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Velocity Requirements for a Soft Landing on Phobos or Deimos in 1969

Prepared by J. M. BAKER
Electronics Division

July 1968

Prepared for SPACE AND MISSILE SYSTEMS ORGANIZATION
AIR FORCE SYSTEMS COMMAND
LOS ANGELES AIR FORCE STATION
Los Angeles, California

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The information in a Technical Operating Report is developed for a particular program and is therefore not necessarily of broader technical applicability

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ABSTRACT

The velocity requirements to soft land on either Phobos or Deimos, the small Martian moons, are determined for the 1969 launch opportunity. A single impulse retro and plane change maneuver is applied at periapsis of the hyperbolic trajectory to achieve the desired final orbit about Mars. Less ΔV is required to attain the orbit of Phobos. This fact, combined with the lower orbital altitude, makes Phobos the preferred target for a soft landing. The minimum ΔV for a soft landing does not coincide with the minimum injection energy for a ballistic trajectory to Mars, so that a trajectory can be found that maximizes the payload landed on Phobos.

CONTENTS

FIGURES

SYMBOLS

$\underline{b}$	miss parameter vector
C_3	twice the energy/unit mass
D	denominator defined in Eq. (10)
$\hat{e}$	unit vector
i_s	inclination of spacecraft hyperbolic trajectory about Mars with respect to earth's equatorial plane
J	inclination of Phobos' or Deimos' orbit plane
J_1	function defined after Eq. (2) or (4)
$\hat{L}$	unit vector along line of intersection of spacecraft trajectory plane and orbit plane of either Phobos or Deimos
$\hat{M}$	unit vector along line of nodes of spacecraft trajectory plane and earth's equatorial plane
N	ascending node of either Phobos' or Deimos' orbit plane
N_1	function defined after Eq. (2) or (4)
η	numerator defined in Eq. (10)
R	orthogonal transformation matrix
r_p	periapsis radius relative to Mars
$\hat{S}$	unit vector parallel to asymptote of incoming hyperbolic trajectory relative to Mars
t	arrival date in Julian centuries of 365.25 days duration
V_h	hyperbolic excess velocity relative to Mars
ΔV	velocity required to circularize at radius r_p and change plane through angle ϵ
X, Y, Z	vernal equinox, earth equatorial coordinate system

SYMBOLS (contd.)

x, y, z	coordinate system with z normal to spacecraft hyperbolic trajectory plane
α_s	right ascension of $\hat{S}$
β	function defined after Eq. (2) or (4)
δ_s	declination of $\hat{S}$
ϵ	angle between spacecraft hyperbolic trajectory plane and orbit plane of either Phobos or Deimos
θ	rotation angle in target parameter plane
μ	Mars' gravitational constant = 4.29778×10^4 km^3/sec^2
ν	angle in spacecraft hyperbolic trajectory plane
ξ, η, ζ	coordinate system with ζ normal to orbit plane of either Phobos or Deimos
Ω_s	ascending node of spacecraft hyperbolic trajectory

SECTION I

INTRODUCTION

Early in 1969 the energy required for a mission to Mars falls within the capabilities of present day launch vehicles. Various missions, such as a lander, an orbiter, or a flyby, have been studied in the past. Each of these missions has a particular set of problems associated with it; e.g., entry into the tenuous Martian atmosphere imposes stringent restrictions on the trajectory. A mission that alleviates some of these problems involves a soft landing on Phobos or Deimos, the small Martian moons. This mission combines the advantages of an orbiter and a lander, and could be of significant scientific value. Some of the possible beneficial results are discussed in Reference 1 and are briefly outlined below. For example, a study of the moon's surface material could provide information about the origin of Mars. Photographic observation of the Martian surface and repeated occultation experiments could also be performed. Also, Doppler tracking data would provide an accurate determination of the moon's orbit, from which the first coefficients of Mars' gravitational potential in spherical harmonics could be precisely computed. Moreover, these data would provide an opportunity to test the hypothesis that Newton's constant varies with the gravitational potential (Ref. 1).

A soft landing on Phobos or Deimos is not planned by the United States; however, it may be an attractive mission for the Soviet Union to attempt during the 1969 opportunity. In addition to the scientific results mentioned above, the mission would be another space "first" for the Soviets, thereby providing further propaganda for their claim of technological leadership.

This report determines the velocity requirements to soft land on Phobos or Deimos as a function of launch date and flight time. The minimum ΔV required for a soft landing generally will not coincide with the minimum energy required for a ballistic trajectory to Mars. Thus, the possibility exists of maximizing the payload that is soft landed on either of the Martian moons. The results of this study, together with the payload exchange ratios for a particular vehicle configuration, can be used to determine the optimum combinations of launch date and flight time.

SECTION II

ANALYSIS

To minimize the ΔV requirements for a soft landing on either Phobos or Deimos, one would like the spacecraft plane of motion relative to Mars to coincide with the orbit plane of either Phobos or Deimos. A convenient coordinate system for determining the relative orientation of the orbit planes is the vernal equinox, earth equatorial coordinate system shown in Figure 1. In this system, X points toward the vernal equinox, Z is normal to the earth's equatorial plane, and Y completes the right-handed system. The orbit plane orientation of either Phobos or Deimos can be determined by the ascending node N and the inclination J, as shown in Figure 1. For Phobos, N and J are obtained as a function of date from (Ref. 2)

$$N = N_1 + [1.588^\circ + 0.00015^\circ (t - 1950.0)] \sin \beta \tag{1}$$

$$J = J_1 + 0.958^\circ \cos \beta \tag{2}$$

where

$$N_1 = 47.609^\circ + 0.00784^\circ (t - 1950.0)$$
$$J_1 = 37.114^\circ - 0.00411^\circ (t - 1950.0)$$
$$\beta = 277.6^\circ - 158.0^\circ (t - 1950.0)$$

and where t is measured in Julian years of 365.25 days duration.

Similarly, N and J for Deimos are given as a function of date by (Ref. 2)

$$N = N_1 + [2.917^\circ + 0.00028^\circ (t - 1950.0)] \sin \beta \tag{3}$$

$$J = J_1 + 1.733^\circ \cos \beta \tag{4}$$

where

$$N_1 = 46.553^\circ + 0.00762^\circ (t - 1950.0)$$

$$J_1 = 36.450^\circ - 0.00404^\circ (t - 1950.0)$$

$$\beta = 35.5^\circ - 6.374^\circ (t - 1950.0)$$

Define a ξ, η, ζ coordinate system, for Phobos, where ξ points toward the ascending node, ζ is normal to Phobos' orbit plane, and η completes the right-hand system. This coordinate system is related to the X, Y, Z system by the orthogonal transformation

$$\begin{pmatrix} \hat{e}_\xi \\ \hat{e}_\eta \\ \hat{e}_\zeta \end{pmatrix} = R_J R_N \begin{pmatrix} \hat{e}_X \\ \hat{e}_Y \\ \hat{e}_Z \end{pmatrix} \qquad (5)$$

where

$$R_J = \begin{pmatrix} 1 & 0 & 0 \\ 0 & \cos J & \sin J \\ 0 & -\sin J & \cos J \end{pmatrix}$$

$$R_N = \begin{pmatrix} \cos N & \sin N & 0 \\ -\sin N & \cos N & 0 \\ 0 & 0 & 1 \end{pmatrix}$$

The values of N and J are those given by Eqs. (1) and (2). Obviously, a similar coordinate system can be defined for Deimos. In this case, the values of N and J are obtained from Eqs. (3) and (4).

The spacecraft motion in the vicinity of Mars can be approximated by neglecting the gravitational attraction of all other bodies within Mars' sphere of influence. At entry into this sphere (Fig. 2), the spacecraft motion relative to Mars is specified by a unit vector $\hat{S}$ parallel to the asymptote of the incoming hyperbola, and by a miss vector $\underline{b}$ in the target parameter plane normal to $\hat{S}$. The direction of $\hat{S}$ is given in terms of the right ascension α_s and declination δ_s relative to the X, Y, Z coordinate system. The magnitude of $\underline{b}$ is nearly equal to the miss distance that would occur if Mars was massless.

