Maneuver Sequence Design for the Post-Jupiter Leg of the Pioneer Saturn Mission

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A sequence of midcourse maneuvers has been designed for the Pioneer 11 post-Jupiter trajectory to provide Saturn targeting options to aimpoints either inside or outside the visible ring system. An interim aimpoint was selected which retains these options until late 1977 and improves the navigation accuracy to the finally selected target. An arrival time of September 1, 1979, selected to reduce interference due to solar conjunction, required the use of preprogrammed open-loop maneuvers with a brief communication loss to reach the interim aimpoint. Subsequent targeting will be achieved in continuous communication using simpler and more accurate maneuvers near the Earth-line. This paper develops the maneuver sequence, with emphasis on overall efficiency, flexibility, and navigation system accuracy.

I. Introduction

SUBSEQUENT to launch in April 1973, Pioneer 11 was targeted[1] to an interim Jupiter aimpoint to await the encounter results of the sister spacecraft, Pioneer 10. A year after launch, following the successful Pioneer 10 encounter, Pioneer 11 was retargeted[2] for a close retrograde flyby of Jupiter. The successful flyby in December 1974 subsequently placed Pioneer 11 on a post-Jupiter flight path on which it will encounter Saturn in late 1979.

The uncorrected ballistic trajectory resulting from the Jupiter flyby would cause Pioneer 11 to pass 2 million km behind Saturn, arriving Sept. 3, 1979 (Fig. 1). A sequence of midcourse maneuvers, implemented over a 3-yr period, is planned to achieve the selected targeting objectives. This paper defines the options within the targeting objectives and presents the design of the maneuver sequence. The first part of this sequence, already implemented with maneuvers in December 1975 and May 1976, also will be described.

II. Targeting Objectives

The Saturn targeting objectives have been selected from a wide choice of candidate flyby strategies.[3] Particular emphasis was placed on defining those objectives that might be achieved uniquely with the Pioneer 11 spacecraft and would be complementary to the objectives of two future Mariner Jupiter/Saturn flights past Saturn in 1981. Because Pioneer is a precursor to Mariner, specific encounter strategies were developed, considering planet and ring viewing opportunities, occultation characteristics, satellite encounters, and flight-path hazards near the ring system.

The targeting options have been reduced to a pair of aimpoints, each providing a close posigrade flyby either inside or outside the visible ring system. The inside aimpoint, designated $\bar{\mu}_A$, provides good science return,[3,4] including ring occultation and favorable conditions for celestial mechanics investigations. The unknown hazards of flying between the planet surface and the visible rings have resulted in the selection of a more conservative alternate target near, but outside, the visible ring system. This target, designated $\bar{\mu}_B$, does not provide a ring occultation, and the quality of the celestial mechanics investigation, especially for defining the ring mass, is reduced. The relationships of these aimpoints to Saturn and the ring system are compared to Fig. 2. The final target selection will result from an evaluation of the relative science value of the two aimpoints, including the influence of estimated flight-path hazards. Studies (e.g., Ref. 5) of the ring system may lead to better estimates of the flight-path hazards.

An arrival time during Sept. 1, 1979, was selected for both target options to provide a balance between 1) maximum separation from the subsequent superior conjunction for reduced solar interference, and 2) minimum total mission fuel requirements, and to provide a Titan encounter after Saturn closest approach. Separation of this arrival date from solar conjunction is apparent in Fig. 3, indicating a Sun-Earth-Saturn angle of ~8°. Achievement of a larger separation with an earlier arrival date would have required significantly more propellant. Placement of closest approach inside the tracking overlap (Fig. 4) of the Deep Space Stations at Madrid, Spain (DSS 63) and Goldstone, Calif. (DSS 14) provides essential redundancy. Important approach science will be obtained prior to exposure of the spacecraft to the uncertain hazards of ring plane crossings by placing closest approach at 18:00 GMT within this tracking overlap. Safe passage through the ring plane will bring Pioneer 11 within ~500,000 km of Titan one day later, during the next overlap of the tracking station view periods. This additional encounter objective could improve the estimate of Titan's mass and ephemeris, measure its temperature, and obtain modest images of surface features.

III. Maneuver Capabilities and Constraints

The targeting objectives constrain the time of Saturn arrival, and the anticipation of a better understanding of Saturn's ring hazards requires that the aimpoint options remain open as long as practical. The goal of the maneuver sequence design is to achieve these objectives accurately in the presence of spacecraft and operational constraints including available propellant, communication limitations, and orbit redetermination characteristics.

