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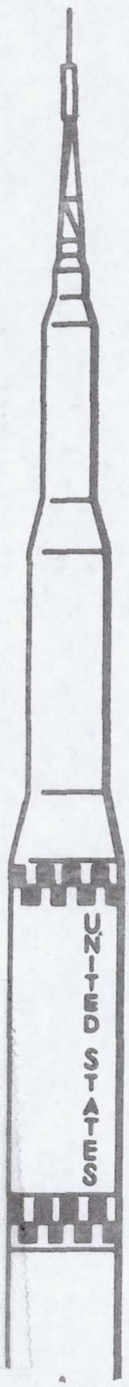

FIFTH SPACE CONGRESS

COCOA BEACH, FLORIDA

MARCH 11,12,13,14,1968

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FOREWORD

Each spring the Canaveral Council of Technical Societies sponsors a symposium devoted to the accomplishments of our space programs and plans for future activities. These proceedings provide a permanent record of the papers presented at our Fifth Space Congress held in Cocoa Beach, Florida, March 11 - 14, 1968.

The Fifth Space Congress theme, "Our Goals in Space Operations", was chosen to provide a forum for engineers and scientists to express individual and corporate views on where our nation should be heading in space operation. The papers presented herein depict the broad and varied views of the industrial organizations and government agencies involved in space activities.

We believe that these proceedings will provide technical stimulation and serve as a valuable reference for the scientists and engineers working in our space program.

On behalf of the Canaveral Council of Technical Societies, I wish to express our appreciation to the authors who prepared and presented papers at the Fifth Space Congress.

Edward P. Wynne

EDWARD P. WYNNE
General Chairman
Fifth Space Congress

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*Note: Inability to obtain the required authorization in time to meet the printing deadline prohibited the inclusion of some papers. In other cases, oral presentations were given but formal papers for publication were not written by the authors.

ELLIPTIC CAPTURE ORBITS FOR MISSIONS TO THE NEAR PLANETS

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Summary

Elliptic capture orbits around Mars and Venus have often been considered as means for reducing arrival and departure energy requirements for two-way missions. It had also generally been feared that the energy savings obtained by capturing a spacecraft into a highly elliptical orbit (rather than a near circular orbit of the same periapsis) would largely be offset by the penalties incurred in aligning the semi-major axis of the ellipse in such a way as to obtain the proper orientation of the departure hyperbola. This paper presents the results of an analysis which takes into consideration the penalties arising from the requirement to match the orientation of the elliptical orbit with the asymptote of the departure hyperbola. The scientific aspects of elliptical orbits around the target planet are discussed, and it is shown that such orbits exhibit characteristics which may be considered advantageous or disadvantageous depending on the purpose of the mission.

Alignment of the semi-major axis of the capture ellipse relative to the asymptote of the escape hyperbola was found not to be a critical requirement since the kinetic energy remains high over a substantial portion of the elliptical capture orbit. This means that the escape stage can operate efficiently even when ignited at some angle from the true periapsis point. Considerable freedom in choosing this angle is available at little propulsive cost. The resulting latitude in the choice of angles between arrival and escape asymptotes makes it possible to consider a wide variety of interplanetary transfers and planetary staytimes without the need for separate propulsive maneuvers to realign the capture ellipse before departure. Special consideration has also been given to plane change maneuvers around the planet. These may be required for reasons of orbit dynamics or scientific experimentation and are not uniquely tied to elliptical captures. The sensitivity of the mass of the excursion module to the eccentricity of the capture orbit is discussed and mass-penalty diagrams are presented. It is shown that these penalties do not materially offset the large gains obtained through the use of the elliptical capture mode.

Introduction

The capture of a spacecraft around a planet requires an amount of energy which is a function of the characteristics of the capture orbit. For round-trip missions, low circular orbits have most frequently been considered. These orbits are the most demanding in terms of capture and departure energy. In terms of compatibility with the departure hyperbola, on circular orbits, the kinetic energy is uniform at all points and, hence, the position at which the escape maneuver takes place is noncritical; i.e., as long as the escape asymptote lies in the plane of the capture orbit, all directions are equally favored. When the escape asymptote does not lie in the plane of the orbit, a plane change is necessary. Since the energy required for a plane change increases with the kinetic energy of the point at which this plane change is to be undertaken, it is evident that it is more economical to undertake a plane change at the apoapsis of an elliptical orbit than at a point on a low circular orbit. Even in the case of those planetary round-trip missions which incorporate low circular capture orbits, plane change requirements often make it desirable to consider elliptical pre-escape orbits as a means of reducing total energy requirements. It follows that elliptic orbits may be used around the planets in order to minimize capture energy and departure energy and also to minimize energy requirements for plane changes. Since the alignment of the semi-major axis of the capture ellipse and that of the pre-escape ellipse will usually not be identical, methods for "turning" the semi-major axis before escape were found and have been described in the literature.[1, 2] These methods have been found to be generally costly in terms of energy and, thus, were felt to have reduced the attractiveness of elliptical capture orbits.

