NASA TECHNICAL NOTE

NASA TN D-6176

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NASA TN D-6176

COMPARISON OF TRAJECTORY PROFILES AND NUCLEAR-PROPULSION-MODULE ARRANGEMENTS FOR MANNED MARS AND MARS-VENUS MISSIONS

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NATIONAL AERONAUTICS AND SPACE ADMINISTRATION • WASHINGTON, D. C. • FEBRUARY 1971

1. Report No. NASA TN D-6176	2. Government Accession No.	3.
4. Title and Subtitle COMPARISON OF TRAJECTORY PROFILES AND NUCLEAR-PROPULSION-MODULE ARRANGEMENTS FOR MANNED MARS AND MARS-VENUS MISSIONS		5. Report Date February 1971
		6. Performing Organization Code
7. Author(s) Edward A. Willis, Jr.		8. Performing Organization Report No. E-5854
		10. Work Unit No. 124-08
9. Performing Organization Name and Address Lewis Research Center National Aeronautics and Space Administration Cleveland, Ohio 44135		11. Contract or Grant No.
		13. Type of Report and Period Covered Technical Note
12. Sponsoring Agency Name and Address National Aeronautics and Space Administration Washington, D. C. 20546		14. Sponsoring Agency Code
15. Supplementary Notes		

16. Abstract

A brief comparison is made between opposition class (with three impulse return), conjunction class, Venus swing-by to Mars and Mars-Venus double stopover trajectories. Vehicle configurations based on NERVA-I engines of varying life and restart capability are considered. It is found that the opposition and Venus swingby trajectories to Mars yield competitive mission performance; the double stopover entails about 30-percent penalties in initial mass and trip time; and that the minimum-weight vehicle uses a single, restartable engine.

17. Key Words (Suggested by Author(s)) Interplanetary trajectories; Nuclear propulsion; Propulsion module; Interplanetary missions; Mars; Venus; Manned space flight; Vehicle configuration	18. Distribution Statement Unclassified - unlimited

19. Security Classif. (of this report)	20. Security Classif. (of this page)	21. No. of Pages	22. Price*
Unclassified	Unclassified	30	$3.00

*For sale by the National Technical Information Service, Springfield, Virginia 22151

COMPARISON OF TRAJECTORY PROFILES AND NUCLEAR-PROPULSION-MODULE ARRANGEMENTS FOR MANNED MARS AND MARS-VENUS MISSIONS

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SUMMARY

In this study, approximate comparisons are made among several trajectory modes and nuclear-propulsion-module configurations that might reasonably be considered for manned missions to Mars or Mars and Venus between 1980 and 1995.

One recoverable and three expendable vehicle configurations are studied. In comparison with the lightest expendable configuration, the recoverable version involves an initial mass penalty of about 40 percent and places heavy demands on the nuclear rocket engine in terms of multiple restarts and very long operating life. The expendable vehicles differ primarily in the degree of engine reuse needed (up to six starts). Initial mass decreases by about 25 percent in going from one shot to restartable engine configurations.

Four interplanetary trajectory profiles are evaluated: (1) the Mars opposition-class with a double conic (three impulse) return leg, (2) the Venus swing-by to Mars, (3) the Mars-Venus double stopover, and (4) the conjunction-class Mars mission. The first two appear to be of primary interest if Mars alone is the manned space-flight goal, and, in desirable launch years, they are competitive in terms of trip times and initial masses. The double stopover is a reasonable contender if Venus, in addition to Mars, is regarded as a major objective for manned reconnaissance. It requires greater initial mass and trip time than fast missions to Mars only or to Venus only (by about 30 percent and 40 percent, respectively), but it represents an attractive savings in comparison with the sums for the two individual missions. The Mars conjunction mission offers the lowest initial mass of any mode studied herein, but it requires trip times exceeding 900 days. The saving in initial mass (about 20 percent compared with the next best mode under present ground rules) does not appear to be a decisive advantage for early missions when weighed against the 300 to 400 day trip-time penalty.

INTRODUCTION

Continuing progress in trajectory analysis and nuclear engine research makes appropriate a brief review of trajectories and nuclear propulsion configurations suitable for manned Mars and Venus missions. This general subject has been extensively studied in the past. Many of the prior studies, however (such as refs. 1 to 11), were primarily concerned with basic questions such as the need for nuclear engines and atmospheric braking, the necessity of solar-flare and meteoroid shielding, the usefulness of fast missions, etc. Relatively simple trajectories and vehicle configurations were usually employed.

Other contributions relate to the development or use of advanced trajectory techniques to obtain lower ΔV's, reduced gravity losses, or other advantages. Examples include the use of double conic (or three impulse) return trajectories for opposition-class Mars missions (fig. 1(a) and refs. 12 to 14), the Venus swing-by to Mars (fig. 1(b) and refs. 15 and 16), the Mars-Venus double stopover (fig. 1(c) and ref. 17), multiburn Earth escape maneuvers (fig. 2(a) and refs. 18 to 21), and elliptic planetary parking orbits (fig. 2(b) and refs. 22 to 26). These advances taken together imply significant reductions of both initial mass (based on equal payloads and technology inputs), vehicle accelerations needed and, hence, a sharp decrease in engine thrust requirements. In addition, there is a good prospect that NERVA-I will eventually be developed into a long-life restartable engine. As shown in reference 27, unconventional vehicle or staging arrangements may be necessary to take full advantage of these capabilities.

Several comparative studies are now available (e.g., refs. 15 and 28 to 31) that explore the advantages of certain of these advanced trajectory modes. Nevertheless, several of the techniques have not been evaluated fully or in the most advantageous combinations, while in other cases comparisons are hindered by major differences in study ground rules. Moreover, it has not been definitely established that any of the fast mission profiles are actually more desirable than the conjunction-class or double Hohmann mode (fig. 1(d) and refs. 6 and 32) which offers absolute minimum ΔV's in return for long (~1000 day) trip times.

For instance, reference 30, though very thorough in its treatment of topics covered, did not consider the possibility of using double-conic (three impulse) trajectory legs or the Mars-Venus double stopover trajectory mode. It did not consider multiple-burn Earth escape maneuvers or vehicle configurations which are made possible by that technique. Finally, the elliptic parking orbit analysis was limited to a single example (1984 opposition-class mission); despite the significant initial mass savings that were found, no effort was made to take across-the-board advantage of the technique. Therefore, the present report offers a brief and approximate, but consistent, comparison between these trajectory modes, and between vehicle staging configurations that reflect various degrees of engine-life - restart capability.

METHOD OF ANALYSIS

Configurations Studied

As indicated in figure 3, four basic vehicle staging configurations are studied:

(1) Tandem-staged vehicle (fig. 3(a)); nuclear Earth escape with three NERVA-I engines; individual nuclear stages for each planetary escape or capture maneuver; Earth aerobraking

(2) Single upper stage vehicle (fig. 3(b)); nuclear Earth escape with three NERVA-I engines (two-burn escape maneuver); single nuclear upper stage used for all interplanetary maneuvers; Earth aerobraking

(3) Single-engine vehicle (fig. 3(c)); one NERVA-I engine used for Earth escape and all interplanetary maneuvers; Earth aerobraking

(4) Single-engine vehicle with recoverable command module and engine (fig. 3(d)); same as (3), except the NERVA-I engine is also used for propulsive Earth retro into low parking orbit.

Configuration (1) is of interest because, by using fresh engines for every main maneuver, there is no requirement for multiple restart capability or long total burn times and no residual radiation problem. Configurations (2) to (4) involve varying degrees of engine reuse and are studied to evaluate the potential benefits of extended nuclear engine lifetime and restart capability.

Initial Mass

Vehicle masses were analyzed with the aid of a simple scaling law which may be readily derived from the classical rocket equation.

$$M_{j^{th}\,stage} = \frac{(1 + k_{ls})M_{pay} + N_e M_e}{1 - k_p(1 + k_{ps})} \qquad (1)$$

(Symbols are defined in appendix A.) If the numerator is defined as the effective payload M_j' and the denominator as the stage payload ratio $1/h_j$, equation (1) may be written as

$$M_{j^{th}stage} = M_j' h_j$$

The propellant fraction k_p is given by

$$k_p = 1 - \exp\left(-\frac{\Delta V_j}{gI_j}\right)$$

and it is understood that M_{pay} is to include all upper stages as well as jettisonable systems and supplies consumed. Thus for a mission with J propulsive maneuvers, the equation for initial mass in Earth orbit is

$$\text{IMEO} = (\ldots (M_J' h_J + M_{J-1}')h_{J-1} + \ldots + M_1')h_1 \tag{2}$$

Assumptions

The following assumptions were used in the calculations:

(1) Successive two-body trajectories with elliptic, coplanar planet orbits are used.

