N72-12788

ASTRO SCIENCES

Report No. T-29

MARS SURFACE SAMPLE RETURN MISSIONS VIA SOLAR ELECTRIC PROPULSION

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by

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This work was performed for the Jet Propulsion Laboratory, California Institute of Technology, as sponsored by the National Aeronautics and Space Administration under Contract NAS7-100, Task Order No. RD-26

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August 1971

FOREWORD

This Technical Report is the final documentation on all data and information required by Task 7: Mars Surface Sample Return Missions. The work herein represents one phase of the study, Support Analysis for Solar Electric Propulsion Data Summary and Mission Applications, conducted by IIT Research Institute for the Jet Propulsion Laboratory, California Institute of Technology, under JPL Contract No. 952701. Tasks 9 and 10 of this study will be reported separately.

SUMMARY AND CONCLUSIONS

This report describes the characteristics and capabilities of solar electric propulsion (SEP) for performing Mars Surface Sample Return (MSSR) missions. The scope of the study emphasizes trajectory/payload analysis and the comparison of mission/system tradeoff options. Questions concerning mission science objectives, instrumentation, operations and spacecraft design are not treated herein. Subsystem weights and scaling relationships used in the present study are based on previous independent studies.

The MSSR mission is examined only for the 1981-82 launch opportunity. This opportunity seems to be realistic in light of current schedules for Mars exploration and SEP technology development. Several other study constraints which bear directly on the results obtained are: (1) return samples in the range 5-25 kg, (2) use of lifting (offset C.G.) atmospheric entry at Mars which allows a low ratio (1.25) of entry weight to landed weight, and (3) rendezvous and docking in Mars orbit.

Major results of the study are presented as performance curves of Earth departure mass versus sample size for a number of different mission/system options. These options represent a spectrum of trip time, launch vehicle capability, combinations of low-thrust and ballistic maneuvers, chemical retro type, and Earth recovery mode. Six mission concepts or baseline examples are selected from the parametric data. Table S-1 summarizes the pertinent aspects of these baseline examples. All assume the direct entry option for the Mars lander vehicle, the Earth orbit capture mode for sample capsule recovery (555 x 9000 km altitude orbit), and the solid propulsion system for retro maneuvers.

TABLE S-1

MSSR MISSION SELECTION SUMMARY

BASELINE MISSION	1	2	3	4	5	6
TRAJECTORY PROFILE						
MISSION DURATION (DAYS)	1155	1055	950	960	680	600
MARS STAY TIME (DAYS)	34	49	22	310	10	20
EARTH-MARS TRANSFER	SEP	SEP	SEP	BALLISTIC	BALLISTIC	BALLISTIC
MARS CAPTURE	SEP	SOLID RETRO	SEP	SOLID RETRO	SOLID RETRO	SOLID RETRO
MARS ESCAPE	SEP	SEP	SEP	SOLID RETRO	SOLID RETRO	SOLID RETRO
MARS-EARTH TRANSFER	SEP	SEP	SEP	SEP	SEP	SEP / VENUS SWINGBY

- DIRECT MARS ENTRY -

- EARTH ORBIT CAPTURE RECOVERY -

DELIVERY SYSTEM						
LAUNCH VEHICLE	TITAN III D / CENTAUR	TITAN III D (7)/ CENTAUR	TITAN III D (7)/ CENTAUR	TITAN III D/ CENTAUR	INT-20	INT-20/ CENTAUR
NO. OF LAUNCHES	1	1	1	2	1	1
SAMPLE SIZE (KGS)	10	10	10	20	10	25
EARTH DEPARTURE MASS (KGS)	5278	6425	5116	4037/3225	11300	15800
SEP POWER, 1 AU (KW)	18.5	22.5	20.5	3.9	27.8	19.2
SEP PROPULSION TIME (DAYS)	784	668	586	287	245	157

Examples 1 through 4 are distinguished by the use of Titan class launch vehicles, a mission duration of 2.5 to 3 years, and SEP being used for most mission phases. Examples 5 through 6 require Intermediate-20 class vehicles, have a shorter trip time of 1.5 to 2 years, and use SEP only for the return interplanetary transfer.

It is possible to return a 10 kg sample using the Titan IIID/Centaur single launch mode provided that SEP is employed for both Mars capture and escape maneuvers (Example 1). The capture spiral time is 98 days; this is approximately the time lag between lander separation and the rendezvous/docking maneuver. The stay time of 34 days refers to the time spent in a 1000 km Mars orbit by the orbiter bus. Example 3 is similar except that a Titan IIID(7)/Centaur is required and the mission duration is 200 days shorter. A hybrid option (Example 2) also employs the 7-segment Titan/Centaur but uses a chemical retro for Mars capture. This would alleviate the problem of orbiter bus/lander communications and the time lag between lander separation and rendezvous. The SEP power requirement for the first three mission concepts is about 20 kw and the propulsion on-time is 60-70% of the mission duration. The dual-launch mode (Example 4) uses a small (4 kw) SEP stage only for the return transfer to Earth. This type of mission could be performed ballistically with two Titan IIID/Centaur vehicles; the flight time is only 100 days longer.

The shorter mission examples (Examples 5 and 6) require a relatively high energy Earth-Mars transfer. SEP is not recommended for this phase of the mission since the power requirements are prohibitively high for large Earth departure mass. Even when SEP is used only for the return transfer the power requirement is at least 19 kw. Example 6 is a 600-day mission which will return a 25 kg sample. This mission uses a Venus swingby with the SEP system operating for only 157 days on the Mars-Venus leg. The required launch vehicle is the Intermediate-20/Centaur; the margin of launch vehicle capability is about 4000 kg.

In conclusion, the study has shown that solar electric propulsion can be used effectively to accomplish the MSSR mission. Performance advantages over all-ballistic (chemical propulsion) systems are <u>either</u> a smaller launch vehicle requirement for comparable trip time and sample size, <u>or</u> a significant reduction in trip time for comparable launch vehicles and sample size. The latter advantage is not generally available when a Venus swingby opportunity is employed.

TABLE OF CONTENTS

LIST OF FIGURES

LIST OF FIGURES (Continued)

LIST OF TABLES

MARS SURFACE SAMPLE RETURN MISSIONS VIA SOLAR ELECTRIC PROPULSION

1. INTRODUCTION

1.1 Study Background

A logical follow-up of the Viking project would be a mission to return samples of the Martian surface to Earth. The recent success of the Soviet Luna 16 mission has demonstrated that automated sample return is a technically feasible concept. Thus, there is renewed interest within NASA in automated Mars surface sample return (MSSR) missions.

Previous studies by Niehoff (1967) and Odom (Feb. 1970) have dealt with MSSR missions in the mid to late '70's. These studies were concerned primarily with ballistic-type missions using the Saturn V class launch vehicle. A follow-on study by Odom (Nov. 1970) included the use of solar electric propulsion (SEP) and smaller classes of launch vehicle, emphasizing mission opportunities in the mid to late '70's. The present study described in this report is based, in part, on unpublished work initiated in November 1970 for the Planetary Programs Office, OSSA.

1.2 Study Objectives and Approach

The objectives of this study are the following:

- Determine the capability of solar electric propulsion for performing Mars surface sample return (MSSR) missions.

- Identify various mission and system design options and show their performance tradeoffs.

- Match mission requirements with candidate launch vehicles of the Titan III and Intermediate-20 class.

- Present results in terms of such significant parameters as sample size, flight time, SEP power required, and propulsion on-time.

- Define the most promising application of solar electric propulsion for reducing mission duration and/or launch vehicle requirements.

The set of guidelines and constraints used throughout the study are the following:

- 1981-82 launch opportunity.

- Solar electric stage used for at least one propulsive phase of the mission. Assume 3500 sec I_{sp} for all SEP stages.

- Return samples in the range 5-25 kg.

- Mars orbit rendezvous mode.

- Mars orbiter and lander science is secondary to primary objective of sample return.

- Use of lifting entry (offset C.G.) at Mars.
- Use of Northrop mass scaling assumptions for lander/ascent vehicle.
- Earth storable propellants for Mars ascent stage (e.g., N_2O_4/Aerozine-50, I_{sp} = 310 sec).
- Limit Earth reentry speed to 40,000 ft/sec.
- Utilize existing trajectory date where possible.

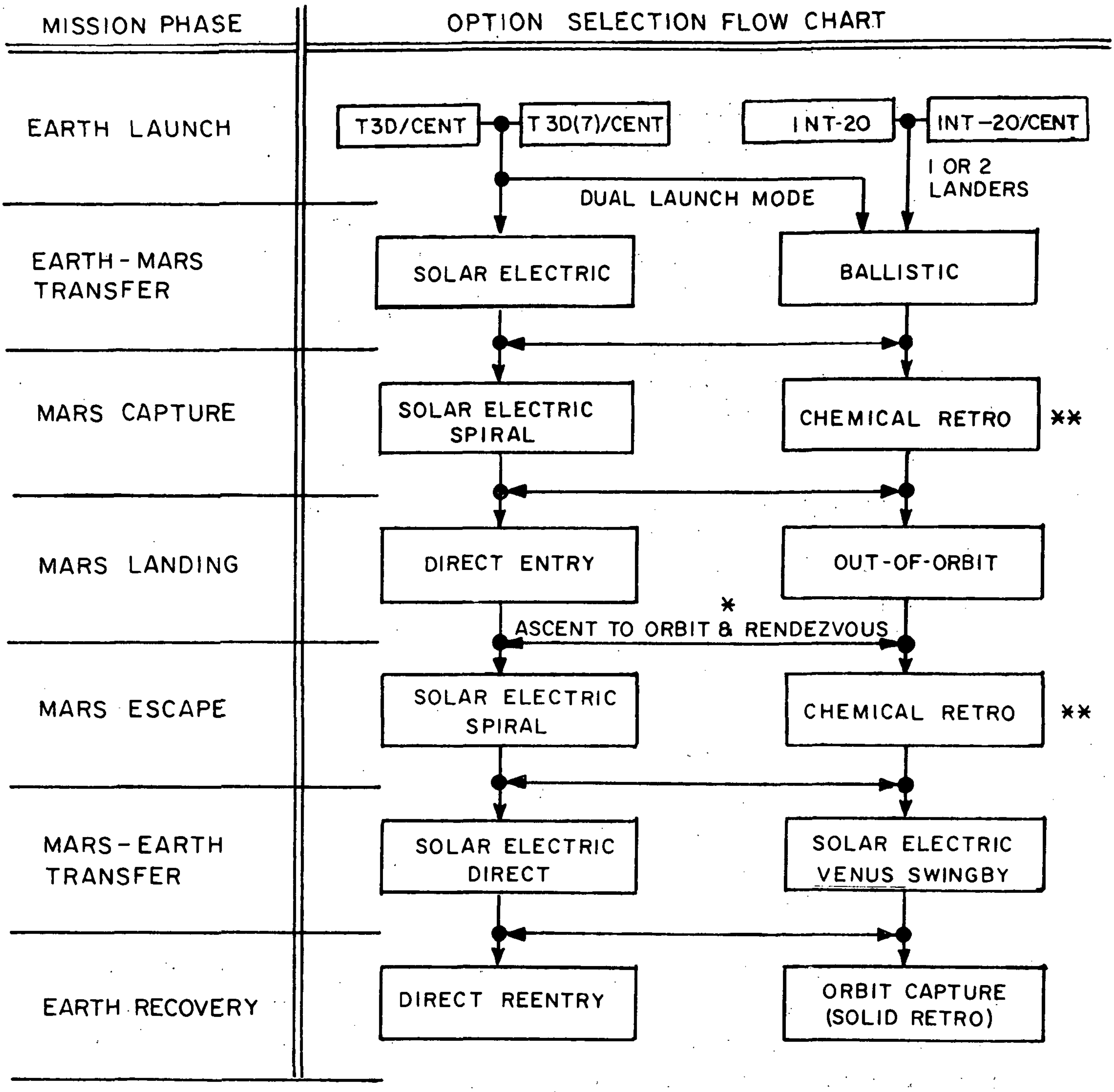
1.3 Mission Phase Options

The MSSR mission profile was separated into the following distinct phases:

Earth launch,
Earth-Mars transfer,
Mars capture,
Mars landing,
Mars escape,
Mars-Earth transfer, and
Earth recovery.

Figure 1 depicts, in flowchart form, the options which can be associated with each phase of the mission profile. The selection of various options for each phase was made keeping in mind the study constraints listed above. It will be seen in Sections 4 and 5 that not all possible combinations of options suggested in Figure 1 were considered in this study.

FIGURE I. OPTION ARRAY SET



* ASCENT PROPULSION - EARTH STORABLE PROPELLANT

** CHEMICAL RETRO OPTIONS
1.) SOLID RETRO, 2.) SPACE STORABLE PROPELLANT

Earth Launch

The options considered for the Earth launch phase were the Titan IIID/Centaur, Titan IIID(7)/Centaur, Intermediate-20 and Intermediate-20/Centaur launch vehicles using a 100 N.M. parking orbit.

A "sub-option" which was considered for the Titan III option is that of the dual launch. With this concept, the Mars lander and Earth return stage are launched in separate launch vehicles. This concept will be discussed in more detail in Section 4.

Earth-Mars Transfer

Two types of interplanetary transfers were considered for this phase; solar electric low-thrust and ballistic. Each of these options can be classified by either of two types of transfers: the so-called direct and indirect solar electric transfers, and the opposition and conjunction type ballistic transfers. The main difference between direct and indirect, and also opposition and conjunction, is that the former type transfer can be characterized as having relatively higher Earth escape and Mars approach velocities, and shorter flight times than the latter type transfers. Also, indirect SEP transfers are characterized by a heliocentric travel angle greater than 360° (i.e., more than one revolution about the Sun).

Note that for Earth-Mars transfers using the Intermediate-20 class vehicle, SEP was not needed to achieve the desired outbound payload.

Mars Capture

All systems options require a Mars orbiting bus for the return phase of the missions.

The two options considered for capture into Mars orbit were a SEP low-thrust spiral maneuver, and a chemical high-thrust retro maneuver. For the chemical retro case, both solid propellant and space storable liquid propellant stages were considered. Capture velocity requirements for the assumed orbit are discussed in Section 2.

Mars Landing

Two options were considered for Mars entry. For direct entry, the lander enters the Martian atmosphere directly from the hyperbolic approach trajectory, having separated from the orbiter bus before it maneuvers into Mars orbit. The second option is to have the lander enter Mars orbit with the orbiter and then descend from orbit to the Martian surface.

As compared to the orbit capture option, direct entry would have more critical approach and entry guidance and control requirements and no landing site selection from orbit, but a lower orbit capture stage requirement. It was decided that savings in capture stage weight far outweighed critical guidance and lack of site selection. Therefore, the direct entry option was used almost exclusively throughout this study.

System scaling assumptions for the entry vehicle are discussed in Section 2.

Mars Escape

The same options were considered for escape from Mars orbit as for capture into orbit, i.e., SEP low-thrust spiral or chemical high-thrust retro. Escape velocity requirements are discussed in Section 2.

Mars-Earth Transfer

The return-to-Earth transfers considered in this study are essentially of the direct-type SEP low-thrust. A Venus swingby mode was also examined in which a SEP Mars-Venus transfer was matched to a Venus-Earth free return trajectory*.

Earth Recovery

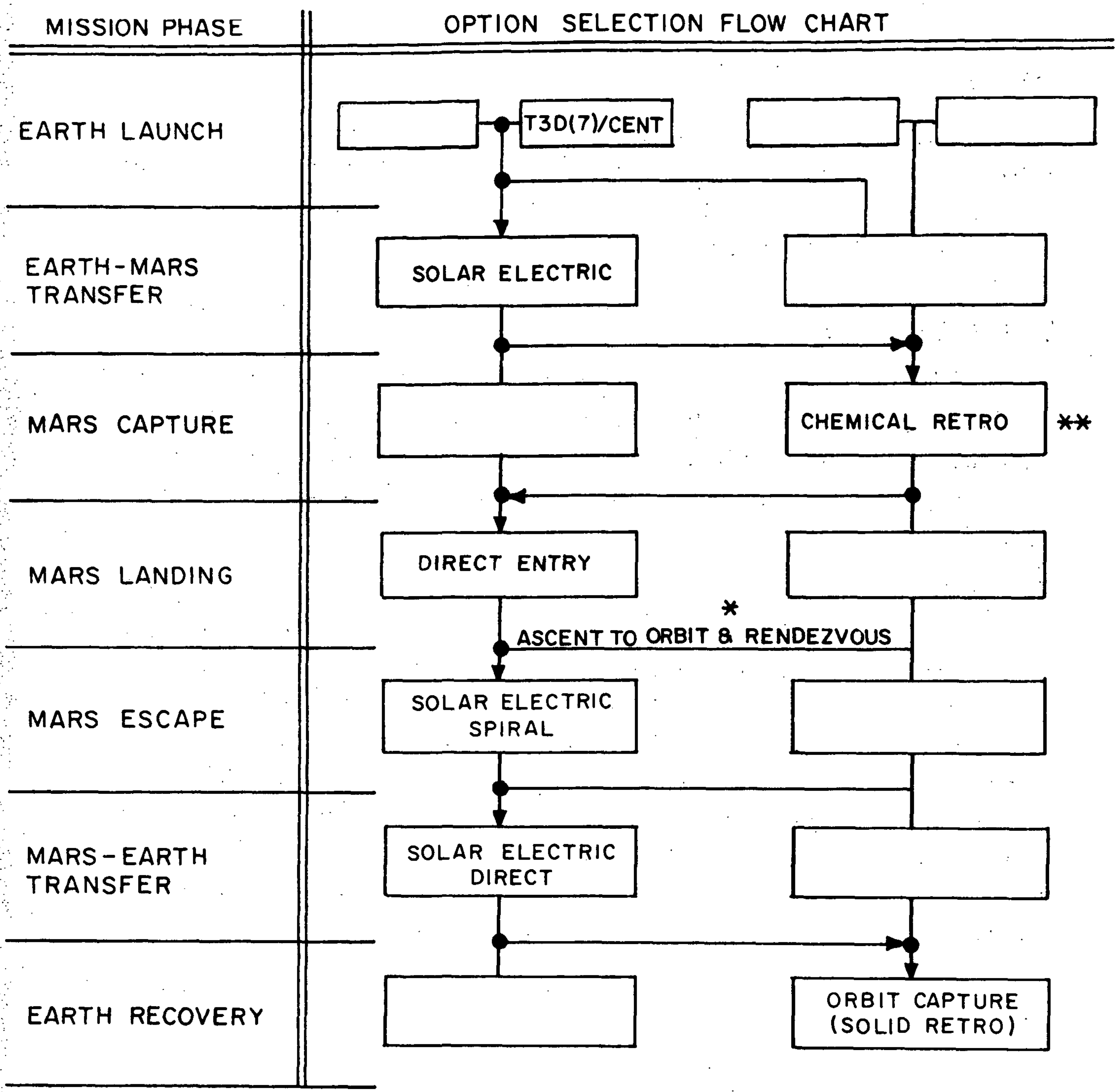
Two methods for recovery of the sample container were considered. The first, direct entry, assumes that the sample container enters the Earth's atmosphere directly from the hyperbolic approach trajectory. No consideration was given as to whether the capsule should be air snatched or surface recovered. The other available option is to have the capsule put into a loose Earth orbit via a chemical retro stage, and then recovered from orbit, perhaps by a manned vehicle. Only a solid propellant stage was considered for performing this maneuver.

Option Selection Example

Figure 1A presents an example of how the options may be selected for the various phases of a mission. The particular example shown uses a Titan IIID(7)/Centaur single launch with an SEP Earth-Mars transfer. Mars orbit is via chemical retro and

*Searches for Venus-Earth SEP transfers were not made due to limited time available for the study.

FIGURE 1A. OPTION SELECTION EXAMPLE



* ASCENT PROPULSION-EARTH STORABLE PROPELLANT

** CHEMICAL RETRO OPTIONS

1.) SOLID RETRO, 2.) SPACE STORABLE PROPELLANT the landing is by direct entry. Escape from Mars orbit is by SEP spiral and the Mars-Earth transfer is SEP. Finally, orbit capture of the sample container is selected for the Earth recovery phase.

Report Organization

The remaining sections will discuss in detail the analysis of MSSR missions. Section 2 presents system scaling assumptions and mission velocity requirements used throughout the study. Section 3 will show characteristics of the solar electric low-thrust transfers and maneuvers that apply to MSSR missions. Section 4 presents a set of conceptual mission characteristics in parametric data form. And Section 5 contains design-point mission examples using the data from Section 4.

