Technical Report No. 32-464

The Determination and Characteristics of Ballistic Interplanetary Trajectories Under the Influence of Multiple Planetary Attractions

Michael A. Minovitch

T. W. Hamilton

T. W. Hamilton, Chief
Systems Analysis Section

JET PROPULSION LABORATORY
CALIFORNIA INSTITUTE OF TECHNOLOGY
PASADENA, CALIFORNIA

October 31, 1963

amounts, it is possible to reduce these high return energies. The flight times are short because the Venus–Mars and Mars–Earth transfers are both Type I, while the Earth–Venus transfers are Type II. For the 1970 trajectories this situation is reversed.

Although the energy requirements for these trajectories are somewhat higher than those of the previous period, these missions would be ideally suited for manned vehicles. The planetary configuration for these manned reconnaissance missions appears in Fig. 33. Figures 34 and 35 describe the May 27 trajectory as it passes Venus and Mars.

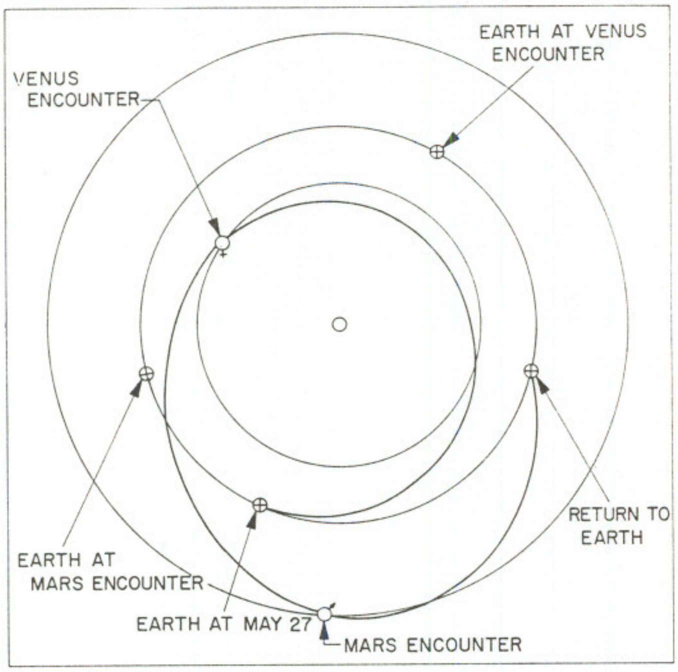

Fig. 33. Planetary configuration for Earth–Venus–Mars–Earth, 1972 (May 27 trajectory)

When more attention is directed toward manned interplanetary flights of the near future, the above reconnaissance trajectories of the 1970 and 1972 launch periods should warrant serious consideration.

5. Manned Landings on Venus and Mars Utilizing Multiple Planetary Trajectories (The *Saturn* 5 Possibility)

We shall now give another example of how multiple planetary trajectories can be applied to early manned interplanetary missions. In particular, let us consider the trajectories for manned landings on Venus and Mars. Tables 18 and 19 describe the optimum Venus–Earth and Mars–Earth transfer trajectories, respectively. If we compare these Tables with Tables 1 and 3, we find that the