It is feasible to avoid this "turning" requirement in a wide variety of cases by allowing the capture and escape maneuvers to take place at some angle away from the periapsis point. This possibility has been analyzed extensively in a paper by Luidens and Miller[3] under the simplifying assumption of purely impulsive thrusting. This assumption is entirely adequate for general mission analysis purposes. For the analysis of the propulsive systems and the operational factors involved, however, one has to take into account the duration of the thrusting phase so that gravity losses and engine thrust levels may be studied.

In addition to studying the operational problems of elliptic capture and escape, it is necessary to examine in detail the impact upon the mission objectives of the use of elliptical orbits.

The purpose of this paper is thus to examine some of the propulsive requirements for planetary capture into, and escape out of, elliptical orbits, and to examine some of the requirements and restrictions imposed upon landing and global mapping operations through the use of such orbits.

Analysis

Encounter Geometry

Out of the many elliptical capture modes considered in the literature,[3] we have singled out as an example the case in which a capture orbit about Mars is coplanar with both the incoming and the departing asymptote. As may be seen in Figure 1, where the trajectory of the planet is indicated by the long curved line at the top, the spacecraft trajectory (indicated by "S") generally crosses that of the planet in an outbound direction during the arrival phase. At encounter, the spacecraft heliocentric velocity is generally less than that of the planet. This arrival situation is depicted on the right half of Figure 1, while on the left is described, in similar fashion, the departure situation. Again, in the general case, the spacecraft leaves the planet with a velocity which is smaller than that of the planet and directed inward. The two velocity diagrams indicated in the lower half of Figure 1 show that for Mars, in the planetocentric system of reference, the spacecraft velocity vectors at arrival and departure are approximately normal to each other. The angle between these two vectors, ψ, is called the "difference angle." Arriving at Mars in a generally non-Hohmann trajectory, and capturing into a posigrade elliptical orbit, will generally result in a situation such as described in Figure 2. The ellipse labeled "a" owes its orientation to an impulsive deceleration maneuver performed at the periapsis of the arrival hyperbola (labeled "A"). Since the direction of the departure hyperbola (labeled "D") is prescribed by the required heliocentric transfer trajectory, an ideally oriented predeparture ellipse would have its periapsis in common with that of the departure hyperbola and is shown at "d."

Off-Pericenter Maneuver

The situation just described would allow capture into the elliptic orbit "a," with a minimum of energy and would also allow departure from the elliptic orbit labeled "d" with a minimum of energy; however, an intermediate orbit labeled "i" would now be required to "turn" the periapsis direction from the orientation of "a" to the pre-departure orientation shown in "d." Considerable velocity savings may be obtained by employing only one elliptical orbit, the periapsis of which does not coincide with the periapsis of either hyperbola. This situation is described in Figure 3, which is shown in the same heliocentric orientation as Figures 1 and 2. Again, a posigrade capture orbit around Mars has been chosen as a typical example. The angle θ, which was not particularly emphasized in the previous figures, is defined as the angle between the major axis of the ellipse and the asymptote of the arrival or escape hyperbola. It is called the "turning angle" and does not necessarily have to be equal for the incoming and departing trajectories. The maneuvers required to obtain the situation just described are referred to as "off-pericenter thrust" maneuvers. To indicate where thrusting would take place for an escape maneuver, the powered portion of the escape trajectory is shown as a broken line on the diagram.