(2) The Earth assembly and parking orbit is circular with a radius of 400 kilometers.

(3) The parking orbits at Mars and Venus are elliptic (periapse radius, 1.1 planet radii; eccentricity e_{po}, 0.9). Optimum, off-periapse, nontangential impulses are used at capture and escape to orient the ellipse properly with respect to the hyperbolic asymptote (refs. 24 and 25).

(4) Atmospheric braking is used at Earth return for configurations (1) to (3). The maximum braking or atmosphere re-entry speed is 15.8 kilometers per second (2 times the circular speed or 52 000 ft/sec). Chemical retropropulsion is used, if necessary, to decrease entry speed to this value.

(5) For configuration (4), the command module and the propulsion module are captured at Earth return into a low circular parking orbit using nuclear propulsion.

(6) The payload and stage input parameters used are based on references 29 and 33, and are listed in table I.

(7) Representative gravity-loss ΔV corrections of 5 percent at Earth escape and 2 percent at Mars and Venus approach or departure are used except in the "Effect of propulsion system variations" section where exact corrections from references 16, 17, 21, and 23 are applied.

(8) The following reserve ΔV budget was arbitrarily incorporated: (a) midcourse correction, 45.7 meters per second (150 ft/sec) per leg and (b) Venus swingby maneuver, 305 meters per second (1000 ft/sec). Chemical propulsion (see table I) is used for corrective maneuvers.

(9) No further contingency ΔV's were allowed for launch window requirements, engine performance dispersions, and the like.

(10) Nuclear engine aftercooling losses were not accounted for in the main calculations. The effect of these losses on initial mass in Earth orbit (IMEO) is estimated in appendix B, however, and discussed in the next section.

RESULTS AND DISCUSSION

Comparison of Vehicle Configurations

The Venus swing-by offers an attractive combination of low $\Sigma \Delta V$'s with moderate trip times and is used here as a representative trajectory mode for comparing different vehicle types. The vehicles studied (shown in fig. 3) are representative configurations that reflect four distinct levels of nuclear engine restart and total lifetime capability.

Effect of burn time and restart capability. - As previously mentioned, the configurations studied differ primarily in their engine lifetime requirements. This is illustrated in figure 4, which relates IMEO, maximum burn time (accumulated on any one engine), and number of burns required for a 1984 Venus swing-by Mars mission.

The baseline tandem staged vehicle (circular symbols) requires five engines, of which none requires restart capability or more than a 30-minute burn. The corresponding IMEO is about 590 000 kilograms (1300 klbm).

The square symbols in figure 4 represent the baseline single-upper-stage configuration, which uses two engines for Earth escape by means of a two-burn injection maneuver, and a single upper stage for both capture and escape at Mars. In this case a significant IMEO reduction, to about 500 000 kilograms (1100 klbm), results from providing a single-restart capability and extending the maximum burn time to 45 minutes.

A further IMEO decrease (to about 480 000 kg or 1050 klbm) results for the baseline single-engine configuration (triangular symbols) in return for multiple restart capability and a maximum burn time approaching 2 hours.

The final configuration shown (diamond shaped data symbols) involves the recovery of the single nuclear engine and the command module by propulsive braking into a low Earth circular parking orbit. There is a significant IMEO penalty for doing this (about 40 percent compared with the lightest configuration). The added launch costs associated with increased IMEO would have to be balanced against the possible savings due to reusing the engine and command module, and eliminating the high speed re-entry vehicle, to determine the cost effectiveness of this configuration (not attempted herein). It should be pointed out also that the total burn time and number of burns shown in figure 4 for the recoverable configuration are for one mission only and that these values must be multiplied by the planned number of uses to define a total requirement. For example, for three mission uses, the engine would require a total of 9 hours burn time distributed over 21 burns, during a time interval of about 6.4 years (three consecutive launch opportunities).

The vehicle configurations studied thus far are based on a straightforward compromise between performance and complexity. Many alternatives exist, however, and some of them could offer appreciably better performance if their attendant design elaborations are tolerable. For example, the three lower modules in the tandem staged vehicle could be rearranged to give either a two-stage, two-burn Earth escape maneuver (ref. 19) or a three-stage, three-burn maneuver (ref. 20). Both are compatible with the baseline one-start, 30-minute engine capability. Similarly, the single upper stage vehicle could be modified to use a two-stage, two- (or more) burn Earth escape maneuver. Both single engine vehicles could use subdivided propellant tankage with multiple jettison points. Such measures could yield IMEO reduction in the 5- to 10-percent range, but, if applied consistently, it would not affect the present comparisons.

Effect of launch year. - The same general conclusion applies in launch years other than 1984, as may be judged from figure 5. There, IMEO is plotted against departure date for optimum trip times, still using Venus swing-by trajectories and input data from table I. The trend seen in figure 5 for 1984, that is, high IMEO's for the single-engine and recoverable configurations and distinctly lower values for the single-engine and single-upper-stage configurations, is clearly seen throughout the launch years considered. Burn times, listed in table II for the two extreme cases, follow the same pattern as the IMEO's.

The IMEO's shown in figure 5 cover a range of about 2 to 1 from best to worst. With the present inputs, however, all the resultant IMEO's appear to be reasonably within the capacity of Saturn V class launch vehicles (assuming orbital assembly of a few large, modular stages, and payload packages). Under these conditions, IMEO differences may be less decisive than variations in cost effectiveness, mission safety or reliability, or scientific data return. (For heavier inputs, however, the IMEO differences may be of greater concern.)

Effect of aftercooling losses. - Aftercooling propellant losses, neglected in the body of this report, are evaluated in appendix B. For a typical (1984 Venus swing-by) trajectory mode, aftercooling losses with no thrust recovery produce IMEO penalties of up to 8 percent for the single-engine vehicle and 4.2 percent for the tandem-staged vehicle. (These penalties can be roughly halved by employing aftercoolant thrust recovery.) Although such penalties are not negligible in practice, the 2- to 4-percent relative error caused by neglecting them is not great enough to significantly affect the comparisons drawn herein.

Effect of postfiring engine radiation. - Reference 27 shows that the postfiring nuclear engine residual radiation need not interfere seriously with planned extravehicular activity (EVA) such as the undocking and departure of a Mars excursion module. On the other hand, it would not be possible for the astronauts to conduct in person maintenance, repair, or inspection operations in a hot nuclear engine compartment. This could be an important disadvantage for any of the reusable-engine configurations. It is not known how much remote sensing and manipulating equipment would be required to provide the necessary maintenance capability. The mass of this equipment, however, cannot logically exceed the mass of a spare Nerva-I installation.[1] For the 1984 Venus swing-by single-engine configuration this would imply an IMEO penalty of 44 000 kilograms (97 000 lbm) or about 9.4 percent. A very detailed analysis would be necessary to decide whether such measures could properly be applied to one configuration but not another. Nevertheless, it is interesting to note that even if the maximum aftercoolant differential (see appendix B) is assumed, the single-engine configuration with a spare engine is still about 10 percent lighter (550 000 vs 605 000 kg or 1209 vs 1335 klbm) than the tandem configuration with no spares.

Effect of propulsion system variations. - The results presented thus far are based on a standard 334 000-newton (75 000-lbf) thrust, 11 680-kilogram (25 750-lbm) mass, 825-second-specific-impulse nuclear engine (see table I). The minimum IMEO vehicle configuration used only one of these engines, with four burns required for Earth escape. The effects of perturbing these parameters one at a time are shown in figure 6 for a representative mission (1986 Venus swing-by).

In figure 6(a), thrust is varied from 222 400 to 444 800 newtons (50 000 to 100 000 lbf) with the other parameters held at their baseline values. Compared with the baseline 334 000-newton (75 000-lbf) thrust, this results in an IMEO penalty of about 20 000 kilograms (44 000 lbm) for the decreased thrust and a saving of about 5440 kg (12 000 lbm) for increased thrust. This relatively small variation is entirely due to changes in the gravity loss ΔV requirements, primarily at Earth escape. As stated in the Assumptions section, accurate gravity loss corrections per references 18, 19, 23, and 24 are used in this section only for calculating propulsion system variations. The use of a multiple-burn Earth escape maneuver results in a relatively low IMEO to thrust tradeoff of about 0.1 kilogram per newton thrust (1 lbm per lbf thrust).

