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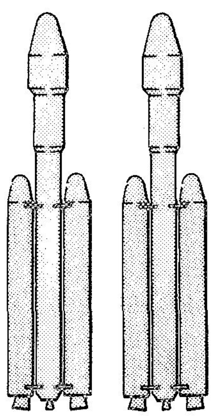
PRELIMINARY STUDY OF MINIMUM PERFORMANCE APPROACHES TO AUTOMATED MARS SAMPLE RETURN MISSIONS

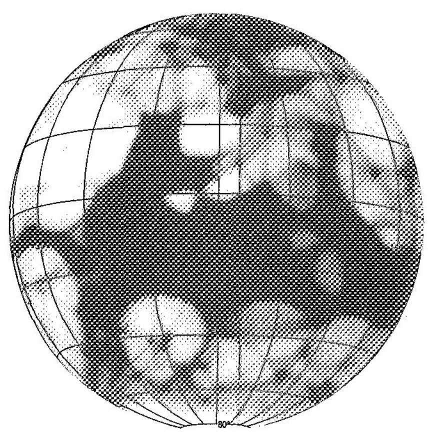
FINAL REPORT

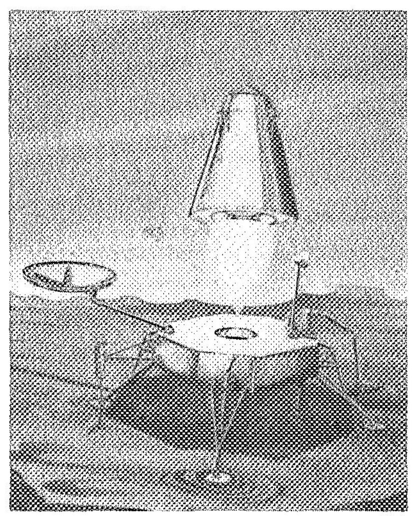
PREPARED FOR:

NATIONAL AERONAUTICS AND SPACE ADMINISTRATION
GEORGE C. MARSHALL SPACE FLIGHT CENTER
Huntsville, Alabama

UNDER CONTRACT NAS8-26656







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November 1970

by

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FOREWORD

This report documents the results of a preliminary five-week investigation of minimum performance approaches to automated Mars surface sample return missions. The study was conducted for the George C. Marshall Space Flight Center (MSFC) during the period 19 October 1970 to 20 November 1970. The contract called for a level of effort of approximately 800 man-hours. Mr. J. P. Hethcoat, PD-SA-P, was the NASA/MSFC Contracting Officer's Representative. Northrop wishes to acknowledge the helpful assistance provided by Messrs. Hethcoat and C. Guttman, PD-SA-P, throughout the study.

SUMMARY

A preliminary study was performed to identify and define promising alternative mission/system approaches to automated Mars surface sample return (MSSR) missions based on utilization of Titan III or Saturn Intermediate-20 (S-IC/S-IVB) class launch vehicles. The 1975, 1977, and 1979 launch opportunities were considered in conducting broad parametric performance analyses to establish gross earth departure weight requirements as a function of sample return weight for candidate mission modes employing all-chemical or solar-electric/chemical spacecraft propulsion. Three earth departure modes were considered: (1) single launch, (2) earth orbit rendezvous where the MSSR spacecraft is mated with a Centaur orbital transplanetary injection stage in earth orbit, and (3) multiple or dual interplanetary departure where the lander/ascent probe(s) and orbiter/bus return vehicle are launched to Mars as separate payloads.

Parametric mission/system performance data were generated using the computer program for all-chemical systems developed under Contract NAS8-24714, and a new program developed for the solar-electric/chemical systems. Based on a 10-pound sample return requirement and the Titan III and Intermediate-20 launch vehicles specified by NASA for consideration, a comparative evaluation of alternative mission/system approaches was performed to select three promising concepts for further preliminary point design analysis. The evaluation indicated that if the direct reentry/sample recovery mode is acceptable at Earth return, then the potentially lowest cost approach is a dual departure, all-chemical concept employing two Titan IIID/Centaur vehicles. If the orbital capture/sample recovery mode is groundruled in the mission, then two promising alternative approaches are: (1) a dual departure, solar-electric/chemical concept employing two Titan IIID/Centaur vehicles, or (2) a single launch, all-chemical concept using the Intermediate-20/Centaur vehicle. The all-chemical missions are based on conjunction class heliocentric profiles with mission durations of around 1000 days and Mars stopover times of roughly one Earth year. The solar-electric/chemical mission duration is of the order of 1000 days with a stopover time of typically 200 days, 150 days of which are spent in low-thrust spiral-out escape for earth return. All other planetocentric maneuvers are performed by chemical systems. The dual departure concepts employ direct

(hyperbolic) entry of the lander/ascent probe at Mars. The single launch, all-chemical concept employs an entry out-of-elliptical-orbit mode.

A preliminary point design analysis of each of the above three alternative concepts was performed to define the approaches in terms of mission profile, preliminary system configuration, weights summary, cursory development schedule, cursory costs, and summary of supporting technology requirements. The direct reentry/recovery, dual departure, all-chemical concept requires gross earth departure weights of 8500 and 9000 pounds for the lander/ascent probe and orbiter/bus return vehicle payloads, respectively, based on 1971 technology. The departure weight of the orbiter/bus vehicle in the orbital capture/recovery, solar-electric/chemical approach is 9300 pounds and the power required at 1 AU is approximately 15 kilowatts. The gross departure weight required for the single launch, all-chemical concept is approximately 34,000 pounds.

The concepts identified in the study are promising alternative approaches to automated MSSR missions in the 1975-1980 time period using the Titan IIID/Centaur or Intermediate-20/Centaur vehicles. The 1975 mission particularly would require maximum utilization of Viking/Mariner technology and hardware. A very intensive Phase A/Phase B program would be required in CY 1971 with a program commitment by January 1972. The 1977 mission schedule should also be based on a Phase A study in CY 1971.

The dual departure mission/system concepts offer mission flexibility and lander-orbiter/bus interface advantages over single launch concepts. The dual or multiple departure approach offers the possibility of using multiple lander/ascent probes to sample more than a single Mars site. The probability of mission success would be increased over a single probe approach. Further, the lander or orbiter/bus could perform missions as independent payloads. The probe ascent vehicle could be replaced by a 2000-pound class mobile surface laboratory and launched as a mission in itself. With regard to orbiter-lander interface, the dual departure concept minimizes performance and system interdependences between the lander/ascent probe and orbiter/bus vehicle.

Recommendations for follow-on activities based on this study are summarized in Section IX of the report.

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Section I
INTRODUCTION

A candidate concept for continuing the exploration of Mars in the 1970's and early 1980's is an automated system to return physical samples of the planetary surface to Earth. The automated return of an extraterrestrial sample has been demonstrated by the recently successful Soviet Luna 16 mission. The basic technologies required to acquire and return Mars samples with automated spacecraft will be available in the 1970's to both the United States and Russia. Whether or not the Mars surface sample return (MSSR) goal is pursued by either program will depend on the priority assigned to the mission on the basis of its merits scientifically and otherwise, including its contribution to international technological leadership.

The concept of a fully automated MSSR system based on use of the Saturn V class launch capability was studied by Northrop during the period July 1969 through February 1970 under Contract NAS8-24714 (refs. 1 through 4). The mission was considered as a potential early (1977-82) operational application of the Saturn V vehicle with a nuclear (NERVA) third stage. Comparison was made with an all-chemical Saturn V approach. Candidate mission/system concepts were developed and a baseline approach selected and defined. The baseline system performance would provide a total sample return capability of approximately 160 pounds employing an integrated orbiter/lander/rover concept with dual lander/ascent probes. The orbiter would act as a high data rate relay and provide planet-wide synoptic imagery and multispectral mapping. These data would be used with the ground truth established by the dual landers and rovers to biologically, geophysically, and meteorologically characterize the planet's surface and atmosphere.

The increasing Soviet emphasis on sophisticated automated exploration has prompted the study reported in this document. There is renewed NASA interest in MSSR mission possibilities in the mid to late 1970's, particularly using smaller systems and launch vehicles to reduce the cost. The sections to follow present the results of a preliminary investigation to identify and define promising minimum performance (10-pound sample return) approaches to the MSSR mission. Emphasis is on analysis of the 1975 mission possibility using 1971 technology and the Titan III and Saturn Intermediate-20 (S-IC/S-IVB) classes of launch vehicles with and without a Centaur upper stage.

Section II summarizes the study objectives and guidelines and outlines the technical approach. Section III defines the mission/system alternative approaches considered. Section IV defines the flight profile characteristics and associated mission requirements for the candidate mission/system alternatives. Section V defines and describes the system design approaches and weight scaling data employed in the parametric performance analyses. Section VI presents and discusses the results of a broad parametric performance analysis of the alternative mission/system concepts under consideration. Section VII gives a comparative evaluation of the mission/system alternative approaches and identifies the most promising concepts. Section VIII summarizes the results of preliminary point design analyses of the three most promising concepts selected in Section VII. A mission profile summary, brief system description, weights summary, cursory program schedule, cursory costs estimate, and identification of supporting technology requirements are given for each selected concept. Section IX completes the report with a summary of the conclusions based on the study results and gives specific recommendations for follow-on activities.

Section II

STUDY OBJECTIVES, GUIDELINES, AND TECHNICAL APPROACH

2.1 OBJECTIVES

The objectives of the present investigation are as follows:

- Parametrically determine minimum-weight MSSR systems for candidate mission modes employing all-chemical or solar-electric/chemical propulsion
- Establish candidate launch vehicles which match required mission capability
- Identify significant performance trades and mission/system design approaches which minimize weight and performance requirements
- Define and recommend alternative mission/system concepts which should be studied in a follow-on activity
- Provide cursory program schedule and cost estimates for most promising alternatives.

2.2 GUIDELINES

The overall study guidelines and constraints specified by NASA were as follows:

- The 1975, 1977, and 1979 Mars launch opportunities are to be considered.
- The Titan III family of launch vehicles and selected Saturn V derivatives such as the Intermediate 20/Centaur are to be considered as directed by NASA.
- The primary mission objective will be to return a minimum Mars surface sample to Earth. Mission science will be minimized to only that essential to support the primary objective.
- Solar electric systems are to be considered for primary spacecraft propulsion in addition to all-chemical systems approaches.
- New mission modes which may be introduced by Titan class launch vehicle performance constraints shall be considered such as:
  - Dual launches to Mars with the lander/ascent probe and return orbiter/bus carried as separate payloads.
  - Dual launches to Earth orbit for mating of MSSR spacecraft and departure injection stage.
- Sterilization should be enforced at Mars through the 20-year period beginning 1 January 1969.

2.3 TECHNICAL APPROACH

The technical approach is shown in Figure 2-1. The study was performed in four major tasks. These are briefly discussed in the following paragraphs.

Task A. Definition of Mission/System Alternatives and Assembly of Mission Mode and System Characteristics Data

This task required, first, the definition of the mission/system alternatives which appeared to warrant consideration in the study. The results of the work performed in Contract NAS8-24714 were used as a point of departure to establish the principal mission mode alternatives which offer the best potential for Earth launch performance limited systems. Two new mission mode alternatives were introduced through consideration of limited Earth launch capability: (1) multiple interplanetary launches to Mars with the lander/ascent probe(s) and return orbiter/bus carried as separate payloads; and (2) dual launches to Earth orbit for mating of the MSSR spacecraft and departure injection stage (Earth orbit rendezvous mode).

As will be described in Section III, eight categories of system concept alternatives were identified including four all-chemical propulsion concepts and four solar-electric/chemical propulsion concepts. A preliminary analysis of mission characteristics and system functional requirements was performed to define the major spacecraft elements for each of the eight categories of system concept alternatives.

Based on the results of the above analysis, mission and system design characteristics data required for parametric performance evaluation were assembled and analyzed for each of the identified mission/system alternatives. Maximum utilization of available existing mission and systems data was made. The data base was built primarily around the results of Northrop's work under NAS8-24714; available information on Mariner '69, Mariner '71, and Viking; Voyager Phase B studies; and past solar-electric mission and spacecraft studies documented in the open literature. Two technology bases were considered in establishing weights and weight scaling parameters for the performance analyses: (1) 1971 technology dictated by the 1975 launch opportunity mission, and (2) 1974 technology consistent with the 1977 and 1979 launch opportunities.

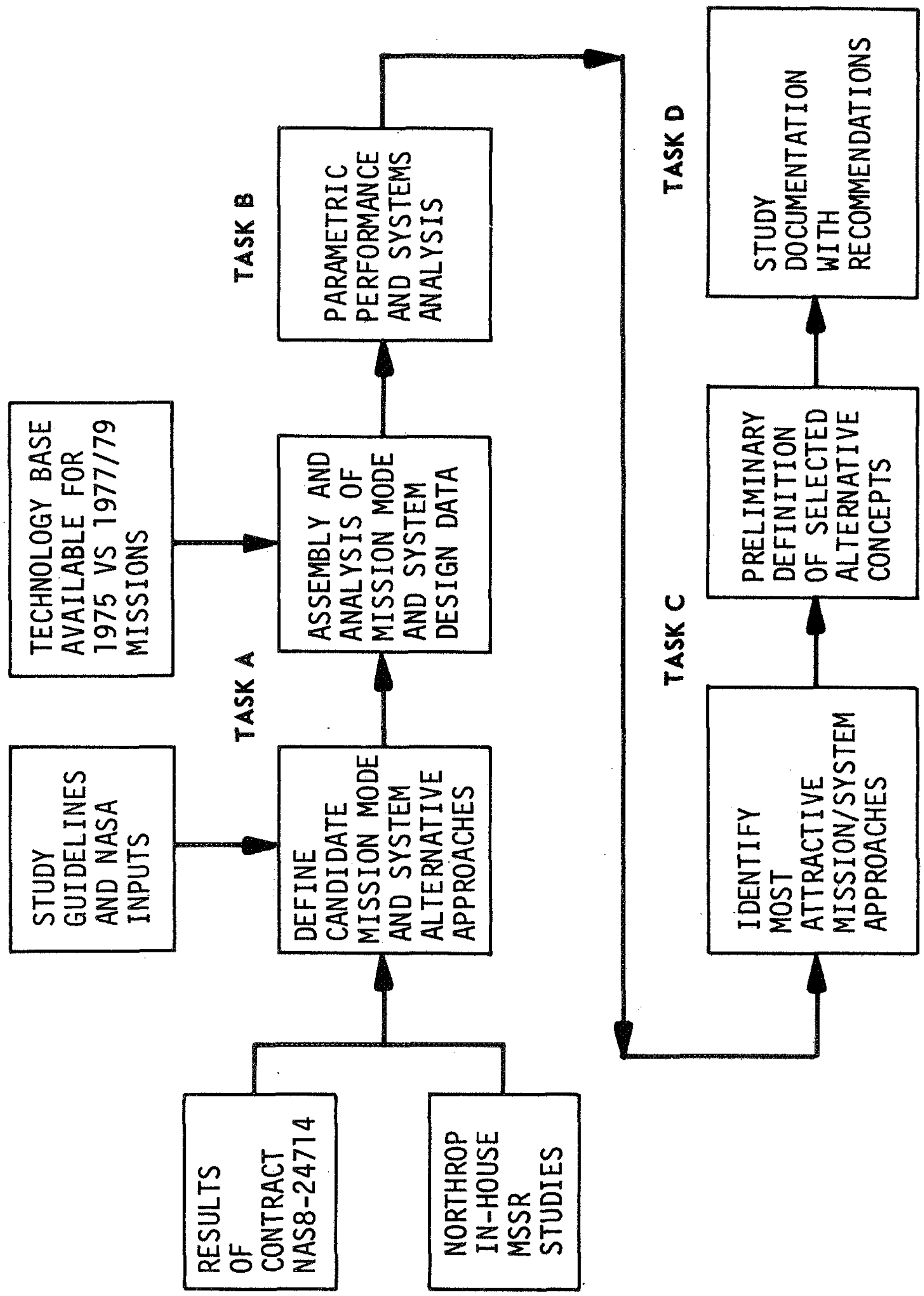


Figure 2-1. STUDY APPROACH

Two MSSR mission/system performance evaluation models were employed in the study. First, the versatile performance evaluation computer program developed under NAS8-24714 was modified to handle the additional all-chemical system mission modes considered in the present study. This program was then used for generation of performance data for the all-chemical mission/system concepts. For the solar-electric/chemical propulsion concepts, a new computer program model was established. Both of these computer models were set up for use in the parametric studies and for use in iterative analysis of preliminary point designs of selected systems.

Task B. Parametric Performance and Systems Analysis

The data and performance models prepared in Task A were used to conduct a series of parametric mission performance analyses covering the eight categories of system concept alternatives identified in Task A. Weight requirements for all major spacecraft system elements were generated for sample return weights from 0 to 50 pounds. Parametric data were generated to show overall performance and weight trades for particular mission/system design parameters of interest including: 1971 technology base versus 1974 technology base; effect of Mars entry mode; effect of adding additional lander science weight; effect of adding additional orbiter science; effect of solar array and ion thruster module specific weights (solar-electric propulsion concepts), etc. Parametric mission/system performance data were prepared in graphical form such that major alternative approaches can be compared and matched to selected Titan III family and Saturn V derivative launch vehicle capabilities. The January 1970 Launch Vehicle Estimating Factors book published by NASA/OSSA (ref. 5) was used as a baseline reference for candidate launch vehicle performance capabilities. Additional launch vehicle data were supplied by NASA/MSFC as required. The vehicles of primary interest are the Titan IIID/Centaur, Intermediate-20, (S-IC/S-IVB), and Intermediate 20/Centaur. The Titan IIID was considered for the Earth orbit rendezvous concepts, and the possibility of using the Space Shuttle with a Centaur transmars injection stage was also considered. The required Mars entry/lander probe diameters were compared to the Titan III/Centaur payload shroud envelope given in reference 5 and the Intermediate-20 shroud configuration supplied by NASA/MSFC.

Throughout the above analyses, iterations were made between the computer performance models and relatively detailed system weights and configuration sizing checks. This approach was used to ensure realistic trends in the parametric data. This was of particular importance in the cases based on the 1971 technology base (1975 launch opportunity). Here, for example, propulsion module mass fractions were not as "rubberized" as for the advanced technology modules using new development engines. The 1971 technology vehicles were based on use of existing engines with defined weights and design characteristics.

Task C. Selection of Most Promising Candidate Mission/System Approaches

This task was to evaluate the results of tasks A and B and select the candidate mission/system approaches that appear to warrant more detailed study. The three most promising alternative concepts were defined and summarized in terms of a mission profile summary, brief system description, weights summary, cursory program schedule, cursory costs estimate, and identification of supporting technology requirements. The above summaries were based on a brief preliminary point design analysis of each selected concept.

Task D. Documentation

The final task was the study documentation and preparation of briefing materials. Conclusions were summarized on the basis of overall study results and recommendations prepared to provide a basis for follow-on study activities.

Section III

MISSION/SYSTEM ALTERNATIVES

This section defines the alternative mission/system approaches considered in the present study.

3.1 MISSION AND SYSTEM OPTIONS

The principal mission/system options which can be considered for automated MSSR concepts are summarized in Table 3-1. The options that are not self explanatory are briefly defined in the following subsections.

3.1.1 Earth Launch/Departure Mode

3.1.1.1 Single Launch. In this mode, the MSSR spacecraft is launched with a single Earth launch vehicle.

3.1.1.2 Multiple Interplanetary Launch. In this departure mode, the lander/return (ascent) probe and orbiter/bus return vehicle are launched as separate payloads onto separate Earth-Mars trajectories. In this concept, more than one lander/ascent probe could be launched and returned by a single orbiter/bus return vehicle.

3.1.1.3 Earth Orbit Rendezvous. In this approach, the MSSR spacecraft and transmars injection stage are launched into low Earth orbit by separate launch vehicles. The spacecraft and injection stage perform rendezvous and dock in parking orbit. The spacecraft is injected onto the transplanetary escape trajectory by the injection stage.

3.1.2 Heliocentric Profile

3.1.2.1 Conjunction Class. The Conjunction Class heliocentric profile is characterized by low-energy transfer trajectories both outbound Earth-to-Mars and inbound Mars-to-Earth. Mars arrival generally occurs relatively near Earth/Mars conjunction. The Mars stopover time must be of the order of an

Table 3-1. MSSR MISSION/SYSTEM OPTIONS

(1) Earth Launch/Departure Mode	(a) Single Launch (b) Multiple Interplanetary Launch (c) Earth Orbit Rendezvous
(2) Earth Launch Vehicle	(a) Titan IIIC (b) Titan IIID (c) Titan IIID/Centaur (d) INT-20 (S-IC(4-F1)/S-IVB) (e) INT-20/Centaur (f) Shuttle/Centaur
(3) Heliocentric Profile	(a) Conjunction Class (b) Opposition Class (Direct and Venus Swingby)
(4) SEP Spacecraft Planetocentric Capture/Escape Modes	(a) Chemical Propulsion (b) Spiral Out/In (c) Combination
(5) Planetary Vehicle Concept	(a) Direct Probe (b) Orbiter/Bus-Probe
(6) Orbiter/Bus Primary Propulsion	(a) All-Chemical (b) Solar-Electric/Chemical
(7) Number Lander/Return Probes	(a) Single (b) Dual
(8) Mars Entry Mode	(a) Direct (Hyperbolic) (b) Out-of-Circular Orbit (c) Out-of-Elliptical Orbit
(9) Entry Concept	(a) Ballistic (b) Lifting (Offset C.G.)
(10) Mars Descent/Landing System	(a) Full Aerobraking (b) Full Propulsion (c) Aerobraking with Terminal Propulsion
(11) Mars Surface Rover	(a) None (b) Lander-Dependent Class (c) Lander-Independent Class
(12) Mars Ascent/Earth Return Mode	(a) Direct Ascent to Escape (b) Direct Return Via Mars Parking Orbit (c) Return Via Mars Orbit Rendezvous
(13) Earth Intercept/Recovery Mode	(a) Direct Reentry/Recovery (b) Orbital Capture/Recovery

Earth year to achieve low-energy transfer trajectories both outbound and inbound. The required stopover time results in total mission durations of the order of 1000 days.

3.1.2.2 Opposition Class. Opposition Class missions are characterized by 35 to 60 percent less total mission duration than the conjunction class missions but at the expense of less favorable velocity characteristics. The outbound and inbound trajectories are direct in the so-called Standard Opposition Class missions. The high energy characteristics of this class of mission make the Standard Opposition profiles generally unattractive as a candidate mode for MSSR. Stopover times at Mars are generally 10 to 30 days for Standard Opposition missions.

Depending on mission opportunity, either the outbound or inbound trajectory of an opposition class profile may permit use of a Venus swingby to improve the energy characteristics of the mission for roughly a 25 percent increase in total mission duration. The total duration is typically of the order of 600 days.

3.1.3 Solar-Electric Spacecraft Planeocentric Capture/Escape Modes

3.1.3.1 Chemical Propulsion. In this option, the interplanetary transfers outbound and inbound are performed with solar-electric propulsion. The escape at Earth is performed by the Earth launch vehicle. The capture and escape maneuvers at Mars and the capture maneuver at Earth return (if the orbital capture/recovery mode is employed) are performed with chemical propulsion systems.

3.1.3.2 Spiral Out/In. In this option, the escape and capture maneuvers at Earth and Mars are all performed by low-thrust spiraling about the planet to either build energy for escape or decrease energy relative to the planet for capture.

3.1.3.3 Combination. A combination of the above two modes can be considered. A combination of particular interest is chemical escape from Earth, chemical capture at Mars, spiral-out escape at Mars, and chemical capture at Earth.

3.1.4 Planetary Vehicle Concept

3.1.4.1 Direct Probe. In the Direct Probe concept, the entire spacecraft lands on the Mars surface and returns to Earth directly from the surface.

3.1.4.2 Orbiter/Bus-Probe. The Orbiter/Bus-Probe alternative introduces an orbiting element into the system. In the Mars orbit rendezvous Earth return mode, the orbiter/bus provides the return transportation. The orbiter can also support the Mars surface mission from orbit as a communications relay and surface mapper.

3.1.5 Mars Entry Mode

3.1.5.1 Direct (Hyperbolic). In this mode,the lander/return probe enters the Mars atmosphere directly from the hyperbolic approach trajectory.

3.1.5.2 Out-of-Circular Orbit. The probe, in this mode, is carried into circular capture orbit by the Mars braking stage of the orbiter/bus vehicle. The probe is separated in orbit for de-orbit and entry.

3.1.5.3 Out-of-Elliptical Orbit. In this mode, the probe is carried into an initial elliptical capture orbit and is separated for de-orbit and entry. The orbiter/bus vehicle later adjusts down to a circular operational orbit to support the surface mission.

Obviously, not all combinations of the numerous options in Table 3-1 are feasible based on weight and performance considerations. In the present study, emphasis was placed on selection of approaches which offer feasible missions within the launch capability of the Titan III and Intermediate-20 (INT-20) class vehicles. An initial screening of mission/system options was made based on the results of work under Contract NAS8-24714 (refs. 1 through 4). This screening resulted in selection of the following mission/system options for analysis:

- All three earth launch/departure modes shown in Table 3-1
- Consider all earth launch vehicles shown in Table 3-1

- Conjunction class missions; consider Venus swingby missions for INT-20/ Centaur class vehicle (all-chemical spacecraft propulsion)
- For Solar-Electric Propulsion (SEP) Concepts:
  Low thrust heliocentric transfer; chemical propulsion for planetocentric escape/capture except consider SEP spiral-out Mars escape as an alternative
- Earth return via the Mars orbit rendezvous (MOR) mode
- Orbiter/bus-probe planetary vehicle concept (Orbiter/bus needed for MOR mode)
- Consider both all-chemical and solar-electric/chemical propulsion alternatives for the orbiter/bus
- Single lander/return (ascent) probe
- Direct (hyperbolic) entry at Mars
- Lifting (offset c.g.) Mars entry
- Aerobraking with terminal propulsion for Mars descent/landing
- Small (150-pound) lander-independent class rover
- Consider both direct reentry and orbital capture Earth recovery modes.

The out-of-elliptical and out-of-circular orbit entry modes at Mars were carried through the parametric performance analyses as alternatives for comparison to the direct mode.

3.2 SUMMARY OF CANDIDATE MISSION/SYSTEM CONCEPT ALTERNATIVES

On the basis of the options defined in the preceding subsections, eight categories of MSSR spacecraft concepts were defined covering all the baseline mission mode and major system options of interest. These eight concepts are summarized in Figure 3-1 in terms of Earth departure mode, orbiter/bus primary propulsion concept, Earth recovery mode, and the various major system elements required by the given concept based on an analysis of functional mission requirements. The first four concepts listed in Figure 3-1 are all-chemical systems. The last four are solar-electric/chemical systems. Solar-electric propulsion is used only in the orbiter/bus vehicle since the lander/return (ascent) probe must necessarily employ chemical propulsion systems.

In order to give a quicker grasp of the eight categories of spacecraft systems under consideration, schematic illustrations of each concept are given in Figures 3-2 through 3-9. Each figure gives: (1) a system concept sketch which identifies the major spacecraft system elements, (2) a mission profile summary description, and (3) an identification of potential candidate Earth launch vehicles.

The definitions of mission/system alternatives given in this section serve as a basic framework for the analyses and presentation of study results in the sections to follow.

S/C CONCEPT	EARTH DEPARTURE MODE	ORBITER/BUS PROPULSION	EARTH RECOVERY MODE	LANDER/ RETURN PROBE	SPACECRAFT BUS	ORBITER/ BUS MODULE	MARS BRAKING STAGE	MARS DEPARTURE STAGE	EARTH BRAKING STAGE	EARTH REENTRY CAPSULE	SEP MODULE	EOR DOCKING ADAPTER	ELV ADAPTER
1	SINGLE LAUNCH	ALL-CHEM	DIRECT	X		X	X	X		X			X
2	SINGLE LAUNCH	ALL-CHEM	CAPTURE	X		X	X	X	X				X
3	EOR	ALL-CHEM	DIRECT	X		X	X	X		X		X	
4	MULTIPLE DEPARTURE	ALL-CHEM	DIRECT	X(1)	X	X	X	X		X			X
5	SINGLE LAUNCH	CHEM/SEP	DIRECT	X		X	X(2)	X(2)		X	X		X
6	SINGLE LAUNCH	CHEM/SEP	CAPTURE	X		X	X(2)	X(2)	X		X		X
7	EOR	CHEM/SEP	CAPTURE	X		X	X(2)	X(2)	X		X	X	
8	MULTIPLE DEPARTURE	CHEM/SEP	CAPTURE	X(1)	X	X	X(2)	X(2)	X		X		X

(1) CONCEPT MAY INCLUDE MULTIPLE INTERPLANETARY LANDER/RETURN (ASCENT) PROBE LAUNCHES (ONE PROBE PER LAUNCH)

(2) MARS BRAKING AND DEPARTURE STAGES MAY BE COMBINED.

Figure 3-1. SPACECRAFT CONCEPT DEFINITION MATRIX

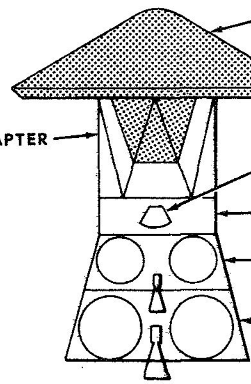
SYSTEM CONCEPT SCHEMATIC	BASELINE MISSION PROFILE SUMMARY	CANDIDATE LAUNCH VEHICLES
	EARTH DEPARTURE: SINGLE LAUNCH THROUGH 185-KM PARKING ORBIT HELIOCENTRIC PROFILE: CONJUNCTION CLASS OR VENUS SWINGBY MARS ENTRY: DIRECT OFF HYPERBOLIC APPROACH TRAJECTORY; PROBE SEPARATES FROM VEHICLE AND PERFORMS SMALL DEFLECTION MANEUVER. MARS CAPTURE: SINGLE IMPULSE CAPTURE INTO CIRCULAR ORBIT USING MARS BRAKING STAGE; EMPTY STAGE JETTISONED IN ORBIT. PROBE DESCENT/LANDING: AEROBRAKING WITH TERMINAL PROPULSION SURFACE MISSION: ACQUIRE SAMPLES AND ENVIRONMENTAL DATA; EXPLORE AREA AROUND LANDER AND SAMPLE WITH ROVER; STAY TIME 300-400 DAYS FOR CONJUNCTION CLASS MISSION OR 1 TO 30 DAYS FOR VENUS SWINGBY MISSION. ORBIT OPERATIONS: ORBITER/BUS OPERATES AS COMMUNICATIONS RELAY AND SURFACE MAPPER. PROBE ASCENT/RENDEZVOUS: ON STORED COMMANDS FROM EARTH, PROBE ASCENT VEHICLE PERFORMS ASCENT THROUGH PHASING ORBIT TO ORBITER/BUS STABLE ORBIT. ORBITER/BUS PERFORMS RENDEZVOUS WITH PROBE, DOCKS, RETRIEVES SAMPLE CANISTER INTO EARTH REENTRY CAPSULE AND JETTISONS PROBE ASCENT VEHICLE STAGE. MARS DEPARTURE: MARS DEPARTURE MANEUVER PERFORMED BY THE MARS DEPARTURE STAGE; STAGE ALSO PROVIDES PROPULSION FOR MARS ORBIT RENDEZVOUS AND INBOUND MIDCOURSE CORRECTIONS. EARTH INTERCEPT/RECOVERY: THE EARTH REENTRY CAPSULE IS SEPARATED ON EARTH APPROACH TRAJECTORY AND PERFORMS A DIRECT REENTRY. RECOVERY IS BY AIR SNATCH WITH WATER RECOVERY BACKUP.	• INTERMEDIATE 20 OR SHUTTLE/CENTAUR FOR CONJUNCTION CLASS MISSIONS • INTERMEDIATE 20/CENTAUR FOR VENUS SWINGBY MISSIONS

Figure 3-2. ALL-CHEMICAL, SINGLE LAUNCH, DIRECT REENTRY/RECOVERY CONCEPT

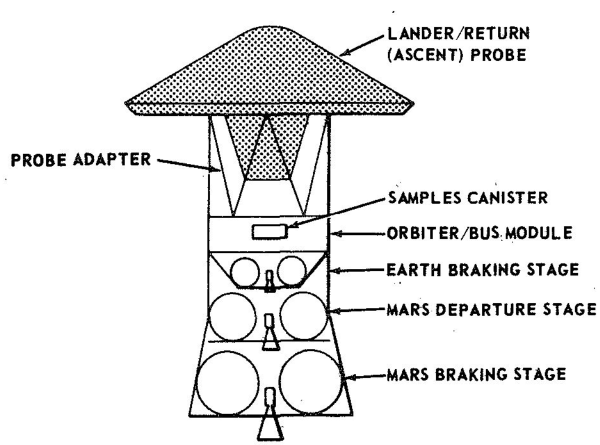
SYSTEM CONCEPT SCHEMATIC	BASELINE MISSION PROFILE SUMMARY	CANDIDATE LAUNCH VEHICLES
	EARTH DEPARTURE: SINGLE LAUNCH THROUGH 185-KM PARKING ORBIT HELIOCENTRIC PROFILE: CONJUNCTION CLASS MARS ENTRY: DIRECT OFF HYPERBCLIC APPROACH TRAJECTORY; PROBE SEPARATES FROM VEHICLE AND PERFORMS SMALL DEFLECTION MANEUVER. MARS CAPTURE: SINGLE IMPULSE CAPTURE INTO CIRCULAR ORBIT USING MARS BRAKING STAGE; EMPTY STAGE JETTISONED IN ORBIT. PROBE DESCENT/LANDING: AEROBRAKING WITH TERMINAL PROPULSION SURFACE MISSION: ACQUIRE SAMPLES AND ENVIRONMENTAL DATA; EXPLORE AREA AROUND LANDER AND SAMPLE WITH ROVER; STAY TIME 300-400 DAYS FOR CONJUNCTION CLASS MISSION ORBIT OPERATIONS: ORBITER/BUS OPERATES AS COMMUNICATIONS RELAY AND SURFACE MAPPER. PROBE ASCENT/RENDEZVOUS: ON STORED COMMANDS FROM EARTH, PROBE ASCENT VEHICLE PERFORMS ASCENT THROUGH PHASING ORBIT TO ORBITER/BUS STABLE ORBIT. ORBITER/BUS PERFORMS RENDEZVOUS WITH PROBE, DOCKS, RETRIEVES SAMPLE CANISTER INTO EARTH REENTRY CAPSULE AND JETTISONS PROBE ASCENT VEHICLE STAGE. MARS DEPARTURE: MARS DEPARTURE MANEUVER PERFORMED BY THE MARS DEPARTURE STAGE; STAGE ALSO PROVIDES PROPULSION FOR MARS ORBIT RENDEZVOUS. STAGE IS JETTISONED AFTER MARS DEPARTURE MANEUVER. EARTH INTERCEPT/RECOVERY: SINGLE IMPULSE EARTH CAPTURE MANEUVER INTO ELLIPTICAL ORBIT PERFORMED BY THE EARTH BRAKING STAGE; STAGE ALSO PROVIDES PROPULSION FOR INBOUND MIDCOURSE CORRECTIONS. SAMPLES CANISTER, FILM CASSETTES, ETC, ARE RETRIEVED IN ORBIT BY MANNED VEHICLE AND TRANSPORTED TO SPACE STATION/BASE FOR PRELIMINARY ANALYSIS.	INTERMEDIATE 20/CENTAUR

Figure 3-3. ALL-CHEMICAL, SINGLE LAUNCH, EARTH CAPTURE/RECOVERY CONCEPT

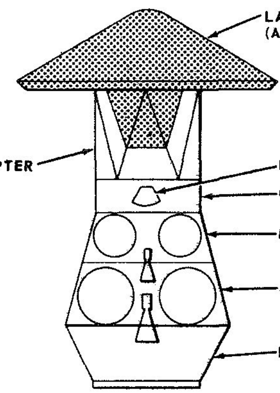
SYSTEM CONCEPT SCHEMATIC	BASELINE MISSION PROFILE SUMMARY	CANDIDATE LAUNCH VEHICLES
	EARTH DEPARTURE: EARTH ORBIT RENDEZVOUS MODE; THE MSSR SPACECRAFT AND TRANSMARS INJECTION STAGE ARE LAUNCHED INTO EARTH PARKING ORBIT WITH SEPARATE LAUNCHERS. THE SPACECRAFT PERFORMS RENDEZVOUS AND DOCKS WITH THE INJECTION STAGE WHICH IS USED TO INJECT THE SPACECRAFT ONTO THE TRANSMARS ESCAPE TRAJECTORY. HELIOCENTRIC PROFILE: CONJUNCTION CLASS MARS ENTRY: DIRECT OFF HYPERBOLIC APPROACH TRAJECTORY; PROBE SEPARATES FROM VEHICLE AND PERFORMS SMALL DEFLECTION MANEUVER. MARS CAPTURE: SINGLE IMPULSE CAPTURE INTO CIRCULAR ORBIT USING MARS BRAKING STAGE; EMPTY STAGE JETTISONED IN ORBIT. PROBE DESCENT/LANDING: AEROBRAKING WITH TERMINAL PROPULSION SURFACE MISSION: ACQUIRE SAMPLES AND ENVIRONMENTAL DATA; EXPLORE AREA AROUND LANDER AND SAMPLE WITH ROVER; STAY TIME 300-400 DAYS FOR CONJUNCTION CLASS MISSION ORBIT OPERATIONS: ORBITER/BUS OPERATES AS COMMUNICATIONS RELAY AND SURFACE MAPPER. PROBE ASCENT/RENDEZVOUS: ON STORED COMMANDS FROM EARTH, PROBE ASCENT VEHICLE PERFORMS ASCENT THROUGH PHASING ORBIT TO ORBITER/BUS STABLE ORBIT. ORBITER/BUS PERFORMS RENDEZVOUS WITH PROBE, DOCKS, RETRIEVES SAMPLE CANISTER INTO EARTH REENTRY CAPSULE AND JETTISONS PROBE ASCENT VEHICLE STAGE. MARS DEPARTURE: MARS DEPARTURE MANEUVER PERFORMED BY THE MARS DEPARTURE STAGE; STAGE ALSO PROVIDES PROPULSION FOR MARS ORBIT RENDEZVOUS AND INBOUND MIDCOURSE CORRECTIONS. EARTH INTERCEPT/RECOVERY: THE EARTH REENTRY CAPSULE IS SEPARATED ON EARTH APPROACH TRAJECTORY AND PERFORMS A DIRECT REENTRY. RECOVERY IS BY AIR SNATCH WITH WATER RECOVERY BACKUP.	• TRANSMARS INJECTION STAGE: CENTAUR • LAUNCH TO EARTH ORBIT: TITAN III C FOR SPACECRAFT LAUNCH; TITAN III D/CENTAUR FOR INJECTION STAGE LAUNCH. ALTERNATIVE IS TWO LAUNCHES OF SPACE SHUTTLE (PARTIAL LOADS).

Figure 3-4. ALL-CHEMICAL, EARTH ORBIT RENDEZVOUS, DIRECT REENTRY/RECOVERY CONCEPT

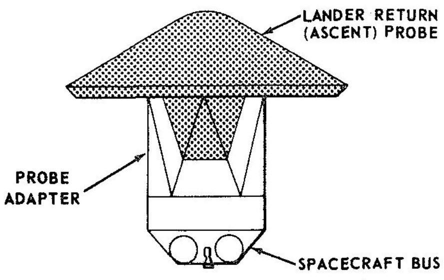
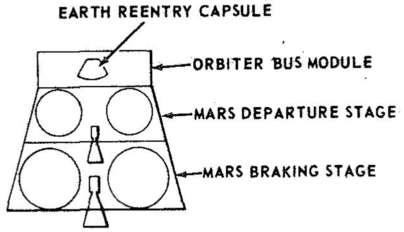
SYSTEM CONCEPT SCHEMATIC	BASELINE MISSION PROFILE SUMMARY	CANDIDATE LAUNCH VEHICLES
LAUNCH NO. 1 LANDER/RETURN (ASCENT) PROBE  LAUNCH NO. 2 ORBITER/BUS RETURN VEHICLE  	EARTH DEPARTURE: LANDER/RETURN (ASCENT) PROBE AND ORBITER/BUS RETURN VEHICLE ARE LAUNCHED AS SEPARATE PAYLOADS ONTO SEPARATE EARTH-MARS TRAJECTORIES; EACH DEPARTURE IS THROUGH 185-KM PARKING ORBIT. HELIOCENTRIC PROFILE: CONJUNCTION CLASS MARS ENTRY: DIRECT OFF HYPERBOLIC APPROACH TRAJECTORY; SPACECRAFT BUS PERFORMS SMALL DEFLECTION MANEUVER AND FLIES PAST MARS INTO HELIOCENTRIC ORBIT. MARS CAPTURE: SINGLE IMPULSE CAPTURE INTO CIRCULAR ORBIT USING MARS BRAKING STAGE; EMPTY STAGE JETTISONED IN ORBIT. PROBE DESCENT/LANDING: AEROBRAKING WITH TERMINAL PROPULSION SURFACE MISSION: ACQUIRE SAMPLES AND ENVIRONMENTAL DATA; EXPLORE AREA AROUND LANDER AND SAMPLE WITH ROVER; STAY TIME 300-400 DAYS FOR CONJUNCTION CLASS MISSION ORBIT OPERATIONS: ORBITER/BUS OPERATES AS COMMUNICATIONS RELAY AND SURFACE MAPPER. PROBE ASCENT/RENDEZVOUS: ON STORED COMMANDS FROM EARTH, PROBE ASCENT VEHICLE PERFORMS ASCENT THROUGH PHASING ORBIT TO ORBITER/BUS STABLE ORBIT. ORBITER/BUS PERFORMS RENDEZVOUS WITH PROBE, DOCKS, RETRIEVES SAMPLE CANISTER INTO EARTH REENTRY CAPSULE AND JETTISONS PROBE ASCENT VEHICLE STAGE. MARS DEPARTURE: MARS DEPARTURE MANEUVER PERFORMED BY THE MARS DEPARTURE STAGE; STAGE ALSO PROVIDES PROPULSION FOR MARS ORBIT RENDEZVOUS AND INBOUND MIDCOURSE CORRECTIONS. EARTH INTERCEPT/RECOVERY: THE EARTH REENTRY CAPSULE IS SEPARATED ON EARTH APPROACH TRAJECTORY AND PERFORMS A DIRECT REENTRY. RECOVERY IS BY AIR SNATCH WITH WATER RECOVERY BACKUP.	TWO TITAN III D/CENTAUR VEHICLES NOTE: THIS MISSION/SYSTEM CONCEPT IS ADAPTABLE TO CONSIDERATION OF DUAL LANDER/ASCENT PROBES WITH A SINGLE ORBITER/BUS RETURN VEHICLE. THE DUAL PROBE CONCEPT WOULD REQUIRE AN ADDITIONAL TITAN III D/CENTAUR VEHICLE.

Figure 3-5. ALL-CHEMICAL, MULTIPLE INTERPLANETARY LAUNCH, DIRECT REENTRY/RECOVERY CONCEPT

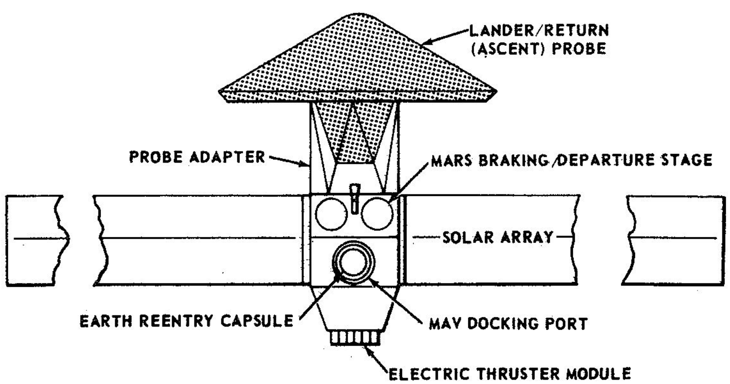
SYSTEM CONCEPT SCHEMATIC	BASELINE MISSION PROFILE SUMMARY	CANDIDATE LAUNCH VEHICLES
	EARTH DEPARTURE: SINGLE LAUNCH WITH DEPARTURE THROUGH 185-KM PARKING ORBIT HELIOCENTRIC PROFILE: CONJUNCTION OR OPPOSITION CLASS; LOW-THRUST SEP TRANSFERS OUTBOUND AND IN-BOUND MARS ENTRY: DIRECT OFF HYPERBOLIC APPROACH TRAJECTORY; PROBE SEPARATES FROM VEHICLE AND PERFORMS SMALL DEFLECTION MANEUVER. MARS CAPTURE: CAPTURE MANEUVER INTO CIRCULAR ORBIT IS PER-FORMED BY CHEMICAL PROPULSION MARS BRAKING/DEPARTURE STAGE. SOLAR ARRAYS ARE RETRACTED PRIOR TO MANEUVER AND RE-DEPLOYED IN MARS ORBIT. PROBE DESCENT/LANDING: AEROBRAKING WITH TERMINAL PROPULSION SURFACE MISSION: ACQUIRE SAMPLES AND ENVIRONMENTAL DATA; EXPLORE AREA AROUND LANDER AND SAMPLE WITH ROVER; STAY TIME 100-200 DAYS FOR CONJUNCTION CLASS MISSION OR 1 TO 30 DAYS FOR OPPOSITION CLASS MISSION. ORBIT OPERATIONS: ORBITER/BUS OPERATES AS COMMUNICATIONS RELAY AND SURFACE MAPPER. HIGH POWER AVAILABLE FROM SEP SYSTEM SOLAR ARRAYS PROBE ASCENT/RENDEZVOUS: ON STORED COMMANDS FROM EARTH, PROBE ASCENT VEHICLE PERFORMS ASCENT THROUGH PHASING ORBIT TO ORBITER/BUS STABLE ORBIT. ORBITER/BUS PERFORMS RENDEZVOUS WITH PROBE, DOCKS, RETRIEVES SAMPLE CANISTER INTO EARTH REENTRY CAPSULE AND JETTISONS PROBE ASCENT VEHICLE STAGE. MARS DEPARTURE: MARS DEPARTURE MANEUVEP PERFORMED BY CHEMICAL BRAKING/DEPARTURE STAGE; STAGE ALSO PROVIDES PROPULSION FOR MARS ORBIT RENDEZVOUS. STAGE IS JETTISONED AFTER DEPARTURE MANEUVER. ALTERNATIVE DEPARTURE MODE IS TO JETTISON CHEMICAL STAGE AFTER RENDEZVOUS/DOCKING IN ORBIT AND USE SEP FOR SPIRAL-OUT ESCAPE. EARTH INTERCEPT/RECOVERY: THE EARTH REENTRY CAPSULE IS SEPARATED ON THE EARTH APPROACH TRAJECTORY AND PERFCRMS A DIRECT REENTRY. RECOVERY IS BY AIR SNATCH WITH WATER RECOVERY BACKUP.	● INTERMEDIATE 20 ● SHUTTLE/CENTAUR

Figure 3-6. SOLAR-ELECTRIC/CHEMICAL, SINGLE LAUNCH, DIRECT REENTRY/RECOVERY CONCEPT

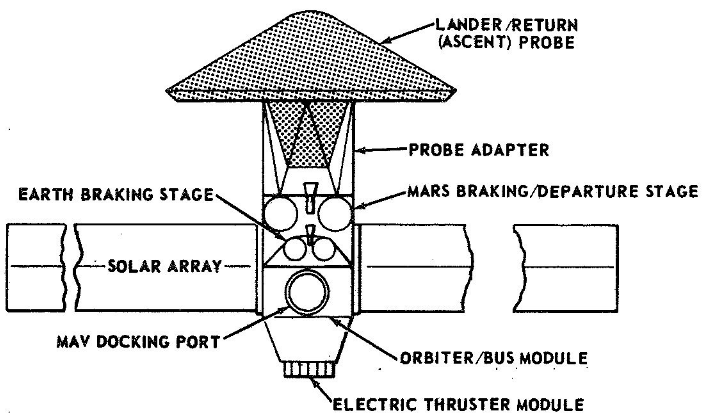
SYSTEM CONCEPT SCHEMATIC	BASELINE MISSION PROFILE SUMMARY	CANDIDATE LAUNCH VEHICLES
	EARTH DEPARTURE: SINGLE LAUNCH WITH DEPARTURE THROUGH 185-KM PARKING ORBIT HELIOCENTRIC PROFILE: CONJUNCTION OR OPPOSITION CLASS; LOW-THRUST SEP TRANSFERS OUTBOUND AND IN-BOUND MARS ENTRY: DIRECT OFF HYPERBOLIC APPROACH TRAJECTORY; PROBE SEPARATES FROM VEHICLE AND PERFORMS SMALL DEFLECTION MANEUVER. MARS CAPTURE: CAPTURE MANEUVER INTO CIRCULAR ORBIT IS PERFORMED BY CHEMICAL PROPULSION MARS BRAKING/DEPARTURE STAGE. SOLAR ARRAYS ARE RETRACTED PRIOR TO MANEUVER AND RE-DEPLOYED IN MARS ORBIT. PROBE DESCENT/LANDING: AEROBRAKING WITH TERMINAL PROPULSION SURFACE MISSION: ACQUIRE SAMPLES AND ENVIRONMENTAL DATA; EXPLORE AREA AROUND LANDER AND SAMPLE WITH ROVER; STAY TIME 100-200 DAYS FOR CONJUNCTION CLASS MISSION OR 1 TO 30 DAYS FOR OPPOSITION CLASS MISSION. ORBIT OPERATIONS: ORBITER/BUS OPERATES AS COMMUNICATIONS RELAY AND SURFACE MAPPER. HIGH POWER AVAILABLE FROM SEP SYSTEM SOLAR ARRAYS. PROBE ASCENT/RENDEZVOUS: ON STORED COMMANDS FROM EARTH, PROBE ASCENT VEHICLE PERFORMS ASCENT THROUGH PHASING ORBIT TO ORBITER/BUS STABLE ORBIT. ORBITER/BUS PERFORMS RENDEZVOUS WITH PROBE, DOCKS, RETRIEVES SAMPLE CANISTER INTO EARTH REENTRY CAPSULE AND JETTISONS PROBE ASCENT VEHICLE STAGE. MARS DEPARTURE: MARS DEPARTURE MANEUVER PERFORMED BY CHEMICAL BRAKING/DEPARTURE STAGE; STAGE ALSO PROVIDES PROPULSION FOR MARS ORBIT RENDEZVOUS. STAGE IS JETTISONED AFTER DEPARTURE MANEUVER. ALTERNATIVE DEPARTURE MODE IS TO JETTISON CHEMICAL STAGE AFTER RENDEZVOUS/DOCKING IN ORBIT AND USE SEP FOR SPIRAL-OUT ESCAPE. EARTH INTERCEPT/RECOVERY: EARTH CAPTURE IS PERFORMED BY THE CHEMICAL EARTH BRAKING STAGE INTO AN ELLIPTICAL ORBIT. SAMPLES CANISTER, FILM CASSETTES, ETC, ARE RETRIEVED IN ORBIT BY MANNED VEHICLE AND TRANSPORTED TO SPACE STATION/BASE FOR PRELIMINARY ANALYSIS.	• INTERMEDIATE 20 • INTERMEDIATE 20/CENTAUR • SHUTTLE/CENTAUR

Figure 3-7. SOLAR-ELECTRIC/CHEMICAL, SINGLE LAUNCH, EARTH CAPTURE/RECOVERY CONCEPT

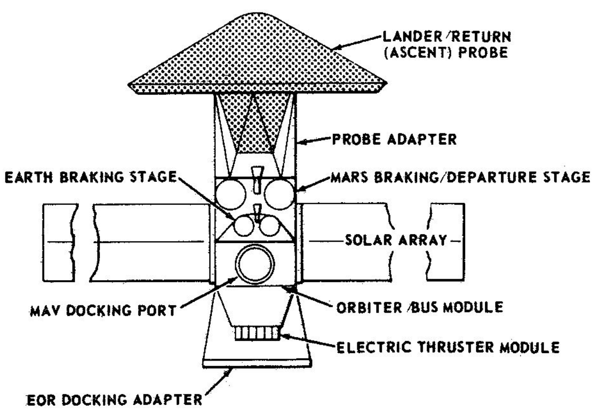
SYSTEM CONCEPT SCHEMATIC	BASELINE MISSION PROFILE SUMMARY	CANDIDATE LAUNCH VEHICLES
	EARTH DEPARTURE: EARTH ORBIT RENDEZVOUS MODE; THE MSSR SPACECRAFT AND TRANSMARS INJECTION STAGE ARE LAUNCHED INTO EARTH PARKING ORBIT WITH SEPARATE LAUNCHES. THE SPACECRAFT PERFORMS RENDEZVOUS AND DOCKS WITH THE INJECTION STAGE WHICH IS USED TO INJECT THE SPACECRAFT ONTO THE TRANSMARS ESCAPE TRAJECTORY. HELIOCENTRIC PROFILE: CONJUNCTION OR OPPOSITION CLASS; LOW-THRUST SEP TRANSFERS OUTBOUND AND IN-BOUND MARS ENTRY: DIRECT OFF HYPERBOLIC APPROACH TRAJECTORY; PROBE SEPARATES FROM VEHICLE AND PERFORMS SMALL DEFLECTION MANEUVER. MARS CAPTURE: CAPTURE MANEUVER INTO CIRCULAR ORBIT IS PER-FORMED BY CHEMICAL PROPULSION MARS BRAKING/DEPARTURE STAGE. SOLAR ARRAYS ARE RETRACTED PRIOR TO MANEUVER AND RE-DEPLOYED IN MARS ORBIT. PROBE DESCENT/LANDING: AEROBRAKING WITH TERMINAL PROPULSION SURFACE MISSION: ACQUIRE SAMPLES AND ENVIRONMENTAL DATA; EXPLORE AREA AROUND LANDER AND SAMPLE WITH ROVER; STAY TIME 100-200 DAYS FOR CONJUNCTION CLASS MISSION OR 1 TO 30 DAYS FOR OPPOSITION CLASS MISSION. ORBIT OPERATIONS: ORBITER/BUS OPERATES AS COMMUNICATIONS RELAY AND SURFACE MAPPER. HIGH POWER AVAILABLE FROM SEP SYSTEM SOLAR ARRAYS PROBE ASCENT/RENDEZVOUS: ON STORED COMMANDS FROM EARTH, PROBE ASCENT VEHICLE PERFORMS ASCENT THROUGH PHASING ORBIT TO ORBITER/BUS STABLE ORBIT. ORBITER/BUS PERFORMS RENDEZVOUS WITH PROBE, DOCKS, RETRIEVES SAMPLE CANISTER INTO EARTH REENTRY CAPSULE AND JETTISONS PROBE ASCENT VEHICLE STAGE. MARS DEPARTURE: MARS DEPARTURE MANEUVER PERFORMED BY CHEMICAL BRAKING/DEPARTURE STAGE; STAGE ALSO PROVIDES PROPULSION FOR MARS ORBIT RENDEZVOUS. STAGE IS JETTISONED AFTER DEPARTURE MANEUVER. ALTERNATIVE DEPARTURE MODE IS TO JETTISON CHEMICAL STAGE AFTER RENDEZVOUS/DOCKING IN ORBIT AND USE SEP FOR SPIRAL-OUT ESCAPE. EARTH INTERCEPT/RECOVERY: EARTH CAPTURE IS PERFORMED BY THE CHEMICAL EARTH BRAKING STAGE INTO AN ELLIPTICAL ORBIT. SAMPLES CANISTER, FILM CASSETTES, ETC, ARE RETRIEVED IN ORBIT BY MANNED VEHICLE AND TRANSPORTED TO SPACE STATION/BASE FOR PRELIMINARY ANALYSIS.	• TRANSMARS INJECTION STAGE: CENTAUR • LAUNCH TO EARTH ORBIT: TITAN III C FOR SPACECRAFT LAUNCH; TITAN III D/CENTAUR FOR INJECTION STAGE LAUNCH. ALTERNATIVE IS TWO LAUNCHES OF SPACE SHUTTLE (PARTIAL LOADS).

Figure 3-8. SOLAR-ELECTRIC/CHEMICAL, EARTH ORBIT RENDEZVOUS, EARTH CAPTURE/RECOVERY CONCEPT

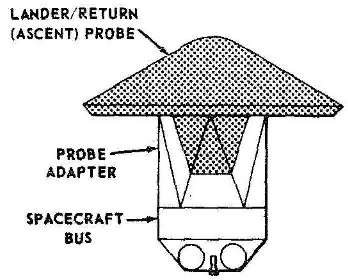
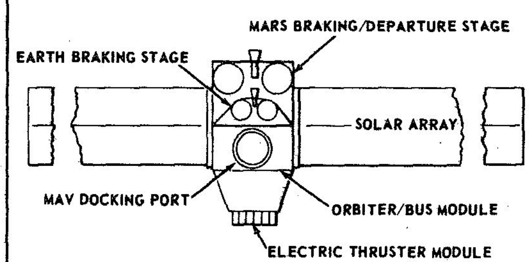
SYSTEM CONCEPT SCHEMATIC	BASELINE MISSION PROFILE SUMMARY	CANDIDATE LAUNCH VEHICLES
LAUNCH NO. 1 LANDER/RETURN (ASCENT) PROBE  LAUNCH NO. 2 ORBITER/BUS RETURN VEHICLE  	EARTH DEPARTURE: LANDER/RETURN (ASCENT) PROBE AND ORBITER/BUS RETURN VEHICLE ARE LAUNCHED AS SEPARATE PAYLOADS ONTO SEPARATE EARTH-MARS TRAJECTORIES; EACH DEPARTURE IS THROUGH 185-KM PARKING ORBIT. HELIOCENTRIC PROFILE: CONJUNCTION OR OPPOSITION CLASS; LOW-THRUST SEP TRANSFERS OUTBOUND AND IN-BOUND MARS ENTRY: DIRECT OFF HYPERBOLIC APPROACH TRAJECTORY; SPACECRAFT BUS PERFORMS SMALL DEFLECTION MANEUVER AND FLIES PAST MARS INTO HELIOCENTRIC ORBIT MARS CAPTURE: CAPTURE MANEUVER INTO CIRCULAR ORBIT IS PERFORMED BY CHEMICAL PROPULSION MARS BRAKING/DEPARTURE STAGE. SOLAR ARRAYS ARE RETRACTED PRIOR TO MANEUVER AND RE-DEPLOYED IN MARS ORBIT. PROBE DESCENT/LANDING: AEROBRAKING WITH TERMINAL PROPULSION SURFACE MISSION: ACQUIRE SAMPLES AND ENVIRONMENTAL DATA; EXPLORE AREA AROUND LANDER AND SAMPLE WITH ROVER; STAY TIME 100-200 DAYS FOR CONJUNCTION CLASS MISSION OR 1 TO 30 DAYS FOR OPPOSITION CLASS MISSION. ORBIT OPERATIONS: ORBITER/BUS OPERATES AS COMMUNICATIONS RELAY AND SURFACE MAPPER. HIGH POWER AVAILABLE FROM SEP SYSTEM SOLAR ARRAYS PROBE ASCENT/RENDEZVOUS: ON STORED COMMANDS FROM EARTH, PROBE ASCENT VEHICLE PERFORMS ASCENT THROUGH PHASING ORBIT TO ORBITER/BUS STABLE ORBIT. ORBITER/BUS PERFORMS RENDEZVOUS WITH PROBE, DOCKS, RETRIEVES SAMPLE CANISTER INTO EARTH REENTRY CAPSULE AND JETTISONS PROBE ASCENT VEHICLE STAGE. MARS DEPARTURE: MARS DEPARTURE MANEUVER PERFORMED BY CHEMICAL BRAKING/DEPARTURE STAGE; STAGE ALSO PROVIDES PROPULSION FOR MARS ORBIT RENDEZVOUS. STAGE IS JETTISONED AFTER DEPARTURE MANEUVER. ALTERNATIVE DEPARTURE MODE IS TO JETTISON CHEMICAL STAGE AFTER RENDEZVOUS/DOCKING IN ORBIT AND USE SEP FOR SPIRAL-OUT ESCAPE. EARTH INTERCEPT/RECOVERY: THE EARTH REENTRY CAPSULE IS SEPARATED ON EARTH APPROACH TRAJECTORY AND PERFORMS A DIRECT REENTRY. RECOVERY IS BY AIR SNATCH WITH WATER RECOVERY BACKUP.	TWO TITAN III D/CENTAUR VEHICLES NOTE: THIS MISSION/SYSTEM CONCEPT IS ADAPTABLE TO CONSIDERATION OF DUAL LANDER/ASCENT PROBES WITH A SINGLE ORBITER/BUS RETURN VEHICLE. THE DUAL PROBE CONCEPT WOULD REQUIRE AN ADDITIONAL TITAN III D/CENTAUR VEHICLE.

Figure 3-9. SOLAR-ELECTRIC/CHEMICAL, MULTIPLE INTERPLANETARY LAUNCH, EARTH CAPTURE/RECOVERY CONCEPT

Section IV
MISSION ANALYSIS

This section defines the mission profile characteristics required for analysis of MSSR performance for each candidate mission/system alternative under consideration. The all-chemical missions are considered first, followed by discussion of the solar-electric/chemical missions. Reference performance capabilities of the the candidate Earth launch vehicles considered in the present study are summarized at the end of the section.

4.1 ALL-CHEMICAL MISSIONS

The mission analysis for all-chemical concepts was based on use of the results of work under Contract NAS8-24714 (ref. 2) with the development of additional data where required.

4.1.1 Heliocentric Trajectories and Mission Characteristics

The following guidelines and constraints were used in the selection of baseline interplanetary trajectories:

- Conjunction Class and Opposition Class flight profiles corresponding to the 1975, 1978, and 1980 Earth-Mars oppositions are considered.
- Nominal Earth departure period is 20 days.*
- Launch will be from Cape Kennedy at a launch azimuth between 80 and 100 degrees (relaxed for 1975 Inbound Venus Swingby mission).
- In Venus swingby missions, the swingby distance at Venus should not be less than approximately 1.1 planet radii.
- Mars stopover times are considered as follows:
  - One day for Opposition Class missions (near minimum energy missions)
  - From optimum stopover time to 100 days less than optimum for Conjunction Class missions
- Earth reentry speed is limited to 50,000 ft/sec (15.24 km/sec) for direct reentry/recovery mode missions
- The principal criterion for selection of reference trajectory and mission characteristics is the minimization of total mission velocity requirements.

**In the dual departure mission concepts, the departure period would possibly have to be increased if both launches were to be made from the same pad.*

4.1.1.1 Conjunction Class Missions. Figure 4-1 summarizes the selected baseline Conjunction Class mission time characteristics. The 1982, 1984, and 1986 missions are added in the figure for comparison. As is well known for this class of mission profile, the total mission duration is of the order of 1000 days for all opportunities. The Mars stopover time ranges from 340 to 370 days for the 1975, 1978, and 1980 missions of interest.

Table 4-1 summarizes the Conjunction Class mission velocity requirements and characteristics. The Earth departure energy requirements for a 20-day departure period ranges from 11 km^2/sec^2 for the 1980 mission up to 17 km^2/sec^2 for the 1975 mission. The departure asymptote declination is less than 30 degrees for these missions; therefore, launch azimuths generally within a few degress of due east (90 degrees) can be used to achieve acceptable daily firing windows during the mission departure period. For the three mission opportunities of particular interest (1975, 1978, 1980), the Mars capture and departure impulse requirements are relatively invariant with mission. The impulse requirements shown are based on a reference 600-km circular Mars orbit. The selection of this altitude will be discussed in subsection 4.1.2.

The Earth capture impulse requirements given in Table 4-1 are determined by assuming the Apollo Command and Service Module (CSM) is employed for sample retrieval. This assumption is discussed in subsection 4.1.3. The reentry speed at Earth for the direct reentry/recovery mode is given in the last column in the table. The speeds are relatively low ranging from about 37,600 to 40,500 ft/sec depending on mission opportunity.

A brief parametric analysis was performed to investigate the velocity penalties associated with shortening the conjunction class mission by either shorter return flight time trajectories or departing Mars at an earlier date (shortening the stopover time). Figures 4-2, 4-3, and 4-4 present the results of the analysis.

Figure 4-2 shows the Mars departure impulse penalty as a function of the total savings in mission duration. Data are given for the 1975 and 1978 missions. Each point on the curves represents the optimum (minimum ΔV) combination of Mars departure date and Mars-Earth flight time to produce the indicated net

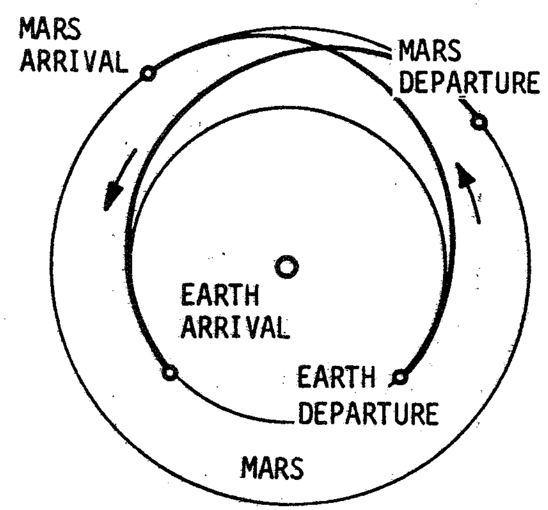
REPRESENTATIVE FLIGHT PROFILE	MISSION	EARTH DEPARTURE PERIOD (DAYS)	TOTAL MISSION DURATION (DAYS)	MARS STOPOVER TIME (DAYS)	MARS ARRIVAL DATE
	1975	20	980	340	20 AUG 76
	1978	20	1000	340	30 AUG 78
	1980	20	1020	370	8 SEP 80
	1982	20	1030	420	28 SEP 82
	1984	20	990	500	7 OCT 84
	1986	20	940	535	16 NOV 86

Figure 4-1. CONJUNCTION CLASS MISSION CHARACTERISTICS

Table 4-1. CONJUNCTION CLASS MISSION VELOCITY REQUIREMENTS

- 600-KM CIRCULAR MARS CAPTURE ORBIT
- 555 x 9100-KM EARTH CAPTURE ORBIT

MISSION	EARTH DEPARTURE C_3(1) (km^2/sec^2)	MARS CAPTURE ΔV(2) (km/sec)	MARS ENTRY SPEED (km/sec)	MARS DEPARTURE ΔV(2) (km/sec)	EARTH CAPTURE ΔV(3) (km/sec)	REENTRY SPEED (km/sec)
1975	17	2.07	5.50	2.23	2.2	11.45
1978	13	2.01	5.45	2.12	2.3	11.57
1980	11	2.10	5.54	2.01	2.9	12.03
1982	11	2.35	5.78	1.95	3.1	12.34
1984	13	2.65	6.03	2.00	2.4	11.68
1986	11	2.39	5.79	2.43	2.3	11.54

(1) C_3 = TWICE THE TOTAL ENERGY PER UNIT SPACECRAFT MASS
(2) ΔV'S INCLUDE 3% CONTINGENCY
(3) EARTH CAPTURE IMPULSE ALLOWS ORBITAL RETRIEVAL OF SAMPLES BY APOLLO CSM WITH 10% ΔV RESERVE

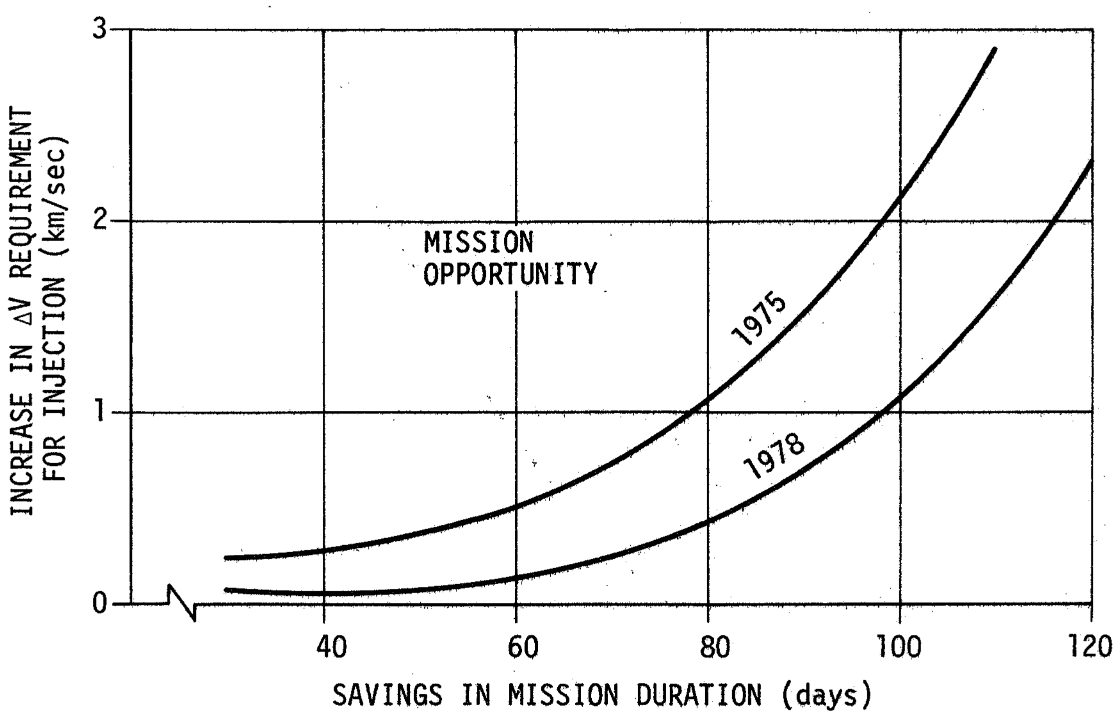


Figure 4-2. TRANSEARTH INJECTION ΔV PENALTY FOR SHORTENED MISSION

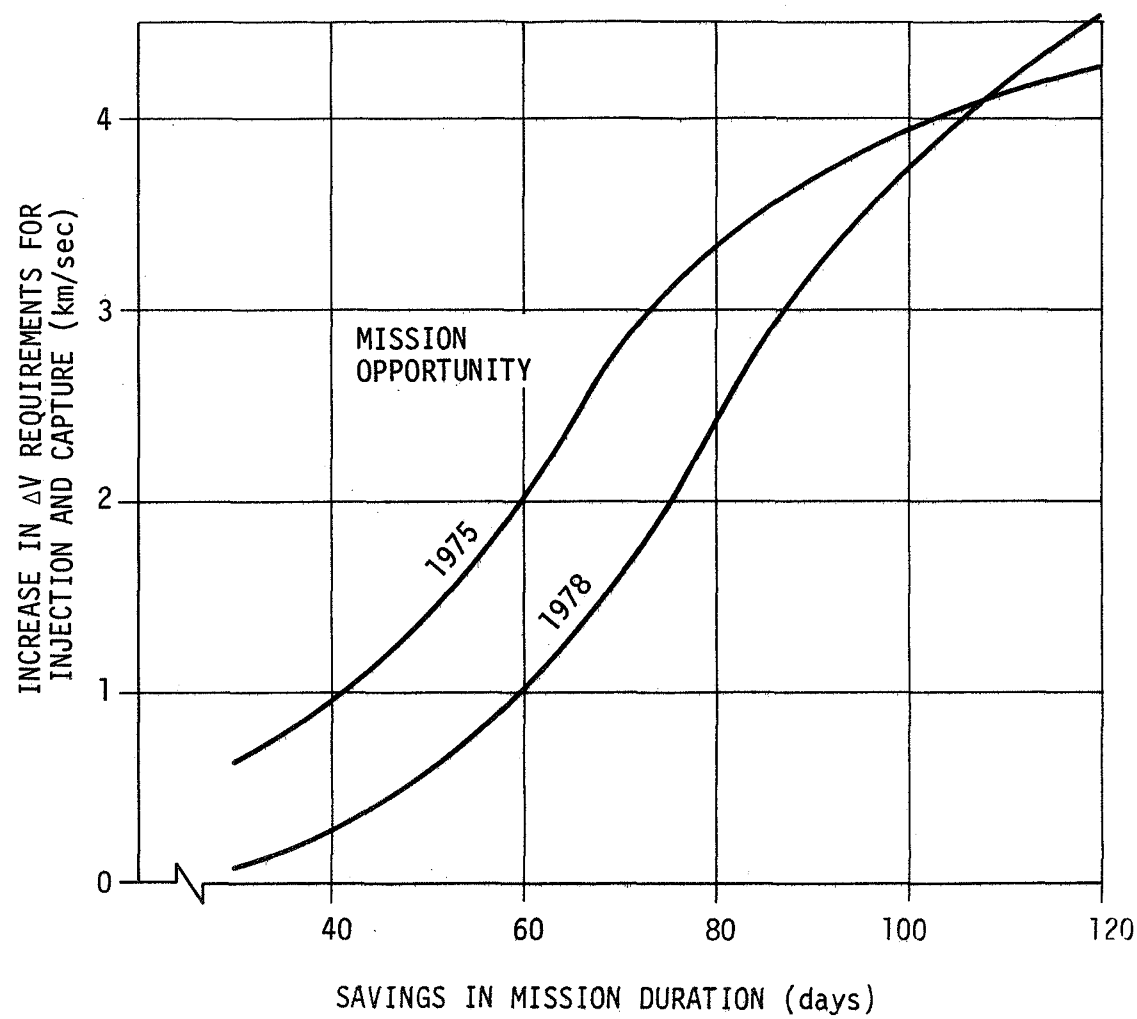


Figure 4-3. TOTAL ΔV PENALTY FOR SHORTENED MISSION

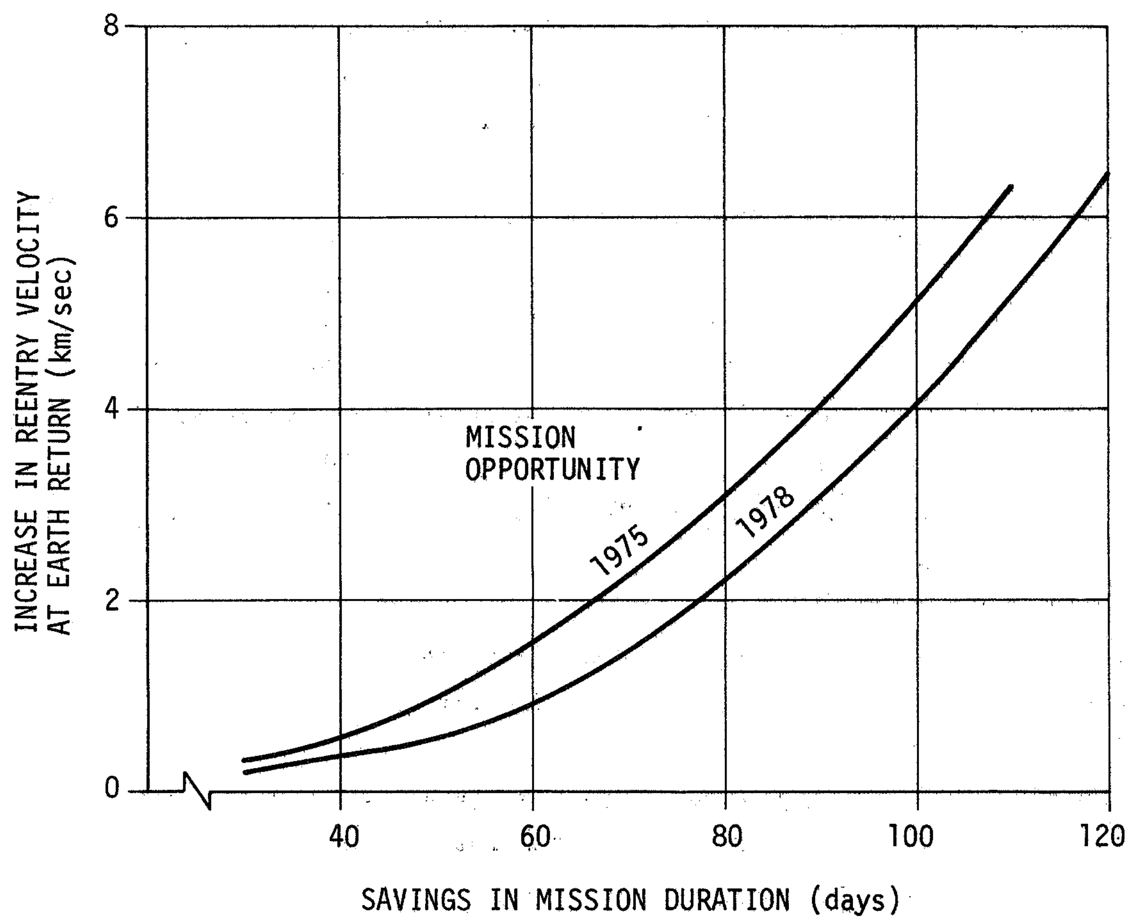


Figure 4-4. EARTH REENTRY SPEED PENALTY FOR SHORTENED MISSION

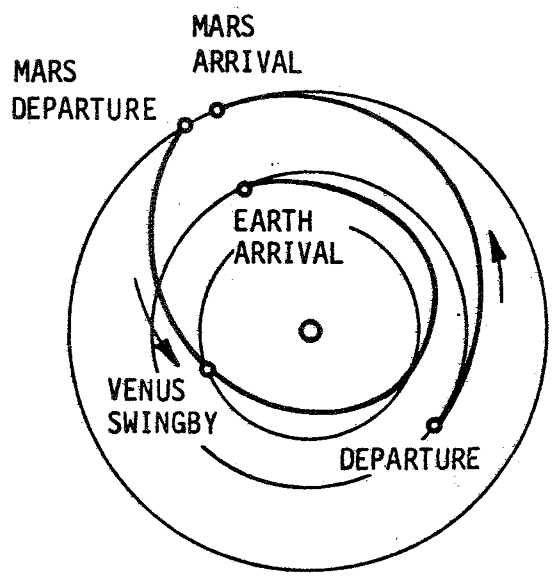
mission duration savings. The curves indicate that an additional 500 m/sec impulse capability would produce a 60-day savings in the 1975 mission or approximately an 83-day savings in the 1978 mission. The data show that the velocity penalty becomes quite large to significantly shorten the mission.