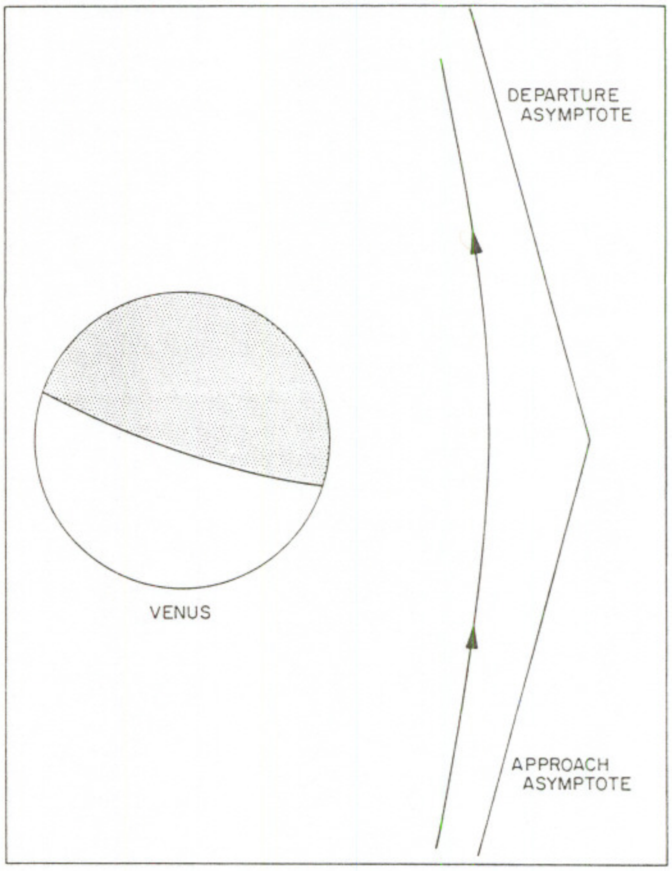

Fig. 34. May 27, 1972, Earth–Venus–Mars–Earth trajectory at Venus intercept

optimum Venus–Earth return trajectories occur 20 to 40 days after the optimum Earth–Venus departure trajectories. The optimum Mars–Earth return trajectories occur 40 to 60 days *before* the optimum Earth–Mars departure trajectories!

A brief study of manned missions to Venus and Mars during 1970–1975 was carried out to obtain some characteristic properties of the most favorable trajectory profiles. The results appear in Tables 20 and 21. The symbol ΔT represents the number of days spent on Venus or Mars. One immediately notices the unusually high hyperbolic excess velocities associated with the return Mars missions. This is a characteristic property of all missions to Mars of this type where only one vehicle is involved. These high velocities occur because the Mars launch date will always fall much later than the optimum launch dates for the Mars–Earth transfers. Thus, in general, only a relatively small amount of time can be devoted to the exploration of Mars. These properties are also generally true for the mission to Venus. It is important to note, however, that it is not always advantageous, from an