Individual In-Plane Requirements

The results of a numerical analysis of the thrust angle requirements and the velocity requirements as a function of turning angle θ are shown in Figure 4. While the horizontal refers to the turning angle θ, the right-hand scale on the diagram shows the planetocentric angle ("true anomaly") at which the thrust maneuver has to begin for departure or terminate for arrival in order to obtain the desired turning angle. While for the case of infinite thrust the propulsive maneuver has to take place at periapsis in order to obtain the minimum velocity requirements, the application of finite thrust causes this maximum efficiency point to shift some 20° away from the periapsis as indicated by the thin straight lines. The velocity requirements (shown on the left vertical scale) have been normalized to the velocity requirement encountered at the optimum thrust angle. The velocity requirement for the infinite thrust case is shown by the solid line, while that for finite thrust is shown by the broken line. It is interesting to note that for mission analysis purposes the impulsive approximation is indeed excellent; however, for systems analysis purposes it will be necessary to take gravity losses and finite thrusting time into account. Our computation of the latter case takes gravity losses into account and optimizes the thrust-to-weight ratio of the system for each individual turn angle, i.e., the thrust-to-weight ratio changes along the ΔV curve. An investigation of this optimized thrust-to-weight requirement is presented in Figure 5. A specific impulse of 450 seconds has been assumed, tankage and engines are of variable size and follow generally accepted scaling laws. On this diagram, the planetocentric angle at which the thrust maneuver has to begin for departure or terminate for arrival is shown on the horizontal while the ratio of engine thrust to spacecraft initial gross mass is shown on the vertical. It may be seen that, since efficient thrusting maneuvers have to take place in the vicinity of the periapsis, the optimum thrust-to-weight ratios required are similar to those required for escape from a low circular orbit.

Sum of Velocity Requirements

If all the elements of the Mars arrival, capture and departure orbits are assumed to be in the same plane, the sum of the velocity penalties of arrival and departure may be plotted against the difference angle between the arrival and departure asymptotes such as shown in Figure 6. As previously stated, this angle is generally in the vicinity of 90°; the velocity penalty, therefore, is approximately half a kilometer per second.

Up to this point, we have analyzed strictly coplanar maneuvers. Plane change maneuvers will generally be necessary and have been analyzed extensively in a number of other papers, among which Reference 3 presents probably the most comprehensive results. In Reference 3 and similar papers, it has been shown that the velocity penalties required to accommodate plane change maneuvers can be kept small by always applying the plane change at the apoapsis of an intermediate orbit of high ellipticity. Such an intermediate orbit could be considered a standard phase of the escape maneuver. Adding plane change penalties to the turn angle penalties analyzed in this paper, we can now compose a diagram that shows the sum of all velocity penalties for arrival and departure at Mars as a function of the eccentricity of the orbit. Such a diagram is shown in Figure 7 where the first increment (ΔV_a) indicates the velocity required to capture into an orbit of given eccentricity; the second increment (ΔV_d) indicates the departure velocity requirement; the third increment (ΔV_p) is a 90° plane change requirement; and the fourth increment (ΔV_t) is due to the turn angle requirement. As would be expected, the turn angle requirement is zero for a circular orbit and increases with orbit eccentricity. The results shown in Figure 7 were computed for a specific Mars round-trip mission and are representative of a wide variety of cases.

The relationship between initial mass requirements and turn angle is illustrated on Figure 8. Shown on the vertical is the ratio of mass requirement at any particular turn angle to mass requirement at the optimum turn angle. While this "normalized mass" is plotted on the ordinate, the abscissa is again the turn angle. The turn angle region typical of posigrade captures is shown by the shaded area marked "P" while the turn angles typical of retrograde captures are marked by the shaded area "R." The broken horizontal line indicates the mass requirement for capture into a low circular orbit of equal periapsis altitude. The case shown was computed for a chemical cryogenic system having a specific impulse of 450 seconds. The inert mass fractions follow conservative scaling laws and take into account micrometeoroid protection, propellant boiloff and thermal insulation penalties. Although the precise amount of mass saved by the utilization of the elliptical capture mode depends also on a number of additional factors such as excursion module penalties, precession and regression of the capture orbit during the staytime, etc., Figure 8 makes it evident that very significant savings are potentially available through the use of this particular capture mode.

Other Planetary Applications

Estimates of initial mass in Earth orbit are shown on Figure 9 for orbiting missions to Mercury, Venus, Mars and Jupiter. The ordinate is a measure of the initial mass required in Earth orbit before departure for a complete round-trip mission to the planets shown. The unshaded areas indicate the application of chemical cryogenic systems throughout the entire mission, while the shaded areas indicate the application of nuclear rockets throughout the entire mission. Earth entry is assumed to be achieved by aerobraking without the assist of propulsive retro maneuvers. The left boundary of every area shown refers to circular captures; the right boundary refers to elliptical captures with an eccentricity of .7. The vertical range corresponds to the range of requirements over a cycle of mission opportunities. As would be expected, elliptic captures are especially effective in reducing the initial mass requirements for the more massive of the four planets shown, Venus and especially Jupiter.