Figure 4-3 introduces consideration of the velocity impulse penalty for orbital capture at Earth return. The figure gives the sum of the impulse increases for Mars departure and Earth capture as a function of net savings in total mission duration. For systems employing the Earth orbital capture/recovery mode, the spacecraft performance penalty is seen to increase rapidly for significant savings in mission duration.

For the direct reentry/recovery mode at Earth return, the increase in reentry speed as a function of total net savings in mission duration is shown in Figure 4-4. The curves indicate that a savings of approximately 87 days in the 1975 mission would push the reentry speed up to 50,000 ft/sec. Similarly, a savings of about 96 days in the 1978 mission would increase the reentry speed to 50,000 ft/sec.

In summary, the Conjunction Class missions cannot be shortened significantly without large penalties in velocity requirements.

4.1.1.2 Venus Swingby Missions. Figure 4-5 and Table 4-2 summarize the selected baseline Venus swingby mission characteristics and velocity requirements. One-day stopover times were selected as a reference for all missions. This would be representative of a minimum energy mission and would allow a bare minimum of time on the surface for sampling. Stopovers up to several days could be considered without significant mission energy penalties. The 1975 mission is an inbound swingby opportunity. Both outbound and inbound swingby opportunities exist for the 1978 mission; however, the inbound mission is characterized by high energy requirements and the outbound case has a very narrow departure window to achieve acceptable Venus passage conditions. The 1978 outbound mission is shown for a 10-day Earth departure window. The Earth reentry speed is approximately 56,700 ft/sec and exceeds the 50,000 ft/sec study constraint by 13.4 percent. The Earth departure energy (C_3) requirement is high at 42 km^2/sec^2. The 1980 mission opportunity is an outbound swingby with overall mission characteristics comparable to the 1975 mission.

REPRESENTATIVE FLIGHT PROFILE	MISSION	EARTH DEPARTURE PERIOD (DAYS)	TOTAL MISSION DURATION (DAYS)	MARS STOPOVER TIME (DAYS)	MARS ARRIVAL DATE	VENUS SWINGBY DISTANCE * (km)
	1975 INB. SWINGBY	20	520	1	1 APR 76	1920
	1978 OUTB. SWINGBY	10	640	1	11 JAN 78	1590
	1980 OUTB. SWINGBY	20	640	1	13 OCT 79	2260

* AVERAGE OVER EARTH DEPARTURE PERIOD

Figure 4-5. VENUS SWINGBY MISSION CHARACTERISTICS

Table 4-2. VENUS SWINGBY MISSION VELOCITY REQUIREMENTS

- 600-KM CIRCULAR MARS CAPTURE ORBIT
- 555 x 9100-KM EARTH CAPTURE ORBIT

MISSION	EARTH DEPARTURE C_3 (1) (km^2/sec^2)	MARS CAPTURE $\Delta V^{(2)}$ (km/sec)	MARS ENTRY SPEED (km/sec)	MARS DEPARTURE $\Delta V^{(2)}$ (km/sec)	EARTH CAPTURE $\Delta V^{(3)}$ (km/sec)	REENTRY SPEED (km/sec)
1975 INB. SWINGBY	21	3.00	6.39	3.99	4.3	13.46
1978 OUTB. SWINGBY	42	3.19	6.55	3.57	8.2	17.25
1980 OUTB. SWINGBY	30	3.73	7.08	2.42	4.2	13.34

(1) C_3 = TWICE THE TOTAL ENERGY PER UNIT SPACECRAFT MASS
(2) ΔV'S INCLUDE 3% CONTINGENCY
(3) EARTH CAPTURE IMPULSE ALLOWS ORBITAL RETRIEVAL OF SAMPLES BY APOLLO CSM WITH 10% ΔV RESERVE

It was necessary in the case of the 1975 Venus swingby mission to consider launch azimuths as far North as 45 degrees because of the large declinations reached by the departure asymptote during the Earth departure period. This causes launch vehicle performance penalties and requires launches at azimuths near the most northeasterly range safety limit which potentially could be approved for launches from KSC.

Of the three opposition years, the 1975 inbound swingby and the 1980 outbound swingby missions were selected for investigation in the mission/system performance studies.

4.1.2 Mars Orbit Selection

A circular operational orbit is desirable for the MSSR orbiter/bus vehicle because of the requirement for orbital rendezvous with the probe ascent vehicle. For purposes of the present study, orbit altitude was selected primarily on the basis of lifetime requirements to satisfy planetary quarantine. The quarantine groundrule was taken to be the same as that directed by NASA for the study under Contract NAS8-24714; i.e., quarantine is to be enforced for a 20-year period beginning January 1, 1969. Thus, orbit minimum lifetimes must extend from Mars arrival through January 1, 1989. In the previous study a constraint was imposed which required sterilizable propellants to be used below 1000 km altitude. This groundrule was relaxed in the present study.

4.1.2.1 Orbit Altitude. Based on the arrival dates at Mars, Table 4-3 summarizes the minimum orbit lifetime requirement for each reference mission under consideration. The requirement is seen to range from 8.3 to 12.8 years depending on the mission. The orbit lifetime data given in reference 2 indicate that for an orbiting vehicle effective ballistic coefficient of 0.1 slug/ft^2, the minimum orbit altitude ranges from about 560 to 600 km depending on mission. An effective ballistic coefficient of roughly 0.1 slug/ft^2 would be representative of a spent chemical spacecraft propulsion stage such as the Mars Braking Stage. Therefore, for purposes of the parametric performance studies, a baseline operational orbit altitude of 600 km was assumed for the orbiter/bus vehicle.

4.1.2.2 Orbit Inclination. The primary concern in this study is the analysis of mission/system performance; therefore, orbit inclination is not a critical parameter for consideration. It suffices here to state that selection of orbit inclination for the all-chemical mission/system concepts must satisfy the following fundamental requirements:

- The inclination must satisfy Mars arrival and departure asymptote geometry constraints. Conjunction Class missions generally permit selection of high inclinations up to near polar. Venus Swingby missions place relatively severe constraints on choice of inclination because of the short stopover time.
- The inclination must equal or exceed the latitude of the probe landing site to avoid dog-leg or plane change requirements for ascent and rendezvous.

Table 4-3. MARS ORBIT MINIMUM LIFETIME REQUIREMENTS

MISSION	MARS ARRIVAL DATE (JULIAN 244-)	MINIMUM ORBIT LIFETIME REQUIREMENT (YEARS)
1975 Conjunction Class	3010	12.4
1975 Inbound Venus Swingby	2870	12.8
1978 Conjunction Class	3750	10.4
1980 Conjunction Class	4490	8.3
1980 Outbound Venus Swingby	4160	9.2

- The orbit should be posigrade to avoid probe ascent vehicle launches against Mars' rotation.

4.1.2.3 Elliptical Capture Orbit. In the Mars entry out-of-elliptical orbit mode, the lander/ascent probe is carried into an initial elliptical capture orbit by the orbiter/bus vehicle. The probe is separated in the ellipse and a deorbit maneuver is performed to achieve atmospheric entry. The orbiter/bus vehicle later fires the braking stage a second time to maneuver from the ellipse into a 600-km operational circular orbit.

For purposes of this study, a Mars synchronous ellipse with a 600-km periapsis altitude was assumed for the initial orbit in missions employing this capture mode. The velocity decrement required for capture is a function of the Mars arrival hyperbolic excess speed. Figure 4-6 gives the required ΔV as a function of arrival excess speed. The figure presents curves for periapsis altitudes of 600 km and 1000 km, and also 600-km and 1000-km circular orbits for comparison. The velocity impulse requirement is seen to be a weak function of periapsis altitude. The difference between impulses for initial elliptical and circular capture maneuvers is seen to be of the order of 1 km/sec and is relatively independent of arrival excess speed.

The velocity impulse required for maneuver from an initial ellipse down to a 600-km circular orbit is shown in Figure 4-7 as a function of the ratio (r_a/r_p) of apoapsis to periapsis radii of the ellipse. Data for a 1000-km periapsis altitude are shown for comparison. The r_a/r_p ratio for a Mars synchronous period (24.6 hours) ellipse is approximately 9.2 for a 600-km periapsis altitude and 8.3 for a 1000-km altitude. The maneuver impulses corresponding to these cases are approximately 1.13 km/sec and 1.05 km/sec, respectively.

Table 4-4 summarizes the velocity impulse requirements for the elliptical capture mode for each reference mission under consideration.

4.1.2.4 Orbit Periods. For reference purposes, the periods of circular and elliptical orbits about Mars are given in Figure 4-8 and 4-9, respectively.

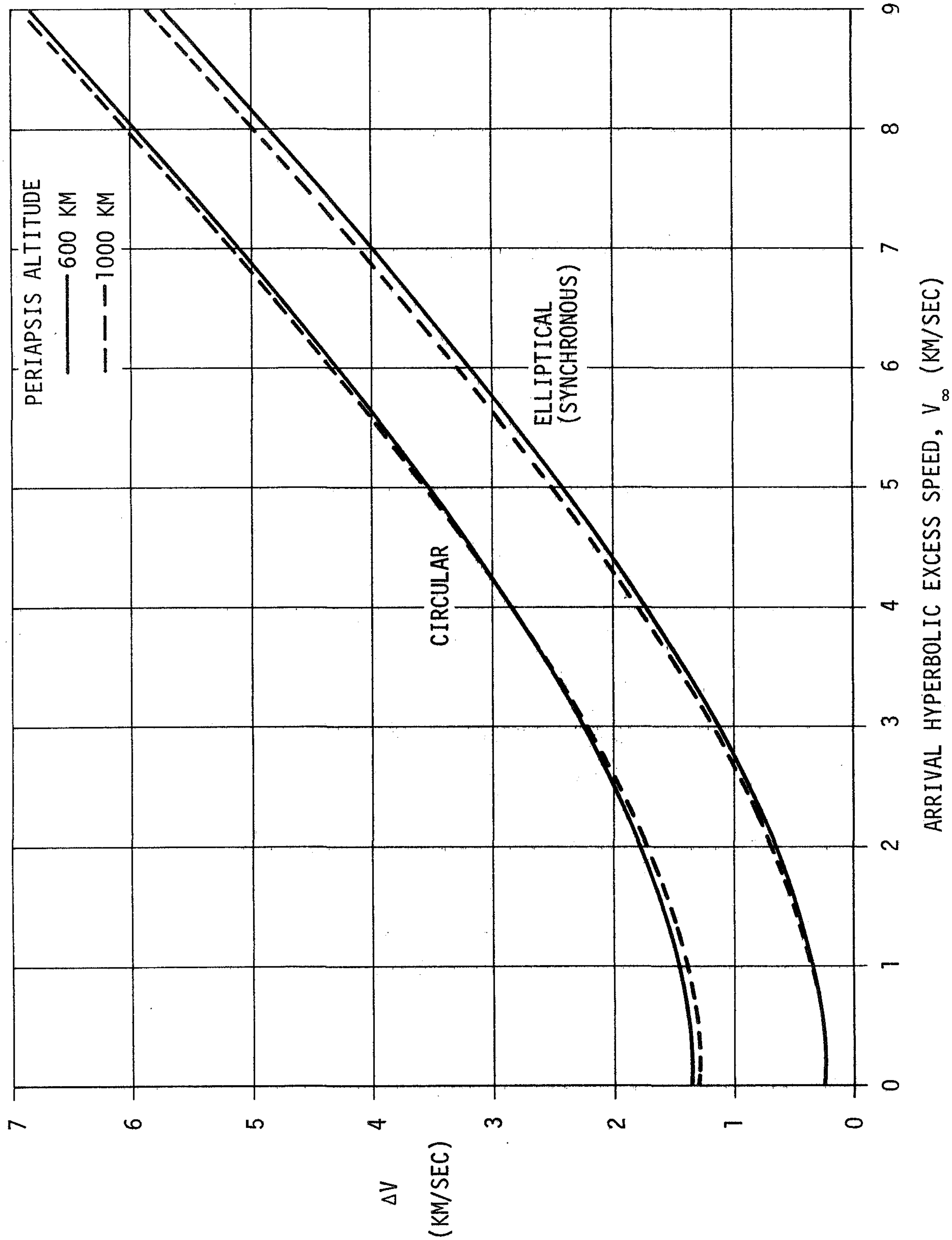


Figure 4-6. MARS CAPTURE ΔV REQUIREMENT

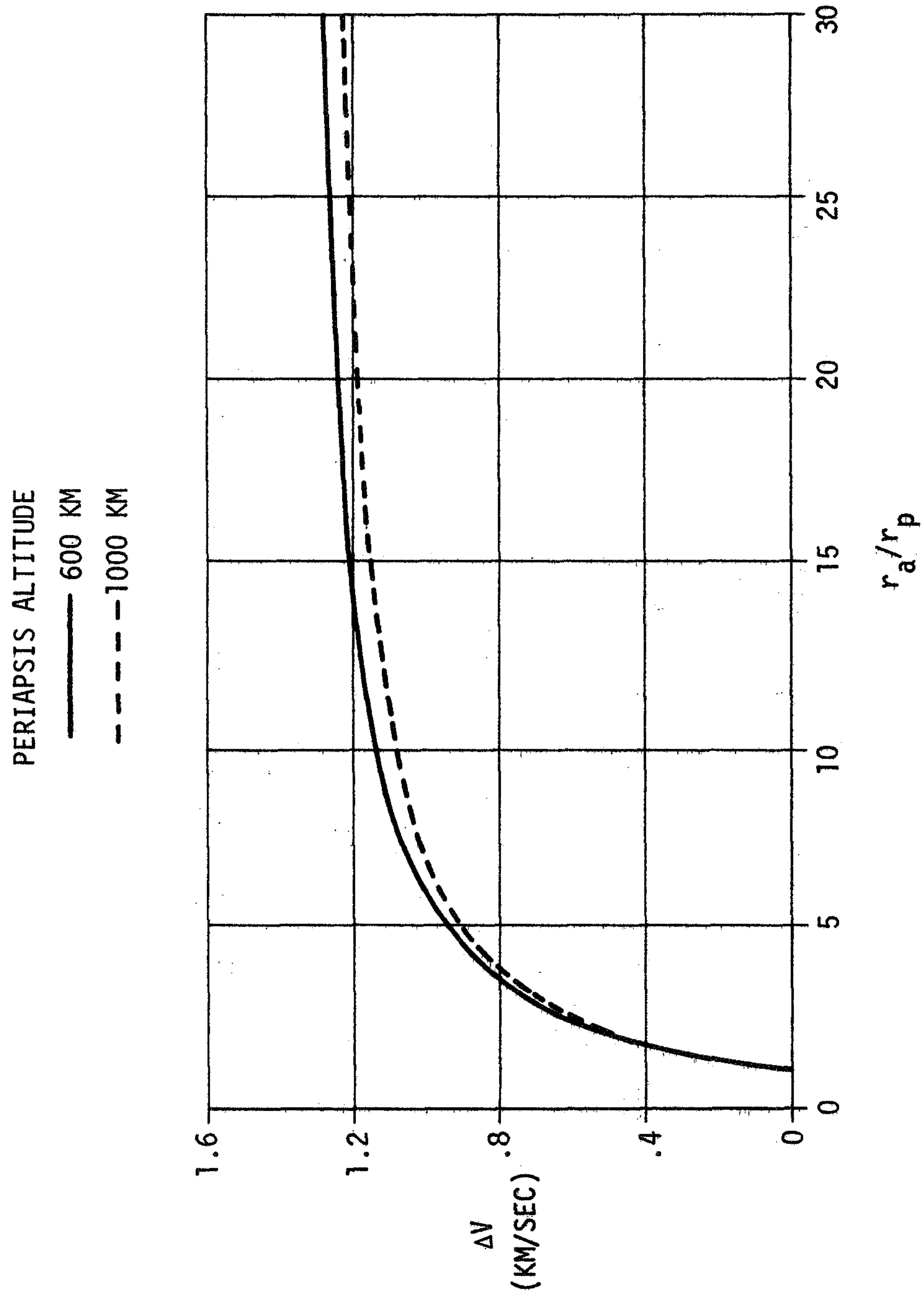


Figure 4-7. ΔV FOR MANEUVER FROM ELLIPTICAL TO CIRCULAR ORBIT AT MARS

Table 4-4. MARS ELLIPTICAL CAPTURE IMPULSE REQUIREMENTS

MISSION	V_∞ AT MARS ARRIVAL (KM/SEC)	ΔV FOR INITIAL CAPTURE[1] [2] (KM/SEC)	ΔV FOR MANEUVER TO 600-KM CIRCULAR ORBIT[1] (KM/SEC)
1975 Conjunction	2.59	.93	1.16
1975 Inbound Venus Swingby	4.11	1.86	1.16
1978 Conjunction	2.44	.88	1.16
1980 Conjunction	2.62	.96	1.16
1980 Outbound Venus Swingby	5.13	2.60	1.16

(1) All Numbers include a 3% Contingency

(2) Initial capture is into a Mars synchronous ellipse

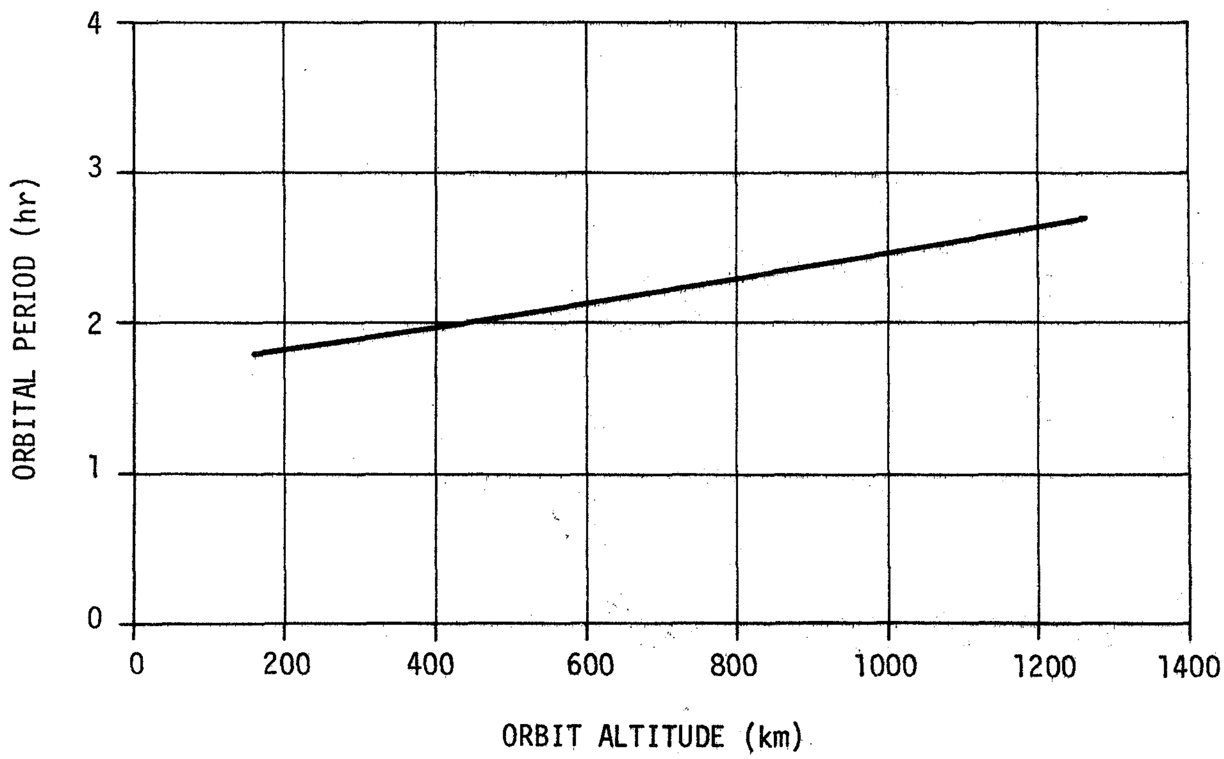


Figure 4-8. PERIOD OF CIRCULAR ORBITS ABOUT MARS

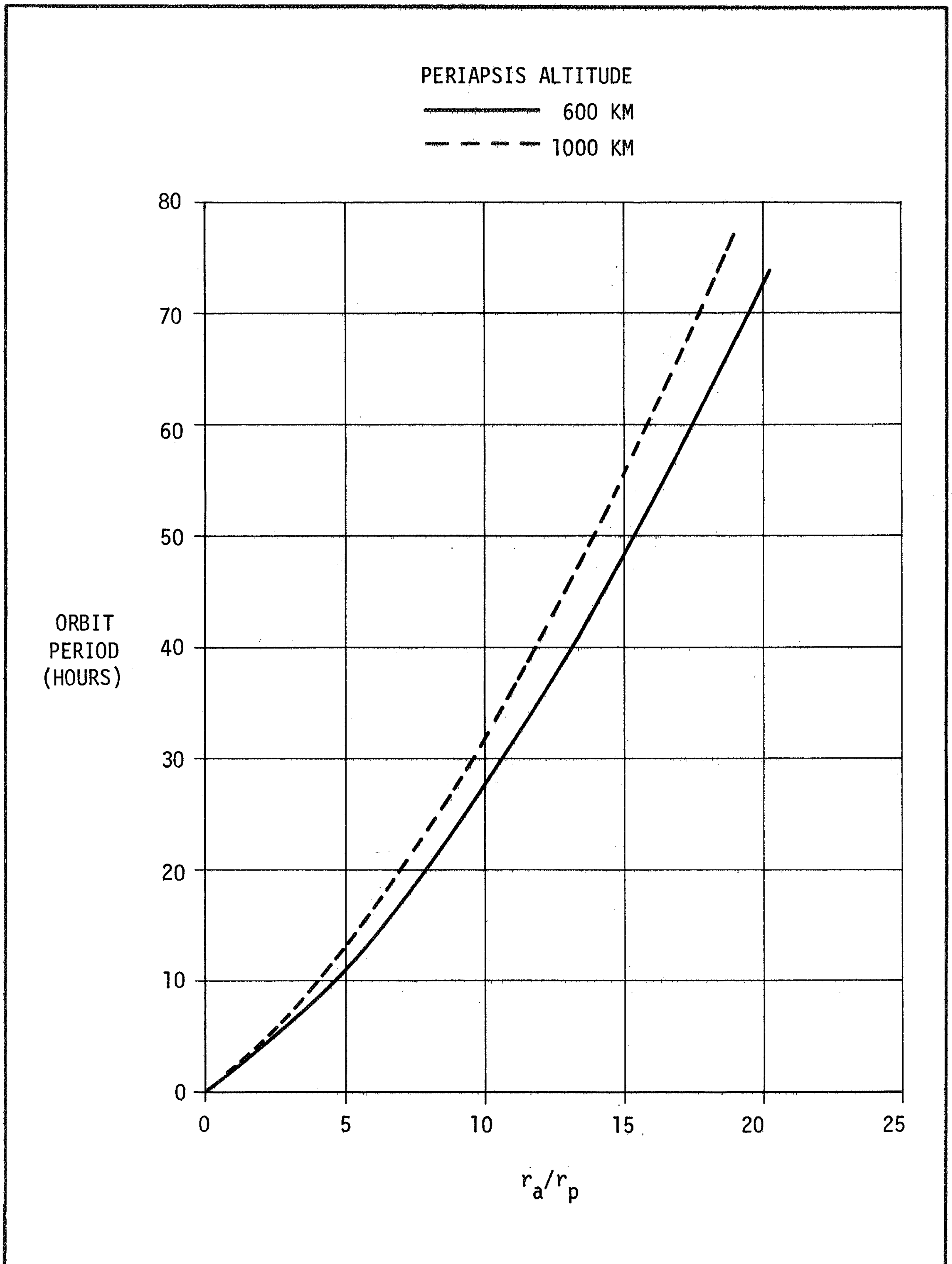


Figure 4-9. PERIOD OF ELLIPTICAL ORBITS ABOUT MARS

4.1.3 Earth Capture Orbit Selection

In the orbital capture Earth intercept/recovery mode missions, the MSSR spacecraft, or a separated portion of it, must perform a propulsive braking maneuver to place the samples canister and necessary equipment into an orbit about the Earth. The samples canister may be retrieved by a manned vehicle or an automated system such as an automated space tug. The samples would be transported to an orbiting facility for initial quarantine and analysis or perhaps encapsulated in a special container for return to Earth laboratories directly.

In any event, the method of orbit retrieval impacts the selection of Earth capture orbit and, therefore, the braking impulse requirements placed on the MSSR spacecraft.

Figure 4-10 shows representative orbit selection trades based on the use of an orbit-launched, fully loaded Apollo Command and Service Module (CSM). The maximum allowable apogee altitude for the MSSR spacecraft capture orbit is shown as a function of the out-of-plane angle required for the retrieval operation and the total recovery weight. (It is possible that the entire spacecraft could be retrieved.) The data in the figure are based on a 300-nautical mile (555-km) perigee altitude and assumes a 10 percent performance contingency in the CSM. The allowable apogee altitude is seen to be a strong function of the out-of-plane angle and a weak function of the recovery weight.

The CSM is perhaps the most reasonable choice of recovery vehicle for the 1975 mission opportunity. A space tug could become the prime choice for recovery operations of this type in the late 1970's and early 1980's. For purposes of the present study, an orbit-launched CSM recovery vehicle was assumed for all missions because it exists and provides a reasonably conservative approach with regard to the performance impact on the MSSR system.

The data shown in Figure 4-10 can be translated directly into velocity impulse requirements placed on the MSSR spacecraft for Earth orbit capture. A representative capture ellipse of 300 x 4900 nautical miles (555 x 9100 km) was selected for mission analysis purposes. This orbit allows for approximately 3 degrees of out-of-plane maneuvering and retrieval of around 2000 pounds. The equivalent performance could be represented by smaller recovery weights and

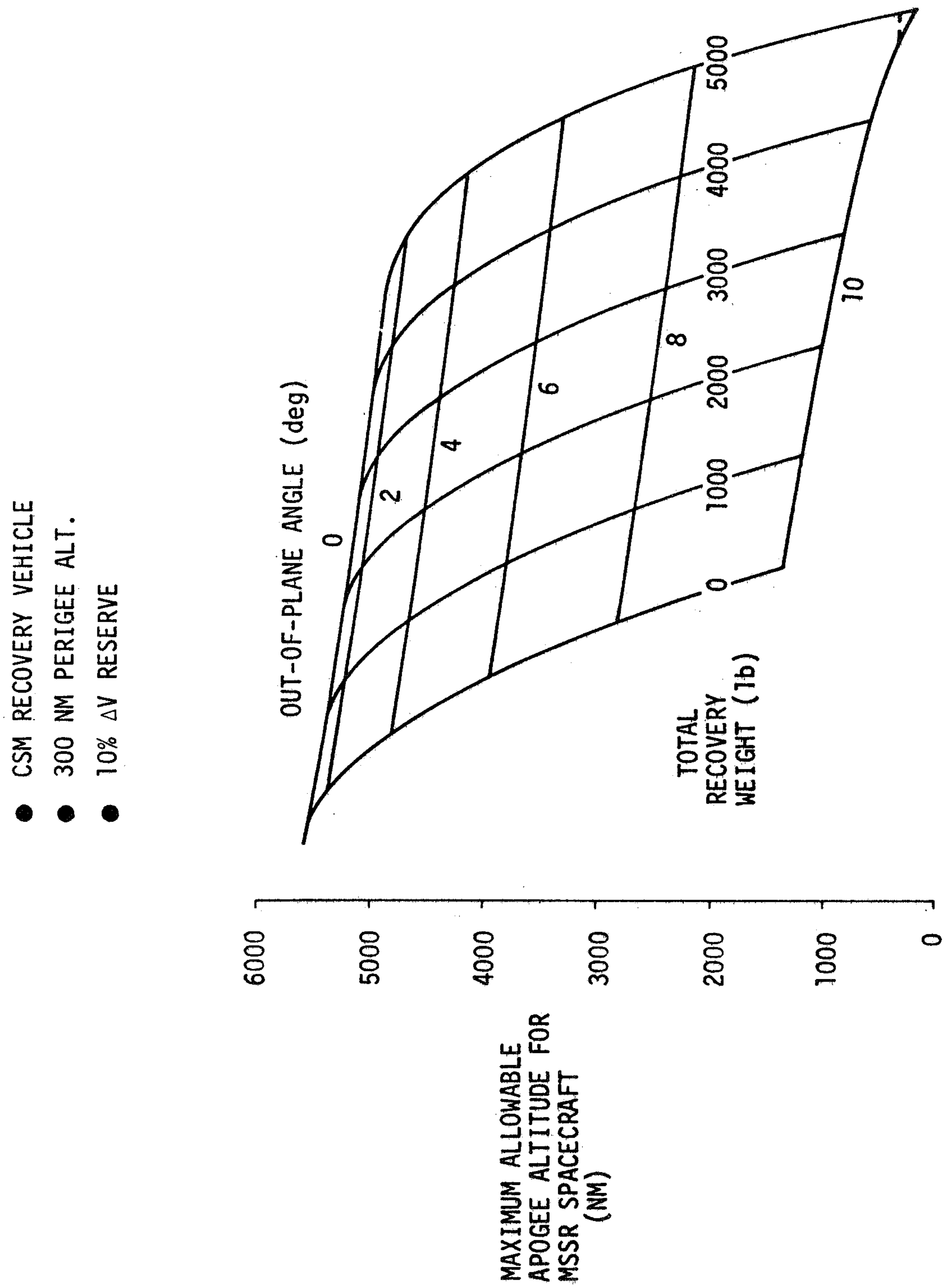


Figure 4-10. EARTH CAPTURE ORBIT SELECTION TRADES

slightly more out-of-plane capability or larger recovery weights and slightly less out-of-plane capability.

Figure 4-11 shows the capture velocity impulse requirement for the MSSR spacecraft as a function of Earth arrival hyperbolic excess speed. Both Apollo CSM and representative orbit-launched space tug recovery vehicle curves are shown. The capture impulse requirements are spotted in the figure for the various reference missions under consideration.

4.1.4 Mars Ascent Vehicle Velocity Requirements

Based on Northrop's past studies of Mars Ascent Vehicle (MAV) trajectories and performance, a representative curve of ascent characteristic velocity as a function of orbiter/bus orbit altitude is given in Figure 4-12. A two-stage coast ascent profile is assumed with rendezvous phasing through a 300-km circular orbit. For the selected baseline orbiter/bus altitude of 600 km, the MAV characteristic velocity requirement is 4340 m/sec. The curve includes a 5 percent contingency.

4.1.5 Secondary Mission Velocity Requirements

On the basis of Northrop's past studies, Table 4-5 summarizes the velocity budgets adopted for the various secondary spacecraft maneuver requirements throughout the MSSR mission.

4.1.6 Earth Orbit Rendezvous Mode Requirements

The mission concept assumed for analysis of the earth orbit rendezvous departure mode is outlined as follows:

- The Centaur stage, assumed to be the transmars injection stage, is launched into a low earth orbit and separated from the launch vehicle upper stage.
- The Centaur is programmed to automatically hold a fixed attitude relative to the velocity vector around the orbit.
- The second launch brings up the MSSR spacecraft equipped with propulsion, astrionics equipment, and a docking adapter system for rendezvous and docking with the orbiting Centaur stage.
- With ground control assistance, the rendezvous and docking operation is performed within two to three orbital passes with the MSSR spacecraft as the active vehicle.
- After a final orbital check is made on spacecraft and Centaur systems, the Centaur engines are ignited to perform the transmars injection maneuver. The Centaur stage is jettisoned after completion of the departure burn.

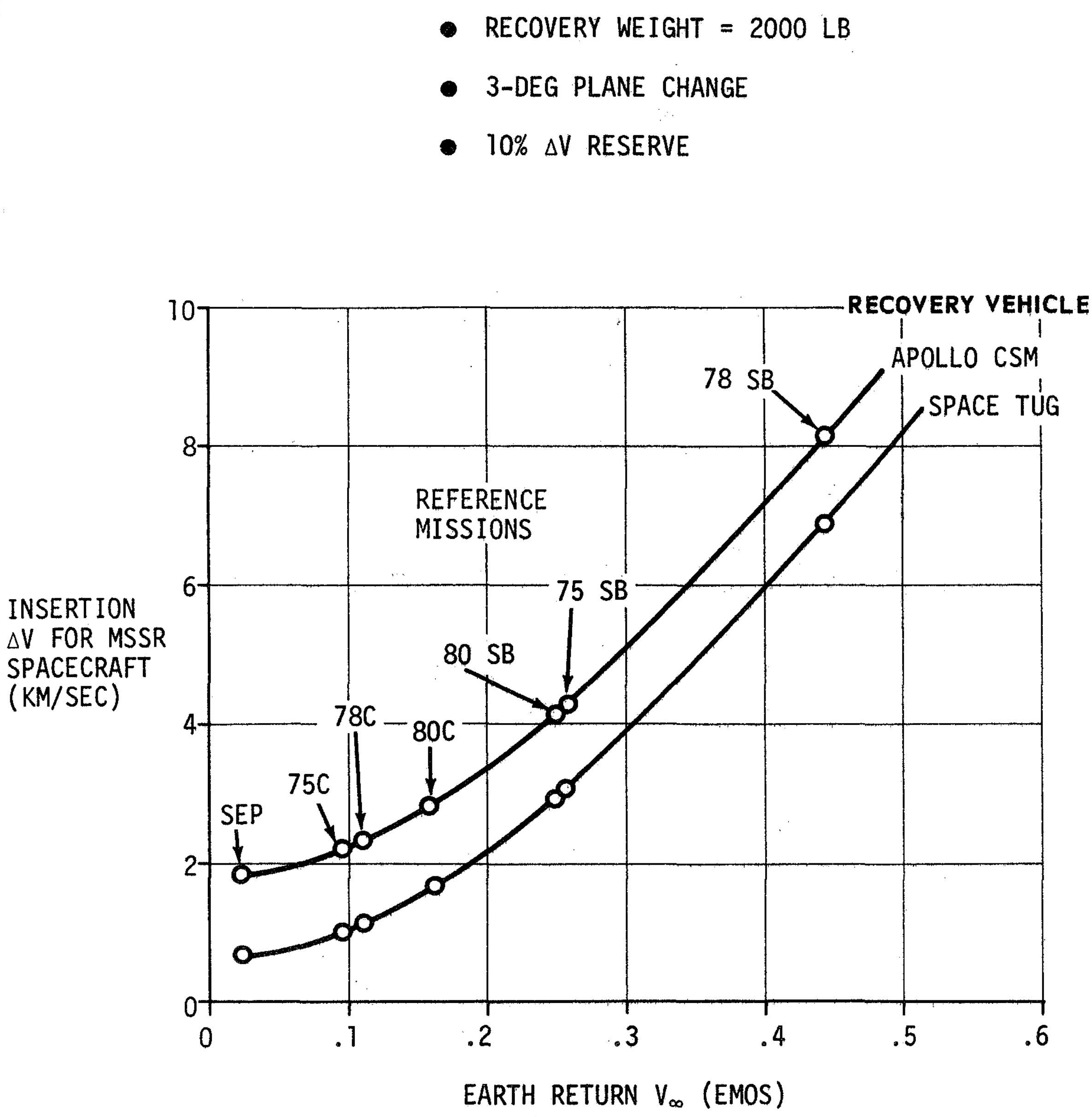


Figure 4-11. EARTH CAPTURE ΔV REQUIREMENT

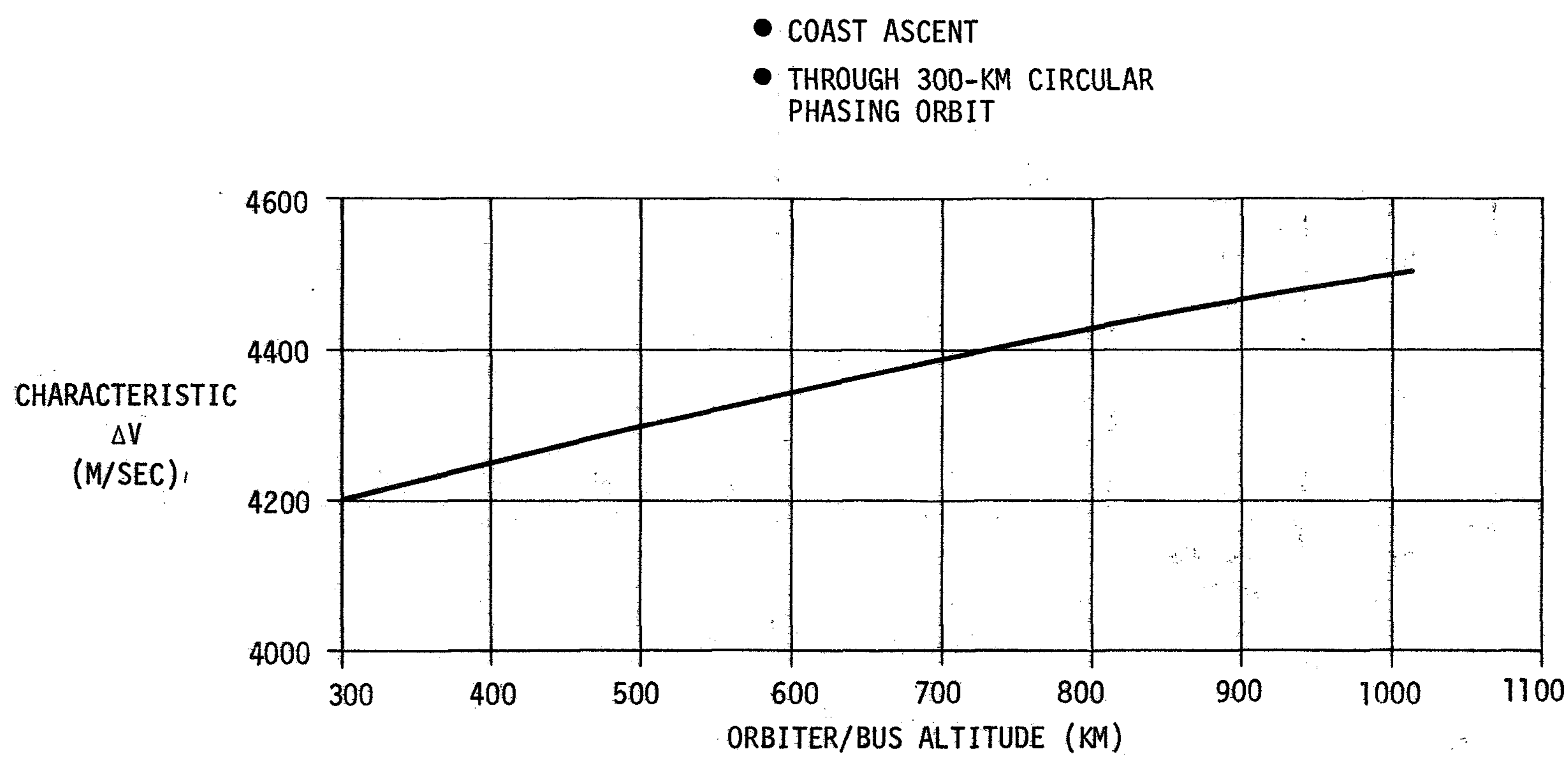


Figure 4-12. MAV CHARACTERISTIC VELOCITY

Table 4-5. SECONDARY MISSION VELOCITY REQUIREMENTS (3σ)

REQUIREMENT	CONJUNCTION CLASS ALL-CHEMICAL MISSIONS (m/sec)	VENUS SWINGBY ALL-CHEMICAL MISSIONS (m/sec)	
		1980 OUTBOUND SWINGBY	1975 INBOUND SWINGBY
OUTBOUND INTERPLANETARY MIDCOURSE CORRECTIONS	25	200	30
MANEUVER FOR DIRECT ENTRY	90	155	125
MARS DEORBIT MANEUVER			
● OUT OF CIRCULAR ORBIT	300	300	300
● OUT OF SYNCHRONOUS ELLIPSE	100	100	100
MARS ORBIT TRIM	150	150	150
MARS ORBIT RENDEZVOUS	150	150	150
INBOUND INTERPLANETARY MIDCOURSE CORRECTIONS	100	110	300

For mission and performance analysis purposes, it is assumed that launch on time capability from KSC would permit the rendezvous to be accomplished in a minimum of time following the second launch. A 185-km circular orbit is assumed to be adequate for the mission based on an anticipated short lifetime requirement. The velocity budget for rendezvous and docking maneuvers by the spacecraft is estimated to be of the order of 100 m/sec based on Northrop's past studies of automated Mars orbit rendezvous and docking. This impulse allocation was used in the parametric performance analysis to be presented in Section VI.

4.2 SOLAR-ELECTRIC/CHEMICAL MISSIONS

The determination of mission and performance characteristics of solar-electric/chemical systems is significantly more complex than for the all-chemical, ballistic concepts. In the solar-electric missions, the spacecraft system and mission trajectory characteristics are interdependent and cannot be decoupled in the sense possible with all-chemical, ballistic missions.

4.2.1 Heliocentric Mission Characteristics

The currently available parametric data on Earth-to-Mars solar-electric trajectories and performance characteristics are very limited. A search for data early in the present study indicated that practically no Mars-to-Earth information is available. Because of the very limited time frame of the present study, it became necessary to adapt existing outbound data to the inbound problem. The approach used in this analysis is outlined below.

4.2.1.1 Basic Data Source. The set of Earth-Mars trajectory and performance data employed by Northrop was the recent NASA Contractor's Report by Horsewood and Mann, Optimum Solar Electric Interplanetary Trajectory and Performance Data, CR-1524, April 1970 (ref. 6). Mission analysis data are presented for flyby and orbiter missions to Mars and other solar system targets. Optimal mission data are presented for six launch vehicles including Titan III/Centaur and SIC/SIVB Centaur vehicles. Data in the report are based on heliocentric trajectories which maximize net spacecraft mass.

The following quantities are presented in the report as functions of Earth-Mars flight time: the initial and net spacecraft masses, the electric propulsion system and propellant masses, the retro-stage propellant mass (for orbiter missions), the maximum and reference power levels, and the propulsion system total operating time. Other data presented are the minimum and maximum spacecraft-sun distances, the heliocentric transfer angle, the Earth departure and target arrival hyperbolic excess speeds, the velocity impulse for planetary capture (for orbiter missions), and the initial Lagrange multipliers.

The planets are assumed to travel in circular, coplanar orbits about the sun. The radii of the orbits are taken to be the semi-major axes of the actual planetary orbits. The authors state that these assumptions lead to performance estimates that essentially are average values of the cyclic variations in performance over several mission opportunities noted when the actual three-dimensional, non-circular planetary ephemerides are employed.

The propulsion system is assumed to operate at constant jet exhaust speed c, with a thrust subsystem efficiency η expressed by

$$\eta = \frac{bc^2}{c^2 + d^2}$$

where c is in units of km/sec and b and d are constants with assumed values of 0.769 and 14.3 km/sec, respectively. Figure 4-13 (from ref. 6) shows the value of η as a function of c.

The variation of electrical power as a function of distance from the sun is given in reference 6 by

$$P = P_{1AU} \frac{1}{r^2} \sum_{i=0} a_i r^{-i/2}$$

where P_{1AU} is the power at 1 AU solar distance and r is the distance of the spacecraft from the sun. This function was recommended by the Jet Propulsion Laboratory and accounts for the effects of the space environment on solar cells. Figure 4-14 shows a plot of the power ratio (P/P_{1AU}) as a function distance r from the sun in AU. The coefficients a_i are defined as follows: $a_o = 0.6270$, $a_1 = 5.3054$, $a_2 = -10.0376$, $a_3 = 7.1073$, and $a_4 = -2.0021$.

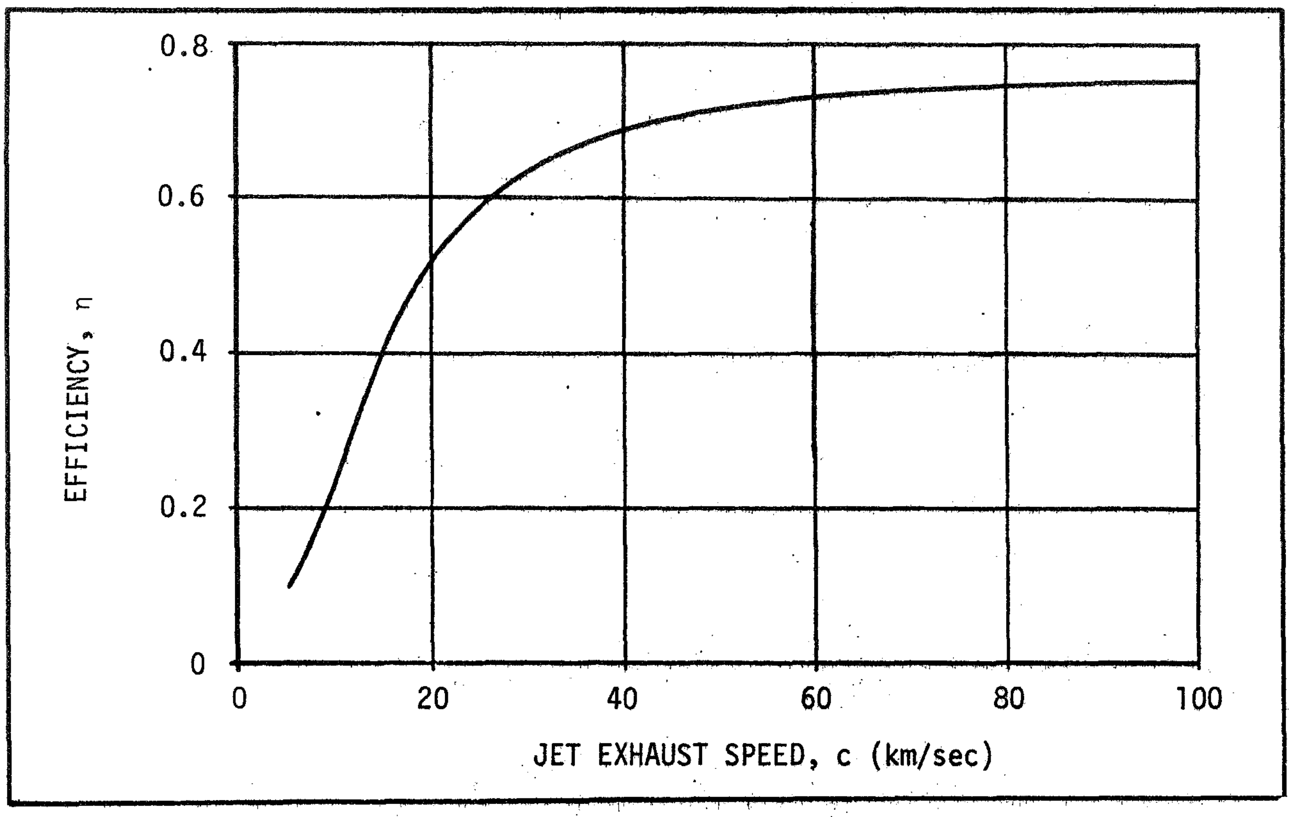


Figure 4-13. PROPULSION SYSTEM EFFICIENCY

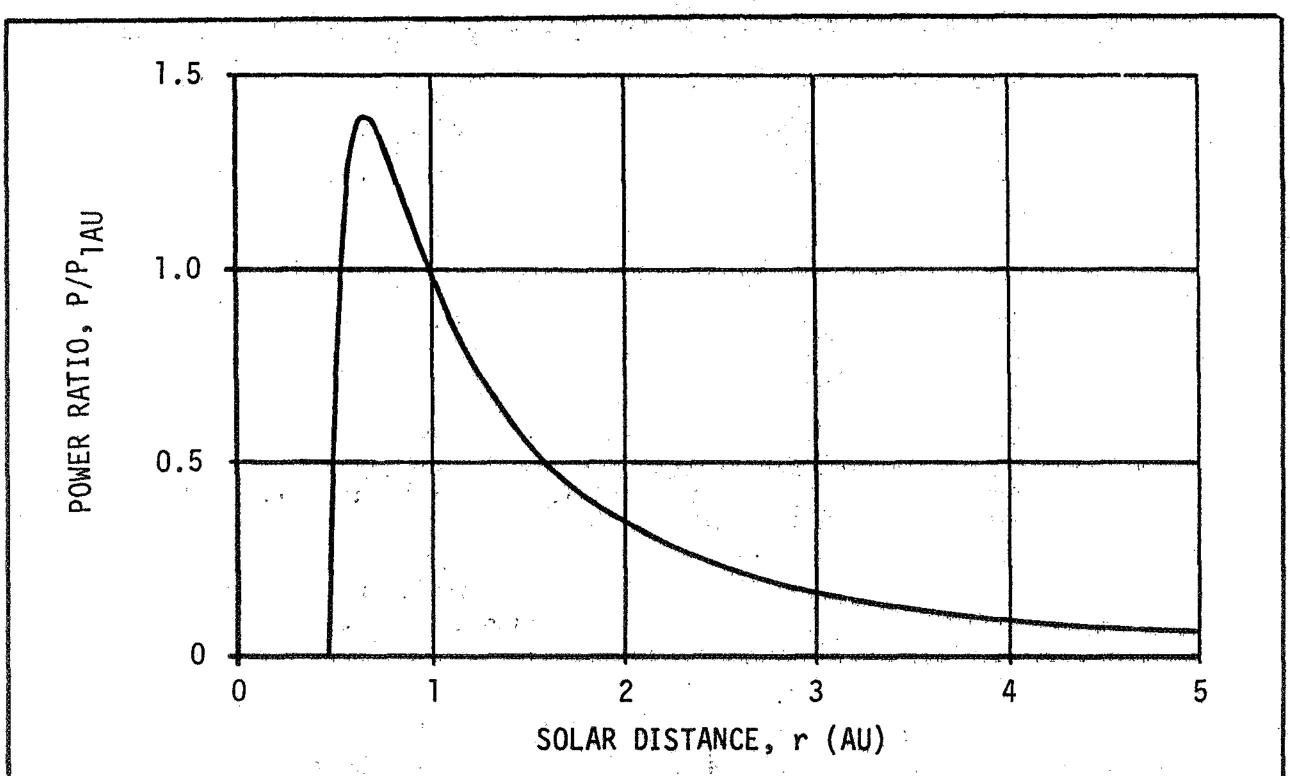


Figure 4-14. POWER VARIATION WITH SOLAR DISTANCE

4.2.1.2 Northrop Analysis Procedure. It was observed in the above data that the principal mission parameters and certain system parameters and parameter ratios plotted versus flight time are relatively insensitive to the choice of launch vehicle. From the data for Mars orbiter missions, plots were made of the following parameters versus flight time:

- Earth departure energy (C_3)
- Initial spacecraft acceleration
- Solar-electric propulsion (SEP) system specific impulse
- Propulsion time
- SEP propellant fraction (ratio of SEP propellant to Earth departure weight)
- Optimum heliocentric transfer angle
- Hyperbolic excess speeds at Mars and Earth.

These plots are given in Figure 4-15 through 4-21, respectively.

The above parametric data were used to generate mission requirements and characteristics for both the outbound and inbound legs of the MSSR mission. The assumption was made that a near-minimum energy class of solar-electric round trip missions can be approximated by employing mirror image return trajectories based on the characteristics of the parametric outbound data. Under this assumption representative round trip mission profiles were constructed as follows:

- A plot of Earth and Mars heliocentric longitude versus Julian date was prepared for each of the three mission opportunities of interest (1975, 1978, and 1980).
- The plot of optimum heliocentric transfer or travel angle was used in conjunction with the above plots to iteratively construct round trip mission profiles by matching inbound/outbound flight times and Mars stopover time to planetary positions. A systematic search scheme was used to converge on round trip profile "solutions" satisfying heliocentric transfer angle/flight time characteristics and planetary positions. The criterion used to generate reference missions was to maximize net spacecraft mass within the shortest total mission duration consistent with near-maximum performance.
- Once the reference outbound (Earth-Mars) and inbound (Mars-Earth) flight times were established, it was then possible to determine the associated mission and system performance characteristics from the data in Figures 4-15 through 4-21. The performance model developed

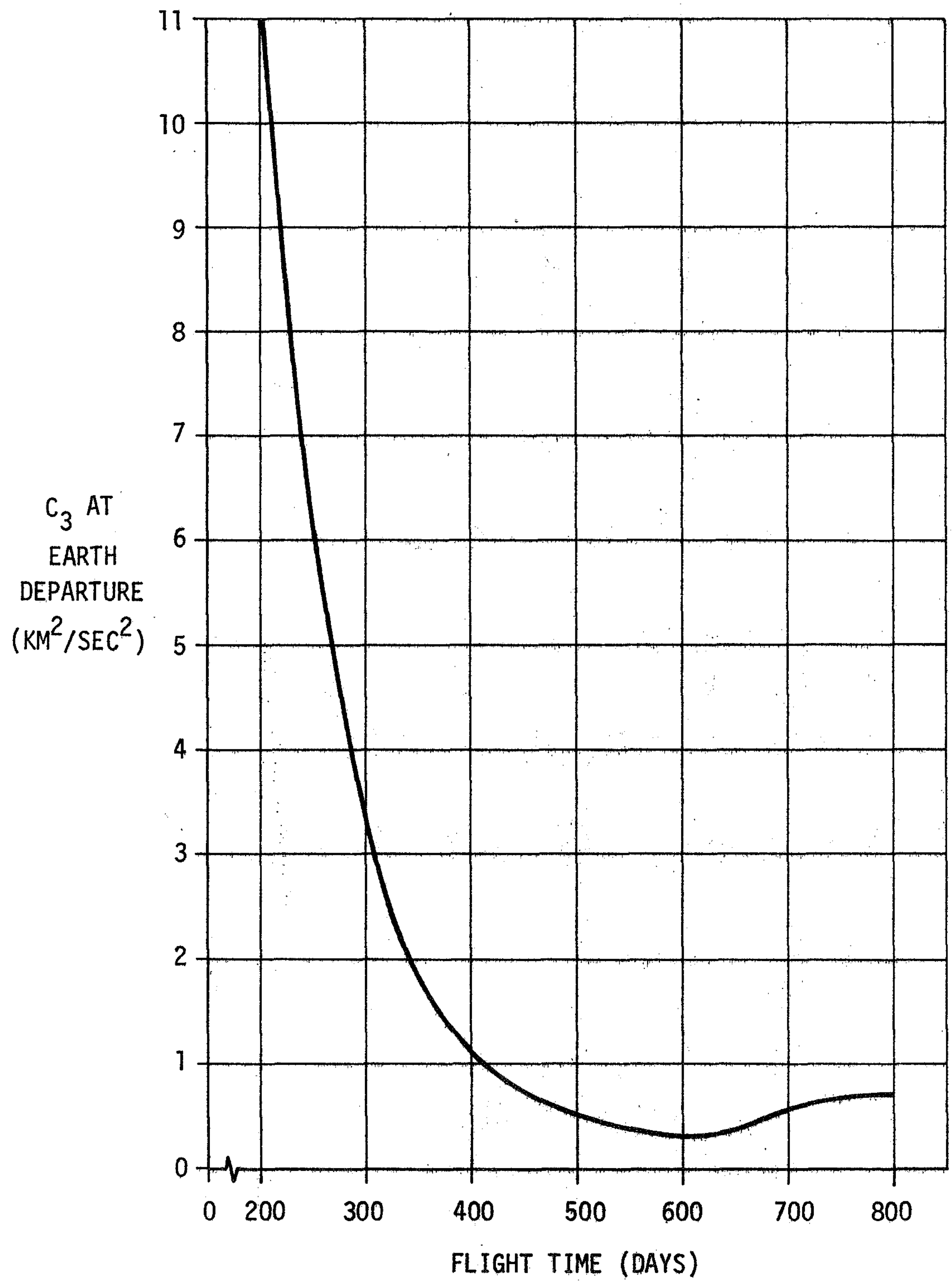


Figure 4-15. EARTH DEPARTURE ENERGY FOR SEP MISSIONS

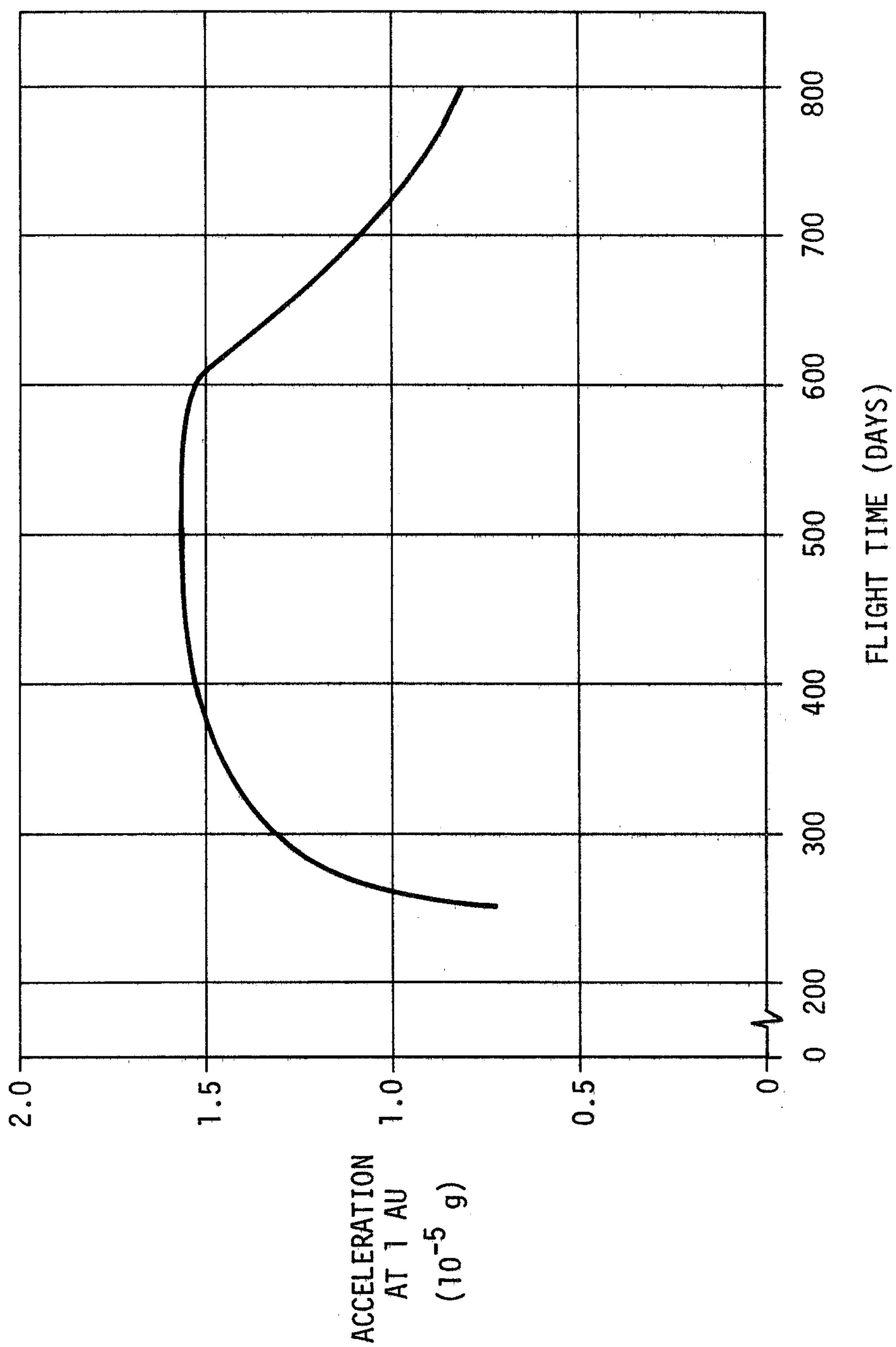


Figure 4-16. OPTIMUM SEP SPACECRAFT THRUST-TO-WEIGHT RATIO AS A FUNCTION OF FLIGHT TIME

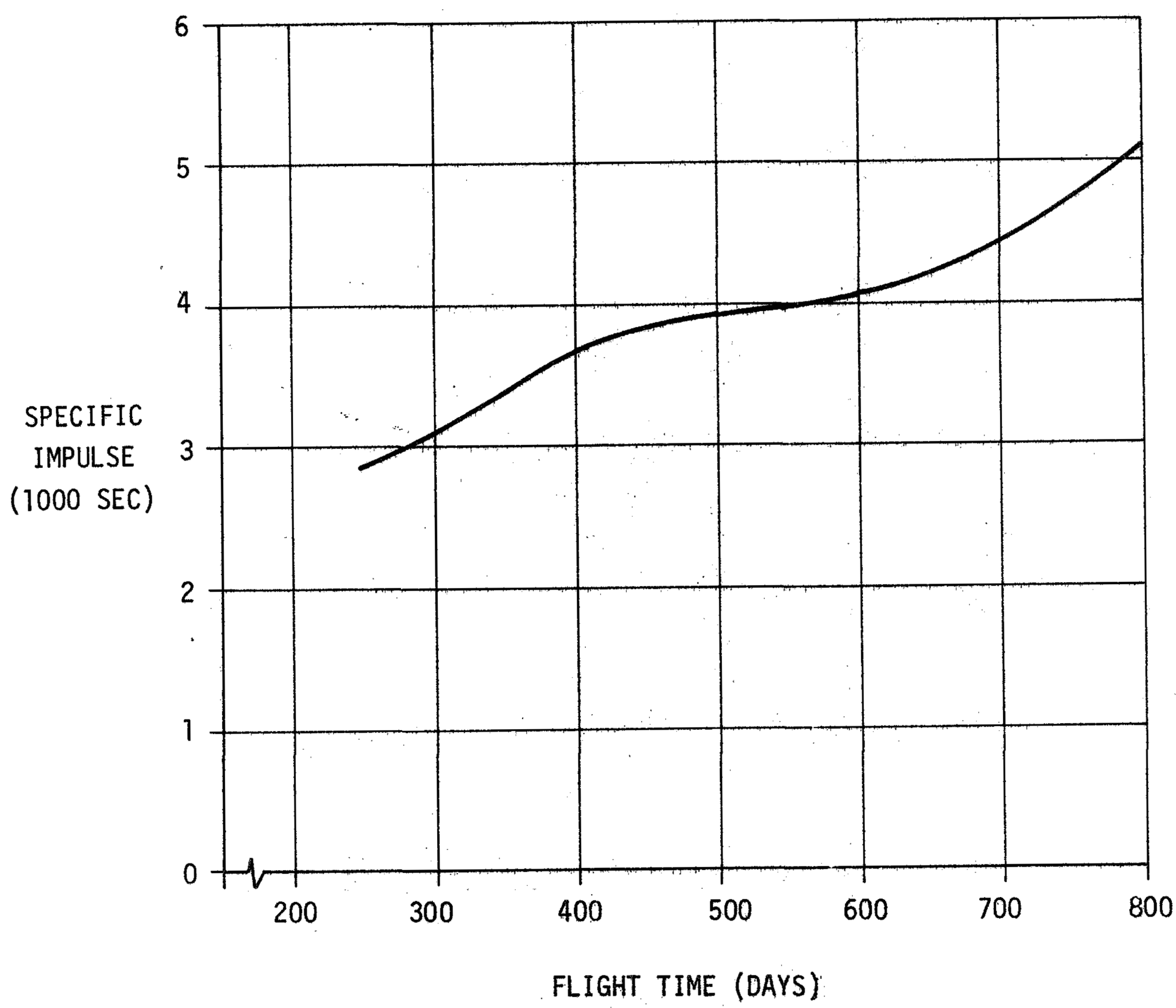


Figure 4-17. OPTIMUM SEP SPECIFIC IMPULSE AS A FUNCTION OF FLIGHT TIME

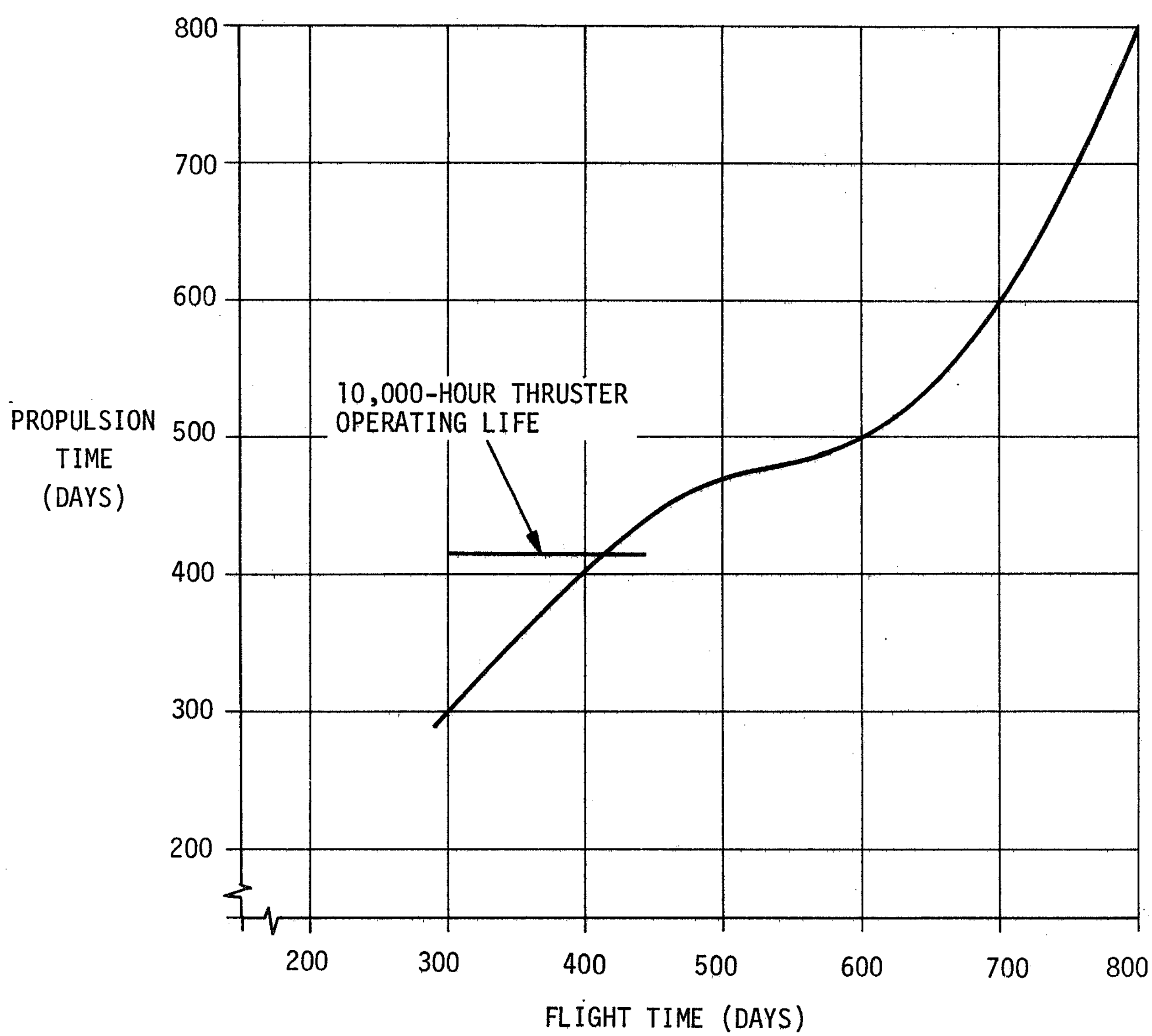


Figure 4-18. OPTIMUM SEP PROPULSION TIME AS A FUNCTION OF TOTAL FLIGHT TIME

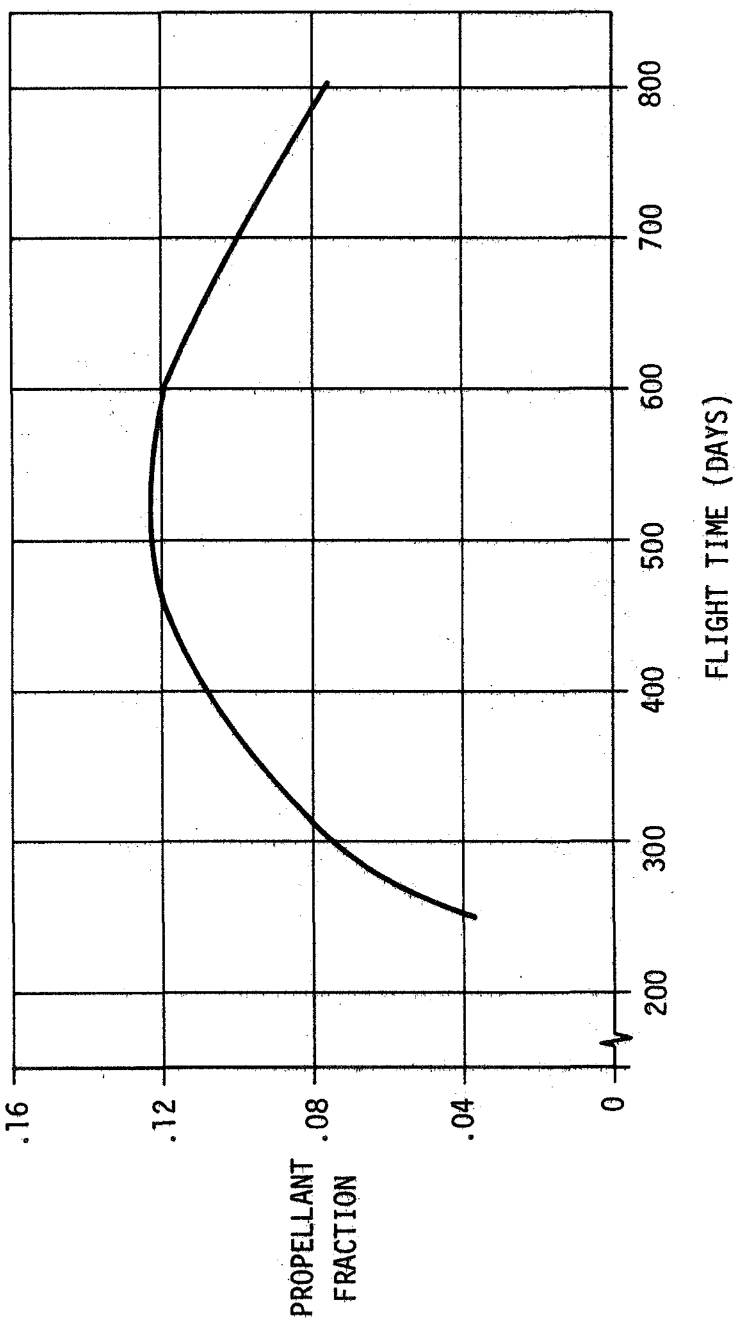


Figure 4-19. OPTIMUM SEP PROPELLANT FRACTION AS A FUNCTION OF FLIGHT TIME

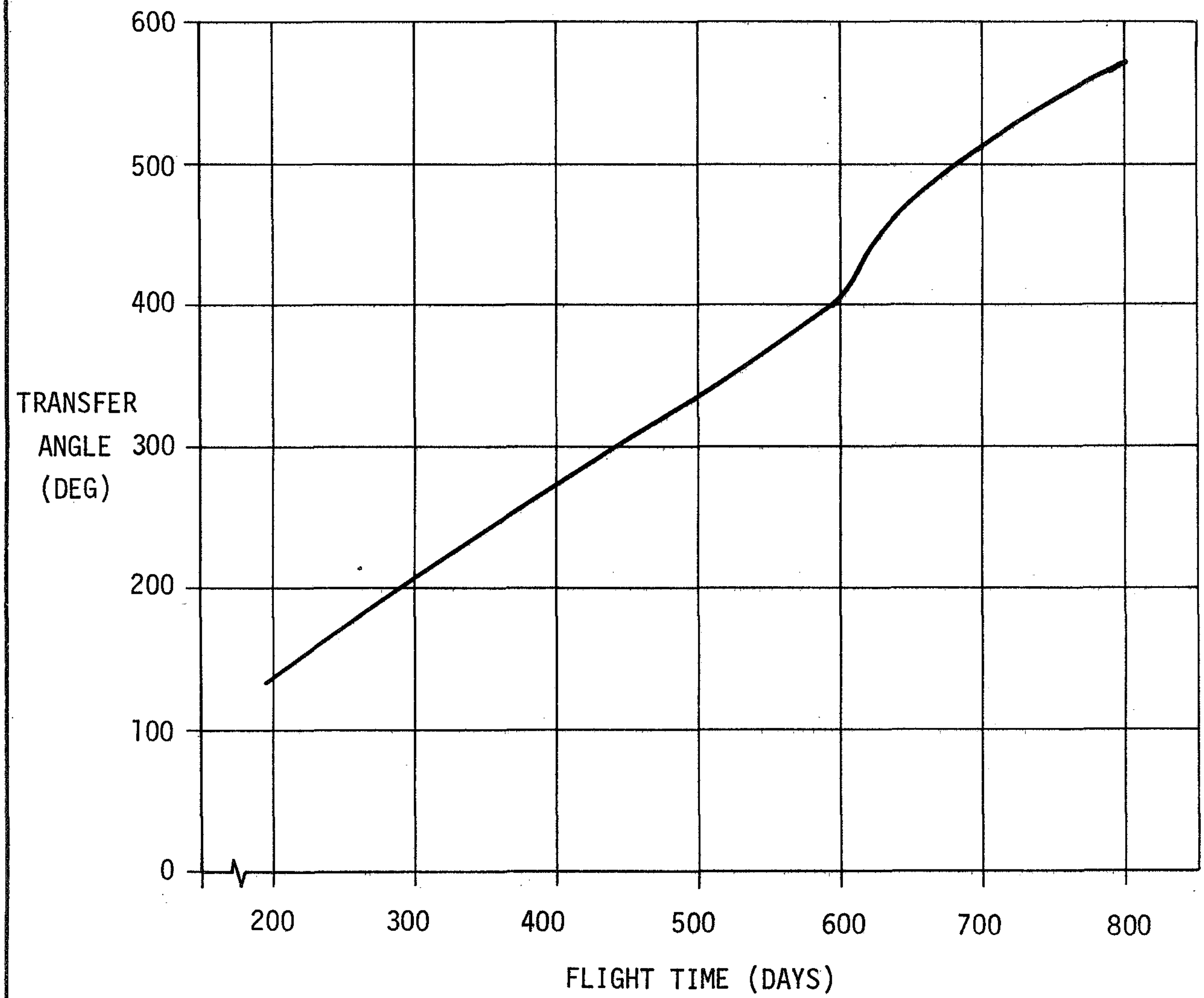


Figure 4-20. OPTIMUM HELIOCENTRIC TRANSFER ANGLE AS A FUNCTION OF FLIGHT TIME

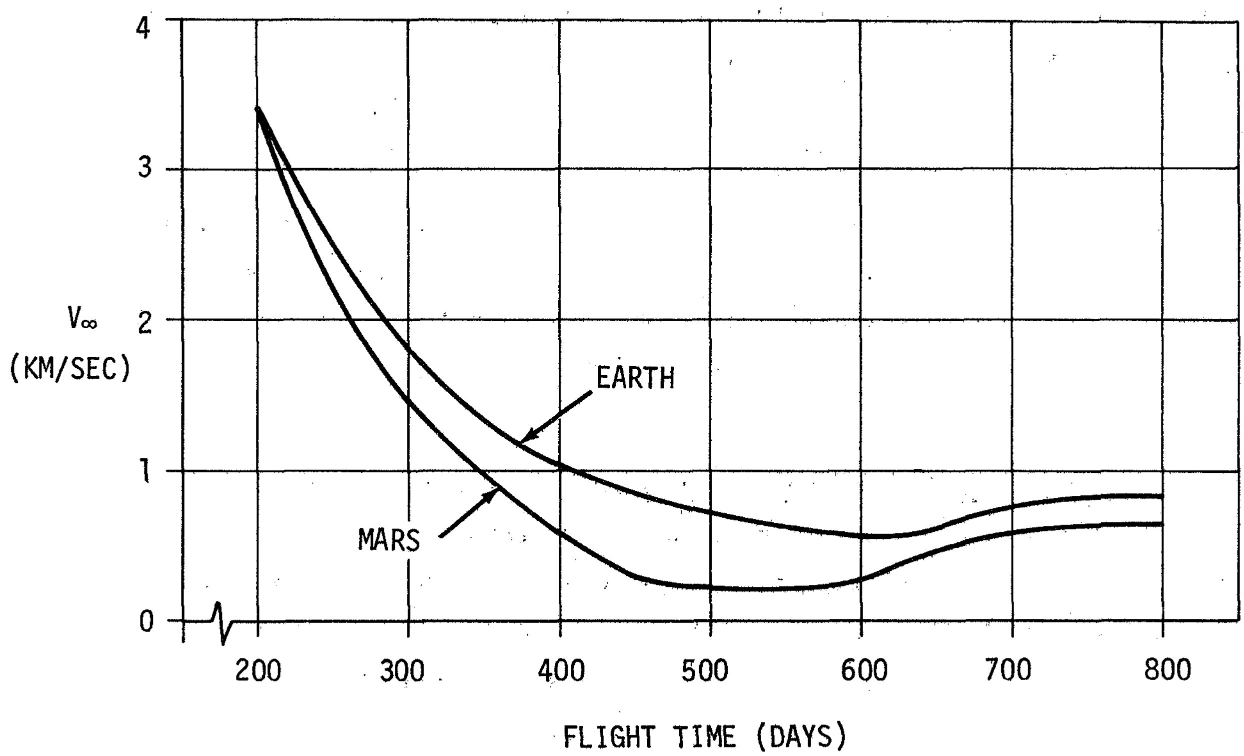


Figure 4-21. OPTIMUM HYPERBOLIC EXCESS SPEEDS AS A FUNCTION OF FLIGHT TIME

for parametric analysis of alternative solar-electric mission/system concepts will be discussed in Section VI. The model was designed to enforce all outbound and inbound parameters consistent with the mission requirements defined by the above procedure based on the parametric data contained in reference 6. The SEP propellant fractions were increased by 10 percent to account for uncertainties in the mission analysis assumptions.

