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A Summary of the Pioneer 10 Maneuver Strategy

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The Pioneer Project[1] placed a number of interesting and precise requirements on the navigation of the Pioneer 10 spacecraft flyby mission to Jupiter during 1972–1973. To satisfy these requirements the Pioneer Navigation Team employed a number of versatile computer programs to evaluate the strategies and maneuver sequences required to execute midcourse corrections. This article summarizes the Pioneer 10 mission objectives and the midcourse strategies used to satisfy these *objectives.*

Introduction

Two Pioneer flyby missions to Jupiter during 1972–1975 may have a significant influence on the future exploration of the solar system. These missions will be the first to venture beyond the orbit of Mars, fly through the asteroid belt, and encounter Jupiter, the largest planet in the solar system. If the spacecraft survive the 7-mo journey through the asteroid belt, intense radiation belts at Jupiter may test the spacecraft performance. A hostile environment at Jupiter could influence future missions, either for scientific observation, or for using its gravity to accelerate spacecraft to regions beyond.

The mission objectives are designed to provide scientists with the opportunity to examine the interplanetary region between Earth and Jupiter, in addition to the near-Jupiter environment. The primary mission objectives include exploratory investigations beyond the orbit of Mars, examination of the asteroid belt, and studies of the environmental and atmospheric characteristics of Jupiter (Reference 1). Secondary objectives are to assess the hazards of long duration in interplanetary space and flight through the asteroid belt. Scientific objectives include penetration of the

[1] The Pioneer Project is managed for NASA by the Ames Research Center. Jet Propulsion Laboratory responsibilities in the project include preflight navigation analyses and real-time navigation support.

Jupiter radiation belts and provision for good instrument viewing conditions of Jupiter prior to closest approach.

A tertiary objective of the first mission is to attempt spacecraft occultation by Io, a satellite of Jupiter that may possess a tenuous atmosphere. The effects on the radio signals from the spacecraft at entry to, and exit from, occultation will help describe the characteristics of a possible ionosphere, or atmosphere. This objective places the most stringent requirements on the spacecraft navigation to assure both a precise trajectory position and time at Jupiter closest approach.

Pioneer Spacecraft

The two identical Pioneer spacecraft are spin-stabilized and will carry the same experiments. The 256-kg spacecraft, pictured in Figure 1, includes a 29-kg payload for 11 scientific instruments, and 27 kgs of propellant for attitude control and midcourse corrections. The overall length is 2.4 m; its widest dimension is that of the 2.7-m diameter high-gain parabolic dish antenna. Two 2.7-m extendable booms isolate radioisotope thermoelectric generator (RTG) nuclear power sources away from the instrument package. A 6.4-m boom mounts a magnetometer remotely from each RTG and the remainder of the spacecraft.

The liquid hydrazine propulsion system is capable of making velocity changes totaling 200 m/s. Two pairs of thrusters on opposite sides of the antenna dish are used for attitude and velocity changes. For velocity changes, these thrusters operate in either a continuous mode or, when small trims are desired, a pulsed mode. A third pair of thrusters is used for spin rate adjustment.

Two-way communication is provided through the high-gain dish antenna (38 dB) and a medium-gain horn antenna (12 dB). The horn antenna is used near the Earth where two midcourse corrections were scheduled. The high-gain antenna, with a narrow 3-deg beam width, is used after the spacecraft is nearly 47 million km (60 days) from Earth.

The four RTG units, fueled with uranium-238 dioxide, replace the solar panels that have been used on other interplanetary spacecraft for power generation. Solar panels for Pioneer would be prohibitively large, since Jupiter is over 5 AU from the Sun. At launch, the RTGs are expected to produce 160 W of electrical power, at least 134 W at Jupiter, and nearly 120 W 5 yr after launch.

A description of the 11 scientific instruments, and the experiments each will perform, is summarized below. These instruments, shown on the spacecraft in Figure 1, are:

(1) Helium vector magnetometer, to measure magnetic field components along three axes.

(2) Plasma analyzer, to map the solar wind.

