180°, 225°, 270°, 315°, and 360°. As the secular torque characteristics repeat every 180°, only the torque values for $0 < \theta_s < 180°$ are shown. For a spacecraft orbit of 200 naut miles ($K = 3.90 \times 10^{-6}$ rad/sec²) and the example spacecraft ($I = 2.66 \times 10^4$ slug-ft²), the peak secular torque is 2.60×10^{-2} ft-lb.

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Maneuver Design and Implementation for the Mariner 9 Mission

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The maneuver strategy and operational techniques employed in controlling the Mariner 9 flight path from Earth launch, through interplanetary space, Mars orbit insertion, and the subsequent orbital trim maneuvers are presented. It is shown how the maneuver strategy was tailored to meet the mission requirements with maximum reliability in the presence of launch vehicle injection, orbit determination, and spacecraft maneuver execution errors as great as 3σ.

I. Introduction

THE first section of this paper presents a description of the mission design as it relates to the development of specific navigation requirements. Orbit determination and maneuver execution accuracy statistics are used in conjunction with the total velocity increment constraint to determine specific maneuver requirements.

The second section describes the design of each maneuver, the selection of target parameters, the minimization of the effects of execution errors, and the actual inflight results. The impact of each maneuver on subsequent maneuvers is discussed.

II. Maneuver Strategy

Mission Design

One of the primary objectives of the Mariner 9 mission to Mars was to perform a detailed mapping of the Martian surface with contiguous television pictures having a resolution of 1 km or better over the latitude band from −60° to +40°. Due to the 0.6 hr difference between the rotational periods of Earth and Mars, it is possible to synchronize the period of revolution of a Mars orbiter with an Earth tracking station and also have a migration of the orbiter's surface track which is very favorable for mapping—about 9° of Mars longitude per day.

The Mariner 9 mission design capitalized on this fortuitous circumstance by selecting an orbital period of about 12 hr so that every other periapsis passage would occur near the middle of the Mars view period from the tracking station at Goldstone, Calif. The view periods were just long enough to play back a full spacecraft tape recorder load of 32 pictures to the 64-m Goldstone antenna, record another 32 picture swath near periapsis, and then immediately play back the new load before the end of the tracking period. The spacecraft tape recorder was then empty and ready to record another 32 picture load at the next periapsis, which would not occur during a Goldstone view period.

The spacecraft propellant load required to achieve the desired 12 hr Mars orbit stressed the Atlas/Centaur launch vehicle's payload capability for the mission to the point that only 60 m/sec spacecraft velocity capability was available for midcourse and orbital trim maneuvers. Therefore, minimization of the total velocity requirement was a prime consideration in determining the maneuver strategy.

Navigation Requirements

For reasons of over all reliability it was a goal to minimize the number of maneuvers. Furthermore, to avoid disrupting the mapping activity during the prime orbital mission, which by definition was the first 90 days in orbit, any necessary orbital trim maneuvers were to be completed during the 8 days immediately following orbit insertion. In order to achieve satisfactory synchronization with the Goldstone view period it was required that the time of periapsis passage following the trim maneuver(s) occur within the 1 hr "window" immediately following Goldstone zenith. Because the duration of the Goldstone view period would increase gradually throughout

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the 90-day orbital mission, a slight drift in the periapsis passage time would be tolerable. To keep the drift within the acceptable tolerance, an orbital period control accuracy of 0.3 min was required. This was by far the most stringent flight path control requirement.

The periapsis altitude and inclination were specified so as to provide the desired mapping coverage. Since the inclination of the approach trajectory could be easily controlled to be within the required tolerance with midcourse maneuvers, a coplanar orbit insertion strategy was selected in order to minimize the required insertion velocity increment. The apsidal rotation angle (the angle measured at the center of Mars from the spacecraft hyperbolic approach direction to the orbital periapsis direction) was to be $140° \pm 5°$ primarily to provide for high altitude global TV coverage of Mars. Specification of the apsidal rotation angle (ψ) and orbital inclination (i) together with the knowledge of the orientation of the approach a symptote ($\hat{S}_I$) and the fact that the orbit insertion maneuver was to be a coplanar transfer uniquely determined the two remaining classical orbital elements, namely, longitude of node (Ω) and argument of periapsis (ω). Mariner 9 was to achieve the following orbit to the accuracy specified: orbital period—11.98 hr ± 0.3 min, periapsis altitude—1200 km—1500 km, inclination—65° $\pm 5°$, apsidal rotation—140° $\pm 5°$, time of periapsis passage—Goldstone zenith $+ 1.0$ hr, -0 hr, longitude of node—that resulting from planar orbit insertion.

