Orbit Trim Strategies for the 1975 Mars Viking Mission

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Propulsive maneuver strategies have been developed for the Mars orbital phases of the Viking mission, including the activities before and after lander deployment. These strategies have been formulated as fixed sequences of orbit parameter-correction maneuvers which will be performed on specified spacecraft revolutions. Certain adaptive features are also available to ensure efficient use of propellants. Numerical simulations demonstrate that it is possible (to within a very high probability) to satisfy mission requirements on orbital navigation, using these strategies and Viking technology.

I. Introduction

THE United States will send two unmanned spacecraft to Mars during the 1975 launch opportunity. Each of these spacecraft will orbit Mars and deploy a soft-lander to the surface of the planet as part of the Viking Project. These objectives place a number of interesting and stringent requirements on the control of the satellite orbit to obtain near periapsis reconnaissance of the landing site and to prepare for lander release. As a result, the preflight navigation analyses and maneuver strategy development have been considerably increased in scope and complexity over previous missions. Reference 1 describes the Viking mission objectives and overall navigation profile from trans-Mars injection through the postlanding station-keeping phase.

During the Mars orbit trim (MOT) phase of each flight, a sequence of orbit trim spacecraft propulsive maneuvers is performed to remove the effects of orbit determination and maneuver execution errors, plus any intentional biases, remaining after earlier maneuvers in the flight. The sequence of trim maneuvers is performed according to a predetermined strategy designed to achieve the mission objectives.

This paper describes the maneuver objectives and the resulting strategies for the Mars orbital phases of the Viking mission, including the activities before and after lander deployment. The prelanding activities are treated in Sec. II–IV, with the consideration of postlanding activities being reserved for Sec. V. Numerical results are included in Sec. IV and V to demonstrate that the strategies satisfy all mission requirements which have been identified

The preflight navigation analyses for Viking have included the analyses of the two typical missions defined in Table 1. The typical candidate for the first spacecraft is referred to in this paper as Mission 1, while the candidate for the second spacecraft is called Mission 2. These example missions are considered in this paper to clarify techniques and to provide realistic numerical results. They are not intended to contribute a parametric analysis of all possible Viking missions.

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Index categories: Navigation, Control and Guidance Theory; Lunar and Interplanetary Spacecraft Systems, Unmanned; Spacecraft Navigation, Guidance, and Flight-Path Control Systems.

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Table 1 Selected parameters for Missions 1 and 2

Parameter	Mission 1	Mission 2
Launch date	8/11/75	8/21/75
Encounter date	6/18/76	8/7/76
Sun elevation angle at touchdown, deg	30	125
Latitude of landing site	19.5°N	44.3°N
Longitude of landing site	34°W	10°W
Period,[a] hr	24.6	24.6
Height at periapsis,[a] km	1500.0	1500.0
Inclination,[a] deg	33.4	48.8
Longitude of ascending node,[a] deg	122.8	22.3
Argument of periapsis,[a] deg	44.0	75.0
PER, deg	−6.7	−6.9

[a] This parameter is a spacecraft orbital parameter. The value given is the nominal value at lander touchdown with angles referenced to the aerocentric Mars mean equator and equinox of date coordinate system.

II. Prelanding Trim Maneuver Objectives

The orbit trim problem is to attain certain mission, spacecraft, and operations dependent objectives while coping with delivery, satellite orbit determination, and maneuver execution errors. The objectives of the prelanding orbit trim strategies for Viking are as follows.

1) To satisfy the requirements for landing. The orbit of the spacecraft must be controlled to pass through a prescribed spacetime region from which the lander can maneuver to the desired landing site without violating any design constraints. The orbit is specified for that revolution during which the Viking lander separates from the orbiter. The five parameters to be controlled are shown in Fig. 1 and include the downrange *DR* and cross-

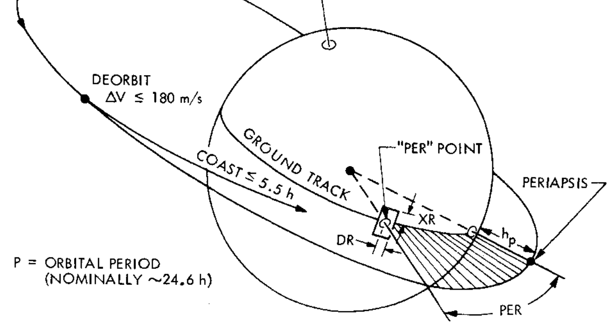

Fig. 1 Satellite orbit control for lander separation.