Following Jupiter encounter, the spacecraft had propellant sufficient to provide velocity changes totaling ~88 m/sec. The Pioneer project desired a 20-m/sec propellant reserve for contingencies and to assure good control of propulsion performance as the propellant neared depletion. This guideline allowed 68 m/sec to be used for targeting.

The spacecraft has four thrusters, which are used for axial velocity change and spin axis precession. There are fore- and aft-pointing thrusters mounted at each of the two mounting locations on opposite sides of the antenna dish.[1] By selecting

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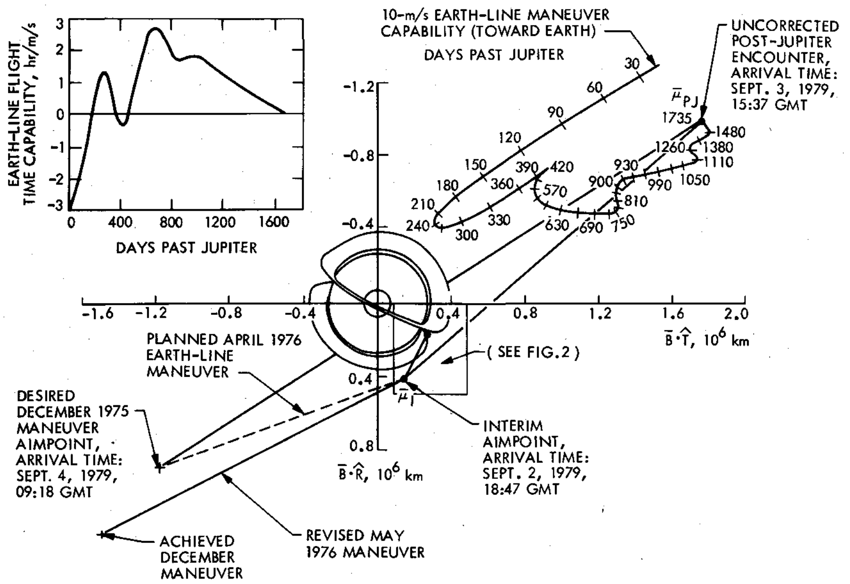


Fig. 1 Saturn-relative *B*-plane target data and Earth-line maneuver capability.

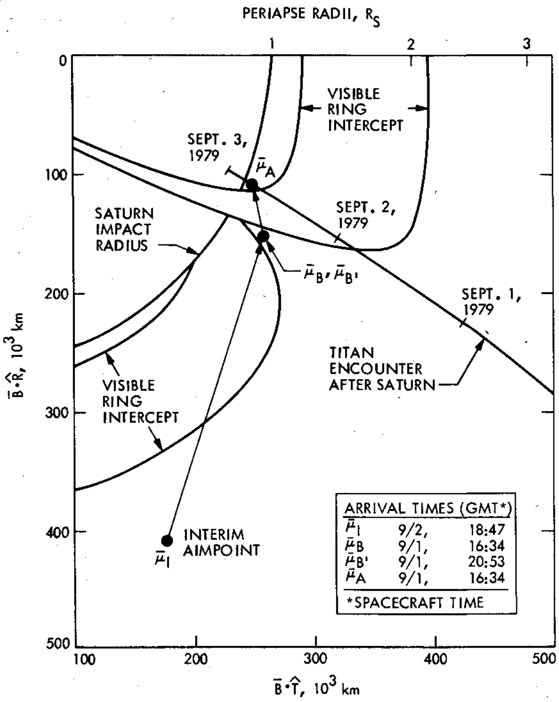


Fig. 2 Target geometry relative to Saturn.

