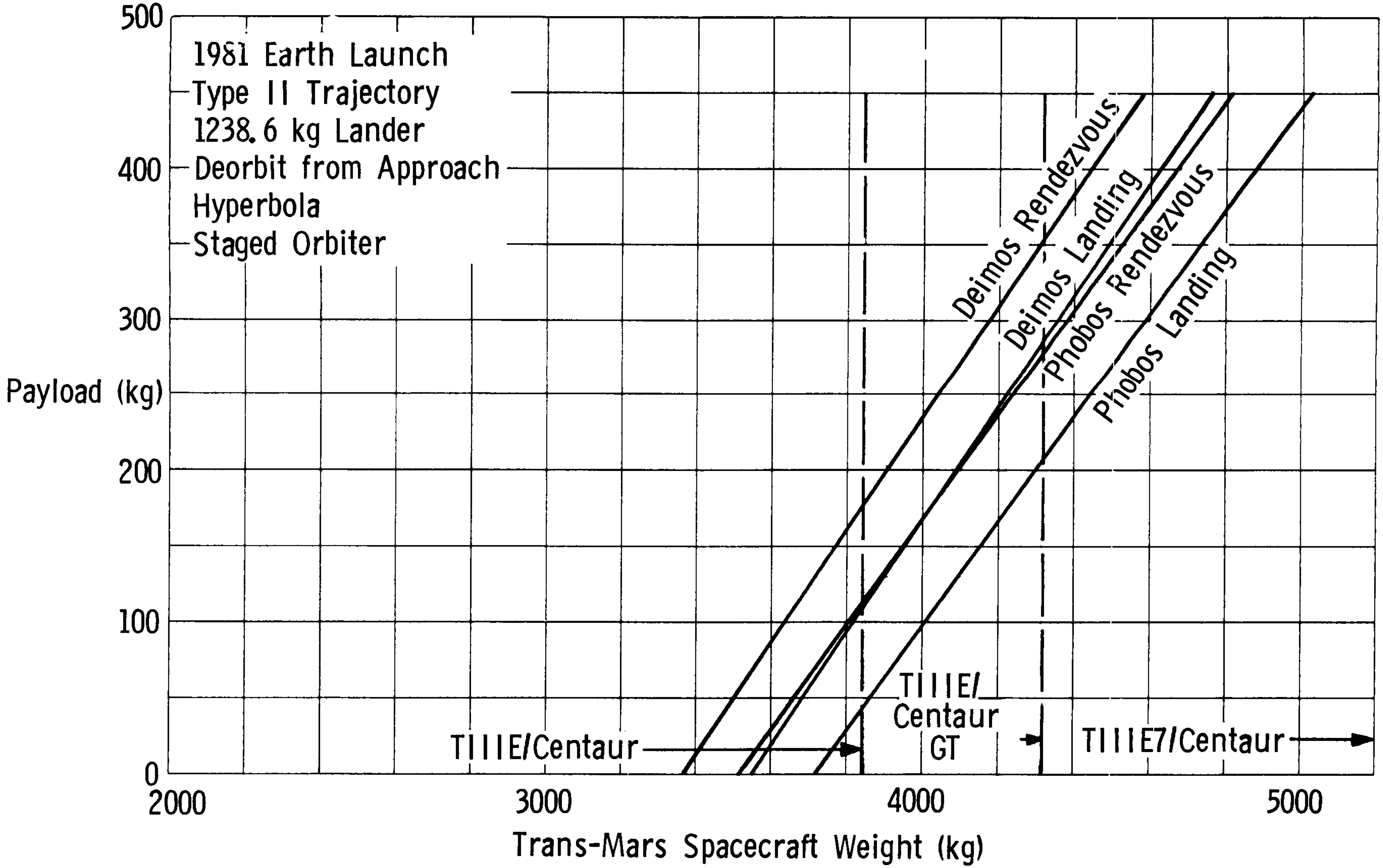


Figure II-29 Trans-Mars Spacecraft Weight vs Payload - 1981 - Direct Entry

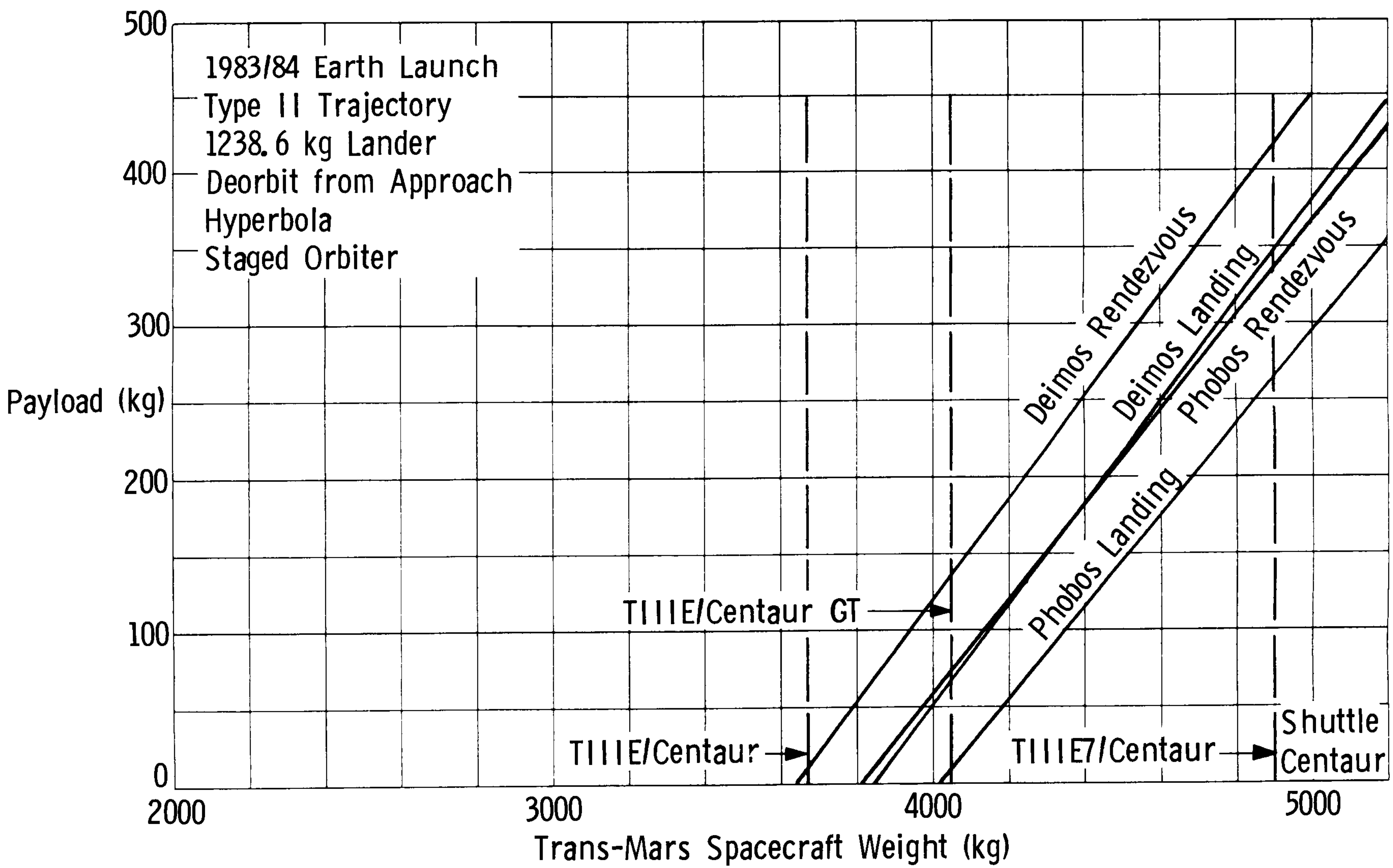


Figure II-30 Trans-Mars Spacecraft Weight vs Payload - 1983/84 - Direct Entry

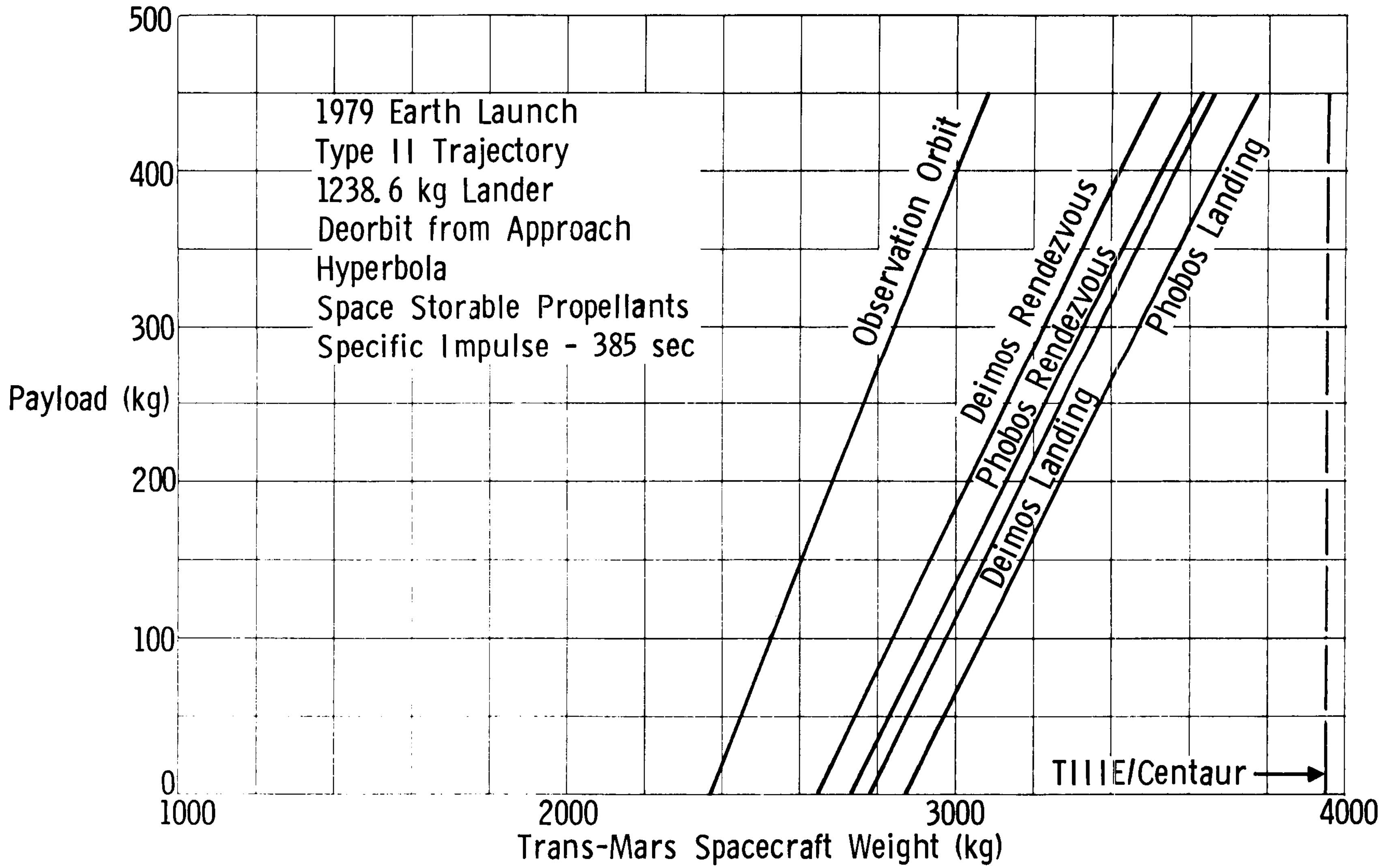


Figure II-31 Trans-Mars Spacecraft Weight vs Payload - 1979 - Direct Entry

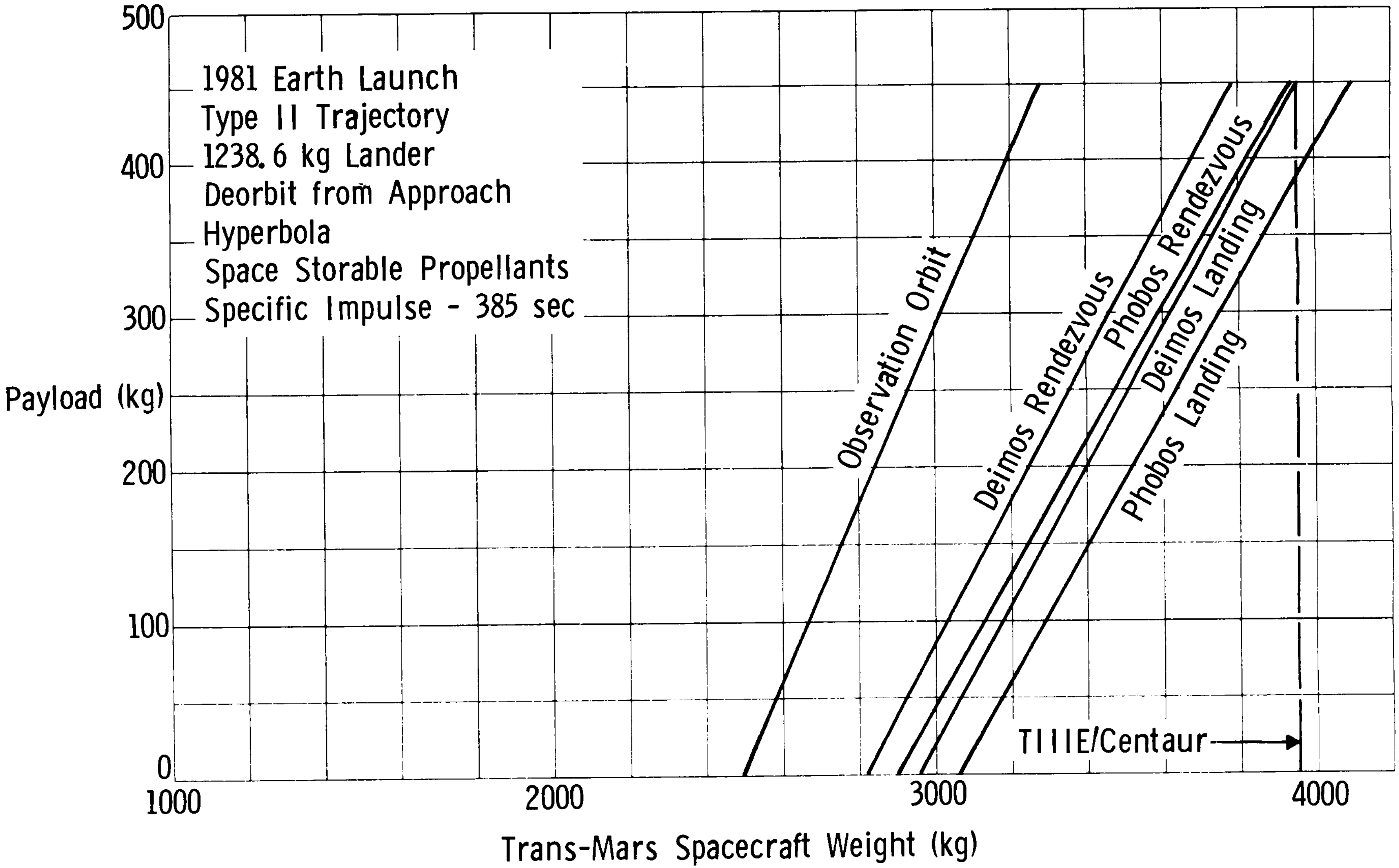


Figure II-32 Trans-Mars Spacecraft Weight vs Payload - 1981 - Direct Entry

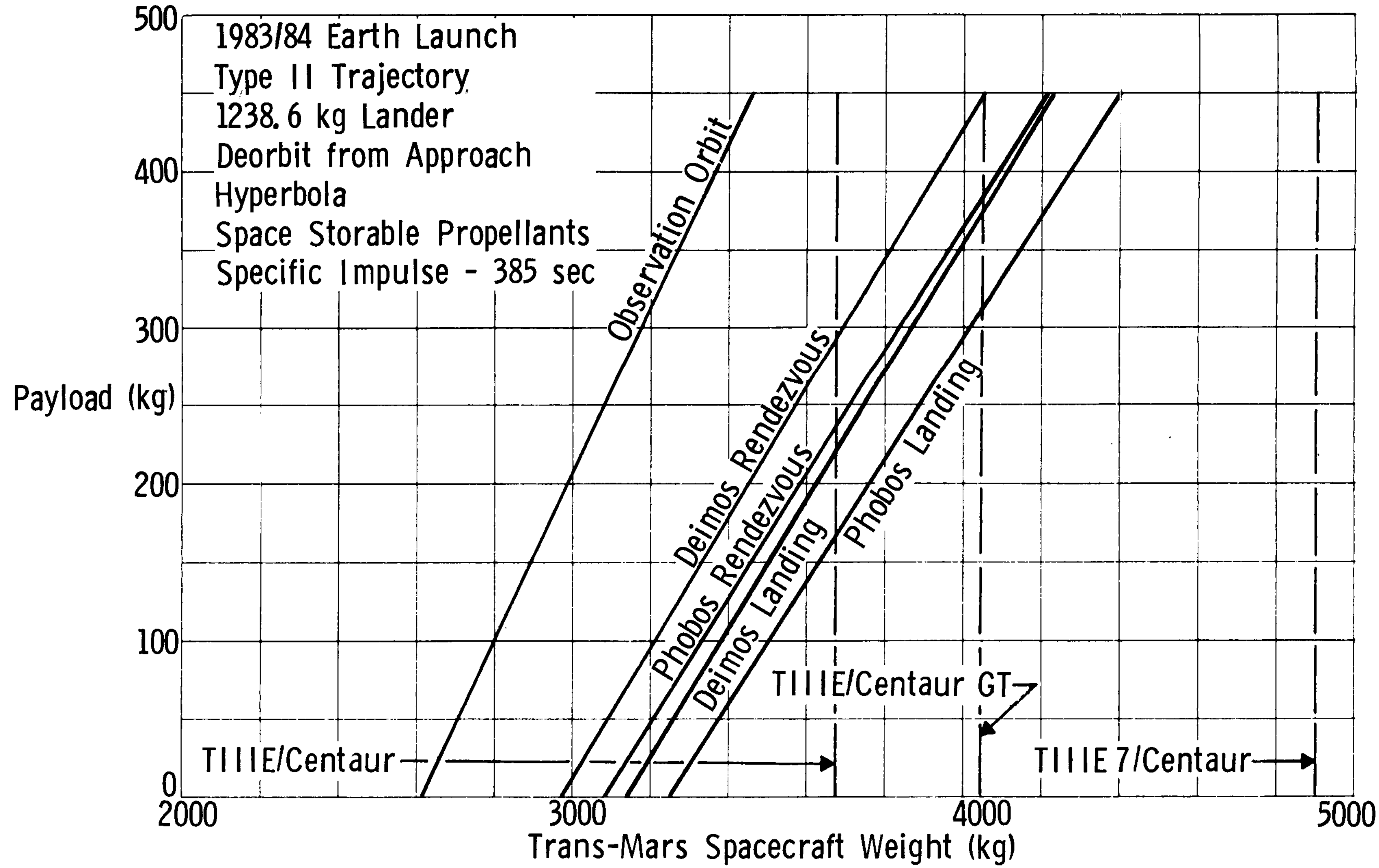


Figure II-33 Trans-Mars Spacecraft Weight vs Payload - 1983/84 - Direct Entry

Table II-6 Baseline Mission Performance Description

Event	ΔV (MPS)	Post Maneuver Weight (kg)	Time
Trans-Mars Injection		3945	October 9, 1979
Midcourse Correction	25		October 19, 1979
Midcourse Correction		3910	August 22, 1980
Mars Orbit Insertion	870	2752	September 1, 1980
Lander Separation		1629	September 8, 1980
Plane Change and Raise Periapsis	85	1530	September 10, 1980
Transfer to Phasing Orbit	300		September 16, 1980
Transfer to Observation Orbit		1395	September 18, 1980
Raise Periapsis to Phobos Altitude	200	1302	October 18, 1980
Transfer to Phobos Orbit	430	1102	October 20, 1980
Rendezvous and Landing	100	1043	October 21, 1980
Additional ΔV Requirements (gravity, steering, and navigation losses)	225		
TOTAL	2235		

Table II-7 Navigation Results

Mission	Statistical ΔV (MPS)	DCA (km)
Phobos Rendezvous or Landing	125	22
Deimos Rendezvous or Landing	100	22
15.15 Hour Observation Orbit	25	---

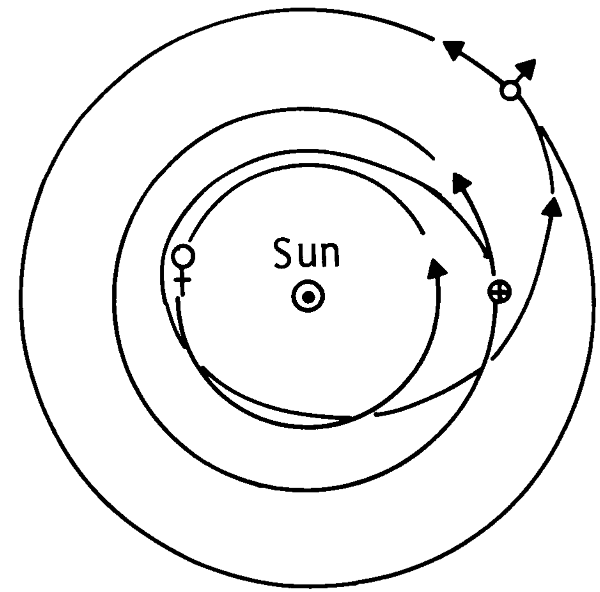
<u>Assumptions</u>

1. TV sightings at 1/10 min rate in observation orbit only.
2. Optimal consider Kalman/Schmidt filter.
3. A priori satellite errors based on astronomical data only* (600 km, 35 m/s)

* Mariner 9 has shown that pre-encounter TV sightings can reduce satellite errors to 10 km and l m/s level.

EARTH-VENUS-MARS

	1978	1980	1981	1983
• Launch Date	09-09-78	02-01-80	10-20-81	05-20-83
• C_3 $(km/sec)^2$	200	38.4	27.0	16.5
• Venus Arrival Date	04-24-79	06-11-80	01-24-82	10-07-83
• Mars Arrival Date	06-17-79	12-30-81	03-17-83	09-01-84
• VHE at Mars (km/sec)	29.1	12.4	13.1	7.8



MARS-VENUS-EARTH

- The Theoretical Lower Limit for the C_3 at Mars to Transfer to Venus (Hohmann Transfer) is 16 $(km/sec)^2$ and is Therefore Not Feasible for Sample Return Missions

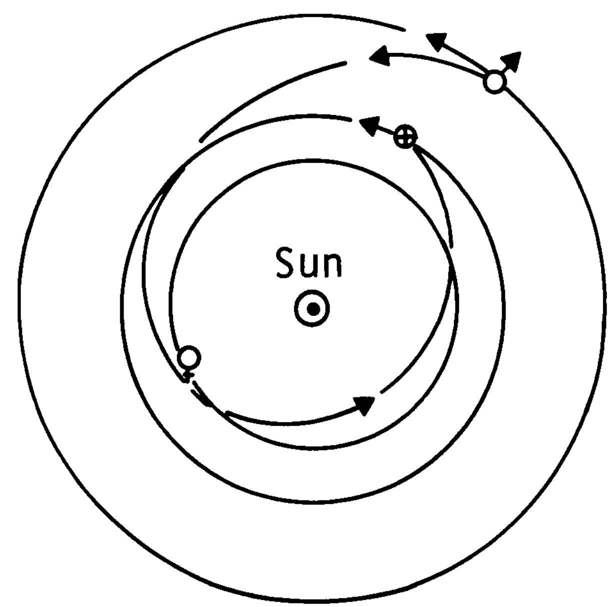


Figure II-34 Venus Swingby Trajectory Features to Mars and Mars to Earth "Venus Swingbys" require more energy than the current missions. It is very possible that using higher energy propellants, a swingby of Venus could be used to reduce the trip time for a sample return mission.