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2. SYSTEM SCALING ASSUMPTIONS

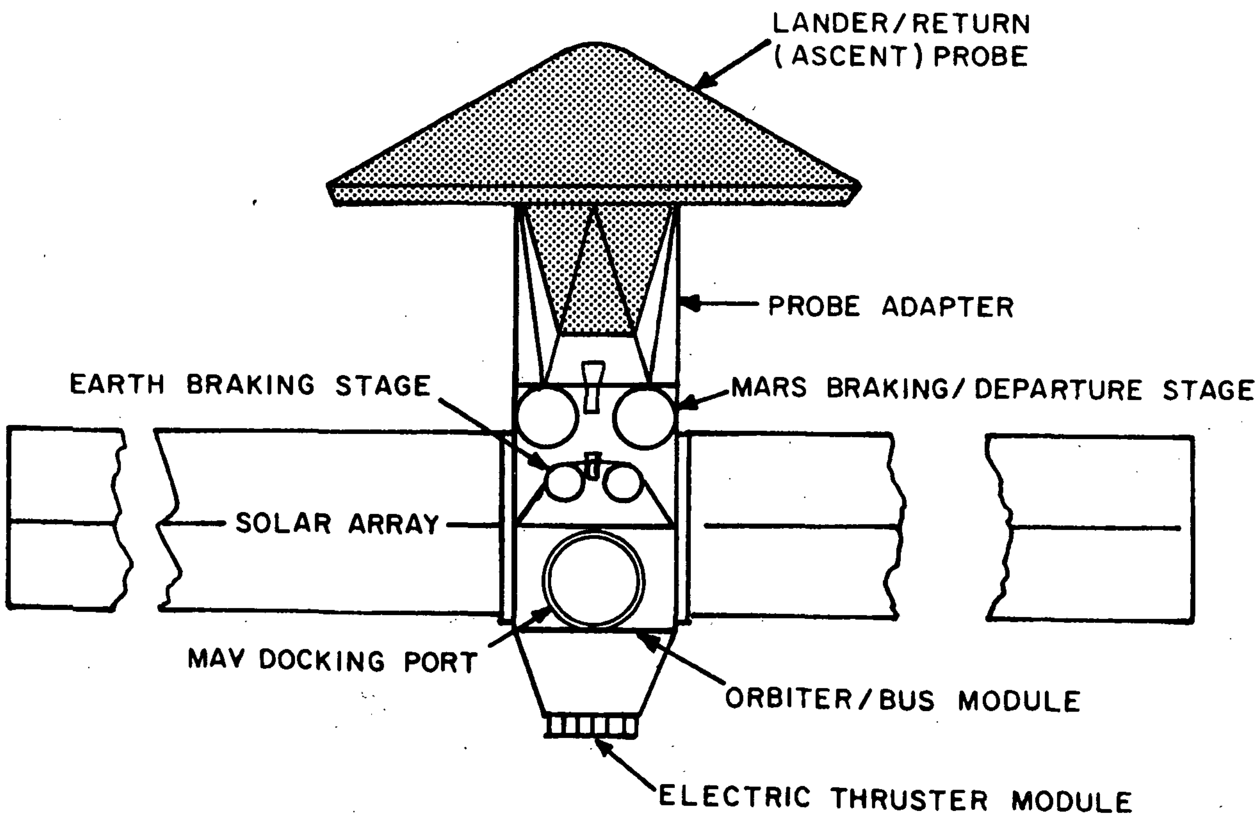
2.1 Launch Vehicle Data

Figure 2 presents curves of maximum injected mass as a function of hyperbolic launch velocity for the four launch vehicles used in this study. The data for the Titan III class vehicles was provided by the Jet Propulsion Laboratory, and the data for the Intermediate-20 class vehicles was taken from the 1971 Launch Vehicle Estimating Factors Handbook.

2.2 Stage Mass Data

Table 1 presents data which was assumed for scaling of various systems for MSSR missions.

The following sketch presents a possible system configuration concept for a MSSR mission. The system shown would employ SEP for both outbound and inbound interplanetary transfers, direct entry of the Mars lander/ascent probe, chemical propulsion for both capture and escape at Mars, and Earth orbit recovery of the sample container. The schematic is taken from Odom (Nov. 1970).



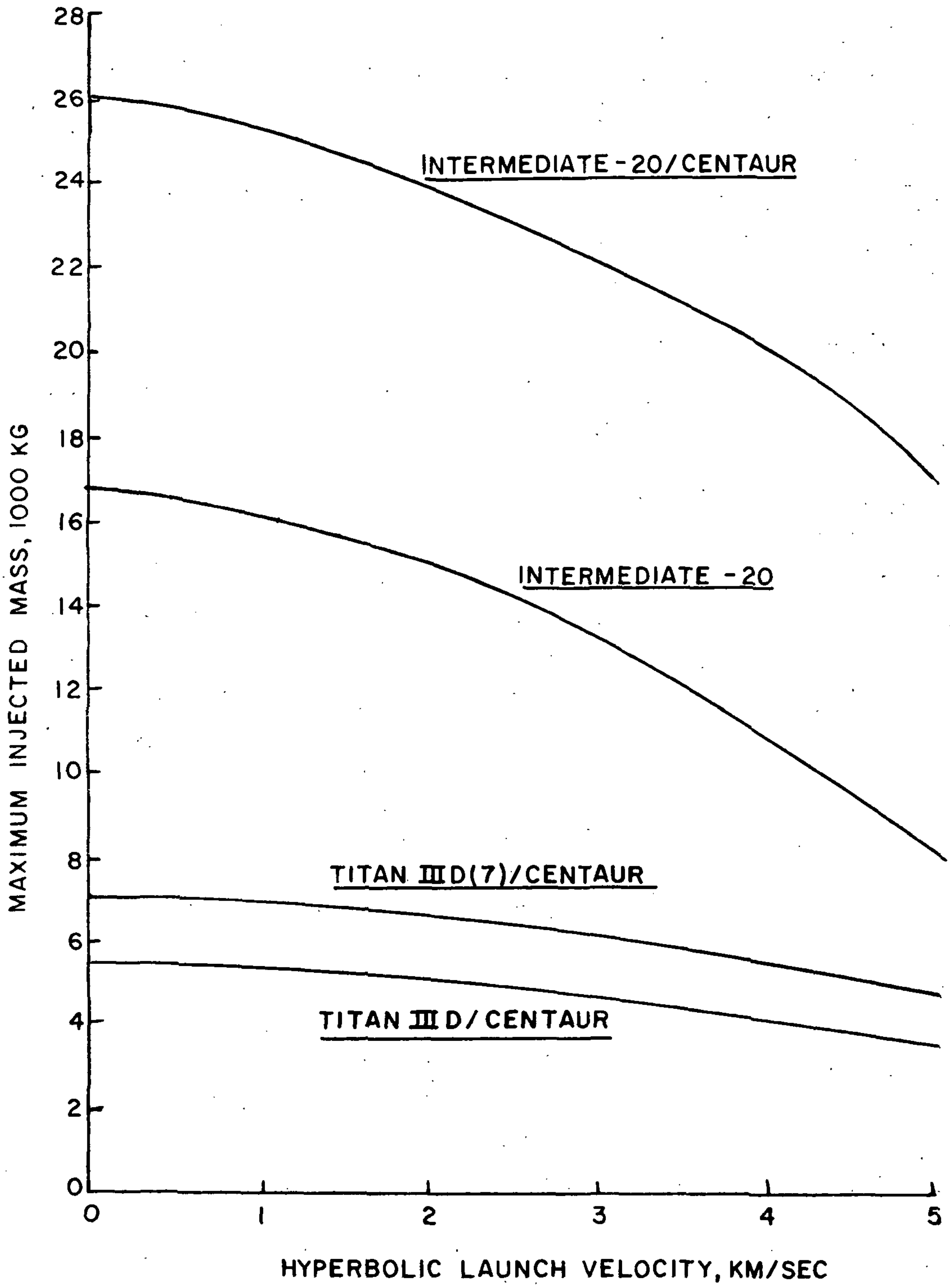


FIGURE 2. LAUNCH VEHICLE PERFORMANCE CURVES

TABLE 1

SYSTEM MASS SCALING RELATIONSHIPS

SOLAR ELECTRIC PROPULSION		
Specific Mass	30 kg/kw	
Tankage Factor	3% of Propellant Loading	
Specific Impulse	3500 sec	
Overall Efficiency	66%	
CHEMICAL RETRO STAGES	**SPECIFIC IMPULSE**	**INERT FRACTION**
Solid Propellant	300 sec	0.11
Space Storable Propellant	400 sec	0.20
MARS ASCENT VEHICLE		
Earth Storable Propellant	310 sec	0.20 (1st stage) 0.25 (2nd stage)
LANDER/PROBE SUPPORT MASSES		
Sterilization Canister Probe Mounting Structure	12% of Entry Vehicle Mass (25% for Two Landers)	
MARS ENTRY/DESCENT MASS RATIO		
Lifting (Offset C.G.) Entry	$\frac{\text{Entry Mass}}{\text{Gross Landed Mass}} = 1.25$	
SPACECRAFT EQUIPMENT MODULE		
Interplanetary Cruise and Orbiter/Bus	453 kg (outbound) 340 kg (inbound)	

The propulsion stage data, both SEP and chemical, is representative of current to mid-1970's technology. The specific impulse of 3500 seconds for SEP is a constraint of the study*. As mentioned previously, both space storable and solid propellant stages were considered for the retro capture and escape maneuvers at Mars. As will be seen in Section 4 space storable propulsion systems provide better performance, but based on current technology, are more costly to develop than solid propulsion systems. Thus a tradeoff based on cost-effective performance would have to be made prior to selection of a particular system. The use of an Earth storable two-stage system for Mars ascent is based on results of a previous study by Niehoff (1967).

The sterilization canister, or bioshield, provides contamination protection to the lander/ascent vehicle from sterilization at Earth to Mars arrival. The probe mounting structure provides the mechanical interface between the lander/ascent vehicle and the orbiter bus. It also supports the sterilization canister. The combined mass of the two systems is taken as 12% of a single lander/ascent probe's mass.

The entry technology assumed in this study was that derived from the entry analysis performed by Northrop (Odom, Nov. 1970). The deceleration system employs a blunt cone aeroshell utilizing lifting entry, an attached inflatable decelerator, and a terminal liquid propulsion system. The entry weight to landed weight mass ratio is assumed to be 1.25; this low mass ratio is a critical factor in allowing the use of Titan class launch vehicles.

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*A 3500 sec specific impulse is representative of current ion thruster development. This value may not be optimum for the MSSR mission; the effect of changing I_{sp} should be studied.

The 453 kg spacecraft equipment module (outbound) allows for such items as structure, telecommunications, navigation and attitude control, and a certain amount of orbiter science. For the inbound transfer, 113 kgs is discarded prior to leaving Mars orbit. This would include such items as the now-unnecessary docking mechanism and structure, and the orbiter science instruments and associated equipment.

Figures 3 and 4 present the masses of various stages of the Mars lander vehicle as functions of surface sample mass. The scaling of all stages was assumed to be linear with sample size. Note that the Earth recovery systems, i.e., the aerobraking system for direct reentry, or the solid propulsion stage for orbit capture mode, remain with the orbital bus in Mars orbit. The sample container is then transferred to the recovery system upon rendezvous of the Mars ascent vehicle with the orbiter bus.

As an example of sizing the various systems of the lander/ascent probe, consider a sample size of 10 kgs. From Figure 3, the total entry vehicle mass is 2803 kgs and the gross landed mass is 2345 kgs. Thus, the Mars aerobraking system and descent propellants total 558 kgs. The ascent vehicle mass is 1330 kgs, which then allows 915 kgs for the lander. Some of the lander subsystems and their approximate masses (Odom, Nov. 1970) are: structure and landing gear, 300 kg; guidance, control and communication, 140 kg; power, 100 kg; terminal descent propulsion hardware, 75 kg. A portion of the lander's mass may be allocated for in situ science instruments and perhaps a small surface rover.

2.3 Mission Velocity Data

Table 2 presents the velocity data which were used in this study for the scaling of various systems. The data for the Earth-Mars SEP transfers were obtained by scanning the Earth-Mars transfer data from Horsewood (1970) for transfers with

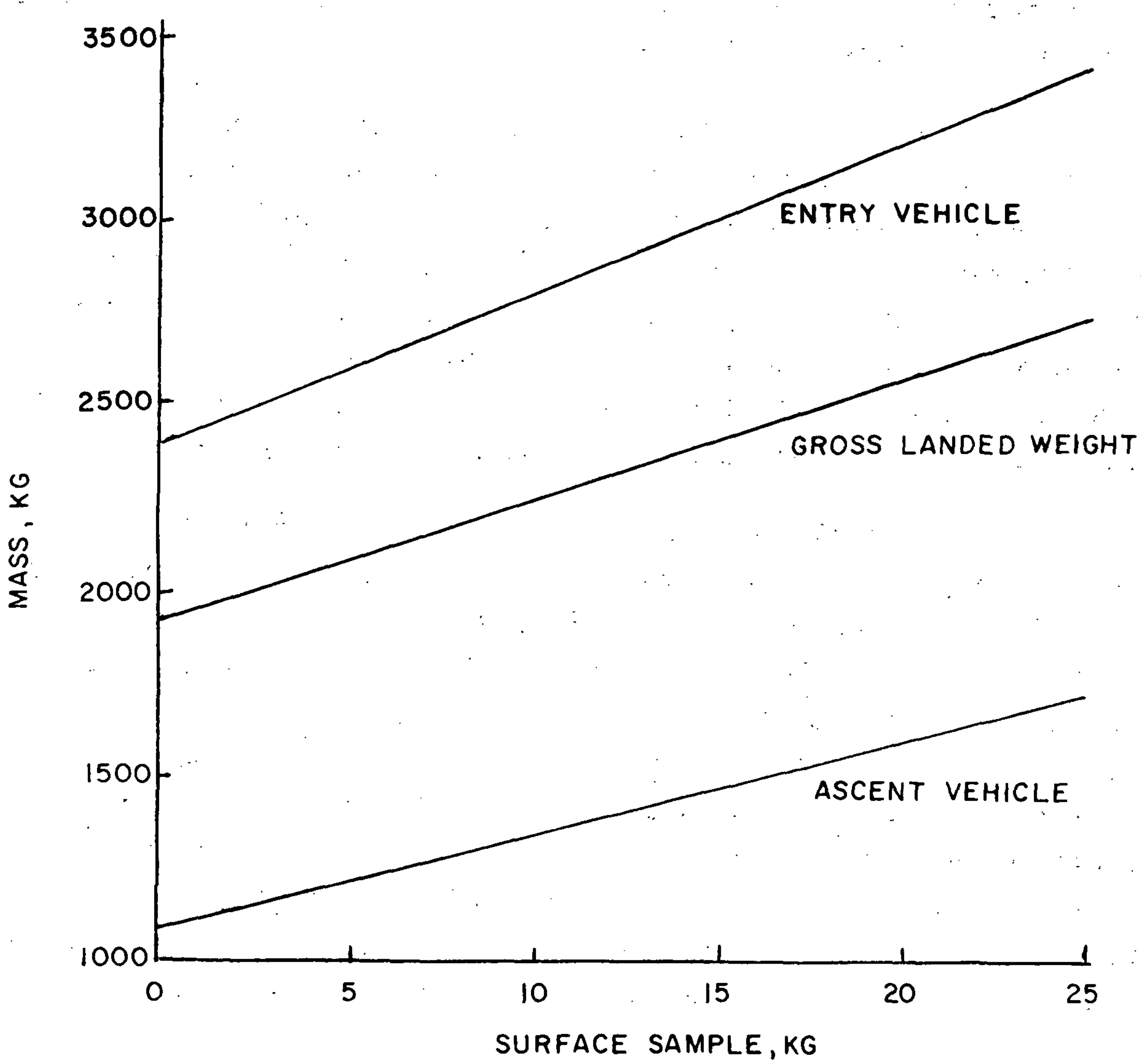


FIGURE 3. MASS SCALING FOR LANDER/ASCENT PROBE
MARS SAMPLE RETURN MISSION

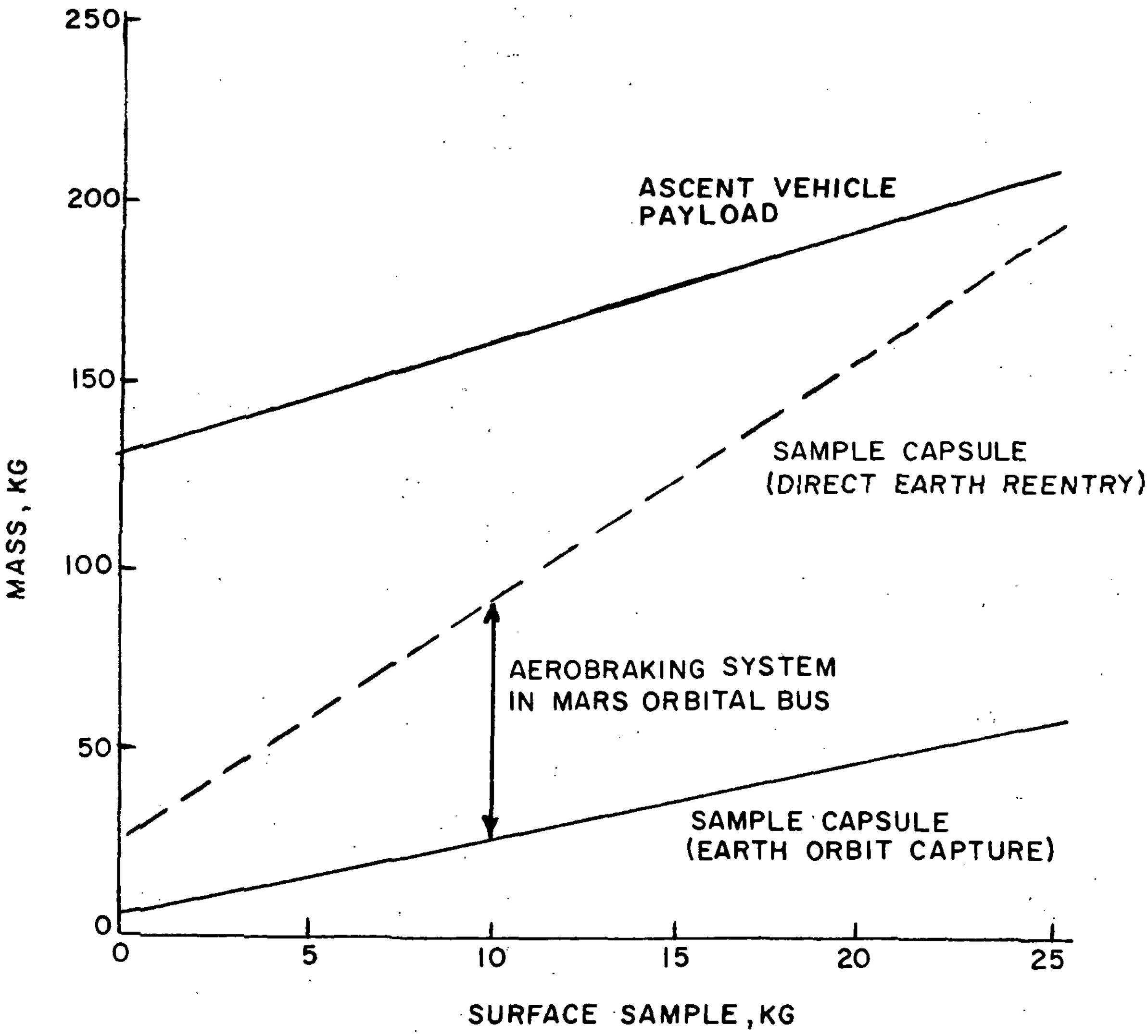


FIGURE 4. MARS SAMPLE CAPSULE AND ASCENT VEHICLE PAYLOAD

TABLE 2

MISSION VELOCITY DATA

EARTH-MARS TRANSFERS	SOLAR ELECTRIC (DIRECT)	SOLAR ELECTRIC (INDIRECT)	BALLISTIC (CONJUNCTION)	BALLISTIC (OPPOSITION)
Departure Hyperbolic Velocity	3.0 km/sec	0.91 km/sec	3.25 km/sec	3.35 km/sec
Arrival Hyperbolic Velocity	1.0	0.17	3.0	4.75
Mars Entry Speed	5.0	5.0	5.8	6.8
Capture ΔV (1000 km orbit)	1.41	1.30	2.22	3.37

MARS ASCENT TO ORBIT (1000 km) Characteristic $\Delta V = 4.6$ km/sec

MARS-EARTH TRANSFERS	DIRECT SEP	SEP TO VENUS SWINGBY
Departure Hyperbolic Velocity	0 km/sec	2.0 km/sec
Escape ΔV from 1000 km Velocity	1.30 km/sec	1.73 km/sec
Venus Hyperbolic Velocity	--	11.08 km/sec
Venus Swingby Distance	--	1.45 Planet Radii
Earth Arrival Hyperbolic Velocity	5.0 km/sec	5.0 km/sec
Earth Reentry	40,000 ft/sec	40,000 ft/sec
Orbit Capture ΔV (555 x 9000 km)	2.95 km/sec	2.95 km/sec

appropriate trip time and velocities. The SEP trajectories will be discussed in more detail in Section 3.