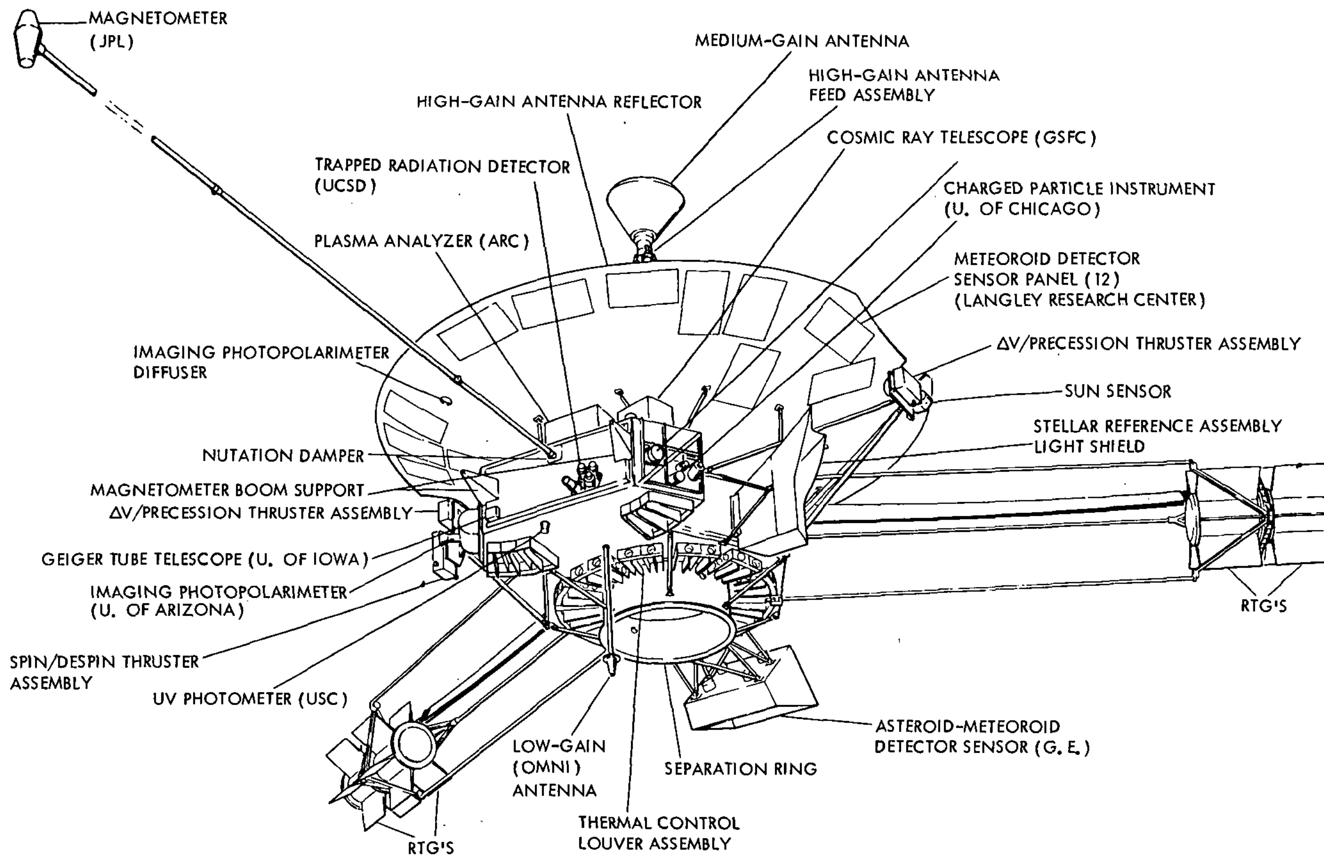


Figure 1. Pioneer spacecraft

(3) Charged-particle detector, to identify 8 of the first 16 elements of the periodic table.

(4) Cosmic ray telescope, to monitor solar and galactic cosmic ray particles.

(5) Geiger tube telescope, to survey electrons and protons in Jupiter's magnetosphere.

(6) Trapped radiation detector, to help correlate trapped particle data with Jovian radio signals.

(7) Asteriod–meteoroid detector, to survey solid material between Earth-orbit and 15 AU.

(8) Meteoroid detector, to detect the distribution in space of particles too small to be seen by light scattering.

(9) Ultraviolet photometer, to determine the ratio of molecular hydrogen to helium in Jupiter's atmosphere and measure the amount of neutral hydrogen in the heliosphere.

(10) Imaging photopolarimeter, to obtain two-color images of light from Jupiter and zodiacal light from interplanetary space.

(11) Infrared radiometer, to measure Jupiter thermal balance, temperature distribution in outer atmosphere, and helium/hydrogen ratio.

Mission Profile

Pioneer F was successfully launched from Cape Kennedy, Florida, at 01:49 GMT, on March 3, 1972, and designated Pioneer 10. The Atlas/Centaur/TE364-4 launch vehicle combination placed the spacecraft in a heliocentric orbit to Jupiter with a direct ascent trajectory. Pioneer 10 is expected to encounter Jupiter with closest approach on December 4, 1973, at 02:26 GMT, after a 640-day, one billion-km journey. An identical Pioneer G spacecraft is scheduled to be launched in April 1973, 8 mo before the Jupiter encounter of Pioneer 10, and is intended to explore a different region about Jupiter.

The nominal Earth-to-Jupiter mission profile, consistent with the primary mission objectives, is shown in Figure 2.

