REPORT GD | A-AOK64-006
31 JANUARY 1964

A STUDY OF MANNED INTERPLANETARY MISSIONS

CONTRACT NO. NAS8-5026 (FOLLOW-ON)

CONDENSED SUMMARY REPORT

Prepared for
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(Future Projects Office)
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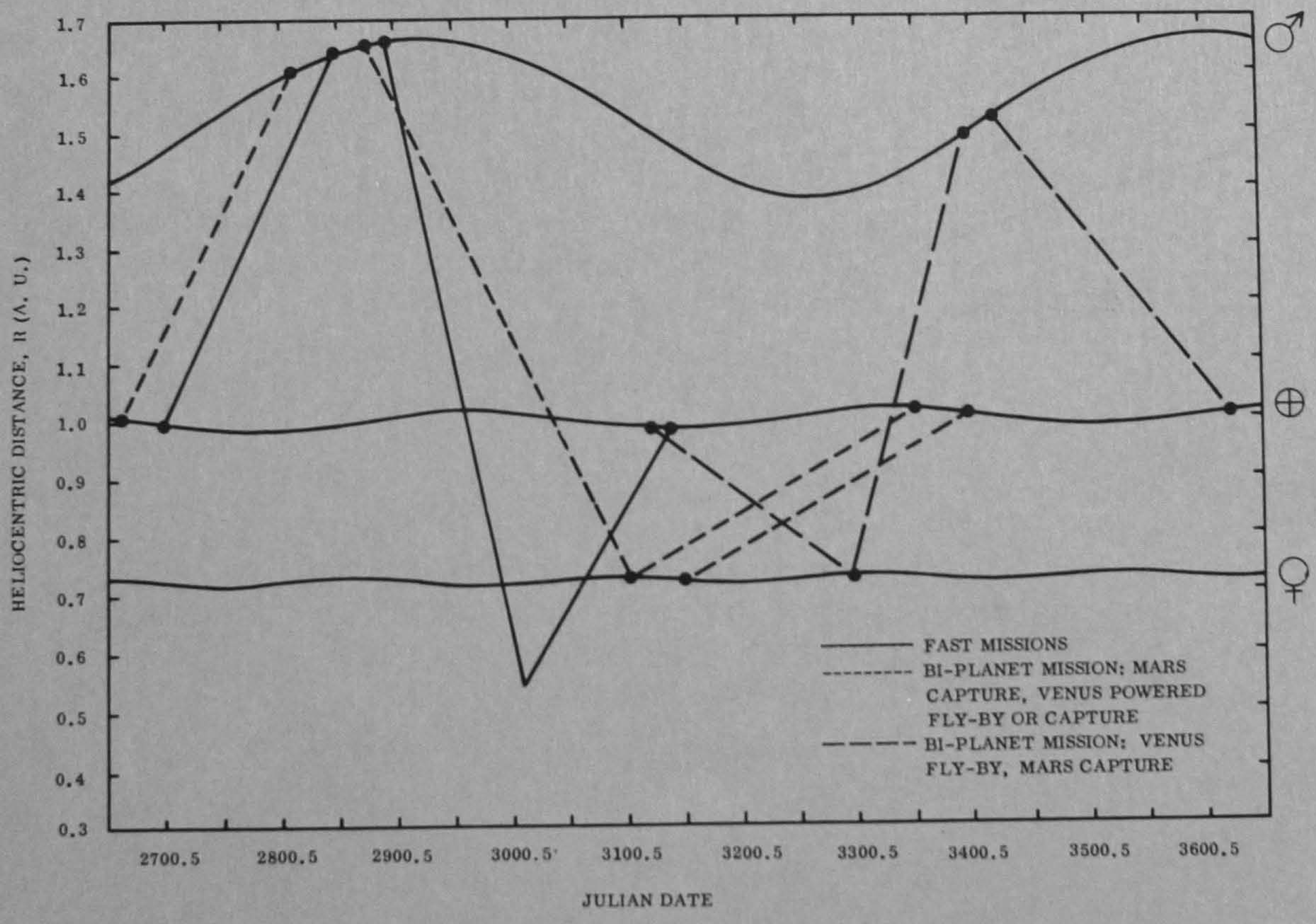

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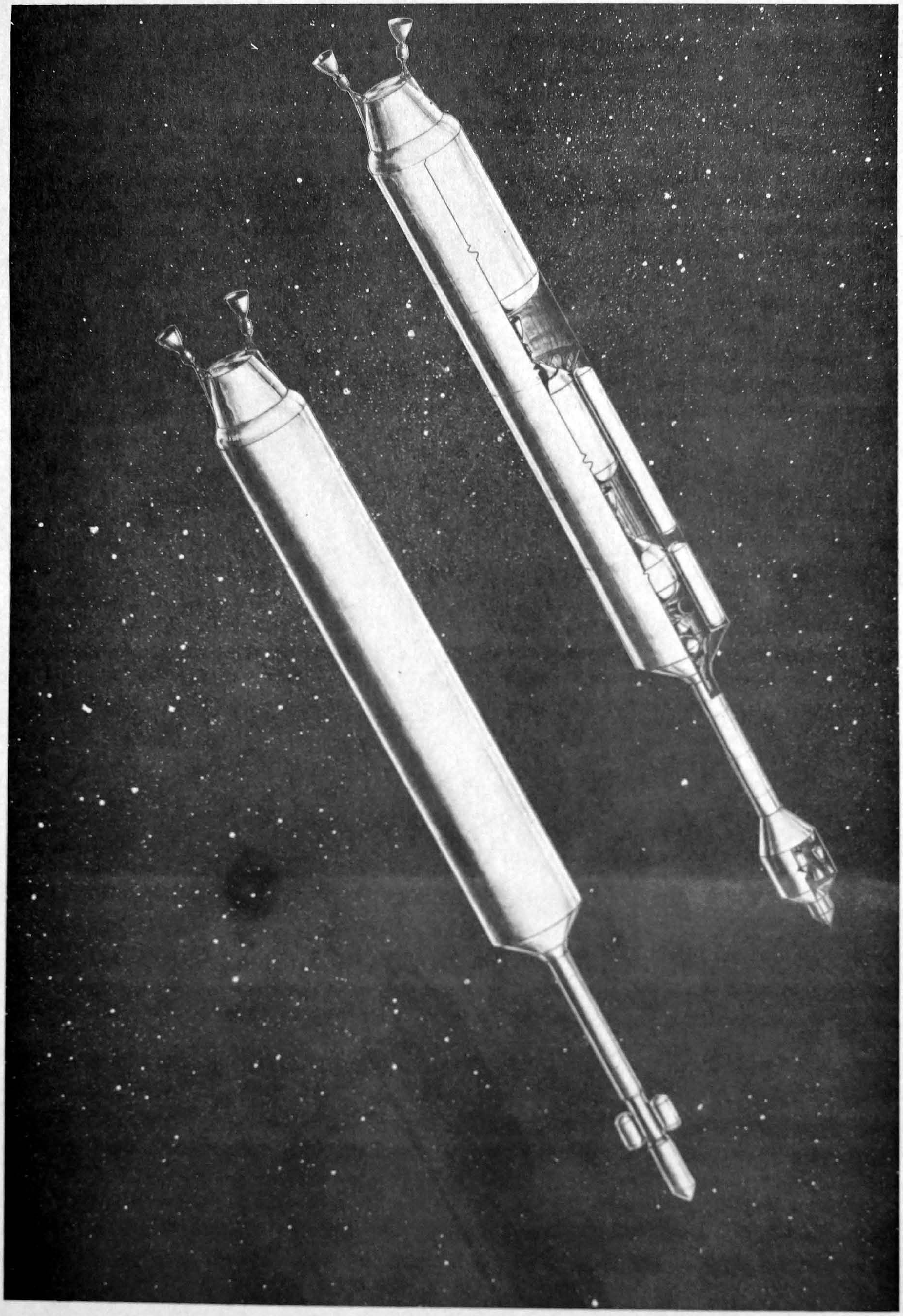

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A Study of
EARLY MANNED
INTERPLANETARY MISSIONS
(EMPIRE Follow-On)

Condensed Summary Report

Prepared for

GEORGE C. MARSHALL SPACE FLIGHT CENTER
(FUTURE PROJECTS OFFICE)
HUNTSVILLE, ALABAMA

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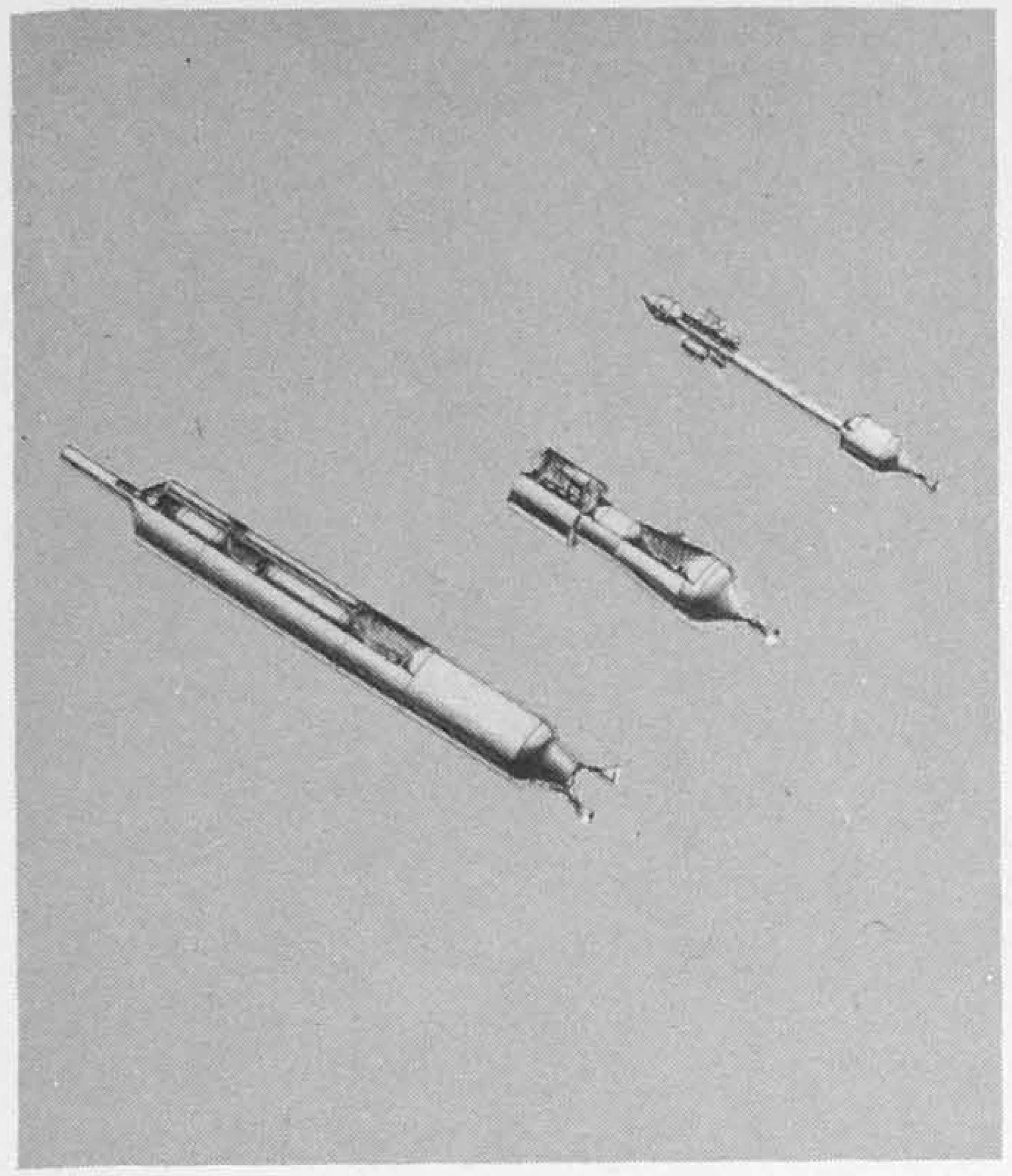
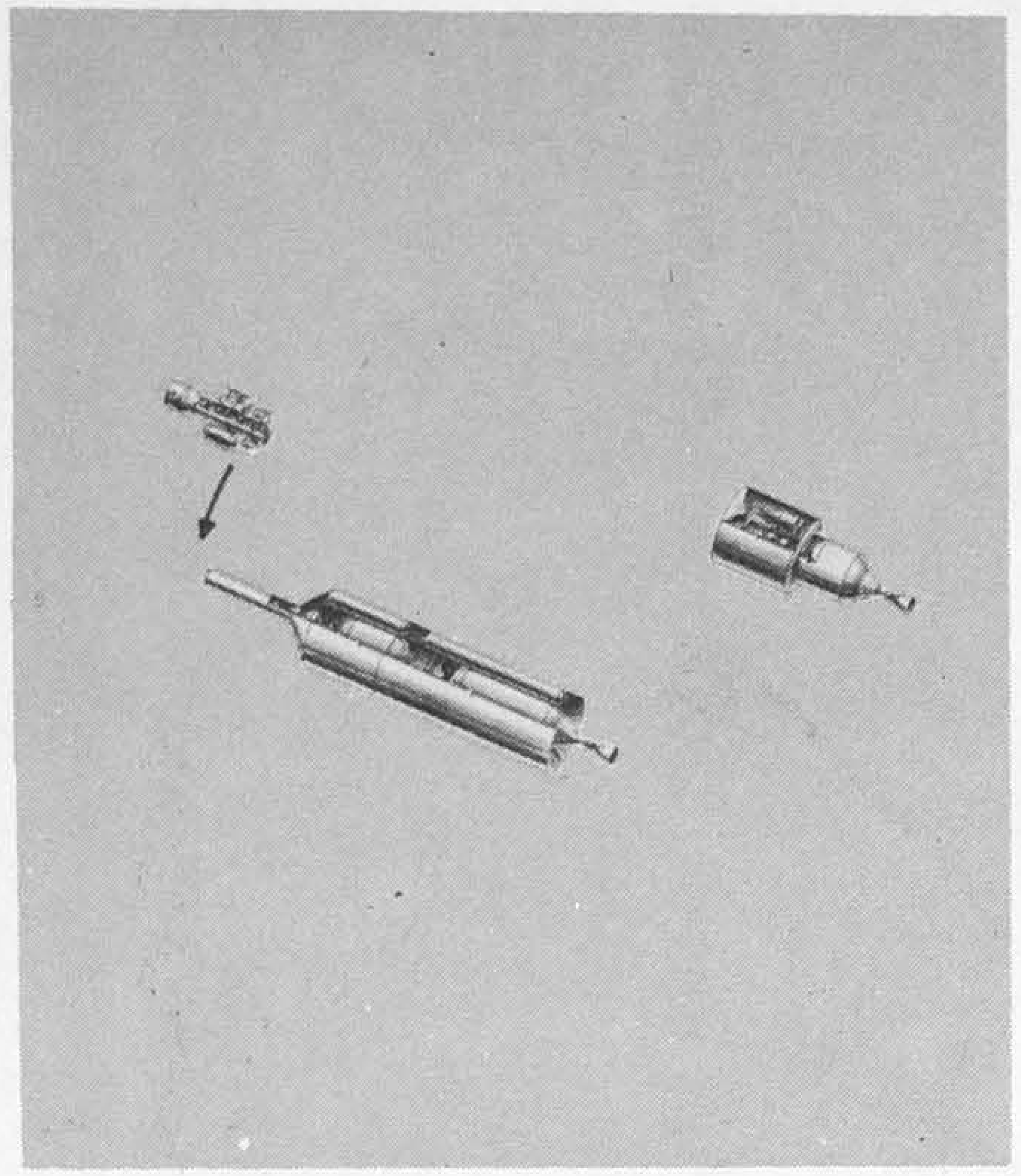
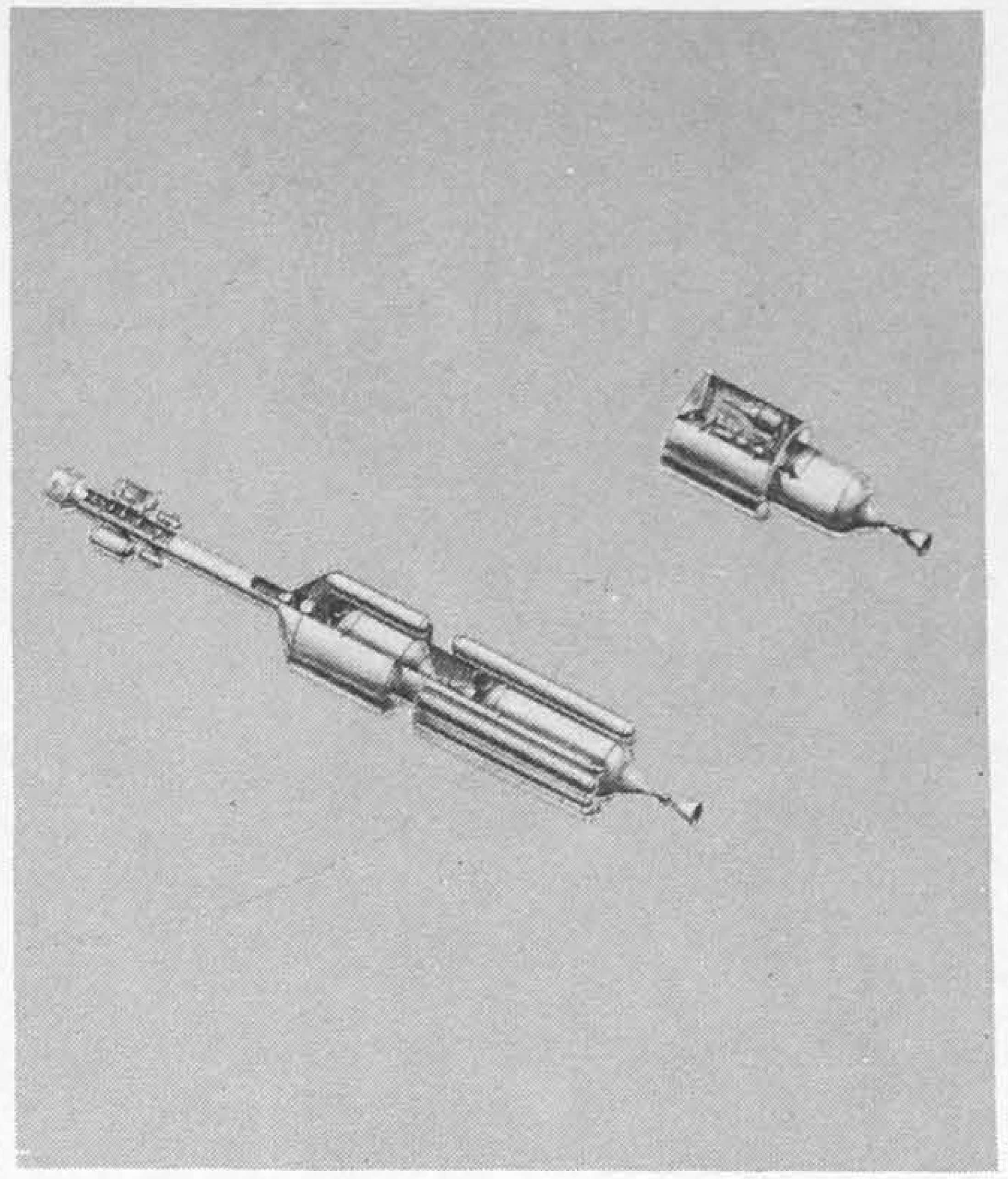


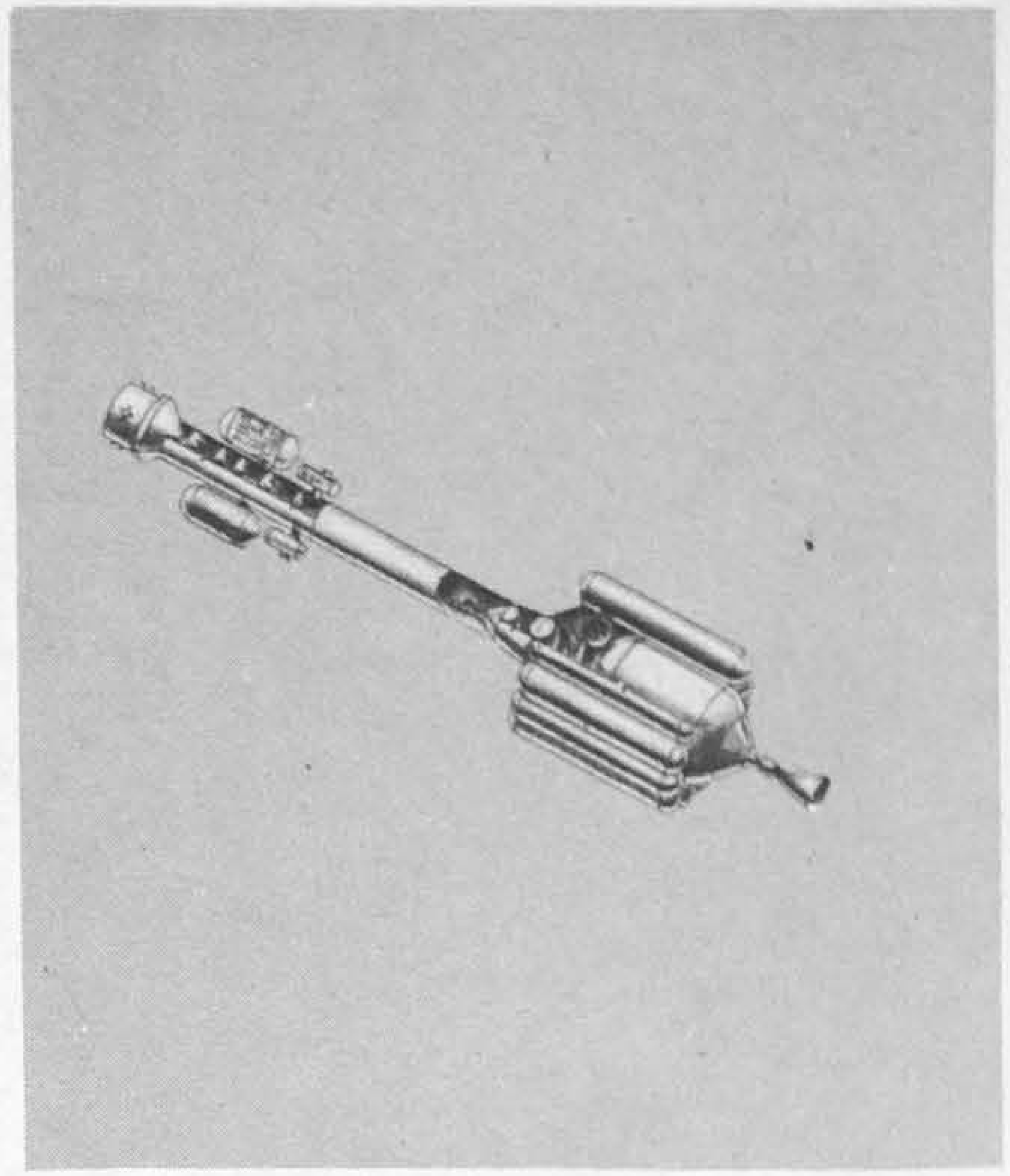
Earth Departure Condition



Transfer of LSS During
Outbound Coast



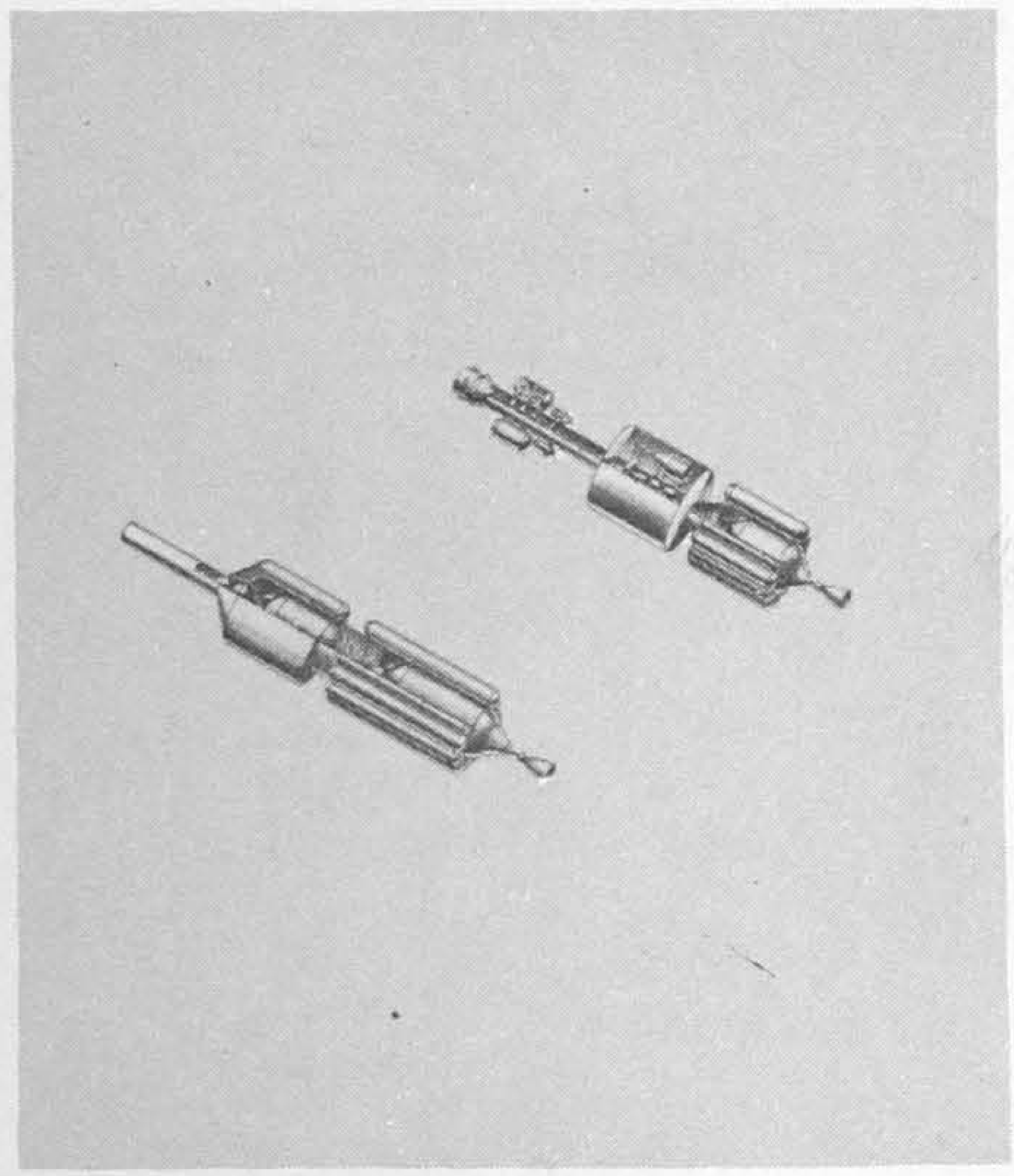
Mars Arrival Condition



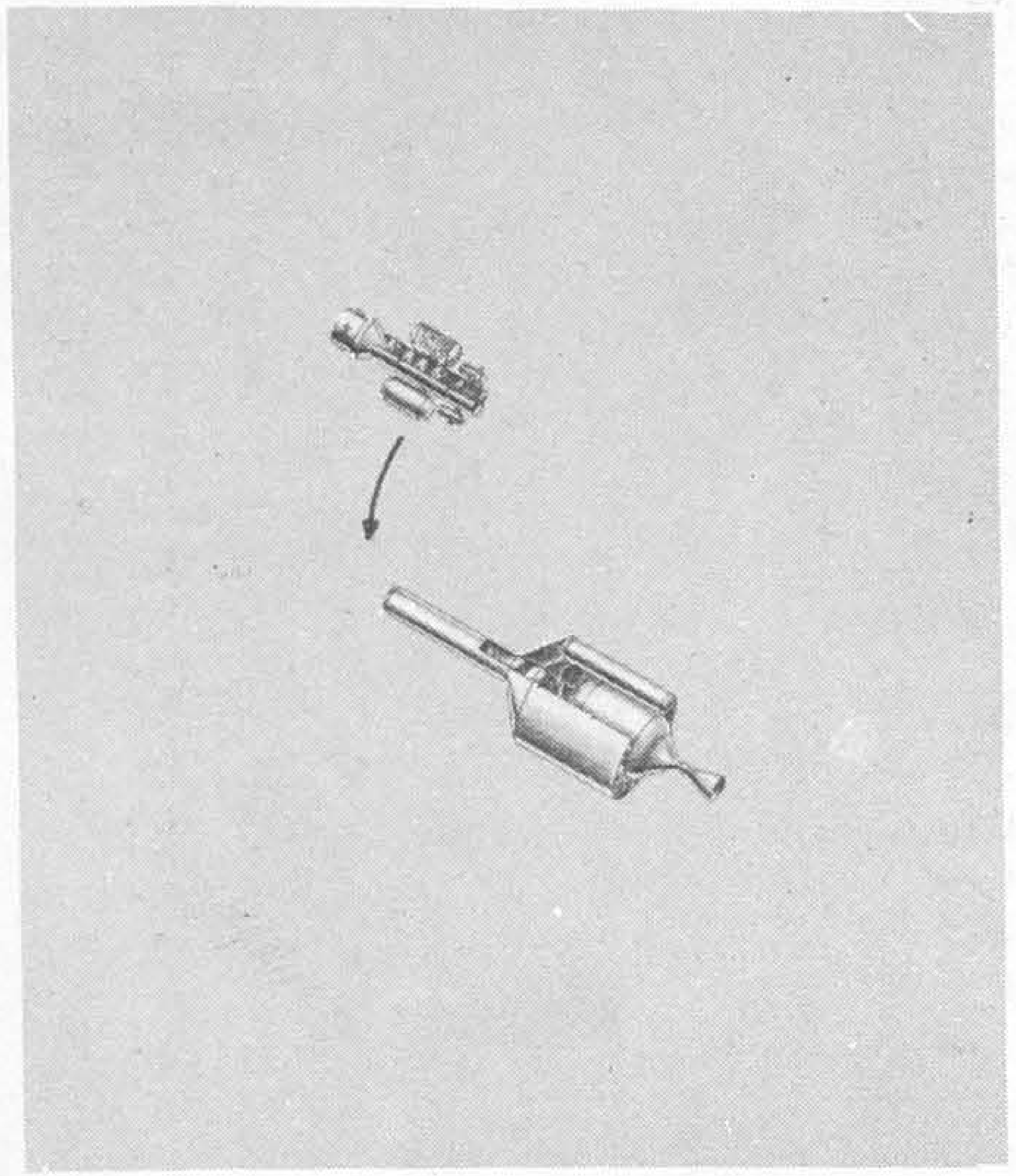
Mars Departure Condition

Fig. 5-15

IVAM (2)