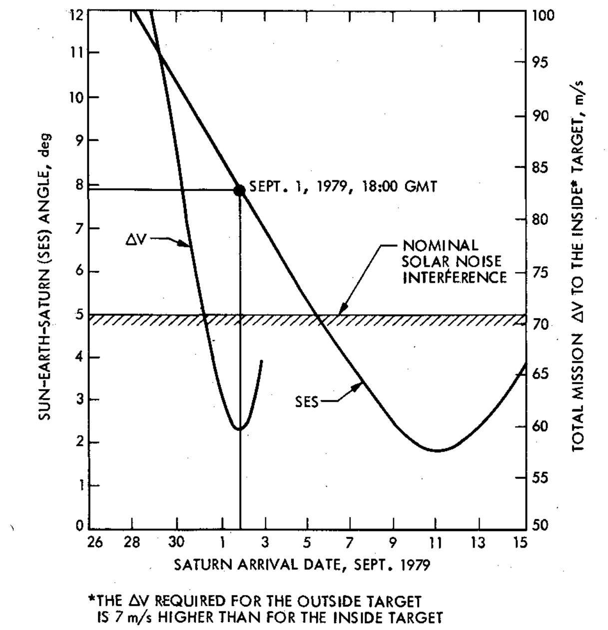


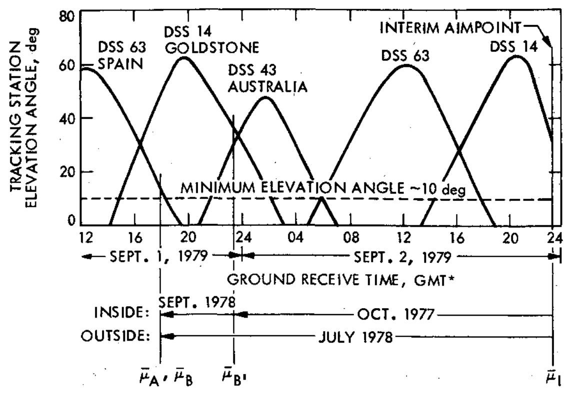
Fig. 3 Effect of Saturn arrival date on Sun-Earth-Saturn angle and total mission ΔV requirements.

the appropriate thruster pair, the desired velocity change or precession can be executed. For velocity changes, the thrusters operate in either a continuous mode for a programmed duration or, when small corrective trims are desired, a pulsed mode. A third pair of thrusters mounted normal to the spin axis is used for spin-rate adjustment.

The standard midcourse maneuver is achieved by precession of the spin axis to the required pointing direction, followed by the firing of a thruster pair which adds a velocity increment in the desired direction along the spin axis. This standard mode was used during near-Earth maneuvers in April 1973[1] to provide efficient targeting, while continous communication was maintained using the medium-gain antenna.

Beyond 60 days after launch, the spacecraft requires the axially aligned high-gain antenna to be pointing nearly toward Earth to sustain communication. Maneuver pointing directions outside the 2° cone of this antenna require nearly all of the maneuver sequence to be executed without communication, using a stored command sequence. This "blind" maneuver strategy entails objectionable risks and requires a significant period of orbit redetermination before execution errors can be resolved. Within ~25° of the Earth-line, sufficient Earth-based signal strength could be received by the spacecraft medium-gain antenna to allow an onboard homing system called Conscan to restore Earth alignment, should a spacecraft failure occur using the onboard sequencer. This capability tends to limit maneuvers using the preprogrammed sequencer to pointing directions within ~25° of the Earth-line.

To achieve maximum execution accuracy and maintain high reliability, strategies have been developed using maneuvers near the Earth-line. In this attitude, maneuvers are implemented either toward or away from Earth by proper thruster pair selection. The velocity magnitude may be trimmed precisely by monitoring the Doppler shift in the spacecraft signal and using small pulses to increment the velocity change until the desired value is reached. To maintain



*SPACECRAFT TIME IS 01^h26^m EARLIER (ONE-WAY LIGHT TIME)

Fig. 4 DSS tracking station coverage at Saturn encounter.

adequate signal strength and take full advantage of the Earth-line maneuver capabilities, it is necessary to constrain maneuver pointing directions inside a 2° cone about the Earth-line.

When the Earth look angle (ELA) is constrained to zero, the B-plane maneuver capability can be defined as a function of time, as shown by the contour in the upper right of Fig. 1. The vector from the origin to each point on this contour represents the direction and magnitude of the target plane change achieved by a 10-m/sec velocity directed toward Earth. The opposite change is achieved when the velocity is directed away from Earth. The corresponding flight time capability is shown in the upper left of Fig. 1. Since the relationships between the capabilities to change $\bar{B}\cdot\hat{R}, \bar{B}\cdot\hat{T}$ and flight time are fixed at each epoch, it is necessary to find an epoch that has the appropriate combination or to perform a series of Earth-line maneuvers at different epochs to achieve the desired target change. If such combinations are not available, continuous communication still can be maintained using a two-component vector mode strategy.