The data for the ballistic Earth-Mars transfers were taken from the Northrop study as representative of ballistic transfers for the launch opportunity.

By comparing the velocity data between the ballistic and SEP transfers, it can be seen that the SEP mode has, in general, lower velocity requirements. In particular, this is most evident with the indirect SEP transfer. This effect is mainly due to the longer flight times of the SEP transfers considered for this study, and also to the general nature of SEP trajectories.

The characteristic ΔV of 4.26 km/sec for ascent to a 1000 km altitude circular orbit from the Martian surface is the result of a numerically integrated trajectory solution. A circular orbit is desirable for the orbiter bus because of the requirement for automated rendezvous with the ascent probe. The 1000 km altitude was chosen both as a rough tradeoff point between capture stage and ascent stage requirements and because of a sterilizable propellant constraint below 1000 km.

For the Mars-Earth transfers, VHL at Mars was arbitrarily set to 0 km/sec for direct SEP transfers, and 2 km/sec for the SEP/Venus swingby transfer mode. The VHP at Earth was set to 5 km/sec to correspond with an Earth reentry speed of 40,000 ft/sec.

Finally the capture orbit at Earth of 555 km x 9000 km altitude is similar to that used in the Northrop study. The orbit selection was based on the use of an orbit-launched, fully loaded Apollo CSM, or a system such as the proposed Earth Orbital Space Tug (if it is operational by the early '80's), for retrieval of the sample container.

3. SOLAR ELECTRIC TRAJECTORY REQUIREMENTS

3.1 Interplanetary Transfers

The analysis of round-trip missions requires a survey of compatible outbound and return trajectories. SEP trajectory data was generated to satisfy the 1981-82 launch opportunity as specified in the list of study constraints. The CHEBYTOP computer program was employed for this purpose (Hahn, et al. 1969).

A convenient way of presenting the trajectory energy requirements is shown in Figure 5. The energy measure used is "J" which is given by the time-integral of $a^2/G(R)$, where $a(t)$ is the thrust acceleration magnitude and $G(R)$ is the normalized solar power (relative to $R = 1$ a.u.) available to the thrust subsystem. The parameter J is related to the propellant expenditure; suffice it to say that the lower the J value the lower the propellant expenditure. Figure 5 shows constant J contours plotted in a grid of Earth launch and arrival dates (abscissa) and Mars arrival and departure dates (ordinate). The outbound transfers are of the direct type with the exception of the 550-day indirect transfer point shown. Return transfers to the right of the slanted broken line are direct while those to the left are indirect. This type of data map is convenient for determining suitable launch and arrival dates and the effect of varying trip time and stay time at Mars. A 950-day mission is shown as an example, departing Earth on Julian date 2444950 (Dec. 11, 1981), arriving Mars 2445300 (Nov. 26, 1982), staying 240 days, departing Mars 24445540 (July 28, 1983), and returning to Earth on 2445900 (July 18, 1984). It will be noted that both the outbound and return legs are near-minimum energy direct transfers. Furthermore, the steep-ridge characteristic of the J contours indicates that an attempt to reduce trip time below 950 days will

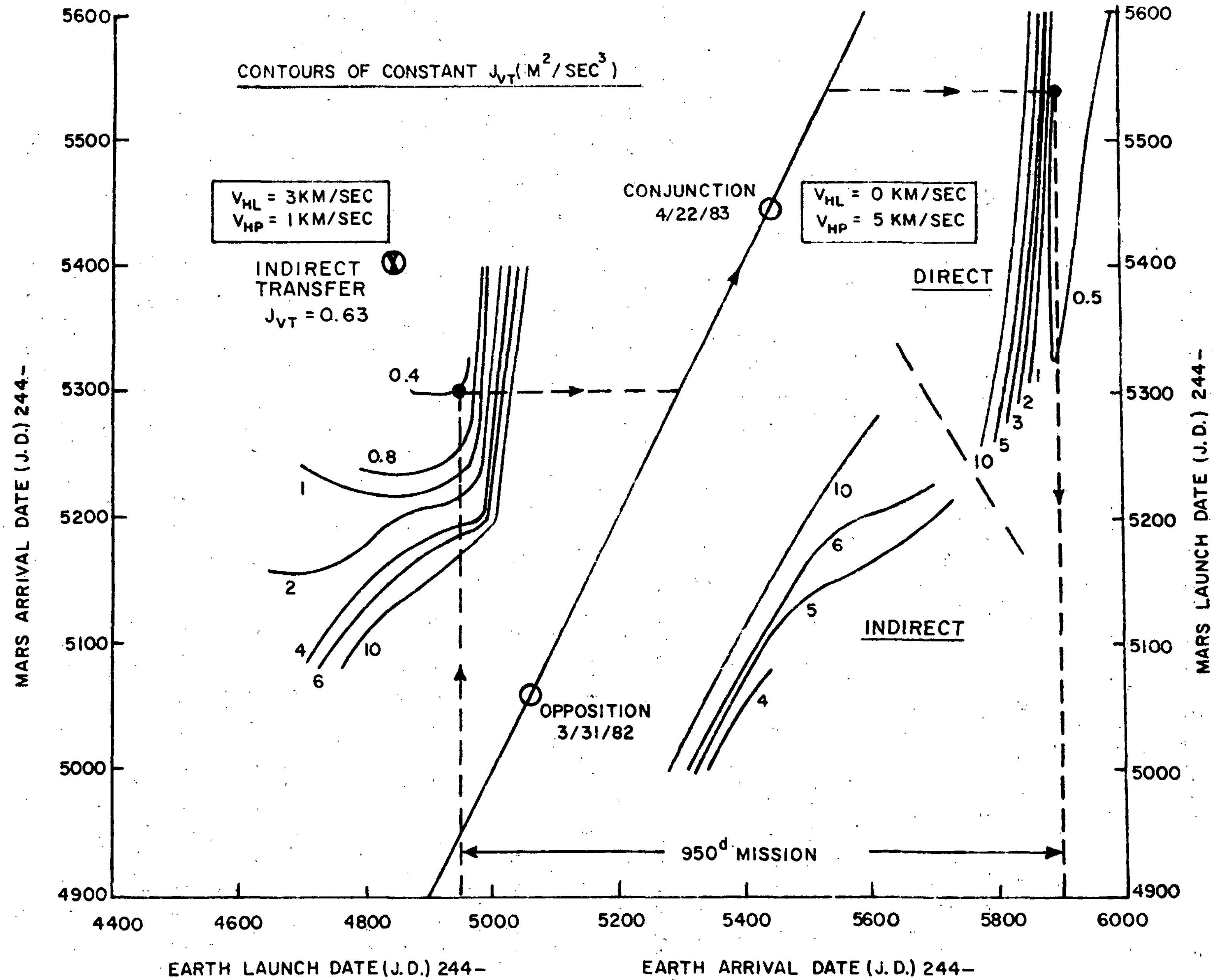


FIGURE 5. SOLAR ELECTRIC TRAJECTORY REQUIREMENTS FOR MARS SAMPLE RETURN MISSION, 1981-82 OPPORTUNITY

meet with a rapidly increasing energy requirement. It will be shown subsequently that the shorter missions require a much larger spacecraft mass at Earth departure than would be possible using Titan/Centaur launch vehicles.

Upon examining the J contour map, a set of several outbound and return transfers were selected for further analysis of sample return capability. These are shown in Figure 6 where net mass fraction is plotted as a function of normalized power/mass ratio. For the return transfers, m_o is the initial mass of the return vehicle after Mars escape, but P_o is still the SEP power referred to a distance of 1 a.u. It is desirable to choose a design point providing a maximum value of net mass fraction. As seen from Figure 6 this generally occurs near the minimum value of P_o/m_o, below which the thrust acceleration is insufficient to accomplish the trajectory in the given time of flight. Design points to the right of this cut off would have decreasing propulsion on-time requirements as P_o/m_o increases. In the case of the return transfers the design point may not be chosen arbitrarily. For example, if the same SEP system is utilized for both the outbound and return legs, then P_o is fixed and the ratio P_o/m_o is determined by the resulting mass at Mars departure. In such cases P_o/m_o is typically well to the right of the minimum acceleration cut off. While this may not be detrimental to the mission objectives, it does raise the possibility of considering a staged SEP design. In Section 4 of this report several combinations of the selected outbound (SEP and ballistic) and return transfers will be described as to their sample return capability.

3.2 Mars Spiral Capture and Escape

The introductory remarks on mission phase options mentioned both chemical retro and SEP modes for orbit capture

TRANSFER		LAUNCH(J.D.)	FLIGHT TIME	V_∞(LAUNCH)	V_∞(ARRIVAL)
1	EARTH-MARS	244-4950	350 DAYS	3 KM/SEC	1 KM/SEC
2	EARTH-MARS	4845	550	0.91	0.17
3	MARS-EARTH	5540	360	0	5
4	MARS-VENUS	5200	190	2	11.08
5	MARS-EARTH	5240	560	0	5
6	MARS-EARTH	5240	380	0	5
7	MARS-EARTH	5640	360	0	5

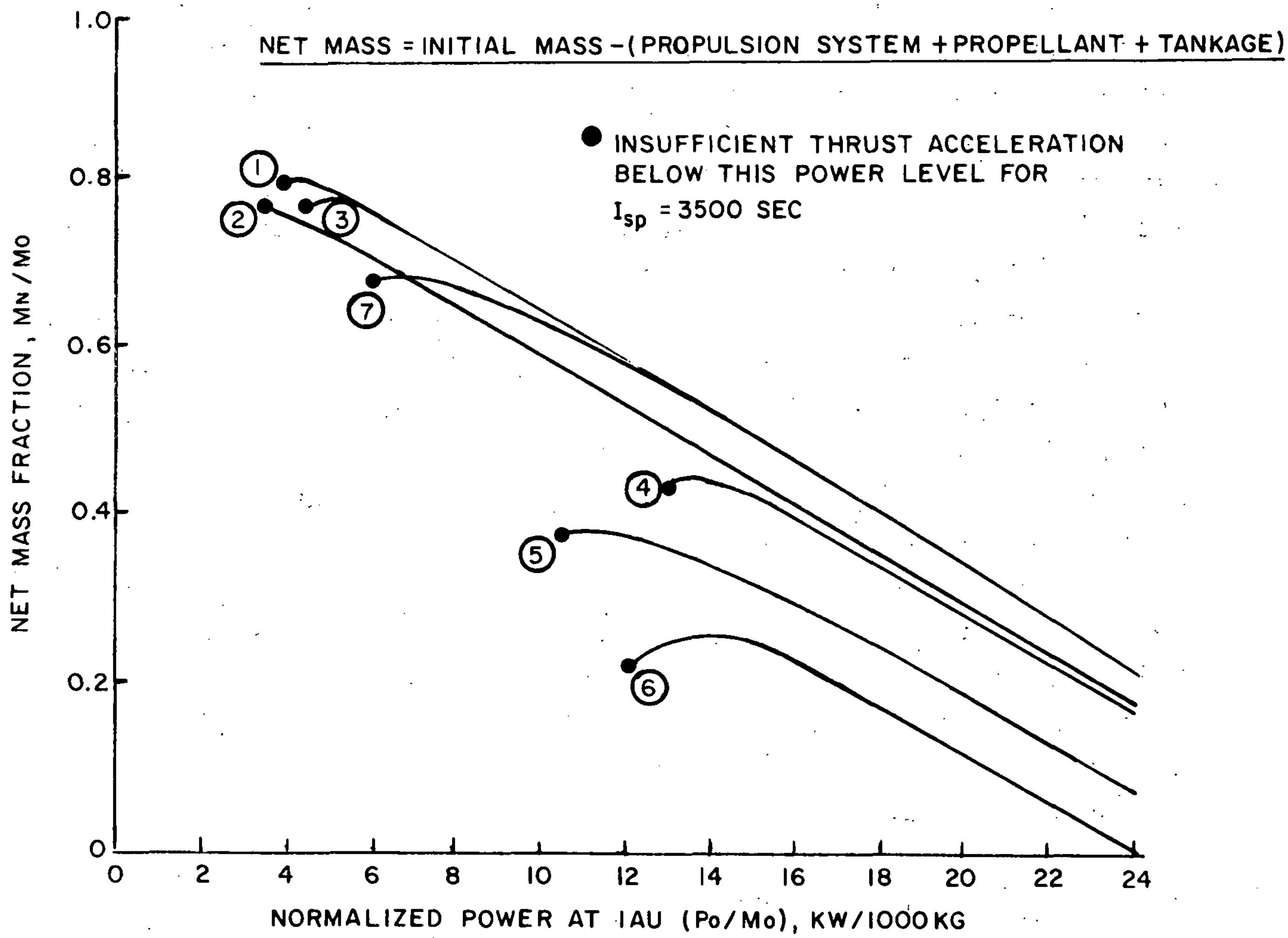


FIGURE 6. NET MASS FRACTION VERSUS SOLAR ARRAY POWER

and escape maneuvers. In the SEP mode these maneuvers are characterized by multi-revolution spirals about Mars due to the low-thrust acceleration levels. The spiral requirements are shown in Figures 7 to 10, are based on analytical solution formulas (Ragsac 1967). Spiral maneuver time and final/initial mass ratio are given as a function of thrust acceleration. For the capture spiral a_h is the initial acceleration available on the hyperbolic approach asymptote. For the escape spiral a_c is the initial acceleration available upon leaving the circular orbit about Mars. Three values of orbit altitude are shown for comparison purposes, but only the 1000 km orbit is used in the subsequent mission analysis. It should be noted that the acceleration value used must take into account the actual value ($P_oG(R)$) of solar power available at Mars distance (approximately 1.5 a.u. but variable as a function of date).

A typical thrust acceleration a_h would be 3×10^{-4} m/sec^2. The capture time is then 130 days and the final mass fraction is 0.902 (or, 0.098 propellant fraction). As an example, suppose a_c is somewhat higher at 4×10^{-4} m/sec^2 due to a reduction in mass. The escape time is then 80 days and the final mass fraction is 0.921 (or, 0.079 propellant fraction). Because the SEP system operates with a specific impulse about an order of magnitude higher than the chemical retro systems, the resulting propellant fraction is very significantly lower. The penalty incurred is a long and somewhat complex (solar array pointing) maneuver. Another disadvantage is the long time that the Mars lander must wait on the surface or in orbit before the SEP return stage reaches the rendezvous altitude of 1000 km. Nevertheless, if these operational difficulties can be tolerated, the SEP capture and escape modes can be expected to yield a large performance advantage over chemical retros; this is particularly important when Titan/Centaur launch vehicles are employed. This point will be shown in the following section.

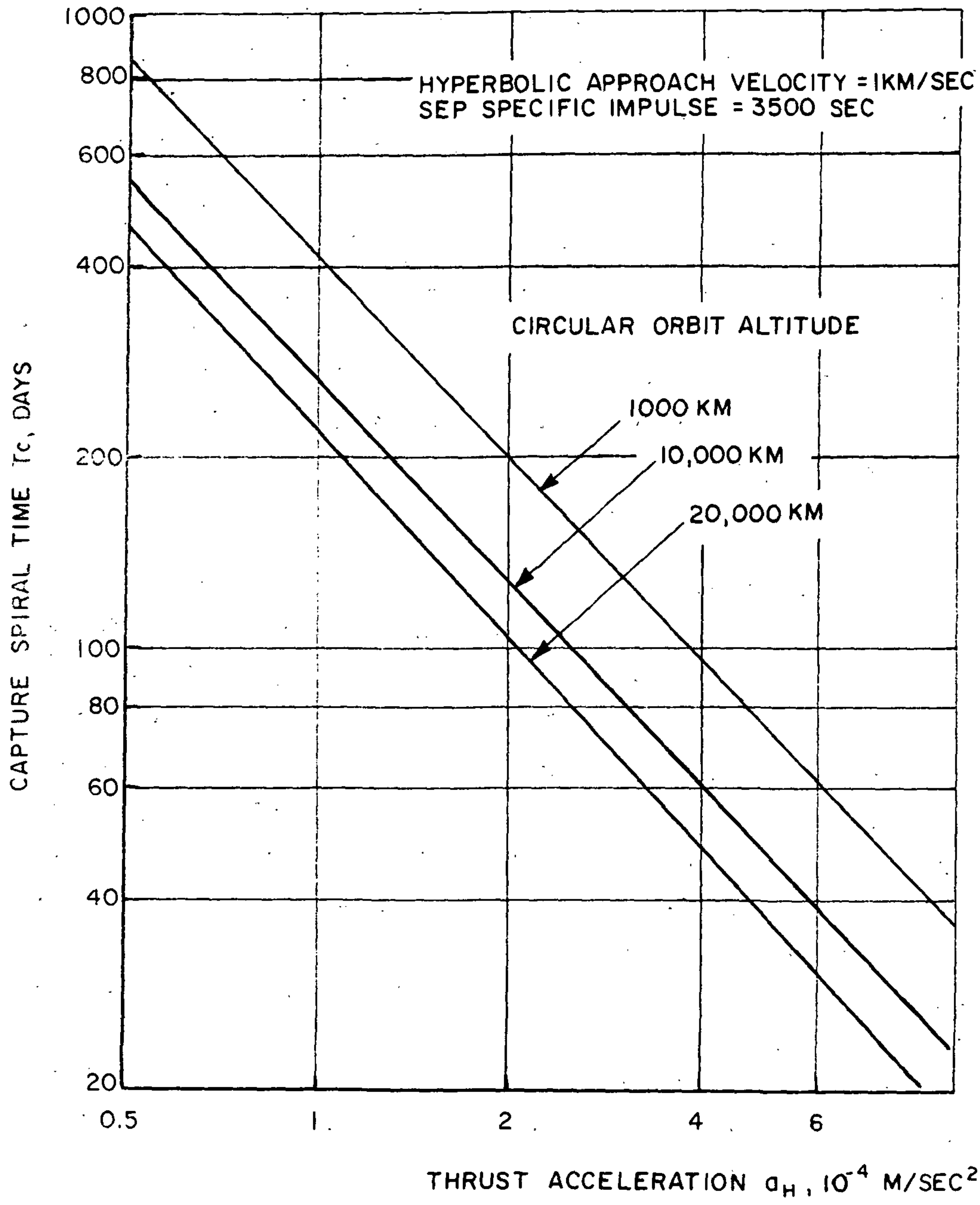


FIGURE 7. MARS LOW THRUST CAPTURE SPIRAL REQUIREMENTS (SPIRAL TIME)

