Mars Round-Trip Mission Analysis for the 1975–1985 Time Period

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The availability of desirable interplanetary transfers for manned missions to Mars in the 1975–1985 period is complicated by the large energy requirements associated with departures in the late 1970's and high solar activity with concomitant weight penalties experienced during the early 1980's. Promising missions, selected on the basis of minimum gross weight in Earth orbit, are available with long-duration class transfers (800 to 1000 days) during the entire period investigated but are restricted to the 1980's if short-duration (360–600 day) missions are desired. For the long missions, stay times of 300–400 days give minimum gross weight, whereas for the fast trips, stay time should be as short as practical (<40 days). The concept of multimission vehicle design is developed, and the vehicle's capability is determined for several synodic periods. This vehicle, with a constant structural weight and tank size, is capable of performing round-trip missions of various durations during several synodic periods. Weight-saving techniques, such as utilizing elliptical orbits at Earth departure and at Mars, planetary atmospheric braking, and high-speed Earth entry, have been evaluated with regard to feasibility, weight, and cost; weight savings are not sufficient to justify the complexities of such techniques.

Nomenclature

I_{sp} = specific impulse, sec
PL_i = payload weight of ith stage, kg
r_p = radius of periplanet, km
S_∞ = hyperbolic excess velocity, km/sec
t_m = total mission duration, days
t_s = stay time at planet, days
V_T = total round-trip velocity, emos (1 emos = 29,776 km/sec)
ΔV = incremental velocity, km/sec
W_E = burnout weight, kg
W_G = gross weight, kg
λ_i' = step propellant mass fraction of ith stage

Introduction

MANNED exploration of Mars is commonly considered the logical follow-up to the successful manned exploration of the moon. On the one hand, it has become increasingly clear that nuclear engines suitable for large-scale planetary exploration will not be available until the late 1970's or early 1980's, and on the other hand, this time period is "unfavorable" from an energy standpoint and from the concomitant shielding required due to increased solar activity. The present paper surveys the energy and gross weight requirements of vehicle systems for the manned exploration of Mars during the 1975–1985 period. The results indicate the magnitude of difficulty associated with these missions as well as possible means of alleviating this difficulty. The concept of multimission vehicle design is developed, and vehicle capability is summarized for several missions during this period.

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Velocity requirements are obtained using a Keplerian solution of the equations of motion[1-3] and a "patched conic" approach. The computer program developed allows simultaneous determination of interplanetary transfer parameters, total energy requirements, and resulting vehicle sizes for one-way and round-trip missions. A realistic solar system, using elliptical planetary motions with appropriate relative inclination of the orbit plane to the ecliptic, is used. Promising missions are determined using a baseline missions profile consisting of launch from a 325-km (175-mile) circular Earth orbit, capture and departure from a 555-km (300-mile) circular Mars orbit, and direct entry upon return to Earth at a velocity of 12.2 km/sec (40,000 fps). Mission durations of 360, 460, 560, 720, and 880 days are examined. Stay times are varied from 0 to 60 days for short-duration missions and up to 400 days for long-duration missions. This range of mission duration and stay times allows the effects of solar flare activity, biotechnological problems, environmental hazards, and energy requirements to be integrated for proper mission selection. Effects of communication and perihelion

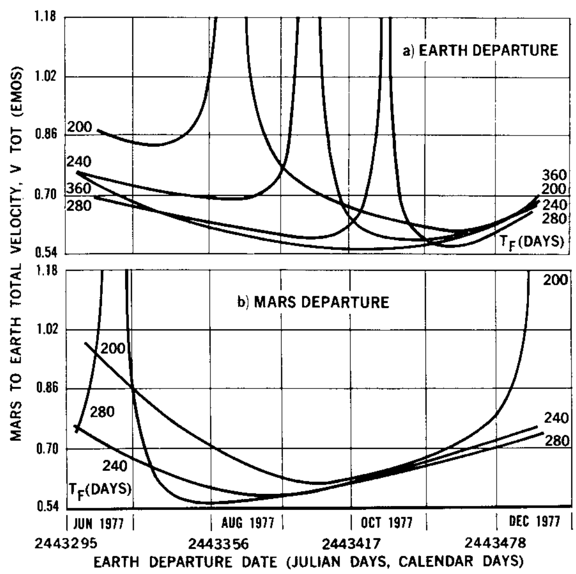

Fig. 1 Total velocities for 1977.

