intermediate orbit other than $r_2$, then all that is known of the optimum scheme (see corollary 3) is that it is comprised of a set of nested monotone sequences. For instance, the outermost orbit must be visited in the first monotone sequence. If $r_2$ is not selected in the first sequence, then it must be visited in the second sequence, etc. Figure 1 shows one allowable path between an extreme orbit ($r_1$) and an intermediate orbit ($r_3$). Note that each monotone sequence has the maximum possible swing.

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# Optimization of Interplanetary Stopover Missions


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**A method of interplanetary trip selection is described, using a 10-day orbital stopover mission at Mars as an example. Analysis of numerous possible interplanetary orbits integrated with transportation system design established a set of minimum-mass requirements on Earth parking orbit. The selection of best trips is shown to rely on the relationship between the parameters in transportation system design and mission trajectories. Corresponding trajectory information uniquely defines a best trip once the on-orbit mass is determined. In general, trip selection on the basis of trajectory parameters alone, such as departure velocities, is insufficient for ascertaining the best trip in terms of vehicle mass required.**


## Scope of Study

An analysis of interplanetary missions involves a multitude of considerations, ranging from transfer trajectories to spacecraft design. Preliminary mission evaluation requires a basis from which an entire system can be defined. This paper presents such a basis.

To develop a method of trip selection which will provide values for the major design parameters, the mass required on Earth orbit was selected as the basic criterion for trip comparison. A best trip is defined as one that minimizes this mass. The primary independent variables are the arrival and departure velocities for both Earth and the destination planet. These are integrated with an assumed transportation system to define the mass on Earth orbit. The absolute magnitudes of the results are secondary and should be used only as representative values.

An orbital stopover at Mars was selected as an example mission to demonstrate the trip selection method. The mission requirements are limited to a three-man crew and a 10-day orbital stopover at Mars for the August 10, 1971 opposition. Various other assumptions are made as to the type of planetary entry, propulsion systems, structural factors, and life support systems.

The method is presented in sufficiently complete form so as to be adaptable to other applications, such as different reentry systems, different oppositions and conjunctions, advanced system concepts, and more ambitious missions.

## Technical Approach

The interorbital trajectory data used in this study are taken from Ref. 1. A chart for the Mars opposition of August 10, 1971 is given in Figs. 1a and 1b. This chart displays four sets of hyperbolic excess speed contours normalized with respect to Earth's mean orbital speed: one for departure from Earth, one for arrival at Mars, one for departure from Mars, and one for arrival at Earth. The outbound departure and arrival speed contours are superimposed on Fig. 1a, whereas the inbound departure and arrival contours are superimposed on Fig. 1b. All departure contours are shown as solid lines, whereas all arrival contours are shown as broken lines.

The contour charts are read in the following manner. Suppose one wishes to depart from Earth on 244 0880 and arrive at Mars on 244 1130. The required hyperbolic excess speed of departure is then 0.3 EMOS (Earth's mean orbital speed), and the arrival speed at Mars is 0.2 EMOS (Fig. 1a, outbound). If the stay time at the destination planet is one month, one then leaves at about 244 1160 (Fig. 1b, inbound). If one elects to return home on 244 1280, the Mars departure speed is thus 0.3 EMOS, and the arrival speed at Earth is 0.2 EMOS. Of course, the inbound and outbound leg durations and total trip time is the difference between the appropriate Julian dates. The basic assumptions for the example computation are listed below for reference:

1) August 10, 1971 opposition, 10-day orbital stopover at Mars with no manned landing.

2) $O_2/H_2$ Earth escape propulsion, estimated pre-1970 state of technology, storable propulsion system for Mars escape.

3) Drag brake of Apollo-type command module at Earth (three men), drag-brake capture at Mars.

4) Arrival velocity limitations: 50,000 fps at Earth, 40,000 fps at Mars. Earth departure from 200-km orbit, Mars departure from 500-km orbit.

