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THE EMPIRE DUAL PLANET FLYBY MISSION

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Abstract

The investigations and results of the EMPIRE Study Program performed for NASA by Aeronutronic in 1962 are summarized. Mission analyses and spacecraft syntheses are reported for various methods of accomplishing a manned mission with close approach to both Venus and Mars and return to the Earth in the 1970-72 period. Major systems requirements are defined and a Development Plan and Funding Schedule are presented for accomplishing the dual planet flyby using a nuclear engine for interplanetary injection from Earth orbit, taking up to 631 days, and providing for Earth return with a lifting re-entry vehicle. Accelerated developments required to support such a mission are specified.

Introduction

General

The EMPIRE* studies form a unique program to examine the problem areas associated with the goals of early manned interplanetary missions in the early 1970's. Aeronutronic has concentrated its efforts on the dual planet flyby missions under contract NAS8-5025 for the Future Projects Office, Marshall Space Flight Center, Huntsville, Alabama. The work, which is summarized only briefly in this paper, led to definition of feasible systems and pinpointed problem areas requiring further attention for manned flights by Mars and Venus and safe return to the surface of the Earth.

It should be noted that the interest in dual planet missions precluded a detailed exploration of other planetary missions in the present study. This does not mean that single planet flyby missions, planetary orbital missions, and planetary landing missions are less important. However, having obtained the results oriented toward low and medium energy trips in a free-fall interplanetary transit for the dual flyby mission, the relative cost and utility of the latter versus the larger energy demands of landing and orbiting missions can be evaluated.

Much credit must be given to the forward thinking approach shown by the NASA effort on this program in 1962. By attacking the areas of interest at this early date, it was possible to obtain a clearer picture of the requirements for early manned planetary and interplanetary flight. Thus the nation's resources, and the NASA and other United States space programs, can be oriented toward long range goals at an early date. The

* Early Manned Planetary-Interplanetary Round-trip Expedition.

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efforts reported here represent an unusually early attack for this type of analysis. The conclusions, which indicate the need for acceleration of certain related development programs to meet the earliest possible low energy Venus-Mars flyby launch date of July 1970, justify this early definition of potential problem areas.

A Development Plan and Funding Schedule for the design point mission is presented below. Use of the selected mission for this purpose does not indicate that such a program will be undertaken and funded in time to utilize the July-August 1970 Symmetric Orbit Launch Window. In fact, work should have started by 1 July 1963 to meet the schedule milestones for this earliest mission. However, this plan defines in some detail enough of the development, test, and funding requirements to show the critical areas. This work has provided information for appraisal of technical and financial considerations in discussions and for decisions related to the early manned planetary programs.

Scope of the Aeronutronic EMPIRE Study

The primary goals of the EMPIRE Study Program were to establish requirements for the Nova booster development program, to provide inputs to the nuclear rocket program, and explore any advanced space operational concepts necessary for implementation of the missions under study. To define the requirements in these areas a detailed systems analysis, mission analysis, and program planning exercise was performed. The final goals of these analyses are the Development and Funding Plans mentioned before.

In order to obtain the above results it was necessary to concentrate on several areas for analysis and system integration. The areas requiring specific attention included:

1. Trajectories - Crocco* and other useful interplanetary trajectories
2. Propulsion - Nuclear and/or chemical rockets
3. Operations - Orbital operations and mission staging development requirements
4. Earth Re-entry
5. Crew Requirements
6. Scientific Aspects
7. Life Support Requirements
8. Cryogenics
9. Other Subsystems Definition - Electronics, Guidance, Control and Power Supply Requirements
10. Emergency Operations

* See Figure 2 and associated text for a definition of the mission trajectories.

As the study progressed, some of the specific areas required more concentrated investigation due to the critical nature of the system and mission constraints, due to the lack of current technological solutions, or due to other characteristics tending to render the particular area a "problem area." Some of these concentrations of effort became passe' for the final mission choice when the nuclear injection Symmetric Mission was selected as the design mission over the Crocco Nuclear, Symmetric Chemical I (I_{sp} = 410 seconds), and the Symmetric Chemical II (I_{sp} = 300 seconds) to be discussed in more detail. The selection of the lightest payload in Earth orbit for the dual planet flyby mission eliminated the requirement for long term cryogenic storage, and for orbital operations, such as rendezvous, assembly, and fueling, <u>if a Nova Earth launch vehicle is used</u>.

<u>Approach to the EMPIRE Study</u>

As historic background it might be useful to outline the course of the reported study program. In the early phase an effort was made to examine the Crocco orbit in a three dimensional analysis to confirm its desirability for a dual planet flyby mission. A launch window was obtained in 1971 which fell in the 1970-72 era under investigation. Velocity requirements, launch dates and flight times around 400 days were derived. A search was made for alternate trajectories and the symmetric trajectory was found to have a launch window in the middle of 1970 with a much lower velocity requirement for injection, but a higher return velocity after about 600 days. The detailed investigations on the two missions resulted in the data presented in the Trajectory Section.

In addition, the crew requirements, life support, radiation protection, and environment analyses were performed. Also, emergency operations were considered with the derivation of abort trajectories for return from an unsatisfactory injection phase of the Earth departure leg. A detailed analysis of the re-entry technique including evaluation of the applicability of Apollo developed technology was performed. A selection of a re-entry vehicle of the high lift-to-drag aerodynamic type with some rocket deceleration was made. This study and its results are discussed in the Re-entry Section.

The problems of guidance and navigation with the related mid-course and planetary approach velocity vector corrections were studied to define the overall system requirements. The goal was to derive design data to allow a system synthesis of suitable interplanetary spacecraft. All of the above detailed studies, and several other areas, were integrated to provide inputs for the interplanetary vehicles. These design considerations and various system parametric studies are discussed in the Design Section. It should be emphasized that four useful systems were provided in the various designs and that the all chemical powered Crocco vehicle was investigated and discarded as impractical due to excessive size, weight, and cost.

Examination of the various approaches to an early manned interplanetary vehicle for the missions studied led to the choice of the Nuclear

Symmetric Mission of 1970, as indicated above. This allowed a more detailed subsystem and vehicle definition and provided a program for the operational analysis and the Development Plan and Funding Program. Figure 1 shows the EMPIRE Interplanetary Spacecraft for the Nuclear Symmetric Mission. The lower portion pictures the nuclear rocket injection phase from Earth orbit to the first interplanetary leg and the upper part of the picture shows the deployed spacecraft approaching Mars. The Earth Re-entry Vehicle is attached at the forward end of the spacecraft.

The general conclusions and relevant recommendations based on this study serve to focus attention on the feasibility of the mission considered and the necessary developments to accomplish a useful program.

Mission Considerations

Ground Rules for EMPIRE

In the present study several points were established to provide guidelines. The trajectory was to provide a multiple planet flyby of Venus and Mars. The launch time was set for the early 1970's. Propulsion was to be nuclear and/or chemical. The boosters to be considered for Earth launch were Saturn and/or Nova to be launched from AMR with mission staging to be investigated in Earth orbit or in interplanetary transit.

It was specified that survival of the crew was essential and that the design should allow for a 3 percent growth in velocity requirements and a 10 percent growth in payload weight when booster and injection systems are selected.

Also, the scientific aspects were to be considered from the point of view of desirability, compatibility with the mission profile, and only in sufficient detail to establish spacecraft support requirements essential to the overall mission design.

Use was to be made of systems, concepts, and techniques currently under development with an effort to avoid extension of capabilities beyond those developments programmed for Apollo. Although not specifically set as a ground rule, ability to actually perform the mission selected in this study required a conservative approach to selection of system parameters and capabilities for the 1970 period and this approach was taken for all areas of this study where feasible.

Scientific Experiments

While a detailed system integration was not performed on the scientific payload, an investigation was made in order to arrive at a useful weight and power assignment for spacecraft synthesis. Investigation of past and future experiment requirements in the unmanned military and space programs with synthesis of several experiment combinations into payload packages supported the finding that a useful payload weight of 1000 to 2500 pounds and an average power of 200 to 300 watts represents a reasonable estimate of the requirements for an integrated payload package in 1970. This considers three phases of experimentation in five steps: interplanetary (and Earth departure), Mars approach and departure, interplanetary, Venus approach and departure, interplanetary (and Earth approach).

Many decisions remain in the choice of experiments, delivery techniques, data processing, data transmission, and other areas. The detailed selections will undoubtedly be based largely on data to be obtained in the 1962-63 planetary experiment programs using unmanned spacecraft and automatic equipment, therefore a general allocation with a minimum of 1000 pounds and 300 watts for spacecraft support requirements was adopted. For more detailed discussion of possible experiments you are referred to G. de Vaucouleurs[1].

Trajectory Studies

Several trajectories were investigated using a patched conic computer program. The trajectories included the one suggestion by Crocco[2] for an orbit by Venus and Mars with an unperturbed period of one year and a semi-major axis of approximately 1 astronomical unit (a.u.). Investigations showed that a detailed three dimensional analysis[3] using the actual orbital elements of the planets led to somewhat different conclusions although such a journey did have a launch window in 1971. The energy requirements for injection were high and proved to require an excessively large vehicle due to primary injection boost requirements even with a nuclear engine.

The Crocco mission and the alternate, or Symmetric, mission trajectories are indicated in Figure 2. The Crocco trajectory can be readily correlated with the upper diagram. The Symmetric mission of about 600 days derives its name from the approximately symmetric launch and arrival positions of the Earth with symmetry about the longitude position of Mars at the midpoint of the spacecraft trajectory (neglecting planetary perturbations).

Launch windows, characteristic velocities required as a function of time, the hyperbolic excess arrival and departure velocities at Mars and Venus, and the Earth re-entry velocities were computed using a Planetary Transfer Program and a Hyperbolic Turn Program for both the Crocco and Symmetric cases. A comparison of the more important features of each is given in Table 1. Here q is the periapse distance of closest approach to the center of the planet in planetary radii. The characteristic launch velocity is the impulsive velocity increment required to inject from a 300 kilometer Earth parking orbit into the required interplanetary transfer orbit. The comparison between 5.3 km/sec and 10.1 km/sec for the Symmetric versus the Crocco mission readily indicates the difference in injection energy requirements. The design point mission is the symmetric trajectory passing Venus and Mars in 613 to 631 days with launch from a 300 km Earth orbit between July 19, 1970 and August 16, 1970.

It is evident that q increases with launch date for the Symmetric mission with a value of 1.3 Mars and Venus radii at the beginning of the possible launch dates and a value of 2.0 radii a month later. The legs of the mission include the following times:

Earth to Venus about 97 to 102 days
Venus to Mars about 191 to 199 days
Mars to Earth about 312 to 343 days

The total mission will run from 613 to 631 days, as shown.

Abort Capabilities

Single impulse abort maneuvers from a geocentric escape trajectory have been investigated to determine the velocity increment necessary to accomplish a successful return to Earth in a specified time after abort. Escape trajectories have been assumed to have a perigee radius of 1:1 Earth radii, with hyperbolic excess velocities varying between zero and sixteen km/sec. Abort trajectories are assumed to be elliptical, with a perigee radius of 1.005 Earth radii. The escape trajectory and abort trajectory are assumed to be co-planar with the same rotational sense. Abort action is initiated at distances between 1.5 and 10.0 Earth radii. Times from abort to arrival at perigee (an indication of flight time) vary from 15 minutes to 10,000 minutes. Parametric plots have been formed which relate the required velocity increment to abort radius, escape trajectory excess velocity, and time from abort to arrival at perigee. Re-entry velocities are, of course, less than parabolic in magnitude, and the re-entry flight path angles are mainly between 5 and 7 degrees.

The problem of mission abort at less than Earth escape velocity has already received considerable treatment in the open literature[4]. For the EMPIRE mission, however, it is necessary to extend the abort capability to escape trajectories with a considerable excess velocity. Since the abort propulsion requirement will be governed by the extent of this capability, it was necessary to parametrically investigate the propulsive requirements for such aborts. In performing the study, it has been assumed that a single thrust application would be used, that the abort trajectory is co-planar with the escape trajectory and that the re-entry corridor is specified by an unperturbed perigee location. Abort trajectories are assumed to be elliptical, and with the same rotational sense as the escape trajectory. No attempt has been made to specify landing locations.

Figure 3 shows the relevant geometry for performing an abort maneuver from a hyperbolic (or parabolic) escape trajectory. The escape trajectory is defined by its perigee radius q_1, and hyperbolic excess velocity V_h. The abort ellipse is defined by its perigee radius q_2, and major semi-axis a. The dashed abort ellipse is identical in geometry to the solid one but differs in that the flight path angle at abort is negative, resulting in a quicker return to Earth at the expense of additional abort velocity. The other two solutions, those involving retro-grade motion, have not been considered in view of their greater propulsive requirements. The velocity vector diagram defines the abort velocity requirement ΔV.

Figure 4 parametrically defines the abort propulsive requirements for one of the abort radii investigated. Time from abort to arrival at perigee is used as the abscissa in deference to its importance in the abort problem. The parameter is the hyperbolic excess velocity for the escape trajectory, V_h. The range shown is sufficient to apply to almost any currently envisioned space mission. An abort radius of 3.0 Earth radii is used for this figure. The following conclusions may be drawn.

1. Except for very small excess velocities and abort radii, a reduction in the time parameter always results in an increased velocity increment.
2. For moderate abort radii, a significant reduction in the time parameter may be obtained with a modest increase in velocity increment.
3. Regardless of excess velocity, the abort velocity requirement will ultimately increase with abort radius for a given value of the time parameter.

Item 1 suggests that there is a practical lower limit on the time from abort to arrival at perigee. Item 2 suggests, however, that a fairly low time to perigee (say 1000 minutes) can be achieved without an excessive abort propulsion requirement. For example, for a typical Symmetric class trajectory ($q_{Venus} = q_{Mars} = 1.3$) with an excess velocity of 7.1 km/sec, an abort velocity capability of 4 km/sec would permit a return to Earth in 1000 minutes, if abort action were taken at a distance of 3 Earth radii. At 5 Earth radii, this velocity increment would still permit return to Earth in about 5000 minutes or about [illegible] days.

An additional consideration which affects the abort problem is the radius at which thrusting along the escape trajectory is completed. Except for very low thrust trajectories (e.g., electrical propulsion) or very large escape velocities, it is anticipated that burning will be completed within less than 3 Earth radii[5].

Guidance and Navigation

The following "modus operandi" has been hypothesized as most appropriate for the EMPIRE missions.