III. System Design Trade Studies

III. SYSTEM DESIGN TRADE STUDIES

A. INTRODUCTION

Several major systems level trade studies were conducted in order to properly assess the design impact of the many variables that were identified during the mission design trade studies discussed in Chapter II. The major trades that were performed are identified in Table III-1. In addition,

Table III-1 System Design Trade Studies

- Propulsion system evaluation
- Impact of mission mode selections on spacecraft subsystem design
- Cost trade studies of spacecraft configuration options
- Schedule comparison of candidate missions

several ancillary trades were also performed in support of the mission design studies. Some of the more important ones included the determination of the effect of capture orbital periods on lander total weight and the determination of lander weight as a function of various targeting strategies. This section will describe the major design trade studies that were performed, cite the most significant results of those studies, and discuss briefly the design aspects of the selected baseline mission configuration.

1. Propulsion System Evaluation

Four basic propulsion system modules were considered for the combined mission application. These were: a stretched Viking Orbiter two tank configuration; a four tank, two engine Viking Orbiter; a staged system; and a space storable propellant propulsion module. Each of the configurations was designed to land the orbiter with a satellite science module on Phobos as well as to deploy a modified Mars lander from the Mars capture orbit.

Integration layouts were prepared depicting possible packaging arrangements of a Mars lander and Phobos/Deimos payloads, and the appropriate orbiter delivery systems. These layouts were made in order to evaluate the maximum amount of growth that was possible to be made to the orbiter propulsion system and still allow a Mars lander and a reasonable Phobos/Deimos payload to be housed within the standard Viking fairing.

a. Stretched Two Tank Configuration - Design analyses were conducted to determine how much the propellant capacity of the present Viking Orbiter propulsion system could be increased without seriously impacting the design of the basic Orbiter or exceeding the design limitations of some component of the propulsion system. Our studies indicated that we could increase the propellant capacity up to 50% over the basic Viking Orbiter propellant capacity of 1404 kg before extensive design modifications were required. The results of our study were compatible with the results of earlier studies that were conducted independently by JPL in 1970.

b. Four Tank Configurations - Increasing the propellant requirements beyond 50% dictated that we utilize a four tank propulsion system module in order to maintain the spacecraft CG within acceptable limits in the launch configuration and to avoid encroaching into the launch vehicle dynamic envelope. With the change to a four tank configuration it was necessary also to add an additional engine in order not to exceed the burn time limitation of the 300 lb thrust engine.

The four tank configuration drastically changes the structure of the Orbiter by requiring a new truss system for support of the tank and to accommodate the additional engine. The location of the pressure spheres force the lander capsule to be moved further forward and the adapter truss for the lander to be redesigned. Our study results reveal that we could increase the propellant loading approximately 100% over the basic Viking Orbiter propellant capacity by utilizing this concept.

c. Staged Configurations - Staged concepts were studied in an attempt to improve the spacecraft mass fraction efficiency by staging off the major part of the tankage after the Mars insertion orbit burn. The resulting configuration approximates a Viking Orbiter propulsion system for Stage I and a Mariner '71 propulsion system for Stage II. The staged orbiter configuration requires the design of an adapter ring to provide an interface between the two stages and to support the Stage I tank trusses. The basic propellant loading assumptions used in this study were:

1) First stage propellant loading is twice the second stage propellant loading;
2) First stage propellant loadings studied varied from 1134 kg to 1814 kg.

d. Space Storable Configuration - Five space storable propellant combinations were evaluated for use in the combined missions study. The candidate combinations considered are summarized in Table III-2.

Table III-2 Candidate Space Storable Propellant Combinations

- Fluorine/Hydrazine (F_2/N_2H_4)
- Oxygen Difluoride/Diborane (OF_2/B_2H_6)
- Flox/Monomethylhydrazine ($FLOX/CN_2H_6$)
- Flox/Methane ($FLOX/CH_4$)
- Flox/Light Hydrocarbons (Ethane, Ethylene & Propane)

Of these propellants, fluorine/Hydrazine was chosen for application to the combined missions study because of propulsion system performance, operational flexibility and state of propulsion system development.

The space storable propulsion system utilizes the same two tank, one engine concept as is used for the stretched tank concepts. The fluorine and helium pressurant tanks are insulated with approximately 5.1 cm thick foam and the hydrazine tank is insulated with 1.27 cm thick aluminized mylar. In addition, appropriate solar and inner tank radiation shields are provided.

e. Propulsion Parametric Weight Equations - Based on the analyses and design studies just described, parametric weight equations were developed for each of the propulsion system concepts. These equations are summarized in Table III-3.

Table III-3 Propulsion System Inert* Weights (kg)

Propulsion Subsystem Type	Weight Equation
Growth Viking '75 Orbiter Propulsion System - Two Tank, up to 50% Growth	$223.5 + .129\ (W_p - 1404)$
Growth Viking '75 Orbiter Propulsion System - Four Tank, 50 to 100% Growth	$371.9 + .117\ (W_p - 2107)$
Staged Orbiter Propulsion System (Assumptions: $W_{P_1} = 2W_{P_2}$; First stage, propellant loading from 1134 kg to 1814 kg)	$W_{T_1} = 209.4 + .109\ (W_{P_1} - 1134)$ $W_{T_2} = 165.2 + .167\ (W_{P_2} - 566.9)$
Space Storable (F_2/N_2H_4) Propulsion System	$W_T = 54.4 + .145\ W_p$
* Inert weights also include pressurant gas and residual (trapped) propellant	

2. Impact of Mission Mode Selection on Space Subsystem Design

Several design trade studies were conducted in order to evaluate the design changes necessary to be made to the basic Viking '75 Lander to accommodate the mission mode studies conducted in Chapter II. The studies that were performed and the results of these studies are described in the subsequent sections.

a. Comparison of Lander Entry Conditions - Entry parameters used in evaluating the design changes to be made to the basic Viking '75 Lander are tabulated in Table III-4 for the two orbital periods considered and for the direct entry condition. As shown, the entry velocity for the 97-hour orbital period lander is 4831 mps (15,850 fps), or approximately 200 mps greater than the basic Viking '75 Lander, which corresponds to the 24.6-hour period column. This increase in entry velocity manifests itself in the form of increased deceleration g levels and increased maximum dynamic pressure values.

The entry corridor width and nominal entry flight path angle for the 97-hour orbital period are identical to the basic 24.6-hour orbital period for Viking '75.

The direct entry concept produces significantly greater entry velocity and much steeper entry angles. These result in increased heat inputs, larger dynamic pressures and greater peak deceleration levels. A direct comparison of the four most critical parameters is shown in Figure III-1. As shown, the peak heating rate is up by a factor of almost three, but this is still at an acceptable level. Total integrated heat load is up by a factor of about two over that of the out-of-orbit entry. Also, the peak dynamic pressure and deceleration g's are up by factors of two and one-half, and two, respectively. These greater loads reflect themselves in a 75 kg (165 pound) increase in aeroshell structure and heat shield weight over that of the baseline Viking '75.

Table III-4 Comparison of Lander Entry Conditions

	Out-of-Orbit (24.6 Hr Period)	Out-of-Orbit (97 Hr Period)	Direct Entry
Entry Velocity, MPS	4572-4625	4831	6096
Entry Corridor Width, Deg	-15° to -19°	-15° to -19°	-20° to -25°
Inertial Flight Path Angle at Entry, γ_E Nominal	-17°	-17°	-22.5°
Entry Weight, KGS	934.2	941.9	1049.0

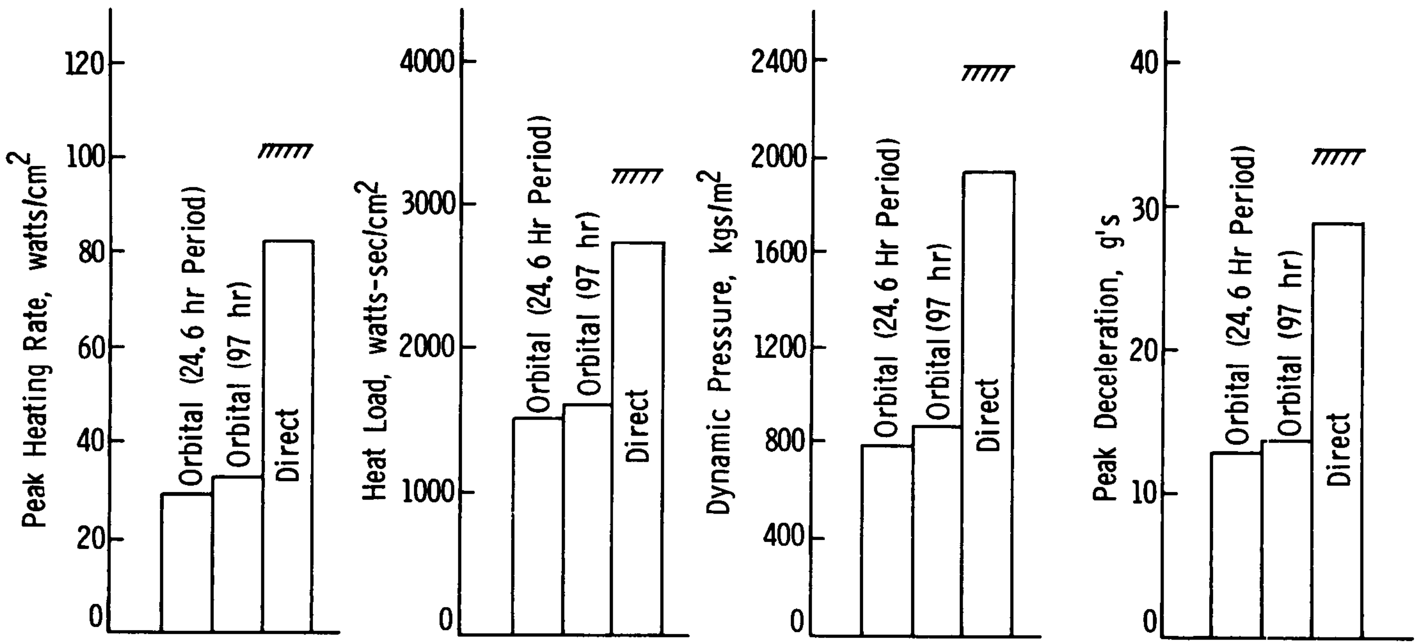


Figure III-1 Comparison of Direct and Orbital Entry Environments

The weight penalty associated with a lander landing out of a 97-hour orbital period is relatively insignificant, amounting to only 8.7 kg. The bulk of this weight is due to the additional thickness of ablator required to withstand the increased heat load.

The cross-hatched areas in Figure III-1 indicate the value of the parameters that resulted from an earlier direct entry Viking Lander study (Alternate Viking '75 Mission Mode Study, December 1970).

b. Impact of Entry Conditions on Lander Design - As discussed previously, the primary effect of the 97-hour orbital entry mode on the lander design is in the structural area, with minor design impacts in the propulsion and aerodecelerator subsystems.

The increase in total heat load for the 97-hour orbit, which dictates the required ablator thickness from 1498 watts-sec/cm^2 to 1580 watts-sec/cm^2 requires the thickness of the ablator to be increased resulting in a weight increase of 4.8 kg. In addition, because the entry weight of the 97-hour period vehicle has increased from 934.2 kg (basic Viking Lander weight) to 941.9 kg, the aerodecelerator capability must be enhanced, resulting in a weight increase of 0.7 kg. The separated and landed weight has increased 4.9 kg and 1.5 kg, respectively, necessitating an increase of 1.8 kg in usable propellants. The result of these changes reflect themselves in a total loaded weight increase of 8.7 kg for the 97-hour period lander when compared to the baseline Viking Lander.

The direct entry lander presents quite a number of serious design problems, resulting in a total loaded weight increase of some 123 kg over the baseline Viking Lander weight. This weight increase stems largely from the requirement for higher density heat shield material and ablator thickness, and the increased structural loads on entry.

Heat shield design is influenced by two basic requirements:

1) Surface recession (as a result of aerodynamic heating) and shear forces must be predictably minimized;
2) Peak aeroshell structural temperature must be limited to acceptable limits.

High surface recession rates increase the uncertainties associated with heat shield design. Surface recession can be controlled by ablator material formulation. Recession of the ablator char decreases with ablator and char density and carbon content. However, the ablator thermal efficiency is generally a decreasing function of ablator density. Optimum heat shield design involves tailoring a material with the highest thermal efficiency that exhibits minimal surface recession for the design environment. The maximum heating rate trajectory governs the heat shield material selection, while the entry trajectory with the maximum total heat load will determine the required ablator thickness.

As shown in Figure III-1 the peak convective heating rate (82.3 watts/cm^2) occurs in the maximum surface density model atmosphere at an entry angle of -25°. The maximum total convective heat load (2758 watts-sec/cm^2) occurs in the maximum density scale height model atmosphere at an entry angle of -20°.

The severity of the direct entry environment relative to the Viking baseline orbital entry environment is shown in Figure III-1. The SLA 561V and SLA 220V Viking baseline ablative materials have been tested for peak values of heating rates of 100 watts/cm^2 with minimal surface recession. This heating rate corresponds quite closely with the peak convective heating rate predicted for direct entry. However, the test pressures were an order of magnitude less than predicted for the direct entry case. Tests were recently conducted in the MMC Plasma Arc Facility in support of the Viking program's Option B direct entry study, to investigate the recession behavior of the SLA materials at combined heating rates and pressures representative of the direct entry environment. Results of these tests indicated that a modified SLA ablative formulation consisting of the addition of carbon fillers to improve the recession characteristics would be adequate for the direct entry mode. Density of the modified formulation was approximately .48 grams/cm^3 (30 lbs/ft^3).

Summarizing then, the increased heat loadings and increased structural loads on entry resulted in a weight increase of 81.2 kgs in the structural subsystem.

Lander mounted subsystem component equipment weight has also increased approximately 5% because of the 38 g (30 g limit deceleration times 1.25) deceleration (qualification) level.

As in the case of the 97-hour orbit period lander, the aerodecelerator and propulsion subsystem weights have increased, in this case by 2.5 kgs and 15.7 kgs, respectively.

A weight summary, by subsystem, for each of the lander concepts analyzed is given in Table III-5. A weight summary, by mission function (e.g., launch, separated, entry and landed weight) is presented in Table III-6 for the three lander concepts that were considered.

Midway through our Phase III studies we received our first input from the Mariner 9 spacecraft mission. The initial data indicated that the Martian atmosphere closely resembled the nominal atmosphere as defined in the Mars Engineering Design Criteria document. The basic Viking '75 Lander is presently designed to accommodate the most severe environment imposed by the five model atmospheres specified in the above referenced document. A study therefore, was initiated to evaluate the weight savings that could be realized if the lander system was designed to only the nominal atmosphere criteria. The resulting weight savings as shown in Figure III-2

Table III-5 Lander Concept Weight Comparison, kgs

	Out-of-Orbit 24.6 hr Period	Out-of-Orbit 97 hr Period	Direct Entry
Structures & Mechanical	324.3	329.1	405.5
Propulsion - Inerts	81.2	81.2	82.0
Parachute & Mortar	52.6	53.3	55.1
Pyro	11.8	11.8	11.8
Thermal Control	45.4	45.4	45.4
Power	110.2	110.2	110.2
Telemetry	21.3	21.3	21.9
Guidance & Control	64.0	64.0	67.0
Communications	34.0	34.0	35.6
Wire Harness	37.6	37.6	39.2
Science	62.6	62.6	64.8
Contingency	112.3	113.1	124.3
Dry Weight	957.3	963.6	1062.8
Residual	16.8	17.1	18.3
Pressurant	9.5	9.8	10.6
Burnout Weight	983.6	990.5	1091.7
Expendables	132.0	133.8	146.9
Loaded Weight	1115.6	1124.3	1238.6

Table III-6 Comparison of Lander Capsule Weights, kgs

	Out-of-Orbit (24.6 Hr Period)	Out-of-Orbit (97 Hr Period)	Direct Entry
Loaded Weight	1115.6	1124.3	1238.6
Separated Weight	1006.8	1011.7	1129.8
Entry Weight	934.2	941.9	1049.0
Landed Weight *	575.5	577.0	619.5

* Same Payload in Each Case

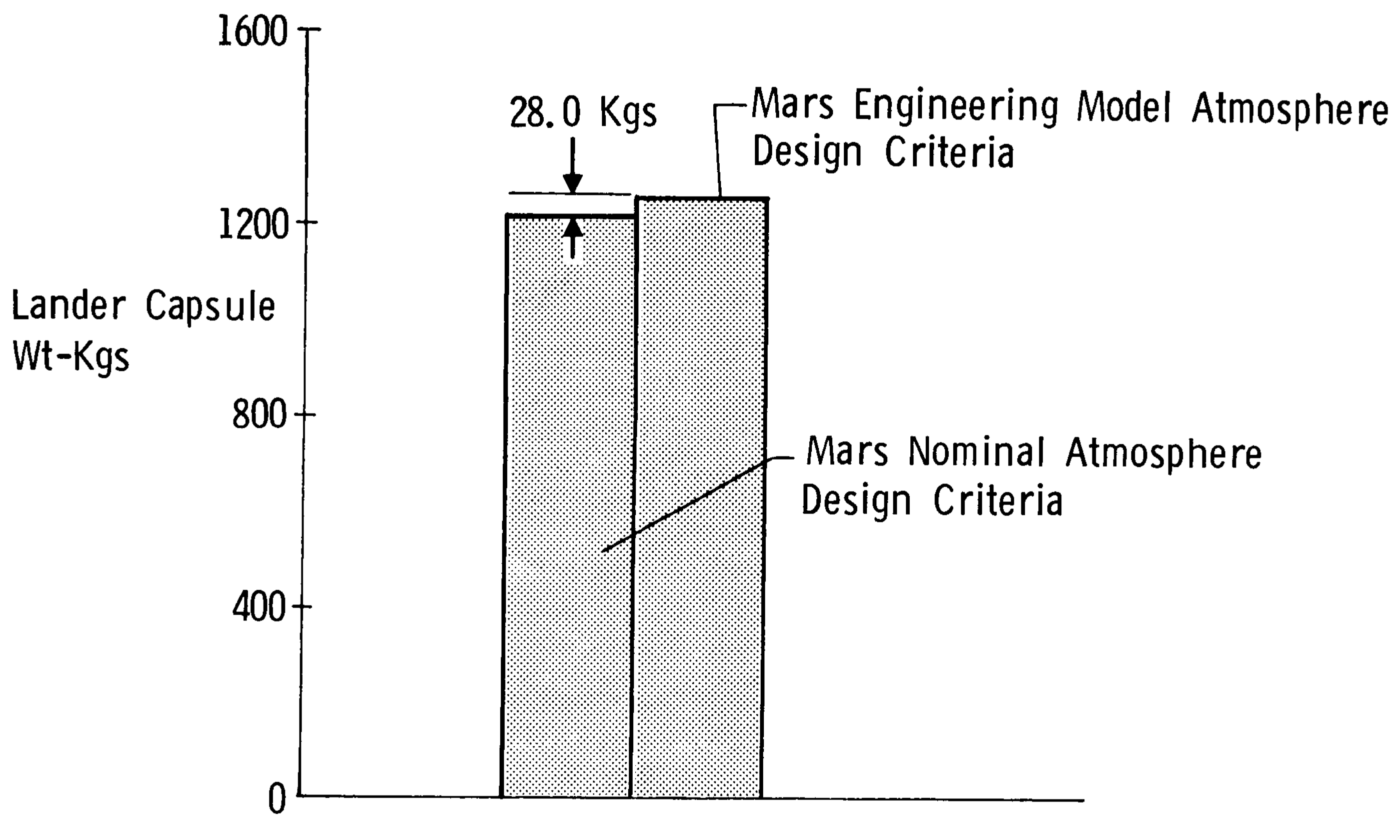


Figure III-2 Sensitivity of Lander Systems Weight to Mars Design Atmosphere Variation was found to be 28 kg for the direct entry concept. Based on this result then, and the fear that later data may invalidate this earlier finding, the decision was made to use the same criteria presently in use on the Viking Project, that is, to use the most severe environment dictated by the model atmospheres specified in the present design criteria document.

An ancillary study was also conducted to determine and evaluate the lander capsule weight as a function of nominal entry angles. Results of this study are shown in Figure III-3. It was determined that entry angles in excess of approximately -24° produced deceleration g levels and aerodynamic heating loads that required an extensive equipment requalification program as well as a different heat shield material formulation, resulting in prohibitive weight penalties. Thus, all study entry angles were limited to angles no steeper than -24°.

c. Impact of Mission Mode Selection on Orbiter Design - Orbiter inert weights for the three mission modes investigated are shown in Table III-7.

The main weight changes occur in the structures, guidance and navigation, science and communications subsystems.

The structures subsystem weight for the observation mode orbiter is essentially unchanged from the basic Viking Orbiter. The structures subsystem weight for the station-keeping mode orbiter has increased 6.0 kg due to the increased structural modifications required to be made to the orbiter bus to handle the larger propellant system modules associated with this mission mode. The 42.3 kg weight increase in the structures subsystem for the landed mode reflects the changes required to adapt the orbiter to a landed role.