Navigation Terminology

For analysis and targeting purposes the hyperbolic encounter trajectory is defined in terms of the osculating conic at unbraked (i.e., ballistic) closest approach as illustrated in Fig. 1. The asymptote of the incoming branch of the hyperbola is referred to as the incoming asymptote ($\hat{S}_I$). The vector from the center of the planet (i.e., the focus) defining the perpendicular distance to the incoming asymptote is known as the B-vector. Midcourse maneuvers do not significantly change the direction of the incoming asymptote, consequently, the B-vector remains in the plane which contains the planet's center and is perpendicular to the incoming asymptote. This plane is known as the B-plane. The intersection of the B-plane and a reference plane such as the planet's equatorial plane defines a reference direction called the T-axis. As illustrated in Fig. 1, the B-vector can then be defined by its orientation (θ) relative to the T-axis and its magnitude (B). Variations in the angle θ correspond to rotating the trajectory plane about an axis ($\hat{S}_I'$) which is parallel to the incoming asymptote and passes through the center of the planet. In this case the inclination, argument of periapsis, and longitude of node change while there is no change in the inplane trajectory profile (e.g., altitude vs time). When only the magnitude B is varied the converse is true except for argument of periapsis, which changes in either case. Since the Mariner 9 orbit insertion was to be coplanar, the midcourse maneuver(s) would be used to control θ such that the hyperbolic encounter trajectory inclination would be within the acceptable range for the postinsertion Mars orbit. B was to be controlled so that it would be possible to perform the coplanar transfer from the hyperbolic encounter trajectory to the desired elliptical Mars orbit within the spacecraft insertion velocity budget.

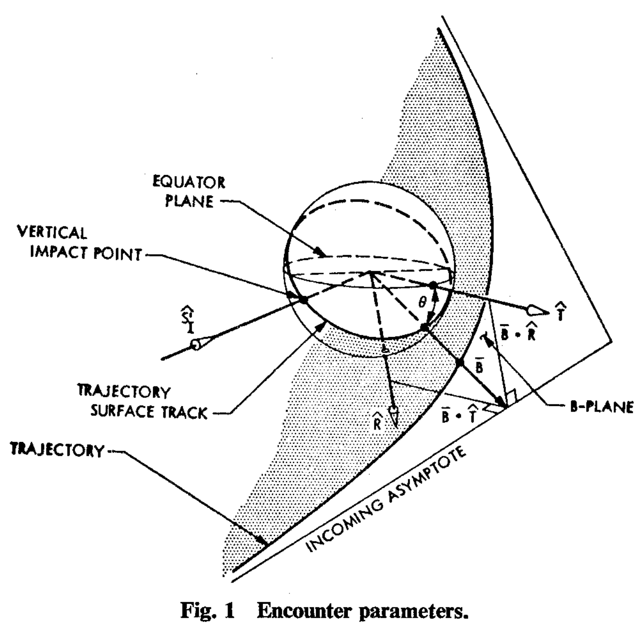

Fig. 1 Encounter parameters.

The terms "control" and "knowledge" used herein require definition. Control refers to the ability of the navigation system to achieve a desired flight path, and is determined by errors in estimating the trajectory at the time the maneuver is calculated, and also by execution errors associated with the maneuver. The final delivery errors thus are seen to arise from executing an imperfect maneuver (execution errors) on a trajectory other than the one for which the error-free maneuver was calculated (orbit determination errors). Knowledge is the accuracy with which the orbit determination process can estimate parameters.

Strategy Development

The ability of a fixed-magnitude orbit insertion maneuver to reduce the period of the resulting ellipse varies inversely with the altitude at which it is performed. Therefore knowledge errors, which result in calculating the maneuver parameters for the wrong altitude, map directly into errors in the achieved period. For Mariner 9 the sensitivity was about 0.2 min/km. For example, suppose B is estimated to be 8300 km, but the actual value is 8200 km. The insertion velocity increment would be calculated based on 8300 km. When this increment was executed by the spacecraft on the lower 8200 km trajectory, the resulting orbital period would be 20 min less than desired. The knowledge of B magnitude at the time of the final orbit insertion calculations was the largest of all orbit insertion error sources for Mariner 9.

For reasons of reliability, project policy stipulated that the orbit insertion maneuver commands were to be loaded in the spacecraft computer at least 24 hr before execution. Consequently, the expected orbit insertion accuracy was dominated by the orbit determination uncertainty at 24 hr before encounter—prior to the time where improvement in the orbit determination accuracy could be gained by observing the influence of the Mars gravitational attraction on the spacecraft.[1] The predicted uncertainty was 300 km (3σ) in B magnitude, which maps to over 1 hr in orbital period. Hence, achieving the specified period control required orbital trims. The other insertion error sources were of considerably less importance.

The control of the post insertion period was independent of normal control errors in the approach trajectory (i.e., delivery errors) since these known errors could be compensated for when selecting the insertion velocity increment magnitude and direction. This was not in general the case for periapsis altitude and apsidal rotation. The orbit insertion strategy was therefore to target for the desired period using a velocity increment of 1610 m/sec or less and allowing the other elements (altitude and orientation) to vary as appropriate. These variations are shown in Fig. 2 as a function of the achieved B. It is seen that for B values outside of the range from about 8150 to 8250 km, the apsidal rotation will vary from the nominal target value of 140°, and for B greater than about 8320 km, the postinsertion target altitude will increase. Within the range where $\psi = 140°$, velocity savings of up to 15 m/sec may be realized. The optimum B for midcourse