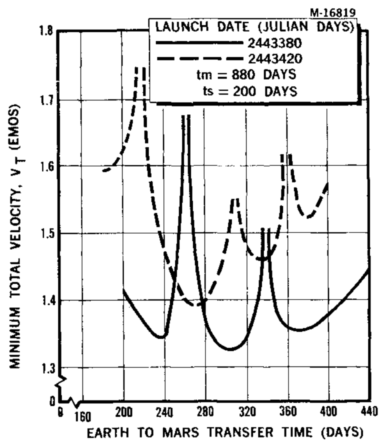

Fig. 2 Effect of transfer time on minimum total velocity, 1977.

passage distances during the transfer are considered. For one-way missions, only the Earth launch and Mars approach velocity requirements are needed to isolate desirable launch regions[4]; for round-trip missions, Mars launch and Earth approach velocities must also be included. The total velocity (the sum of Earth departure, Mars approach, Mars departure, and Earth approach velocities) is used to describe energy requirement as a function of Earth departure date, mission duration, stay time at the planet, and perihelion distance.

Mission Requirements

Earth-Mars and Mars-Earth trajectories for various outbound and return transfer times are shown in Fig. 1. This type of parametric variation has been obtained for the entire 1975–1985 period. Since two minimum velocities will be discussed, the differences between them should be noted. The daily minimum velocity is determined from combinations of outbound and return transfer times for a given mission duration and stay time on a selected Earth departure date, as shown in Fig. 2. Cross-plotting of these daily minimums vs Earth departure date gives the minimum velocity for the selected mission duration, as shown in Figs. 3 and 4 for the 1977 and 1983 synodic periods. A comparison of these figures indicates that the minimum total velocity¶ drastically varies with mission duration, Earth departure date, and synodic period. In Figs. 3 and 4, the perihelion passage distance is restricted to values greater than 0.6 a.u. Figure 5 shows that this restriction eliminates the use of desirable low-energy regions (indicated by the cross-hatched area). The

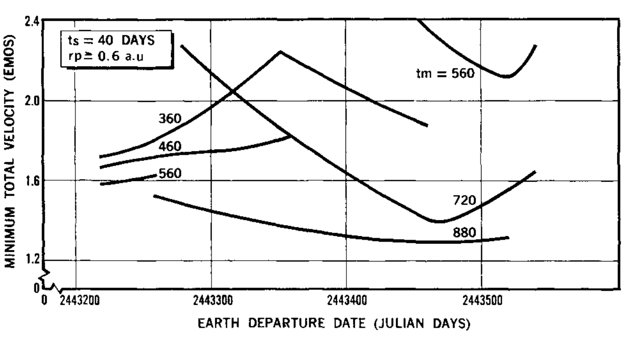

Fig. 3 Minimum total velocity for various mission durations, 1977.

¶ The velocity values shown are not incremental velocity requirements. To obtain incremental velocity requirements, subtract 0.75 emos from each value to account for parking orbit, capture orbit, and entry velocities used in the mission profile.

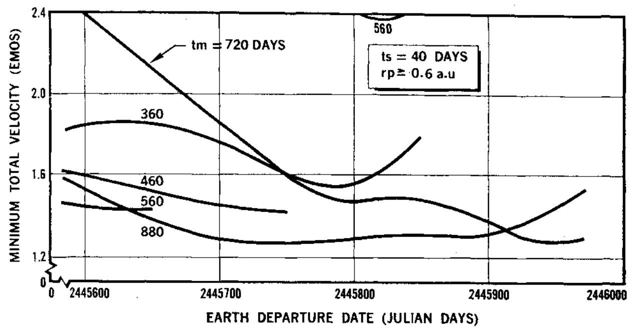

Fig. 4 Minimum total velocity for various mission durations, 1983.

perihelion restriction was dropped when thermal protection penalties were determined to be small when compared with the weight saved using small perihelion values.