In figure 6(b), the role of multiburn Earth escape maneuvers is further illustrated. With baseline engine performance, IMEO drops by 68 000 kilograms (150 000 lbm) in going from one to two burns. Further but progressively smaller savings are seen for trajectories with three to eight burns. The optimum number of burns appears to be between four and seven when aftercooling losses are taken into account.

The effect of varying engine mass is displayed in figure 6(c). An engine mass increase of 4540 kilograms (10 000 lbm) causes an IMEO penalty of about 14 500 kilograms (32 000 lbm), and a 4540-kilogram (10 000-lbm) engine mass decrease yields a 14 500-

[1]The spare engine system would be remote from the main engine and could thus be inspected and maintained without the special remote sensing and manipulating equipment.

kilogram (32 000-lbm) IMEO saving; that is, the IMEO to engine mass tradeoff ratio for the case shown is about 3.2 to 1.

Finally, a ±25-second specific-impulse perturbation is displayed in figure 6(d). A 15 000-kilogram (33 000-lbm) penalty or savings occurs, respectively, when I is decreased or increased by 25 seconds. The IMEO to specific-impulse tradeoff ratio here is about -590 kilograms (-1300 lbm) IMEO to 1 second I or roughly -1 percent IMEO per 1 percent I.

These relatively low sensitivities are due in part to the use of a minimum IMEO vehicle as a reference point. For other vehicle configurations or different trajectory modes, the sensitivities increase roughly in proportion to IMEO.

Comparison of Mission Modes

Although the Venus swing-by trajectory may be a popular candidate for an early Mars mission, there are actually several other reasonable choices as described in the INTRODUCTION. These are compared in figure 7 for the minimum IMEO, single-engine vehicle configuration discussed in the preceding section.

Fast mission to Mars. - In figure 7(a), IMEO is plotted against launch year for the selected trajectory profiles. Note that the opposition-class trips contain two subcases, according to whether a conventional (two-impulse) or double-conic (three-impulse) return leg is used. These alternatives are indicated in the figure by ▢ and △, respectively. A comparison shows that the double-conic version has significant IMEO advantages in the difficult launch years but only minor gains in the easy years such as 1984 or 1986. Both versions require 450- to 550-day trip times. In brief, the double-conic option is never worse than the conventional one in terms of IMEO and is usually better. In addition, it will be shown in the section Effect of Trajectory Perturbations that the double-conic option is less penalized by the application of constraints such as extended Mars staytime or a lower maximum allowable Earth atmospheric re-entry speed. Its primary disadvantage is that an additional nuclear engine burn is required to perform one midcourse maneuver. This, however, is not a major problem when restartable engines are available. For these reasons the double-conic mode is selected as the standard type of opposition-class mission.

Consider next the comparison between the opposition - double-conic and the Venus swing-by trajectories. In general, these two models are quite competitive. The opposition class has a slight IMEO advantage in 1984 and 1986; in other years, the swing-by offers the lower IMEO's but also requires longer optimum trip times (fig. 7(b)). As will be illustrated later, the opposition - double-conic mode also yields the lower IMEO in most launch years if trip time is limited to 480 days or less. On the other hand, it does require one more engine restart. An advantage of the swingby mode is that its associated Earth re-entry speeds are moderate (e.g., 14.3 km/sec or 47 000 ft/sec). The opposition - double-conic mode invariably uses the assumed maximum of 15.8 kilometers per second (52 000 fps).

Based on these considerations, there seems to be no clear cut choice between the double-conic and Venus swingby modes. The final decision between them could more appropriately depend on factors such as reliability, mission objectives capability, compatibility with standard vehicles, etc, than on IMEO performance.

Low energy missions to Mars. - Historically, the fast profiles were studied as means of decreasing the very long trip times and staytimes associated with minimum energy (conjunction class) round trips. In order to evaluate the penalty implied by this, the IMEO's for a sequence of conjunction-class trips are presented as the lower curve in figure 7(a). Compared with this, the previously discussed fast missions bring with them IMEO penalties averaging 100 000 kilograms (220 000 lbm) in return for a trip time saving of about 400 days (see fig. 7(b)). This would appear to be a reasonable trade-off for early missions. On the other hand, it should be recognized that:

(1) The IMEO penalties could be much larger under different study ground rules - in particular, if chemical propulsion and circular rather than elliptic parking orbits were used. Hence, the conjunction-class mission is perhaps of some value as a backup.

(2) Based on Apollo experience, it may be expected that the first manned missions to Mars will be primarily of an engineering nature. Although such missions will also undoubtedly result in some valuable scientific information, it seems doubtful whether conclusive, fundamental investigations can be accomplished by two to four men in 30 days. Thus, the long stay times of the conjunction class mission may eventually prove to be a scientific advantage. (This assumes that at least some of the crew members are scientifically trained and that appropriate apparatus is provided.)

These considerations suggest that the conjunction-class mission be viewed as a backup mode for early missions, and as a leading contender for later, more ambitious, missions. A more detailed discussion is provided in references 6, 28, and 32.

Missions to both Mars and Venus. - The preceding modes are of primary interest if Mars is considered to be the only major objective for manned interplanetary space flight in the foreseeable future. On the other hand, if Venus is also of interest, there is no reason to assume that both planets can best be explored by separate missions. The alternative, which is to stop over at both Mars and Venus before returning to Earth, was shown in reference 17 to offer total ΔV and trip time advantages when compared with the sums for Mars only and Venus only trips. Its IMEO performance for present assumptions is shown by the circular data symbols in figure 7(a). As might be expected, the resultant IMEO's are higher, typically by about 150 000 kilograms (330 000 lbm) or 33 percent, than for the "fast" trips to Mars only. There is also a requirement for two additional engine restarts and about 30 minutes additional total burning time. Trip times as illustrated in figure 7(b) also tend to be significantly larger.

These penalties, however, are smaller than would be incurred by adding a separate Venus orbiter mission (ref. 29) to one of the previously discussed Mars only missions; that is, using input assumptions roughly comparable to the present ones, reference 29 estimated the mass of a typical Venus manned orbiter at about 272 000 kilograms (600 000 lbm) for trip times in the 450- to 550-days range. When these values are added to those shown previously for the Mars only mission, it may be seen that the Mars-Venus double stopover mode yields savings on the order of 136 000 to 181 000 kilograms (300 000 to 400 000 lbm) total IMEO and 200 to 400 days total trip time. In addition, this mode halves the requirement for astronaut crews and for several major cost items, such as command modules and nuclear propulsion modules. On this basis of comparison, the main disadvantage of the double stopover is that its trip times (which apply to a single crew) tend to be uncomfortably long. It should be noted, however, that a trip of about 650 days is available once in every 3.2 years (approximately). This is little greater than the longest Venus swing-by Mars mission.

Effect of trip time constraints. - Up to this point, the profiles considered have been discussed primarily in terms of their optimum mission time requirements. The effect of placing various arbitrary upper bounds on this parameter is illustrated in figure 8. Initial mass in Earth orbit is plotted against the allowable mission time for each of the four basic modes considered. Launch years are 1984 (part a), 1986 (part b), and 1988 (part c).

In 1984 and 1986, as previously noted, the opposition - double-conic mission yields smaller IMEO's than the swing-by; in 1986 there is also a substantial optimum trip time reduction. It is clear that both trajectories can be shortened to about 420 days in 1984 before major IMEO penalties are incurred. Conjunction-class missions in 1984 cannot be shortened much below 900 days without giving up their IMEO advantage. There is no region in which the 1984 double stopover yields the lowest IMEO; however, the penalties shown are not intolerable and do not become so until trip times below 600 days are sought.

In 1986, the double conic has both a trip time and IMEO advantage and can be shortened to about 400 days without an excessive penalty. The swing-by would require a 550-day trip for the same IMEO. Neither the conjunction-class nor the double stopover trajectory can be shortened significantly below 800 days in this year.

In 1988, the swing-by yields lower IMEO than the opposition double-conic mode for optimum trip times, but this advantage decreases and finally vanishes as the trip time is decreased. In contrast to 1984 and 1986, the conjunction mission in this case can be shortened to about 700 days before major IMEO penalties set in. This is, apparently, a peculiarity of the 1988 opportunity. The double stopover can be shortened to about

600 days, as was the case in 1984.

At present, the question of allowable trip time is unclear. It has not been objectively established whether 500 days (e.g.) is acceptable or whether 1000 days is too long. Clearly however, this choice, which has yet to be made, will have a major effect on the selection of a mission mode.