Now define an xyz coordinate system where x is directed opposite to $\hat{S}$, y is directed along $\underline{b}$, and z completes the right-hand system. The angle θ, shown in Figure 2, is measured counter clockwise from a line in the target parameter plane parallel to the earth's equator. The x, y, z system is related to the X, Y, Z system by the orthogonal transformation

$$\begin{pmatrix} \hat{e}_x \\ \hat{e}_y \\ \hat{e}_z \end{pmatrix} = R_\theta R_{\delta_s} R_{\alpha_s} \begin{pmatrix} \hat{e}_X \\ \hat{e}_Y \\ \hat{e}_Z \end{pmatrix} \tag{6}$$

where

$$R_\theta = \begin{pmatrix} 1 & 0 & 0 \\ 0 & \cos\theta & \sin\theta \\ 0 & -\sin\theta & \cos\theta \end{pmatrix}$$

$$R_{\delta_s} = \begin{pmatrix} \cos\delta_s & 0 & -\sin\delta_s \\ 0 & 1 & 0 \\ \sin\delta_s & 0 & \cos\delta_s \end{pmatrix}$$

$$R_{\alpha_s} = \begin{pmatrix} -\cos\alpha_s & -\sin\alpha_s & 0 \\ \sin\alpha_s & -\cos\alpha_s & 0 \\ 0 & 0 & 1 \end{pmatrix}$$

It is seen in Figure 2 that the xy plane defines the spacecraft plane of motion relative to Mars. Using Eq. (6), the inclination of this plane relative to the earth's equator is

$$i_s = \cos^{-1}(\hat{e}_z \cdot \hat{e}_Z) = \cos^{-1}[\cos\delta_s \cos\theta] \tag{7}$$

Define a unit vector $\hat{M}$ in the direction of the ascending node of the spacecraft plane of motion by

$$\hat{M} \equiv \frac{1}{\sin i_s}(\hat{e}_Z \times \hat{e}_z)$$

Then the ascending node Ω_s of the spacecraft plane of motion relative to the X, Y, Z system is

$$\sin\Omega_s = \hat{M} \cdot \hat{e}_Y = \frac{-1}{\sin i_s}[\sin\alpha_s \sin\theta + \cos\alpha_s \sin\delta_s \cos\theta] \tag{8a}$$

$$\cos \Omega_s = \hat{M} \cdot \hat{e}_X = \frac{-1}{\sin i_s} [\cos \alpha_s \sin \theta - \sin \alpha_s \sin \delta_s \cos \theta] \qquad (8b)$$

The angles i_s and Ω_s are indicated in Figure 1. Equations (7) and (8) show that i_s and Ω_s can be varied within limits by varying θ, the angle that defines the point of entry into Mars' sphere of influence. For example, the minimum inclination is equal to the magnitude of δ_s, and is obtained when θ is zero. To minimize retro velocity requirements, θ is chosen to minimize the angle ϵ between the spacecraft plane of motion and the orbit plane of either Phobos or Deimos. The angle ϵ is found from

$$\cos \epsilon = (\hat{e}_\zeta \cdot \hat{e}_z) \qquad (9)$$

To find the angle θ that minimizes ϵ, differentiate Eq. (9) with respect to θ and equate the result to zero. Using Eqs. (5) and (6), the result is

$$\tan \theta \Big|_{\min \epsilon} = \frac{-\sin J \cos (\alpha_s - N)}{\sin J \sin \delta_s \sin (\alpha_s - N) + \cos J \cos \delta_s} \equiv \frac{\mathcal{N}}{D} \qquad (10)$$

where it is understood that the appropriate values of N and J must be used, depending on which Martian moon a landing is to be made. To solve for the correct quadrant for θ, note from Eqs. (2) and (4) that, regardless of the date, the motion of both Phobos and Deimos is direct with respect to the Z axis. To get the spacecraft motion in the same sense, it is seen from Figure 2 that θ must be restricted to

$$-\frac{\pi}{2} \leq \theta \leq \frac{\pi}{2}$$

Thus, from Eq. (10)

$$\cos\theta\Big|_{\min\epsilon} = \frac{D}{\sqrt{n^2 + D^2}} \geq 0 \qquad (11a)$$

$$\sin\theta\Big|_{\min\epsilon} = \frac{n}{\sqrt{n^2 + D^2}} \qquad (11b)$$

When Eqs. (11) are substituted into Eqs. (9), (5), and (6), the following result is obtained

$$\epsilon_{min} = \cos^{-1}\left[\left(n^2 + D^2\right)^{1/2}\right] \qquad (12)$$

For $\epsilon_{min} \neq 0$, define a unit vector $\hat{L}$ along the line of intersection between the spacecraft plane of motion and the orbit plane of either Phobos or Deimos by

$$\hat{L} \equiv \frac{1}{\sin\epsilon}(\hat{e}_\zeta \times \hat{e}_z) \qquad (13)$$

Because $\hat{L}$ is in the spacecraft plane of motion, it will have no z component in the x, y, z coordinate system. By allowing both positive and negative values for ϵ, the y component of $\hat{L}$ can always be made positive, regardless of the orientation of $\hat{e}_\zeta$ and $\hat{e}_z$.

When the angle ϵ is minimized (but not equal to zero), it can be shown that $\hat{L}$ is aligned with the positive y axis. That is, Eqs. (5), (6), and (10) can be used to verify that

$$\hat{L}\cdot\hat{e}_x\Big|_{\min\epsilon} = \frac{1}{\sin\epsilon}(\hat{e}_\zeta \times \hat{e}_z)\cdot\hat{e}_x\Big|_{\min\epsilon} = \frac{1}{\sin\epsilon}(\hat{e}_\zeta\cdot\hat{e}_y)\Big|_{\min\epsilon} = 0 \qquad (14)$$

Figure 3 shows the spacecraft hyperbolic trajectory prior to retro, together with the direction of $\hat{L}$ when ϵ is minimized and non-zero. To transfer into the orbit plane of Phobos, for example, the miss vector $\underline{b}$ is chosen so that the resultant hyperbolic periapsis radius r_p is identical to the orbit radius of Phobos. At periapsis, a retro ΔV is initially applied to circularize the spacecraft orbit. Then a second ΔV is subsequently applied at the line of intersection $\hat{L}$ (or $-\hat{L}$) to rotate the orbit plane through the angle ϵ_{min}. The total ΔV required to circularize and change plane is

$$\Delta V_2 = \left(V_h^2 + 2\frac{\mu}{r_p}\right)^{1/2} - \left(\frac{\mu}{r_p}\right)^{1/2}\left[1 - 2\sin\left(\frac{\epsilon_{min}}{2}\right)\right] \quad (15)$$

where

V_h = hyperbolic excess velocity

μ = Mars' gravitational constant

Alternatively, when $\epsilon_{min} \neq 0$, the retro and plane change maneuvers can be combined into a single maneuver by specifying that $\hat{L}$ point toward periapsis of the hyperbolic trajectory. However, the angle ϵ in this case is no longer minimized. To find the new value for ϵ, note first from Figure 3 that for $\hat{L}$ pointing toward periapsis,

$$\hat{L} = -\sin\nu\,\hat{e}_x + \cos\nu\,\hat{e}_y \quad (16)$$

where the angle ν can be found from

$$\sin\nu = \left(1 + \frac{r_p V_h^2}{\mu}\right)^{-1} \quad (17)$$

Using Eqs. (13) and (16), the scalar product of $\hat{L}$ and $\hat{e}_y$ is

$$\cos \nu = \frac{1}{\sin \epsilon} (\hat{e}_\zeta \times \hat{e}_z) \cdot \hat{e}_y = \frac{-(\hat{e}_\zeta \cdot \hat{e}_x)}{\sin \epsilon} \tag{18}$$

Since this equation is valid for arbitrary values of ν, it holds also when $\nu = 0$. However, $\nu = 0$ implies that ϵ is minimum from Eq. (14). Thus, one may write

$$\sin \epsilon = \frac{\sin \epsilon_{min}}{\cos \nu} \tag{19}$$

where

$$\sin \epsilon_{min} = -\hat{e}_\zeta \cdot \hat{e}_x = \cos J \sin \delta_s - \sin J \cos \delta_s \sin (\alpha_s - N) \tag{20}$$

The aiming point in the target parameter plane to align $\hat{L}$ with the hyperbolic periapsis can be determined by taking the scalar product of $\hat{L}$ and $\hat{e}_x$. Using Eqs. (13) and (16), obtain

$$-\sin \nu = \frac{1}{\sin \epsilon} (\hat{e}_\zeta \times \hat{e}_z) \cdot \hat{e}_x = \frac{\hat{e}_\zeta \cdot \hat{e}_y}{\sin \epsilon}$$

From the previously developed equations, the new angle θ in the target parameter plane is

$$\theta - \theta\Big|_{min\ \epsilon} = \sin^{-1}\left[\tan \epsilon_{min} \tan \nu\right] \tag{21}$$

where $\theta\Big|_{min\ \epsilon}$ is obtained from Eq. (10). With the value of ϵ calculated in Eq. (19), the ΔV required to perform the orbit transfer and plane change in a single maneuver is given by

$$\Delta V_1 = \left[V_h^2 + 3\frac{\mu}{r_p} - 2\sqrt{\frac{\mu}{r_p}}\sqrt{V_h^2 + 2\frac{\mu}{r_p}} \cos \epsilon \right]^{1/2} \qquad (22)$$

Note that for $\epsilon = \epsilon_{min} = 0$, Eqs. (15) and (22) give identical results. For $\epsilon_{min} \neq 0$, it can be shown that for the trajectories in this report, the single impulse retro maneuver requires less ΔV than the two-impulse maneuver. The actual ΔV magnitudes for Phobos and Deimos are given in the following section.