The two-component capability provides three-coordinate target control using vectorial combinations of velocity changes along and normal to the Earth-line. The axial thrusters are used to implement the Earth-line component, and the normal velocity vector is accumulated by a series of timed pulses using the spin control thrusters. However, repeated spin axis realignment is necessary to maintain communication, as torques resulting from the center of gravity being offset from the spin-thrust plane act to precess the spin axis away from Earth alignment. This control technique requires ~28% more fuel for the normal component because of pulse mode inefficiency, cosine losses resulting from motion during the pulse, and the cost of restoring the pointing direction. An additional inefficiency results from combining the ΔV in separate components rather than a single ΔV in the desired direction. Still, this maneuver class would allow the entire maneuver sequence to be implemented while remaining pointed near Earth.

The maneuver epochs are constrained by the ability to redetermine the spacecraft orbit following each maneuver. This time depends on the tracking geometry, the quality of the tracking data, and the ability to model the previous maneuver in the redetermination process. Nominally, this process will require 3 to 4 months,[6] and in these analyses a minimum of 120 days is assumed between maneuvers.

The change in the inertial positions of the Earth and Sun relative to the spacecraft affects attitude reference but has minor influences on the maneuver strategy. As the Earth-probe-Sun (EPS) angle decreases below ~10° during periodic solar conjunctions, poor geometry degrades the Sun sensor reference accuracy for attitude control. After mid-1977, the spacecraft-Earth distance will exceed ~6 AU, resulting in an EPS angle that will remain below 10° for the balance of the mission. Prior to this time, it will be possible to use either the Sun or the star sensor for attitude reference; after this time, the star sensor will provide the most reliable method.

IV. Maneuver Design Concept

The most desirable design sequence would exclusively employ the accurate and reliable capabilities of Earth-line maneuver strategies. When targeting requirements preclude this approach, the two-component vector mode or standard maneuver strategies are available to supplement the design. The increased risk and higher execution errors of these alternate strategies require that they be minimum in number and implemented at the beginning of the design sequence. This approach restores the maneuver sequence to Earth-line strategies, increases the time available to remove execution errors, and significantly improves the overall delivery accuracy. Highest delivery accuracy is achieved by a maneuver sequence that uses progressively smaller maneuvers, each of which demonstrates a capability to correct the execution errors of the previous maneuvers.

An interim aimpoint was placed in the maneuver sequence which allows the target options to be retained until late 1977 and provides access to either the inside or outside target with relatively small Earth-line maneuvers. This design concept also satisfies the modest planetary quarantine constraints[7] and separates the maneuver sequence into two distinct parts. The first part includes those maneuvers required to reach the interim aimpoint. The second part of the sequence contains those maneuvers necessary to accurately retarget from the interim aimpoint to the selected target. The exclusive use of maneuvers near the Earth-line in the second interval limits the execution errors to values significantly less than the predicted orbit determination uncertainties.[6] The result is a final delivery accuracy that is dependent primarily on the orbit determination process and Saturn ephemeris errors and much less dependent on terminal maneuver accuracy.

A total flight time reduction of almost 2 days is required to arrive at the selected target on Sept. 1. This large change from the uncorrected post-Jupiter trajectory is most efficiently made early in the maneuver sequence. However, the Earth-line maneuvers at the end of the sequence provide part of this change, and it is necessary to bias the arrival time at the interim aimpoint to account for these planned changes.

The maneuver sequence design will begin with the selection of the interim aimpoint and a description of how the targeting options will be exercised from this aimpoint. The design and implementation of maneuvers to the interim aimpoint follow in Secs. VI and VII. Finally, the results of a two-maneuver sequence to the interim aimpoint are described.

V. Interim Aimpoint Design

Objectives

The primary reasons for selecting an interim aimpoint are to retain the targeting options as long as practical and to provide the highest achievable navigation accuracy. To accomplish this, the coordinates and arrival time of the interim aimpoint were designed to permit retargeting to the selected target using Earth-line maneuvers. In addition, these maneuvers are designed for epochs that are far enough apart to provide good orbit determination between maneuvers.

The inside and outside target options have the same arrival time requirements (Fig. 2) but different B-plane coordinates. The inside target, with uncertain flight path hazards near the visible rings, requires the greater navigation accuracy. To achieve this increased accuracy, a two-maneuver sequence from the interim aimpoint was designed for the inside option. The second part of this sequence requires a relatively small maneuver to minimize execution errors and to assure a delivery accuracy that is limited only by the orbit determination process. The less demanding navigation requirements to the outside target can be satisfied with a single maneuver near the Earth-line.