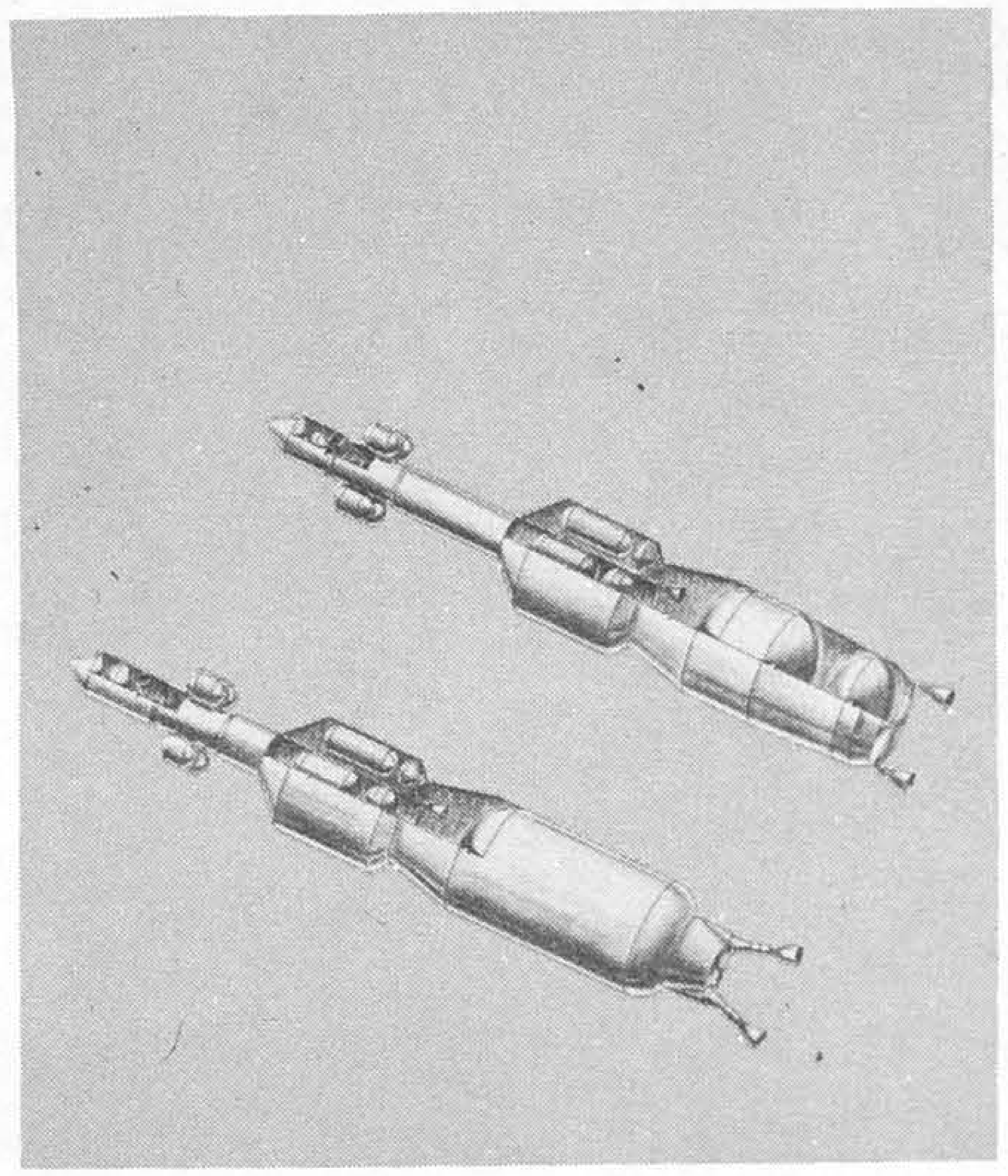
Earth Departure Condition



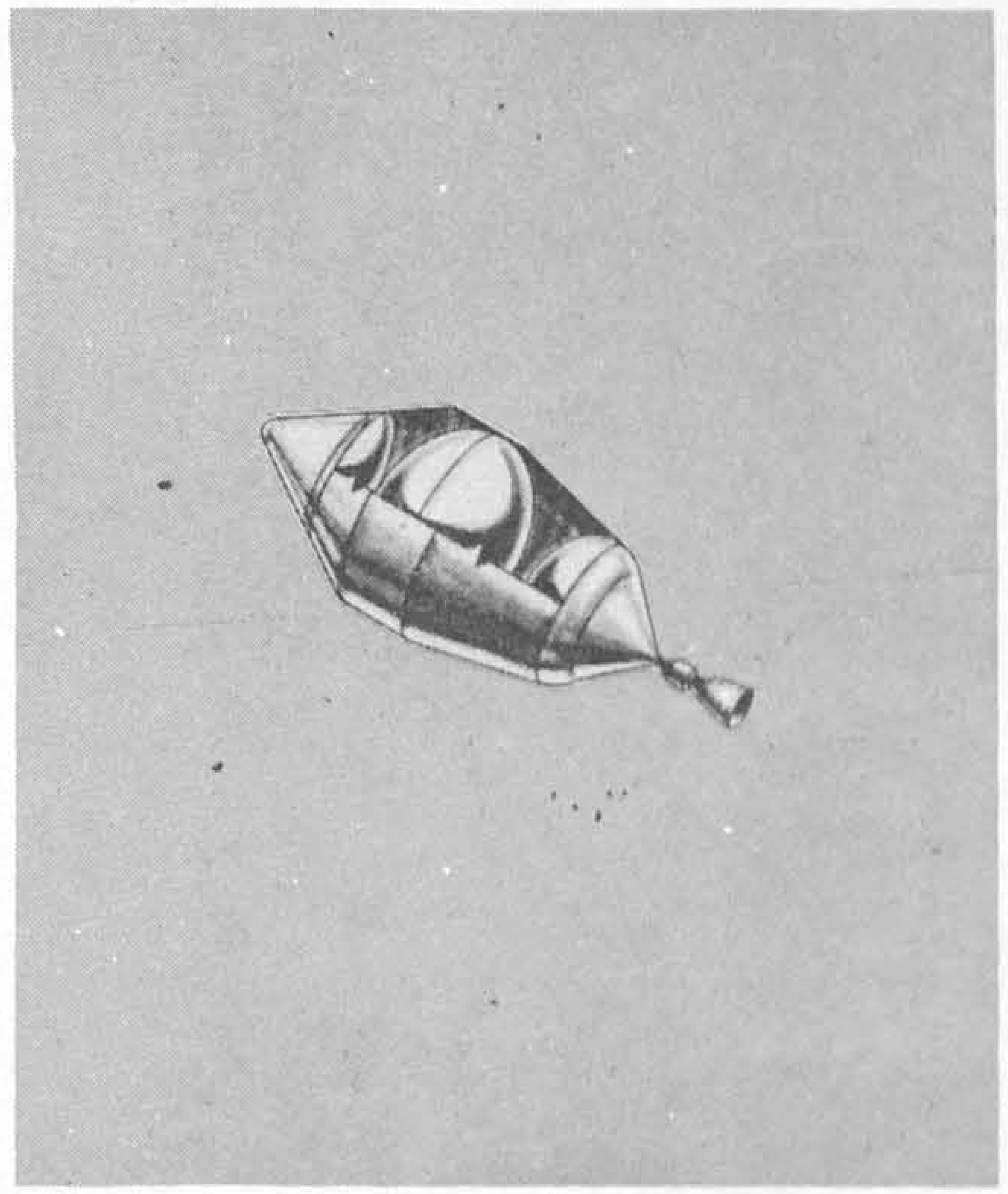
LSS Transfer to PM-3 and
PM-4 in Mars Capture Orbit

Fig. 5-16

COVAM (2)



Nuclear Powered Vehicle,
-23 Config., to Mars
Chemical Vehicle to Venus



Earth Return Condition

OVAM

Fig. 5-13

FOREWORD

This report represents a condensed summary of the work performed under Contract NAS8-5026 and is submitted in partial fulfillment of technical documentation of the study. The work was performed by the Advanced Studies Office, General Dynamics/Astronautics under the cognizance of Dr. H. H. Koelle, Director, Future Projects Office, NASA/MSFC, and Dr. H. Ruppe, Deputy Director, Future Projects Office and Technical Manager of Contract NAS8-5026. The comments and recommendations by members of the Future Projects Office have been most helpful.

INTRODUCTION

A manned capture mission to Venus or Mars represents a key mission in preparation for the goal of the first phase of a long-range program of manned planetary exploration and base establishments.

Instrumented probes are very important in preparing the way for manned flights, but cannot replace manned exploration of the solar system or even our neighboring planets, if for no other reason than that they lack the judgement and superior reliability characteristics contributed by man, without which a task of such enormity cannot be accomplished.

In the course of this study, the following principal conclusions were obtained:

(1) Fast round-trip missions to Venus (360-420 d) are characterized by short outbound transfer, elliptic capture at $n = r_A/r_P \geqq 8$ and a long return orbit. Hyperbolic entry velocities are under 50,000 ft/sec without the aid of a retro-maneuver.

(2) Most economic fast ($\prec$ 500 d) round-trip capture mission to Mars consists of a short outbound transfer in a favorable window, circular capture, close-perihelion return transfer orbit, slow-down near the perihelion (perihelion brake), using solar-thermal propulsion thereby reducing the Earth approach velocity and a small Earth retro-maneuver, using fuel which served as shielding, followed by hyperbolic entry at 40-45,000 ft/sec. By means of this mission profile, the difference between favorable and unfavorable mission years can be greatly reduced.

(3) Bi-Planet round-trip missions to Venus and Mars offer superior flexibility in timing. Bi-Planet capture missions, following favorable transfer windows shown mission velocities similar to those for fast round-trip mission to Mars alone.

(4) Combinations of capture and powered fly-by (PFB) in bi-planet missions can be used to reduce significantly the arrival velocity at Earth upon return from Mars. The mission period is longer than in (2), namely close to 500 days, but this mission offers the advantage of visiting both planets in one mission.

(5) Investigation of powered maneuvers during fly-by was found to increase the number of available mission windows, to increase mission windows from days to weeks and to reduce mission velocity as well as mission period, compared to non-powered fly-by.

(6) For a Venus capture mission, using circular capture and retro-thrust to Apollo entry condition, (36,200 ft/sec), and nuclear stages with I_{sp} = 765,

- (a) elimination of Apollo entry condition and entry at 46,000 ft/sec reduces the orbital departure weight (ODW) by 11%
- (b) changing from circular to elliptic capture at n = 8 reduces the orbital departure weight by 21%. This figure takes into account the likely need for rotating the major axis of the ellipse.
- (c) omitting retro-thrust to Apollo entry and changing to elliptic capture, reduces the orbital departure weight by 29%
- (d) applying atmospheric braking reduces the initial orbital weight by 27%
- (e) applying atmospheric braking and omitting Apollo reduces the orbital departure weight (ODW) by 48%.

(7) For a Mars mission in 1975, using an all-nuclear vehicle with 900 sec in the planetary stages, and capturing in a circular orbit with retro to Apollo conditions at Earth return,

- (a) increase in entry velocity to 50,000 ft/sec reduces the ODW by 21%
- (b) increase in entry velocity to 60,000 ft/sec reduces the ODW by 32%
- (c) increase in entry velocity to 70,000 ft/sec reduces the ODW by 40%
- (d) application of perihelion braking, using solar-thermal propulsion, reduces the ODW by 45%
- (e) application of aerodynamic braking reduces the ODW by 44%
- (f) return from Mars with powered fly-by near Venus reduces the ODW by 51%
- (g) application of aerodynamic braking and perihelion brake reduces the ODW by 48%

(8) Enlarging the diameter of Saturn V to 50 ft is of greater importance to its use as ELV for hydrogen-carrying for interplanetary vehicles than increase in payload by 10-25 percent.

(9) Based on the determination of a characteristic gross number of binary bits of information gathered on each of these missions and based on the gross mission cost listed above, the mission yields, if expressed in terms of gross number of binary bits per gross dollar expended, were found to compare as follows: Venus: fly-by or powered fly-by: ~6; elliptic capture (20 days): ~ 36.5. Mars: powered fly-by: ~9; circular capture (30 days): ~75; circular capture and surface excursion: ~81. Mars-Venus bi-planet powered fly-by: ~14; Mars circular capture and Venus fly-by: ~ 89.

(10) Considering a Mars base as the principal goal of the first phase of man's exploration of this solar system, the evaluation of the different mission types, exclusive of surface excursion mission, can be summarized as follows:

- (a) On the basis of lowest cost and highest probability of mission success, Venus powered fly-by mission rates highest.
- (b) On the basis of cost, early feasibility and relevance of information gathered relative to the establishment of a Mars base, highest rating goes to the Mars powered fly-by mission.
- (c) On the basis of variety of information, the preference goes to the Mars-Venus powered fly-by mission.
- (d) On the basis of total amount and cost of information and of mission operational relevance to later capture and surface excursion missions to Mars, highest rating goes to the Venus elliptic capture mission.
- (e) Highest mission yield, highest mission cost, and largest orbital departure weight characterize the Mars capture mission. Comparison between the two principal mission profiles shows that perihelion brake yields the shortest mission period (460-490 days); return via Venus powered fly-by yields slightly lower orbital departure weights, higher mission yield and longer mission period (about 600 days).

(11) A planetary mission evaluation matrix was developed, employing the following groups of evaluation criteria: Required state of the art; minimum ELV required; technologically (earliest) feasible schedule; programmatically feasible schedule; relative development risk; mission risk under conditions of programmatically feasible schedule; required launch facilities; development cost and operating cost; prime objectives of mission; expected mission yield; comparative evaluation of using instrumented probe for the same mission objective in terms of technological feasibility, mission yield and cost; compatibility with follow-on mission objectives; harmony of mission with national space program in the time period in question. This set of evaluation criteria was applied to the evaluation of a variety of missions (cf. Vol. II: Summary).

2. STUDY OBJECTIVES

The second phase of a Study of Early Manned Planetary Missions has been completed for the Future Projects Office of the NASA George C. Marshall Space Flight Center, Huntsville, Alabama. The primary study objectives were defined as follows:

A. A detailed definition of the mission profile of a fast trip to Mars in the 1975 time period. The auxiliary vehicles (i.e., manned landers, unmanned probes, etc.) to complete this mission profile should be considered as a secondary objective.

B. A preliminary design of a space vehicle system suitable for this mission profile, including Earth launch requirements, orbital operations requirements, nuclear engine requirements, scientific mission requirements and atmospheric re-entry requirements.

C. A compatibility study of this space vehicle system for other missions within the national space program.

D. The growth potential of the proposed space vehicle system.

The expected results are to include the following:

A. Refinements of the analysis of the four basic mission modes investigated by GD/A in the first part of this study.

B. Refinement of the basic mission requirements in terms of weight, volume, power and other critical elements.

C. Refinement of launch window specifications for Earth and target planet.

D. Definition of abort and abort possibilities throughout the mission. Check list of the more probable emergency-type situations and how to cope with them.

E. Refinement and implementation of previous work done in convoy vehicle design and systems analysis.

F. Continued investigation of crew requirements.

G. Detailed study of the development plan for this mission. The preliminary development plan shall contain a cost estimate for the total mission.

3. RELATIONSHIP TO OTHER NASA EFFORTS

The relationship of manned planetary round-trip missions to other NASA efforts is surveyed in Fig. 3-1. The interrelation was divided into 6 basic areas.

(1) Destination payload, especially orbital reconnaissance equipment, data processing equipment, a variety of probes and the Mars excursion module (MEM)

(2) Propulsion system, design criteria and configuration of the interplanetary vehicle (I/V)

(3) Earth return conditions, particularly the state of the art in hyperbolic entry into the Earth atmosphere and in hyperbolic rendezvous with the returning I/V

(4) Earth launch vehicle (ELV) availability and characteristic constraints

(5) The supporting instrumented probe program with reference to Mariner, Voyager and roving interplanetary probes (RIP's)

(6) The manned space station program as the principal instrument for orbital development and testing of the ecological system and other life support equipment and for long-duration training of the mission crew. The manned space station (or the orbital laboratory) is the principal means of orbital development and testing of practically the entire operational payload of the I/V.

The individual areas are detailed further in Fig. 3-1. A distinction is made between contributory developments which presently add to the relevant state of the art and required research and development, both based on conditions of FY-64. The contributory developments represent the principal foundation for an early "minimum-type" manned planetary mission.

Fig. 3-2 shows key research and development requirements in preparation of manned planetary missions which are specified in Tab. 3-1, on last page.

It was established that development of a chemonuclear or all-nuclear I/V and the preparation of manned planetary flights would furnish the following contributions to other areas of astronautics:

1. At least one type of long-duration ecological system for a crew of about 8 persons operating over a period of 450 to 600 days.

2. Complete life support sections, modularized, which can be assembled in orbit to form a space station or on the Moon to form the nucleus of a base.

3. A lunar shuttle vehicle of a variable payload capability, depending primarily on the number of stages of the I/V configuration used.

4. Providing mission specifications and particular incentives for the development of nuclear engines.

5. Providing incentives and specifications for modifications of Saturn V and for the Post-Saturn ELV. Specifically, it was found that enlarging the diameter of Saturn V, in order to increase the length and volume of its payload section, is more important than increasing its payload by 10-20%, assuming hydrogen-based I/V's are being used.

4. METHOD OF APPROACH AND PRINCIPAL ASSUMPTIONS

The study was divided into three phases (Fig. 4-1). The first phase involved study of various vehicle configurations and concepts, leading to the selection of preferred configurational concepts for the propulsion modules (PM) and the life support section (LSS). General payload requirements and mission characteristics were established, basic performance requirements determined and mission modes were defined. This phase was completed during the first portion of the study contract NAS8-5026. The second and third phase were completed during the present study period.

In the second phase, the selected configurations were studied in detail. From these studies relatively accurate weight scaling coefficients were derived in order to provide a reliable basis for the parametric analyses in the third phase. Special attention was given to the radiation protection of the crew and to vehicle/engine integration problems, including the analysis of interaction between openly clustered nuclear engines (solid core, graphite). Operational problems were treated in greater detail. Various vehicle assembly modes were investigated and coordinated with Earth launch vehicle (ELV) requirements. Ground launch operations, orbital pre-departure operations and mission operations were normalized to permit a systematic and consistent analysis. In the area of mission analysis, guidance and navigational aspects were analyzed. Capture mission studies to either planet were continued. Powered fly-by (PFB) missions to either planet were investigated, as well as capture missions to both planets (bi-planet capture missions) and "hybrid" missions, involving PFB near one planet and capture at the other (PFB/C missions); or vice versa (C/PFB missions). While the weight determination deliberately was kept conservative, to conform with the expected realities of practical development requirements considerable emphasis was placed in the mission analysis area, on measures to increase the attainable payload fraction for vehicles with given engine specific impulses, by reducing the mission requirements.

In the third phase, mission analysis and vehicle systems design and analysis were integrated, resulting in the development of several nomographic methods, coordinating mission velocity requirements (by individual maneuvers) with vehicle stage weight determination. These methods are based on integration of the scaling coefficients to mass fraction coefficients and therefore permit a comparatively rapid determination of the orbital departure weight (or initial payload fraction) for a given mission. A weight determination computer program was developed, based directly on the scaling coefficients. In the operations analytical area, vehicle assembly modes were integrated with mission profiles; ELV selection and launch requirements determined, taking reliabilities for orbit delivery, orbital mating and orbital fueling into account. The resulting procurement figures represent input into the cost analysis. In the area of program analysis, critical development problems and ground and flight test programs for the most important components had to be evaluated before development schedules could be established. Development schedules and launch requirements, in turn, form the basic inputs for the analysis of development cost, indirect cost and direct mission cost data.

The overall study was based on the following principal assumptions:

1. Target planets: Venus or Mars or both.
2. Reference mission years: 1975-1977, with parametric extension of mission characteristics into the late seventies and early eighties.
3. Reference mission group: Capture, with secondary consideration of fly-by and surface excursion.
4. Reference mission objectives (MiO.):

   MiO-1: Orbital reconnaissance of planetary surface (minimum objective)

   MiO-2: MiO-1, plus deployment of auxiliary vehicles, such as: environmental satellites (ES), atmospheric high-speed entry probes (AEP) atmospheric slow-descent or buoyant probes (Floaters), landing probes (Landers), landing probes capable of returning to the I/V (Returners) and Mars moon probes (Phopro, Deipro)

   MiO-3: MiO-1, plus MiO-2, plus manned surface excursion capability

MiO-2 was used as the principal or reference mission objective in determining destination payloads.

5. Interplanetary vehicle (I/V) propulsion systems assumed to be available are listed in Tab. 4-1. The thrust level of the "second generation" 250 k nuclear engine was determined to be near-optimum in the first phase of this study. The "advanced" nuclear engine (fast neutron spectrum) was used for Mars capture fast missions in combination with either the 250 k engine or the 700 k engine for Earth departure.
6. Interplanetary vehicles (I/V) considered were all of the hydrogen- or oxygen/hydrogen carrying type, in accordance with the engine systems considered. Two basic vehicle types were assumed: the convoy vehicle (CV) and the multiplex vehicle (MV), primarily the duplex vehicle (DV).

   In the convoy mode, at least two vehicles depart successfully from orbit. The various loads are distributed over the convoy vehicles. In the duplex mode, two vehicles are coupled, instead of traveling separately as in the convoy mode. Operational and destination payloads are jointly mounted in the duplex. The destination payload and part of the operational payload are jettisonable.
7. The following vehicle assembly modes were assumed:

   DFM = direct flight mode (i.e. complete assembly and operational readiness on the ground)

   OVAM = orbital vehicle-assembly mode (module mating and/or fueling in Earth orbit)

   IVAM = interplanetary vehicle-assembly mode (Life support section (LSS) is mated during heliocentric interorbital coast with the propulsion modules required to complete the mission)

   COVAM = capture orbit vehicle-assembly mode (LSS is mated in capture orbit with the propulsion modules required to complete the mission)
8. The following capture modes near the target planet were assumed:

   (a) Retrothrust capture into elliptic orbit (Venus)

   (b) Retrothrust capture into circular orbit (Mars)

   (c) Aerodynamic capture into ellipse of $n = r_A/r_P = 49$; subsequent retro-thrust into circular orbit (Venus, Mars)

   (d) Aerodynamic capture and slow-down to near-circular velocity (Venus, Mars)
9. Powered fly-by modes were considered for both planets. PFB involves a maneuver near the periapsis of the planet, to change from the arrival hyperbola into a suitable departure hyperbola. The powered maneuver is used as a means to modulate the effect of the hyperbolic encounter with planetary field (which does the main job in changing the heliocentric orbit of the interplanetary vehicle) as required in the particular situation.
10. The following Earth return modes were assumed:

    (a) Retro-thrust to Apollo entry conditions

    (b) Retro-thrust to specified hyperbolic entry conditions (HE_x; x = specified velocity)

    (c) Hyperbolic capture and slow-down to high subparabolic speed, with subsequent Apollo entry (2-pass return mode)

    (d) Direct hyperbolic entry

    (e) Hyperbolic rendezvous (HR). The returning I/V meets in its hyperbolic orbit with a (manned) pick-up vehicle (PUV) sent from Earth. The crew transfers and returns to Earth in the PUV.
11. The following Earth launch vehicles (ELV) were assumed to be available:

    (a) Saturn V (Apollo configuration)

    (b) Saturn VM (Saturn V with 50-ft diameter, but unchanged payload weight capability)

    (c) Post-Saturn (ELV with 10^6 lb payload capability and practically no limits on diameter, length or volume of the 10^6 lb payload)

    It is realized that a Post-Saturn vehicle is unlikely to be available in 1975. It was considered here for purposes of comparison and in compliance with the work statement.
12. Weight assumptions: Even at the risk of arriving at "unattractive" vehicle weights and, consequently, launch requirements and direct cost figures, no compromise was made with an earnest attempt to keep the weight analysis realistic. Because of many intricacies and detail assumptions which enter the weight analysis and which can not always be spelled out in detail, weight determinations are to a degree a matter of trust. Every attempt has been made to avoid weight figures which could be seriously misleading regarding the practicality of a particular mission and/or the adequancy of a particular ELV.

5. BASIC DATA GENERATED AND SIGNIFICANT RESULTS

Most of the significant results of this study concern capture missions to the target planets; combinations of capture and powered fly-by modes in bi-planet missions; multi-stage H_2-carrying vehicles using solid core reactor nuclear engines; crew sizes between 4 and 10 with their associated life support sections and shielding provisions; vehicle assembly modes; convoy investigation, orbital operations and associated ELV analysis; emergency analysis; mission planning, mission evaluation techniques; and schedule and cost studies.

5.1 Basic Data Generated

A large amount of data was generated which can be regarded as basic in that they are applicable to studies other than this one. The data are subsequently described briefly.

5.1.1 Mission Analysis

5.1.1.1 Heliocentric Transfer Orbits. Determination of favorable transfer windows (Fig. 5-1) for

Earth ⇄ Venus,	1973-84	Transfer times in
Earth ⇄ Mars,	1973-84	the 150 to 250 day
Venus ⇄ Mars,	1976-84	range.

Hyperbolic excess velocities and transfer orbit elements were computed.

5.1.1.2 Heliocentric Transfer Orbits. Determination of favorable transfer windows for slow synodic missions for Earth ⇄ Mars and Earth ⇄ Venus, 1973-84. Hyperbolic excess and transfer orbit elements were computed.

5.1.1.3 Fast Round-Trip Mission Computations. Determination of impulsive velocity changes Earth (CC; $r^* = 1.1$) and Mars (CC; $r^* = 1.3$); of associated atmospheric entry velocities at Mars and Earth; and Earth return-retro maneuvers for slow-down to hyperbolic entry velocities of 70,000, 60,000, 50,000 ft/sec and Apollo conditions.

5.1.1.4 PFB Mission Computations. Determination of Earth departure windows and mission data for PFB missions Earth ⇄ Mars in spring 1975 & 1977 and in fall 1975.

5.1.1.5 Bi-Planet Mission Computations. Determination of mission data for

Earth - Venus (PFB) - Mars (CC) - Earth
Earth - Mars (CC) - Venus (PFB) - Earth
Earth - Mars (CC) - Venus (EC) - Earth
Earth - Venus (EC) - Mars (CC) - Earth

5.1.1.6 Navigation. Determination of flight conditions at the escape point of the Earth's activity sphere and at the "impact" point of the Martian activity sphere for Earth departure periods Feb/March 1975 with the constraint that the hyperbolic approach plane (hence, capture orbit plane) coincide with the plane of the departure hyperbola 30 days later.

5.1.1.7 Finite Thrust/Weight Ratio Computations. Effect of finite thrust/weight ratios on burning time and gravitational losses in the range of 0.05 to 0.5 g initial or terminal acceleration for departure and for arrival maneuvers, respectively, involving Venus, Earth and Mars.

5.1.1.8 IBM-7090 Computer Programs (FORTRAN). For the computations sub 5.1.1.4 and 5.1.1.5, a powered fly-by computer program was developed and tied into the existing interplanetary 2-body transfer orbit program.