The guidance and navigation weight increase of 1.1 kg for the station-keeping mode orbiter is the result of additional attitude control gas. The landed mode orbiter guidance and navigation

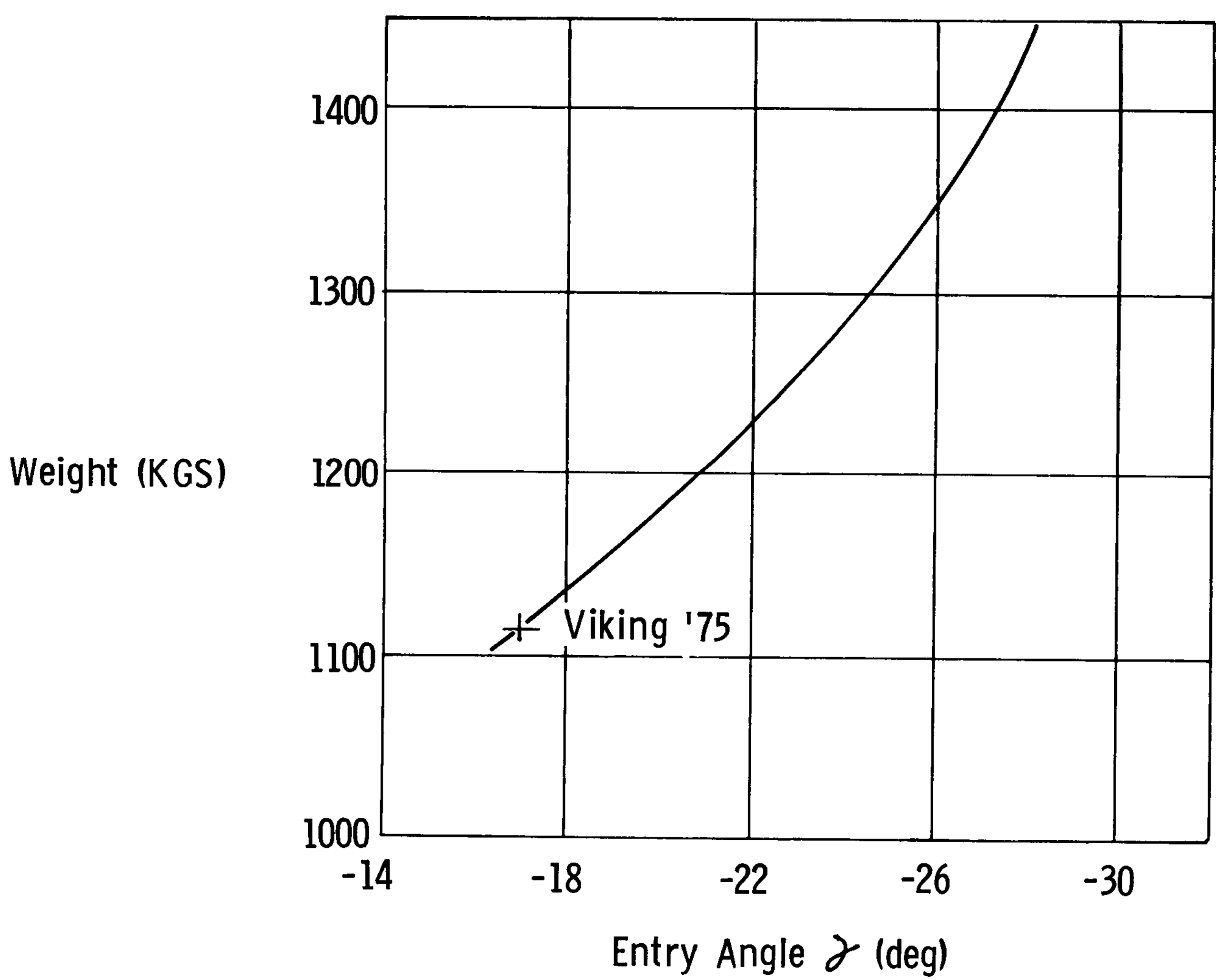


Figure III-3 Sensitivity of Lander Capsule Systems Weight to Entry Angle Variation

Table III-7 Orbiter* Concept Weight Comparison, kgs

	Landed Mode	Observation Orbit Mode	Station-Keeping Orbit Mode
Structures & Mech Devices	295.0	252.7	258.7
Communications	46.7	55.5	55.5
Power	131.6	131.6	131.6
Data Handling	45.5	45.5	45.5
Pyrotechnics	5.2	5.2	5.2
Cabling	47.7	47.7	47.7
Guidance & Navigation	87.4	75.4	76.5
Propulsion - Descent - Inert	28.9	-----	-----
Science	67.5	44.0	62.1
Orbiter Reserve	20.3	17.7	18.1
VLC Adapter	15.9	15.9	15.9
Inert Weight	791.7	691.2	716.8
Inert Weight, Less Science	724.2	647.2	654.7

*Less Primary Propulsion System Inerts

subsystem weight has increased by 12.0 kg due to increased attitude control gas and the addition of a rendezvous radar.

Science weight increases for the stationkeeping and landed mode orbiters are due to additional science instruments on board these vehicles.

The landed orbiter communications subsystem is lighter because the relay radio link has been deleted.

3. Spacecraft Configuration Evaluation

As discussed earlier, integration layouts were prepared to evaluate the potential packaging and design problems associated with the propulsion system modules, lander and orbiter concepts that were studied. Figure III-4 presents the three most attractive configurations that were developed during this exercise. Each configuration shown is designed to land the orbiter with a science module on Phobos. Modified solar panel assemblies, landing legs, and the addition of the descent propulsion system and rendezvous radar are common to all configurations. The results of this design study revealed that the two tank, stretched orbiter concept, combined with an out-of-a 97 hour orbit Mars lander represented our recommended baseline configuration. This particular configuration from strictly a design aspect, required minimum design modifications to be made to the existing Viking spacecraft (both Lander and Orbiter), while at the same time, presenting fewer and cleaner (design-wise) integration problems.

A science instrument complement was recommended for use in each of the mission modes studied. These representative complements are shown in Table III-8.

4. Cost Trade Studies of Spacecraft Configuration Options

A cost trade study was performed to determine relative cost data for the most attractive mission/system options that were

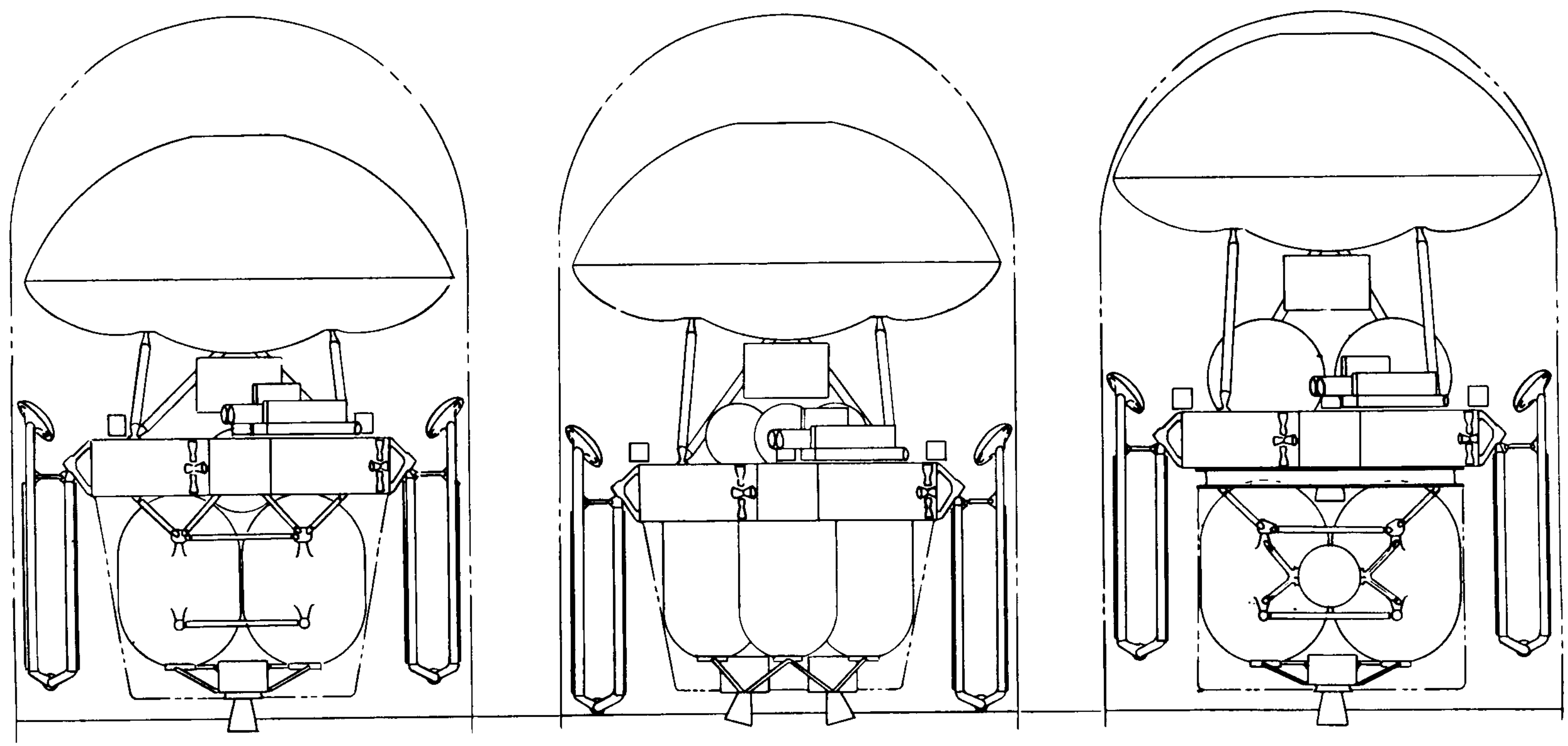
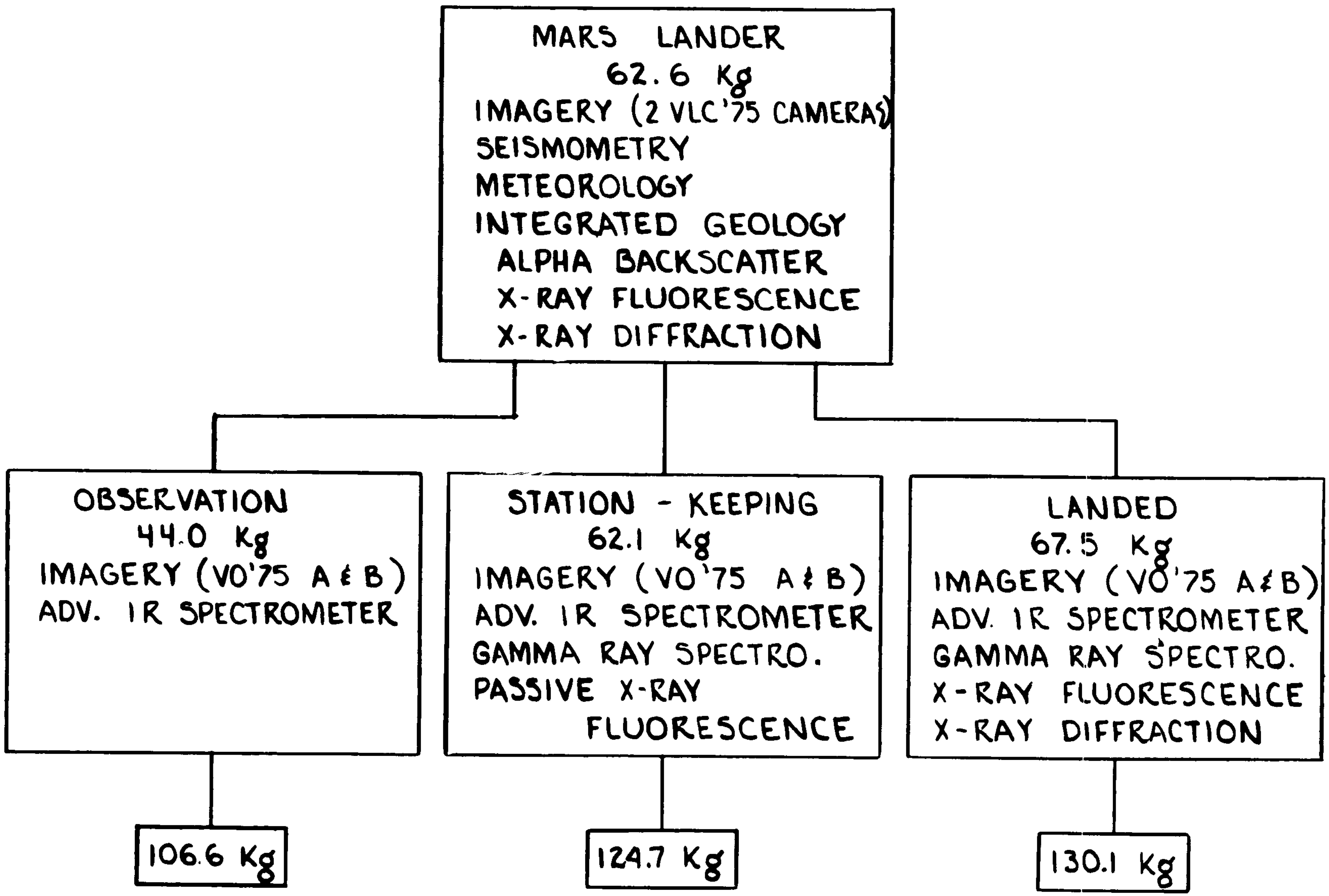


Figure III-4 Combined Missions Spacecraft Configurations

Table III-8 Nominal Science Payload Allocation



identified in the mission and system trade studies. A total of nineteen (19) mission options were examined involving twelve (12) orbiter and three Mars lander configurations.

The method employed in costing these mission options was accomplished with a parametric cost model. This cost model utilizes inputs by subsystems such as subsystem weight, power requirements, thrust levels, etc. and a subjective assessment of the percentage of new design and development work required. Additional inputs such as number of modules, number of high risk subsystems, mission time (in months) and sterilization (lander only) requirements, provide the capability to generate total program costs through the use of the parametric model.

By costing each such mission option in this manner, we have obtained a consistent application of cost factors and the generation of relative cost data which was used to determine the most cost effective mission from the mission options considered technically acceptable.

Results of this trade study are presented in Table III-9. The relative program cost data for each option was normalized to the selected baseline mission (Mars landing out-of-orbit plus Phobos/Deimos landing in 1979).

5. Schedule Comparison of Candidate Missions

A comparison of program schedules for each of the six candidate missions utilizing the 1979 launch opportunity are summarized in Figure III-5.

Program go-ahead for the Mars landing out-of-orbit mission modes can be initiated as late as mid calendar year 1975 and still support an October 1979 launch date. As can be seen from the figure, the fabrication and assembly time period is somewhat

Mission Description	Relative Cost		
	1979	1981	1983
Mars Landing Out-of-Orbit Plus			
Phobos/Deimos Observation Orbit	0.87	0.88	0.90
Phobos/Deimos Station-Keeping	0.91	1.01	1.13
Phobos/Deimos Landing	1.00	1.08	1.20
Mars Direct Entry Plus			
Phobos/Deimos Observation Orbit	0.91	0.91	0.91
Phobos/Deimos Station-Keeping	0.92	0.92	0.92
Phobos/Deimos Landing	1.04	1.04	1.05

Table III-9 Trade Studies - Comparative Program Costs Summary

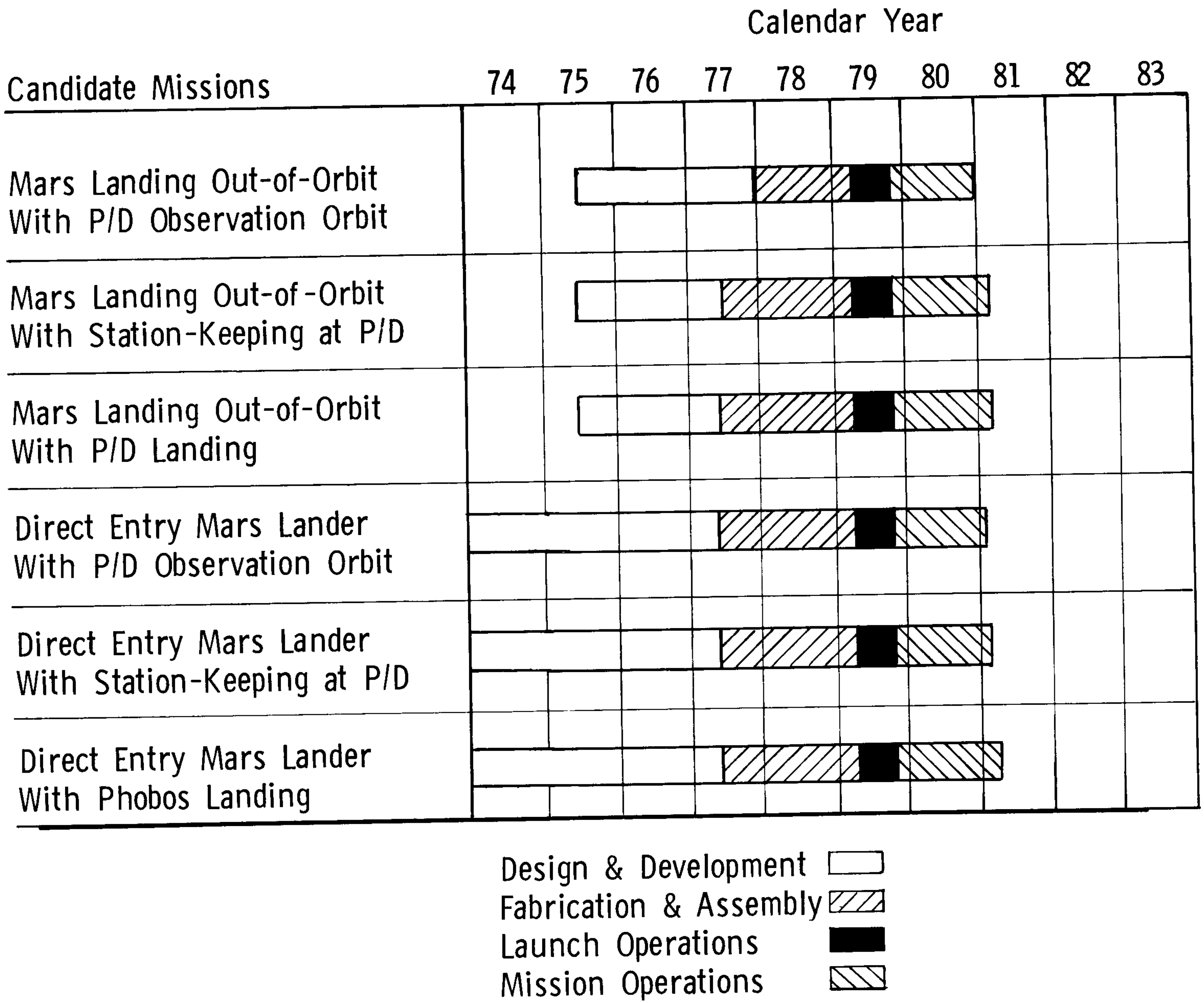


Figure III-5 Mission Mode Study Program Schedule - 1979 Launch Opportunities longer for the Mars landing out-of-orbit missions combined with the Phobos/Deimos station-keeping and landing modes, than for the observation orbit mode. This is due to the more extensive modifications which must be made to the orbiter vehicle.

Those missions employing a direct entry lander concept must be started earlier, by January 1974, to allow adequate time to accomplish the lander modifications.

Program schedules for 1981 and 1983/84 launches would be similar except for the difference in mission cruise time, approximately 8½ months for 1981, and 9 months for 1983/84, as opposed to approximately 11 months for 1979 missions.

IV. System Description

IV. SYSTEM DESCRIPTION

A. SYSTEM OVERVIEW

This section presents the overview of the selected baseline spacecraft and briefly describes the major modifications required to be made to the Mars Viking spacecraft to perform the combined Mars and Phobos/Deimos mission. Specific details of the subsystems are presented in subsequent sections of this chapter.

The 3945 kg spacecraft fits within the standard Viking fairing on the Titan IIIE/Centaur. This configuration is packaged so that the entire spacecraft is within the allowable dynamic envelope. The orbiter/launch vehicle adapter truss supports the combined missions spacecraft at four symmetrical points and is attached to the modified Viking Orbiter with ordnance operated bolts and springs. This is the spacecraft/launch vehicle separation plane. The modified Viking Lander is attached to the modified Viking Orbiter by another truss adapter. The forward end of the truss attaches to the modified lander at three symmetrical points, as on the baseline Viking. The aft end of the truss attaches to the modified orbiter, at four symmetrical points, mating with the same attachments on the orbiter as is presently utilized for the baseline Viking '75 mission. Again separation is provided by means of ordnance operated bolts and springs at the lander interface. The adapter truss remains attached to the orbiter after lander separation and is subsequently jettisoned prior to landing of the orbiter on Phobos. The launch dynamic environment is similar to that of the baseline Viking and was treated as such.

The orbiter configuration is essentially the same as that presently conceived for the Viking '75 Orbiter with some modifications made to meet the 1979 combined missions study. The most significant modifications are:

1) Addition of four integrated solar panel/landing legs
2) Addition of a hydrazine terminal descent system
3) Addition of a rendezvous radar to assist in landing operations
4) Addition of flip covers mounted over the existing Viking Orbiter thermal control louver system and addition of internally mounted phase change material
5) Addition of the geoscience instrument payload
6) Primary propulsion system growth (26% increase).

The 26% "stretch" of the orbiter propulsion capability is achieved by increasing the two propellant tanks 7.6 cm (3.0 inches) in length and 7.6 cm in diameter and by increasing the pressurization sphere 1.8 cm in diameter.

Oribter science instruments are mounted on a scan platform similar to the design used on the Viking '75 Orbiter, and in a new satellite science module which is mounted adjacent to the scan platform. The two TV cameras have been retained. The IR thermal mapper and the Mars atmospheric water detector currently on board the Viking Orbiter have been replaced by an advanced IR spectrometer and gamma mass spectrometer. The satellite science module houses the X-ray fluorescence spectrometer and X-ray diffractometer instruments.

The terminal descent propulsion system consists of a titanium fuel tank mounted within the orbiter bus structure, a series/parallel ordnance valve package, fuel filter, and four quad-thruster and solenoid valve assemblies. Each of the thruster assemblies are mounted on the side of the orbiter bus and in line with the cold gas attitude control thrusters that are mounted on the solar panel extremeties.

With the exception of the subsystems described in the foregoing discussion, all other Viking Orbiter subsystems can be used as they are presently conceived.

The Mars lander is essentially a Viking '75 Mars Lander, the only modifications required to adapt it to this mission being:

1) Increase in the aeroshell heat shield ablator thickness to be compatible with the increased heat loads imposed by the lander being deployed out of a 97 hour orbital period.
2) Minor increase in parachute design capability to handle increased entry weight
3) Slight increase in propellant loading (1.8 kg) to accommodate increased landed weight
4) Change to a geoscience payload, and
5) Increase of the thermal mass of the equipment mounting plate by the use of phase change material.

The weight breakdown for the modified Phobos/Deimos orbiter and lander is shown in Table IV-1 and Table IV-2, respectively.

The spacecraft weight buildup by systems is shown in Table IV-3. Mars Viking allocated weights are also presented to facilitate comparisons between the two missions. As can be seen, the injected payload weight for the Phobos/Deimos combined missions spacecraft is 4154 kg compared to 3664 kg for the Viking '75 spacecraft, both being within the injected weight capability of 4157 for the Titan IIIE/Centaur launch vehicle.

B. GUIDANCE AND CONTROL

1. Cruise and Orbital Phases

The cruise and orbital injection maneuvers for the combined missions will be executed the same as described in Phase I. The spacecraft executes an initial rendezvous maneuver to put the vehicle in a co-orbit with the satellite. The navigational uncertainties at the end of this initial rendezvous maneuver are

Table IV-1 1979 Recommended Baseline Orbiter Weight Statement, KG

ORBITER		
• Structures & Mech Devices	295.0	
• Communications	46.7	
• Power	131.6	
• Data Handling	45.5	
• Pyrotechnics	5.2	
• Cabling	47.7	
• Guidance & Navigation	87.4	
• Propulsion - Descent - Inert	34.8	
• Propulsion - Primary - Inert	263.7	
• Science	67.5	
• Orbiter Reserve	20.3	
• VLC Adapter	15.9	
• Dry Weight		1061.3 (2340 lb)
• Expendable Propellant	1759.6	
• Loaded Weight		2820.9 (6220 lb)
• Landed Weight		1043.0 (2300 lb)

Table IV-2 Recommended Baseline Lander Weight Statement, KG

LANDER

• Structures & Mechanisms	329.1	
• Propulsion - Inerts	81.2	
• Parachute & Mortar	53.3	
• Pyro	11.8	
• Thermal Control	45.4	
• Power	110.2	
• Telemetry	21.3	
• Guidance & Control	64.0	
• Communications	34.0	
• Wire Harness	37.6	
• Science	62.6	
• Contingency	113.1	
• Dry Weight		963.6 (2125 lb)
• Residuals	17.1	
• Pressurant	9.8	
• Burnout Weight		990.5 (2184 lb)
• Expendables	133.8	
• Loaded Weight		1124.3 (2479 lb)

Table IV-3 1979 Recommended Baseline Spacecraft Weight Summary, KG

	Baseline	Viking '75
• Lander Capsule Loaded Weight	1124.3	1115.6
• Orbiter Loaded Weight	2820.9	2340.0
• Spacecraft Loaded Weight	3945.2	3455.6
• Project Reserve	43.0	43.0
• Launch Vehicle Mission Peculiar	104.3	104.3
• V-S/C Adapter	61.2	61.2
• Injected Payload Weight	4153.7 (9159 lb)	3664.1 (8080 lb)

estimated to produce a 22 km uncertainty in the distance of closest approach. This closest approach uncertainty will not be exceeded in 99% of the cases according to Monte Carlo simulations performed in Phase II of this study. This analysis assumed the orbiter TV imaging system is used to reduce the navigational uncertainties by recording satellite images against star backgrounds for Earth based processing.