The Earth drag-brake system consists of an Apollo-type command capsule, a large drag brake, and a propulsion system to provide vehicle lift (Fig. 2). The capsule is decelerated from its approach speed down to 36,000 fps, the designed entry speed for Apollo. The Mars capture system also consists of a drag brake and a propulsion system to augment

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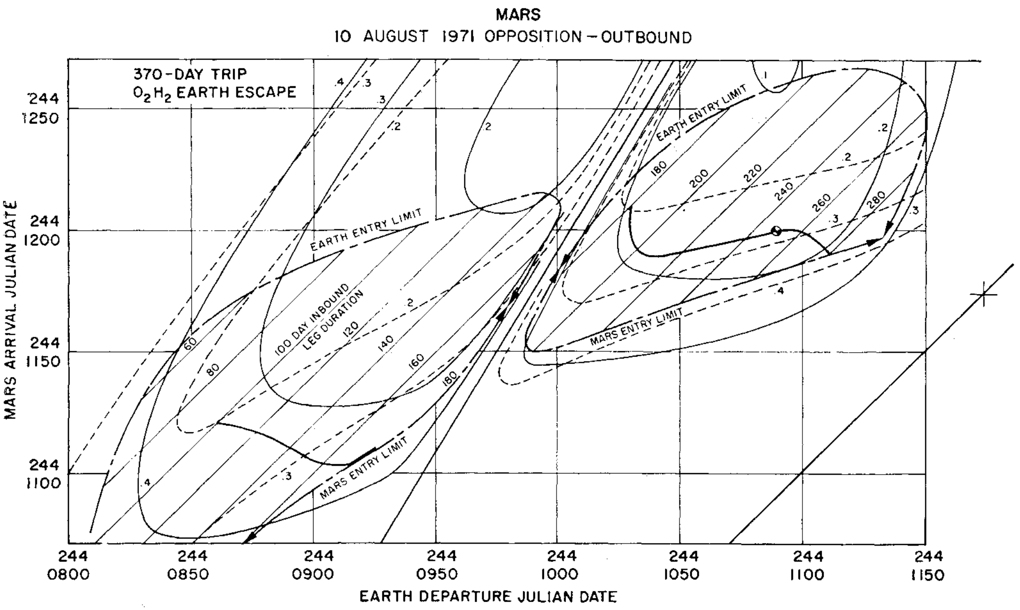


**Fig. 1a Regions of analysis for the outbound leg, overlaid on contours of hyperbolic excess speed in EMOS.**

vehicle lift (Fig. 3). The vehicle is decelerated to near-circular velocity by the drag brake, with the orbit being established by attitude control and midcourse propulsion systems.

The variable subsystems accounted for in this analysis were the life support weight and propulsion step inert weight factors. Information on the weight of a nearly closed life support system was obtained for total trip times of from 330 to 550 days for three men. The amount of overboard waste from the life support system also was accounted for. Structural data for propulsion steps using storable, cryogenic ($O_2/H_2$), or nuclear systems were determined as a function of impulse propellant. These equations allowed the computation of mass on Earth parking orbit to be machine programmed.

The minimizing procedure is outlined below and may be followed graphically by reference to Fig. 1. For a selected total trip time of 370 days, a series of constant inbound leg durations are superimposed on each side of the chart within a region defined by the Mars- and Earth-entry velocity limits. A point on the inbound portion of the chart is displaced with respect to the corresponding outbound point by 10 days, the time of stopover at the planet.

For each possible Earth departure date that satisfies a given inbound leg, the mass on Earth orbit is computed using the model transportation system. For the given inbound leg, there is one Julian date that corresponds to the minimum mass. By investigating all the available inbound legs, it is possible to establish one that minimizes mass for the given total trip time. The locus of minimums for the example case in Fig. 1 is indicated by the heavy solid curve. Of these minimums, the lowest value of mass is seen to occur at about 250 days, inbound leg, for the selected total trip time of 370 days. This entire procedure is repeated for all total trip times of interest.