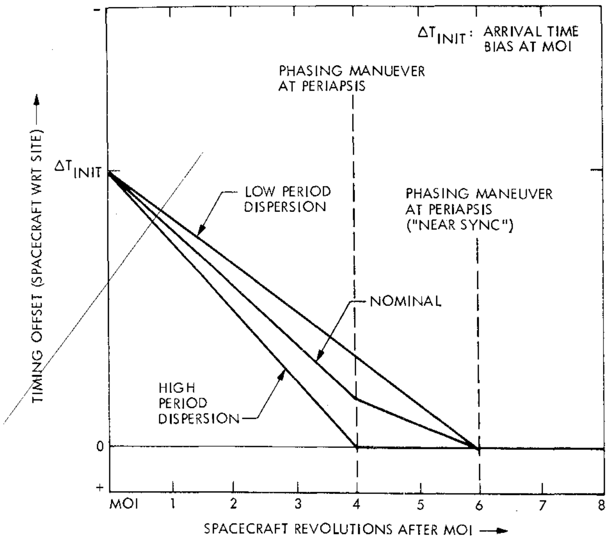

Fig. 2 MOI targeting biases for Mission 2. Although the timing offset is defined only at those times when the orbiter is at the *PER* point, continuous lines are used in figures of this type to enhance their readability.

range XR errors of the PER point† with respect to the desired landing site, the orbital period P, the periapsis altitude h_p, and the timing at lander touchdown. The current specified tolerances for these five control parameters are given in Table 2.

Table 2 Final 99% control tolerances and dispersions

Parameter	Ref. 2 tolerances	Latest desired tolerances	Final control 99% dispersions: Mission 1	Mission 2
Period, min	±5	±5	−2.5 to 2.5	−1.3 to 1.2
h_p, km	±300	±200	−58 to 33	−45 to 49
Timing, min	±5	*a*	−1.3 to 1.9	−1.9 to 1.7
DR, deg	±1.75	±2	−1.0 to 0.9	−1.6 to 1.6
XR, deg	±4	±3	−1.3 to 1.4	−2.4 to 2.4

a The permissible timing error is included in the $DR \times XR$ tolerances.

2) To provide near-periapsis site reconnaissance as soon as possible following the orbit insertion maneuver. This objective is especially important for the first spacecraft for mission design reasons. For a site certification reconnaissance sequence, it is necessary to observe the landing site at an emission angle‡ of less than 45°.

3) To provide adequate orbit determination and command generation time between maneuvers by allowing at least 36 hr between the orbit insertion maneuver and the first trim and 32 hr between successive trims. Also, one and one-half spacecraft revolutions must be allowed between the last prelanding trim and lander separation.

4) To make efficient use of propellant capability. In particular, the (mission dependent) velocity increment budget for all prelander release orbit trims, together with the variation from the nominal velocity magnitude of the orbit insertion maneuver, must not be exceeded. Also, the magnitude of each prelanding trim shall not exceed 145 m/sec.

Each orbit trim maneuver strategy is analyzed by means of a Monte Carlo error analysis. If the four objectives above are satisfied for 99% of the Monte Carlo samples, the strategy is acceptable.

III. Prelanding Trim Strategies

MOI Targeting Strategy

The Mars orbit insertion (MOI) maneuver is the first of a sequence of Mars orbital flight path changes designed to attain a synchronous (~24.6 hr) orbit from which the Viking lander can make a successful landing. The MOI strategy[3] takes the estimated spacecraft trajectory and targets a finite burn maneuver to achieve the desired postinsertion elliptical orbit.

The first facet of a Mars orbit trim (MOT) strategy affects the MOI maneuver in the selection of the initial satellite orbit period and time of arrival biases. To determine the post-MOI target period, it is necessary to trade off early site reconnaissance of the primary landing site (objective 2) against the statistical navigation velocity cost $\Delta VSTAT$.§ This situation results from the fact that a post-MOI target period which is sufficiently supersynchronous reduces the $\Delta VSTAT$ cost since it ensures that each of the period-change maneuvers reduces the size of the spacecraft orbit. That is, each of these maneuvers takes energy out of the system. However, such a biased period, together with an initial timing bias, also decreases the probability of satisfying the site reconnaissance requirements on early orbits. A timing bias introduces a shift between the longitude of the landing site and that of the spacecraft at the PER point.

The preflight analyses for Viking have demonstrated that the first spacecraft can be targeted to a synchronous period with no timing bias for all candidate missions. That is, the navigation velocity cost is within budget for all such missions.

However, the need for early reconnaissance is not as important for the second spacecraft as for the first and the $\Delta VSTAT$ cost can be large for candidate missions of the second spacecraft. Since $\Delta VSTAT$ is minimized by selecting the supersynchronous period and arrival time bias which causes the fan of dispersed post-MOI cases to slope downward (see Fig. 2), the optimal values of the period and arrival time biases are determined as the solution of two linear equations in two unknowns to be 35.6 hr and −3.6 hr, respectively. To calculate these optimal target values, it is necessary to know the expected post-MOI period dispersions and the fact that the first two time-phasing maneuvers are performed at the fourth and sixth periapsis passages. In this calculation, the period change introduced by the first trim MOT1 is neglected. (See Fig. 3, which indicates the trim maneuver timeline for Mission 2. All of the maneuvers are discussed in the next section.) For periods dispersed outside the fan of Fig. 2, additional ΔV is required because energy must be added to the system in order to perform the time-phasing with maneuvers on the indicated periapses. The final selection of a 31-hr period and a −6.4-hr arrival time bias (equivalently, longitude offset) was made for mission design reasons such as the desire to overfly the primary landing site at the first periapsis.

The data given in Sec. IV reflect these post-MOI targeted values. Additional details concerning this targeting selection process are given in Ref. 4.