The space navigator would be supplied an updated ephemeris by means of a suitable network of ground tracking stations. A self-contained system consisting of an on-board stellar tracker and associated auxiliary equipment will be available as the redundant system for the transit trajectory determination. This optical part of the system would also be employed as the principal guidance sensor for the planetary approach phase. An additional benefit would be the availability of vehicular attitude information as a redundant sensor to the attitude control system.

The mission may be expected to be initiated by the placement of the vehicle into a suitable, precomputed Earth parking orbit. Injection into the heliocentric transfer orbit will be initiated as the vehicular position and velocity vectors are appropriate to the specified initial conditions. As a consequence of the uncertainties at injection, due primarily to the errors in monitoring the thrusting phase and the errors in determining the parking orbit ephemeris, one may expect a significant variation of the nominal heliocentric orbit, hence, some mid-course maneuver would be required to redirect the vehicle onto a new nominal orbit so as to accomplish the prescribed terminal conditions, i.e., the intercept of the target planet. Also, as a consequence of the uncertainties in determining the actual vehicular heliocentric orbital parameters and the monitoring of the applied characteristic velocity during the mid-course maneuver, one would further expect some trajectory variations at the target planet. To reduce this error a planetary approach corrective maneuver for the EMPIRE missions has been included.

The mid-course correction required for each leg of the mission leads to a propulsive capability of approximately 200 feet per second. This provides a total mid-course correction ΔV of 600 feet per second and gives an approach corridor for the next planet intercept of approximately 6000 kilometers. To trim this to an acceptable value, a planetary approach guidance system is required. This system[6,7], which is based on an angle measurement between two stars and the target planet along with a diameter measurement on the planet disc, provides navigation information to guide the spacecraft into a suitably close impact to execute the desired turn angle at the planet. A conservative standard deviation optical sensor accuracy of σ = .0002 radians for the early 1970 time period leads to a requirement for the following incremental velocity corrections:

Mars	ΔV	=	330 fps
Venus	ΔV	=	650 fps
Earth	ΔV	=	360 fps
Total	ΔV	=	1340 fps

Coupled with the mid-course corrections a total capability to provide velocity corrections of approximately 2000 fps, or 0.61 km/sec, has been included in the EMPIRE spacecraft design.

An examination of the several system components required for the guidance and control functions for the EMPIRE missions was conducted. In particular, the subsystem reliability requirements have been derived and an estimate of the present day reliability standards for each subsystem determined.

Assuming a subsystem reliability, R, of 0.95, the necessary mean-time to failure was calculated using the well known relationship

$$R = e^{-\frac{\text{operating time } (t)}{\text{mean time to failure (MTF)}}} \qquad (1)$$

An added measure of reliability may be achieved through the use of backup systems either by allowing complementary systems to back each other up or by multiple systems. In that event the reliability is expressed as

$$R = 1 - \left(1 - e^{-\frac{t}{MTF}}\right)^n \qquad (2)$$

where n is the number of backups.

The following components will form the principal guidance subsystems for the EMPIRE mission.

Attitude Control Package - Consisting of rate and displacement gyro sensors, the subsystem will provide the required attitude signals during the thrusting phases of the injection, mid-course and planetary approach maneuvers as well as the re-entry phase. Since each of the two forms of sensors may be required to provide an alternate output (rate of displacement) in addition to its principal function, the MTF of the entire attitude control package was increased by a factor of two.

Star Tracker and Computer - This subsystem will operate in two principal modes of operation. Basically the star tracker and computer will provide the space navigator the capability to establish the vehicular ephemeris during the heliocentric transfer and the planetary approach maneuvers.

Sun and Planet (Earth) Tracker - These sensors provide the orientation data required to stablize the omni-directional antenna in the Earth-vehicle communication link.

A summary of the reliability estimates of the several subsystems is contained in Table 2.

Earth Return

The final payload affecting system design is the re-entry vehicle. A detailed study of this subsystem has been made for the flyby trajectories passing both Mars and Venus with minimum energy requirements for the 1970-72 time period. These results are not directly related to a particular interplanetary mission, but apply equally to Earth re-entry from all missions where the entry velocities range from 11.2 to 16 km/sec.

Generally, the re-entry velocity has been treated as a variable with the result that heat protection and propulsion requirements may be estimated throughout the velocity range of interest for interplanetary missions.

As shown in Figure 5, three basic types of vehicles Apollo-Type, High L/D (Lift Drag), and Drag Brake have been considered for re-entering the Earth's atmosphere at initial velocities well above the Earth escape velocity. These vehicles were sized for a crew of six. The nominal re-entry velocities for the EMPIRE Crocco and Symmetric orbits are 13.5 km/sec and 15.8 km/sec, respectively for a nonrotating Earth. At these velocities, radiative aerodynamic heating from ionized gas just ahead of the vehicle is present in addition to the convective heating for which Mercury, Gemini, and Apollo are designed. Radiative heating is minimized if vehicles are more pointed, whereas convective heating requires blunting of the vehicle.

An Apollo-Type configuration would enter with its edge tilted in the direction of travel for two reasons. The first reason is that of reducing the radiative heating at the expense of some increase in convective heating. The other reason for the upward pitched attitude of Apollo is that of creating lift through an offset center-of-gravity. This is necessary to achieve a satisfactory re-entry corridor depth.

A High L/D vehicle would open up the re-entry corridor considerably. Again, it would have a pointed nose to minimize the radiative heating. As was the case with Apollo, the High L/D vehicle would initially enter at a high angle-of-attack (α) until a certain g-limit is reached. After that, the vehicle would gradually pitch forward toward zero α maintaining this constant g-limit. Subsequently, it would maneuver to select a suitable landing site. This modulated lifting program would enable re-entry at high velocities with greater certainty of survival, provided the crew has been preconditioned to one g or so for some time during the last few weeks of interplanetary flight. A 10 g maximum deceleration is experienced for approximately one minute during the re-entry maneuvers investigated in this study.

An entirely different re-entry concept is that of the Drag Brake system which modulates its drag near the overshoot corridor to maintain a constant g-load and to maintain control. Also, distinctly different is the concept that the surface of the Drag Brake reradiates heat almost as rapidly as it receives it, whereas the other two vehicles dissipate the heat through ablation of heat shield material. One difficulty with the Drag Brake configuration is that the highest heating is taking place on the face of the Apollo requiring a large heat shield there. Pointing the body and trimming back the brake would minimize the heating, but this would mean higher brake weight to maintain the same ballistic coefficient. In addition, there are the problems of effecting a satisfactory heat reradiation at these high velocities and the difficulty of controlling the flight path angle with sufficient accuracy to avoid dipping too deeply into the atmosphere or, alternately, that of skipping out when not desired.

Minimum system weights were found to exist for a combination of aerodynamic and retro-rocket braking. Consequently, retro-thrusting is used initially to decelerate from higher re-entry velocities. The minimum total weight, including the tradeoff of heat shield and retro unit is determined for both the Apollo and the High L/D vehicle. The Drag Brake was analyzed for complete aerodynamic braking along the overshoot boundary without a deliberate skipout maneuver.

The higher L/D of the aerodynamic lifting vehicle offers advantages in entry corridor depth over the Apollo-Type and Drag Brake vehicles. Figure 6 shows the corridor for Earth re-entry as a function of the entry velocity. The High L/D vehicle has L/D $\cong$ 2.0, the Apollo-Type has L/D $\cong$0.5, and the Drag Brake has L/D $\cong$ 0.3. It is evident that the High L/D vehicle provides a corridor with much less stringent constraints on terminal approach guidance related to altitude and attitude errors.

The difficulty of sensing and maintaining a fraction of a degree angle-of-attack required to control the Drag Brake configuration, coupled with the problems of reradiative materials for the skirt, propulsion to prevent skipout, and lack of terminal maneuver capability tended to eliminate this approach. The Apollo-Type system with its deeper entry corridor and roll modulation with offset center-of-gravity gives a better system approach, but the requirement for maintaining precise control over angle-of-attack, and the smaller landing site selection capability tended to discriminate against this approach. In the final analysis the High L/D vehicle with deepest entry corridor, widest latitude in site selection and conventional landing capability appeared to provide the best approach although all three vehicles are competitive in weight requirements. For this reason, the EMPIRE Earth Re-entry Vehicle used the High L/D configuration.

Figure 7 shows the results of retro-propulsion and aerodynamic braking tradeoff studies. It is evident that for this study and the systems proposed for Earth return a combination of propulsion and aerodynamic braking provides a minimum weight for both the Apollo and High L/D vehicles. The minimum point on the weight versus entry velocity for pure aerodynamic braking shows a requirement to retro down to 15.8 km/sec for the Apollo-Type and to 13 km/sec for the High L/D. In the design point vehicle this gives a ΔV = 15.8 - 13.0 or 2.8 km/sec for the propulsion system. A liquid propulsion system with storable propellants was employed for this maneuver.

A summary comparison of the three vehicles is indicated in Table 3.

The design vehicle for this study is shown in its pitched up entry mode in Figure 8. The small nose radius tends to minimize total heating (radiative plus convective) for the design entry mode. It should be noted that a detailed discussion of the entry studies may be found in the proceedings of the Symposium on the Exploration of Mars[8], where recommendations for additional study and experimentation are given.

Spacecraft Design

General

Conceptual design of the EMPIRE spacecraft for the dual planet flyby mission was accomplished for several types of propulsion systems and for the Crocco and Symmetric missions. The design criteria were based on mission constraints, environmental effects, and crew requirements with subsystem choice based on reliability, availability, and weight.

The main mission constraints are the time and energy: 404 days for the Crocco Mission and 631 days for the Symmetric Mission with energy based on velocity increments previously discussed.

The design choice for the detailed analysis became the nuclear injected symmetric spacecraft based on the constraints of reliability, availability, weight in Earth orbit required, and cost. The spectrum of conceptual designs leading to this decision were all based on the same design criteria exclusive of mission energy requirements. The major factors were environmental and crew requirements. The environment included meteoroid protection based on estimates of near Earth impact rates and theoretical damage mechanisms[9]. Also, all systems were capable of operation at zero gravity, although the spacecraft would have provision for rotation to provide 0.3 g for the long term crew exposure. The space vacuum effects on materials and the thermal control of the spacecraft to provide reasonable temperatures were investigated. The crew requirements and life support considerations, both physiological and psychological, were included. The daily requirements per man included O_2 - 1.8 lb/day, H_2O - Food and Drink 4.84 lb/day, H_2O - other 20.0 lb/day. Wastes produced per man included: Urine - 3.08 lb/day, CO_2 - 2.25 lb/day, H_2O - 2.20 lb/day[10].

An analysis was performed on the crew duty cycle and crew size and it was found that on missions in excess of 2 to 4 weeks a minimum crew of five persons was required to perform the required tasks on a regular basis without loss of efficiency. In order to insure adequate allowance for all functions and include emergency capability a minimum crew of six was established. The crew and duties are specified in Table 4 and a typical activity schedule covering 24 hours is shown in Table 5, where each letter represents a different member of the crew. In light of the importance of crew number to the overall mission and spacecraft design, it is extremely important that detailed investigation in the area of interplanetary crew requirements be continued in depth. Presently discussed concepts have included from three to eight crew members for various Mars and Venus missions with a resultant wide variation in weights and systems required for otherwise similar missions. The conclusions from our study appear valid also for other missions. That is, the minimum crew for any extended mission should be <u>at least five persons</u> with a normal requirement of six astronauts. Landing on Mars, for example, would require five or six plus the landing crew if descent from Mars orbit is employed or the possibility exists that the return trip could be an impossible task for the crew remaining in the spacecraft.

The living volume includes 750 cubic feet of free volume per man with only 50 cubic feet required for short periods of a few days in the radiation "storm cellar." The allowable radiation dose was taken as 200 Rem for the entire mission with the most stringent requirement a restricted short term dose from an extreme solar flare. Also, this total dose criterion provides a high probability that the trip will not leave the astronauts without allowance for additional exposure from normal lifetime sources before and after the mission. The shield includes 50 cm (46.5 gm/cm^2) of polyethylene plus the additional tank and vehicle structure shielding of approximately 20 gm/cm^2. Optimization of material in the shield (i.e., loading with material having a high absorption cross-section for cascade neutrons) was not performed in this study. Such an approach should allow some reduction in weight for a given exposure. The total mission dose comes from the trapped radiation, solar flare protons, cosmic radiation, and nuclear reactors in the propulsion and power systems. For this spacecraft with a SNAP-8 auxiliary power unit the Van Allen passage and reactor dose plus cosmic rays gives approximately 40 Rem. A single flare of the 10 May 1959 type would add 70 to 80 Rem with the shielding provided, and smaller, more numerous, 3 and 3^+ flares would deliver another 80 to 90 Rem for a total of 200 Rem during the Symmetric Mission based on estimates for 1970-72.

The communication system was taken as a minimum system for voice and digital data with a power requirement of 150 to 200 watts, redundant spacecraft antennas having 40 db gain, 4 kc information bandwidth, and operating in the S-band frequency range. More elaborate systems could be used with only nominal increase in total weight of the spacecraft.

Attitude control is provided by a reaction control propulsion system. This is used for spinning and despinning and also pointing the spacecraft prior to propulsive velocity corrections. The estimated weight of the system is 1500 pounds.

Auxiliary power would be provided by a SNAP-8 nuclear reactor power system with a backup unit for the required mission lifetime. A preferred system could be a solar collector with a turbo electric generator on which maintenance could be performed. The 15 KW system under development by Sunstrand Corporation would meet the mission requirements.

The propulsion studies included nuclear engines for injection from Earth orbit and storable chemical systems for injection and/or subsequent maneuvers. Two ranges of chemical system performance were considered with specific impulses of I_{sp} = 410 sec and I_{sp} = 500 sec. The extrapolated availability of high specific impulse tri-propellant or other storable chemicals led to a choice of I_{sp} = 410 sec for the mission spacecraft. Solid propulsion was eliminated due to the lower performance and problems of long term protection against meteoroids as well as capability for test and checkout during the mission.

The subsystem weights are summarized for both missions in Table 6.

Spacecraft Configuration

The design point configuration developed during the vehicle weight synthesis is shown in Figure 9. The in-board profile depicts the interior arrangement of the trajectory spacecraft. The Earth re-entry vehicle is oriented so that injection and separation forces will be received by the crew in the optimum manner (chest to back). The entry retro-pack is a two-stage chemical propellant system which also functions as the abort system in the event of a failure within a 3 Earth radii range. The command center/radiation shelter is located centrally from the living modules, which are extended through telescoping cylinders. Protection for the crew from radiation is provided by 50 cm of polyethylene which completely surrounds the command center. An area set aside for an inertial platform and weightless experimentation is located just aft of the command center. Four chemical propellant tanks which provide trajectory correction fuel are placed around the command center.