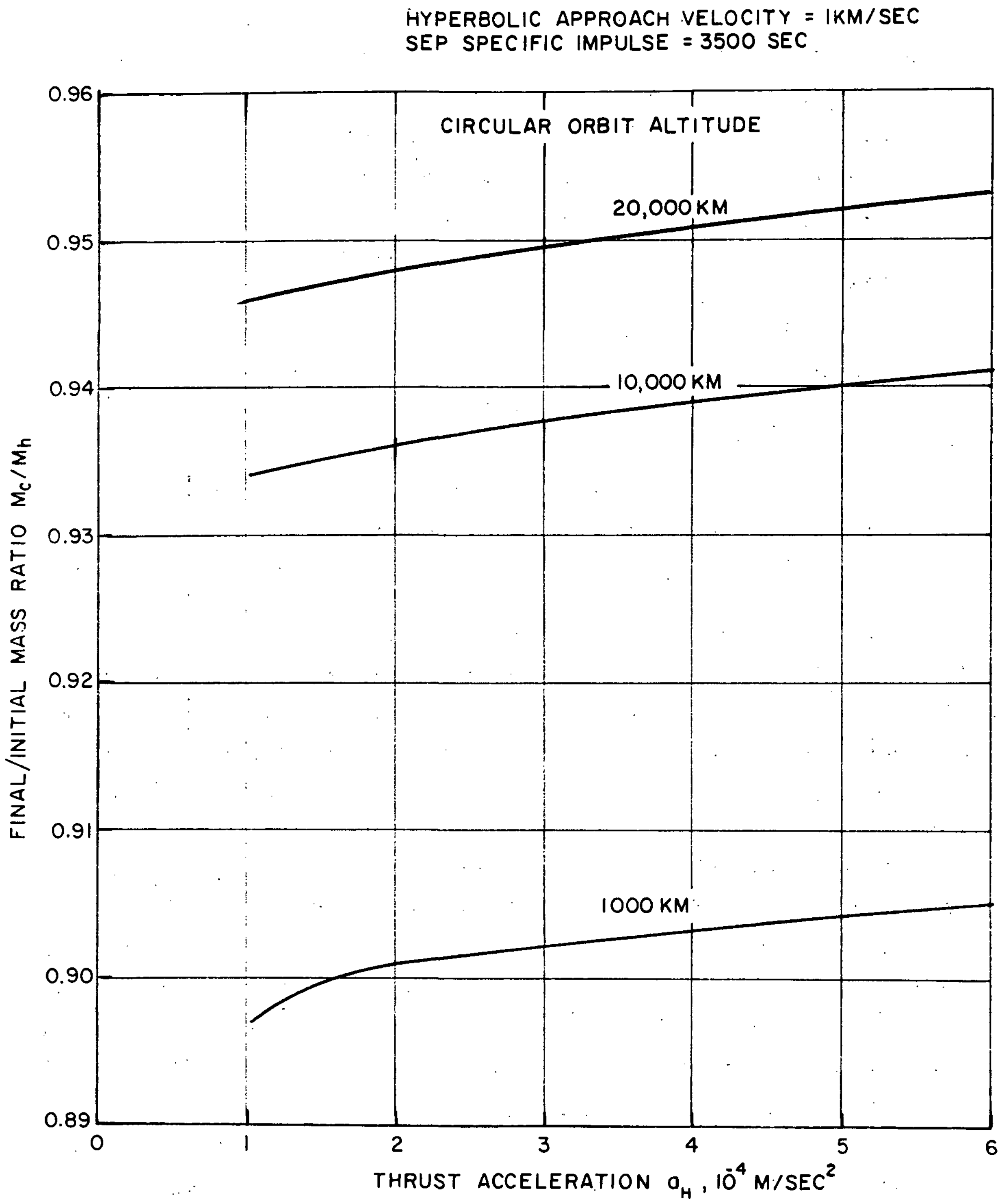


FIGURE 8. MARS LOW THRUST CAPTURE SPIRAL REQUIREMENTS (MASS RATIO)

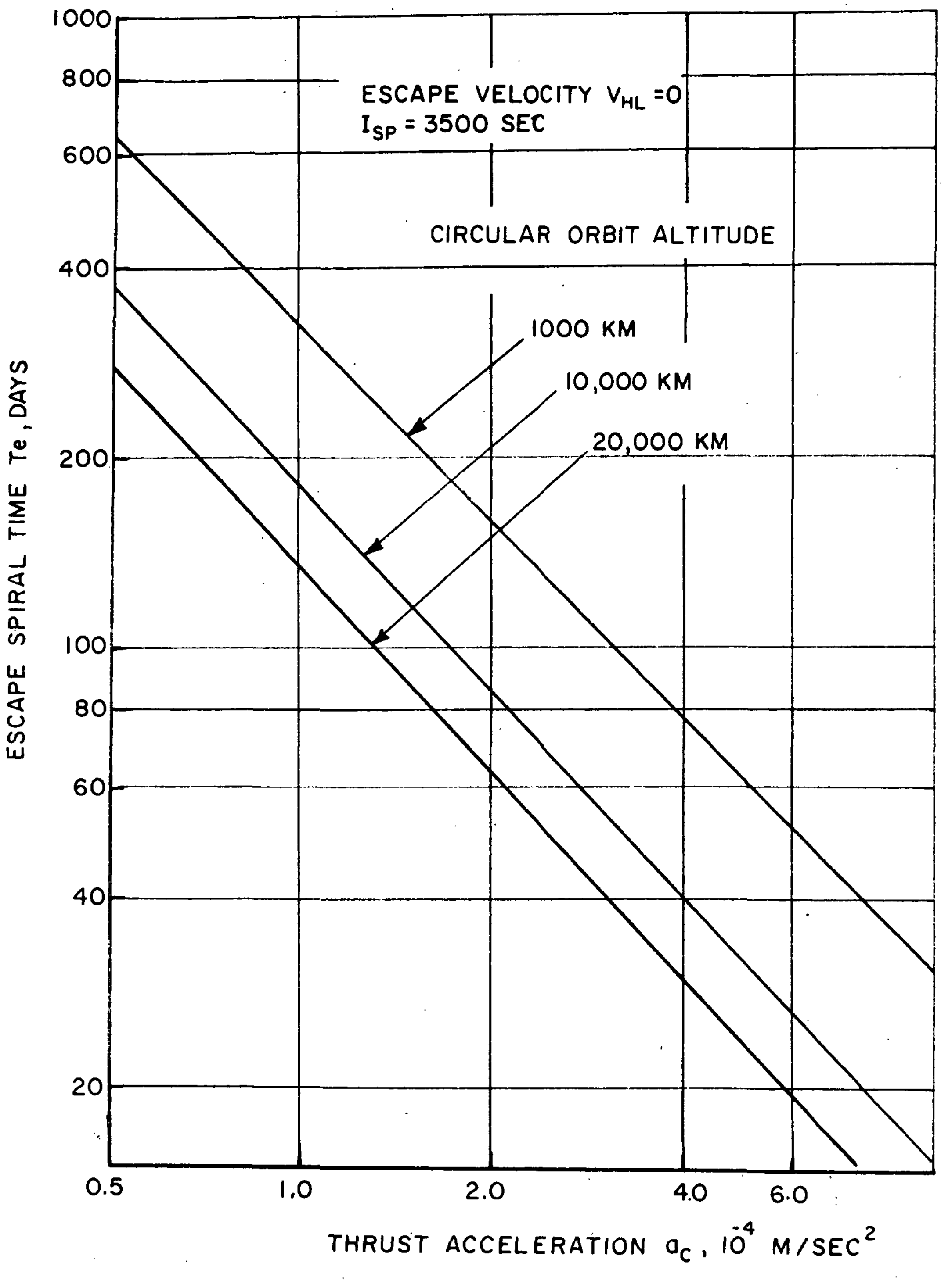


FIGURE 9. MARS LOW THRUST ESCAPE SPIRAL REQUIREMENTS (SPIRAL TIME)

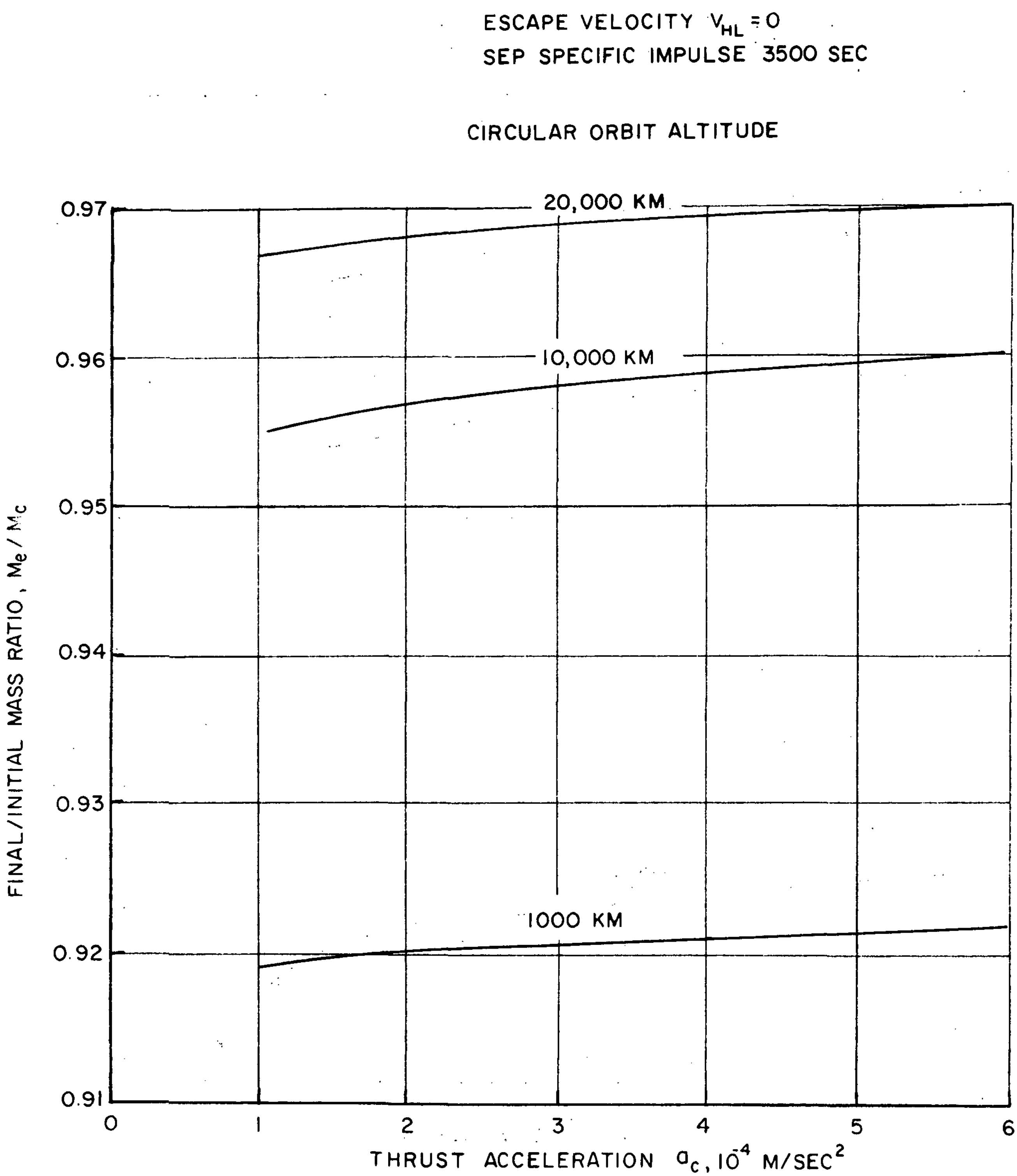


FIGURE 10. MARS LOW THRUST ESCAPE SPIRAL REQUIREMENTS (MASS RATIO)

4. MISSION PERFORMANCE CHARACTERISTICS

This section describes a set of possible MSSR missions, arrived at using the guidelines, trajectory and system data from the previous sections. The approach taken in this study was to selectively match outbound and inbound transfers to create a set of mission trajectory profiles. From the set of mission phase options (Figure 1), combinations of selected options were considered for each mission profile. The mass fraction and system scaling data were then used to size the total system requirements, depending on phase option, for each mission profile.

The results of this approach are shown in Figures 11 through 26. The set of mission profiles are depicted by polar heliocentric trajectory plots; associated launch and arrival dates are indicated on each diagram. The figure(s) following each trajectory plot presents system mass data dependent on the selected phase options for the mission profile. This data is in parametric form: Earth departure mass as a function of desired sample size. On each payload curve, the range of power required at 1 a.u. for the SEP stage(s) is indicated. Also, the launch vehicle capability at the particular Earth departure VHL is shown.

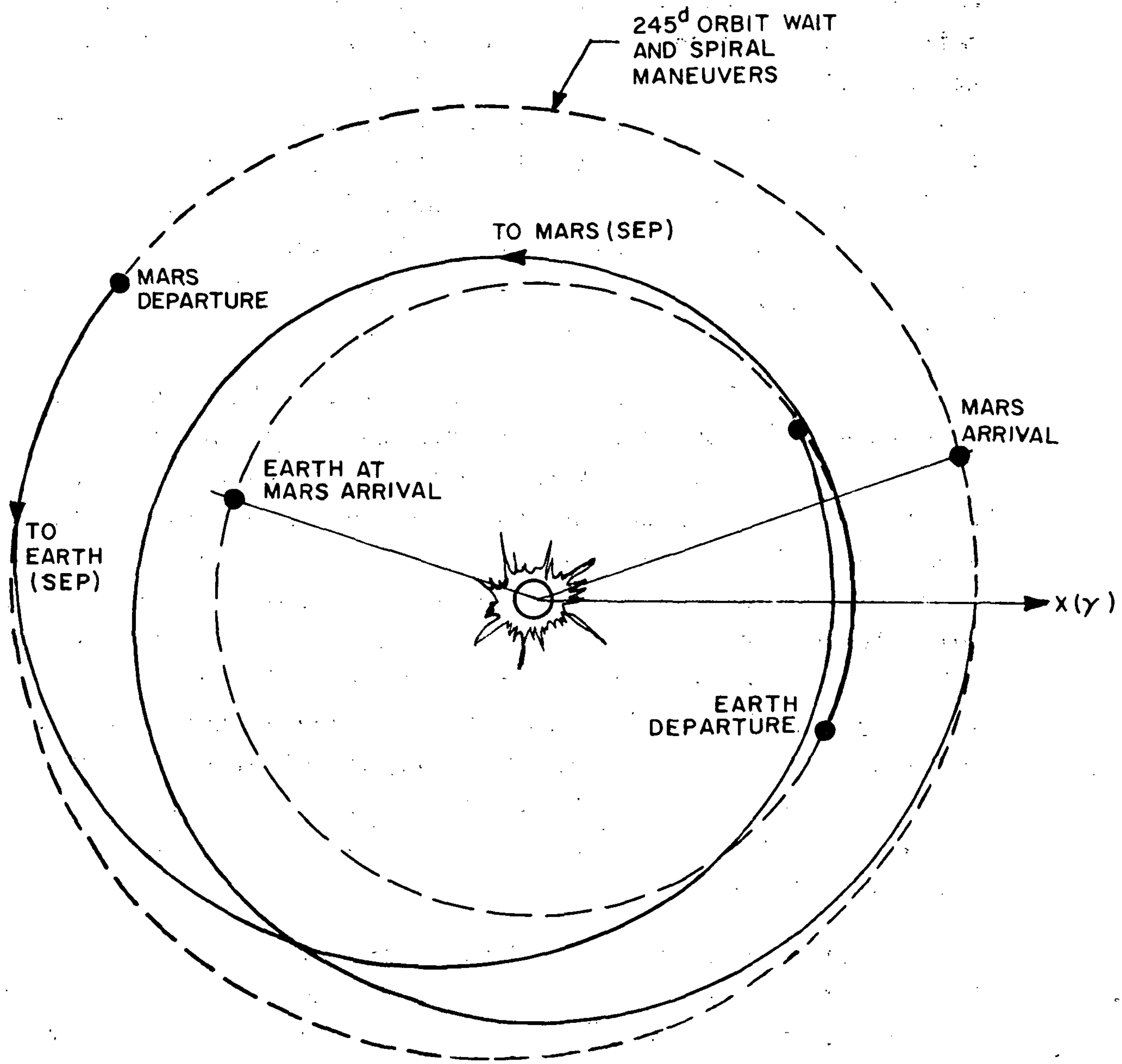
As an aid in determining the combinations of phase options for the missions considered in this study, Table 3 presents which combination of options relates to which payload figure(s). Whether the chemical retro option, where used, is solid or space storable propellant, and whether Earth recovery is by direct reentry or orbit capture, is indicated on each of the figures.

Figure 11 shows the trajectory profile for an 1155-day MSSR mission. The outbound and inbound transfers are both SEP low-thrust, the outbound leg being of the indirect mode. A

TABLE 3

MISSION PHASE OPTION SELECTION GUIDE

EARTH-MARS TRANSFER	MARS ORBIT CAPTURE	MARS LANDING	MARS ESCAPE	MARS-EARTH TRANSFER	REFERENCE FIGURES
SEP	SEP	DIRECT	SEP	SEP	12, 17
SEP	CHEMICAL	DIRECT	SEP	SEP	14, 15, 17
SEP	SEP	DIRECT	CHEMICAL	SEP	18
SEP	CHEMICAL	DIRECT	CHEMICAL	SEP	14, 15, 18
BALLISTIC	CHEMICAL	DIRECT	CHEMICAL	SEP	23, 24
BALLISTIC	CHEMICAL	OUT-OF-ORBIT	CHEMICAL	SEP	23, 24
BALLISTIC (DUAL LAUNCH)	CHEMICAL	DIRECT	CHEMICAL	SEP	22
BALLISTIC	CHEMICAL	DIRECT	CHEMICAL	SEP/VENUS SWINGBY	26



EARTH DEPARTURE	AUG. 29, 1981
MARS ARRIVAL	MAR. 2, 1983
MARS DEPARTURE	NOV. 2, 1983
EARTH ARRIVAL	OCT. 27, 1984

FIGURE 11. TRAJECTORY PROFILE FOR 1155^d MSSR MISSION

245-day stop-over time occurs at Mars which can be used either totally for orbit wait or for SEP spiral maneuvers plus orbit wait. Figure 12 shows the injected payload data for this mission if SEP spiral maneuvers are used for both Mars capture and escape. All systems are launched by a single vehicle, and the same SEP stage is used for the outbound and inbound transfers and the spiral maneuvers. The launch date for this mission is August 29, 1981. It will be noted that the Titan IIID/Centaur launch vehicle is capable of returning a 10 kg sample in either Earth recovery mode, and the SEP power requirement is about 18 kw.

Figure 13 shows a 1055-day mission which uses the same Earth-Mars transfer shown in Figure 11. For this mission, only 145 days are available at Mars. This amount of stop-over time did not allow for both SEP spiral capture and escape. Figure 14 presents payload data for two mission concepts using this profile. One assumes a space storable chemical retro escape, while the other uses a SEP spiral escape. Both use space storable retro capture. Again, this is for a single launch without SEP staging. Figure 15 shows the same concept using solid chemical retro stages. The Titan IIID(7)/Centaur launch vehicle and the spiral escape mode are both necessary to return a nominal sample of at least 5 kg. A typical SEP power requirement would be 20-25 kw.

Figure 16 shows a 950-day mission. Both outbound and inbound transfers are of the direct SEP type. A stop-over time of 240 days are allowed at Mars, giving time to use spiral maneuvers for capture and escape. The launch date is December 11, 1981. Figure 17 shows payload data for concepts which use either a solid retro capture or SEP spiral capture; both concepts use spiral escape. The data is for a single launch without SEP staging. The Titan IIID(7)/Centaur and spiral escape provide a return sample of at least 5 kg.