Effect of trajectory perturbations. - The results presented up to this point were based on several trajectory assumptions, namely, (1) elliptic planetary parking orbit with an eccentricity of 0.9, (2) Mars (or Venus) stopover of at least 30 days, and (3) atmospheric braking at Earth return with re-entry speed limited to 15.8 kilometers per second (52 000 ft/sec). Figure 9 shows the effect of more conservative assumptions in these areas, based on fast mission profiles in 1986. The opposition trips are represented by two solid curves, the upper for the conventional (single-conic return) option and the lower for double-conic trajectories. Both modes have 450 to 500 days optimum trip times. The long- and short-dashed curves represent Venus swing-bys with optimum trip times (600 to 700 days) and with trip times held to 500 days, respectively.

The effect of varied parking orbit eccentricity e_{po} is shown in figure 9(a). In each case, the use of a highly elliptic parking orbit yields an IMEO saving of about 113 400 kilograms (250 000 lbm) in comparison with the circular orbit. Also, the saving varies almost linearly with e_{po}. The orbital period (represented by the long-short dashed curve and applicable to all mission modes), however, increases sharply as e_{po} passes a knee at about 0.75. The eccentricity of 0.9 as used herein corresponds to an orbital period of about 60 hours or $2\frac{1}{2}$ days and is probably a practical upper limit in terms of Earth return launch window requirements.

Figure 9(b) illustrates the influence of the stay time at Mars. Clearly, this may be extended to up to perhaps 90 days for either the optimum-time swing-by or the opposition-class trajectory, without encountering prohibitive IMEO penalties. The optimum-time swing-by is less sensitive but requires longer trip times than the opposition class (700 as opposed to 500 days).

The effect of decreasing the allowable Earth re-entry speed V_{AE} is shown in figure 9(c). As was mentioned earlier, Venus swing-by trajectories often have re-entry speeds below the assumed maximum of twice circular speed (15.8 km/sec or 52 000 ft/sec), whereas opposition trips invariably use the largest permissible value. Thus the opposition trips are penalized[2] for any speed reduction, but the swing-bys are not penalized at all until V_{AE} reaches $1.52\ V_C$ for time-optimum swing-bys or $1.73\ V_C$ for the

[2]As suggested in ref. 26, however, the opposition trips would be less penalized if the Earth retropropellant could also be used for biological shielding.

500-day swing-by. Thus, a crossover between opposition trips and the optimum-time swing-by occurs near $V_{AE} = 1.6\ V_C$ (about 12.6 km/sec) with swing-bys yielding lower IMEO's at lower V_{AE}'s. On the other hand, the swing-by requires over 100 days longer trip time than the opposition-class trajectory. When these two trajectories are compared on the basis of trip times not exceeding 500 days, the opposition-class trip retains some IMEO advantage down to parabolic entry ($V_{AE} = 1.414\ V_C$ or about 11.02 km/sec), especially if the double-conic option is used.

Attribution of Initial Mass in Earth Orbit Savings

It may be recalled from figure 7 that the years 1982-1988 are especially favorable for fast (450 to 600 days) trips to Mars. Even when due allowance is made for neglected items such as after-coolant losses and launch-window reserves, it appears that IMEO's need be no greater than about 500 000 kilograms (1100 klbm). Such values are relatively light in comparison with the payloads used (121 000 kg or 267 klbm) on a Mars mission. In retrospect, this observed fact may be attributed primarily to five features or ground rules that were combined in the present analysis: (1) the availability of a long-lived restartable nuclear engine with performance at least equalling the present NERVA-I engine goals, (2) multiple-burn Earth escape maneuvers and an appropriate vehicle configuration, (3) the use of a highly elliptic parking orbit at Mars, (4) the use of Venus swing-by or double-conic interplanetary trajectories (with the choice possibly depending on launch year or allowable trip times), and (5) the use of moderately high-speed aerobraking for Earth return.

It will be recalled from the discussions of figures 4, 6, and 10, that the IMEO penalty for seriously compromising any one of these features would not, in general, exceed 25 percent. On the other hand, the IMEO could easily be doubled or tripled by simultaneously abandoning all five of these features.

CONCLUSIONS

Although the preceding study is preliminary in nature, several tentative conclusions may be drawn:

1. By taking full advantage of expected engine restart and life capabilities and presently known trajectory techniques, the NERVA-I nuclear rocket engine can efficiently support a major manned interplanetary mission program. The key trajectory techniques are (a) multiburn Earth escape maneuvers, (b) elliptic parking orbit at Mars or Venus, (c) Venus-swingby or three-impulse interplanetary transfers, and (d) moderate to high-

speed atmospheric braking at Earth return.

2. The launch years from 1982 to 1988 appear especially attractive for fast missions to Mars. Although it is understood that many factors will temper the choice as to when the first such mission will be performed, it must be pointed out that a similarly favorable stretch of launch opportunities will not recur until 1998.

3. Among the fast mission modes, both the opposition-class trajectory with a double-conic (three-impulse) return leg and the Venus-swing-by mode seem attractive and yield comparable Mars mission performance. The swing-bys usually have a slight initial-mass advantage while the double-conic opposition trajectories have shorter trip times. In neither case is the difference great enough to be decisive. Further studies, using reliability, cost effectiveness, or mission objectives capability as criteria, will be required to resolve the competition between these alternatives.

4. If Venus as well as Mars is of major interest as a manned space flight goal, then the Mars-Venus double stopover trajectory is an interesting possibility and deserves more detailed study. Under present assumptions, this profile involves less total mass and total trip time than the sum of individual missions to Venus and Mars. On the other hand, its initial mass and trip time are about 30 percent higher than for the best Mars only modes. These penalties must be balanced against the added mission objectives that might be accomplished by a manned orbital reconnaissance of Venus.

5. The conjunction mission yields an initial mass saving of about 20 percent or 10^5 kilograms, compared with the best fast mode, and usually requires trip times in excess of 900 days. Although certainly not negligible, the initial mass savings alone does not appear to be decisive when compared with the trip time penalty of 300 to 400 days. On the other hand, the extra 300 to 400 days is spent entirely at Mars; this long stay time could develop into a major advantage if Mars-science objectives (as opposed to technological objectives) are heavily emphasized in the mission plan. Moreover, the conjunction mission's initial mass saving could increase greatly under the following conditions: (a) if there is a sharp increase in mission payload, (b) if chemical rather than nuclear propulsion must be used, and (c) if low circular rather than highly elliptic planetary parking orbits are required. The status of conjunction-class missions should be re-examined if any of the above circumstances occur.

6. Among the expendable vehicles, the lowest masses were obtained for those using multiple start, long-life engines. There is a significant initial mass reduction in going from a system with single-shot engines to one using a single engine started six times (for a total burn time of around 2 hr). The single engine configuration is also attractive because of its mechanical simplicity and should be studied in greater detail.

7. Although it appears feasible to recover the command module, the engine system, and a part of the hydrogen tankage into a low Earth orbit, this entails an initial mass

penalty approximating 40 percent. It will be necessary to balance the added launch costs implied by this against the value of the recovered equipment.

Lewis Research Center,
National Aeronautics and Space Administration,
Cleveland, Ohio, September 16, 1970,
124-08.

APPENDIX A

SYMBOLS

AERO	abbreviation for aerodynamic braking at Earth return
e	parking orbit eccentricity, dimensionless
F/W	engine thrust to engine weight ratio
g	acceleration due to gravity, 9.8 m/sec^2 (32.17 ft/sec^2)
h	stage mass growth factor (see eq. (1), dimensionless
I	specific impulse, sec
IMEO	initial mass in Earth orbit, kg (lbm)
k_{ls}	deadweight fraction proportional to stage payload, dimensionless
k_p	propellant fraction, dimensionless
k_{ps}	deadweight fraction proportional to stage propellant, dimensionless
M	stage initial mass, kg (lbm)
M_{pay}	stage payload mass, kg (lbm)
M_{prop}	stage propellant mass, kg (lbm)
M_e	nuclear engine mass, kg (lbm)
M'	effective stage payload mass (see eq. (1)), kg (lbm)
N_e	number of nuclear engines on stage
N-I	abbreviation for NERVA-I type solid-core nuclear rocket engine
V	velocity, km/sec (in./sec)
ΔV	propulsive velocity increment, km/sec (mile/sec)
T_{max}	maximum allowable postshutdown nuclear rocket core temperature
1N, 2N, 3N. . .	code indicating number of nuclear engines expended per maneuver
1T, 2T, 3T. . .	code indicating number of propellant tanks expended per maneuver
Subscripts:	
AE	atmospheric entry
arr	arriving
C	circular

J	top (Earth return) stage
j	general stage
lv	leaving
po	parking orbit
⊕	Earth
♀	Venus
♂	Mars

APPENDIX B

APPROXIMATE ANALYSIS OF INITIAL MASS IN EARTH ORBIT PENALTY DUE TO AFTERCOOLING LOSSES

Figure 12 of reference 25 gives total aftercoolant mass as a function of (1) total impulse propellant and (2) the maximum temperature T_{max} to which the engine can safely be heated after shutdown, for a NERVA type of core. The data given there may without major error be represented by the following empirical equation:

$$M_{aftercoolant} \approx \left(\frac{1350^{o}\ R}{T_{max}}\right)^{1.4} \left(M_{main\ stage\ propellant}\right)^{0.8} \tag{B1}$$

(Here as in ref. 25, U.S. Customary units are used). A single-burn nuclear engine need only be kept structurally intact to prevent a disintegrating core from damaging or contaminating the nearby space vehicle. In this case, a T_{max} as high as 2500° R may be allowable. For a reusable engine, however, a lower value - for example, 1500° R as suggested in reference 25 - is necessary to preserve functional as well as structural integrity.