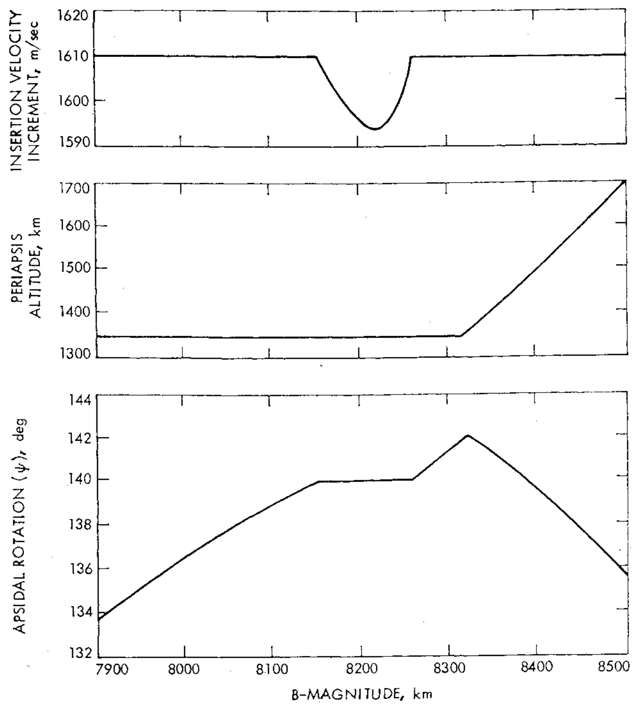

Fig. 2 Mars orbit insertion target criteria.

targeting was judged to be 8200 km because it would 1) provide an insertion velocity savings of about 15 m/sec over the maximum of 1610 m/sec, and 2) maximize the probability of achieving the nominal target value of apsidal rotation—a very important consideration.

Planetary quarantine requirements precluded targeting to the desired B vector at launch. In fact, planetary quarantine required a bias in the B plane of about 31,000 km. Further biasing was required to meet the spacecraft propulsion system constraint that the first midcourse maneuver be at least 5.6 m/sec. Consequently, the spacecraft execution errors for the first midcourse alone were predicted to be greater than 1000 km (3σ) in B magnitude.

Due to the accuracy requirements on control of B just discussed, it was very likely that a second midcourse maneuver would be required. By executing the second midcourse near Mars (30 days or less before encounter) it is possible to control the B vector to an accuracy essentially equal to the orbit determination accuracy. The expected (3σ) delivery accuracy based on performing a second midcourse was about 325 km in the direction of $\bar{B}$ and about 400 km normal to $\bar{B}$, the latter being equivalent to a 3° error in the inclination of the approach trajectory. Errors in arrival time were of no concern. The probability was 0.99 that 15 m/sec velocity capability would be sufficient for the interplanetary maneuvers.

Table 1 compares the orbit control accuracy requirements with the expected errors in the post insertion orbit parameters based on the interplanetary and insertion strategy previously described. It is seen that inclination and aspsidal rotation would be within the specified tolerances following the insertion maneuver, therefore, correction of these parameters was not

Table 1 Expected [3σ] errors after insertion maneuver

Parameter	3σ Error	Final required accuracy
Orbital period	± 75 min	± 0.3 min
Periapsis altitude	± 250 km	± 150 km
Inclination	$\pm 4°$	$\pm 5°$
Apsidal rotation	$\pm 5°$	$\pm 5°$

required of the trim strategy. Because the postinsertion inclination would be within tolerance, and the corruption of inclination and longitude of node due to orbit trim maneuvers would be negligible, the orbit trim strategy development was reduced to an inplane problem. Preflight analysis also demonstrated that the only significant orbit determination errors relating to the trim strategy were those in estimating orbital period and time of periapsis passage.

The trim strategy was designed to synchronize the orbit with the Goldstone view period with a single period trim maneuver in 64% of all cases, while never requiring more than two period trims to synchronize. A third trim was to be performed in the unlikely event that periapsis altitude was out of tolerance after insertion. In order to minimize the trim velocity requirements, synchronizing maneuvers were to be performed near periapsis and the altitude trim maneuver near apoapsis. Furthermore, all trim maneuvers were to be executed essentially colinear with the local velocity vector, which not only minimizes the velocity increment required, but also virtually eliminates the effect of thrust pointing errors on the orbit parameters.

To satisfy the Goldstone synchronization requirement, it was necessary to perform the final period trim when the time of periapsis passage was within 1 hr window immediately following Goldstone zenith. Project policy required that trims be performed between the 4th and 16th revolutions following insertion. The diagram shown in Fig. 3 proved to be very useful in designing the trim strategy to meet these criteria. It was used to graph the time delay between Goldstone zenith and each even numbered periapsis passage. A negative delay indicates that periapsis passage occurs before Goldstone zenith. For a given period, a sequence of points may be plotted to the scale of Fig. 3 indicating this timing delay for each periapsis passage. These points will lie on a straight line for a constant period, and will have zero slope for a synchronous period. Periods greater than synchronous will give points on a line with negative slope, and less than synchronous periods have a positive slope. Straight lines are used to determine these discrete points in Fig. 3. The intercept of the pretrim line is the delay for the zeroth periapsis, which is defined by the conditions at the end of the orbit insertion maneuver. The slope is determined by the postinsertion orbital period. The period trim maneuvers produce an instantaneous change in slope corresponding to the amount of period change produced. Zero slope indicates the orbit is synchronized with Goldstone. To satisfy the criteria stated earlier, the final period trim maneuver must be performed at an even numbered periapsis for which the time delay is between 0 and +1 hr. This maneuver must produce the synchronous period. If the postinsertion period is such