Trim Strategies

Following the spacecraft insertion into orbit, a sequence of orbit trimming maneuvers is performed to satisfy the landing and site reconnaissance constraints defined in objectives 1 and 2. The timeline for these trims and some of the various tradeoffs required in selecting the timeline are considered for each of the strategies in turn. The strategies described in this section are motivated by geometrical factors (discussed below) plus the need for opera-

† The PER angle is the true anomaly of the point in the orbit that is nominally placed directly above the landing site. The PER point is the subspacecraft point at the PER angle.

‡ Emission angle is defined as the angle between the local vertical at the point of interest on the surface of the planet and the vector from this point to the spacecraft.

§ $\Delta V_{MOI\text{-}REF}$ is the nominal finite burn MOI ΔV for attaining a synchronous orbit with a 1500.0 km periapsis altitude and a specified orientation. $\Delta VSTAT$ is the 99% navigation velocity cost for MOI and preseparation trims for acquiring the primary landing site, less the value of $\Delta V_{MOI\text{-}REF}$.

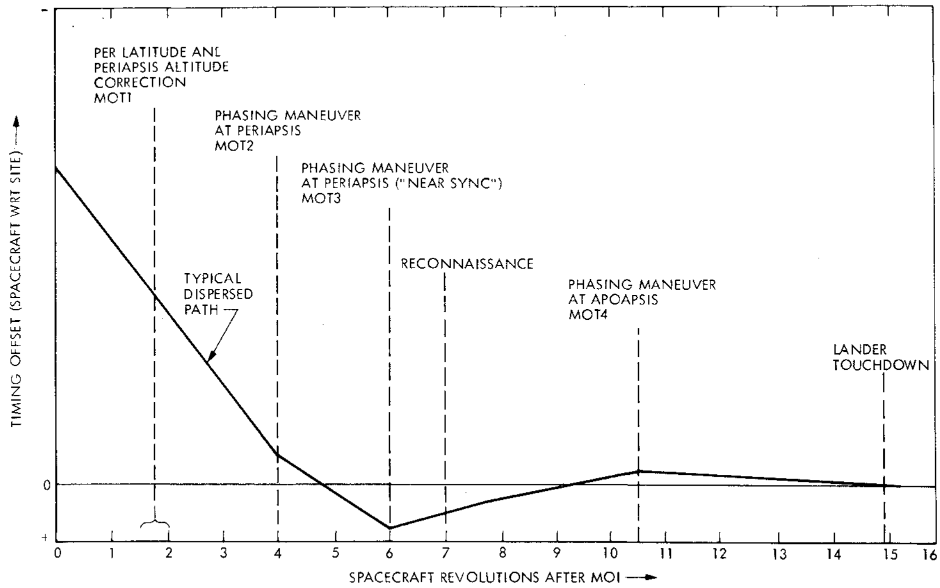

Fig. 3 Delayed reconnaissance strategy.

tional flexibility and simplicity. These strategies have been formulated as fixed sequences of orbit parameter-correction maneuvers (see Tables 3 and 4) which will be performed on specified spacecraft revolutions. Certain adaptive features are also available to ensure efficient use of propellants. The algorithms required to simulate these strategies are specified in Ref. 5.

Table 3 Parameter-correction capabilities

1) Correct the height of periapsis (h_p) and the latitude of the *PER* point to within acceptable tolerances by a ΔV-optimal maneuver.

2) Change period by an in-plane maneuver performed at a specified point near periapsis to produce a desired timing offset on a later spacecraft revolution or to obtain a desired period.

3) Change period by an in-plane maneuver performed at a specified point near apoapsis to time-phase, subject to the constraint that the period and height at periapsis remain within tolerances.

4) Change period and the argument of periapsis to correct the latitude of the *PER* point to within acceptable tolerances by a ΔV-optimal in-plane maneuver.

5) Change inclination to correct the latitude of the *PER* point to within acceptable tolerances without changing period or h_p by a maneuver performed approximately normal to the orbital plane at a Mars equator crossing point.

6) Retarget the latitude of the *PER* point and time-phase in a nearly ΔV-optimal manner.

Table 4 Prelanding maneuver sequences

Strategy	Trim No. 1	Trim No. 2	Trim No. 3	Trim No. 4
Nominal site acquisition strategies:				
1) Delayed reconnaissance	1[a]	2	2	3
2) Early reconnaissance—Type *L* (typically used for *l*ow inclination orbits)	2	2	5	3
3) Early reconnaissance—Type *M* (typically used for *m*edium inclination orbits)	2	2	1	3
4) Early reconnaissance—Type *H*	4	2	3	...
(typically used for *h*igh inclination orbits)	2	4	3	...
5) Secondary site	2	2	3	...
	1	2	2	3
Alternate mission strategy:				
6) Alternate mission	6	2	3	...

[a] The number given for each trim specifies a maneuver type defined in Table 3.

1) Delayed reconnaissance strategy

This strategy was designed for missions having large expected post-MOI periapsis altitude or orientation dispersions, a leisurely site reconnaissance schedule with a relatively long time between MOI and lander touchdown, and a need for ΔV conservation. The maneuver timeline for this strategy is illustrated in Fig. 3 and the maneuver sequence is given in Table 4. A detailed description of the maneuvers follows.