This entire core is surrounded by eight second-stage-injection tanks. These tanks are retained throughout the mission to provide meteoroid protection for the propellant systems contained within the core.

The top view depicts the vehicle before and during first stage injection. First stage injection propellant is contained in seven cylindrical tanks. The center tank also serves as structural support for the injection engine assembly. The living modules are stowed, as they will be during Earth launch and interplanetary injection. The antennas and SNAP-8 power supplies are also stowed until the mission trajectory is attained.

Figure 10 shows the staging sequence employed from injection to Earth entry. The vehicle is shown during first stage injection with propellant from the seven tanks being fed into the main engine. The crew is located in the re-entry (abort) vehicle. At the completion of first stage burning, the six peripheral tanks are jettisoned, leaving the center tank as structural carry-through between the injection engine and the remainder of the vehicle.

During second stage of the injection, the propellant in the second stage tanks is fed into the main engine. At the completion of second stage burning, the main engine and first stage center tank are jettisoned.

After the transfer trajectory has been established, one SNAP-8 power supply and the two living modules are deployed, and spacecraft rotation of approximately 3 rpm begins. Mid-course corrections and planetary approach corrections are accomplished with small systems fed from four tanks located around the radiation shelter. A second SNAP-8 is carried to insure adequate power for the entire mission. In the event of a malfunction, the first SNAP-8 is jettisoned with its attendant radiation problem and the second deployed.

After Earth approach corrections are complete, the crew enters the re-entry vehicle and the re-entry system is separated by small posigrade solid units. The entry retro-pack decelerates the re-entry vehicle to a velocity at which full aerodynamic entry is feasible.

Injection Engine Burning Time

Two of the most important parameters in nuclear engine performance are thrust and burning time. In the time remaining for EMPIRE mission development before 1970, nuclear engine thrusts above approximately 50,000 pounds appear to be questionable, therefore the burning time was calculated based upon that value to give 2873 seconds for the Symmetric injection ΔV of 5.3 km/sec. This exceeds current estimates of burning time, which are limited by reactor material and design considerations. The effect on burning time and spacecraft weight with variations in nuclear engine thrust were calculated. The data are presented in graphical form in Figure 11. The curve shows that the nuclear engine burning time decreases with increasing thrust, while the overall increase in spacecraft weight for the wide range in thrust is not severe.

If one were free to select an optimum thrust and burning time the design mission could be best performed in light of the extrapolated technological capabilities for 1970 with a nuclear rocket engine having a thrust of 200,000 pounds capable of burning for approximately 800 seconds. For the 1972 dual planet flyby mission an increase to approximately 1000 seconds burning time would be required. Later missions up to 1983, when Mars is in an even less favorable position at opposition, require much larger total impulse.

Configuration Summary for Mission Spacecraft

To provide a comparison of the possible spacecrafts for performing dual planet flyby missions envelope drawings and some of the major parameters are shown in Figure 12. The four design vehicles which were investigated in detail are shown in proportion to an Atlas-Mercury system as it would appear ready for launch. The enormity of the Nuclear Crocco spacecraft is evident and is primarily due to the large amount of low density hydrogen fuel required (4.4 lbs/cu.ft.). The chemical propulsion systems having more dense fuels (approximately 60 lbs/cu.ft.) occupy smaller volumes for the same total weight.

Note that the Nuclear Symmetric spacecraft with a weight, W_o, of 375,000 lbs., a thrust, F, of 50,000 lbs., and an injection burning time, t_b, of 2800 seconds would represent almost two Saturn C-5 payloads in Earth orbit. The Chemical Symmetric vehicle (I_{sp} = 410) would be about four C-5 payloads and the Nuclear Crocco spacecraft would be about ten to twelve C-5 payloads.

Launch and Orbital Operations

An analysis was performed to derive success probability for the design mission in 1970 based on anticipated developments in Earth launch vehicles (ELV), orbital operations, and nuclear injection engines.

The minimum number of ELV's was established for each mission with the Saturn C-5, Nova and Super Nova boosters included in the analyses. The ELV's were defined by a nominal payload capability to the orbital altitude for interplanetary injection (300 km). The C-5 was assigned a 200,000 to 240,000 pound capability, the Nova capability was taken to be 500,000 pounds, and the Super Nova was taken at 700,000 pounds (here direct injection of the 152,400 pound Interplanetary Spacecraft Payload would be possible).

Reliability trends were extrapolated based on development programs consistent with present conditions and the requirement for the 1970 mission[11]. Orbital operations are a major part of the consideration and estimates were made for the 1970 time period based on recent orbital launch operations studies[12]. To summarize the results of this investigation, Figure 13 shows the projected ELV reliability trends where reliability indicated the probability of satisfactory boost performance in a single test. Figure 14 shows the projected orbital launch and operations success probabilities. Combination of the elements for the mission in 1970 gives the results in Figure 15 where the number of ELV's for successful mission accomplishment is shown on each curve allowing for no backup or redundant systems. Based only on these considerations the use of a single Super Nova is apparently the best route for high probability of a successful mission. Of course, this provides for liquid injection of the spacecraft into interplanetary transit and eliminates the low reliability factor for the nuclear rocket in 1970.

The addition of redundant nuclear engines to improve reliability and acceleration of the nuclear engine development program leads to the results shown in Figure 16. The gain in using more Saturn C-5's to provide backup for Earth launch is also reflected here. The use of four C-5's and two redundant nuclear rocket engines places the probability of success at 72 percent for the 1970 Symmetric Mission. Even with this approach the requirement for abort capability from all boost phases of the mission is evident.

Similar studies were performed for the Nova and also for the Symmetric Mission in 1972 with development work starting at the present time. The results of this analysis for 1972 is shown in Figure 17.

The conclusions reached from the foregoing are: Saturn C-5's are practical ELV's with the nuclear injection stage, an all chemical mission is possible although it would require extensive orbital operations, abort capability in all phases is required to provide adequate manned survival probability, backup ELV's are required to allow for sufficient mission success probability, redundant nuclear injection units are desirable for these early missions, and accelerated nuclear engine development is desirable.

Schedule and Funding

A detailed Development Plan was formulated for the 1970 launch date in order to bring into focus the critical development areas and to establish the supporting schedules for subsystem development. The general constraints included requirements to use Apollo hardware and technology where feasible, to launch at the Atlantic Missile Range, and to use hardware already under development where possible. The funding schedule should include all EMPIRE program costs except those research and development costs for nuclear rocket engines, nuclear auxiliary power supplies, Saturn ELV, Nova ELV, Apollo, and other equipment under planned development for the 1970 time schedule. Although it is necessary to expedite some of these developments, the ground rules have been followed and only the actual hardware and R&D costs for the EMPIRE spacecraft, subsystems, and production boosters have been included with the above mentioned program hardware costs for the 1970 mission.

The schedule for the design mission is shown in Figure 18. Note that some of the developments (Nuclear Rocket Engine and Nuclear Auxiliary Power System) were scheduled to start in January 1963. These could have started in July 1963 and still not required undue acceleration. It is now late and a crash development would be required. The design mission also calls for a Nova booster to eliminate the need for orbital operations and improve success probability. The Nova development could start in mid-1964 and still not pace the program if the F-1, M-1, and J-2 engines under present development for Saturn were employed. Also, note that the EMPIRE program is slightly accelerated in the early phases compared to a conventional development program with prototype subsystem hardware deliveries 1 to 2 years after initiation.

In keeping with the funding criteria and the detailed development plan summarized by the schedule of Figure 16, program costs have been estimated and a funding schedule is presented in Figure 19. The EMPIRE costs do not include the captive test model and four development Nova ELV's shown on the development schedule since they are considered to be a part of a separate development program and are shown only to demonstrate the subsystem development requirement. The cost estimate is based on the flight hardware costs plus supporting development and test operations exclusive of post injection operation costs for ground based support systems. The total cost is 12.6 billion dollars for a single mission in 1970. This estimate is based on a typical program funding distribution as outlined by Koelle.

Conclusions

Requirements to Accomplish the 1970-72 Launch

It would appear that even now a launch in July 1970 is within technological reach of a plausible development program--provided certain problem areas are attacked immediately. The most critical areas include nuclear rocket engines, auxiliary power supplies of the SNAP or solar turbo electric varieties, development of the Nova booster or C-5 Earth orbital operations capability, development of storable noncryogenic (or mild cryogenic) tri-propellants with I_{sp} ~ 410 seconds, re-entry materials and technology improvements, and immediate allocation of funds. Rapid developments would be essential to a successful launch in the 1970 window. Slippage beyond August of 1970 would not be acceptable without uprating of the system to meet a 1972 launch for a similar mission and for this reason the program would have very stringent delivery date requirements which must be successfully met.

The early definition of a suitable advanced nuclear rocket engine (at least 200,000 pounds thrust burning for 600 seconds or 50,000 pounds thrust burning for 3000 seconds) must be made and development should have started before July 1963 to avoid an accelerated development program. Immediate development of a high I_{sp} storable liquid propellant and oxidizer is also necessary to meet the EMPIRE requirements. For Nova, continued development of the F-1, M-1, and J-2 engines is called for with the chosen engines to be integrated into a booster system starting in mid-1964. Of course, the alternative of using Saturn C-5 with rendezvous, docking, assembly, and fueling orbital operations is also feasible. The areas of interest not specified as critical also require active development, but are represented by programmed developments or have apparent solutions within reach.

Use of Programmed Hardware

In keeping with a conservative approach, it would be desirable to use the Saturn C-5, Apollo Command Module, NERVA nuclear rocket engine, and other hardware under development. Much of this approach has been integrated into this EMPIRE study. However, use of the Apollo re-entry system, long term objective. This would leave the flyby mission objectives to be accomplished before the Mars landing by automatic or <u>unmanned</u> probes, orbiters, and flyby spacecraft. The proper conclusions can be reached only after the results of the integrated planning and mission study program become available.

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13. H. H. Koelle, Ed., <u>Handbook of Astronautical Engineering</u>, Table 1.12, p. 1-64, McGraw-Hill, 1961.

TABLE 1

TYPICAL EMPIRE MISSION CHARACTERISTICS

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SUMMARY OF SUBSYSTEM WEIGHTS

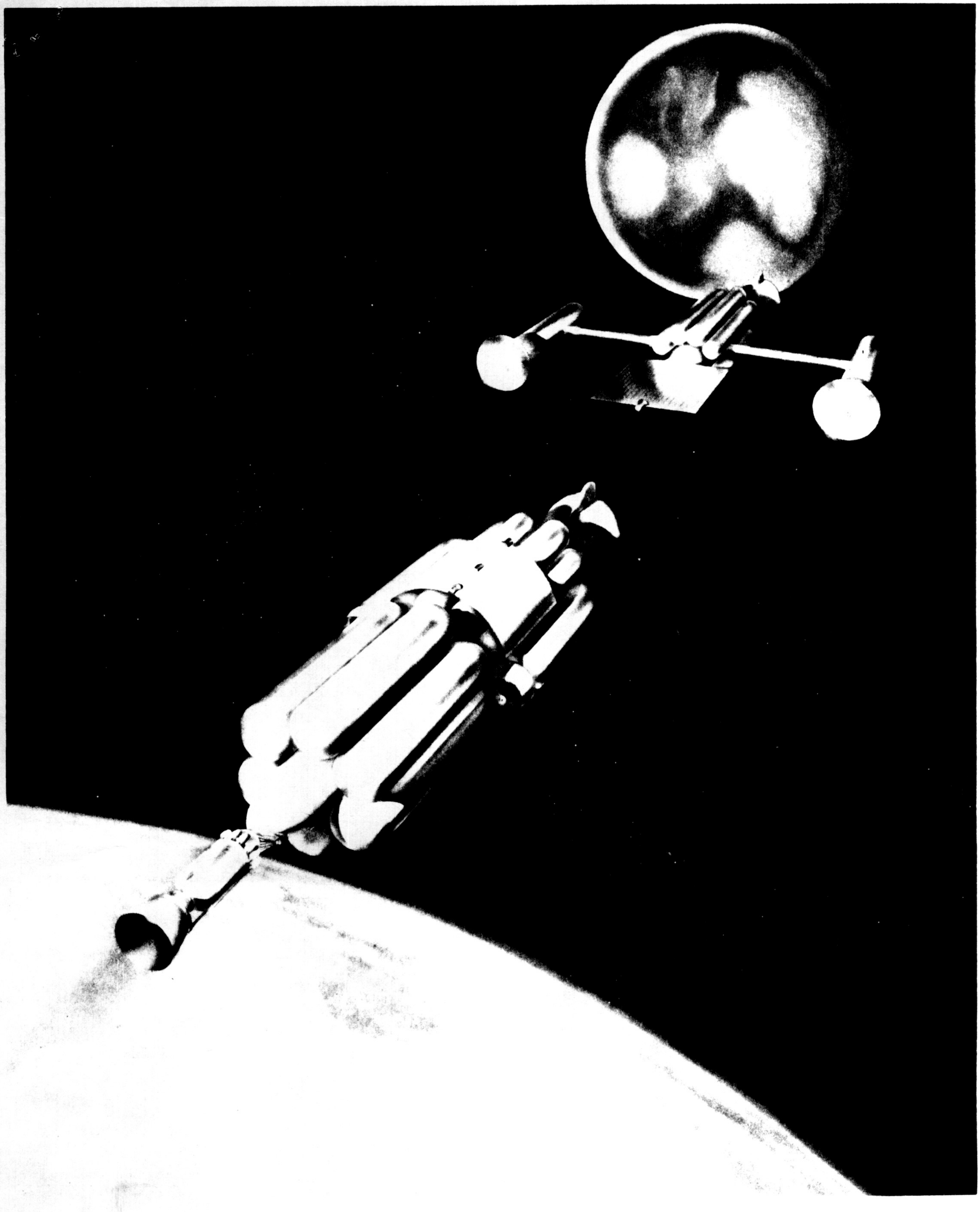
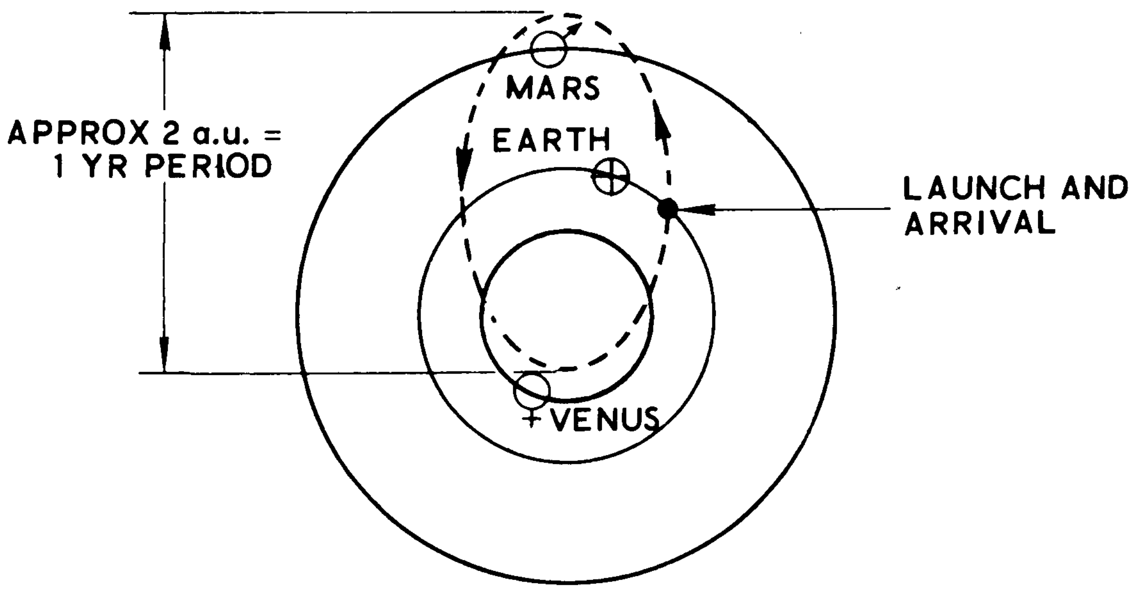
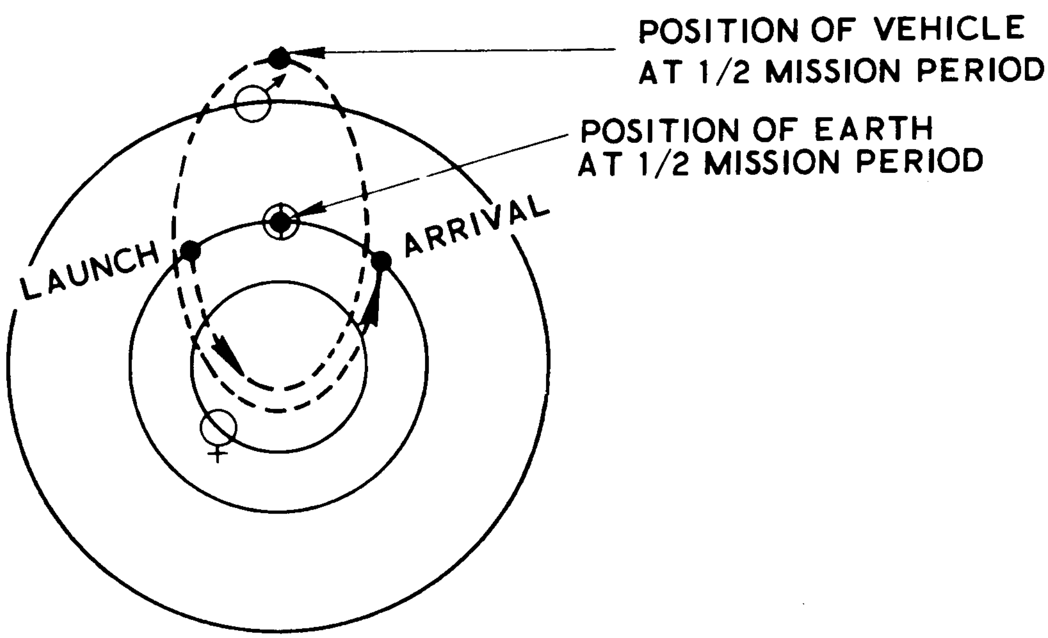