After the initial rendezvous maneuvers, the vehicle will be in a co-orbit with the satellite; where the spacecraft orbit will have the same semi-major axis as the satellite's orbit with a small eccentricity. The spacecraft will follow a small station-keeping co-orbit around the satellite.

The spacecraft orbit can be determined by using two-way tracking of the vehicle from Earth during its first 13 hours in orbit. Two orbits will be needed to determine the orbit when rendezvousing with Phobos (period = 7.65 hours) and one orbit will be needed for Deimos (period = 30.3 hours). From the spacecraft and satellite emphemeris data, the time of radar acquisition of the satellite and the spacecraft attitude needed to point at the center of the satellite at acquisition can be predicted.

2. Terminal Rendezvous and Landing Phases

Figure IV-1 shows how the terminal rendezvous and landing phases are executed. Fifteen minutes before the determined acquisition time, the vehicle is commanded to the predetermined rendezvous attitude. Sixty minutes prior to this time, the rate gyros were turned on to warm up. Five minutes before acquisition time, the rendezvous radar is turned on and should acquire the satellite when the vehicle is within radar range. The satellite should be well within the 70 degree field-of-view (FOV) of the rendezvous radar (RR) because the vehicle can be pointed to within $\pm$ 5 degrees in the large limit cycle ACS pointing mode. If the

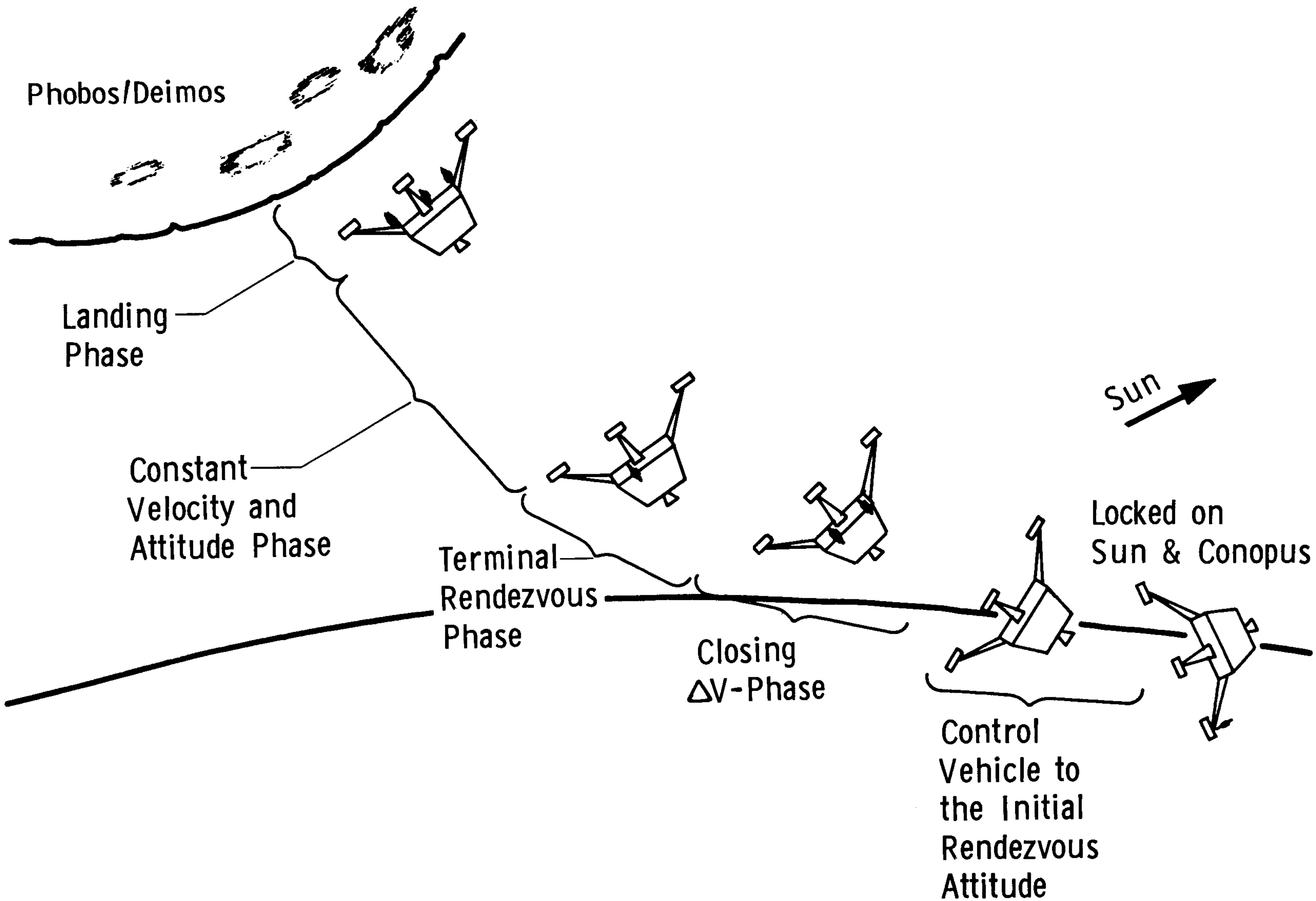


Figure IV-1 Terminal Rendezvous and Landing Phases vehicle's attitudes are trimmed by the ACS in the small limit cycle mode, the vehicle can be pointed if needed to within $\pm$ 0.25 degrees. Pointing with this accuracy is not necessary to acquire the satellite.

After rendezvous radar acquisition, the control computer initiates the terminal rendezvous and landing phases, which consists of four subphases:

a) The closing ΔV phase;
b) The terminal rendezvous phase;
c) The constant velocity and attitude phase;
d) The landing phase.

After the radar acquisition discrete is issued by the control computer, the vehicle starts executing the closing ΔV phase. During this phase, a closing velocity of 50 meters-per-second is imparted to the vehicle along the line-of-site vector by the body-mounted RCS engines. This phase is terminated when the vehicle's axial accelerometer indicates the additional velocity is reached.

During the next phase, which is the terminal rendezvous phase, the vehicle's thrust is controlled by optimum thrust control logic and the vehicle's attitudes are controlled to point along the LOS vector. The vehicle's body mounted RCS engines are used to control the vehicle during the terminal rendezvous and landing phases. The terminal rendezvous phase will be described later, when the results of a digital simulation of the rendezvous will be described. The vehicle descends to within 30 meters of the satellite during the terminal rendezvous phase. At a 30 meters altitude as indicated by the RR, the control computer issues a discrete to initiate the constant velocity and attitude phase.

During the constant velocity and attitude phase, the spacecraft descends at a constant velocity to within 2 meters of the surface and the vehicle attitudes are kept constant throughout the phase, using inertial navigation. The difference between the vehicle velocity and the satellite surface velocity can be compensated for as described in Phase I.

The body-mounted RCS engines facing upward are fired continuously during the landing phase to produce an artificial gravity and damping to the spacecraft, so the spacecraft will settle into a smooth landing. Thrusting during the landing phase is needed, because the lander would bounce under the low gravity of the satellite. During these later two phases, the vehicle would descend at a velocity of 1.5 m/sec $\pm$ 1.0 m/sec. Inertial navigation is used to guide the vehicle during the last two phases, because the rendezvous radar will operate marginally below 30 meters altitude.

Figure IV-2 shows the suggested G&C mechanization for the baseline vehicle, which is the landed orbiter for the combined missions. The G&C system is mechanized similar to the Viking Lander system to give an inertial navigation capability during the later phases of flight and during loss of one or more radar beams from the rendezvous radar. The RR is added to the existing G&C subsystem components of the Viking Orbiter (VO) to mechanize a terminal rendezvous G&C system. The existing VO control computer and sequencer (CC&S) may be marginal to handle the additional computations needed for the terminal rendezvous and landing phases. It is impossible to size the computer and sequencer until the total mission sequence is defined. Probably a slightly upgraded CC&S will be needed, which will mean adding some additional storage and computational capability. The CC&S can be upgraded easily because of its modular design. The small modifications required to upgrade the CC&S for a rendezvous system would require little additional weight, if any, so the existing CC&S weights were used in the weight statements. In addition, the cost to upgrade the present CC&S would be small.

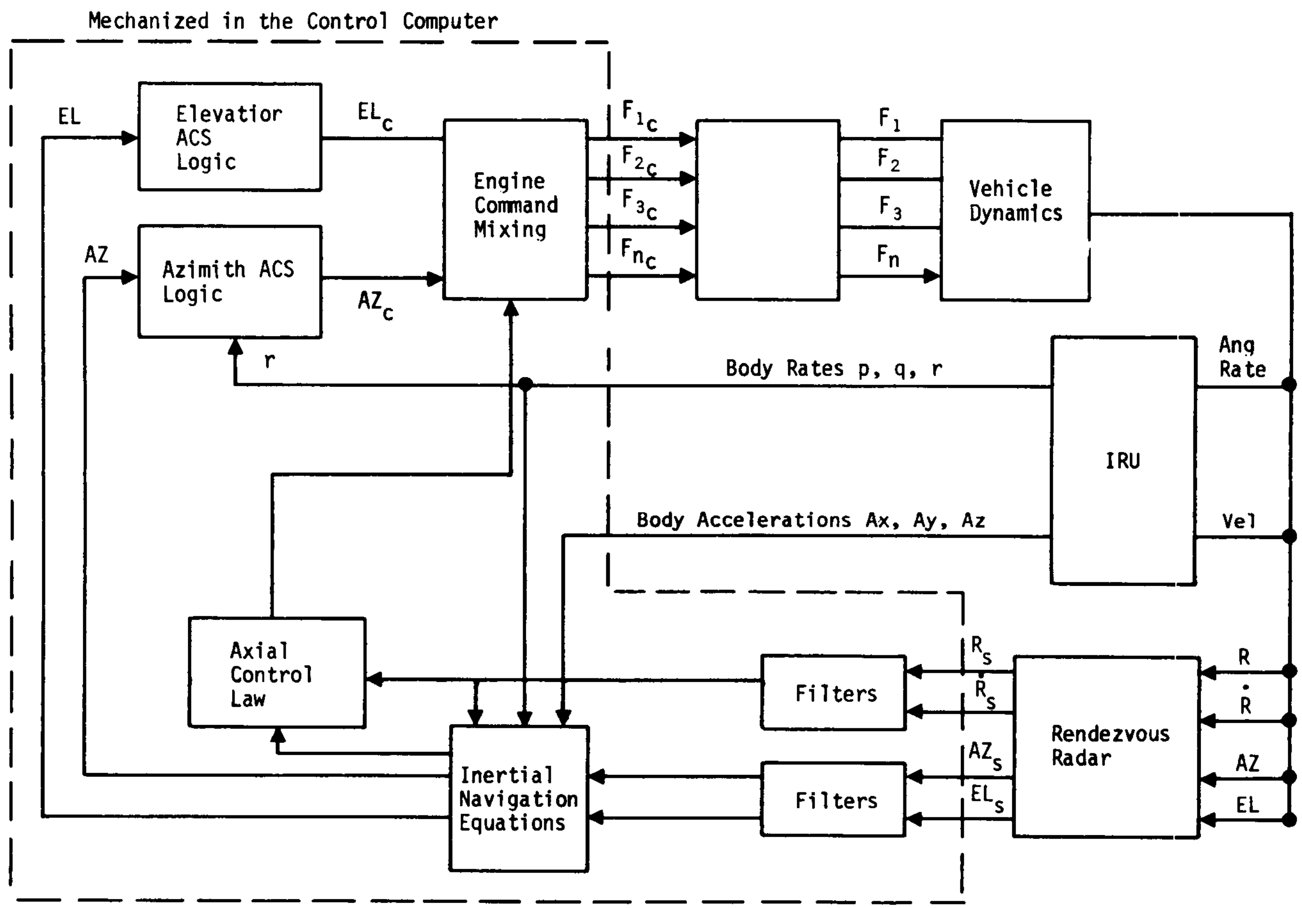


Figure IV-2 Landed Orbiter G&C Subsystem Mechanization

A rendezvous radar and two additional accelerometers are added to the VO G&C system to give the spacecraft the capability to rendezvous and land on the satellite. These components are not needed, if the vehicle does not rendezvous and land. The ACS logic, axial control laws, engine command mixing, filter equations and inertial navigation equations are mechanized in the control computer as shown in Figure IV-3. The ACS logic is similar to that used in the Viking Lander. The axial control laws will be described later where the rendezvous simulation is described.

As shown in this last figure, the vehicle dynamic motion is sensed by the RR and the IRU, which consists of a three-axis strapped down gyro and accelerometer systems. The data from these sensors are used to generate outputs u, r, w, R and $\dot{R}$ from the radar aided inertial navigator as shown below:

$$\dot{u} = A_x - qw + rv + g\,A_{13} + K_u\,(u_r - u)$$

$$\dot{v} = A_y + pw - ru + g\,A_{23} + K_v\,(v_r - v)$$

$$\dot{w} = A_z - pv + qu + g\,A_{33} + K_w\,(\dot{R} - w)$$

$$\dot{R} = A_{13}\,u + A_{23}\,v + A_{33}\,w$$

In these equations, u, v_r, and w are the body-axis velocity components; u_r, v_r are surface velocity components as determined previously and used only during the constant velocity and landing phase; p, q, r are the body attitude rates; A_x, A_y, A_z are the body acceleration components; A_{13}, A_{23}, A_{33} are the direction cosines; g is the acceleration due to gravity; K_u, K_v, K_r, K_w are adjustable gains; and $\dot{R}$ is the vehicle's range rate.

The rendezvous radar suggested for the landed orbiter is the same as used in Phase I and is described in that section. The rendezvous radar is shown in Figure IV-3 which is a 10 to 15

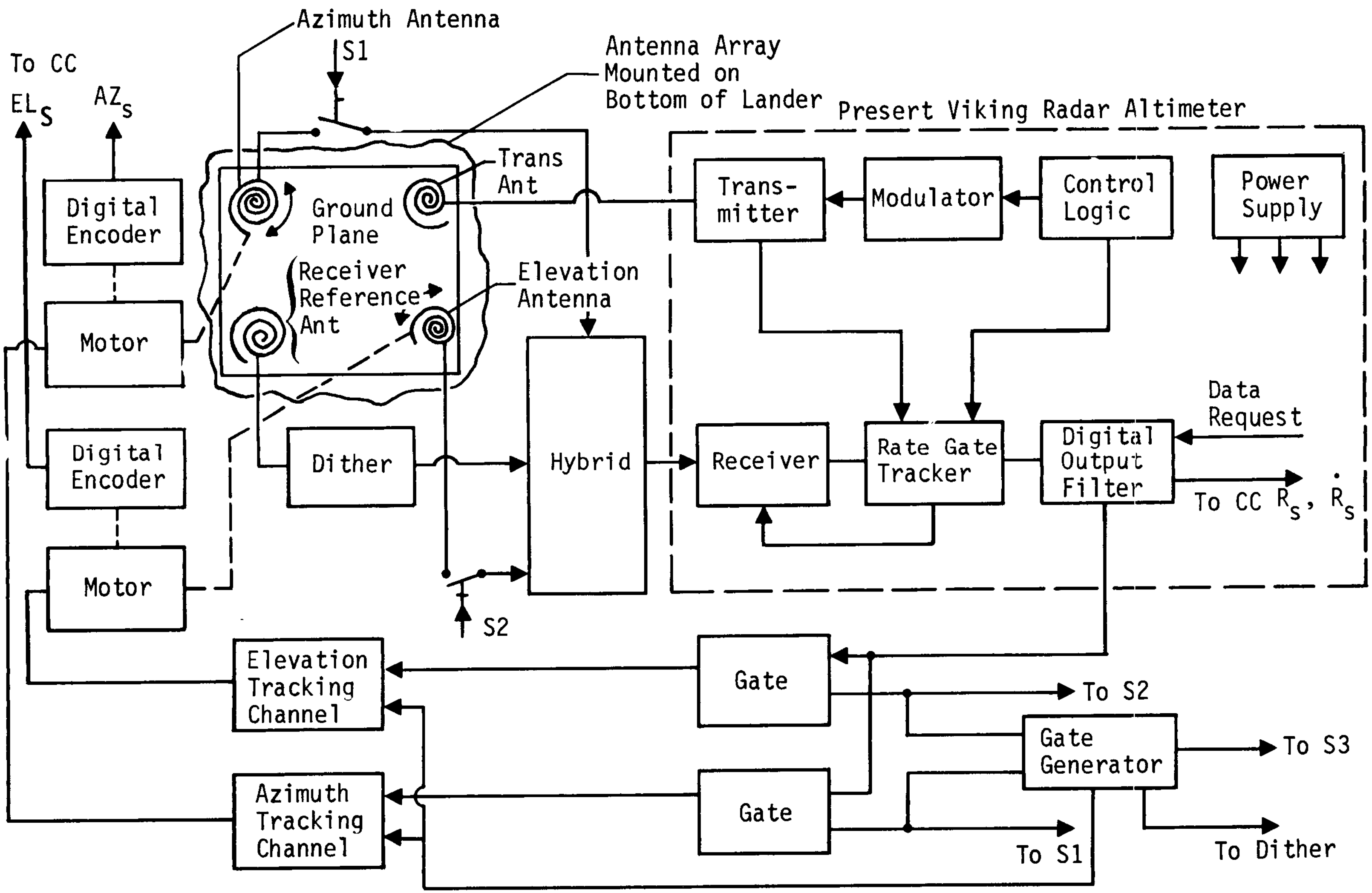


Figure IV-3 Suggested Rendezvous Radar

percent modification of the radar altimeter used on the Viking Lander. The use of an existing radar altimeter saves development time and costs. A lightweight antenna system is added to the radar altimeter to mechanize the rendezvous radar system. The recommended antenna system is four spiral wound antennas printed on epoxy boards and mounted in a ground plane on the bottom of the lander. Two of these antennas are driven with lightweight instrument type servomotors to shift the phase of the incoming signal, so the phase differences between these antennas and the receiver reference antenna can be nulled out by rotating the antennas. The line-of-sight (LOS angles are read directly from the digital encoders, when the phase difference between the two signals is nulled out.

Figure IV-4 shows a block diagram of the interferometer tracking system, and how this implementation operates as a landing site selection system at low altitudes. The figure on the right shows how the signals of each channel are compared to command the servo to null out the phase difference between the channels. The shift encoder output (θ) is proportional to the phase differences (∅) as shown by the equation on this figure.

At low altitude, the interferometer tracking system will control the vehicle so that the longitudinal axis or LOS vector will be perpendicular to the average slope within the rendezvous radar field-of-view. If the vehicle lands on the side of a hill or cliff as illustrated in the figure, the component of the thrust vector will tend to guide the vehicle off the hill or cliff and if possible to a level landing site. The vehicle is guided as it descends to keep its LOS vector perpendicular to the average surface slope within the radar's FOV, so that the vehicle will rotate into the hill or cliff.

Table IV-4 shows the RR worst case gains and losses in the radar range equation for a rendezvous radar utilizing 4.4 kw of

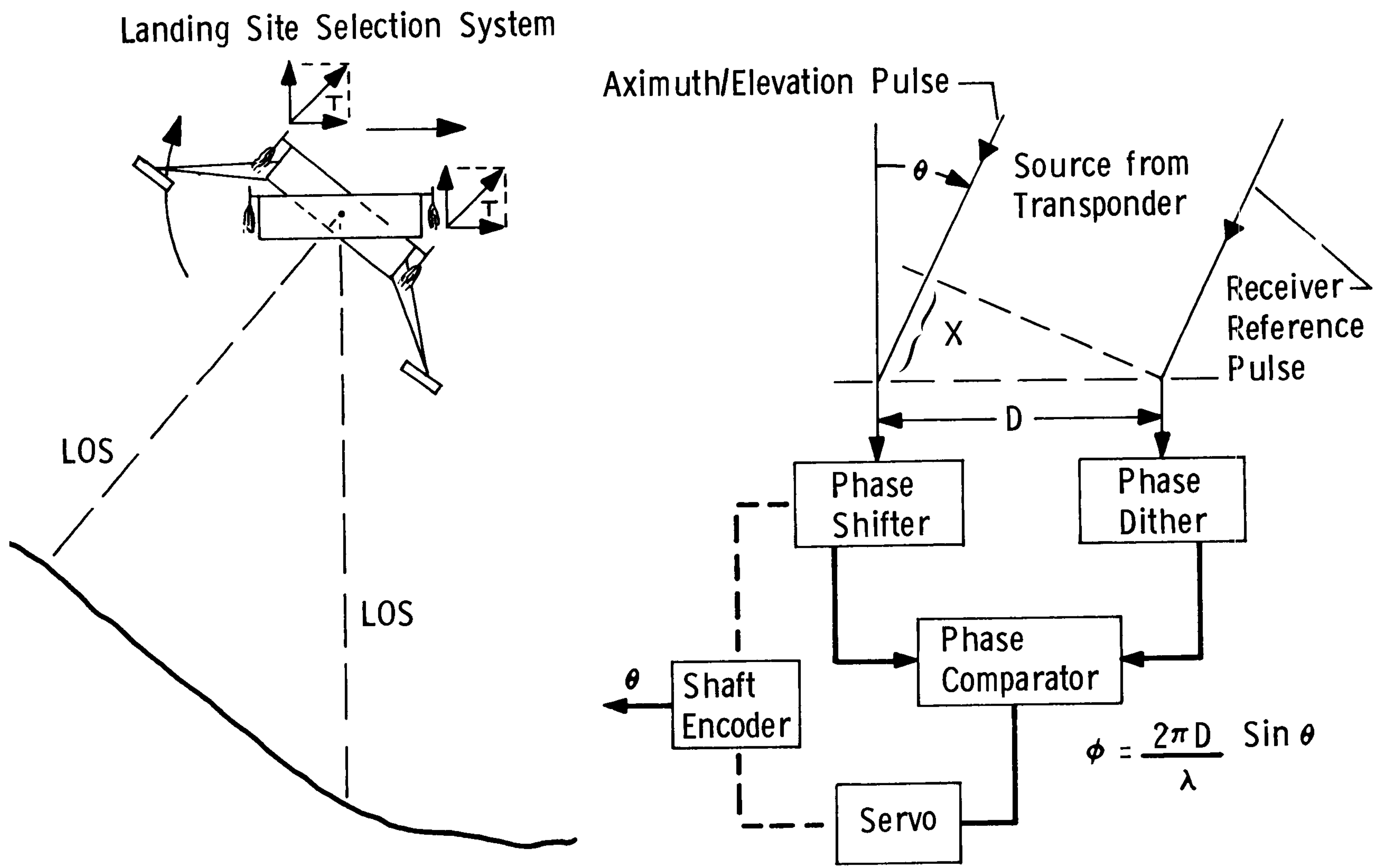


Figure IV-4 Interferometer Tracking System

Table IV-4 Rendezvous Radar Maximum Range

Freq = 1 GHZ
PRF = 256 HZ
Pulse Width = $6(10^{-6})$ Sec

$3.5 \leq \theta \leq 9.0$ (Used 3.5)
Deimos
Swerling Model V

$$R^4 = \frac{G^2 \lambda^2 \sigma L P_t}{(S/N)_{Req} (4\pi)^3 KTB(NF)}$$

	NUM	DEN
Antenna Gain (G)	+10.0 db	
Wave Length (λ)	-10.4	
Radar Cross Section (σ)-Deimos	+69.0	
Losses (L)	-10.0	
Transmitter Power (Pt)	+36.4	
$(S/N)_{Req}$		+8.0 db
$(4\pi)^3$		+33.0
KT		-204.0
B		+62.0
NF		+9.0
Numerator & Denominator Total	+95.0 db	-93.0 db
Total		+188.0

Maximum Range = 50 km
Peak Transmitter Power = 4.4 kw
Average Transmitter Power = 6.7 watts
Input Power Requirement (20% Eff) = 34 watts

peak radiated power. The RR utilizes a S-band frequency of 1 GH_z; because the Viking radar altimeter, which is a major component of the rendezvous radar, operates in this band. The pulse repetition frequency (PRF) of 256 Hz was chosen to be the same frequency as used by the radar altimeter. A surface reflectivity of 3.5 was used as the worst case. A reflectivity of 3.5 is equivalent to a surface with a talcum powder texture. A radar cross section for Deimos was used because the satellite was the smallest. The Viking radar altimeter pulse width of 6 microseconds was used. The rendezvous radar with above stated specifications would have a maximum range of 50 km and consumes 34 watts from the power supply assuming a 20 percent efficiency. This range is half that of the radar designed and recommended for Phase I. More refined navigation analyses have indicated that the satellite will be well within 50 km at time of encounter.