4.2.1.3 Comments on Mission Analysis Procedure. While the procedure described above is not as accurate as would have been desired, it is believed that reasonably realistic mission performance trends were established consistent with the objectives and intent of the present investigation. Personal communication with the authors of reference 6 indicated that they felt the overall approach was reasonably valid for preliminary investigative purposes.

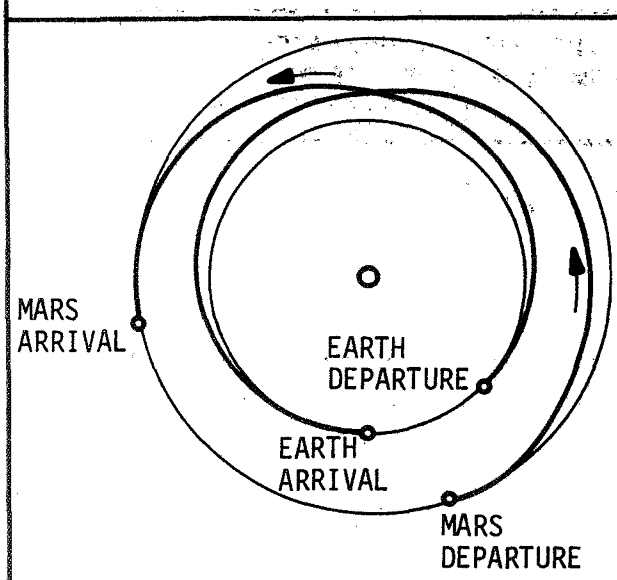
It is noted that the available data did not allow consideration of "opposition" type Mars-Earth return trajectories that typically cross inside Earth's orbit. Missions with durations less than the 1000-day class low-energy profiles would require this type of return leg.

4.2.1.4 Resulting Mission Characteristics. Based on the procedure which has been described, Figure 4-22 summarizes representative characteristics for the three mission opportunities under consideration. The total mission durations are similar to the Conjunction Class ballistic missions. It was found that a reasonable outbound leg for all missions was a 350-day trajectory. The inbound flight time can be traded with the Mars stopover time; however, the total mission duration was found to be relatively invariant.

The inbound flight time for the cases shown in Figure 4-22 is 600 days for the indicated stopover times of 130 to 170 days. It was later found in the parametric performance studies that a slightly longer stopover time of around 200 days was desirable if spiral-out escape is employed at Mars. In this mode, much of the stopover time is used in the spiral-out operation. For these missions, a 500-day inbound trajectory was found to be characteristic.

Based on the inbound and outbound trajectories corresponding to the missions in Figure 4-22, the mission velocity and system performance parameters are summarized in Table 4-6. The Earth departure energy requirement C_3 is approximately 2 km^2/sec^2. The velocity impulses for Mars capture and chemical escape are seen to be relatively low at less than 1.5 km/sec. As will be discussed later, the reference orbit at Mars is based on an altitude of 1000 km. The Earth capture impulse requirement is approximately 1.9 km/sec to allow recovery by an orbit-launched Apollo CSM.

Sufficient data were not available to account for the effects of Earth departure period on mission characteristics and requirements. Therefore, the performance margin between the required gross weight of the spacecraft at Earth departure (for a given sample return weight requirement) and the launch vehicle performance capability must provide for departure period effects.



REPRESENTATIVE FLIGHT PROFILE	MISSION	TOTAL MISSION DURATION (DAYS)	MARS STOPOVER TIME (DAYS)	MARS ARRIVAL DATE
	1975	1080	130	21 JUN 76
	1978	1090	140	5 AUG 78
	1980	1120	170	18 SEP 80

Figure 4-22. SOLAR ELECTRIC MISSION CHARACTERISTICS

Table 4-6. REPRESENTATIVE SOLAR-ELECTRIC MISSION/SYSTEM REQUIREMENTS

- 1000-KM MARS CAPTURE ORBIT
- 555 x 9100-KM EARTH CAPTURE ORBIT

EARTH DEPARTURE C_3 (km^2/sec^2)	MARS CAPTURE $\Delta V^{(1)}$ (km/sec)	MARS DEPARTURE $\Delta V^{(1)}$ (km/sec)	EARTH CAPTURE $\Delta V^{(1)}$ (km/sec)	REENTRY SPEED (km/sec)
1.82	1.45	1.34	1.87	11.1

(1) ΔV'S INCLUDE A 3% CONTINGENCY

SYSTEM PERFORMANCE CHARACTERISTIC

MISSION LEG	INITIAL ACCELERATION (g's)	SPECIFIC IMPULSE (sec)	SEP PROPELLANT FRACTION (10% CONTINGENCY)
OUTBOUND	1.46×10^{-5}	3400	.103
INBOUND	1.53×10^{-5}	4100	.131

The required initial spacecraft acceleration both outbound and inbound is approximately 1.5×10^{-5} g or about 1.5×10^{-6} m/sec^2. The optimal specific impulses are in the 3000 to 4500 seconds range. The SEP propellant fractions are approximately 10 percent outbound and 13 percent inbound including a 10 percent contingency over the values determined from the data by Horsewood and Mann in reference 6.

4.2.2 Mars Orbit Selection

Generally the discussion of Mars orbit selection considerations given in subsection 4.1.2 for the all-chemical missions applies to the solar-electric/chemical missions with the exceptions discussed in the following paragraph.

The orbit altitude based on lifetime considerations generally must be increased because of the characteristically lower effective ballistic coefficient of the solar-electric orbiter/bus vehicle. A representative worst case ballistic coefficient is estimated to be roughly 0.03 slug/ft^2. This would require an 800-km orbit for the maximum anticipated lifetime requirements based on the 20-year quarantine groundrule discussed in subsection 4.1.2. Early in the study a 1000-km circular orbit was adopted for preliminary mission analysis purposes. The mission velocity impulse requirements based on the 1000-km orbit were increased by a 3 percent contingency factor. This was later found to be a sufficient increase in ΔV to allow capture into the lower 800-km orbit sized by lifetime considerations. In addition, the characteristic velocity of the Mars ascent vehicle was sized early in the study with adequate performance margin (five percent reserve for launch to 600-km orbit) to permit ascent to orbits as high as 1000-km. Therefore, the assumption of a 1000-km orbit for purposes of parametric performance analysis of solar-electric/chemical missions was maintained throughout the study. The lander/ascent vehicle was sized based on essentially common performance requirements between the all-chemical and solar-electric/chemical concepts (for a given Mars entry mode). This permitted consideration of a common lander/ascent vehicle sizing dependent primarily on sample return payload and Mars entry mode (direct or out-of-orbit).

4.2.2.1 Mars Planetocentric Maneuver Impulse Requirements. Figure 4-23 shows the velocity impulse required for Mars capture or escape as a function Earth-Mars or Mars-Earth flight time, respectively. Curves are shown for 1000-km and 600-km

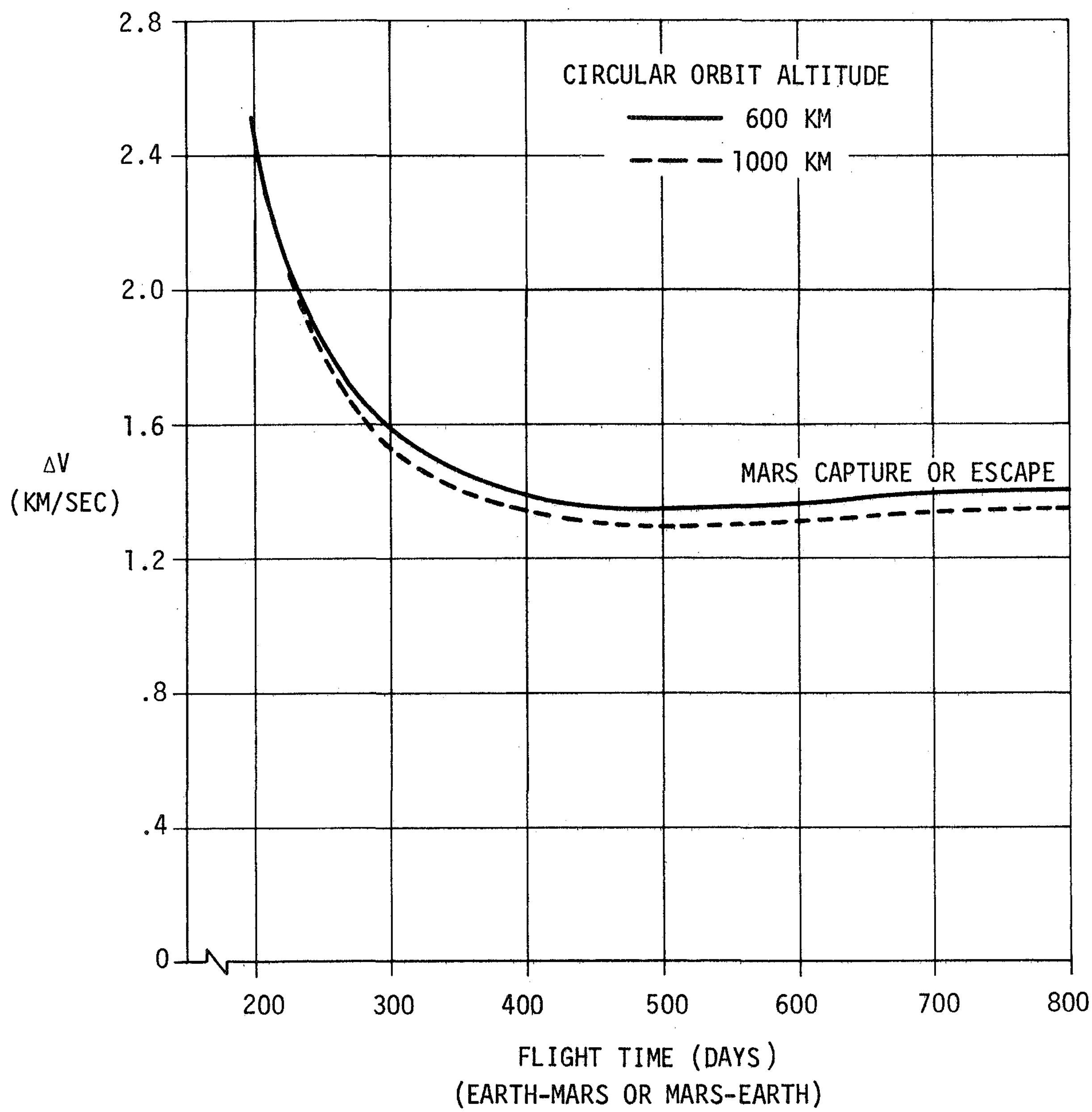


Figure 4-23. SEP MISSION ΔV REQUIREMENTS AT MARS

circular orbits. The baseline outbound Earth-Mars flight time is 350 days. The velocity impulse required for Mars capture (1.45 km/sec including 3 percent contingency) is seen to be in the knee of the curves. The Mars-Earth flight time is typically 500 to 600 days. The Mars escape impulse for chemical propulsive departure (1.34 km/sec including 3 percent contingency) is seen to be relatively independent of flight time for times greater than about 450 days.

The elliptical capture orbit mode at Mars may be used in the single-launch solar-electric/chemical concepts. The hyperbolic excess speed at Mars arrival is near 1 km/sec for the outbound 350-day trajectory. This low energy level at arrival results in a requirement of only 260 m/sec (including 3 percent contingency) to achieve capture into a Mars synchronous ellipse. The maneuver required to lower the orbit to circular after separation of the lander/ascent probe for entry is the same as that for the chemical spacecraft concepts.

4.2.2.2 Mars Spiral-Out Escape Mode Analysis. The performance characteristics of the low-thrust spiral-out escape mode alternative at Mars can be reasonably well analyzed by a method developed by Melbourne (ref. 7). Melbourne's method employs constant tangential thrust to derive analytical expressions for trajectory and performance parameters. A correction factor is then applied to the analytical expressions based on a numerical correlation between true optimal steering program results obtained from integration of the equations of motion and the analytical approximations.

The two key performance parameters of interest in the analysis of spiral-out escape at Mars are the propellant fraction required and the time required for escape. These two parameters may be computed by the following expressions based on Melbourne's method. The SEP propellant fraction may be written as

$$\mu_{pf} = 1 - \frac{1}{1 + \frac{J\, W_{mo}}{2000\ \eta\ P_{mo}}} \tag{1}$$

where

μ_{pf} = ratio of SEP propellant to total spacecraft mass at initiation of spiral-out

J = integral of acceleration squared over the spiral-out maneuver ($\int a^2 dt$) (m^2/sec^3)

W_{mo} = total spacecraft mass at initiation of spiral-out (kg)

η = thruster subsystem efficiency

P_{mo} = total power in Mars orbit (kw)

The trajectory characteristic J is given by the equation

$$J = 2000\eta \left(\frac{P_{mo}}{W_{mo}}\right) \left(\frac{\Gamma\xi}{1 - \Gamma\xi}\right) \tag{2}$$

where Γ is Melbourne's correction factor expressed by

$$\Gamma = 1 - .76382 \, (A_o)^{.24323}$$

where A_o is the initial thrust acceleration in units of g at the parking orbit. The parameter ξ in equation (2) is a function of the parking orbit circular velocity V_c and the jet exhaust speed c of the SEP system. The expression is

$$\xi = 1 - e^{-V_c/c}.$$

The SEP propellant mass W_{pso} required for spiral-out escape is given by

$$W_{pso} = \mu_{pf} W_{mo} \tag{3}$$

The time T required for the spacecraft to achieve escape velocity is given by the equation

$$T = \frac{c^2 \, \Gamma\xi \, W_{mo}}{172.8 \, \eta \, P_{mo}} \tag{4}$$

where T is in units of days. All parameters in the equation have been defined.

The above equations were incorporated into the solar-electric mission/ system performance computer program used to generate the parametric data presented in Section VI.

4.2.3 Earth Capture Orbit Selection

The discussion of Earth capture orbit selection considerations given in subsection 4.1.3 for the all-chemical missions applies to the solar-electric/

chemical missions. The velocity impulse requirement for capture is less for the SEP missions because of the low hyperbolic excess speed at Earth arrival. Figure 4-24 shows the capture impulse requirement as a function of Mars-Earth flight time. For the 500- to 600-day Earth return times under consideration, the impulse required is approximately 1.82 km/sec.

4.2.4 Mars Ascent Vehicle Velocity Requirements

As indicated in subsection 4.2.2 in discussion of Mars orbit altitude selection, the Mars ascent vehicle characteristic velocity requirements were sized such that a common vehicle could be considered for both solar-electric/ chemical and all-chemical missions. The requirement of 4340 m/sec for the nominal 600-km orbit of the all-chemical missions includes a 5 percent contingency or reserve. This is sufficient to achieve the higher 1000-km orbit of the solar-electric missions with less contingency.

4.2.5 Secondary Mission Velocity Requirements

Table 4-7 summarizes the secondary mission velocity requirements for the solar-electric/chemical concepts. All of these maneuvers could be performed by chemical propulsion systems. However, in the present study it is assumed that the interplanetary midcourse correction and Mars orbit trim requirements can be handled by the solar-electric primary propulsion system of the orbiter/ bus vehicle. All other maneuvers shown in the table are performed with chemical propulsion.

4.2.6 Earth Orbit Rendezvous Mode Requirements

The Earth orbit rendezvous departure mode concept for solar-electric/ chemical missions is essentially identical to that described in subsection 4.1.6 for the all-chemical missions. The spacecraft would utilize electrical power from batteries or small solar arrays during Earth orbit operations to avoid deployment of primary solar arrays prior to the transmars injection maneuver.

4.3 CANDIDATE EARTH LAUNCH VEHICLE PERFORMANCE CAPABILITIES

The groundrule regarding Earth launch vehicle baseline performance in the present study was to use data contained in the NASA/OSSA Launch Vehicle Estimating Factors book dated January 1970. In cases where data were not given in the OSSA book, NASA/MSFC or Northrop-generated data were used for reference.

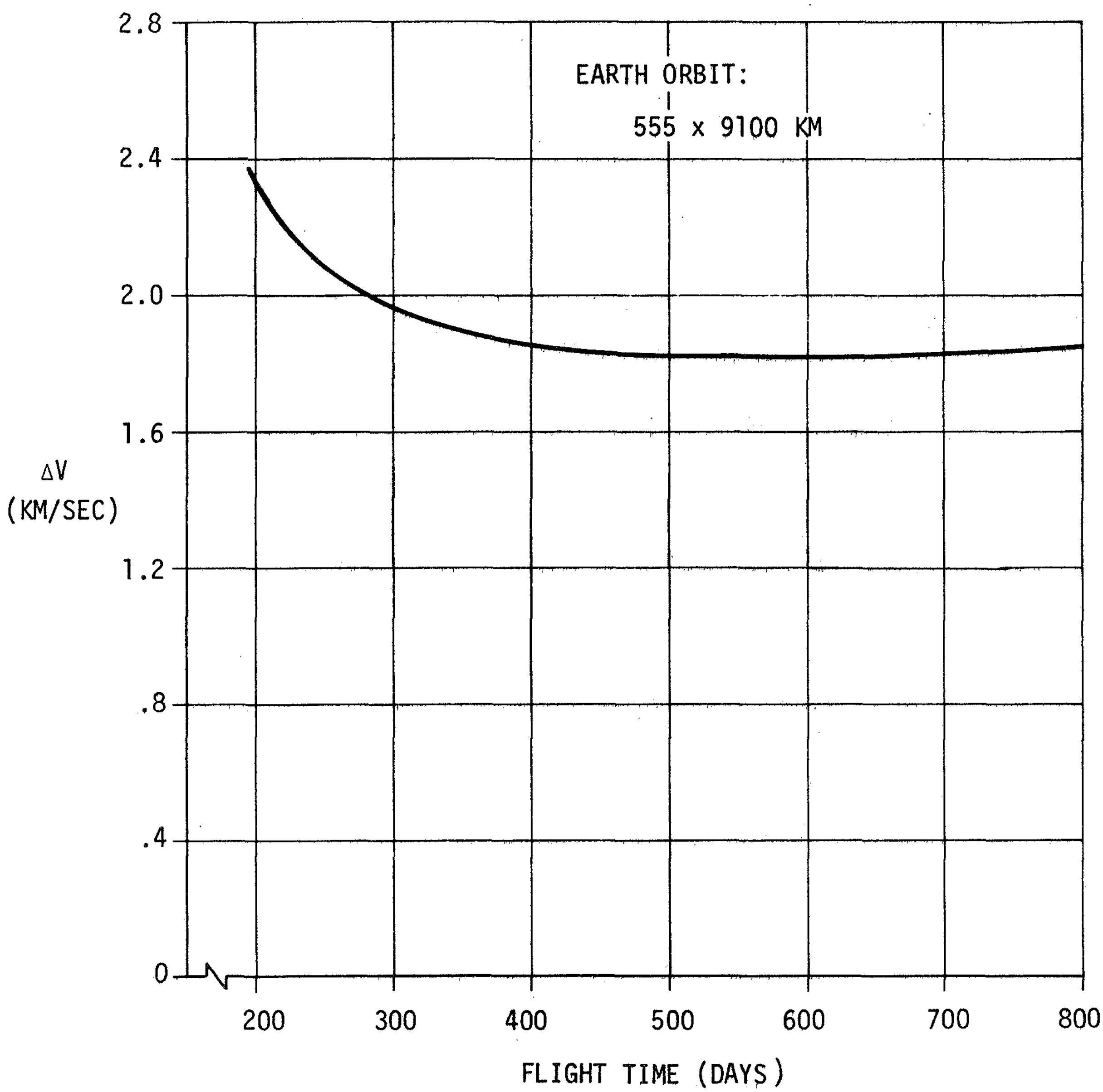


Figure 4-24. SEP MISSION ΔV REQUIREMENT FOR EARTH CAPTURE

Table 4-7. SECONDARY VELOCITY REQUIREMENTS (3σ) FOR SOLAR-ELECTRIC MISSIONS

REQUIREMENT	ΔV (m/sec)
OUTBOUND INTERPLANETARY MIDCOURSE CORRECTIONS	(20)[1]
MANEUVER FOR DIRECT ENTRY	200
MARS DEORBIT MANEUVER	
● OUT OF CIRCULAR ORBIT	300
● OUT OF SYNCHRONOUS ELLIPSE	100
MARS ORBIT TRIM	(100)
MARS ORBIT RENDEZVOUS	150
INBOUND INTERPLANETARY MIDCOURSE CORRECTIONS	(75)

(1) THE REQUIREMENTS IN PARENTHESES MAY BE PERFORMED BY THE SOLAR-ELECTRIC PROPULSION SYSTEM AS AN ALTERNATIVE TO THE USE OF CHEMICAL PROPULSION.

Table 4-8 summarizes baseline payload performance capabilities of the six launch vehicles specified by NASA for consideration in this study. Net payload capabilities are shown for each vehicle for the various mission applications of interest. The low Earth orbit capabilities of selected vehicles are shown because of the consideration of Earth orbit rendezvous departure mode concepts.

The Centaur stage characteristics used in analysis of injection performance for the Earth orbit rendezvous missions and Shuttle/Centaur performance are as follows:

- Propulsion system specific impulse = 444 seconds
- Propellants capacity = 30,000 pounds
- Stage jettison weight = 4600 pounds

Shuttle/Centaur Availability for MSSR Missions

A current flight schedule for the Space Shuttle is shown in Figure 4-25. The first four flights are primarily development flights. The first operational flight is shown for the first of calendar year 1978. This would occur after the 1977 Mars launch opportunity. This means that the first opportunity for consideration of use of the Shuttle for a MSSR mission would be the 1979 opportunity.

Table 4-8. SUMMARY OF BASELINE EARTH LAUNCH VEHICLE PERFORMANCE CAPABILITIES

VEHICLE	NET PAYLOAD CAPABILITY IN POUNDS					
	185-KM EARTH ORBIT	EARTH ESCAPE FOR SOLAR-ELECTRIC MISSION (C_3=2 km^2/sec^2)	1978/80 CONJUNCTION CLASS MISSIONS (C_3=13 km^2/sec^2)	1975 CONJUNCTION CLASS MISSION (C_3=17 km^2/sec^2)	1975 INBOUND VENUS SWINGBY MISSION (5) (C_3=21 km^2/sec^2)	1980 OUTBOUND VENUS SWINGBY MISSION (C_3=30 km^2/sec^2)
TITAN IIIC	27,500 (1)	5,000				
TITAN IIID	(31,000) (2)	(4,200)				
TITAN IIID/CENTAUR	35,000	11,900	10,000	9,300	8,100	7,400
TITAN IIID(7)/CENTAUR	46,000	15,300	13,000	12,000	10,900	9,800
INTERMEDIATE-20 (3)		34,500	27,000	24,000	20,000	16,500
INT-20/CENTAUR		(46,500)	(38,500)	(36,200)	32,700	(30,100)
SHUTTLE/CENTAUR	(47,000)	17,300 (4)	15,000 (4)	14,200 (4)	13,000 (4)	12,000 (4)

(1) NUMBERS WITHOUT PARENTHESES ARE FROM NASA/OSSA LAUNCH VEHICLE ESTIMATING FACTORS, JANUARY 1970.

(2) NUMBERS IN PARENTHESES ARE BASED ON DATA FROM NASA/MSFC.

(3) SIC (4-F1)/SIVB VEHICLE

(4) NORTHROP COMPUTED PERFORMANCE.

(5) PERFORMANCE REFLECTS PENALTY FOR 45-DEGREE LAUNCH AZIMUTH REQUIRED BY DECLINATION OF DEPARTURE ASYMPTOTE

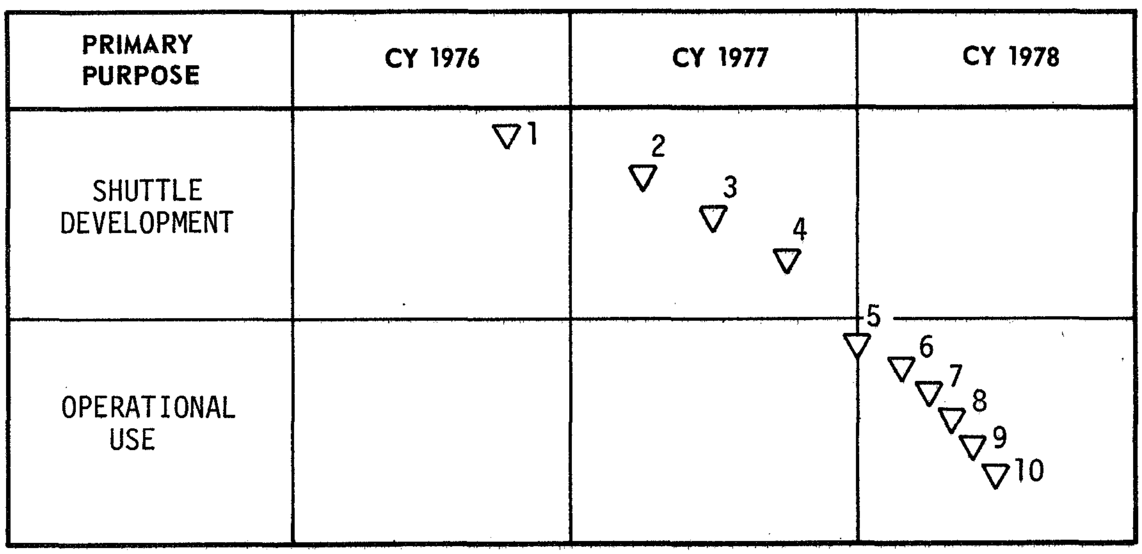


Figure 4-25. REPRESENTATIVE SHUTTLE FLIGHT SCHEDULE

Section V
SYSTEMS ANALYSIS

This section describes the systems analysis accomplished during the study. The MSSR system concepts investigated were those identified in Section III. The output of this task was the system design parameters and weights data used in the parametric performance and preliminary point design analyses.

5.1 SYSTEMS ANALYSIS GUIDELINES

The primary objective of the systems analysis was to define the spacecraft system characteristics required to return a minimum Mars surface sample. From the previous study under Contract NAS8-24714 a minimum sample weight was defined as 6 pounds. For purposes of the present effort a 10-pound sample return requirement was assumed.

The overall systems design approach was to minimize the gross earth departure weight in order to utilize smaller class launch vehicles, principally the Titan III and intermediate Saturn class vehicles. Payload envelopes of the principal launch vehicles under consideration are presented in Figures 5-1 and 5-2. Figure 5-2 gives the payload envelope for the Titan IIID/Centaur Viking bulbous shroud. The shroud envelope for the Saturn INT-20/Centaur is presented in Figure 5-2.

A constraint on the lander/ascent probe was the requirement for sterilization. The probe designs include a bioshield that completely encapsulates the probe from the time of sterilization prior to Earth launch until Mars encounter.

The approach employed in the systems analysis is illustrated in Figure 5-3. The similarity of the systems analyzed during the previous study under Contract NAS8-24714 provided a realistic basis for systems weight scaling. An existing computer program was used to evaluate the chemical systems performance and a new program was developed for the solar-electric/chemical systems.

The mass histories generated by the performance programs were used to design each systems module. Preliminary configurations were developed to

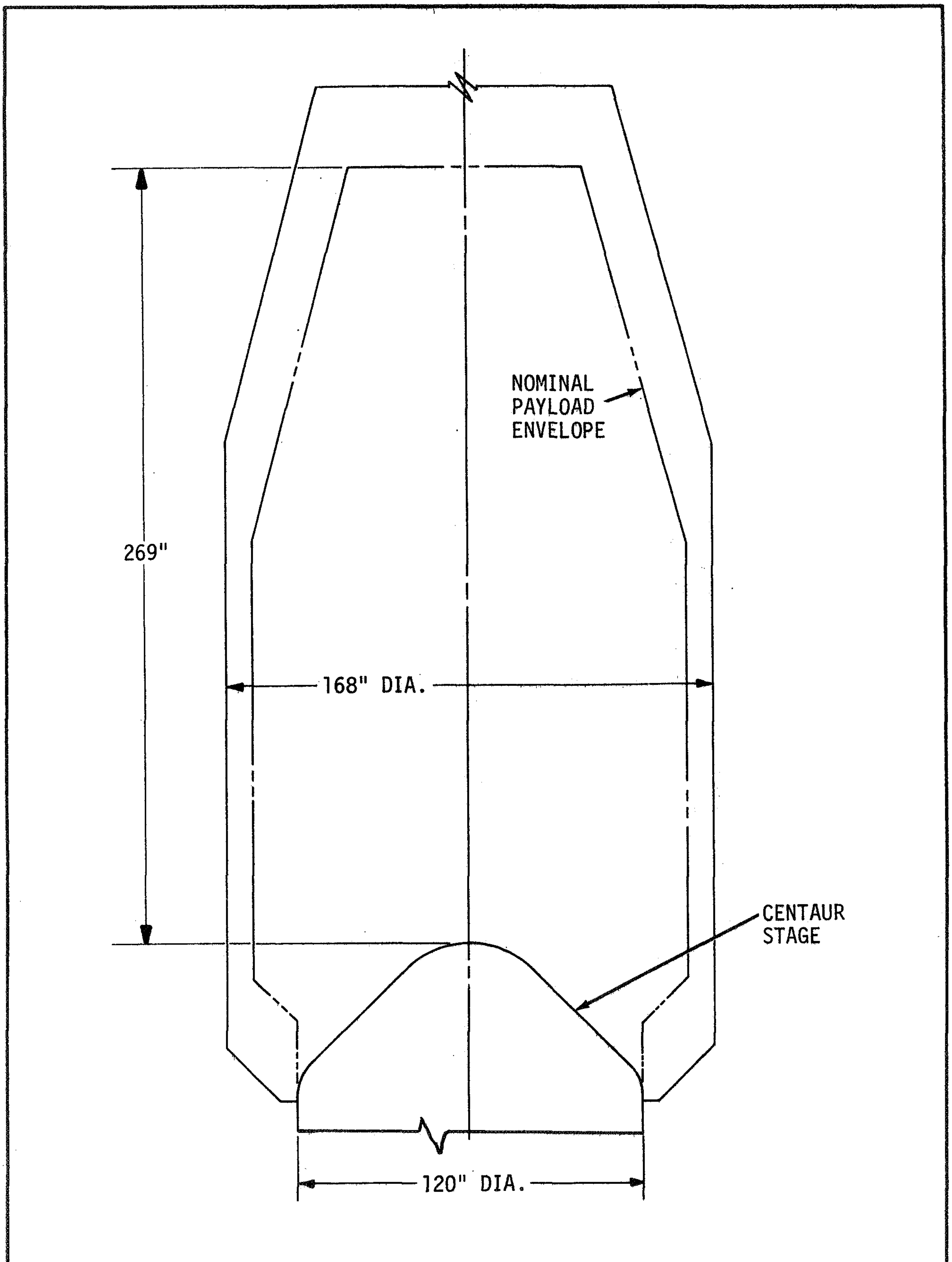


Figure 5-1. TITAN IIID/CENTAUR VIKING BULBOUS SHROUD ENVELOPE

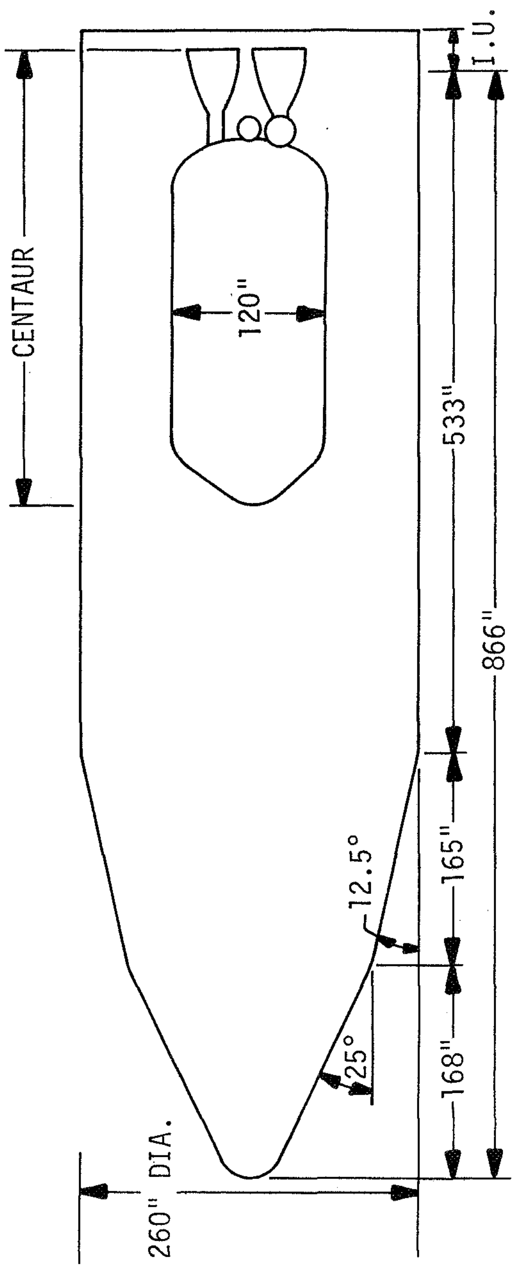


Figure 5-2. SATURN INT-20/CENTAUR PAYLOAD ENVELOPE

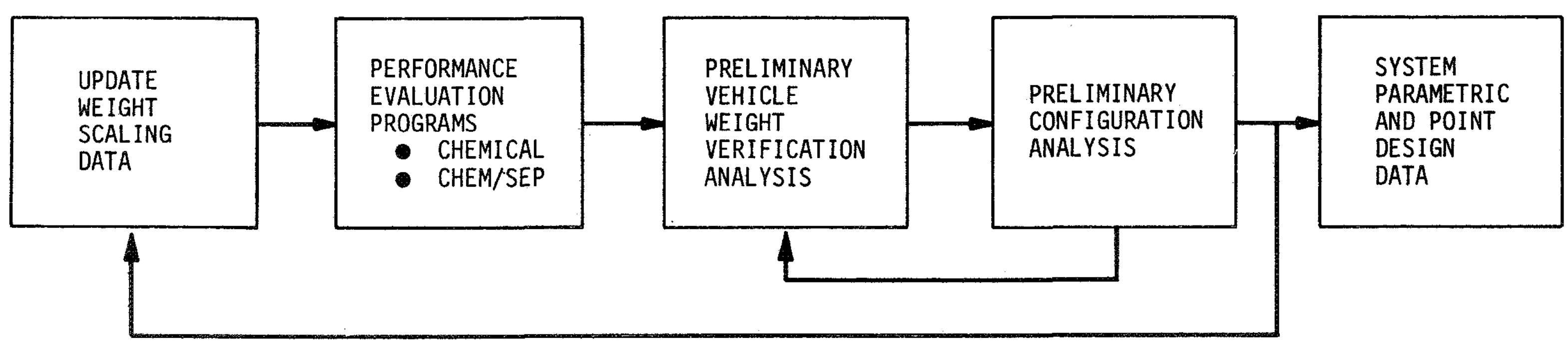


Figure 5-3. SYSTEM ANALYSIS APPROACH

verify packaging feasibility and overall vehicle envelopes. This procedure was repeated using updated input data until the output matched the system point designs required for a 10-pound sample. The required study parametric data were then generated using the iterated design data.

5.2 DEFINITION OF SYSTEM FUNCTIONAL REQUIREMENTS

System functional requirements were defined for the spacecraft system, system module, and subsystem levels for both the all-chemical and solar-electric/chemical concepts.

5.2.1 Spacecraft Functional Requirements

The definition of spacecraft functional requirements is a continuation of the mission/system definitions in Section III. Table 5-1 presents the mission characteristics that define the spacecraft system concepts for both the all-chemical and solar-electric/chemical systems.

5.2.2 System Module Functional Requirements

This subsection presents functional descriptions of all the major hardware elements that comprise both the all-chemical and solar-electric/chemical systems identified in Table 5-1. System elements are described from the launch vehicle interface forward.

5.2.2.1 Earth Launch Vehicle Adapter. The Earth launch vehicle adapter is the mechanical interface between the upper stage of the launch vehicle and the lower stage of the planetary vehicle.

5.2.2.2 Mars Braking Stage. The Mars Braking Stage performs the outbound midcourse trajectory correction maneuvers and the maneuver required to brake the orbiter/bus into orbit about Mars. The MBS also provides attitude control for the planetary vehicle during the outbound leg of the mission.

5.2.2.3 Mars Departure Stage. The Mars departure stage performs the rendezvous maneuvers in Mars orbit, the Mars departure maneuver, and the inbound midcourse correction maneuvers.

Table 5-1. SPACECRAFT CONCEPT DEFINITION MATRIX

S/C CONCEPT	EARTH DEPARTURE MODE	ORBITER/BUS PROPULSION	EARTH RECOVERY MODE	LANDER/ RETURN PROBE	SPACECRAFT BUS	ORBITER/ BUS MODULE	MARS BRAKING STAGE	MARS DEPARTURE STAGE	EARTH BRAKING STAGE	EARTH REENTRY CAPSULE	SEP MODULE	EOR DOCKING ADAPTER	ELV ADAPTER
1	SINGLE LAUNCH	ALL-CHEM	DIRECT	X		X	X	X		X			X
2	SINGLE LAUNCH	ALL-CHEM	CAPTURE	X		X	X	X	X				X
3	EOR	ALL-CHEM	DIRECT	X		X	X	X		X		X	
4	MULTIPLE DEPARTURE	ALL-CHEM	DIRECT	X(1)	X	X	X	X		X			X
5	SINGLE LAUNCH	CHEM/SEP	DIRECT	X		X	X(2)	X(2)		X	X		X
6	SINGLE LAUNCH	CHEM/SEP	CAPTURE	X		X	X(2)	X(2)	X		X		X
7	EOR	CHEM/SEP	CAPTURE	X		X	X(2)	X(2)	X		X	X	
8	MULTIPLE DEPARTURE	CHEM/SEP	CAPTURE	X(1)	X	X	X(2)	X(2)	X		X		X

(1) CONCEPT MAY INCLUDE MULTIPLE INTERPLANETARY LANDER/RETURN (ASCENT) PROBE LAUNCHES (ONE PROBE PER LAUNCH)

(2) MARS BRAKING AND DEPARTURE STAGES MAY BE COMBINED.

5.2.2.4 Orbiter/Bus Module. The orbiter/bus module receives and executes commands for control of the planetary vehicle throughout the mission.

5.2.2.5 Probe Mounting Structure. The probe mounting structure is the mechanical interface between the upper stage of the planetary vehicle and the lander/ascent probe.

5.2.2.6 Sterilization Canister. The sterilization canister is a bioshield that encapsulates the lander/ascent probe from Earth to Mars.

5.2.2.7 Probe Deorbit Module. The probe deorbit module provides the velocity decrement to deorbit the lander/ascent probe after it separates from the planetary vehicle in Mars orbit (entry out of orbit modes only).

5.2.2.8 Lander/Ascent Probe. The lander/ascent probe consists of the aerobraking system, lander, rover, and the Mars ascent vehicle.

Aerobraking System

The aerobraking system decelerates the probe from entry velocities to near zero velocity at touchdown on the planet surface.

Lander

The lander contains the subsystems required to land and support the lander science and rover.

Rover

The rover is a lander-independent surface-roving vehicle that will gather and return surface samples to the lander for loading into the ascent vehicle.

Mars Ascent Vehicle

The Mars ascent vehicle (MAV) transports the samples payload from the surface of Mars into orbit about the planet for rendezvous and docking with the orbiter/bus return vehicle.

5.2.2.9 Lander/Ascent Probe Bus. The lander/ascent probe bus is used to transport the lander/ascent probe to Mars in the dual departure concepts and consists of a propulsion module and a spacecraft module.

Propulsion Module

The propulsion module is used to perform the mid-course trajectory correction maneuvers on the Earth-Mars leg of the mission. It is also used to deflect the probe bus from the approach path at Mars onto a flyby heliocentric trajectory after the lander/ascent probe has been separated.

Spacecraft Module

The spacecraft module contains the subsystems required to deliver the probe onto its entry trajectory at Mars and is deflected onto a flyby heliocentric trajectory after probe separation.

5.2.2.10 Earth Orbit Rendezvous (EOR) Adapter. The EOR adapter is used to dock the earth departure stage to the planetary vehicle in earth orbit. The EOR adapter replaces the conventional launch vehicle adapter for those concepts requiring earth orbit rendezvous.

5.2.2.11 Solar-Electric Propulsion (SEP) System. The SEP system consists of the solar cell arrays, power conditioning, propellant and thrusters. The SEP system is part of the orbiter/bus vehicle for all solar electric propulsion system concepts. The SEP system provides low-thrust propulsion for the Earth-Mars transfer, for the Mars spiral-out departure maneuver (spiral-out mode) and for Mars-Earth transfer.

5.3 SYSTEM DESIGN APPROACH ALTERNATIVES

The overall system design approach alternatives are primarily a function of the launch opportunity. The 1975 opportunity requires that existing technology be employed in the system design. The later launch opportunities in 1977 and 1979 allow the use of technological advances until 1974. The latter date is consistent with the performance data generated under the previous study (Contract NAS8-24714) while the current (1971) technology case requires development of performance data for existing hardware components and subsystems where possible.

5.3.1 All-Chemical Concepts

5.3.1.1 System Design Approach for 1975 Opportunity. The spacecraft systems approaches and weights employed for this mission rely heavily on components from Viking and Mariner. Where development items were required, the approach was based on existing technology. The subsystems characteristics selected for this mission opportunity are presented in Table 5-2 and in the following paragraphs.

Science

The Viking orbiter and lander science payloads were used without change in the MSSR orbiter and lander. The other systems modules do not contain science instruments.

Telecommunications, Command and Control, Data Storage and Power

For the orbiter/bus and lander these subsystems were taken directly from the Viking systems. For the probe/bus (dual departure concepts) they were based on Mariner Mars '69, and for the Mars ascent vehicle the weights from the previous study under Contract NAS8-24714 were used with some performance degradation assumed to account for application of present technology.

Guidance

The Viking lander guidance system is not considered adequate for the more demanding requirements associated with direct lifting entry at Mars; therefore, some modification will be required. The Viking orbiter guidance system will also require modification to meet the approach guidance requirements for MSSR. The same is true of the Mariner guidance system used in the probe/bus.

Structure and Mechanical Systems

The structure and mechanical systems must be developed for all the major MSSR system modules but no problems are anticipated using existing technology.

Landing Gear

This subsystem was assumed to be similar in technology to that on the Viking lander.

Table 5-2. SUBSYSTEMS FOR 1975 BASELINE ALL-CHEMICAL MISSIONS

	LANDER/ASCENT PROBE	ORBITER/BUS	LANDER/ASCENT PROBE BUS	MARS ASCENT VEHICLE
SCIENCE	VIKING LANDER	VIKING ORBITER		
TELECOMMUNICATIONS			MARINER/MARS 1969	1971 TECHNOLOGY
COMMAND AND CONTROL				
DATA STORAGE				
POWER				
GUIDANCE	MODIFICATION REQUIRED FOR LIFTING ENTRY	APPROACH GUIDANCE REQUIRED FOR DIRECT ENTRY AT MARS	APPROACH GUIDANCE REQUIRED FOR DIRECT ENTRY AT MARS	STRAPDOWN SYSTEM
STRUCTURE AND MECHANICAL SYSTEMS	NEW DEVELOPMENT	NEW DEVELOPMENT	NEW DEVELOPMENT	NEW DEVELOPMENT
LANDING GEAR	NEW DEVELOPMENT			
AEROBRAKING SYSTEM	NEW DEVELOPMENT			
PROPULSION CHEMICAL	N_2O_4/MMH BELL 8570 ENGINE	N_2O_4/MMH BELL 8570 ENGINE	N_2O_4/MMH BELL 8570 ENGINE	N_2O_4/MMH BELL 8570 ENGINE
THERMAL CONTROL	BASED ON EXISTING HARDWARE	BASED ON EXISTING HARDWARE	BASED ON EXISTING HARDWARE	BASED ON EXISTING HARDWARE
ATTITUDE CONTROL	BASED ON EXISTING HARDWARE	BASED ON EXISTING HARDWARE	BASED ON EXISTING HARDWARE	BASED ON EXISTING HARDWARE

Aerobraking System

The aerobraking system was based on the entry analysis performed during the previous study and employs an aeroshell, an attached inflatable decelerator (AID) or ballute and terminal liquid propulsion.

Propulsion

The selection of propulsion system performance characteristics received special attention because of its impact on the overall system weight. Several existing engines were considered as summarized in Table 5-3. The Bell 8570 engine was selected for use throughout the system based on its weight and specific impulse. The engine is used either singly or in clusters of up to 13 engines to meet all the propulsion requirements of the MSSR system.

Table 5-3. PROPULSION CHARACTERISTICS

ENGINE	MARINER '69	ROCKETDYNE	RS-2101 ROCKETDYNE	BELL 8570	RS-18 ROCKETDYNE
PROPELLANT	N_2H_2	N_2O_4/MMH	N_2O_4/MMH	N_2O_4/MMH	N_2O_4/A-50
VACUUM THRUST (lb)	50	100	300	325	3500
NOZZLE EXPANSION RATIO	44	60	40	50/100	40
ISP (sec)	231	300	287	300/306	305
P_c (psi)	198	140	118	124	120
WEIGHT (lb)	2.5	5.5	17	12.3/15.2	210
LENGTH X WIDTH (in.)	7.9 x 2.9		18.9 x 8.7	19.7 x 9.7	51 x 31
RATED DURATION (seconds)	INDEFINITE	4800	4800	1105	520

The engine is used in two configurations, one for the lander and Mars ascent vehicle (MAV) and the other for the orbiter/bus and associated stages. For the lander/MAV the engine has an Isp of 300 seconds, expansion area ratio of 50, and a thrust of 325 pounds. For the latter applications, the engine has an Isp of 306 seconds, an expansion area ratio of 100 and a thrust of 336 pounds.

Thermal Control

The thermal control systems are based on existing technology and hardware (insulation, heaters, louvers, controls, etc.).

Attitude Control

The attitude control systems are also based on existing technology and hardware (thrusters, pressurant tanks, valves, etc.).

5.3.1.2 Systems Design Approach for 1977/1979 Opportunities. The systems design approach selected for the 1977/1979 mission opportunities was essentially identical to that selected during the study conducted under Contract NAS8-24714. This approach was based on 1974 technology. Detailed subsystem performance studies were conducted in the previous study and documented in reference 3.

The subsystem characteristics for the all-chemical systems are summarized in Table 5-4.

The direct entry technology was derived from the entry analysis performed during the previous study. As in the case of the 1975 mission/system concept, the deceleration system employs a blunt cone aeroshell (SLA 561/SLA 220 heat protection materials), an attached inflatable decelerator, and terminal liquid propulsion.

The subsystem characteristics having the greatest impact on system weight are, again, the propulsion characteristics which are presented in Table 5-5.

Table 5-4. SUBSYSTEMS FOR 1977/1979 CHEMICAL MISSIONS

	LANDER/ASCENT PROBE	ORBITER/BUS	LANDER/ASCENT PROBE BUS	MARS ASCENT VEHICLE
SCIENCE	100 POUNDS	125 POUNDS		
TELECOMMUNICATIONS	1974 TECHNOLOGY	TWO 50-WATT TWT AMPLIFIERS	MARINER/MARS 1969	NiCd BATTERY
COMMAND AND CONTROL	1974 TECHNOLOGY	16,000 WORD STORAGE		1974 TECHNOLOGY
DATA STORAGE	4 x 10^7 BITS/ RECORDER	10^8 BITS/ RECORDER		
POWER	2.0 W/LB RTGs	2.0 W/LB RTGs		
GUIDANCE	1974 TECHNOLOGY	1974 TECHNOLOGY DIRECT ENTRY AT MARS	APPROACH GUIDANCE REQUIRED FOR DIRECT ENTRY AT MARS	STRAP-DOWN SYSTEM
STRUCTURE AND MECHANICAL SYSTEMS	NEW DEVELOPMENT	NEW DEVELOPMENT	NEW DEVELOPMENT	NEW DEVELOPMENT
LANDING GEAR	NEW DEVELOPMENT			
AEROBRAKING SYSTEM	NEW DEVELOPMENT LIFTING AEROSHELL/ AID/PROPULSION			
PROPULSION	N_2O_4/MMH NEW DEVELOPMENT ENGINES Isp = 320 SEC	N_2O_4/MMH NEW DEVELOPMENT ENGINES Isp = 320 SEC	N_2O_4/MMH NEW DEVELOPMENT ENGINES Isp = 320 SEC	N_2O_4/MMH NEW DEVELOPMENT ENGINES Isp = 310 SEC
THERMAL CONTROL	1974 TECHNOLOGY	1974 TECHNOLOGY	1974 TECHNOLOGY	1974 TECHNOLOGY
ATTITUDE CONTROL	BASED ON EXISTING HARDWARE	BASED ON EXISTING HARDWARE	BASED ON EXISTING HARDWARE	BASED ON EXISTING HARDWARE

Table 5-5. PROPULSION SYSTEM CHARACTERISTICS FOR 1974 TECHNOLOGY BASE

APPLICATION	SPECIFIC IMPULSE (sec)	PROPELLANTS COMBINATION
Mars Braking Stage	320	N_2O_4/MMH
Mars Departure Stage	320	N_2O_4/MMH
Earth Braking Stage (Orbiter/Bus)	320*	N_2O_4/MMH
Mars Ascent Vehicle (1st Stage)	315	N_2O_4/MMH
Mars Ascent Vehicle (2nd Stage)	315	N_2O_4/MMH
Deorbit/Lander	300	N_2O_4/MMH

**An alternative is a solid motor with a delivered specific impulse of 290 seconds.*

5.3.2 Solar-Electric/Chemical Concepts

5.3.2.1 System Design Approach for 1975 Opportunity. The subsystems selected for the 1975 mission opportunity are presented in Table 5-6. These subsystems are essentially identical to those selected for the all-chemical system except for the addition of the solar-electric propulsion system.

Because of weight and packaging advantages, roll-out type solar cell arrays were assumed for the baseline design approach. The specific weight for the roll-out solar arrays is shown in Table 5-6 to be 33 pounds per kilowatt. This was increased to 38 lb/kw to account for degradation of the solar cells due to radiation.

The design point specific impulse for the outbound leg of the mission was 3400 seconds. The inbound specific impulse was 3900 seconds. The specific weight of the power conditioning and thrusters was 20 lb/kw based on the use of 2.5 to 3-kilowatt mercury-electron bombardment engines.

5.3.2.2 System Design Approach for 1977/1979 Opportunities. The system design approach selected for the 1977/1979 mission opportunities is similar to that selected for the all-chemical concepts with the addition of the solar-electric

Table 5-6. SUBSYSTEMS FOR 1975 BASELINE SOLAR-ELECTRIC/CHEMICAL MISSIONS

	LANDER/ASCENT PROBE	ORBITER/BUS	LANDER/ASCENT PROBE BUS	MARS ASCENT VEHICLE
SCIENCE	VIKING LANDER	VIKING ORBITER		
TELECOMMUNICATIONS			MARINER/MARS 1969	1971 TECHNOLOGY
COMMAND AND CONTROL				
DATA STORAGE				
POWER		SOLAR CELL ARRAYS (SEP)		
GUIDANCE	MODIFICATION REQUIRED FOR LIFTING ENTRY	APPROACH GUIDANCE REQUIRED FOR DIRECT ENTRY AT MARS	APPROACH GUIDANCE REQUIRED FOR DIRECT ENTRY AT MARS	STRAPDOWN SYSTEM
STRUCTURE AND MECHANICAL SYSTEMS	NEW DEVELOPMENT	NEW DEVELOPMENT	NEW DEVELOPMENT	NEW DEVELOPMENT
LANDING GEAR	NEW DEVELOPMENT			
AEROBRAKING SYSTEM	NEW DEVELOPMENT			
PROPULSION:				
CHEMICAL	N_2O_4/MMH BELL 8570 ENGINE	N_2O_4/MMH BELL 8570 ENGINE	N_2O_4/MMH BELL 8570 ENGINE	N_2O_4/MMH BELL 8570 ENGINE
SEP		SOLAR CELL ARRAY ROLL-OUT: 33 LB/KW ENGINE MERCURY-ELECTRON BOMBARDMENT: 20 LB/KW THRUST-TO-WEIGHT $1\text{-}2 \times 10^{-5}$g		
THERMAL CONTROL	BASED ON EXISTING HARDWARE	BASED ON EXISTING HARDWARE	BASED ON EXISTING HARDWARE	BASED ON EXISTING HARDWARE
ATTITUDE CONTROL	BASED ON EXISTING HARDWARE	BASED ON EXISTING HARDWARE	BASED ON EXISTING HARDWARE	BASED ON EXISTING HARDWARE

propulsion characteristics and associated configuration influences. The specific weight of the solar arrays was taken to be 33 lb/kw including the effects of environmental degradation. The specific impulse for both the outbound and inbound legs of the mission were the same as those for the 1975 opportunity. The specific weight of the thrusters and power conditioning was assumed to be 16 lb/kw. The subsystem characteristics are summarized in Table 5-7.

5.4 WEIGHT SCALING ANALYSIS

The data for the system weight analysis was derived from three principal sources, actual component weights, calculated component weights, and estimated or scaled weights.

5.4.1 Actual Component Weights

The weights used for the science, telecommunications, command and control, data storage, power, and guidance subsystems were the existing or current weights reported for Viking and Mariner systems.

5.4.2 Calculated Weights

Weights were calculated for most of the subsystems where existing component weights were not available. Engine and pressurization system weights were calculated using computer programs developed during the previous Northrop MSSR study under Contract NAS8-24714.

A computer program was written to calculate propellant tank weights based on material properties, design acceleration loading, design tank pressure, and propellant mass.

Structure weights were calculated for the spacecraft propulsion stages using preliminary loads analyses, and assuming skin-stringer-ring frame sidewalls with truss-frame secondary structure for mounting engines and propellant tanks.

Heatshield weights were calculated using aeroshell area, estimated heatshield thickness and properties for SLA 561 and SLA 220 heat protection materials.

Table 5-7. SUBSYSTEMS FOR 1977/1979 SOLAR-ELECTRIC/CHEMICAL MISSIONS

	LANDER/ASCENT PROBE	ORBITER/BUS	LANDER/ASCENT PROBE BUS	MARS ASCENT VEHICLE
SCIENCE	100 POUNDS	125 POUNDS		
TELECOMMUNICATIONS	1974 TECHNOLOGY	TWO 50-WATT TWT AMPLIFIERS	MARINER/MARS 1969	NiCd BATTERY
COMMAND AND CONTROL	1974 TECHNOLOGY	16,000 WORD STORAGE		1974 TECHNOLOGY
DATA STORAGE	4 x 10^7 BITS/ RECORDER	10^8 BITS/ RECORDER		
POWER	2.0 W/LB RTGs	2.0 W/LB RTGs		
GUIDANCE	1974 TECHNOLOGY	1974 TECHNOLOGY DIRECT ENTRY AT MARS	APPROACH GUIDANCE REQUIRED FOR DIRECT ENTRY AT MARS	STRAP-DOWN SYSTEM
STRUCTURE AND MECHANICAL SYSTEMS	NEW DEVELOPMENT	NEW DEVELOPMENT	NEW DEVELOPMENT	NEW DEVELOPMENT
LANDING GEAR	NEW DEVELOPMENT			
AEROBRAKING SYSTEM	NEW DEVELOPMENT LIFTING AEROSHELL/ AID/PROPULSION			
PROPULSION CHEMICAL	N_2O_4/MMH NEW DEVELOPMENT ENGINES Isp = 320 SEC	N_2O_4/MMH NEW DEVELOPMENT ENGINES Isp = 320 SEC	N_2O_4/MMH NEW DEVELOPMENT ENGINES Isp = 320 SEC	N_2O_4/MMH NEW DEVELOPMENT ENGINES Isp = 310 SEC
SOLAR ELECTRIC		SOLAR CELL ARRAY ROLL-OUT: 33 LB/KW ENGINE MERCURY-ELECTRON BOMBARDMENT: 16 LB/KW THRUST TO WEIGHT 1-2 x 10^{-5} g		
THERMAL CONTROL	1974 TECHNOLOGY	1974 TECHNOLOGY	1974 TECHNOLOGY	1974 TECHNOLOGY
ATTITUDE CONTROL	BASED ON EXISTING HARDWARE	BASED ON EXISTING HARDWARE	BASED ON EXISTING HARDWARE	BASED ON EXISTING HARDWARE

5.4.3 Weight Scaling Equations

Weight scaling equations were employed where existing weights were not available and the weights could not be calculated directly.

The aeroshell weight less heatshield was determined from the following equation modified from reference 14:

$$W_{AS} = 1.655 \times 10^{-4} (W_E)^{1.5} (A/g_{max})^{.4}$$

where

W_{AS} = weight of the aeroshell (lb)

W_E = weight of the probe at entry (lb)

A = area corresponding to the aeroshell base diameter (ft^2)

g_{max} = the maximum deceleration g loading encountered during entry

The weight of the attached inflatable decelerator (AID) was determined from the following equation from reference 15:

$$W_{AID} = 1.55\ D^2\ (.368 + .353 \times 10^{-3} D\ q)$$

where

W_{AID} = weight of the AID (lb)

D = diameter of the aeroshell (ft)

q = dynamic pressure at AID deployment conditions (lb/ft^2)

The propellants weight for lander/ascent probe terminal descent were determined from the following equation:

$$W_P = .0809\ W_{GL}$$

where

W_P = weight of terminal descent propellants (lb)

W_{GL} = gross landed weight (lb)

The constant in the equation was determined based on the terminal velocity characteristic of the entry trajectory at ignition and accounts for propulsive descent losses.

A brief literature survey was conducted during the study in order to ascertain SEP propellant tankage weight coefficients, and specific weights of the solar-electric power plant subsystems which are compatible with 1971 and 1974 technology bases.

Survey papers by various authors (references 8, 9, 10, 11, 12, 13) were reviewed and data tabulated as shown in Tables 5-8 and 5-9. Based on these results the following specific weights and coefficients were selected as representative of the 1971 and 1974 technology bases:

TECHNOLOGY TIME FRAME REFERENCE	SOLAR PANEL SPECIFIC WEIGHT (lb/kw)	SOLAR PANEL DEGRADATION	THRUSTER SUBSYSTEM (lb/kw)	TANKAGE AND FEED SYSTEM FACTOR
1971	Roll-out @ 33 Fold-out @ 50	15%	20	3% of Hg Propellant Weight
1974	Roll-out @ 28 Fold-out @ 43	15%	16	3% of Hg Propellant Weight

The fold-out array data are based primarily on experimental and projected data by Boeing/JPL studies (references 9 and 12). Roll-out solar array data are based on experimental efforts at General Electric, Ryan, and Fairchild under auspices of the Jet Propulsion Laboratory in Pasadena (references 9, 12, and 13).

Ion thruster subsystem specific weight for a 2.5 to 3 kw mercury electron bombardment type thruster (including the power conditioning equipment) was selected on the basis of studies and experimental research conducted at the NASA Lewis Research Center and the JPL/Hughes Aircraft Company work on power conditioning equipment and thrusters.

Table 5-8. SURVEY OF SOLAR CELL ARRAY TECHNOLOGY

AUTHORS	REFERENCE	TYPE DATA & DATA SOURCE	TECHNOLOGY TIME FRAME REFERENCE	ARRAY SIZE AND TYPE	SPECIFIC WEIGHT (lb/kw)	PREDICTED DEGRADATION OF SOLAR ARRAY DUE TO RADIATION
Mullin Barber Zola	8	Analytical study by Ratcheson	1964	50kw fold-out	Predicted 50.6	--
		Recent analysis by Barber	1969	14kw fold-out	Predicted 48.5	18%
		Analysis by Carlson	Advanced technology assuming electroforming	-	Predicted 26.4	--
		Experimental Results by Ryan, G. E. & Fairchild	1969-70	Roll-out	Predicted 33	--
Friedlander	9	Estimated by author	1970	10-15kw Roll-out	33	20%
		Boeing/JPL 1966 design goal	1970	50kw Fold-out	50	
		North American Rockwell study, 15 Mar 1970, JPL Contract No. 952394	1970	2.5 kw Roll-out	30.8	--
Sauer	10	Sandstrom, J. D. paper, Aug. 13, 1968	1971	8.4kw	58.7	18%
Lazar	11	Ratcheson W. I. paper, Oct. 65	Current technology	to 50kw	30-50	--
Barber Goldsmith Edberg	12	Boeing data	Current technology	to 50kw Fold-out	50	--
		JPL data	Early 1970 technology	10kw Roll-Out	33	--

Table 5-9. THRUSTER SUBSYSTEM TECHNOLOGY

AUTHORS	REFERENCE	TYPE DATA & DATA SOURCE	TECHNOLOGY TIME FRAME REFERENCE	POWER CONDITIONING EQUIPMENT		THRUSTER		TVC, STRUCTURE & CONTINGENCY		OVERALL THRUSTER SUBSYSTEM	TANKAGE FACTOR (INCL. FEED SYSTEM)	
				DESCRIPTION	SPECIFIC WEIGHT	TYPE	SPECIFIC WEIGHT	TYPE TVC	SPECIFIC WEIGHT	SPECIFIC WEIGHT	TYPE PROPELLANT	TANKAGE COEFFICIENT
Mullin Barber Zola	8	SERT II, and other current program goals	Demonstration by 1970; Flight by mid 1970's	2:1 power ratio η = 92%	11 lb/kw (overall power range 5-20 kw)	Mercury electron bombardment (hollow cathode)	4.4 lb/kw (15cm Dia) @ 2-2.5 kw thruster module	gimbal-translator type	6.6 lb/kw	22 lb/kw	Liquid Mercury	3% of Propellant weight
Friedlander	9	Estimated by author	1970			2-3 kw module 30cm mercury electron bombardment				26.4 lb/kw		3% of Propellant Weight
		TRW Systems Group Study, 15 March 1970, JPL Contract No. 952394	1970			2.25 kw 3:1 throttling range	9.2 lb/kw includes feed system				Liquid Mercury	
		Hughes Aircraft Co./ JPL Test	Mid 1960's	2.5 kw unit η = 90%	14.1 lb/kw							
		Author	Current Technology Goals	2.5 kw η = 90-95% 10,000-hr. life	8.8 lb/kw							
		Author	Current Technology					Mechanical Actuator ±10° Gimbal & Two Axis Translation	3.9 lb/kw			
Sauer	10	AIAA Paper 67-698	1971	η = 93%		2.5 kw mercury electron bombardment 70% thruster efficiency				16.1 lb/kw	Liquid Mercury	3% of Propellant Weight
Kerrisk Bartz	13	JPL Report No. ASD 760-18	1971		Goal: 8 lb/kw; experi-mentally verified 12 lb/kw	2.5 kw mercury electron bombardment	5.4 lb/kw			~14 lb/kw	Liquid Mercury	3% of Propellant Weight

A moderate improvement in specific weight of the power plant which amounts to about 15 percent each for the solar array subsystem and the thruster subsystem was assumed for the 1974 technology base. These estimates appear realistic based on the general trends of SEP research and progress over the past decade. Additionally, a range of power plant specific weight was considered parametrically during the study for selected missions in order to ascertain the overall performance impacts of reductions or increases in specific weights.