The determination of vehicle mass on Earth orbit starts with the Earth entry velocity. From this, an Earth-entry system weight is obtained (Fig. 2). To this mass is added a mission module, a power supply system, radiation shielding, and a life support system, the weight of which varies as a function of the total trip time. Compensation also is made for overboard waste expended during the trip. This net mass constitutes the payload to be boosted from the Mars orbit. The Mars escape propulsion then is determined by the payload mass, specific impulse, and the incremental velocity required for departure. The return leg mass then is defined. To this mass is added the weight of the scientific probes in order to establish the mass on Mars orbit. The drag-brake

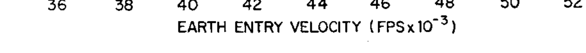


**Fig. 2 Apollo drag-brake capture system.**

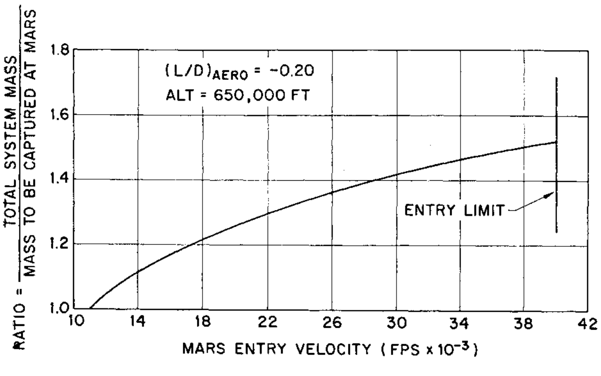


**Fig. 3 Mars drag-brake capture system.**

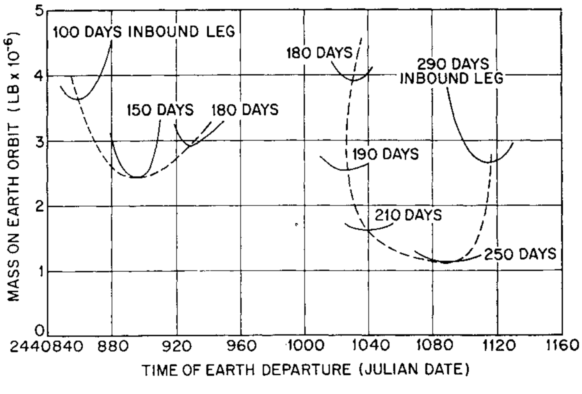


**Fig. 4 Determination of locus of minima and optimum trip (370 days total trip time).**

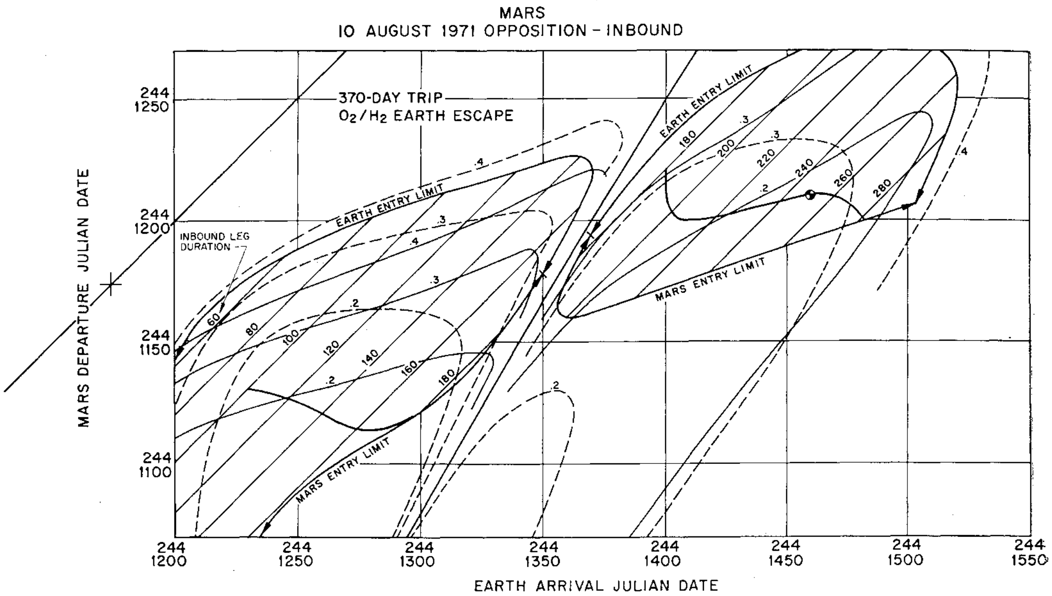


**Fig. 1b Corresponding regions for the inbound leg, overlaid on hyperbolic excess speed contours.**

system required for Mars capture is determined from the mass on Mars orbit and the entry velocity (Fig. 3). The outbound expendables are added to the Mars approach mass to obtain the payload leaving Earth. The Earth departure propulsion then is defined by this mass and the incremental velocity required to escape from a 200-km orbit. A specific impulse of 430 sec is used for the cryogenic system, 333 sec for the storable system, and 830 sec for the nuclear.

In this sample design, representative subsystem weights were used. These design values were selected only for demonstrating the trip selection method. The mission module weights 25,000 lb. This is composed of an 8000-lb module, 1500-lb power supply, 5000-lb radiation shielding, 5000 lb of equipment and scientific probes, and 5500 lb of tie-in structure and onboard mission equipment. The life support system varies with total trip time and ranges from 10,000 to 14,000 lb. The storable propulsion system for Mars escape is sized for both incremental and midcourse velocity requirements. The escape propulsion systems, storable and $O_2/H_2$ or nuclear, include 5% and 8% velocity losses, respectively.