For the computations sub 5.1.1.7 a powered flight path integration program was developed for tangential thrust. Data outputs include important orbital characteristics of the instantaneous osculating Kepler orbit, including hyperbolic excess velotity following attainment of positive orbital energy (central force field).

A comprehensive interplanetary mission information computer program (IMI COMP) was developed providing the following outputs: (a) 2-body heliocentric transfer data; (b) space vehicle position data (in orbit plane) during heliocentric transfer; (c) local solar constant and time integral of solar constant over mission; (d) capture orbit data, including data required for computations sub 5.1.1.6.

A navigational computer program was developed for computing the heliocentric transfer orbit backwards from a specified impact point on the target planet's activity sphere as independent variable to a point (dependent variable) at the limit of the Earth's activity sphere. Purpose of the program is to determine the exact escape point on the Earth's activity sphere connected with a specified impact condition on the target planet's activity sphere, derived from specifications of target planet capture conditions.

5.1.2 Interplanetary Vehicle Systems

5.1.2.1 Propulsion Structure and Vehicle Spine Structure. Detailed analytical justification of the adopted structural arrangements of the I/V propulsion modules and of the vehicle spine.

5.1.2.2 Clustered Nuclear Engine Analysis. Two computer programs were developed.

The first program (completed) involves computation of the number of neutrons intercepted by one reactor from the other (IBM-7090, FORTRAN).

The second program (not completed) concern determination of the fraction of neutrons, and their associated energies, that penetrate the target reactor pressure vessel and reflectors.

5.1.2.3 Life Support Sections. Detailed analysis of two life support sections for a crew of 8, using a semi-closed inorganic ecological system with oxygen generation from water and LO_2 removed by means of the reverse water gas process, water reclamation from utility water, urine and atmospheric humidity and control of atmospheric contaminants by absorption in activated charcoal as well as by catalyzed burning.

5.1.2.4 Detail analysis of crew radiation protection requirements, as function of solar activity, mission profile and mission duration.

5.1.2.5 For the computations sub 2.4 a computer program was developed for IBM-7090 determining the mean flare nuclear radiation fluxes, gamma shielding data and cosmic ray fluxes.

5.1.2.6 Detail analysis of fuel conservation systems and meteoroid protection shields. The following fuel conservation systems were considered, either singly or in combination: superinsulation; on-board LH_2 refrigeration and re-liquefaction; shadow shields.

5.1.2.7 Determination of weight scaling coefficients k_f for thrust-dependent weights, for the following nuclear engine-tank configurations:

50,000 lb thrust metal carbide engines on a tank of 20 ft, 33 ft and 50-60 ft dia. respectively

53,000 lb thrust graphite engines on a tank of 33 ft and of 50-60 ft dia., respectively

250,000 lb thrust graphite engines on a tank of 50-60 ft and 65-75 ft dia., respectively

in the following arrangements: single, and clusters of 2, 3 and 4.

5.1.2.8 Determination of weight scaling coefficients k_p for propellant dependent structural weights for the following four basic configurations (Fig. 5-2).

-22 Tank cluster arrangement, consisting of control tanks (surrounded by satellite tanks Thrust is provided by separate graphite core engines for each major maneuver, M-2, M-3 and by chemical engines for M-4 (if any).

-23 Tank cluster arrangement, similar in principal to those of -22, except that the clustered tanks are more nearly of equal diameter. Thrust is provided by one pair of metal core engines, common to the maneuvers M-2, M-3 and M-4 (i.e. all major maneuvers, except Earth departure)

The propulsion module PM-1 for Earth departure (maneuver M-1) consists of a single tank powered by one or more graphite core reactor engines. The overall diameter of the PM-1 tank and the clustered tanks is equal to, or in excess of 50 ft.

-28 Single tank arrangement in tandem for all propulsion modules, including PM-1. Separate graphite engines are used for each principal maneuver. Tank diameter: 50 ft.

-28V Same as -28, but tank diameter is restricted to 33 ft to make it compatible with Saturn V.

The Earth departure module consists of a single tank structure.

5.1.2.9 Determination of the mass fraction coefficients x, derived from the above scaling coefficients.

5.1.2.10 Nomographic Weight Determination. Two methods were developed, of determining rapidly the weight of the individual I/V stages and eventually the orbital departure weight, taking into account weight reductions between the principal maneuvers.. Both methods are based on the mass fractions referred to sub 5.2.2.7; but one method can be expounded to use the scaling coefficients directly.

5.1.2.11 Development of a weight determination program on IBM-7090. The program which uses the weight scaling coefficients referred to above is capable of operating on the basis of impulsive velocity changes as well as finite thrust/weight ratios of any value.

5.1.3 Operations Analysis

5.1.3.1 A reliability matrix system was developed to enable rapid determination of the number of ELV's necessary to support a variety of I/V configurations. The matrix permits immediate assessment of the number of ELV launchings necessary to assemble (a) one vehicle, (b) a convoy of three vehicles in orbit, depending on the following independent variables:

1. Number of matings
2. Number of fuelings (tanker launchings)
3. ELV delivery reliability
4. Mating success probability
5. Fueling success probability
6. Interchangeability, or lack thereof, of modules of a given vehicle and where a convoy of 3 vehicles is involved:
7. Based on 1. - 5.: Modules and vehicles interchangeable
8. Based on 1. - 5.: Modules not interchangeable; vehicles interchangeable
9. Based on 1. - 5.: Neither modules nor vehicles interchangeable.

5.1.3.2 Characteristic ground operations, orbital operations and mission operations models were developed.

5.1.4 Program Analysis

5.1.4.1 A mission planning model was established, formalizing the treatment of the principal schedule-controlling items defined in Fig. 3-1 above.

5.1.4.2 A mission evaluation model was established, taking into account the criteria shown in Fig. 5-3.

5.2 Significant Results

5.2.1 Fast Round-Trip Missions (Single Planet)

Fig. 5-1 shows clearly that for flights between Earth ⇄ Mars and Earth ⇄ Venus the favorable transfer windows are not in harmony. Upon arrival at the respective target planet (Venus, Earth or Mars), the opportunity for a favorable return flight has passed. For mono-elliptic transfers directly to the target planet one has therefore a choice either to depart ahead of one's favorable transfer window or to return after the respective favorable return transfer window. It was found that the latter case is comparatively less disadvantageous from an overall mission and vehicle systems point of view.

5.2.2 Bi-Planet Missions

In the course of further investigations it was found that the transfer window constraint can be relieved significantly by eliminating the requirement of direct transfer and permitting "detours" via a second target planet (bi-planet missions). Thereby a greater number of favorable transfer windows becomes available. Fig. 5-1 shows that shortly after Mars arrival in early 1976 a favorable transfer window opens up to Venus which matches neatly with a favorable window to Earth. Then, in early 1977, there exists an opportunity to transfer from Earth to Venus, from Venus to Mars and then from Mars to Earth. Similar opportunities, some better, some less good are seen in Fig. 5-1 to exist also for other years. It was found that the bi-planet missions offer opportunities to visit, by capture mode, both target planets (bi-planet capture missions) at no greater amount of overall mission velocity than needed for a single-planet round-trip with an unfavorable return flight. On the other hand, bi-planet mission can be flown with PFB near one planet and capture at the other.

5.2.3 Powered Fly-By (PFB)

The investigation of FB missions established that powered fly-by, in contrast to simple FB, broadens the Earth departure window considerably, results, in many cases, in shorter overall mission period and lower velocity requirements, primarily because a greater amount of orbit change can be effected than is possible with the comparatively weak g-fields of these planets. Thereby return orbits become available which have a lower hyperbolic excess velocity at Earth return. The PFB gives the astronauts the practical advantage of modulating by thrust the planetary field which they encounter, in analogy to the throttle and the brake with which the car driver "modulates" the effect of uphill and downhill slopes. The velocity requirements for the powered maneuver at fly-by need not be large (Fig. 5-4). Generally a Δv_{PFB} of 10 to 20% of the hyperbolic excess velocity involved is sufficient.

5.2.4 Mars Capture and Venus PFB Mission

The development of bi-planet missions, of powered fly-by missions; and the effectiveness of a heliocentric orbit change if negotiated within the gravitational field of a planet, led to the combination of these facts to reduce the Earth arrival velocity when returning from Mars in what otherwise would be an unfavorable transfer window. A powered fly-by near Venus on the way from Mars to Earth was found to be very effective in lowering the Earth arrival hyperbolic excess velocity.

5.2.5 Definition of Reference Missions

The above described investigations led to the accumulation of 6 reference missions, shown in Figs. 5-5 and 5-6, except for Mission I which is shown in Fig. 5-5 only. Tab. 5-1 presents their principal characteristics. The missions are special cases in their respective Earth departure windows. Their overall velocities are the result of the particular combination of individual maneuvers. Their variation can alter the overall mission velocities within comparatively wide limits. The principal factors affecting the magnitude of the maneuvers are reviewed in the subsequent 5 paragraphs.

5.2.6 Earth Return Velocity

In missions to Venus, favorable Earth-Venus transfer windows can be used without encountering excessive velocities at Earth return ($.2 \leq v_\infty^* .35$), if capture periods are kept short (20-40 d). In missions to Mars, use of favorable Earth-Mars transfer windows is associated with small perihelion distances and high Earth return velocities, caused by comparatively steep intersection of the return orbit with the Earth's orbit. Fig. 5-7 shows that the unfavorable mission years to Mars (roughly 1975-79) are due primarily to high Earth return velocities. If the technological state of the art permits return into the Earth atmosphere at very high velocity, the difference in the sum of Δv_1 through Δv_3 between favorable and unfavorable mission years is greatly reduced.

5.2.7 Perihelion Braking (PB)

The path intersection angle at return crossing of the Earth orbit can be reduced significantly, and the return velocity lowered correspondingly, by slowing the vehicle down at the perihelion passage. At the small perihelion distances encountered (.45 to .55 AU), perihelion braking by 5000 to 8000 ft/sec causes a reduction in Earth arrival velocity by 16,000 to 24,000 ft/sec. Weight-wise, the effectiveness of PB is reduced by the fact that a heavier payload must be slowed down than near Earth where everything except the EEM is jettisoned.

5.2.8 Atmospheric Braking (AB)

Atmospheric braking was applied at both planets. At Venus, the velocity was reduced to near-circular, followed by a small powered maneuver to establish circular orbit at 1.1 radii distance. At Mars the same mode was applied to establish a circular orbit at 1.3 radii. For purposes of comparison, partial braking was used where the hyperbolic approach velocity is reduced by drag to elliptic speed, followed by powered maneuvers to establish circular orbit at 1.3 radii. The magnitude of potential velocity reductions in the case of Mars can be deduced from Fig. 5-7; it is about 12,000 ft/sec at Venus. Again, the weight saving effect is reduced by the need for carrying a heavy drag brake through the Earth departure maneuver. Operationally and technologically a realistic assessment of the potential weight savings suffers from the uncertainties in the present knowledge of both planetary atmospheres.

5.2.9 Elliptic Capture Orbit (EC)

Capture in an elliptic orbit was found to be effective only at Venus, because the gravity field of Mars is not strong enough for comparable velocity reductions. At Venus, capture in an elliptic orbit of $n = r_A/r_P = 8$ ($r_P^* = 1.1$) results in a velocity reduction by about 7000 ft/sec.

5.2.10 Hyperbolic Rendezvous (HR)

In the HR mode (Fig. 5-8), the interplanetary crew is met by a pick-up vehicle (PUV), launched from Earth orbit to rendezvous with the incoming I/V in its hyperbolic orbit. Originally conceived as an emergency measure, in case the pre-planned return mode should fail, it became apparent that the development of the PUV and the HR mode would be too extensive to be treated as mere back-up effort of limited reliability. Typical velocity requirements for the PUV range from 18.3 km/sec (60,000 ft/sec), if the hyperbolic excess velocity of the incoming I/V is about 0.42 EMOS, to 35 km/sec (115,000 ft/sec), if the hyperbolic excess is about 0.72 EMOS; this at a hyperbolic entry velocity of 15.3 km/sec (50,000 ft/sec). At such velocities the orbital departure weight of the PUV ranges from 750,000 lb to several million lb, in spite of the fact that the initial payload weight of the PUV is only about 1/6 of that of the I/V. In spite of several disadvantages to be discussed in the main report, the use of HR can be justified primarily on the basis that HR is the only way to provide for the returning crew a measure of insurance against involuntary re-escape, should their own capture mode fail.

5.2.11 Variation of Reference Missions

The effect of applying the above modes to the individual maneuvers of the reference missions, Tab. 5-1, leads to the variations of overall mission velocity indicated in Figs. 5-9 and 5-10. The corresponding variation in orbital departure weight of a number of 8-man I/V's with nuclear and chemical propulsion modules is shown in Fig. 5-11. In missions to Venus, elliptic capture is a significant weight saving mode. The effect of atmospheric braking appears to be less significant, because of the large mass of the drag brake. In missions to Mars, the effect of increasing hyperbolic entry speed is significant, in spite of increasing EEM weight, especially up to 60,000 ft/sec. HR yields the lowest weight in this group; but at considerable penalty in PU weight, because the approach velocity is $v_\infty^* = 0.59$ EMOS in this case. The PB, reducing the entry velocity to 50,000 ft/sec, was found to be very effective, in fact, even more so than Venus PFB (Mission IV), in spite of the fact that, at perihelion, the payload was 87,600 lb. PB is seen to be similarly effective as complete AB. A combination of PB and complete AB (II J) yields the lowest weight, short of complete AB and HR. In Mission IV the effect of Venus PFB greatly reduces the orbital departure weight (ODW) compared to Mission II A. The weight can be reduced further by adding complete AB at Mars which, in this case is particularly effective, since the weight of the vehicle is much smaller than under Mission II conditions. Mission VI in 1977 has a similarly beneficial effect as Mission IV in 1975, compared to Mission II A.

5.2.12 Conclusions

On the basis of the results so far it is concluded that Venus missions can be flown as single-planet missions in 400 days round-trip time, without encountering unduly high Earth entry velocities. Elliptic capture reduces the mission energy requirements and, due to the characteristics of radar mapping, interferes less with orbital reconnaissance than with optical reconnaissance at Mars. At later missions it will become possible to use the planetary atmosphere for capture and establishment of a circular orbit. The most attractive return orbits lead through the orbit's aphelion at distances greater than one A. U. (Fig. 5-6).

For Mars, circular capture is found preferable. The preferred outbound transfer orbit is short and lies in a favorable transfer window. Return via mono-elliptic transfer assures shortest mission period and highest degree of freedom in timing, but one must accept small perihelion distances. Lowest orbital departure weight without AB at Mars is attained by HR at the highest velocity for which this mode is developed at the time preceded by PB if the incoming velocity is higher than that attainable by the PUV. If a longer mission period is acceptable the best return flight from Mars is via Venus PFB, significantly reducing the Earth approach speed at little cost in energy. Because of the use of powered maneuvers at fly-by, an adequate degree of freedom in timing the return flight can be maintained (cf. the computer results in a subsequent volume). Once the Mars atmosphere is better known, it may be found useful for aerodynamic braking.

5.2.13 Single and Multiplex I/V Configurations

Conceptual vehicle studies for capture missions, using chemical and solid core nuclear reactor engines, have led to the definition of six vehicle configurations, distinguished by the structural arrangement of their propulsion modules. Four of these belong to single vehicles, defined in Par. 5.1.7.

The residual two classes are multiplex vehicles. The multiplex vehicle concept was developed as a alternative to the single vehicle, traveling in a convoy in which crew vehicles and cargo carrying service vehicles are separate. In the multiplex mode, the individual convoy vehicles are clustered to form one vehicle which can be taken apart . . . if portions are damaged and must be abandoned . . . without necessarily impeding the capability of the remaining system to function as crew vehicle. A typical duplex vehicle design is shown in Fig. 5-12.

The multiplex vehicle, compared to the multi-vehicle convoy, offers the advantages of simplified engine control and flight control; good accessibility to the auxiliary vehicles and other cargo, since they are located in the same vehicle; and it avoids crew module transfer from ship to another in case of an emergency.

The single vehicles must be employed in a convoy of at least two. With the performance-limited vehicles presently under consideration, the largest amount of destination payload weight is obtained by carrying the crew in one vehicle and most of the destination payload in the other, while the heavily protected LSS of the crew vehicle is transferable to the back-up vehicle in case of emergency. The frontispiece depicts a convoy of 2 vehicles consisting of a crew vehicle and a service vehicle (cut-away).

Convoy modes generally have the advantage over multiplex vehicles of lower overall vulnerability in case of catastrophic failures, hence offer high assurance that the back-up vehicle will be available to the crew in an emergency.

The conclusion reached from the study of both modes is that the duplex mode should be investigated in greater detail.

5.2.14 Crew Size and Distribution

Factors affecting the mission crew size are: Vehicle oriented tasks; mission oriented tasks; duration of nominal capture period; size of landing party on the surface of Mars; and overal mission period. Investigations in the course of this study have led to the following results:

Crew size for capture mission (400-450 d) with optional landing capability of 2, using the 2-vehicle convoy mode: 8-10. For the same, but with duplex: 7-9; and without optional landing capability, 6-8 and 5-7, respectively. A crew size of 8 was selected as reference.

In a nuclear convoy, the vehicles must, during powered flight, either be a considerable distance (15-20 km) apart to protect each other's crews from nuclear radiation (assuming the reactors have no significant side shielding), or the crew must be concentrated in one vehicle and the service vehicle must be lined up behind the crew vehicle. All other solutions involve severe weight penalties. These two alternative requirements cause some of the problems in convoy control mentioned sub para. 5.2.13.

An important conclusion of the above mentioned crew distribution analysis is that the LSS as a section must be transferrable from one ship to the other.

5.2.9 Elliptic Capture Orbit (EC)

Capture in an elliptic orbit was found to be effective only at Venus, because the gravity field of Mars is not strong enough for comparable velocity reductions. At Venus, capture in an elliptic orbit of $n = r_A/r_P = 8$ ($r_P^* = 1.1$) results in a velocity reduction by about 7000 ft/sec.

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An important conclusion of the above mentioned crew distribution analysis is that the LSS as a section must be transferrable from one ship to the other.

5.2.15 Interplanetary Vehicles

A variety of vehicle configurations was developed. All principal capture mission configurations to Mars are propelled by nuclear engines. Vehicles to Venus involve nuclear, combinations of chemical and nuclear propulsion modules and and all-chemical vehicles. The design principles which are similar in all cases are shown in Fig. 5-13. The vehicle consists of propulsion section and life support section or service section. In the early manned planetary vehicles, the life support section contains essentially the operational payload and the intransit payload; the service section contains the destination payload.

Hydrogen containers, or combinations of tanks and engines are jettisoned as the tanks are emptied. Each propulsion module is surrounded by a combination heat and meteorite protection shield which is jettisoned just prior to ignition of the particular module. By this means, a high mass fraction is obtained for the operating propulsion module.

The weight analysis of the propulsion structure has reached a level on which the remaining uncertainty tolerance becomes relatively unimportant compared to the effect of potential variations in mission velocity on the ODW. For example, in investigating the importance of jettisoning clustered tanks (Config's -22 & -23) compared to keeping a given PM unchanged during its burning period, it was found that the ODW can be reduced by 2 to 3%, if all satellite tanks are jettisoned at the same time (using I_{sp} = 825 sec engines); and by an additional percent, if they are jettisoned in pairs as they are depleted. The benefits of jettisoning would increase at lower I_{sp}, however.

5.2.16 Vehicle Assembly Modes

The purpose of studying various methods of assemblying I/V's which cannot be transported into orbit in operational condition, was to find means of combining operational and service payload within the convoy mode; to assure compatibility of vehicle weight with engines of limited thrust and operating life; and to reduce the number of ELV's required, as well as the extent of associated orbital operations. Reference system was a 2-vehicle convoy, an 8-man crew payload around 131,000 lb and a service payload of equal weight.

If the S/V is to accompany the C/V back to Earth, OVAM is comparatively the most attractive mode for the following reasons: (a) weight to be transported into orbit is lowest; (b) number of orbital matings of modules is smallest (this is based on Saturn V-type ELV's); the degree of module interchangeability is highest. Interchangeability is the most important factor influencing the rate of increase in the probability of success when redundancies are added (Fig. 5-14).

If the S/V is to remain in the capture orbit (i.e. in the case of return flight of the C/V without back up vehicle), OVAM (2) requires significant less transportation into orbit than OVAM; whereas IVAM (2) (Fig. 5-15) and COVAM (2) (Fig. 5-16) are comparable to OVAM. This is shown in Fig. 5-17 which depicts the launch requirements for assemblying 3 Venus vehicles in orbit with the following weight distribution: OVAM: 3 I/V @ 10^6 lb; OVAM (2): C/V = 10^6 lb; S/V = 488,000 lb; (OVAM (2): CS/V = 843,000 lb; PM-3 Carrier = 775,000 lb; IVAM (2): C/V = 262,000 lb; S/V = 374,000 lb; PM-3 Carrier = 805,000 lb. The unfavorable effect of lack of module interchangeability on the number of launchings required for (OVAM (2) and IVAM (2) is quite apparent. COVAM (2), however, has a number of operational advantages because LSS and service section are combined in one vehicle (the CS/V) during flight to the target planet.

5.2.17 Life Support

Already during the first phase of the study a distinction was made between "dry" and "wet" LSS, the latter using propellant for shielding. Among the dry LSS a radial and horizontal version was developed. In comparing integrated with modularized structures, the latter were chosen, in spite of a slight weight penalty because in the performance-limited vehicles considered here, the option to jettison non-vital LSS-modules in an emergency represents a valuable performance reserve for the crew. The final versions of the radial and horizontal systems are shown in Fig. 5-18. Both versions have minimum-size polyethylene borate radiation shelters, for reasons of weight from which the command module can be operated during the solar storm. The horizontal version appears superior for the following reasons: Problem of canal sickness of the crew can be minimized; feeling of confinement is less pronounced than in radial version; artificial gravity is constant, because of single-floor arrangement; piping and plumbing is simplified and lighter. The gross volume available to the crew is 800 ft^3/person at a net volume of 550 ft^3/person; slightly larger than the volume provided in a nuclear submarine.

Among the "wet" versions, a life support section partly submerged in the PM-4 LH_2 tank was investigated during the first study phase. Additional propellant components were compared with polyethylene for the dry LSS. The results are summarized in Tab. 5-2 for a shelter with internal dimensions: 8.5 ft dia., 5 ft height. Using 50% of the MMH stored in shelter walls yields a propellant weight of 22,000 lb. With a terminal payload (EEM) of 9400 lb for Apollo-type entry, this propellant provides 12,000 ft/sec for terminal braking, reducing PB to about 4000 ft/sec and resulting in further lowering of the ODW.

Tab. 5-2 SHIELDING EFFECTIVENESS OF PROPELLANT COMPONENTS

	Wall Thickness (ft)	Wt. of Material (lb)	Wt. of Propellant (lb)
Polyethylene	.635	11,450	—
Monomethyl Hydrazine	.725	11,920	41,700(ox.=OF_2)
RP-1	.79	11,980	57,000 " "
CH_4	1.66	14,840	89,000 " "
OF_2	.533	18,000	23,150(w. MMH)

5.2.18 Earth Launch Vehicle (ELV)

Two ELV's were considered originally: Saturn-V and Post-Saturn (10^6 lb, no volume restrictions). It soon became apparent that with hydrogen-carrying I/V modules as orbital freight, the principal constraint of Saturn V was the volume of its payload section, rather than its payload weight. Thus, a hypothetical Saturn V M of 50 ft dia. with unchanged payload weight was added (Fig. 5-19). A comparison of the three ELV's is shown in Fig. 5-20 on the example of establishing a convoy of 3 identical Mars vehicles @ 2 · 10^6 lb apiece in orbit. The study showed that enlargement of the diameter of Saturn V to about 50 ft would be very worthwhile, if it is to be used as ELV for LH_2-carrying interplanetary vehicles.

5.2.19 Ground Handling and Launch Operations

The success of the manned interplanetary mission will be the result of a well coordinated effort to completely prepare the I/V for its mission. Although the launch site will not be the final departure point for I/V, it is here that the tasks needed to assure the vehicle's readiness can be accomplished most easily. Because of the I/V's complexity and for reasons of basic economy, as many operations as possible must be performed on the ground, a minimum in orbit. Therefore, it was felt necessary in this study to be concerned with ground operations and launch facilities required to support the preparation of an interplanetary expedition. The impact and the compatibility of I/V requirements on facilities and on operations primarily designed to support the ELV were studied.

A launch facilities requirements matrix for Saturn V, a facility operational schedule and a ground operations schematic were established.

5.2.20 Orbital Operations

Orbital operations begin following cargo delivery which nominally is completed with the attainment of rendezvous in the immediate vicinity of the orbital launch preparation complex. Orbital operations involve primarily mating and fueling of space vehicle modules, orbital inspection, testing and checkout operations culminating in a comprehensive mission readiness test immediately preceding orbital departure. Each of the major orbital operations was investigated and an orbital operations model developed (Fig. 5-21).

5.2.21 Mission Operations

The investigation of mission operations plays an important part in the development of a mission risk analysis, failure analysis and emergency analysis. A mission operations model was established for several different missions, showing primary operations and the sequence in which they are performed. A mission operations model for a fast round-trip mission to Mars (CC/SE) was combined with the operations model and is presented in Fig. 5-21.

5.2.22 Ground and Flight Test Integration

A model for integrating ground and flight (orbital) testing in preparation of a planetary mission was established. Primary emphasis was given to economic as well as engineering aspects, showing how such a test program can be planned to simultaneously benefit the national space program in general while benefitting to a maximum degree from events in the national space program in economizing the development of the I/V. In this respect, three focal areas were found which deserve considerable consideration by space planners in the near future: (a) Utilization of the Apollo Program and of the capabilities generated by it for orbital testing, cislunar flight testing and partly for direct application to the planetary mission; (b) Saturn V growth as ELV for initial manned planetary missions; (c) space station development and development of a life support section (LSS) for a lunar base which initially, in effect, is nothing but a "stationary interplanetary vehicle", as far as operating life and most of its environmental conditions are concerned. The development of amulti-purpose largely standardized LSS for long-operating life space stations, lunar base and interplanetary vehicle appears highly worthwhile from the standpoint of economy, time and operating reliability, compared to separate LSS for orbital launch facilities, space stations, lunar base and I/V. One attractive way to translate this concept into practice is to use the LSS of an I/V and use it in combination with others for establishing a space station, an orbital launch facility and a test bed for the interplanetary development team. One of several orbital systems concepts developed during the study is shown on the inside of the back-cover. The system is Saturn V-compatible. For more discussion cf. the subsequent volumes.

5.2.23 Availability Schedule and Schedule Confidence

Based on the analysis of the principal schedule controlling items (cf. Fig. 3-1), an availability schedule was established, showing the expected data of availability (Fig. 5-22; black triangle; and Fig. 5-23*) and the estimated earliest and latest availability. On this basis a schedule confidence model was established, showing the probability of successfully meeting a planned schedule for a particular mission as function of time (Fig. 5-24*).

5.2.24 Cost Analysis

Cost data were divided into 4 categories. The direct development cost includes design and testing from the component level up to the complete vehicle; the cost of establishing and maintaining test facilities and of the test operations; the cost of the launch vehicles for the flight tests and of the flight test operations; the cost of ELV modifications or of a new Post-Saturn ELV. The indirect development cost includes supporting scientific research and the direct operating costs of space probes carried out in support of the particular manned program. The direct operating cost include procurement of launch vehicles, launch cost, procurement of space vehicles and spares, all based on given reliabilities. The indirect operating cost includes establishment of additional launch pads and the cost of orbital labor as well as in ground support. A cost survey is presented in Fig. 5-25.

5.2.25 Mission Evaluation

A matrix has been developed for the purpose of evaluating planetary missions as consistently and objectively as possible. The model is discussed in detail in the subsequent volumes. One of the aspects of mission evaluation, the approximate gross amount of binary bits of information per dollar acquired during a given mission is compared for 7 missions in Fig. 5-26.

6. STUDY LIMITATIONS

A principal limitation of the study has been one of time to study in greater depth some of the great number of parameters, variables and trade-offs involved in obtaining a true optimization from a programmatic point of view, both as to its integration into the national space program as to the mission in its own right. It is certainly only a very first step, for example, to minimize the orbital departure weight when uncertainties in the mission planning (e.g. reliabilities, deployment of HR etc.) imply uncertainties in total weight that may have to be transported into orbit which are many times the weight variation in a departure window around the minimum weight. An interesting example for the need for more careful study of important details was the finding, gained from computer data, that minimum orbital departure weight does not necessarily coincide with minimum transportation cost, if the effects of reliabilities and in interchangeability of modules are taken into account. On the other hand, the very fact that this limitation could be brought into sharp focus, pointing at the need for studies in considerably greater depth is felt to be an important and valuable result.

A secondary limitation has been the limited effort which could be devoted to the aspects of planetary exploration by the crew during the capture period. Both, the destination payload weight (auxiliary vehicles) and the required crew size are affected.