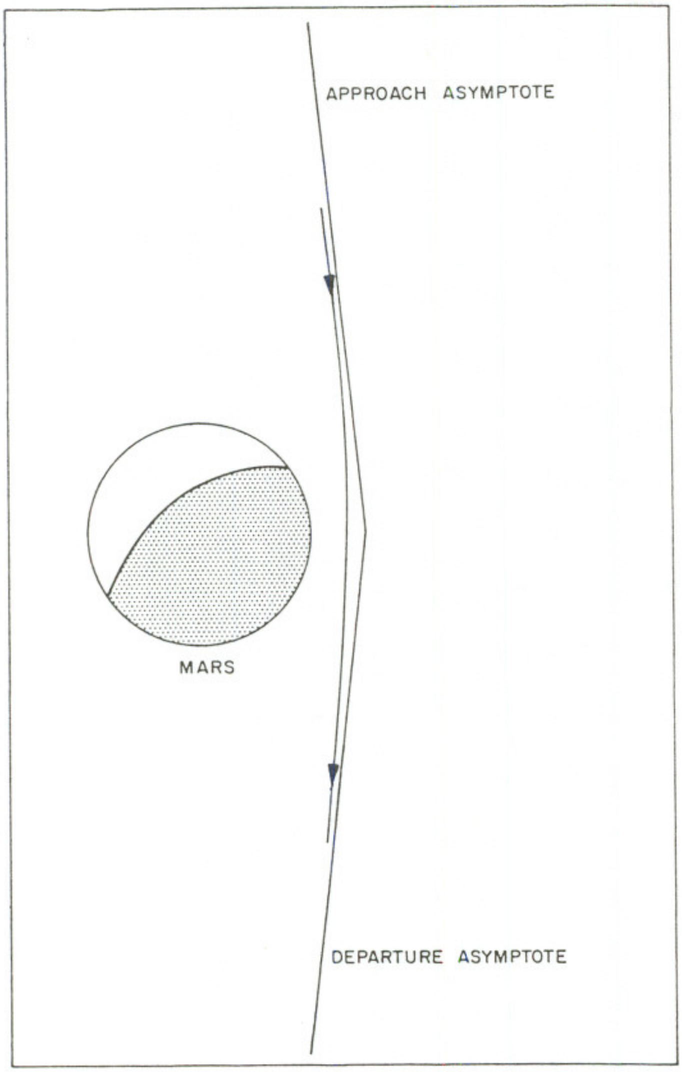

Fig. 35. May 27, 1972, Earth–Venus–Mars–Earth trajectory at Mars intercept

energy point of view, to start the return trip as soon as possible. For example, the Venus launch energies for Type II Venus–Earth trajectories actually decrease from December 22, 1970, to January 15, 1971, although the optimum launch date is October 21, 1970. This situation does not appear in the Mars missions. Type I Venus–Earth and Mars–Earth transfers require much higher launch energies than the Type II return trajectories and hence are of little interest.

Now it may appear that the manned exploration of Mars will require more powerful launch vehicles than those required for the Venus missions; however, since the escape velocities of Venus and Mars are 10.40 and 5.03 km/sec, respectively, the missions to Venus will actually require the more powerful rockets. In view of the low Mars escape velocity, it appears that, if we can find low Earth launch-energy trajectories that will take a vehicle to Mars such that the Mars arrival date is sufficiently close to the favorable Mars–Earth launch dates, it may be possible to conduct manned missions to Mars before nuclear rockets become available. *Such trajectories do indeed exist.* As a matter of fact, we have already calculated some – the Earth–Venus–Mars trajectories. By utilizing these trajectories as our Earth-to-Mars transfer trajectories, it may be possible to save a great deal of energy. Table 22 contains trajectory profiles of the 1971 and 1973 manned missions to Mars using Earth–Venus–Mars trajectories as the Earth-to-Mars transfers. These trajectories clearly require far less weight-lifting capability of the primary Earth launch vehicle. Unfortunately, the Earth–Venus–Mars trajectories of 1973 and of 1975 require much longer flight times and, hence, are unattractive Earth-to-Mars transfer trajectories.

Although the alternative trajectory profiles given in Table 22 represent a considerable improvement over the conventional trajectory profiles given in Table 21, large *Nova*-type launch vehicles, along with advanced nuclear propulsion systems, will still be necessary. These large launch vehicles will probably not be available before 1973. *There is, however, a way that would permit a manned exploration of Mars using a conventional non-nuclear Saturn 5 launch vehicle.*

Recent studies regarding manned interplanetary missions to Mars show that a Mars landing can be carried out with almost total atmospheric braking (see Ref. 6 and 7). These studies show that the dry weight of the smallest possible vehicle of this type (including the heat shield) would be approximately 13,000 lb. This vehicle, called a Mars Excursion Module, could accommodate a crew of four men. It has also been calculated that the weight of an interplanetary mission module carrying all the life-support equipment and related supplies to last a crew of five for a 1-yr period would be in the neighborhood of 40,000 lb. It was also determined that an *Apollo*-type (six-man) Earth re-entry vehicle re-entering at 13.3 km/sec and employing total atmospheric braking would weigh approximately 13,180 lb.

Before attempting an actual manned landing mission, it would probably be desirable first to carry out a manned Mars fly-by mission. This mission could provide an ideal means of testing man's endurance under prolonged interplanetary space flights. For this reason, the August 12, 1970 Earth–Venus–Mars–Earth trajectory appearing in Table 16 may be highly suited for this purpose. Let us