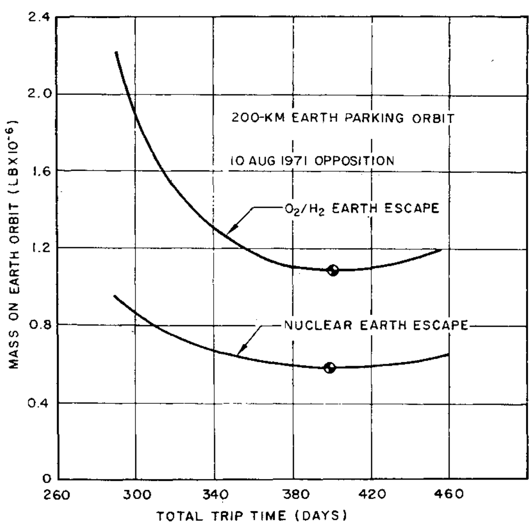


**Fig. 5 Minimum required mass for Mars 10-day stopover missions.**

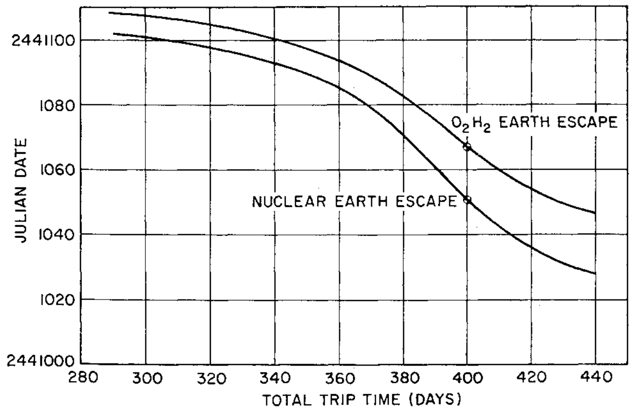


**Fig. 6 Optimum Julian departure dates for Mars 10-day stopover missions (August 10, 1971 opposition).**

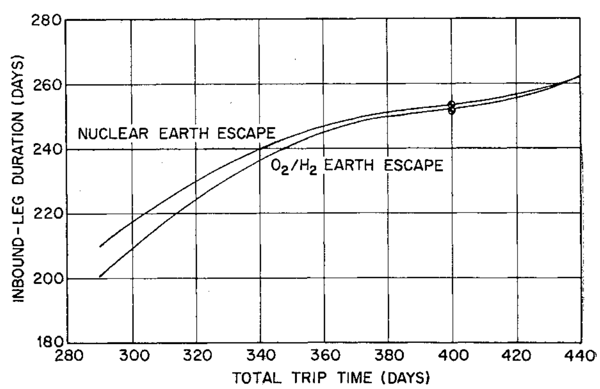


**Fig. 7 Optimum inbound-leg durations for Mars 10-day stopover missions (August 10, 1971 opposition).**

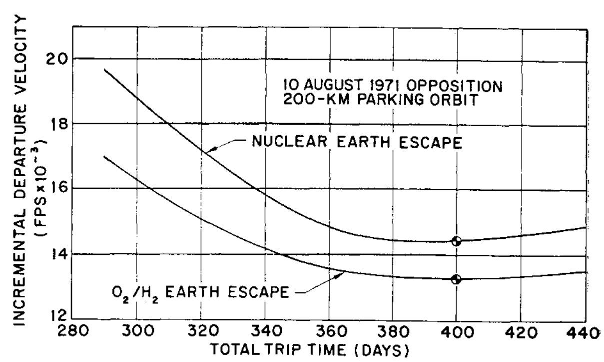


**Fig. 8 Optimum Earth departure velocities for Mars 10-day stopover missions.**

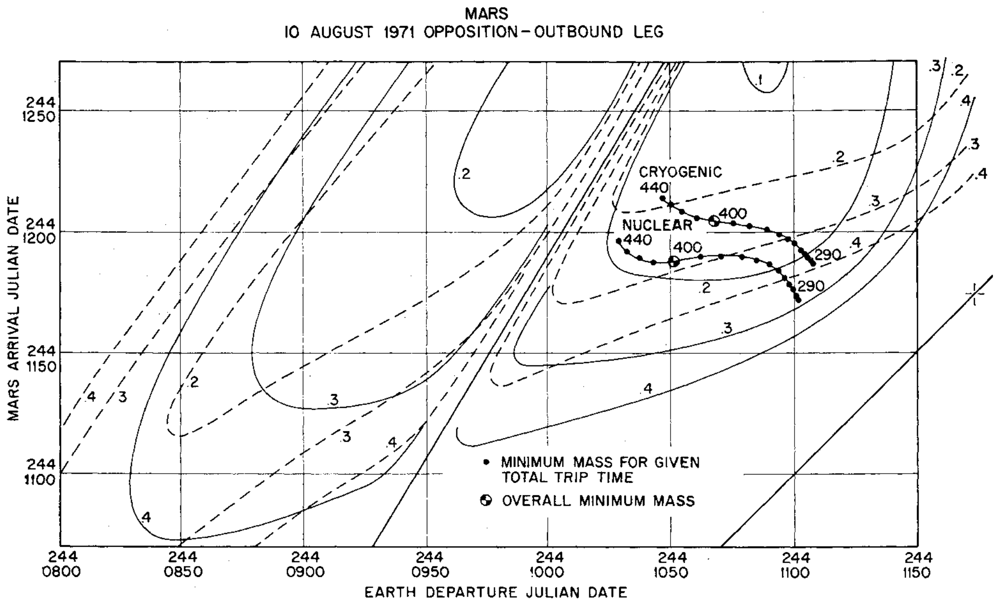


**Fig. 9 Favored departure region for minimum mass trips (Mars August 10, 1971 opposition).**

## Discussion of Results

For each total trip time, a curve such as given in Fig. 4 may be drawn to establish the minimum condition. For the example case of 370 days total trip time, an inbound leg of about 250 days on a trip departing Earth at 244 1090 requires the lowest mass. The two dotted curves of Fig. 4 correspond to the two possible departure "valleys" shown in Fig. 1a. To identify the favored region of departure, the locus of minimums is plotted in Fig. 1 and the minimum point noted. For this trip time, the upper departure region is the favored one. As will be seen later, this area also yields minimum mass requirements for all other total trip times.