Table IV-5 is a weight statement for a guidance and navigation (G&N) system. Camera A, which is the Viking Orbiter wide angle television camera, is included in the weight statements because the camera is used to reduce the navigational uncertainties when the vehicle is in the observation orbit. The additional weight needed for a landed mission is also shown in this table.

A digital computer simulation of the terminal rendezvous phase was developed to study the problems associated with various rendezvous techniques. This program is described in the Phase I Appendix. A number of rendezvous algorithiums were studied to determine the best method to execute the Phobos/Deimos rendezvous. Proportional navigation rendezvous technique appeared to be a near optimum type of rendezvous and still be easy to implement in the control computer.

Figure IV-5 shows thrust control curves used in this scheme to turn the RCS engines on or off. This figure shows the optimum thrust control curves for the landed orbiter configuration used in

Table IV-5 Landed Orbiter Guidance and Navigation Description

	Weight, Kgs	Added Wt For Landed Orbiter
Attitude Control Components	36.2 (79.8 lb)	
Electronics Assys		
IRU		
Reaction Control Assy		
Canopus Tracker		
Sun Sensors		
Stray Light Sensors		
Computer Command	18.1 (40.0 lb)	
CC&S Memory, Output, Processor and Power Supply		
Camera A	12.2 (27.0 lb)	
Rendezvous Radar	10.4 (23.0 lb)	+10.4
Nitrogen Gas	10.0 (22.0 lb)	+10.0
Additional Accelerometers (2)	0.4 (1.0 lb)	+ 0.4
Total	87.3 (192.8 lb)	+20.8

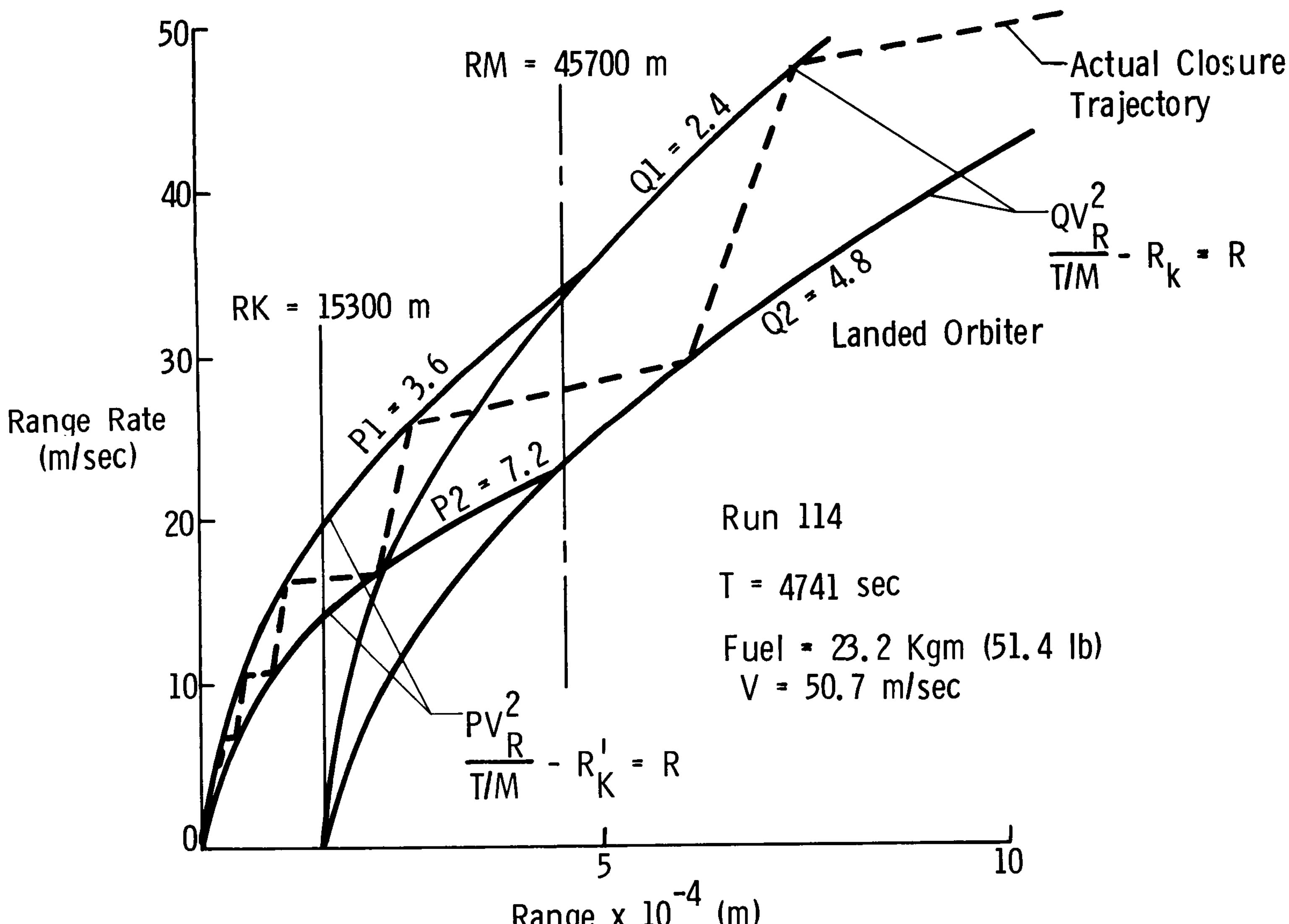


Figure IV-5 Rendezvous Control Curves

the rendezvous radar thrust control algorithm in the control computer. The dotted line represents the range vs range rate trajectory during the terminal rendezvous. The control curves are represented by solid lines. The upper control curves are thrust-on curves, which turn the RCS engines on and have control gains of Q1 and P1 depending on whether the spacecraft is above or below the control gain change altitude RM respectively. The lower control curves are thrust-off curves, which turn the RCS engines off and have control gains Q2 and P2 depending whether the spacecraft is above or below the altitude RM. The equations of the control curves are shown on this figure, where P and Q are the control gains and R_K is the control curves asymtotic altitude. The altitude R_K for the higher altitudes is 15300 meters and for lower altitudes is 20 meters. Eight thrusting periods are needed for the spacecraft to rendezvous with the satellite. The terminal rendezvous takes 4741 seconds and uses 23.2 kg of propellant. A near optimum rendezvous was achieved as only 1 kgm more fuel was used than the most optimum case where the vehicle executes a two impulse rendezvous.

Figure IV-6 shows the in-orbit and out-of-orbit rendezvous trajectory for a landed orbiter rendezvous with Deimos. The spacecraft was 18 km out the Deimos orbit when the vehicle started its rendezvous with the satellite. The thrusting periods are also shown as well as when the thrusting periods are initiated. The out-of-orbit trajectory shows how the vehicle would rendezvous as viewed in the orbital plane. The in-orbit trajectory shows how the vehicle would rendezvous as viewed from a position out of the orbital plane.

Figure IV-7 shows how the range, range rate, and line-of-sight (LOS) rate varies during the rendezvous. The times of thrusting are shown on the range rate profiles. As can be seen by this figure, the total line-of-sight rate, which is the vector sum of the elevation

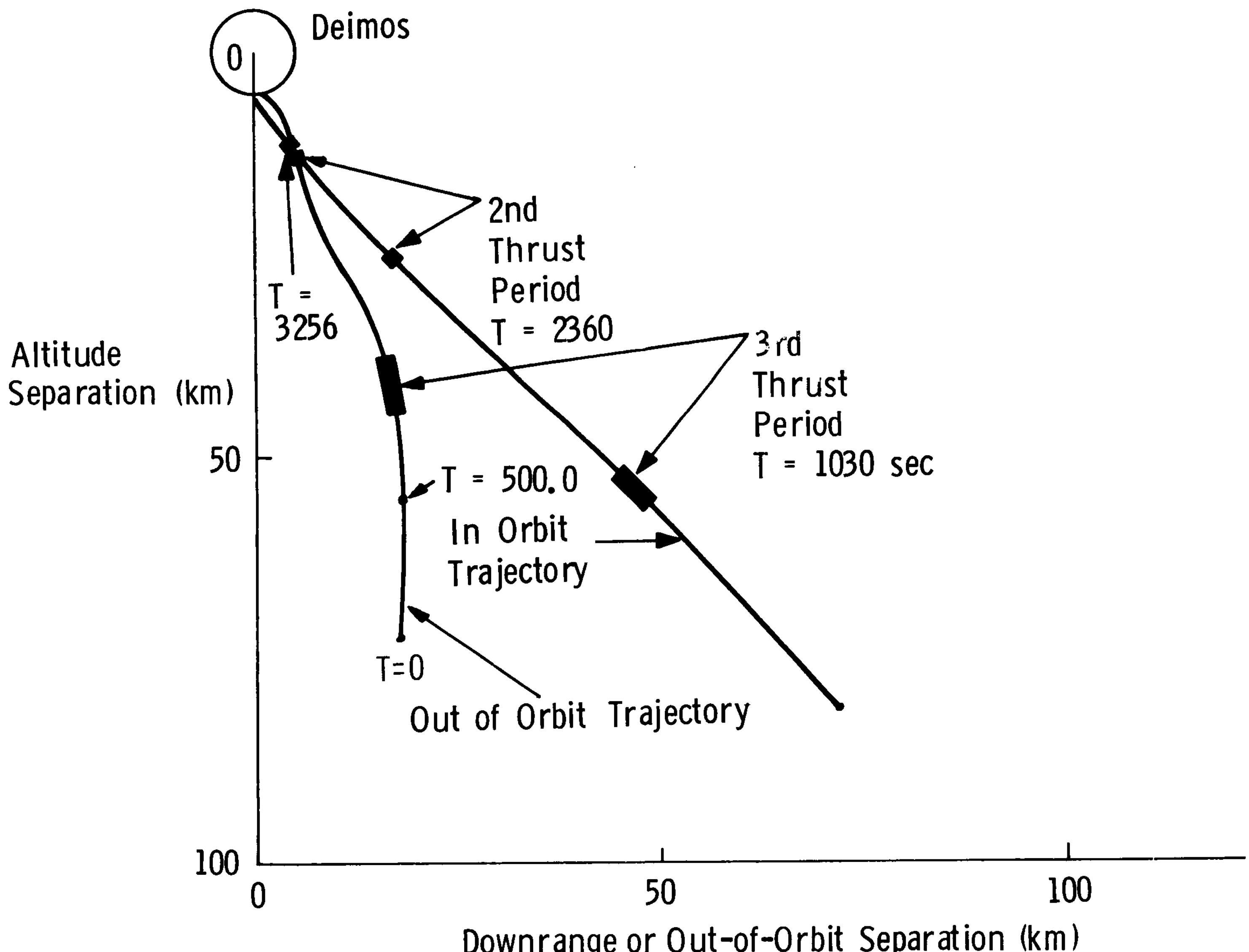


Figure IV-6 Rendezvous Trajectory - Landed Orbiter

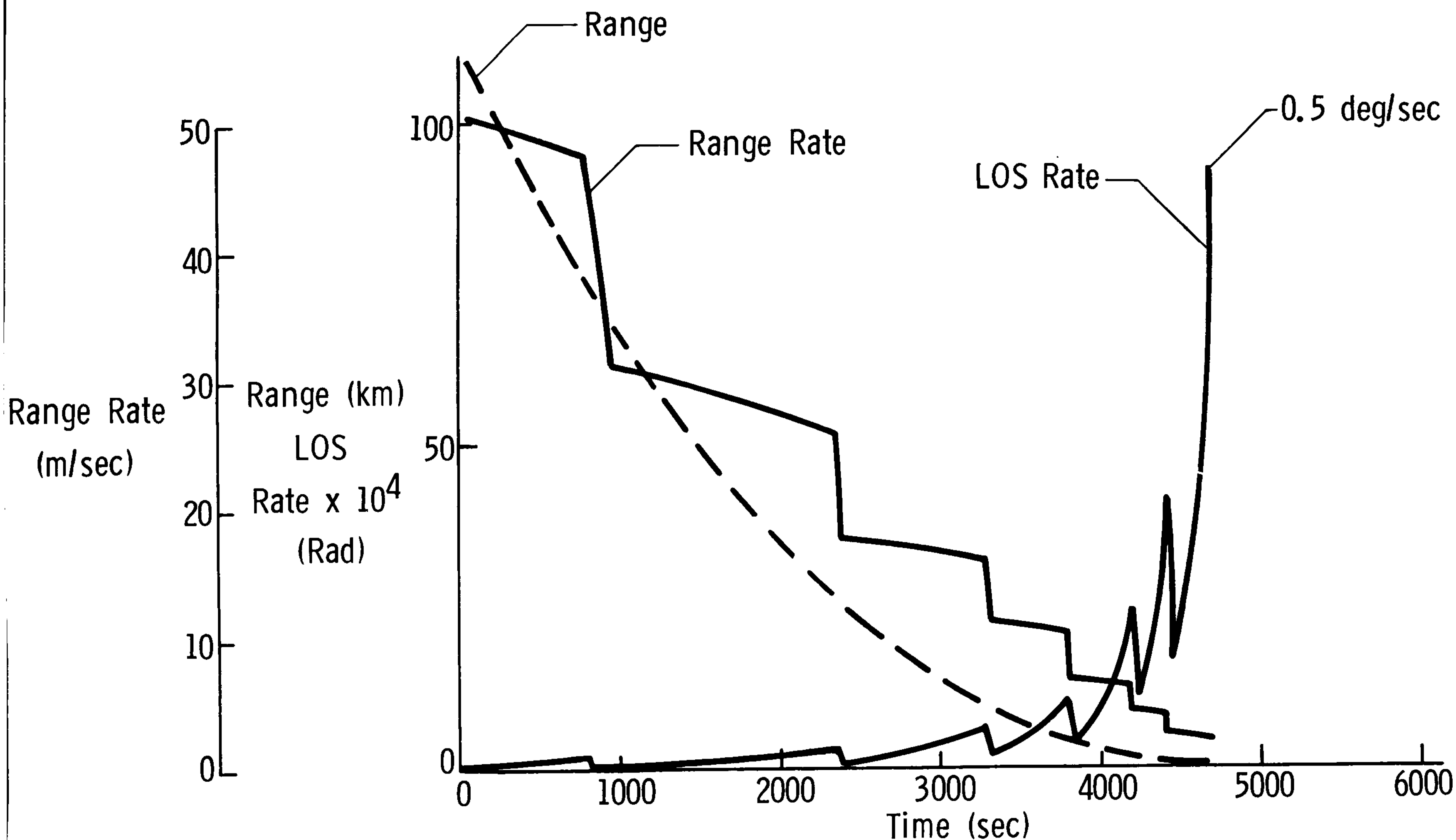


Figure IV-7 Rendezvous Time Profile - Landed Orbiter and azimuth rates, increases throughout the flight. If the LOS rates can be kept low, a more optimum rendezvous can be achieved. At the end of rendezvous, the maximum rate during the flight was only 0.5 degrees/second; which was about 20% of the rates achieved by the separable lander in Phase I. This is why the landed orbiter executes a more optimum rendezvous than the separable lander.

C. STRUCTURAL DESIGN

1. Design Approach

Structural design and systems installation for the spacecraft and launch vehicle adapters were studied in detail to define the necessary revisions and additions to the baseline Mars Viking spacecraft to perform the combined Mars landing and Phobos/Deimos landing mission.

The structural configuration selected for the baseline combined missions spacecraft is a modified Viking '75 Orbiter, a modified Viking '75 Lander and associated truss adapters. All structural members utilize state-of-the-art processes and materials to ensure high reliability and to minimize costs.

The addition of the "growth" propulsion system module to the orbiter plus reducing the height of the launch vehicle adapter serves to provide a lower center of mass for the spacecraft relative to the launch vehicle (12.2 cm closer).

The injected weight of the combined missions spacecraft is now 4154 kgs compared to the injected weight of 3664 kg of the Mars Viking '75 spacecraft. Thus, the structural loadings have increased somewhat but these are partially offset by the reduction in bending moment resulting from the lower center of mass. Structural design and systems installation of the spacecraft and adapter trusses were studied in order to identify the impact of this additional loading. The study results indicated the need for heavier adapter truss members and minor modifications to the orbiter bus and truss components, as well as minor changes required to be made to the lander. A summary of the major modifications to the orbiter and lander is shown in Figure IV-8.

a. Modified Orbiter - The increased Phase III loadings required that four vertical stiffeners be added to the orbiter side beams at the four attachment points of the lander/orbiter adapter truss. These stiffeners are required in order to transfer the vertical load in the truss members (higher member loads because of increased lander weight) directly to the corresponding orbiter truss member. This approach to stiffening the orbiter bus by using "add-on" members rather than redesigning the existing structure minimizes the impact of the modification. The orbiter truss members, in turn, need to be structurally stiffened to accommodate the increased loads This is accomplished by increasing the wall thickness of each tubular truss member.

The propulsion module truss members, which have been increased in diameter and length, but to a lesser degree than in Phase I or II, to accommodate the increased (26%) propellant loading, tie into the orbiter lower ring structure at the same points at which the adapter truss members attach, thus transferring loads to the adapter in a direct load path. Local "beef-up" of the orbiter lower ring structure is necessary to handle the increased weight of the propulsion module.

As in Phase II, the orbiter science instruments are mounted on the scan platform and in a new satellite science module, which is mounted adjacent to the scan platform.

Four landing legs, similar to the Phase II concept, are provided. These solar panel/landing legs pick up the same "hard"

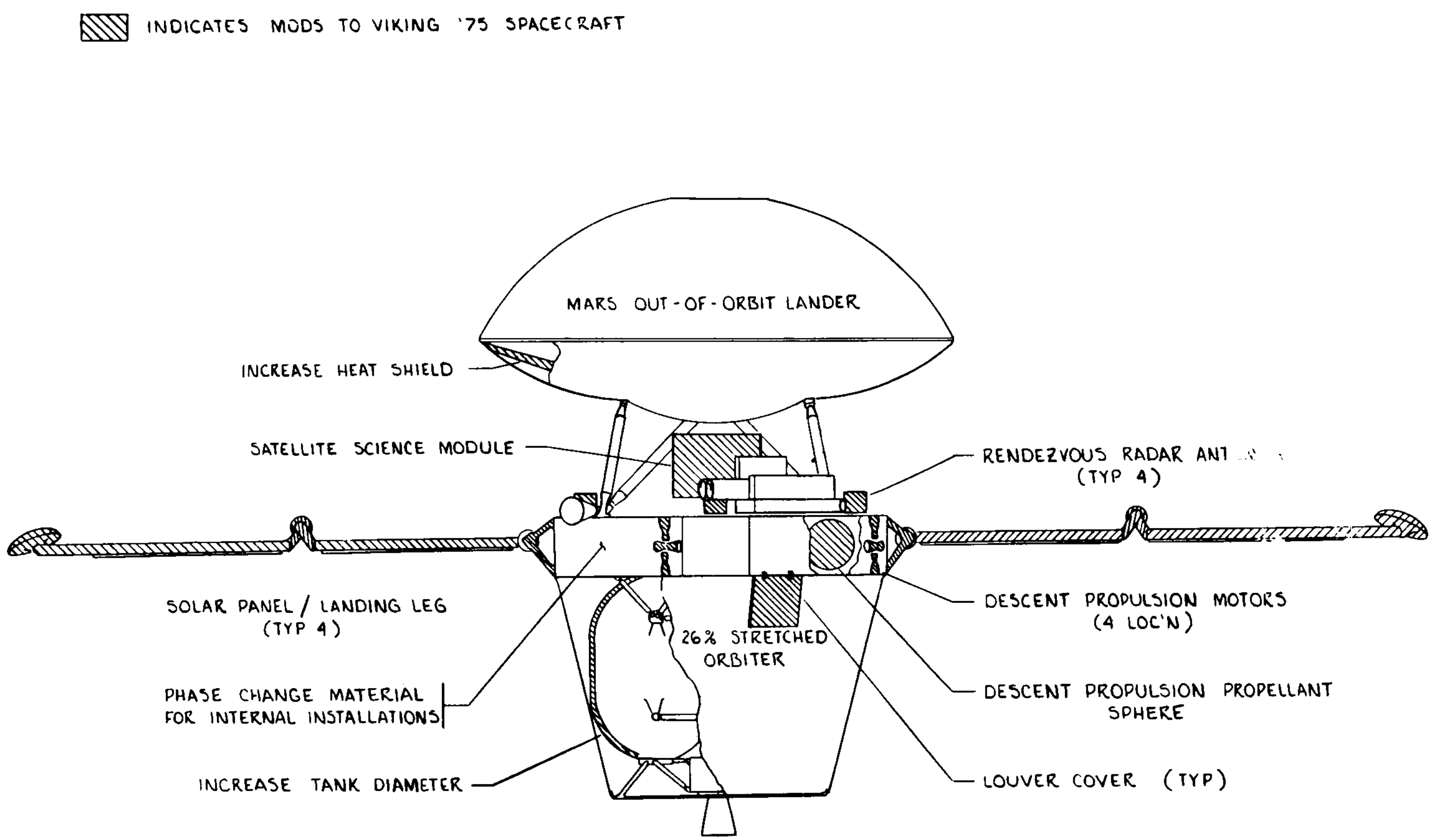


Figure IV-8 Combined Missions Baseline Spacecraft Modifications points on the orbiter bus that formerly served to support the four outriggers that attached the fan-like array of four solar panels to the Viking Orbiter bus. Since the landing dynamics program indicated that the total leg load is somewhat less than 68 kg at impact, no structural modifications are required to be made to the basic bus structure to handle this load.

Provision has been made to incorporate the terminal descent propulsion tank into the basic propulsion module. Hard points have been provided on the bus side beams to accommodate the terminal descent thruster assemblies.

b. Modified Lander - Structurally the lander body will remain essentially unchanged from the basic Viking '75 Lander.

Higher heat loads have caused an increase in the heatshield ablator thickness. The change to a geoscience payload, which is composed of the two Viking '75 facismile cameras, an advanced seismometry experiment, meteorology and an integrated geology experiment are arranged within the lander body as shown in Figure IV-9 . The arrangement of these components are such that an L/D of approximately 0.20 is maintained.