FIGURE 1. EMPIRE INTERPLANETARY SPACECRAFT



(a) UNPERTURBED NON-SYMMETRICAL (CROCCO) TRAJECTORY



(b) UNPERTURBED SYMMETRICAL TRAJECTORY

FIGURE 2. CROCCO AND SYMMETRIC TRAJECTORIES

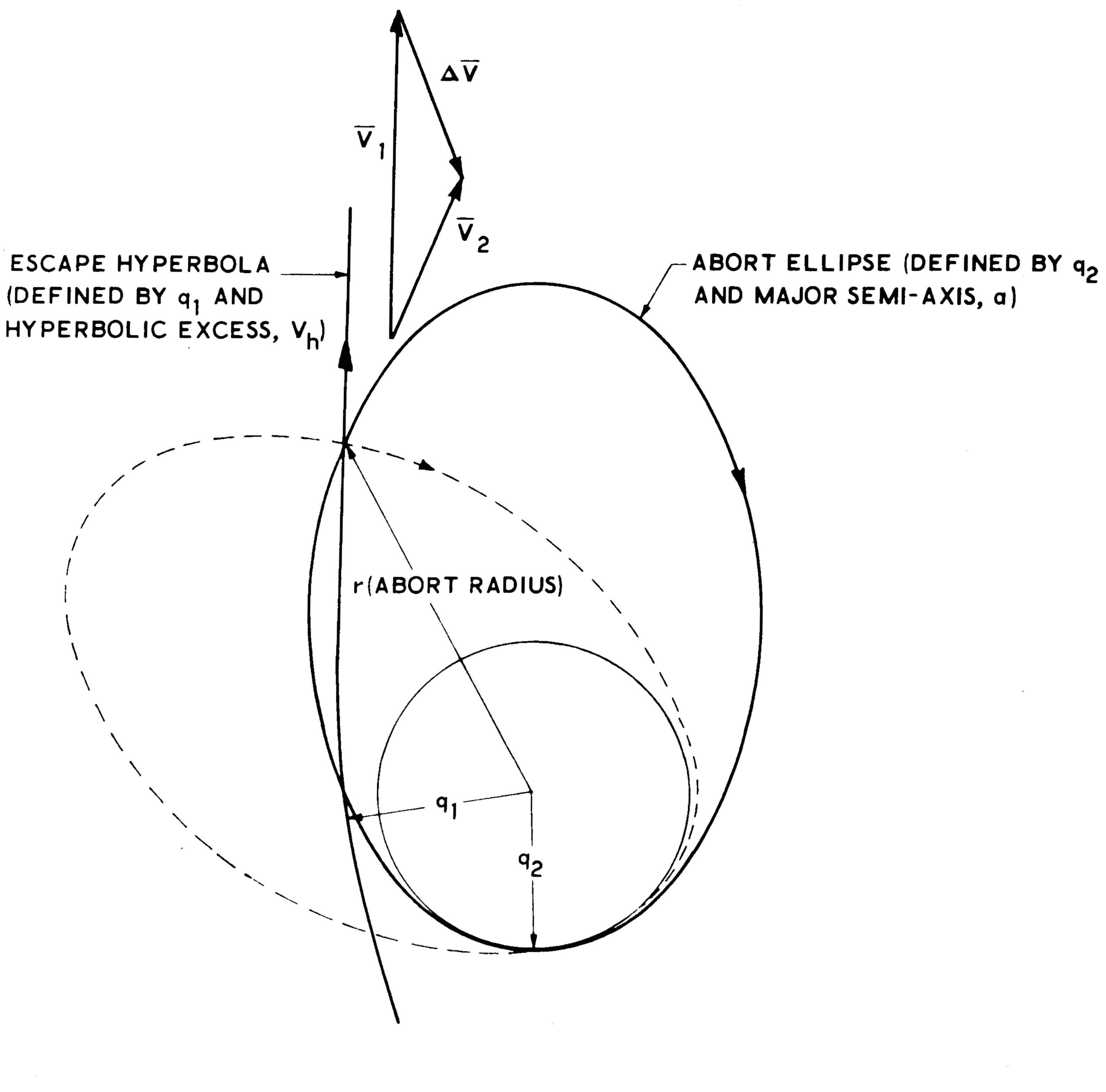


FIGURE 3. GEOCENTRIC ABORT MANEUVER GEOMETRY

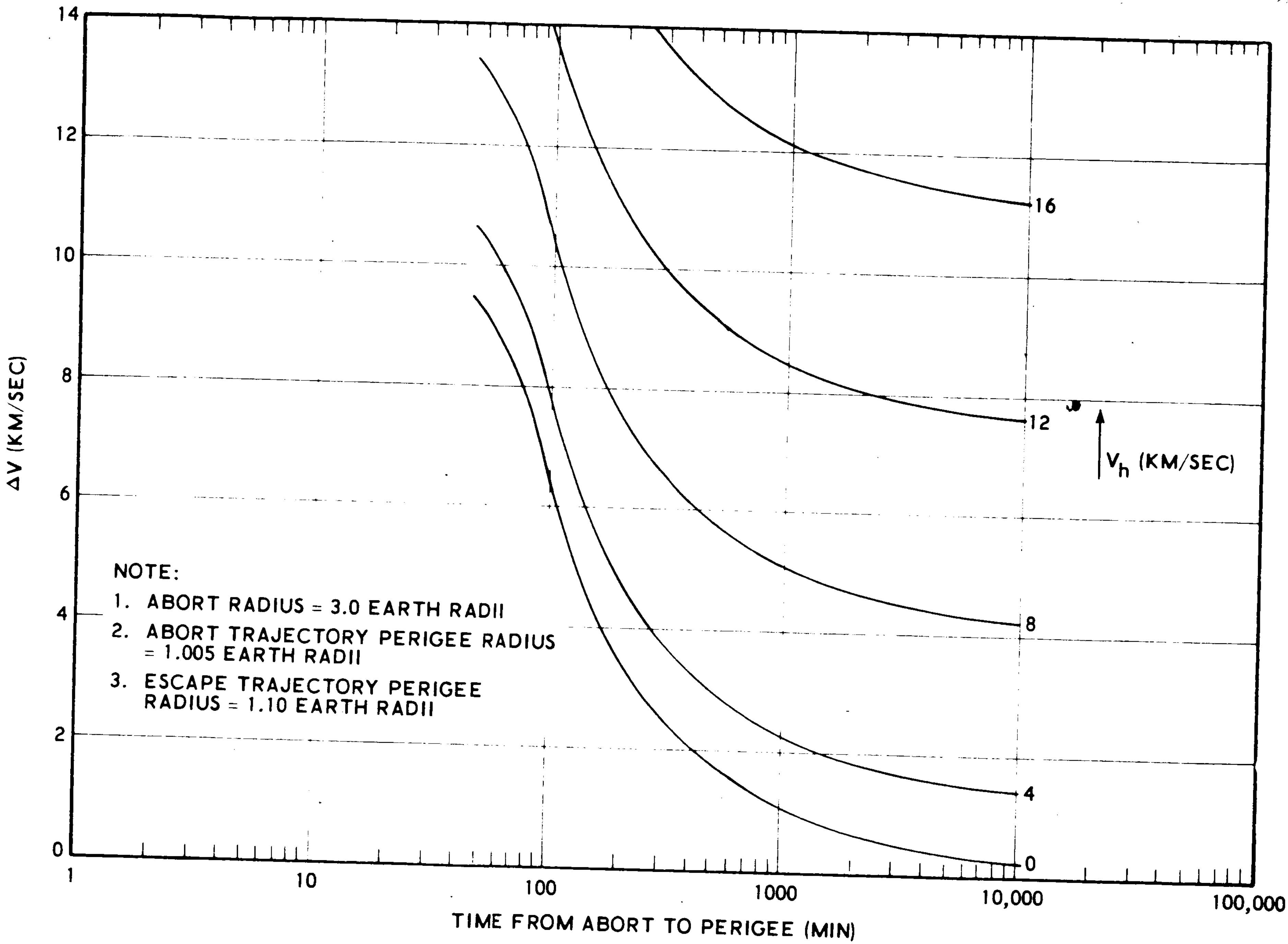


FIGURE 4. ABORT VELOCITY REQUIREMENT

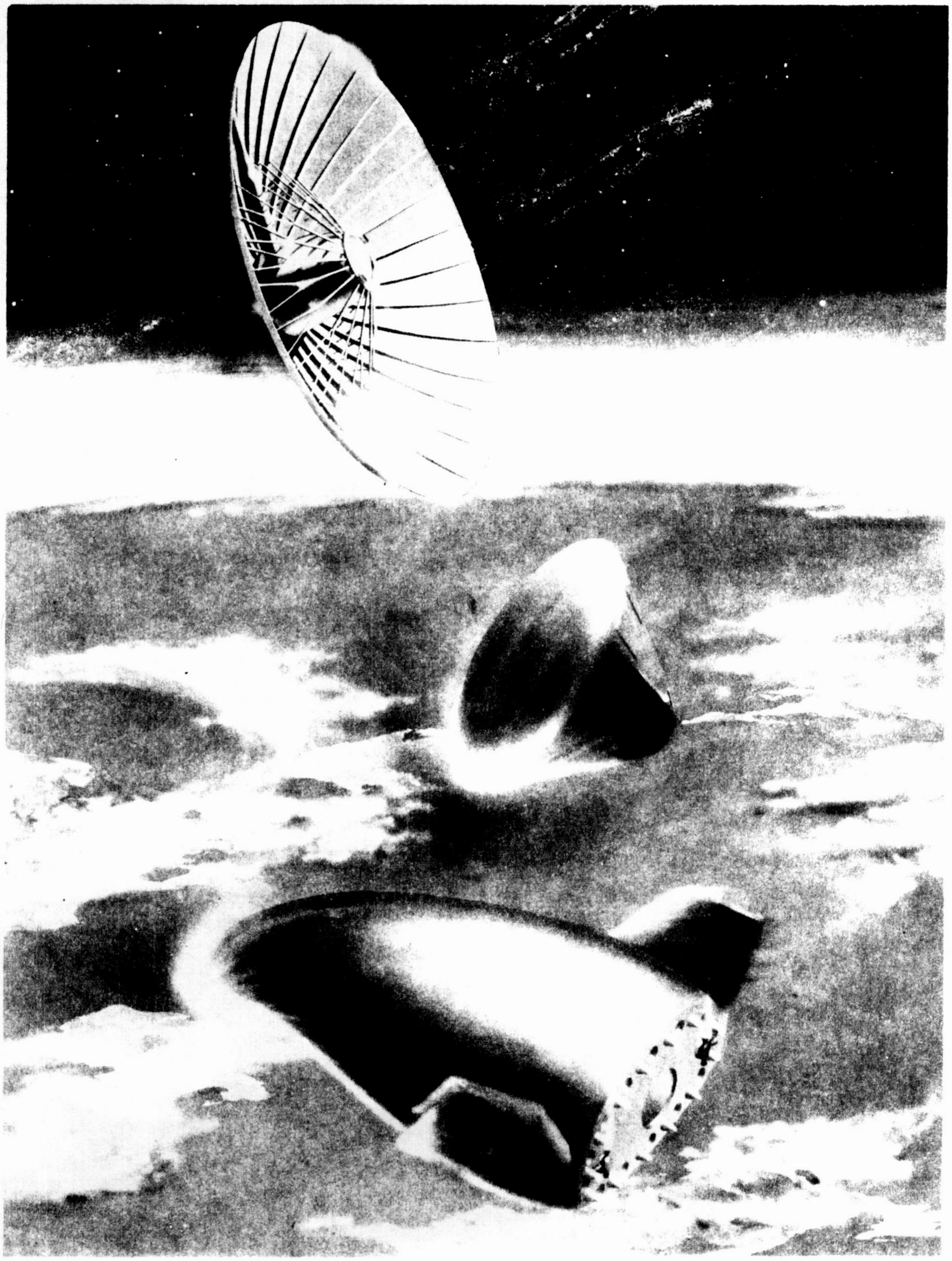


FIGURE 5. TYPES OF EARTH RE-ENTRY VEHICLES

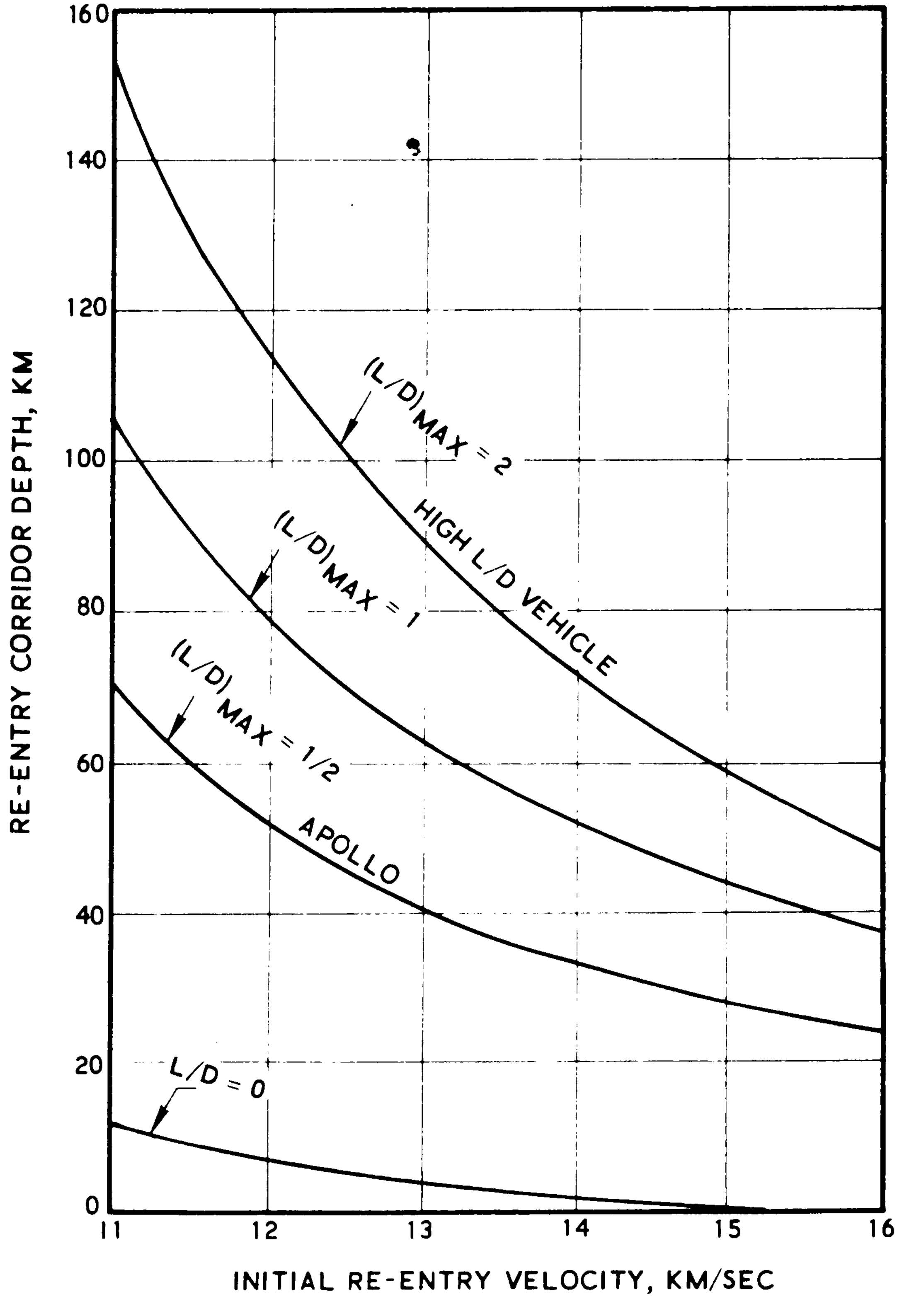


FIGURE 6. RE-ENTRY CORRIDORS – LIFT MODULATED, 10 g MAXIMUM

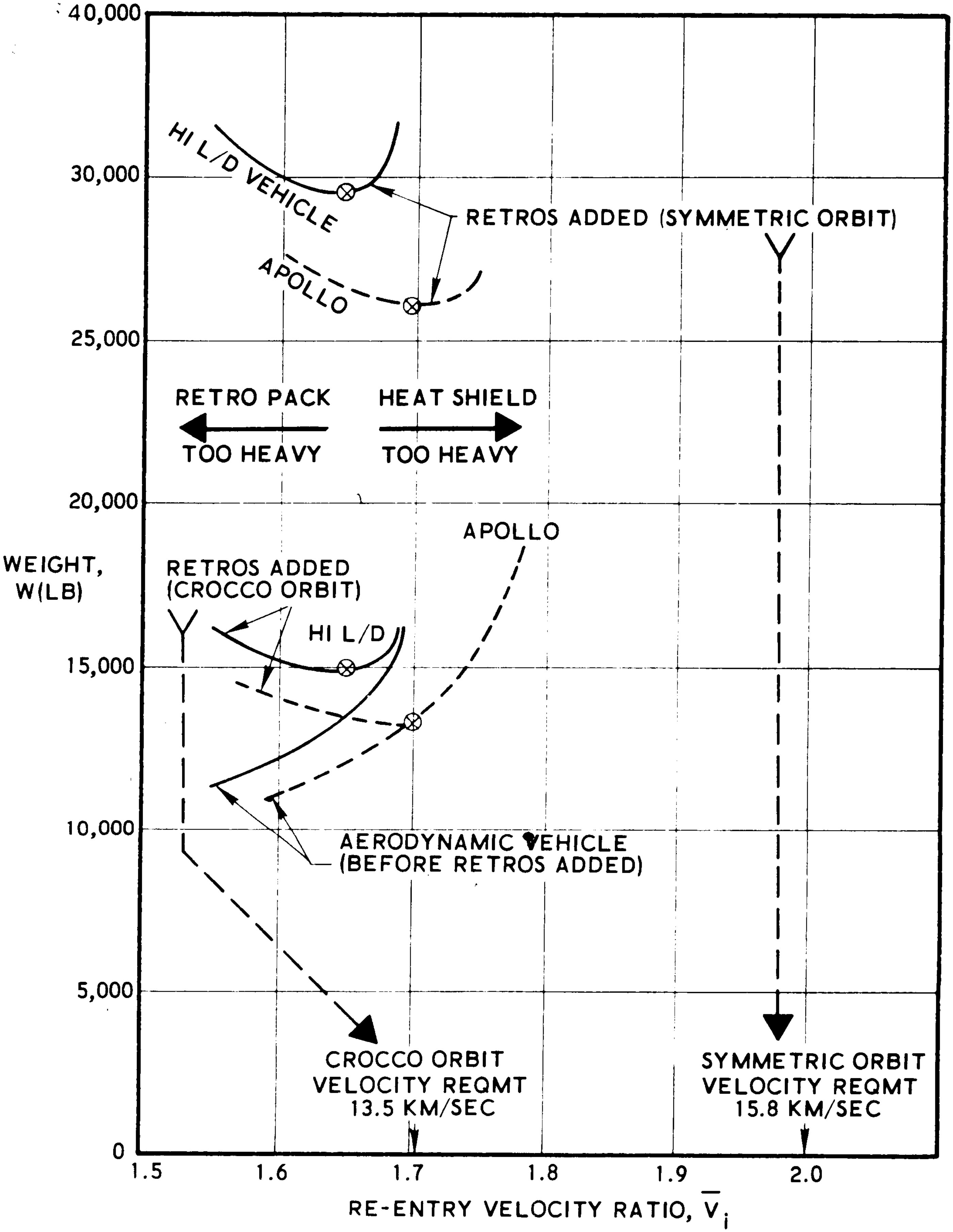


FIGURE 7. RETRO-ASSISTED RE-ENTRY VEHICLES

FIGURE 8. HIGH L/D RE-ENTRY VEHICLE

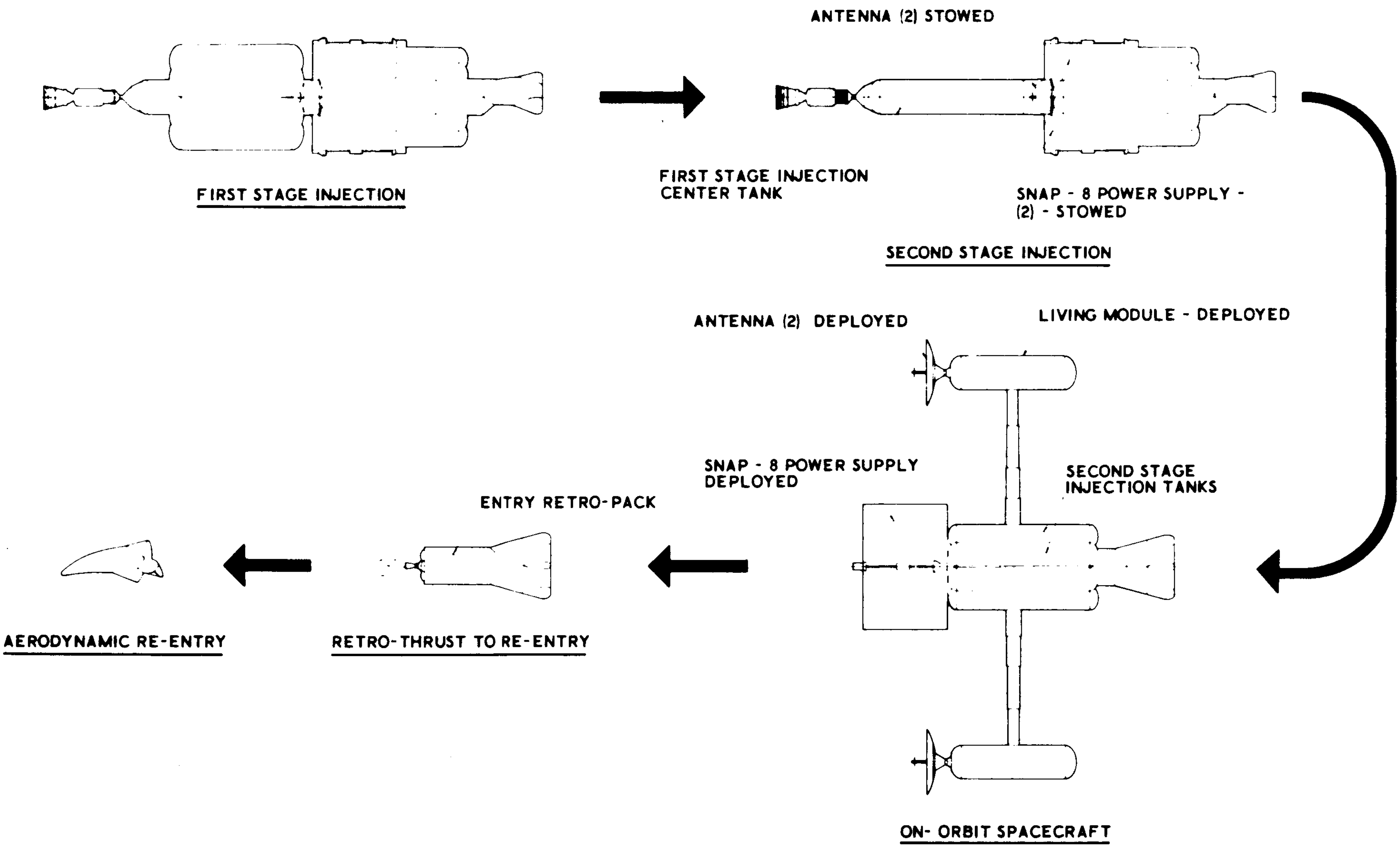


FIGURE 9. EMPIRE VEHICLE – SYMMETRIC TRAJECTORY – NUCLEAR INJECTION

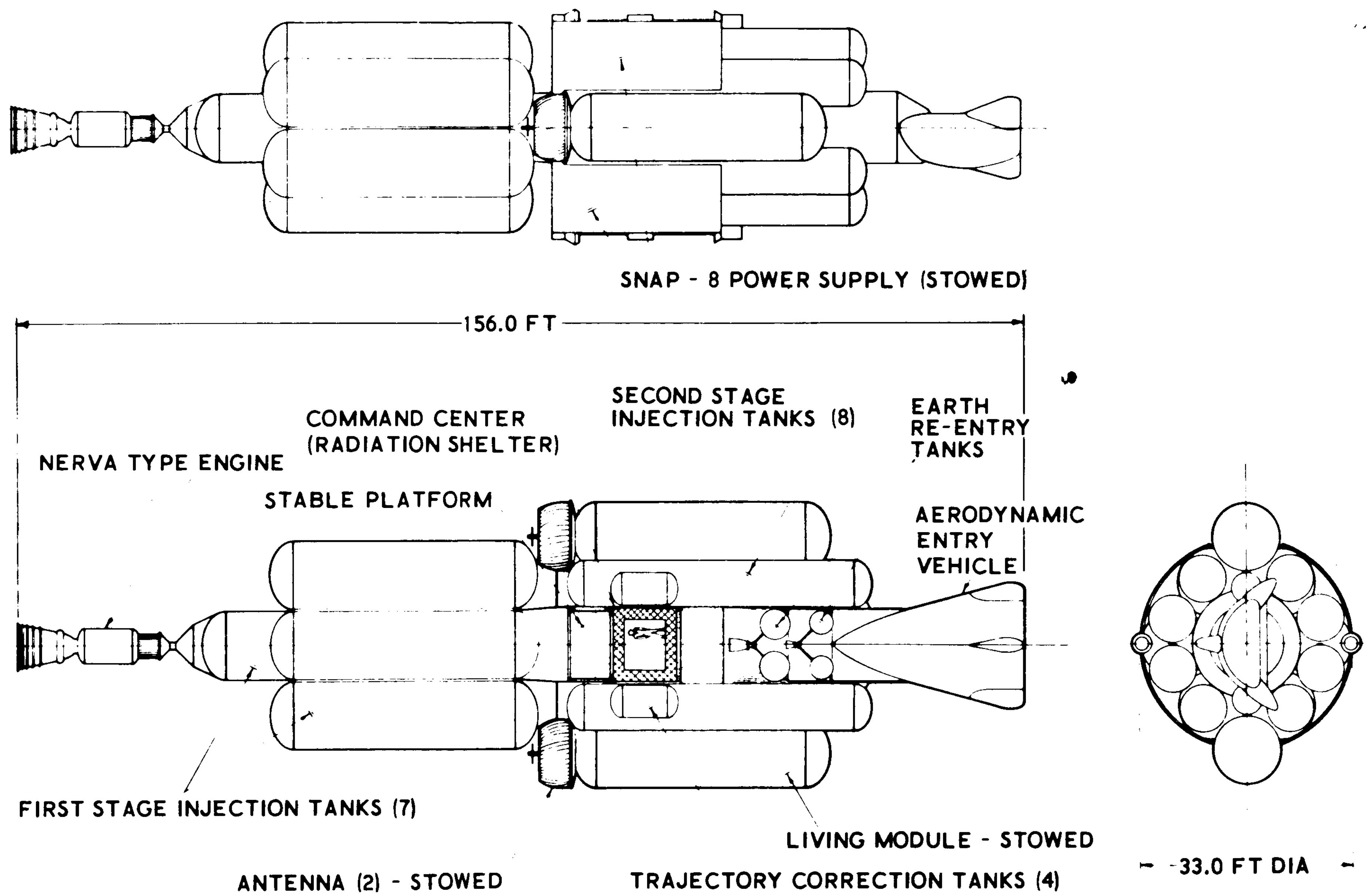


FIGURE 10. EMPIRE VEHICLE STAGING SEQUENCE – NUCLEAR SYMMETRIC

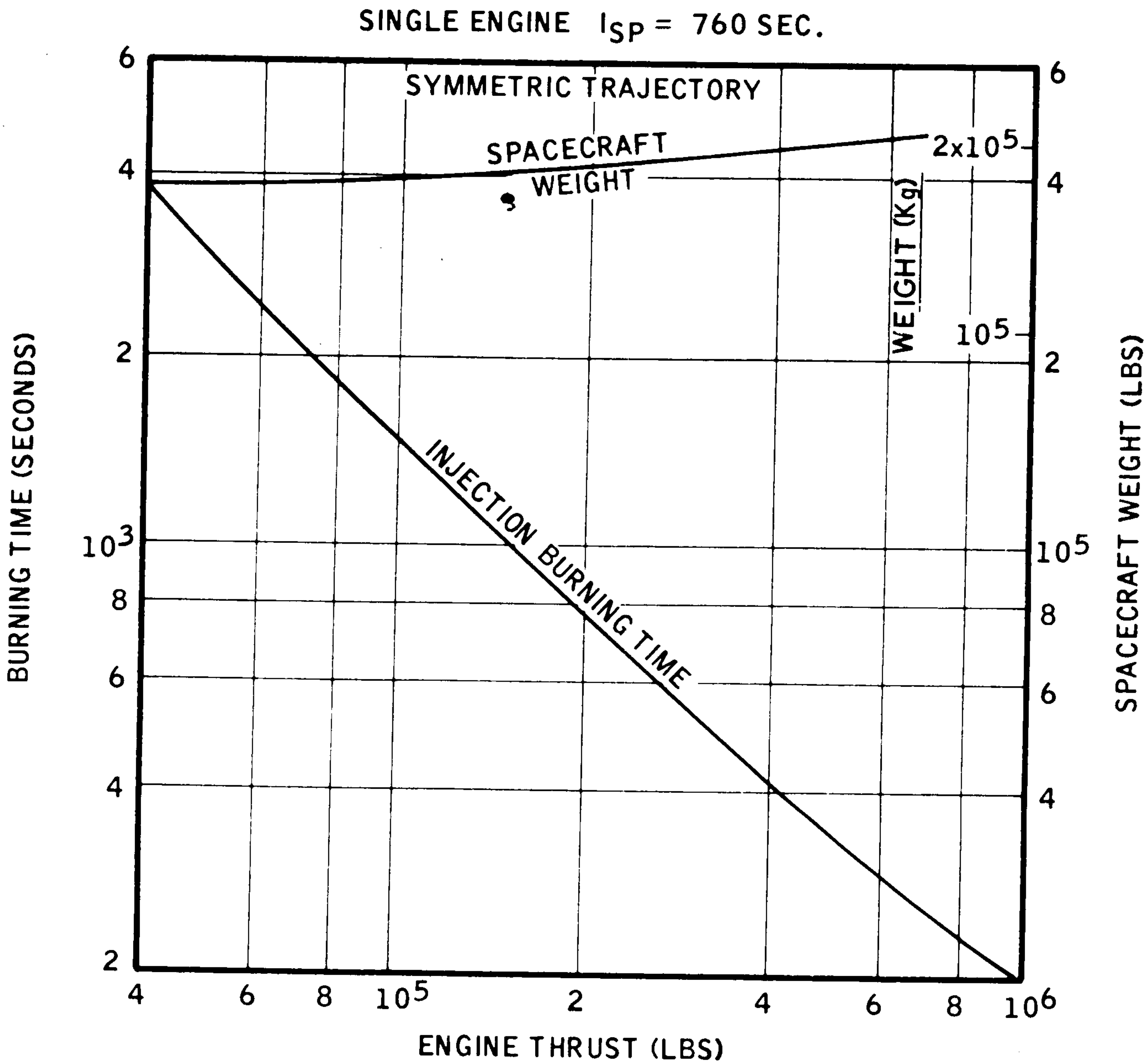


FIGURE 11. NUCLEAR ENGINE BURNING TIME AND SPACECRAFT WEIGHT VERSUS FUNCTIONS OF ENGINE THRUST

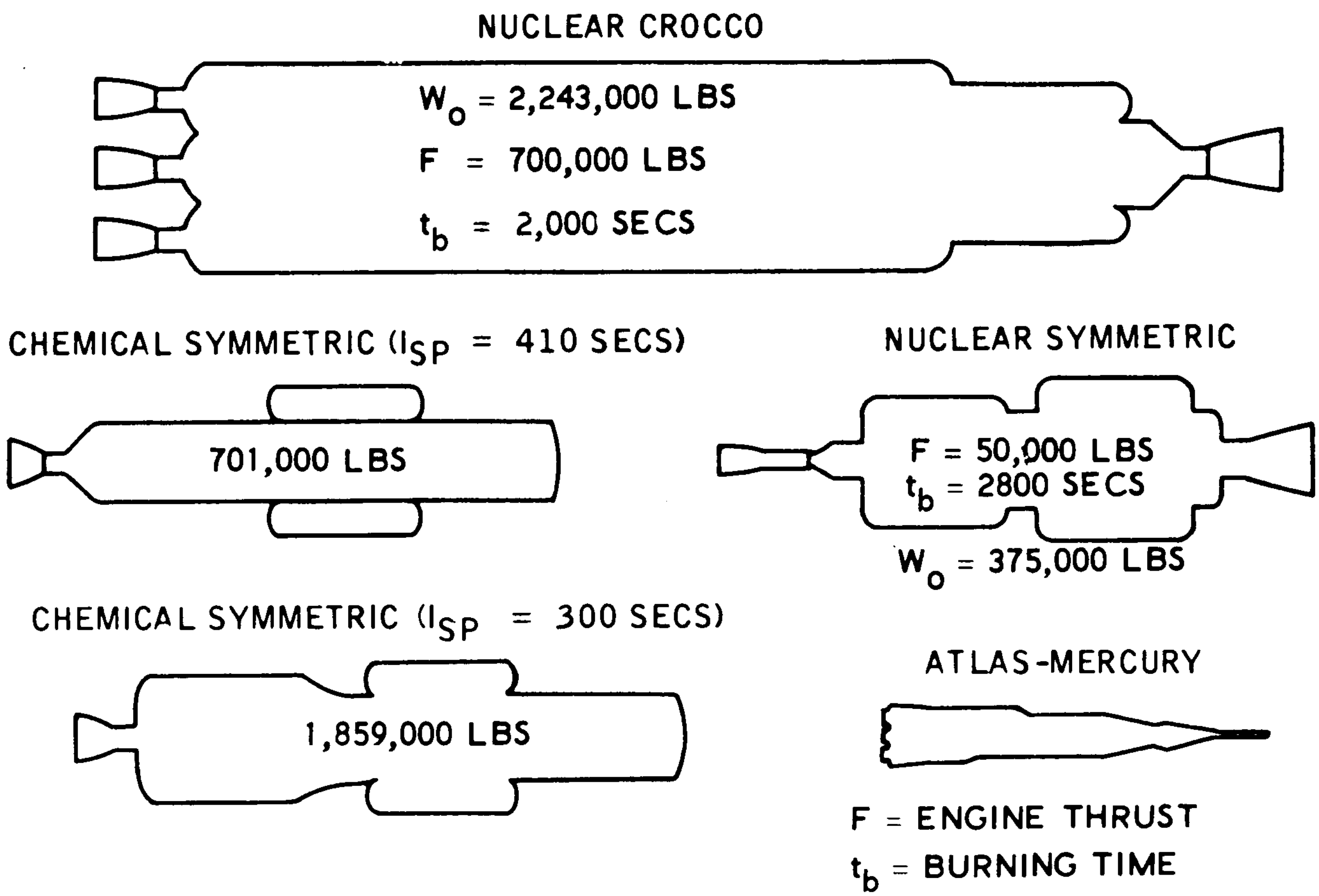