The first trim maneuver MOT1 is designed to correct the periapsis altitude to the nominal value of 1500 km and to place the *PER* point at the required latitude to within tolerances. Large post-MOI dispersions in periapsis altitude h_p dictate that this parameter must be corrected early to provide satisfactory reconnaissance at the seventh¶ periapsis (P7) after MOI. (Note that h_p is not corrected until the last trim in the Early Reconnaissance Strategies described below.) The h_p correction is combined with an orientation change because it is possible to correct h_p with very little additional velocity cost. If no orientation correction is required, only h_p is corrected by a maneuver performed at apoapsis. The large dispersions in these parameters also require that they be corrected prior to the phasing maneuver to avoid degradation of the final period and timing control by errors experienced from this maneuver. Only the latitude of the *PER* point is corrected since the longitude error can be interpreted as a timing offset and corrected more economically in the phasing sequence (MOT2 and MOT3) to follow.

The trim MOT1 is performed in a ΔV-optimal manner, while being restricted to occur between the second apoapsis and second periapsis passage after the orbit insertion maneuver. The maneuver direction and position in the orbit is determined to obtain a combination of changes in inclination and argument of periapsis that corrects the latitude of the *PER* point for minimum velocity cost. If the post-MOI target period is biased to be supersynchronous, then another ΔV-optimization feature results from the fact that h_p and orientation can be corrected for less cost in a supersynchronous orbit than in a (smaller) nearly synchronous one.

Once the periapsis altitude has been corrected to within tolerances of the desired value, it is possible to correct the period and timing errors by changing the height h_a at apoapsis only. These corrections are made in two time-phasing maneuvers performed at periapsis. The trim MOT2 is performed at P4 to achieve an orbital period such that the desired site longitude offset is produced at P6, where MOT3 is performed to achieve a

¶ To simplify the discussion, the location of each maneuver is given as that required for the Missions 1 and 2. The positioning of the maneuvers is actually mission dependent.

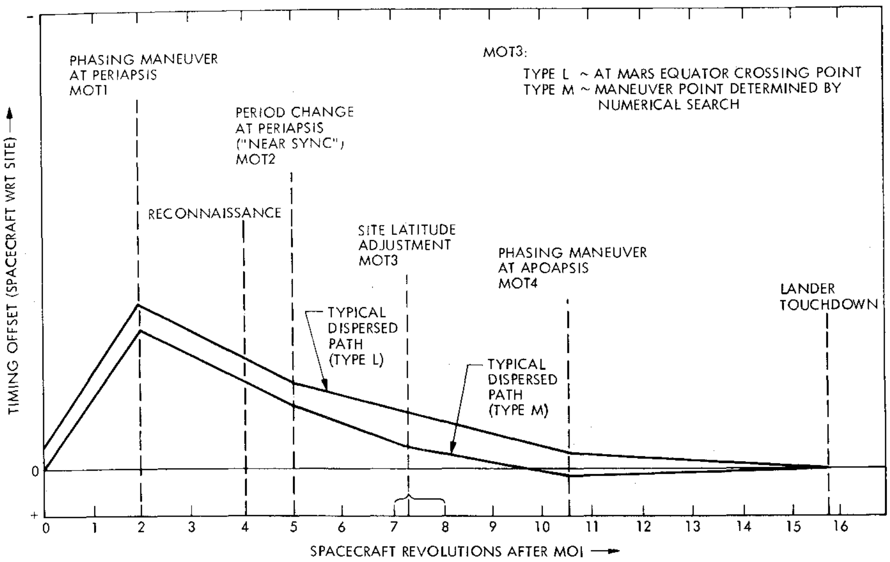

Fig. 4 Types L and M early reconnaissance strategies.

nearly synchronous orbit. In the absence of errors, these maneuvers would synchronize the *PER* point to the landing site longitude, except for a minor adjustment which reflects the fact that the lander must lead the orbiter between deployment and touchdown. However, once MOT2 has been simulated, MOT3 is determined to nullify the final timing error at landing. The strategy is designed so that this phasing sequence always shrinks the size of the orbit if possible. For those dispersed orbits for which this is not possible at both MOT2 and MOT3, MOT2 is selected to achieve the supersynchronous or subsynchronous post-MOT2 period which minimizes $\Delta V_{\mathrm{MOT2}} + \Delta V_{\mathrm{MOT3}}$.

The locations of the maneuvers MOT2 and MOT3 have been chosen to satisfy objectives 2 and 3. It should also be noted that performing them at periapsis minimizes the velocity increment needed for period changes. It also permits making the maneuvers along the orbiter's velocity vector, which nullifies the first-order effects of pointing errors, an important error source in the control of period and timing. However, in the flight operations, MOT2 and MOT3 may be performed at a specified point near but not at periapsis in order to achieve precise period control using a ground-based maneuver start command.[6]

The fourth and final prelander-release trim is performed at the eleventh apoapsis after MOI to achieve accurate timing control of the separation orbit. This maneuver removes the timing error which accumulates during the $4\frac{1}{2}$** orbits after MOT3. Performing MOT4 at apoapsis reduces the effect of fixed execution errors to about $\frac{1}{8}$ that of a maneuver at periapsis. Hence, having neutralized the effects of the only execution errors which are significant at MOT4, the timing error at touchdown is approximately $4\frac{1}{2}$** times the uncertainty in estimating the orbital period at MOT4.