The extent to which these limitations made themselves felt, was related to the complexity of the study subject, which grew as more was learned about it. To the broad scope of this study and the knowledgeable management by the Future Projects Office of NASA/MSFC goes the credit for keeping limitations to a minimum.

7. IMPLICATIONS FOR RESEARCH

Reference is made to Fig. 3-1 in which areas of importance to advanced research and technology are especially marked; and to Fig. 3-2 which shows key research requirements in preparation of manned planetary missions. To these should be added:

Research on long duration ecological systems; search for combinations of inorganic and organic systems which may be superior to either pure system.

Extensive and systematic testing of human crews under controlled low-gravity and zero-gravity conditions for extended time periods. Behavioral Research on space crews over long periods of time to establish in predictable form the degree to which crew members will retain their proficiency and reliability.

The most critical areas of research for missions in the 1975/79 time period are those which potentially require the longest lead time but without which adequate mission planning, preparation and execution is not possible:

1. Nuclear propulsion systems.

* For explanation of abbreviations see nomenclature in back of report.

2. Hyperbolic entry into the Earth atmosphere at 50 to $60 \cdot 10^3$ ft/sec.

3. Exploration by instrumented probes of the atmosphere of the target planet and of the meteoritic density in Earth-Mars space as well as near Venus and Mars.

8. SUGGESTED ADDITIONAL EFFORT

The following subjects are suggested as now deserving increased attention:

(1) Bi-Planet Missions: Systematic evaluation of Mars ⇌ Venus transfer windows as related Earth ⇌ Mars and Earth ⇌ Venus transfer windows involving capture on both planets, or capture on one, PFB on the other, or PFB on both planets (1975-1985).

(2) Powered Fly-By Missions: Systematic search for PFB mission windows to Mars and Venus (1975-1979).

(3) Perihelion Braking: Determination of minimum energy flight paths involving PB.

(4) Hyperbolic Rendezvous: Continued investigation and evaluation of the HR mode; including the establishment of practical limitations on the maximum hyperbolic excess with which the PUV is capable to cope if powered by solid core reactor engines.

(5) Multiplex Vehicles: Extension of structural analysis and determination of scaling coefficients with particular emphasis on duplex vehicles.

(6) Multiplex Vehicles: Continuation of the comparison of convoy mode versus multiplex mode.

(7) Planetary Exploration: Crew activities, auxiliary vehicles and other techniques for planetary exploration during the capture period, including the following cases:
   (a) for a methodology of Mars and Venus reconnaissance from orbit only
   (b) for a methodology of Martian reconnaissance from orbit and at surface during a breif surface excursion
   (c) for a methodology of establishment and operation of a synodic base on Mars.

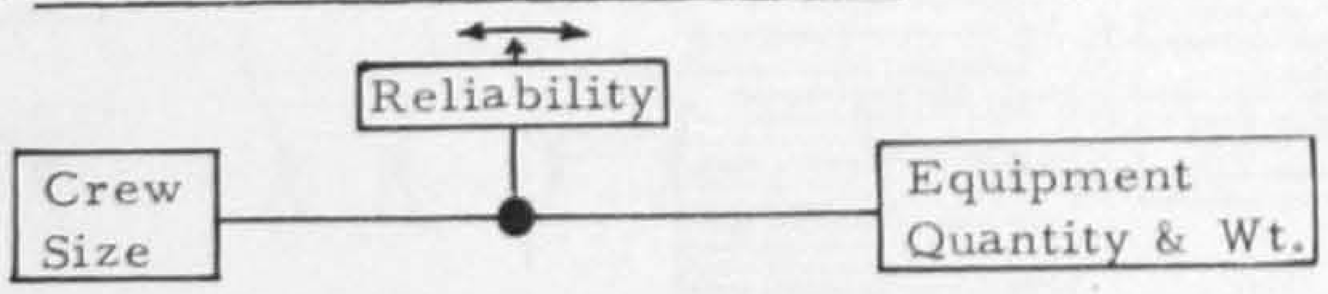
(8) Study of diagnostic requrements pertaining to the I/V:
   (a) methods of damage detection
   (b) methods of determining precise location, type and extent of damage
   (c) methods of damage repair

(9) Continuation of the analysis of OVAM(2) and COVAM(2).

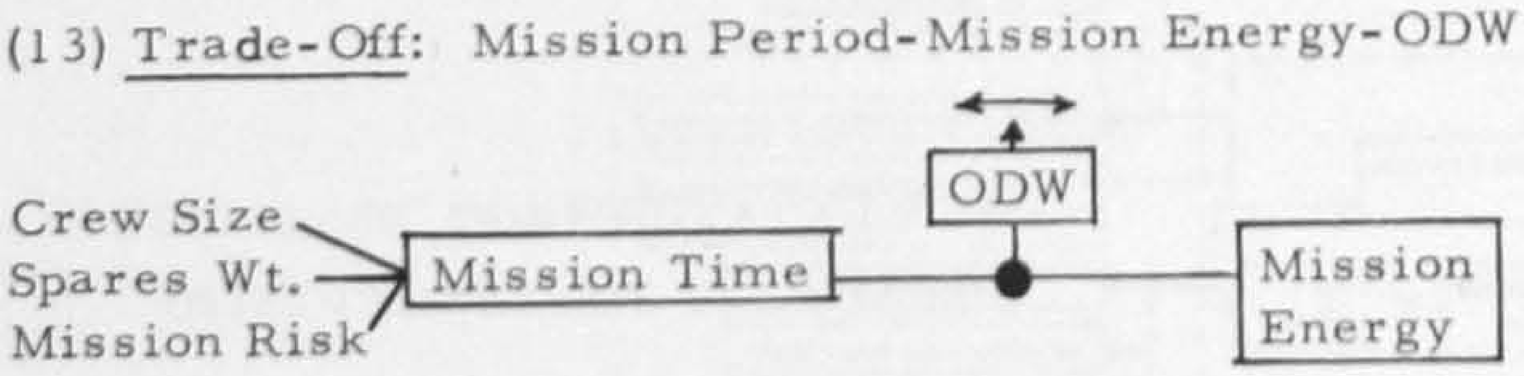
(10) Development of a failure probability model and determination of spares.

(11) Crew Sizing analysis, based on items (8), (9), (10) and (11) above.

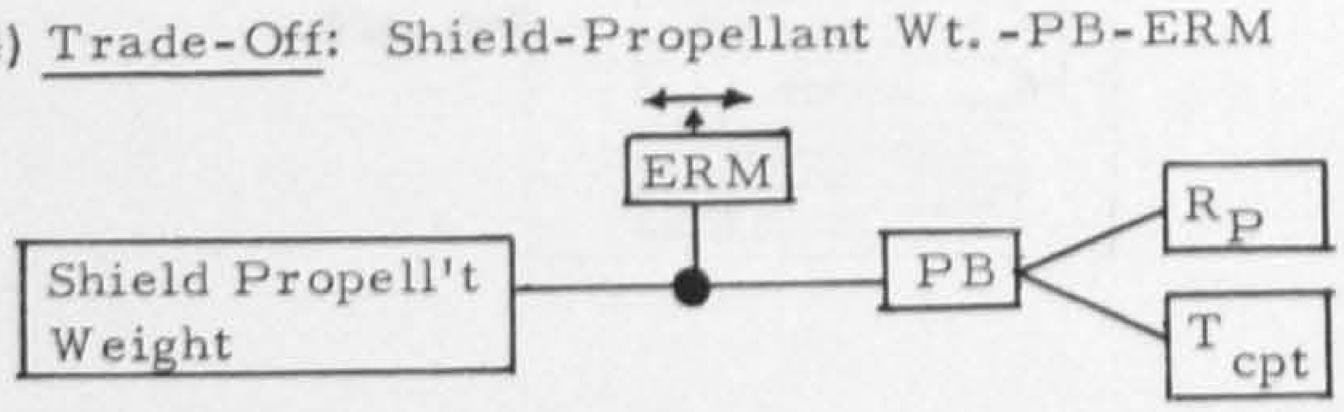
(12) Crew vs. Equipment Trade-Off



(13) Trade-Off: Mission Period-Mission Energy-ODW

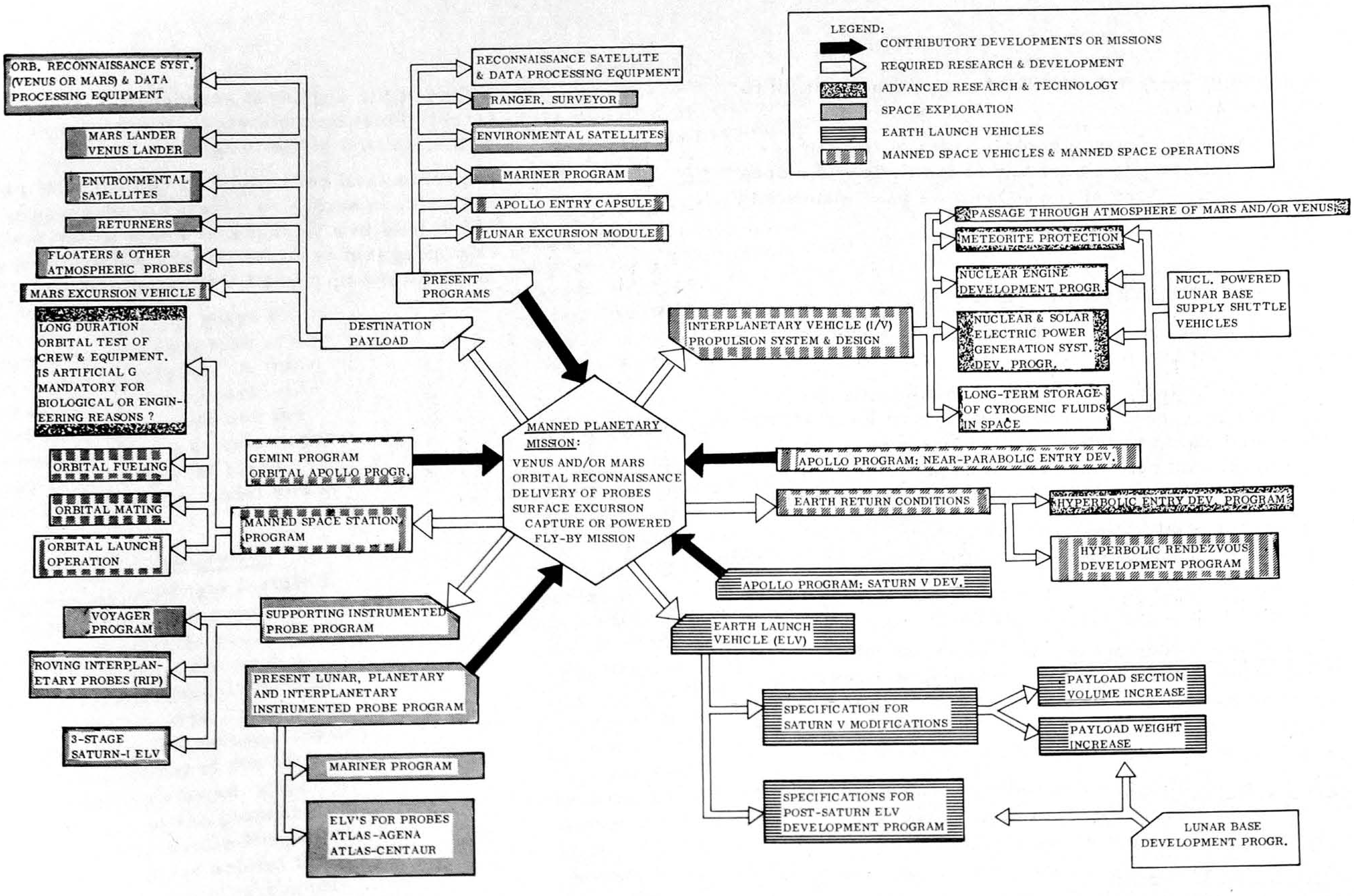


(14) Trade-Off: Shield-Propellant Wt.-PB-ERM



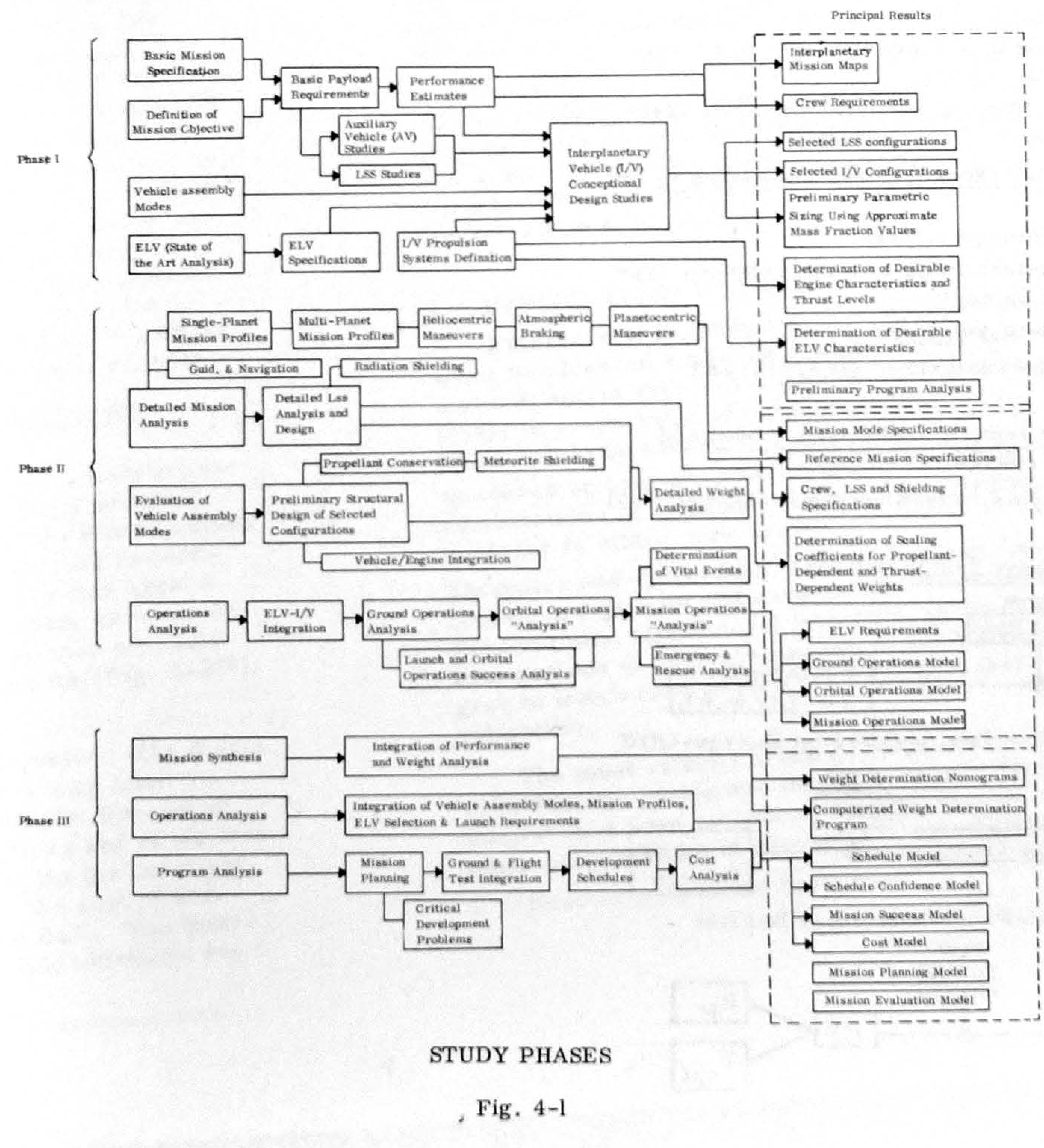
(15) Effect of the degree of acrodynamic braking at the target planet on the structural design and weight of the I/V and of the drag body.

(16) Expansion and refinement of comparative program analysis, considering several different programs which lead to a first manned expedition; and analyzing each program as to its interrelation with space data, lunar follow-up and ELV development requirements.



RELATIONSHIP OF MANNED PLANETARY ROUND-TRIP MISSIONS TO OTHER NASA EFFORTS

Fig. 3-1



STUDY PHASES

Fig. 4-1

SPECIFICATIONS OF I/V PROPULSION SYSTEMS USED IN STUDY

Type	Base	Thrust (10^3 lb)	Spec. Imp. (sec)	No. of Restarts	Theoret. Operating Life (hrs)	Mission Application
Chemical	O_2/H_2	75	450	N/R 1)	N/R	Ve (C)
NERVA	Classif.	→	→	→	→	Ve (PFB)
Nuclear	Graphite	63	765	2	1	Ma (PFB) PB
Nuclear	"	250	825	2	1	Mars
Nuclear	"	700	845	2	1	Mars
Nuclear	Metal	50	900	N/R	N/R	Mars

1) N/R = Not Restricted

Tab. 4-1

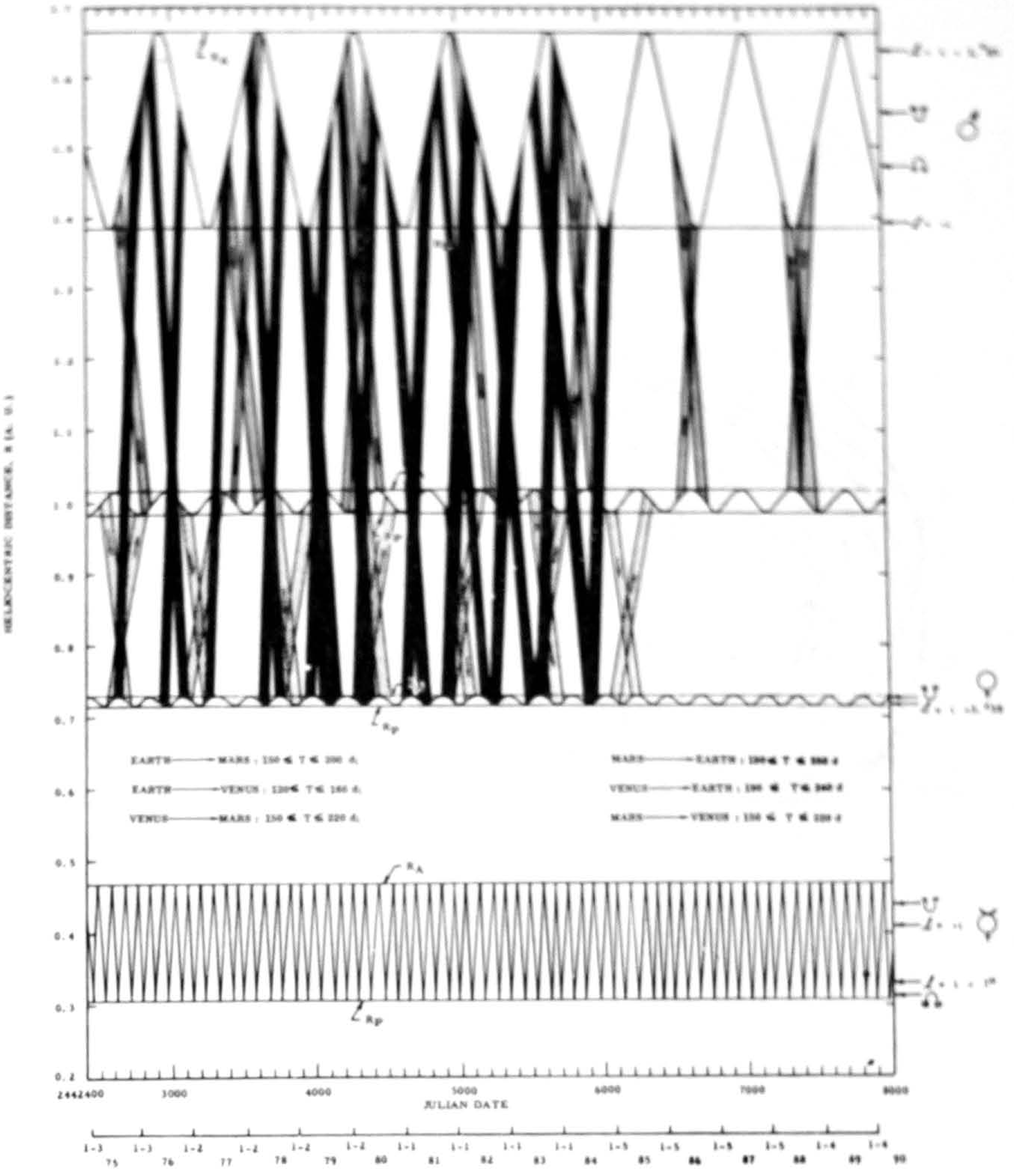
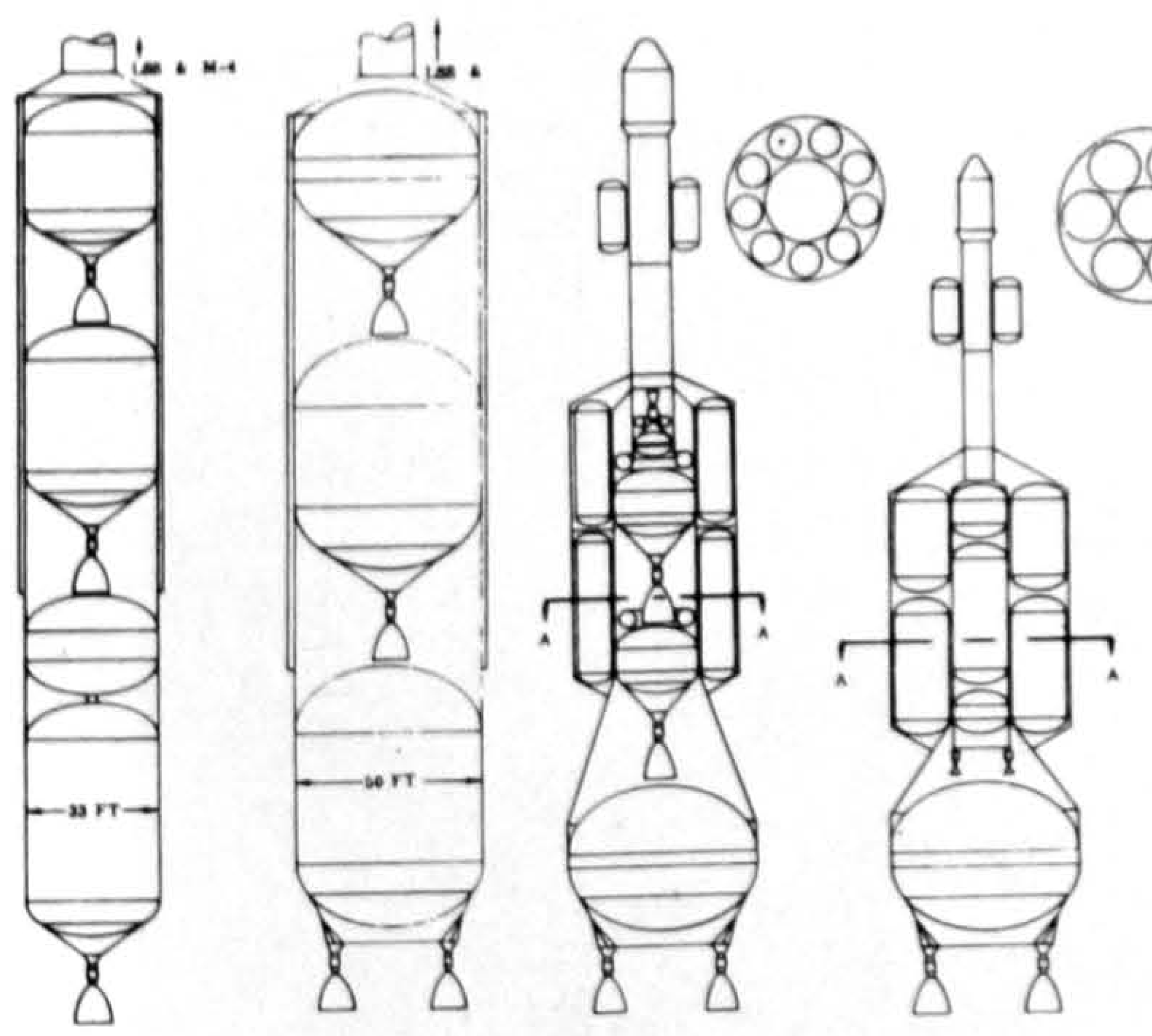


Fig. 5-1



FOUR BASIC CONFIGURATIONS FOR INTERPLANETARY VEHICLES

Fig. 5-2

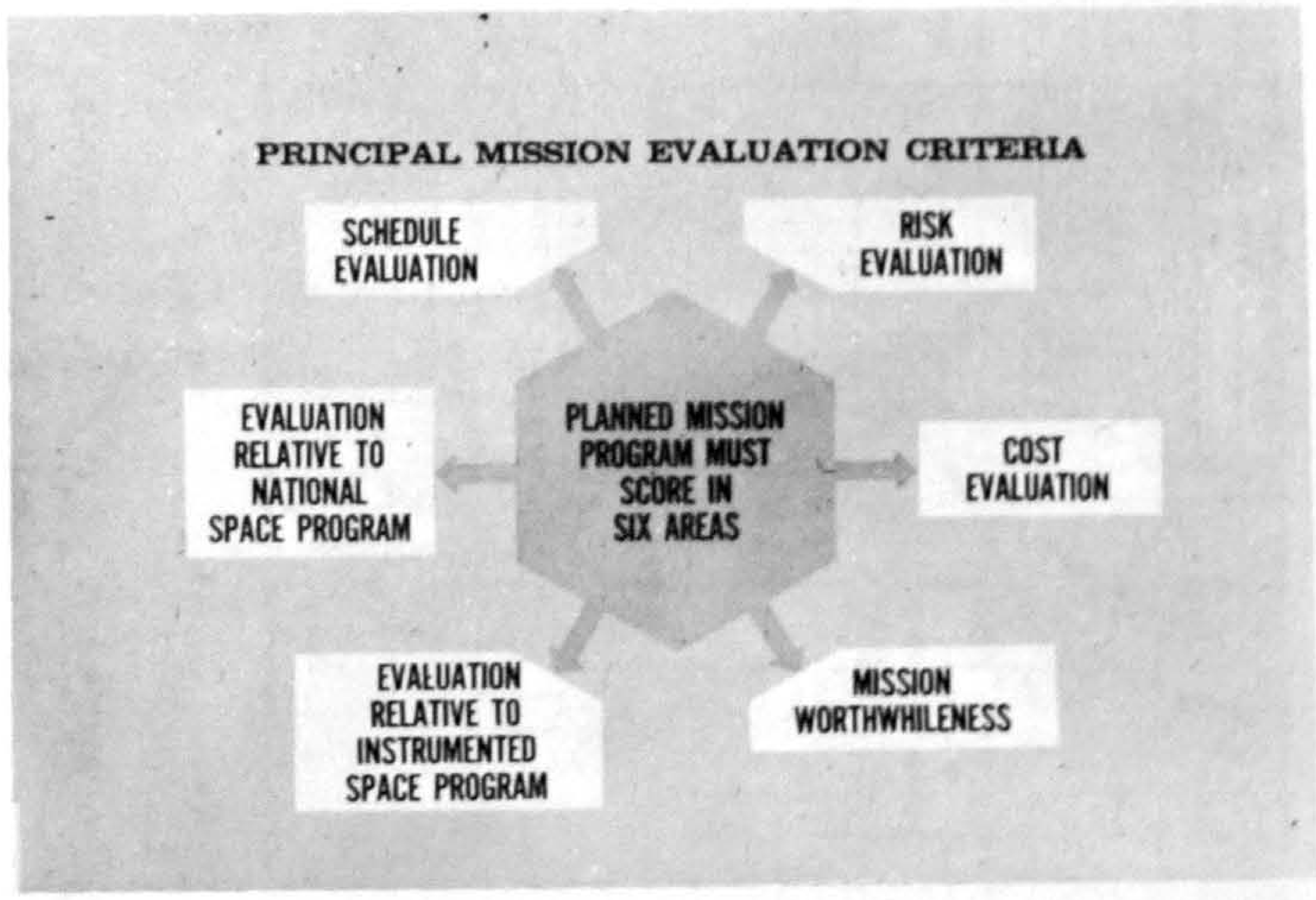
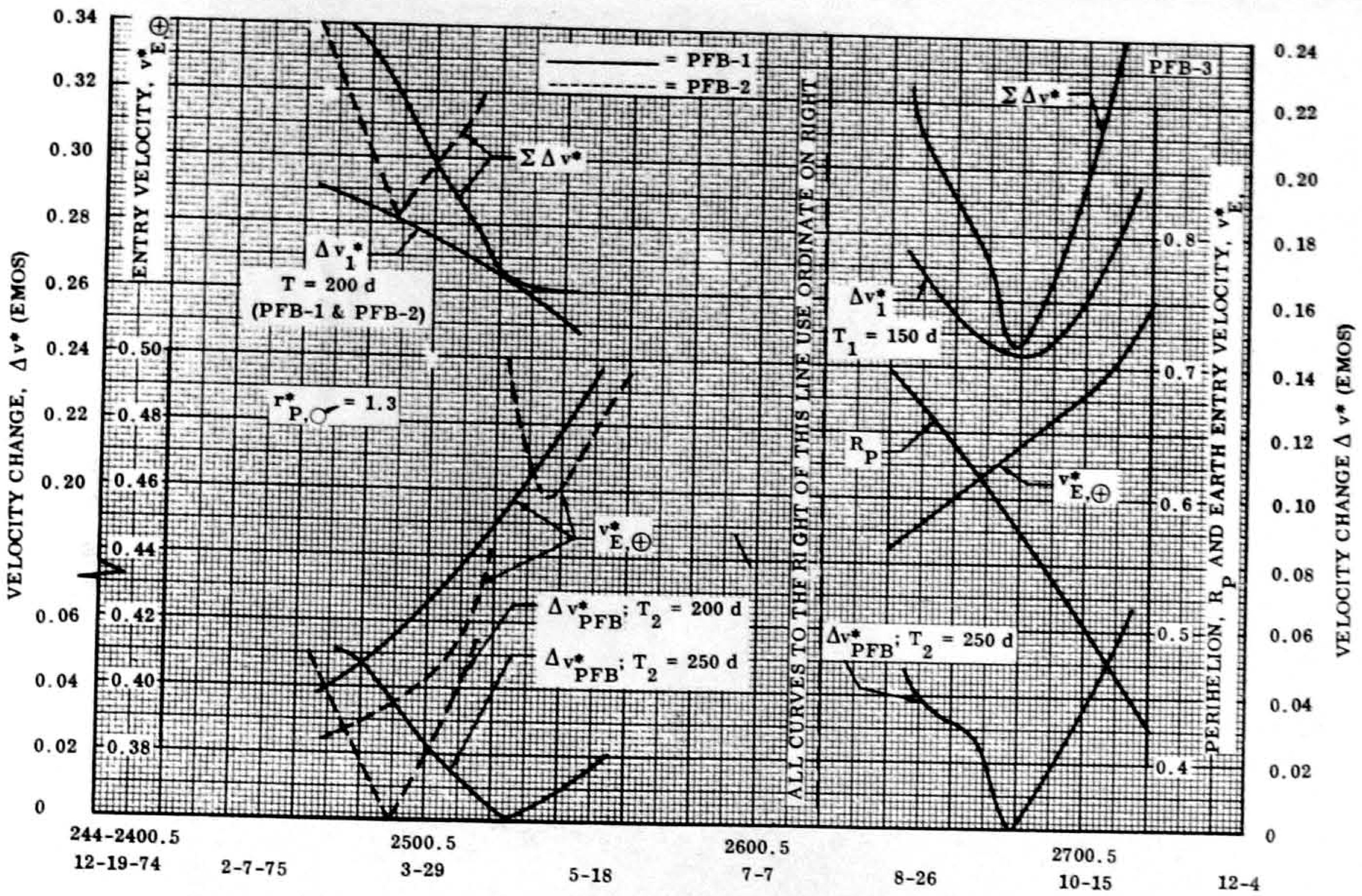