FIGURE 12. CONFIGURATION SUMMARY

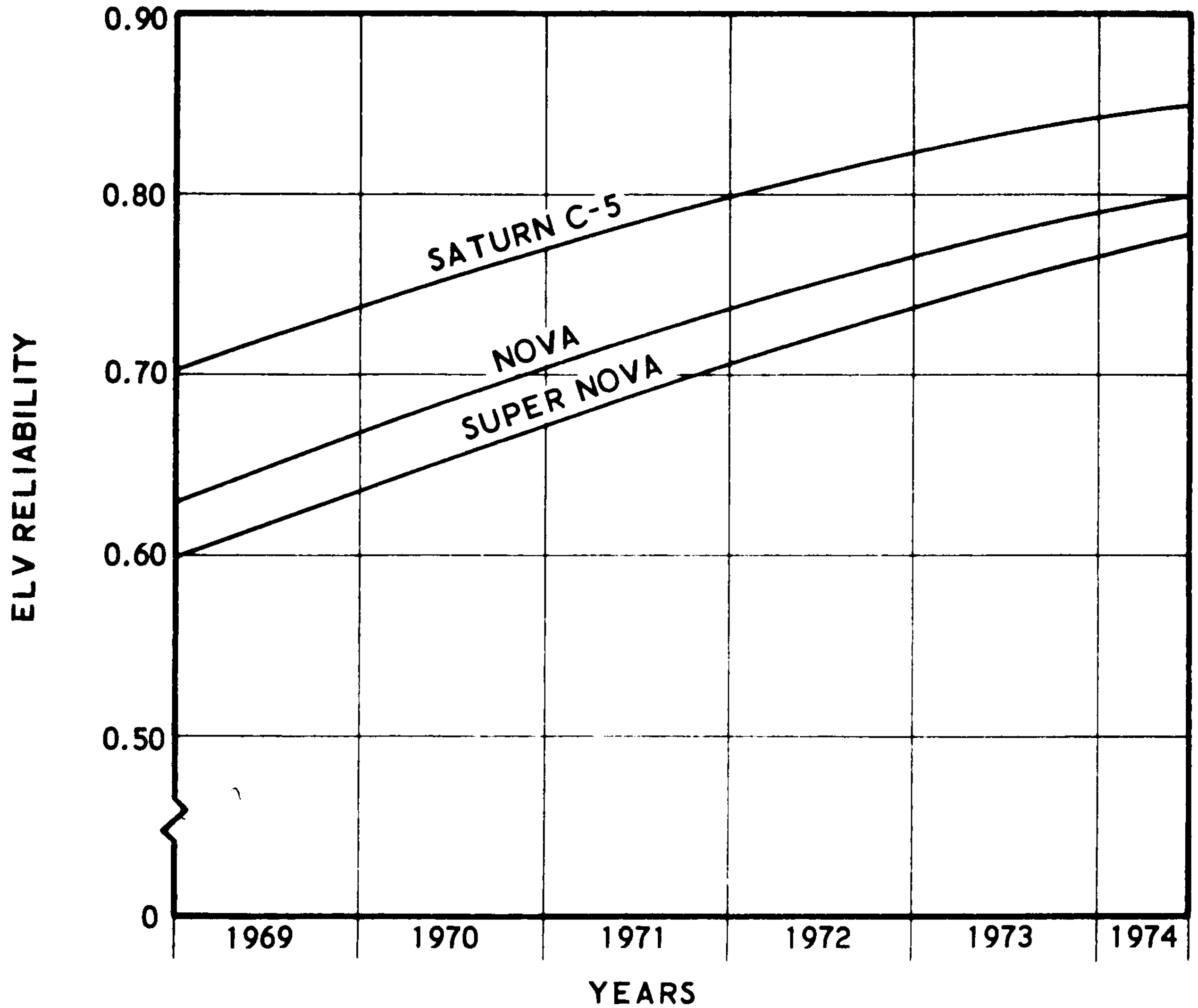


FIGURE 13. PROJECTED EARTH LAUNCH VEHICLE RELIABILITY

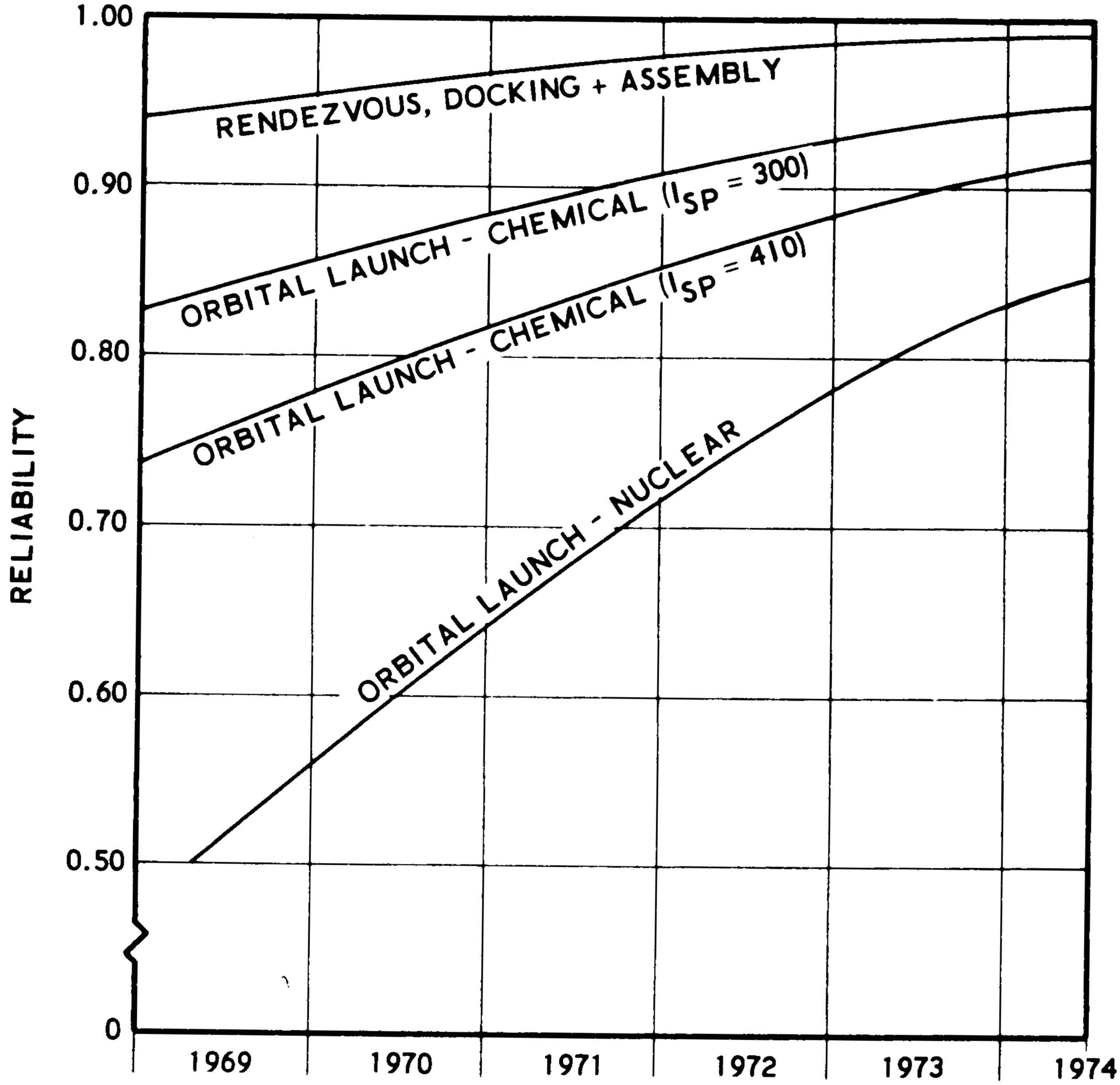


FIGURE 14. PROJECTED ORBITAL OPERATIONS RELIABILITY TRENDS

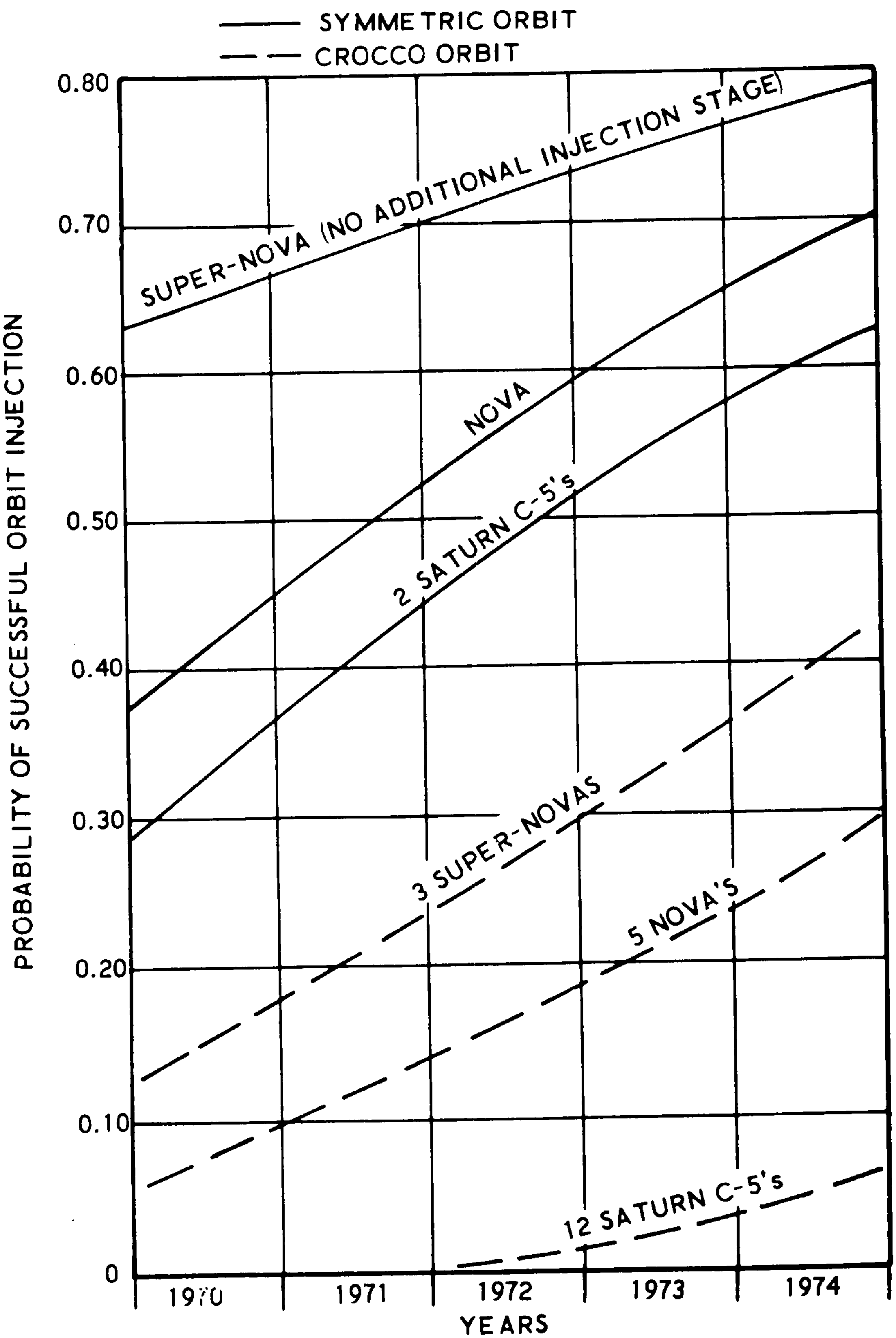


FIGURE 15. MISSION SUCCESS FOR NUCLEAR SYSTEMS

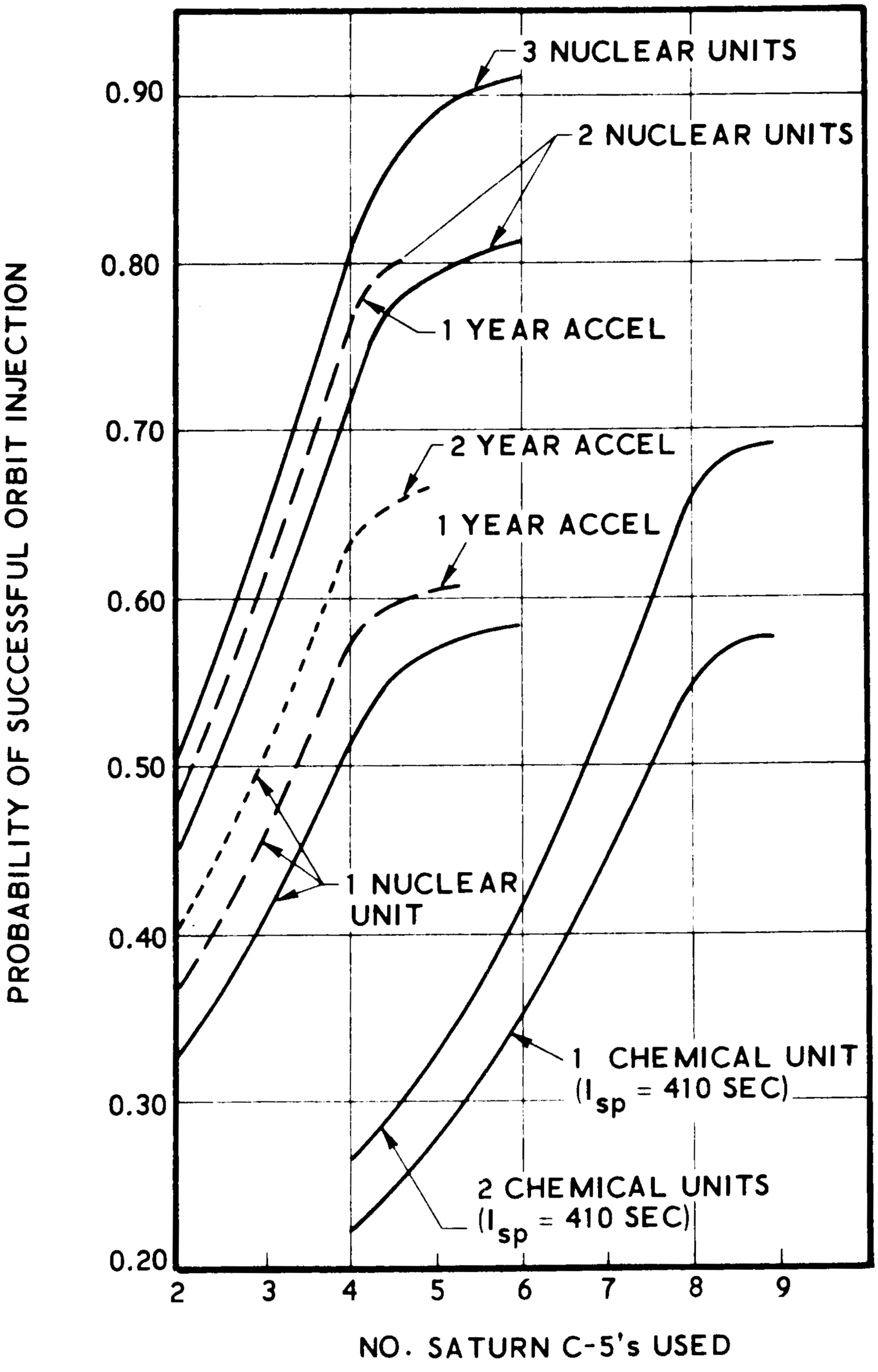


FIGURE 16. SYMMETRIC MISSION SUCCESS IN 1970 USING SATURN C-5'S

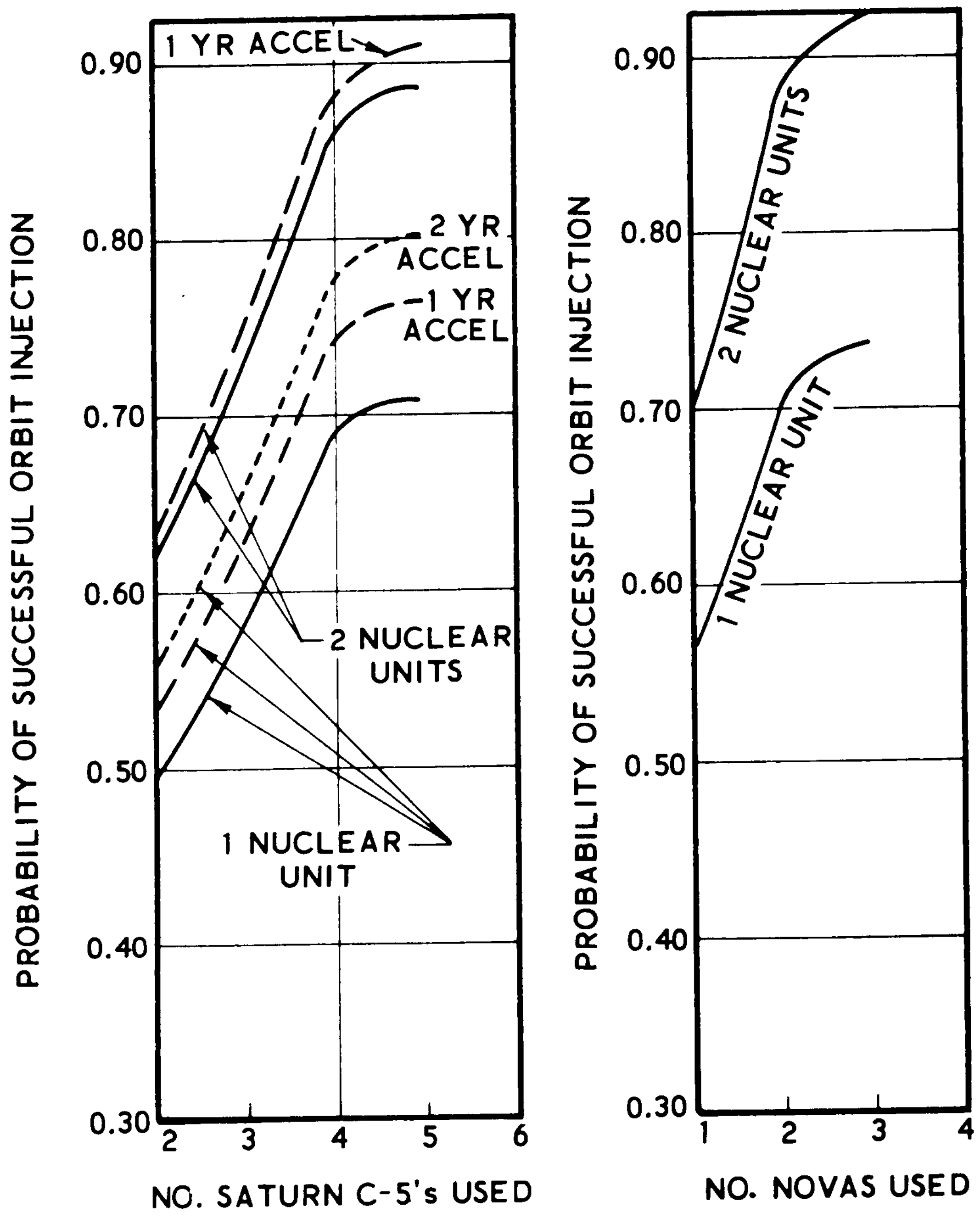
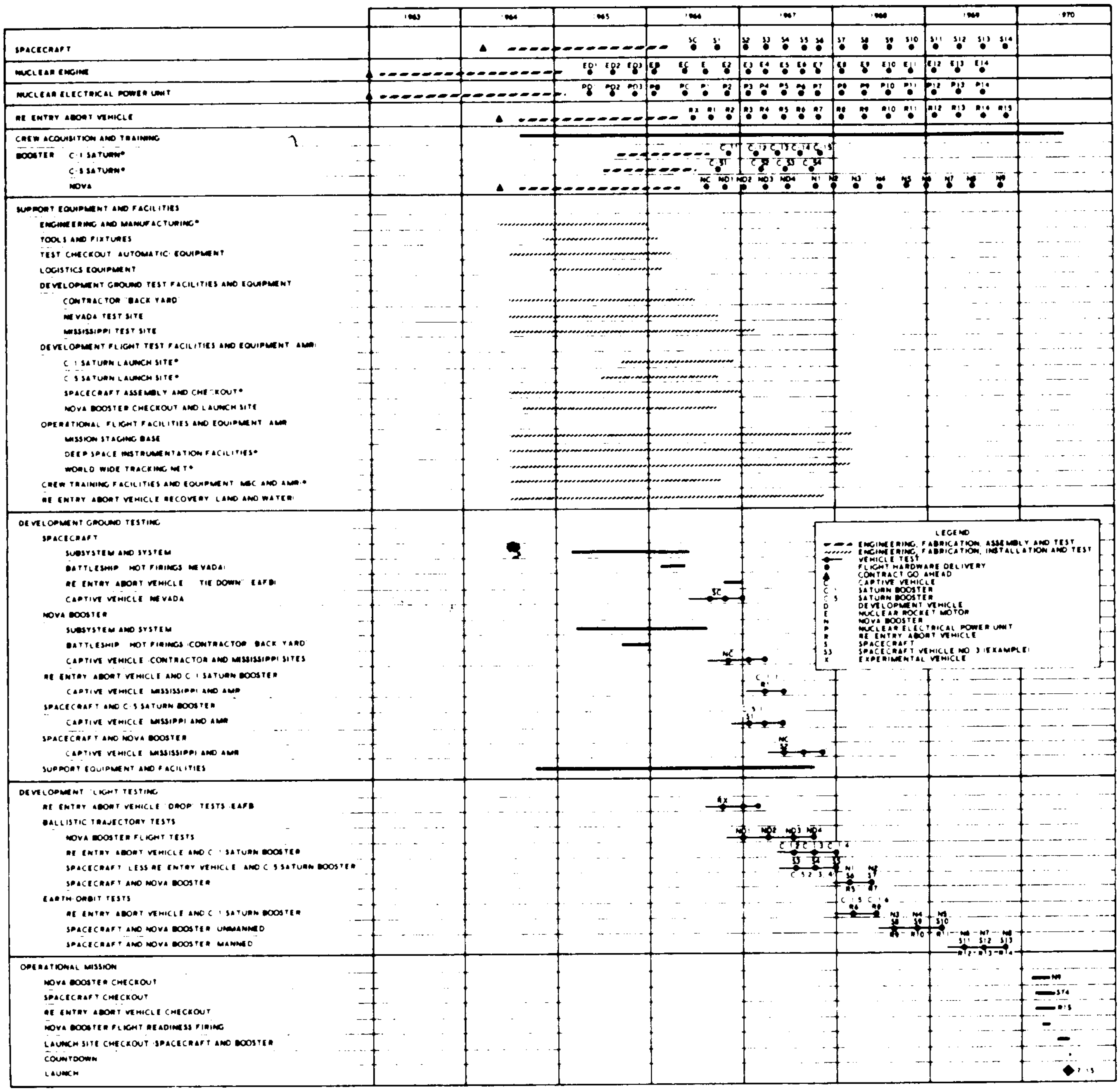


FIGURE 17. SYMMETRIC MISSION SUCCESS IN 1972



* MODIFICATION OF EXISTING EQUIPMENT AND FACILITIES

FIGURE 18. PROJECT EMPIRE DEVELOPMENT SCHEDULE – NUCLEAR SYMMETRIC MISSION

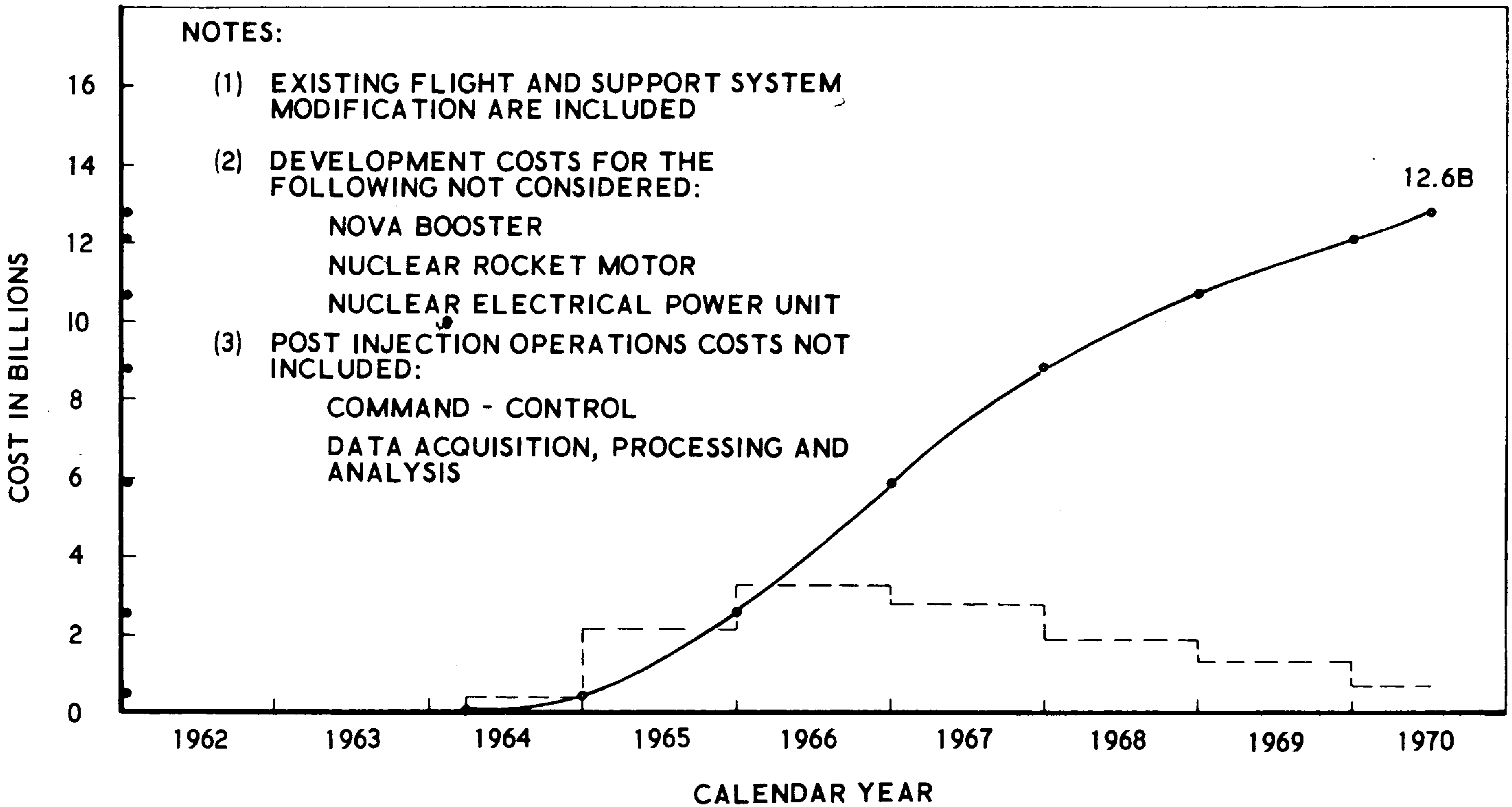


FIGURE 19. NUCLEAR SYMMETRIC MISSION FUNDING REQUIREMENT FOR 1970 LAUNCH

TABLE 1

TYPICAL EMPIRE MISSION CHARACTERISTICS

	CROCCO		SYMMETRIC	
$q_{MARS} = q_{VENUS}$	1.3	1.5	1.3	2.0
CHARACTERISTIC LAUNCH VELOCITY REQUIREMENT (300 KM PARKING ORBIT)	10.1 KM/SEC	8.9 KM/SEC	5.3 KM/SEC	3.9 KM/SEC