The placement of MOT4 must be properly balanced between P6 and P15. Location too soon after P6 permits too much time for timing error growth (which accumulates linearly in time) by P15, violating the 5-min timing control requirement of objective 1. Location too near P15 would not allow enough time before P15 to correct the accumulated timing error without violating the tight period and h_p tolerances shown in Table 2. This concern for period and h_p is introduced by the fact that the maneuver at apoapsis adjusts h_p to change the period in removing the timing error. Selection of the midpoint (eleventh) apoapsis was determined to achieve the best tradeoff between the two extremes.

Observe that only the first trim is made in a direction out of the spacecraft's orbital plane. The other trims change orbit size and shape parameters only.

** This value is mission dependent. See earlier footnote¶ and Fig. 3.

2) *Type* L *early reconnaissance strategy*

This strategy was designed for missions having a low inclination orbit, small post-MOI orientation dispersions, and a need for very early site reconnaissance data. The maneuver timeline is illustrated in Fig. 4 and the maneuver sequence is given in Table 4. The wait time between MOI and lander touchdown in Fig. 4 is about 16 spacecraft revolutions. However, for candidate missions late in the set of nominal missions, this wait time reduces to only 12 revolutions, necessitating a much more compact maneuver timeline.

To provide reconnaissance at the fourth periapsis, the first two maneuvers are time-phasing trims and the landing site latitude correction is delayed until MOT3. The third maneuver corrects the *PER* point to an acceptable latitude by means of a pure inclination change at one of the Mars equator crossing points. The fourth maneuver is a periapsis altitude and time-phasing maneuver at apoapsis to improve the final timing control by minimizing the effect of execution errors. In somewhat more detail, the purpose of each of the Type L trims may be explained as follows.

The objective of the timing strategy is to make (in the absence of errors) the final maneuver a synchronizing one performed at apoapsis that corrects the periapsis altitude. Thus, immediately after MOT4, the timing error must be zero and the height h_a at apoapsis must assume the nominal value $h_{a_{\mathrm{nom}}}$, the value for h_a which corresponds to a synchronous period with $h_p = 1500.0$ km. Hence, the period just prior to MOT4 must be P_4, where P_4 is the period of an orbit having the nominal value $h_{a_{\mathrm{nom}}}$ for the height at apoapsis and the uncorrected value h_p for the height at periapsis. These requirements essentially determine the point (left-hand limit) at MOT4 and the slope of the line just before this trim (see Fig. 4). The slope and point then determine the desired timing offset at MOT2, which in turn determines the target period for MOT1.

To implement this strategy, MOT1 is a time-phasing maneuver made at periapsis passage to minimize ΔV usage. This maneuver must phase in such a manner that the h_p change in MOT4 and the plane change in MOT3 are anticipated.

The second trim MOT2 is made at P5 to achieve an orbital period equal to 24.6 hr minus the period change ΔP_4 anticipated at MOT4. (MOT3 is a pure inclination-change maneuver and does not, except for errors, alter period as shown in Fig. 4. There is, however, a change in the timing offset at MOT3 introduced by the instantaneous change in the right ascension of the *PER* point with respect to the landing site. For simplicity, this small change is not illustrated in Fig. 4. The same practice has been followed in Fig. 3.) The MOT2 corrects apoapsis altitude to the nominal value.

The purpose of MOT3 is to correct the latitude of the orbiter *PER* point to an acceptable latitude band determined by the target $DR \times XR$ tolerance zone centered about the *PER* point (see Fig. 1). This maneuver can be performed as a pure inclination change at one of the two Mars equator crossing points for non-equatorial orbits. The ΔV-optimal point is used, unless some other constraint necessitates making the maneuver at the non-optimal point. As MOT1 and MOT2 have dealt only with time-phasing and synchronizing, the effects of pre-MOI approach orbit determination errors normal to the desired orbital plane have not been corrected. This rotation error in the satellite orbit could be corrected in the ΔV-optimal manner discussed for the Type M case below. However, most of the correction tends to be in the inclination direction for low inclination orbits. Also, the complexity of the algorithm[5] and the computer running time (for certain missions) can be significantly reduced by this method, which is important for Monte Carlo analyses. The method does require that the argument of periapsis ω not be dispersed sufficiently toward the equator that it is not possible to correct the latitude of the *PER* point without changing ω.

In the presence of errors, the final trim MOT4 is made as a time-phasing maneuver to improve the final timing control. MOT4 is performed subject to the control requirements on the height at periapsis and orbital period given in Table 2.

3) Type M *early reconnaissance strategy*

This strategy is a generalization of the Type *L* strategy considered above. The timing strategy is the same as that for Type *L*, except that there may be a period change at the time of MOT3, which might occur anywhere in an orbital segment. The apoapsis altitude is targeted to the nominal value and h_p is allowed to float. Figure 4 shows the change in slope at MOT3 for a typical dispersed path.