Tankage and feed system coefficients for mercury propellant are quoted by several authors (references 8, 9, and 10) to be about 3 percent of propellant weight. Therefore, this coefficient was used throughout the study for both the 1971 and 1974 technology bases.

The maximum operating times for thrusters was assumed to be 10,000 and 12,500 hours for the 1971 and 1974 technology bases, respectively.

Section VI
PARAMETRIC PERFORMANCE ANALYSIS

This section presents the results of a broad parametric analysis of performance characteristics of the alternative mission/system concepts under consideration. The principal aim of this analysis was to define gross earth departure weight requirements for the MSSR system as a function of sample return weight and primary mission mode and system options. Both the all-chemical and solar-electric/chemical orbiter/bus approaches were analyzed. The two technology bases discussed in Section V are reflected in the analysis.

The candidate earth launch vehicle payload capabilities are superimposed on the required mission performance to identify and establish the mission/system approaches which appear to be feasible and attractive from a performance standpoint.

The following subsections describe the method of analysis and present the results first for the all-chemical mission/system concepts and then for the solar-electric/chemical concepts.

The order of presentation of results is directly correlated with the eight spacecraft concepts which were defined in Section III in Figures 3-3 through 3-10. The order of data presentation is as summarized in Table 6-1.

6.1 METHOD OF ANALYSIS

The method of data generation for both the all-chemical and solar-electric/chemical concepts was to use computerized mission/system performance models. The computer programs were employed in an iterative mode whereby preliminary point design checks were made to verify the performance and system weights generated by the mission/system models. The mission and systems characteristics and weight scaling data discussed in Sections III, IV, and V were used as inputs into the parametric studies.

6.1.1 All-Chemical Mission/System Performance Program

The performance program used for analysis of the all-chemical concepts was

Table 6-1. SPACECRAFT CONCEPTS FOR PARAMETRIC PERFORMANCE ANALYSIS

SPACECRAFT CONCEPT	EARTH DEPARTURE MODE	ORBITER/BUS PROPULSION	EARTH RECOVERY MODE
1	SINGLE LAUNCH	ALL-CHEM	DIRECT
2	SINGLE LAUNCH		CAPTURE
3	EARTH ORBIT RENDEZVOUS		DIRECT
4	DUAL DEPARTURE		DIRECT
5	SINGLE LAUNCH	CHEM/SEP	DIRECT
6	SINGLE LAUNCH		CAPTURE
7	EARTH ORBIT RENDEZVOUS		CAPTURE
8	DUAL DEPARTURE		CAPTURE

developed under Contract NAS8-24714. Slight modifications were required to handle the new dual departure modes introduced for the first time in the present study. The program is documented in detail in reference 2. The input to the program consists of:

- Mission mode options
- All primary and secondary mission velocity impulse requirements
- Specific impulses of all propulsion systems
- Weight scaling constants and coefficients
- Mars lander probe ballistic and drag coefficients
- Selection of parameters to be parametrically varied.

The program computes a complete mass history of the mission for the selected combination of mission mode options and provides a listing of all major system weights and the required Mars entry probe diameter for each value of the parameter selected for variation, such as sample weight.

6.1.2 Solar-Electric/Chemical Performance Program

A performance model was derived and programmed for analysis of the solar-electric/chemical mission/system concepts under consideration. A detailed description of this model is given in the Appendix. Briefly, the inputs are as follows:

- Mission mode options
- Outbound/inbound accelerations and specific impulses
- Chemical maneuver velocity impulses
- Specific impulse of chemical systems
- SEP outbound/inbound propellant fraction requirements (heliocentric transfer)
- Specific weights of solar arrays and thrust subsystem
- Weight scaling constants and coefficients
- Mars orbit radius to initiate spiral-out escape
- Sample return weight requirement.

The program computes a complete mission mass history for the input combination of mission mode options for a specified sample return weight. Sample weight can be automatically varied over a specified range to generate a series of mission cases. An iteration is made on the required orbiter/bus module weight to determine the earth departure weight requirement based on the input parameters. An iteration is incorporated in the program to jettison the necessary portion of solar arrays at Mars to match the power, acceleration and specific impulse parameters specified for the Mars-Earth return leg of the mission.

The output of the program consists of:

- All major system weights
- Design power and solar array area
- Time to spiral-out escape Mars (if this mode used).

6.2 PARAMETRIC DATA FORMAT

The principal computed parameter of interest in the broad parametric performance analysis of mission/system alternatives is the gross earth departure weight of the MSSR spacecraft as a function of sample return weight. Thus, the primary format for data presentation in the present report is gross departure weight required versus sample weight for given mission mode options, system options, mission opportunities, and spacecraft propulsion technology base (1971 or 1974). All basic performance analysis results are presented in this form with sample weight as the independent parameter. The impact of special weight items such as lander science, rover, and orbiter/bus equipment module weights on gross earth departure weight is shown in the same format. The power required at 1 AU for the solar-electric/chemical orbiter/bus concepts is also plotted versus sample weight as the independent parameter.

The departure weight versus sample weight format provides for direct matching of launch vehicle capabilities to mission requirements. The comparative evaluation of mission/system alternatives based on performance considerations is relatively straightforward using the data as presented.

6.3 PARAMETRIC RESULTS FOR ALL-CHEMICAL MISSION/SYSTEM CONCEPTS

6.3.1 Single Launch Concepts

Figure 6-1 shows the gross Earth departure weight required for single launch systems based on Conjunction Class missions, the direct reentry/recovery mode, and 1971 propulsion technology. Curves for the three Mars entry modes are shown. It is seen that the INT-20 vehicle can perform the mission using the entry-out-of-elliptical orbit mode. The spacecraft departure weight for a 10-pound sample return is approximately 19,700 pounds. The figure shows more than a 4000-pound performance margin for this mission.

Figure 6-2 gives similar data based on 1974 technology propulsion systems. The curves are generally shifted down by roughly 2000 pounds, and because here we are considering the 1977 and 1979 Mars opportunities, the performance capabilities of the launch vehicles are increased due to the lower departure energy requirements. The Shuttle/Centaur is seen to be capable of performing the single launch mission using a direct entry at Mars while the INT-20 vehicle has capability for an entry-out-of-circular-orbit at Mars. The gross Earth departure weight of the spacecraft for a 10-pound sample using the direct entry mode at Mars is only around 14,400 pounds. The departure weight penalty for the out-of-elliptical-orbit entry mode is seen to be about 2600 pounds for a 10-pound sample return.

Figure 6-3 gives Earth departure weight requirements for single launch concepts employing the orbit capture recovery mode at Earth. The Earth capture impulse requirement for the spacecraft is based on use of the Apollo CSM for the recovery vehicle. It is seen that the INT-20/Centaur vehicle can perform this mission using either the out-of-elliptical-orbit or direct entry mode at Mars. The INT-20 vehicle without Centaur does not have adequate capability for this particular mission approach. The circle symbol in the figure indicates a representative design point which will be discussed in Section VIII.

Similar data for the above concept based on 1974 propulsion technology is shown in Figure 6-4. Because of the lower departure energy requirements for the later launch opportunities (1977, 1979), the INT-20 vehicle can perform the mission if direct entry or possible entry-out-of elliptical-orbit at Mars is employed.

The impact on Earth departure weight of variation in the lander science and supporting systems is shown in Figure 6-5. The baseline lander science weight is 100 pounds. A total weight variation from -100 to +200 pounds is shown in the Figure. The total variation in departure weight over this range is approximately 800 pounds.

Similar data are given in Figures 6-6 and 6-7 for variations in Mars rover weight and orbiter systems weight (excluding propulsion), respectively. The impact of rover (and supporting systems) weight on Earth departure weight is essentially the same as the lander science and supporting systems. In Figure 6-7, it is seen that the orbiter/bus module weight (weight in Mars orbit) can increase from the baseline of approximately 1550 pounds to around 2400 pounds and still be within the INT-20/Centaur capability.

Figure 6-8 shows the results of an analysis of the 1975 inbound Venus swingby mission using the single launch concept with direct Earth reentry/recovery. Curves are shown for both Earth storable and space storable Mars braking stage propulsion. The gross Earth departure weight requirements indicate that the INT-20/Centaur vehicle is marginally capable of this mission if space storable propulsion is used in the Mars braking stage. The weight penalty imposed by the use of Earth storables exceeds the capability of the INT-20/Centaur vehicle by several thousand pounds. The use of space storable stages for Mars braking and escape appears to be an area for further study because of the potential for reducing the MSSR total mission duration by approximately 400 days employing the Venus swingby mode.

6.3.2 Earth Orbit Rendezvous Concepts

The next few figures summarize the results of parametric study of the Earth orbit rendezvous departure mode concepts. Figure 6-9 shows the gross Centaur earth surface launch weight as a function of gross MSSR spacecraft weight for the 1975 and 1977 conjunction class missions. The maximum launch weight based on the Centaur stage propellant capacity is approximately 35,700 pounds. This limits the gross spacecraft weight for the Earth orbit rendezvous mission to approximately 15,500 pounds for the 1975 conjunction class mission or 16,800 pounds for the 1977 mission. The gross spacecraft weight includes the necessary Earth orbit rendezvous equipment.

Figure 6-10 shows the Earth surface launch weight of the MSSR spacecraft as a function of the gross spacecraft weight less Earth orbit rendezvous equipment. Two curves are shown. The top curve gives the launch weight of the spacecraft, and the bottom curve shows the gross weight of the spacecraft at time of

Centaur-stage ignition for the trans-Mars injection maneuver.

Figure 6-11 gives the Earth surface launch weight of the spacecraft as a function of sample weight for the three potential entry modes at Mars. These data are based on the direct Earth reentry/recovery mode and 1971 propulsion technology for the spacecraft. The Titan IIIC vehicle is seen to have more than sufficient capability to launch the spacecraft into low Earth orbit (185 km circular) for either the out-of-elliptical-orbit or direct entry modes at Mars.

The Earth surface launch weight for the Centaur stage as a function of sample weight and Mars entry mode of the spacecraft is shown in Figure 6-12. The Titan IIID/Centaur vehicle is seen not to have sufficient performance capability even for the direct Mars entry mode. Further it is seen that the present Centaur propellant capacity of 30,000 pounds limits performance capability below the required launch weight for the Mars direct entry mode. The Space Shuttle vehicle capability to launch the loaded Centaur to orbit, as indicated in the Figure, would be more than adequate if the Centaur propellant capacity were increased. The shuttle vehicle, however, would not be available until the 1979 Mars opportunity.

Figures 6-13 and 6-14 present similar data for the Earth orbit rendezvous concept based on 1974 spacecraft propulsion technology. Figure 6-13 shows that the Titan IIIC vehicle again has more than adequate performance for launch of the spacecraft to orbit. Because of the reduced Earth departure energy requirements and the improved propulsion performance characteristics, the Titan IIID/Centaur vehicle becomes adequate for launch of the Centaur injection stage into low Earth orbit. As Figure 6-14 indicates, the launch could be made within the Centaur propellant capacity. It is seen that the mission must be based on the direct Mars entry mode.

6.3.3 Dual Interplanetary Launch Concepts

Figure 6-15 shows the gross Earth departure weight required for the lander/ascent probe payload as a function of sample weight for the 1971 spacecraft propulsion technology base. The top curve shows the gross departure weight requirement and the bottom curve gives the corresponding probe diameter based on preliminary sizing analysis. The departure weight required for a 10-pound sample return is approximately 8500 pounds. This is seen to be well within the capability of the Titan IIID/Centaur vehicle and provides an 800-pound performance margin. The probe diameter is slightly over 14 feet. This is near the maximum diameter (14 feet) of the Titan/Centaur bulbous Viking shroud. The circle symbol in the figure indicates the preliminary design point discussed in Section VIII.

Figure 6-16 presents similar data based on 1974 propulsion technology for the spacecraft. The gross Earth departure weight requirement is reduced to approximately 7500 pounds for a 10-pound sample return. The probe diameter is about 13.2 feet. Because of the reduced Earth departure energy for the 1977 and 1979 missions, the Titan IIID/Centaur performance capability is increased to approximately 10,000 pounds. Thus, a performance margin of about 2,500 pounds exists for a 10-pound sample return mission.

Figures 6-17 and 6-18 present the gross Earth departure weight requirements for the orbiter/bus return vehicle payload based on the direct Earth reentry/recovery mode. The departure weight required for a 10-pound sample return is approximately 9000 pounds based on 1971 propulsion technology as shown in Figure 6-17. This leaves a 300-pound performance margin for the Titan IIID/Centaur vehicle which has a departure weight capability of 9300 pounds. The circle symbol in the figure indicates the preliminary design point discussed in Section VIII. Figure 6-18 shows that the performance margin is increased to 2500 pounds for the later missions using 1974 spacecraft propulsion technology.

Figures 6-19 and 6-20 present the results of an analysis of the Earth capture intercept/recovery mode. Figure 6-19 is for the 1971 technology base and covers the 1975 mission opportunity. Three curves are shown. The top curve shows the departure weight required if the orbiter/bus vehicle is maneuvered into an elliptical orbit at Earth return to allow sample retrieval by an Apollo CSM vehicle. The departure weight required for a 10-pound sample mission is 22,000 pounds and is in the INT-20 class performance capability range. The middle curve assumes that a sub-module of the return orbiter/bus is separated and propulsively captures into the elliptical orbit for sample retrieval by the CSM. A considerable departure weight savings is achieved with this system approach at the expense of some added spacecraft complexity. The departure weight required for the 10-pound sample mission becomes approximately 11,700 pounds. This is well beyond the Titan IIID/Centaur vehicle capability of 9300 pounds. Although the Titan IIID(7)/Centaur vehicle was not considered officially in the present study groundrules, it is of interest to show the vehicle capability in the figure. It is seen that this vehicle, if developed, could potentially perform a minimum sample return mission using the sub-module system approach for earth capture/recovery. The direct reentry/recovery curve is repeated in Figure 6-19 for comparison to the capture recovery mode requirements.

Data similar to the above are presented in Figure 6-20 for the 1974 technology base and 1977, 1979 mission cases. The Titan IIID/Centaur becomes capable of potentially providing Earth capture/recovery capability using the sub-module capture approach. The Titan IIID(7)/Centaur would have roughly a 1500-pound performance margin for this mission mode as seen in the figure.

It is of interest to determine the gross Earth departure weight increase required to allow Mars entry-out-of-elliptical-orbit in the dual departure mission concept. Figure 6-21 shows the departure weight requirement for this entry mode as a function of sample return weight using direct reentry/recovery and 1971 spacecraft propulsion technology. Curves for both Earth storable and space storable Mars braking stages are shown. The performance capability of the Titan IIID/Centaur vehicle is exceeded in all cases. As a matter of interest, the Titan IIID(7)/Centaur vehicle capability is indicated in the figure. This vehicle is seen to have adequate capability to permit consideration of the out-of-elliptical-orbit entry mode. Figure 6-22 gives similar data for the 1977 and 1979 missions using 1974 spacecraft propulsion technology. The reduced gross Earth departure weight requirement is low enough to consider use of the Titan IIID/Centaur vehicle if space storable propulsion is used for the Mars braking stage.

The next three figures present the results of parametric analyses to determine the sensitivity of gross Earth departure weight to variations in lander science weight, Mars surface rover weight, and in the case of the orbiter/bus, variations in orbiter systems weight. Figure 6-23 shows the impact of variation in the lander science and supporting systems weight on Earth departure weight. Curves are shown for variations of -100 and +200 pounds around the baseline system based on the current Viking science weight allocation of 100 pounds. For a 10-pound sample return mission an increase over the baseline weight of an additional 100 pounds results in an increase in Earth departure weight of approximately 340 pounds. The departure weight requirement is still within the Titan IIID/Centaur capability of 9300 pounds.

In Figure 6-24 similar data are shown for a variation in Mars rover weight from 0 to 400 pounds. For a 10-pound sample return mission, the performance capability exists to carry a 400-pound rover within the capability of the Titan IIID/Centaur vehicle. The baseline rover weight for the parametric performance analyses is 150 pounds.

Figure 6-25 presents performance sensitivity data for variations in the total orbiter/bus systems module weight. The baseline system weighs 1545 pounds. Variations are shown in this weight from 1400 to 2200 pounds. As seen in the figure, weight growth in the orbiter subsystems is critical because of the impact on gross departure weight. For example, a weight growth of roughly 200 pounds over the baseline would exceed the performance capability of the Titan IIID/Centaur vehicle for a 10-pound sample return mission. Thus, considerable emphasis in a Phase A study of the all-chemical dual departure concept should be placed on analysis of the orbiter subsystem weights.

A more critical weight is the weight of the orbiter/bus return vehicle subsystems module after the excess equipment (not needed for return) is jettisoned in Mars orbit. (See Section VIII, pages 8-15 and 8-16.) Figure 6-26 shows the sensitivity of earth departure weight to a variation in the return orbiter/bus system module weight from 600 to 900 pounds. The baseline weight in the present study is 730 pounds (excluding the Earth reentry capsule) as indicated in the figure. Roughly a 10-percent growth in the return module weight over the baseline of 730 pounds would increase the departure weight requirement to exceed the Titan IIID/Centaur capability. The baseline weight of 730 pounds includes contingencies; however, Figure 6-26 points out the criticality and need of a very close analysis of the return orbiter/bus module weight in a Phase A study of the dual departure concept.

6.3.4 Summary of Parametric Performance Results for All-Chemical Concepts

The results of the parametric performance analysis of all-chemical mission/system alternative approaches may be summarized as follows based on a requirement of returning a minimum sample weight of 10 pounds.

6.3.4.1 Single Launch Concept

- The INT-20 vehicle can perform a conjunction class, direct reentry/recovery mission employing Mars entry out of orbit and 1971 spacecraft propulsion technology.
- The INT-20 can marginally perform a conjunction class, orbital capture/recovery mission using 1974 technology and employing capture of the complete Earth return spacecraft. An approach based on capture of only a sub-module of the return spacecraft could be used to increase the performance margin.
- The INT-20/Centaur vehicle can perform a conjunction class, orbital capture/recovery mission with Mars entry out of elliptical orbit using 1971 technology.
- A 1975 Venus swingby, direct reentry/recovery mission appears to be possible using the INT-20/Centaur vehicle if space storable propulsion is used in the Mars braking stage.

6.3.4.2 Dual Departure Concept

- It appears that two Titan IIID/Centaur vehicles can perform a conjunction class, direct reentry/recovery mission employing direct Mars entry and 1971 technology.
- The lander/ascent probe diameter for a 10-pound sample return is near the current Viking bulbous shroud diameter of 14 feet. (It is believed that sizing can be reduced by a detailed further analysis of the entry/descent/landing system for direct entry using lift.)
- It appears possible to achieve the Earth capture/recovery mode within the capability of a Titan IIID(7)/Centaur using 1971 technology only if a sub-module of the return spacecraft is captured into Earth orbit.
- It appears possible to achieve a capture/recovery mission in the late 1970's (1977, 1979 missions) using the Titan IIID/Centaur and the sub-module earth capture approach.

6.3.4.3 Earth Orbit Rendezvous Concept

- The conjunction class, direct reentry/recovery mission with direct Mars entry appears to be marginally possible using Titan vehicles and a Centaur transplanetary injection stage.
- The Space Shuttle, after it becomes available (1979 mission), could be used to transport either or both the spacecraft and loaded Centaur injection stage to Earth Orbit.

6.4 PARAMETRIC RESULTS FOR SOLAR ELECTRIC/CHEMICAL CONCEPTS

The following three subsections present the results of parametric performance analyses of the solar-electric/chemical mission approaches of interest. Recall that the performance differences between the all-chemical and solar electric/chemical systems are reflected through the orbiter/bus vehicle only. The lander/ascent vehicles are essentially identical between corresponding approaches.

Primary interest in the present study was in the near-term (1971) solar electric technology base reflected principally by a 58 lb/kw (26.3 kg/kw) effective powerplant specific weight and 306-second specific impulse earth storable chemical propulsion systems. The parametric data for these systems will be presented and discussed. Similar data for a projected late 1970's technology base will be included for reference at the end of the section. These data are based on a powerplant specific weight of 49 lb/kw (22.2 kg/kw) and 320-second specific impulse chemical propulsion systems for major planetocentric maneuvers.

6.4.1 Single Launch Concepts

Figures 6-27 and 6-28 present gross Earth departure weight requirements for a single launch concept using the direct Mars entry mode. Figure 6-27 is for the direct reentry/recovery mode at Earth return. The Titan IIID/Centaur vehicle capability is seen not to be adequate for this mission. The gross departure weight required for a 10-pound sample return ranges from 15,200 pounds to about 16,100 pounds depending on the Mars escape mode employed. Data are shown for both escape modes. It is indicated in the figure that the Titan III(7)/Centaur vehicle, if developed, could marginally perform the mission using the SEP spiral-out escape mode.

Figure 6-28 presents similar data for the Earth orbital capture/recovery mode. The departure weight requirement for a 10-pound sample return ranges from 18,300 pounds to approximately 22,600 pounds depending on the Mars escape mode. Among the launch vehicles under consideration, the INT-20 would be required for Earth launch.

The power required at 1 AU for the direct reentry/recovery mission concepts is given in Figure 6-29. Similar data are given in Figure 6-30 for the orbital capture/recovery mode. The power level required for the mission ranges from 24.7 kw up to 36.6 kw depending on mission mode combination (10-pound sample).

Figure 6-31 gives gross departure weight requirements for single launch missions employing entry-out-of-elliptical-orbit at Mars. Data are shown for the two Earth intercept/recovery modes and the two Mars escape modes. The gross departure weight for a 10-pound sample return ranges from 16,700 pounds to 24,600 pounds depending on mission mode combination. Again, the INT-20 vehicle capability would be required for Earth departure. The power requirement at 1 AU for these mission concepts is presented in Figure 6-32. For a 10-pound sample return, the requirement is seen to range from 26.9 kilowatts to 39.4 kilowatts depending on mission mode combination.

Data analogous to that presented in Figures 6-31 and 6-32 are given in Figures 6-33 and 6-34 for single launch missions using Mars entry out of circular orbit. The departure weight requirement for a 10-pound sample return is seen to vary from approximately 23,400 pounds to 31,700 pounds and is within the INT-20 vehicle capability. The corresponding power levels required at 1 AU range from 37.6 kilowatts to 50.7 kilowatts depending on mission mode combination.

6.4.2 Earth Orbit Rendezvous Concept

Parametric data for the solar-electric/chemical Earth orbit rendezvous mission concept are given in Figure 6-35. The curves are based on Earth orbital capture/recovery, direct Mars entry, and the spiral-out Mars escape mode. The results show that the Titan IIID vehicle has more than adequate capability for launch of the MSSR spacecraft into low Earth orbit. The top curve shows the Earth surface launch weight of the Centaur orbital injection stage. It is seen that the Titan IIID/Centaur vehicle has adequate capability for launch of the stage to low Earth orbit. The dashed horizontal line in the figure indicates the Centaur propellant capacity as it would limit the size spacecraft that could be injected onto the transplanetary escape trajectory.

6.4.3 Multiple Interplanetary Launch Concepts

Figures 6-36 and 6-37 present gross Earth departure weight requirements for the dual-launch solar-electric/chemical orbiter/bus system. Figure 6-36 is for the direct reentry/recovery mode at Earth. Curves are shown for both the SEP spiral-out and chemical Mars escape modes. The departure weight required for a 10-pound sample return ranges from approximately 5600 pounds to 7600 pounds depending on Mars escape mode. This requirement exceeds the capability of the Titan IIIC vehicle, but is well within the Titan IIID/Centaur capability. Similar data for the Earth orbital capture/recovery mode are shown in Figure 6-37. The departure weight required for a 10-pound sample return ranges from approximately 9300 pounds to 14000 pounds depending on Mars escape mode. The Titan IIID/Centaur vehicle is seen to be more than adequate for the mission if the SEP spiral-out escape mode is used at Mars. The launch vehicle performance margin for a 10-pound sample return using the Titan IIID/Centaur vehicle is approximately 2400 pounds. The circle symbol in the figure indicates the preliminary design point discussed in Section VIII.

It is of interest to consider the effect of solar array specific weight on Earth departure weight requirements. Figure 6-38 presents data for the Earth orbital capture/recovery mission using spiral-out Mars escape, showing the impact of a variation in solar array specific weight from 33 pounds per kilowatt to 50 pounds per kilowatt. The 33 pounds per kilowatt figure is characteristic of currently predicted specific weights of roll-out arrays under development. The 50-pound per kilowatt figure is representative of predicted specific weights of the fold-out type of array currently under development. The increase in Earth departure weight of the spacecraft going from the 33 pound per kilowatt to the 50 pound per kilowatt arrays is roughly 1200 pounds. The departure weight is still well within the Titan IIID/Centaur vehicle capability.

Figure 6-39 shows the power required at 1 AU for the dual launch solar-electric orbiter/bus as a function of the Earth intercept/recovery and Mars escape mode combination. The power at 1 AU for a 10-pound sample return mission is seen to range from 9.2 kw to 22.6 kw depending on mission mode combination. The circle symbol in the figure indicates the preliminary design point discussed in Section VIII.

A parametric analysis was performed to determine the effect of Mars orbit altitude on gross earth departure weight and power required for the orbiter/bus vehicle. Figures 6-40 and 6-41 present the results. The effect on earth departure weight is shown in Figure 6-40. For a given sample return weight, the slopes of the curves indicate roughly a 120-pound increase in earth departure weight per 100-km decrease in Mars orbit altitude. Figure 6-41 shows the corresponding impact of orbit altitude on the orbiter/bus power required at 1 AU. The slopes of the curves indicate an increase in power requirement of the order of 160 watts per 100-km decrease in Mars orbit altitude. As a matter of interest, the impact of orbit altitude on time required for spiral-out escape is relatively small. For a given sample return weight, the variation in escape time was found to be of the order of a week over an altitude range from 600 to 1200 kilometers. The escape time for the baseline case for a 1000-km orbit and 10-pound sample return is approximately 148 days.

6.4.4 Summary of Parametric Results for Solar/Electric Concepts

The results of the parametric performance analysis of solar-electric/chemical mission system alternatives are summarized as follows based on a requirement of returning a minimum sample weight of 10 pounds.

6.4.4.1 Single Launch Concept

- The Titan IIID/Centaur vehicle does not have adequate capability for a single launch solar-electric/chemical MSSR mission.
- Low-energy, orbital capture/recovery missions using direct Mars entry are possible with the INT-20 vehicle capability. A considerable performance margin exists regardless of Mars escape mode.
- The power requirement at 1 AU for the above missions ranges from approximately 31 kw to 37 kw depending on Mars escape mode.
- The above mission with Mars entry out of circular orbit is possible with the INT-20 vehicle regardless of Mars escape mode. The power required at 1 AU ranges from approximately 44 kw to 51 kw for a 10-pound sample return depending on Mars escape mode.

6.4.4.2 Earth Orbit Rendezvous Concept

- Low-energy, orbital capture/recovery missions using direct Mars entry and spiral-out Mars escape are possible using Titan IIID/Centaur for launch-to-orbit of the Centaur injection stage and the Titan IIIC or Titan IIID vehicle for launch of the spacecraft to orbit.

- Approximately 31 kw of power are required at 1 AU for a 10-pound sample return mission.

6.4.4.3 Dual Departure Concept

- A low-energy, orbital capture/recovery mission using direct entry at Mars and spiral-out Mars escape is possible using two Titan IIID/ Centaur vehicles.
- The power requirement at 1 AU is approximately 15.3 kw for a 10-pound sample return system.
- A direct reentry/recovery mission is possible using chemical Mars escape. The power required at 1 AU is approximately 12.5 kw.

6.4.5 Reference Parametric Data for Advanced Technology Base

For reference purposes, Figures 6-42 through 6-49 give representative parametric performance data based on advanced SEP technology projected for potential missions in the late 1970's. The data cases and format are similar to the data which have been presented in the preceding subsections. The curves may be used to roughly estimate the potential decreases in earth departure weight for given sample return requirements.

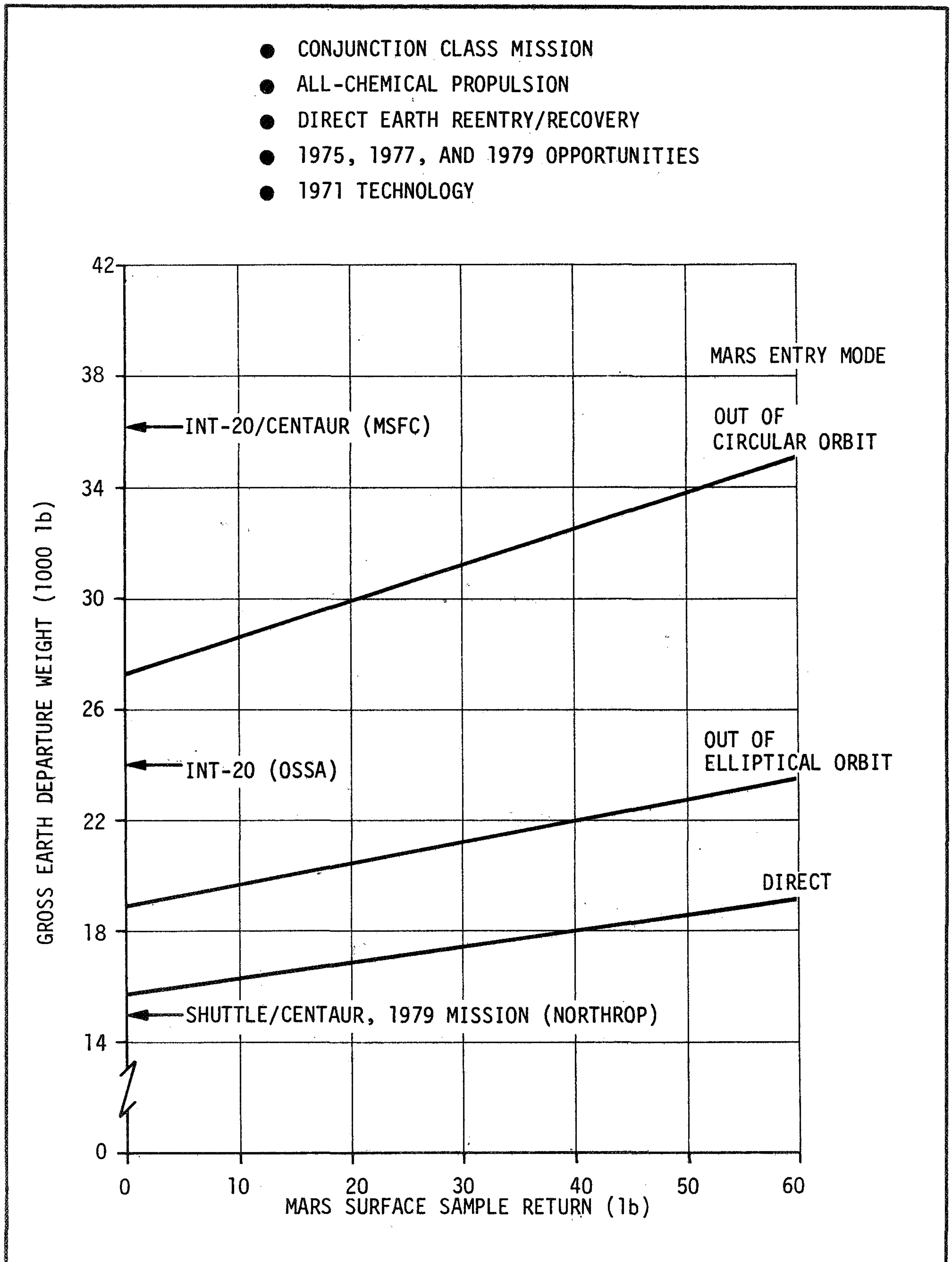


Figure 6-1. SINGLE LAUNCH CONCEPT PERFORMANCE (DIRECT REENTRY)

- CONJUNCTION CLASS MISSION
- ALL-CHEMICAL PROPULSION
- DIRECT EARTH REENTRY/RECOVERY
- 1977 AND 1979 OPPORTUNITIES
- 1974 TECHNOLOGY

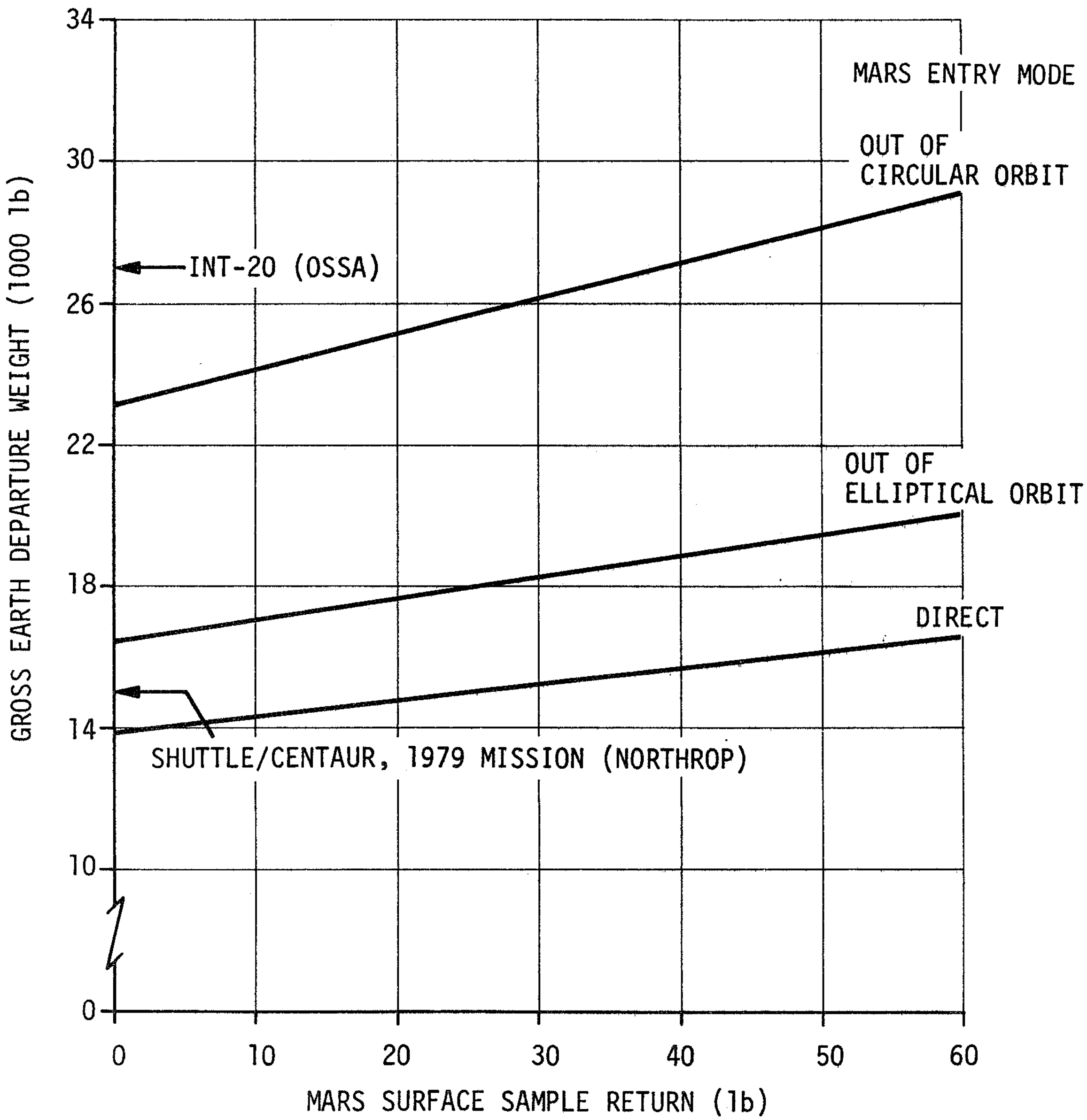


Figure 6-2. SINGLE LAUNCH CONCEPT PERFORMANCE (DIRECT REENTRY; 1974 TECHNOLOGY)

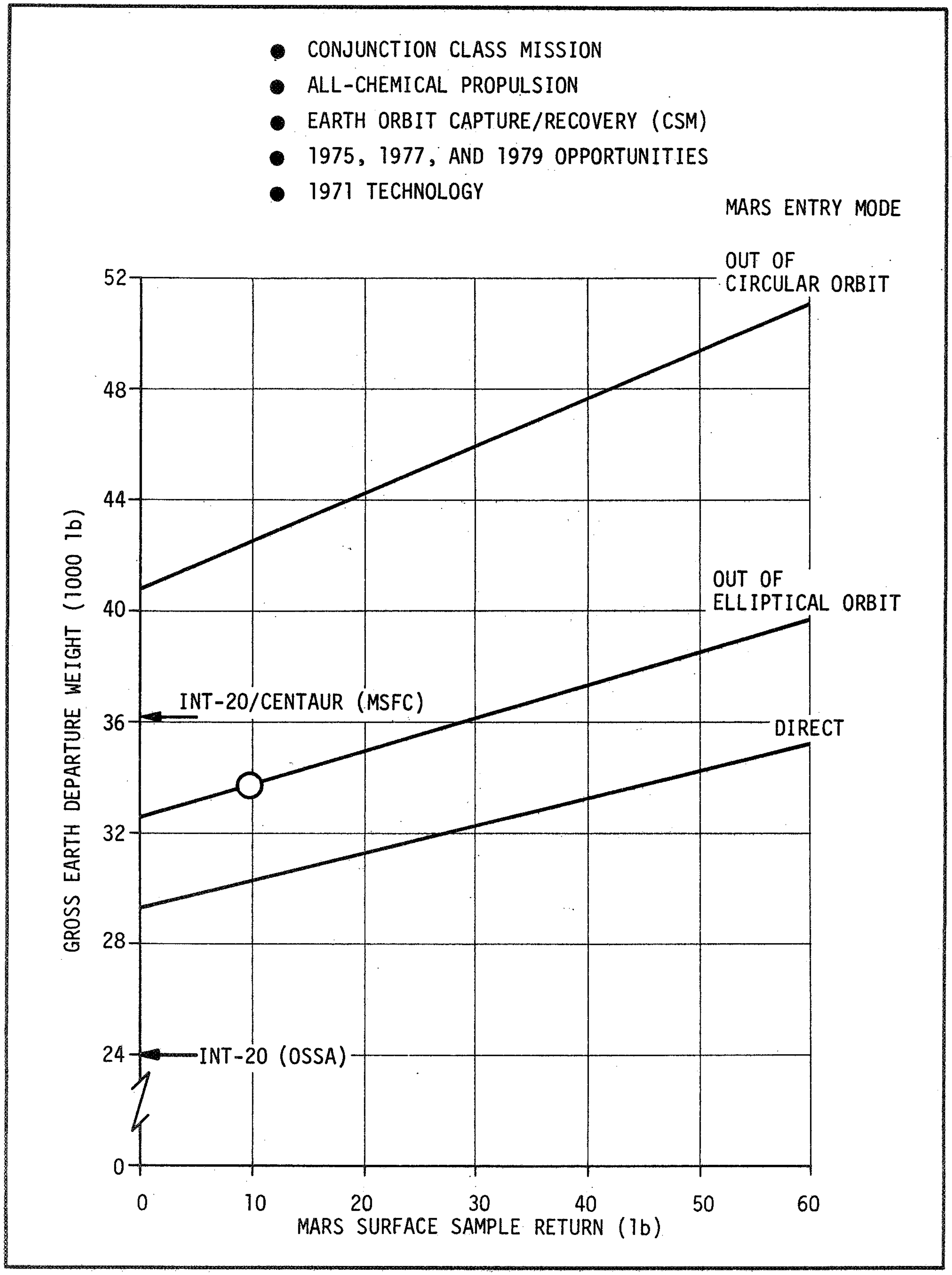


Figure 6-3. SINGLE LAUNCH CONCEPT PERFORMANCE (CAPTURE RECOVERY)

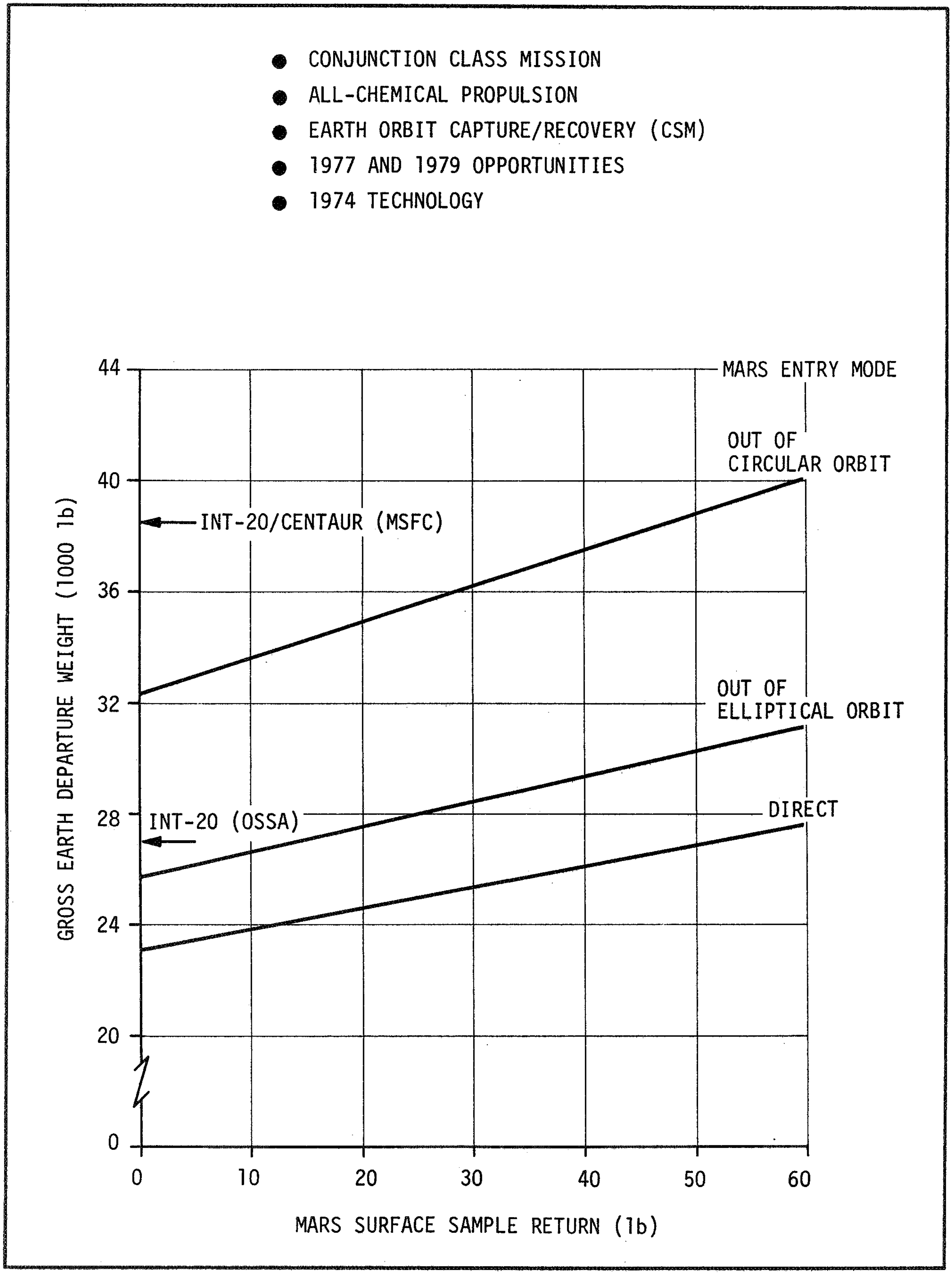


Figure 6-4. SINGLE LAUNCH CONCEPT PERFORMANCE (CAPTURE RECOVERY; 1974 TECHNOLOGY)

- CONJUNCTION CLASS MISSION
- ALL-CHEMICAL PROPULSION
- EARTH ORBIT CAPTURE/RECOVERY
- MARS ENTRY OUT OF ELLIPTICAL ORBIT (CSM)
- 1975, 1977, AND 1979 OPPORTUNITIES
- 1971 TECHNOLOGY

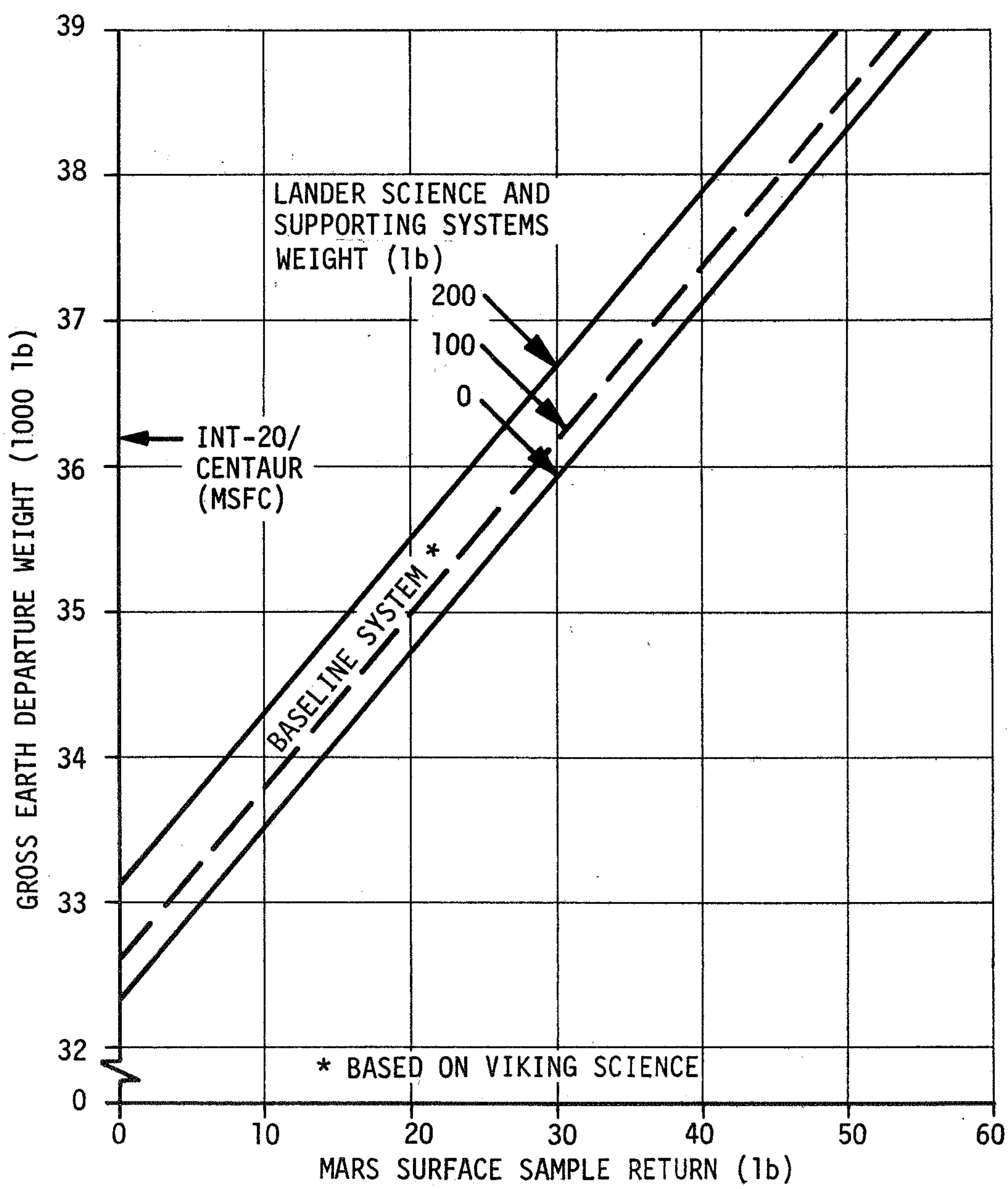


Figure 6-5. SINGLE LAUNCH CONCEPT - EFFECT OF LANDER SCIENCE WEIGHT ON PERFORMANCE

- CONJUNCTION CLASS MISSION
- ALL-CHEMICAL PROPULSION
- EARTH ORBIT CAPTURE/RECOVERY (CSM)
- MARS ENTRY OUT OF ELLIPTICAL ORBIT
- 1975, 1977, AND 1979 OPPORTUNITIES
- 1971 TECHNOLOGY

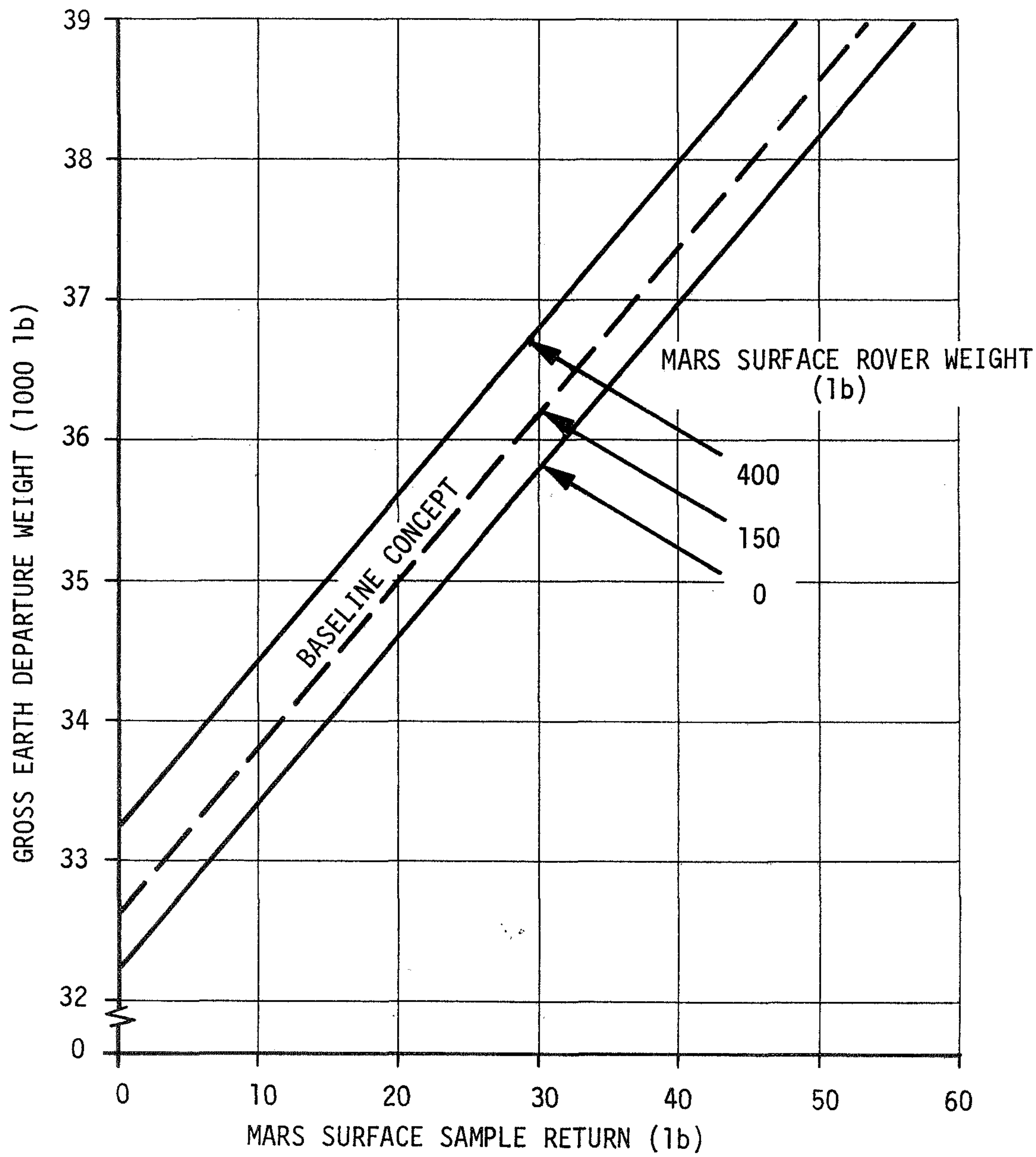


Figure 6-6. SINGLE LAUNCH CONCEPT - EFFECT OF ROVER WEIGHT ON PERFORMANCE

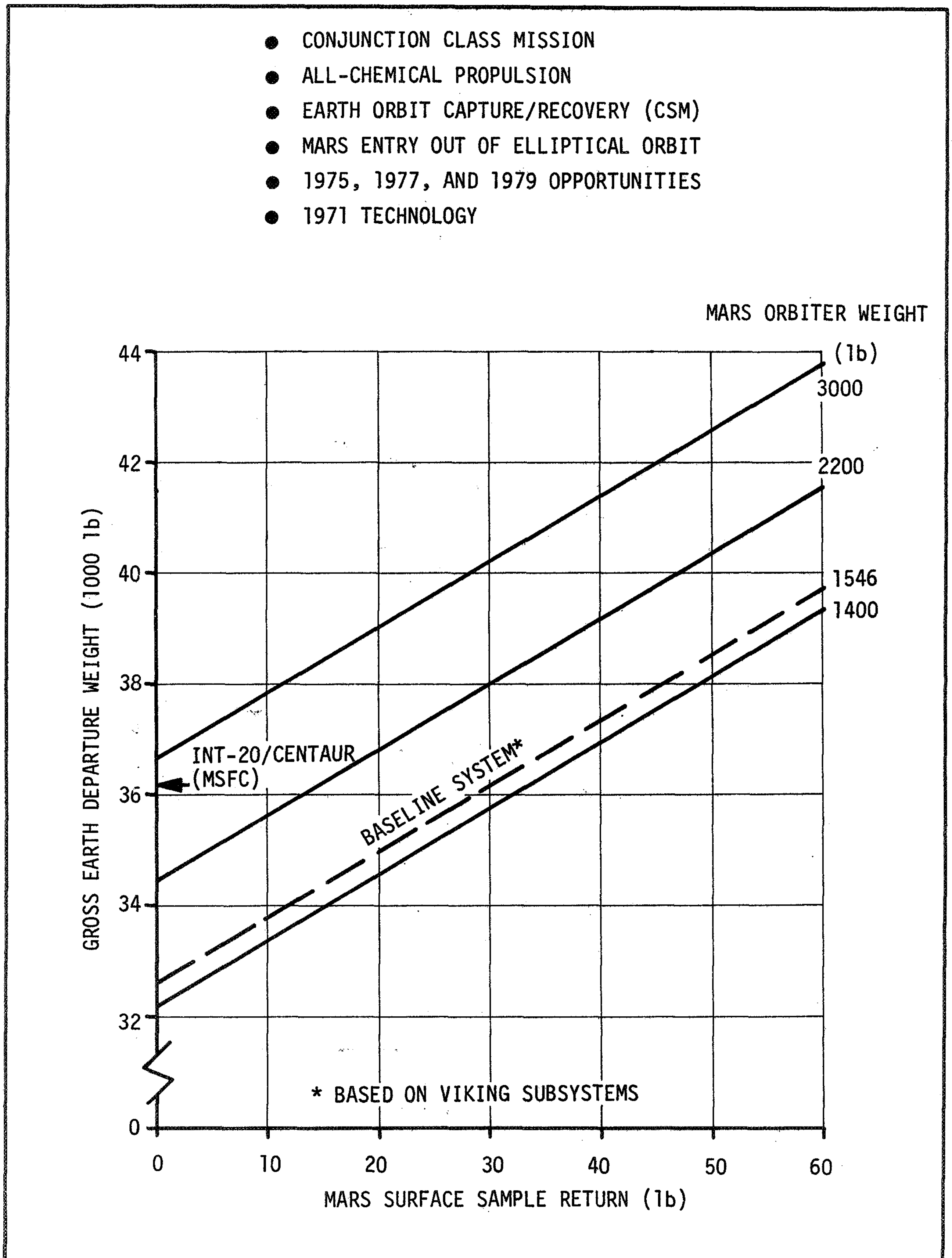


Figure 6-7. SINGLE LAUNCH CONCEPT - EFFECT OF ORBITER EQUIPMENT MODULE WEIGHT ON PERFORMANCE

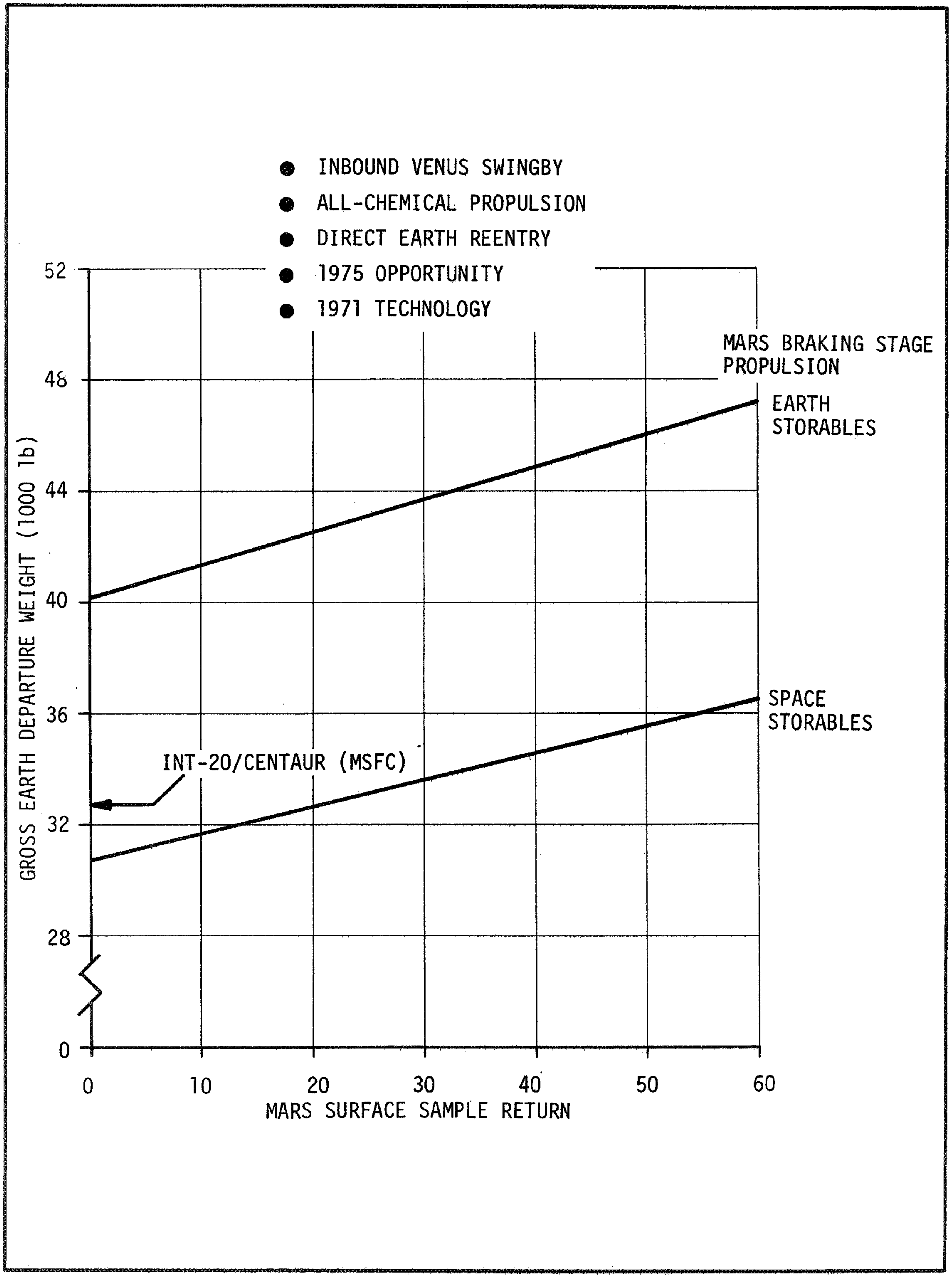


Figure 6-8. SINGLE LAUNCH CONCEPT FOR 1975 INBOUND VENUS SWINGBY

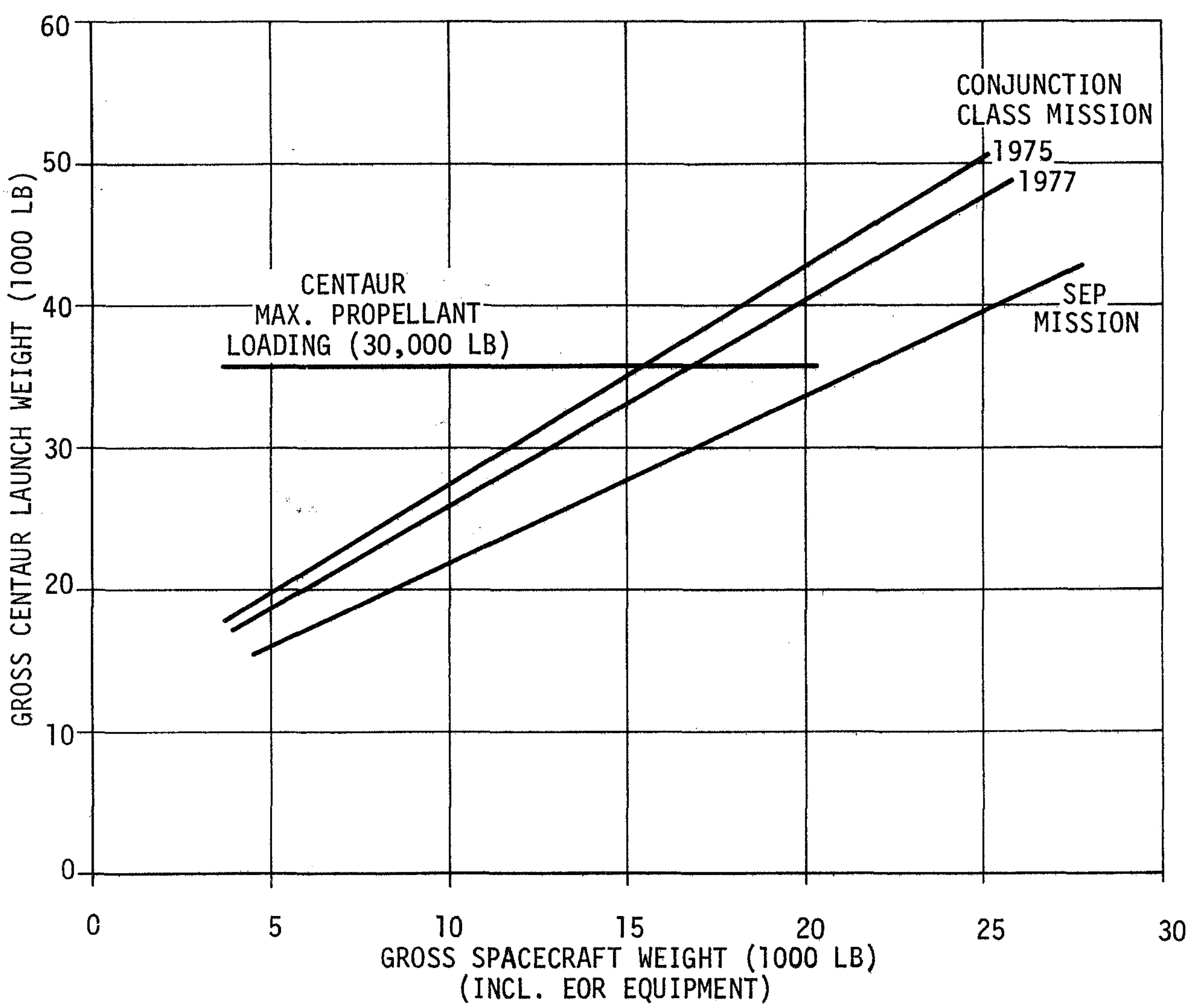


Figure 6-9. EARTH SURFACE LAUNCH WEIGHT OF CENTAUR FOR USE AS TRANS-MARS INJECTION STAGE IN EOR CONCEPT

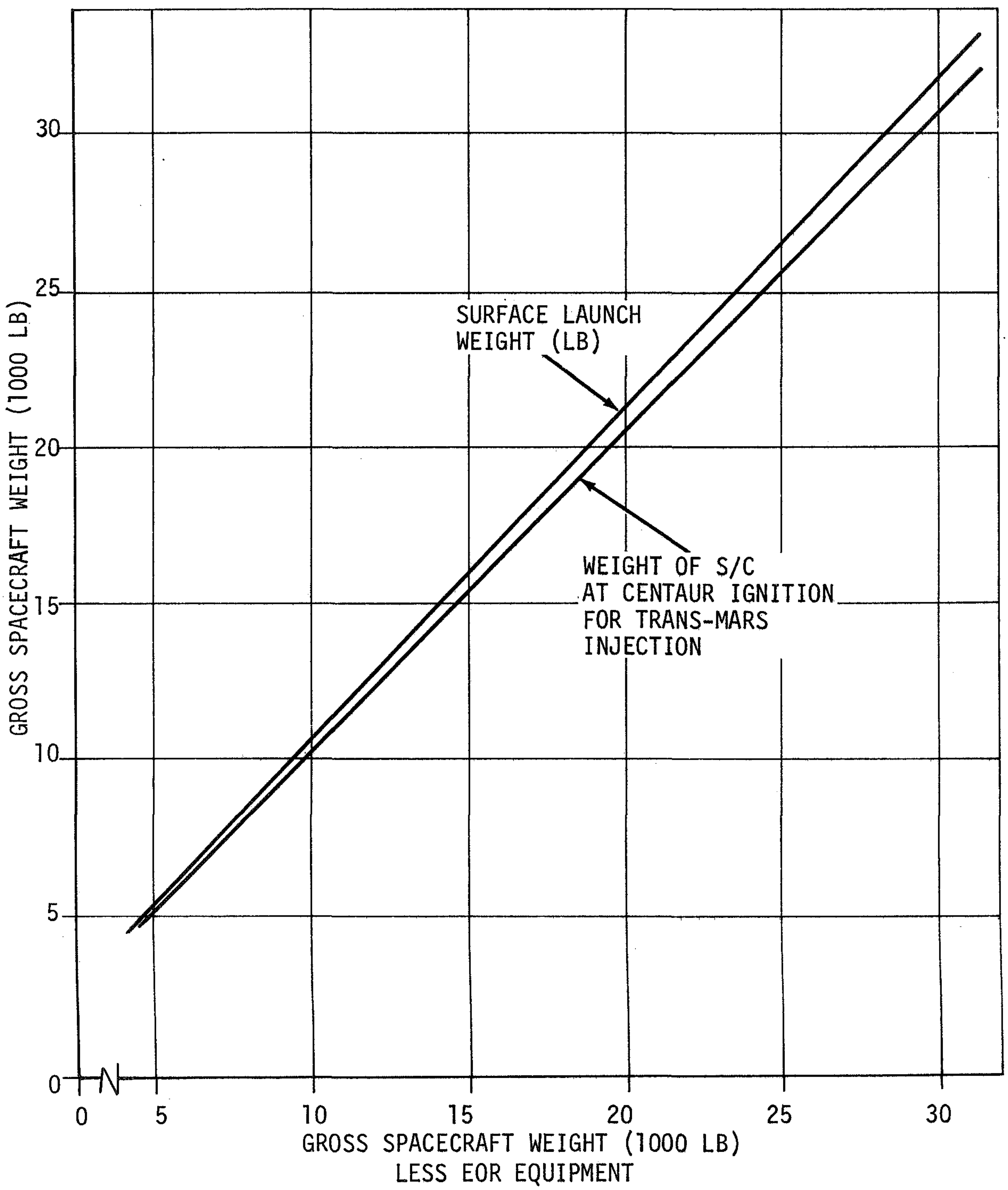


Figure 6-10. EARTH ORBIT RENDEZVOUS MODE - SPACECRAFT SURFACE LAUNCH WEIGHT AS A FUNCTION OF SPACECRAFT WEIGHT LESS EOR EQUIPMENT