Shortly after closest approach, the spacecraft will pass behind Jupiter, interrupting communications with Earth for about 1 h. Jupiter's gravity will strongly deflect the trajectory and accelerate the spacecraft to a velocity that will permit escape from the solar system. If Pioneer 10 continues to return data until it reaches the present predicted limit of communication with Earth, the spacecraft would be more than halfway between the orbits of Saturn and Uranus. At that point, nearly 8 yr after launch, it will be 2.4 billion km (16 AU) from the Sun. Pioneer will pass the orbit of Uranus (20 AU) with a velocity of 15 km/s relative to the Sun. From this point, it will

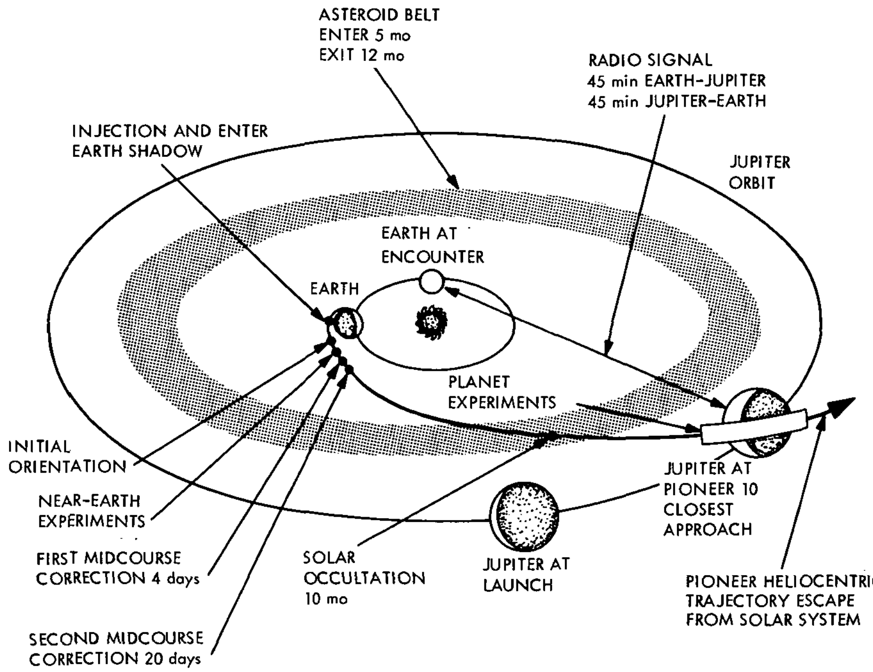


Figure 2. Pioneer 10 mission profile

travel away from the Sun in virtually a straight line, approaching a Sun-relative velocity of 11.5 km/s in the direction of the constellation Taurus. Scientists have placed a pictorial plaque on Pioneer 10 to identify its Earth-origin, should the spacecraft contact another intelligence.

The Pioneer 10 target coordinates at Jupiter were selected to provide arrival conditions best suited for the primary mission objectives, including good instrument viewing conditions prior to closest approach, examination of the radiation belts, and spacecraft occultation by Jupiter. The encounter conditions that satisfy these mission objectives require a closest approach within 3 Jupiter radii of the planet center, and a trajectory plane inclined 14 deg below a parallel to the ecliptic through the planet center. The radius of closest approach consistent with these objectives was to be controlled within ¼ Jupiter radius (Reference 2). There are two secondary navigation requirements. One is to control arrival time within ±½ h of the center of one of the daily 5-h periods when the spacecraft is simultaneously in view of two tracking stations of the Deep Space Network (DSN). The second requirement is for tighter control of closest approach distance (±0.05 Jupiter radii) to control imaging, radiation, and magnetic field survey characteristics near the optimum closest approach of 3 Jupiter radii.

The tertiary objective of spacecraft occultation by the Jupiter satellite, Io requires even tighter control of arrival conditions. The navigation constraints of this objective are discussed later in the article.

Target coordinates at Jupiter were selected to provide an opportunity for achieving each of these objectives. These target coordinates are illustrated in the B-plane hyperbolic encounter coordinate system in Figure 3. For the purposes of evaluating midcourse strategies, the B-plane target coordinates are specified by:

$$\bar{B} \cdot \bar{R} = 209{,}200 \text{ km}$$

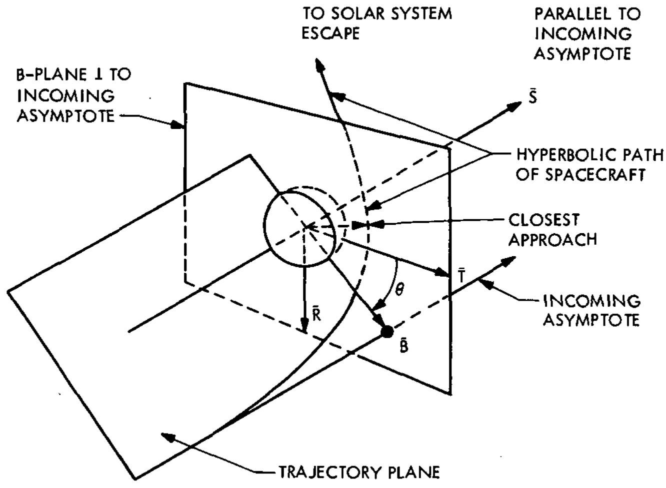
$$\bar{B} \cdot \bar{T} = 838{,}900 \text{ km}$$

Time of closest approach (TCA) = December 4, 1973, 02:26 GMT

Pioneer Navigation Team

The Pioneer Navigation Team provides mission analysis support, performs maneuver analyses, determines orbits, evaluates trajectories, and defines navigation alternatives within the constraints and guidelines specified by the Pioneer Project. The Pioneer 10 mission is the first interplanetary mission to require midcourse corrections of a spin-stabilized spacecraft.