When velocity requirements are compared for the 1975–1985 period, one concludes that 1) requirements are highest in 1977 and lowest in 1985, 2) short-duration missions require more velocity than long-duration missions, 3) the 1981, 1983, and 1985 synodic periods generally have lower requirements than 1975, 1977, and 1979, 4) minimum total velocity requirements associated with various mission durations do not occur at the same Earth departure date, and 5) perihelion restrictions limit available minimum-velocity regions.

Vehicle Performance

Incremental velocities were obtained for each transfer and vehicle performance analysis conducted using a step propellant mass fraction (λ') of 0.85 and a specific impulse value of 850 sec for all steps. The achievable propellant mass fraction values were refined as more data became available and the vehicle systems became better defined. The sizing analysis used gross weight as the dependent variable. A constant 8000-kg payload package comprised of an Earth entry module (EEM) was assigned for all missions. A 25,000-kg Mars excursion module (MEM) transported a portion of the crew from the Martian capture orbit to the surface and back to the basic transport vehicle. The MEM was then left in orbit at Mars. A variable life-support system (LSS) was utilized to account for biotechnological system weight variations with mission duration. The crew was transferred from the LSS to the EEM prior to Earth approach, and the LSS unit was jettisoned. Using the foregoing model, the initial gross weight at Earth departure for a four-stage vehicle was determined from the relationship

$$W_G = \frac{\lambda_1'\lambda_2' W_{MEM}}{(\lambda_1' - 1 + e^{-C_1})(\lambda_2' - 1 + e^{-C_2})} + \frac{\lambda_3' W_{LSS}}{(\lambda_3' - 1 + e^{-C_3})} + \frac{\lambda_4' PL_4}{(\lambda_4' - 1 + e^{-C_4})}$$

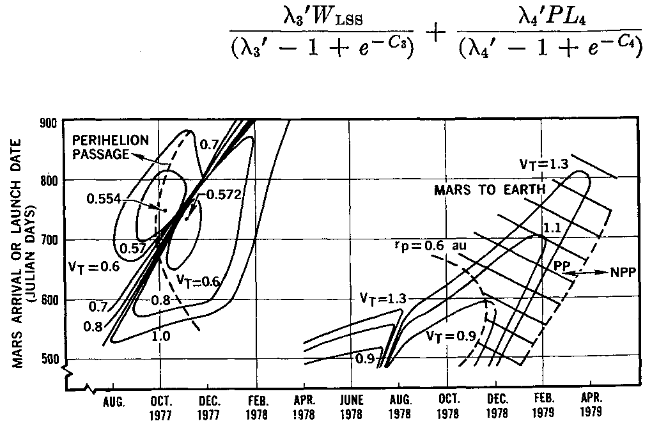

Fig. 5 Interplanetary transfer analysis, 1977.

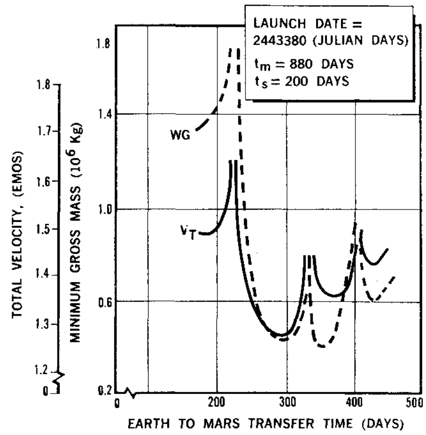

Fig. 6 Variation of velocity and mass with transfer time, 1977.

where

$$C_i = \Delta V_i/gI_{spi} \qquad i = 1, 2, 3, 4$$

$$\lambda_i' = W_{Pi}/W_{Gi} - PL_i$$

$$\Delta V_i = gI_{spi}\log_e(W_{Gi}/W_{ei})$$

The performance analysis (for the previously determined minimum total velocity regions) indicated that the vehicle gross weight in Earth orbit is strongly affected by the stage velocity distribution associated with each mission. Therefore, the minimum total velocity requirement does not necessarily yield minimum gross weight ($W_{G_{min}}$), as seen in Fig. 6. When the stage velocity distribution is analyzed, it becomes evident that the magnitude of Earth departure velocity is most influential in determining $W_{G_{min}}$. As a result, $W_{G_{min}}$ generally corresponds to minimum first-stage weight.