The post-MOT1 target is determined in a manner analogous to the procedure in 2 above. Note, however, that it is necessary to compute MOT3, based on expected premaneuver orbital parameters, prior to determining MOT1. Thus, the expected final orientation parameters and the location of the period change at MOT3 are obtained for use in the timing strategy. Since the third trim might change h_p, a 5×3 numerical search procedure is used to minimize the sum

$$\Delta V = \Delta V_{\text{MOT3}} + \Delta V_{\text{MOT4}}$$

The independent variables for the search are the eccentric anomaly (location) and the three components of the velocity increment vector for MOT3, plus the magnitude of the velocity increment for MOT4. The three dependent variables are the target latitude of *PER* and the apoapsis radius for MOT3 and the target periapsis radius for MOT4. The remaining degrees of freedom are used to minimize ΔV. To reduce the ΔV cost, the optimal combined change in the argument of periapsis and inclination is determined for MOT3 rather than attempting to attain pre-MOI nominal values for these parameters.

After MOT1 and MOT2 are implemented as for the Type *L* strategy, MOT3 is redetermined from pre-MOT3 estimated parameters, fixing the maneuver location and the target value for the inclination. MOT4 corrects the final timing error.

4) Type H *early reconnaissance strategy*

This strategy was designed for missions having a high inclination orbit, a need for early site reconnaissance data, and large post-MOI orientation errors. The maneuver sequence is described in Table 4.

If no orientation change is required, this strategy reduces to the timing strategy discussed under strategy 2 above. When an orientation change is required, it is combined with one of the two near periapsis maneuvers as an in-plane rotation. Performing the orientation change early in the maneuver sequence prevents the degradation of final period control which could result from execution errors experienced in correcting a large orientation dispersion. By combining this correction with a large period-change maneuver (MOT1 or MOT2), the velocity cost is further

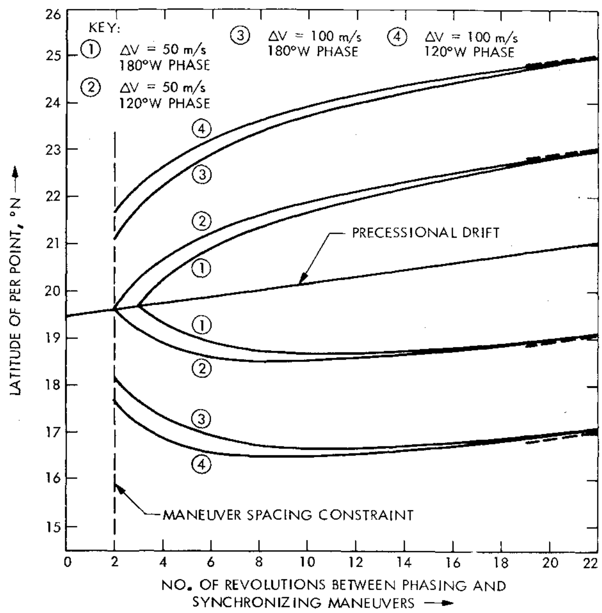

Fig. 5 Retargeting capability for Mission 1.

minimized. Reducing the sequence to three trims also improves overall reliability and permits a more compact timeline than can be handled by the Type *L* and Type *M* strategies.

For high inclination orbits, the orientation correction can be performed efficiently as an in-plane maneuver. By limiting the strategy to two dimensions in this manner, the algorithm is simplified and computer running times required for analyses are reduced. In two dimensions, the problem reduces to a condition for the intersection of two confocal ellipses in a plane.[5,7]

5) Secondary site strategies

In the event that reconnaissance information demonstrates that the primary site is unacceptable for landing, it will be necessary to retarget to a secondary site. The Delayed Reconnaissance Strategy can be utilized for such a retargeting (see Table 4). Figure 5 provides a deterministic assessment of the latitudes which can be reached if Mission 1 is retargeted after having acquired the primary site. This in-orbit retargeting capability is given as a function of the wait time between the phasing (MOT2) and synchronizing (MOT3) maneuvers. Both of these maneuvers are assumed to be performed at periapsis. In Fig. 5, the curve labeled ① indicates the edge of a latitude "band" which can be reached while phasing to a secondary landing site 180° W of the primary site with a fixed ΔV budget of 50 m/sec. The curve labeled ② indicates the "band" which is accessible while phasing 120° W with the same ΔV budget. Curves ③ and ④ are the same as ① and ②, respectively, except that the ΔV budget is increased to 100 m/sec.

As the wait time between maneuvers increases, each of the curves approaches a line parallel to the one which indicates the latitude change due to precessional effects only. For example, the upper half of curve ① in Fig. 5 approaches a dashed line about two degrees above the line due to precession. This asymptotic behavior reflects the fact that the ΔV cost for phasing decreases as the wait time increases. For a long wait time, nearly all of the ΔV budget can be used for latitude retargeting. For Mission 1, about two degrees of latitude change can be obtained for 50 m/sec since $\partial(\Delta V)/\partial(\text{LAT } PER) = 24$ m/sec/deg. Therefore, curve ① for this case approaches the dashed line indicated in the figure.