Using these values of T_{max}, equation (B1) was applied to the tandem-staged and single-engine configurations for the 1984 Venus swing-by mission by computing linear perturbations about equation (2):

$$\begin{aligned}\Delta(\mathrm{IMEO}) = h_1 M_{aftercoolant, lv\oplus} &+ h_1 h_2 h_3 M_{aftercoolant, arr ♂} \\ &+ h_1 h_2 h_3 h_4 M_{aftercoolant, lv ♂}\end{aligned} \tag{B2}$$

Here, it is assumed that the aftercoolant flow is entirely wasted. Pertinent details for the evaluation of equations (B1) and (B2) are given in table III. This gives, for the single-engine vehicle, an IMEO penalty of about 8 percent or 36 300 kilograms (80 klbm) bringing the total to 504 000 kilograms (1110 klbm).

For the tandem-staged vehicle, the penalty is approximately 4.2 percent or 24 400 kilograms (53.9 klbm) bringing its total to 605 000 kilograms (1335 klbm).

These penalties may be decreased significantly, perhaps by a factor of two, by taking credit for thrust due to aftercoolant flow in calculating the propulsive maneuvers.

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TABLE I. - INPUT DATA

Payload mass, kg (lbm):	
Returned to Earth (ref. 31):	
Entry vehicle	7890 (17 400)
Electric power system	3710 (8170)
Total	11 600 (25 570)
Mission module jettisoned at Earth (ref. 31)	37 600 (82 900)
Expendables, kg/day (lbm/day)	22.70 (50)
Used at Mars:	
Mars excursion module[a]	43 220 (95 290)
Probes and auxiliary equipment	11 100 (24 480)
Structure and enclosures	9520 (21 000)
Circular to elliptic parking orbit crew transfer vehicle[b]	4420 (9750)
Total	68 260 (150 520)
Used at Venus (ref. 27):	
Orbital equipment	13 600 (30 000)
Probes	4540 (10 000)
Total	18 140 (40 000)
Stage weights (NERVA-I engines (ref. 31)), kg (lb):	
Nuclear engine (includes internal shield and thrust structure)	11 680 (25 750)
External shielding (top nuclear stage only)	908 (2000)
Tank mass (dry), kg/module (lbm/module)	7070 (15 586) [+0.064 × M_{prop}]
Residuals (gas and liquid), kg/engine (lb/engine)	908 (2000)
Interstage structure ($M_{pay\ total}$)	0.015
Nuclear engine performance:	
Thrust, N (lb)	334 000 (75 000)
Specific impulse, sec	825
Chemical stage parameters:	
Specific impulse, Earth retromaneuver (O-H, pump-fed), sec	460
Specific impulse, midcourse maneuver (O-H, pressure-fed), sec	425
Tank fraction, k_{ps}	0.050
Interstage fraction, k_{ls}	0.015
Engine, F/W (including thrust structure)	25
Vehicle initial acceleration, g	0.2

[a]This includes an ascent stage capable of rendezvousing in a low circular parking orbit.

[b]This is an auxiliary vehicle placed in a low circular parking orbit before the MEM descends to Mars' surface. Upon completion of surface operation, it will first rendezvous with the MEM's ascent stage. Then, (with crew and samples aboard) it will transfer from the circular orbit to the ellipse and rendezvous with the main space vehicle. A similar scheme is analyzed and discussed in pp. 49 and 50, fig. 24, and appendix C of ref. 28.

TABLE II. - BURN TIME BREAKDOWN FOR VENUS-SWINGBY MISSIONS TO MARS

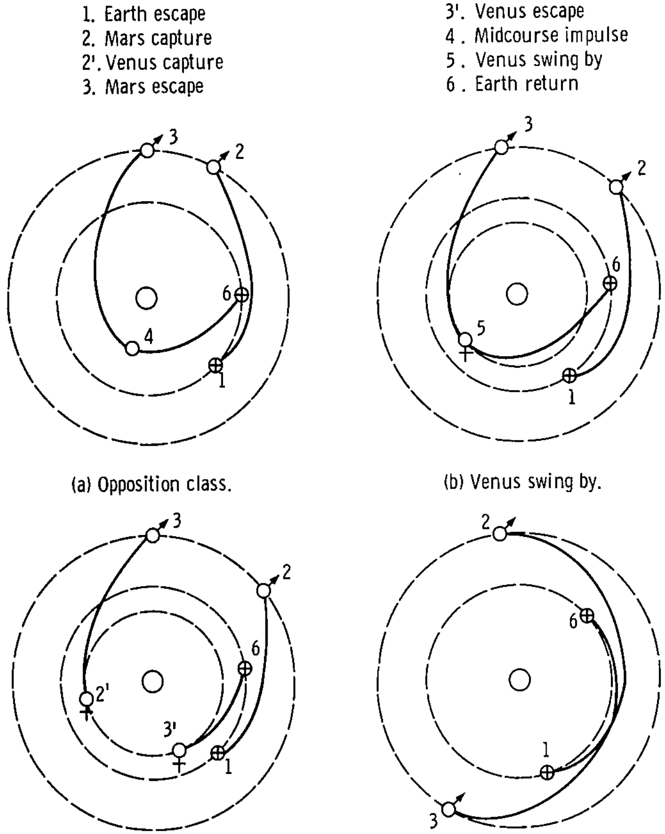
Year	Type	Earth escape burn, min	Mars capture burn, min	Mars escape burn, min
Tandem staged vehicle (3N3T/1N1T/1N1T/AERO)				
1980	Outbound	[a]34.1	24.0	5.6
1982	Inbound	25.8	16.8	17.9
1984	Inbound	27.9	24.2	24.1
1986	Outbound	32.0	27.0	4.5
1988	Inbound	26.4	12.5	15.4
1990	Outbound	39.8	29.0	10.0
1993	Outbound	37.6	38.6	5.6
1995	Inbound	30.0	23.8	19.8
Single engine vehicle (2T/1N1T/AERO)				
1980	Outbound	[b]80.0	21.4	5.6
1982	Inbound	61.1	15.8	17.8
1984	Inbound	67.5	22.1	23.8
1986	Outbound	74.8	27.5	4.7
1988	Inbound	62.0	11.5	15.0
1990	Outbound	95.5	26.2	10.3
1993	Outbound	89.6	34.3	5.65
1995	Inbound	72.5	22.0	23.1

[a]Values in column are for single burn.
[b]Values in column are the total of four Earth escape burns.

TABLE III. - DATA FOR EVALUATING EQUATIONS (B1) AND (B2)

Maneuver	Julian date, 244-	Propulsive velocity increment, ΔV		Growth factor (cumulative)	Main-stage propellant mass		Aftercoolant mass (eq. (B1))		Notes
		km/sec	mile/sec		kg	lbm	kg	lbm	
Single engine; initial mass in Earth orbit, 467×10^3 kilograms (1.03×10^6 lbm)									
Leave Earth	5723	3.57	2.22	1.613	167×10^3	368×10^3	11.2×10^3	24.7×10^3	(a), (b)
Arrive at Mars	5946	1.80	1.12	2.085	54.7	120.6	4.54	10.0	(a)
Leave Mars	5976	4.43	2.75	3.79	59	130	2.36	5.2	(c)
Tandem-stated configuration; initial mass in Earth orbit, 582×10^3 kilograms (1.281×10^6 lbm)									
Leave Earth	5725	3.55	2.21	1.611	207×10^3	456×10^3	6.44×10^3	14.2×10^3	(c)
Arrive at Mars	5946	1.80	1.12	2.085	60	132	2.40	5.29	(c)
Leave Mars	5976	4.45	2.77	3.820	59.4	131	2.38	5.25	(c)

[a]Maximum allowable postshutdown rocket nuclear core temperature, 1500° R.
[b]Four-burn escape maneuver.
[c]Maximum allowable postshutdown rocket nuclear core temperature, 2500° R.