Note that, for the overlapping case of 180 days in Fig. 4, a saving of approximately $1 \times 10^6$ lb is obtained by using the earlier departure date. Thus, there is a point at which it becomes advantageous to change departure valleys if shorter inbound legs are sought. Of additional importance is the fact that, if one had selected only the earlier departure valley for analysis, the minimum established there (150 days, inbound leg) would be construed erroneously as an overall minimum.

By determining the minimum mass conditions for each total trip time, the curves of Figs. 5–8 are obtained. The parameters defining a trip are plotted as a function of total trip time. Each curve represents the locus of minimum mass requirements under the constraints outlined previously. Specification of total trip time, inbound leg duration, and Julian departure date uniquely establishes the transfer orbits. These curves also may be used for first-order perturbation studies and related vehicle sizing.

It is interesting to note how these parameters vary by comparing the cryogenic and nuclear Earth escape systems. For the cryogenic case, the mass requirement is 1,080,000 lb on Earth parking orbit, with a total trip time of 400 days. The Julian departure date is 244 1067 (April 27, 1971), with an incremental velocity of 13,300 fps. The trip requires 138 days to Mars and 252 days to return. The nuclear system requires 580,000 lb with the same total trip time. However, the departure date is 16 days earlier, and the outbound duration is a day less. The incremental velocity is greater at 14,400 fps.

It should be emphasized that trip selection should be based on an integration of the transfer trajectories and vehicle design to incorporate the sensitivity of all parameters involved. This assures that the minimum mass will be established and that excessive requirements on escape propulsion will be avoided by eliminating arbitrary selection of trips.

This philosophy becomes clear by inspection of Fig. 1. The regions to be investigated are established by the entry velocity limitations for both Earth and Mars. Because the interplanetary trajectories have been computed for elliptical Earth and Mars orbits mutually inclined, the Hohmann minimum energy criterion for circular coplanar transfers disappears, and a "ridge" of high departure and arrival velocities separates two possible departure "valleys."[2] However, low departure velocities still are possible. A departure date of 244 1090 with an inbound leg of 192 days intersects the 0.1 departure contour and arrives at Mars with less than 0.2. The Mars departure speed is 0.35, with an Earth arrival of 0.39. This point is far from the minimum-mass point for this inbound leg, which has a departure date of 244 1030. Hence, lowest incremental velocities for Earth and Mars departure will not assure that the required mass is the lowest achievable.

For the entire range of total trip times, it can be seen in Fig. 9 that the ideal departure region for this opposition is identified firmly. This means that, of all the possible trips, there is only one departure region that should be used if minimum system mass is desired. Note that the minimum mass points for the nuclear system also fall in this area.

Within the constraints of the mission selected and the vehicle design assumptions, the overall optimum indicated in Figs. 5 and 9 is the best that could be expected. The approach of computing the mass requirements for all possible trips assures one that this optimum is unique for this opposition. Because of the complex relationship between the trajectory parameters, it is necessary to apply this optimization method at each opposition or conjunction. This is especially true for Mars, where, in succeeding oppositions after 1971, all the velocity requirements became quite high due to the eccentricity of its orbit.

An approach as presented here becomes increasingly important as mission objectives become more ambitious. For manned surface landings, the mass required on Earth orbit will increase many-fold and will necessitate multiple rendezvous if large boosters are not available. These requirements, however, may be minimized to a large extent by use of this optimizing procedure.

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