SECTION III

RESULTS

As shown in the previous section, the spacecraft retro and plane change maneuvers can be combined into a single maneuver when $\hat{L}$ points toward periapsis of the hyperbolic trajectory. The spacecraft hyperbolic trajectory is specified by the periapsis radius r_p, the hyperbolic excess velocity V_h, the right ascension α_s and declination δ_s of the incoming asymptote, and the orientation angle θ.

To determine r_p, note that the eccentricities of Phobos' and Deimos' orbits about Mars are 0.019 and 0.003, respectively (Ref. 3). Thus, it is assumed with little error that a transfer is made from a hyperbolic orbit to a circular orbit whose radius r_p is equal to the semimajor axis of either Phobos' or Deimos' orbit. The semimajor axes are computed from the orbital periods (Ref. 4), and are

$$r_p = 9384.6 \text{ km (Phobos)}$$

$$r_p = 23484.3 \text{ km (Deimos)}$$

For reference, these orbits are roughly at 2.8 and 6.9 Mars radii, respectively.

The hyperbolic excess velocity and the direction of $\hat{S}$ were obtained from microfilm data generated by JPL. Constant V_h contours are shown on a launch date-flight time graph in Figure 4. The launch dates and flight times encompass the region of minimum energy ballistic trajectories to Mars in 1969. If aerodynamic braking in the Martian atmosphere is disregarded, the minimum arrival velocity at the surface of Mars would be 6.190 km/sec (20,308 fps). It will be shown subsequently, that the ΔV requirements to soft land on Phobos or Deimos can be considerably less than this value. The lowest values for V_h will generally give the lowest ΔV requirements.

Values for α_s and δ_s are given in Figures 5 and 6, respectively. There is relatively little variation in the arrival direction. For most of the trajectories of interest, Figure 5 shows that α_s = 255 deg ± 10 deg; similarly, Figure 6 shows that δ_s = -10 deg ± 10 deg. Note that for a given launch date and flight time, the arrival date is specified. For this date,* N and J for Phobos or Deimos are found using Eqs. (1) and (2) or Eqs. (3) and (4). Using these data, the angle ϵ between the hyperbolic trajectory plane and the orbit plane of either Phobos or Deimos can be determined from the previously developed equations. The results for Phobos are shown in Figure 7. For most of the trajectories of interest, it is seen that the plane change angle is less than 10 deg.

The results for Deimos, shown in Figure 8, are quite similar. This is not surprising, because Eqs. (1) through (4) indicate that N and J for the two moons are very similar. It should be mentioned that the values for ϵ in Figures 7 and 8 are very close to ϵ_{min}. The reason can be seen from Eqs. (17) and (19). For the hyperbolic excess velocities given in Figure 4, and for the r_p's noted above, the angle ν is usually less than 10 deg. Hence, ϵ is only slightly greater than ϵ_{min}, and it is concluded that the single ΔV maneuver with $\hat{L}$ pointed toward the hyperbolic periapsis is an attractive orbit transfer technique.

The ΔV magnitude to attain the orbit of Phobos is shown in Figure 9. Note the similarity between these curves and the constant V_h contours in Figure 4. The minimum ΔV is 2.574 km/sec (8445 fps) for a launch on March 30, 1969 with a 200 day flight time. The ΔV requirements are greater than 3 km/sec (9843 fps) for any launch prior to March 5, regardless of flight time.

The ΔV required to attain the orbit of Deimos is shown in Figure 10. The minimum ΔV is 2.735 km/sec (8973 fps) for a launch on March 30 with a

*α_s and δ_s are given with respect to the mean equinox and equator of launch date, whereas N and J are with respect to the mean equinox and equator of arrival date. For the accuracies necessary in this report, the two coordinate systems are essentially identical.

198 day flight time. The ΔV magnitudes are somewhat lower to land on Phobos than on Deimos. The reason is that the final circular radius r_p is less, requiring a smaller retro maneuver for a fixed value of hyperbolic excess velocity. This fact, coupled with the better viewing of Mars due to the lower orbital altitude, makes Phobos the preferred target for a soft landing.

The minimum ΔV to attain the orbit of Phobos does not necessarily give the maximum landed payload. The energy required to inject the spacecraft into a ballistic interplanetary trajectory must also be considered. This injection energy is given as a function of launch date and flight time in Figure 11. The relationship between injection energy, injection altitude, and injection velocity (in fps) is indicated in Figure 12. Figure 11 shows that the minimum injection energy occurs for a launch on March 2 with a 178 day flight time, and does not coincide with the minimum ΔV to soft land on Phobos. Thus, for a given launch vehicle, the launch date-flight time combination can be found that maximizes the soft-landed payload. This combination will lie somewhere between the minimum ΔV and minimum injection energy. That is, the launch date for maximum landed payload will be between March 2 and March 30, 1969; the flight time will be between 178 and 200 days.

There are additional factors besides maximum payload that influence the trajectory selection process. For example, all of the Soviet lunar and interplanetary trajectories for the past 30 months have used a near-earth inclination of about 52 deg. If this is an operational constraint, the Soviets would exclude those trajectories above the limit line in Figure 11. Furthermore, this study is limited to heliocentric trajectories that have central angles from launch to arrival less than 180 deg. For a given injection energy, these trajectories generally have shorter flight times and correspondingly smaller communication distances at arrival than trajectories with central angles greater than 180 deg.

Many other parameters should be considered before a trajectory is ultimately chosen. Some of these, such as tracking visibility at injection, and lighting at arrival are discussed in Reference 5. A complete feasibility study of this mission is beyond the scope of the present investigation.

The ΔV requirements to attain the orbit plane of Phobos include only that required for retro and plane change. The correct phasing to perform a rendezvous with Phobos has not been considered. A possible method of correcting for improper phasing would be to apply small midcourse impulses in the $\hat{S}$ direction. These corrections mainly affect the arrival time, so that some control of spacecraft position with respect to Phobos is possible prior to the retro maneuver.

Finally, the trajectories in this report are limited to major thrusting periods only at the beginning and end of each trajectory. However, it may be advantageous to have a third major impulse midway through the trajectory (Refs. 6 and 7). A study of the class of trajectories is beyond the scope of the present investigation.

SECTION IV

CONCLUSIONS

The ΔV requirements to soft land on either of the two Martian moons have been determined for the launch opportunity in 1969. The method of orbit transfer is a single impulse retro and plane change maneuver applied at periapsis of the spacecraft hyperbolic trajectory.

Less ΔV is required to land on Phobos than on Deimos. Coupled with the additional advantage of a much lower orbital altitude about Mars, this result makes Phobos the preferred moon for a soft landing.

The launch date-flight time combination that gives the minimum ΔV for a soft landing is not a minimum-energy ballistic interplanetary trajectory. For a given launch vehicle, an optimum launch date-flight time combination can be found that maximizes the payload landed on Phobos.

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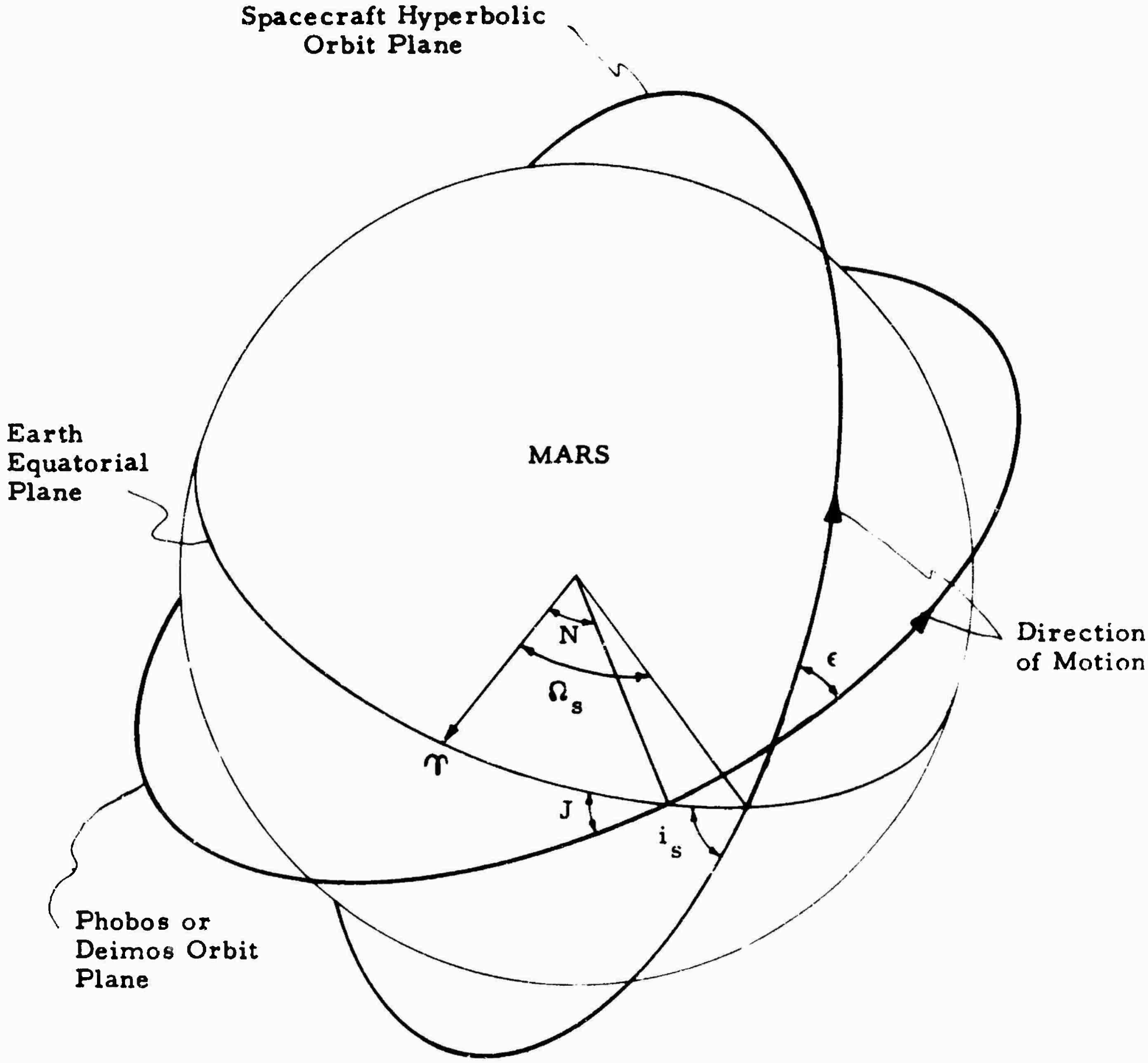