Fig. 5-3

Fig. 5-4



THREE POWERED FLY - BY MISSIONS TO MARS IN 1975: PFB-1, PFB-2 AND PFB-3

REFERENCE MISSIONS I THROUGH V

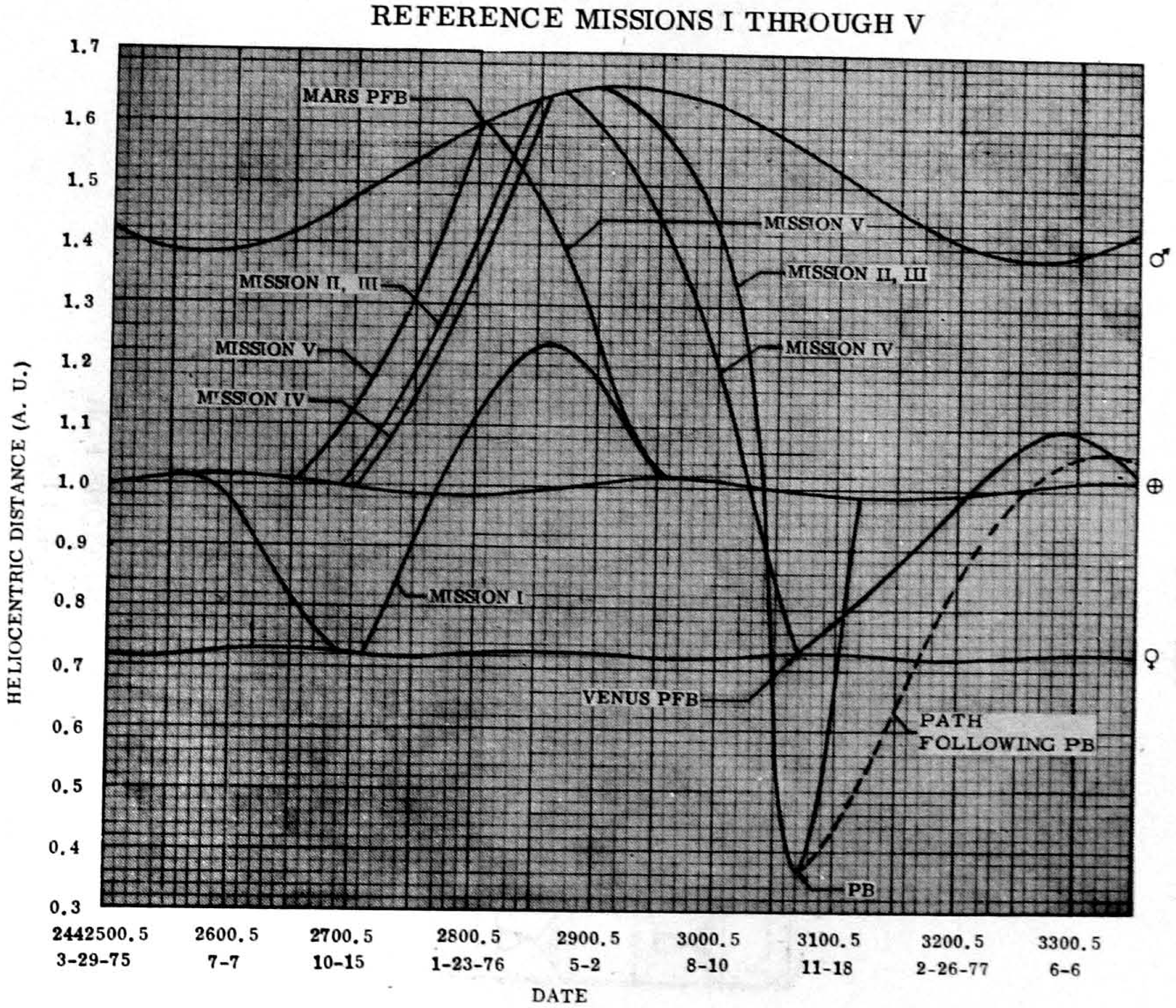
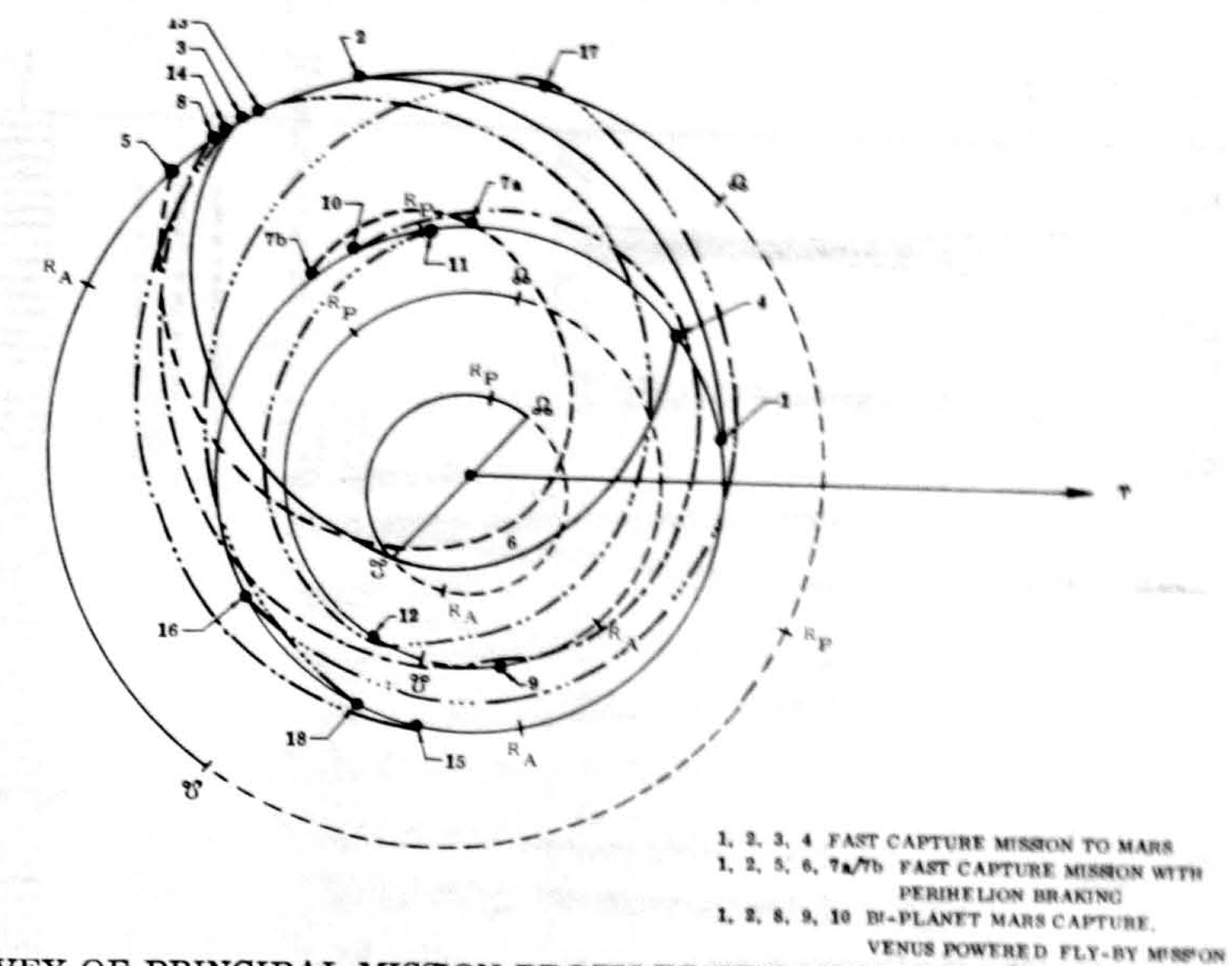


Fig. 5-6

Fig. 5-5



SURVEY OF PRINCIPAL MISSION PROFILES INVOLVING VENUS POWERED FLY-BY, MARS CAPTURE AND MARS POWERED FLY-BY

SURVEY OF FAVORABLE TRANSFER WINDOWS BETWEEN EARTH AND MARS, BETWEEN EARTH AND VENUS AND BETWEEN VENUS AND MARS, 1975 THROUGH 1990

Mission	I	II	III	IV	V	VI
Target Planet	Venus	Mars	Mars	Ma/Ve	Mars	Ve/Ma
Dep. Earth	5-20-75	9-5-75	→	10-15-75	8-31-75	1-27-77
Transfer Period, T_p (d)	140	160	→	160	150	150
Planet Mode	EC	CC	CC/SE	CC	PFB	PFB
Capture Period, T_{cpt} (d)	20	30	→	20	0	0
Departure Window, (d)	0	20	→	0	-	-
Transfer Period, T_2 (d)	-	-	→	200	-	200
Planet Mode	-	-	-	PFB	-	CC
Capture Period, T_{cpt} (d)	-	-	-	0	0	0
Departure Window, (d)	-	-	-	-	-	0
Transfer Period to Earth, (d)	240	220		200	250	220
Mission Period, T (d)	400	440	→	590	400	590
Earth Dep. Δv_1^* (EMOS)	.132	.153	→	.164	.166	.1605
Target Pl. Arr. Δv_2^* (EMOS)	.0501	.1735	→	.1163	-	-
PFB, Δv^*_{PFB} (EMOS)	-	-	-	-	.0359	.0171
Target Pl. Dep., Δv_3^* (EMOS)	.1922	.200	→	.1725	-	-
Target Pl. Ar., Δv_4^* (EMOS)	-	-	-	-	-	.106
PFB, Δv^*_{PFB} (EMOS)	-	-	-	.00505	-	
Target Pl. Dep., Δv_5^* (EMOS)						.169
Unbroken Earth Entry Vel., $v^*_{E,\oplus}$ (EMOS)	.47	.74	→	.415	.592	.547
Earth Ar. Maneuver at Entry Vel. limited to $v^*_{E,\oplus} = .512$, Δv_6^* (EMOS)	0	.236	→	0	.0825	.036
Mission Velocity $\Sigma\Delta v^*$ (EMOS)	.3743	.7625	→	.4578	.2844	.3886
(ft/sec)	36,600	74,500	→	44,600	27,800	38,000
(km/sec)	11.2	22.7	→	13.6	8.47	11.6

Table 5-1

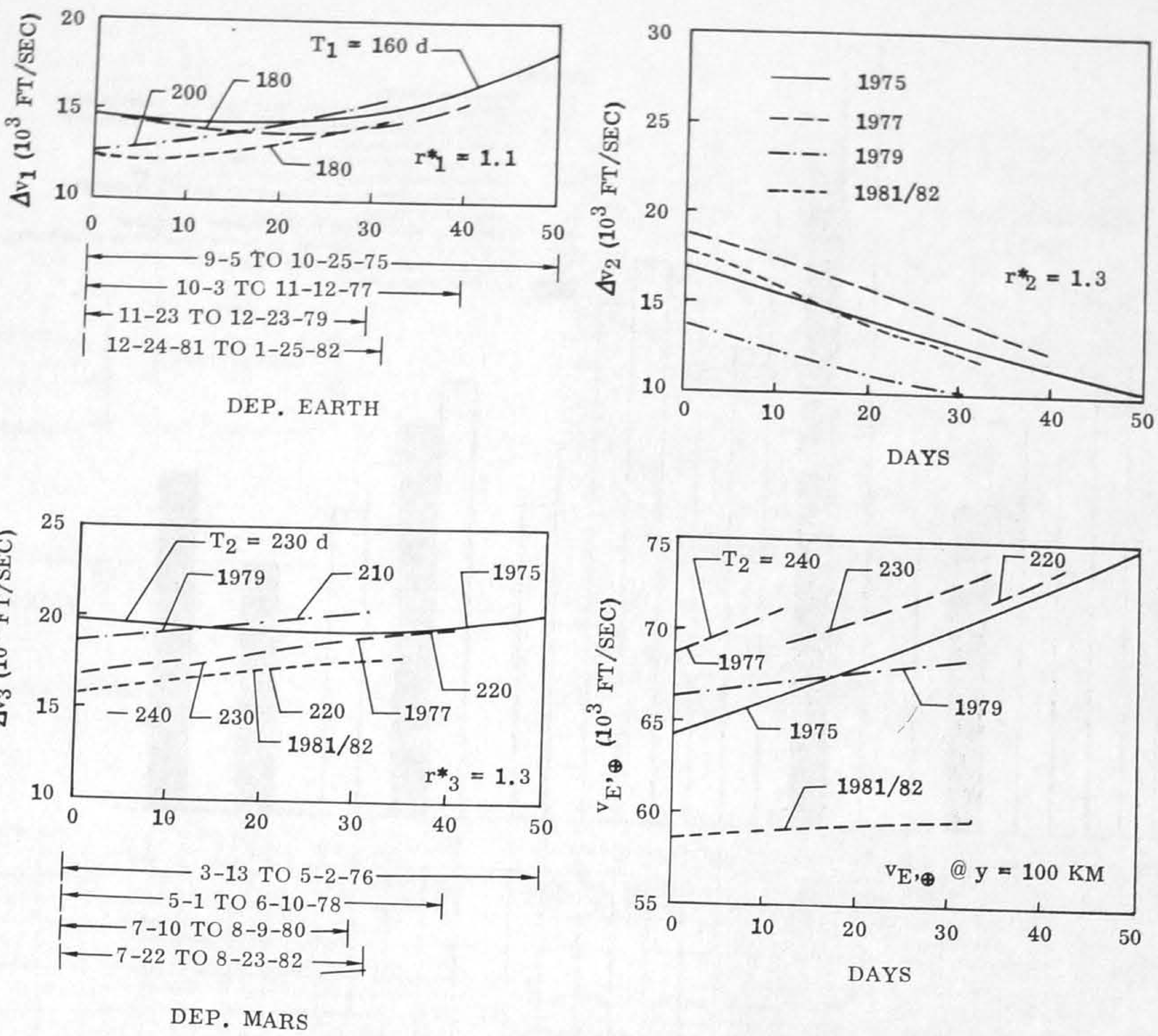


Fig. 5-7

VARIATION OF MANEUVERS Δv_1, Δv_2 and Δv_3 AND OF EARTH ENTRY VELOCITIES $v_{E,\oplus}$ FOR FOUR MARS MISSION YEARS, 1975-1982 WITH EARTH-MARS TRANSFER DURING FAVORABLE WINDOW

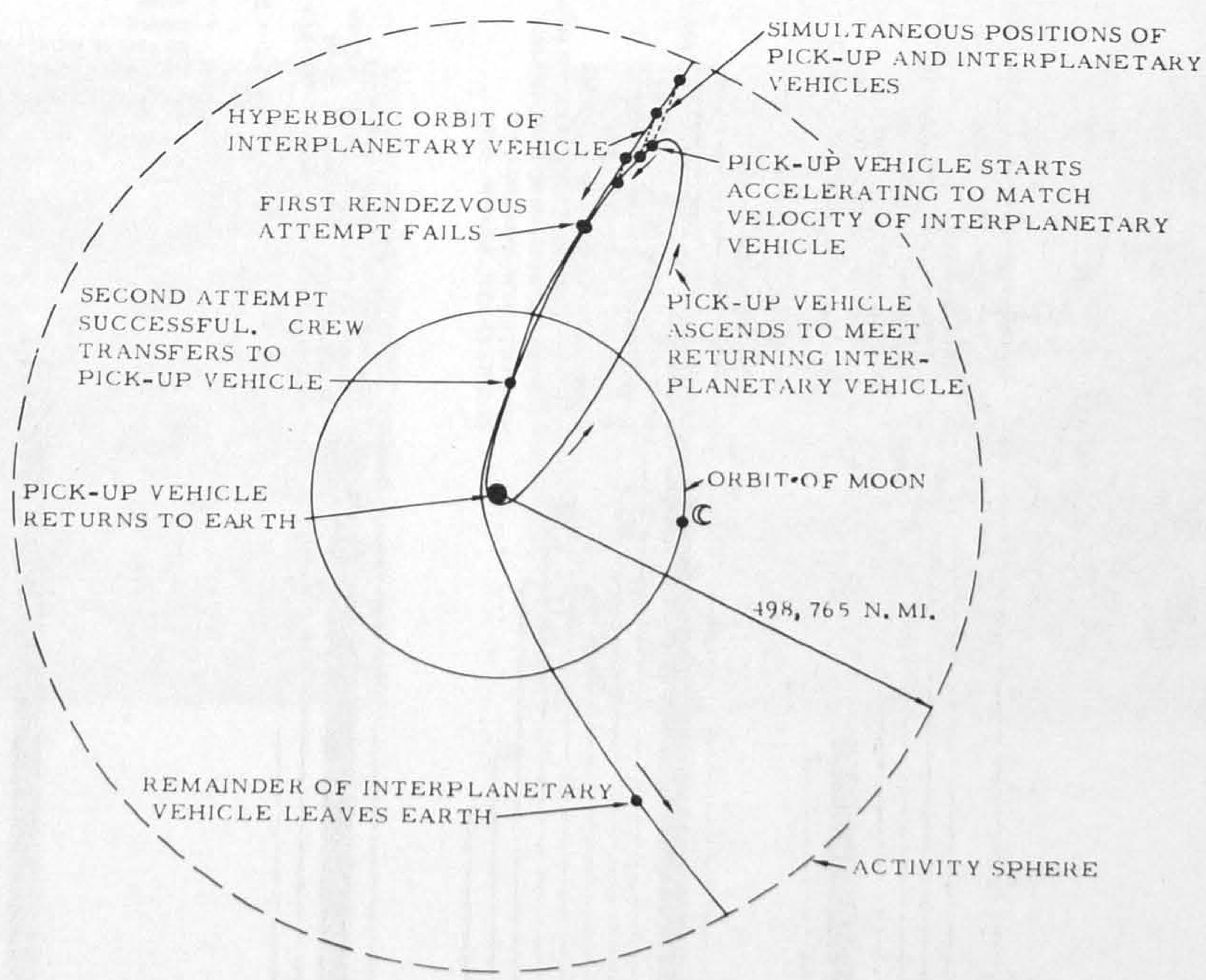
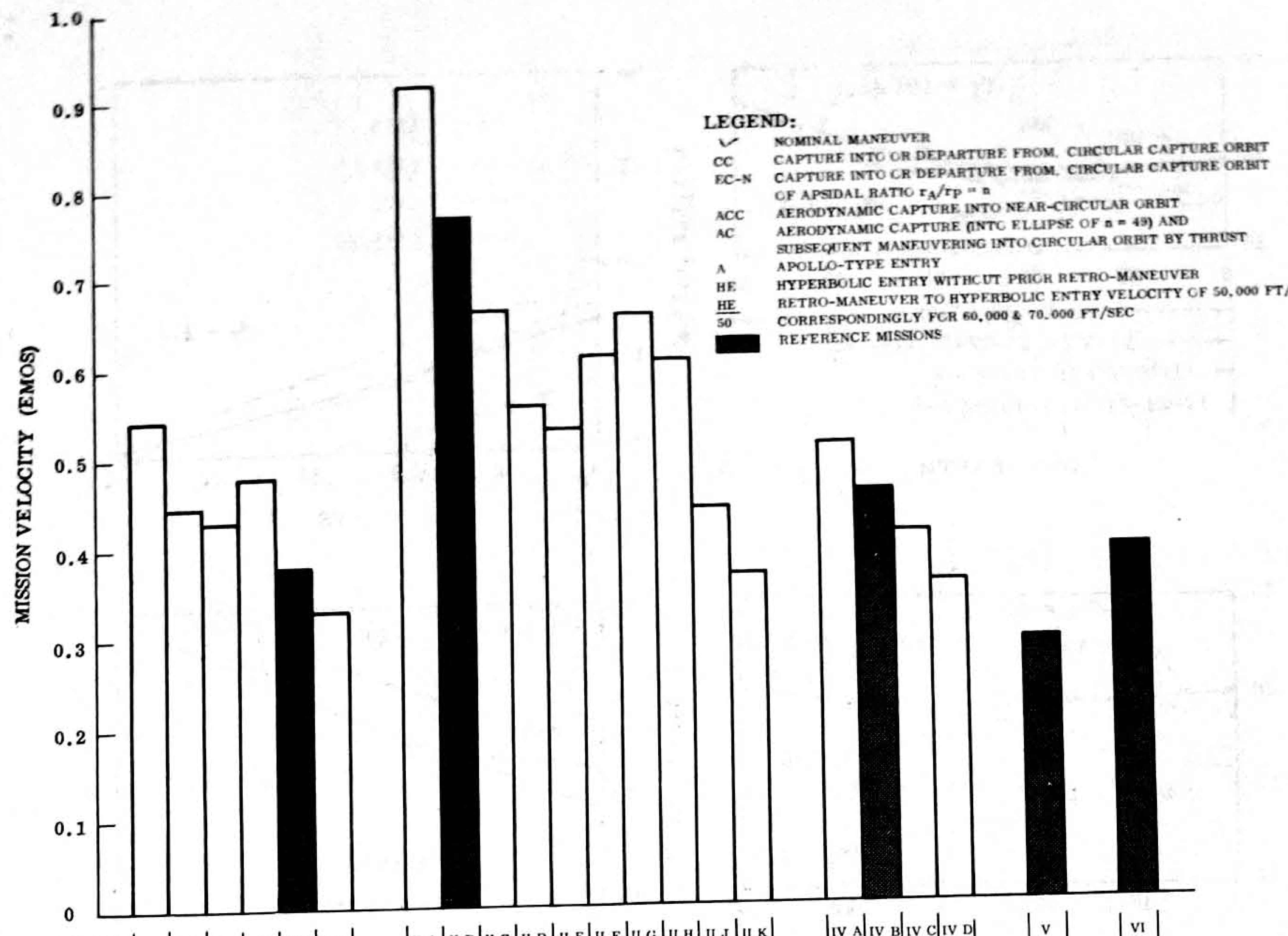


Fig. 5-8

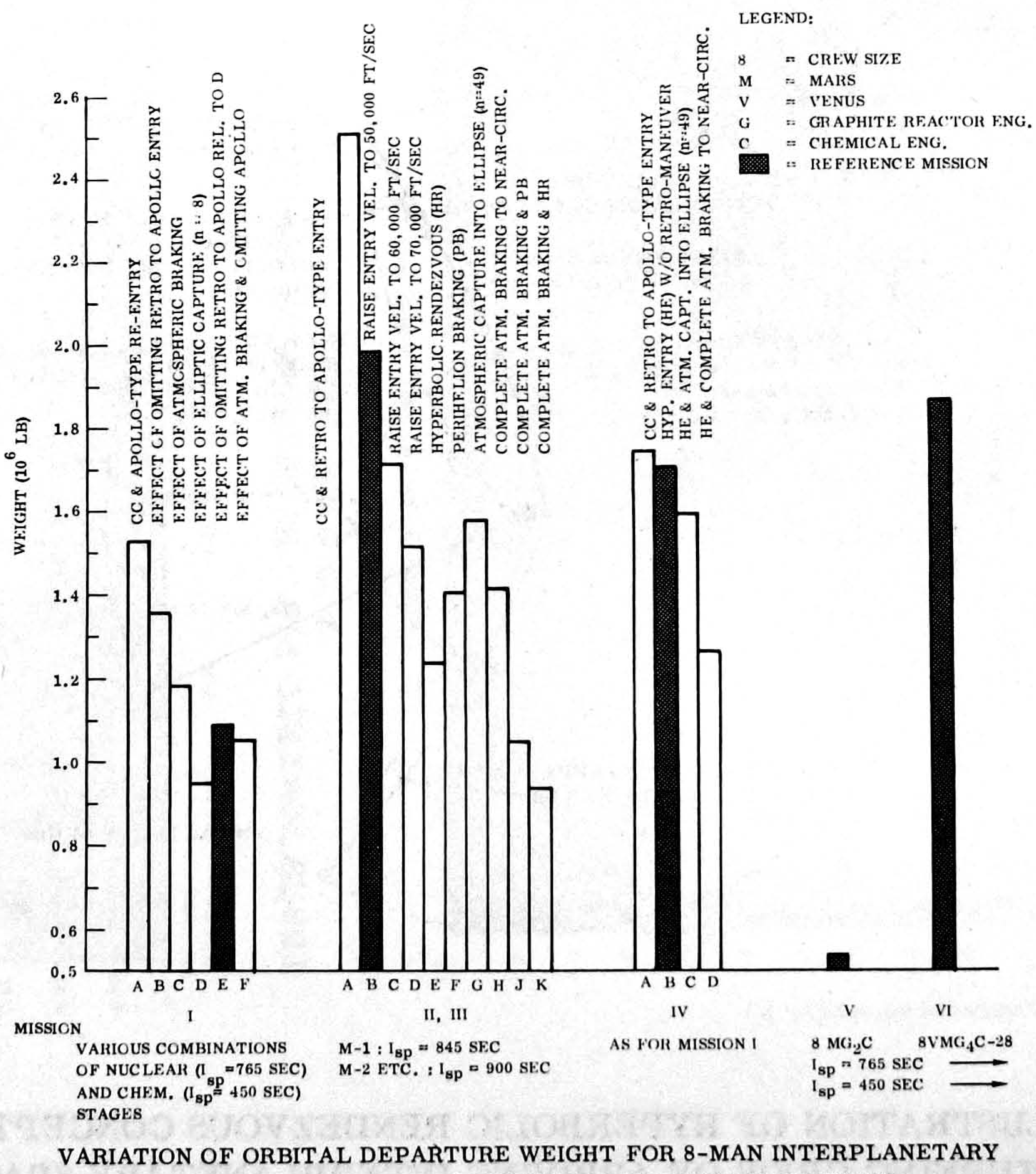
ILLUSTRATION OF HYPERBOLIC RENDEZVOUS CONCEPT FOR RETURNING CREW OF ARRIVING INTERPLANETARY SPACE VEHICLE TO EARTH