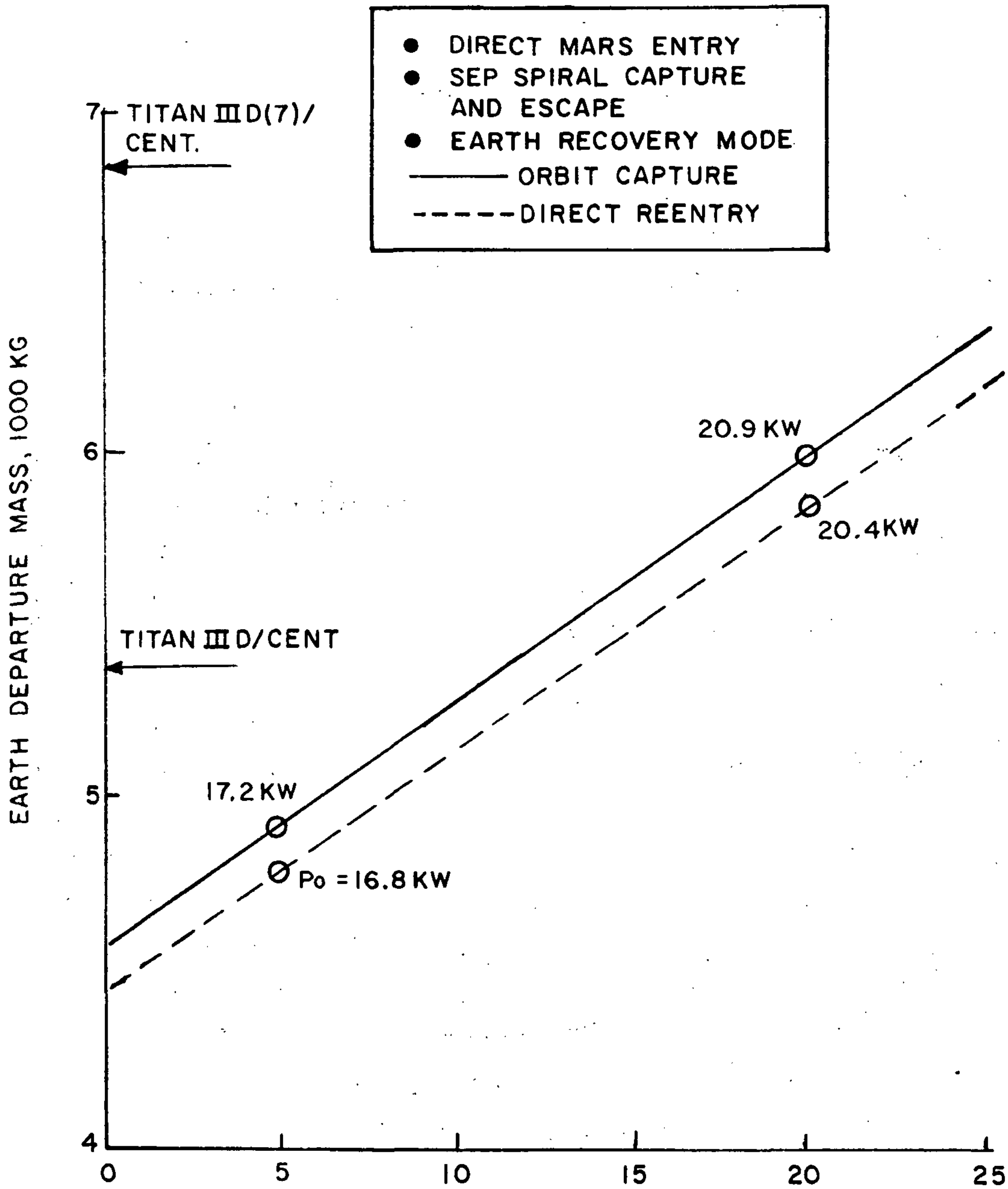
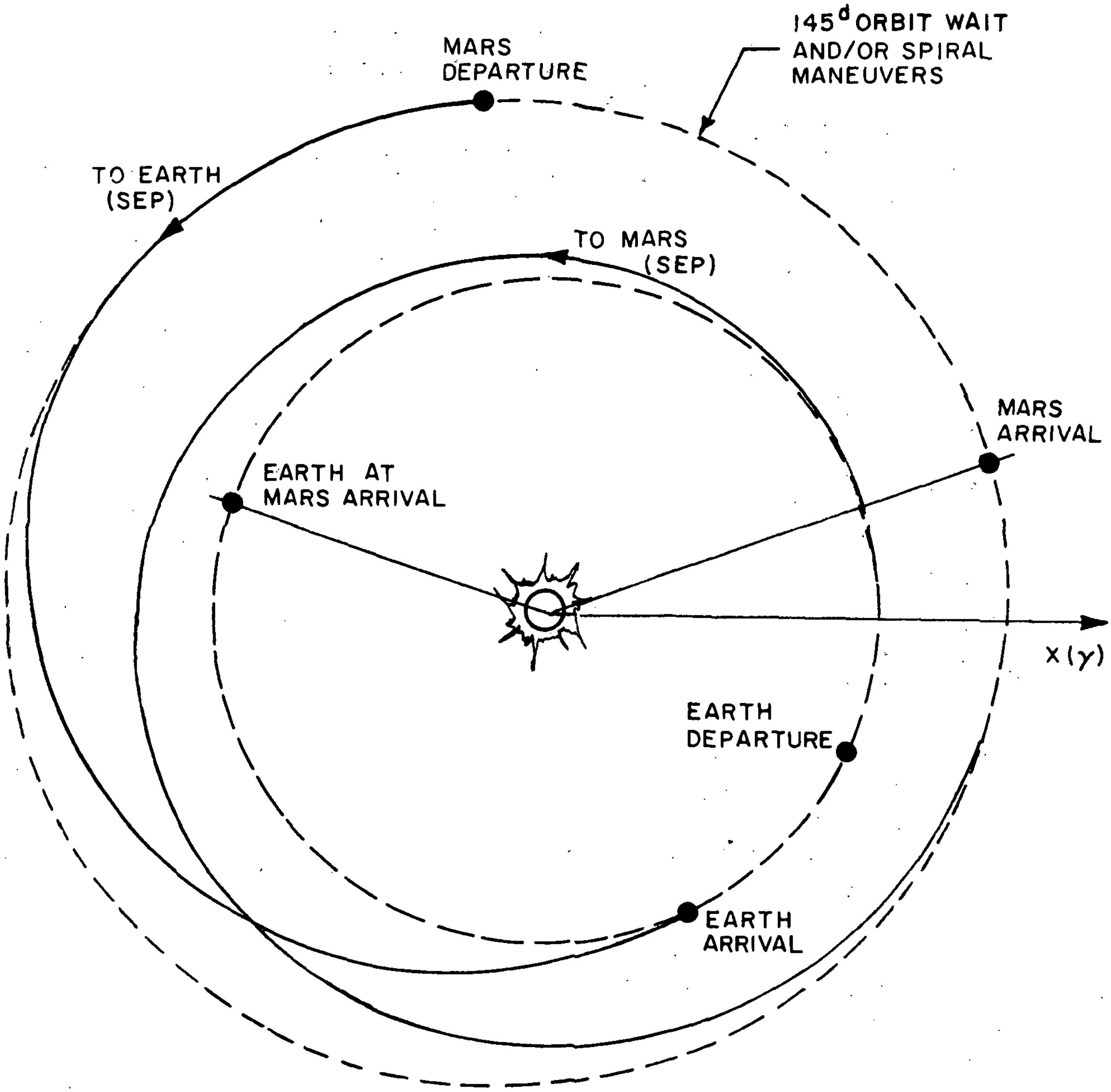


FIGURE 12. SINGLE LAUNCH SOLAR ELECTRIC PERFORMANCE FOR 1155-DAY MSSR MISSION, SEP NOT STAGED



EARTH DEPARTURE	AUG. 29, 1981
MARS ARRIVAL	MAR. 2, 1983
MARS DEPARTURE	JULY 24, 1983
EARTH ARRIVAL	JULY 18, 1984

FIGURE 13. TRAJECTORY PROFILE FOR 1055^d MSSR MISSION

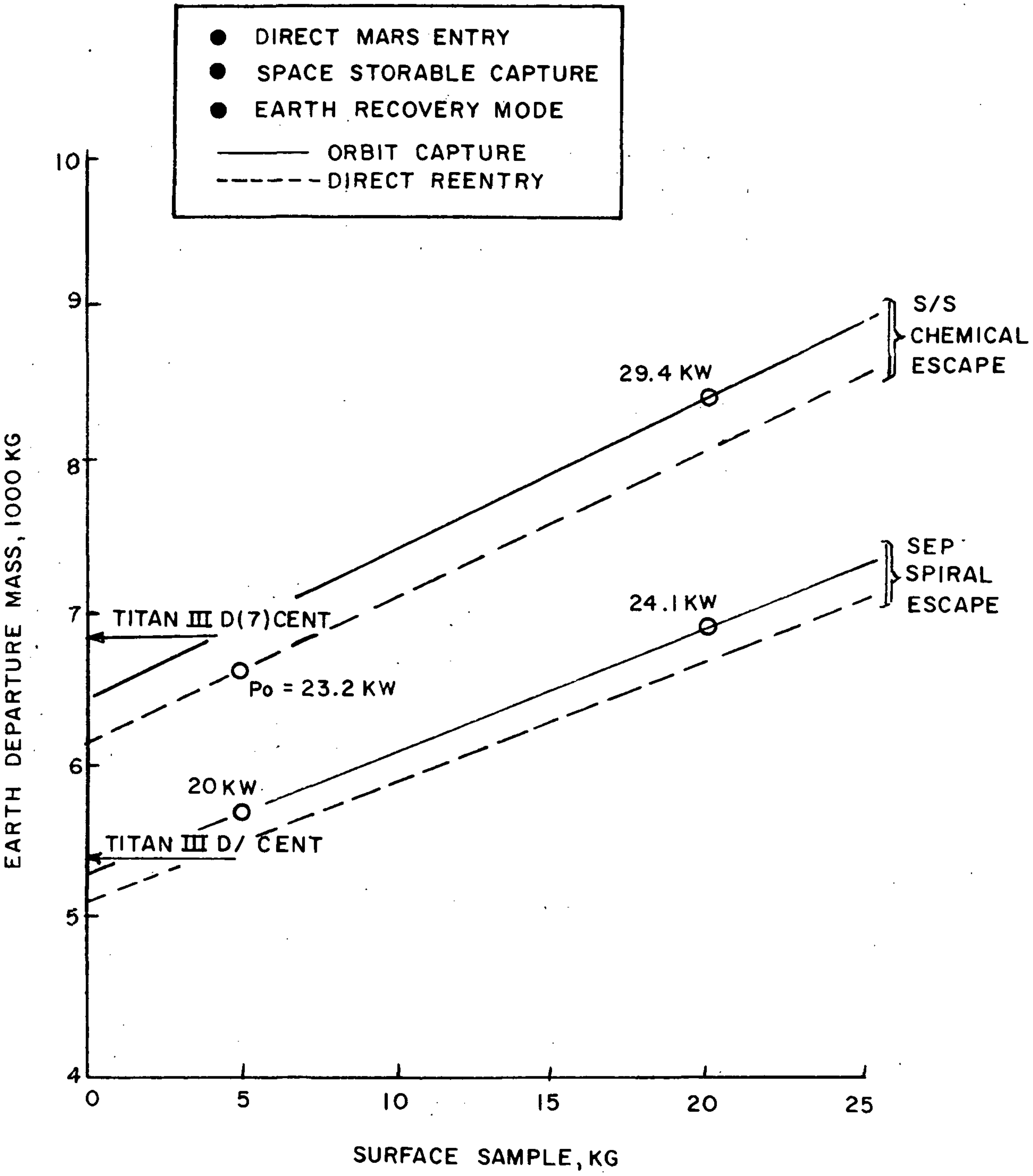


FIGURE 14. SINGLE LAUNCH SOLAR ELECTRIC PERFORMANCE FOR 1055-DAY MSSR MISSION, SEP NOT STAGED (SPACE STORABLE RETRO)

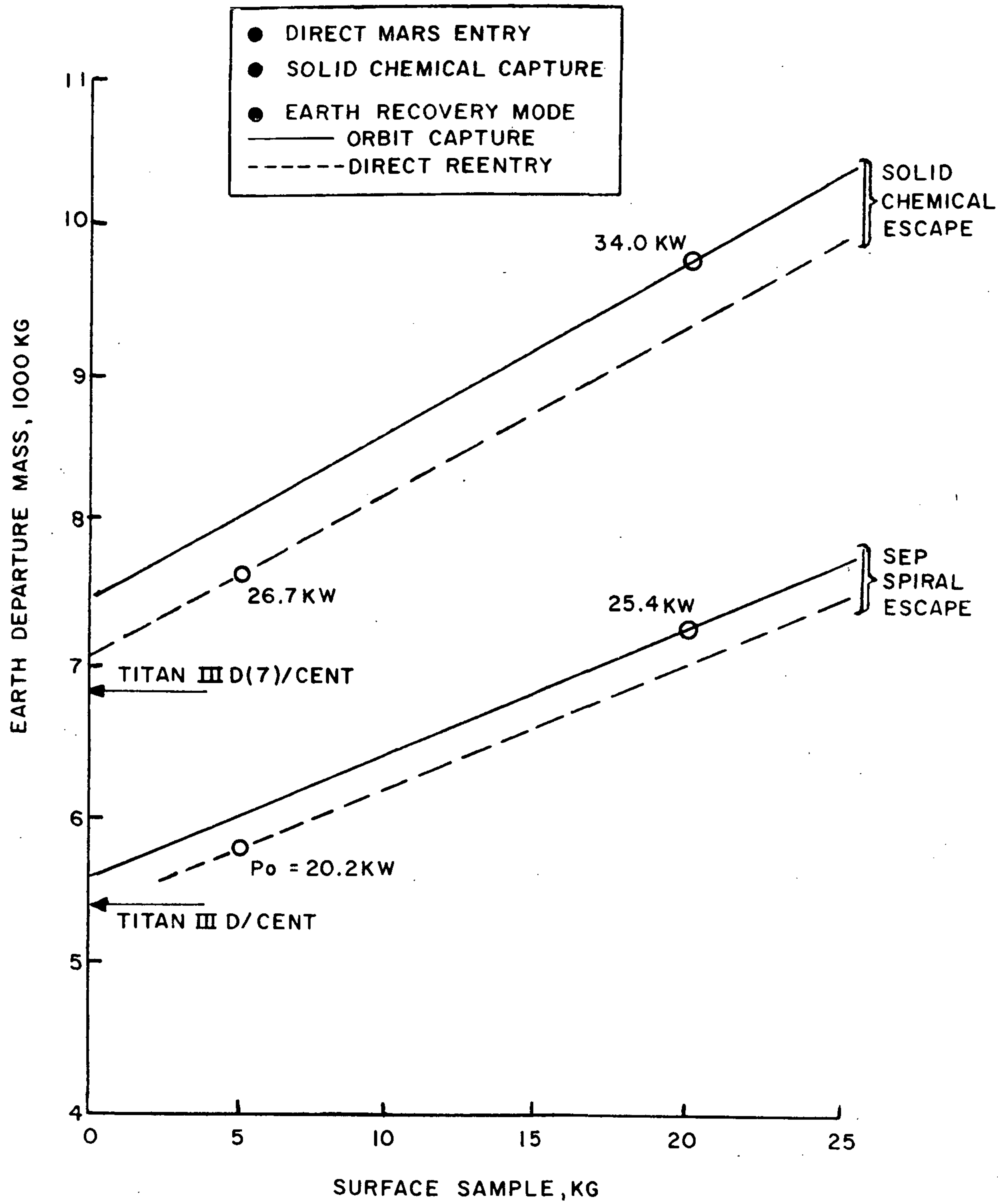
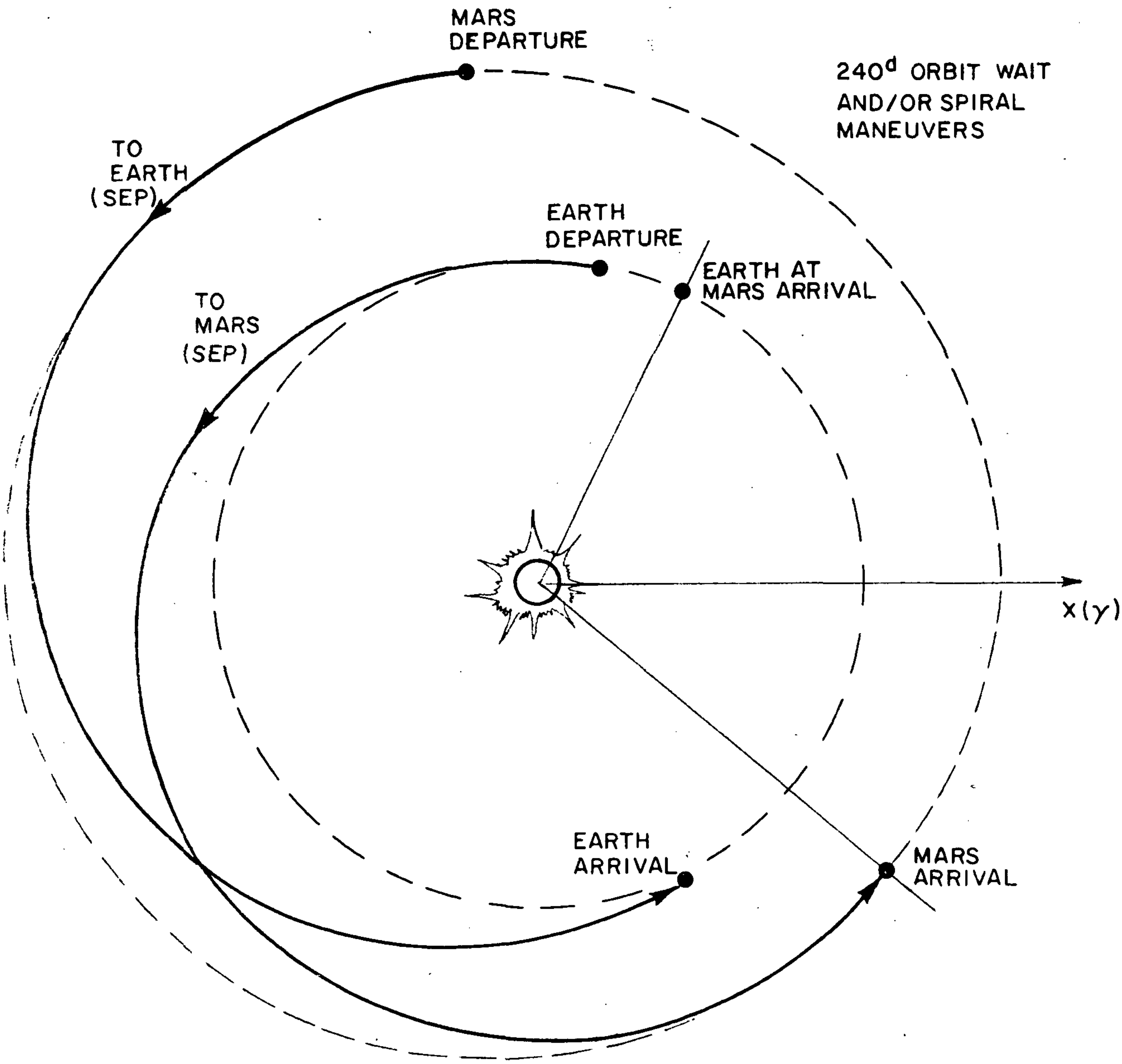


FIGURE 15. SINGLE LAUNCH SOLAR ELECTRIC PERFORMANCE FOR 1055-DAY MSSR MISSION, SEP NOT STAGED (SOLID RETRO)



EARTH DEPARTURE	DEC. 11, 1981
MARS ARRIVAL	NOV. 26, 1982
MARS DEPARTURE	JULY 24, 1983
EARTH ARRIVAL	JULY 18, 1984

FIGURE 16. TRAJECTORY PROFILE FOR 950^d MSSR MISSION

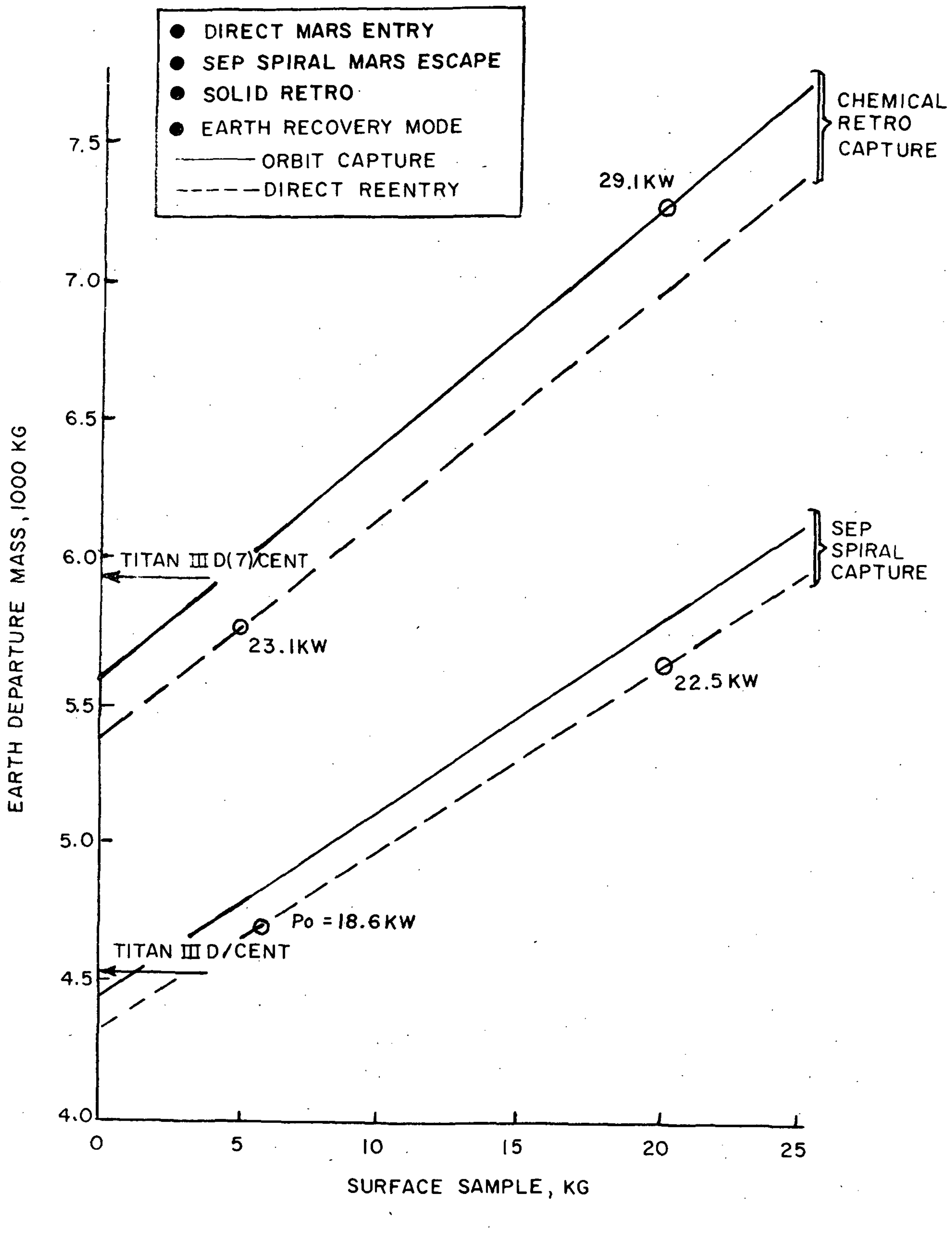


FIGURE 17. SINGLE LAUNCH SOLAR ELECTRIC PERFORMANCE FOR 950-DAY MSSR MISSION, SEP NOT STAGED

Figure 18 shows data for concepts which stage the SEP module. Both chemical (solid retro) and SEP capture options are considered, but the Mars escape maneuver is via chemical retro. SEP escape was not examined because the relatively low power rating of the second SEP stage did not allow sufficient acceleration to perform the spiral maneuver in a reasonable amount of time.