Contingency Operations

The relationship between true anomaly for thrust initiation and optimum thrust to weight suggests the use of elliptical capture orbits as a possible measure of contingency for unexpected propulsive failures. One such example is given in Figure 10. In this figure, it is assumed that it was planned to achieve a close circular capture around Mars at an altitude of 300 km. It is also assumed that, early during the arrival phase, half the planned thrust was lost due to a partial failure. Under these circumstances, the close circular capture orbit is no longer attainable, but, instead, it is possible to go into an elliptical capture orbit with an eccentricity of .7 and a periapsis altitude of 300 km. There are no mass penalties for this contingency maneuver and it could therefore easily be planned into the mission as a safety factor. As will be seen later, a mass penalty would have to be planned into an excursion vehicle to accommodate the higher requirements for entry from elliptical orbit and return to it.

Excursion Module Operations

The above discussion has identified the potential weight saving advantages of the elliptical capture mode as applied to orbital missions to the near planets. These advantages are somewhat degraded if a planetary landing and subsequent rendezvous with the orbiting spacecraft are contemplated. This degradation is caused by the increased landing and ascent propellant requirements associated with high eccentricity orbits. The landing requirements from elliptical orbits can, however, be reduced if a planetary atmosphere exists and if it is dense enough to be used to advantage for terminal deceleration. The amount of terminal deceleration is, of course, a function of the density of the atmosphere and the amount of mass to be landed.

The very tenuous nature of the Martian atmosphere creates a problem for an excursion vehicle used at that planet in its terminal descent to the surface. Previous studies[4, 5] have described the technological problems associated with conventional descent concepts using aerodynamic decelerators such as drogues and parachutes as applied to the landing of large payloads. Recently, attention has turned to propulsive descent as a means of overcoming these technological difficulties. The use of some propulsive retardation, of course, increases the penalty associated with use of an elliptical parking orbit since the ascent propellant must be decelerated propulsively to the surface. A schematic of the terminal descent maneuver is shown in Figure 11. During a typical landing, the entry vehicle would initiate its entry from orbit and would achieve constant altitude flight at an altitude of approximately 10 km through the use of aerodynamic lift and by roll or pitch modulation of that lift force. Eventually, sufficient deceleration would occur to where constant altitude flight could no longer be maintained. The vehicle would then lose altitude and speed in a gliding manner until an altitude of about 3 km is reached. At this point, a rocket would fire in a constant thrust mode until nearly zero velocity is reached near the surface. The rocket is then throttled in such a way as to provide a low rate of descent and hover and the touchdown is then accomplished.

Excursion Module Mass Requirements

Two descent trajectories have been illustrated in Figure 11. The first indicates a trajectory for entry from circular orbit and the second illustrates entry from elliptical orbit. It can be seen that once the velocity has decreased to below circular speed, the descent trajectories are identical. Thus, the only penalty associated directly with the landing maneuver from elliptical orbit is the greater aerodynamic heating and load environment encountered during the super-circular part of the descent trajectory.

If an ascent and rendezvous maneuver is planned for the excursion system, the increased ascent propellant requirement for elliptical orbit rendezvous will require more terminal descent propellant than that associated with descent from circular orbit. The total weight of an excursion system for Mars is shown as a function of parking orbit eccentricity in Figure 12. These results are for a four-man crew and a 30-day staytime on the Martian surface, and are based upon the results of a recent study by the North American Rockwell Corporation for NASA.[6] The conceptual vehicle as illustrated in its landing configuration by the pictorial insert in Figure 12 is shaped similar to the Apollo command module and is about 10 m in diameter. It can be seen that about a 25% weight penalty is incurred for the highly elliptic orbits.

Science Implications

Utilization of elliptic orbits for planetary missions has a considerable impact upon the scientific observational capabilities of a particular mission. Elliptic orbits are characterized by long orbital periods and by short close-approach observation times. In addition, the long period reduces the number of close-approach passes for a given staytime. The long period is not, however, always a disadvantage, since a large amount of data can be obtained during a pass and the time spent during apoapsis passage can be used to advantage for the processing and transmitting of that data.

Orbital imaging experiments are affected to the greatest degree by the eccentricity of the parking orbit. Average resolution is, of course, degraded and for optical imaging it becomes difficult to reconcile the requirement for desirable surface lighting conditions with the planetary approach (and possibly departure) conditions and still obtain as complete a coverage as possible.