MISSION	I A	I B	I C	I D	I E	I F	II A	II B	II C	II D	II E	II F	II G	II H	II J	II K	IV A	IV B	IV C	IV D	V	VI
M-4	A	HE	A	A	HE	HE	A	HE 50	HE 60	HE 70	HR	HE 50	HE 50	HE 50	HE 50	HE 50	HE 50	HE 50	HE 50	HE 50	HE 50	HE 50
PB	-	-	-	-	-	-	-	-	-	-	-	✓	-	-	✓	✓	-	-	-	-		-
PFB	-	-	-	-	-	-	-	-	-	-	-	-	-	-	-	-	✓	✓	✓	✓	-	-
M-3	CC	CC	CC	EC	EC	CC	CC	CC	CC	CC	CC	CC	CC	CC	CC	CC	CC	CC	CC	CC	-	CC
M-2	CC	CC	ACC	EC-8	EC-8	ACC	CC	CC	CC	CC	CC	CC	AC	ACC	ACC	ACC	CC	CC	AC	ACC	-	CC
PFB	-	-	-	-	-	-	-	-	-	-	-	-	-	-	-	-	-	-	-	-	✓	✓
M-1	✓	✓	✓	✓	✓	✓	✓	✓	✓	✓	✓	✓	✓	✓	✓	✓	✓	✓	✓	✓	✓	✓

VARIATION OF MISSION VELOCITY I, II AND IV FOR DIFFERENT SETS OF MANEUVERS

Fig. 5-9



VARIATION OF ORBITAL DEPARTURE WEIGHT FOR 8-MAN INTERPLANETARY VEHICLES AS FUNCTION OF MISSION VELOCITY VARIATIONS

Fig. 5-11

Fig. 5-10

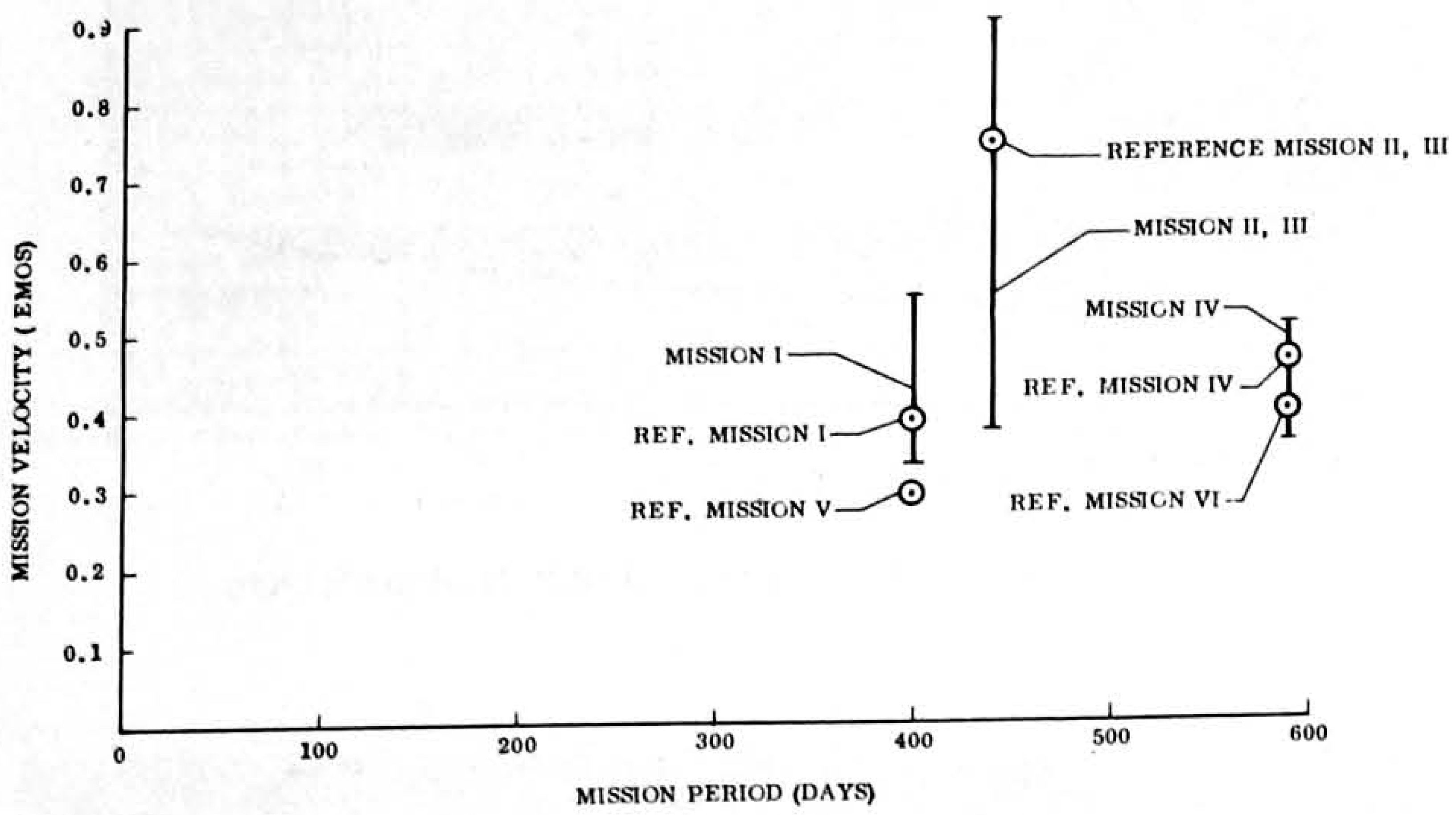


Fig. 5-10 MISSION VELOCITIES AND MISSION PERIOD FOR MISSIONS I THROUGH VI

Fig. 5-12

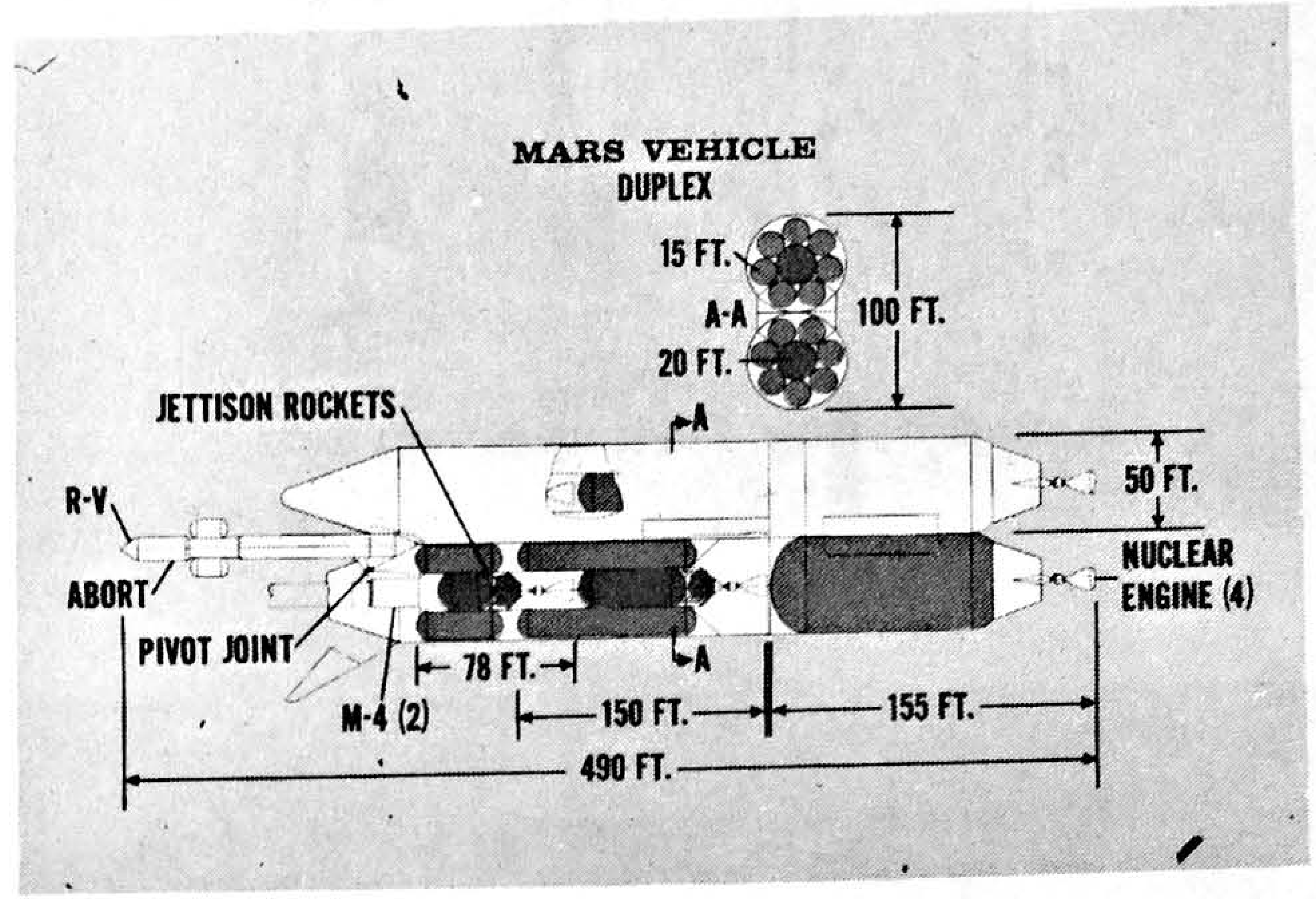
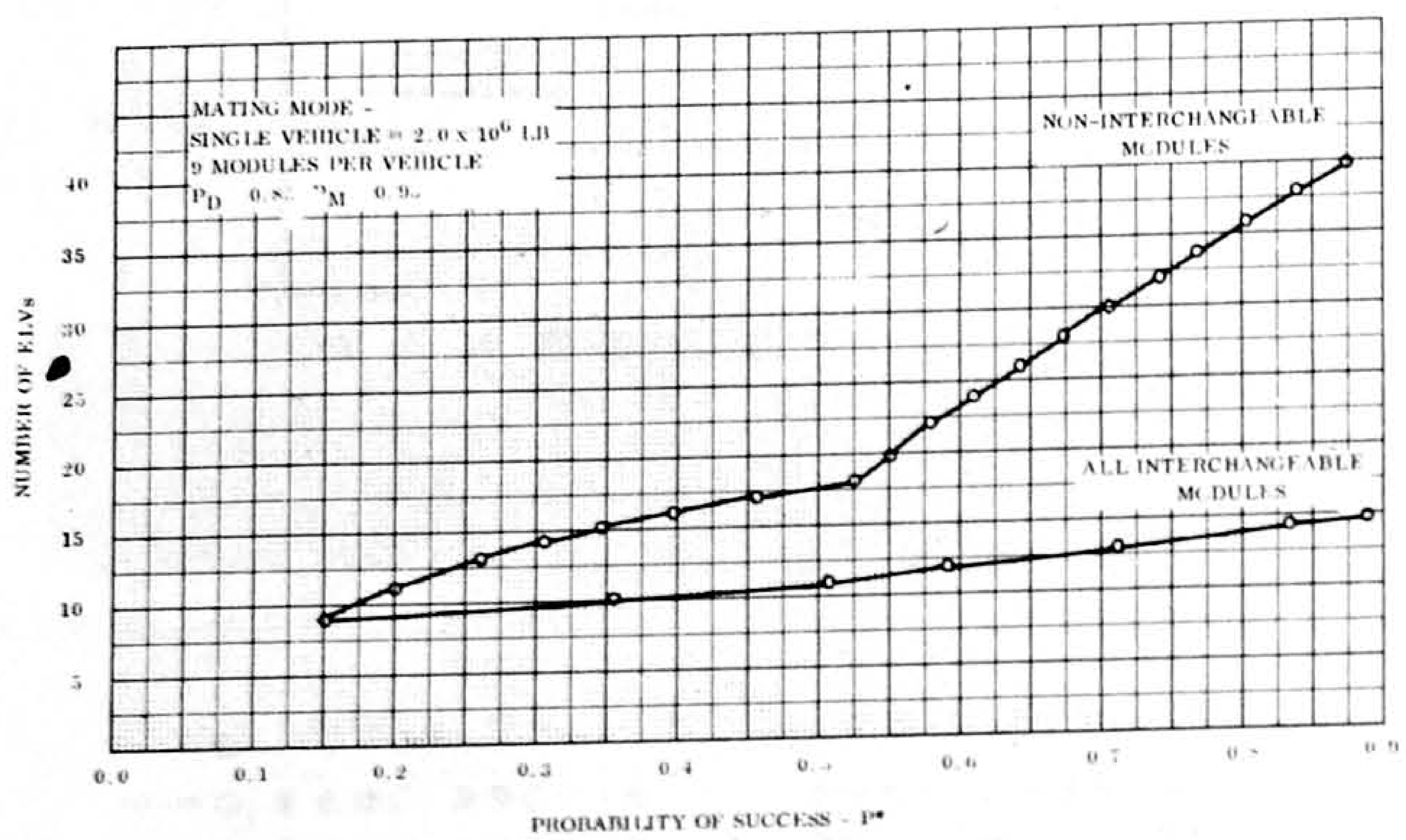
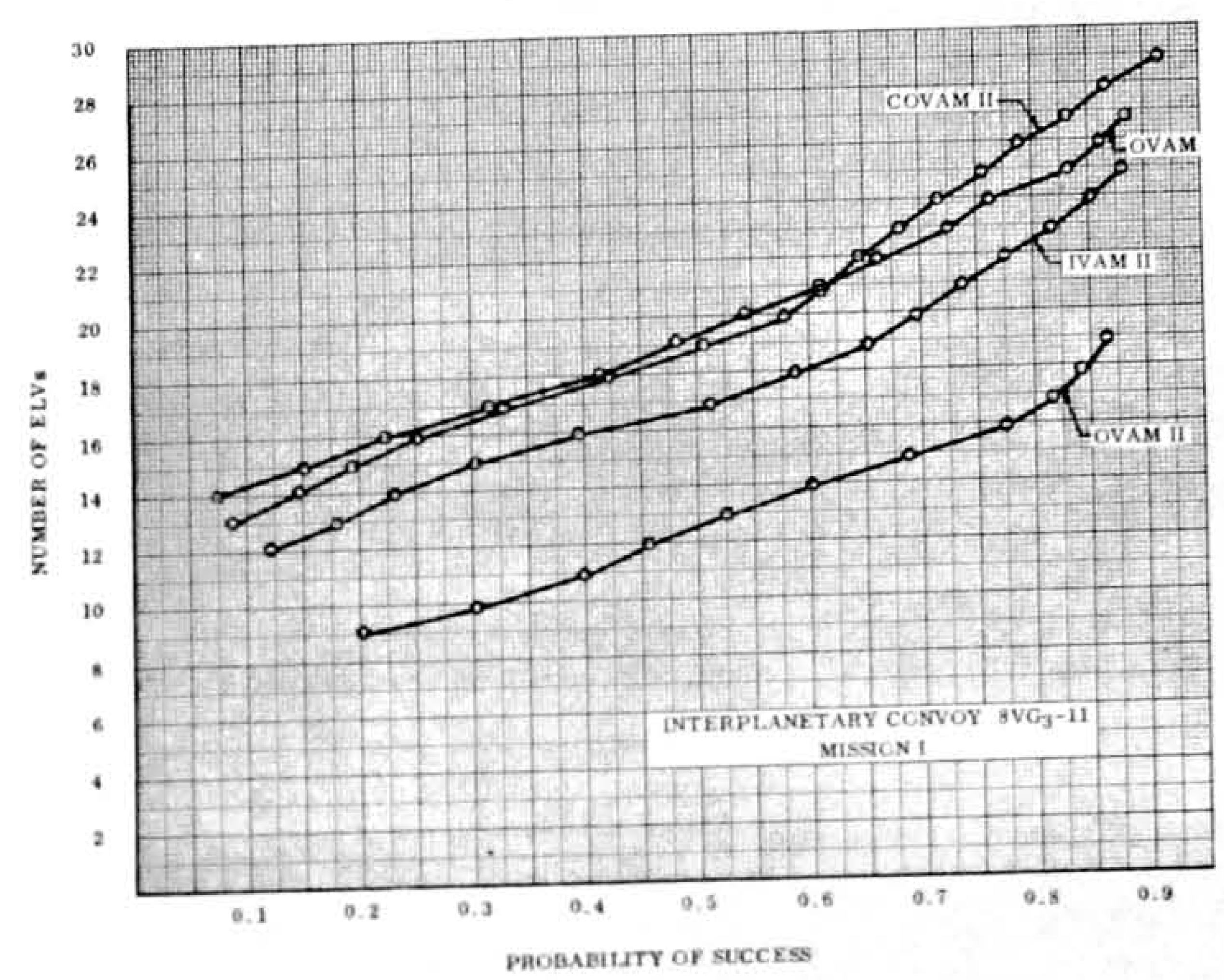


Fig. 5-14



EFFECT OF MODULE INTERCHANGEABILITY ON ELVs

Fig. 5-17



EFFECT OF MISSION MODE ON ELV REQUIREMENTS

Fig. 5-18a

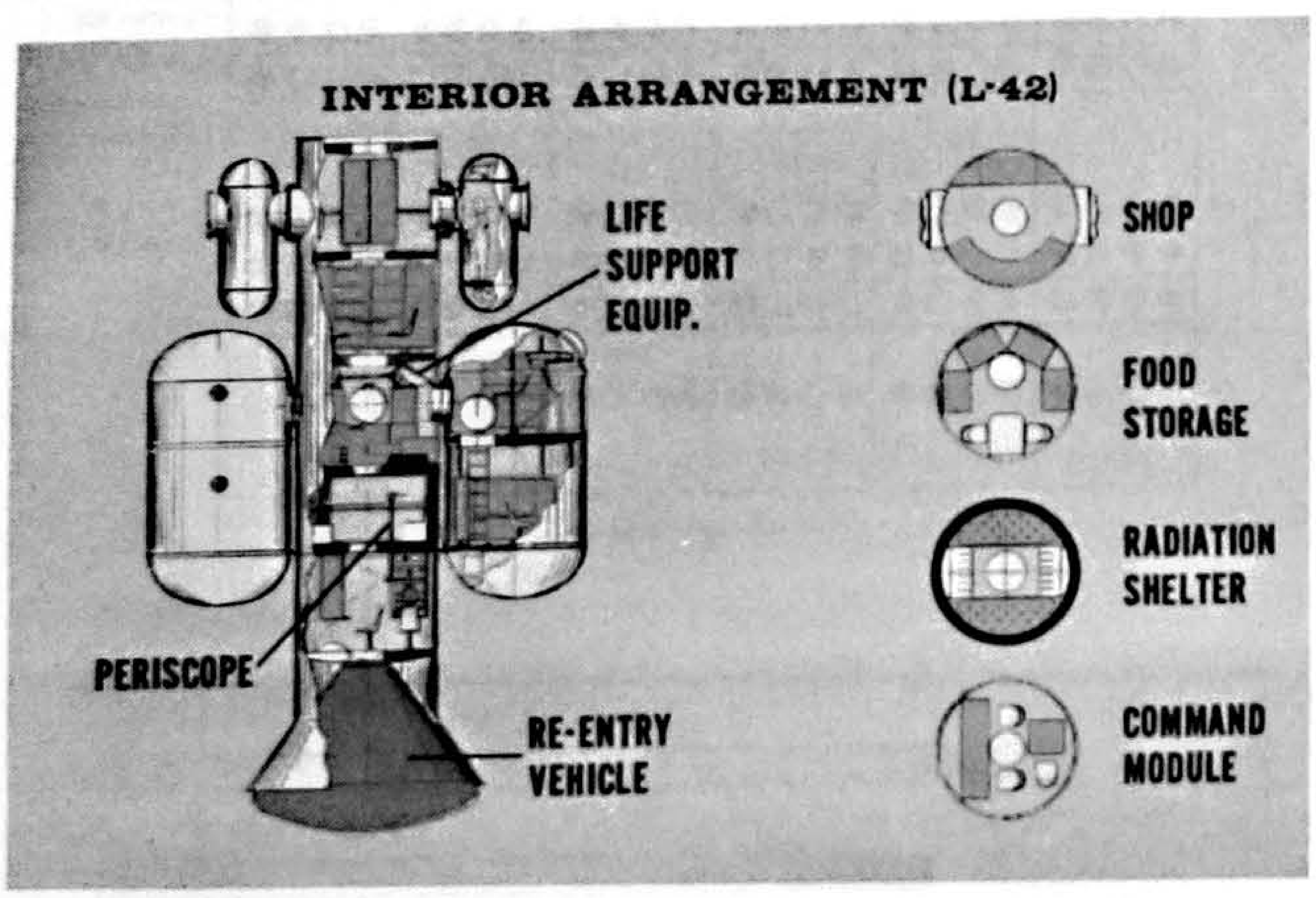
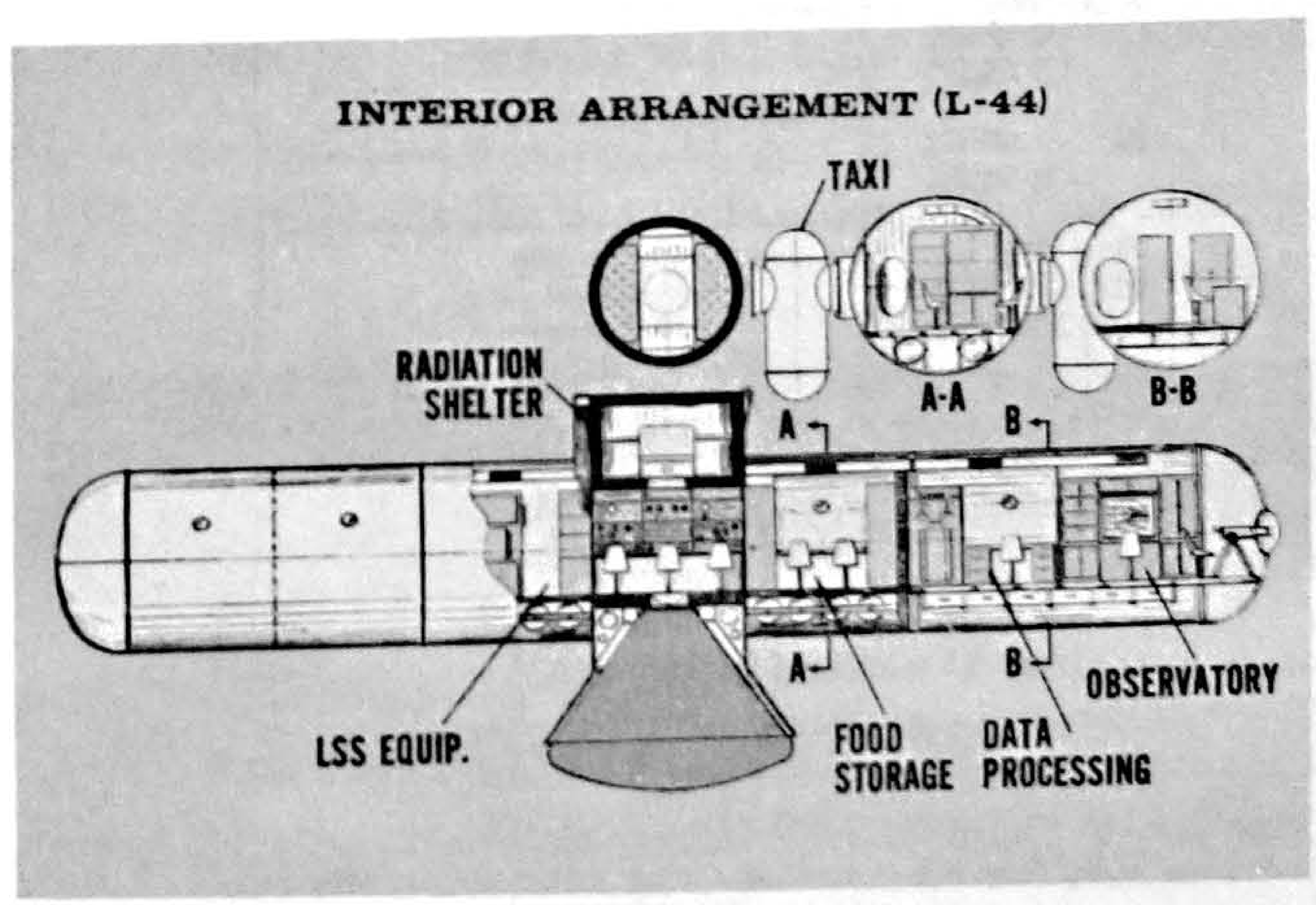


Fig. 5-18b



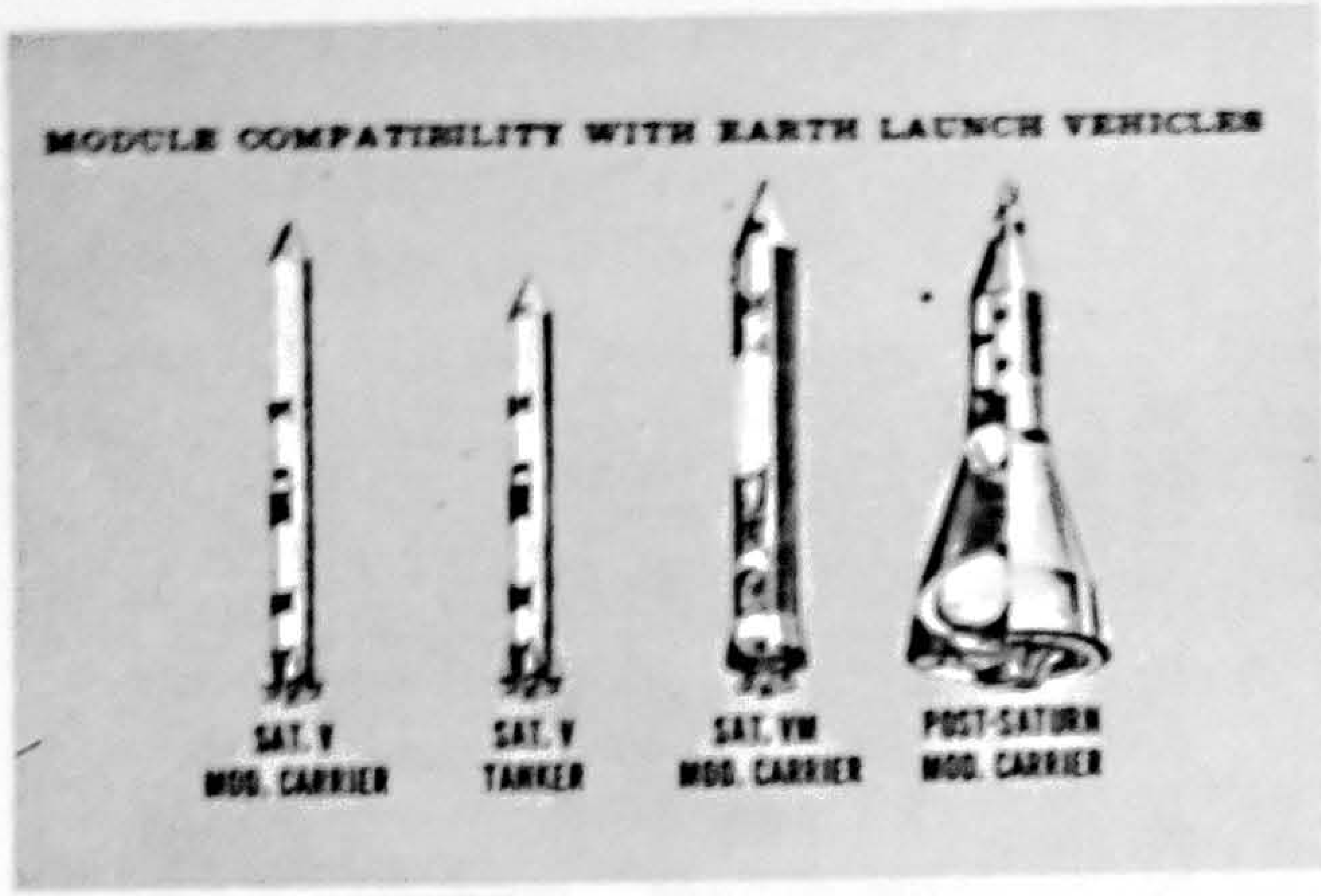
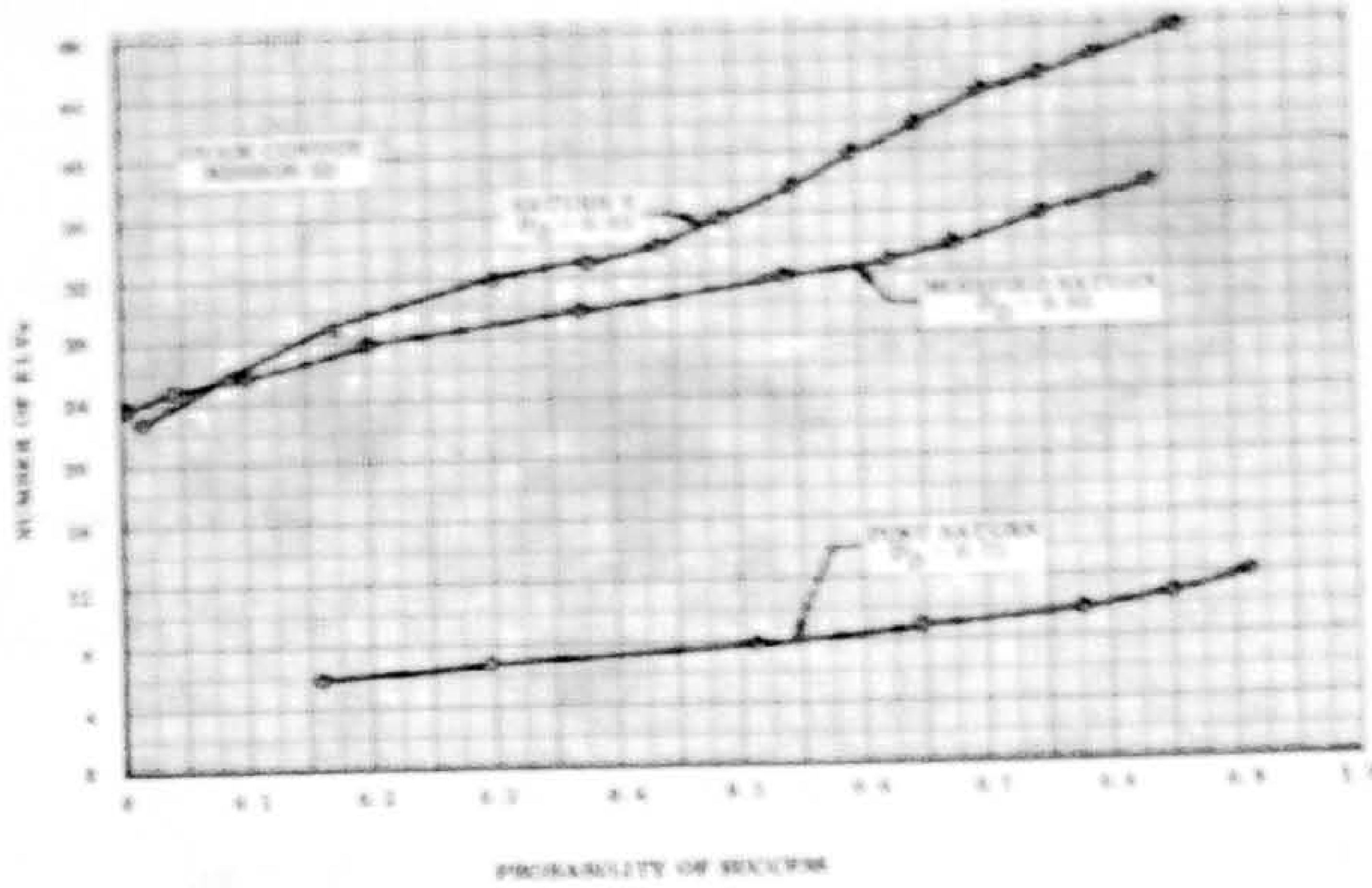