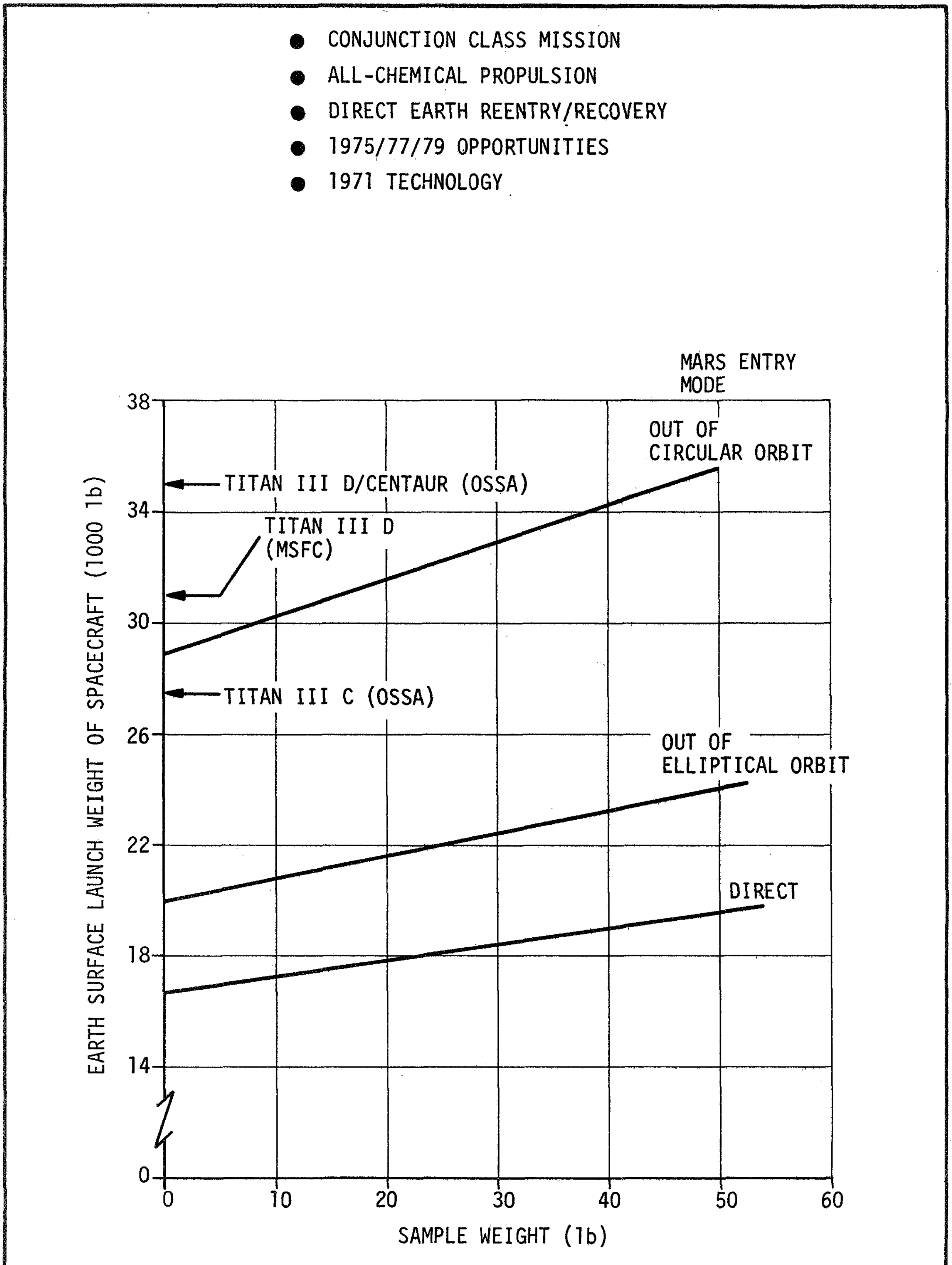


Figure 6-11. EARTH ORBIT RENDEZVOUS CONCEPT - SPACECRAFT LAUNCH TO ORBIT

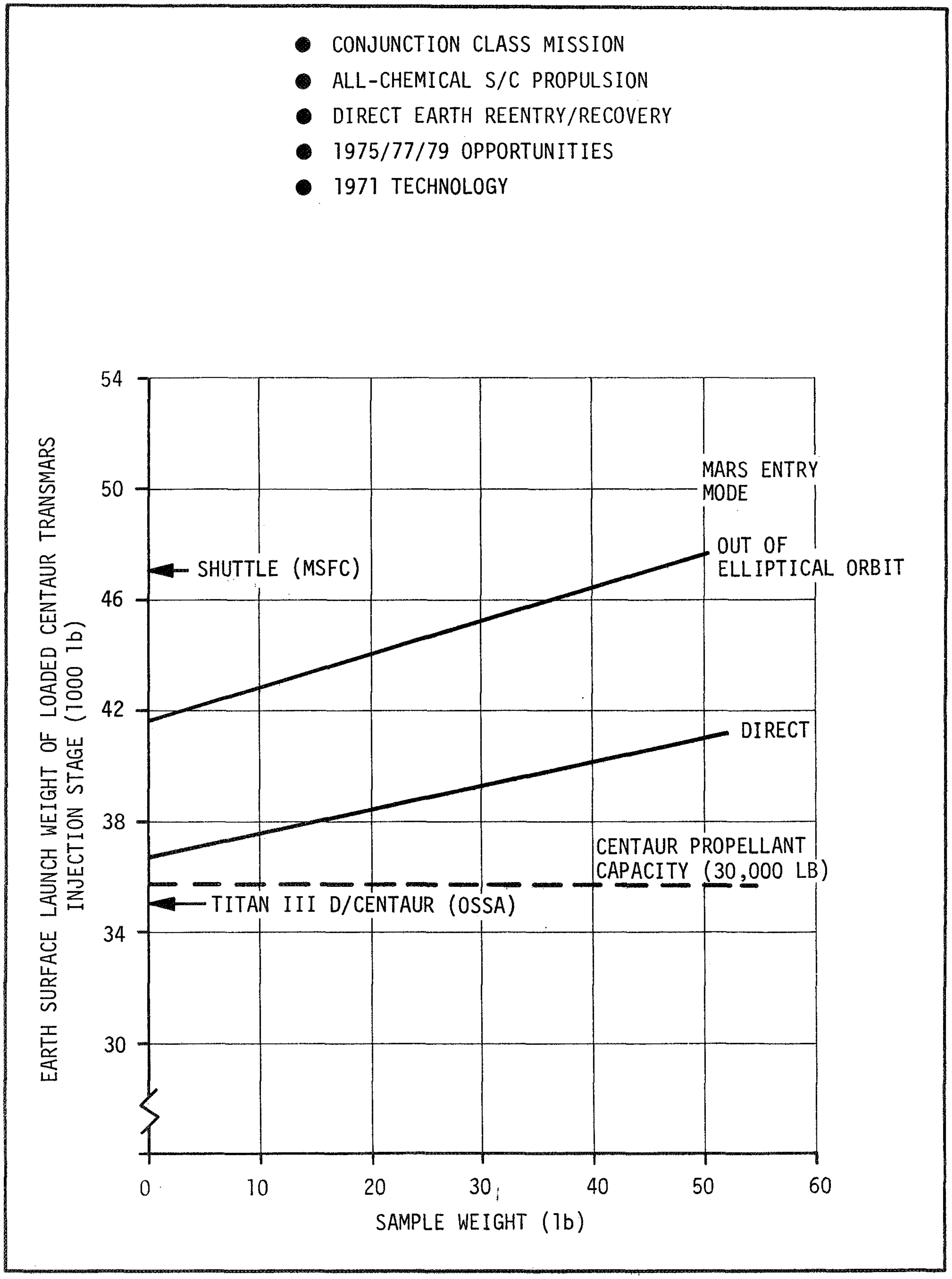


Figure 6-12. EARTH ORBIT RENDEZVOUS CONCEPT - CENTAUR INJECTION STAGE LAUNCH TO ORBIT

- CONJUNCTION CLASS MISSION
- ALL-CHEMICAL PROPULSION
- DIRECT EARTH REENTRY/RECOVERY
- 1977/79 LAUNCH OPPORTUNITIES
- 1974 TECHNOLOGY

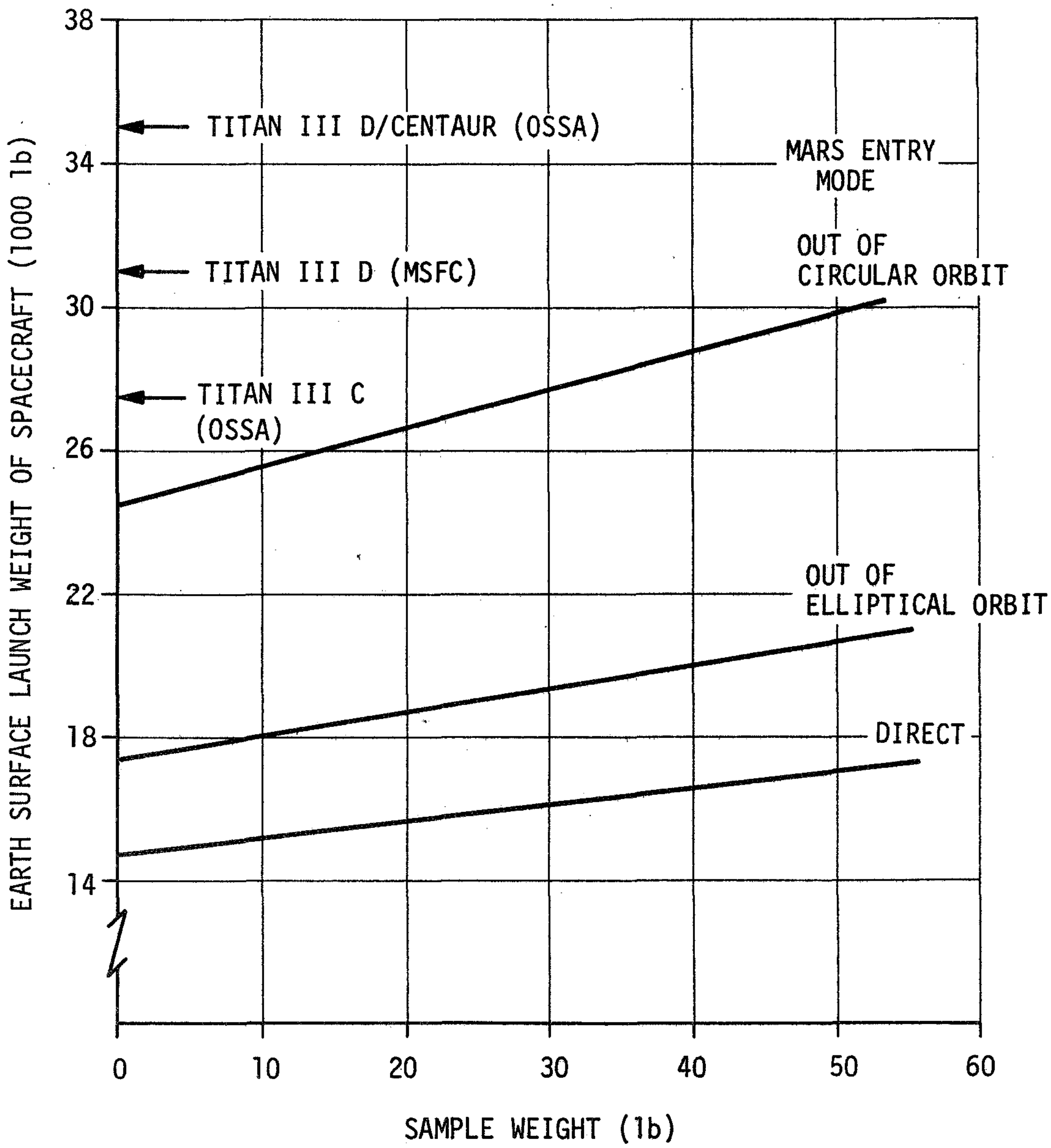


Figure 6-13. EARTH ORBIT RENDEZVOUS CONCEPT - SPACECRAFT LAUNCH TO ORBIT (1974 TECHNOLOGY)

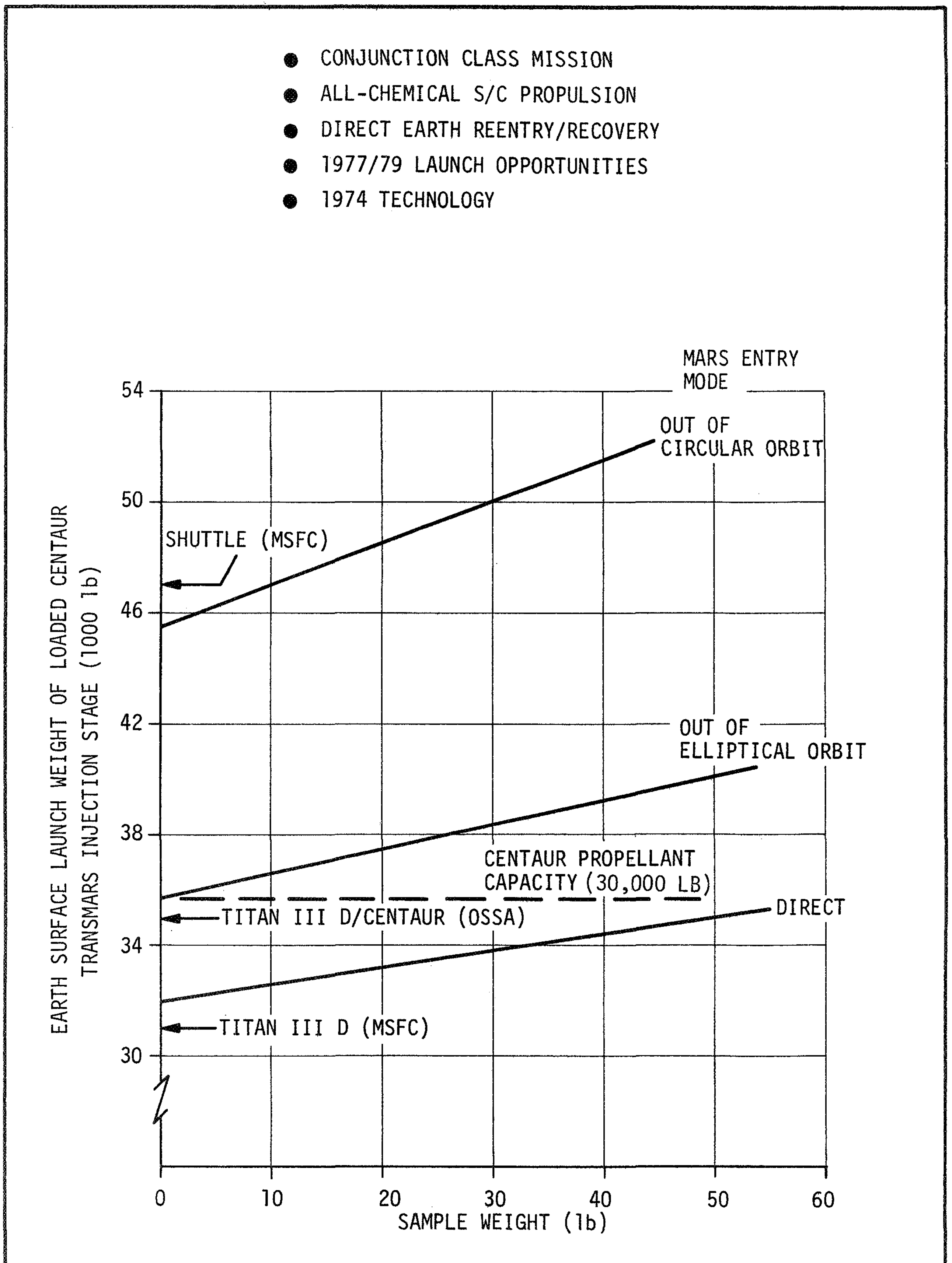


Figure 6-14. EARTH ORBIT RENDEZVOUS CONCEPT - CENTAUR INJECTION STAGE LAUNCH TO ORBIT (1974 TECHNOLOGY)

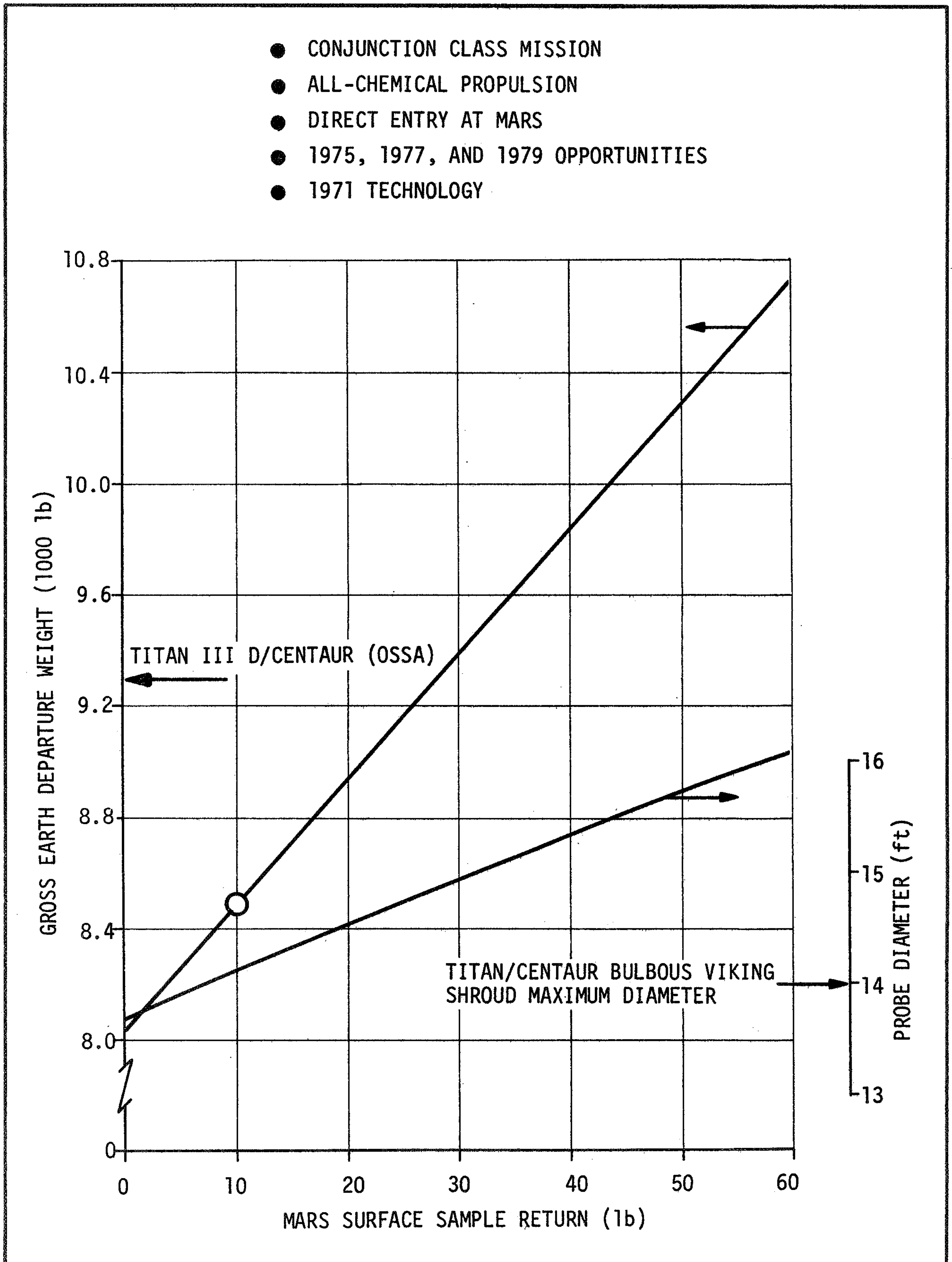


Figure 6-15. DUAL LAUNCH LANDER/RETURN PROBE PERFORMANCE

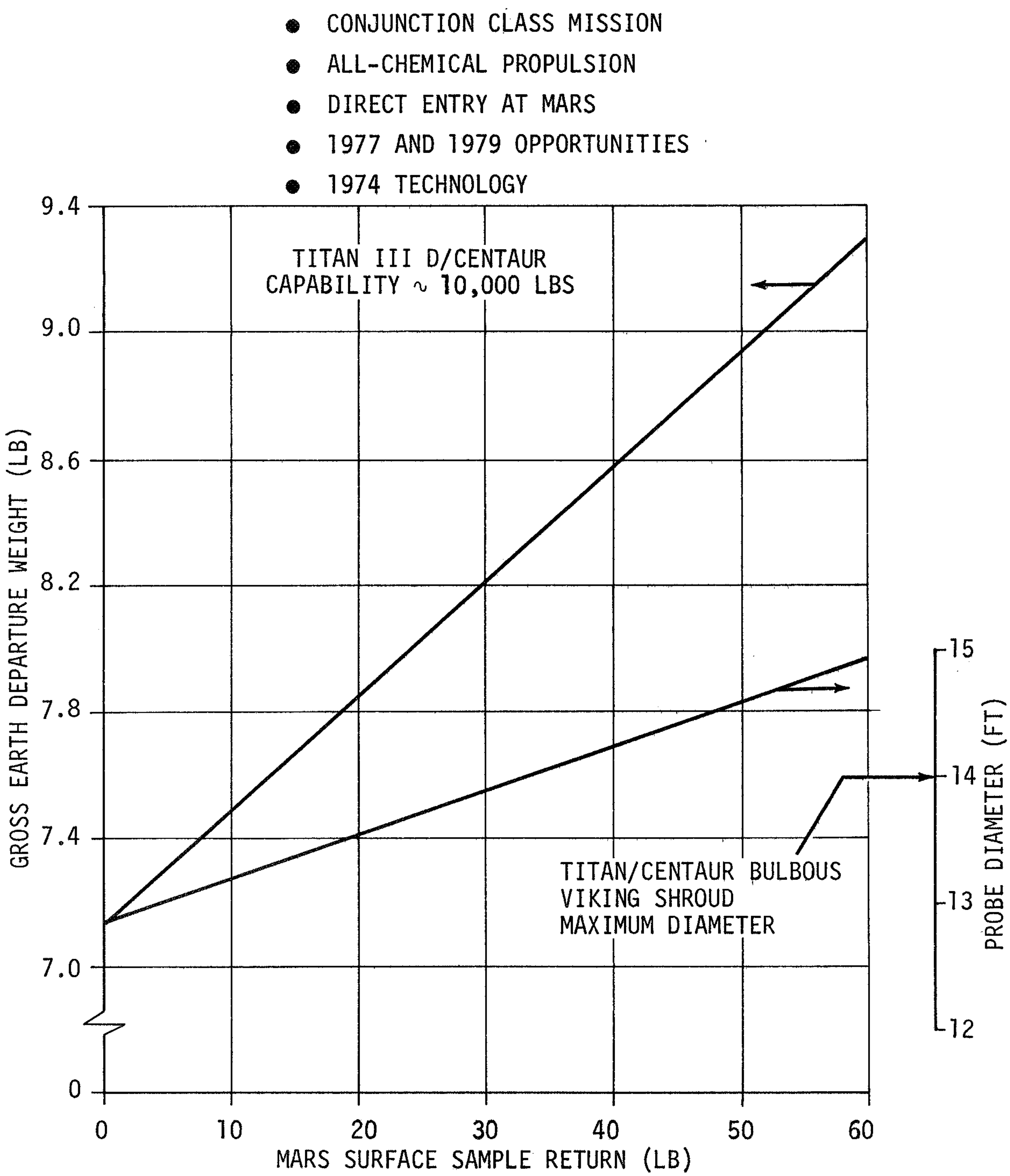


Figure 6-16. DUAL LAUNCH LANDER/RETURN PROBE PERFORMANCE FOR 1974 TECHNOLOGY BASE

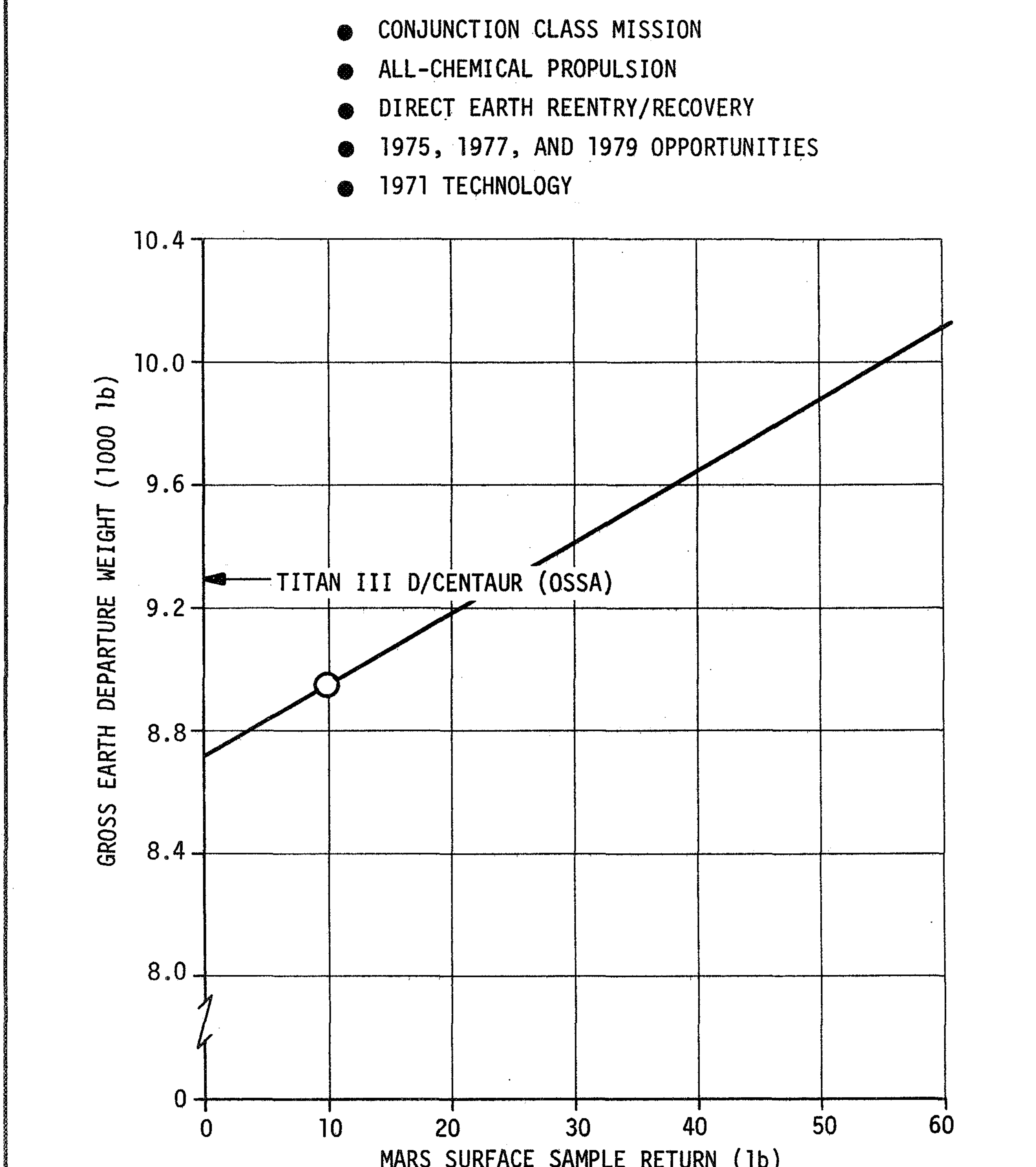


Figure 6-17. DUAL LAUNCH ORBITER/BUS PERFORMANCE (1971 TECHNOLOGY)

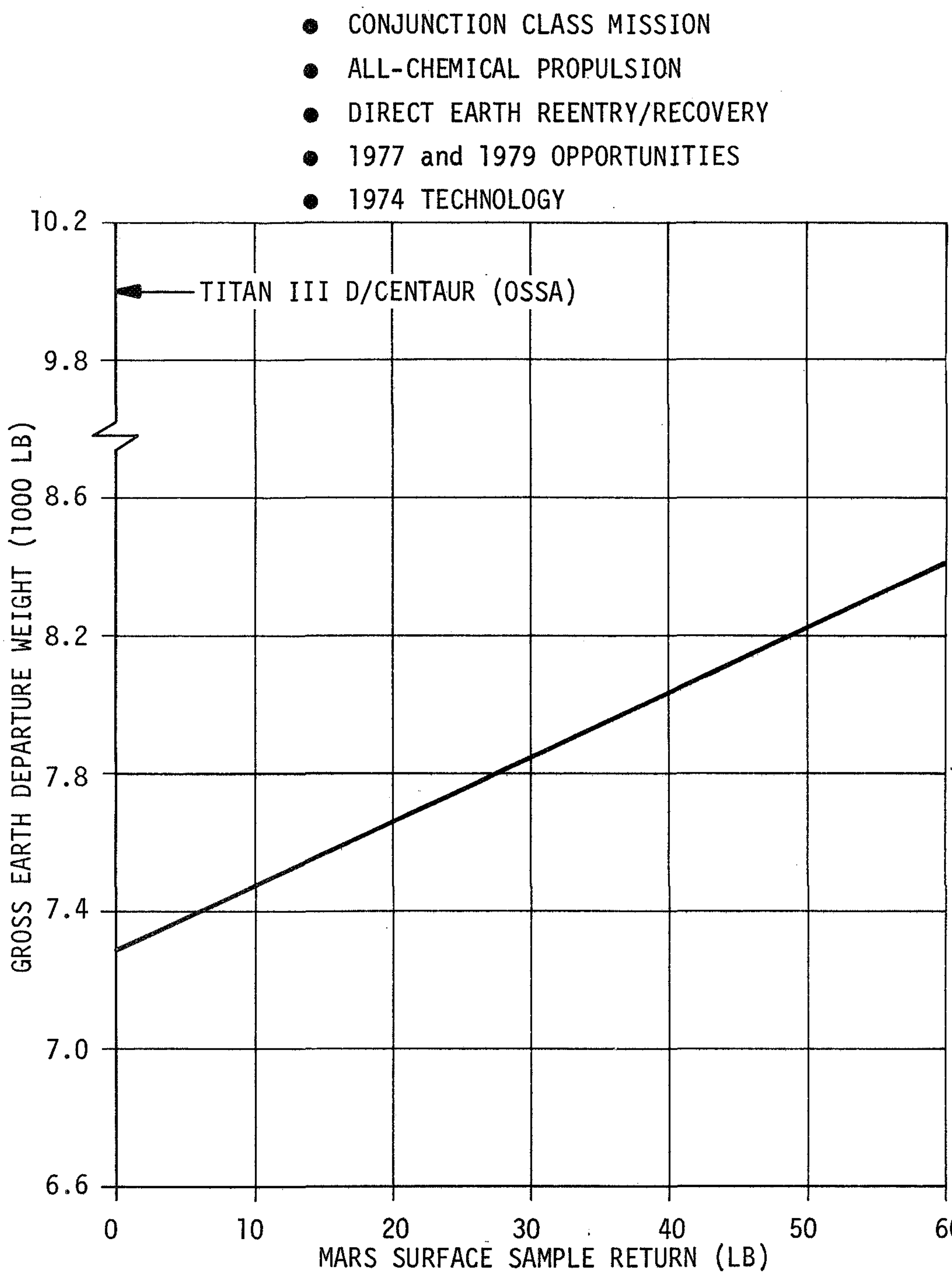


Figure 6-18. DUAL LAUNCH ORBITER/BUS PERFORMANCE (1974 TECHNOLOGY BASE)

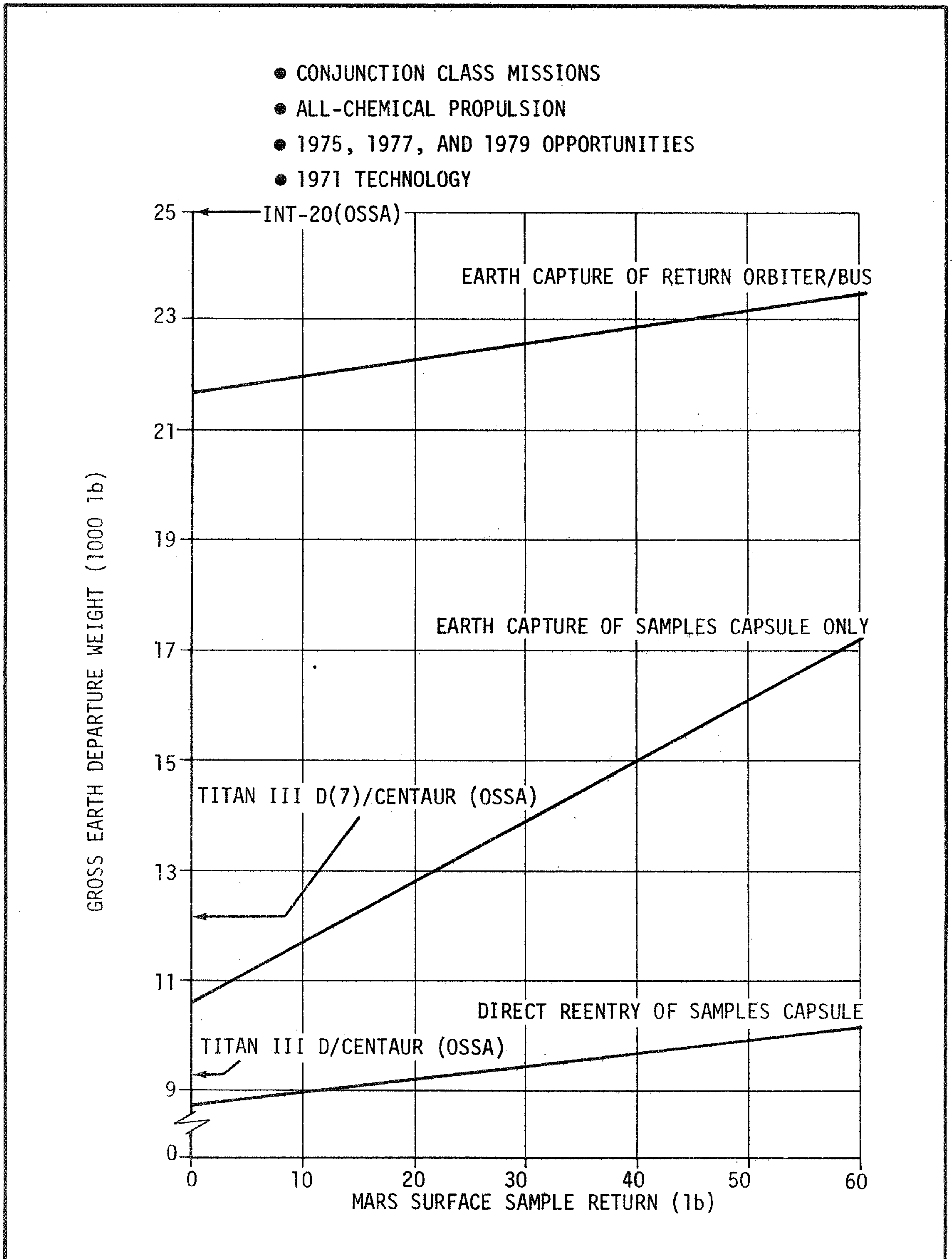


Figure 6-19. DUAL LAUNCH ORBITER/BUS - COMPARISON OF EARTH INTERCEPT/ RECOVERY MODES (1971 TECHNOLOGY BASE)

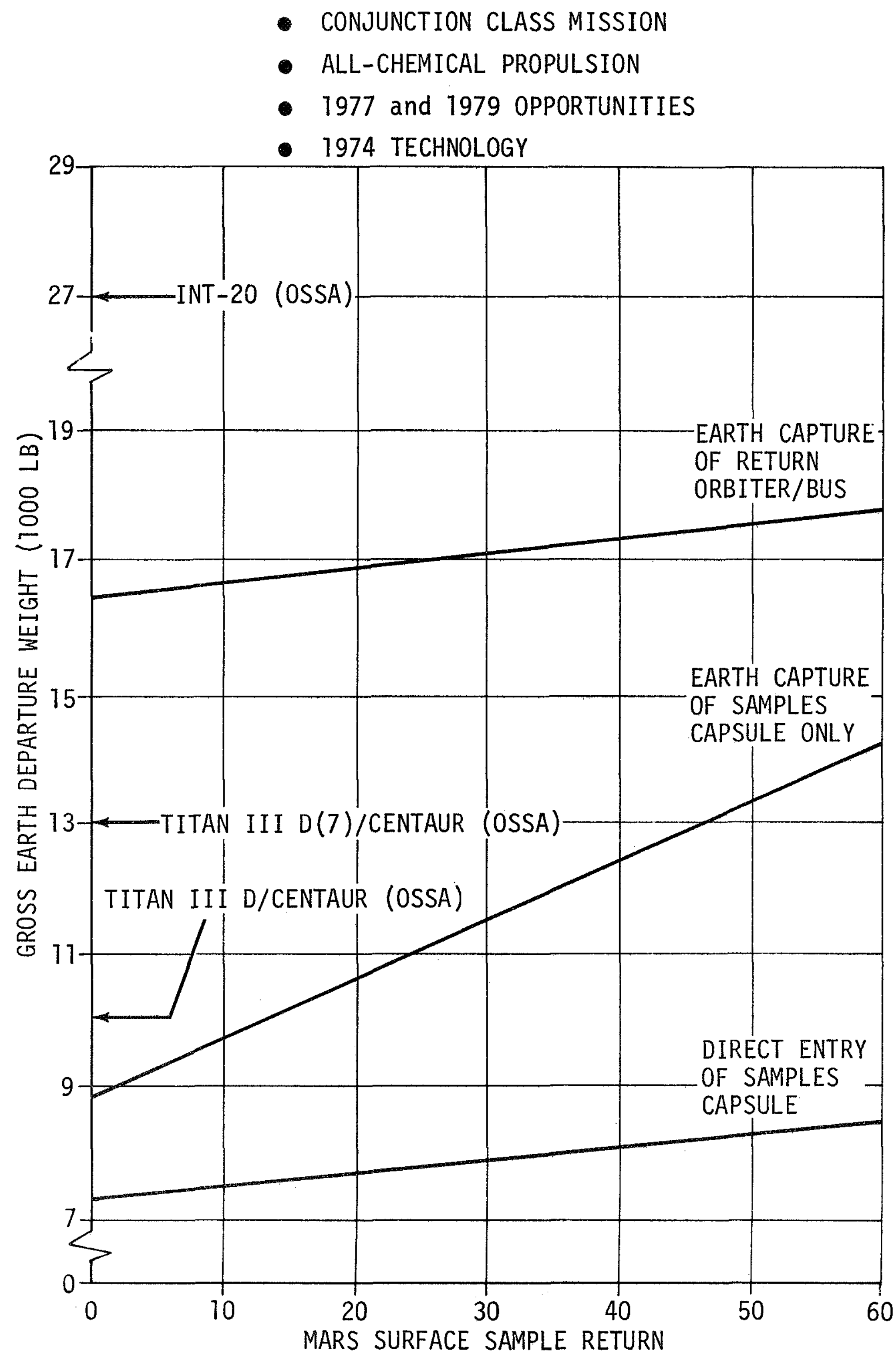


Figure 6-20. DUAL LAUNCH ORBITER/BUS PERFORMANCE - COMPARISON OF EARTH INTERCEPT/RECOVERY MODES (1974 TECHNOLOGY)

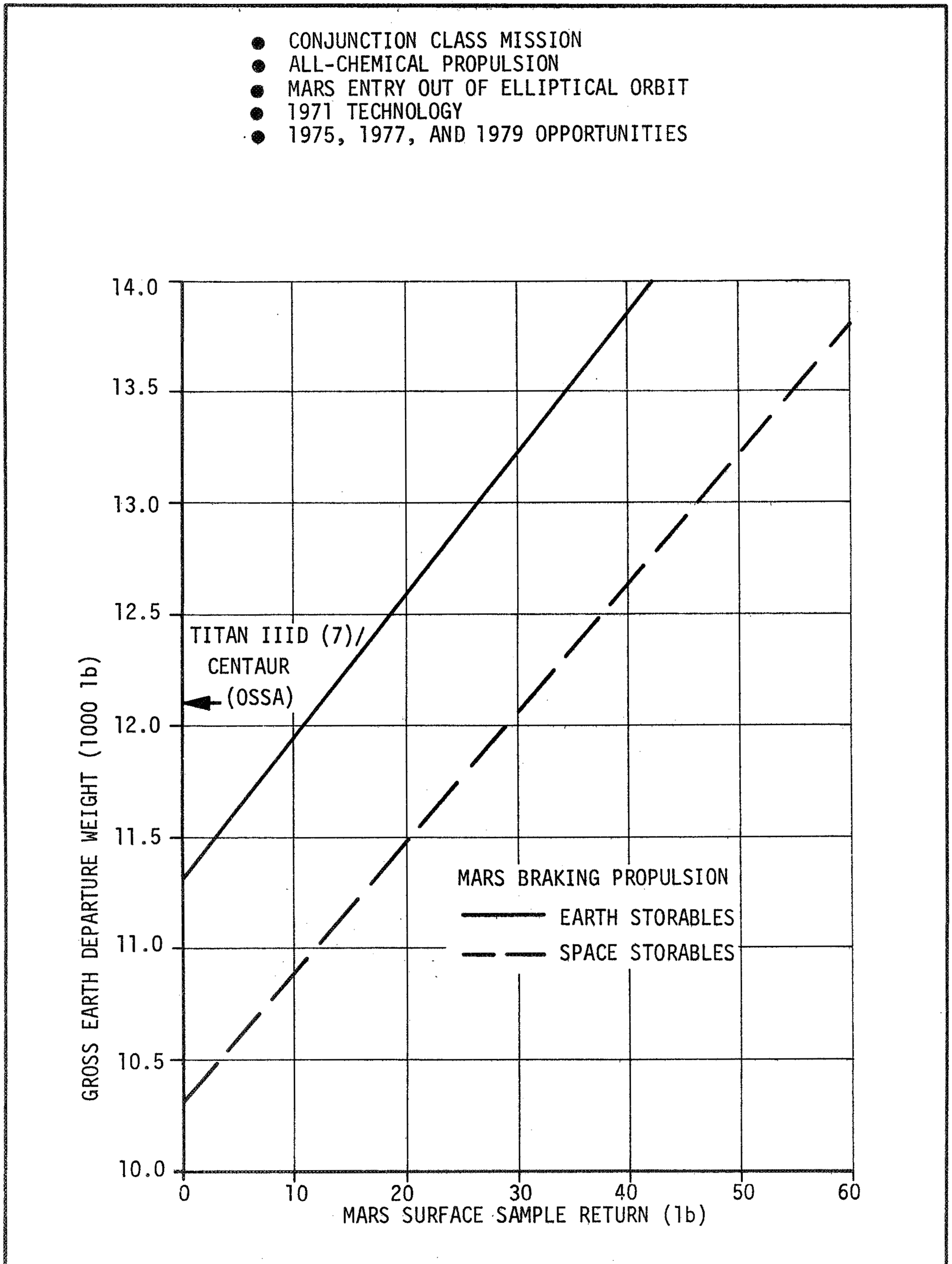


Figure 6-21. DUAL LAUNCH LANDER/RETURN PROBE - PERFORMANCE FOR MARS ENTRY OUT OF ORBIT

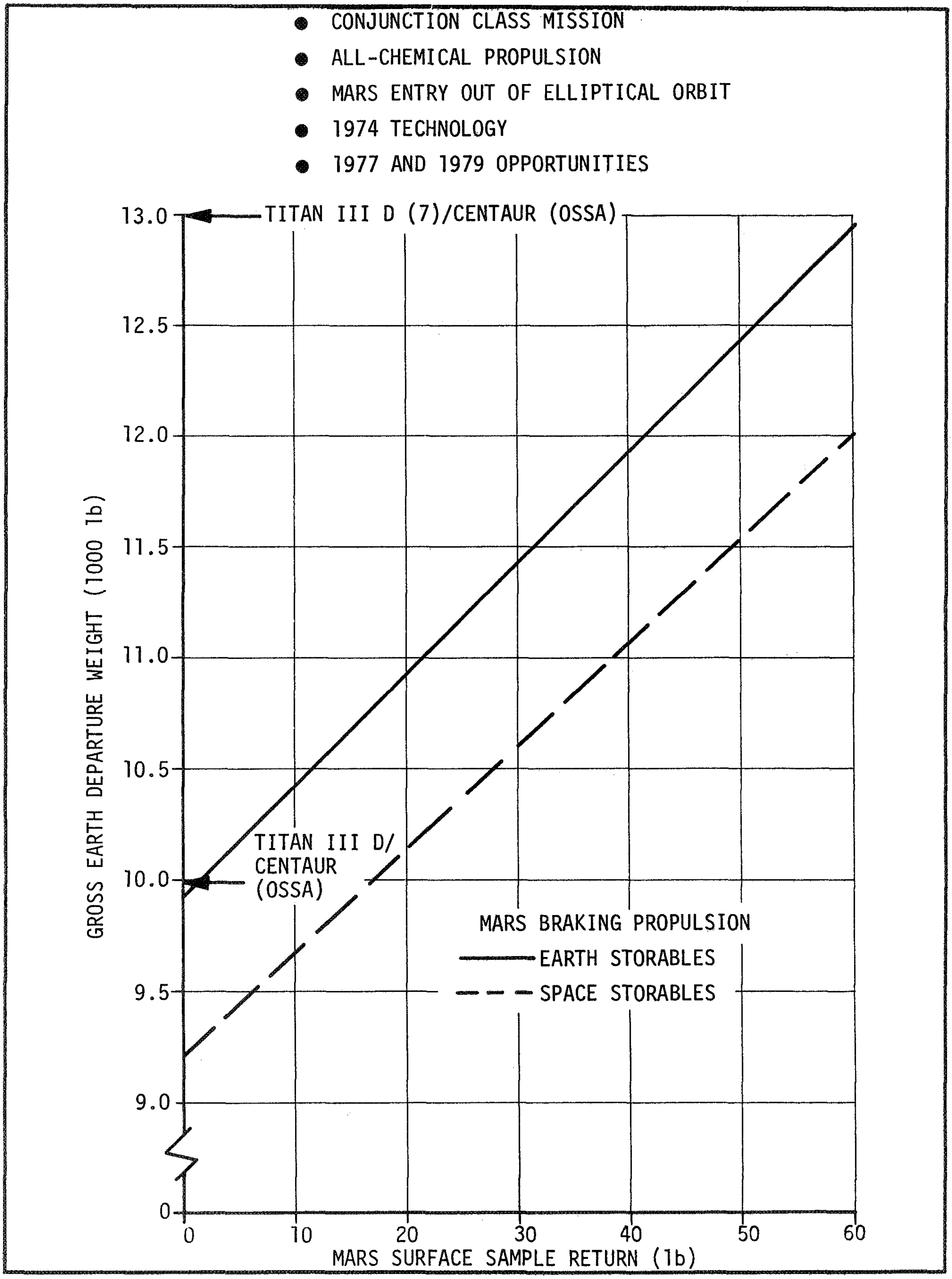


Figure 6-22. DUAL LAUNCH LANDER/RETURN PROBE PERFORMANCE FOR MARS ENTRY OUT OF ORBIT (1974 TECHNOLOGY)

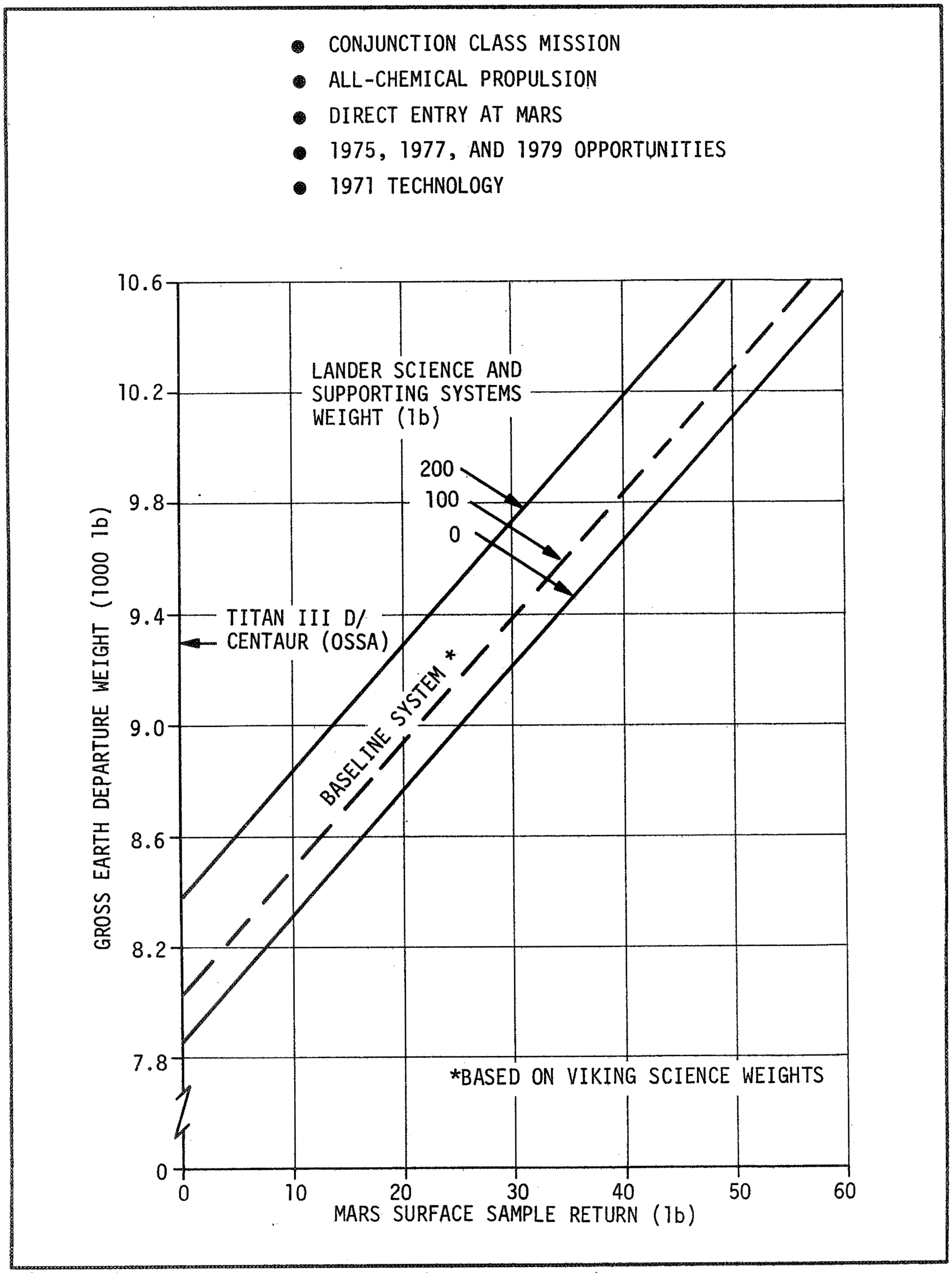


Figure 6-23. DUAL LAUNCH LANDER/RETURN PROBE - EFFECT OF LANDER SCIENCE WEIGHT ON PERFORMANCE

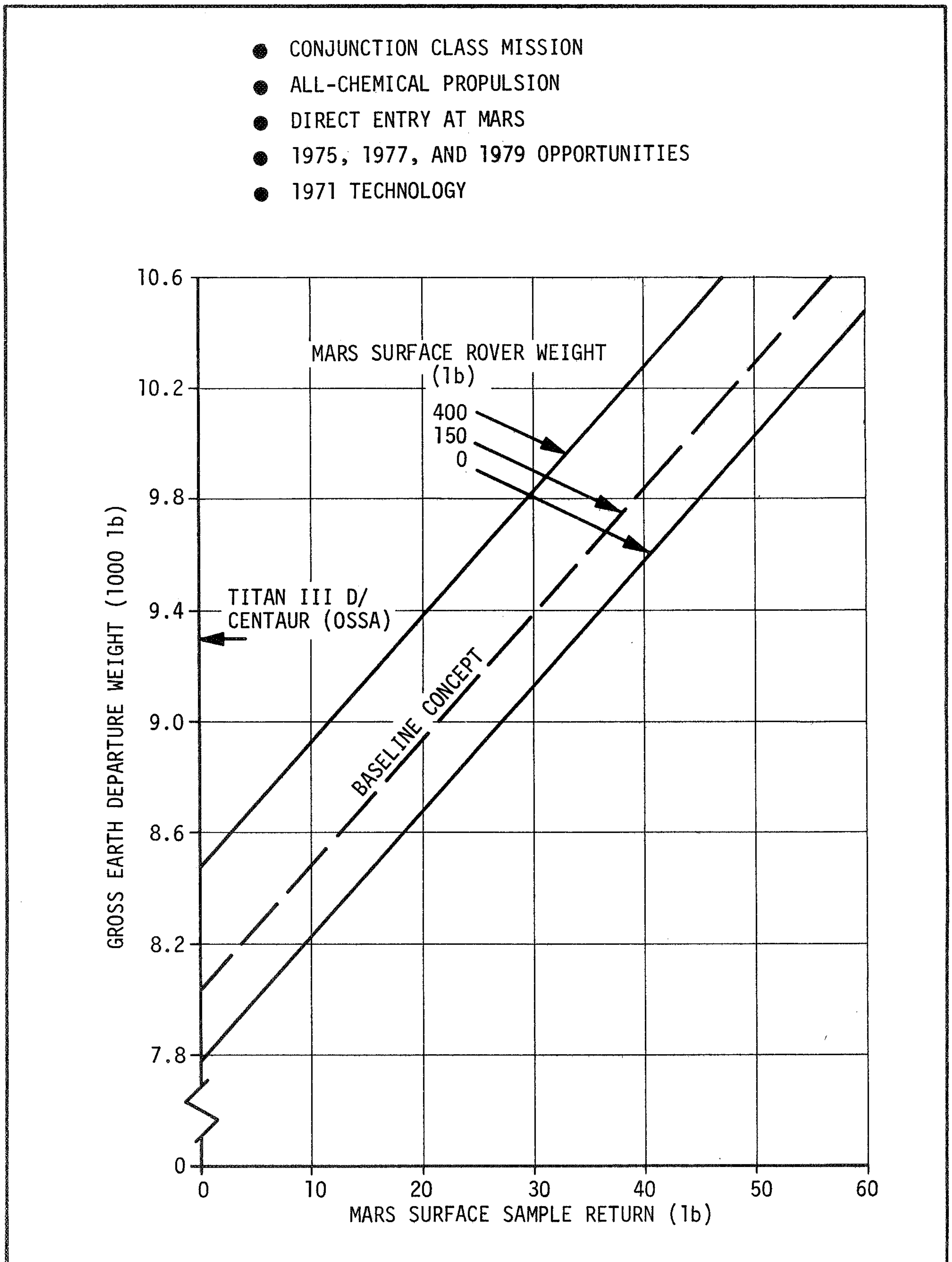


Figure 6-24. DUAL LAUNCH LANDER/RETURN PROBE - EFFECT OF ROVER WEIGHT ON PERFORMANCE

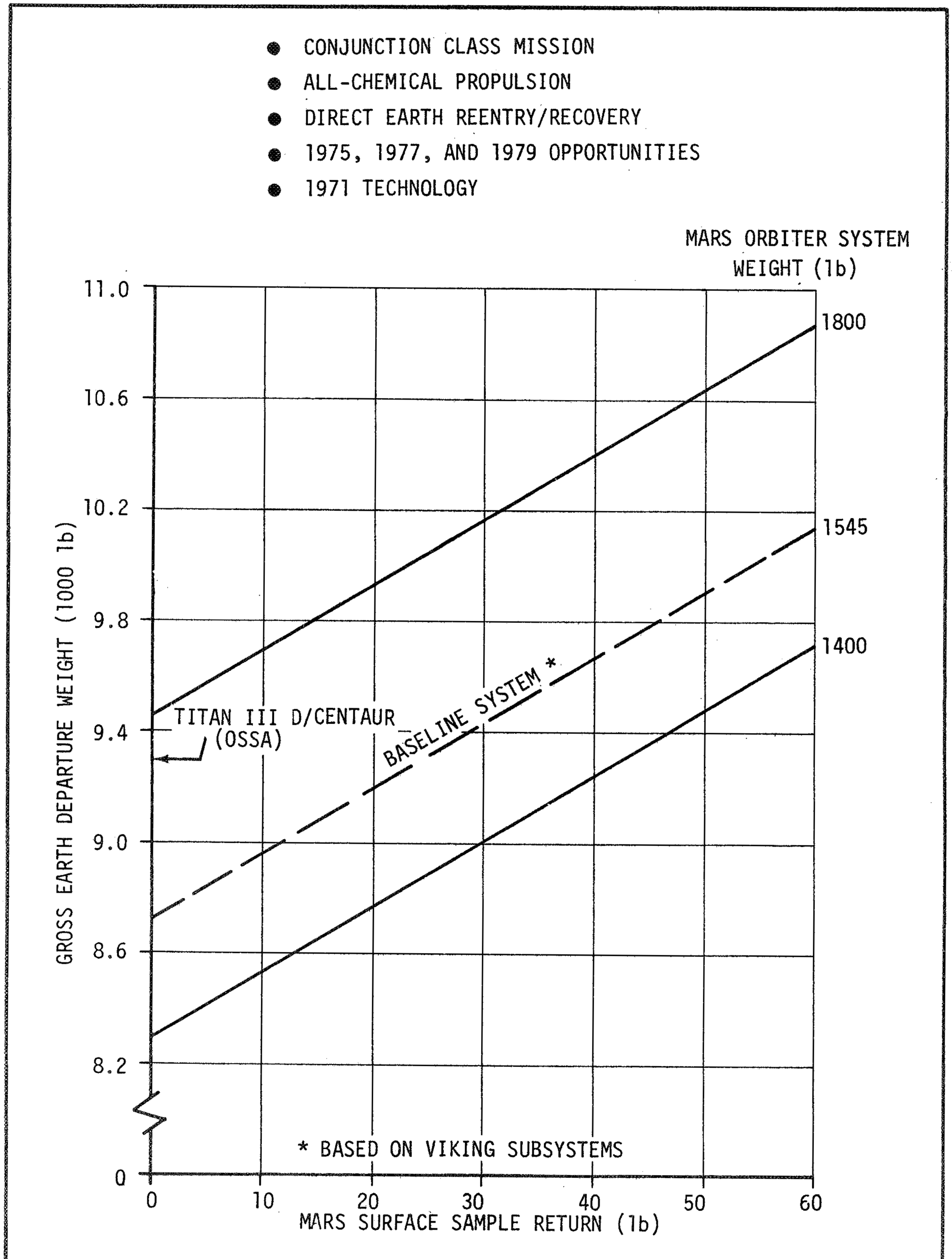


Figure 6-25. DUAL LAUNCH ORBITER/BUS - EFFECT OF SUBSYSTEMS WEIGHT ON PERFORMANCE

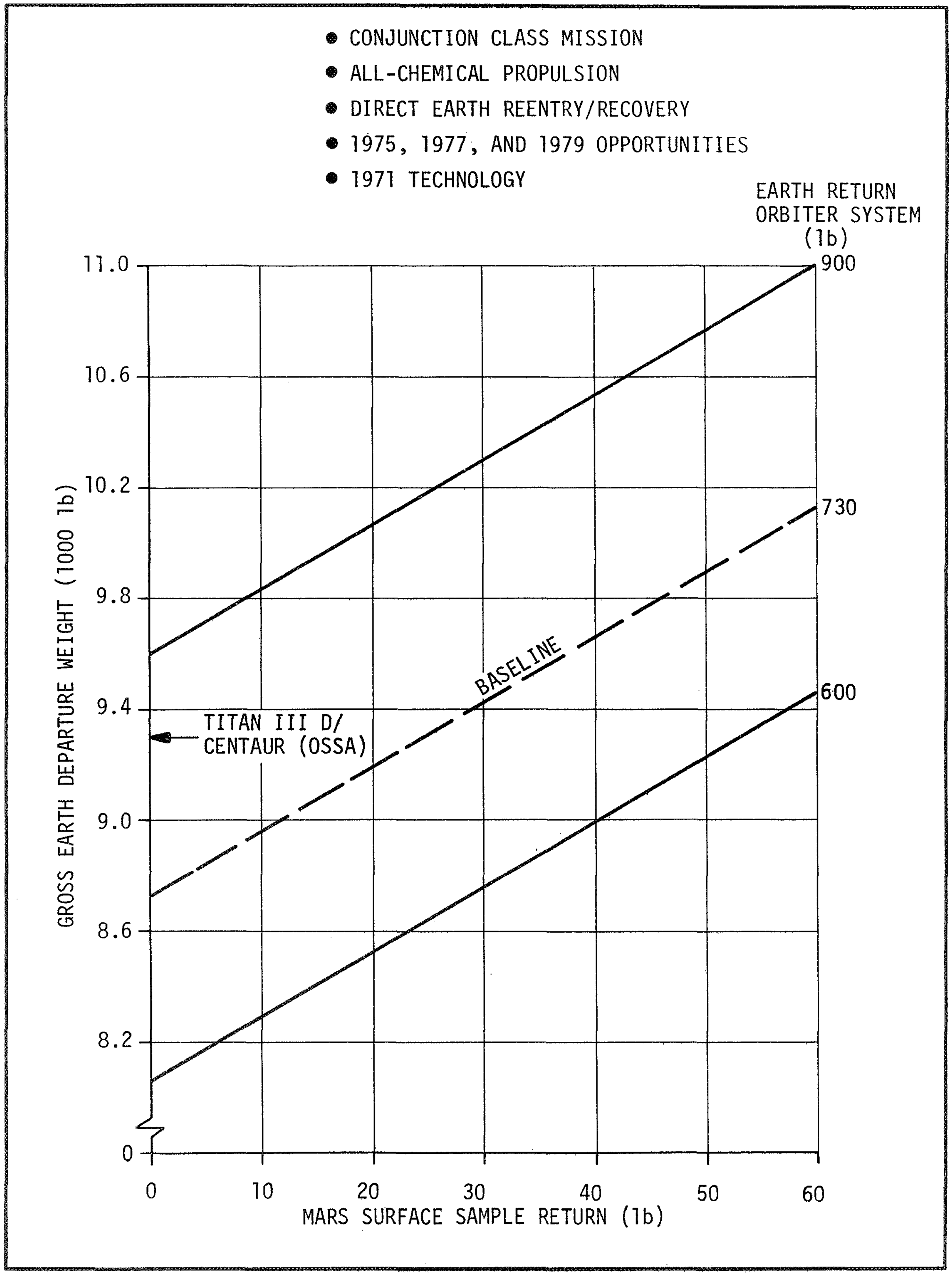


Figure 6-26. DUAL LAUNCH ORBITER/BUS - EFFECT OF EARTH RETURN ORBITER/BUS WEIGHT ON PERFORMANCE

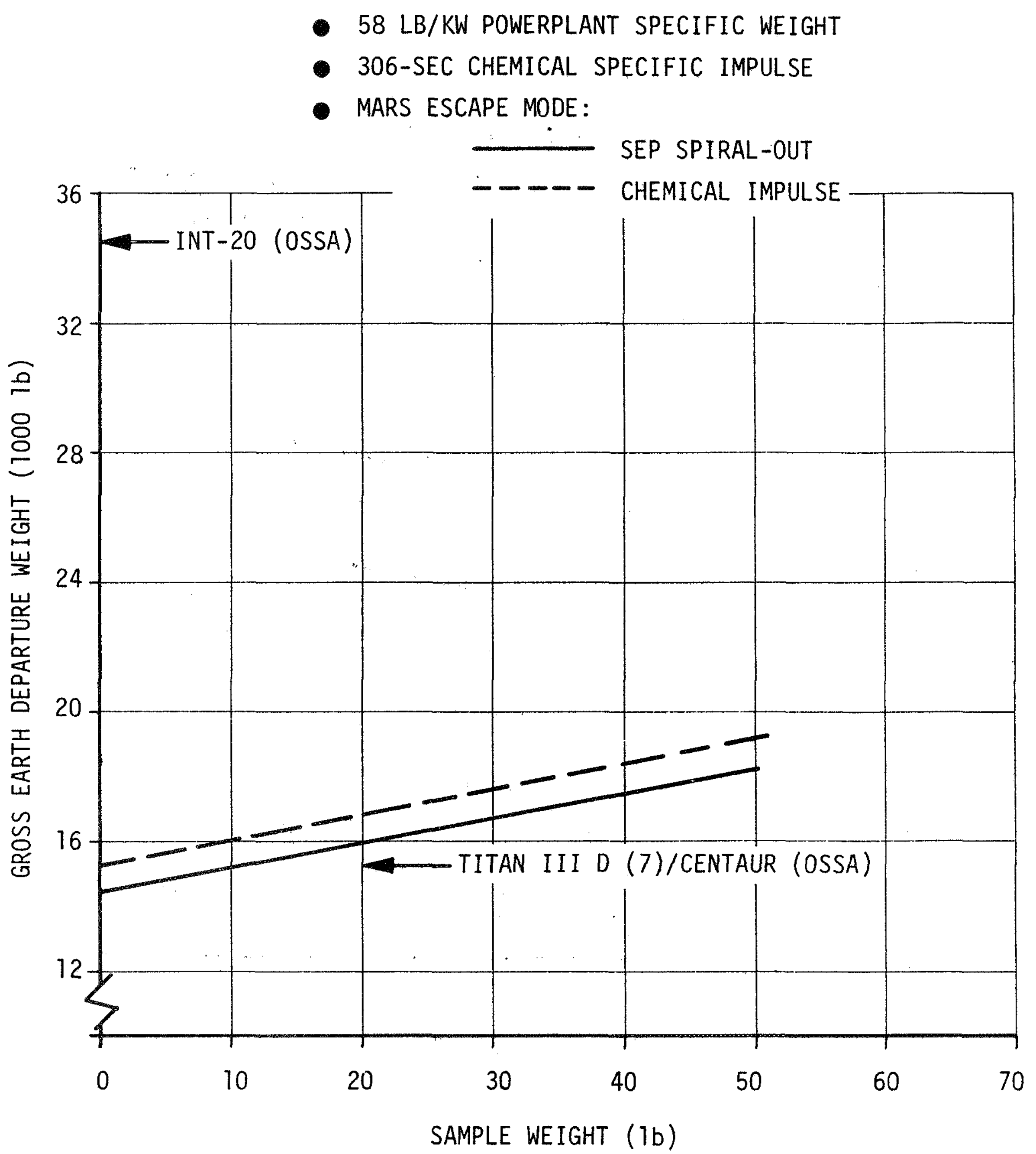


Figure 6-27. SINGLE LAUNCH SOLAR-ELECTRIC CONCEPT PERFORMANCE FOR DIRECT MARS ENTRY (DIRECT REENTRY/RECOVERY)

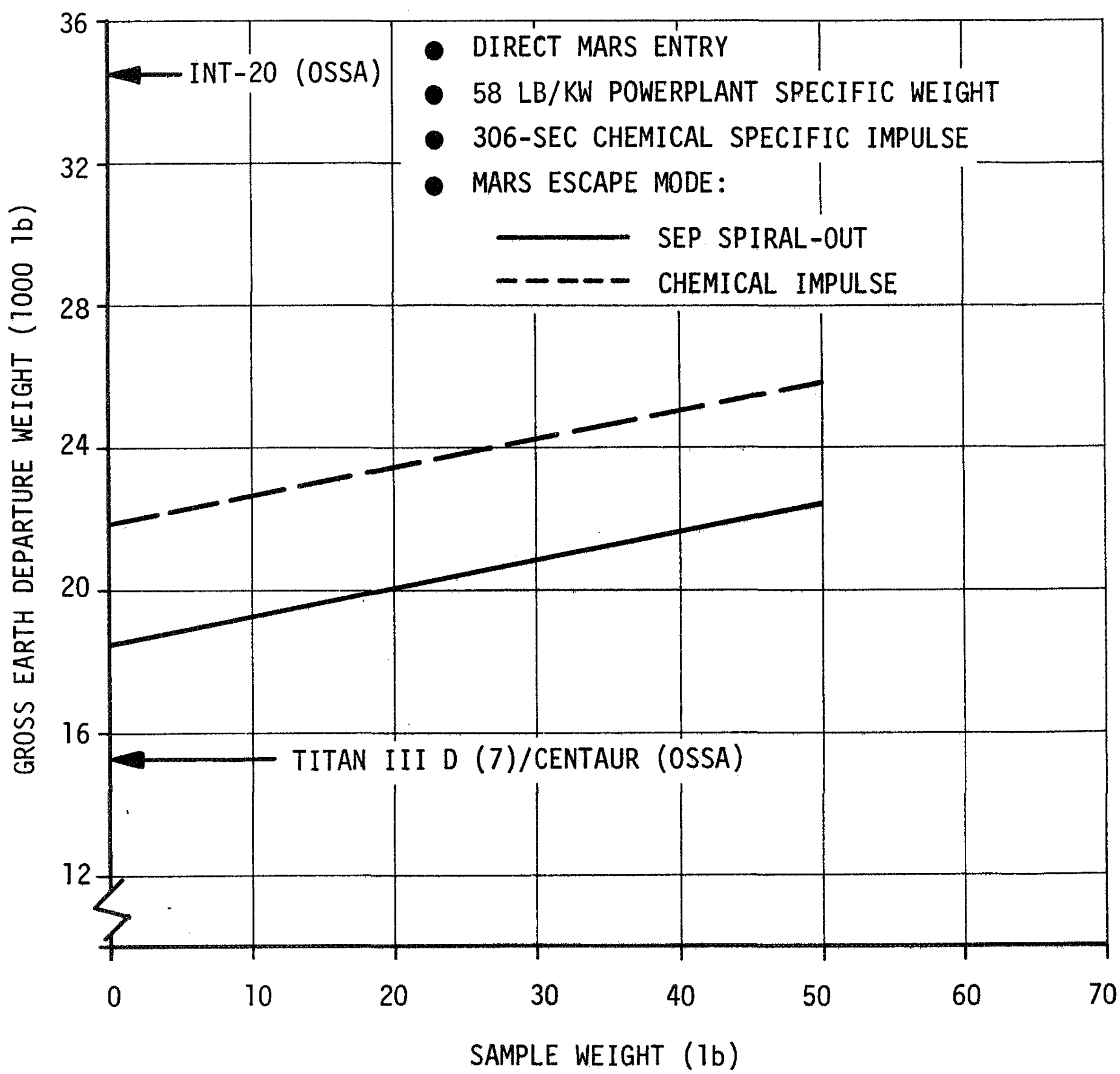


Figure 6-28. SINGLE LAUNCH SOLAR-ELECTRIC CONCEPT PERFORMANCE FOR DIRECT MARS ENTRY (ORBITAL CAPTURE/RECOVERY)

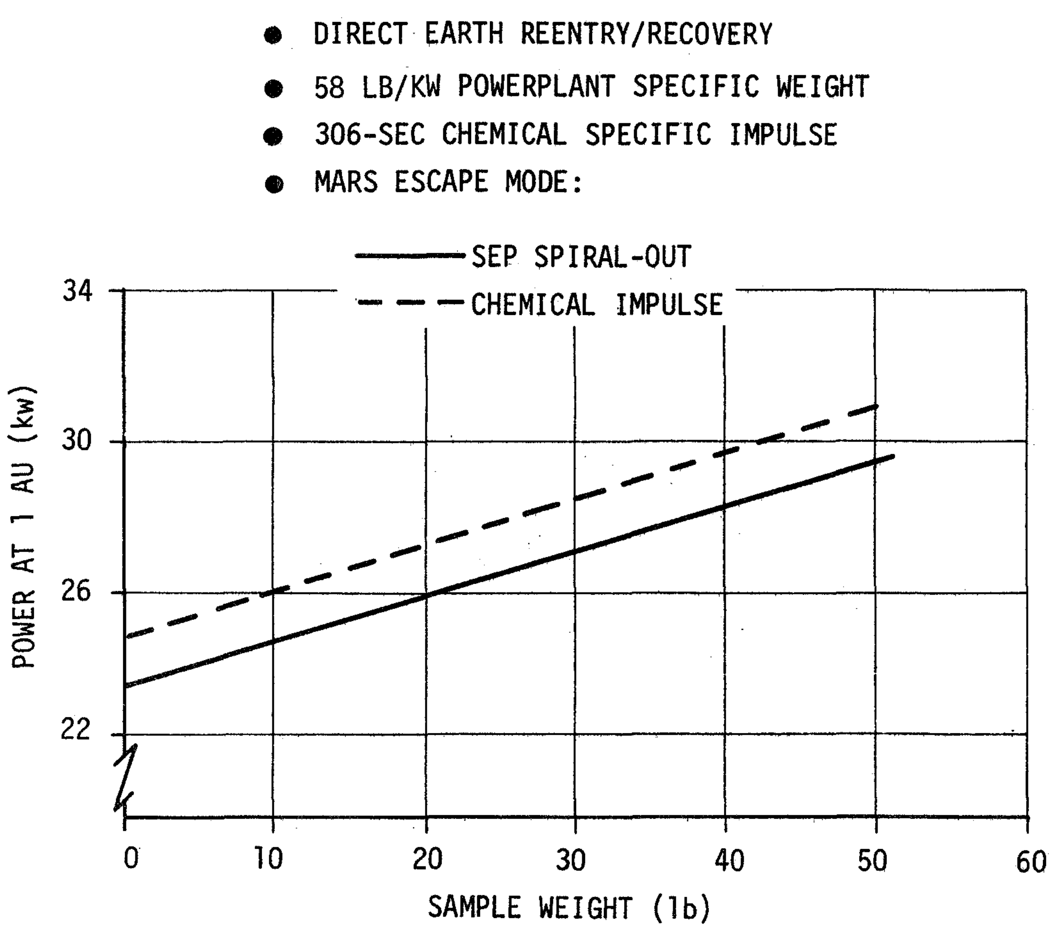


Figure 6-29. SINGLE LAUNCH SOLAR-ELECTRIC CONCEPT - POWER REQUIRED AT 1 AU (DIRECT MARS ENTRY, DIRECT REENTRY/RECOVERY)

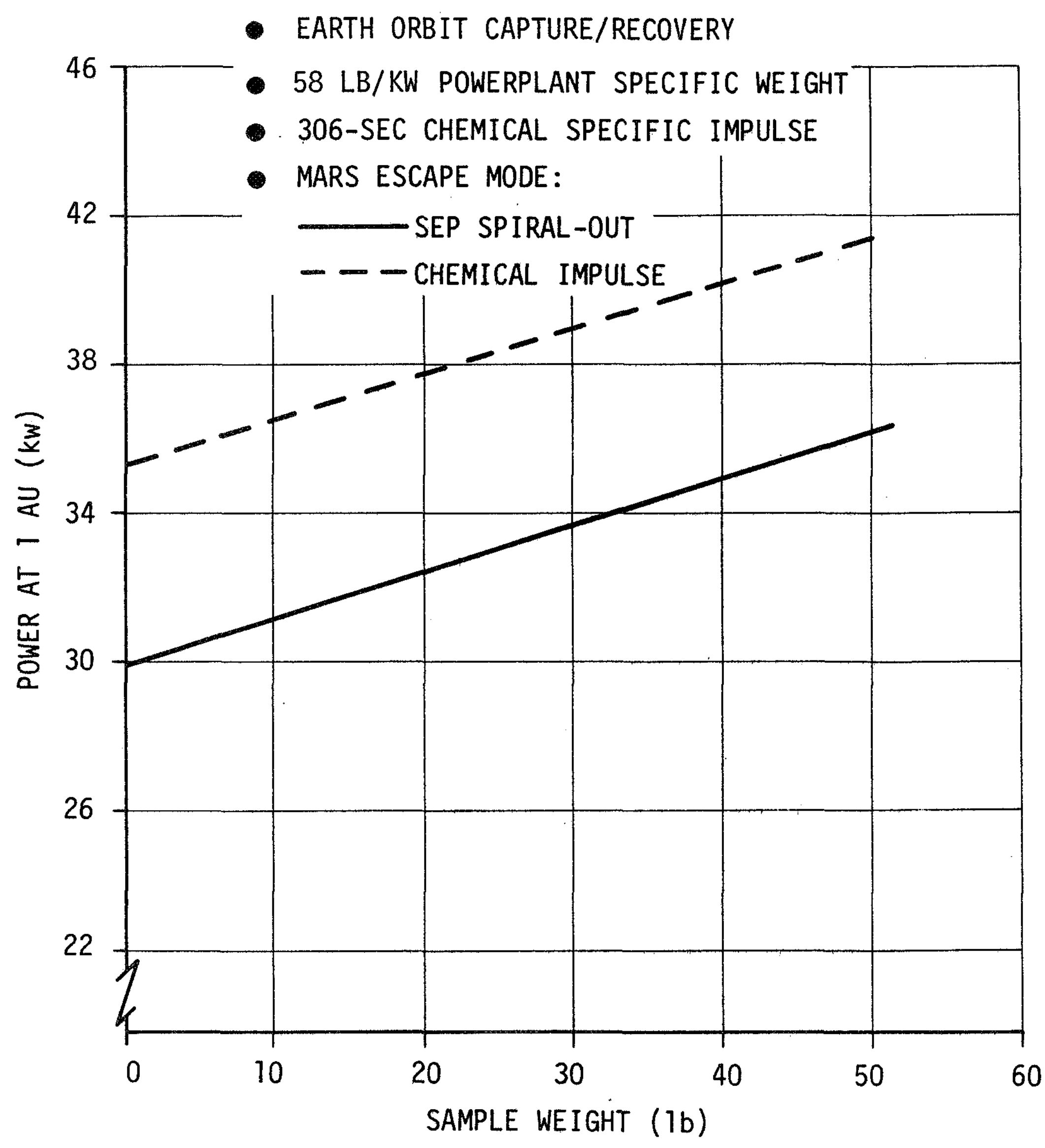


Figure 6-30. SINGLE LAUNCH SOLAR-ELECTRIC CONCEPT - POWER REQUIRED AT 1 AU (DIRECT MARS ENTRY, ORBITAL CAPTURE/RECOVERY)