Table 18. Some important properties of optimum Venus–Earth transfer trajectories

Launch period	Min. HEV_1 km/sec	Max. HEV_1 km/sec	Min. T_{12} days	Max. T_{12} days	Min. HEV_2 km/sec	Max. HEV_2 km/sec
	Type I					
8/6/70 –10/17/70	3.21	6.15	94	122	3.65	5.87
3/24/72– 5/21/72	4.04	6.32	92	128	3.27	7.91
11/6/73 – 1/15/74	4.12	6.38	96	162	2.71	7.60
	Type II					
7/15/70– 3/22/71	3.47	5.93	152	348	2.85	10.48
3/8/72 –10/20/72	3.13	5.78	174	348	3.04	9.77
10/17/73– 5/9/74	2.96	5.58	166	330	3.59	10.04

Table 19. Some important properties of optimum Mars–Earth transfer trajectories

Launch period	Min. HEV_1 km/sec	Max. HEV_1 km/sec	Min. T_{12} days	Max. T_{12} days	Min. HEV_2 km/sec	Max. HEV_2 km/sec
	Type I					
7/7/70 – 7/8/71	2.55	5.57	172	360	3.05	9.02
12/21/72– 8/14/73	3.53	5.55	168	296	2.93	4.93
	Type II					
3/3/70 – 9/22/71	2.75	5.48	242	508	3.45	9.00
3/22/72–11/20/73	3.71	5.61	260	516	3.00	11.56

now perform for this trajectory some rough calculations based on the before-mentioned vehicle design parameters.

We recall the well-known formula

$$\frac{M_1}{M_2} = \exp\left(\frac{\Delta V}{c}\right)$$

where M_1 and M_2 are the masses of the vehicle before and after obtaining a velocity change of ΔV km/sec, using a rocket engine with an exhaust velocity of c km/sec. In carrying out our calculations we shall assume simple single-stage burning with an exhaust velocity of 3.6 km/sec (i.e., $I_{sp} = 367$ sec). Now, assuming a crew of three or four, the weight of the primary interplanetary mission module could be taken to be 43,000 lb. Since our trajectory's Earth approach hyperbolic excess velocity is 9.34 km/sec and since the Earth's escape velocity is about 11.0 km/sec, the re-entry module would re-enter the Earth's atmosphere at about

$$\left(9.34^2 + 11.00^2\right)^{1/2} \text{ km/sec} = 14.43 \text{ km/sec}$$

Thus, we shall employ partial rocket braking to permit our 13,000-lb module to re-enter the Earth's atmosphere at 13.3 km/sec. Consequently, the mass of the module before retro will be

$$13{,}500 \exp\left(1.13/3.6\right) \text{ lb} = 18{,}400 \text{ lb}$$

where we assume the weight of the retro engine is 500 lb. Hence the total weight at the beginning of the fly-by mission will be about 62,000 lb. The *Saturn 5* launch vehicle will have the capacity of sending 90,000 lb on an

Table 20. Some important properties of near-optimum trajectory profiles for manned exploration of Venus

Launch date	HEV_1 km/sec	T_{12} days	HEV_2 km/sec	ΔT days	Venus launch date	HEV_2 km/sec	T_{23} days	HEV_3 km/sec	Arrival date	Total time
Aug. 18, 1970	2.91	116	5.43	32	Jan. 13, 1971	4.81	302	8.65	Nov. 11, 1971	450
Mar. 26, 1972	3.50	112	6.16	20	Aug. 5, 1972	4.59	284	8.51	May 16, 1973	416
Nov. 10, 1973	3.66	106	4.89	14	Mar. 10, 1974	4.49	284	8.34	Dec. 19, 1974	404

Table 21. Some important properties of near-optimum trajectory profiles for manned exploration of Mars

Launch date	HEV_1 km/sec	T_{12} days	HEV_2 km/sec	ΔT days	Mars launch date	HEV_2 km/sec	T_{23} days	HEV_3 km/sec	Arrival date	Total time
May 19, 1971	3.53	135	5.52	9	Oct. 10, 1971	5.78	264	9.86	June 30, 1972	408
July 27, 1973	3.90	175	3.62	9	Jan. 27, 1974	6.81	258	15.77	Oct. 12, 1974	442

Table 22. Some important properties of alternative trajectory profiles for manned exploration of Mars