Fig. 5-19



EFFECT OF ELV ON LAUNCH REQUIREMENTS

Fig. 5-20

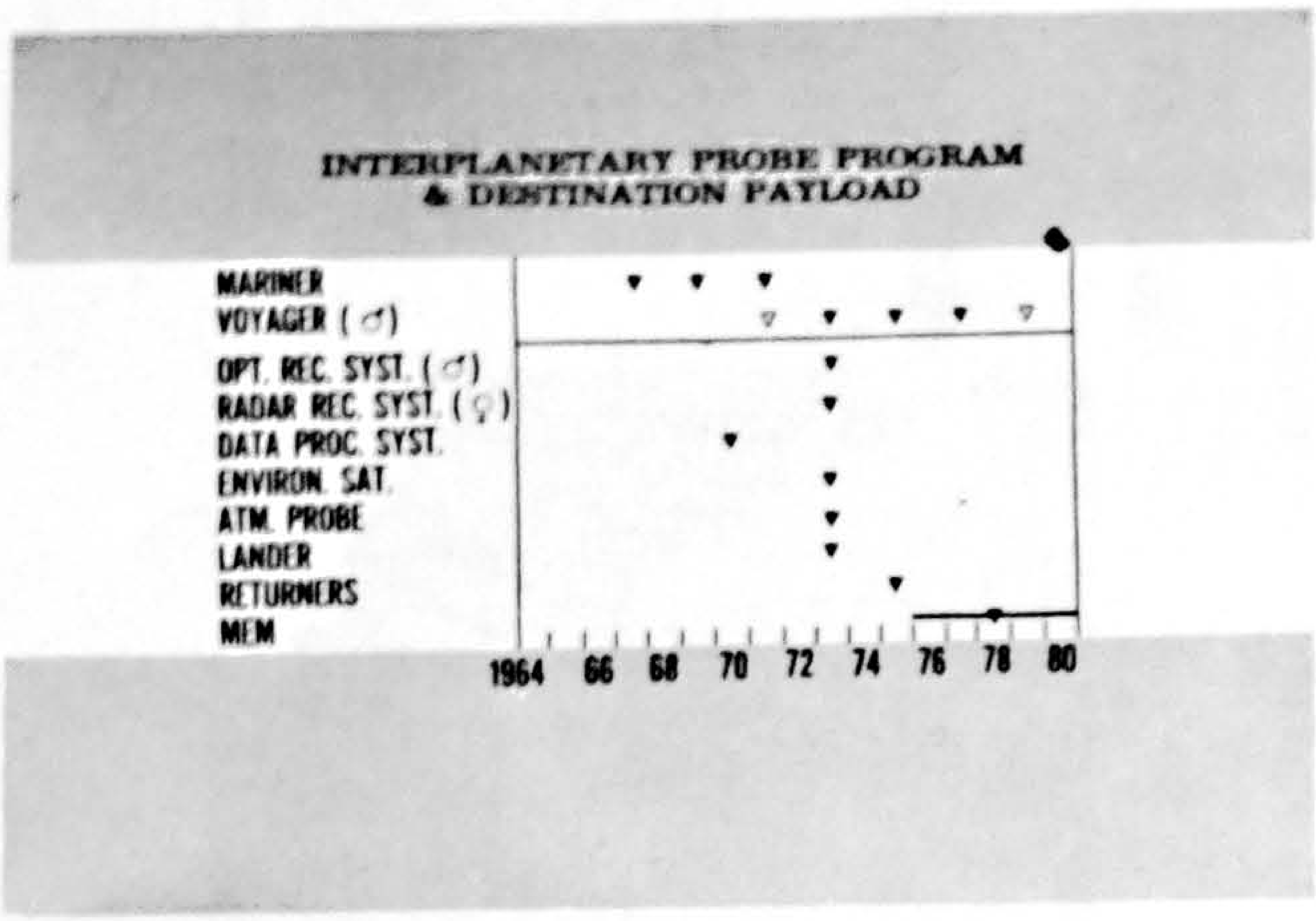


Fig. 5-22 a

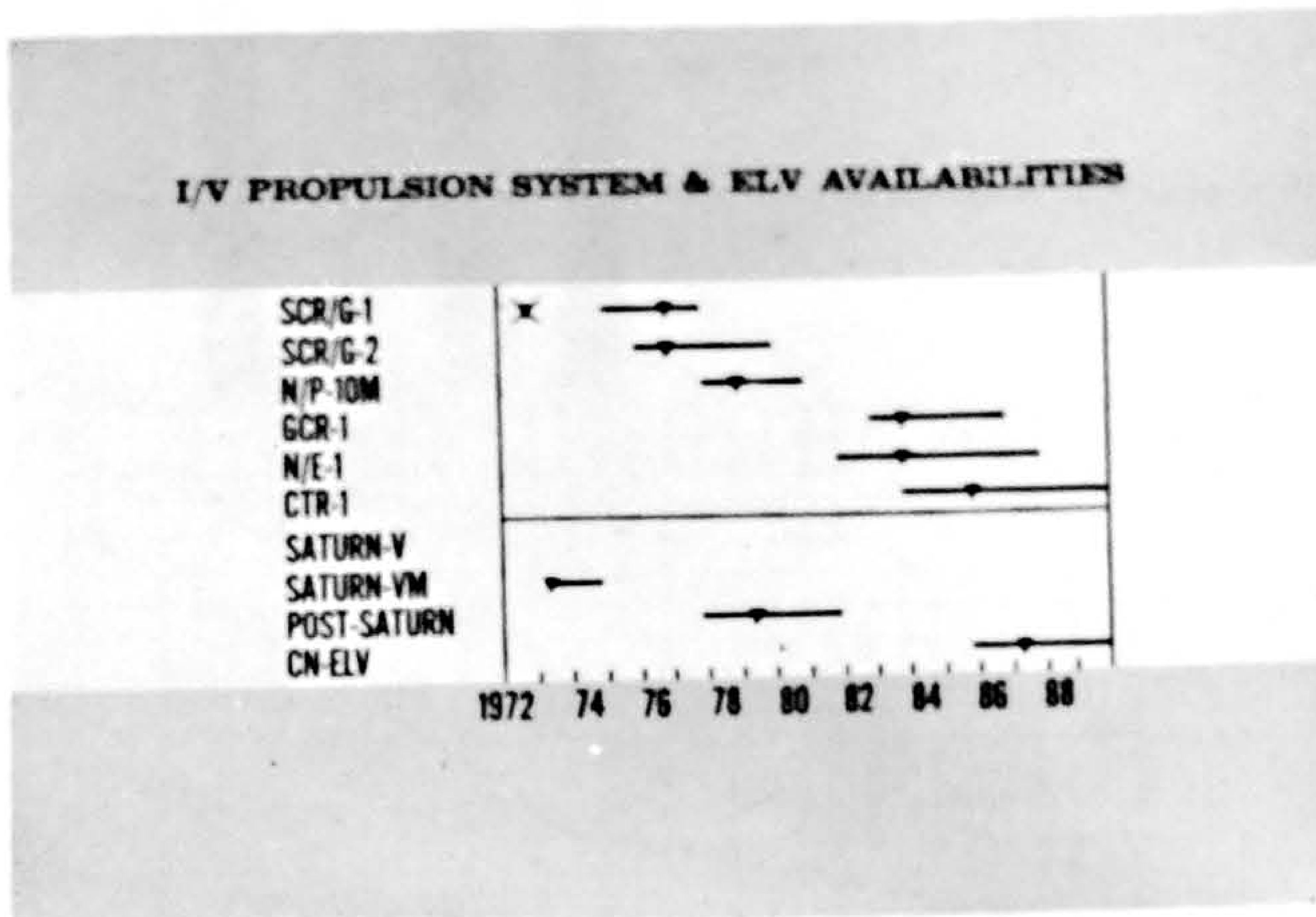


Fig. 5-22b

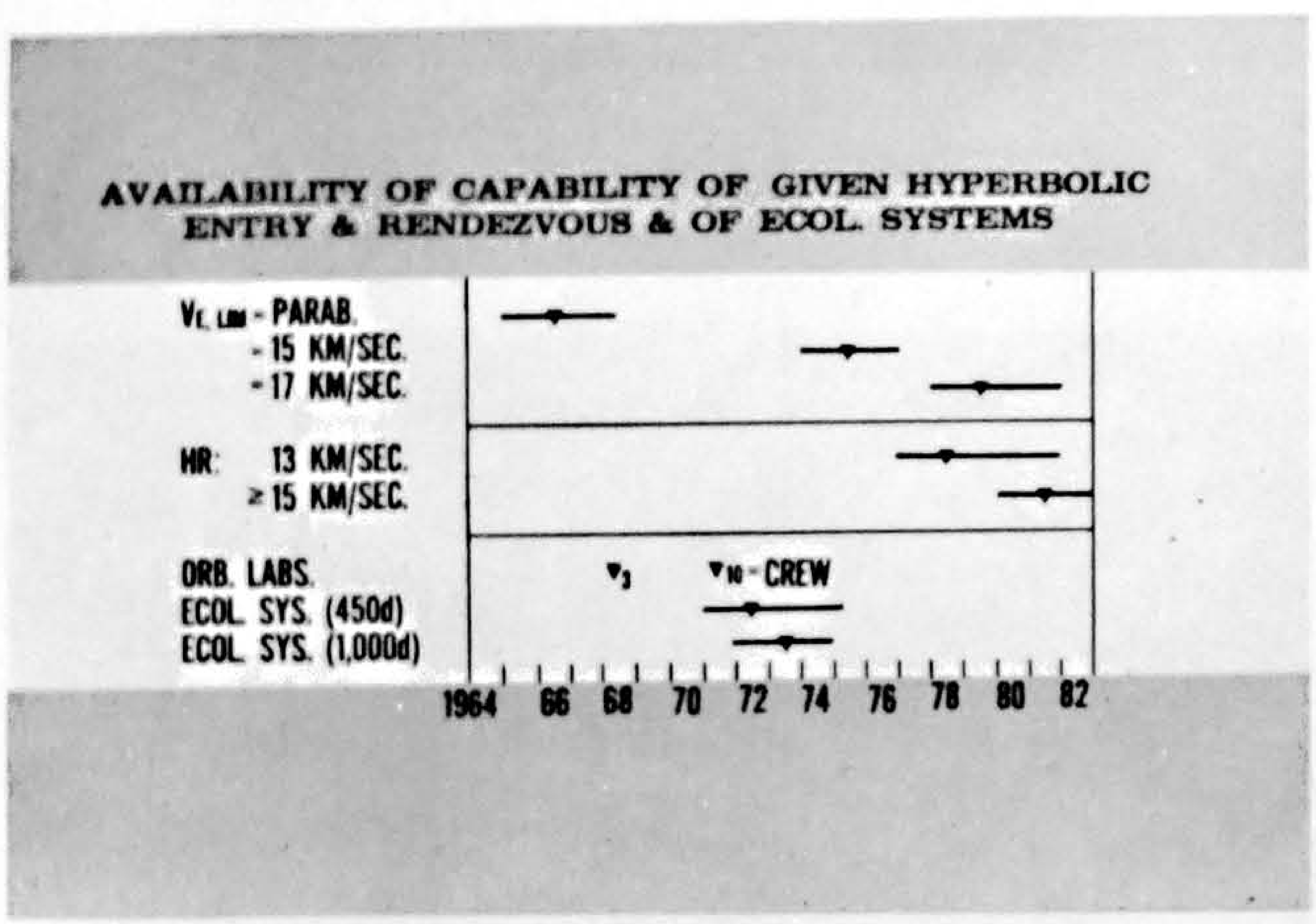
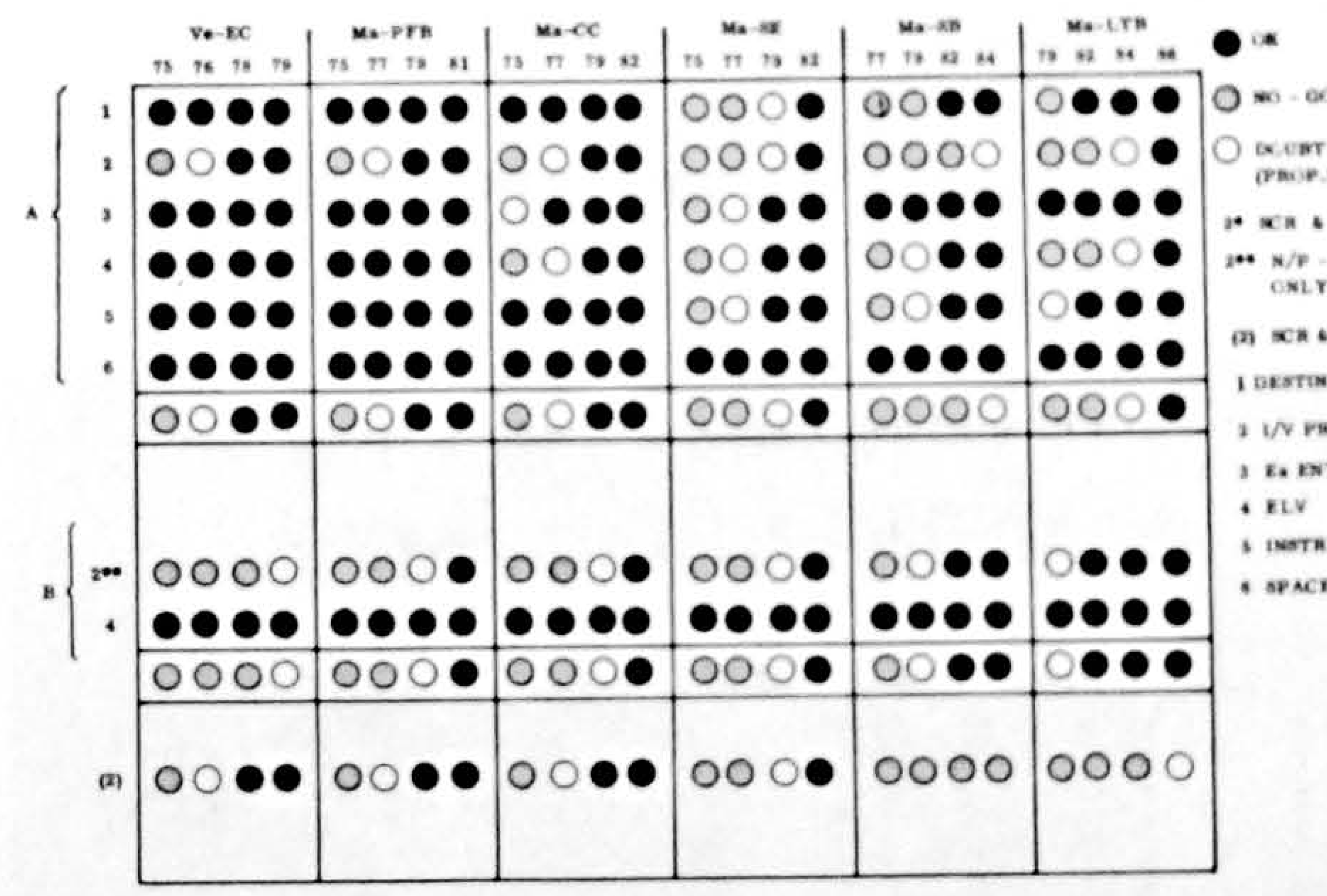


Fig. 5-22c



MISSION FEASIBILITY EVALUATION

Fig. 5-23

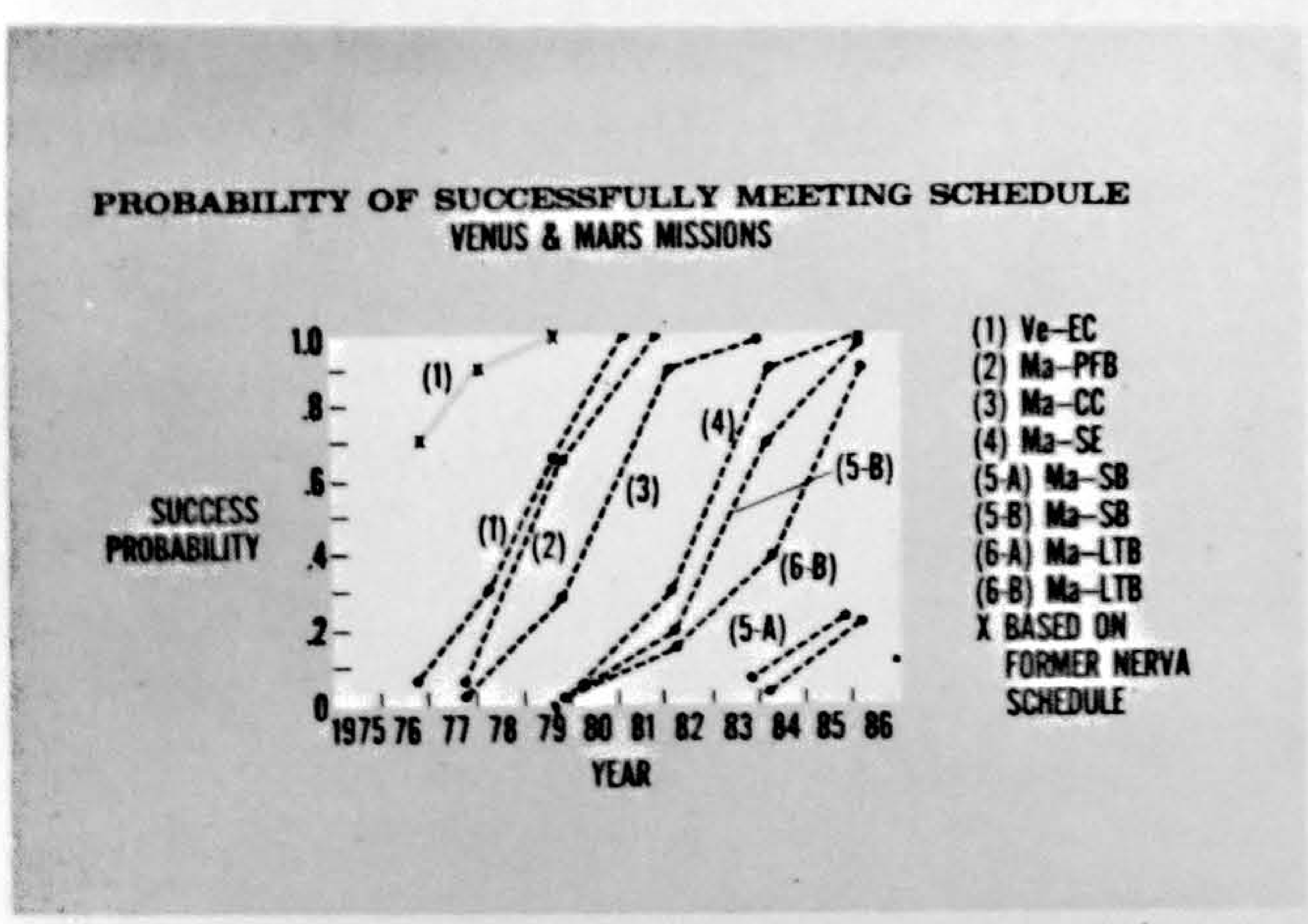


Fig. 5-24

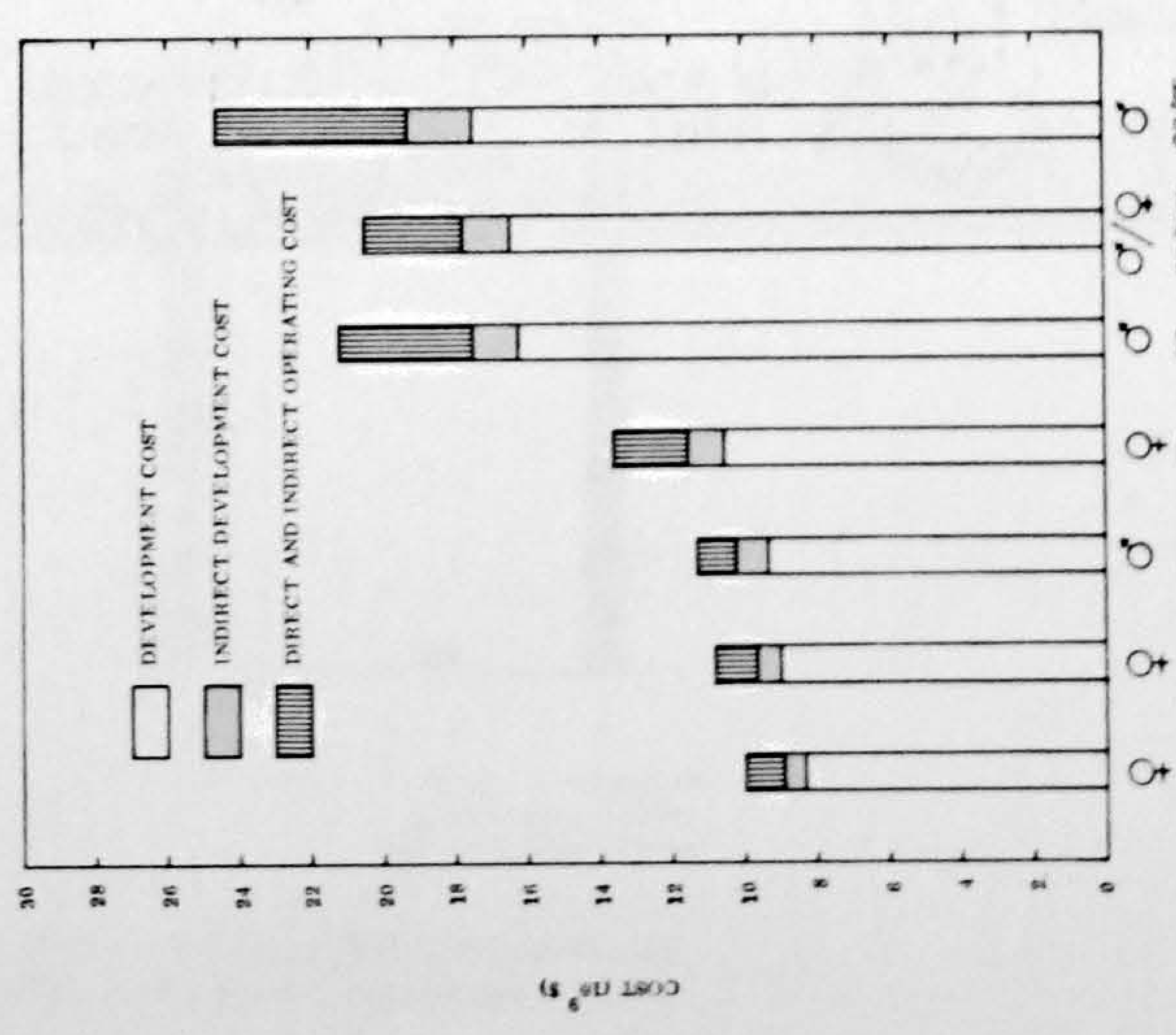
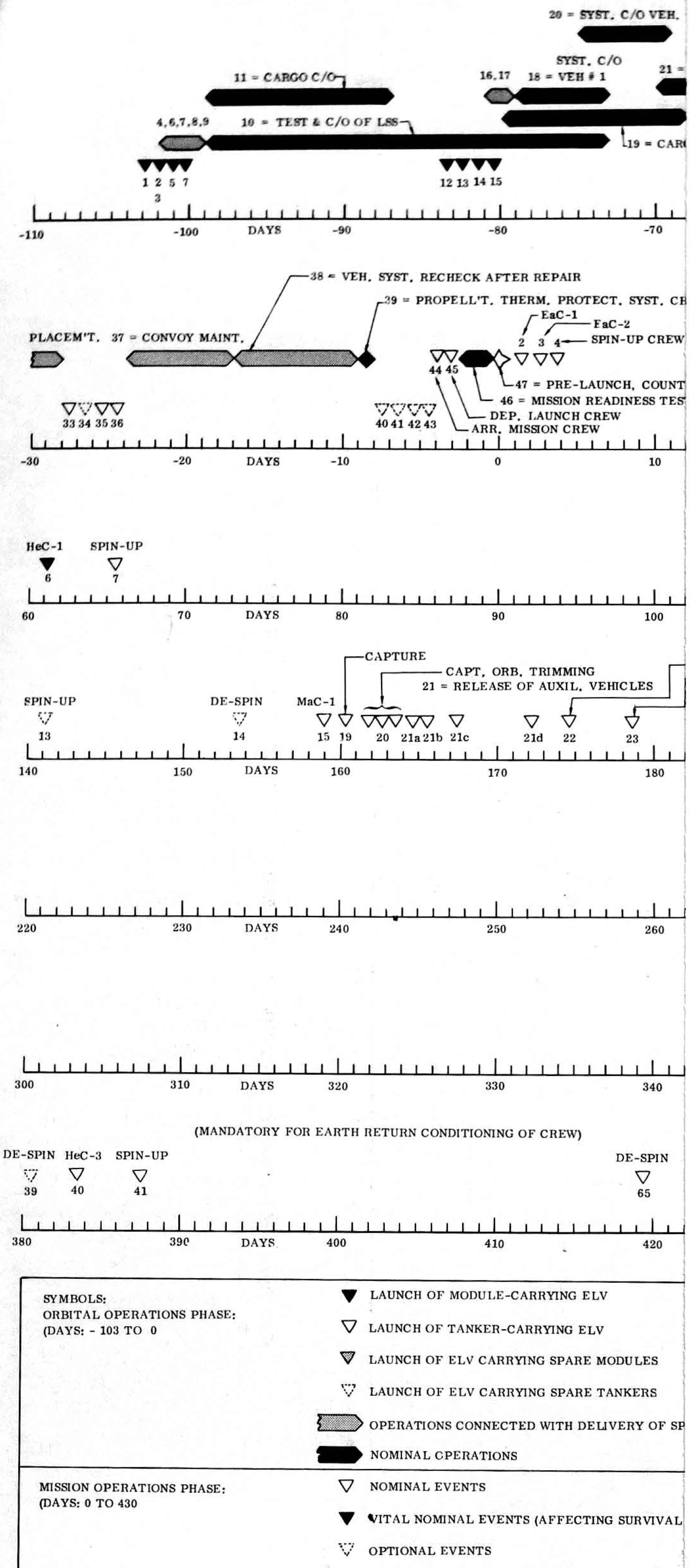