RE-ENTRY VELOCITY (NEGLECTING EARTH ROTATION)	13.5 KM/SEC	13 KM/SEC	15.8 KM/SEC	14.7 KM/SEC
TOTAL TRIP TIME	398.2 DAYS	403.8 DAYS	631.0 DAYS	613 DAYS
EARTH LAUNCH DATE	8-29-71	8-13-71	7-19-70	8-16-70

TABLE 2

RELIABILITY ESTIMATES OF EMPIRE GUIDANCE SUBSYSTEMS

SYSTEM	OPERATING TIME (HOURS)	SINGLE SYSTEM MTF (R=0.95) (HOURS)	SYSTEM WITH STANDBY BACKUP MTF (R=0.95) (HOURS)	STATE-OF-THE-ART MTF (HOURS)	REMARKS
ATTITUDE CONTROL PACKAGE	80	3,200	LESS THAN 1,000	GREATER THAN 1,000	DISPLACEMENT AND RATE GYRO REFERENCE
SUN SENSOR	14,500	290,000	65,900	75,000	
PLANET (EARTH) SENSOR	14,500	290,000	65,900	75,000	RELIABILITY MAY BE ENHANCED BY CARRYING REPLACEMENT UNITS
STAR TRACKER	1,450	29,000	6,590	80,000 (FOR R=0.90)	
GUIDANCE COMMUNICATION LINK	1,450	29,000	6,590	10,000 (NARROW B.W.)	
GUIDANCE COMPUTER	1,450	29,000	6,590	10,000	MINUTEMAN PROGRAM SPECIFICATIONS

TABLE 3

RE-ENTRY VEHICLE COMPARISONS

	APOLLO-TYPE		HIGH L/D VEHICLE		DRAG BRAKE
	SYMMETRIC	CROCCO	SYMMETRIC	CROCCO	SYMMETRIC
TOTAL WEIGHT (LB)	26,100	13,300	29,500	14,290	30,000
RETRO WEIGHT (LB)	12,920	-0-	16,300	1,675	-0-