6) Alternate mission strategy

An alternate mission is one that results from an anomalous situation, e.g., a hardware failure. An anomaly might make it necessary to acquire the landing site very rapidly. The Alternate

Mission Strategy, which is being considered mainly for retargeting to a secondary site, combines the beginning of the phasing sequence with the latitude correction maneuver (see Tables 3 and 4). In this strategy, the numerical search minimizes the performance index

$$PI = \sum_{i=1}^{3} c_i \Delta V_{\mathrm{MOT}i} + c_4 |\Delta T_{\mathrm{final}}|$$

where $\Delta V_{\mathrm{MOT}i}$ denotes the magnitude of the velocity increment for the ith trim, $i = 1, 2, 3$; $\Delta T_{\mathrm{final}}$ denotes the final timing error; and c_i denotes a weighting constant, $i = 1, 2, 3, 4$.

IV. Summary of Numerical Results and Conclusions

The orbit trim computer program provides a Monte Carlo error analysis for the selected trim strategy. This program has the capability to target to an input downrange by crossrange ($DR \times XR$) tolerance zone (see Fig. 1). If the $DR \times XR$ error for a particular dispersed orbit can be made acceptable by a longitude change only, the latitude change (in MOT3 for Mission 1 and MOT1 for Mission 2) is omitted. Such tolerance zones are considered below.

The input parameters for the orbit trim program were chosen to be compatible with Ref. 2 and current hardware and orbit determination capabilities. The Type M Early Reconnaissance Strategy was used in the analysis of Mission 1 and the Delayed Reconnaissance Strategy was used for Mission 2. The performance of the selected strategy relative to each of the four trim maneuver objectives is evaluated next.

In the preliminary analyses of Missions 1 and 2, it was determined that targeting to a 1.5° × 2.5° $DR \times XR$ tolerance zone would obtain an acceptable combination of final timing and downrange and crossrange errors (see Table 2). When the orientation-change trim was targeted for this zone, the 99% dispersions for the five control parameters of the first objective were determined. The values are stated in Table 2, which shows that the separation orbit control requirements of objective 1 are met.

The requirement for site reconnaissance is satisfied by 99% of the Monte Carlo samples by the fourth periapsis passage for Mission 1 and by the seventh one for Mission 2. A significant number of samples provide reconnaissance on earlier orbits, especially for Mission 1 because of the synchronous targeting at MOI. This situation is considered to satisfy objective 2.

Objective 3 is satisfied by the maneuver timelines selected for Missions 1 and 2 (see Figs. 3 and 4).

The total of the velocity increments required for corrective navigation is shown in Fig. 6 as a function of the size of the targeted $DR \times XR$ tolerance zone for dimensions $k° \times 2k°$. This figure shows that the corrective navigation ΔV budget (125 m/sec for Mission 1 and 150 m/sec for Mission 2) is adequate for targeting to a 0° × 0° tolerance zone for the two particular missions considered in this paper. Hence, the budget requirement of objective 4 is met.

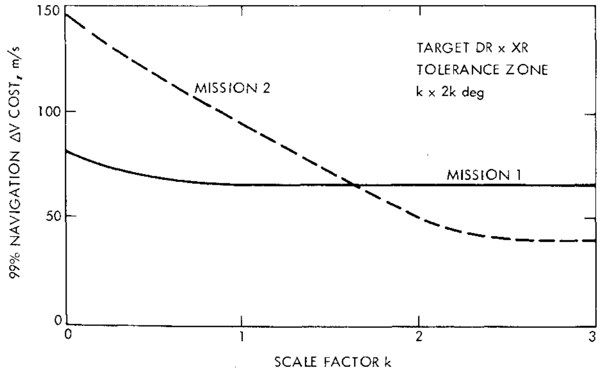

Fig. 6 Navigation cost vs targeted $DR \times XR$ tolerance zone size.

Table 5 99% MOT velocity increment requirements[a]

Trim No.	ΔV, m/sec Mission 1	ΔV, m/sec Mission 2
1	34	65
2	19	54
3	0	56
4	22	2

[a] These requirements are for a 1.5° × 2.5° target $DR \times XR$ tolerance zone.

Even though Mission 2 was targeted to a supersynchronous orbit to conserve ΔV, large navigation velocity costs were experienced for small $DR \times XR$ tolerance zones. This situation reflects the fact that the pre-MOI delivery errors map more out-of-plane for this mission than for Mission 1. The aimpoint at Mars for Mission 2 causes the pre-MOI delivery ellipse to line up in a more out-of-plane direction, resulting in about six times larger post-MOI inclination errors than those for Mission 1.

The 99% high ΔV values required for each of the four trim maneuvers when targeting to a 1.5° × 2.5° tolerance zone are given in Table 5. Each trim uses less than the 145 m/sec limit of objective 4. Note that the phasing trims MOT2 and MOT3 for Mission 2 require much more ΔV than the corresponding trims MOT1 and MOT2 for Mission 1. This large difference reflects the MOI targeting strategy which requires that the Mission 2 trims remove a (deterministic) 6.4 hr period bias.