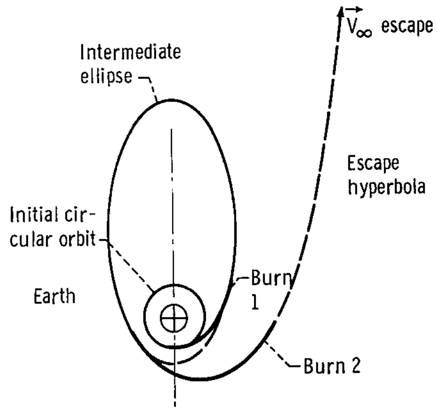
(a) Opposition class.

(b) Venus swing by.

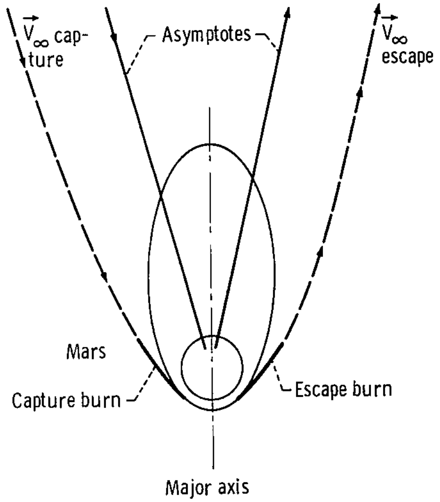
(c) Venus-Mars double stopover.

(d) Conjunction class.

Figure 1. - Interplanetary trajectory profiles.

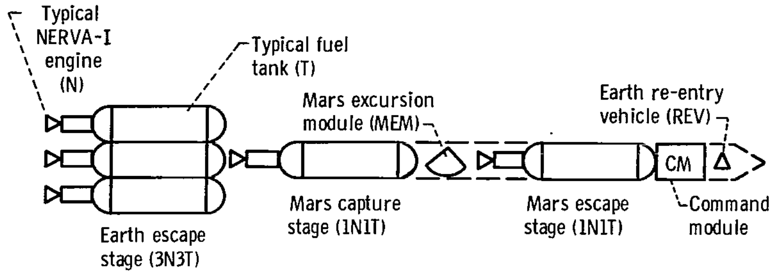


(a) Two-burn and multiburn Earth escape maneuvers.

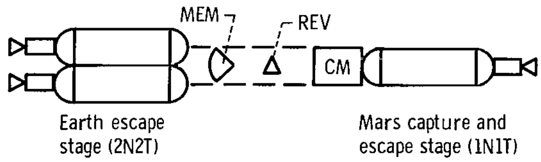


(b) Elliptic planetary parking orbit.

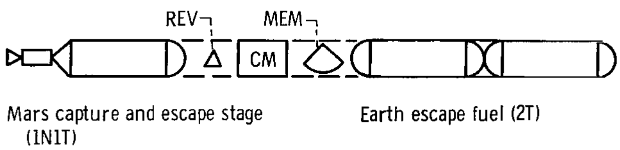
Figure 2. - Planetocentric trajectory profiles.



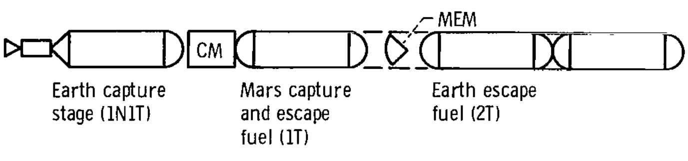
(a) Configuration 1. Tandem-staged vehicle (3N3T/1N1T/1N1T/AERO). Three-engine Earth escape stage; individual single-engine stages for Mars capture and escape; aerodynamic Earth retro.



(b) Configuration 2. Single upper stage (2N2T/1N1T/AERO). Two-engine Earth escape stage; one single-engine stage for both capture and escape at Mars; aerodynamic Earth retro.



(c) Configuration 3. Single engine vehicle (2T/1N1T/AERO). Single engine Earth escape, Mars capture, and Mars escape; tank staging at Earth escape and Mars escape; aerodynamic Earth retro.



(d) Configuration 4. Single engine vehicle with recoverable engine and command module (2T/1T/1N1T). Single engine Earth escape, Mars capture and escape, and Earth retro. Tank staging at Earth escape and Mars escape. (Command module, engine, and Earth capture propellant tank are recovered into low circular Earth parking orbit, Earth re-entry vehicle not required).

Figure 3. - Vehicle configurations (illustrated for Venus swingby model).

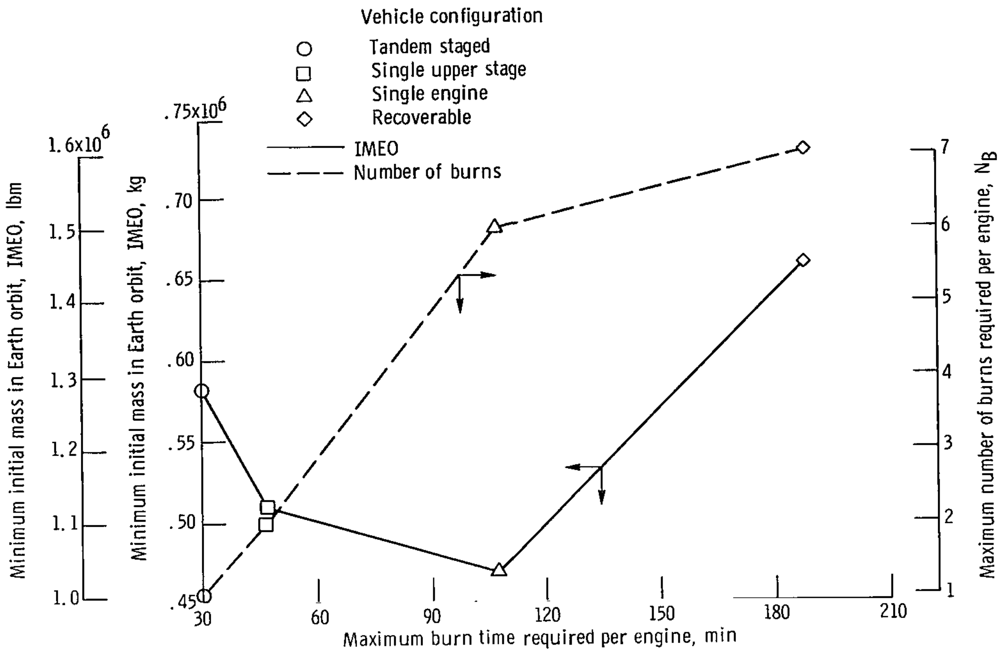


Figure 4. - Effect of burn time and restart capability of NERVA-I propulsive stages. See table I for input data. Inbound Venus swing-by in 1984; 500-day trip; 30-day stopover.

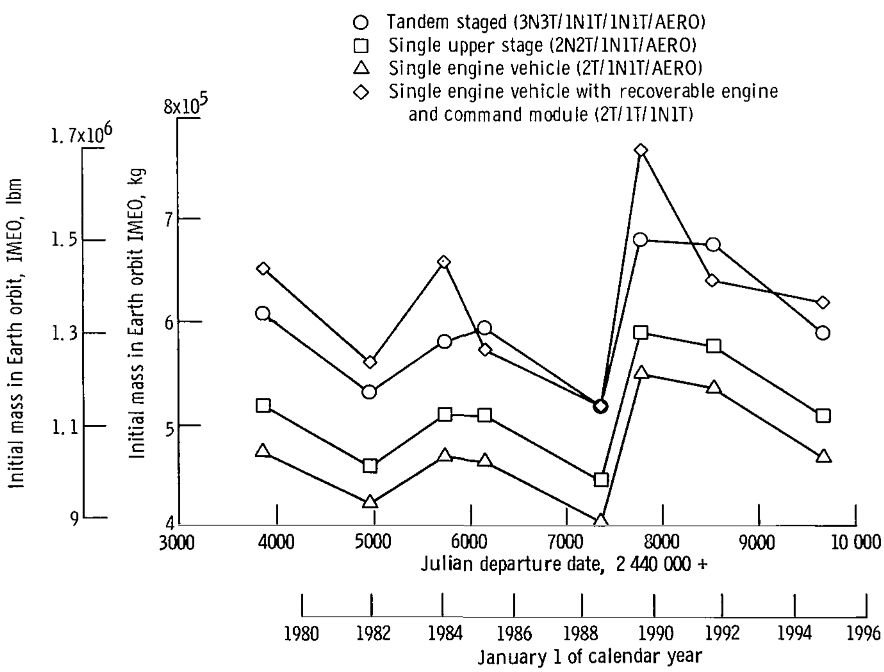


Figure 5. - Effect of launch year on initial mass. NERVA-I propulsive stages. See table I for input data. Venus swing by trajectories, 30-day stopover at Mars; optimum trip times.