The maneuver function of the Pioneer Navigation Team evaluates the spacecraft navigation requirements and identifies maneuver strategies necessary to achieve the mission objectives. A set of computer programs has been developed to aid in this process. The principal program is the Pioneer Maneuver Operations Programming System (PMOPS). This program considers operational constraints, determines a variety of acceptable



$\bar{B}$ = TARGET PARAMETER
θ = ORIENTATION OF $\bar{B}$
$\bar{S}$ = PARALLEL TO INCOMING ASYMPTOTE
$\bar{T}$ = PARALLEL TO ECLIPTIC PLANE AND $\perp$ TO $\bar{S}$
$\bar{R} = \bar{S} \times \bar{T}$

Figure 3. Pioneer hyperbolic encounter coordinate system

maneuver strategies within those constraints, and, after a maneuver is selected, generates command parameters required to maneuver the spacecraft. PMOPS was developed in accordance with the functional and operational requirements set forth in References 3 and 4. An essential input to PMOPS is an accurate estimate of the spacecraft trajectory, provided by the orbit determination function of the Pioneer Navigation Team. Some of the principal Navigation Team analyses prior to launch are reported in References 5 through 8.

Pioneer Maneuver Strategy

The Pioneer spacecraft standard maneuver is achieved by precession of the spin axis to the required direction, followed by the firing of a thruster pair that adds a velocity increment along the spin axis. The spin axis is initially pointed toward the Earth with a nominal spin rate of 4.8 rpm. When performing a precession, a reference direction to the Sun is established by an on-board sensor as the Sun crosses its field-of-view. The spin axis is then precessed to the desired pointing direction by pulsing coupled precession thrusters at a fixed time increment after the Sun crosses the sensor field-of-view. The spin axis is thus stepped in small increments in a direction fixed relative to that of the Sun. The spacecraft thruster pairs can provide velocity changes in either direction along the spin axis. This capability requires turns of no more than 90 deg from Earth-alignment.

The maneuver parameters that specify the Pioneer precession sequence are the geometric rhumb angle, α, and the precession magnitude, ψ. As used in Pioneer navigation terminology, the geometric rhumb angle is the angle between the Sun's projection onto a plane perpendicular to the spin axis and the desired precession direction. The precession magnitude is the total arc length traversed by the spin axis during the rhumb-line precession. The geometric rhumb angle and the precession magnitude are illustrated in Figure 4.

A Pioneer maneuver is subject to execution errors. These errors are modelled as a combination of pointing and velocity magnitude errors. The pointing error has components that are a function of initial pointing accuracy, quantization, and systematic and random errors. The pointing errors are a strong function of the precession magnitude and comprise the dominant maneuver execution error. The velocity magnitude error has both proportional and fixed characteristics. The proportional error is a function of the velocity magnitude, and the fixed error is a function of the thruster shutoff characteristics. The velocity thrusters operate in either a continuous or pulse mode. The pulse mode operation provides much more accurate shutoff characteristics than the continuous mode. The pulse mode is particularly effective when the spacecraft is in an Earth-pointing attitude and the Earth-line component can be directly observed in the tracking data during velocity addition. This trimming feature of the Pioneer 10 spacecraft provides very accurate control of velocity magnitude.