As an example, the effects on imaging experiments of elliptical orbits will be examined in detail for an orbiter mission to Mars during a specific opportunity. The 1971 opportunity was chosen as typical and serves to illustrate the approach to orbit selection and the major effects of orbit eccentricity. The selected heliocentric transfer trajectory for this mission is a type I transfer.

The geometry of planetary arrival is shown in Figure 13 which shows a stereographic meridional projection of the planet surface, projected for convenience upon the sub-solar point at the time of arrival. The outer edge of the hemispherical projection thus represents the terminator. The planets' south pole is seen in this projection and the equator and the 0° and -90° meridian lines are shown for reference (longitude is defined similar to right ascension). The plane of Mars' orbit is also indicated in the figure. The shaded area represents the surface region where the lighting constraints imposed by the optical imaging experiment are satisfied (i.e., sunlight conditions between 30° and 60° off-zenith). The arrival velocity vector relative to the planet is also indicated and the vector head is shown in the projection.

The orbit selection process is initially limited by considerations of orbit insertion penalties to coplanar insertions only. That is, the orbit must be coplanar with the arrival hyperbolic asymptote; therefore, an orbit trace shown on the projection must pass through the head or tail of the arrival vector.

In 1971, the arrival date corresponds to slightly after midsummer in the southern hemisphere. The areas undergoing a maximum color change are in the southern hemisphere at latitudes near 30° south.[7] The orbit which best satisfies the lighting constraints for the planet surface areas of interest, including considerations of the apparent motion of the Sun and the nodal regression and apsidal progression of the orbit, appears to be a 58° inclined orbit achieved by an on-periapsis southern insertion from the approach asymptote. This orbit allows for optical coverage under the stated lighting constraints from about 20° south latitude to 58° south latitude for the first few days after arrival.

The selection of the orbit eccentricity is dictated primarily by considerations of orbital lifetime, insertion velocity requirements, and optical coverage. Desirable optical coverage is most systematically obtained by picking an orbital period which is nearly in synchronism with the planet's rotation. That is, if an orbit period is chosen as slightly different than an integer fraction of a Martian day, then the orbit traces can be made to apparently drift either east or west at any desired rate. The desired drift rate is determined by the width of the camera field of view and the time in which complete longitude coverage is to be obtained.

The three factors influencing the choice of orbit eccentricity are shown graphically in Figure 14. In this figure, apoapsis altitude is shown as a function of periapsis altitude for various constant values of orbital period and insertion velocity requirements. The orbital periods are shown for various integer fractions of a Martian day. The velocity requirements shown are for typical mission arrival conditions. In addition, the shaded area to be left indicates the region of orbit characteristics which produce orbit lifetimes less than 50 years (the sterilization criteria). In order to stay well outside the 50-year lifetime constraint without an undue sacrifice in surface resolution, a 1,000 km periapsis altitude was chosen for our example. In addition, a camera system with a 9° total field of view was chosen which results in an imaging swath width on the surface at periapsis of approximately 3°. Since it is desirable to complete the longitude coverage because of favorable lighting conditions in approximately 60 days, an orbit of period equal to nearly one half a Martian day is required. This orbit produces an imaging swath in the eastern hemisphere followed by a similar swath in the western hemisphere nearly 180° away from the first in longitude. The third swath is then positioned 3° eastward from the first. This positioning requires that the orbital period be approximately six minutes longer than one half a Martian day. Complete coverage of the eastern and western hemispheres then requires about 60 days. Considerable overlap is obtained for pictures taken prior to and after periapsis passage, and overlap at periapsis does not appear necessary. The apoapsis altitude for such an orbit is approximately 18,000 km and the insertion ΔV requirement is approximately 1.3 km per second. The orbit eccentricity is approximately 0.66.

The amount of optical coverage obtained from the best optical coverage orbit with the above characteristics is shown in Figures 15 and 16. The limits imposed upon the coverage are that the resolution will not be degraded from that obtained at periapsis by greater than a factor of 3 (i.e., $H/H_p < 3$) and the solar angle shall be between 30° and 60° off-zenith. In Figure 15, the first two orbit swaths are shown. In addition, the apparent motion over the planet surface of periapsis of the odd numbered orbits is shown. Due to the inclination of this orbit, the progression of the line of apsides is very small. A similar motion of periapsis would, of course, be apparent for the even numbered orbits. Complete longitude coverage is obtained in 60 Martian days or 61.5 Earth days. This coverage is illustrated in Figure 16. The slight sweep of the coverage in the northernly direction is caused primarily by the apparent motion of the Sun toward the northern hemisphere and by the progression of the line of apsides.