Figures 7 and 8 depict the variation in $W_{G_{min}}$ with Earth departure date for selected mission durations in 1977 and 1983. These data were generated for a stay time of 40 days. The short missions generally require a larger gross weight than do the long missions. As was previously mentioned, the restriction of perihelion passage to 0.6 a.u. greatly curtails the launch window and eliminates many attractive launch dates. When this restriction is removed, noticeable weight reductions are realized, as shown by the dashed curves in Figs. 7 and 8; for 1977 (Fig. 7; also true in 1979), $W_{G_{min}}$ is sharply reduced for the short-duration missions, thereby increasing the number of launch dates available for a given mass in orbit. This effect on short missions is much smaller during the 1981–1985 period. As illustrated in Fig. 8 for the 1983 synodic period, the r_p restriction has no effect on the absolute minimum gross weight, but the number of launch dates available is noticeably increased. For the longer mission durations (720–880 days), the r_p restriction has no effect in any synodic period. It may be of interest to note that the perihelion passage distance encountered for some of the absolute $W_{G_{min}}$ missions was as low as 0.4 a.u. The gross weights shown in the dashed regions on the plots do not account for the additional thermal insulation that would be required for close solar approach; however, spot checks of thermal requirements indicate that large weight savings are still obtained.

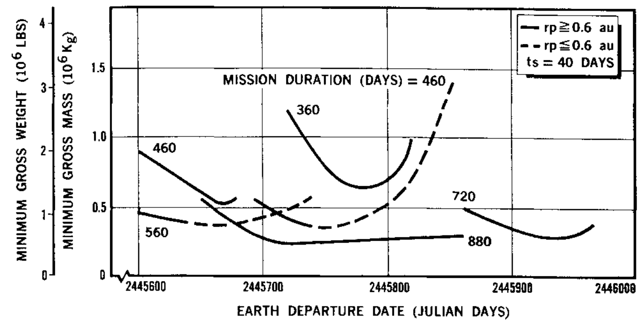

Fig. 8 Minimum gross weight for various mission durations, 1983.

Having established a preliminary set of weight requirements based on a 40-day stay time at Mars (t_s), the effect of t_s is next examined; $W_{G_{min}}$ varies, since the entire transfer at a given mission duration is altered. Figure 9 shows that, for a 460-day mission in 1977, the $W_{G_{min}}$ for a zero stay time is one-half that for a 40-day stay. Relaxation of the perihelion restriction results in an additional 25% weight reduction for the zero-stay case, and the effect of t_s is less pronounced. Long-duration missions (Fig. 10) react in just the opposite manner, t_s = 300 days is desirable from a weight standpoint. For these missions, the one-way transfer time of the long-duration mission is still in the order of 220 days, but now the return leg is made in another synodic period in which desirable low-energy transfers can be obtained. Perihelion restrictions do not affect the long-duration missions at all. Figure 11 summarizes these effects for the various missions investigated in 1977 and 1983. (Similar graphs have been developed throughout the 1975–1985 period.) In general, gross weight increases significantly with t_s when t_m = 560 days; for 720-day missions, there is still a small adverse effect of stay time; but for 880-day missions in 1977, an opposite (i.e., favorable) effect of t_s is seen.

The effect of synodic period on gross weight is shown in Fig. 12. The points for the particular mission durations investigated are connected by straight lines rather than faired curves because data were generated for small intervals in mission duration only in 1981. The effect of synodic period is very noticeable for short missions, the later years having much lower weight requirements. As previously indicated, the weight requirements for long missions are rather constant throughout the years.

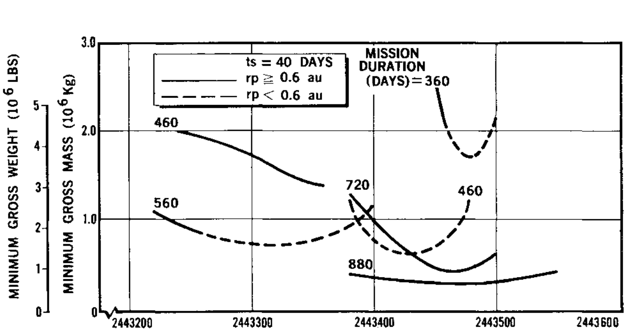

Fig. 7 Minimum gross weight for various mission durations, 1977.