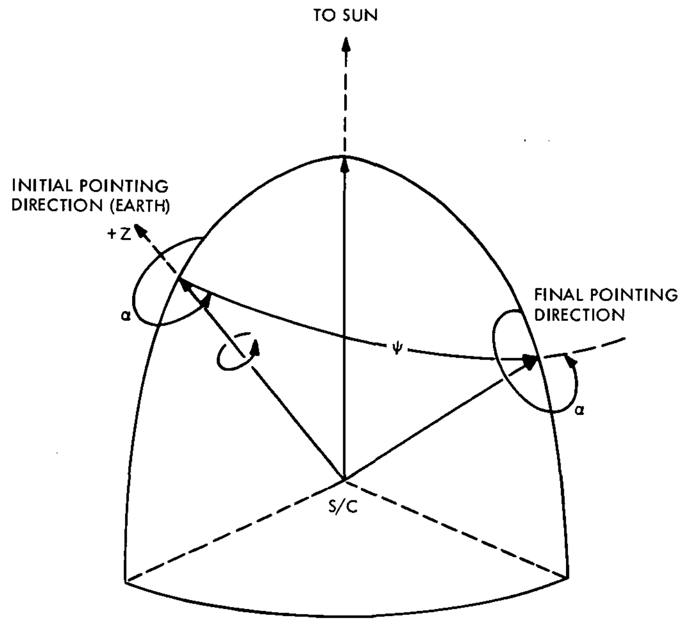


Figure 4. Standard rhumb-line maneuver geometry

The Pioneer maneuver strategy is influenced by a number of spacecraft operational constraints. One is the Sun look angle (SLA) constraint, defined as the angle between the spacecraft-Sun direction and the spacecraft spin axis. This constraint has both an upper and lower limit. The upper limit is a function of spacecraft backside heating and was established as 120 deg for the first and second maneuvers of the Pioneer 10 mission. The lower limit, a function of spacecraft Sun sensor reference accuracy, was established as 10 deg.

Another spacecraft constraint is the maximum Earth look angle (ELA), defined as the angle between the spacecraft–Earth direction and the spacecraft spin axis. This constraint is imposed to maintain adequate signal strength from the medium-gain antenna during maneuvers. For the first maneuver, 4 days after launch, the maximum ELA constraint was established as 45 deg. The second maneuver, 20 days after launch, had a reduced maximum ELA constraint of 24 deg.

The Sun and Earth look angle constraints are illustrated on the surface of a sphere centered at the spacecraft (S/C) in Figure 5. The region A, inside the maximum ELA constraint boundary, but outside the maximum SLA boundary, is an unconstrained pointing direction for the spacecraft spin axis. Also shown in Figure 5 is a region B that is diametrically opposite to region A. When the required velocity vector direction lies in region A, the velocity correction is applied along the positive spin axis. Velocity corrections requiring pointing directions inside region B are achieved using thruster pairs which add velocity along the negative spin axis.

The maneuver strategy must be selected to produce the required encounter conditions at Jupiter without violating these constraints. The selected strategy must also provide a trajectory adjustment within the total multimaneuver capability of 200 m/s allocated for Pioneer 10.

When a single maneuver would violate one or more of the spacecraft constraints, the original required velocity vector is broken into two equivalent unconstrained velocity vectors. Typically, one of the two components is selected along the Earth-line ($\alpha = 0$, $\psi = 0$), while the other is selected near one of the constraining boundaries. An Earth-line maneuver is a highly desirable component, since greatest execution accuracy is attainable in this attitude by making pulsed trim corrections based on the observed change in the doppler tracking signal.

First Maneuver Sequence

The first Pioneer 10 maneuver was scheduled for 4 days after launch to provide an early opportunity to remove injection errors from the trajectory. This epoch, near the Earth, provides flexibility in spacecraft pointing while maintaining adequate spacecraft communication, but is at a sufficient time after launch to permit accurate trajectory estimates. A mean first maneuver of 30 m/s is required to remove the predicted launch vehicle injection errors

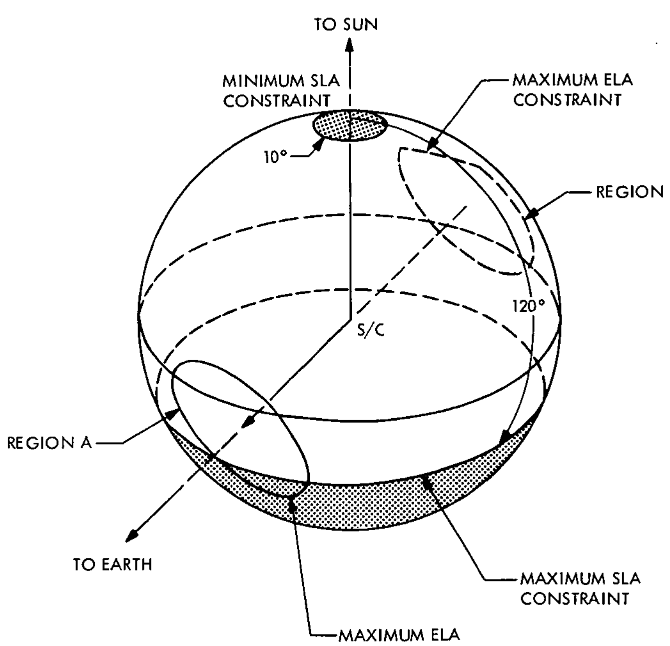


Figure 5. Spacecraft maneuver constraints

(Monte Carlo analysis, Reference 6). The execution errors of this maneuver map to the B-plane as an ellipse with a semi-major axis of approximately 25,000 km (one sigma). A second maneuver of approximately 1 m/s, scheduled for 20 days after launch, is required to remove this error.

A preliminary maneuver analysis using PMOPS, and an orbit solution obtained at launch plus 3 h, indicated that a 14.45-m/s maneuver would deliver the spacecraft to the specified target. This maneuver, with a rhumb angle of 251 deg and a precession magnitude of 75.7 deg, had a pointing direction with an ELA of 76 deg and an SLA of 106 deg.