Concluding Remarks

This paper has presented the operational aspects of elliptical capture orbits for missions to the near planets. Through the use of off-pericenter thrusting, the misalignment of the arrival and departure asymptotes and the semi-major axis of the parking orbit may be accommodated without a large velocity penalty. Plane changes may be efficiently employed at the apoapsis of elliptical orbits. Total velocity requirements for elliptical orbit missions, including plane changes, are below the range of circular orbit requirements. It has been shown that the mass savings due to elliptical capture orbits are large for the more massive planets, Venus and Jupiter and are still significant for the planet Mercury. The mass requirements of an excursion module increases only moderately with increased eccentricity.

Contingency planning suggests the alternate use of elliptical orbits in the event of propulsion system failure during circular orbit capture.

The optimal location of initiation or termination of finite thrust is removed from pericenter and lies some 20° before periapsis in the cases examined. The optimal thrust-to-weight ratio is highest when thrusting takes place about the periapsis and is an order of magnitude lower for thrusting about the apoapsis.

Scientific imaging experiments are not seriously impaired by elliptical captures. Compatible orbits can be defined, such as the 12-hour orbit with 0.66 eccentricity for complete coverage of the Martian east and west hemispheres in 60 days.

References

1. Bossart, Karel J.: "Techniques for Departure and Return in Interplanetary Flight." Convair Aeronautics; 1958.

2. Ehricke, Krafft A.: "Space Flight. I. Environment and Celestial Mechanics." D. Van Nostrand Co., Inc.; 1960.

3. Luidens, Roger W. and Miller, Brent A.: "Efficient Planetary Parking Orbits With Examples for Mars." NASA TN D-3220; January, 1966.

4. Syvertson, C. A. and Swenson, B. L.: "Lifting Entry for Manned Planetary Missions." Presented at USAF/ASSET Conference, Miami, Florida; December, 1965.

5. Swenson, B. L., Carlson, R. W. and Tindle, E. L.: "Some Requirements on Lifting Vehicles for Manned Mars Operations." Presented at AIAA/AAS Stepping Stones to Mars, Baltimore, Maryland; March 28-30, 1965.

6. Cannetti, G. S., et al: "Definition of Experimental Tests for A Manned Mars Excursion Module." Contract NAS 9-6464, North American Rockwell Corp., Rep # SID 67-755-1.

7. Swan, Paul R. and Sagan, Carl: "Martian Landing Sites for the Voyager Mission." AIAA J. Spacecraft; Jan.-Feb., 1965, pp. 18-25.

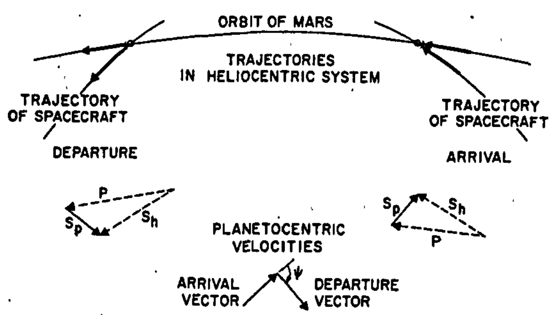

Figure 1.-Planetary encounter geometry showing heliocentric, planetocentric and mutual orientation of the arrival and departure vectors.

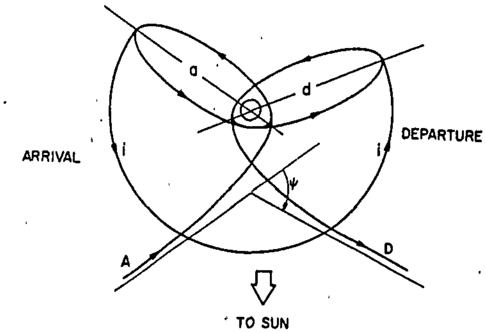

Figure 2.-Braking from an arrival hyperbola "A" into an elliptic capture orbit "a" with minimum energy, and departing optimally oriented ellipse "d" to obtain the departure hyperbola "D."