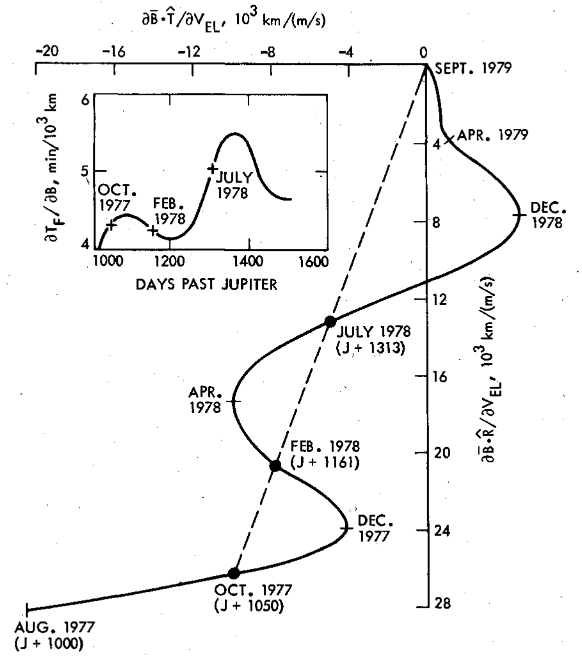


Fig. 5 Earth-line maneuver capability for retargeting.

Maneuver Capability

The interim aimpoint design was aided by the Earth-line maneuver capability shown in Fig. 5. This figure presents an expanded view from Fig. 1, showing the maneuver capability between mid-1977 and Saturn encounter in September 1979. The broad "snake-like" pattern of this capability, reflecting the spacecraft view of the yearly cycle of the Earth about the Sun, adds considerable flexibility to the maneuver design. An important characteristic of this capability is illustrated by the dotted line that crosses the capability contour three times, indicating that the same target change can be implemented at each crossover epoch, but with proportionally larger velocity requirements at the later epochs and with different arrival times. The insert in Fig. 5 shows the corresponding flight time maneuver capability, expressed in terms of B-plane change. Here, the relative efficiency of different epochs for making flight time changes is compared for a fixed B-plane change. For example, a B-plane change of 100,000 km made on either day 1050 or day 1313 will achieve the same B-plane coordinates, but with an arrival time difference of over 1 hr. In this case, day 1313 has the superior flight time capability but has a velocity requirement that is larger by a factor of 2.

Aimpoint Design

The accuracy requirements of the inside target option dictate a two-maneuver sequence from the interim aimpoint. It was decided that the aimpoint for the first of these maneuvers has the same B-plane coordinates as the outside target $\bar{\mu}_B$ but a different arrival time. This intermediate target has been designated $\bar{\mu}_{B'}$ (Fig. 2). Targeting from the B-plane coordinates of $\bar{\mu}_{B'}$ to the inside target can be accomplished with an Earth-line maneuver in September 1978 (day 1388). This maneuver requires 4.2 m/sec and results in an arrival time 04:19 earlier than $\bar{\mu}_{B'}$. Therefore, it is necessary to achieve an arrival time at $\bar{\mu}_{B'}$ on Sept. 1, 1979, at 20:53 GMT to finally achieve the inside target at the desired arrival time. An interim aimpoint that provides access to both $\bar{\mu}_B$ and $\bar{\mu}_{B'}$ at the respective arrival times is required.

The dynamic changes in the Earth-line maneuver capability (Fig. 5) were helpful in defining these coordinates. Day 1050 (October 1977) was selected as the nominal epoch to maneuver from the interim aimpoint to $\bar{\mu}_{B'}$. This selection centers the slope of the required B-plane change within the "snake" to provide a powerful error-correcting capability with modest epoch changes. Combined use of epoch change and pointing directions within 2° of the Earth-line can effectively remove dispersions introduced by prior maneuvers.