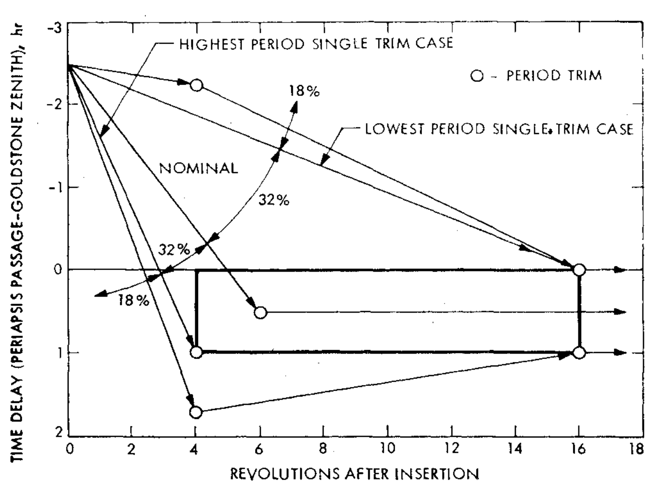

Fig. 3 Orbit trim strategy.

that a time delay between 0 and +1 hr will not occur at any even numbered periapsis between the 4th and 16th, a trim would be performed at the 4th periapsis to adjust the orbital period such that the required delay would be achieved at the 16th periapsis and a second trim would then be performed to achieve synchronization. By separating these two trims to the maximum extent (i.e., periapsides 4 and 16), the period change required of each is minimized and, thereby, the velocity requirement is minimized. Figure 3 illustrates some representative trim situations.

If the arrival time was adjusted to cause the zeroth periapsis passage to occur in exactly the middle of the acceptable window (i.e., time delay of 0.5 hr.), the probability of being able to synchronize between revolutions 4 and 16 with a single trim was only 24% based on the insertion accuracy presented in Table 1. In most cases (76%) the postinsertion period error would cause the passage time to migrate outside of the acceptable window prior to the 4th revolution thereby requiring the one trim on the 4th revolution to reverse this migration and cause periapsis passage to occur inside the window on the 16th revolution. The second trim would then be performed accomplishing the synchronization. For 50% of the cases the postinsertion period would be dispersed below the synchronous value which would be doubly wasteful of propellant since the insertion maneuver has been too large and the first trim maneuver is then required to restore energy to the orbit with an amount of propellant equal to or greater than that already wasted at insertion. However, by optimally adjusting the arrival time to cause the zeroth periapsis passage to occur 2.5 hr before Goldstone zenith and targeting to an orbital period of 12.5 hr at orbit insertion, the probability of synchronizing with a single trim was maximized at 64% and in virtually all cases the trim maneuver(s) would be required to remove energy (i.e., decrease the orbital period) which means that no propellant would be wasted. As illustrated in Fig. 3 this optimization resulted in the nominal trim situation being a single trim at the sixth periapsis in the center of the acceptable window. Note that 32% of the cases (orbital periods dispersed above nominal) will still permit a single trim at periapsis 4 or 6, whereas, 32% (periods below nominal) will permit the single trim at a periapsis between 6 and 16. In cases where an altitude trim maneuver was required, an exception to the project policy would have been allowed to permit this final trim as late as 20 revolutions after insertion. Because of the high correlation of postinsertion period error and periapsis altitude error, any case requiring an altitude trim would virtually always require two synchronizing trims. In this event the strategy was to bias the second trim maneuver such that the period change resulting from the altitude trim would precisely correct the period to the synchronous value. The advantage in performing the altitude trim maneuver last was that since it would be performed at apoapsis the effect of the maneuver execution errors on orbital period would be minimized. Because of the biasing at the second trim, this final maneuver could actually be targeted to precisely correct orbital period since the resulting periapsis altitude would easily be within tolerance. All of the Mariner 9 orbital trim maneuvers would be sufficiently small that the fixed magnitude error would be the dominant execution error source. Consequently, performing the last period adjustment at apoapsis where the sensitivity of period to velocity is least would give the best accuracy. The 99% velocity capability requirement for the trim maneuvers was 50 m/sec.

III. Maneuver Design and Implentation

Mariner 9, launched on May 30, 1971, was targeted to a close approach at Mars on Nov. 14, 1971. The Atlas-Centaur guidance system was sufficiently accurate to make the spacecraft targeting at launch an important part of the overall guidance strategy. Three constraints governed targeting: 1) satisfaction of the planetary quarantine constraint, 2) the propulsion subsystem requirement that the first maneuver magnitude exceed 5.6 m/sec, and 3) insuring a spacecraft attitude during the first maneuver that would permit communications throughout the maneuver. The first two constraints required biasing the aim point at launch away from the final desired aim point; the third provided a criterion for determining this bias is an optimum manner. The optimum spacecraft orientation for communications is to direct the antenna axis directly to Earth. Since the motor axis and the low gain antenna axis are parallel, this orientation corresponds to that required to accelerate the spacecraft along the Earth to spacecraft direction. The launch bias which nominally requires such a maneuver is readily determined. An analysis of the launch vehicle injection statistics indicated that 1) a bias at launch such that a perfect injection would require an 8 m/sec midcourse maneuver was necessary to insure that the first maneuver would exceed 5.6 m/sec with 99% probability, and 2) a bias this large would control the first maneuver direction sufficiently that no in-tolerance injection error would cause a violation of the antenna pointing constraint. Verification that an 8 m/sec bias along the Earth to spacecraft direction yielded launch target values outside the region constrained by planetary quarantine completed the launch targeting strategy. The 1σ injection dispersion ellipse and the final desired, the targeted, and the achieved injection aim points are shown in Fig. 4 using the B plane coordinates discussed previously. The miss at injection is seen to be about a 1σ error.