Launch date	HEV_1 km/sec	T_{12} days	HEV_2 km/sec	ΔT days	Mars launch date	HEV_2 km/sec	T_{23} days	HEV_3 km/sec	Arrival date	Total time
Aug. 12, 1970	3.26	309	6.75	19	July 6, 1971	4.19	246	6.44	Mar. 8, 1972	574
June 4, 1972	4.32	335	6.19	61	June 19, 1973	3.54	190	3.42	Dec. 26, 1973	586

escape trajectory (parabolic); thus this rocket will be able to send approximately

$$90{,}000 \exp\left[\frac{11.00 - \left(11.00^2 + 3.26^2\right)^{1/2}}{3.6}\right]\text{lb} = 78{,}300\ \text{lb}$$

on the required trajectory! The *Saturn 5* should be a highly reliable standard launch vehicle by 1970.

We shall now describe the method by which a Mars-landing mission can be carried out using only the standard non-nuclear *Saturn 5* launch vehicle. To carry out this mission, two vehicles will be used. The first, which we shall denote by A, shall be a simple Earth–Venus–Mars–Earth fly-by vehicle. The second vehicle, denoted by B, shall be launched on an Earth–Venus–Mars trajectory, and, in place of an Earth re-entry module, it shall carry a small Mars excursion module. The mission profile consists of launching A and B on particular trajectories that will bring B to Mars several days before A makes its Mars closest approach. During this time the crew of B can be exploring the surface of Mars. Then, as A begins to make its closest approach, the crew of B launches their small Mars excursion vehicle to rendezvous with A whereupon they abandon the excursion module to complete the journey back to Earth in A.

As in the case of the 1970 fly-by mission, we shall consider a definite trajectory profile. The particular trajectories that we shall choose do not appear in any of the tables given in this Report. They have, however, been carefully calculated. The Earth–Venus–Mars trajectory that we select for B has the following important characteristics:

T_1 = May 31, 1972

T_2 = November 17, 1972

T_3 = May 12, 1973

HEV_1 = 4.27 km/sec

DOCA_2 = 9,223 km

HEV_3 = 6.03 km/sec

T_{12} = 170.00 days

T_{23} = 175.65 days

TFT = 345.64 days

The Earth–Venus–Mars–Earth fly-by trajectory that we select for A is described by the following characteristics:

T_1 = June 4, 1972

T_2 = November 19, 1972

T_3 = May 23, 1973

T_4 = October 17, 1973

HEV_1 = 4.33 km/sec

DOCA_2 = 9,164 km

HEV_3 = 5.97 km/sec

DOCA_3 = 609 km

HEV_4 = 9.51 km/sec

T_{12} = 167.56 days

T_{23} = 185.44 days

T_{34} = 146.67 days

TFT = 499.67 days

For the re-entry profile associated with this trajectory, a retro thrust must be applied, as in the previous case, to dissipate about 1.23 km/sec to enable the re-entry to take place at 13.3 km/sec. Consequently, before retro, the module will weigh approximately 19,000 lb. Since the interplanetary mission module of A will be occupied by a crew of only one or two persons during the first 353 days, which represents about 70% of the total flight time, we can take its weight to be about 41,000 lb. The *Saturn 5* will be able to send about 71,700 lb on this trajectory. This is 11,700 lb more than required.

Now just before the Mars excursion module is launched from the surface of Mars to rendezvous with A, it is literally stripped of all equipment (heat shield, etc.) that is not absolutely necessary to effect a successful rendezvous. Thus, at the moment it joins A, we may assume that it weighs 12,000 lb. Consequently, prior to launch it would weigh

$$12{,}000 \exp\left[\frac{(5.03^2 + 5.97^2)^{1/2}}{3.6}\right]\text{lb} = 106{,}500\ \text{lb}$$

Assuming that 1.5 tons of supplies and equipment are left on Mars, we find that B's weight at the beginning of its flight will be approximately 145,000 lb. The *Saturn 5* will be able to send about one-half of this weight on B's trajectory. This means that only two *Saturn 5*'s would be necessary to send B on its required trajectory.