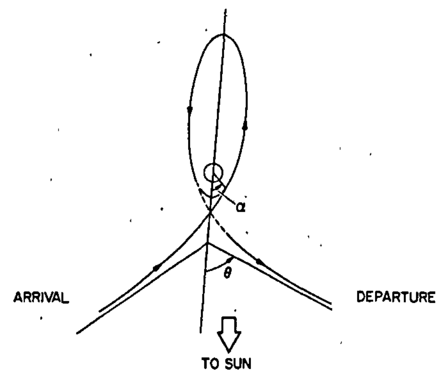

Figure 3.-Common capture ellipse accommodating both arrival and departure hyperbolas. Thrusting maneuvers required for capture and departure now occur away from the periapsis region of the ellipse.

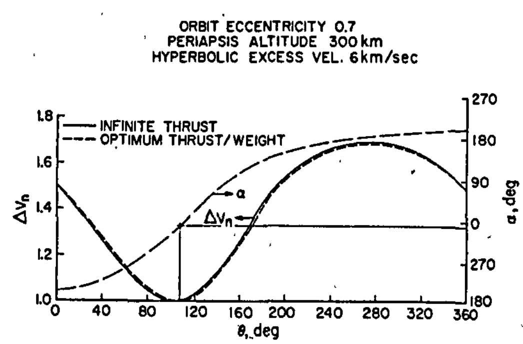

Figure 4.-True anomaly, α, at which a propulsive maneuver has to be initiated in order to obtain the desired turning angle, θ. On the left scale is shown the ratio of the velocity requirement for any turning angle to that for the minimum energy case.

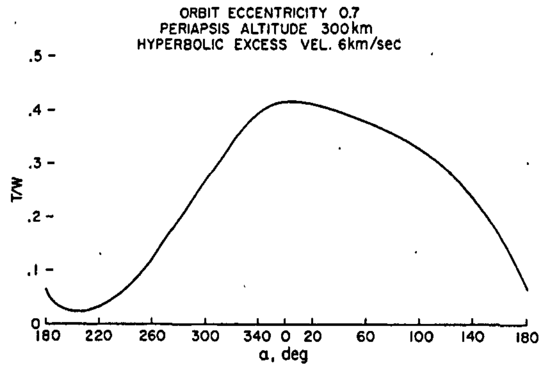

Figure 5.-Optimized thrust-to-weight ratio for escape maneuvers starting at different true anomaly, α, along the ellipse.

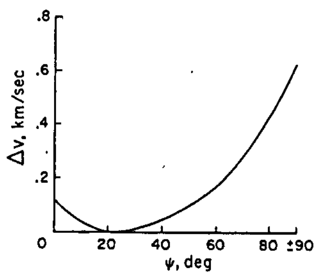

Figure 6.-Sum of velocity penalties for coplanar capture and escape. Orbit eccentricity of 0.7, and periapsis altitude of 300 km around Mars.

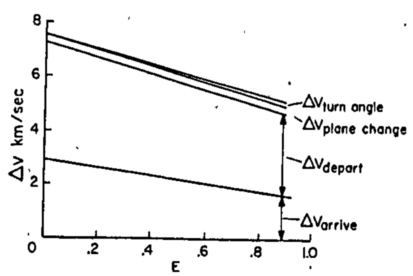

Figure 7.-The four major velocity requirements needed to accomplish capture and depart from Mars shown as a function of orbit eccentricity "E." Hyperbolic excess velocity 6 km/sec for Mars departure.

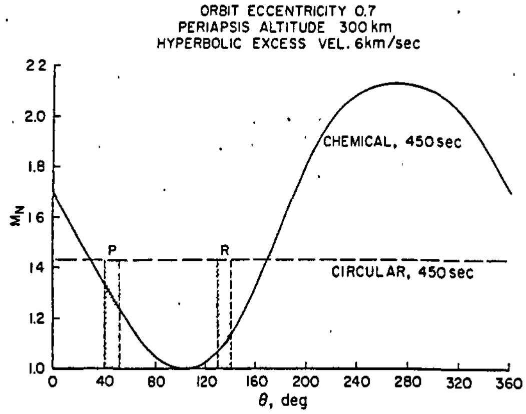

Figure 8.-Mass requirements normalized at the turning angle requiring least energy. The shaded areas, "P" and "R," denote the turning angle regions typical of posigrade and retrograde captures, respectively.