The definition of a coordinate system frequently used in maneuver analysis will facilitate the following discussion of maneuver design. The gradients of $B \cdot R$ and $B \cdot T$ expressed in velocity space at the maneuver epoch define a plane with the property that maneuvers performed in this plane give maximum change in miss at encounter. This plane is referred to as the critical plane, and maneuvers perpendicular to this plane have no effect, to first order, on miss. This perpendicular direction is known as the noncritical direction. The flight time gradient does not in general coincide with the noncritical direction, but does approach it for late maneuvers. Two basis vectors for this coordinate system are defined to lie along the gradient of R and along the noncritical direction. The third completes a right handed set.

It was a virtual certainty that at least one interplanetary maneuver would be required to remove both random errors and the bias associated with the launch vehicle injection. For this mission, the a priori probability of requiring a second interplanetary maneuver to correct the orbit determination and execution errors affecting the accuracy of the first maneuver so that mission objectives would be met was about 80%. In order to minimize the final delivery errors at Mars, it was

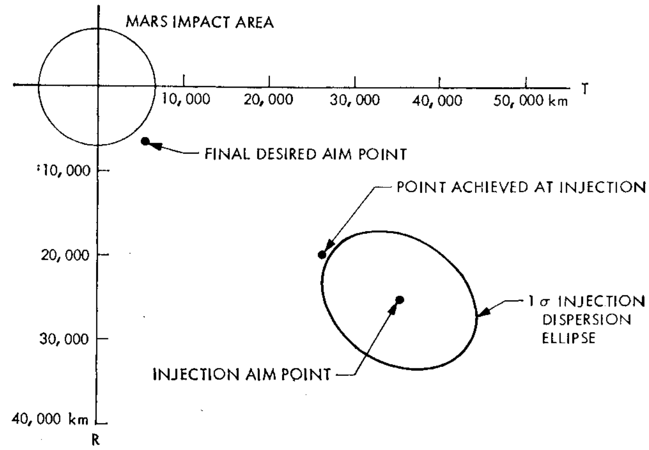

Fig. 4 Mariner 9 injection results and statistics.

necessary to analyze the second maneuver design with respect to spacecraft execution errors when selecting target values for the first maneuver.

Maneuver execution errors may be placed in two categories; errors in pointing and errors in magnitude. Each of these may be further subdivided into fixed errors independent of the maneuver, and errors proportional to the maneuver magnitude. For the maneuvers as small as typical second interplanetary corrections on this mission (<1 m/sec), the fixed errors are dominant and may be quite significant. For example, for a second Mariner 9 maneuver performed 3 to 4 weeks before encounter, the 3σ fixed magnitude error for a maneuver in the critical plane corresponds to a B plane error of about 200 km. The same error in a maneuver along the noncritical direction yields an error of about 1 min in arrival time. The primary consideration determining the arrival time is the requirement to maximize the likelihood of needing only one trim maneuver. However, variations in arrival time of up to ± 30 min would not appreciably change the one-trim probability, provided the orbit insertion target period was adjusted appropriately. Because the fixed pointing errors for the Mariner 9 spacecraft are essentially zero, the optimum interplanetary maneuver strategy was to bias the first maneuver target arrival time in such a manner that the second maneuver would be mostly along the noncritical direction. This has the effect of minimizing the projection of the fixed error onto the critical plane, thereby minimizing its effect on $B \cdot R$ and $B \cdot T$. A constraint on this biasing is that it be kept small enough that a second maneuver would not be required if the first maneuver was sufficiently accurate in correcting the miss. The geometry involved is indicated in Fig. 5.

The projection of the fixed magnitude error on the critical plane can be made arbitrarily small for a given miss correction by increasing the noncritical component of the velocity increment. However, as the magnitude becomes larger, so do the effects of proportional errors. The proportional pointing errors which project onto the critical plane are of particular interest. The optimum maneuver design corresponds to that flight time correction which minimizes the execution errors in the critical plane. Figure 6 shows the contributions of the fixed magnitude error and one component of the two-dimensional pointing error in the critical plane as a function of velocity increment along the noncritical direction for a fixed miss correction of 0.75 m/sec. These error sources were combined in an rss manner, yielding the minimum error at just over 2 m/sec along the noncritical direction. The 0.75 m/sec miss correction corresponded to a 99% high value, and the minimum error point did not vary appreciably for smaller values. The 2 m/sec minimum point corresponded to an arrival time change of about 25 min. Since a second maneuver to decrease the flight time would give excellent telecommunications, the first maneuver was targeted to arrive 25 min later

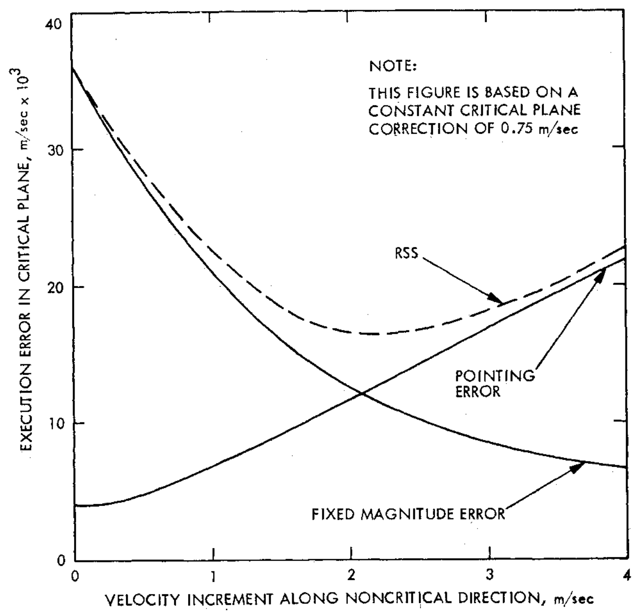

Fig. 6 Execution error analysis for second midcourse design.