During spacecraft reorientation to Earth-alignment after injection, telemetry dropouts were experienced in the ELA region where interference between the forward and aft spacecraft antennas exists. This condition, predicted in prelaunch analysis, confirmed the decision to restrict the maneuver to an ELA of less than 45 deg. Although the required single-maneuver direction did not violate the SLA constraints, it did violate the 45-deg ELA constraint. An equivalent two-component first maneuver strategy was developed within this constraint. In addition, it was decided to bias the aimpoint for this maneuver to leave a 0.5-m/s maneuver along the Earth-line for the second maneuver, 20 days after launch, on March 23, 1972. The planned Earth-line component would tend to reduce the pointing direction off the Earth-line and result in improved execution accuracy for the March 23 maneuver. The biased aimpoint for the first maneuver sequence was:

$$\bar{B}\cdot\bar{R} = 216{,}444 \text{ km}$$

$$\bar{B}\cdot\bar{T} = 842{,}564 \text{ km}$$

$$\text{TCA} = \text{December 4, 1973, 00:16:34 GMT}$$

and is shown together with the nominal aimpoint in the B-plane diagram in Figure 6.

The spacecraft sequences for the two-component first maneuver were generated using PMOPS and an updated trajectory estimate based on all doppler tracking data available 12 h before the scheduled maneuver. The required maneuver was reduced in magnitude to 13.45 m/s as a result of the updated trajectory estimate, but the adopted maneuver strategy was not affected.

The first component was initiated by precessing 45 deg from the Earth-line into a plane containing the required velocity (13.45 m/s) and the Earth-line, and adding 18.46 m/s to the spacecraft velocity. Following execution of the initially commanded sequence, the velocity projection along the Earth-line was trimmed to within 0.033 m/s of the desired value using the pulse mode operation of the velocity thrusters, and observing the change in the doppler tracking signal. The expected encounter conditions resulting from this maneuver alone were:

$$\bar{B}\cdot\bar{R} = 353{,}844 \text{ km}$$

$$\bar{B}\cdot\bar{T} = 967{,}188 \text{ km}$$

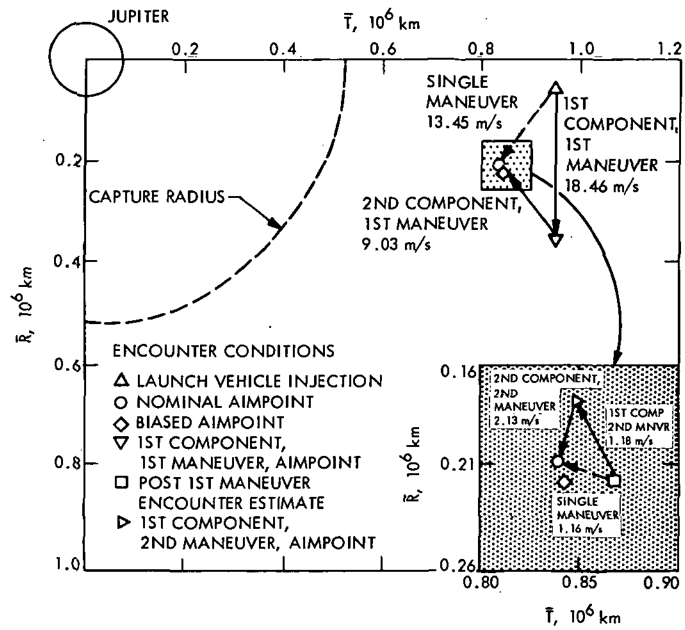


Figure 6. B-plane encounter conditions

TCA = December 2, 1973, 02:48:44 GMT

as shown in Figure 6. These encounter conditions provide for Jupiter arrival within the tracking overlap capabilities of the DSN.

The second component was initiated 7 h after completion of the first component by returning to Earth-alignment and adding 9.03 m/s toward Earth. The desired Earth-line component of velocity change was exceeded by 0.05 m/s but was not trimmed since a residual Earth-line velocity was consistent with the strategy selected for the second maneuver sequence scheduled on March 23.

The total velocity required for both components was 27.49 m/s, compared to 13.45 m/s for the single maneuver. This increase in maneuver velocity magnitude permitted a trajectory adjustment to the biased aimpoint without violation of the 45-deg ELA constraint.

Prior to the first maneuver, the spacecraft battery temperature was observed to be above its preflight prediction. A decision was made to precess the spacecraft backside away from the Sun after the first maneuver to better shade and protect the battery. A cruise attitude with the spacecraft spin axis 24 deg off the nominal Earth-line orientation was selected to maintain the desired signal strength using the medium-gain antenna (Reference 9). This orientation was maintained until re-alignment toward Earth at the second maneuver epoch.