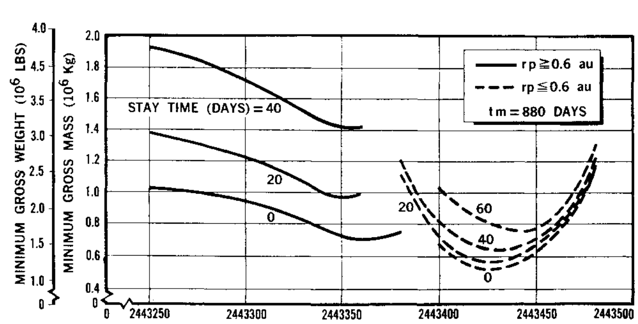

Fig. 9 Minimum gross weight for various stay times, 1977.

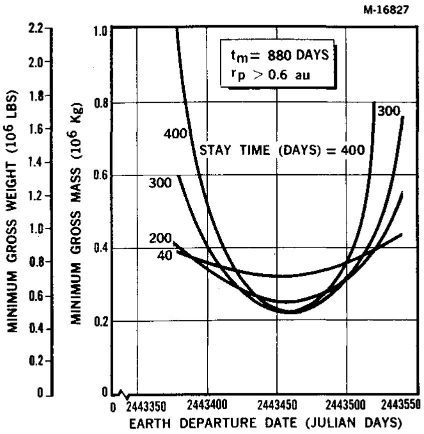

Fig. 10 Minimum gross weight for various stay times, 1977.

Weight-Saving Techniques

Various techniques might provide improved vehicle performance or a saving of initial weight in Earth orbit. Several of these techniques have been investigated[5,6] with regard to their effectiveness as an integral part of a vehicle system. Aerodynamic braking during Mars entry may be used in two ways: 1) to establish the capture orbit for a system that uses an excursion module for the landing phase, or 2) to permit direct landing, thus eliminating the excursion module and the associated rendezvous. Aerodynamic orbit attainment proved to be more beneficial. Allowing for an increase in thermal protection, a reduction of up to 20% in W_G in Earth orbit was still possible, whereas direct Mars entry resulted in doubling the W_G requirement.[5]

The use of elliptic capture orbit at Mars is another possibility. If the periplanets of the approach and departure orbits are coincident, large weight savings may be realized with the use of elliptic capture orbits. However, because of planet oblateness this is generally not possible. When nodal regression and apsidal advance are accounted for, the weight savings are relatively small, amounting to about 6% for an apoplanet to periplanet ratio of 2 to 4.[5]

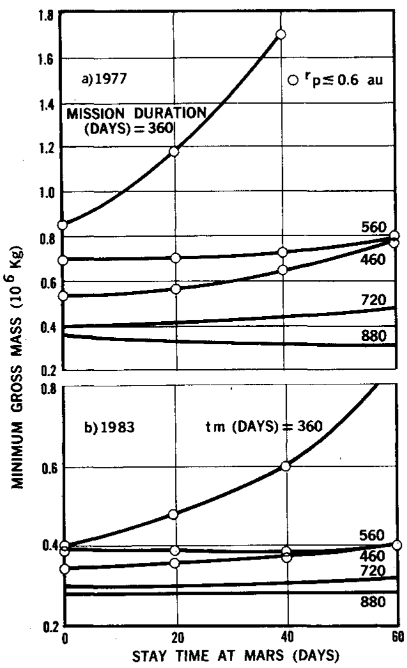

Fig. 11 Minimum gross weight for various mission durations in 1977 and 1983.

It will be recalled that the baseline mission profile placed a limit on the Earth entry velocity of 12.2 km/sec. For missions where the approach velocity was higher, a propulsive stage would be used to reduce the approach velocity. Since a weight savings in the fourth stage reflects very strongly on the W_G required in Earth orbit, the feasibility of direct entry at Earth with higher velocity was examined. Using a vehicle with an L/D ratio of 1.3 and a composite material (phenolic refrasil and low-density, high-temperature insulation) for thermal protection, velocities of up to 18.3 km/sec were investigated. This variation on the baseline mission profile proved most attractive for short mission durations in the late 1970's. Savings of up to 24% in Earth orbit weight could be realized.[6]

The largest weight savings were obtained with a mass fraction improvement scheme that discards the meteoric and thermal insulation shield of each stage just prior to stage ignition. A performance gain of 38% is possible with this technique.[6] Other techniques investigated[5] included utilizing lunar gravitational attraction to perturb the trajectory, lunar-based vehicles, elliptical departure orbit at Earth, orbital altitude decay, midcourse plane change, and other advanced concepts. Although weight savings can be obtained with several of these methods, all add complexity to the system. Hence, it was concluded[6] that the original baseline profile offers the most desirable system for first-generation Mars landing operations.