than nominal, well within the constraint not to jeopardize the one trim probability if the second maneuver were not required. Figure 7 indicates all the factors that determined the final first maneuver target arrival time of Nov. 14, 00:29 GMT. The desire to have the first elliptic periapsis occur 2.5 hr before Goldstone zenith has been discussed previously. An additional 10 min bias is required since the hyperbolic periapsis time, which this maneuver is targeted to, would occur 10 min before the first elliptic periapsis. The 25 min bias just discussed for the second maneuver completes the determination of the target arrival time. As discussed earlier, the optimum B-plane target parameters were a B magnitude of 8200 km and a θ corresponding to an inclination of 65° to the Mars equator.

The Mariner 9 spacecraft implements maneuvers by counting a specified number of pulses for each maneuver parameter. Each pulse corresponds to 0.03 m/sec in velocity increment (0.96 m/sec for insertion) and 0.18° in turns, yielding maximum resolution errors of 0.015 m/sec in velocity increment (0.48 m/sec for insertion) and 0.09° in turns. The effect of the velocity resolution error is minimized by adding (or subtracting) a small velocity increment along the noncritical direction to yield a total velocity magnitude corresponding to

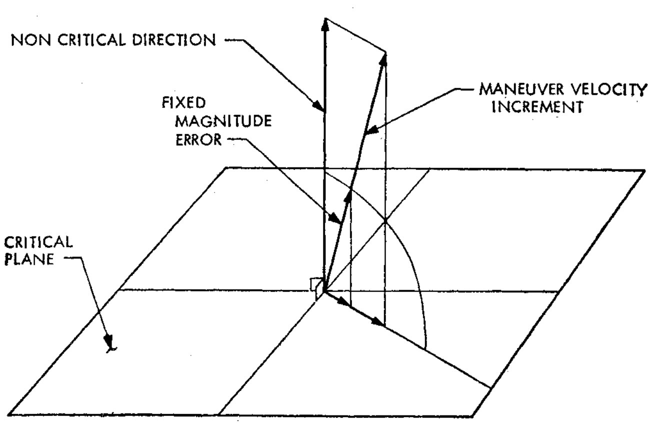

Fig. 5 Critical plane geometry for second midcourse maneuver design.

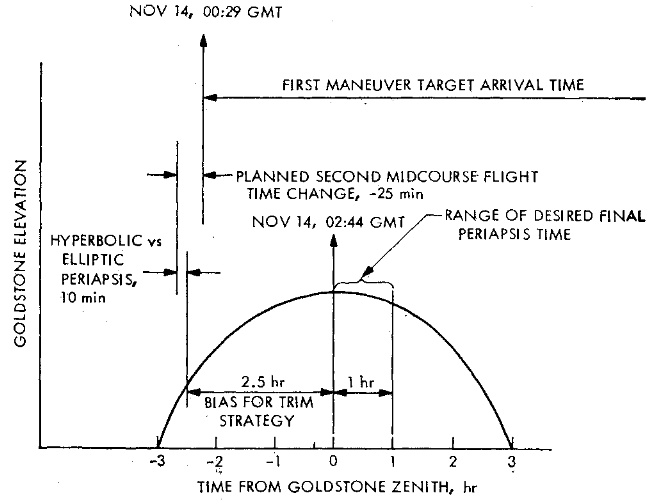

Fig. 7 Selection of first maneuver target arrival time.

Table 2 First interplanetary maneuver delivery results

	Target	Achieved	Error	A priori 1 sigma[a]	Error Sigma
B-magnitude	8200 km	8261 km	61 km	400 km	0.2
Inclination	65°	64.23°	−0.77°	3°	0.3
Arrival time	11/14 00:29:00	11/14 00:31:09	$2^m 9^s$	7.5^m	0.3

[a] Based on one-maneuver statistics

Table 4 Near insertion trajectory estimates and resulting errors

	Maneuver calculation	Update decision point	Final
B	8209	8235	8261 km
INC	63.87	64.02	64.23°
TCA	00:31:07	00:31:03	00:31:09 GMT
		Resulting errors	
Period		$7^m 24^s$	$14^m 52^s$
Altitude		21	40 km
Apsidal rotation		−0.1	−0.1°

an integer number of pulses. This is appropriate since miss errors are more critical than arrival time errors. The error due to quantization of the turns is generally less than that due to the velocity increment resolution, and further, it is small relative to the expected execution errors. For these reasons, it was not compensated for beyond ensuring that the quantization was done in the best manner. Rounding to the next higher pulse in both roll and yaw corresponded not only to the minimum miss, but also to a miss almost entirely in the inclination direction, a desirable situation since control of B magnitude was more critical than control of inclination. Table 2 shows the first maneuver delivery results, and a comparison to the a priori capability predictions. The achieved results shown are from an orbit determination solution made soon after implementation of the maneuver, and differ from later solutions due primarily to the presence of nongravitational forces and different Mars ephemeris estimates. The use of later trajectory estimates for determining the maneuver performance is not appropriate since it is then impossible to separate the effects of execution errors from nongravitational forces and ephemeris errors.