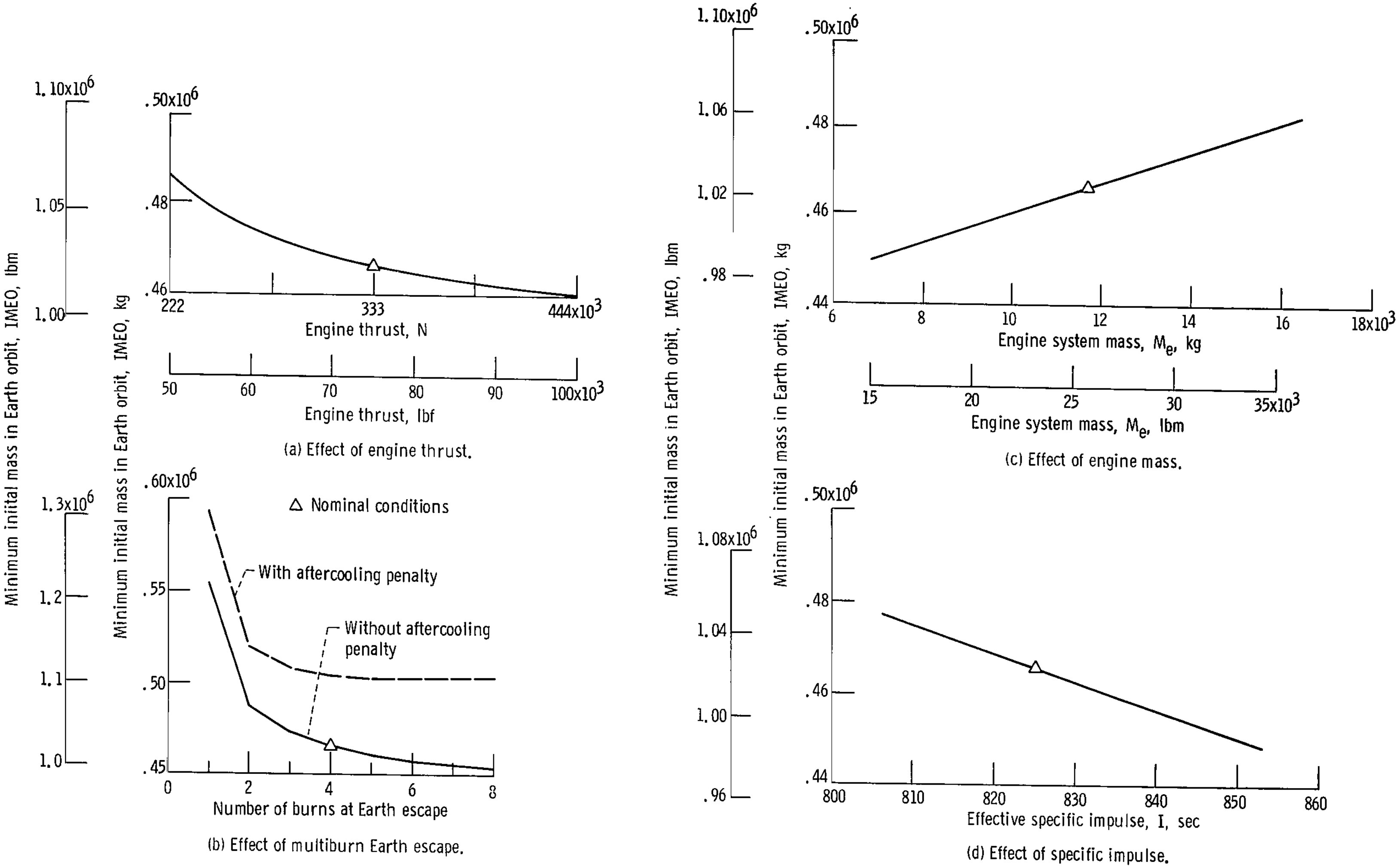


Figure 6. - Effect of propulsion system characteristics. For single engine vehicle input data, see table I. Venus swing-by trajectory in 1986. Optimum trip time; 30-day stopover at Mars.