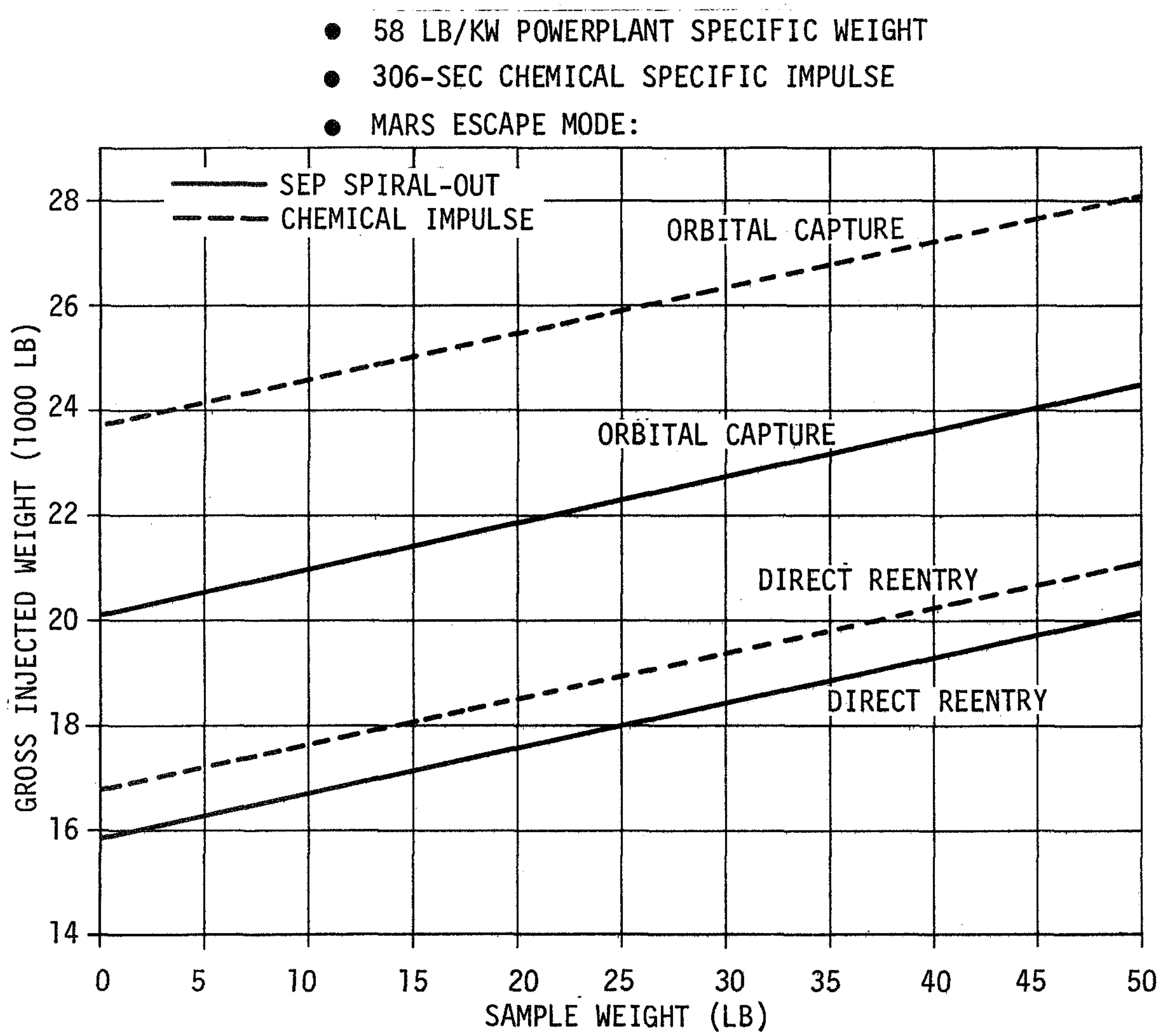


Figure 6-31. SINGLE LAUNCH SOLAR-ELECTRIC CONCEPT PERFORMANCE FOR MARS ENTRY OUT OF ELLIPTICAL ORBIT

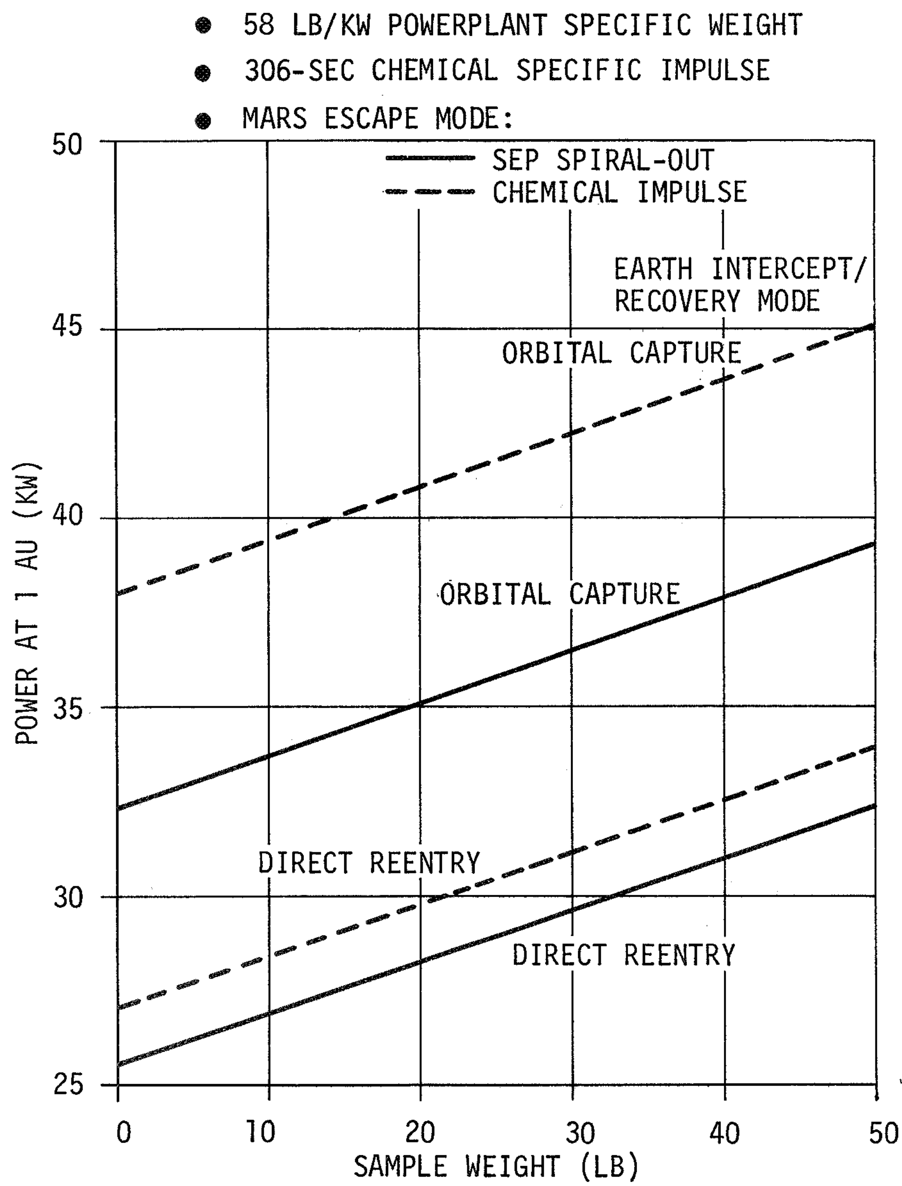


Figure 6-32. SINGLE LAUNCH SOLAR-ELECTRIC CONCEPT - POWER REQUIRED AT 1 AU (MARS ENTRY OUT OF ELLIPTICAL ORBIT)

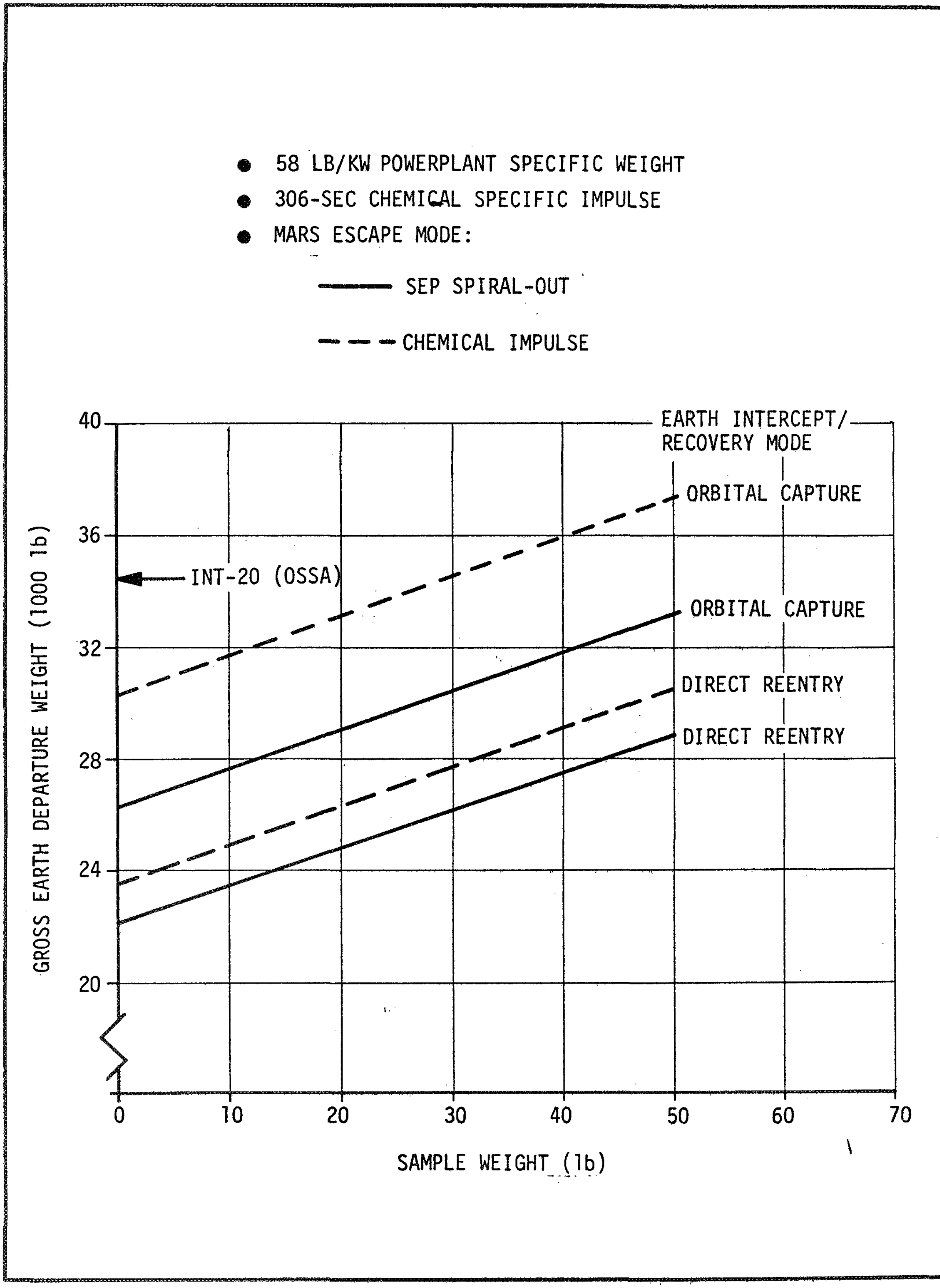


Figure 6-33. SINGLE LAUNCH SOLAR-ELECTRIC CONCEPT PERFORMANCE (MARS ENTRY OUT OF CIRCULAR ORBIT)

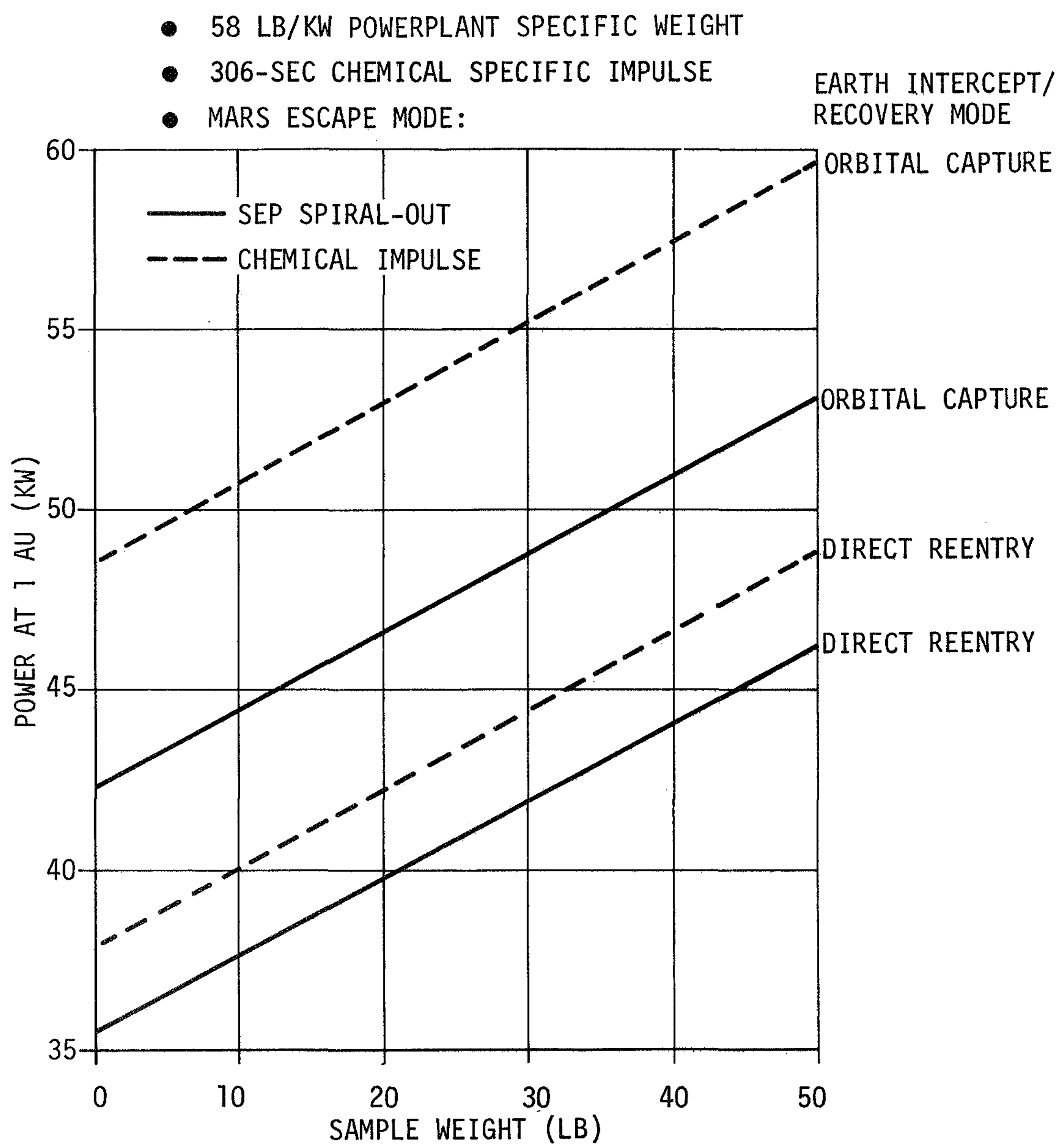


Figure 6-34. SINGLE LAUNCH SOLAR-ELECTRIC CONCEPT - POWER REQUIRED AT 1 AU (MARS ENTRY OUT OF CIRCULAR ORBIT)

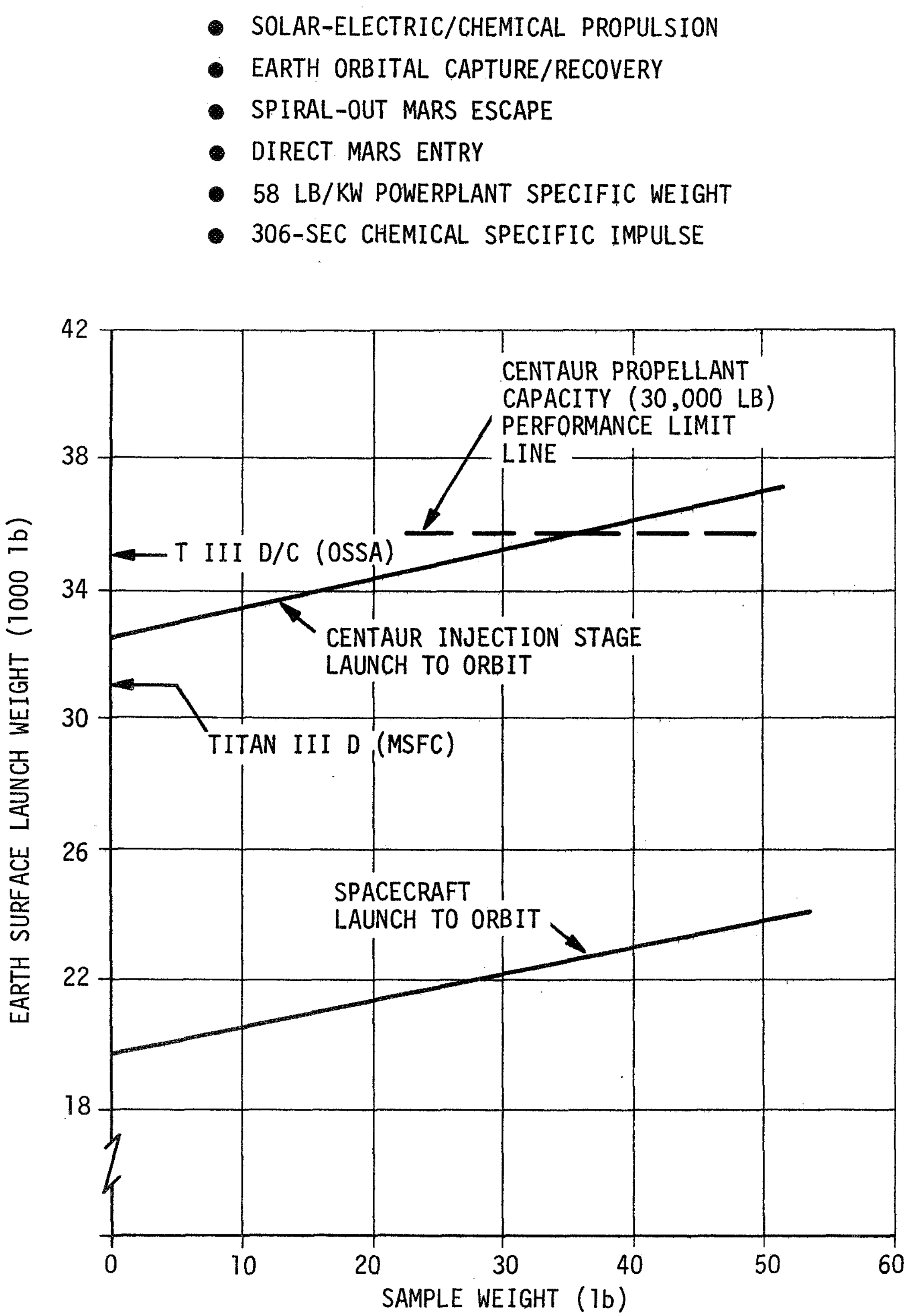


Figure 6-35. EARTH ORBIT RENDEZVOUS CONCEPT PERFORMANCE FOR SOLAR ELECTRIC SYSTEMS

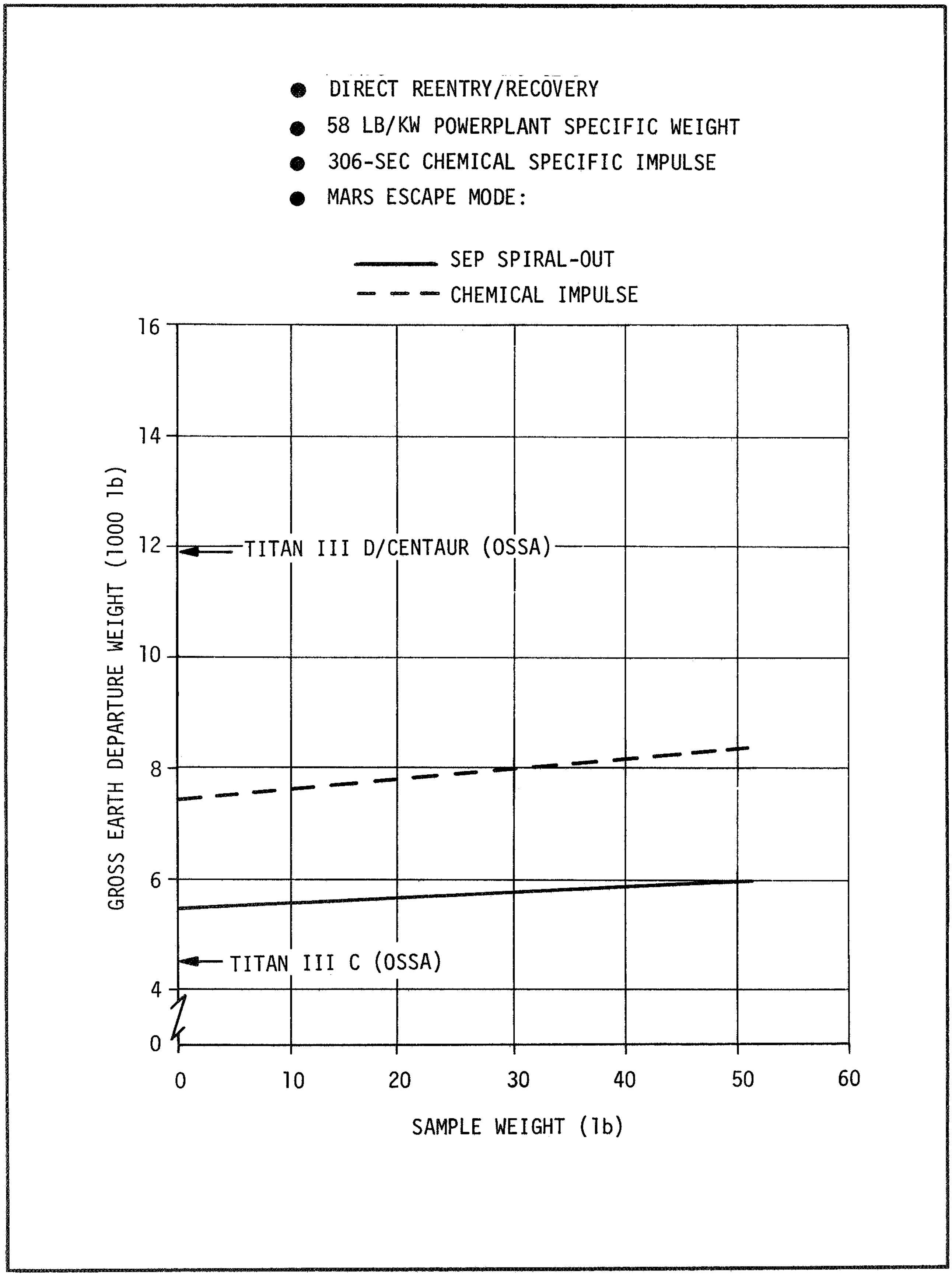


Figure 6-36. DUAL LAUNCH SOLAR-ELECTRIC ORBITER/BUS PERFORMANCE (DIRECT REENTRY/RECOVERY MODE)

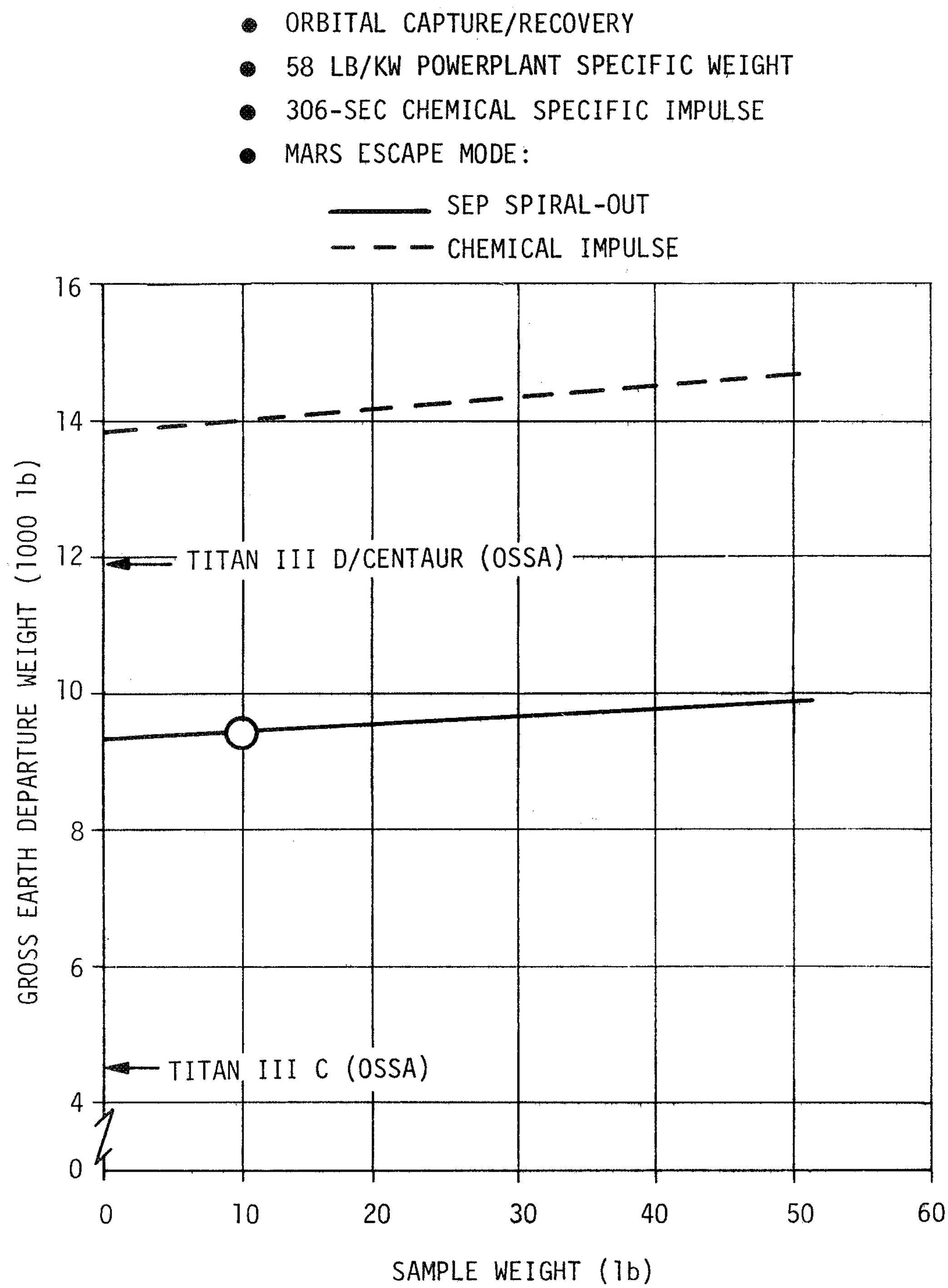


Figure 6-37. DUAL LAUNCH SOLAR-ELECTRIC ORBITER/BUS PERFORMANCE (ORBITAL CAPTURE/RECOVERY MODE)

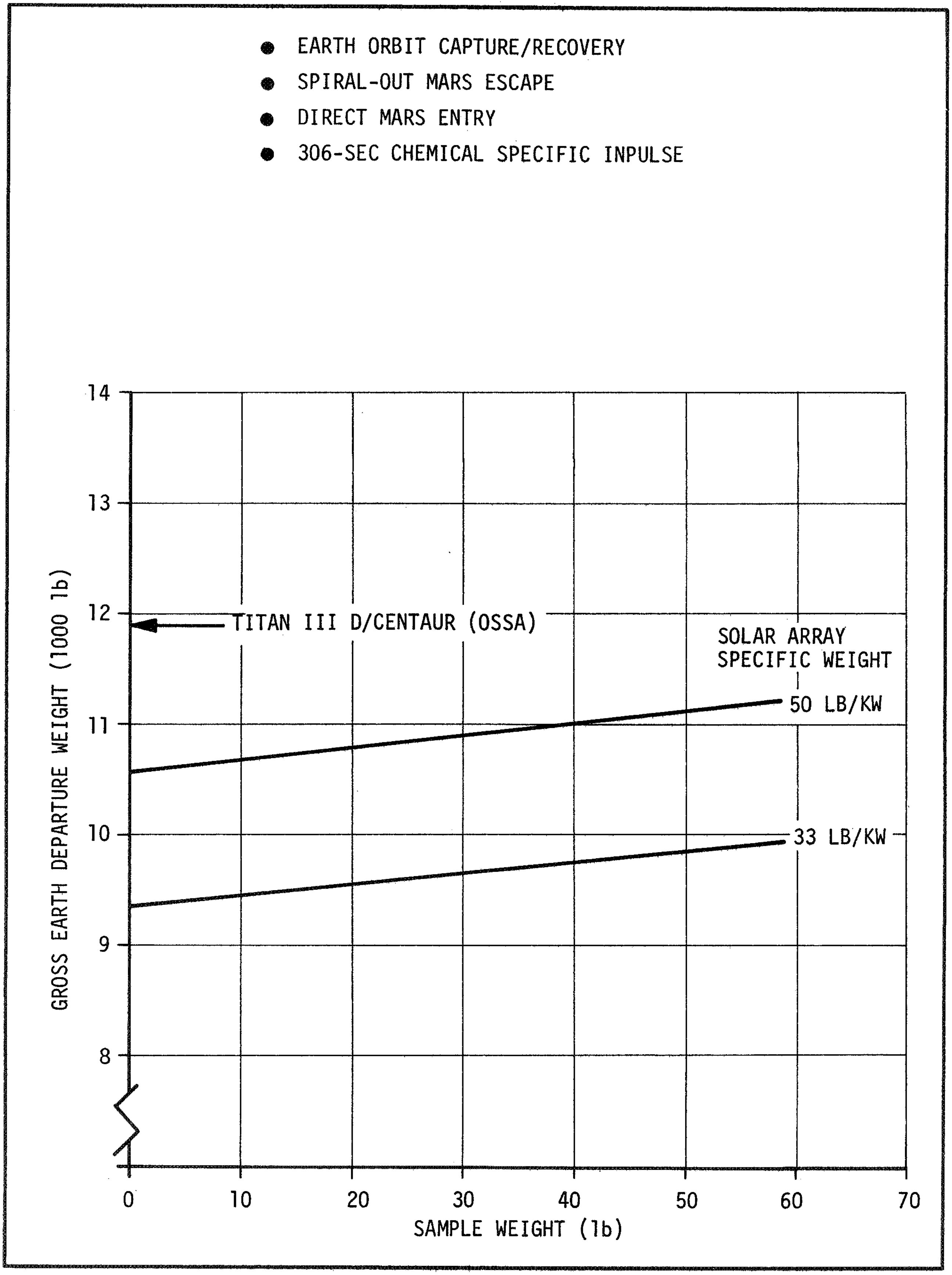


Figure 6-38. DUAL DEPARTURE CONCEPT - EFFECT OF ARRAY SPECIFIC WEIGHT ON SOLAR-ELECTRIC ORBITER/BUS PERFORMANCE

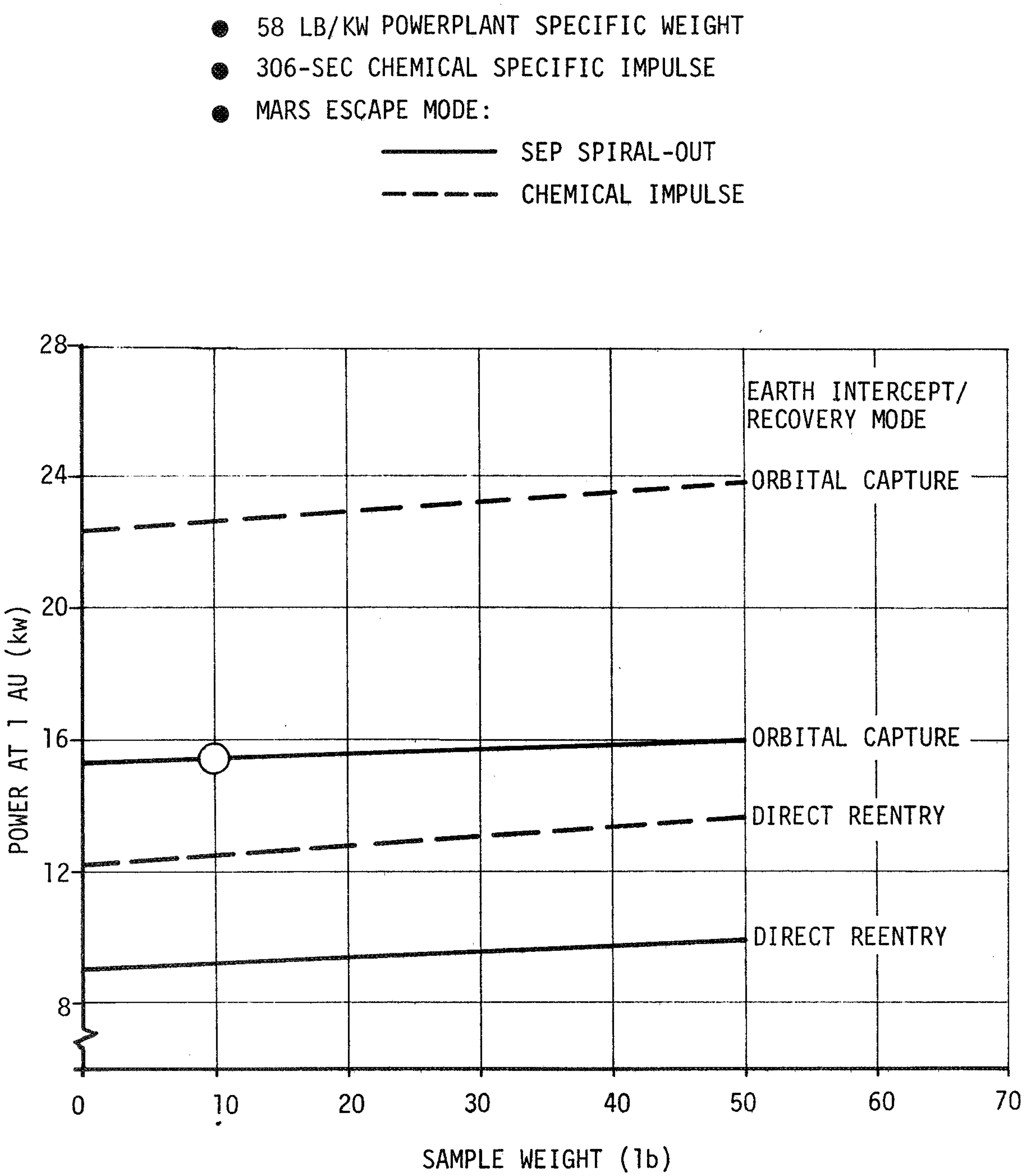


Figure 6-39. DUAL LAUNCH SOLAR-ELECTRIC ORBITER/BUS POWER REQUIRED AT 1 AU

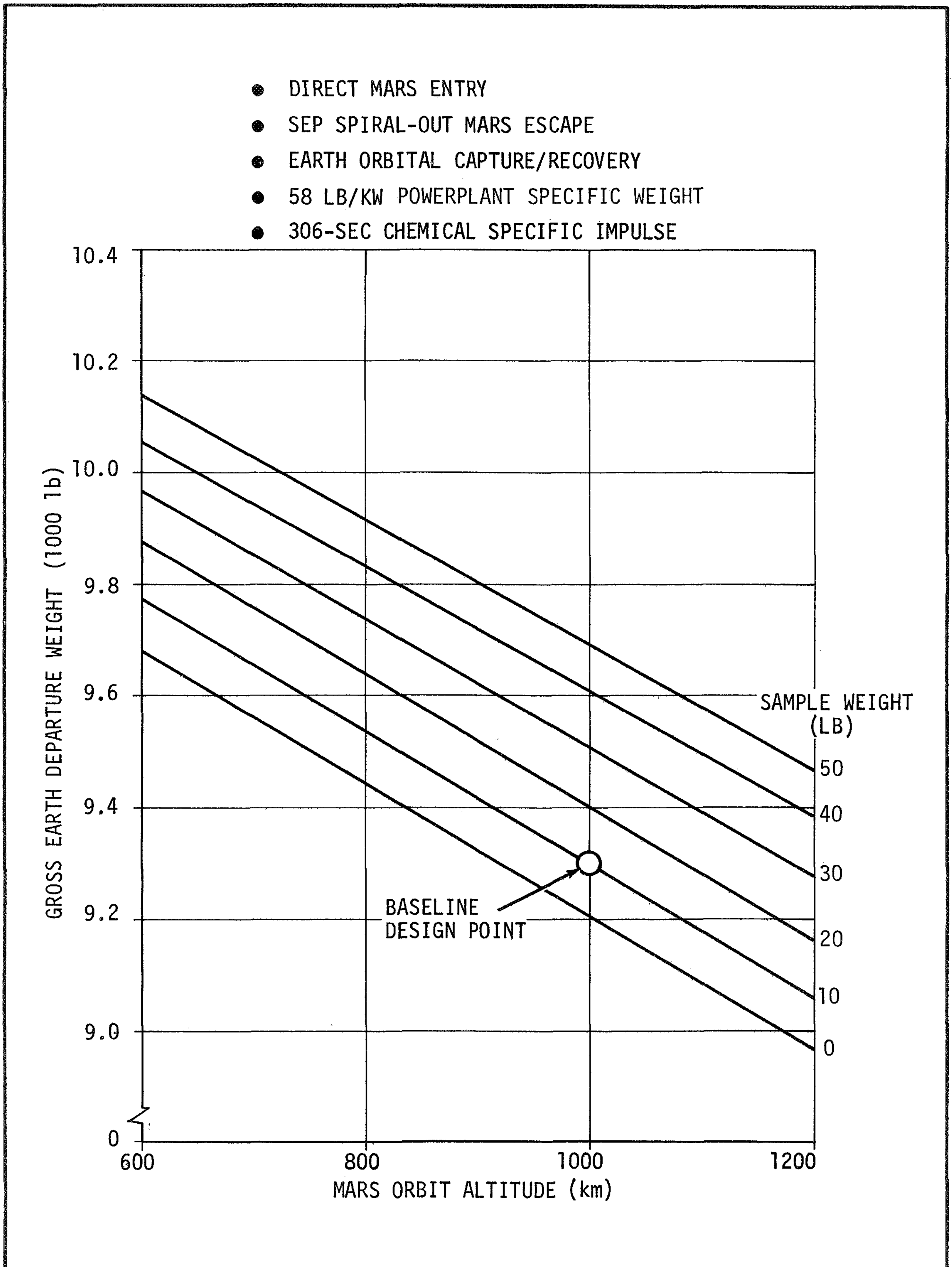


Figure 6-40. EFFECT OF MARS ORBIT ALTITUDE ON PERFORMANCE OF DUAL DEPARTURE SOLAR-ELECTRIC/CHEMICAL ORBITER/BUS

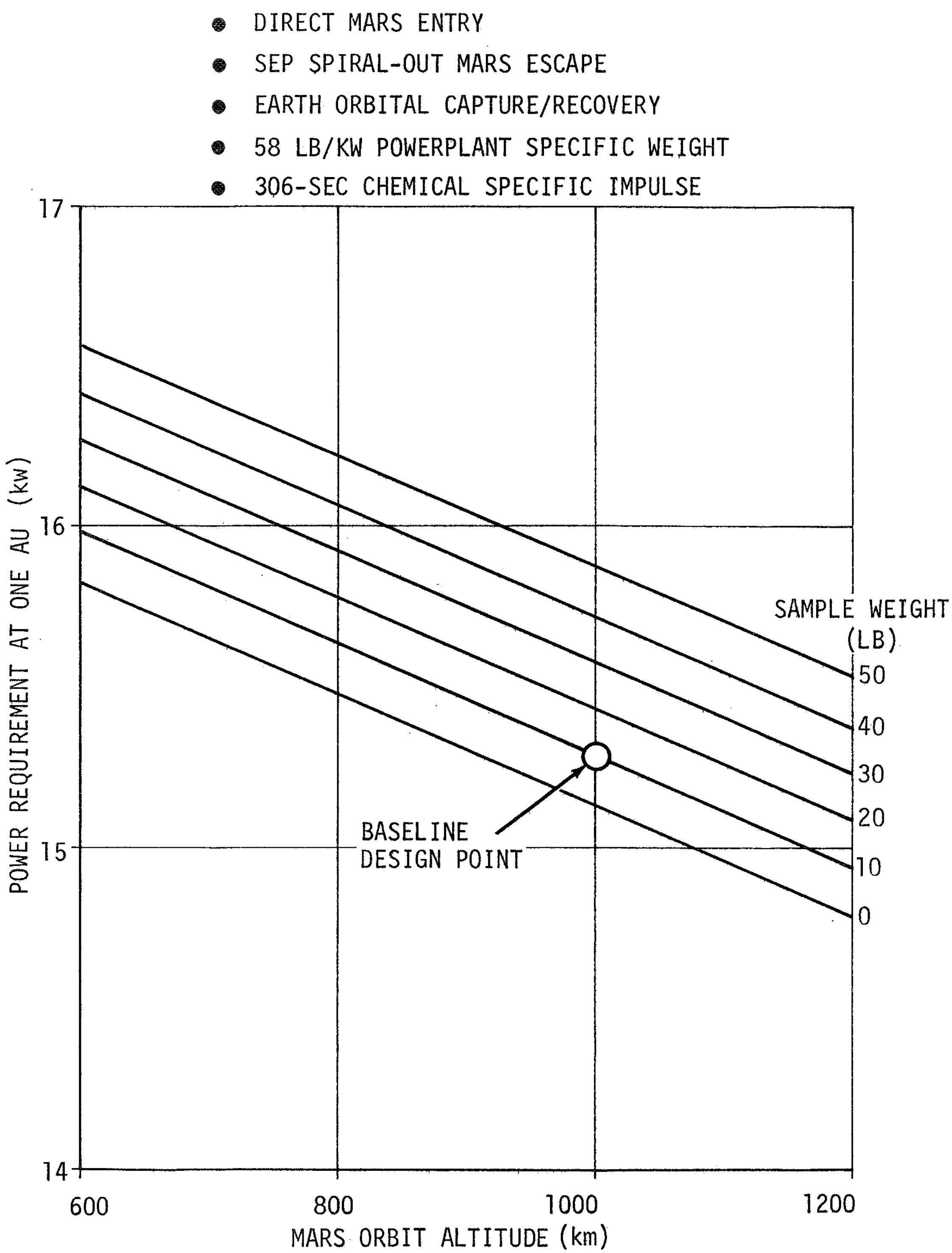


Figure 6-41. EFFECT OF MARS ORBIT ALTITUDE ON POWER REQUIRED AT 1 AU FOR DUAL DEPARTURE, SOLAR-ELECTRIC/CHEMICAL ORBITER/BUS

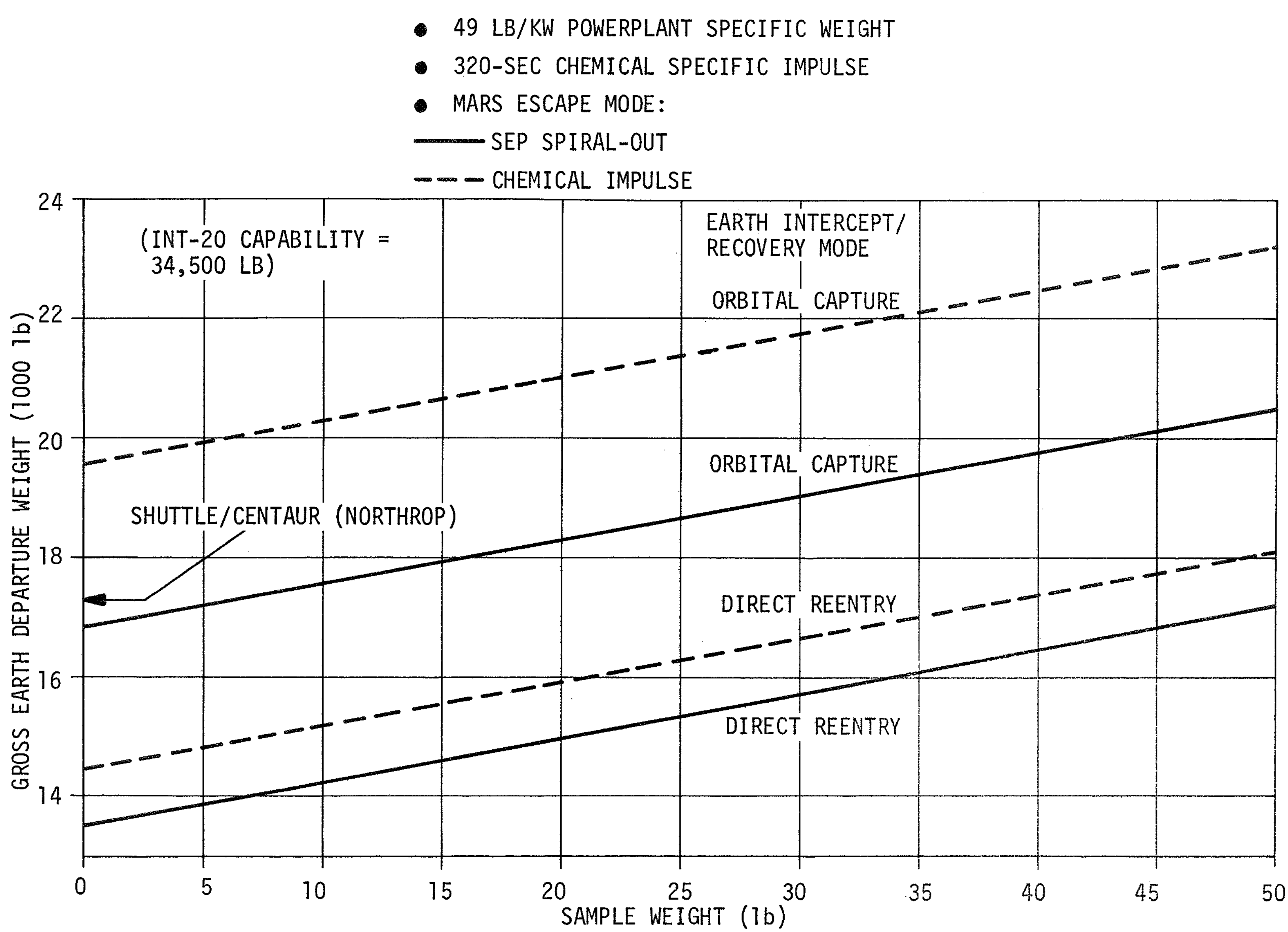


Figure 6-42. SINGLE LAUNCH SOLAR-ELECTRIC CONCEPT PERFORMANCE FOR DIRECT MARS ENTRY AND 1977-79 MISSIONS

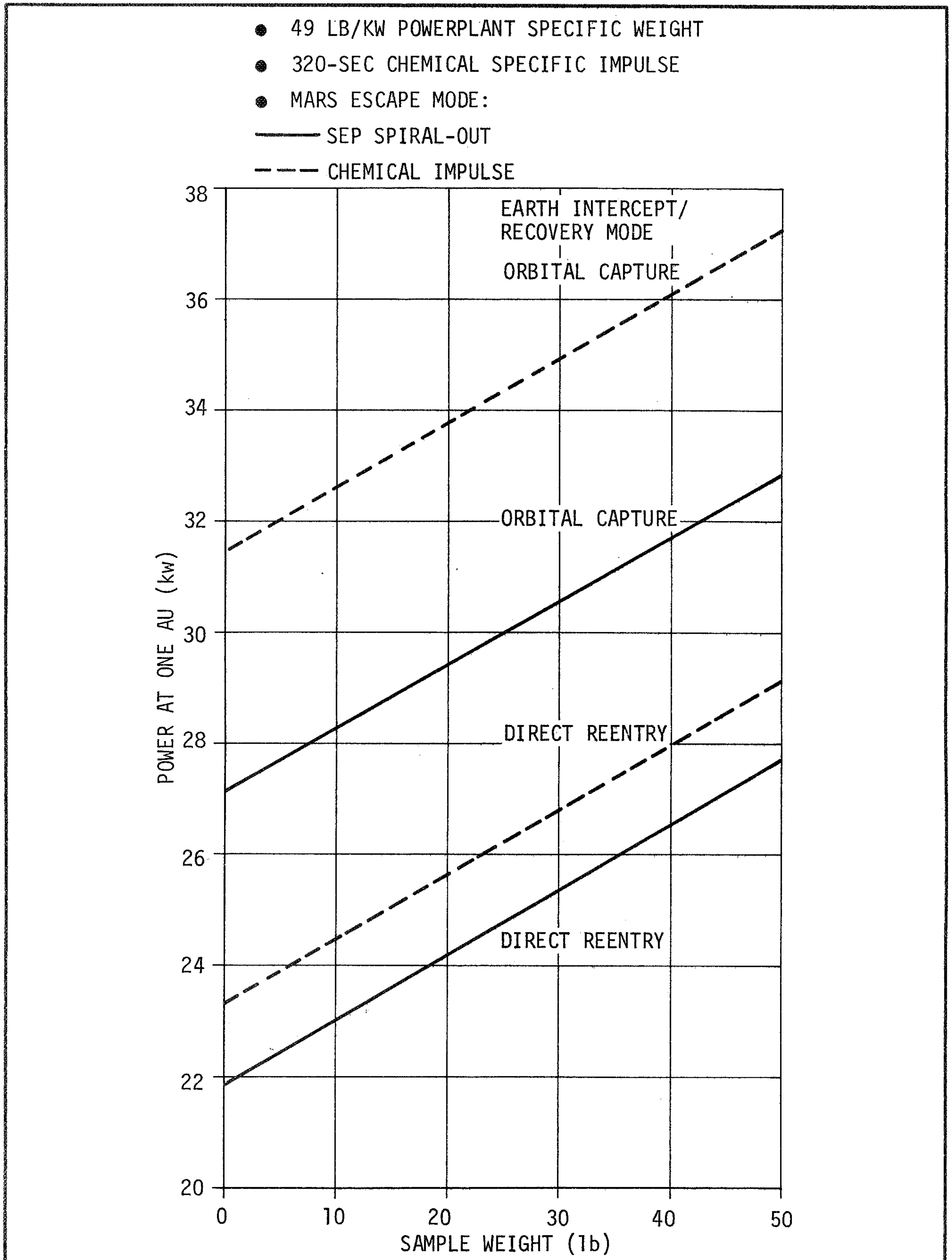


Figure 6-43. SINGLE LAUNCH SOLAR-ELECTRIC CONCEPT POWER REQUIRED AT ONE AU (DIRECT MARS ENTRY, 1977-79 MISSIONS)

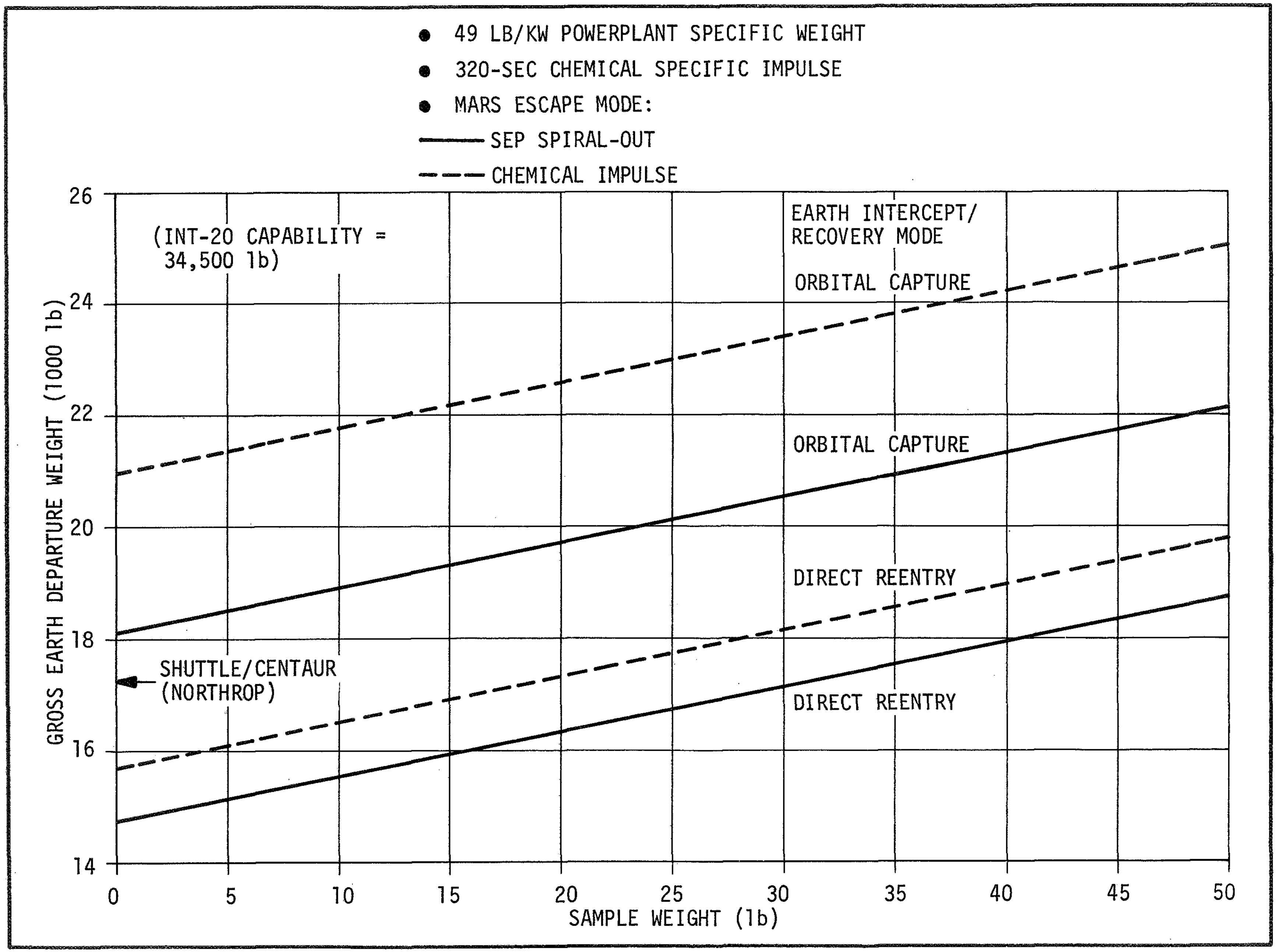


Figure 6-44. SINGLE LAUNCH SOLAR-ELECTRIC CONCEPT PERFORMANCE FOR MARS ENTRY OUT OF ELLIPTICAL ORBIT (1977-79 MISSIONS)

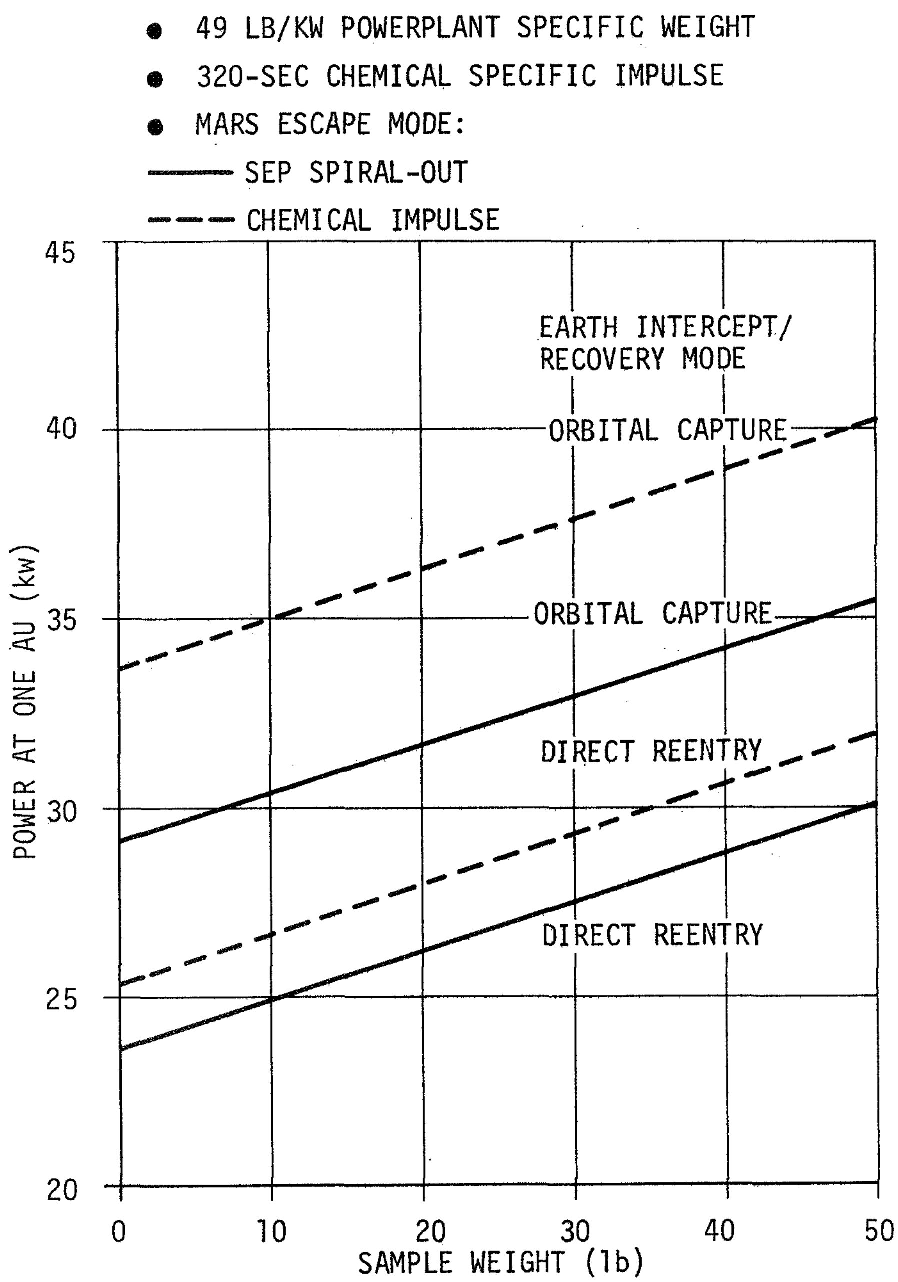


Figure 6-45. SINGLE LAUNCH SOLAR-ELECTRIC CONCEPT - POWER REQUIRED AT ONE AU (MARS ENTRY OUT OF ELLIPTICAL ORBIT, 1977-79 MISSIONS)

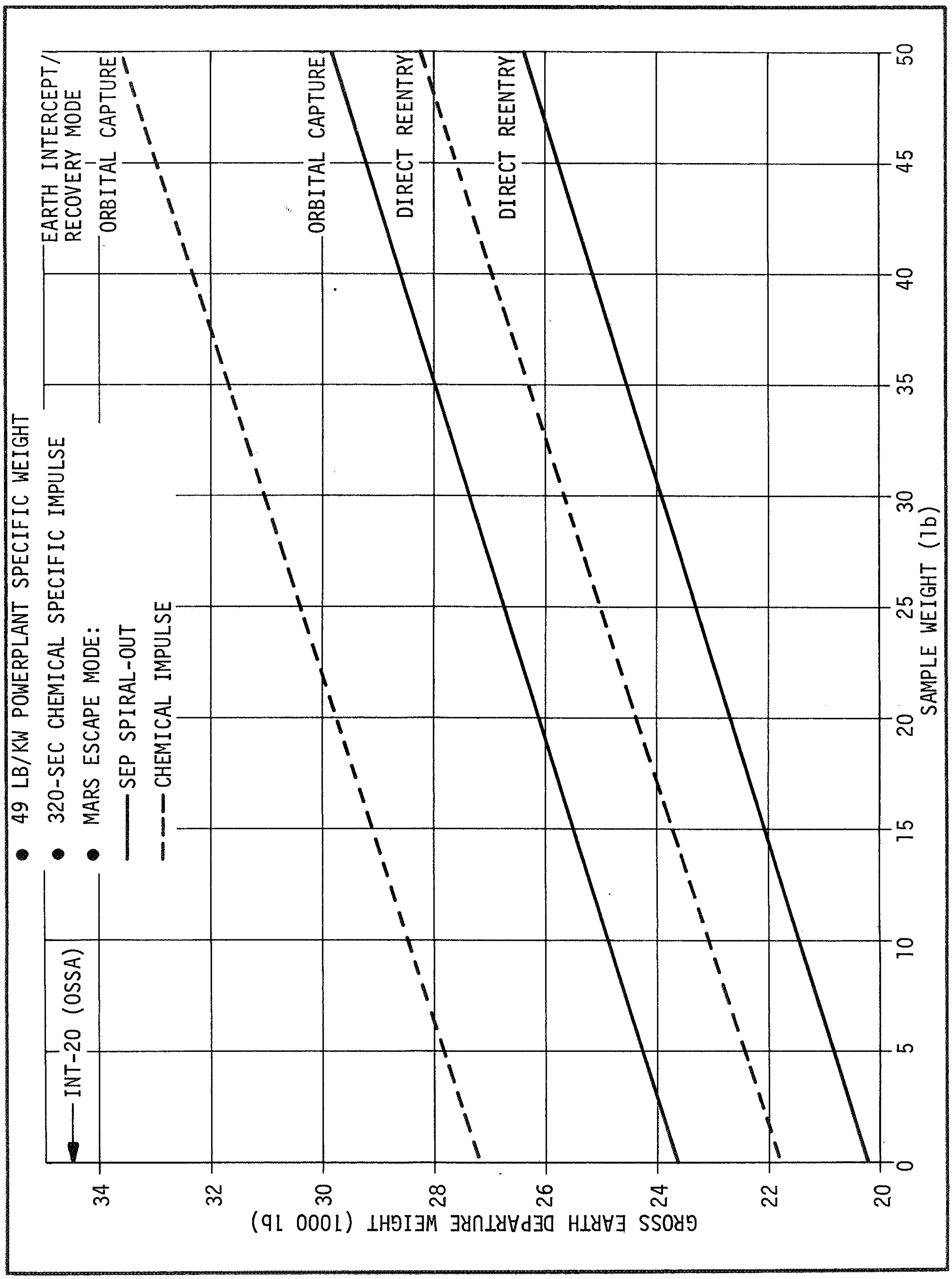


Figure 6-46. SINGLE LAUNCH SOLAR-ELECTRIC CONCEPT PERFORMANCE FOR MARS ENTRY OUT OF CIRCULAR ORBIT (1977-79 MISSIONS)

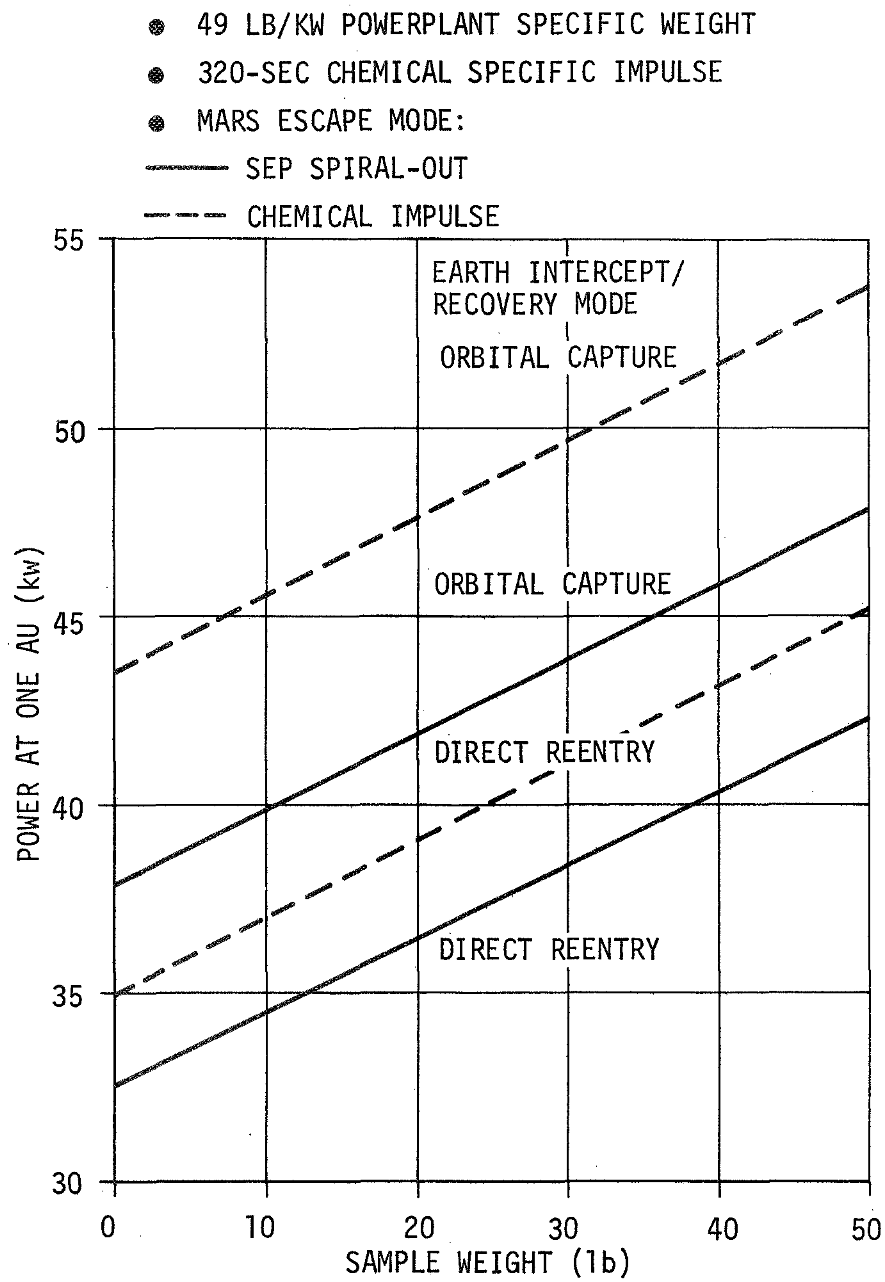


Figure 6-47. SINGLE LAUNCH SOLAR-ELECTRIC CONCEPT POWER REQUIRED AT ONE AU (MARS ENTRY OUT OF CIRCULAR ORBIT, 1977-79 MISSIONS)

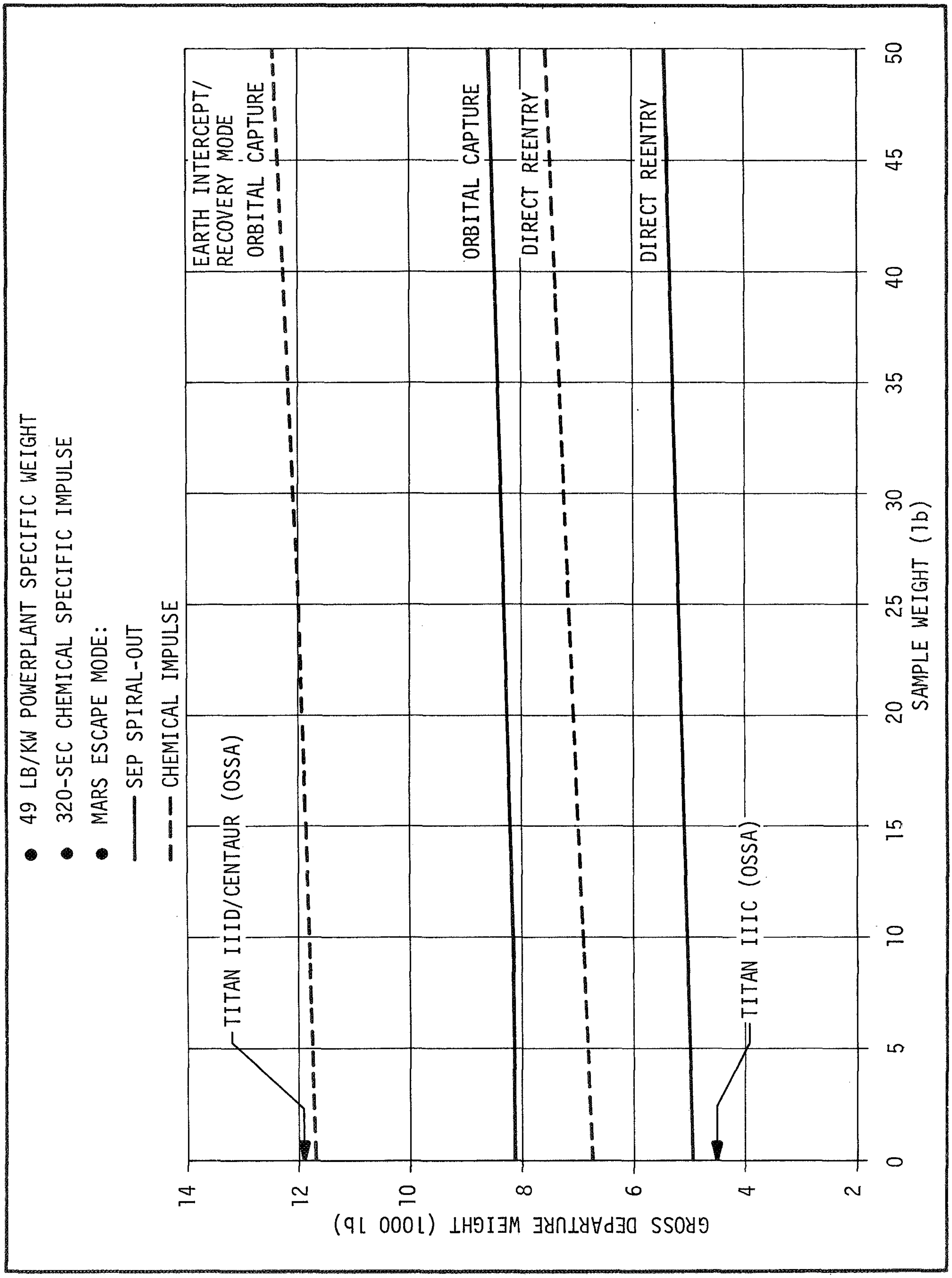


Figure 6-48. DUAL DEPARTURE SOLAR-ELECTRIC ORBITER/BUS PERFORMANCE FOR 1977-79 MISSIONS

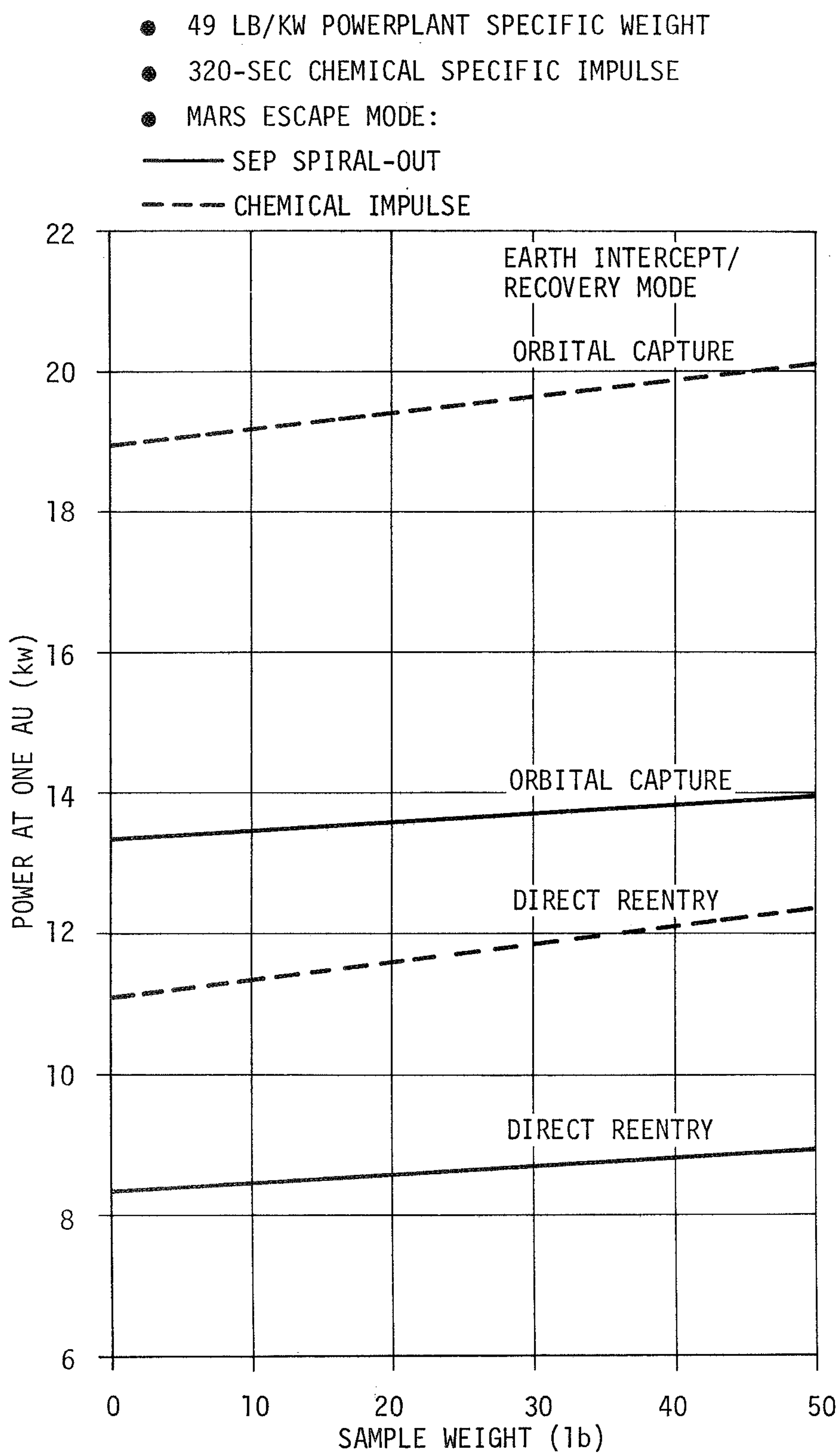


Figure 6-49. DUAL DEPARTURE SOLAR-ELECTRIC ORBITER/BUS POWER REQUIRED AT ONE AU

Section VII

COMPARATIVE EVALUATION OF ALTERNATIVE MISSION/SYSTEM APPROACHES

7.1 EVALUATION CRITERIA

On the basis of the mission/system definitions presented in Sections II through V and the parametric performance analysis in Section VI, a comparative evaluation was performed to select the more promising concepts for preliminary point design analysis. The criteria used in the evaluation were as follows:

- Assume a Minimum Sample Return Mission. It was assumed for purposes of comparative evaluation that a sample weight of 10 pounds is representative of a minimum mission. This assumption is based on the analysis of sample return requirements documented in reference 2. By requiring that all system alternatives perform the 10-pound return mission, the sample return performance requirement was fixed and sample return capability dropped out of consideration as a parameter for comparison except as it reflects growth potential of the system.