About one month before encounter, the first maneuver was judged sufficiently accurate that a second interplanetary maneuver would not be required. The trajectory estimate at this time indicated a B of 8235 km and an inclination of 64.3° The data of Table 3 was generated based on this estimate. The difference in TBIAS (time delay of the zeroth periapsis) of 0.4 hr is the 25 min allowed for the second maneuver to correct. The differences in the target period at insertion with and without the maneuver corresponds to the difference in the optimum slope of the pre trim line of Fig. 3. Two reasons for preferring not to execute a second maneuver are the risk associated with doing a maneuver and the advantage in having an uninterrupted long arc of tracking data for the encounter orbit determination process. Since there was no appreciable advantage to a second maneuver (as Table 3 shows), it was eliminated.

Based on the stated mission requirements and the previously discussed trim strategy, the values to target for at insertion

Table 3 Second interplanetary maneuver tradeoffs

Parameter	Without maneuver	With maneuver
T BIAS $(t_{p_0} - t_{gz})^a$, hr	−2.1	−2.5
MOI target period, hr	12.43	12.50
ΔV total 99% high, m/sec	1661.0	1661.0
Rotation angle, deg	140.0	140.0
Inclination, deg	64.3	65.0
	Trim probabilities, %	
One trim	59	64
One P6 trim	15	15
Two trims	29	30
Three trims	12	6

[a] Time of zeroth periapsis minus time of Goldstone zenith.

were well established; namely, an apsidal rotation of 140°, periapsis altitude of 1350 km, and a period such that the sixth periapsis would occur one-half hour after Goldstone zenith. The final insertion maneuver parameters to accomplish this were computed based on a trajectory estimate obtained about four days prior to encounter, which indicated a B of 8209 km and an inclination of 63.9°. These maneuver parameters were loaded in the spacecraft and verified. Table 4 indicates the trajectory estimates at the time of the maneuver calculation, at the time the last decision was made not to update the maneuver commands, and the final estimate using data up to the point of motor ignition. Also shown are the errors that would result from application of the computed maneuver to both the decision trajectory and the final trajectory. The decision not to update the maneuver was based on the information of Fig. 8, which indicated a near nominal trim sequence, although the mean trim time would now be on the fourth periapsis rather than the sixth. The actual insertion results and their comparison with predicted capabilities are shown in Table 5.

The actual insertion maneuver was sufficiently accurate that only one trim maneuver would be required, and its sole function was to control period. Normally the most efficient manner to perform such a trim maneuver is to add (or subtract) velocity at periapsis parallel to the spacecraft velocity. However, by loading commands computed for a nominal maneuver parallel to the local velocity prior to periapsis, the period correction could be either increased or decreased by simply adjusting the ignition time via a direct ground command. Such a strategy would also cause changes in other orbit parameters, but these changes were shown to be small, and completely negligible relative to the project specified tolerances on these parameters over the range of velocity magnitudes and ignition times of interest. A nominal maneuver time of 20 min before periapsis was chosen because

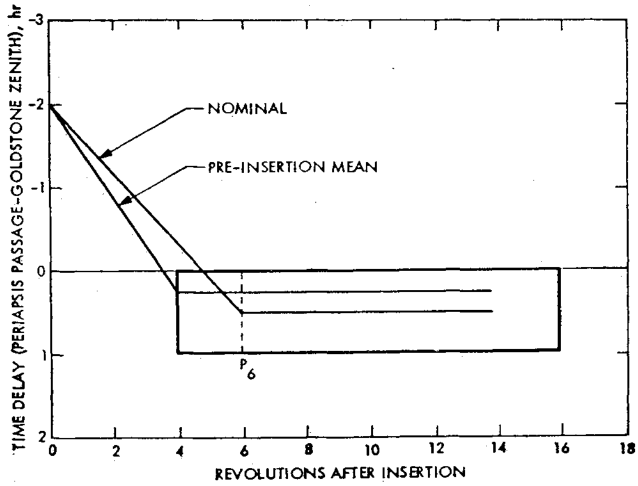

Fig. 8 Predicted trim situation before insertion.

Table 5 Insertion results

	Target	Achieved	Error	A priori 1 sigma	Error Sigma
Period	12:25	12:34	9 min	17 min	0.53
H_p, km	1350	1398	48	69	0.69
Apsidal rotation, deg	140.0	139.7	−0.3	0.6	0.5
Inclination, deg	64.1[a]	64.36	0.26	0.8	0.33

[a]Target inclination is the result of applying the calculated maneuver to the final orbit estimate with no execution errors.

it gave sufficient flexibility in period control by ignition-time change, and because the spacecraft orientation required to align the thrust vector with the spacecraft velocity at that time pointed the spacecraft antenna in a favorable direction for communications during the burn.

At the time the trim maneuver parameters were to be loaded in the spacecraft computer, slightly less than two revolutions of tracking data were available. Later, the final pretrim orbit estimate indicated that the motor ignition should occur exactly 1 min earlier than planned and the time of transmission of the ignition command was adjusted accordingly. The achieved orbit parameters following the trim maneuver are shown in Table 6, along with the estimated errors and a priori capability estimates.