Second Maneuver Sequence

During the period following the first maneuver, the spacecraft was tracked continuously by the DSN. Trajectory estimates were updated as new tracking data became available, and the trajectory changes resulting from the first maneuver were estimated. The 24-deg cruise attitude made data interpretation more difficult because of increased doppler noise caused by the antenna offset from the center of rotation, and an off-nominal solar pressure sailing effect. Solar sailing effects are translations of the spacecraft induced by solar radiation pressure (Reference 8). Torques about the spacecraft center of gravity are also induced by solar pressure, resulting in precession of the spin axis. The effects of solar pressure must be accurately modelled during orbit determination.

The second maneuver sequence was initiated on March 23 at 10:30 GMT. A trajectory estimate based on all doppler tracking data available 12 h before the scheduled maneuver indicated the spacecraft to have the following Jupiter encounter conditions:

$\bar{B}\cdot\bar{R}$ = 218,648 km

$\bar{B}\cdot\bar{T}$ = 869,643 km

TCA = December 3, 1973, 23:33:35 GMT

PMOPS was used to estimate the maneuver required to return the trajectory to the nominal aimpoint. A single maneuver of 1.16 m/s, with a rhumb angle of 1.9 deg, a precession magnitude of 48.6 deg, and a pointing direction with an ELA of 48 deg and an SLA of 13 deg, would provide this adjustment. While the 10- and 120-deg SLA constraints were not violated, the 48-deg ELA required for this maneuver was not acceptable because spacecraft communication requirements restricted the maneuver to an ELA of less than 24 deg. To accommodate the maneuver constraints, an equivalent two-component maneuver strategy was selected.

The first component was executed along the Earth-line by adding 1.18 m/s away from Earth. This orientation was determined to within ¼ deg using the spacecraft high-gain conical scanning capability. The magnitude of the initially commanded velocity change was less than the desired value by approximately 0.024 m/s. A single 1-s pulse was added which resulted in a net overshoot of approximately 0.008 m/s. Although a shorter pulse could be commanded, a decision was made not to trim further, but to account for this overshoot in the computations of the second component.

The second component followed 14 h later by precessing 24 deg from Earth-alignment and applying a velocity of 2.14 m/s to the spacecraft. Computations for this component were updated following the first component execution, and accounted for velocity increments to the spacecraft attributable to:

(1) The unbalanced thrusters used during the 24-deg precession to Earth-alignment prior to the first component.

(2) The first component velocity magnitude as observed in the tracking data, including the 0.008-m/s overshoot.

(3) The unbalanced thrusters used during the 24-deg precession from Earth-alignment to the second component pointing direction.

(4) The unbalanced thrusters used during the 24-deg precession back to Earth-alignment after second component velocity addition.

(5) The unbalanced thrusters used during precession to a 10-deg cruise attitude after completion of the second maneuver sequence.

This strategy allowed for known velocity perturbations.

The Earth-line component of the initially commanded second component velocity magnitude exceeded the desired value by 0.013 m/s. This residual was reduced to 0.004 m/s by trimming in the pulse mode. At the time of this trimming, the Earth direction was 9.3 deg from the direction of maximum flight time sensitivity. This geometry provided for excellent control of flight time, the most sensitive parameter affecting the possible occultation by the Jupiter satellite, Io.

The velocity added during the second maneuver sequence totalled 3.32 m/s, twice the velocity magnitude required for the equivalent single maneuver. The additional velocity provided a trajectory adjustment to the nominal aimpoint without violation of the 24-deg ELA constraint imposed at the time of the March 23 maneuver sequence.

A total of 30.81 m/s of the available 200 m/s was utilized to execute the first and second maneuver sequences.

Spacecraft Occultation by Io

The 12 known satellites of Jupiter can be subdivided into inner and outer groups. The inner group includes five bodies, four of which are quite large, and are called Galilean satellites after their discoverer (1610). One of the smaller of these satellites is Io, about the size of the Earth's Moon and nearly as dense. It revolves in a nearly circular, nearly equatorial, orbit with a period of approximately 42 h, at a distance of approximately 6 Jupiter radii from the center of Jupiter (Reference 10).

Gravitational forces associated with Io have suggested the presence of an atmosphere, although present spectroscopic analyses provide no confirmation. Scientists have noticed a brightening of Io after eclipse by Jupiter, with a return to normal brightness some 15 min later. This brightening has been interpreted to be the result of a temporary deposit of ice on the surface, caused by the lowering of temperature during the eclipse (Reference 10). The occultation of Beta-Scorpii C by Io on May 14, 1971, has allowed scientists to place an upper limit on the atmospheric pressure at 2×10^{-7} atm (Reference 11). Occultation of Pioneer 10 by Io would change the spacecraft radio signals, thus helping to determine possible atmospheric characteristics. The success of the first and second midcourse corrections has placed Pioneer 10 on a trajectory that may provide an occultation opportunity.