- Earth Launch Vehicle Availability. Matching available launch vehicle capabilities to gross earth departure weight requirements provides a straight-forward criterion for selection of candidate mission/system alternatives for further consideration.

- Minimum Program Cost is Desired. An underlying objective of the present study has been to identify and define potential low cost approaches to the MSSR mission. A reasonably valid measure of program cost is the size or weight of the spacecraft systems required since the essential hardware elements of the various system approaches have similar functional requirements. In addition to system weight, the cost of the solar-electric/chemical concepts is significantly impacted by the solar array power required. Thus, minimum cost mission/system approaches will tend to be those which require minimum gross Earth departure weight and power in the case of solar-electric/chemical concepts.

- Require a Low Back Contamination Risk. A low back contamination risk at Earth return must be provided by the MSSR mission/system approach. It is not clear at the time of this study what the actual back contamination risks are for the two Earth intercept/recovery modes (direct reentry and capture/recovery) under consideration. However, it is reasonable to assume that the Earth capture/recovery mode is more desirable until in-depth studies have proven otherwise.

- Growth Potential is Desirable. A secondary consideration in the comparative evaluation was the performance growth potential available in the given mission/system alternative approaches. This can be measured by the performance margin available between the Earth launch vehicle mission capability and the required gross Earth departure weight of the spacecraft.

7.2 COMPARISON OF ALTERNATIVES

Because of particular interest in the 1975 mission possibilities in the present study, the 1971 technology base mission/system alternatives were used for the comparative evaluation. Figure 7-1 presents a matrix of the alternatives in terms of the Earth launch vehicle/mission concept combinations and two orbiter/bus spacecraft propulsion approaches under consideration (all-chemical and solar-electric/chemical). The elements of the matrix show: (1) the gross Earth departure weight requirement for a 10-pound sample return system; (2) the Earth launch vehicle capability for the 1975 mission opportunity; (3) the Earth intercept/recovery mode available assuming capture of the return orbiter/bus vehicle into orbit for retrieval of the samples by an orbit-launched Apollo CSM; and (4) the power required at 1 AU for solar-electric/chemical concepts.

Application of the criteria which have been discussed to a comparative evaluation of the available alternatives lead to the following conclusions:

- The dual departure, all-chemical concept using two Titan IIID/Centaur vehicles offers potentially the lowest cost (including consideration of launch vehicles) approach but depends on the direct reentry/recovery mode.
- If Earth capture/recovery is required or groundruled into the mission, then two promising alternatives appear to be:
  - Dual departure, solar-electric/chemical concept using two Titan IIID/Centaur vehicles. This concept offers good growth potential and requires a reasonable power level of approximately 15 kw at 1 AU.
  - Single launch, all-chemical concept using the INT-20/Centaur vehicle. This is the only all-chemical approach considered which offers the Earth capture/recovery mode. If the sub-module capture approach* described in Section VI can be mechanized, the INT-20 or possibly the Titan IIID(7)/Centaur vehicle (if developed) could be used in this concept.

Based on the above evaluation, the three mission/system alternative concepts selected for preliminary point design analysis were as follows:

- Dual Departure, All-Chemical System launched by two Titan IIID/Centaur vehicles.
- Dual Departure, Solar-Electric/Chemical System launched by two Titan IIID/Centaur vehicles.
- Single Launch, All-Chemical System launched by the INT-20/Centaur vehicle.

**In this system approach, a sub-module of the return orbiter/bus spacecraft is separated during Earth approach for capture into orbit. This significantly reduces the weight of the propulsion system required for capture.*

COMPARISON OF ALTERNATIVES

(1971 TECHNOLOGY)

- WG = GROSS S/C EARTH LAUNCH WEIGHT FOR 10-LB SAMPLE RETURN
- WL = ELV PAYLOAD CAPABILITY
- P1 = SEP POWER REQ'D AT 1 AU
- DIRECT MARS ENTRY MODE

SPACECRAFT PROPULSION	EARTH LAUNCH VEHICLE/MISSION CONCEPTS					
	• TITAN III D • TITAN III D/CENTAUR • EARTH ORBIT RENDEZVOUS • CENTAUR INJ. STAGE	• TITAN III D/CENTAUR • SINGLE LAUNCH	• TITAN III D/CENTAUR • DUAL DEPARTURE	• INT-20 • SINGLE LAUNCH	• INT-20/CENTAUR • SINGLE LAUNCH	• SHUTTLE/CENTAUR • SINGLE LAUNCH
ALL-CHEMICAL	• MARGINAL PERFORMANCE • DIRECT REENTRY/ RECOVERY	NOT FEASIBLE	• WG1 = 8500 LB (PROBE) • WG2 = 9000 LB (ORBITER/BUS) • WL = 9300 LB • DIRECT REENTRY/ RECOVERY	• WG = 16,400 LB • WL = 24,000 LB • DIRECT REENTRY/ RECOVERY	• WG = 30,300 LB • WL = 36,200 LB • ORBITAL CAPTURE/ RECOVERY	• WG = 14,400 LB • WL = 15,000 LB • SHUTTLE NOT AVAILABLE UNTIL 1979 OPPORTUNITY • DIRECT REENTRY RECOVERY
SOLAR ELECTRIC/CHEMICAL	• WG = 20,500 LB • P1 = 31.2 KW • ORBITAL CAPTURE/ RECOVERY	NOT FEASIBLE	• WG1 = 8500 LB (PROBE) • WG2 = 9300 LB (ORBITER/BUS) • WL = 11,900 LB • P1 = 15.3 KW • ORBITAL CAPTURE/ RECOVERY	• WG = 19,300 • WL = 34,500 LB • P1 = 31.2 KW • ORBITAL CAPTURE/ RECOVERY	• WG = 19,300 • WL = 46,500 LB • P1 = 31.2 KW • ORBITAL CAPTURE/ RECOVERY	• MARGINAL PERFORMANCE • SHUTTLE NOT AVAILABLE UNTIL 1979 OPPORTUNITY • ORBITAL CAPTURE RECOVERY

Figure 7-1. COMPARISON OF ALTERNATIVES - 1971 TECHNOLOGY

Section VIII
PRELIMINARY POINT DESIGN SUMMARIES

This section contains the results of preliminary point design analyses for the three most promising candidate mission/system concept alternatives identified in Section VII.

A mission profile summary, brief system description, weights summary, preliminary program schedule, preliminary costs estimate, and identification of supporting technology requirements are given for each selected concept. For purposes of point design summaries, all systems are based on the 1975 mission opportunity and 1971 technology base.

Throughout this section, reported weights have been rounded off to the nearest pound. In many instances this implies a greater level of accuracy than is justified by the level of detail accomplished during the study. This approach was taken, however, in order to be consistent with the computed weights generated by the system performance computer programs.

8.1 DUAL DEPARTURE ALL-CHEMICAL CONCEPT

This system concept requires gross Earth departure weights of approximately 8500 and 9000 pounds for the lander/ascent probe and orbiter/bus payloads, respectively, based on 1971 technology. These performance requirements can be met by two Titan III D/Centaur launch vehicles, each of which has a capability of 9300 pounds for the 1975 launch opportunity.

8.1.1 Mission Profile

The preliminary mission profile developed for the dual departure all-chemical concept is summarized in Figures 8-1 and 8-2. Figure 8-1 illustrates the lander/ascent probe profile. Figure 8-2 shows the orbiter/bus vehicle profile. An analysis to establish the sequence of arrival at Mars of the two spacecraft was not performed in this study. For preliminary purposes it is assumed that the orbiter/bus vehicle would be launched first for arrival a few days ahead of the lander/ascent probe. This would permit the possibility of the orbiter/bus monitoring the lander/ascent probe entry, descent and landing.

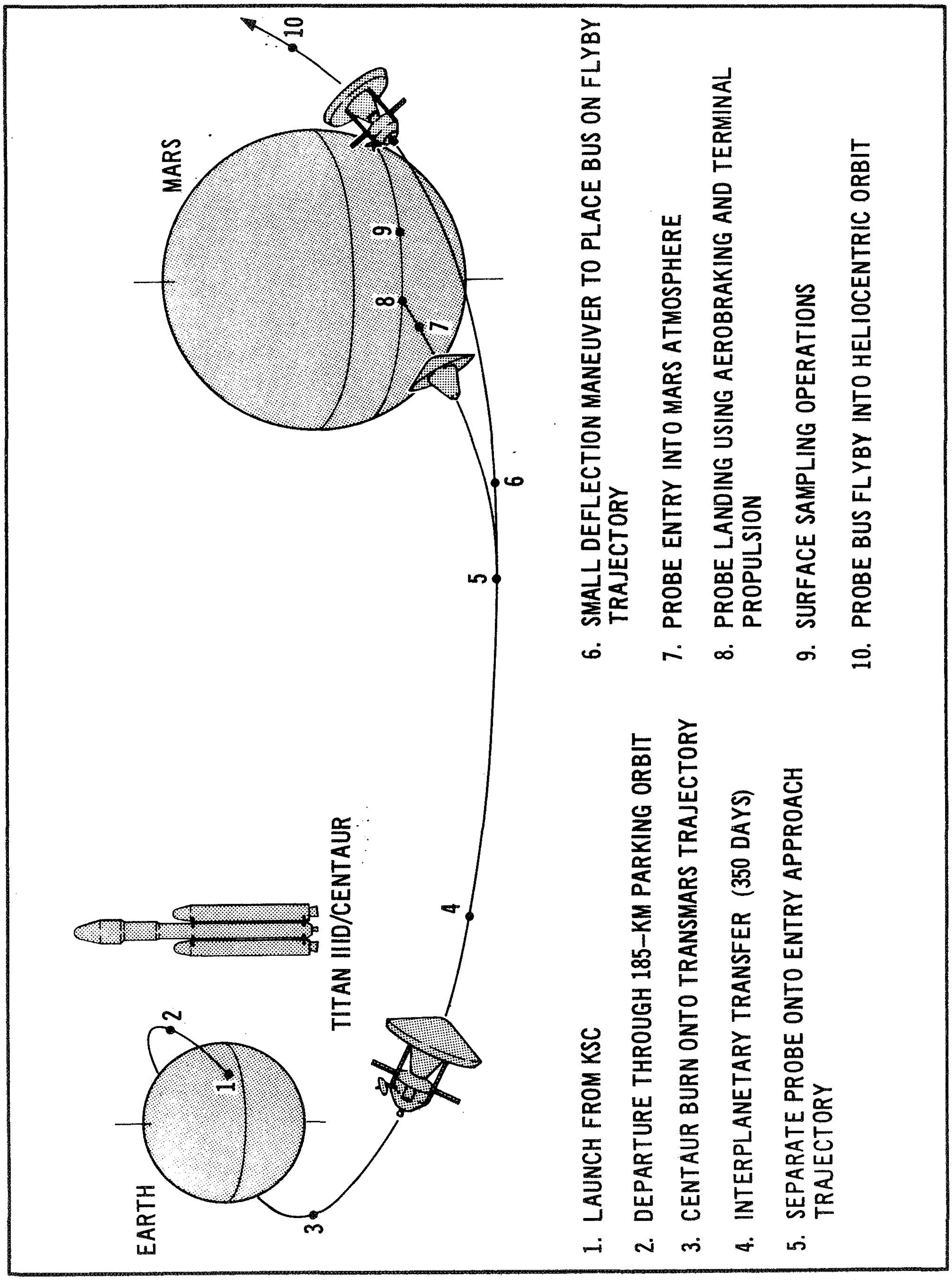
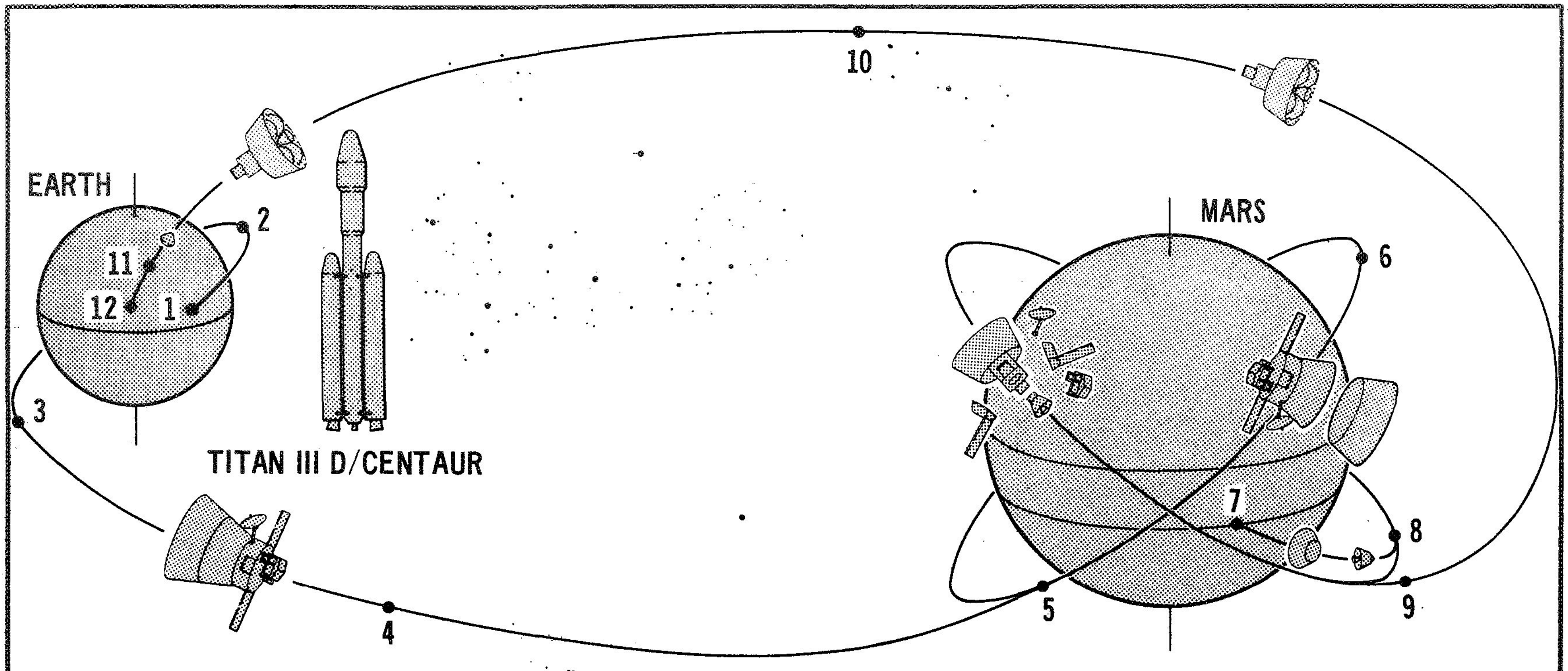


Figure 8-1. DUAL DEPARTURE MSSR CONCEPT - LANDER/ASCENT PROBE PROFILE



1. TITAN IIID/CENTAUR LAUNCH FROM KSC
2. DEPARTURE THROUGH 185-KM PARKING ORBIT
3. CENTAUR BURN ONTO TRANSMARS TRAJECTORY
4. INTERPLANETARY TRANSFER (345 DAYS)
5. CAPTURE INTO 600-KM CIRCULAR ORBIT; JETTISON BRAKING STAGE
6. SUPPORT MARS SURFACE OPERATION
7. MARS ASCENT VEHICLE (MAV) LAUNCH AFTER 30 TO 90 DAYS ON SURFACE
8. RENDEZVOUS/DOCKING WITH ORBITER/BUS; RETRIEVE SAMPLES CANISTER; JETTISON MAV SECOND STAGE AND EXCESS ORBITER EQUIPMENT
9. TRANSEARTH INJECTION AFTER 340-DAY STOPOVER
10. INBOUND INTERPLANETARY TRANSFER (290 DAYS)
11. DIRECT REENTRY OF SAMPLES CAPSULE AFTER EJECTION FROM ORBITER/BUS
12. AIR CATCH RECOVERY (WATER RECOVERY BACKUP)

Figure 8-2. DUAL DEPARTURE MSSR CONCEPT - ORBITER/BUS RETURN VEHICLE MISSION PROFILE

8.1.2 System Description

The dual departure all-chemical system consists of a lander/ascent probe vehicle and an orbiter/bus return vehicle launched as separate payloads by two Titan III D/Centaur vehicles. The following paragraphs briefly describe the characteristics of these systems.

8.1.2.1 Lander/Ascent Probe Vehicle. The lander/ascent probe vehicle consists of an earth launch vehicle adapter, lander/ascent probe bus , probe mounting structure, sterilization canister, and the lander/ascent probe. A preliminary configuration drawing of the overall vehicle is presented in Figure 8-3. The gross Earth departure weight is 8480 pounds. A system weight summary is presented in Table 8-1.

It is noted that the preliminary configuration drawing shows the probe diameter exceeding the current Viking bulbous shroud payload envelope. Because the Viking envelope was considered a relatively "soft" constraint in the present study, iterations in probe sizing were not made to reduce the diameter. If time in the study had permitted, it is believed that iterative entry/descent/landing system tradeoffs could have been exercised to achieve a ten percent reduction in the aeroshell diameter.

Earth Launch Vehicle Adapter

The Earth launch vehicle adapter is the mechanical interface between the Centaur stage and the Mars braking stage of the orbiter/bus return vehicle. The adapter is a tubular truss structure estimated to weigh 291 pounds.

Lander/Ascent Probe Bus

The lander/ascent probe bus is a self-contained, fully independent spacecraft that transports the lander/ascent probe to the vicinity of Mars. The bus consists of a spacecraft module and a propulsion module. A weight summary is presented in Table 8-2.

The spacecraft module receives and executes commands required to place the lander/ascent probe on a direct entry flight path at Mars arrival. The module is essentially a Mariner Mars 1969 spacecraft repackaged in a new

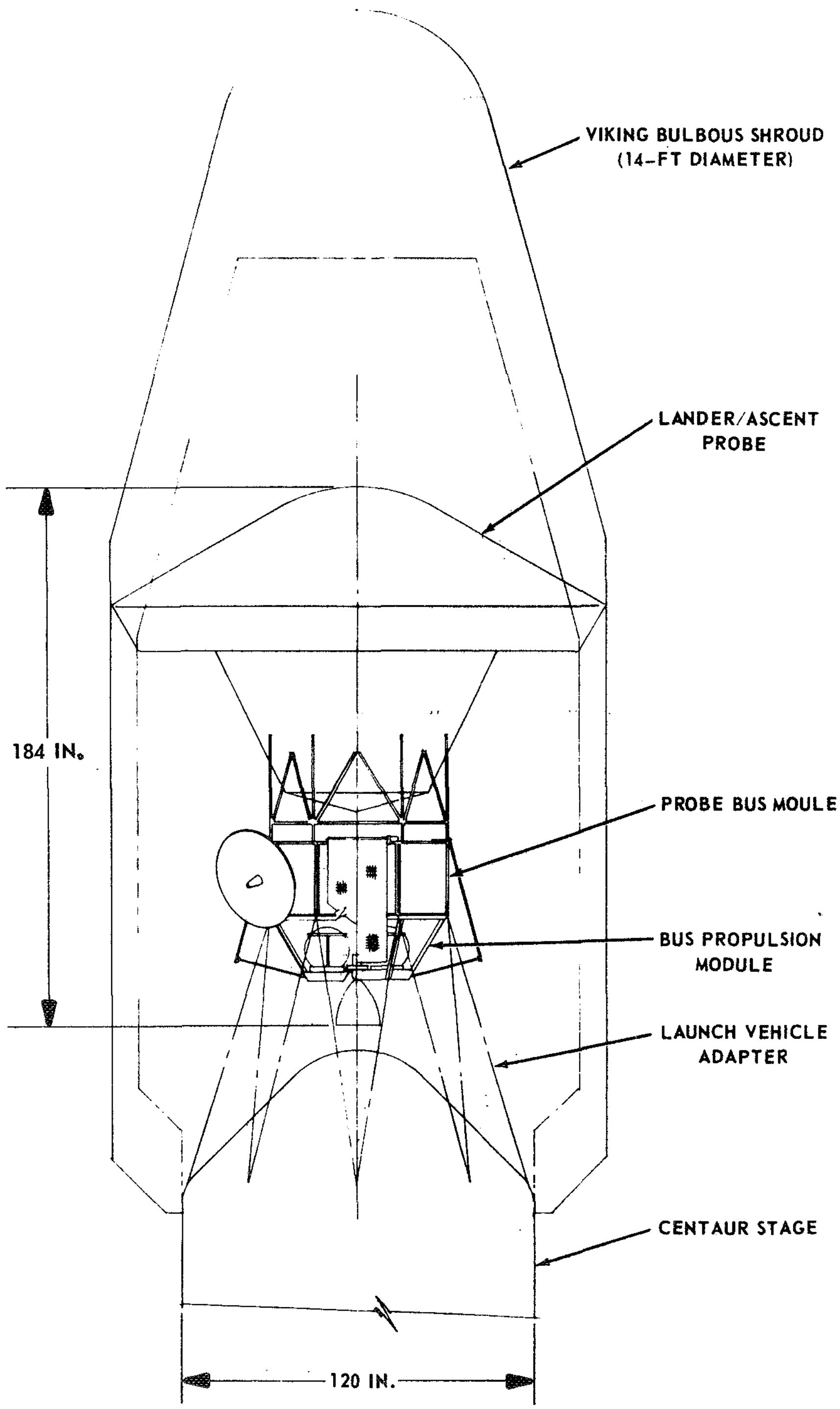


Figure 8-3. DUAL DEPARTURE CONCEPT - LANDER/ASCENT PROBE SYSTEM CONFIGURATION

Table 8-1. LANDER/ASCENT PROBE SYSTEM WEIGHT SUMMARY

LANDER/ASCENT PROBE		6,449 LB
STERILIZATION CANISTER		406
PROBE MOUNTING STRUCTURE		322
LANDER/RETURN PROBE BUS		1,081
SPACECRAFT MODULE	830	
PROPULSION MODULE	181	
TOTAL PLANETARY VEHICLE		8,189
EARTH LAUNCH VEHICLE ADAPTER		291
GROSS EARTH DEPARTURE WEIGHT		8,480 LB

Table 8-2. LANDER/ASCENT PROBE BUS WEIGHT SUMMARY

Spacecraft Module		830 lb
Guidance and Navigation	81	
Transceiver	39	
Antenna	8	
Data Encoder	22	
Video Storage	38	
Television	11	
Command	10	
Computer & Sequencer	11	
Power	150	
Electrical Wiring	56	
Thermal Control	22	
Attitude Control	150	
Structure	94	
Contingency	138	
Propulsion Module		181
Propellants	129	
Residuals	2	
Tankage	6	
Engine	15	
Plumbing	3	
Pressurization	3	
Thermal Control	3	
Structure	14	
Contingency	6	
Total Probe Bus		1011 lb

structure. The Mariner science instruments have been removed and a more sophisticated guidance subsystem added to meet the approach guidance requirements for direct entry at Mars. The spacecraft module has the performance characteristics of the Mariner Mars 1969 spacecraft and a total weight of 830 pounds.

The probe bus propulsion module performs the outbound midcourse correction maneuvers and a deflection maneuver during Mars approach to place the spacecraft onto a flyby trajectory after separation of the lander/ascent probe. The module employs a single 325-pound thrust engine with a specific impulse of 306 seconds and has a total weight of 181 pounds.

Probe Mounting Structure

The probe mounting structure is a 322-pound tubular truss structure that provides mechanical interface between the lander/ascent probe and the probe bus. The mounting structure also supports the sterilization canister.

Sterilization Canister

The sterilization canister or bioshield provides contamination protection to the lander/ascent probe from sterilization at Earth to Mars arrival. The estimated weight of the canister is 401 pounds.

Lander/Ascent Probe

The lander/ascent probe consists of an aerobraking system, lander, rover, and Mars ascent vehicle. A preliminary configuration of the probe is presented in Figure 8-4. The gross weight of the probe at Mars entry is 6450 pounds. A weight summary is presented in Table 8-3.

Table 8-3. LANDER/ASCENT PROBE WEIGHT SUMMARY

Mars Ascent Vehicle	2812 lb	
Rover	150	
Lander	1946	
Gross Landed Weight		4908
Aerobraking System	1542	
Gross Probe Weight at Entry		6450 lb

The aerobraking system consists of the aeroshell, heatshield, attached inflatable decelerator (AID), and attitude control system. The aeroshell is approximately 14.1 feet in diameter based on a ballistic coefficient of 1.0 slug/ft^2 and the drag characteristics associated with the configuration trimmed for a lift-to-drag ratio of 0.3. The AID has a deployed diameter of 35 feet and employs a gas generator for deployment.

A weight summary of the aerobraking system is presented in Table 8-4.

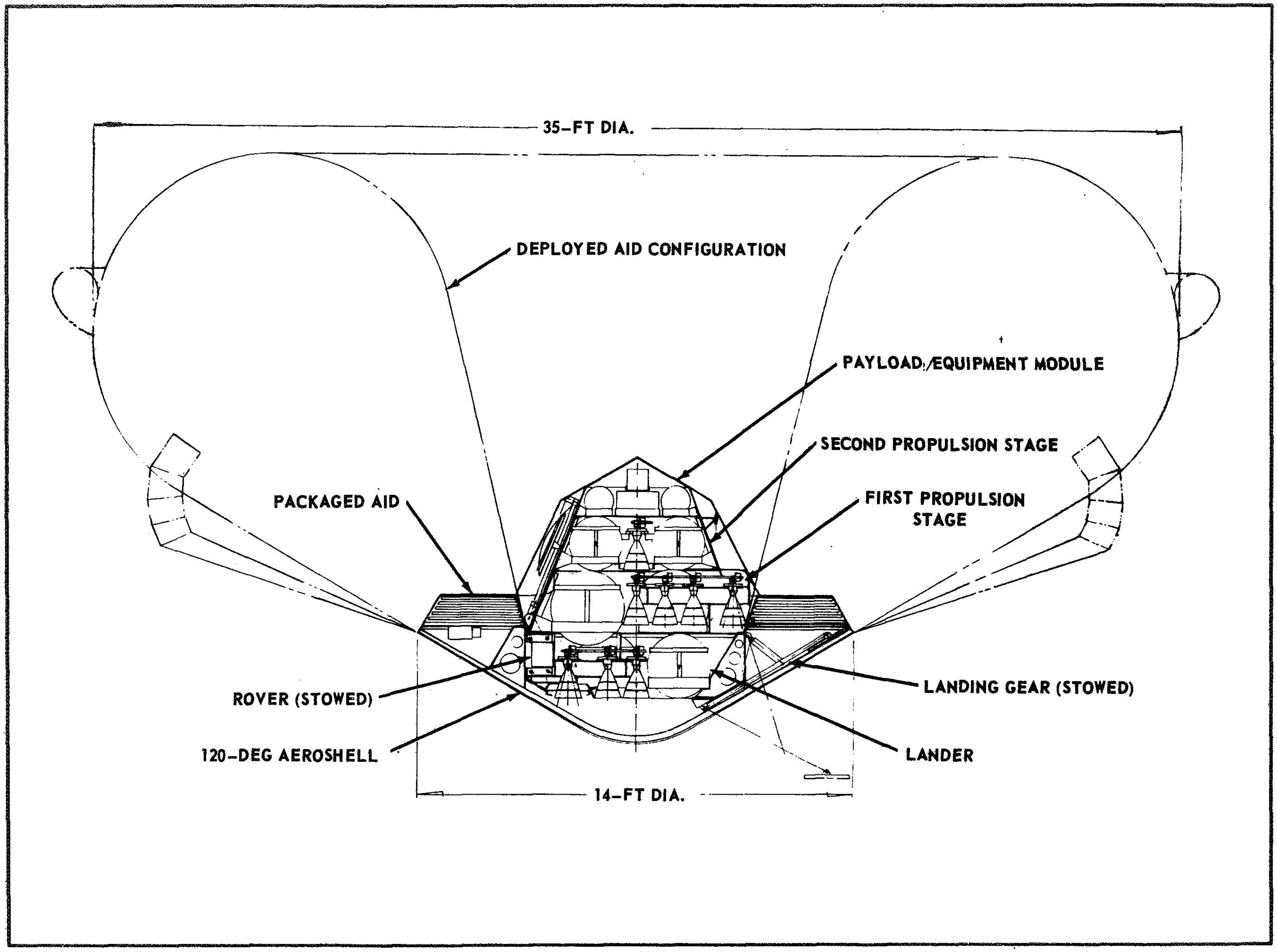


Figure 8-4. LANDER/ASCENT PROBE CONFIGURATION

Table 8-4. AEROBRAKING SYSTEM WEIGHT SUMMARY

Aeroshell	335 lb
Heatshield	132
Attached Inflatable Decelerator (AID)	246
Attitude Control Inerts	74
Attitude Control Propellants	60
Terminal Descent Propellants	397
Contingency	294
Total	1542 lb

Lander

The lander consists of the Viking science instruments, selected Viking subsystems and other subsystems based on existing technology. The subsystems selected from Viking are power, telemetry, guidance and control, communications, command and control, and electrical cabling and harness.

The lander terminal propulsion system employs nine 325-pound thrust engines. The structure, landing gear, thermal control and pyrotechnics weights are based on Viking. A summary of the MSSR lander weights is presented in Table 8-5.

Rover

The rover selected for preliminary purposes is a 150-pound, lander independent vehicle as defined by General Motors and Bendix for lunar application and reported in JPL document, EDP-259 (ref. 16).

Mars Ascent Vehicle

The Mars ascent vehicle (MAV) consists of two propulsion stages, an equipment module and a sample payload module. The second stage employs a single 325-pound thrust engine and has a thrust-to-weight ratio of 0.38. The first stage is designed around a cluster of thirteen 325-pound thrust engines and has a thrust-to-weight ratio of approximately 1.5. Both stages burn N_2O_4/MMH propellants and deliver specific impulses of 300 seconds. A weight summary of the MAV is presented in Table 8-6.

Table 8-5. LANDER WEIGHT SUMMARY

Science	100* lb
Power	212*
Telemetry	67*
Guidance	115*
Communications	95*
Command, Control Sequencer	32*
Cable and Harness	69*
Thermal Control	72
Pyrotechnics	58
Tankage	13
Engines	114
Plumbing	23
Pressurization	7
Residuals	8
Structure	443
Landing Gear	246
Contingency	272
Total Lander Weight	1946 lb

* Viking Lander weights

Table 8-6. MAV WEIGHT SUMMARY

Sample Payload Module		39 lb
Samples	10	
Samples Container	3	
Thermal Control	3	
Docking Adapter	6	
Pyrotechnics	0	
Structure	11	
Contingency	6	
Equipment Module		267
Electrical Power	42	
Cabling	11	
Thermal Control	8	
Docking Transponder	15	
Computer and Sequencer	35	
Guidance and Navigation	37	

Table 8-6. MAV WEIGHT SUMMARY (Concluded)

Transceiver and Antenna	6	
Attitude Control	48	
Structure	41	
Contingency	24	
Second Propulsion Stage		553
Propellants	448	
Residuals	5	
Tankage	14	
Engine	13	
Plumbing	6	
Pressurization	8	
Thermal Control	6	
Structure	40	
Contingency	13	
First Propulsion Stage		1963
Propellants	1472	
Residuals	15	
Tankage	30	
Engine	174	
Plumbing	35	
Pressurization	27	
Thermal Control	14	
Structure	120	
Pyrotechnics and Cabling	30	
Contingency	46	
MAV Total Weight (including 10-pound sample)		2822 lb

8.1.2.2 Orbiter/Bus Return Vehicle. The orbiter/bus return vehicle consists of an earth launch vehicle adapter, Mars braking stage, Mars departure stage, and the orbiter/bus module. The overall vehicle configuration is shown in Figure 8-5. The gross earth departure weight is 8958 pounds. A weight summary is presented in Table 8-7.

Table 8-7. DUAL DEPARTURE CONCEPT ORBITER/BUS SYSTEM WEIGHT SUMMARY

ORBITER/BUS MODULE	1,630 LB
MARS DEPARTURE STAGE	1,711
MARS BRAKING STAGE	5,405
TOTAL PLANETARY VEHICLE	8,746
EARTH LAUNCH VEHICLE ADAPTER	212
GROSS EARTH DEPARTURE WEIGHT	8,958 LB

Earth Launch Vehicle Adapter

The earth launch vehicle adapter is a tubular truss structure that provides the mechanical interface between the Centaur stage and the Mars braking stage. The adapter weight is estimated to be 212 pounds based on 2014-T6 aluminum construction.

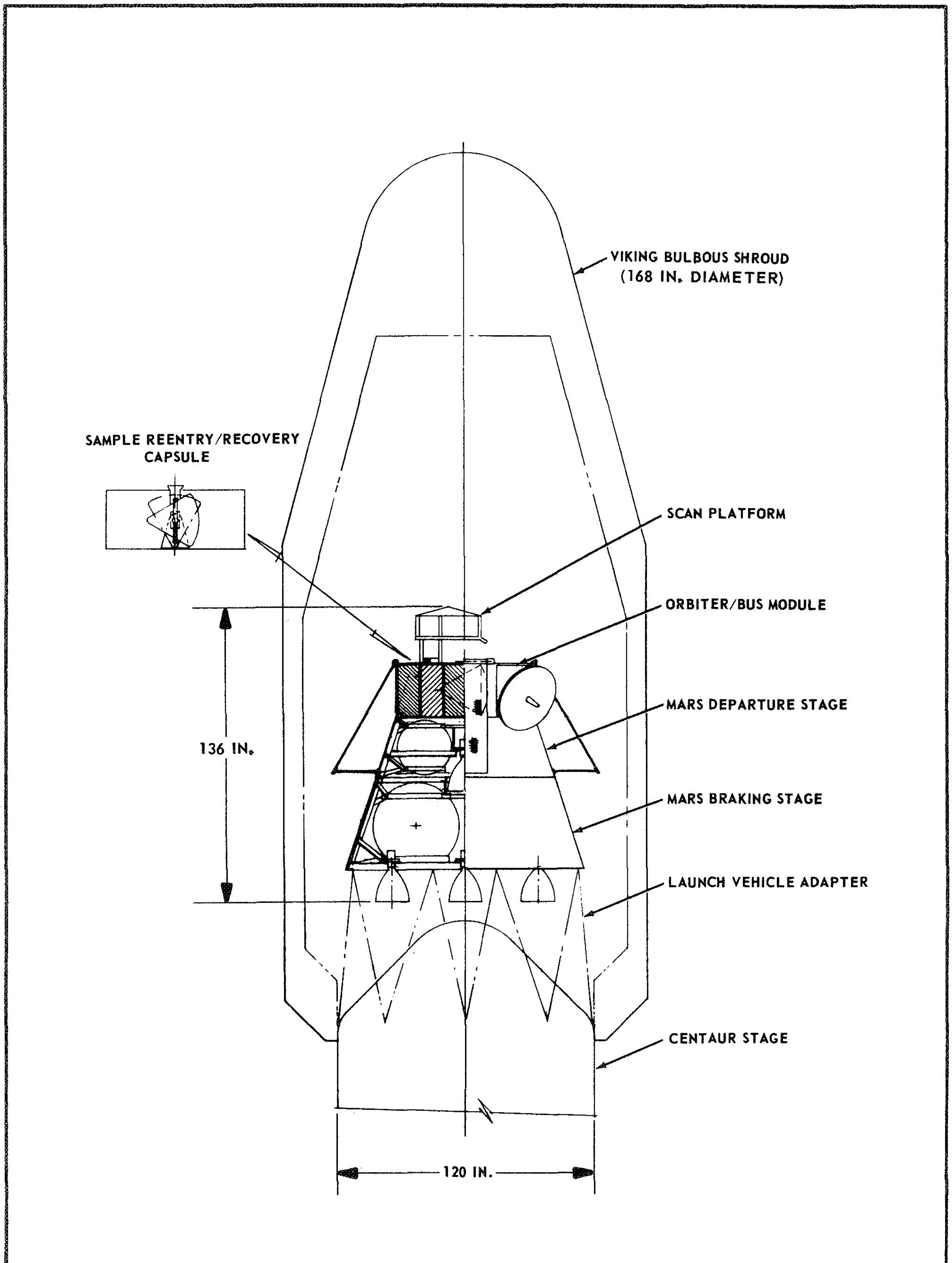


Figure 8-5. DUAL DEPARTURE CONCEPT - CHEMICAL ORBITER/BUS VEHICLE CONFIGURATION

Mars Braking Stage

The Mars braking stage (MBS) performs the outbound midcourse correction maneuvers, the Mars braking maneuver and the orbit trim maneuvers. The MBS contains 4649 pounds of N_2O_4/MMH propellants, 757 pounds of inert systems, and employs five 325-pound thrust engines. The engine characteristics are presented in Table 5-3 of Section V. The MBS also contains the vehicle attitude control system for the outbound leg of the mission. A stage weight summary is presented in Table 8-8.

Table 8-8. MARS BRAKING STAGE

Propellants	4649 lb
Residuals	46
Tankage	83
Engine	91
Plumbing	16
Pressurization	86
Thermal Control	14
Attitude Control	100
Structure	275
Contingency	46
Total Stage Weight	5405 lb

Mars Departure Stage

The Mars departure stage (MDS) performs the Mars orbit rendezvous maneuvers, the Mars departure maneuver, and the inbound midcourse correction maneuvers. The MDS contains 1446 pounds of N_2O_4/MMH propellants, 265 pounds of inert systems, and employs two 325-pound thrust engines. The engine characteristics are identical to those in the Mars braking stage. A weight summary of the MDS is presented in Table 8-9.

Orbiter/Bus Module

The orbiter/bus module consists of the Viking orbiter science, selected Viking subsystems, and other subsystems based on existing technology. The subsystem weights from the Viking orbiter are the communications, power, data storage, computer and command, pyrotechnics, and cabling and harness. The remaining subsystems are based on 1971 technology.

Table 8-9. MARS DEPARTURE STAGE

Propellants	1446 lb
Residuals	14
Tankage	30
Engine	37
Plumbing	10
Pressurization	26
Thermal Control	12
Structure	113
Contingency	23
Total Stage Weight	1711 lb

The orbiter/bus equipment not required for the inbound leg of the mission is jettisoned in Mars orbit. This requires modular packaging of the subsystems.

Weight summaries of both the outbound (Earth-to-Mars) and inbound (Mars-to-Earth) orbiter/bus configurations are presented in Table 8-10.

Table 8-10. ORBITER/BUS EQUIPMENT MODULE WEIGHT SUMMARY

ITEM	OUTBOUND	INBOUND
Science	125 lb	0 lb
Communications	122	36
Power	288	144
Data Storage	94	0
Computer and Command	24	24
Pyrotechnics	31	11
Cabling and Harness	120	60
Temperature Control	37	29
Mechanical Devices	112	56
Attitude Control Inerts	85	85
Attitude Control Propellants	100	40
Docking Adapter	45	10
Rendezvous Radar	24	0
Structure	230	184
Redundancy	35	17
Contingency	72	34
Earth reentry capsule	85	85
Total Weight	1630 lb	815 lb

Earth Reentry Capsule

The earth reentry capsule is an Apollo Command Module shape vehicle approximately 2 feet in diameter. The capsule is spin-stabilized for reentry and de-spun prior to parachute deployment. A weight summary of the capsule is presented in Table 8-11.

Table 8-11. EARTH REENTRY CAPSULE WEIGHT SUMMARY

Samples	10 lb
Samples Container	3
Battery and Cabling	12
Marker Beacon	3
Spin/De-Spin Motors	4
Entry Heat Shield	14
Parachute System	10
Marker Dye	1
Flotation	5
Sequences	5
Structure	20
Contingency	11
Total Capsule Weight	98 lb

8.1.3 Preliminary Program Schedule

The preliminary program schedule prepared for the dual departure all-chemical system is presented in Figure 8-6. This schedule requires a January 1971 Phase A study initiation and a program commitment by January 1972 in order to meet the September 1975 mission launch opportunity.

The system development schedule for the 1975 mission appears to be feasible, but requires a very intensive system definition effort during CY 1971.

For comparison purposes, preliminary schedules were prepared for the 1977 and 1979 mission opportunities. Figure 8-7 gives the system development schedule for the 1977 launch opportunity. The overall Phase C and D time schedule in the program is relatively relaxed and spans about five years from initiation of Phase C. The Phase A study is initiated in CY 1971 as in the 1975 mission schedule.

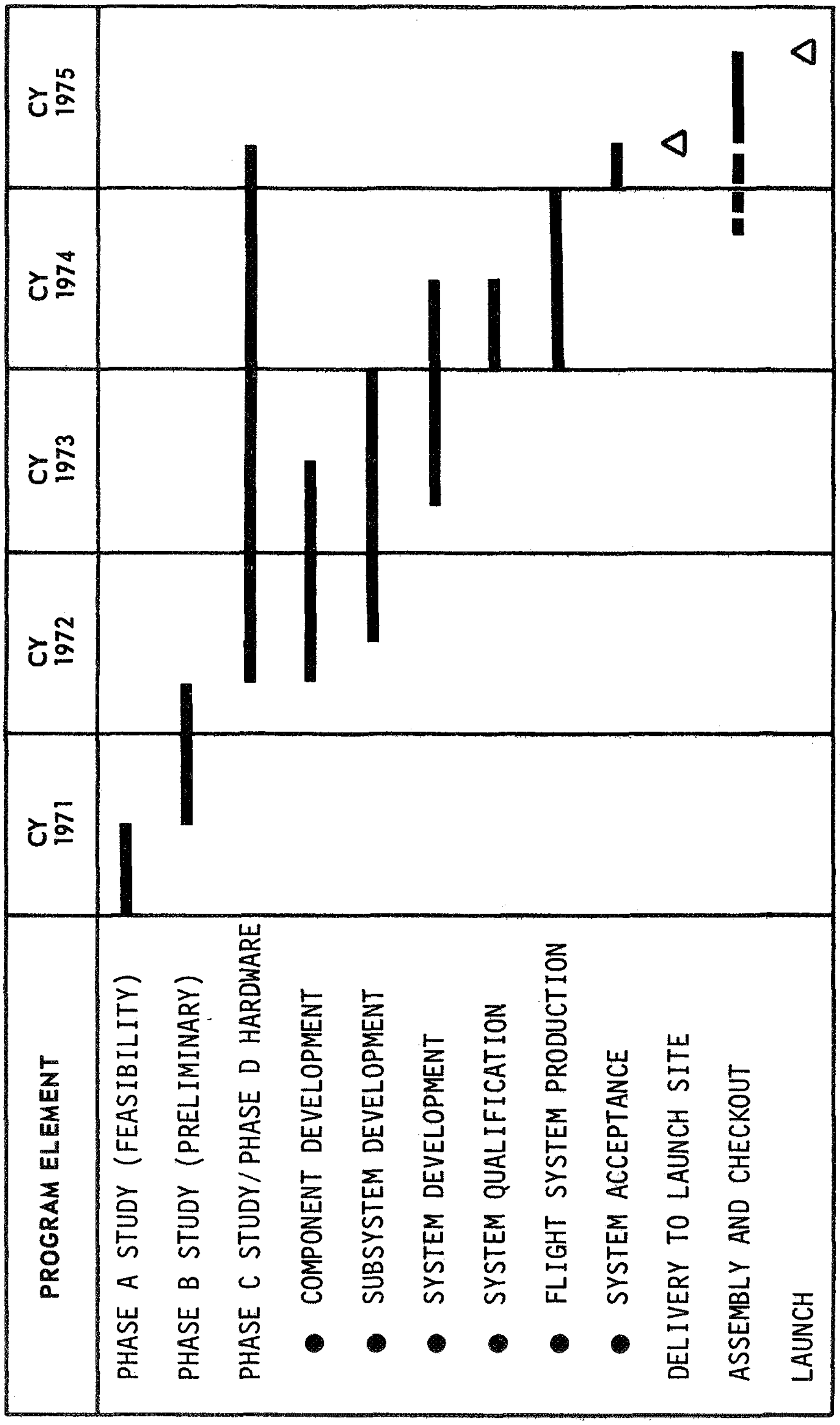


Figure 8-6. MSSR SYSTEM DEVELOPMENT SCHEDULE FOR 1975 LAUNCH OPPORTUNITY

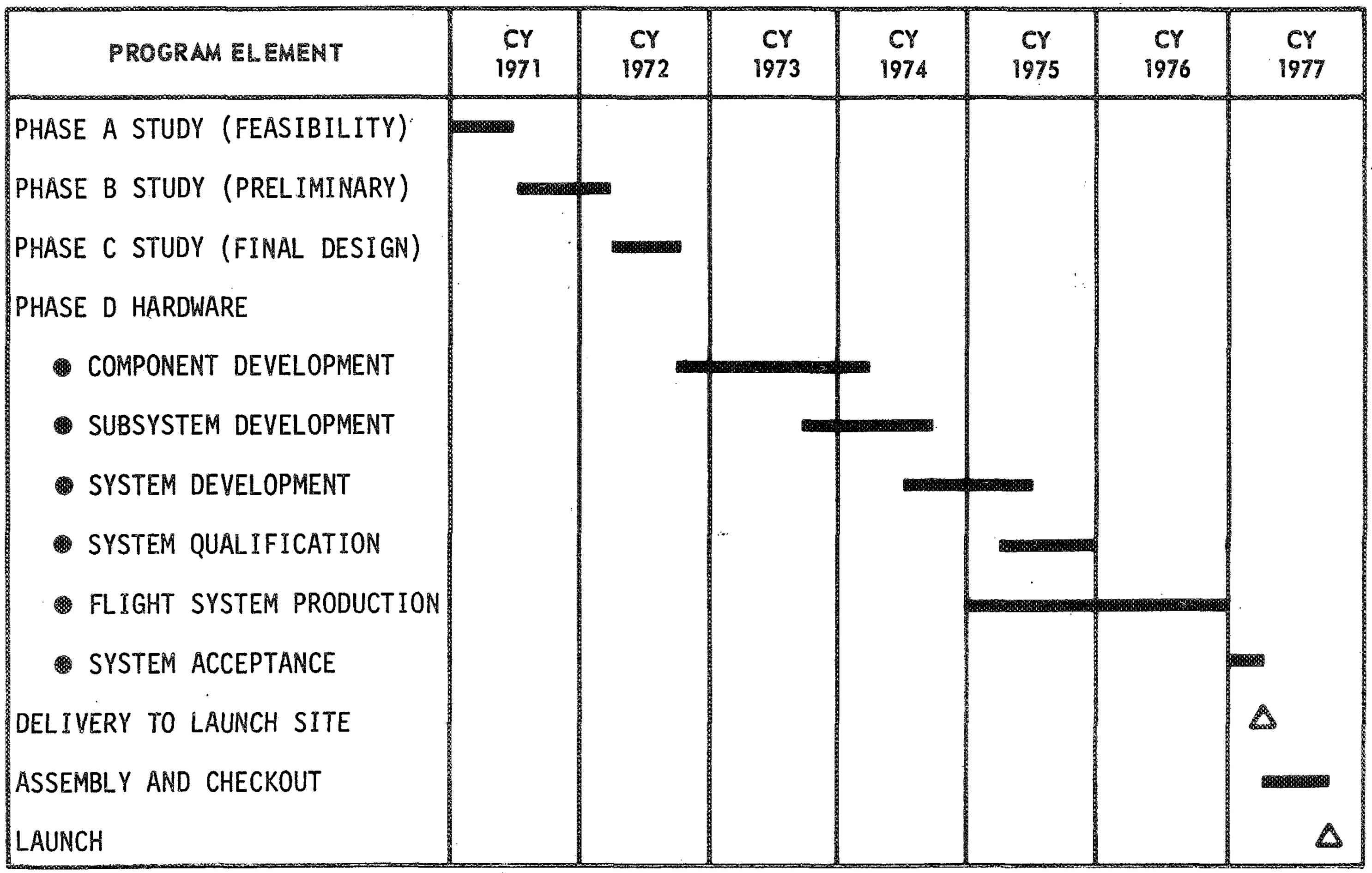


Figure 8-7. SYSTEM DEVELOPMENT SCHEDULE FOR 1977 LAUNCH OPPORTUNITY

Figure 8-8 shows a preliminary schedule keyed on the 1979 launch opportunity. This schedule is identical to the 1977 case except the Phase A start is slipped by two years to CY 1973.

8.1.4 Preliminary Cost Estimate

A preliminary cost estimate was made for the dual departure all-chemical MSSR concept based on the cost data developed under Contract NAS8-24714. The estimate includes launch vehicle costs but does not include recovery costs. The estimate reflects development costs for all subsystems available from the Mariner and Viking programs. The development cost of the propulsion subsystems was reduced because the major development work has been completed on the selected engine used throughout the system.

The production cost estimate includes two flight systems, but only one pair of Earth launch vehicles as required by a single mission.

Table 8-12 summarizes the costs by major cost element. The total program less recovery cost is estimated to be $891 million in 1970 dollars.

Table 8-12. PRELIMINARY COST ESTIMATE FOR DUAL DEPARTURE, ALL-CHEMICAL MISSION/SYSTEM CONCEPT

COST ELEMENT	COST (MILLIONS OF DOLLARS)
Design and Development	615
Production	208
Operations	28
Launch Vehicles	40
Total Cost*	891

* Recovery costs not included.

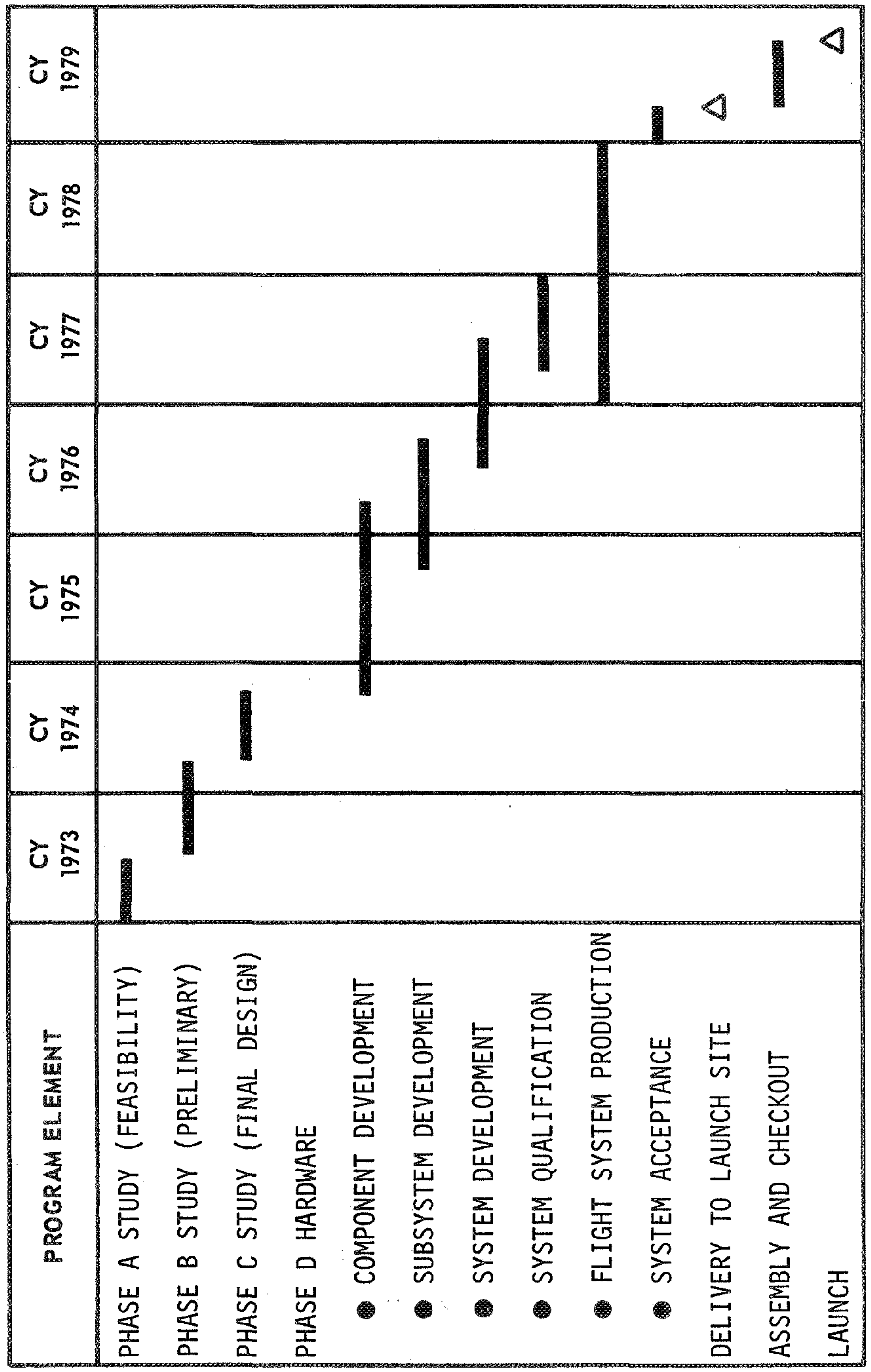


Figure 8-8. SYSTEM DEVELOPMENT SCHEDULE FOR 1979 LAUNCH OPPORTUNITY

8.1.5 Supporting Technology Requirements

The items decribed below are either supporting technology requirements or development items of such magnitude that they deserve special attention prior to a program go-ahead decision. The technology requirements for both the 1975 and the 1977/1979 launch opportunities are described in the following paragraphs. The requirement for the later missions are summarized first.

8.1.5.1 Supporting Technology Requirements for 1977/1979 Launch Opportunities. For these mission opportunities it was assumed that the system could take advantage of technological advances through January 1974.

Engine Development

The design of the baseline MSSR system requires the use of high-pressure, low-flow engines with burn times of 600 to 1000 seconds and thrust levels of 500 to 5000 pounds. The development of one engine within this thrust range will demonstrate the technology. A 1000-pound thrust engine is considered representative and is recommended for development.

The objective of this SRT activity is to perform those tasks required to develop the engine and demonstrate and engine reliability of 0.9. This requires that a total of approximately 845 engine tests be performed. This activity also includes flight qualification.

Propellant Sterilization

The use of sterilizable propellants below 1000 km altitude at Mars was a NASA-imposed groundrule for the MSSR study under Contract NAS8-24714. Although this constraint was relaxed in the present study, sterilization of all lander/ascent probe propellants is required. The guideline for sterilization is to ensure a probability of contamination of the planetary body no greater than 10^{-5}. Current program requirements specify sterilization of the spacecraft (including propellants) before launch by exposure to a terminal dry heat of 135°C for approximately 24 hours. This procedure severely limits the choice of fuels and oxidizers for application to the lander/ascent probe system.

The MSSR baseline system contains three propulsion modules that are subject to the sterilization requirements. These are the deflection (or deorbit) and lander propulsion systems, the Mars ascent vehicle first stage propulsion system, and the Mars ascent vehicle second stage propulsion system, and the Mars ascent vehicle second stage system.

The propellant combination selected for all of these applications was N_2O_4/MMH. The realized Isp of the propellant was reduced from 320 seconds to 315 seconds to account for the effect of sterilization.

The object of this SRT activity is to determine the effects of sterilization on the delivered Isp of N_2O_4/MMH. Because of the improved performance available with space storable propellants, this activity should also determine the feasibility of sterilizing propellant combinations such as FLOX/CH_4 and determine the performance of the sterilized propellant.

Propellant Boiloff Losses

One key parameter influencing the selection of propellants for the various propulsion modules was the percent of propellants lost due to boiloff. The effect of boiloff in the MSSR system analysis was to force the use of Earth storable propellants (Isp ≈ 320 seconds) for the Mars departure stage and the Earth braking stage (in capture/recovery concepts) rather than the higher performance space storable propellants (Isp ≈ 400 seconds). The cause of boiloff as a function of tank volume is illustrated in Figure 8-9. The figure shows the strong dependence on tank support design for tank sizes typical of the MSSR system (10 to 100 ft^3).

The boiloff data used during the MSSR study to select propellants for the various modules were based on extrapolation of test data provided by NASA/MSFC.

The objective of this SRT activity is two-fold: first to accurately determine the boiloff characteristics of a typical space storable propellant stored in a spherical tank nominally four feet in diameter; and second to investigate better methods for supporting the tank to reduce boiloff.

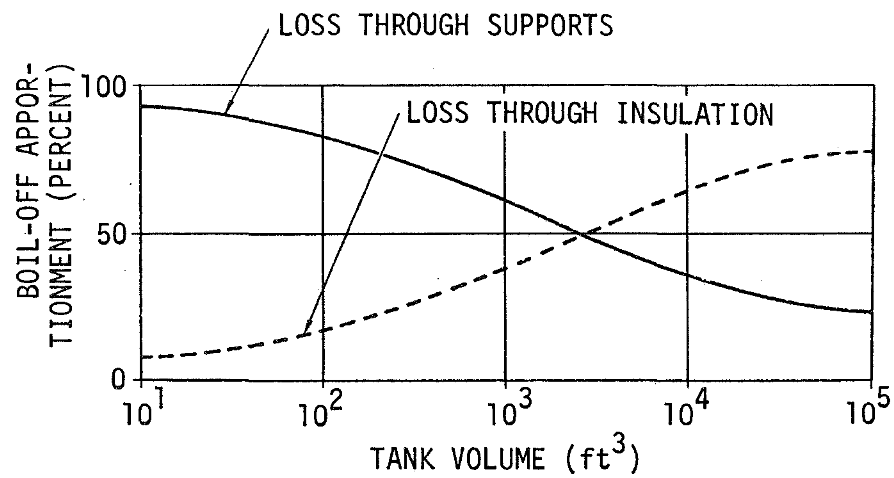


Figure 8-9. CAUSE OF BOILOFF LOSES FOR TYPICAL PROPULSION STAGE

Attached Inflatable Decelerator

Based on the analysis conducted for entry/descent/landing at Mars, the deceleration system utilizes an aeroshell for the high-heating phase, an attached inflatable decelerator (AID) to achieve hypersonic/supersonic deceleration to subsonic conditions, and propulsive terminal deceleration.

The baseline MSSR system requires a 35-foot diameter AID. The AID is deployed at Mach 4 conditions after the entry heating phase terminates. AID development to date has been confined to smaller devices deployed at lower Mach numbers than required for the baseline MSSR system. The current state of AID development is summarized in Table 8-13.

The objective of this SRT activity is to verify the development and stability of large-diameter AID's over a wide range of velocities and dynamic pressures. Aerodynamic and structural design data need to be compiled to provide for optimizing the AID for varying payload weights and entry conditions to accommodate design of the baseline MSSR system.

Electrical Power

The electrical power requirements for the 1974 technology MSSR baseline system are summarized in Table 8-14. Radioisotope Thermoelectric Generators

Table 8-13. ATTACHED INFLATABLE DECELERATOR (AID OR BALLUTE) TECHNOLOGY STATUS

DEVELOPMENT PROGRAM	AID OR BALLUTE DIA (FT)	DEPLOYMENT VELOCITY	RESULTS
NASA LANGLEY RESEARCH CENTER	16.0	LOW SUBSONIC	DETERMINED AERODYNAMIC CHARACTERISTICS; STABILITY; FABRICATION; AND PACKAGEABILITY
	36.0	LOW SUBSONIC	DETERMINED AERODYNAMIC CHARACTERISTICS; STABILITY; FABRICATION; AND PACKAGEABILITY
	5.0	HIGH SUBSONIC	DETERMINED AERODYNAMIC CHARACTERISTICS; STABILITY; VERIFIED NEED FOR BURBLE FENCE; GUST LOADS EFFECTS; DESIGN VERIFICATION
	5.0	MACH 2.2 & 3.0	DETERMINED DEPLOYMENT CHARACTERISTICS, AERODYNAMIC, AND STRUCTURAL DATA OVER A RANGE OF SUPERSONIC SPEEDS. VERIFIED THEORETICAL ANALYSIS WITH TEST DATA

Table 8-14. MSSR ELECTRICAL POWER REQUIREMENTS

MODULE	POWER (Watts)	OPERATING TIMES (Days)	POWER SOURCE
Orbiter/Bus	600	1000	RTG
Lander	200	335	RTG
Rover	200	335	RTG
Mars Ascent Vehicle	690*	1	Ni-Cd Battery

**Watt-hours*

(RTG) were selected for all modules except the Mars ascent vehicle, which uses batteries because of the short (one day) operating time requirement. The RTG's selected for the MSSR system are fueled with Plutonium-238 and nominally produce 2 watts per pound. Current development activities indicate the feasibility of RTG's capable of producing 3 watts per pound.

The General Electric Company is presently developing an RTG for the AEC which is expected to deliver 2.5 watts per pound. This unit is scheduled for delivery to the AEC during CY 1970.

The objective of this SRT activity is to develop a modular 100-watt RTG capable of producing 2.5 to 3 watts per pound. These units should be capable of being combined to meet the 200-watt and 600-watt power requirements of the MSSR system.

Mars Surface Model

The general characteristics of the Mars surface are described in the literature; however, the open literature does not reflect in detail the findings of Mariner '69.

The objective of this SRT activity is to update existing models of the Mars surface with the latest Mariner data and Mariner '71 data when available. The surface models should be prepared with both scientists and engineer users in mind. The model should aid the scientist in the selection of potentially interesting MSSR landing sites. It should also provide engineering data required for design of the lander/return probe(s) including the rover vehicle(s).

Mars Atmosphere Model

Several models of the Mars atmosphere have been developed. The VM-8 atmosphere was used in the MSSR study as required by NASA/MSFC. Preliminary analysis of the Mariner '69 data show some inconsistencies between the VM-8 model and the latest Mariner data. The design of the lander/return probe aeroshell, attached inflatable decelerator and terminal propulsion system are dependent on the characteristics of the model atmosphere.

The objective of this SRT activity is to update existing Mars atmosphere models using the most recent Mariner data.

Meteoroid Environment Model

The MSSR baseline system requires a significant amount of meteoroid shielding in the form of increased skin thickness or meteoroid bumpers based on existing meteoroid environmental data.

The objective of this SRT activity is to provide more accurate meteoroid environmental data both in the vicinity of Mars and between Earth and Mars.

Mars Ascent Vehicle Guidance

Mars ascent vehicle (MAV) guidance presents several problems beyond those previously considered for Mars missions. These problem areas include hardware and software mechanization, and computer initialization. The hardware selected for the MAV guidance is basically the strap-down system selected for the Air Force PRIME lifting reentry development and test program, and consists of an inertial sensor assembly, power supplies, and a computer.

The objective of this SRT activity is to determine the optimum hardware mechanization, develop the guidance software, and determine the best method of initializing and guidance computer.

Rendezvous and Docking

The remote rendezvous and docking of the orbiter/bus and the Mars ascent vehicle (MAV) pose both software and hardware SRT requirements. The present scheme is for the orbiter/bus to be the active element in the terminal phase rendezvous and the docking maneuver. The orbiter/bus contains a rendezvous radar and optical sensors for the terminal docking maneuver. The MAV contains a rendezvous radar transponder and light sources for docking.

The objective of this SRT activity is to simulate the automated rendezvous and docking maneuver between the orbiter/bus and the MAV. The simulation should result in performance specifications for the rendezvous and docking hardware for both the orbiter/bus and the MAV.

Lander-Independent Rover

The ability to sample out to several hundred meters radius around the stationary lander is desirable to enhance the possibility of acquiring a variety of sample types outside the area of possible contamination by the lander. A small, minimum weight system is particularly required in concepts employing the Titan IIID/Centaur launch vehicles.

The objective of this activity is the development and demonstration of a 150- to 200-pound class lander-independent rover for Mars surface sampling around the MSSR landing site.

Sample Acquisition and Handling

Surface Sample Acquisition. Techniques and equipment for the acquisition of surface samples are required based on the latest definition of the Mars surface.

The objective of this activity is the development and test of planetary surface sample acquisition techniques and associated equipment.

Subsurface Sample Acquisition. A core sample at the MSSR landing site is highly desirable. A lightweight, automated core drill is required which can interface with the sample transport/loading requirements.

The objective of this activity would be the development and demonstration of a 1 to 10 meter automated drill for core sampling the Mars surface.

Sample Transport/Loading System. Use of a rover to extend the MSSR sampling radius requires the development of a system for transport and loading of the samples from the rover into the Mars ascent vehicle (MAV) payload compartment. This system should also be capable of loading samples acquired by the stationary lander.

The objective of this activity is the development and demonstration of a rover-to-MAV and lander-to-MAV sample transport/loading system.

Sample Storage. Techniques and equipment are required to provide adequate canning and safe storage of the acquired Mars samples. Canning equipment or devices and canisters must be developed to interface with the sample acquisition/transport/loading systems.

The objective of this activity is the development of planetary sample canning methods and hardware.

Earth Reentry/Recovery Capsule

The direct reentry/recovery mode offers significant performance advantages in the MSSR mission and is required for concepts based on all-chemical propulsion and use of Titan IIID/Centaur class launch vehicles. The feasibility of fail-safe recovery of samples with no significant back contamination risk must be demonstrated prior to commitment to the direct reentry approach.

The object of this activity is the development and demonstration of a fail-safe direct reentry/recovery capsule system for return of extraterrestrial samples.

8.1.5.2 Support Technology Requirements for 1975 Launch Opportunity Mission. The supporting technology requirements for the 1975 mission opportunity are similar to those presented in subsection 8.1.5.1 for the 1977/1979 missions. The major differences occur as a result of using existing subsystems for the 1975 missions.

Chemical engine development and radioisotope thermoelectric generator (RTG) development are not required for this mission. The use of the Bell 8570 engine throughout the system negates the requirement for an engine development program; however, the problems that may arise due to clustering radiation-cooled engines should be analyzed early in the feasibility study. The requirement for the advanced RTG does not exist because the Viking orbiter and lander power systems can be used with minimum modification.

The remainder of the technology requirements described in subsection 8.1.5.1 are common to all launch opportunities; therefore, they apply to the 1975 mission opportunity.

8.2 DUAL DEPARTURE SOLAR-ELECTRIC/CHEMICAL CONCEPT

This mission/system approach is similar to the dual departure all-chemical concept except that the orbiter/bus vehicle employs solar-electric/chemical primary propulsion and provides capability for the orbital capture/recovery mode at Earth return.

The gross Earth departure weight requirements are approximately 8500 and 9300 pounds for the lander/ascent probe and orbiter/bus payloads, respectively. The Titan III D/Centaur has a 9300-pound capability for the probe launch and an 11,900-pound capability for the orbiter/bus vehicle launch. The increased weight capability of the Titan III D/Centaur for the orbiter/bus payload is due to the reduced departure energy requirement for the solar-electric mission profile.

8.2.1 Mission Profile

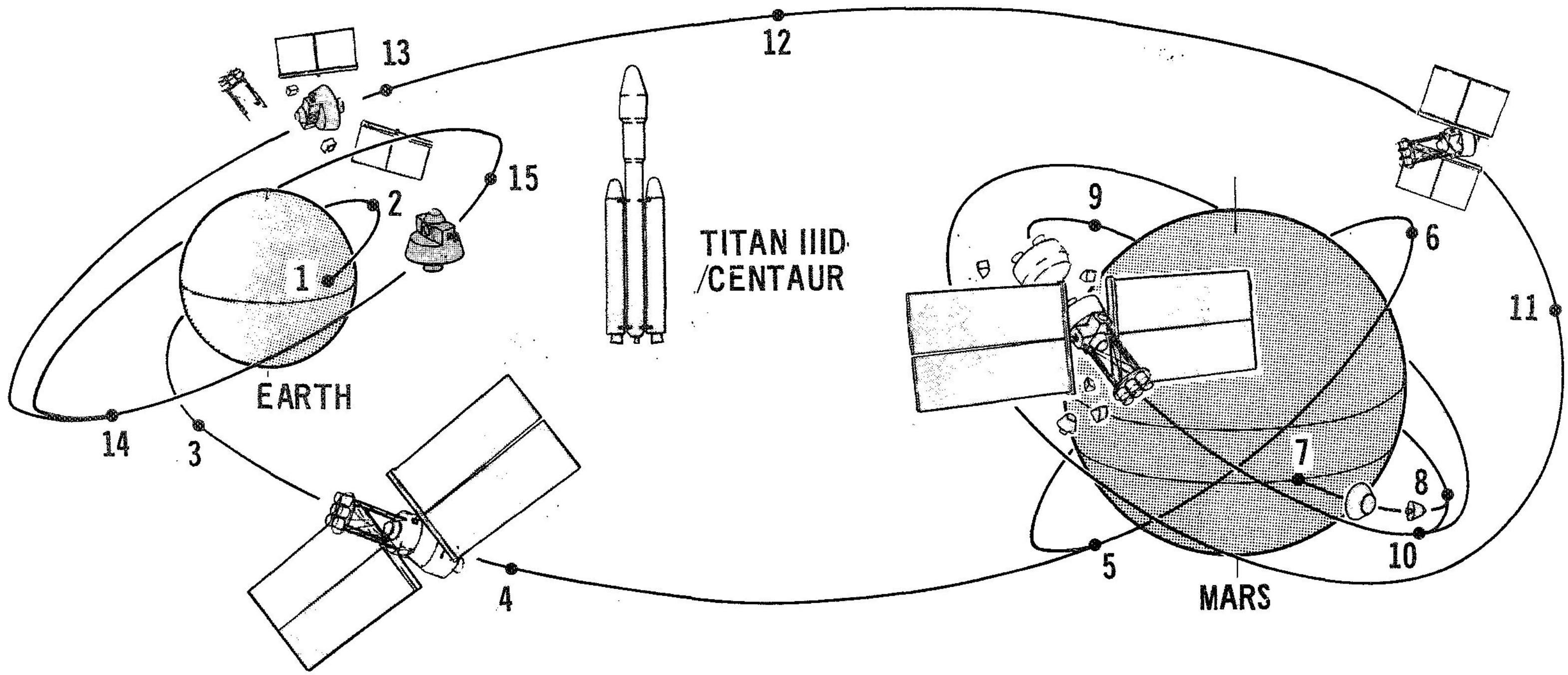
The mission profile for the lander/ascent vehicle is identical to that for the dual departure all-chemical probe as presented in Figure 8-1 (subsection 8.1). The mission profile for the solar-electric/chemical orbiter/bus vehicle is summarized in Figure 8-10.