HEAT SHIELD WT (LB)	3,880	4,000	3,455	3,455	12,000
CONTROL SYSTEMS WT (LB)	500	500	500	500	5,000
INITIAL VELOCITY (KM/SEC)	15.8	13.5	15.8	13.5	15.8
LANDING SITE SELECTION					
LONGITUDINAL	FAIR		GOOD		GOOD
LATERAL	POOR		GOOD		POOR

TABLE 4

CREW COMPOSITION

COMMANDING OFFICER EXECUTIVE OFFICER	FLIGHT SURGEON THREE CREW MEMBERS
WATCH PERSONNEL	RESPONSIBILITY
DUTY OFFICER – ON WATCH IN CONTROL ROOM LOCATED IN SOLAR RADIATION CELLAR	MONITORS ALL SYSTEMS MAINTAINS COMMUNICATIONS RECORDS KEY DATA INITIATES REPAIR ACTIVITY ALERTS SCHEDULED SCIENTIFIC ACTIVITY MAINTAINS SPACECRAFT INTEGRITY
MAINTENANCE AND REPAIR OFFICER	PERFORMS REGULAR PREVENTIVE MAINTENANCE ON ALL SYSTEMS PERFORMS REPAIR SUPERVISES EMERGENCY PROCEDURES ASSISTS IN SCIENTIFIC ACTIVITY
SCIENTIFIC ACTIVITY OFFICER	PERFORMS, RECORDS, AND EVALUATES SCHEDULED SCIENTIFIC ACTIVITY PLANS AND EXECUTES UNSCHEDULED ACTIVITY AS WARRANTED
WASTE MANAGEMENT, MEAL PREPARATION, ETC., SCHEDULED INFORMALLY FOR OFF-DUTY CREW. ALL PERSONNEL PERFORM ALL FUNCTIONS EXCEPT HAZARDOUS REPAIR IS NOT DONE BY COMMANDING OFFICER OR FLIGHT SURGEON.	