Assuming the all-chemical retro option it is seen that the staging concept yields about the same performance as the concept discussed previously where the SEP is not staged and a spiral escape mode is utilized (see Figure 17). The Titan IIID(7)/Centaur is marginal in either case. However, if the SEP capture mode is allowed the Titan IIID(7)/Centaur is capable of a 10 kg sample returned to Earth orbit. The two SEP stage power ratings are about 22.6 kw and 2.9 kw, respectively.

Figures 19, 20 and 21 present mission profiles which use the same conjunction-type Earth-Mars ballistic transfer, with a launch date of December 1, 1981. The 960-day mission profile in Figure 19 uses a direct type Mars-Earth SEP transfer and allows 310 days at Mars. Figure 20 shows an 860-day mission profile with a 10-day orbit wait at Mars and a direct SEP inbound transfer. Figure 21 is for a 680-day mission with a 10-day orbit wait, and a fast, indirect-type Mars-Earth SEP transfer.

Figure 22 presents payload data for dual launch mission concepts using the three mission profiles that have just been described. The first launch injects the Mars lander vehicle onto the trans-Mars trajectory, while the second vehicle injects the orbiter bus/SEP return stage. The two vehicles arrive at Mars at the same time; the planetary vehicle makes a direct entry, while the orbiter/return stage enters orbit. The concepts shown consider only chemical retro capture and escape maneuvers for the

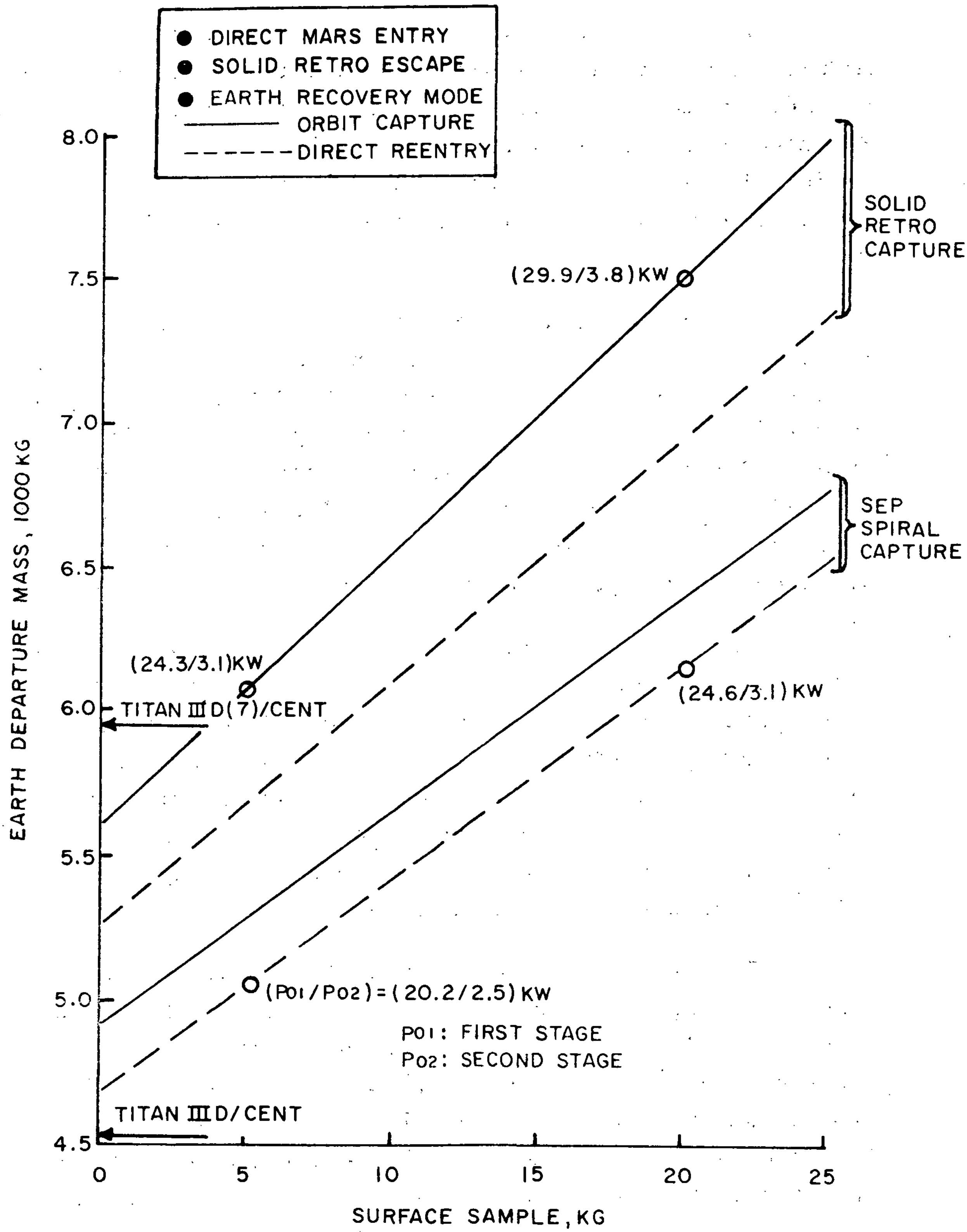
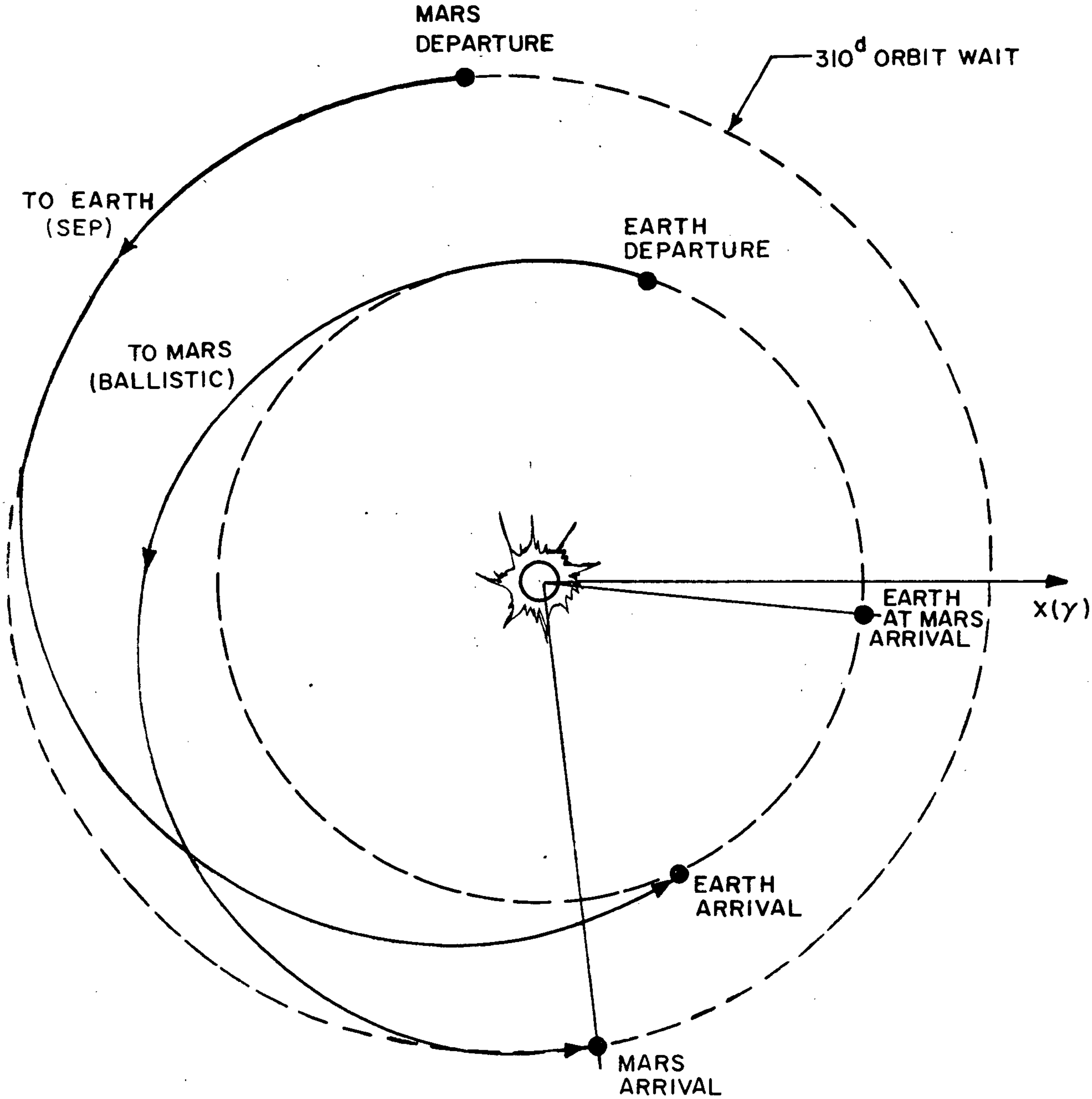


FIGURE 18. SINGLE LAUNCH SOLAR ELECTRIC PERFORMANCE FOR 950-DAY MSSR MISSION WITH SEP STAGING



EARTH DEPARTURE	DEC. 1, 1981
MARS ARRIVAL	SEPT. 17, 1982
MARS DEPARTURE	JULY 24, 1983
EARTH ARRIVAL	JULY 18, 1984

FIGURE 19. TRAJECTORY PROFILE FOR 960^d MSSR MISSION

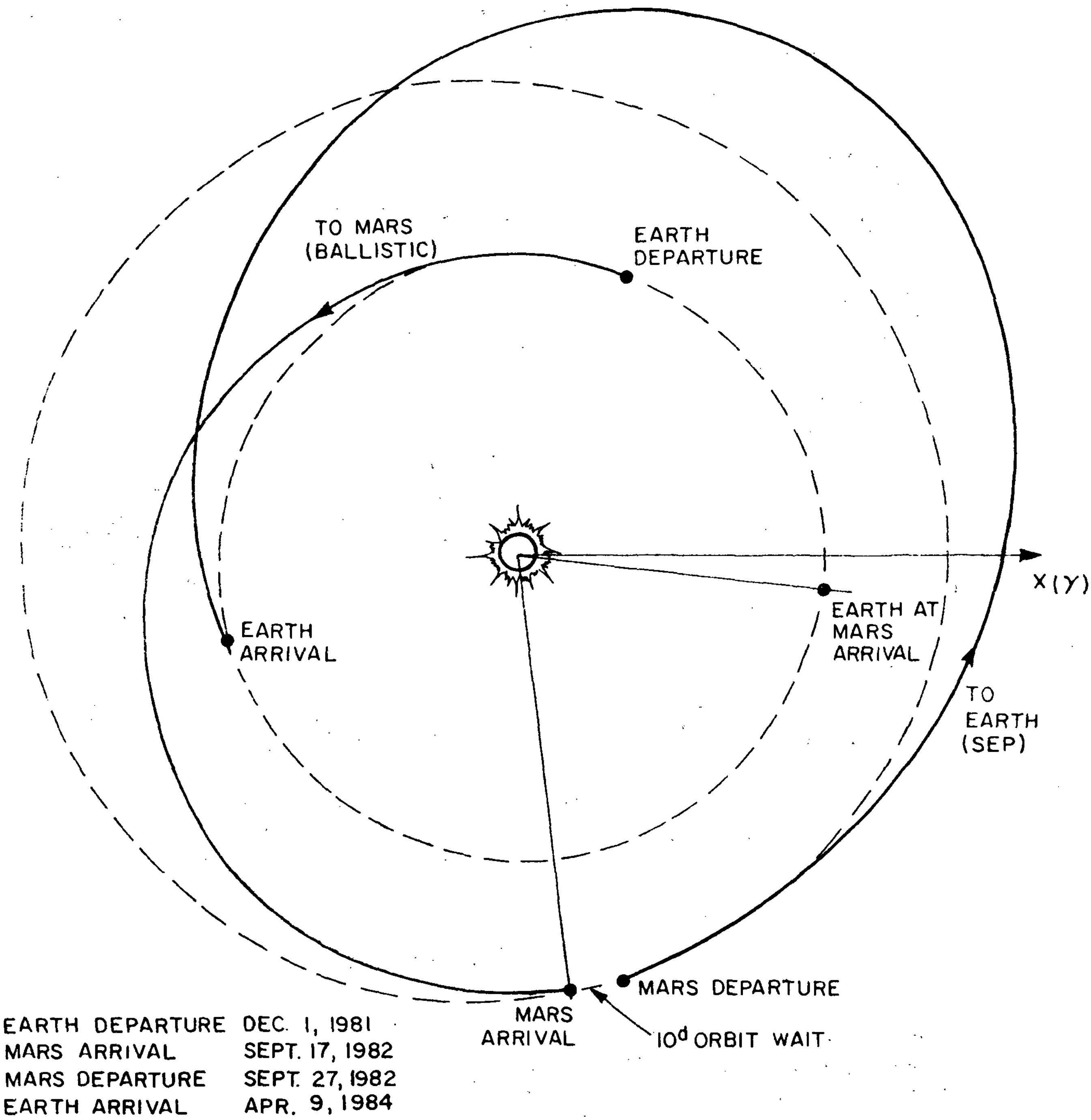
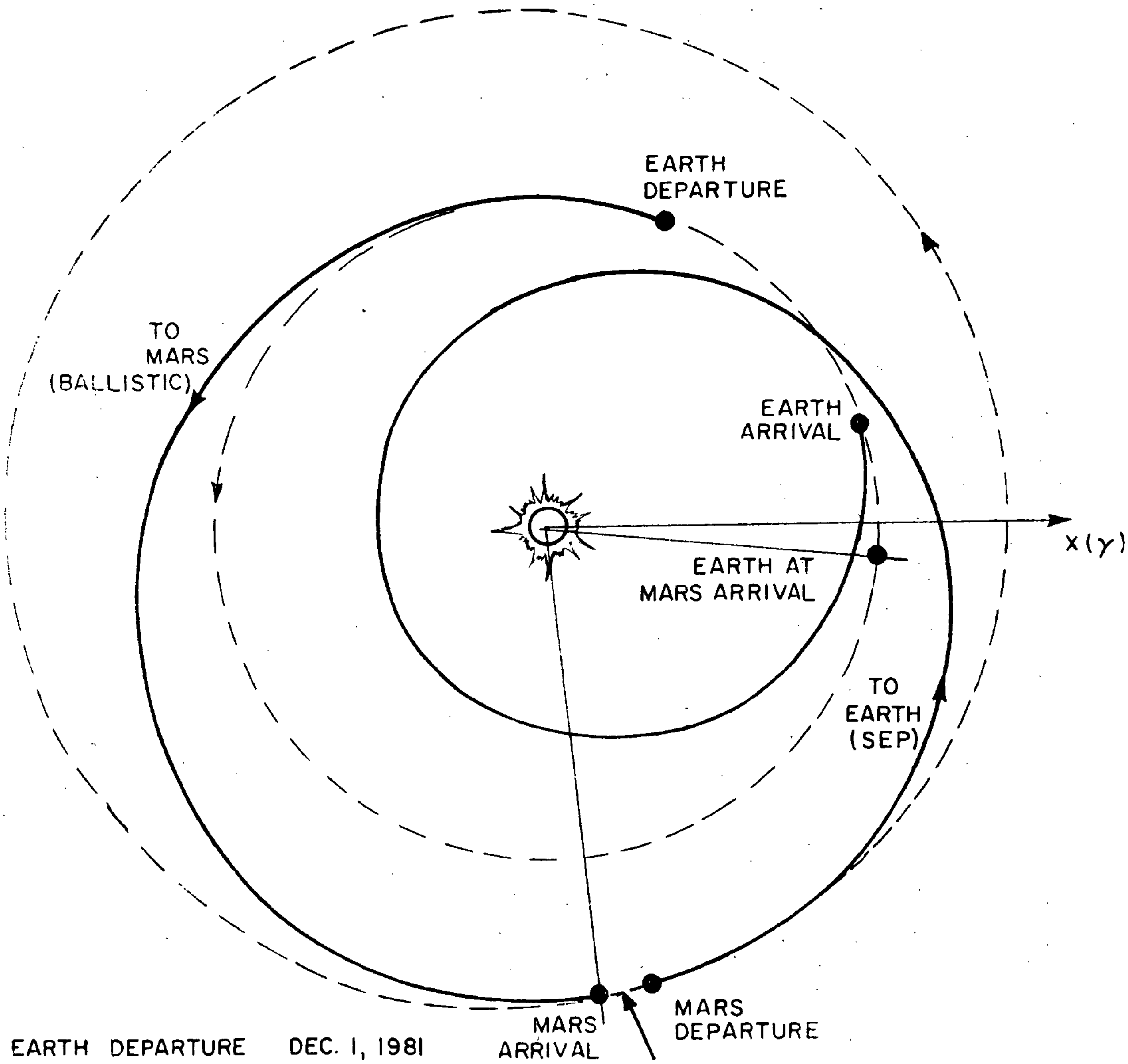


FIGURE 20. TRAJECTORY PROFILE FOR 860^d MSSR MISSION



EARTH DEPARTURE	DEC. 1, 1981
MARS ARRIVAL	SEPT. 17, 1982
MARS DEPARTURE	SEPT. 27, 1982
EARTH ARRIVAL	OCT. 12, 1983

FIGURE 21. TRAJECTORY PROFILE FOR 680^d MSSR MISSION

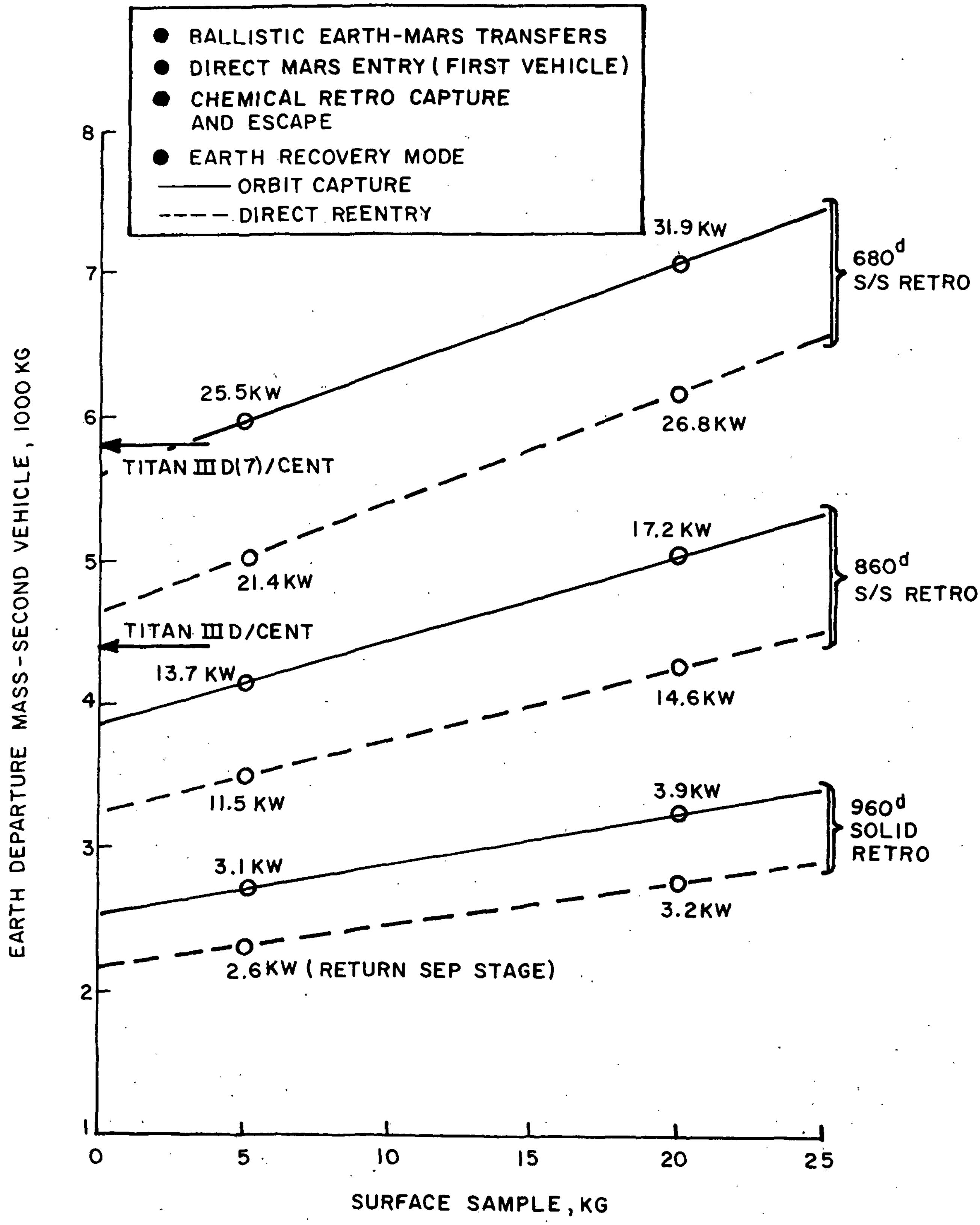


FIGURE 22. DUAL LAUNCH SOLAR ELECTRIC PERFORMANCE FOR MSSR MISSIONS

orbiter/return stage; the propellants considered are as indicated. It will be noted that the Titan IIID/Centaur has adequate performance capability in the dual launch mode for either the 860-day or 960-day missions. The longer mission, in particular, is attractive in that the solid retro system can be employed and the SEP return stage requires less than 4 kw power.