8.2.2 System Description

The system consists of an all-chemical lander/ascent probe vehicle and a solar-electric/chemical orbiter/bus return vehicle.

8.2.2.1 <u>Lander/Ascent Probe</u>. The lander/ascent probe vehicle for this concept is identical to the one previously described for the dual departure, all-chemical concept in subsection 8.1.2.1.

8.2.2.2 <u>Orbiter/Bus Return Vehicle</u>. The orbiter/bus return vehicle consists of an Earth launch vehicle adapter, Mars braking stage, solar-electric propulsion system, Earth braking stage, and orbiter/bus module. A preliminary vehicle configuration is presented in Figure 8-11. The gross Earth departure weight of the vehicle is 9,286 pounds. A weight summary is presented in Table 8-15.



1. TITAN IIID/CENTAUR LAUNCH FROM KSC
2. DEPARTURE THROUGH 185–KM PARKING ORBIT
3. CENTAUR BURN ONTO ESCAPE TRAJECTORY
4. LOW–THRUST INTERPLANETARY TRANSFER (350 DAYS)
5. CAPTURE INTO 1000–KM CIRCULAR ORBIT USING CHEMICAL PROPULSION
6. ORBITAL SUPPORT OF MARS SURFACE OPERATIONS
7. MARS ASCENT VEHICLE (MAV) LAUNCH AFTER 30 DAYS ON SURFACE
8. RENDEZVOUS/DOCKING WITH ORBITER/BUS; RETRIEVE SAMPLES CANISTER
9. JETTISON MARS CAPTURE/RENDEZVOUS STAGE, MAV SECOND STAGE AND EXCESS ORBITER EQUIPMENT
10. INITIATE SPIRAL–OUT ESCAPE WITH SOLAR–ELECTRIC PROPULSION (SEP)
11. ACHIEVE MARS ESCAPE AFTER 150–DAY SPIRAL–OUT; JETTISON EXCESS SEP HARDWARE
12. INBOUND SEP INTERPLANETARY TRANSFER (500 DAYS)
13. JETTISON SEP HARDWARE DURING EARTH APPROACH
14. CAPTURE INTO 555X 9100–KM ELLIPTICAL ORBIT
15. RECOVER SAMPLES AND FILM IN ORBIT

Figure 8-10. SOLAR-ELECTRIC/CHEMICAL ORBITER/BUS MISSION PROFILE

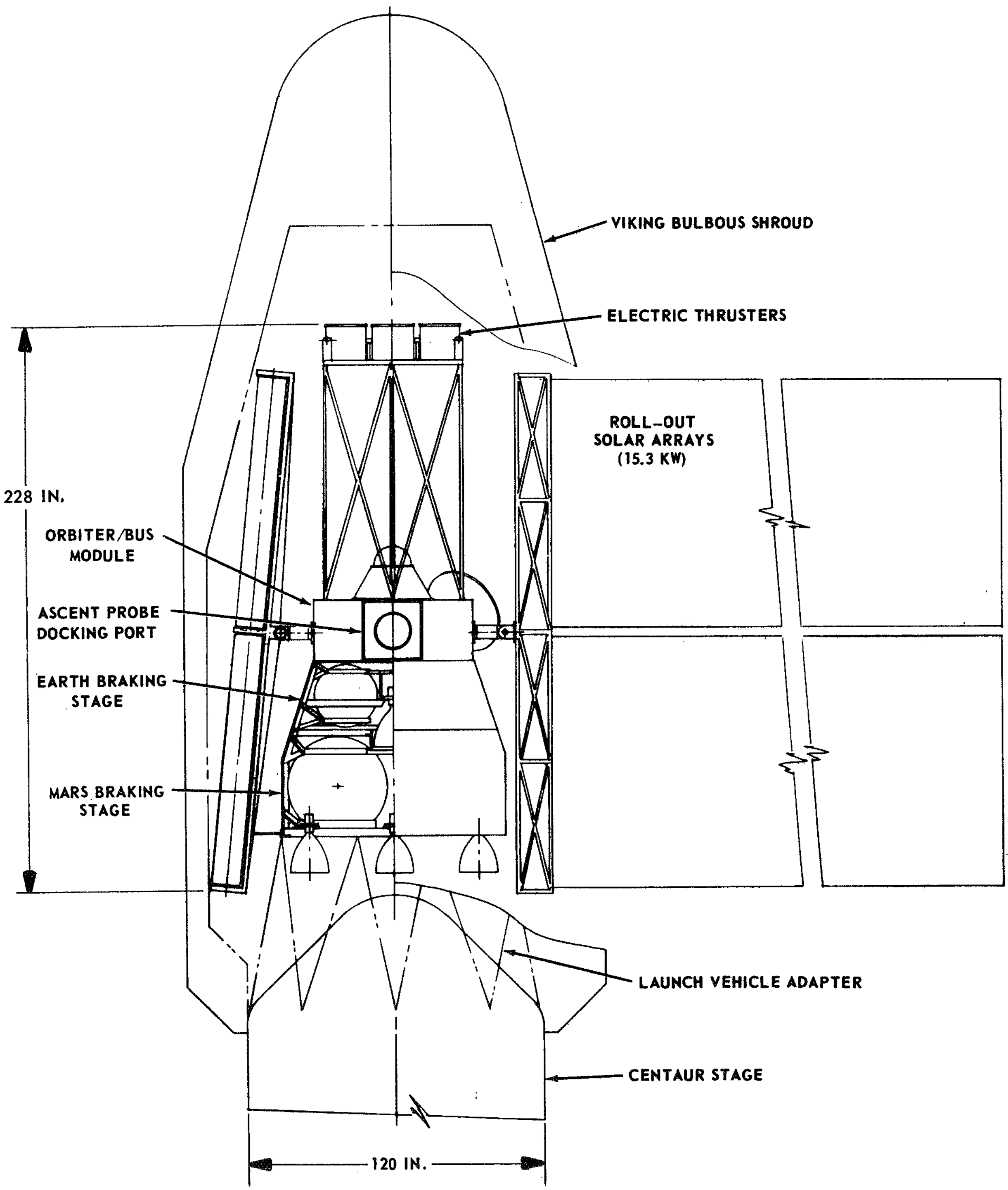


Figure 8-11. DUAL DEPARTURE, SOLAR-ELECTRIC/CHEMICAL ORBITER/BUS VEHICLE CONFIGURATION

Table 8-15. SOLAR-ELECTRIC ORBITER/BUS SYSTEM WEIGHT SUMMARY

ORBITER/BUS MODULE		1,466 LB
SOLAR-ELECTRIC PROPULSION*		2,467
SEP POWER PLANT	928	
SEP PROPELLANT	1,539	
EARTH BRAKING STAGE		1,094
MARS BRAKING STAGE		3,939
TOTAL PLANETARY VEHICLE		8,966
EARTH LAUNCH VEHICLE ADAPTER		320
GROSS EARTH DEPARTURE WEIGHT		9,286 LB

* 15.3 KW POWER AT 1 AU

Earth Launch Vehicle Adapter

The Earth launch vehicle adapter is the mechanical interface between the Centaur stage and Mars braking stage of the orbiter/bus return vehicle. The adapter is a 320-pound tubular truss structure.

Mars Braking Stage

The chemical Mars braking stage (MBS) performs the Mars braking maneuver, orbit trim maneuvers, and the Mars orbit rendezvous maneuvers. The MBS contains a total of 3310 pounds of N_2O_4/MMH propellants in four spherical tanks. The thrust is provided by a cluster of three 325-pound thrust engines. The engine characteristics are presented in Table 5-3 (Section V). The MBS also contains the attitude control system used during the outbound leg of the mission. A stage weight summary is given in Table 8-16.

Earth Braking Stage

The Earth braking stage (EBS) performs the inbound midcourse correction maneuvers and the Earth capture maneuver. The EBS contains 919 pounds of N_2O_4/Aerozine-50 propellants, and employs a single 325-pound thrust engine identical to the MBS engines. A weight summary is presented in Table 8-17.

Solar-Electric Propulsion System

The solar-electric propulsion (SEP) system thrusts continuously during the outbound leg of the mission, during the spiral-out departure maneuver at Mars, and during the inbound leg from Mars to Earth. The SEP system consists of the solar arrays, power conditioning units, ion thrusters, and mercury propellant.

The total area of the solar arrays is approximately 1600 square feet. This includes a 15 percent allowance for radiation degradation of the solar cells and provides an initial output of 15.3 kw at 1 AU. Roll-out type arrays were selected based on current technology with a specific weight of 33 pounds per kilowatt (38 lb/kw including array degradation effects of radiation).

The SEP thrusters were sized at 2.5 kw per thruster based on current technology. The specific weight of the thrusters and power conditioning equipment is 20 pounds per kilowatt.

Table 8-16. MARS BRAKING STAGE WEIGHT SUMMARY

Propellants	3,310 lb
Residuals	33
Tankage	59
Engine	40
Plumbing	8
Pressurization	59
Thermal Control	13
Attitude Control	85
Structure	283
Contingency	49
Total Stage Weight	3,939 lb

Table 8-17. EARTH BRAKING STAGE WEIGHT SUMMARY

Propellants	919 lb
Residuals	9
Tankage	21
Engine	15
Plumbing	3
Pressurization	4
Thermal Control	6
Structure	105
Contingency	12
Total Stage Weight	1,094 lb

The orbiter/bus design concept is based on jettisoning a portion of the solar arrays and thruster modules just after Mars spiral-out escape. The portion to be jettisoned was iteratively determined based on a criterion of matching the required optimal thrust acceleration for the low-energy Earth return (SEP) trajectory. A weight summary of the SEP system is given in Table 8-18.

Orbiter/Bus Module

The basic orbiter/bus module weights for the solar-electric/chemical concept are essentially identical to the all-chemical concept except for the power and mechanical subsystems. The power is provided by the SEP solar arrays and the structure is heavier to carry the loads of the arrays.

Table 8-18. SOLAR-ELECTRIC PROPULSION SYSTEM WEIGHT SUMMARY

SYSTEM ELEMENT	AT EARTH DEPARTURE	AT MARS DEPARTURE
Solar Arrays	583 lb	164 lb
Propellant (Mercury)	1539	334
Tankage	46	46
Thrusters and Power Conditioning	299	84
Total System Weight	2467 lb	628 lb

Table 8-19. ORBITER/BUS MODULE WEIGHT SUMMARY (SOLAR-ELECTRIC/ CHEMICAL CONCEPT)

ITEM	OUTBOUND CONFIGURATION	INBOUND CONFIGURATION
Science	125 lb	0 lb
Communications	122	36
Power (Batteries)	198	99
Data Storage	94	0
Computer and Command	24	24
Pyrotechnics	31	21
Cabling and Harness	120	60
Temperature Control	37	29
Mechanical Devices	112	56
Attitude Control Inerts	85	85
Attitude Control Propellants	47	17
Docking Adapter	45	20
Rendezvous Radar	24	0
Structure	290	231
Redundancy	35	21
Contingency	72	20
Total Weight	1466 lb	719 lb

As in the all-chemical concept, orbiter/bus equipment modules are jettisoned in Mars orbit in order to reduce the gross Earth departure weight requirement. A weight summary of the orbiter/bus in both the outbound and inbound configurations is presented in Table 8-19.

8.2.3 Preliminary Program Schedule

From the level of detail achieved during this study there were no discernable differences between the overall program schedule prepared for the dual departure solar-electric/chemical concept and the dual departure all-chemical concept (Figure 8-6, subsection 8.1.3). Both systems were based on 1971 technology and are virtually identical except for the solar-electric propulsion system. The schedules for the 1977 and 1979 mission opportunities were shown in Figures 8-7 and 8-8.

8.2.4 Preliminary Cost Estimate

The costing groundrules for this concept are identical to those in subsection 8.1.4 for the dual departure all-chemical concept. Table 8-20 summarizes the estimated costs by cost element. The total cost less recovery is $971 million. This is $80 million greater than for the dual departure, all-chemical system (subsection 8.1).

Table 8-20 PRELIMINARY COST ESTIMATE FOR DUAL LAUNCH, SOLAR-ELECTRIC/CHEMICAL MISSION/SYSTEM CONCEPT.

COST ELEMENT	COST (MILLIONS OF DOLLARS)
Design and Development	672
Production	227
Operations	32
Launch Vehicles	40
Total Costs*	971

* Recovery costs not included

8.2.5 Supporting Technology Requirements

The supporting technology requirements for the dual departure solar-electric/chemical system are essentially identical to the dual departure all-chemical concept with the addition of the SEP related requirements.

The requirements for the solar-electric/chemical system then become those stated in subsection 8.1.5 and with the addition of the following items.

Retractable Roll-Out Solar Arrays

Use of solar-electric primary propulsion for the orbiter/bus vehicle has attractive potential for the MSSR mission. This application requires that solar arrays be retracted for a chemical capture maneuver at Mars and re-deployed (at least partially) in Mars orbit. Some mission modes require multiple retraction/deployment cycles. This technology must be demonstrated for relatively long duration missions.

The objective of this activity is the development and demonstration of reliable retractable/deployable roll-out solar arrays for the solar-electric MSSR orbiter/bus.

Low Thrust Trajectories

Solar-electric primary propulsion offers a means to shorten the total MSSR mission duration over that required for low-energy all-chemical conjunction class missions. This is of particular interest for Titan III D/Centaur class system concepts. An in-depth analysis needs to be made of Mars-to-Earth trajectories and performance to determine the mission time savings available. The return concepts must be carefully integrated with the total mission/system approach. The possible use of Venus swingbys should be considered.

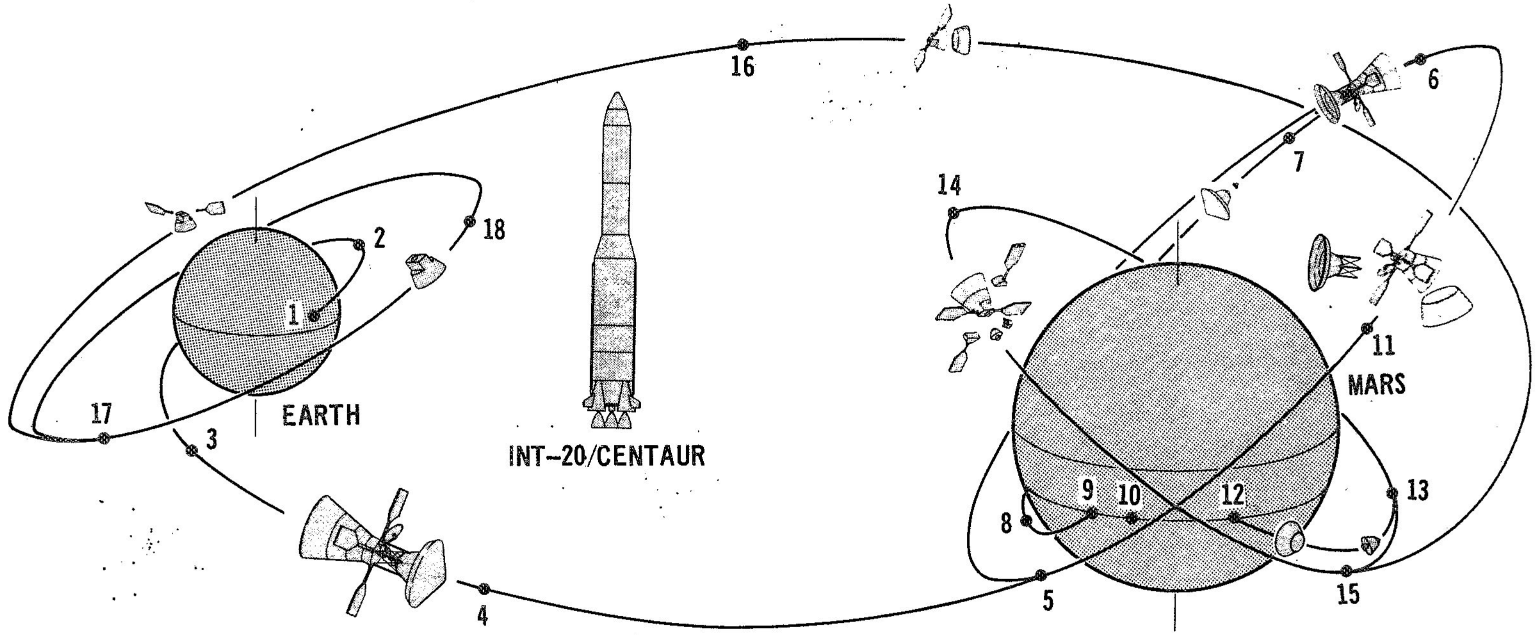
The objective of this activity is to determine the full potential of low thrust solar-electric systems for reducing MSSR mission duration.

8.3 SINGLE LAUNCH ALL-CHEMICAL CONCEPT

This system concept requires a gross Earth departure weight of approximately 34,000 pounds based on 1971 technology. This requirement can be met by the Saturn Intermediate-20/Centaur launch vehicle which has a 36,200-pound capability for the 1975 launch opportunity.

8.3.1 Mission Profile

The preliminary mission profile for the single launch all-chemical concept is presented in Figure 8-12.



1. INT-20/CENTAUR LAUNCH FROM KSC
2. DEPARTURE THROUGH 185-KM PARKING ORBIT
3. CENTAUR BURN ONTO TRANSMARS TRAJECTORY
4. INTERPLANETARY TRANSFER (350 DAYS)
5. CAPTURE INTO 600 X 33,400 KM ELLIPTICAL ORBIT
6. SEPARATE LANDER/ASCENT PROBE; PERFORM DEORBIT MANEUVER
7. PROBE COAST TO ENTRY; JETTISON DEORBIT PROPULSION MODULE
8. PROBE ENTRY INTO MARS ATMOSPHERE
9. LANDING USING AEROBRAKING AND TERMINAL PROPULSION
10. SURFACE SAMPLING OPERATIONS
11. ORBITAL SUPPORT OF MARS SURFACE OPERATIONS
12. MARS ASCENT VEHICLE (MAV) LAUNCH AFTER 90 DAYS ON SURFACE (340-DAY TOTAL STOPOVER)
13. RENDEZVOUS/DOCKING WITH ORBITER/BUS; RETRIEVE SAMPLES CANISTER
14. JETTISON MAV SECOND STAGE AND EXCESS ORBITER EQUIPMENT
15. TRANSEARTH INJECTION; JETTISON MARS DEPARTURE STAGE
16. INBOUND INTERPLANETARY TRANSFER (290 DAYS)
17. CAPTURE INTO 555 X 9100-KM ELLIPTICAL ORBIT
18. RECOVER SAMPLES AND FILM IN ORBIT

Figure 8-12. SINGLE LAUNCH ALL-CHEMICAL CONCEPT MISSION PROFILE

8.3.2 System Description

The system (from the Earth launch vehicle interface forward) consists of an Earth launch vehicle adapter, Mars braking stage, Mars departure stage, Earth braking stage, orbiter/bus module, probe mounting structure, deorbit propulsion module, and the lander/ascent probe. The preliminary system configuration developed for the system is presented in Figure 8-13. A system weight summary is given in Table 8-21.

8.3.2.1 Earth Launch Vehicle Adapter. The Earth launch vehicle adapter is the mechanical interface between the Centaur stage and the Mars braking stage of the planetary vehicle. The adapter is a tubular truss structure estimated to weigh 334 pounds.

8.3.2.2 Mars Braking Stage. The Mars Braking Stage (MBS) performs the outbound midcourse correction maneuvers, the Mars braking maneuver, and the Mars orbit trim maneuvers. The MBS contains 15,240 pounds of N_2O_4/MMH propellants in four spherical tanks. Propulsion is provided by a cluster of thirteen 325-pound thrust engines. The propellant and engine characteristics are presented in Table 5-3 (Section V). The MBS also contains the attitude control system employed during the outbound leg of the mission. A weight summary is presented in Table 8-22.

8.3.2.3 Mars Departure Stage. The Mars departure stage (MDS) performs the Mars orbit rendezvous maneuvers and the Mars departure maneuver. The MDS contains a total of 4407 pounds of N_2O_4/MMH propellants in four spherical tanks. Propulsion is provided by five 325-pound thrust engines. The engine characteristics are identical to the Mars Braking Stage. A weight summary is presented in Table 8-23.

8.3.2.4 Earth Braking Stage. The Earth braking stage (EBS) performs the inbound midcourse trajectory correction maneuvers, and the Earth braking maneuver. The EBS contains a total of 1864 pounds of N_2O_4/Aerozine-50 propellants in four spherical tanks. The thrust is provided by a single 325-pound thrust engine. The engine is the same as that employed in the Mars Braking and Departure Stages. A weight summary is presented in Table 8-24.

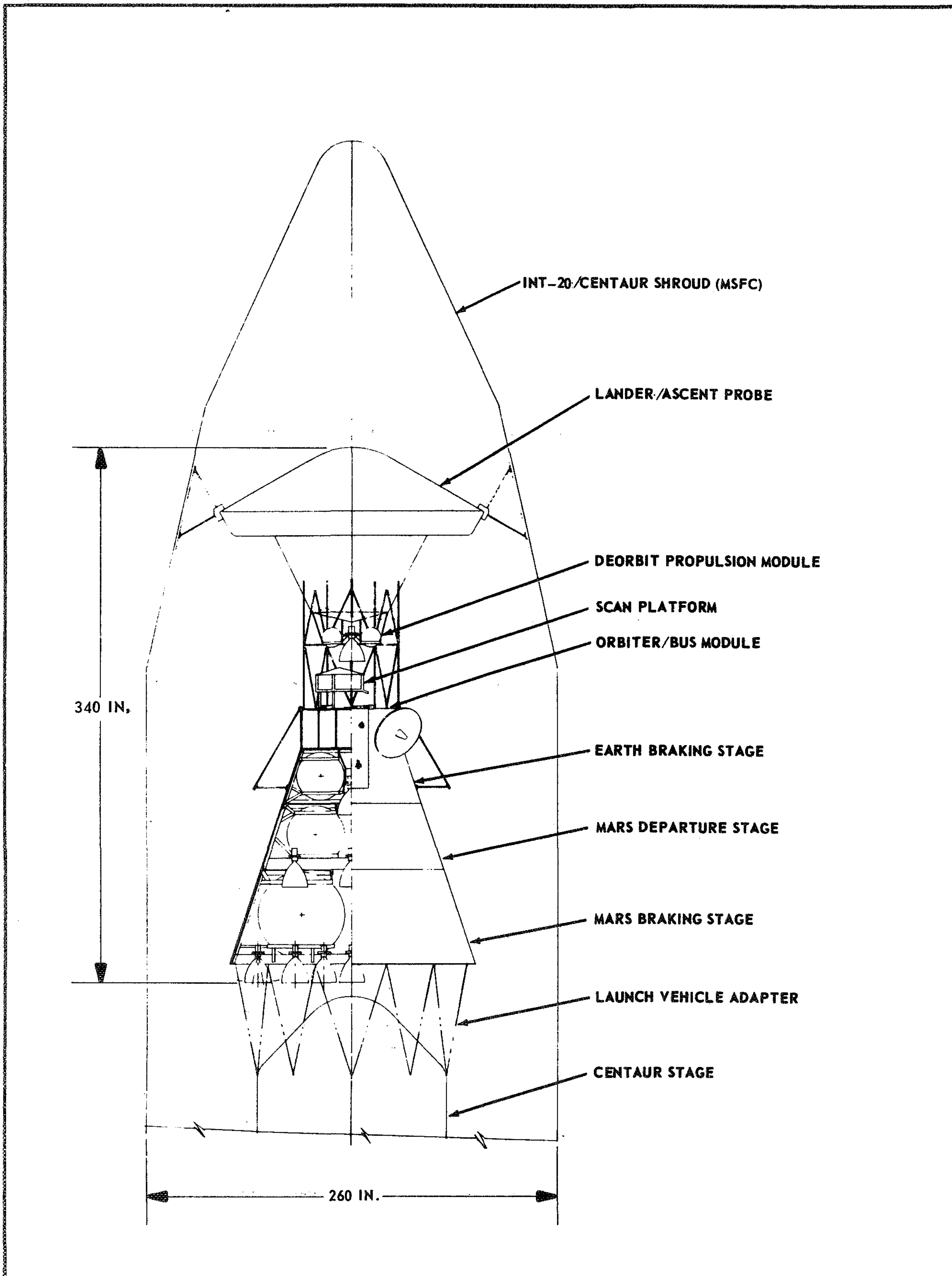


Figure 8-13. SINGLE LAUNCH ALL-CHEMICAL SYSTEM CONFIGURATION

Table 8-21. SINGLE LAUNCH ALL-CHEMICAL SYSTEM WEIGHT SUMMARY

LANDER/ASCENT PROBE		6,086 LB
MARS ASCENT VEHICLE	2,812	
ROVER	150	
LANDER	1,946	
AEROBRAKING SYSTEM	1,178	
DEORBIT MODULE		278
STERILIZATION CANISTER		401
PROBE MOUNTING STRUCTURE		318
ORBITER/BUS MODULE		1,546
EARTH BRAKING STAGE		2,180
MARS DEPARTURE STAGE		5,065
MARS BRAKING STAGE		17,586
TOTAL PLANETARY VEHICLE		33,460
EARTH LAUNCH VEHICLE ADAPTER		334
GROSS EARTH DEPARTURE WEIGHT		33,794 LB

Table 8-22. MARS BRAKING STAGE WEIGHT SUMMARY

Propellants	15,240 lb
Residuals	152
Tankage	273
Engines	237
Plumbing	47
Pressurization	277
Thermal Control	39
Attitude Control (Inert)	82
Attitude Control (Propellants)	268
Structure	711
Contingency	260
Total Stage Weight	15,586 lb

Table 8-23. MARS DEPARTURE STAGE WEIGHT SUMMARY

Propellants	4,407 lb
Residuals	44
Tankage	78
Engine	91
Plumbing	18
Pressurization	81
Thermal Control	14
Structure	258
Contingency	74
Total Stage Weight	5,065 lb

Table 8-24. EARTH BRAKING STAGE WEIGHT SUMMARY

Propellants	1,864 lb
Residuals	18
Tankage	35
Engine	30
Plumbing	6
Pressurization	34
Thermal Control	16
Structure	142
Contingency	35
Total Stage Weight	2,180 lb

8.3.2.5 Orbiter/Bus Module. The orbiter/bus module consists of the Viking orbiter science, selected Viking orbiter subsystems, and other subsystems based on existing technology. The subsystems selected from the Viking orbiter include the communications, power, data storage, computer and command, pyrotechnics, and cabling and harness. The remaining subsystems are based on 1971 technology.

Orbiter/bus equipment not required for the inbound leg of the mission are jettisoned in Mars orbit. This will require packaging most of the subsystems into modules to be retained or jettisoned. A weight summary of the orbiter/bus module in both the outbound and inbound configurations is presented in Table 8-25. The orbiter/bus module described above is essentially identical to that previously described in subsection 8.1 for the dual all-chemical concept.

8.3.2.6 Probe Mounting Structure. The probe mounting structure provides the mechanical interface btween the Earth braking stage and the lander/ascent probe. The probe mounting structure is a tubular truss structure estimated to weigh 318 pounds.

Table 8-25. ORBITER/BUS MODULE WEIGHT SUMMARY

ITEM	OUTBOUND CONFIGURATION	INBOUND CONFIGURATION
Science	125 lb	0 lb
Communications	122	36
Power	288	144
Data Storage	94	0
Computer and Command	24	24
Pyrotechnics	31	11
Cabling and Harness	120	60
Temperature	37	29
Mechanical Devices	112	56
Attitude Control Inerts	85	85
Attitude Control Propellants	100	40
Docking Adapter	45	10
Rendezvous Radar	24	0
Structure	230	184
Redundancy	35	17
Contingency	72	34
Totals	1630 lb	815 lb

8.3.2.7 Sterilization Canister. The sterilization canister or bioshield, provides contamination protection to the lander/return probe from sterilization at Earth to Mars orbit. The canister is estimated to weigh 401 pounds.

8.3.2.8 Probe Deorbit Module. The probe deorbit module provides the ΔV impulse required to deorbit the probe after its separation from the orbiter/bus in Mars elliptical orbit. The module contains a total of 209 pounds of N_2O_4/MMH propellants in four spherical tanks and employs a single 325-pound thrust engine. A weight summary of the module is presented in Table 8-26. It is noted that the deorbit module should be enclosed in the sterilization canister.

8.3.2.9 Lander/Ascent Probe. The lander/ascent probe defined for the single launch, all-chemical system is functionally identical to the probe previously described for the dual departure, all-chemical system in subsection 8.1. The probe consists of an aerobraking system, lander, rover, and Mars ascent vehicle (MAV).

The lander, rover, and MAV are identical to those described in subsection 8.1. The aerobraking system weight is reduced from 1542 pounds to 1178 pounds because of the lower entry velocity, heating rates, and entry "g" levels associated with entry-out-of-orbit versus direct entry from the approach hyperbola. A weight summary of the lander/ascent probe is presented in Table 8-27. A configuration drawing and detailed weights of the lander and MAV were presented in subsection 8.1.

8.3.3 Preliminary Program Schedule

For the level of detail achieved during this five-week study there were no discernable differences between the overall program schedule for the single launch, all-chemical system and the preliminary program schedule previously presented in subsection 8.1.3 for the dual departure all-chemical system. The same (1971) technology base was assumed for system development and many of the subsystems are identical. A parallel launch vehicle development program would be required, however, to make the Saturn INT-20/Centaur vehicle available for the 1975 launch opportunity. The preliminary schedules for the 1977 and 1979 mission opportunities are the same as for the dual departure concept in subsection 8.1.3.

Table 8-26 PROBE DEORBIT MODULE WEIGHT SUMMARY

Propellants	209 lb
Residuals	4
Tankage	8
Engine	15
Plumbing	3
Pressurization	4
Thermal Control	5
Pyrotechnics	4
Structure	16
Contingency	10
Total Module Weight	278 lb

Table 8-27. LANDER/ASCENT PROBE WEIGHT SUMMARY (ENTRY-OUT-OF-ORBIT)

Mars Ascent Vehicle		2,812 lb
Rover		150
Lander		1,946
Aerobraking System		1,178
Aeroshell	206	
Heatshield	102	
AID	140	
Attitude Control Inerts	74	
Attitude Control Propellants	55	
Terminal Descent Propellants	397	
Contingency	204	
Probe Entry Weight		6,086 lb

8.3.4 Preliminary Cost Estimate

The costing groundrules for this concept are identical to those presented in subsection 8.1 for the dual departure, all-chemical concept. Table 8-28 summarizes the estimated costs by cost element. The total program less recovery cost is $1078 million. This is approximately $190 million more than the estimated total cost for the dual departure, all-chemical system (subsection 8.1).

8.3.5 Supporting Technology Requirements

Because of the similarity between the dual departure and single launch all-chemical systems and missions, the supporting technology requirements for the single launch concept are essentially identical to those presented in subsection 8.1.5.

Table 8-28. PRELIMINARY COST ESTIMATE FOR SINGLE LAUNCH, ALL-CHEMICAL MISSION/SYSTEM CONCEPT.

COST ELEMENT	COST (MILLIONS OF DOLLARS)
Design and Development	647
Production	218
Operations	23
Launch Vehicle	190
Development (100)	
Production (90)	
Total Costs*	1078

* Recovery costs not included

Section IX

CONCLUSIONS AND RECOMMENDATIONS

This section summarizes the conclusions drawn from the results of the present study and gives specific recommendations for follow-on activities which appear to be required if the automated MSSR concept is to be pursued by NASA.

9.1 CONCLUSIONS

The conclusions are summarized as follows:

- The selection of spacecraft concept and launch vehicle strongly depends on the Earth return intercept/recovery requirement or groundrule adopted for the mission.
- If direct reentry/recovery is acceptable, then the Dual Departure, All-Chemical Titan IIID/Centaur concept offers a promising low-cost approach.
- If capture/recovery is required, then two promising alternatives to the above approach are:
  * The Dual Departure, Solar-Electric/Chemical, Titan IIID/Centaur Concept
  * The Single Launch, All-Chemical, INT-20/Centaur Concept.
- The dual departure, solar-electric/chemical alternative offers potential cost, mission flexibility, and lander-orbiter interface advantages over the INT-20/Centaur single launch approach.
- With regard to mission flexibility, the dual departure concept offers the possibility of using multiple lander/ascent probes to sample more than a single Mars site. (Probability of mission success can also be increased.) The lander or orbiter could perform missions as independent payloads. Generally, the dual (or multiple) departure concept offers an approach adaptable to a broad range of automated Mars missions. For example, the probe ascent vehicle could be replaced by a 2000-pound class mobile surface laboratory and launched as a separate mission.
- With regard to orbiter-lander interface, the dual departure concept minimizes performance and system interdependence between the lander/return probe and orbiter/bus vehicle.
- The following conclusions are drawn regarding 1975 MSSR mission feasibility:
  * The three selected mission/system concepts identified above appear possible using 1971 technology
  * System development requires maximum utilization of Viking/Mariner hardware

* Program requires January 1971 Phase A start
* Requires January 1972 program commitment
* Requires intensive CY 1971 Phase A/Phase B study program.

9.2 RECOMMENDATIONS

Following is a summary of recommendations based on the results of the present study:

- Further analysis is needed on the application of Mariner/Viking subsystems to the MSSR mission. This is particularly critical if a 1975 mission is to be considered.
- A study should be performed to further analyze alternative lander probe deceleration systems using direct (hyperbolic), lifting entry and high ballistic coefficients (of the order of 1 slug/ft^2).
- An extension of the current effort should be made to develop alternative configurations for the more promising system approaches. This would allow further verification of system weights and mission concepts prior to a Phase A effort.
- System performance exchange ratios should be generated for the promising all-chemical and solar-electric concepts to provide further visibility into the criticality of system design parameter assumptions.
- Because of the possibility of using Venus swingby mission modes to shorten mission duration, further investigation of space storable propulsion systems should be conducted.*
- Further work is needed to analyze the full potential of solar-electric propulsion to shorten mission duration.
- Further analysis is needed to establish alternative design approaches to alleviate probe diameter problems in the promising dual inter-planetary launch concept.
- An analysis should be conducted to establish the compatibility of planned Viking experiments and instrumentation with requirements of an early MSSR mission.
- An in-depth design analysis should be performed to investigate fail-safe direct reentry/recovery system concepts.
- A cost analysis should be performed to compare the direct reentry/recovery and orbital capture/recovery approaches. A preliminary comparison of reliability of alternative recovery approaches should be made.
- A study of the Earth capture/recovery approach using a sub-module of the return orbiter/bus spacecraft should be conducted to further analyze the feasibility of this approach using Titan class launch vehicles in the dual departure concept.

**INT-20/Centaur class systems using all-chemical space storable propulsion for Mars braking (and possibly departure) stage permit consideration of Venus swingby missions.*

Section X

REFERENCES

1. Northrop-Huntsville, "Automated Mars Surface Sample Return Mission/System Study - Final Report", Report No. TR-793-717, Vol. I, February 1970.

2. Northrop-Huntsville, "Automated Mars Surface Sample Return Mission/System Study - Final Report", Report No. TR-793-717, Vol. IIA, February 1970.

3. Northrop-Huntsville, "Automated Mars Surface Sample Return Mission/System Study - Final Report", Report No. TR-793-717, Vol. IIB, February 1970.

4. Northrop-Huntsville, "Automated Mars Surface Sample Return Mission/System Study - Final Report", Report No. TR-793-717, Vol. III, February 1970.

5. "Launch Vehicle Estimating Factors", NASA OSSA, Code SV, Washington, D.C., January 1970.

6. Horsewood, J. L. and Mann F. I., "Optimum Solar Electric Interplanetary Trajectory and Performance Data", NASA CR-1524, April 1970.

7. Melbourne, W. G. and Sauer, C. G., "Performance Computations With Pieced Solutions of Planetocentric and Heliocentric Trajectories for Low-Thrust Missions", JPL Space Programs Summary No. 37-36, Volume IV, December 1965.

8. Mullin, J. P., Barber T., and Zola C., "Solar-Cell-Powered, Electric Propulsion for Automated Space Missions", J. Spacecraft Rockets, Vol. 6, No. 11, November 1969.

9. Friedlander A. L., Solar Electric Propulsion - A Survey, Technology Status and Mission Applications, Report No. M-21, IIT Research Institute, Chicago, Illinois, March 1970.

10. Sauer, Carl G., Jr., "Optimization of a Solar-Electric-Propulsion Planetary Orbiter Spacecraft". AAS Paper, No. 68-104, September 3-5, 1968.

11. Lazar, James, "Review of NASA Programs in Electric Propulsion", AIAA Paper No. 69-248, March 3, 1969.

12. Barber, T. A., Goldsmith, J. V., and Edberg, J. R., "Spacecraft Electric Propulsion - Now?", Astronautics and Aeronautics, June 1968.

13. Kerrisk D. J. and Bartz D. R., "Primary Electric Propulsion Technology - Toward Flight Programs for the Mid - 1970's", Astronautics and Aeronautics, June 1968.

14. Guy, L. D., "Structural and Decelerator Design Options for Mars Entry", AIAA Paper 68-344.

15. Mikulas, M. M. and Bohou, H. L., "Summary of the Development Status of Attached Inflatable Decelerators", AIAA Paper No. 68-929.

16. Divone, L., Mohl, C., Casani, F. and Norris, H., "Surveyor Block II Mission Study", EPD-259, Jet Propulsion Laboratory, Pasadena, California, 15 April 1965.

Appendix
SOLAR-ELECTRIC/CHEMICAL MISSION/SYSTEM PERFORMANCE EVALUATION PROGRAM

This appendix describes the solar-electric/chemical mission/system performance and weight analysis program developed during this study effort. This program was utilized to compute much of the solar-electric/chemical vehicle weights and performance data generated during the study.

The program computes a mission mass history for a specified sample return weight, mission mode combination, specified orbiter/bus systems module weight at earth return, and a specified set of performance parameters for the required chemical and solar electric propulsion systems. During the computation, an iteration was made on the required orbiter/bus module weight to determine the necessary gross earth departure weight. An additional iteration was made during the computation to optimize the fraction of solar panels jettisoned at Mars such that an input optimum thrust-to-weight is achieved for the SEP return leg of the mission.

The program output consists of a summary of all major system weights compatible with the specified mission mode and desired sample return weight. Additionally, the program generates design power requirements; power plant efficiencies, solar array area, power plant weight, and time to spiral-out for a SEP Mars escape (if that mode is specified).

A.1 PHYSICAL MODEL

The system design requirements of the solar-electric/chemical orbiter/bus vehicle are determined by the mission concepts. These are: (1) multiple interplanetary launch with no probe aboard, (2) a single launch which carries the Mars lander/return probe with associated supporting structure, sterilization canister, etc., and (3) Earth orbit rendezvous departure.

The orbiter/bus vehicle was functionally divided into propulsive modules designed to efficiently perform the respective mission velocity increments. Either a chemical module or SEP module was selected based on effectiveness of the particular module in performing its propulsive function without incurring either (1) a large Earth departure weight penalty or (2) excessive mission duration increases.

From the outset of the study it was clear that certain specific propulsive functions of the MSSR mission could not be performed by SEP, for example, maneuvers requiring a specified thrust-to-weight ratio of any significant magnitude, such as escape from the Mars surface. Thus, the lander/ascent probe was necessarily based on all-chemical systems. In contrast, the orbiter/bus vehicle appeared quite open to employment of SEP for almost every major propulsive function with the possible exception of Mars orbit rendezvous.

In order to satisfy the criterion of avoiding excessive increases in mission duration, certain propulsive functions were eliminated from consideration for SEP, even though it appeared that SEP could successfully be utilized if given sufficient time. These were: (1) spiral escape from Earth, (2) spiral Mars capture, and (3) spiral Earth capture. The spiral escape mode at Mars was retained as a potentially attractive means of reducing Earth departure weight requirements without significant impact on total mission duration for the class of low-energy missions under consideration.

Figure A-1 depicts the multiple interplanetary launch mission mode options modeled in the performance program. The four mission/system concepts analyzed by the program are as summarized in Table A-1.

Consideration of the single launch mission mode options resulted in definition of 12 mission combinations which were modeled in the computer program. These are summarized in Table A-2. Figure A-2 outlines the program options.

The earth orbit rendezvous (EOR) departure mode concepts were analyzed by utilizing the output spacecraft weights of the single launch concept option and adding the EOR equipment and appropriate propulsion weights. The Centaur orbital injection stage was added by hand computations.

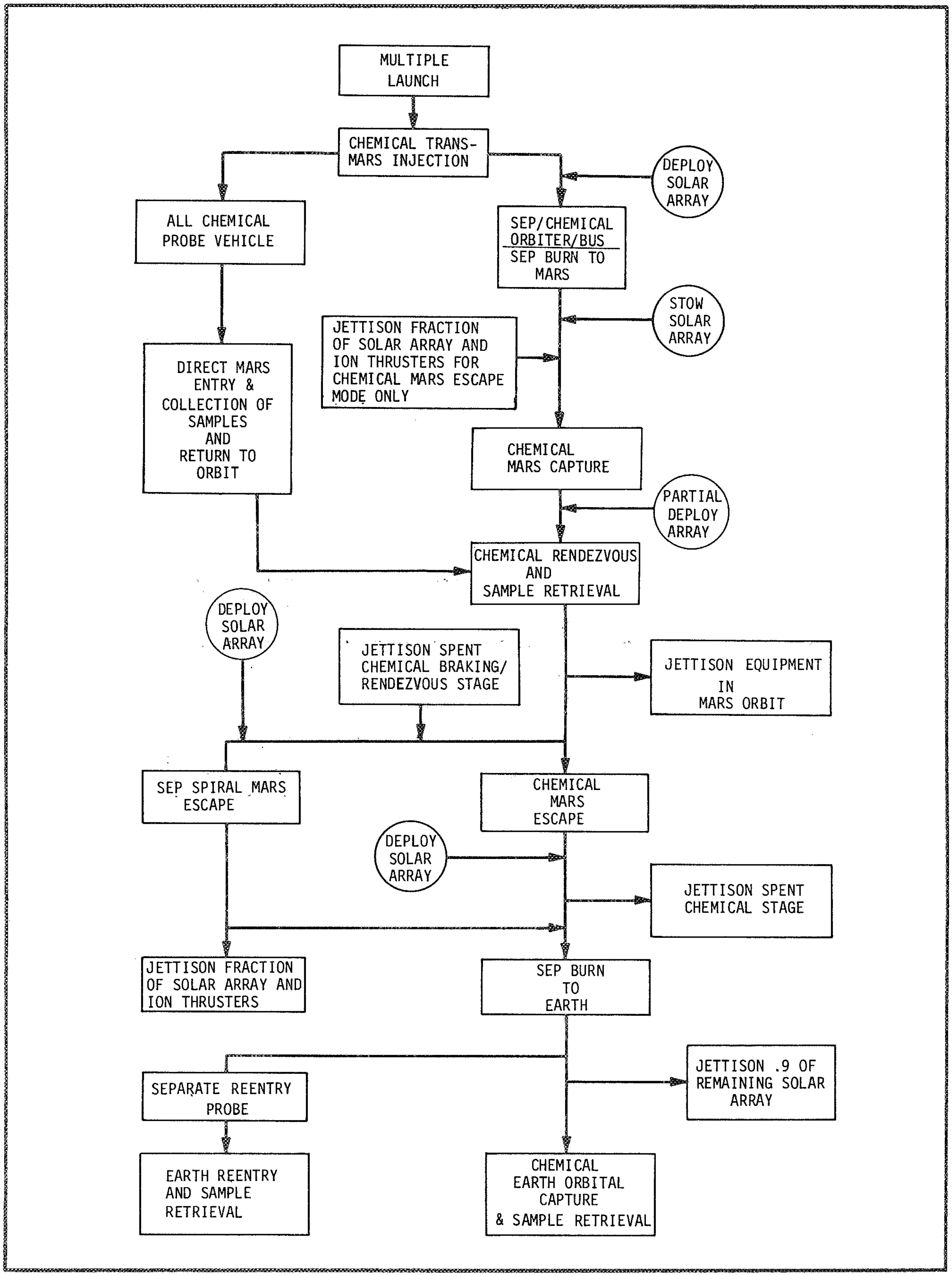


Figure A-1. MULTIPLE LAUNCH CONCEPT OPTIONS

Table A-1. MULTIPLE INTERPLANETARY LAUNCH CONCEPT ALTERNATIVES

- CHEMICAL TRANS-MARS INJECTION
- SEP EARTH-MARS TRANSFER
- CHEMICAL MARS CAPTURE INTO CIRCULAR ORBIT
- CHEMICAL MARS ORBIT RENDEZVOUS
- SEP MARS-EARTH TRANSFER

CONCEPT ALTERNATIVE	MARS ESCAPE MODE	EARTH INTERCEPT/ RECOVERY MODE
M-1	SEP Spiral-Out	Direct Reentry
M-2	SEP Spiral-Out	Chemical Capture
M-3	Chemical Impulse	Direct Reentry
M-4	Chemical Impulse	Chemical Capture

Table A-2. SINGLE LAUNCH CONCEPT ALTERNATIVES

- CHEMICAL TRANS-MARS INJECTION
- SEP EARTH-MARS TRANSFER
- CHEMICAL MARS ORBIT RENDEZVOUS
- SEP MARS-EARTH TRANSFER

CONCEPT ALTERNATIVE	MARS ATMOSPHERE ENTRY MODE	MARS ESCAPE MODE	EARTH INTERCEPT/ RECOVERY MODE
S-1	Direct	SEP Spiral-Out	Direct Reentry
S-2	Direct	SEP Spiral-Out	Chemical Capture
S-3	Direct	Chemical Impulse	Direct Reentry
S-4	Direct	Chemical Impulse	Chemical Capture
S-5	OCO(1)	SEP Spiral-Out	Direct Reentry
S-6	OCO	SEP Spiral-Out	Chemical Capture
S-7	OCO	Chemical Impulse	Direct Reentry
S-8	OCO	Chemical Impulse	Chemical Capture
S-9	OEO(2)	SEP Spiral-Out	Direct Reentry
S-10	OEO	SEP Spiral-Out	Chemical Capture
S-11	OEO	Chemical Impulse	Direct Reentry
S-12	OEO	Chemical Impulse	Chemical Capture

(1) OCO = Entry Out of Circular Orbit

(2) OEO = Entry Out of Elliptical Orbit

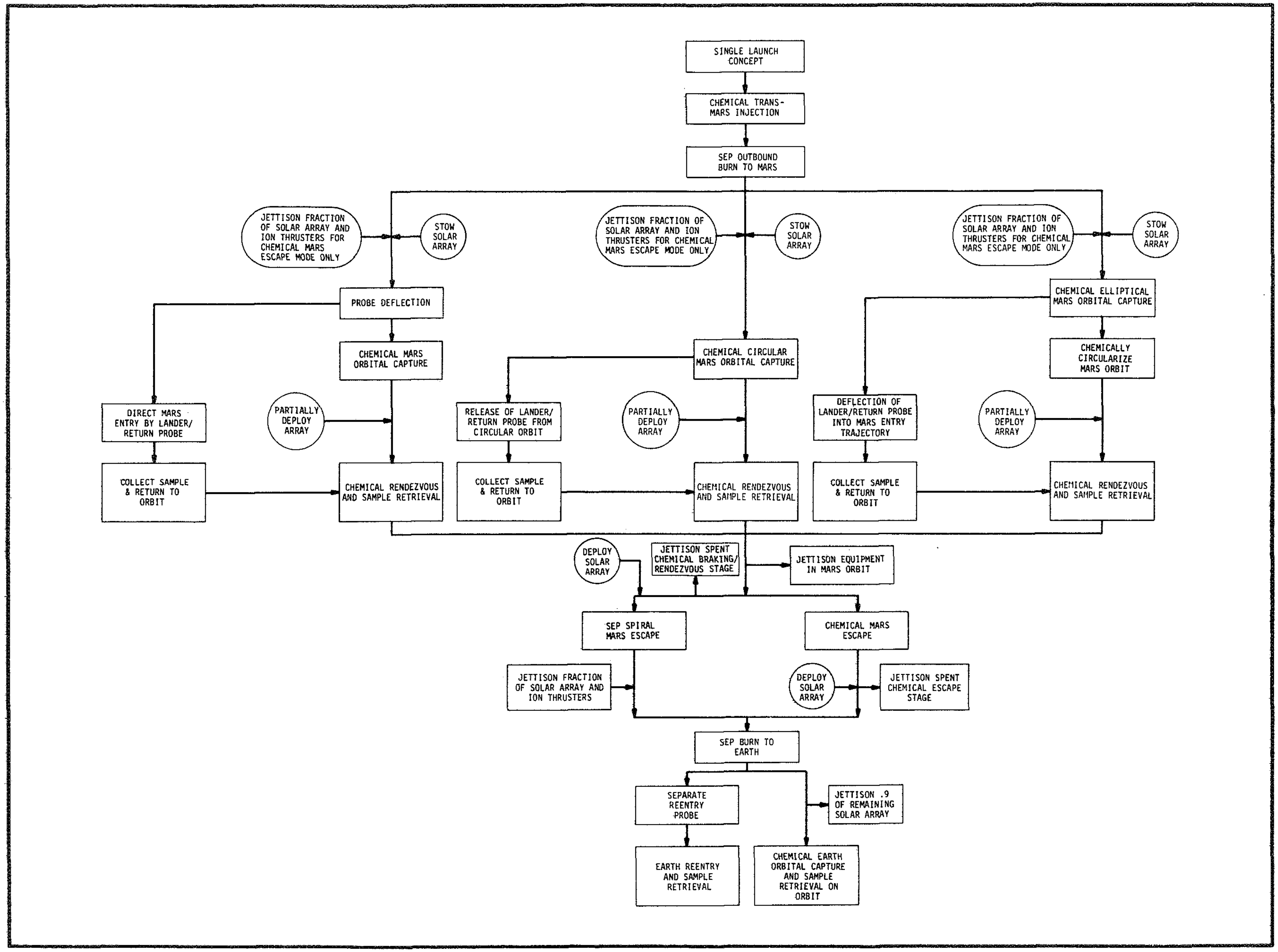


Figure A-2. SINGLE LAUNCH CONCEPT OPTIONS

A.2 MISSION/SYSTEM PERFORMANCE MODEL EQUATIONS

The equations used in the mission/system performance analysis model are outlined in the following paragraphs.

A.3 POWERPLANT EQUATIONS

The SEP power requirement at one AU from the sun is given by

$$P_{1AU} = \frac{C\, a_o\, g_o\, I_{sp}\, W_n}{2\eta} + P_{aux} \qquad \text{(kw)} \qquad \text{(A-1)}$$

where

- C = units conversion factor (.001356)
- a_o = initial thrust acceleration (g)
- g_o = 32.174 ft/sec^2
- I_{sp} = specific impulse of the SEP system (sec)
- W_n = net spacecraft earth departure weight (lb)
- η = efficiency of the thruster subsystem
- P_{aux} = auxiliary spacecraft power requirement (kw)

The efficiency η of the thruster subsystem is related to specific impulse by the following equation from reference 6:

$$\eta = \frac{0.769}{\left[1 + \left(\frac{14.3}{.00981\, I_{sp}}\right)\right]} \qquad \text{(A-2)}$$

The required area of the solar arrays is given by

$$A_{sa} = 100(P_{1AU} + P_{aux}) \qquad \text{(sq ft)} \qquad \text{(A-3)}$$

where power is in kilowatts.

The weight of the power plant W_{pp} is given by

$$W_{pp} = P_{1AU}\,(\alpha_{sp} + \alpha_{it}) - \alpha_{it}\, P_{aux} \qquad \text{(lb)} \qquad \text{(A-4)}$$

where

- α_{sp} = specific weight of solar arrays (lb/kw)
- α_{it} = specific weight of the ion thrusters and associated power conditioning equipment (lb/kw)

The SEP propellant tank and feed system weight is given by

$$W_{SEP_t} = K_t(W_{po} + W_{pi} + W_{pso}) \quad \text{(lb)} \qquad \text{(A-5)}$$

where

K_t = tank and feed system factor for mercury propellant

W_{po} = SEP propellant weight for Earth-Mars transfer

W_{pi} = SEP propellant for Mars-Earth transfer

W_{pso} = SEP propellant for Mars escape (spiral escape mode option)

A.4 OUTBOUND MISSION WEIGHT EQUATIONS

Let W_g be the gross Earth departure weight of the spacecraft. The net departure weight, W_n, is the gross weight less the launch vehicle adapter, or

$$W_n = W_g/(1 + K_{adp}) \quad \text{(lb)} \qquad \text{(A-6)}$$

where K_{adp} is an adapter weight constant.

The outbound SEP propellant weight is defined by

$$W_{po} = K_{po} W_n \quad \text{(lb)} \qquad \text{(A-7)}$$

where K_{po} is the outbound propellant fraction based on trajectory data.

The outbound weight expended for attitude control is defined as

$$W_{aco} = K_{aco} W_n \quad \text{(lb)} \qquad \text{(A-8)}$$

where K_{aco} is the attitude control expendables fraction of the net earth departure weight.

The net Mars arrival weight is computed by

$$W_{na} = W_n - W_{po} - W_{aco} \quad \text{(lb)} \qquad \text{(A-9)}$$

A.5 CALCULATION OF POWERPLANT JETTISON WEIGHT

In the chemical Mars escape option, a portion of the solar array and ion thrusters (including associated power conditioning equipment) are jettisoned prior to Mars capture. The jettison weight is optimized such that the spacecraft thrust-to-weight at Mars departure matches the input thrust acceleration required for the Mars-Earth return trajectory. In the SEP spiral-out Mars escape option, the computed portion of the powerplant weight is jettisoned after the spiral-out maneuver. Determination of the weight to be jettisoned prior to Mars capture (chemical escape option) requires an iterative procedure as follows:

A factor β is defined by the ratio

$$\beta = \frac{(I_{spi}\ \eta_o\ a_i)}{(I_{spo}\ \eta_i\ a_o)} \qquad \text{(A-10)}$$

where a_i and a_o are the input inbound and outbound thrust accelerations, respectively (given in g), I_{spi} and I_{spo} are the input inbound and outbound specific impulses, respectively, and η_i and η_o are the computed inbound and outbound thruster subsystem efficiencies, respectively.

The power plant fraction to be jettisoned is given by the expression

$$F_1 = 1 - (W_{gea}\beta/W_n) \qquad \text{(A-11)}$$

where

W_{gea} = gross Earth arrival weight computed during each iteration.

The solar array weight to be jettisoned is given by

$$W_{sajm} = F_1(\alpha_{sp})\ (P_{1AU}) \qquad \text{(lb)} \qquad \text{(A-12)}$$

The thruster module weight to be jettisoned is

$$W_{itjm} = F_1\ \alpha_{it}\ (P_{1AU} - P_{aux}) \qquad \text{(lb)} \qquad \text{(A-13)}$$

and the ignition weight of the spacecraft for Mars capture is

$$W_{nme} = W_{na} - W_{sajm} - W_{itjm} \qquad \text{(lb)} \qquad \text{(A-14)}$$

The program performs the above computations by making an initial guess of F_1 and iterating on this guess until the desired value of F_1 satisfies a specified accuracy criterion on the spacecraft thrust acceleration for Earth return.

A.6 MARS CAPTURE PERFORMANCE EQUATIONS

A.6.1 Multiple Launch Option (Orbiter/Bus Carries No Lander/Ascent Probe)

The gross capture weight in Mars orbit is given by

$$W_{gc} = W_{na}\, e^{-\Delta V_c / g I_{spc}} \quad \text{(lb)} \qquad \text{(A-15)}$$

where ΔV_c = velocity decrement for capture (m/sec)

I_{spc} = specific impulse for the chemical Mars braking stage (sec)

g = 9.81 m/sec^2

The propellant weight required for capture is computed by the expression

$$W_{pc} = W_{na}\,(1-e^{-\Delta V_c / g I_{spc}}) \quad \text{(lb)} \qquad \text{(A-16)}$$

In the chemical Mars escape option, W_{nme} from equation A-14 is used in place of W_{na} in Equations A-15 and A-16.

Let ΔV_r = velocity budget for orbit trim and rendezvous with the Mars (probe) ascent vehicle. Here we assume $I_{spr} = I_{spc}$; i.e. the same chemical propulsion system is used for Mars capture braking and later for rendezvous. The required orbit trim and rendezvous propellants weight is given by

$$W_{pr} = (W_{gc} - W_{lo})\,(1-e^{-\Delta V_r / g I_{spr}}) \quad \text{(lb)} \qquad \text{(A-17)}$$

where W_{lo} = spacecraft weight loss in Mars orbit including attitude control and other expendables.

The weight for initiation of the Mars escape maneuver is

$$W_{me} = W_{gc} - W_{lo} - W_{pr} + W_{sc} - W_{je} \quad \text{(lb)} \qquad \text{(A-18)}$$

Where W_{je} = excess orbiter/bus equipment jettisoned prior to the Mars escape maneuver (lb)

W_{sc} = weight of the sample canister retrieved from the Mars ascent vehicle (lb)

The sample canister weight is based on a linear scaling law as follows:

$$W_{sc} = W_{scf} + K_{sc} W_{sam} \qquad \text{(lb)} \qquad \text{(A-19)}$$

where W_{scf} is a fixed input weight found from point design analysis and K_{sc} is a coefficient relating W_{sc} to sample weight, W_{sam}, based on design analysis.

A.7 MARS APPROACH AND CAPTURE MANEUVER PERFORMANCE FOR SINGLE LAUNCH OPTION

All single launch Earth departure concepts are handled identically to the multiple departure cases with the exception that the system carries the lander/ascent probe, sterilization canister, and associated supporting structure. The entry mode affects the equations which define Mars arrival and capture weights and the capture velocity impulse requirements differ depending on entry mode. The three Mars entry options are discussed in the following paragraphs.

A.7.1 Direct Entry Option

In this entry mode option, the probe is released and enters Mars' atmosphere directly prior to capture by the orbiter/bus. The net Mars arrival weight is

$$W_{na} = W_n - W_{prob} - W_{padp} - W_{psc} - W_{po} - W_{aco} \qquad \text{(lb)} \qquad \text{(A-20)}$$

where W_{prob} = weight of the lander/ascent probe at separation from the orbiter/bus during Mars approach. The probe weight is defined in terms of sample weight by the scaling equation

$$W_{prob} = K_{pf} + K_{ps} \, (W_{sam}) \qquad \text{(lb)} \qquad \text{(A-21)}$$

where K_{pf} and K_{ps} are constants obtained from point design analysis.

The sterilization canister weight W_{psc} is computed as a function of probe weight by

$$W_{psc} = K_{psc} \, W_{prob} \qquad \text{(lb)} \qquad \text{(A-22)}$$

where K_{psc} is a constant found from point designs.

The probe mounting structure W_{padp} is defined as a function of probe weight by

$$W_{padp} = K_{padp} W_{prob} \qquad (1b) \qquad (A-23)$$

where K_{adp} is also a constant found from point design data.

The W_{prob}, W_{adp}, and W_{psc} weights are computed identically for all single launch mission mode options.

If the chemical Mars escape option is used, equations A-10 through A-14 are employed; propellant requirements and gross capture weight are computed by equations A-15 and A-16.

A.8 MARS ENTRY OUT OF ELLIPTICAL ORBIT OPTION

Net Mars arrival weight is given by equation A-9 where W_n contains W_{prob}, W_{psc}, and W_{padp}.

The gross Mars capture weight is given by equation A-15 except the input velocity decrement is that required for elliptical capture. The propellant required for elliptical capture is given by equation A-16 when the appropriate ΔV is substituted.

After elliptical capture, the lander/ascent probe is separated (followed by the sterilization canister and probe mounting structure) from the orbiter/bus. The probe then deorbits and enters the Mars atmosphere. After release of the probe and associated equipment, a maneuver is performed by the orbiter/bus to achieve a low operational circular orbit. The propellant weight required for this maneuver is given by the expression

$$W_{pcm} = (W_{gc} - W_{prob} - W_{padp} - W_{psc}) \left(1 - e^{-\Delta V_{cm}/gI_{spc}}\right) \qquad (A-24)$$

where ΔV_{cm} = velocity increment required for the orbit circularization maneuver.

The propellants weight for orbit trim and rendezvous maneuvers is

$$W_{pr} = (W_{gc} - W_{prob} - W_{padp} - W_{psc} - W_{pcm}) \left(1 - e^{-\Delta V_r/gI_{spc}}\right) \qquad (A-25)$$

where ΔV_r = the velocity budget for orbit trim and rendezvous.

The spacecraft weight for initiation of the Mars escape maneuver is given by

$$W_{me} = W_{gc} - W_{prob} - W_{padp} - W_{psc} - W_{pcm} - W_{pr} - W_{lo} + W_{sc} - W_{je} \quad \text{(lb)} \qquad \text{(A-26)}$$

Equations A-10 through A-14 are also employed prior to Mars capture if the chemical Mars escape option is used in lieu of the SEP spiral-out mode.

A.9 ENTRY OUT OF CIRCULAR ORBIT OPTION

The net Mars arrival weight is computed by equation A-9 where W_n includes the probe, and probe mounting structure and sterilization canister as in all single launch cases. The gross capture weight in Mars orbit is given by equation A-15 where the velocity decrement is that required for capture into a circular Mars orbit. The probe and associated equipment are carried into orbit. The propellants for capture are given by equation A-16, with the exceptions noted above.

The probe is released from circular orbit for entry and landing. The rendezvous propellants weight is computed by

$$W_{pr} = (W_{gc} - W_{prob} - W_{padp} - W_{psc}) \left(1 - e^{-\Delta V_r / g I_{spr}}\right) \quad \text{(lb)} \qquad \text{(A-27)}$$

The spacecraft weight for initiation of the Mars escape maneuver is

$$W_{me} = W_{gc} - W_{prob} - W_{padp} - W_{psc} - W_{pr} - W_{lo} + W_{sc} - W_{je} \quad \text{(lb)} \qquad \text{(A-28)}$$

Equations A-10 through A-14 are also employed to optimize powerplant jettison weight prior to Mars capture in the case where the chemical Mars escape option is used.

A.10 SEP SPIRAL MARS ESCAPE PERFORMANCE

The algorithm for SEP spiral Mars escape used in the program is that derived by Melbourne (ref. 7). The equations are outlined below.

The power in Mars orbit is given by

$$P_{mo} = P_{1AU}\ (.55) \qquad \text{(kw)}$$

where 0.55 reflects the reduction in solar array power due to the increased distance from the sun.

The initial orbiter/bus vehicle weight to begin spiral escape is given by the relationship

$$W_{mo} = W_{me} - W_j \qquad \text{(lb)} \qquad \text{(A-29)}$$

where the empty Mars braking and orbit rendezvous stage weight W_j is computed by

$$W_j = \left(\frac{1-m_j}{m_j}\right) (W_{pc} + W_{pr}) \qquad \text{(lb)} \qquad \text{(A-30)}$$

where m_j is the stage propellant mass fraction.

The time required to escape Mars by the SEP spiral maneuver is given by

$$T = \frac{c^2 \Gamma \zeta W_{mo}}{172.8\ (\eta) P_{mo}} \qquad \text{(days)} \qquad \text{(A-31)}$$

where Γ = Melbourne's correction factor.

The parameter Γ is given by the empirical equation

$$\Gamma = 1 - .76382(A)^{.24323} \qquad \text{(A-32)}$$

where A is the nondimensional thrust acceleration determined by the expression

$$A = \frac{.00980665\ a_m\ r_{po}^2}{GM_p} \qquad \text{(A-33)}$$

where a_m = thrust acceleration of the spacecraft in Mars orbit (g)
r_{po} = radius of Mars orbit (km)
GM_p = Mars' gravitational constant (km^3/sec^2)

The parameter ζ in equation A-31 is given by

$$\zeta = 1 - e^{-V_c/c} \qquad \text{(A-34)}$$

where V_c is the circular orbit velocity given by

$$V_c = (GM_p/r_{po})^{1/2} \qquad \text{(km/sec)} \qquad \text{(A-35)}$$

and c is the SEP jet exhaust speed in km/sec.

The SEP propellant fraction for escape is expressed as follows:

$$\mu_{pf} = 1 - \frac{1}{1 + \frac{W_{mo} J}{2000 \eta P_{mo}}} \qquad \text{(A-36)}$$

where J is the trajectory characteristic equivalent to the integral of the acceleration squared over the time of spiral-out. The expression for J is

$$J = 2000 \eta \left(\frac{P_{mo}}{W_{mo}}\right) \left(\frac{\Gamma\zeta}{1 - \Gamma\zeta}\right) \qquad (m^3/sec^2) \qquad \text{(A-37)}$$

where all variables have been defined. Finally, the SEP propellant weight for spiral-out escape is simply

$$W_{pso} = \mu_{pf} W_{mo} \qquad \text{(lb)} \qquad \text{(A-38)}$$

A.11 INBOUND PERFORMANCE COMPUTATIONS, SPIRAL ESCAPE OPTION

When the SEP spiral-out Mars escape option is employed, solar panels and ion thrusters (including associated power conditioning equipment) are jettisoned prior to the heliocentric inbound leg of the mission. These weights are defined by equations A-12 and A-13. The net Mars departure weight is given by

$$W_{mdn} = W_{mo} - W_{pso} - W_{sajm} - W_{itjm} \qquad \text{(lb)} \qquad \text{(A-39)}$$

The inbound SEP propellant weight is computed by

$$W_{pi} = K_{pi} W_{mdn} \quad \text{(lb)} \qquad \text{(A-40)}$$

where K_{pi} is the input SEP propellant fraction found from low thrust trajectory analysis.

The inbound attitude control expendables are approximated by

$$W_{iac} = K_{iac} W_{mdn} \quad \text{(lb)} \qquad \text{(A-41)}$$

where K_{iac} is an input attitude control expendables fraction.

The gross earth arrival weight then becomes

$$W_{gea} = W_{mdn} - W_{pi} - W_{iac} \quad \text{(lb)} \qquad \text{(A-42)}$$

A.12 CHEMICAL MARS ESCAPE OPTION

In this option the gross Mars departure weight is given by

$$W_{mdg} = W_{me}\, e^{-\Delta V_{mf}/g\ I_{spmd}} \quad \text{(lb)} \qquad \text{(A-43)}$$

where ΔV_{md} and I_{spmd} are respectively the ΔV increment and chemical specific impulse for Mars departure.

The propellants weight for the departure maneuver is given by

$$W_{pmd} = W_{me}\, (1 - e^{-\Delta V_{md}/g\ I_{spmd}}) \quad \text{(lb)} \qquad \text{(A-44)}$$

The jettison weight of the capture/rendezvous/departure stage is

$$W_j = \frac{1 - m_j}{m_j} (W_{pmd} + W_{pc} + W_{pr}) \quad \text{(lb)} \qquad \text{(A-45)}$$

Here the assumption is made that only a single propulsion module is used for the Mars capture, orbit rendezvous, and departure manuevers. Computation of W_j, the spent stage weight, is identical for all mission options except for the elliptical capture mode. For this option, circularization propellants and elliptical capture propellants are computed separately and are used as follows:

$$W_j = \frac{1 - m_j}{m_j} (W_{pmd} + W_{pcc} + W_{pr} + W_{pcm}) \qquad \text{(lb)} \qquad \text{(A-46)}$$

where W_{pcm} = propellants for elliptical capture maneuver (lb)
W_{pcc} = propellants for orbit circularization (lb)

Finally, the net Mars departure weight is given by

$$W_{mdn} = W_{mdg} - W_j \qquad \text{(lb)} \qquad \text{(A-47)}$$

The SEP propellants and attitude control expendables for the inbound trajectory are computed by equations A-40 and A-41 respectively. Gross earth arrival weight is computed by equation A-42.

A.13 EARTH CAPTURE OPTION

Prior to Earth capture, K_j percent of the remaining solar arrays are jettisoned. This jettisoned weight is computed by

$$W_{jea} = \alpha_{sa} K_j P_{1AU2} \qquad \text{(A-48)}$$

where K_j is the fraction of the solar arrays jettisoned and P_{1AU2} is the remaining power at earth arrival. This power is computed by

$$P_{1AU2} = \frac{C\, a_i\, g_o\, I_{spi}\, W_{gea}}{2\eta} + P_{aux} \qquad \text{(kw)} \qquad \text{(A-49)}$$

where C = .001356 and g_o = 32.174 ft/sec^2.

The SEP propellant tankage and feed system weight is given by

$$W_{sept} = K_T (W_{po} + W_{pi} + W_{pso}) \qquad \text{(lb)} \qquad \text{(A-50)}$$

where K_T is an input factor for mercury propellant (taken to be .03).

The gross earth capture weight is

$$W_{gec} = (W_{gea} - W_{jea})\, e^{-\Delta V_{ec}/g\, I_{spec}} \qquad \text{(lb)} \qquad \text{(A-51)}$$

where ΔV_{ec} is the velocity decrement (m/sec) required for earth capture and I_{spec} is the specific impulse of the chemical earth capture module.

The chemical propellants weight required for capture is

$$W_{pec} = (W_{gea} - W_{jea}) \; (1 - e^{-\Delta V_{ec}/g \; I_{spc}}) \qquad (1b) \qquad (A-52)$$

The net Earth capture weight is given by

$$W_{nec} = W_{gec} - \left(\frac{1 - m_{fec}}{m_{fec}}\right) \; W_{pec} \qquad (1b) \qquad (A-53)$$

where m_{fec} is the propellant mass fraction of the chemical propulsion module.

The net earth capture weight consists of the orbiter/bus spacecraft systems including telecommunications, data, guidance/navigation, attitude control, structure, etc. This weight also includes the sample canister payload plus the remaining portion of the SEP hardware which is defined by

$$W_{sep_c} = W_{pp} - \alpha_{sp} \; K_j \; P_{1AU} + W_{sept} - W_{sajm} - W_{itjm} \qquad (1b) \qquad (A-54)$$

The computed net spacecraft weight in earth orbit is then given by

$$W_{nss} = W_{nec} - W_{sepc} - W_{sc} \qquad (1b) \qquad (A-55)$$

The required spacecraft weight in earth orbit is given by

$$W_{nssc} = W_{fnss} + K_{nss} \; (W_{sam}) \qquad (1b) \qquad (A-56)$$

where W_{fnss} is a specified input fixed systems weight required to be returned to earth orbit. K_{nss} is a constant requiring W_{nssc} to be a function of sample return weight based on point design analysis.

The computer program computes a ΔW where

$$\Delta W = W_{nss} - W_{nssc} \qquad (1b) \qquad (A-57)$$

An iteration is made on the gross earth departure weight (W_g) until the ΔW is less than a specified tolerance.

A.14 DIRECT EARTH REENTRY OPTION

In the direct reentry option the reentry capsule weight is computed as an function of sample weight by the relationship

$$W_{rc} = W_{rcf} - K_{rc} W_{sam} \qquad \text{(lb)} \qquad \text{(A-58)}$$

where W_{rcf} is a fixed weight found from point design analysis. K_{rc} is a coefficient which relates reentry capsule weight to the sample weight (and size).

The net orbiter/bus module weight at earth arrival is then

$$W_{nss} = W_{gea} - W_{rc} - W_{sepc} \qquad \text{(lb)} \qquad \text{(A-59)}$$

In this option, the ΔW is again computed by equation A-57 and an iteration is performed on gross earth departure weight (W_g) until ΔW is less than a specified tolerance.