The difference in arrival time between $\bar{\mu}_{B'}$ and $\bar{\mu}_B$ also can be varied with Earth-line maneuvers by moving the interim aimpoint along the 1050-day Earth-line slope. To achieve the required difference of 04:19, the interim aimpoint would have to be placed such that $\bar{\mu}_{B'}$ could be reached on day 1050 with an Earth-line maneuver of 13.7 m/sec. An additional 4.2 m/sec is required to complete targeting to the inside target. From the same interim aimpoint, the outside target $\bar{\mu}_B$ also could be reached with an Earth-line maneuver of 27.3 m/sec on day 1313. These fuel costs, when combined with the maneuvers required to reach the interim aimpoint, would require a total mission velocity of 70 and 79.4 m/sec, respectively. To reduce the fuel requirements to an acceptable level, the interim aimpoint was moved along the 1050-day Earth-line slope to a point closer to $\bar{\mu}_{B'}$. These coordinates (Fig. 2) provide targeting capability to $\bar{\mu}_{B'}$ for 10 m/sec on day 1050 and require a total of 14.2 m/sec to reach the inside target. To reach the outside target $\bar{\mu}_B$ from this interim aimpoint, a modest maneuver epoch change to day 1316 is required. At this time, the maneuver ELA must be relaxed to an acceptably small 0.9°, and the velocity requirement is reduced to 21 m/sec. Selection of this interim aimpoint allows either the inside or outside target to be reached for 60.4 or 67.2 m/sec, each less than the allowable 68 m/sec.

Having selected this interim aimpoint, it is intended that the inside or outside target option be selected by day 1050. If this is not possible, the coordinates of $\bar{\mu}_{B'}$ can still be achieved by delaying the maneuver until day 1161 (February 1978). At this time, the Earth-line slope available on day 1050 is repeated (see Fig. 5), allowing $\bar{\mu}_{B'}$ to be reached for 12.4 m/sec. A maneuver ELA of 0.6° is required to control the arrival time also. The nominal maneuver planned for the outside target, scheduled for day 1316, remains unchanged.

Overall, the selected interim aimpoint satisfies each of the established criteria for this target. The aimpoint options are accessible, with some flexibility for adaptive modification. The selection of a maneuver strategy to this interim aimpoint was made only after considering a number of possibilities.

VI. Maneuver Design to the Interim Aimpoint

Earth-Line Maneuvers

The Earth-line maneuver capability characteristics (Fig. 1) indicate that spatial targeting to the interim aimpoint could be achieved with a 28-m/sec maneuver implemented near 690 days past Jupiter (October 1976). Although this strategy is both efficient and accurate, it results in an unacceptably late arrival time during Sept. 6, 1979. Equivalent two-maneuver Earth-line strategies provide an earlier arrival time during Sept. 5 but require nearly all of the available maneuver capability (~88 m/sec). Since an earlier arrival time is essential, other maneuver strategies with superior flight time capability are needed in the design sequence.

Maneuver Strategies Using Spin Control Thrusters

A strategy using spin control thrusters also was examined, as it represented a means of accurately implementing velocity changes both along and normal to the Earth-line in combinations required to reach the interim aimpoint, while also remaining in continuous Earth communication. Minimization of the normal component was essential, as fuel savings, reduced execution errors, and shortened implementation time all would be realized. This minimum is ~14 m/sec when implemented during December 1975 and January 1976. Including additional fuel for the Earth-line component and repeated spin axis realignments, a total of ~35 m/sec would be needed to reach the interim aimpoint with this strategy.

Calibration and feasibility exercises were performed to develop an implementation procedure, and these tests showed

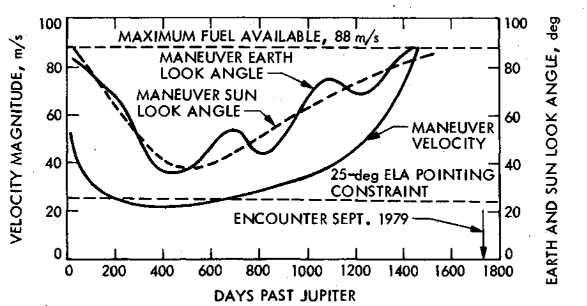


Fig. 6 Single unconstrained maneuver to the interim aimpoint.

that the despin thruster could not be operated. For this reason, the strategy was abandoned.

Single Unconstrained Maneuver

The interim target can be achieved with the standard maneuver mode, using a preprogrammed open-loop sequence. The velocity and pointing requirements for this class of maneuver were evaluated and are shown in Fig. 6. These maneuvers are implemented most efficiently during the period adjacent to perihelion (February 1976), between 200 and 600 days past Jupiter. However, the ELA for these maneuvers is never less than ~37°, and this strategy would always require exceeding the 25° ELA pointing constraint imposed for preprogrammed open-loop maneuvers. A two-epoch maneuver sequence was subsequently designed to accommodate this pointing constraint.