To maintain adequate lander equipment temperatures during the landed portion of the mission, phase change material has been added to the equipment mounting plate to increase its thermal mass to prevent day time equipment overheating.

c. Lander/Orbiter Adapter Truss - The lander/orbiter adapter truss, configuration-wise is the same as the Viking '75 adapter truss. However, because of the increased loading the individual truss members have had to be structurally stiffened and are therefore heavier. Attachment of this truss to the lander at one end and to the orbiter at the other is accomplished in a similar fashion as that employed for the Viking '75 spacecraft.

d. Centaur-Spacecraft Adapter - Preliminary stress analysis indicated that the present design is adequate to handle the Phase III loading.

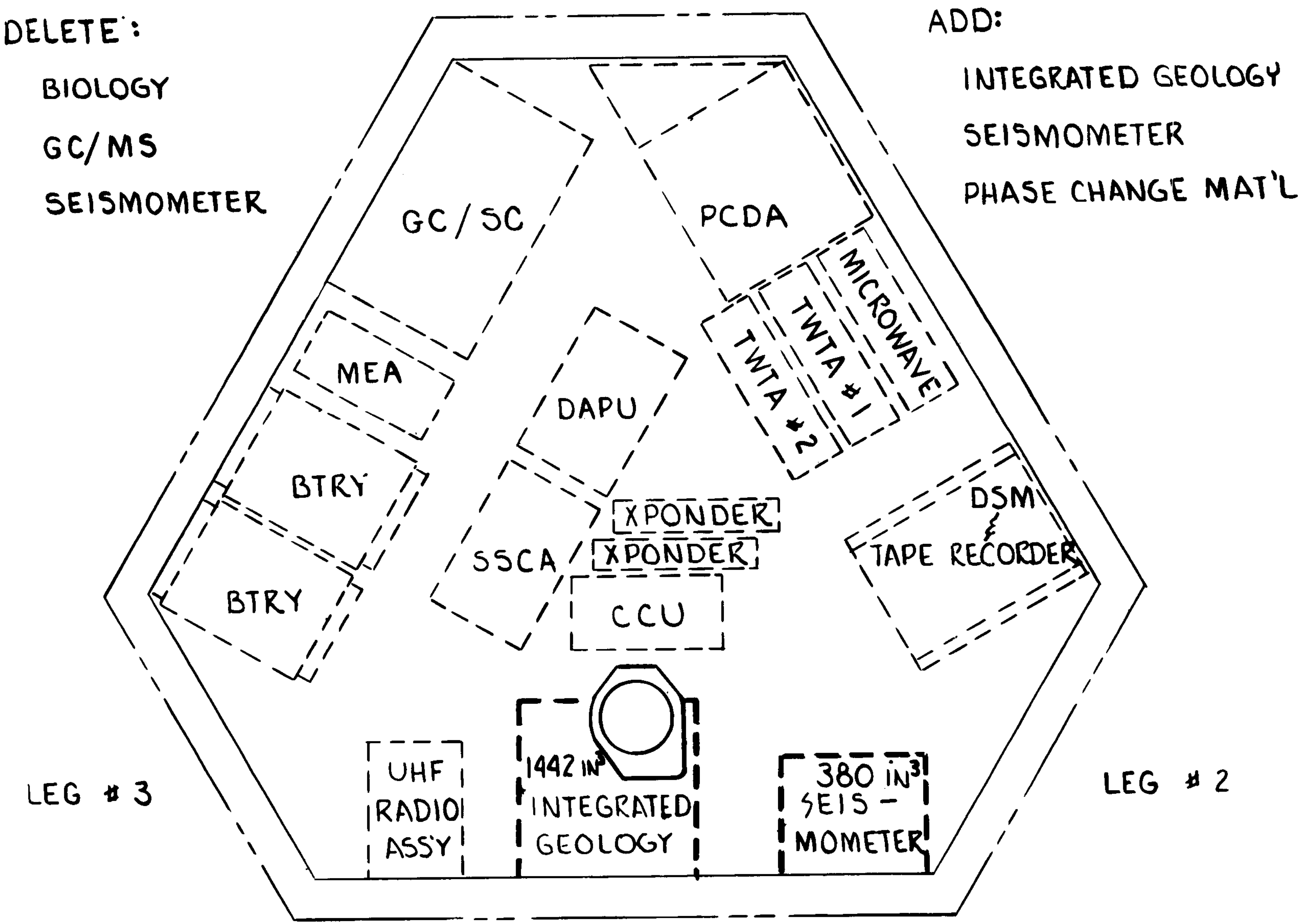


Figure IV-9 Mars Lander Inboard Modifications

2. Dynamic Environment

The acoustic environment to which the Phase III combined missions spacecraft is subjected is identical to that as described in Volume II of this report.

The acceleration and deceleration levels which the baseline spacecraft will experience during the various mission phases are presented in Table IV-6. Viking '75 g levels are also shown for comparison purposes. As can be seen, the entry g's for the study spacecraft are only 0.9 g greater than the basic Viking '75 spacecraft.

The g level experienced by the landed orbiter in landing on Phobos is 0.7 Earth g's, while the launch g's to which the Viking '75 Orbiter has been designed is approximately 10 times this value. This implies then, that adapting the Viking Orbiter to accomplish a landing mission will require at most, only local "beef-up" to the orbiter's bus structure to introduce the loads due to landing into the main load carrying members.

3. Landing Stability

The computer program which was used to determine the stability boundaries outlined during the Phase III combined missions program is that program being utilized for all landing dynamics analyses for the Viking '75 program.

The vehicle analyzed for the Phase III study incorporated the revised mass and inertia of the modified orbiter. Input values made to the computer program are shown along the left side of Table IV-7. A brief description of the study methodology is presented in the following paragraphs.

a. Digital Computer Program - The program used to compute the six degree of freedom dynamic landing behavior employs a piecewise continuous numerical integration procedure to solve the system of non-linear second order differential equations with time and position dependent forcing functions. The mathematics which

Table IV-6 Comparison of Viking '75 and P/D Study Baseline Configuration Acceleration/Deceleration Levels

Mission Phase	P/D Study	Viking '75	Notes
Launch	7.0	7.0	Stage I Burnout
Cruise	0.143	0.125	Mid Course Maneuver
Entry	20.4	19.5	
Terminal Descent	1.44	1.30	TD Engine Firing
Landing			
Mars Lander	30.0	30.0	At C. G.
Landed Orbiter	0.7	-----	

All Values Shown Are Design Level

Table IV-7 Landing Dynamics Analysis - Orbiter

STUDY GUIDELINES	STUDY RESULTS
Max $V_V = V_H = .36$ km/sec (1.5 fps)	Stability -- 100% on 25° Ground Slopes or Less
Coefficient of Friction M = 1.0	
Worst Case Stability Landing 2 Legs Leading Downhill	Leg Load $\angle$ 68 kg (150 lb) Per Leg
Vehicle Weight 1043 kg (2300 lb)	
Vehicle Inertias $I_{xx} = 585.7$ Slug-ft^2 $I_{yy} = 551.3$ Slug-ft^2 $I_{zz} = 803.0$ Slug-ft^2	
Ground Clearance 1.37 m (54 in.)	
R/H = 2.08	
Net Thrust Toward Surface $\sim 7\ lb_f$	

accounts for this motion is described in the Appendix of Volume II of this report.

b. Computer Input/Output - The numerical input to the digital computer program consists of seven sections:

1) Vehicle position and rate;
2) Surface conditions;
3) Mass data;
4) c.g. location;
5) Vehicle leg geometry;
6) Strut characteristics;
7) Leg load characteristics.

The "landing' is started by the computer at time equal to zero and the subsequent motion proceeds in finite time intervals as explained in the Appendix of Volume II of this report.

The program output includes the following information:

1) Time history of translational position, velocity, and acceleration;
2) Time history of angular orientation, velocity, and acceleration (used in determining stability);
3) Time history of loads in all leg members;
4) Maximum vectorial g-load during the landing event;
5) Final main strut lengths and minimum ground clearance.

c. Assumptions in the Stability Analysis -

1) The landing surface is smooth and nonyielding with a constant friction coefficient;
2) The elasticity of the leg and vehicle center body is represented by an equivalent elasticity in the leg only;
3) The structural damping was assumed to be zero; all the elastic stored energy is returned to the vehicle as kinetic energy.

d. Computer Result Verification - The analytical results predicted by the computer program were verified by a one-sixth scale lander model drop test program which was performed in conjunction with the Viking '75 program. The correlation between this analysis and the experimental results obtained from the test program was found to be excellent.

e. Study Results - Results of this study are presented in Table IV-7. As indicated, the landed orbiter is 100% stable on ground slopes of 25° or less. Leg loads do not exceed 68 kgs.

D. THERMAL CONTROL

The Phase III baseline configuration uses the landed orbiter concept for the exploration of Phobos and Deimos, and does not present radically new thermal requirements, as compared to the Phase II approach. The thermal constraints of the mission are essentially the same as for the Phase II baseline configuration, with some added considerations for the Mars lander. The thermal constraints are summarized on Table IV-8.

To meet the requirements indicated by Table IV-8, the following modifications to the thermal control subsystems of the orbiter and lander are proposed:

1) Add flip covers to the orbiter louver system. These will remain open throughout the cruise and orbital phases of the mission, and will be closed during the daytime operations of the landed orbiter. The louvers will also operate in their normal mode during the night on the surface of Phobos. The duration of the "open" position of the flip covers on the satellite surface will be preprogrammed on the basis of thermal analyses, and in conformance with the reduced heat dissipation requirements during landed operations as compared to the interplanetary phases of the mission.

Table IV-8 Phase III Thermal Constraints

- USE VIKING 75 HARDWARE WHERE POSSIBLE	
- LAUNCH THROUGH MARS ORBITAL OPERATIONS	: ORBITER/LANDER THERMAL ENVIRONMENT SAME AS FOR VIKING 75 MISSION
- LANDED OPERATIONS - LANDED ORBITER	: INTERNAL POWER DISSIPATION REDUCED BY 60 - 70 PER CENT AS COMPARED TO CRUISE
	VIEW OF RADIATION HEAT SINK (SPACE) OF LOUVERS CONSTRAINED BY GROUND SURFACE AND ORBITER PROPULSION MODULE
	UNCERTAINTIES IN GROUND RADIATION EFFECTS DUE TO DUST, THERMAL INERTIA, TOPOGRAPHY, ETC.
	DIURNAL SOLAR EXPOSURE OF HEAT REJECTION (LOUVER) SURFACES
- LANDED OPERATIONS - MARS LANDER	: HOT EXTREME TEMPERATURES HIGHER THAN DURING THE 75 MISSION

2) Increase the thermal mass of the orbiter equipment compartment by the use of Phase Change Material (PCM) to absorb the equipment heat generated during the day on Phobos, when the compartment is essentially isolated from the external thermal environment.
3) Increase the thermal mass of the equipment mounting plate of the Viking Lander by the use of PCM. This will prevent possible equipment overheating during the hot extreme environments of the '79 mission.

The Phase III thermal control approach is summarized on Table IV-9.

The use of active control of the orbiter flip covers during the Phase III mission is necessary because of the radiation blockage of the propulsion module; hence equipment heat rejection on the satellite surface is to be the "night mode" as defined in the Phase II thermal control discussion. This compares with the Phase II approach, where the flip covers were closed only once, upon landing. The Phase III system is inherently less reliable because of its active character. However, this is consistent with the shorter duration of the landed phases of the Phase III orbiter mission, since meaningful data may be obtained within a few hours after landing, even if the flip covers are not operating.

E. PROPULSION

The Phase III propulsion studies were directed primarily at determining the suitability of using the Mars Viking spacecraft propulsion and attitude control systems to perform the combined Mars landing and Phobos/Deimos mission.

The Viking Orbiter's primary propulsion system capability must be "grown" by 26% to accomplish the baseline mission delta velocity requirement of 2235 mps. This "growth" is achieved by

Table IV-9 Phase III Thermal Control Approach

LANDED ORBITER MODIFICATIONS :	ADD EXTERNALLY MOUNTED FLIP COVERS TO DEACTIVATE LOUVERS DURING DAY-TIME LANDED OPERATIONS - FLIP COVERS ARE HINGED INSULATED PANELS WITH SOLAR-REFLECTOR EXTERNAL FINISHES - DURING CRUISE, ORBIT, AND AT NIGHT THE FLIP COVERS ARE MAINTAINED IN OPEN POSITION BY PRE-PROGRAMMED ACTIVE CONTROL
	ADD PHASE CHANGE MATERIAL (PCM) TO ABSORB EXCESS EQUIPMENT HEAT DURING DAY-TIME LANDED OPERATIONS
LANDER MODIFICATIONS :	INCREASE THERMAL MASS OF EQUIPMENT MOUNTING PLATE (VIA PCM) TO PREVENT DAY-TIME EQUIPMENT OVERHEATING IN THE HOT EXTREME THERMAL ENVIRONMENT

increasing the two propellant tanks 7.6 cm (3.0 inches) in length and 7.6 cm in diameter and by increasing the pressurization sphere 1.8 cm in diameter.

The orbiter's cold gas attitude control propulsion requires a small amount (0.3 kg) of additional N_2 to accommodate the increased mass and inertia of the baseline configuration. This increase in gas also required that the nitrogen storage tank volume be increased slightly. Total attitude control system weight increased by approximately 0.5 kg.

A terminal descent (Phobos rendezvous and landing) propulsion system is required for the orbiter. Propulsion system selected is a monopropellant blow-down type that utilizes hydrazine. The system consists of a titanium fuel tank, a series/parallel ordnance valve package, fuel filter, and four quad-thruster and solenoid valve assemblies. Each of the thruster assemblies are mounted on the outside of the oribter bus and in line with the cold gas attitude control thrusters that are mounted on the solar panel extremities. This configuration permits use of the three-axis attitude control GCS system.

Titanium (6Al-4V) was selected as tankage material and a 10 mil teflon polymeric bladder for pressurant gas separation and propellant acquisition. One-half of the propellant tank has been allocated for nitrogen pressurant gas at an initial storage pressure of 420 psia resulting in a final blow-down pressure of 210 psia. A pyro-valve package assembly and propellant filter are downstream of the propellant tank. The pyro-valve package consists of two normally open and two normally closed valves overriding positive propellant isolation between rendezvous/ landing burns and future surface maneuver burns. An in-line filter downstream of this package is provided to eliminate particulate matter from the thruster control valves. The thrusters and solenoid control valve assemblies are of the type used on the Mars

Viking Lander for deorbit and terminal descent roll control. The thrusters are a direct catalytic type using Shell 405 catalyst for spontaneous ignition of the hydrazine fuel. The resultant thrust variation for the propellant tank blow-down ratio of 2 to 1 is 9.25 lbs to 5.6 lbs. The average thruster specific impulse is 227.5 seconds.

A schematic of the added terminal descent propulsion system is shown in Figure IV-10. The weight statement for the system is presented in Table IV-10.

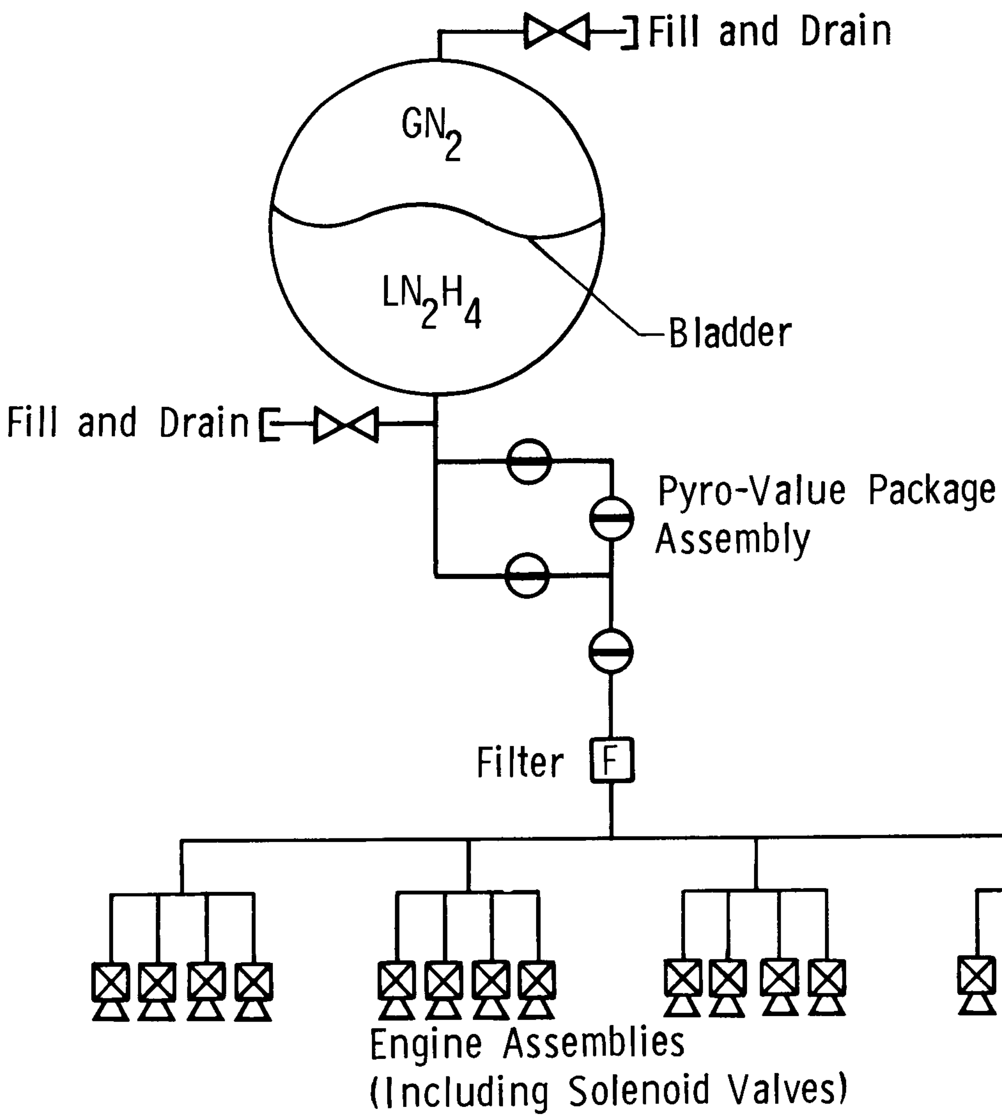
The Viking Lander's deorbit, and terminal descent and landing propulsion subsystems are for all practical purposes adequate to accomplish the mission.

A very slight increase in propellant loading (1.8 kgs) is required to accommodate the modest growth in landed weight.

F. TELECOMMUNICATIONS

The Mars lander communications from the Mars surface will employ both a UHF relay link via the orbiter and a direct to Earth link at S-band. The communications subsystems employed will essentially be the Viking '75 Lander subsystem unchanged. The direct to Earth S-band link will have an increased data rate capability over Viking '75, 500 bits per second instead of 250 bps. This is accomplished by using only one subcarrier on the downlink S-band carrier and transmitting both science and engineering data on the same subcarrier. The S-band transmitter and high gain antenna will be the same. The direct to Earth S-band link will provide over 3 megabits of data return per 24 hours based on 1.75 hours of data transmission.

The UHF relay link through the orbiter will be at 16 kbps, also the same as for Viking '75. The orbiter will be in either a 24.6 hour or a 97 hour orbit with a periapsis altitude of 1500 km. For



System Characteristics

- Hydrazine Monopropellant
- Blow Down Type
- Engines (16)
  - Type Used on VL for Deorbit and Terminal Descent Roll Control
  - Shell 405 Catalyst
  - Thrust Varies from 9.25 to 5.6 lbf
  - 227.5 Seconds Average ISP

Figure IV-10 Orbiter Terminal Descent Propulsion System Schematic

Table IV-10 Landed Orbiter Terminal Descent Propulsion Weight Statement, KG

• Engine Assemblies (16)	15.5	
• Filter (1)	0.5	
• Isolation Valve	0.2	
• Fill/Drain Assemblies	0.2	
• Propellant Tank Assy	8.8	
• Plumbing, Wiring & Structure	7.5	
• Pressurant (GN_2)	1.2	
• Residual Propellant	0.9	
• Total Inert Weight		34.8
• Usable Propellant	49.9	
• Total System Weight		84.7

the 24.6 hour orbit, and a lander-orbiter contact time of 25.42 minutes, a total data volume of 24.4 megabits will be relayed back to Earth. For the 97 hour orbit and a lander-orbiter contact time of 25 minutes, the data volume will be 24 megabits.

The total data volume transmitted to Earth via both links will therefore be in excess of 27 megabits.

It is possible to utilize the UHF relay between the lander on the Mars surface, and the orbiter "landed" on Phobos or near the satellite. The viewing time between the two points will be 1.64 hours per 7.65 hour period of Phobos. Both the lander and orbiter will use the Viking '75 UHF communications subsystems unmodified except for data transmission rate. It will be necessary to reduce the data transmission rate on the link to 4 kbps instead of 16 kbps, since the range will be approximately 7000 km. The Viking '75 system provides adequate margin out to 3500 km at 16 kbps. During the 1.64 hour viewing period per orbit, a total data volume of 23.6 megabits will be transmitted.

For the landed orbiter configuration the communications link will be direct to Earth at S-band and will use the Viking '75 Orbiter communications subsystem. This subsystem consists of the Mariner class S-band equipment with 20 watt output TWTA transmitters, an articulated 58 inch high gain antenna, S-band receiver, and low gain S-band command antenna. The UHF subsystem will be maintained and can be used for reception of data transmitted from a Mars lander.

Transmission time via the direct S-band link to Earth will be 2 hours per 24 hours, with 1.75 hours being used for actual data transmission. The data transmission rate on the direct to Earth link will be 4 kbps as for Viking '75. The total data volume returned to Earth each 24 hours will be over 25 megabits.

G. POWER

Power studies conducted during Phase III where aimed at determining the applicability of the present Viking Orbiter and Lander power systems to accomplish the combined Mars and Phobos/ Deimos mission.

The power output capability of the Viking Orbiter's solar array was computed for various arrival years at Mars corresponding to Earth launches in 1979, 1981 and 1983. The power available for these arrival years is shown in Figure IV-11. Also computed and presented in Figure IV-12 is the power available from the Orbiter's solar array when landed on Phobos. These values were calculated with the solar panel drooped 32 degrees from the horizontal in order to provide a more uniform output during the Sun's transit across the sky.

Power requirements were then developed for the various mission phases and modes to ensure that adequate power was available at all times from the orbiter's solar array.

Figure IV-13 shows the power requirements for the critical phases involved in the transit of the Phobos/Deimos spacecraft from Earth to Phobos rendezvous and subsequent descent and landing. The figure shows that the solar panels are adequate to supply the power needs except when solar orientation is lost. In this case, power is drawn from the orbiter's two 30-ampere hour batteries.

Tables IV-11, IV-12, and IV-13 indicate the critical power requirements for the observation orbit, stationkeeping and landed mission modes, respectively. In each case the total raw power required is alwyas less than the power available.

Power requirements imposed upon the lander's power system was also evaluated. These requirements are presented in Figure IV-14 in the form of a power profile for the Mars Lander mission.