Pioneer 10 occultation by Io will occur when the spacecraft passes behind Io, as viewed from Earth. To maximize the probability of an occultation, a trajectory passing behind the full diameter of Io is desired. A center of Io occultation can be achieved by aiming at a point where the magnitude of $\bar{B}$ is 864,385 km (3 Jupiter radii at closest approach), with a θ of 14 deg, and a TCA of December 4, 1973, 02:26 GMT. A variation in any one of these parameters can degrade the occultation geometry. According to preliminary analyses by Kliore (Reference 12), occultation is most sensitive to the time of closest approach. Further analyses by the Navigation Team (Reference 13) indicated that a variation in TCA of ±7 min can cause a total miss of occultation, whereas variations along or perpendicular to $\bar{B}$ of ±25,000 km are required. As of May 28, 1972, trajectory estimates indicated the spacecraft to be headed for the following Jupiter encounter conditions:

$$\bar{B}\cdot\bar{T} = 208{,}672 \text{ km}$$

$$\bar{B}\cdot\bar{R} = 837{,}285 \text{ km}$$

$$\text{TCA} = \text{December 4, 1973, 02:35:13 GMT}$$

$$|\bar{B}| = 862{,}896 \text{ km}$$

$$\theta = 13.99 \text{ deg}$$

This trajectory will arrive at Jupiter 1489 km short of the nominal aimpoint in $|\bar{B}|$, 150 km high in the θ direction, and 9.2 min late in TCA. The one-sigma uncertainties in these conditions are 1000 km in spatial miss and 6.4 min in TCA, based on current trajectory estimates and Jupiter and Io ephemeris uncertainties. Based on these data, the probability of achieving an occultation by Io is about 40%.

A trim maneuver could be performed to adjust the current trajectory for an occultation across the center of Io. Present estimates indicate a maneuver in the cruise orientation along the Earth-line of approximately 0.12 m/s, on September 21, 1972, would provide this adjustment (Reference 14). This maneuver epoch has been selected to provide an opportunity to reduce the uncertainty in the trajectory estimate with the acquisition of additional tracking data and improved solar radiation modelling. This maneuver is expected to increase the probability of a successful occultation by Io to about 72%.

The execution accuracy of the Earth-line trim maneuver can be controlled to approximately 0.0004 m/s using the pulse mode. This error in velocity magnitude would project to a TCA error of much less than 1 min, well below the current trajectory estimate and Jupiter and Io ephemeris uncertainties.

The Earth-line maneuver could be delayed until much later in the mission when ephemeris uncertainties may be reduced, permitting improvement of the current trajectory estimate relative to Jupiter. The reduced uncertainties in this estimate would improve the accuracy of evaluating the required maneuver and further increase the probability of an occultation. A delayed maneuver would require a larger velocity correction and may also be less attractive from an operational standpoint.

The encounter geometry for an occultation by Io is shown in Figure 7. Occultation would occur 16 min after closest approach to Jupiter, with a maximum duration of 90 s. An hour after occultation by Io, Pioneer 10 will be occulted by Jupiter. The duration of Jupiter occultation is about 1 h.

Future Plans

The present Io occultation analyses are preliminary. Updating the trajectory estimate as additional tracking data become available will make maneuver strategy more certain and improve the probability of a successful occultation by Io.

Recently, the Io occultation geometry has been examined using computer-generated motion pictures representing the appearance of other bodies as seen from the spacecraft as it moves along the flight path. These film sequences have revealed the possibility of a Pioneer 10 occultation by a second Galilean satellite, Ganymede. It may be possible to adjust the current trajectory in both time and position at closest approach such that a dual occultation of the spacecraft by both Io and Ganymede results (Reference 15). This objective is currently being examined in some detail by the Navigation Team to determine possible trajectory adjustments and assess the impact of such changes on the primary mission objectives. These studies will be reported in future Pioneer documentation.

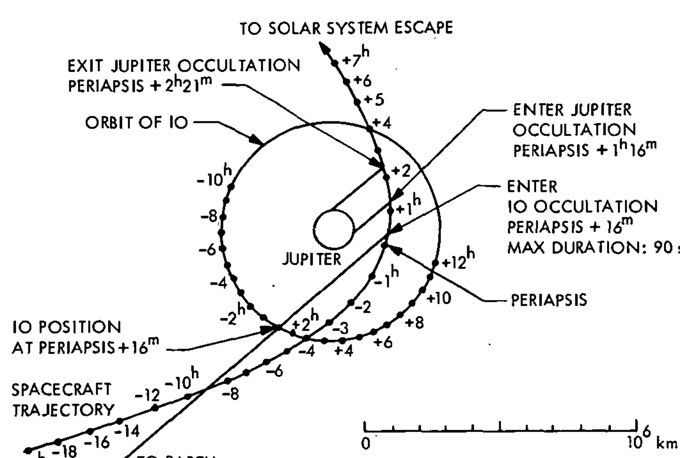


Figure 7. Ecliptic view of Io and Jupiter occultation geometry

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