Figure 1. Orbit Plane Orientations of Spacecraft and Either Phobos or Deimos

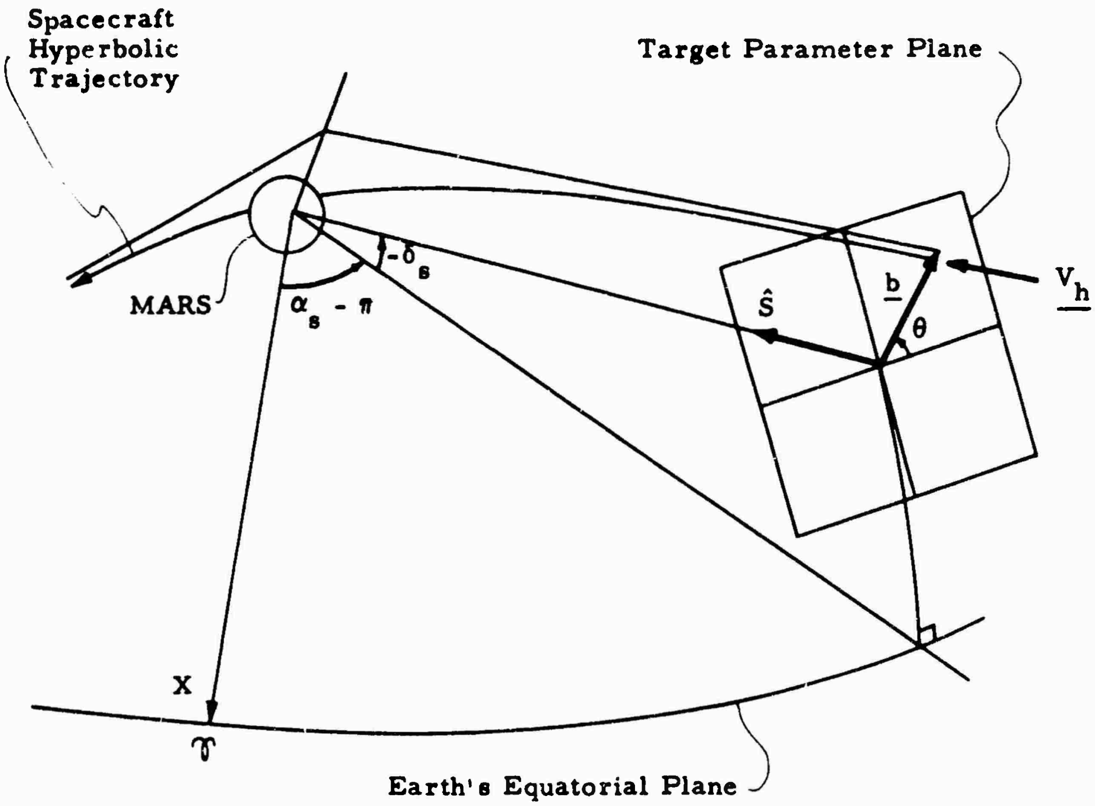

Figure 2. Spacecraft Trajectory Relative to Mars

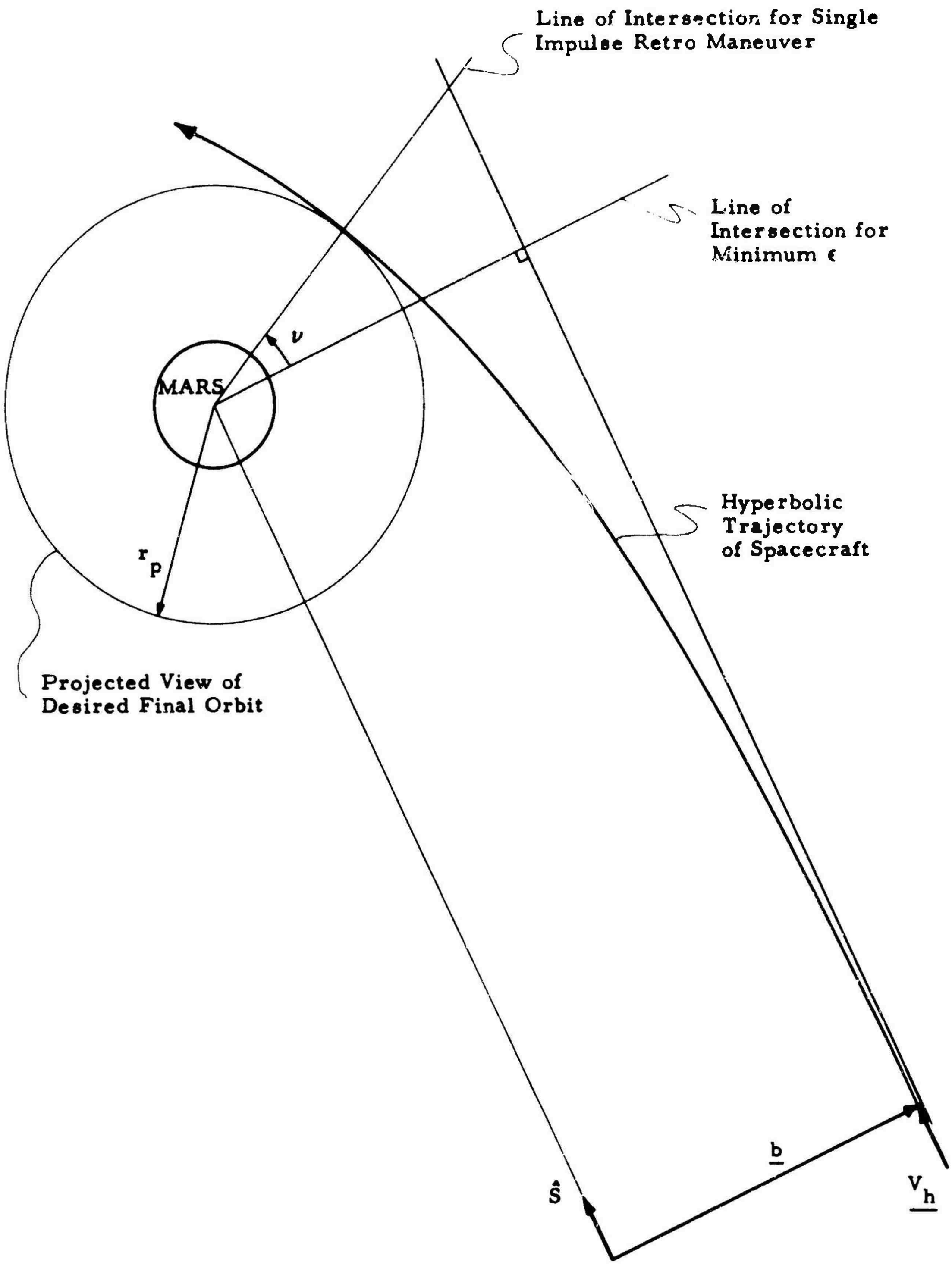

Figure 3. Spacecraft Trajectory Prior to Retro Maneuver

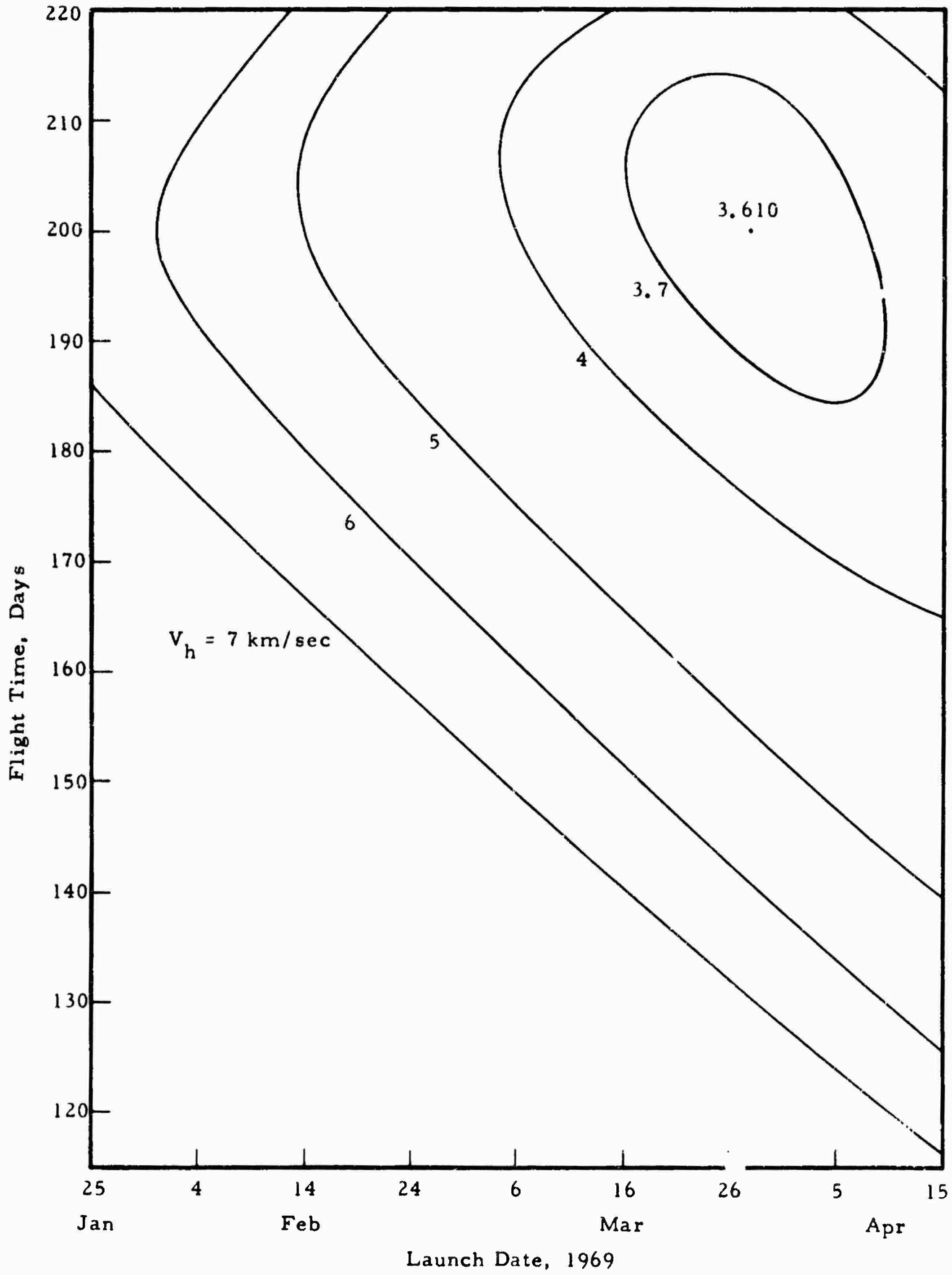

Figure 4. Hyperbolic Excess Velocity Relative to Mars

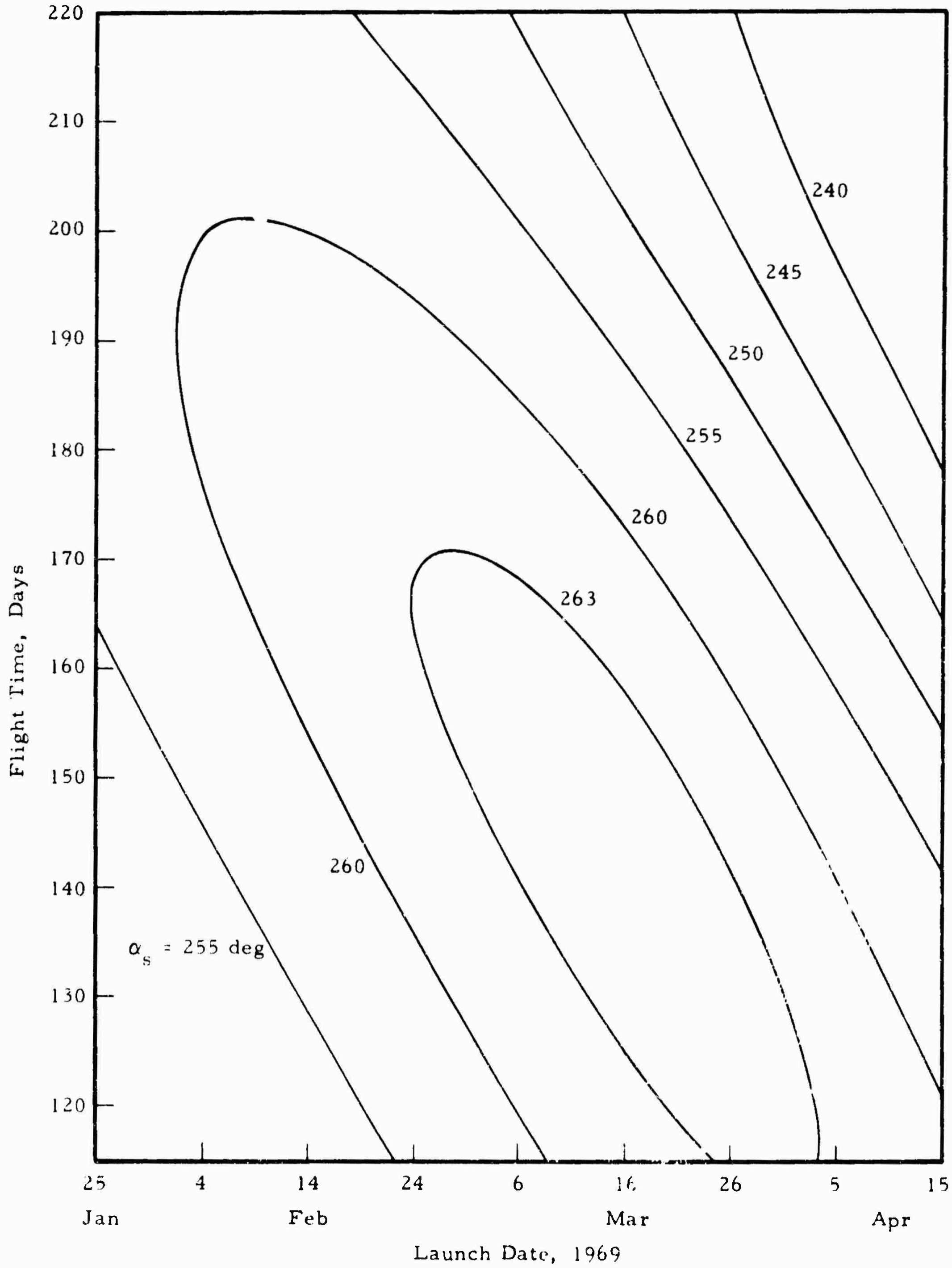

Figure 5. Right Ascension of Incoming Asymptote Relative to Mars