The following conclusions may be drawn from the data presented here and are supported by the detailed results of the simulation.

1) The candidate Missions 1 and 2 can be flown satisfying the requirements of Ref. 2 with current hardware and orbit determination capabilities.

2) For Mission 1, very little ΔV is saved by targeting to a nonzero $DR \times XR$ tolerance zone. However, it is desirable to minimize the number of maneuvers for reasons of overall reliability. Hence, it is important to note that for a 1.5° × 2.5° zone, this strategy reduces essentially to only three maneuvers. Reduction of the size of this tolerance zone is also undesirable for the reason that this reduction tends to degrade the separation orbit control capability for orbital period. Note that this ΔV-insensitivity to the size of the tolerance zone does not hold in general (see Ref. 3).

3) For Mission 2, the ΔV savings for targeting to a nonzero zone can be very significant. However, these significant ΔV savings are not realized in general. For example, a high inclination mission to a landing site near the equator would not experience such large ΔV savings. (See Fig. 5 of Ref. 8 which treats an 80° inclination mission to a site at 30° S latitude. It can also be seen from this reference that the linear behavior exhibited in Fig. 6 of this paper for $k < 2$ does not hold in general.)

V. Postlanding Trim Strategies

In a major portion of the postlanding phase, there is a need to maintain acceptable relay-link telecommunications performance between the orbiter and lander, and to obtain acceptable photography of the landing site. Analyses[1,3,9] have demonstrated that at most one or two station-keeping trims will be required for each spacecraft during the period between landing and the end of mission. The magnitude of these trims will be as small as $\frac{1}{2}$ m/sec if performed at periapsis or as large as 4 m/sec if performed at apoapsis.

The activities after the deployment of the lander also include orbiter excursions away from the vicinity of touchdown. For such an activity, a period-change maneuver will be performed to initiate a "longitudinal walk" for global water vapor mapping via the Mars Atmospheric Water Detector (MAWD). While periapsis is the ΔV-optimal location of such a "desync"

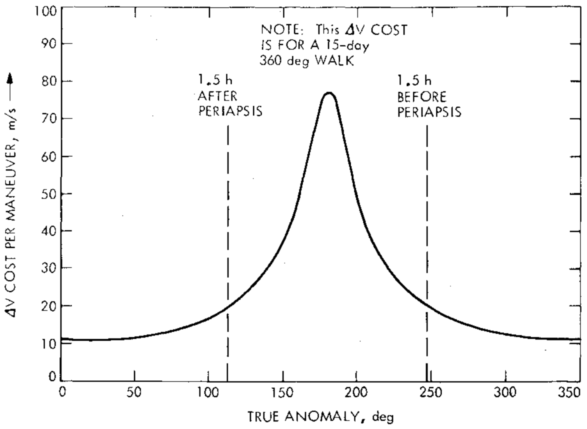

Fig. 7 Desync or resync ΔV cost for supersynchronous longitudinal walk.

maneuver, it is desirable to avoid interruption of the relay link which spans periapsis. Performing the maneuver 1.5 hr (113° in true anomaly) away from periapsis passage would be satisfactory. To complete a global walk in 15 days, the spacecraft's orbital period must be changed 1.64 hr with the longitudinal shift being 24° per revolution. Figure 7 shows the velocity cost for a maneuver made along the spacecraft's velocity vector at any given true anomaly to move to a supersynchronous period of 24.6+ 1.64 hr. The resulting changes in h_p and ω are acceptable.[4] Now the first spacecraft will overfly the dark side of the planet after periapsis passage while the second will do this before periapsis during this phase of the flight. Therefore, a 20 m/sec desync maneuver will be performed at a true anomaly of 113° for the first spacecraft and at 247° for the second, followed 15 days later by another of the same magnitude to resync over the landing site. (To move to a subsynchronous orbit of 24.6 − 1.64 hr, the cost is about 3 m/sec more.) In order to assure the proper time of arrival back at the site, a small trim of less than 1 m/sec will be performed within a few revolutions after desyncing to compensate for execution and orbit determination errors. This small trim can also be performed away from periapsis. When it is not necessary to preserve the relay link, these maneuvers can be performed at periapsis to save about 9 m/sec of additional ΔV for each of the two desync-resync maneuvers.

A large (approx 400 m/sec) postlanding maneuver is being considered for the second spacecraft to move the orbiter to a high inclination for polar observations. A high inclination orbit could be obtained by selecting the proper aimpoint for the interplanetary maneuvers but a large amount of orbiter ΔV would then be required at MOI to obtain the proper sun elevation angle. Once the spacecraft is in orbit about Mars, considerable amounts of propellant are also saved by performing this burn after lander deployment because of the weight reduction due to the detachment of the lander and lander adapter. It is also of interest that the relay link constraints can be satisfied by properly positioning this postlanding maneuver.

VI. Conclusion

The delivery of the Viking landers for soft-landing on Mars, the maintenance of subsequent orbiter-lander communications, and other orbiter functions, offer an exciting challenge which can be met by the Viking navigation team.

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