Multimission Vehicles

The weight-savings potential of several techniques (some of which are briefly discussed) were carefully analyzed with regard to weight saving, program cost, mission risk, and system complexity.[5] It was concluded that 1) the original baseline profile is preferred because of its relative simplicity without large penalties in weight, and 2) for over-all economy, vehicles should be developed which are capable of being launched at various dates on various missions and during several synodic periods.

A multilaunch vehicle concept was investigated for two missions: 1) a 460-day mission with 20-day stay at the planet and 2) an 880-day mission with a 300-day stay. Realistic estimates of step propellant mass fraction, engine size, tankage, etc., were used for all weight comparisons.[6]

Figure 13 shows the W_G variation for a 40-day launch window in each synodic period for these two typical mission durations. (The weights reflect the velocity requirements, of course.) The propellant tanks were sized for the maximum load within the 40-day window and were off-loaded as required for any given launch date within this window. When a short mission is considered, the 1977 synodic period is the

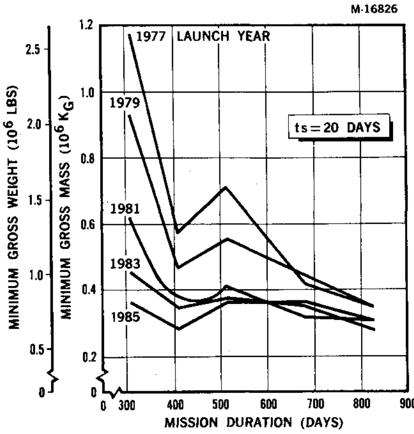

Fig. 12 Minimum gross weight for various launch years, 1977–1985.

most demanding; a vehicle designed for a 460-day mission in 1977 would be greatly oversized for a mission in a later synodic period, but it would offer much larger mission windows. For the 880-day mission, $W_{G_{\min}}$ varies only slightly with synodic period, since the energy requirement is nearly constant.

The multilaunch vehicle size can be further reduced if a smaller mission window is selected for the 1977 period. The vehicle size shown by a point in Fig. 14 was obtained by using the minimum-gross-weight vehicle in 1977 with a 20-day mission window; the vehicle has a multilaunch capability, as shown by the available mission windows for other synodic periods. Although the vehicle has decreased in size by 23%, it has some capability in every period. Similar results for the 880-day mission are shown in Fig. 15. One vehicle configuration was used throughout all periods with propellant off-loaded as required. Since energy requirements are nearly the same for all of these periods, only small variations in W_G are discernible.

It is evident from the results presented that multilaunch capability is easily attained for long-duration missions because little variation in vehicle size is experienced throughout the synodic periods investigated. Since the short-duration missions are very sensitive to synodic period, the multilaunch vehicle is not as easily selected, but the 23% decrease in weight obtained by decreasing the 1977 launch window from 40 days to 20 days is quite significant.

Attainment of multimission capability is further complicated by the stage velocity differences between the fast and slow trips. In general, both fast and slow transfers have large third- and fourth-stage velocity requirements when the first and second stages are moderately small, and vice versa. Since the third- and fourth-stage minimum and maximum velocity requirements are usually larger for fast trips, one must size the vehicle to the maximum velocity requirement for the fastest trip desired as dictated by the mission window constraints. Structural weight can be saved, however, by sizing the third and fourth stages using the maximum velocity requirement of these stages and then off-loading propellant in these upper stages when the first and second stages are sized. This method pays off for the long-duration missions, since the velocity differences between the upper and lower stages are quite large. Few, if any, structural weight savings occur when a multimission vehicle is desired, since these velocity differences are much less pronounced for the shorter missions.