Figure 23 shows several concepts for the 680-day mission profile which uses a single launch and chemical retro capture and escape at Mars. An INT-20/Centaur launch vehicle would be required for the out-of-orbit entry mode. The Centaur upper stage would not be needed for the Mars direct entry mode. The SEP return stage power is greated than 26 kw for a sample size greater than 5 kg.

The performance characteristics for two Mars lander probes launched on the same vehicle are shown in Figure 24, again using the 680-day mission profile. Note that only one orbiter/SEP return stage is employed. This would rendezvous with each Mars ascent vehicle from the two different landing sites. If the Mars direct entry mode is employed an INT-20/Centaur launch vehicle would be capable of returning 50 kg of samples - 25 kg from each landing site. However, the SEP return stage power requirement is greater than 40 kw.

Figure 25 shows a mission profile which uses an opposition-type ballistic transfer to Mars and a Venus swingby return to Earth. The Mars-Venus transfer is SEP and the Venus-Earth transfer is ballistic. The launch date is December 21, 1981 with a total mission time of 600 days and a 20-day stop-over time at Mars. It will be noted that the return trajectory in this case is similar to the all-SEP return on the 680-days mission profile (see Figure 21).

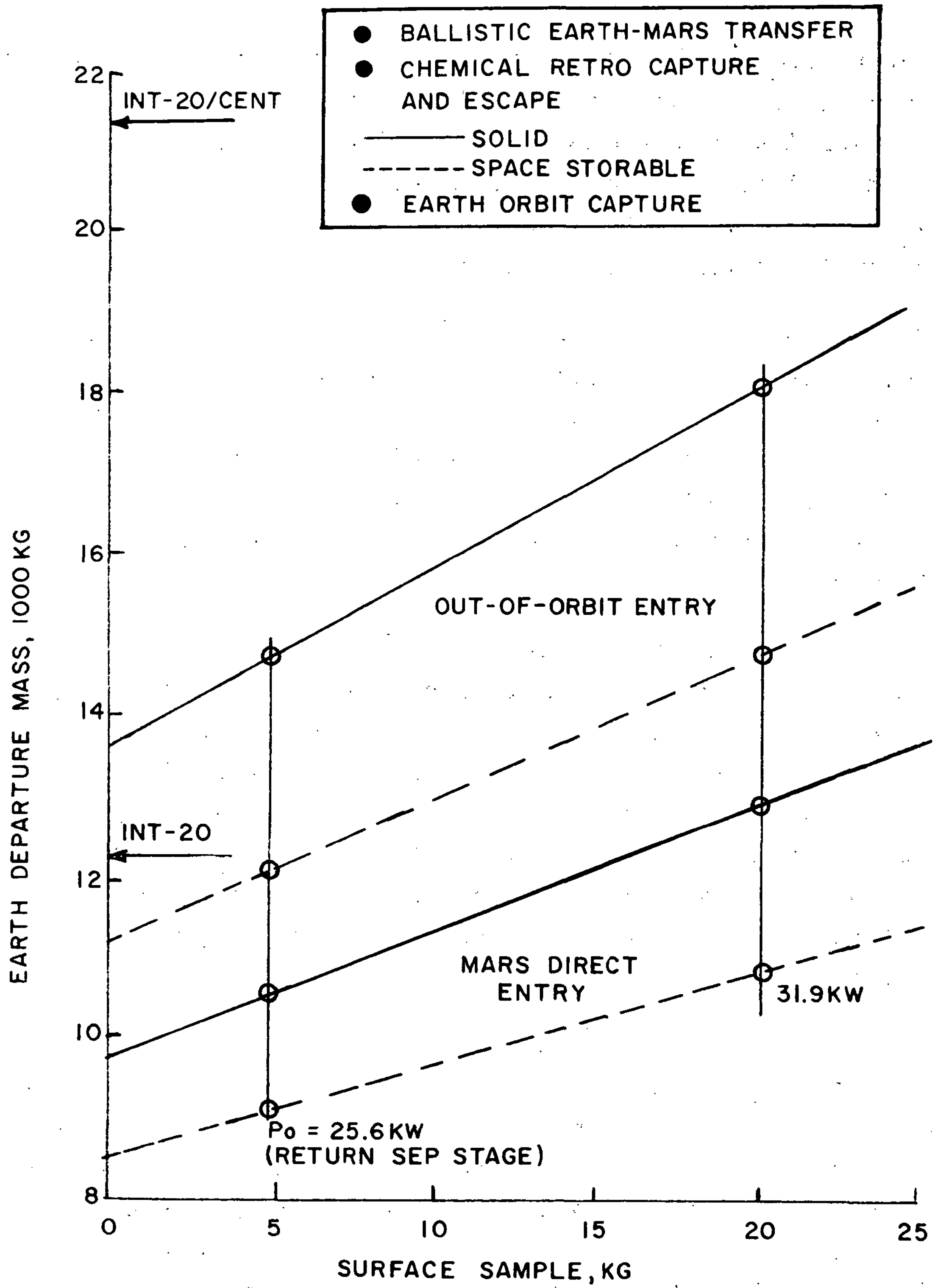


FIGURE 23. SINGLE LAUNCH SOLAR ELECTRIC PERFORMANCE FOR 680-DAY MSSR MISSION

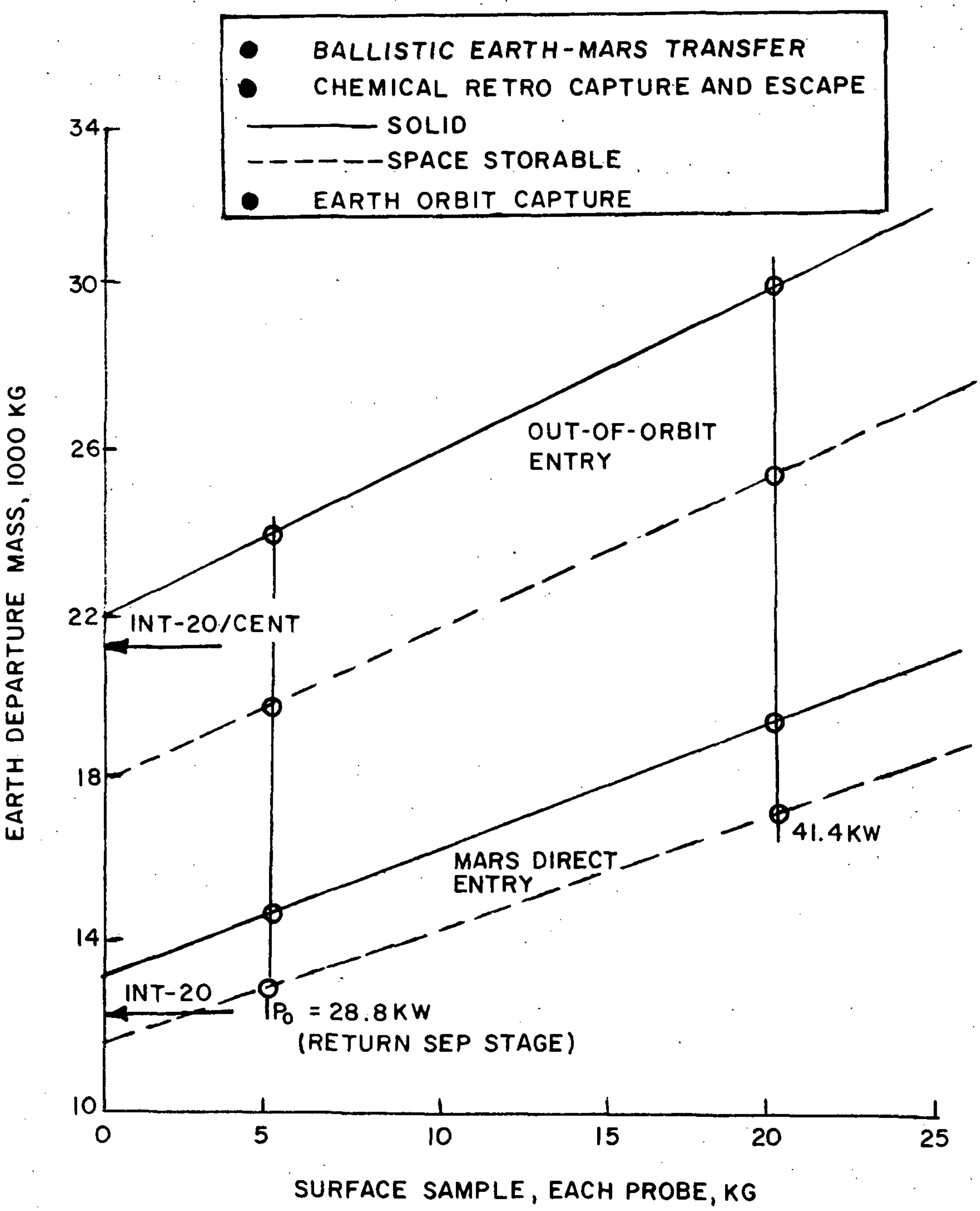
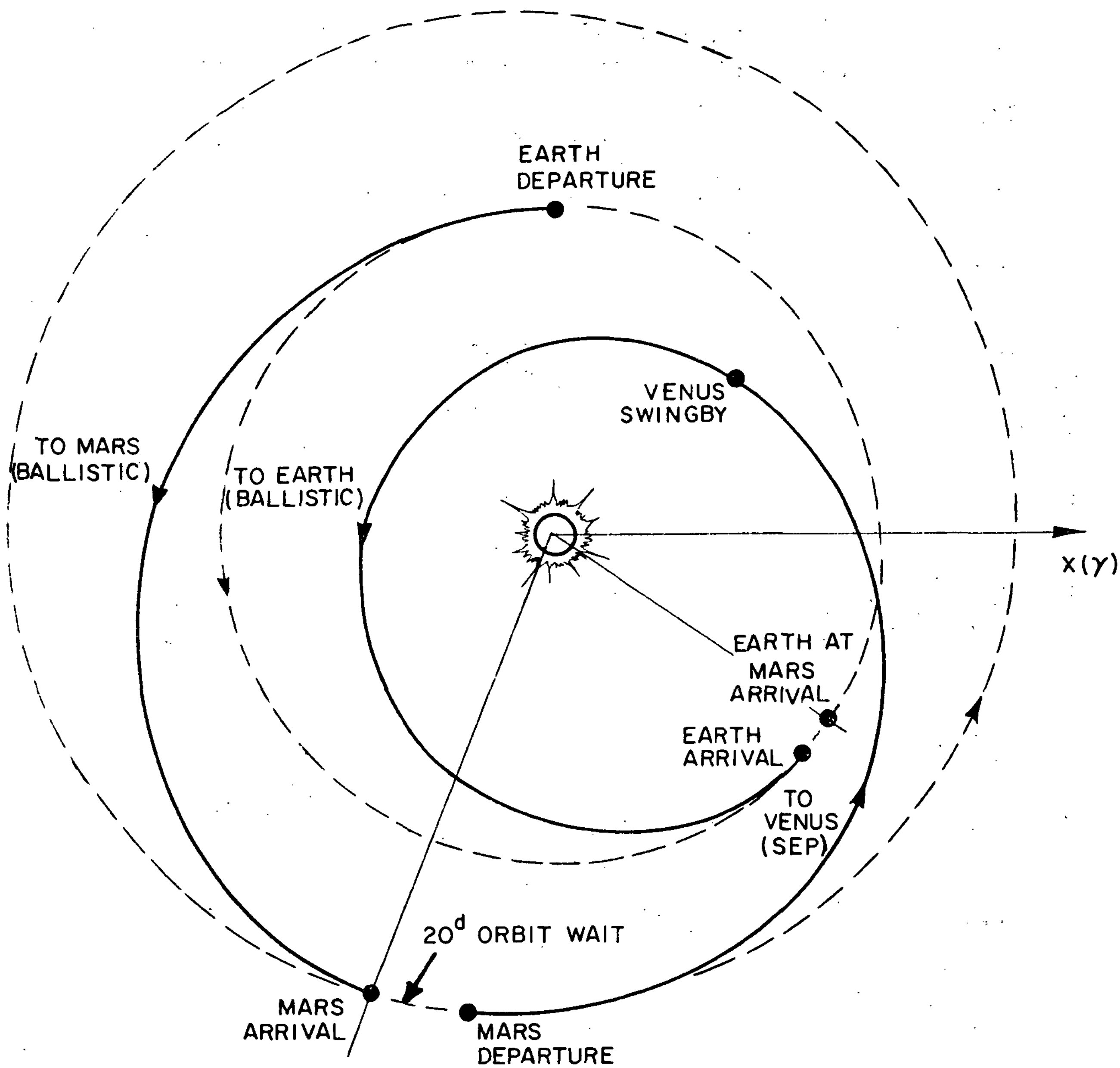


FIGURE 24. SINGLE LAUNCH/TWO PROBE SOLAR ELECTRIC PERFORMANCE FOR 680-DAY MSSR MISSION



EARTH DEPARTURE	DEC 21, 1981
MARS ARRIVAL	JULY 29, 1982
MARS DEPARTURE	AUG. 18, 1982
VENUS SWINGBY	FEB. 24, 1983
EARTH ARRIVAL	AUG. 13, 1983

FIGURE 25. TRAJECTORY PROFILE FOR 600^d MSSR MISSION WITH INBOUND VENUS SWINGBY

Performance data for the Venus swingby concept is presented in Figure 26. A chemical retro stage is used for both Mars capture and escape, and the Earth recovery mode shown is via orbit capture. The INT-20 would be adequate for a space storable retro, but the INT-20/Centaur (off scale) would be needed if a solid retro were utilized. The power of the SEP return stage lies in the range 15-19 kw for a 5-25 kg surface sample.

The next section will summarize several baseline mission examples representing a spectrum of the various concepts and performance data just described.

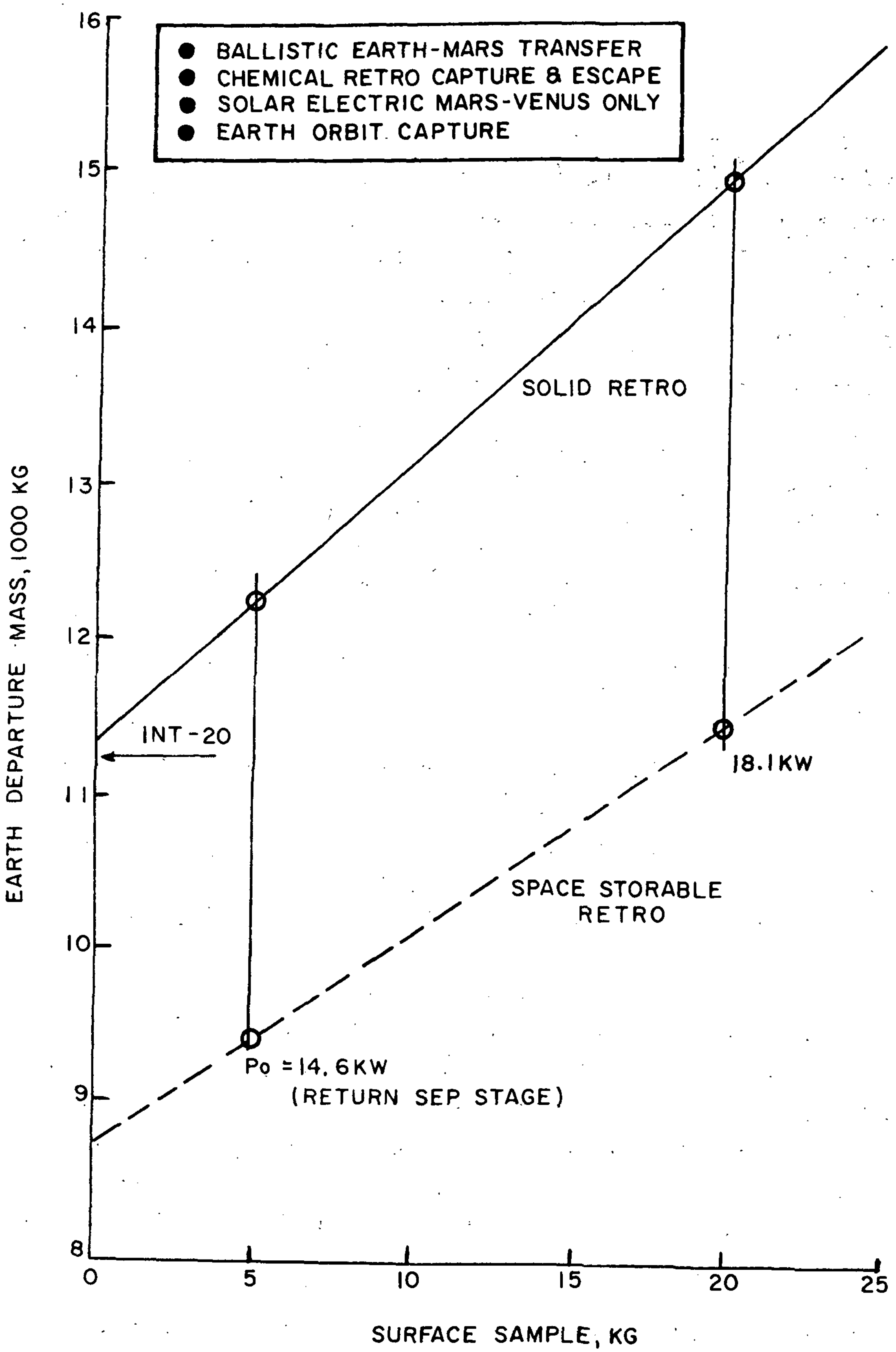


FIGURE 26. SINGLE LAUNCH SOLAR ELECTRIC PERFORMANCE FOR 600-DAY MSSR MISSION WITH INBOUND VENUS SWINGBY

5. BASELINE MISSION EXAMPLES

The previous section presented a set of representative MSSR mission concepts in the form of parametric payload data at Earth departure as a function of sample size. Tables 3 through 8 describe design-point baseline missions taken from the set of mission concepts. The examples selected encompass the spectrum of MSSR missions in terms of flight duration, propulsion modes, launch vehicle, etc. For each baseline mission, a sample size was assumed which would allow a nominal margin between total system mass and launch vehicle capability. The masses of various subsystems were then calculated on the basis of desired sample size. All of the baseline missions considered assume the direct entry option for the Mars lander vehicle and the Earth orbit capture mode for sample capsule recovery.