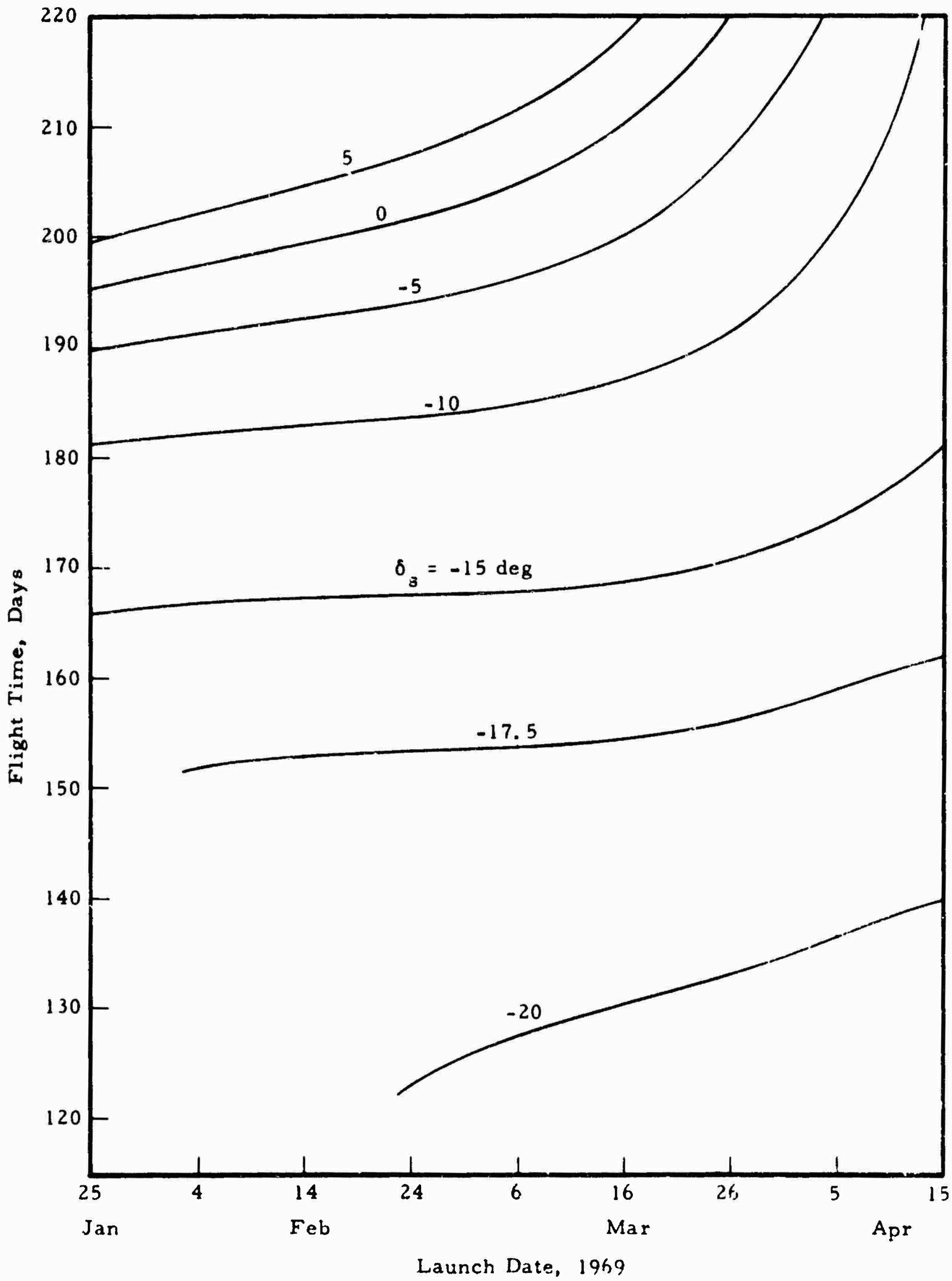

Figure 6. Declination of Incoming Asymptote Relative to Mars

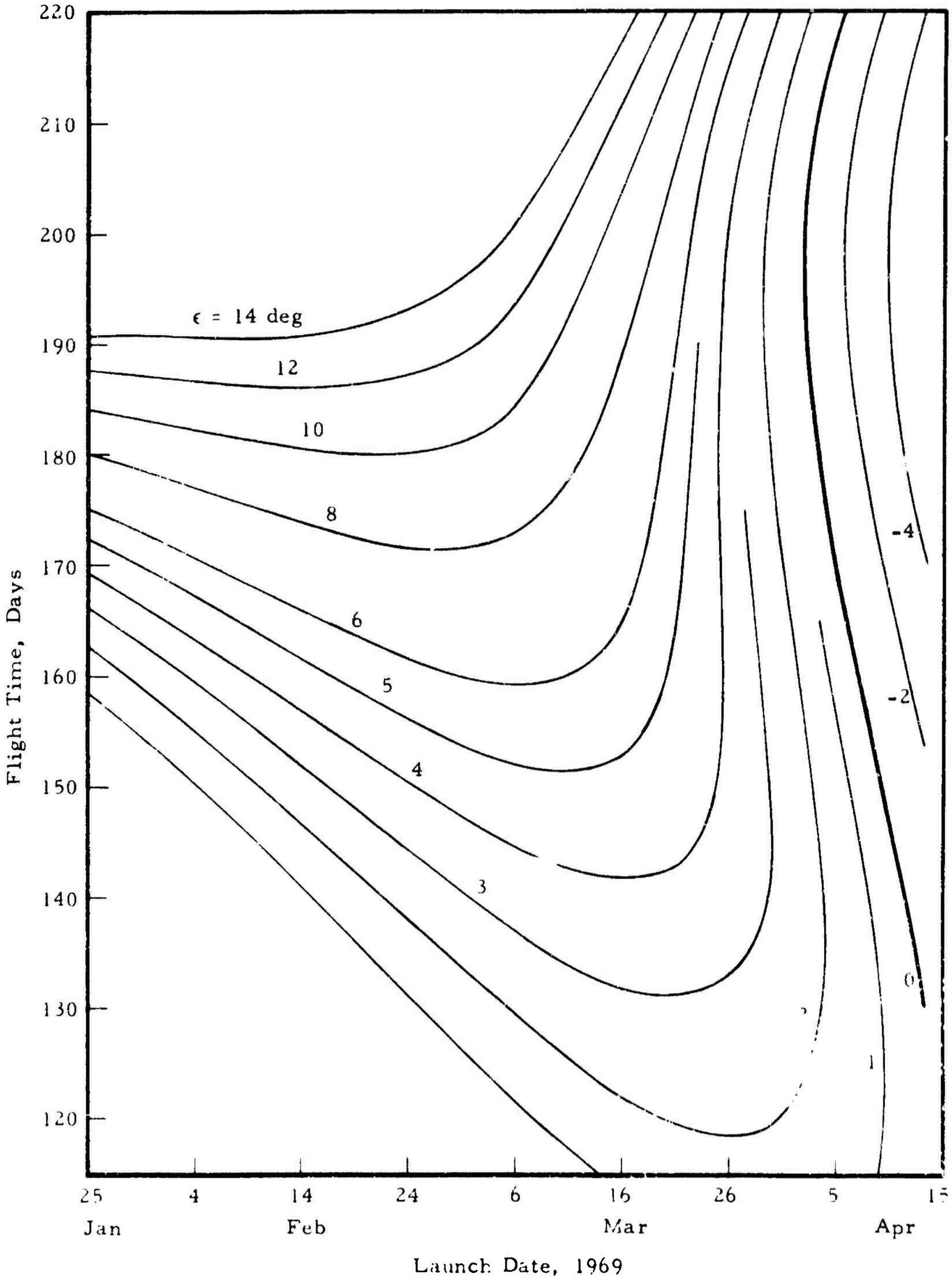

Figure 7. Plane Change Required to Attain Orbit Plane of Phobos

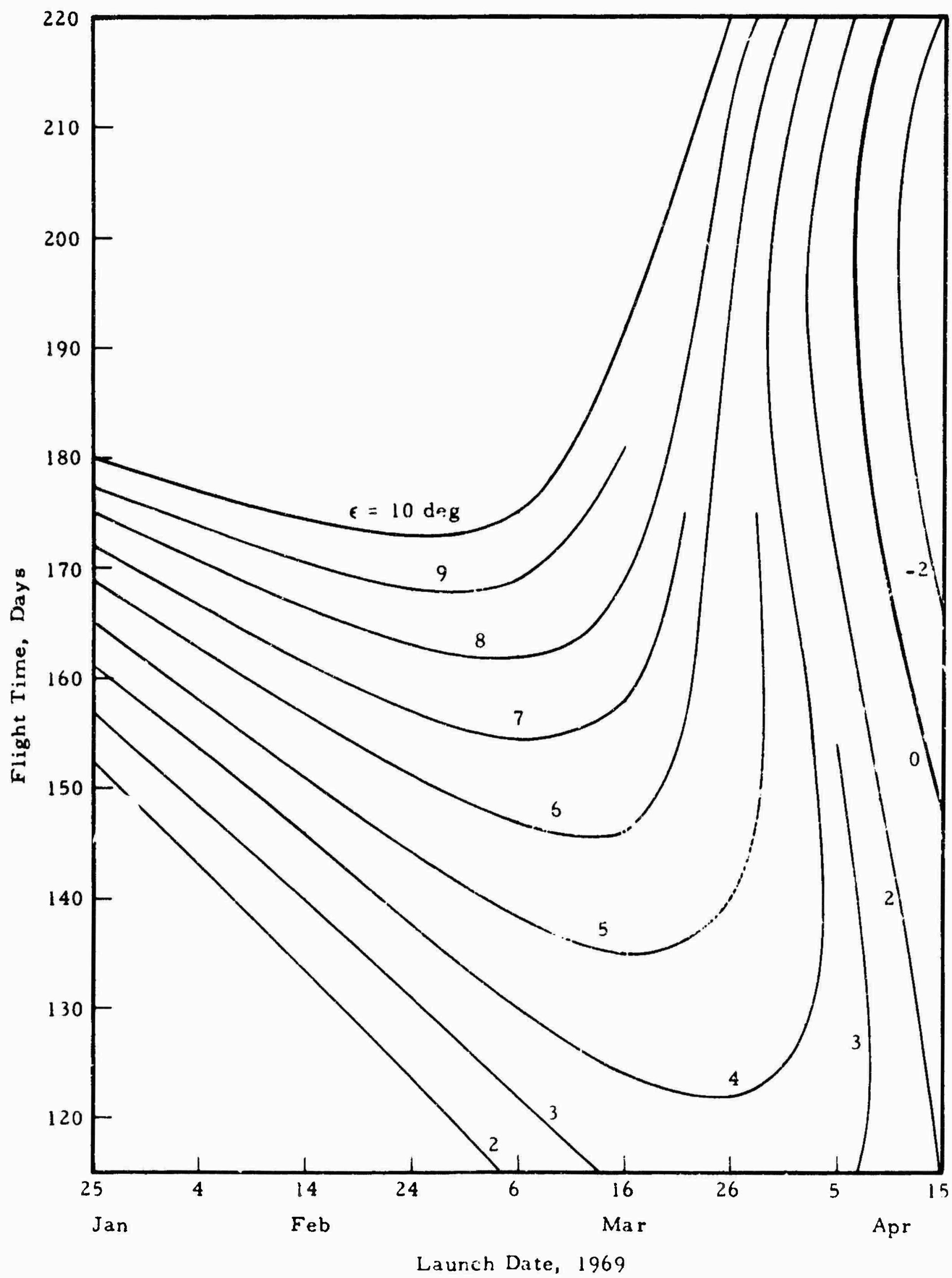

Figure 8. Plane Change Required to Attain Orbit Plane of Deimos

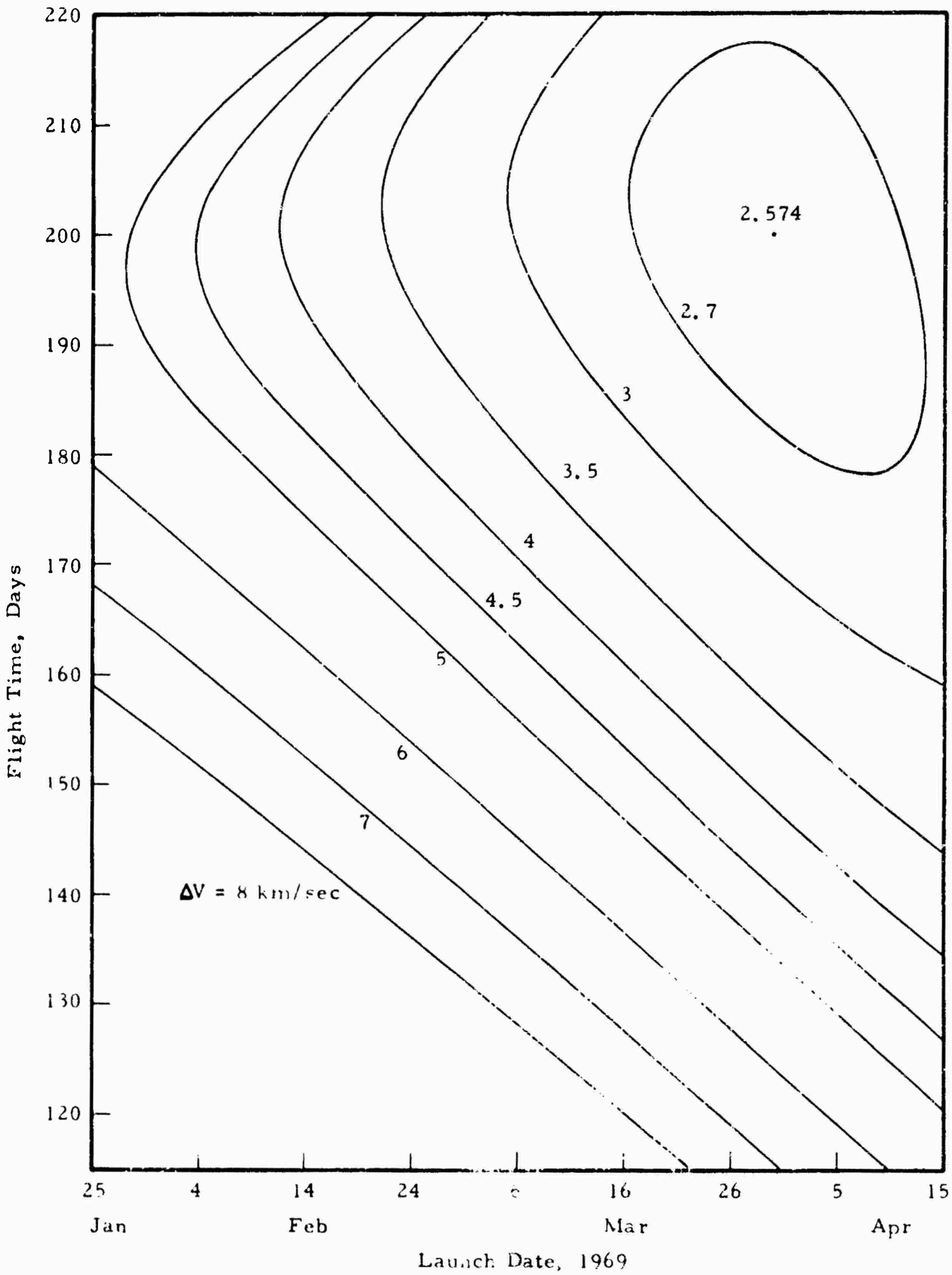

Figure 9. Single Impulse Velocity Requirements to Attain Orbit of Phobos