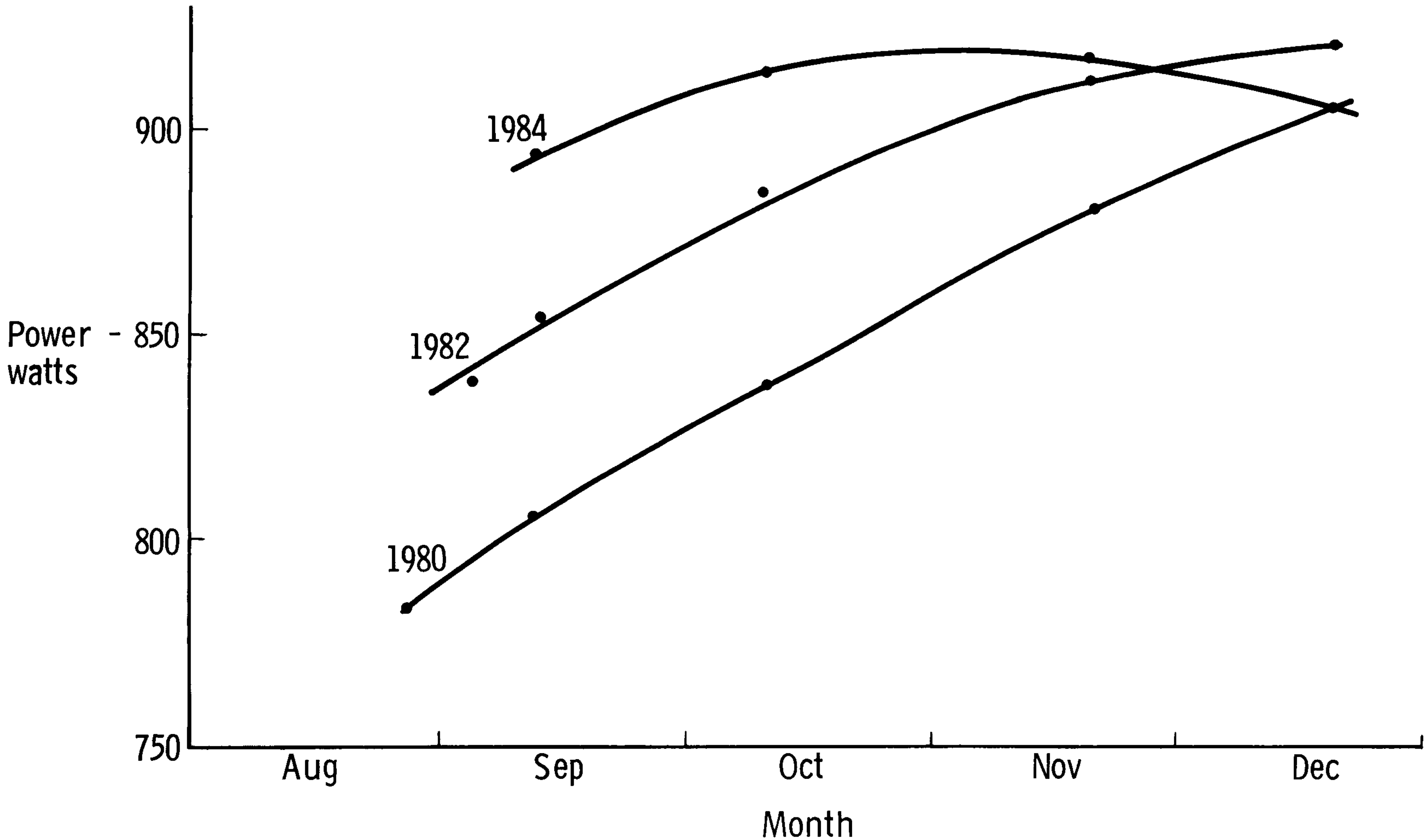


Figure IV-11 Viking Orbiter Power Output for Various Arrival Years

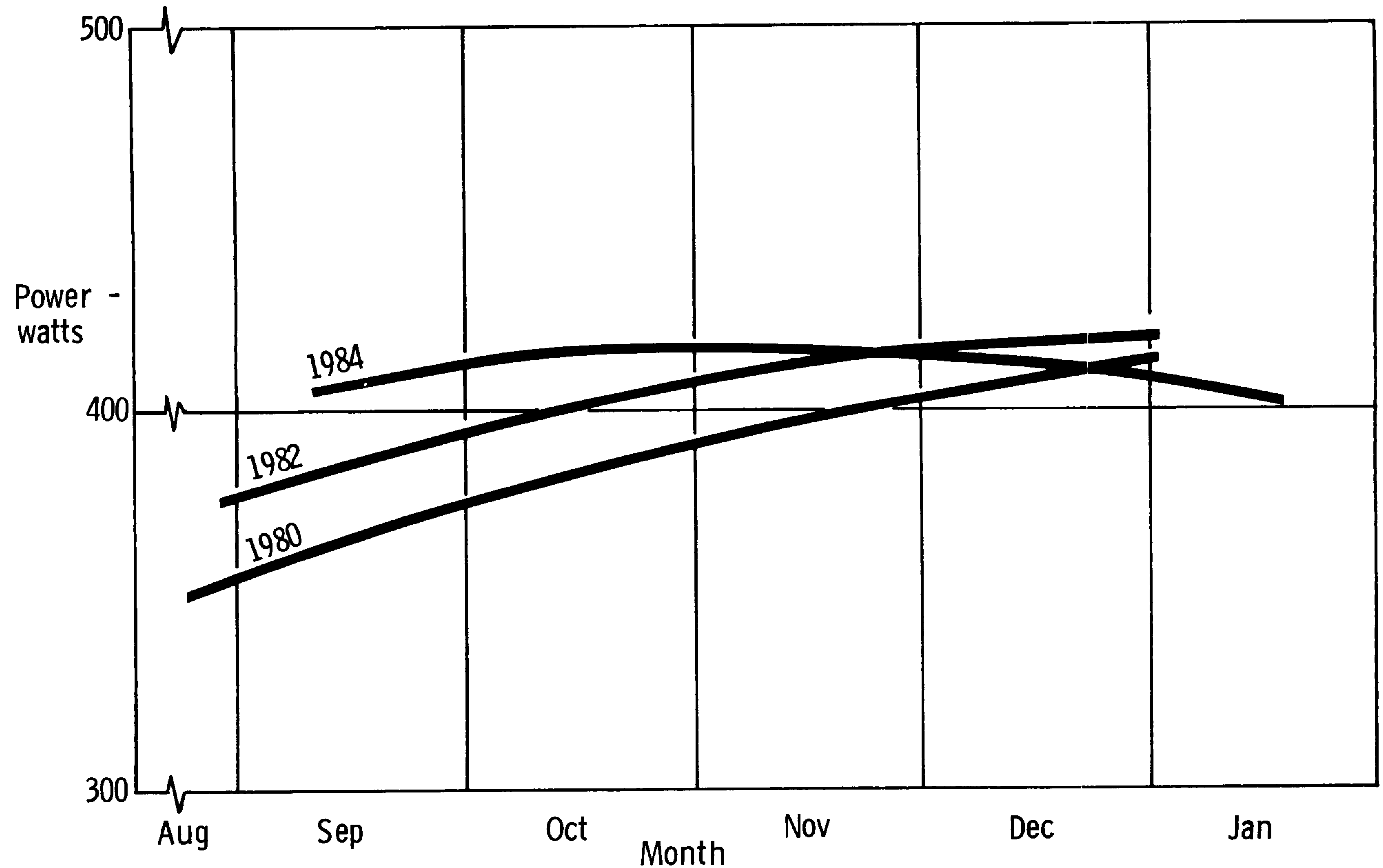


Figure IV-12 Landed Viking Orbiter Power Output

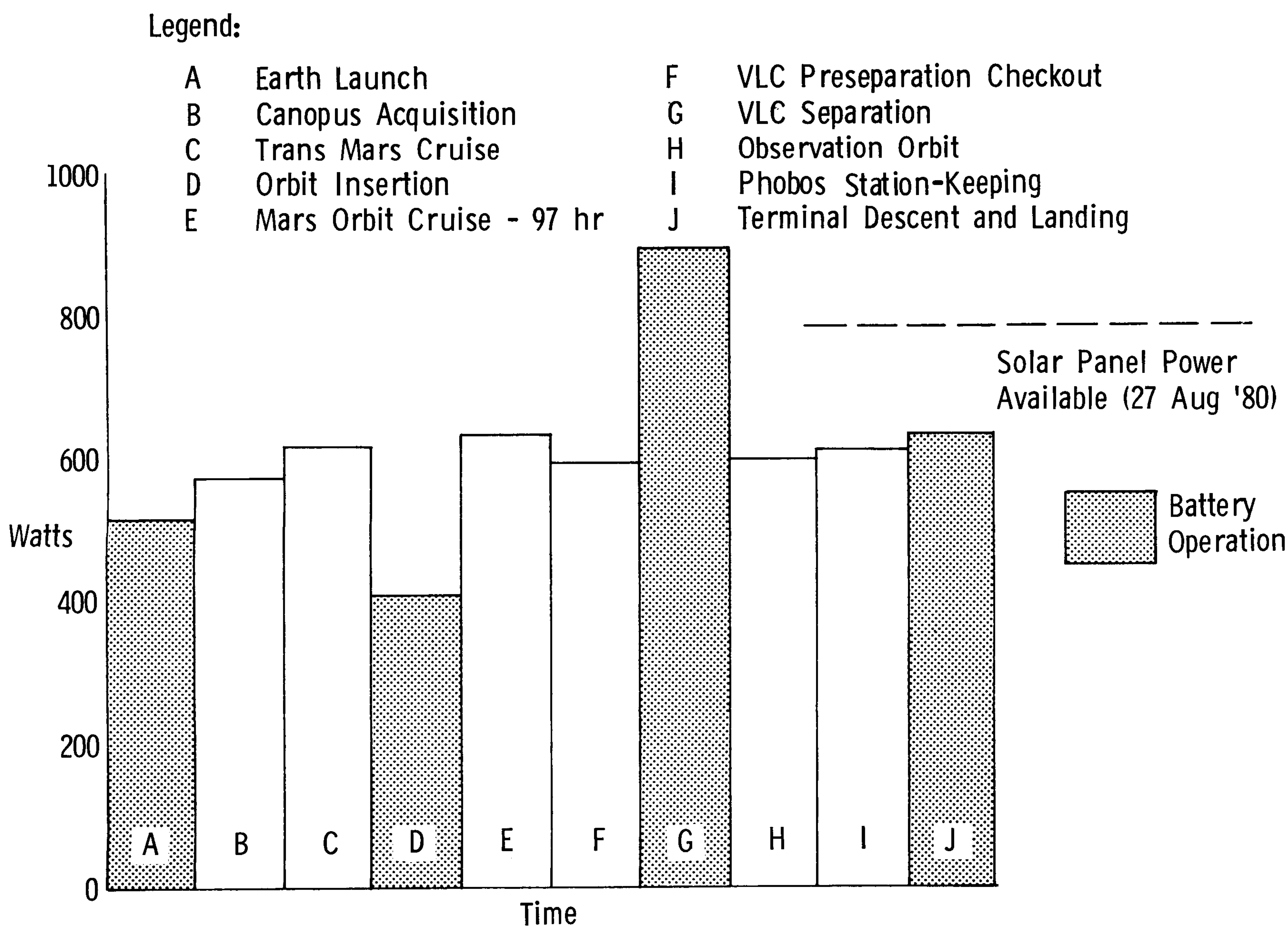


Figure IV-13 Orbiter Power Profile

Table IV-11 Orbiter Power Summary - Observation Orbit

EQUIPMENT	WATTS
2.4 KHZ Inverter	
Engineering Loads	
Malfunction Detection Sys	6.7
Flight Data Subsystem	42.4
Command & Control Seq	15.0
Pyrotechnics	1.0
Power Distribution	13.0
ACS Electronics	12.5
Radio Freq Sys (exc TWT)	28.0
ACS Gyro Electronics	.0
Articulated Control	35.3
Data Storage Subsystem	90.8
Radio Relay Subsystem	.0
Total Eng Load	244.7
Science Loads	
Visual Imaging Subsystem	45.0
Adv IR Spectrometer	20.0
Total Science Load	65.0
Total 2.4 KHz Inverter Load	309.7
400 Hz 3 Ph Inverter	
ACS Gyro Motors	.0
30 VDC Converter	
Engine Valve	.0
Gimbal	.0

EQUIPMENT	WATTS
Booster Regulator	
Total 2.4 KHz Inverter Input	338.5
Total 400 Hz 3 Ph Inverter Input	.0
Total 30 VDC Converter Input	.0
Scan Platform Heaters	42.5
VIS/IR Spec Bay Heaters	.0
DSS1 Bay Heater	.0
DSS2 Bay Heater	.0
Occultation Heaters	.0
Hi Gain Ant Act Heater	14.5
B/R Power Distribution	5.0
Total B/R Load	400.5
B/R Efficiency	.9
Total B/R Input	445.5
Battery Charging	.0
Power Failure Sensor	1.5
TWT System	92.0
VLC	.0
Total Unregulated Power	539.0
PSL Efficiency	.98
Total Raw Power	548.8
2% Off Max Power Point	11.0
System Power Contingency	40.0
Total Raw Power Required	599.8

Table IV-12 Orbiter Power Summary - Phobos Stationkeeping

EQUIPMENT	WATTS
2.4 KHz Inverter	
Engineering Loads	
Malfunction Detection Sys	6.7
Flight Data Subsystem	42.4
Command & Control Seq	15.0
Pyrotechnics	1.0
Power Distribution	13.0
ACS Electronics	12.5
Radio Freq Sys (exc TWT)	28.0
ACS Gyro Electronics	.0
Articulated Control	35.3
Data Storage Subsystem	90.8
Radio Relay Subsystem	.0
Relay Telemetry Subsystem	.0
Total Eng Load	244.7
Science Loads	
Visual Imaging Subsystem	45.0
Adv IR Spectrometer	20.0
Passive Gamma Ray Spectrometer	3.0
Passive X-Ray Fluorescence Sp	3.5
Total Science Loads	71.5
Total 2.4 KHz Inverter Load	316.2
400 Hz 3 Ph Inverter	
ACS Gyro Motors	.0

EQUIPMENT	WATTS
30 VDC Converter	
Engine Valve	.0
Gimbal	.0
Booster Regulator	
Total 2.4 KHz Inverter Input	345.6
Total 400 Hz 3 Ph Inverter Input	.0
Total 30 VDC Converter Input	.0
Scan Platform Heaters	42.5
VIS/IR Spec Bay Heaters	.0
DSS1 Bay Heater	.0
DSS2 Bay Heater	.0
Occultation Heaters	.0
Hi Gain Ant Act Heater	14.5
B/R Power Distribution	5.0
Total B/R Load	407.6
B/R Efficiency	.9
Total B/R Input	542.9
Battery Charging	.0
Power Failure Sensor	1.5
TWT System	92.0
VLC	.0
Total Unregulated Power	546.4
PSL Efficiency	.98
Total Raw Power	557.6
2% Off Max Power Point	11.2
System Power Contingency	40.0
Total Raw Power Required	608.8

Table IV-13 Orbiter Power Summary - Phobos Landing

Equipment Name	Imaging	Transmission (2 hours)	Sample Collection	Sample Processing	Integrated Geology	Transmission (2 hours)	Standby	Transfer to New Site
Engineering Loads								
Malfunction Det Sys	6.7	6.7	6.7	6.7	6.7	6.7	6.7	6.7
Central Computer & Sequencer	15.0	15.0	15.0	15.0	15.0	15.0	15.0	15.0
Power Distribution	13.0	13.0	13.0	13.0	13.0	13.0	13.0	13.0
Radio Freq Sys (ex TWT)	28.0	28.0	28.0	28.0	28.0	28.0	28.0	28.0
Data Storage Subsystem	17.4	17.4	17.4	17.4	17.4	17.4	.0	17.4
Battery Charging	.0	83.1	.0	.0	.0	83.1	.0	.0
Power Failure Sensors	1.5	1.5	1.5	1.5	1.5	1.5	.0	1.5
TWT System	.0	92.0	.0	.0	.0	92.0	.0	.0
System Losses	47.0	53.7	53.0	49.5	47.0	53.7	40.0	45.0
Science Loads								
Visual Imaging Subsys	15.2	.0	.0	.0	.0	.0	.0	.0
Advanced IR Spec	.0	.0	.0	.0	.0	.0	.0	.0
Surface Drill	.0	.0	50.0	.0	.0	.0	.0	.0
Sample Processor	.0	.0	.0	20.0	.0	.0	.0	.0
Integrated Geology	.0	.0	.0	.0	8.5	.0	.0	.0
Total	143.8	310.4	184.6	151.1	137.1	310.4	102.7	126.6

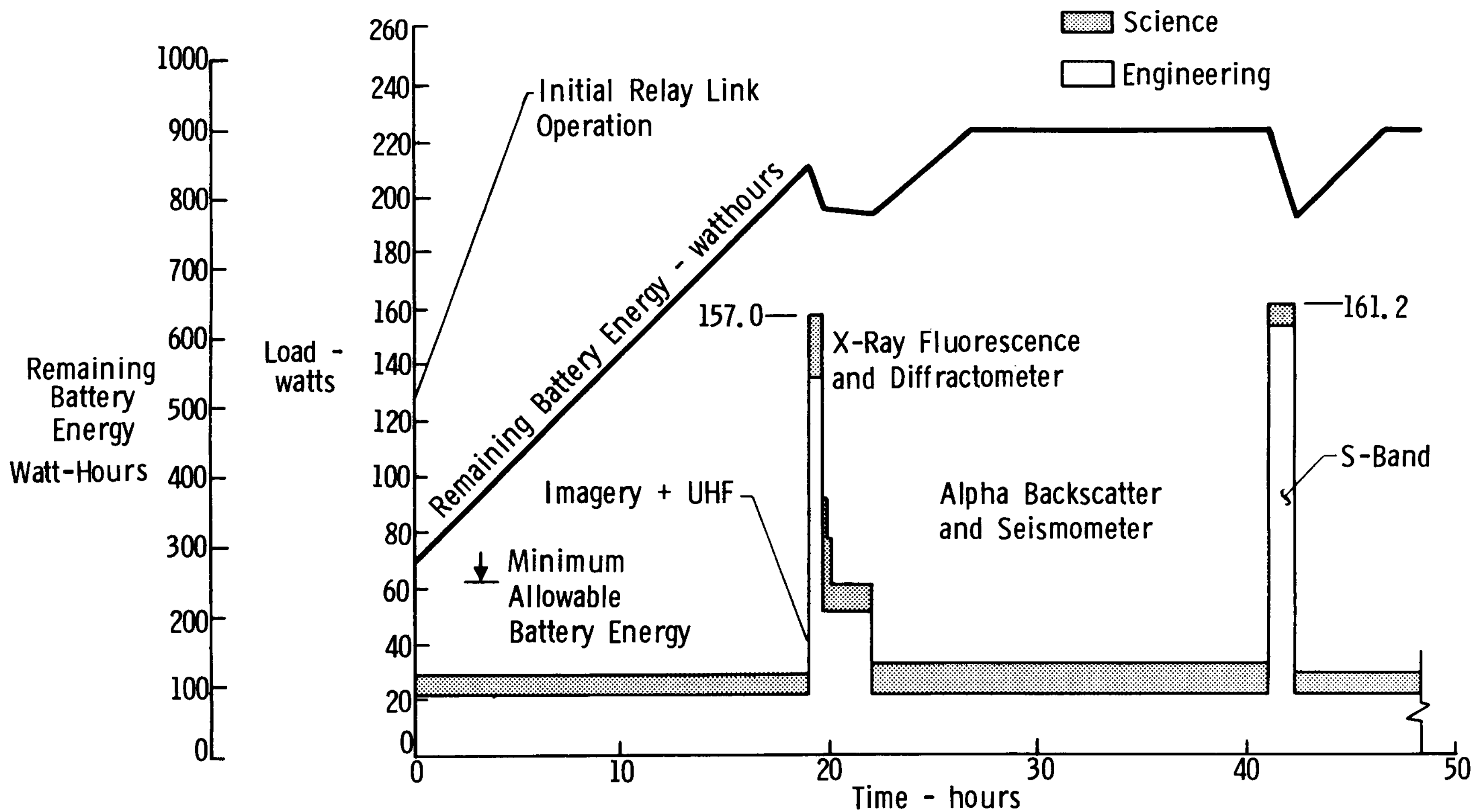


Figure IV-14 Impact of Power Requirements on Mars Lander Power System

H. PHOBOS HARD LANDER

An unguided, free-fall, hard lander which would determine elemental and mineralogical composition at one point on Phobos' surface was considered briefly during Phase III. Such a concept lies, in complexity, science value, and cost, somewhere between a soft lander and no lander.

The hard lander (HL) could be stowed in available space between the orbiter and the Mars lander. After separating from the Mars lander, the orbiter would rendezvous with Phobos and enter a stationkeeping oribt 10-30 km from Phobos' surface. The HL would be released from this altitude with a 1-5 m/sec closure velocity toward Phobos. Under these conditions, free-fall landing velocities will lie between 15 and 20 m/sec.