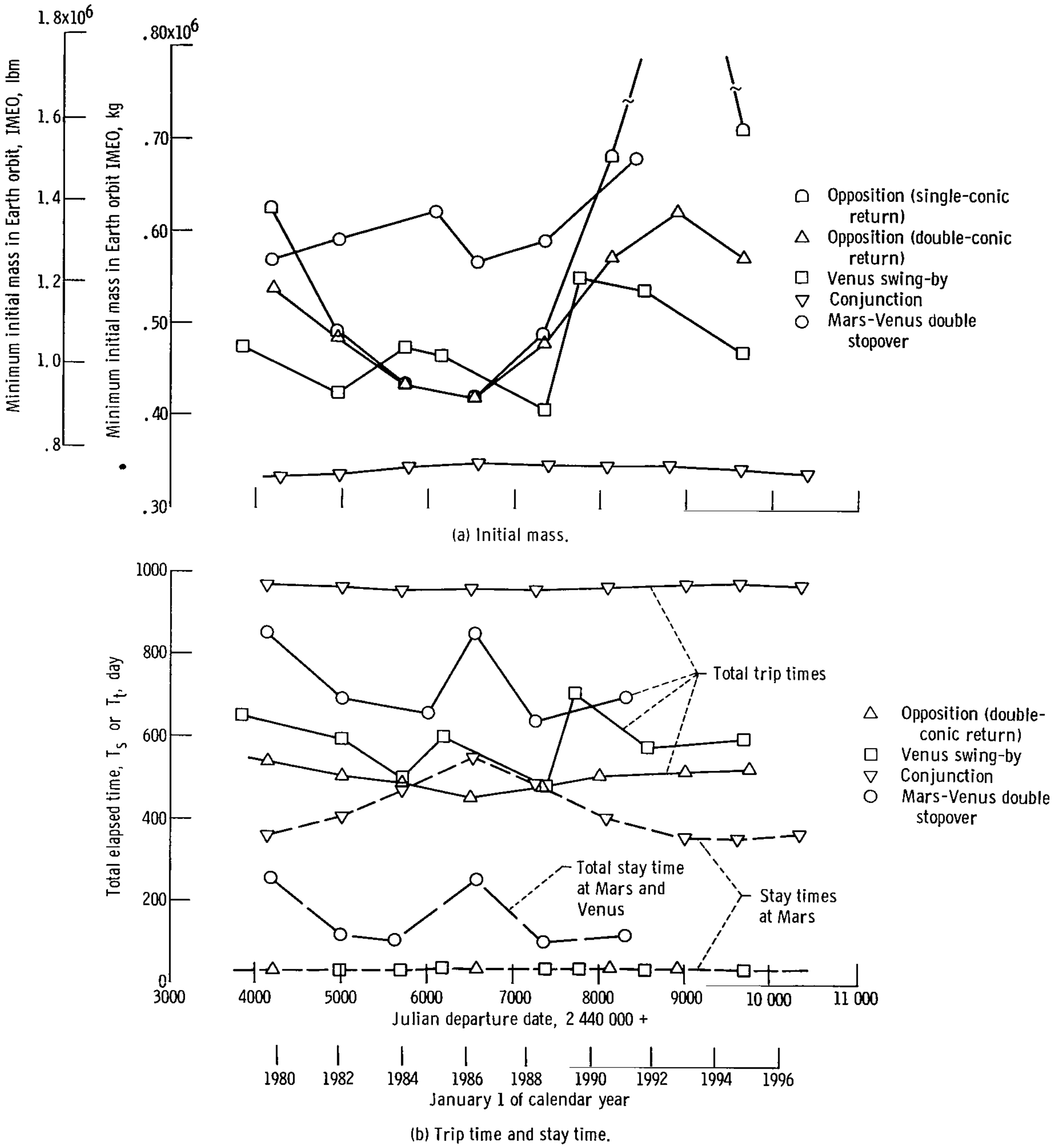
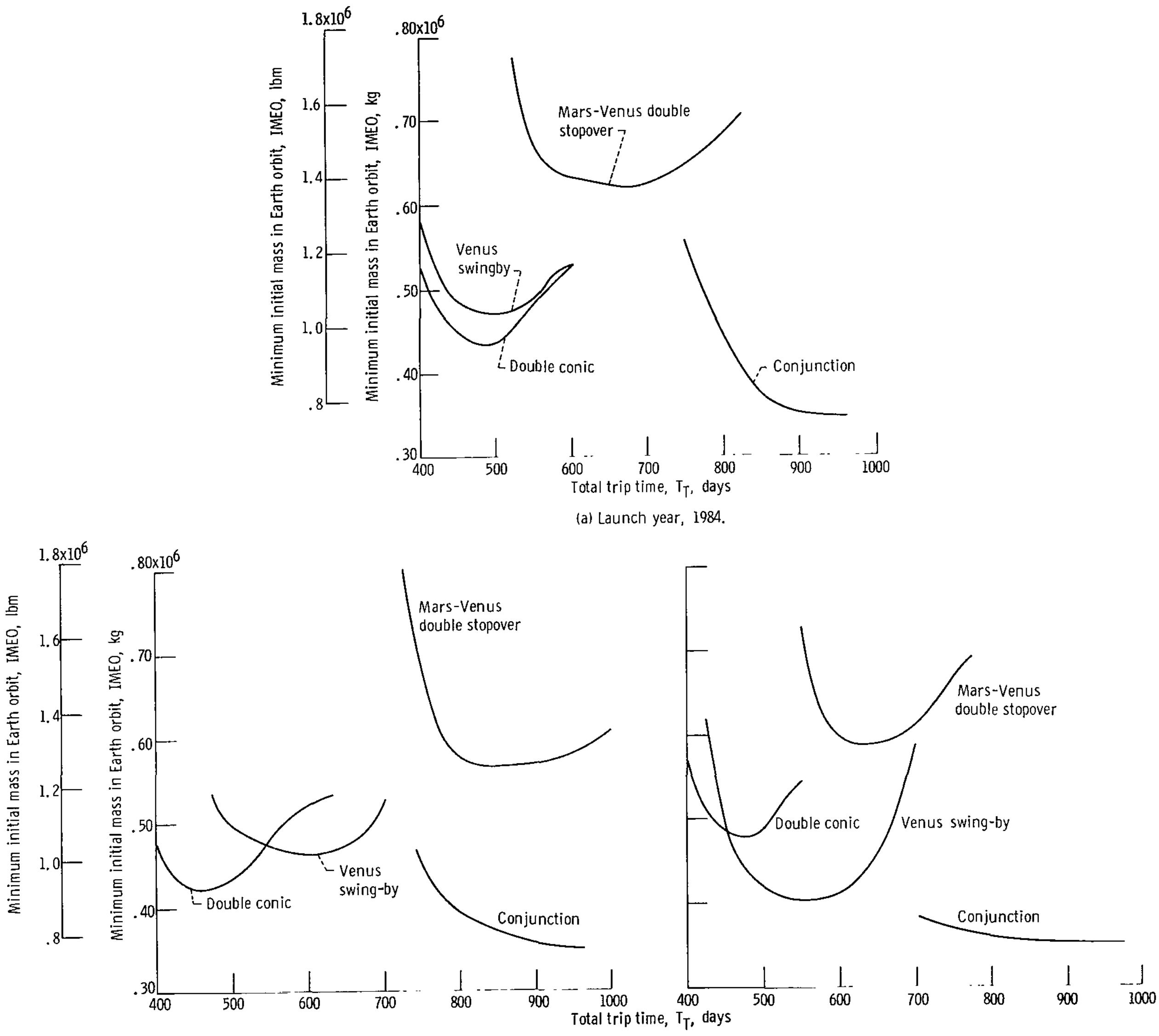


Figure 7. - Comparison of interplanetary trajectory modes. Single-engine vehicle configuration; NERVA-I propulsion. See table I for input data.



(a) Launch year, 1984.

(b) Launch year, 1986.

(c) Launch year, 1988.

Figure 8. - Effect of trip time on initial mass in Earth orbit. Single-engine vehicle; NERVA-I propulsion. See table I for input data.

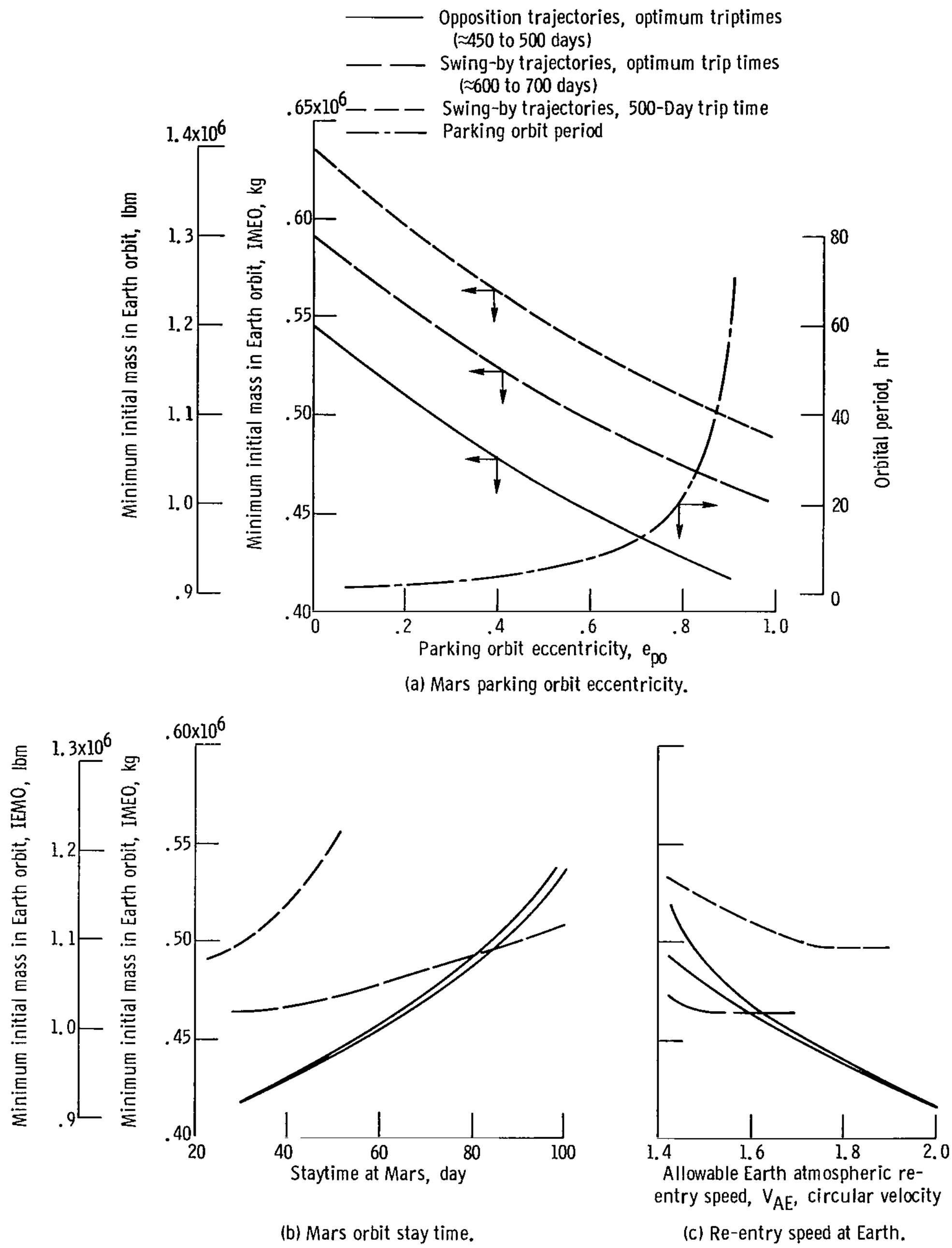


Figure 9. - Effect of trajectory perturbations on fast missions in 1986. Single-engine vehicle; NERVA-I propulsion. See table I for input data.

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