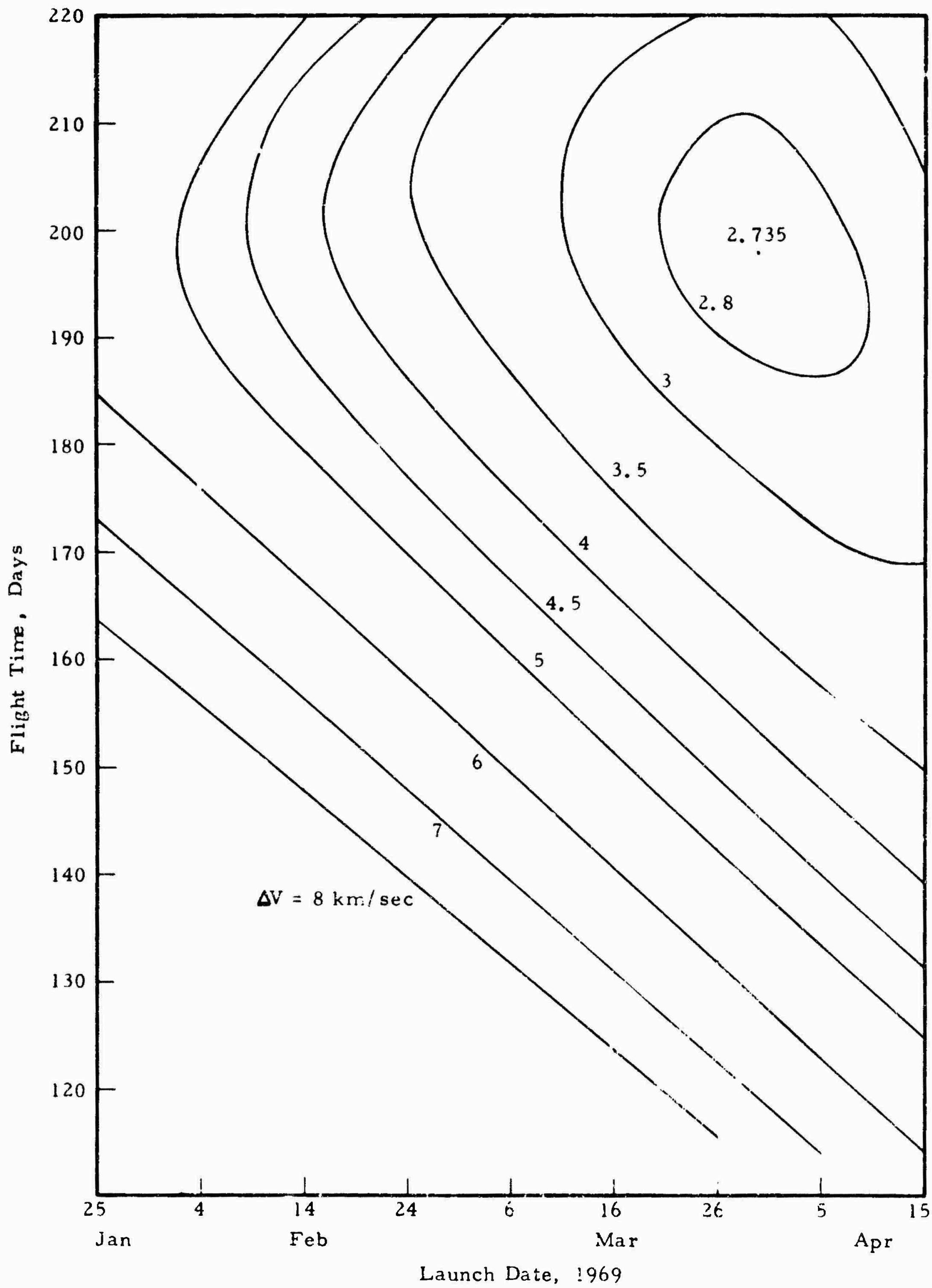

Figure 10. Single Impulse Velocity Requirements to Attain Orbit of Deimos

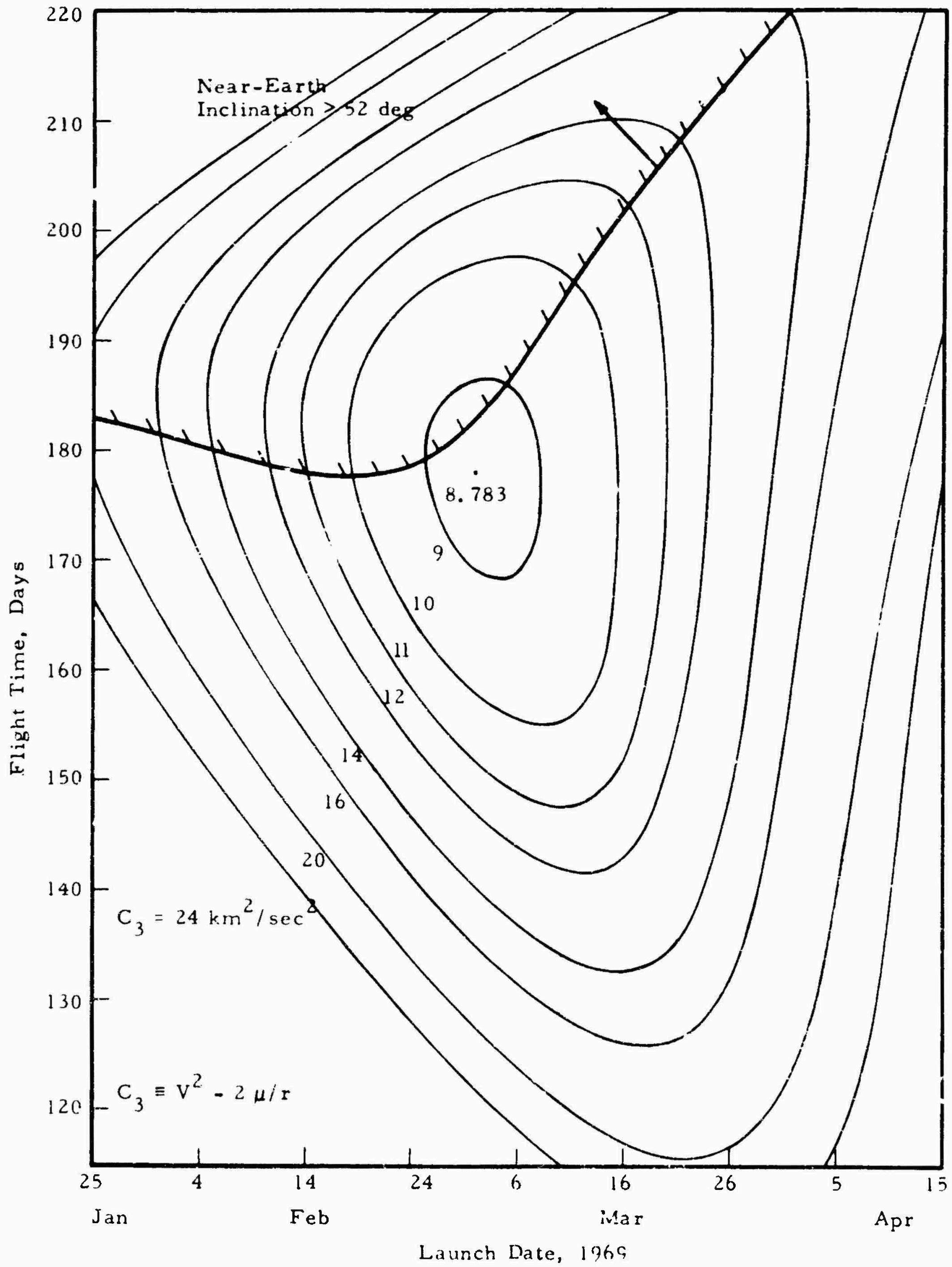

Figure 11. Energy Requirements for Ballistic Trajectories to Mars in 1969

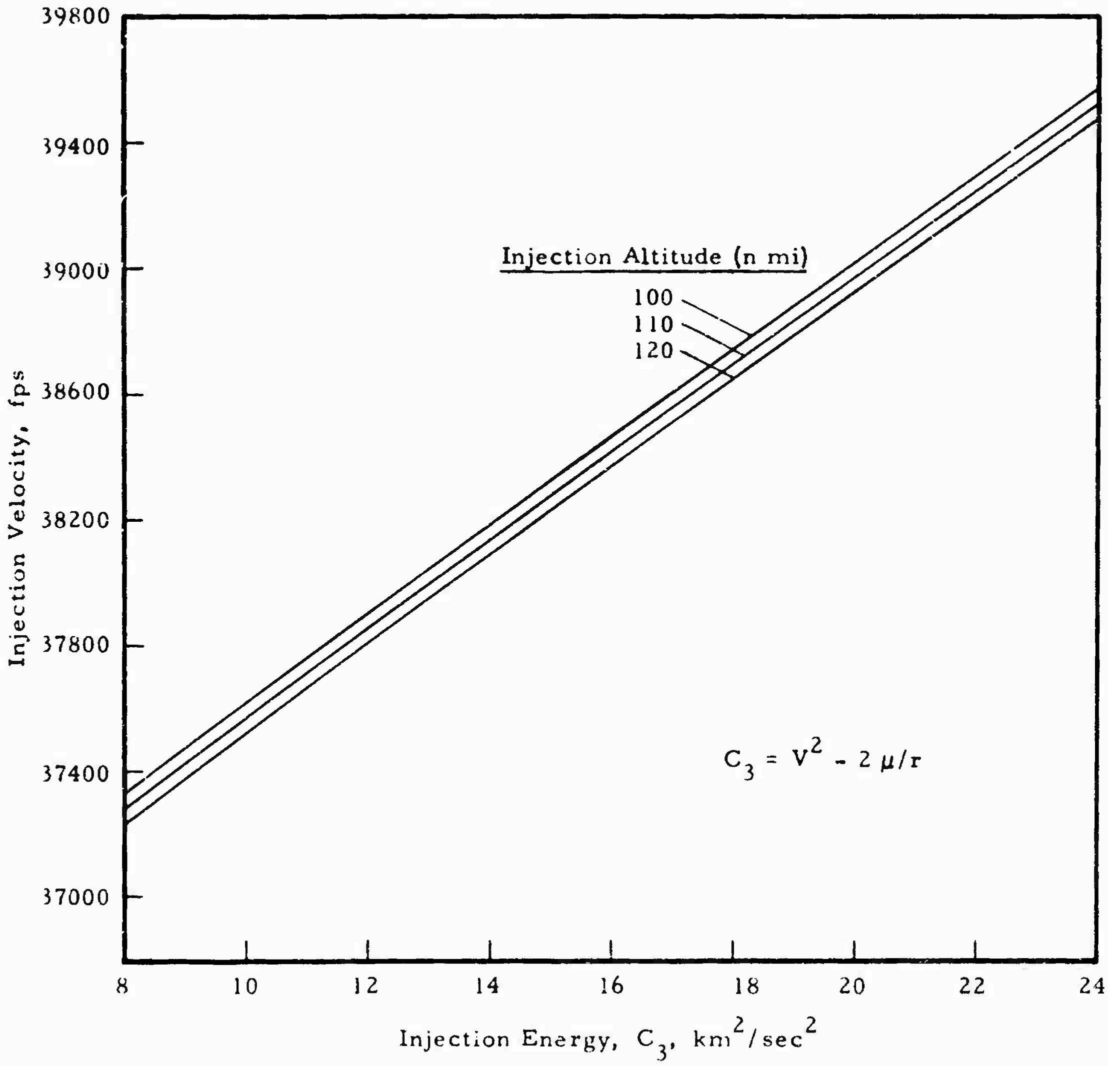

Figure 12. Variation of Velocity with Altitude and Energy at Injection

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13 ABSTRACT

The velocity requirements to soft land on either Phobos or Deimos, the small Martian moons, are determined for the 1969 launch opportunity. A single impulse retro and plane change maneuver is applied at periapsis of the hyperbolic trajectory to achieve the desired final orbit about Mars. Less ΔV is required to attain the orbit of Phobos. This fact, combined with the lower orbital altitude, makes Phobos the preferred target for a soft landing. The minimum ΔV for a soft landing does not coincide with the minimum injection energy for a ballistic trajectory to Mars, so that a trajectory can be found that maximizes the payload landed on Phobos.

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KEY WORDS

Martian Moons
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Deimos
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Martian Moon Soft Landing

Abstract (Continued)