Two-Epoch Maneuver Strategy Constrained by Earth Look Angle

Targeting to the interim aimpoint also can be accomplished by a two-epoch maneuver sequence, restricted to pointing directions within 25° of the Earth-line. The goal of this sequence is to efficiently achieve maximum delivery accuracy to the interim aimpoint in such a way that future maneuvers to the selected aimpoint could be implemented with a 90% likelihood of having a maneuver ELA ≤2°. Achievement of this objective dictates that the second maneuver be capable of nulling the execution errors induced by the first maneuver, while producing minimal execution errors itself. In addition, the epochs must be at least 120 days apart to allow sufficient time for orbit redetermination.

The most desirable and efficient strategy having these combined characteristics requires the initial maneuver to be implemented at the 25° ELA constraint limit, followed at the second epoch by Earth-line maneuver. When the first maneuver is constrained to smaller ELA values, it is necessary to allow the second maneuver ELA to be increased in order to maintain approximately the same overall fuel cost. Such an approach is impractical, as the requirement for a second open-loop maneuver would add unnecessary complexity to the maneuver sequence and may compromise delivery accuracy to the interim aimpoint.

The total two-epoch maneuver velocity was evaluated over a range of the most efficient candidate epochs between late 1975 and mid-1976 and is shown in Fig. 7. Epoch combinations separated by at least 120 days lie on, or above, the locus in Fig. 7. Combinations above the boundary in the upper right are eliminated from consideration, as they require a first maneuver ELA greater than 25°. The maximum maneuver efficiency to the interim aimpoint is available during periods adjacent to perihelion when the first maneuver occurs between 360 and 390 days, and the second maneuver follows between 480 and 510 days.

The sequence selected for implementation was to be initiated with a 25° ELA maneuver on Dec. 18, 1975 (380 days) to the aimpoint shown in Fig. 1. An Earth-line maneuver on April 26, 1976 (510 days) subsequently would complete the targeting to the interim aimpoint. This combination is efficient, requiring 31 m/sec in December and 15 m/sec in April, for a total of 46 m/sec. Good orbit redetermination geometry is available for this combination, as perihelion occurs between the maneuver epochs, and the Earth-relative declination is relatively high.

The accuracy characteristics of this sequence were examined to assess the second maneuver capabilities to null first maneuver execution errors, and to predict the delivery accuracy to the interim aimpoint. Maneuver execution errors for the December sequence were attributable primarily to thruster performance, estimated to have a one-sigma uncertainty of ~1.5%. This performance variation causes pointing errors during spin axis reorientation and velocity magnitude errors during ΔV implementation. These uncertainties dominate the total target dispersions. A random sampling of these dispersions at the April 1976 maneuver epoch indicated that only 77% of the maneuver ELA's required to reach the interim aimpoint were less than 2°. When this epoch is varied by as little as ±15 days, the motion of the spacecraft-Earth direction allows the number of ELA's less than 2° to be increased to well over 90%.

Orbit determination following the December maneuver was expected to reconverge by April 1976 to uncertainties comparable to those existing at the time of the maneuver. If the April Earth-line maneuver is implemented without benefit of trimming, large dispersions at the interim aimpoint may result. Random sampling of the untrimmed dispersions at the retargeting maneuver epoch to $\bar{\mu}_{B'}$ (day 1050) predicted that only 54% of the maneuvers could be implemented with an ELA ≤2°. While epoch change could increase this probability to well over 90%, trimming the April maneuver velocity magnitude would immediately reduce the execution errors to less than the orbit determination uncertainties and restore the nominal maneuver sequence to $\bar{\mu}_{B'}$. Random sampling of the trimmed target dispersions predicted that 98% of the maneuver ELA's would be less than 2°, and 90% would be less than 1.5°.

If the outside target $\bar{\mu}_B$ were selected, the required maneuver from the interim aimpoint in July 1978 (day 1316) also would have a 98% probability of being less than 2° if it followed a trimmed April maneuver.

The accuracy analyses showed that the two-maneuver sequence to the interim aimpoint in December 1975 and April 1976 could be implemented with a satisfactorily high probability that future maneuvers could be executed near the Earth-line. The ΔV totals for the inside (60.4 m/sec) and outside (67.2 m/sec) options each satisfy the nominal guideline that targeting be accomplished for ~68 m/sec.