Additional weight savings can be obtained by eliminating entirely the 1977 region as a launch period. The validity of such action is dependent upon the state of the art of various technologies, development time, program cost, and mission success probability associated with obtaining manned Mars landing missions by the 1977 time period. These factors suggest that 1979 is probably the earliest period of interest as far as manned Mars landings are concerned. However, if development programs are initiated in the near future, launches in the 1977 period can be obtained with a reasonable probability of success. Of primary importance is the fact that multilaunch vehicles can be obtained as a result of compromises made with regard to energy, cost, size, and time.

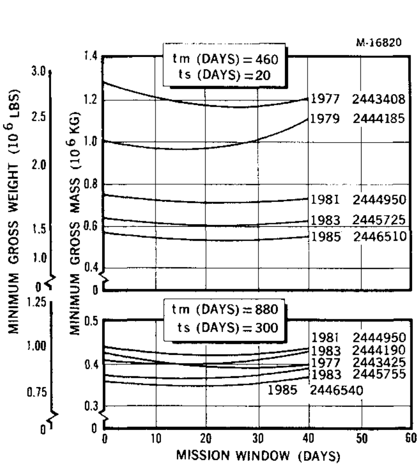

Fig. 13 Multimission performance, 1977–1985.

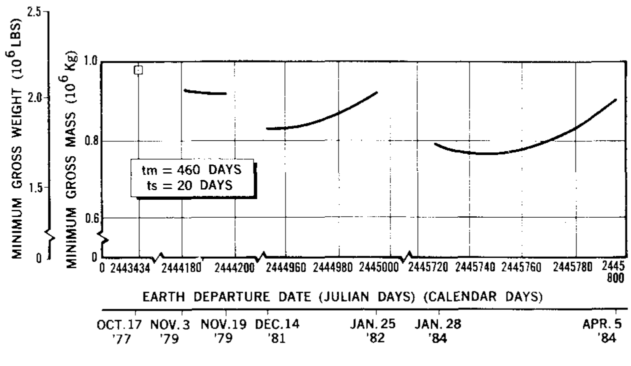

Fig. 14 Multimission performance, short missions, 1977–1984.

Conclusions

The terminology "unfavorable period" associated with the launch years of 1975–1985 has some validity, particularly in the late 1970's, because energy requirements are higher then. The 1977 synodic period requires the largest W_G in Earth orbit for most missions. Long-duration missions are, however, comparable in weight with other years.

Long-duration missions correspond to low energy transfers and consequently to low gross weights. The synodic period has little effect on the W_G requirements. Short-duration missions are highly dependent on synodic period, since departure and return are accomplished in the same period. The astronomical configuration decidedly affects the energy requirements, resulting in appreciable weight variations throughout the years.

Relaxation of perihelion passage distance results in large energy savings for short mission duration (350–560 days), particularly during the late 1970's. Long missions are virtually unaffected throughout all periods.

Gross weights of short-duration missions are strongly dependent on stay time at the planet as well as synodic period. Appreciable weight decreases are experienced with small stay times for most short-duration missions. Long missions are very insensitive to stay time, because the return transfer can be made during the following synodic period at more favorable return dates. Minimum gross weights for the long-duration missions are achieved for stay times in the order of 300–400 days, depending on the synodic period.

Large weight penalties result when the same vehicle is to be used for several synodic periods and must perform a short

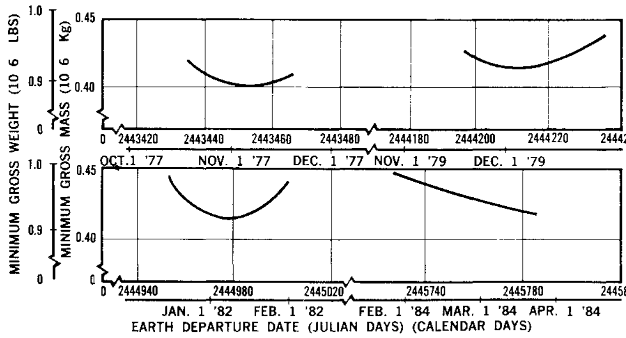

Fig. 15 Multimission performance, long missions, 1977–1984 (t_m = 880 days, t_s = 300 days).

mission, but very small weight penalties result when the same vehicle is to be used for a long mission in various synodic periods.

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