TABLE 5

TYPICAL SCHEDULE FOR SIX MAN CREW

	DUTY OFFICER	MAINTENANCE AND REPAIR	SCIENTIFIC ACTIVITY	SLEEP	REST
0000 - 0200	A	E	D	F	C,B
0200 - 0400	B	D	C	F,E	A
0400 - 0600	A	D	C	F,E	B
0600 - 0800	B	A	D	E	F,C
0800 - 1000	C	A	F	B	E,D
1000 - 1200	D	F	E	A,B	C
1200 - 1400	C	F	E	A,B	D
1400 - 1600	D	C	F	A	B,E
1600 - 1800	E	C	B	D	A,F
1800 - 2000	F	B	A	C,D	E
2000 - 2200	E	B	A	C,D	F
2200 - 2400	F	E	B	C	A,D

TABLE 6

SUMMARY OF SUBSYSTEM WEIGHTS

	WEIGHT (LBS)	WEIGHT (KG)
CROCCO TRAJECTORY		
LIFE SUPPORT	17,705	8,048
THERMAL CONTROL	1,000	455
POWER	10,000	4,545
ATTITUDE CONTROL	1,500	682
GUIDANCE	1,000	455
COMMUNICATION	300	136
FURNISHINGS	500	227
INSTRUMENTATION	1,000	455
EMERGENCY GEAR	1,200	545
SCIENTIFIC PAYLOAD	1,000	455
	35,205	16,003
SYMMETRIC TRAJECTORY		
LIFE SUPPORT	21,810	9,914
THERMAL CONTROL	1,000	455
POWER	10,000	4,545
ATTITUDE CONTROL	1,500	682
GUIDANCE	1,000	455
COMMUNICATION	300	136
FURNISHINGS	500	227
INSTRUMENTATION	1,000	455
EMERGENCY GEAR	1,200	545
SCIENTIFIC PAYLOAD	1,000	455
	39,310	17,869