Fig. 5-25

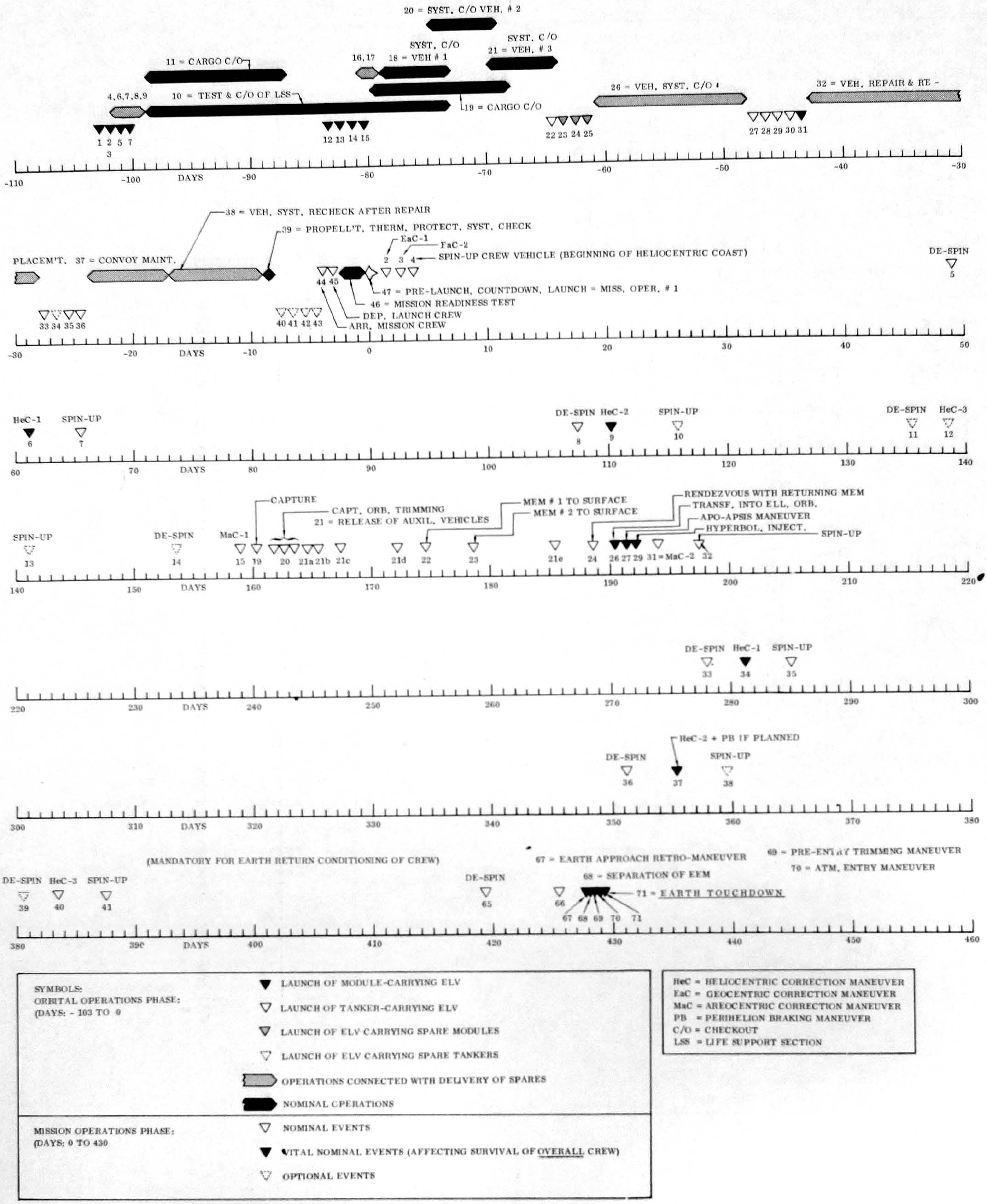


NOTE: OPERATIONS IN ORBITAL PHASE ARE NUMBERED 1 THROUGH 47 = ORBITAL LAUNCH
COUNTING OF MISSION OPERATIONS BEGINS WITH 1 = ORBITAL LAUNCH
DAYS OF ORBITAL OPERATIONS ARE NEGATIVE AND COUNTED DOWN
DAYS OF MISSION OPERATIONS ARE POSITIVE AND COUNTED UP

Fig. 5-21 SEQUENCE OF EVENTS CHART FOR C
AND MISSION OPERATIONS MODELS

NOMENCLATURE

AB	Aerodynamic Braking
AEP	Atmospheric high-speed entry probes
C	Capture
CC	Circular Capture
Crew Payload	= Payload of crew vehicle (primarily, operational payload)
COVAM	Capture Orbit Vehicle-Assembly Mode
COVAM(2)	Same as COVAM except Service Vehicle only goes to target planet
CS/V	Combined crew and service vehicle in COVAM(2)
CV	Convoy Vehicle
C/V	Crew Vehicle
Destination Payload	= Payload to be used at destination
DFM	Direct Flight Mode
DV	Duplex Vehicle
Ea	Earth
EC	Elliptic Capture
EEM	Earth Entry Module
ELV	Earth Launch Vehicle
EMOS	Earth Mean Orbital Speed
ES	Environmental Satellite
FB	Fly-By
HE	Hyperbolic Entry
HR	Hyperbolic Rendezvous
IMICOMP	Interplanetary Mission Information Computer Program
I/V	Interplanetary Vehicle
IVAM	Interorbital Vehicle Assembly Mode
IVAM(2)	Same as IVAM except Service Vehicle only goes to target planet
LSS	Life Support Section
M-#	Maneuver (# indicates maneuver in question)
-1	Earth Departure
-2	Target Planet Arrival
-3	Target Planet Departure
-4	Earth Arrival
Ma	Mars
MEM	Mars Excursion Module
MiO	Mission Objective
MRM	Mars Retro Maneuver
MV	Multiplex Vehicle
ODW	Orbital Departure Weight
Operational Payload	= Payload involving crew, life support section and other equipment required to operate the vehicle or the convoy
OVAM	Orbital Vehicle Assembly Mode
OVAM(2)	Same as OVAM except Service Vehicle only goes to target planet
PB	Perihelion Braking
PFB	Powered Fly-By
PM-#	Propulsion Module (# indicates maneuver in question)
PM-3 Carrier	I/V carrying the Mars or Venus departure propulsion module, to the target planet in IVAM(2) and COVAM(2)
PU	Pick-Up
PUV	Pick-Up Vehicle
R_A	Aphelion Distance
RIP	Roving Interplanetary Probes
R_P	Perihelion Distance
r*	Radial distance from planet (in planet radii)
SE	Surface Excursion
Service Payload	= Payload of service vehicle (primarily, destination payload)
S/V	Service Vehicle
Synodic	Relating to the period between two conjunctions or two oppositions of Earth - Venus and Earth - Mars, respectively
T_{cpt}	Time capture period at target planet
Ve	Venus
Δv_1, Δv_2, Δv_3	= Impulse maneuvers at Earth departure, Mars arrival and Mars departure.



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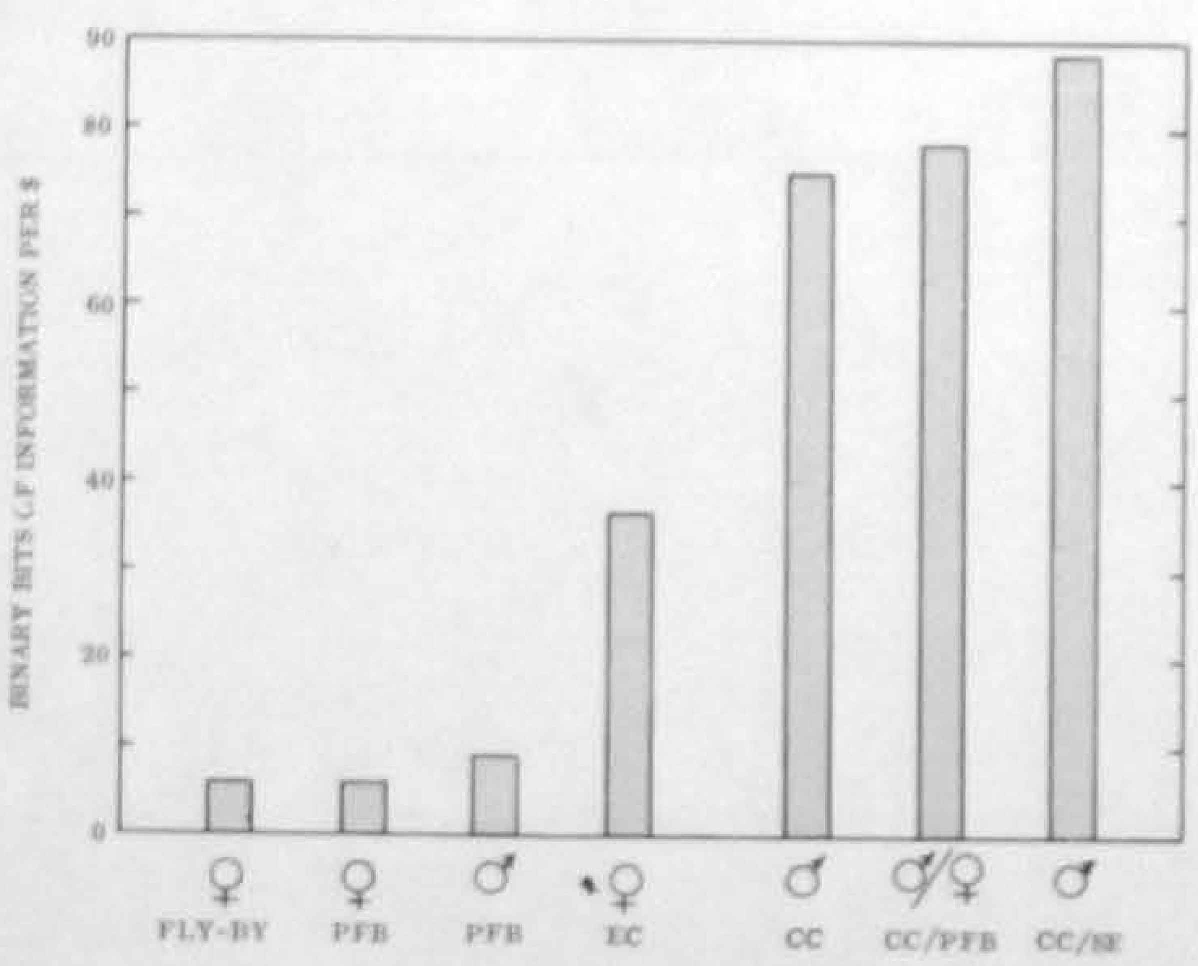
DAYS OF ORBITAL OPERATIONS ARE NEGATIVE AND COUNTED DOWN
DAYS OF MISSION OPERATIONS ARE POSITIVE AND COUNTED UP

Fig. 5-21 SEQUENCE OF EVENTS CHART FOR COMBINED ORBITAL OPERATIONS AND MISSION OPERATIONS MODELS

ABBREVIATIONS
(In Figs. 5-22, 23, 24)

Opt. Rec. Syst. (♂)	Optical reconnaissance system for Mars observation from orbit
Radar Rec. Syst. (♀)	Radar reconnaissance system for Venus observation from orbit
Data Proc. Syst.	Data processing system for handling storage and delayed transmission to Earth
SCR/G-1	Solid core reactor / graphite - First generation
SCR/G-2	Solid core reactor/ graphite - Second generation
N/P-10M	Nuclear Pulse System - First generation
GCR-1	Gas Core Reactor System - First generation
N/E-1	Nuclear Electric System - First generation
CTR-1	Controlled Thermonuclear System - First generation
Saturn-VM	Hypothetical modified Saturn of 50 ft dia.
$V_{E, LIM}$	Limiting atmospheric entry vehicle,
HR	Hyperbolic Rendezvous
Ecol. Syst. (450 d)	Ecological system for 450 d missions
(1) Ve-EC	Mission to Venus - elliptic capture orbit
(2) Ma-PFB	Mission to Mars - powered fly-by
(3) Ma-CC	Mission to Mars - circular capture orbit
(4) Ma-SE	Mission to Mars - surface excursion
(5A) Ma-SB	Mission to Mars - synodic base (use of gas core reactor engines)
(5B) Ma-SB	Mission to Mars - synodic base (using nuclear pulse systems)
(6A) MA-LTB	Mission to Mars - long term base (using gas core reactor engines)
(6B) MA-LTB	Mission to Mars - long term base (using nuclear pulse systems)



MISSION YIELD, MEASURED IN GROSS AMOUNT OF BINARY BITS OF INFORMATION GATHERED DURING THE MISSION PER DOLLAR OVERALL MISSION COST.

Fig. 5-26

NOMENCLATURE

AB	Aerodynamic Braking
AEP	Atmospheric high-speed entry probes
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CC	Circular Capture
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Ve	Venus
Δv_1, Δv_2, Δv_3	= Impulse maneuvers at Earth departure, Mars arrival and Mars departure.

Mission	V-I		V-II	V-III	M-I		M-II	M-III	M-IV	M-V	M-VI	VM-I	MV-I
	V-I. 1	V-I. 2			M-I. 1	M-I. 2							
Target Planet	Venus	Venus	Venus	Venus	Mars	Mars	Mars	Mars	Mars	Mars	Mars	Venus Mars	Mars Venus
Mission Type	Fly-By	Fly-By	Powered Fly-By	Elliptic Capture	Med. Energy Fly-By	Med. Energy Fly-By	Low-Energy Fly-By	Powered Fly-By	Fast Mission Circ. Capt.	Fast Mission Circ. Capt.	Slow Synod. Mission Circ. Capt.	Ell. Capt. & Circ. Capt.	Circ. Capt. & Ell. Capt.
Surface Excursion Capability	No	No	No	No	No	No	No	No	No	Yes	Yes	Yes	No
Departure Earth	Late July 1975	Late March 1977	July 1975	5-20-75	Early March 1975	Early May 1977	Mid-Oct. 1975	Late Sept. 1975	9-5-75	9-5-75	9-15-75	6-19-75	10-5-75
Mission Duration (d)	80+280= 360	80+280 360	100+250= 350	140+(20)+ 240=400	285+285 = 570	285+285 = 570	115+535= 650	140+240= 380	160+(30+20) +230 = 440	160+(30+20) +230 = 440	360+(220) +370 = 950	120+(246)+ 230+(198)+ 210 = 1004	160+(38)+ 190+(64)+ 160 = 612
Velocity Requirements for Principal Maneuvers													
M-1 (10^3 ft/sec)(km/sec)	(16.9)(5.15)	(22.4)(6.84)	(16.6)(5.1)	(12.9)(4.24)	(29.3)(8.94)	(29.3)(8.94)	(17.7)(5.4)	(14.9)(4.53)	(14.9)(4.9)	(14.9)(4.9)	(12.4)(3.93)	(12) (3.66)	(14.24)(4.34)
M-2 (10^3 ft/sec)(km/sec)			(4) (1.2)	(4.9) (1.61)				(3) (0.92)	(17.0)(5.6)	(17.0)(5.6)	(7.4) (2.25)	(3.4) (1.04)	(12.91)(3.72)
M-3 (10^3 ft/sec)(km/sec)				(18.8)(6.14)					(19.4)(6.4)	(19.4)(6.4)	(12.7)(3.86)	(14.2)(4.32)	(13.9)(4.24)
M-4 (10^3 ft/sec)(km/sec)					(12)	(15)			(18.7)(6.15)	(18.7)(6.15)		(12.5)(3.8)	(8.9)(2.72)
M-5 (10^3 ft/sec)(km/sec)												(8.8) (2.68)	(13.5)(4.12)
Overall mission velocity based on M-1 through M-5 without navigational correction maneuvers (10^3 ft/sec) (km/sec)	(16.9)(5.15)	(22.4)(6.84)	(20.6)(6.3)	(36.6)(12)	(29.3)(8.94)	(29.3)(8.94)	(17.7)(5.4)	(17.9)(5.45)	(70) (23)	(70) (23)	(32.5)(10)	(50.9)(15.5)	(63.5)(19.1)
Earth Entry Velocity without retro-thrust (10^3 ft/sec) (km/sec)	(45) (13.7)	(47) (14.3)	(43) (13.1)	(46) (14)	(57.6)(17.6)	(60) (18.5)	(48.5)(14.8)	(47) (14.4)	(68.2)(20.8)	(68.2)(20.8)	(38) (11.6)	(42) (12.7)	(38) (11.6)
Retrothrust Required prior to Earth Entry	No	Probably not	No	Probably not	Yes	Yes	Probably not	Probably not	Yes	Yes	No	No	No
Minimum number of powered maneuvers (not counting navigational correction maneuvers)	1	1	2	3	2	2	1	2	4	4	3	5	5
Operational payload (incl. Earth entry module)	126,000	126,000	120,000	127,000	131,000	131,000	133,000	126,000	128,000	128,000	310,000	310,000	200,000
Orb. Dep. Wt. (lb) conservative	350,000	550,000	450,000	1,100,000	1,000,000	1,000,000	420,000	380,000	2,300,000	2,300,000	2,100,000	3,500,000	3,300,000
optimistic					750,000	750,000			1,500,000	1,500,000	1,600,000	2,300,000	2,100,000

Table 3-1

CHARACTERISTIC MISSION FLIGHT DATA

Table 3-1

CHARACTERISTIC MISSION FLIGHT DATA

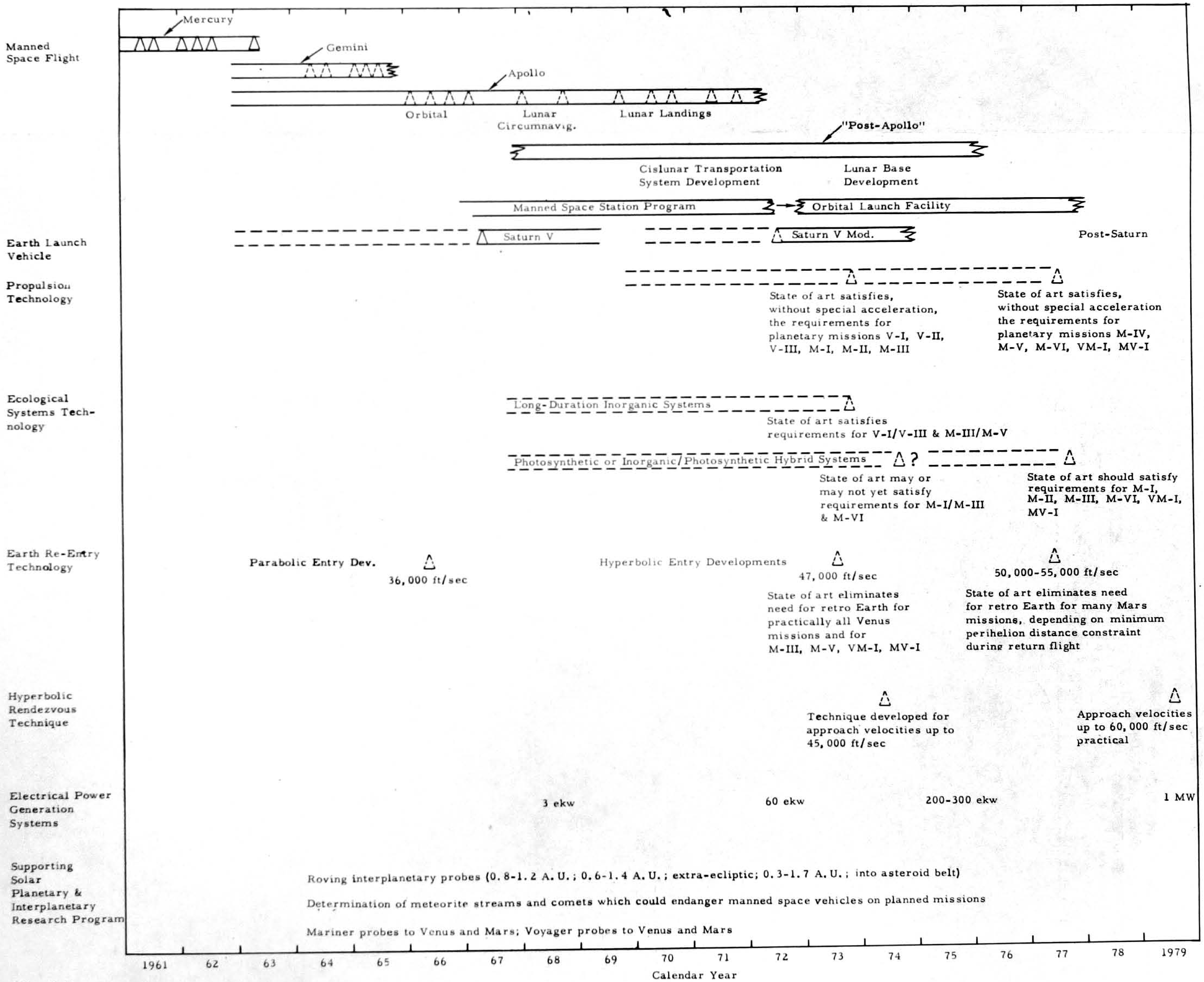
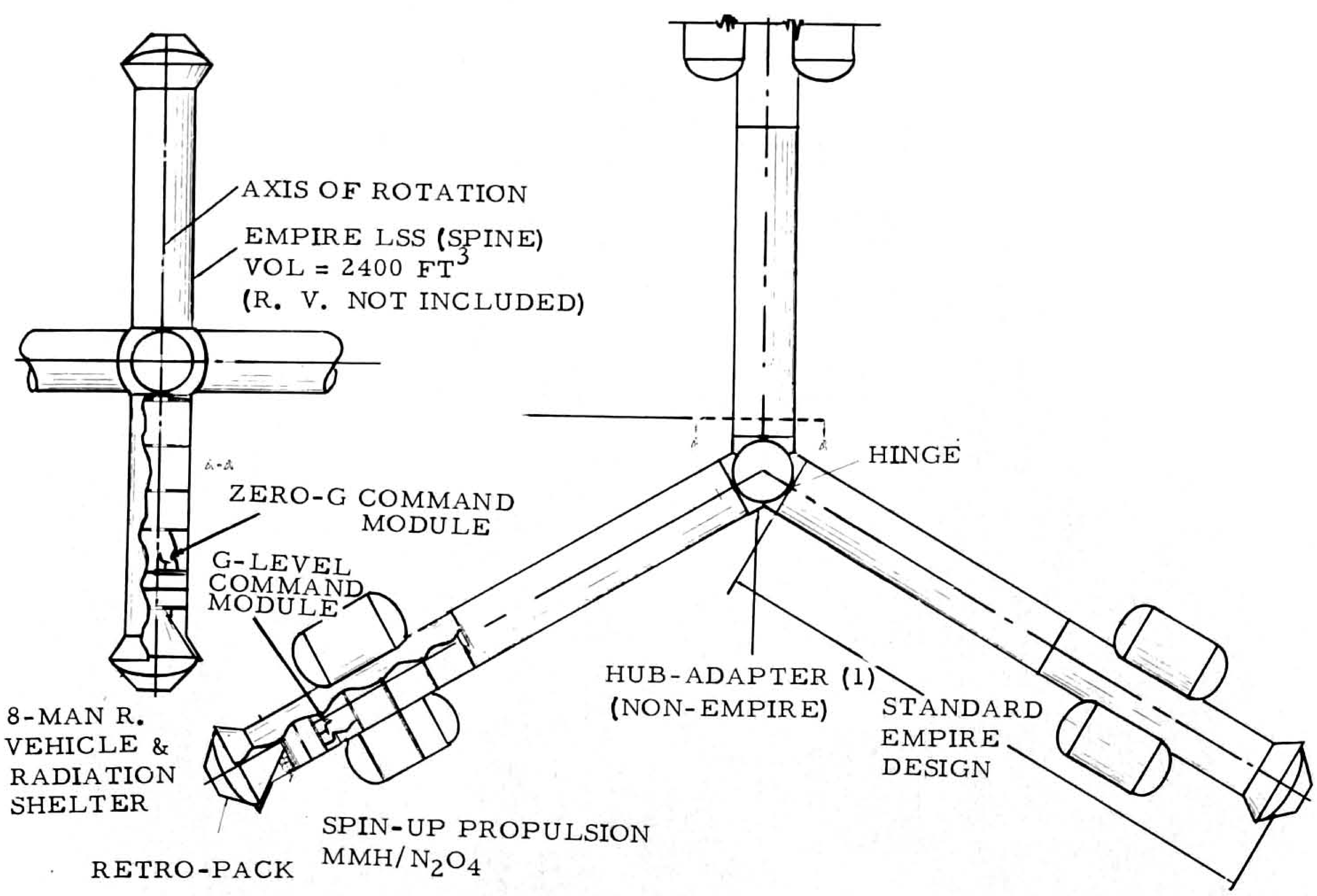


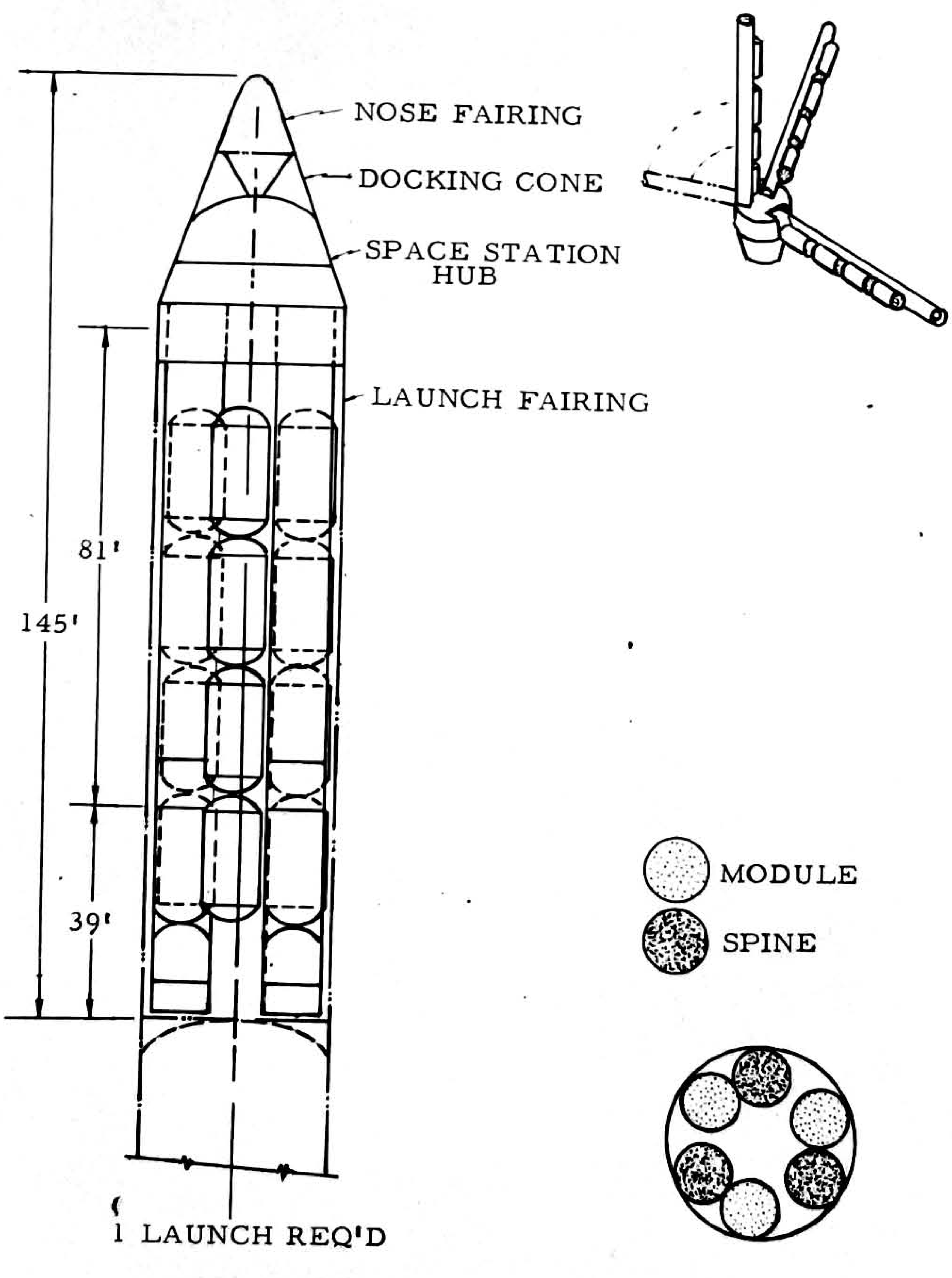
Fig. 3-2

KEY RESEARCH AND DEVELOPMENT REQUIREMENTS IN PREPARATION OF MANNED PLANETARY MISSIONS



EMPIRE TEST SPACE STATION: TRIANGULAR CONFIGURATION

TRIANGULAR SPACE STATION LAUNCH ARRANGEMENT



ASSEMBLE 9 MODULES IN ORBIT