Table IV-14 presents the design constraints and guidelines and the payload for this lander. Figure IV-15 shows the deceleration distances that must be used to provide average deceleration of 500 m/sec^2 at impact. A preliminary concept which could provide these deceleration distances in any landing attitude is shown in Figure IV-16. After deployment from the orbiter, this concept would have a 30 cm (12 inch) payload sphere suspended at the center of a 1.22 m diameter, crushable aluminum sphere. This would provide up to 46 cm of crush (decelration) distance at impact before the payload sphere would reach Phobos' surface. A complete operating sequence from orbiter separation through to payload deployment and data relay to the orbiter using omnidirectional antennas is shown in Figure IV-17. Free-fall descent to the surface would take 20-60 minutes. A period would be allowed for stabilization on the surface after landing in the low-g environment, after which the lander's crush hemispheres would be separated and the payload deployed to the surface. On the order of 50 kilobits of data would be acquired and relayed to the orbiter in

Table IV-14 Phobos Hard Lander Design Considerations

- Weight: < 45 kg (100 lb)
- Stowed Volume:
  - Maximum Diameter = 1.22 m (48 in.)
  - Maximum Height = .51 m (20 in.)
- Payload Weight:

X-Ray Fluorescence Spectrometer	.45 kg (1.0 lb)
X-Ray Diffractometer	4.50 kg (10.0 lb)
Electronics, Sequencer	1.35 kg (3.0 lb)
Battery	.90 kg (2.0 lb)
Payload Support Truss	1.80 kg (4.0 lb)
	9.00 kg (20.0 lb)

- Average Deceleration at Impact: < 500 m/sec^2 (50 g)
- Maximum Deceleration at Impact: < 1000 m/sec^2 (100 g)
- Payload Deployment Requirements:
  - XRFS and XRD "Windows" on Surface
  - Unobstructed Omnidirectional Antenna for Data Relay to Orbiter

- Payload in 30 cm (12 in.) Sphere
- Payload Sphere Supported in Center of 1.22 m (48 in.) Thin wall, Soft Aluminum Sphere
- Crush Distance Between Aluminum Sphere and Payload Sphere = 46 cm (18 in.)
- Crush Distance (S) vs Impact Velocity (V_F) for Average Deceleration of 500 m/sec^2 (50 g)

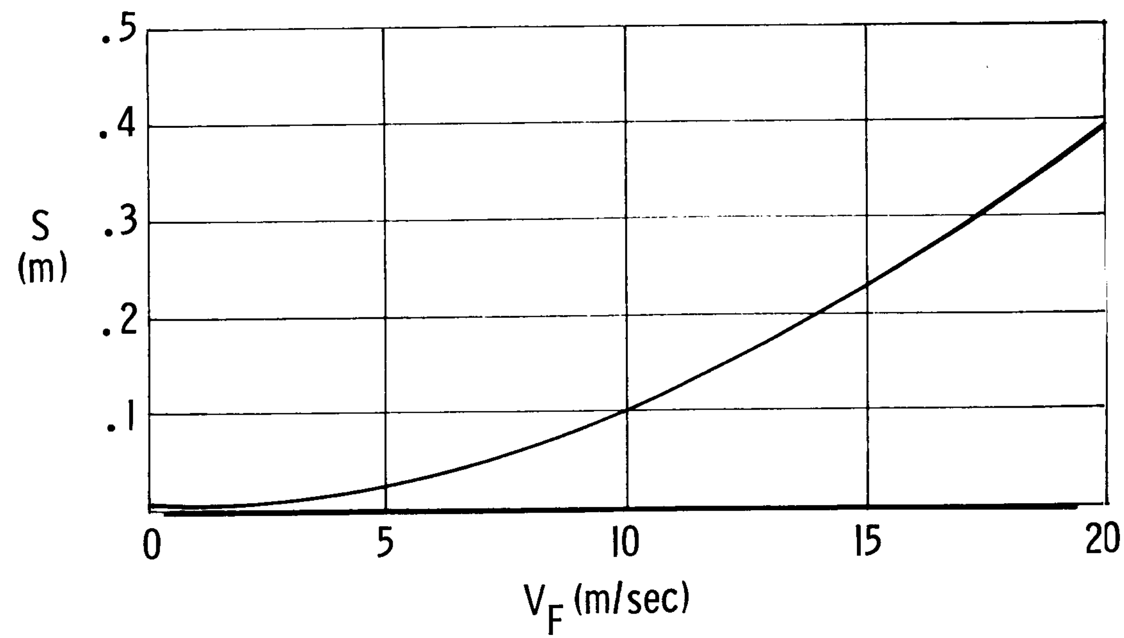


Figure IV-15 Phobos Hard Lander Deceleration Distance (S) vs Landing Velocity (V_F) for Average Deceleration of 500 m/sec^2

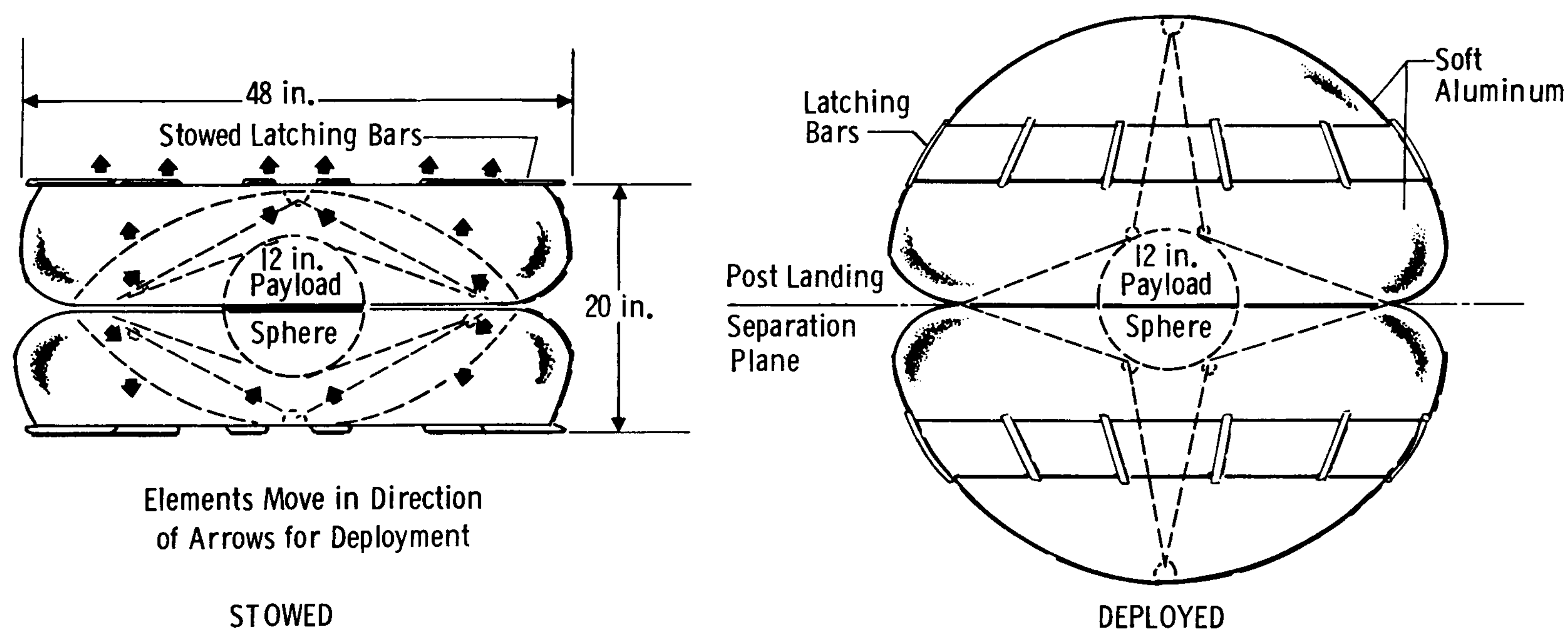


Figure IV-16 Phobos Hard Lander Concept

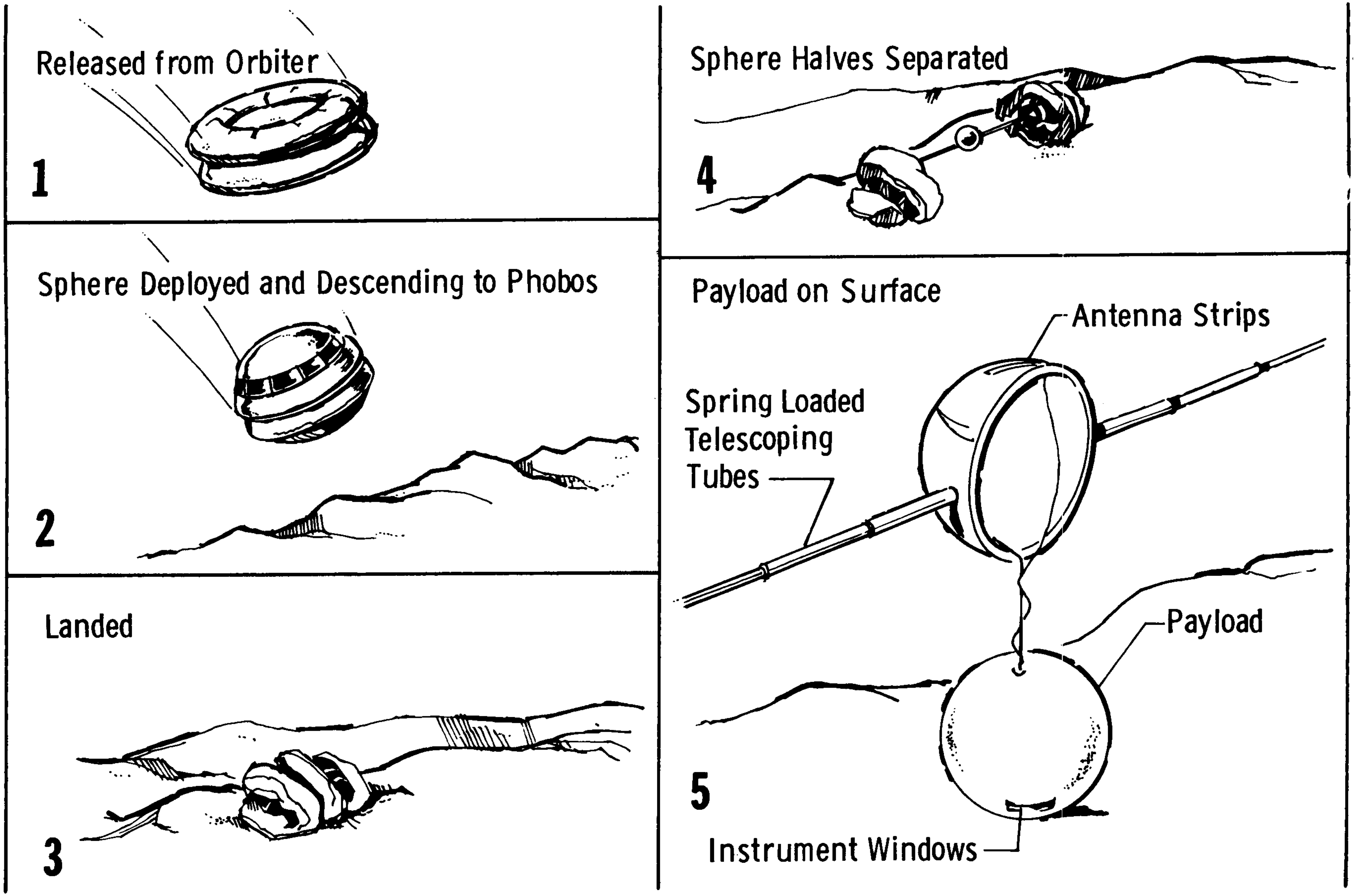


Figure IV-17 Phobos Hard Lander Operating Sequence less than one hour after payload deployment. Lander operations would then cease.

If further work on a Phobos hard lander is initiated in the future, consideration should be given to this concept but alternative concepts should be explored as well. Operating within the constraints imposed on the concept presented here, the following are identified as the critical design problems:

1) provide a lander which can land successfully in any attitude and keep average landing impact acceleration below 50 "g,"
2) stow this lander in a 1.22 m diameter by 0.51 m tall volume,
3) insure that the instrument windows are placed in contact with the surface material, recognizing that in the low-g environment, the slightest interference from the lander's outer body shell can keep the payload from reaching the surface, and
4) insure that the lander's antennas are not obstructed.

V. Program Costs

V. PROGRAM COSTS

The cost summary for the baseline mission in FY 72 dollars is shown in Table V -1. No escalation factors have been added. The baseline mission for which this cost summary was generated is a 1979 earth launch using the 26% stretched Viking Orbiter, the landed orbiter satellite landing mode, and the out-of-orbit (97 hour period) Mars Lander.

The basic groundrules and assumptions that were used in developing these costs are:

1) Two flight and one spare spacecraft are to be developed,
2) Costs are in FY 72 dollars,
3) Titan IIIE/Centaur launch vehicle,
4) One system contractor will have overall system responsibility for the design, development, fabrication, and qualification testing for the mission,
5) Sterilization not required for landed orbiter,
6) Non interference basis with other Viking programs,
7) Maximum inheritance of technology from Viking programs,
8) Use modified Viking '75 ground equipment,
9) Use quality and proof test evaluation units.

The cost estimate has been built up using a work breakdown structure patterned after the Viking '75 Lander system. This work breakdown structure contains over 80 elements of cost. Labor and material estimates were made for each of the WBS elements.

Four previously developed program estimates were used as references and calibrations for the estimate: 1) the Viking '75 program (which would have higher costs for equivalent elements because of the completely new developmental nature of the work),

2) the Viking '77 program (which should be lower for equivalent elements because it involves minimum modification to existing designs), 3) Viking '79 program (which would have a cost higher than a Viking '77 repeat mission due to additional modification of '77 design), and 4) Phobos/Deimos Phase II baseline mission (which is roughly equivalent, in total dollars, to the Phase III baseline mission). This estimated cost of a combined Mars landing and Phobos/Deimos landing mission represents an approximate 14% increase over a Mars-only landing mission performed at the same launch opportunity and using the same costing ground rules and assumptions.

Table V-1 Mars - Phobos/Deimos Baseline Mission

Cost Summary (FY'72 Dollars)
($ in Thousands)

Spacecraft		274
Management and Technical Support	74	
Mission Analysis and Sys. Engr. & Integration	26	
Subsystems Development & Qualification		
Lander (Mars)	47	
Orbiter/Lander (Phobos)	89	
Systems Assembly and Test	12	
Launch and Flight Operations	26	
Science		45
Other NASA Cost		78
Launch Vehicles		44
TOTAL		$441

Funding by Fiscal Year

1976	1977	1978	1979	1980	1981	1982	Total
25	85	111	93	52	40	35	441

VI. Program Schedule

VI. PROGRAM SCHEDULE

The program schedule shown in Figure VI.-1 illustrates the key milestones and span times for the Phase III combined Mars and Phobos/Deimos landing mission from the SRT and MA&D long-lead activities which begin in January 1974, to full go-ahead in mid-1975, through detail engineering, test and launch in October 1979.

The basic assumptions and guidelines which were used in the development of this schedule were:

1) Target launch date is 9 October 1979 with a launch period (nominal) of 30 days,
2) Two flight and one spare spacecraft articles to be developed,
3) One system contractor,
4) Non interference basis with other Viking programs,
5) Modified Viking '75 ground equipment.

The approach that was utilized in scheduling the various program activities was to arrange them so that the adequacy of the basic design modifications would be confirmed as early as possible to allow time for the solution of unpredicted development problems, should they arise. The schedule as presented in Figure VI.-1 is keyed on the early start of science development and mission analysis, which is scheduled to begin some eighteen months before full go-ahead.

The basic philosophy that was used in developing the schedule was to make maximum use of the Mars Viking subsystem and system technology, and hardware development. The schedule as structured in Figure VI.-1 makes maximum advantage of the Mars Viking subcontract buys.

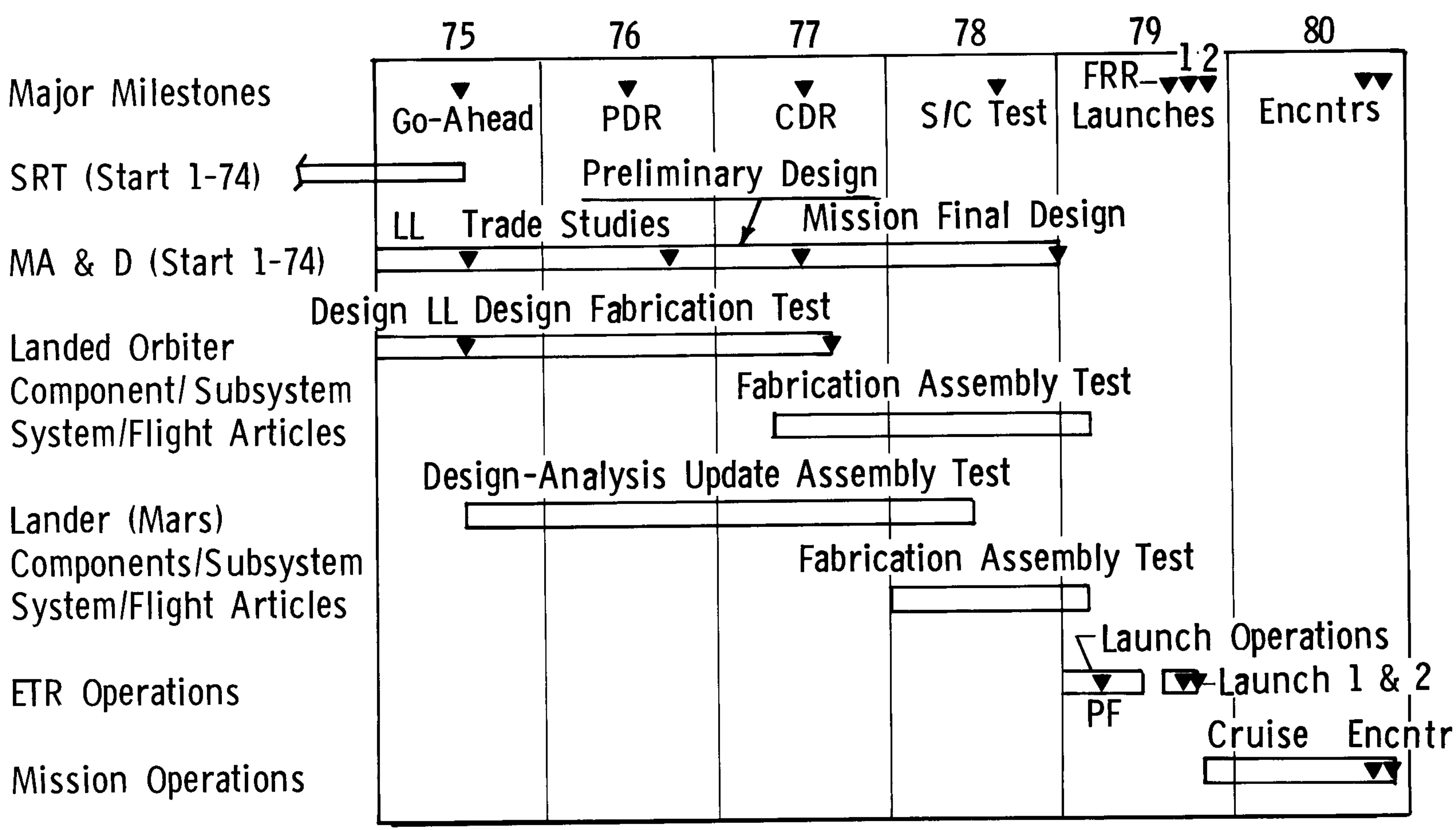


Figure VI-1 Combined Mars and Phobos/Deimos Missions Schedule

VII. Conclusions

VII. CONCLUSIONS AND TECHNOLOGY REQUIREMENTS

The principal conclusions drawn from the Phase III study effort are summarized in this section. Study results indicate that the combined Mars landing and Phobos/Deimos mission is technically feasible and a very cost effective mission to fly in the 1979-1983 time period. A combined mission of this nature in this time period makes maximum utilization of Mars Viking hardware, technology and subcontract buys. No high-risk technology problems were identified in the various subsystem mechanizations selected for the baseline concept.

The modifications necessary to adapt the Mars Viking Orbiter to this mission and to its landed role are nominal and easy to accomplish. The more significant changes are:

1) Propulsion system propellant capacity has increased by 26% to accommodate the additional propellant required for the combined missions,
2) Addition of a terminal descent propulsion system,
3) Incorporation of integrated solar panel/landing legs (4).
4) Addition of a rendezvous radar,
5) Minor thermal control modifications, and
6) Incorporation of a Phobos/Deimos geoscience payload.

The modifications required to be made to the existing Mars Viking Lander to handle the 97 hour orbital period instead of the Viking mission 24.6 hour orbital period are extremely minimal in nature, consisting primarily of; an increase in heat shield ablator thickness; a 1.8 kg increase in propellant loading; and the incorporation of a geoscience payload.

Modifications required to be made to the existing Viking Lander to accomplish a direct entry landing mode, if that option is chosen, are somewhat more extensive. This entry mode requires the following changes:

1) Present heat shield ablator formulation is unacceptable for the direct entry mission because of excessive surface erosion. Therefore incorporation of carbon fillers into the ablator formulation is required to be compatible with the higher heating and dynamic loading.
2) Equipment mounting structure weight has to be increased 5% because of the increased deceleration levels.
3) Propellant loading has to be increased by 14.9 kg to accommodate increased entry and landing weight.
4) Incorporation of a geoscience payload.

In order to build on what has been accomplished in the present study, and to further establish the most effective approach to the combined Mars and Phobos/Deimos mission, as well as to provide more confidence in our recommended baseline, it is recommended that further work be done in the areas listed in Table VII-1. These items are described in the following paragraphs.

Table VII-1 Recommendations for Further Study

Increased Mars Landing Latitude Accessibility
Improved Mars Landing Accuracy
Mars Entry Corridor Analysis
Commonality of Instrumentation for Combined Missions
Add Asteroid Observations to Combined Missions
Small Instrumented Probes for Combined Missions

A. INCREASED MARS LANDING LATITUDE ACCESSIBILITY

The Mars landing latitude capability described in this phase of the study has a relatively limited latitude capability because of the effects of the low ΔV budget for the Phobos/Deimos portion of the combined mission. Additional latitude capability can be obtained by utilizing different orbital maneuver strategies, increasing lander deorbit fuel to obtain additional cross range capability, and using multiple deorbit burns to land further away from periapsis.

B. IMPROVED MARS LANDING ACCURACY

The further exploration of Mars would be enhanced by the capability to land very close to a predetermined landing site. This would allow a mission to be targeted for a small scientifically interesting site with a high probability of success. The primary causes for the current landing footprint are the deorbit maneuver execution errors. The uncertainties in the Martian atmosphere have a smaller effect on the landing site uncertainty. The footprint size can be reduced by 1) reducing basic execution errors by making hardware and software changes, 2) minimizing the effect of these errors by mission design, and 3) combinations of 1) and 2).

C. MARS ENTRY CORRIDOR ANALYSIS

The use of a 4 day orbit at the time of the lander deorbit maneuver yields a slightly higher entry velocity. The effect of this on the lander at the entry corridor extremes should be examined in more detail to minimize the required design changes.

D. COMMONALITY OF INSTRUMENTATION FOR COMBINED MISSIONS

The use of the same instruments to fulfill functions for each of the two missions would enhance the total scientific return. An example of this would be the use of high resolution TV for the exploration of Phobos/Deimos and also for the mapping of the Martian surface.

E. ADD ASTEROID OBSERVATIONS TO COMBINED MISSIONS

The nearness of Mars' orbital path to the asteroid belt suggests that the larger asteroids may be observed by an orbiter mounted long range TV system. The use of a Shuttle launch vehicle could allow the orbiter, after the combined mission is completed, to leave the Martian orbit and establish a trajectory which passes through a large portion of the asteroid belt.

F. SMALL INSTRUMENTED PROBES FOR COMBINED MISSIONS

An unguided, free-fall, hard lander which would determine elemental and mineralogical composition at one point on Phobos' surface was considered briefly during Phase III. Such a concept would be extremely attractive in a combined mission application.

Serious consideration should be given to this concept as well as other alternative concepts in any further study effort.

Appendix A

OBSERV PROGRAM DESCRIPTION

The OBSERV Program is used to determine the relative conditions between a spacecraft in Mars orbit and both Deimos and Phobos during the close encounters. The satellites are propagated using their ephemeris (MARSAT) data. The spacecraft is propagated from the input state vector using a conic propagation. A delta time interval is input into the program and the relative range to each of the satellites is computed. Whenever one of the two ranges is within the input desired maximum range the time increment is decreased by a factor of six and the relative state vector is evaluated to yield the relative range, relative velocity, and the angle from the sun-orientated spacecraft roll axis to the satellite. This information over many day intervals allows the evaluation of the relative observation merits of different orbits.

Appendix B

UD 208 PROGRAM DESCRIPTION

The UD 208 program is a point mass, three-D trajectory simulation which incorporates vehicle aerodynamic characteristics and propulsion simulations. The program is primarily designed for simulating atmospheric entry and powered flight, but is capable of in-vacuum two body trajectory simulation. Planet characteristics simulation includes a rotating spherical or oblate (J2) model with calculation of longitude and geocentric and geodetic latitude.

Initial conditions of velocity, flight path angle and heading may be input in either relative or inertial coordinate systems. Atmosphere model is computed internally and is generated by inputting surface pressure and altitude variations of temperature and molecular weight in tabular form. Wind models are also input in tabular form.

Vehicle aerodynamic lift and drag characteristics are input in the form of C_L and C_D as functions of mach number and angle of attack. Angle of attack and roll angle angle control is available through trim conditions or to any degree of complexity desired through guidance subroutines.

Output includes velocity, flight-path angle and heading angles in the initial (planet centered) coordinate system relative to the rotating planet surface and relative to the atmosphere, including winds. Position parameters available are altitude, latitude and longitude, down and cross range angles and ground traces, angle of attack and roll angle. Mach number, dynamic pressure, lift and drag accelerations, and lift and drag forces are also printed. All output parameters may be stored on a plot tape and plotted by a very flexible UD208 plot program.

The program is specifically designed for eas of input for multi-problem parametric studies. Any number of consecutive problems may be run with a minimum of input per subsequent problem. Each problem may contain any number of phases and provision is made to allow change in aerodynamic or mass properties between stages, such as parachutes and staging components of the vehicle. In the multiple problem cases, subsequent problems may be started at any phase with appropriate changes in vehicle characteristics possible. Propulsion options include, but are not limited to, the Viking terminal phase propulsion system simulation.