VII. Maneuver Implementation to the Interim Aimpoint

December Maneuver

The first of two maneuvers required to reach the interim aimpoint was scheduled for implementation on Dec. 18, 1975. This maneuver was to be executed by precessing 25° from Earth, adding a 30.1-m/sec velocity change, and returning to

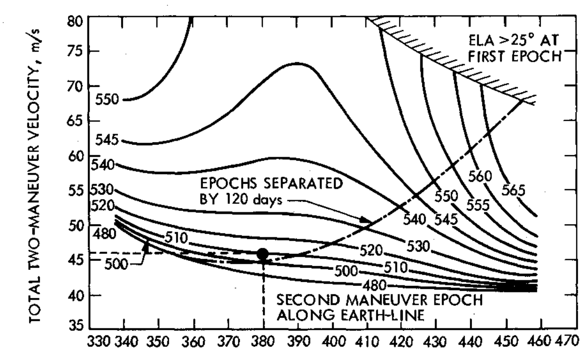


Fig. 7 Two-epoch maneuver sequence to the interim aimpoint.

Earth alignment. To execute this maneuver more accurately, a calibration of the precession and velocity thrusters was performed. The velocity thrusters were calibrated using the observed Doppler shift, whereas calibration of the precession thrusters was based on a playback of the spin axis attitude history, measured by the spacecraft Conscan system in terms of Earth look angle and phase.

The orbit determination following the maneuver indicated that the achieved aimpoint was surprisingly far from the intended aimpoint (Fig. 1). When the maneuver was reconstructed on the basis of orbit determination solutions before and after the maneuver, it was found that the spacecraft had precessed 29° rather than the planned 25°. Further analyses confirmed that Conscan had been providing consistently low readings, but this systematic error was not apparent until the comparatively large Earth look angles of this maneuver were measured. Precession thruster efficiency then was rated 15% higher to match the spin axis pointing history obtained from maneuver reconstruction. An overburn in velocity magnitude of ~3% also was apparent from this analysis.

May Maneuver

As a result of the systematic error in precession thruster calibration from the December maneuver, the plan to use an Earth-line maneuver in April 1976 to reach the interim aimpoint was no longer possible. The best that could be achieved was a maneuver with an ELA of ~5°, available 30 days later on May 26 (day 540). The pointing requirement for this maneuver made it necessary to once again use the open-loop preprogrammed capability of the standard maneuver mode.

The detailed maneuver requirements were evaluated, indicating that a velocity magnitude of 16.6 m/sec at a 4.7° ELA would be required to reach the interim aimpoint. Previous analyses had shown that a trim of the velocity magnitude for the Earth-line maneuver planned in April was necessary to demonstrate a 90% probability that future maneuvers could be implemented with an ELA≤2°. The requirement for an open-loop strategy for the May 26 maneuver increased the need for a trim, as even larger execution errors were anticipated. With Canopus as the selected roll reference, it was determined that expected spin axis pointing errors induced by initial pointing uncertainty, random precession magnitude error, and precession phase error were collectively tolerable, but that control of velocity magnitude was essential. The observed Doppler shift after returning to Earth alignment was determined as a valid measure of the velocity change at maneuver ELA's near 5°. A trim analysis[8] showed that the Earth-line component should not be trimmed to its nominal value but to a value that would minimize the ELA for the maneuver to $\bar{\mu}_{B'}$. This minimum would remain very nearly zero if the maneuver velocity magnitude error were even as great as ±2 m/sec (±12%). The ELA for the maneuver to $\bar{\mu}_B$ also is improved by the trim, and acceptable values are achieved over this range of velocity magnitude error. The trim results in an adjusted interim aimpoint and modest changes in the epoch and magnitude of the maneuver to $\bar{\mu}_{B'}$.

Prior to implementing the maneuver on May 26, the maneuver requirements were re-evaluated, accounting for changes induced by calibration and rating the velocity thrusters ~2% higher than calibration had indicated. This procedure was designed to reduce the likelihood of an overburn and to minimize reverse trim requirements, should an overburn occur. After nominal execution, the observed Doppler shift indicated that the Earth-line component was short by ~0.2 m/sec. Soon afterward, a series of seven 2-sec pulses was commanded to implement an Earth-line trim maneuver of 0.25 m/sec to restore a nominally zero ELA for the maneuver to $\bar{\mu}_{B'}$.

VIII. Conclusions

A maneuver sequence has been designed which provides the required navigation accuracy, while retaining the targeting options. Initial target changes to the interim aimpoint have been implemented successfully with maneuvers in December 1975 and May 1976. Future maneuver requirements remain unchanged, with final target commitment due in late 1977.

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