Initially it appeared that no additional trims would be required. However, as the mission progressed, it became apparent that at least one more trim was required for two reasons. The nominal mission duration of 90 days was to be extended because of the presence of the Mars dust storm which lasted for the first several weeks of orbital operations. This in itself would not have required a second maneuver. However, as more tracking data was processed, it was learned that the unexpected nature of the Mars gravity field was causing the period of the orbit to be sinusoidal with a mean value 25 sec below the originally desired value.[2] This caused the periapsis passage time to migrate across the Goldstone view period at such a rate as to fall outside the view period before the end of the now extended nominal mission. Consequently, about one and a half months into orbital operations, the decision was made to increase the period to move the passage time batis across the view period at a slow enough rate to insure sackfactory communications for the remainder of the mission. Furthermore, the scientific experimenters indicated a desire to raise the periapsis altitude to 1650 km at this time. The most economical way to correct period and periapsis altitude in terms of propellent required is with two maneuvers colinear with the local velocity, one at apoapsis and one at periapsis. The former will correct the altitude and cause some change in period. The latter will then correct to the desired period. The alternative method, performing a single maneuver at either of the two intersections of the current and desired orbits, requires substantially more spacecraft velocity capability. However, due to near nominal performance there was ample propellant remaining at this point in the mission. Consequently, the decision was made to use the one-maneuver option for reliability reasons. An analysis of the geometry involved showed that a maneuver performed at the second intersection (true anomaly >180°) was close to giving good communications. By adding a small out-of-plane component to the maneuver and rotating the target orbit slightly, it was possible to align the medium gain antenna axis along the spacecraft-to-Earth direction for excellent communications. The strategy employed for this maneuver is best explained by noting that the in-plane perpendicular and colinear components of the maneuver velocity relative to the local spacecraft velocity map independently to altitude and period changes, respectively. Further, since the desired changes were about 250 km in altitude and 79 sec in period, the maneuver would be essentially all in the perpendicular direction. Accordingly, the maneuver was designed so that the inertial spacecraft orientation would both align the medium gain antenna with Earth and orient the spacecraft thrust vector perpendicular to the local velocity at the intersection of the pre and post-trim orbits. Now small variations in the

Table 6 First trim results

	Target	Achieved	Error	A priori 1 sigma	Error Sigma
Period	11:58:48[a]	11:58:49	1 sec	3.5 sec	0.29
H_p, km	1387	1387	~0	0.3	~0
Rotation, deg	140.3	140.3	~0	0.03	~0
Inclination, deg	64.37	64.37	~0	0.03	~0

[a] Instantaneous period at the time of the first trim.

Table 7 Second trim results

	Target	Achieved	Error	A priori 1 sigma	Error Sigma
Period	11:58:58[a]	11:58:52	−6 sec	13.2 sec	0.45
H_p, km	1650	1650	~0	0.3	~0
ω, deg	−26.09	−26.02	0.07	0.1	0.7
Inclination, deg	64.43	64.40	−0.03	0.1	0.3

[a] This target value was the instantaneous period that corresponded to a mean period 79 sec larger than the mean of the period achieved by the first trim.

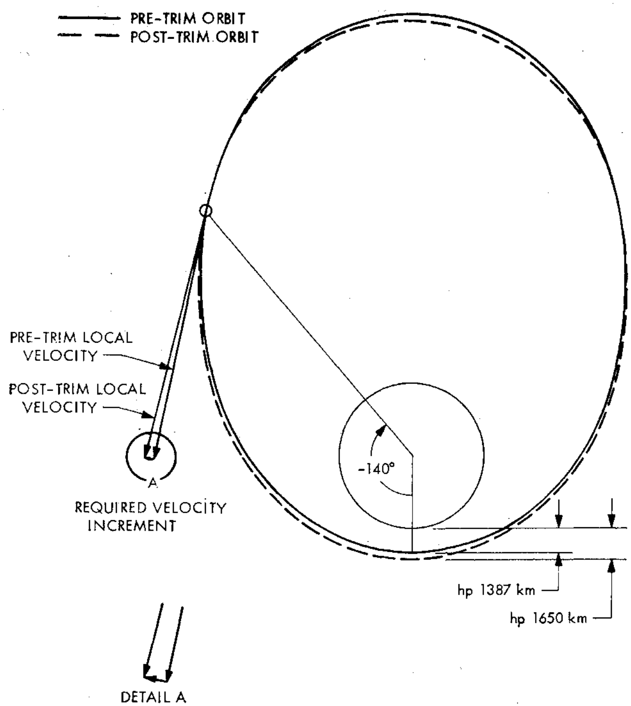

Fig. 9 Mariner 9 orbit trim 2 geometry.

ignition time, allowing the local velocity to rotate, would give the required projection of the correction velocity onto the local spacecraft velocity for the desired period correction. This strategy proved to give excellent results in achieving the desired orbit parameters, as shown in Table 7. The geometry just discussed is depicted in Fig. 9.

IV. Conclusion

The preflight analysis of the total navigation system which was used to deliver Mariner 9 into the prescribed orbit about Mars established that the project navigation requirements could be met within the constraints imposed. This paper has described the maneuver strategy and maneuver design aspects of the Mariner 9 navigation process. It has been shown that the inflight results compare well with the preflight predictions. The strategies and techniques described here are not unique in their application to the Mariner 9 mission, but rather provide a basis for developing more advanced capabilities to meet the needs of future missions.

References

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