Table 4 presents an 1155-day mission which will return a 10 kg sample using a Titan IIID/Centaur single launch. SEP is used for all major propulsion phases and the total SEP thrusting time is 784 days. The power requirement of the SEP stage is 18.5 kw at 1 a.u. The launch vehicle margin is approximately 100 kgs.

Table 5 presents a 1055-day mission to return a 10 kg sample. A solid propellant retro system is used for Mars capture. In comparison with the previous example, note that even though the SEP thrust time has decreased because of no spiral capture, the SEP system requirements (mass and power) have increased. The launch vehicle for this mission is the Titan IIID(7)/Centaur, and the margin is 425 kgs.

Table 6 lists data for a 950-day mission returning a 10 kg sample. All major propulsion phases are again SEP. The total thrusting time has been reduced to 586 days because of the

TABLE 4

BASELINE MISSION 1 - SUMMARY

Sample Size	10 kg (Earth Orbit Capture)
Launch Vehicle	Titan IIID/Centaur (Single Launch)
Mission Duration	1155 Days

	FLIGHT TIME	SEP TIME
Earth-Mars Transfer (SEP)	550 days	496 days
Mars Capture Spiral (SEP)	98	98
Mars Stay Time	34	
Mars Escape Spiral (SEP)	113	113
Mars-Earth Transfer (SEP)	360	77
	1155	784

System Weight Breakdown

Mars Lander/Ascent Probe (Direct Entry)		2803 kg
Aerobraking/Propuslion	558	
Lander	847	
Rover	68	
Ascent Vehicle	1330	
Sterilization Canister		188
Probe Mounting Structure		148
SEP Stage		1540
Propulsion System (18.5 kw)	555	
Propellant + Tankage	985	
Spacecraft Equipment Module		453
Earth Capture Stage (Solid)		146
Earth Departure Vehicle		5278 kg
Titan IIID/Centaur Capability		5380 kg

TABLE 5

BASELINE MISSION 2 - SUMMARY

Sample Size	10 kg (Earth Orbit Capture)
Launch Vehicle	Titan IIID(7)/Centaur (Single Launch)
Mission Duration	1055 Days

		FLIGHT TIME	SEP TIME
Earth-Mars Transfer	(SEP)	550 days	496 days
Mars Stay Time		49	
Mars Escape Spiral	(SEP)	96	96
Mars Earth Transfer	(SEP)	360	76
		1055	668

System Weight Breakdown

Mars Lander/Ascent Probe (Direct Entry)		2803 kg
Aerobraking/Propulsion	558	
Lander	847	
Rover	68	
Ascent Vehicle	1330	
Sterilization Canister		188
Probe Mounting Structure		148
Mars Capture Stage		979
SEP Stage		1708
Propulsion System (22.5 kw)	675	
Propellant + Tankage	1033	
Spacecraft Equipment Module		453
Earth Capture Stage (Solid)		146
Earth Departure Vehicle		6425 kg
Titan IIID/Centaur Capability		6850 kg

TABLE 6

BASELINE MISSION 3 - SUMMARY

Sample Size	10 kg (Earth Orbit Capture)
Launch Vehicle	Titan IIID(7)/Centaur (Single Launch)
Mission Duration	950 Days

		FLIGHT TIME	SEP TIME
Earth-Mars Transfer	(SEP)	350 days	292 days
Mars Capture Spiral	(SEP)	118	118
Mars Stay Time		22	
Mars Escape Spiral	(SEP)	100	100
Mars-Earth Transfer	(SEP)	360	76
		950	586

System Weight Breakdown

Mars Lander/Ascent Probe (Direct Entry)		2803 kg
Aerobraking/Propulsion	558	
Lander	847	
Rover	68	
Ascent Vehicle	1330	
Sterilization Canister		188
Probe Mounting Structure		148
SEP Stage		1378
Propulsion System (20.5 kw)	614	
Propellant + Tankage	764	
Spacecraft Equipment Module		453
Earth Capture Stage (Solid)		146
Earth Departure Vehicle		5116
Titan IIID(7)/Centaur Capability		5995

faster Earth-Mars transfer. The SEP power requirement is 20.5 kw at 1 a.u. The launch vehicle is again the seven-segment Titan/ Centaur with a margin of nearly 880 kgs.

The fourth example, Table 7, presents a 960-day mission which utilizes the dual launch concept. The returned sample size is 20 kgs. Solar electric is used only during the inbound transfer, and the system power requirement is relatively low at 3.9 kw. The vehicle for each launch is a Titan IIID/Centaur, and the weight margins are approximately 360 kgs and 1175 kgs.

Table 8 presents a 680-day mission which will return 10 kgs. SEP is used only for the inbound transfer. The SEP thrusting time is 245 days, and the system power requirement is nearly 28 kw at 1 a.u. The mission utilizes a single launch vehicle, the Intermediate-20, with a weight margin of nearly 1000 kgs.

The final example, Table 9, is a 600-day mission which will return a 25 kg sample. This mission uses a Venus swingby during the inbound transfer, with SEP used only for the Mars-Venus leg. The SEP thrust time is only 157 days and the power requirement is nearly 20 kw at 1 a.u. The required launch vehicle is the Intermediate-20/Centaur with a margin of over 4000 kgs.

Table 10 summarizes the more pertinent aspects of the six baseline missions selected as examples.

For purposes of comparison, Figures 27 and 28 present three all-ballistic mission concepts. The two concepts in Figure 27 use the same conjunction type Earth-Mars and Mars-Earth transfers. Earth departure date is Nov. 23, 1981, with a total mission duration of 1040 days; the Mars stay time is 420 days. As can be seen, with the phase options indicated, the

TABLE 7

BASELINE MISSION 4 - SUMMARY

Sample Size	20 kg (Earth Orbit Capture)
Launch Vehicle	Titan IIID/Centaur (Dual Launch)
Mission Duration	960 Days

	FLIGHT TIME	SEP TIME
Earth-Mars Transfer (Ballistic)	290 days	
Mars Stay Time	310	
Mars-Earth Transfer (SEP)	360	287 days
	960	287

System Weight Breakdown

A. First Launch

Mars Lander/Ascent Probe (Direct Entry)		3200 kg
Aerobraking/Propulsion	630	
Lander	912	
Rover	68	
Ascent Vehicle	1590	
Sterilization Canister		215
Probe Mounting Structure		169
Spacecraft Equipment Module		453
		4037 kg

B. Second Launch

Mars Capture Stage (Solid)		1896 kg
Mars Escape Stage (Solid)		500
SEP Stage		167
Propulsion System (3.9 kw)	117	
Propellant + Tankage	50	
Spacecraft Equipment Module		453
Earth Capture Stage		209
		3225 kg
Titan IIID/Centaur Capability		4400 kg

TABLE 8

BASELINE MISSION 5 - SUMMARY

Sample Size	10 kg (Earth Orbit Capture)
Launch Vehicle	Intermediate-20 (Single Launch)
Mission Duration	680 Days

	FLIGHT TIME	SEP TIME
Earth-Mars Transfer (Ballistic)	290 days	
Mars Stay Time	10	
Mars-Earth Transfer (SEP)	380	245 days
	680	245

System Weight Breakdown

Mars Lander/Ascent Probe (Direct Entry)		2803 kg
Aerobraking/Propulsion	558	
Lander	847	
Rover	68	
Ascent Vehicle	1130	
Sterilization Canister		188
Probe Mounting Structure		148
Mars Capture Stage (Solid)		4807
Mars Escape Stage (Solid)		1294
SEP Stage		1461
Propulsion System (27.8 kw)	830	
Propellant + Tankage	631	
Spacecraft Equipment Module		453
Earth Capture Stage (Solid)		146
Earth Departure Vehicle		11300 kgs
Intermediate-20 Capability		12250 kgs

TABLE 9

BASELINE MISSION 6 - SUMMARY

Sample Size	25 kg (Earth Orbit Capture)
Launch Vehicle	Intermediate-20/Centaur (Single Launch)
Mission Duration	600 Days

	FLIGHT TIME	SEP TIME
Earth-Mars Transfer (Ballistic)	220 days	
Mars Stay Time	20	
Mars-Venus Transfer (SEP)	190	157 days
Venus-Earth Transfer (Ballistic)	170	
	600	157

System Weight Breakdown

Mars Lander/Ascent Probe (Direct Entry)		3400 kgs
Aerobraking/Propuslion	675	
Lander	932	
Rover	68	
Ascent Vehicle	1725	
Sterilization Canister		228
Probe Mounting Structure		180
Mars Capture Stage (Solid)		9066
Mars Escape Stage (Solid)		1421
SEP Stage		835
Propulsion System (19.6 kw)	588	
Propellant + Tankage	247	
Spacecraft Equipment Module		453
Earth Capture Stage (Solid)		217
Earth Departure Vehicle		15800 kgs
Intermediate-20/Centaur Capability		20000 kgs

TABLE 10

MSSR MISSION SELECTION SUMMARY

BASELINE MISSION	1	2	3	4	5	6
TRAJECTORY PROFILE						
MISSION DURATION (DAYS)	1155	1055	950	960	680	600
MARS STAY TIME (DAYS)	34	49	22	310	10	20
EARTH-MARS TRANSFER	SEP	SEP	SEP	BALLISTIC	BALLISTIC	BALLISTIC
MARS CAPTURE	SEP	SOLID RETRO	SEP	SOLID RETRO	SOLID RETRO	SOLID RETRO
MARS ESCAPE	SEP	SEP	SEP	SOLID RETRO	SOLID RETRO	SOLID RETRO
MARS-EARTH TRANSFER	SEP	SEP	SEP	SEP	SEP	SEP / VENUS SWINGBY

- DIRECT MARS ENTRY -

- EARTH ORBIT CAPTURE RECOVERY -

DELIVERY SYSTEM						
LAUNCH VEHICLE	TITAN III D / CENTAUR	TITAN III D (7) / CENTAUR	TITAN III D (7) / CENTAUR	TITAN III D/ CENTAUR	INT-20	INT-20/ CENTAUR
NO. OF LAUNCHES	1	1	1	2	1	1
SAMPLE SIZE (KGS)	10	10	10	20	10	25
EARTH DEPARTURE MASS (KGS)	5278	6425	5116	4037/3225	11300	15800
SEP POWER, 1 AU (KW)	18.5	22.5	20.5	3.9	27.8	19.2
SEP PROPULSION TIME (DAYS)	784	668	586	287	245	157

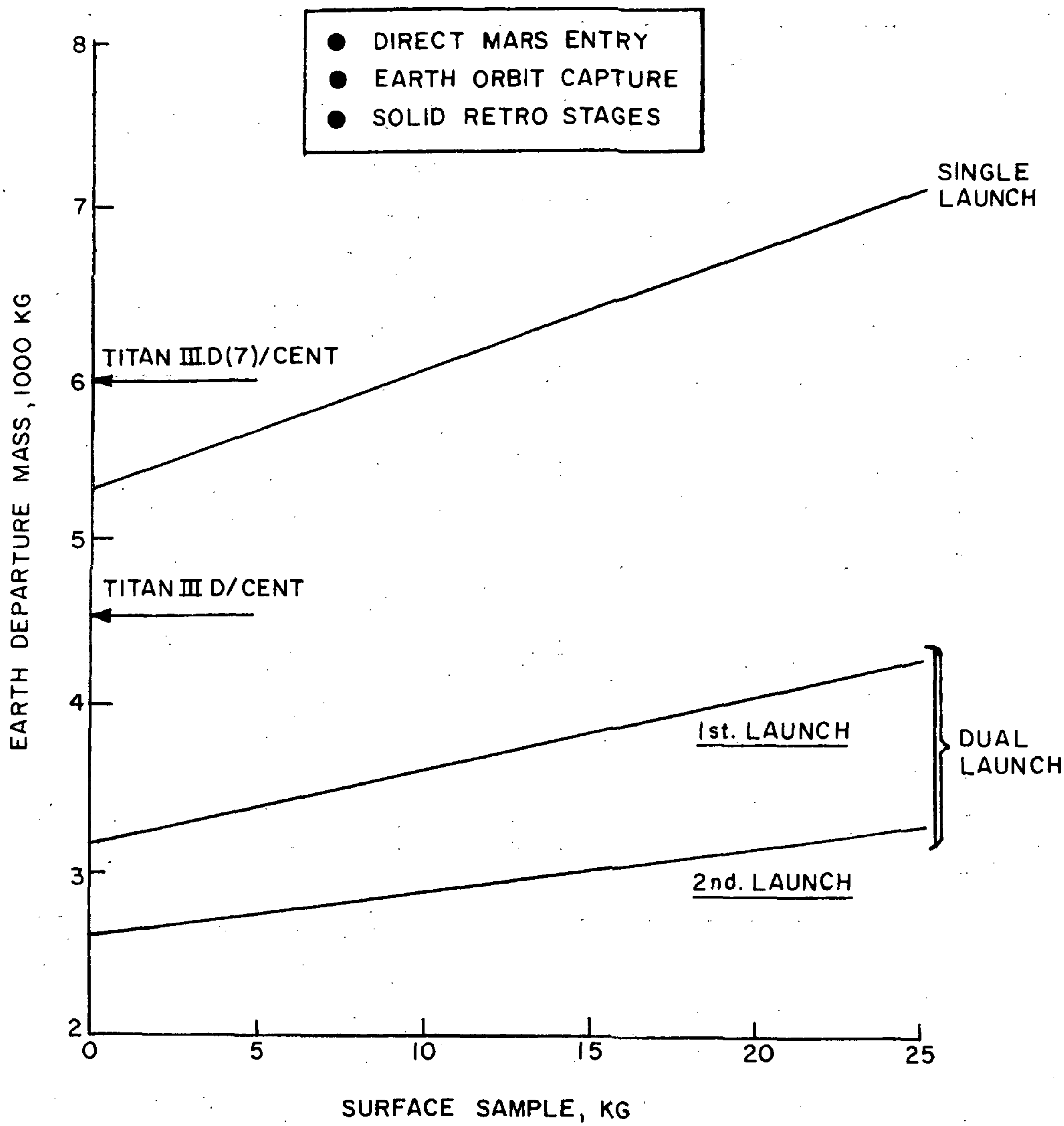


FIGURE 27. ALL-BALLISTIC PERFORMANCE FOR 1040 DAY MSSR MISSION

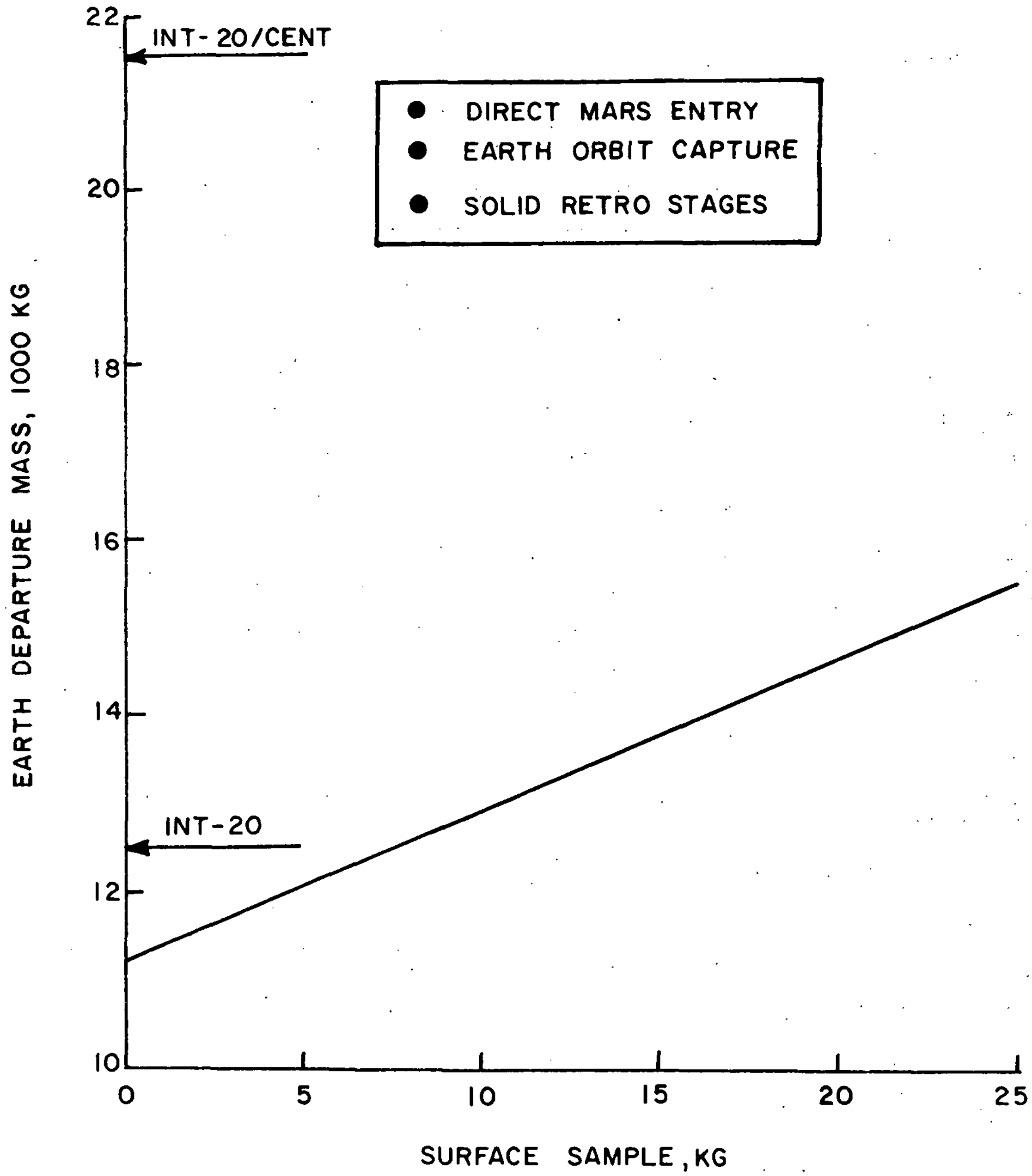


FIGURE 28. ALL-BALLISTIC PERFORMANCE FOR 625 DAY MSSR MISSION WITH INBOUND VENUS SWINGBY

Titan IIID(7)/Centaur single launch can marginally return a 5 kg sample, whereas the Titan IIID/Centaur dual launch concept is capable of returning the full range of sample size.

Figure 28 presents payload data for a mission which uses a fast, opposition type Earth-Mars transfer and a Venus swingby return to Earth. The launch date is Nov. 17, 1981, with a total mission time of 625 days and a 30-day Mars stay time. The Intermediate-20/Centaur is required to return samples greater than about 7 kgs. Comparing this with the 600-day mission which uses a SEP/Venus swingby Earth return and solid retro option (see Figure 26), the all-ballistic mission provides slightly better performance. This is due largely to the fact that the SEP stage is being used only for the Mars-Venus transfer and must be carried as inert mass from the launch phase through the Mars escape maneuver. A fast, opposition type mission with a direct Mars-Earth transfer was also examined, but the energy requirements were much too high for a practical mission application.

CONCLUSIONS

Solar electric propulsion can be used effectively to accomplish the Mars Surface Sample Return mission. Performance advantages over all-ballistic (chemical propulsion) systems are either a smaller launch vehicle requirement for comparable trip time and sample size, or a significant reduction in trip time for comparable launch vehicle and sample size.

The major results of this study are listed below:

(1) A sample of 10-25 kg can be returned to an Earth orbit compatible with manned spacecraft recovery operations.

(2) State-of-the-art chemical propulsion systems may be utilized; solid propellants for retro maneuvers and earth-storable liquid propellants for ascent from the Martian surface.

(3) Titan IIID/Centaur vehicles (5 or 7 segment) can be employed in the single-launch mode provided that SEP is used for both outbound and return interplanetary transfers and, at least, the Mars escape maneuvers.

(4) The above mission concept requires a total trip time of 2.5 to 3 years, a powerplant size of about 20 kw, and a 60-70% propulsion duty cycle.

(5) Shorter missions (1.5-2 years) can be accomplished with the INT-20 or INT-20/Centaur launch vehicles. However, SEP should be used only for the return transfer in order to limit the SEP power requirement.

Since a mission duration of 2.5 years does not seem unreasonable, the best application of SEP may well be Mission Concept No. 3 which utilizes the Titan IIID(7)/Centaur launch vehicle. There is a healthy margin of safety between the Earth departure weight and the launch vehicle capability. The problem areas or reservations concerning this choice are the possible difficulty of mechanizing the thrust steering program during the Mars spiral maneuvers, and the long wait (118 days) between landing and rendezvous with the orbital bus.

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