It should be stressed at this point that the trajectories which we have chosen for A and B are not the best ones available; more desirable ones do exist. It should also be pointed out that we have taken only the *minimum* launch capabilities of the *Saturn 5* as seen at the present time. This vehicle may develop a high growth potential: by 1972 it may be capable of sending payloads weighing considerably more than 90,000 lb on an escape trajectory. On the other hand, these calculations are optimistic in that many small items, which may have a significant cumulative effect, were omitted. Our main purpose for making these calculations is to show that the *Saturn 5* could be used as the launch vehicle for the first manned landing mission to Mars. No *Nova* launch vehicle or advanced propulsion system would be required.

The unusual mission profiles described for the above manned mission to Mars lead us to the last major topic of this Report.

C. Interplanetary Transportation Networks to Support Manned Bases on Venus and Mars

After the first flights to the inner planets, man will naturally construct bases on them. These bases, no matter when they are constructed, will naturally require means by which men and equipment can be taken to and from these bases. In the distant future when propulsion systems far more advanced than those currently being studied are developed, it will probably be possible to make interplanetary voyages, such as Earth–Mars transfers, with flight times as short as one or two weeks. But for the near future all interplanetary transfers will have to be made on near-optimum transfer trajectories with low departure and arrival hyperbolic excess velocities. Consequently, a great deal of life-support equipment will be necessary to transport a few persons from one planet to another. In short, cargo vehicles shall probably be robot-type vehicles carrying no equipment necessary for manned flight, while the manned vehicles will probably carry very limited amounts of cargo. In addition to carrying all the extra equipment for manned flight, these vehicles will also probably be required to be able to induce some artificial gravity. Hence, the transportation of just 10 men, for example, from Mars to Earth should ordinarily require a large and very expensive rocket. Methods of recovery will become a necessity. This problem of economics can be conveniently solved by constructing a long-lasting interplanetary transportation network designed for the sole purpose of transporting personnel from one planet to another.

Preliminary calculations have shown that if the planets P_i are restricted to Mercury, Venus, Earth, and Mars, where P_1 = Earth and $P_i \neq P_{i+1}$ for $i = 1, 2, \cdots, n$, it is possible to find sequences $P_1 - P_2 - \cdots - P_n$ such that the flight times $T_{i+1} - T_i$ are comparable to those required for optimum $P_i - P_{i+1}$ transfers. Moreover, many of these multiplanet trajectories were found to have very low launch energies.

The network can be established by first constructing the many large space vehicles that are to be used in the transportation system. This can be done by methods of prefabrication and orbital assembly. These vehicles can be designed to accommodate 20 to 60 persons, and, since artificial gravity will be highly desirable, the geometry of such a vehicle could resemble a torus with an outside diameter of perhaps 200 to 300 ft. When each individual vehicle is completed, one simply awaits its launch date T_1, when the vehicle (i.e., space bus) is injected into its prescribed interplanetary trajectory. This could be accomplished by convenient strap-on solid-propellant rocket engines. Each vehicle will carry extra provisions and life-support equipment to last until it makes its first Earth rendezvous, whereupon its supply can be replenished to last until it makes its second Earth rendezvous, etc. When such a vehicle approaches a planet P_i, a small excursion module orbiting P_i and containing a few men wishing to go to P_{i+1} can be injected onto an intercept trajectory with the space bus. Upon making rendezvous, the excursion module containing very accurate, automatic planetary approach guidance equipment could be left a few miles behind the space bus on an almost identical interplanetary trajectory to be used to transport the men from the space bus onto an orbit about P_{i+1}. A tanker vehicle, also equipped with accurate automatic planetary approach guidance and also following the space bus, could be used to refuel the planetary excursion modules. Other transportation systems could be established on each separate planet to provide a means for bringing the men from the orbiting excursion modules down to a planet's surface. The modules, however, could be left in respective circular orbits about the planets for future use. Tanker vehicles orbiting the planets could provide a means for refueling the excursion modules, which never actually land on a planet.