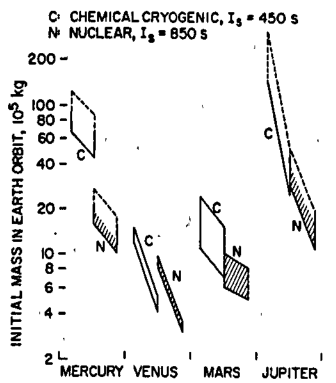

Figure 9.-Initial mass required in Earth orbit for manned stopover missions. The left vertical boundary refers to circular capture; the right to elliptical capture, E = .7. The vertical range indicates the variation of requirements over a complete cycle of opportunities.

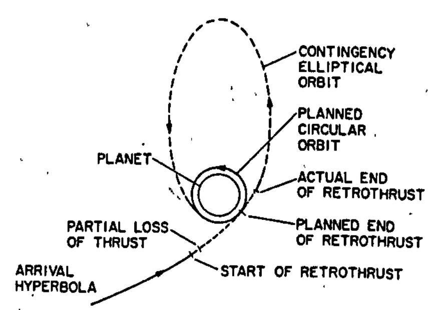

Figure 10.-Example of the use of an elliptical capture orbit as a contingency for a partial propulsive failure.

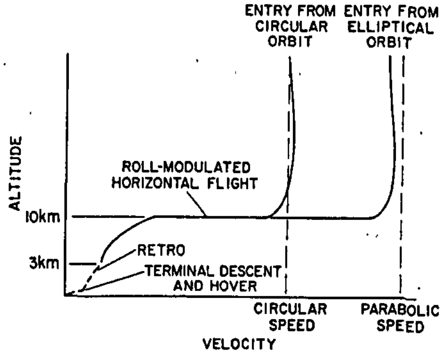

Figure 11.-Terminal descent maneuver at Mars. Periapsis altitude 300 km.

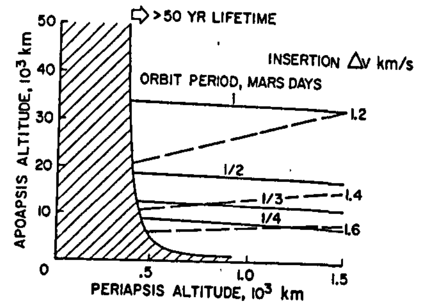

Figure 14.-Factors influencing selection of eccentricity for typical Mars arrival conditions.

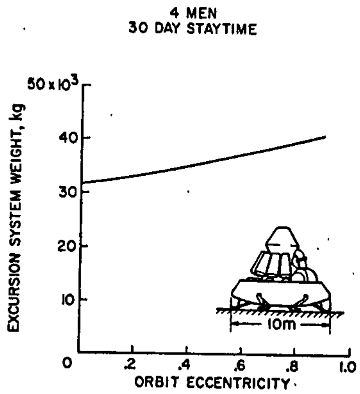

Figure 12.-Variation in excursion module weight with orbital eccentricity. Mars periapsis altitude 300 km.

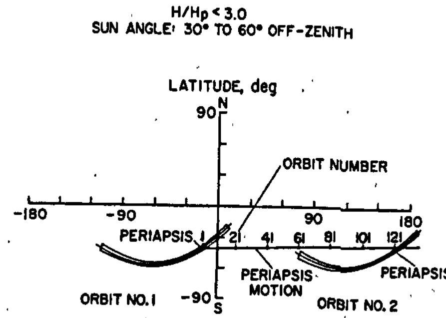

Figure 15.-Apparent motion over Mars surface of periapsis of odd-numbered orbits. First two orbit swaths shown.

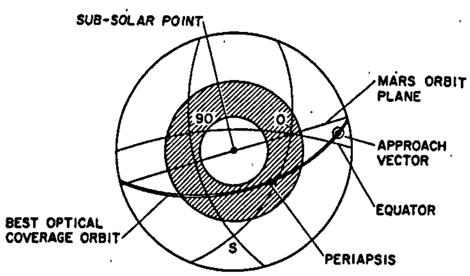

Figure 13.-Projection of planet's surface upon the sub-solar point at time of arrival. Mars 1971.

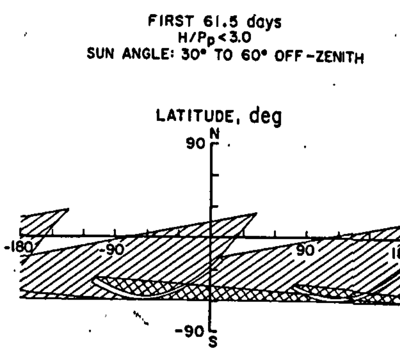

Figure 16.-Complete longitude coverage. Mars periapsis altitude 1000 km. Eccentricity 0.66.