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# NOZOMI CIS-LUNAR PHASE ORBIT DETERMINATION

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The Nozomi spacecraft is the first to achieve an Earth-Mars transfer orbit without a direct hyperbolic insertion maneuver. The trajectory is accomplished by use of multiple lunar gravity assists and a powered Earth flyby. This paper describes the Jet Propulsion Laboratory orbit determination effort in support of the mission. Navigation delivery goals were achieved despite weak data information content, due to spacecraft geometry factors such as large spacecraft range, small absolute velocity and low declination. Also described is the near real time assessment of the post Earth flyby orbit which enabled the design and successful execution of a critical trajectory correction maneuver.

## INTRODUCTION

Japan's Institute of Space and Astronautical Science (ISAS) launched Nozomi, its first mission to the planet Mars, in early July, 1998. In addition to its scientific objectives, the spacecraft also serves as an engineering demonstration of basic technology for planetary exploration. One of the new technologies is a unique trajectory, which uses solar gravitational perturbations at the weak stability boundary as an aid to achieve an Earth-Mars transfer orbit. This trajectory is an extension of a technique employed by two previous ISAS missions in the Earth-moon system[1,2]. It saves approximately 120 m/s of $\Delta V$ compared to direct hyperbolic insertion and is considered an enabling technology for the mission. Nozomi is the first spacecraft to employ this trajectory and provided on-orbit validation of the technique.

Nozomi is a cooperative mission between ISAS and the National Aeronautics and Space Administration (NASA). The NASA contribution includes navigation and tracking services provided by the Jet Propulsion Laboratory (JPL). Orbit determination for Nozomi is performed in parallel by both ISAS and the Multi-Mission Navigation (MMNAV) group at JPL. In this paper, information

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regarding the MMNAV orbit determination effort for the first six months of the mission is presented. The mission plan and spacecraft are characterized, followed by a discussion of the orbit determination process, including models, estimation procedure and data considerations. Results from various mission phases are presented and orbit solutions discussed.

## MISSION OVERVIEW

The Nozomi spacecraft, its launch and mission plan were conceived and executed by ISAS. The scientific objectives of the mission are to study the structure and dynamics of the Martian upper atmosphere and its interaction with the solar wind.

Nozomi was launched on July 3, 1998 using the newly developed M-V launch vehicle. The spacecraft was initially placed in a highly elliptical phasing orbit, with an apoapsis in the vicinity of the moons orbit. Over three months, numerous small maneuvers were performed to correct injection errors and target an outbound swingby of the moon. The late September 1998 lunar flyby increased the orbital period to three months and raised apogee to the vicinity of the weak stability boundary (approximately 1.7 million km). A maneuver at apoapsis of this orbit increased the spacecraft velocity. Nozomi then performed an inbound lunar swingby in late December 1998 to further increase it velocity, followed only two days later by a powered Earth swingby. The Trans Mars Insertion (TMI) burn at the final Earth periapsis was intended to place the spacecraft on a heliocentric transfer trajectory. Table 1 is a list of major events for the nominal mission and Figure 1 shows the cis-lunar trajectory.

| Event | Date (UTC) | Altitude (km) | Maneuver (m/sec) |
|---|---|---|---|
| Launch | 03-JUL-1998 18:12 | | |
| First Lunar Swingby | 20-SEP-1998 10:43 | 628 | |
| Second Lunar Swingby | 18-DEC-1998 07:34 | 2810 | |
| Trans Mars Insertion | 20-DEC-1998 08:10 | 994 | 340 |
| Mars Orbit Insertion | 10-OCT-1999 07:00 | 150 | 1270 |

**Table 1 Major Events In The Nominal Mission Plan**

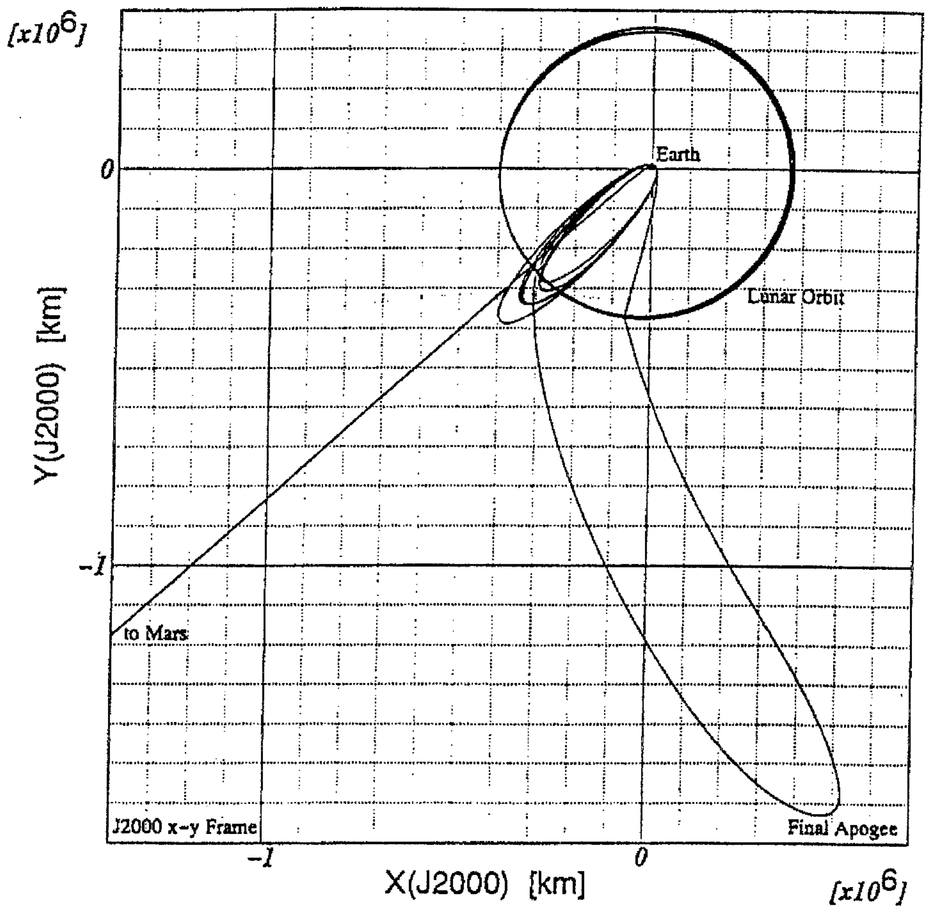


**Figure 1 The Nominal Cis-Lunar Phase Trajectory**

The nominal mission plan called for a Mars orbit insertion on October 11, 1999. This would place the spacecraft in an 150 km by 50,000 km orbit in the ecliptic plane. This orbit allows observations of the martian surface, its lower atmosphere and ionosphere, magnetosphere and intrinsic magnetic field during periapsis passage. Data recorded at that time, when instruments operate at maximum capacity, will be recorded onboard and transmitted to Earth during the remainder of the orbit. Measurements of the solar wind in the vicinity of the sub solar point will be made at apoapsis. The moons Phobos and Deimos can be photographed during untargeted encounters and the vertical structure of the Martian atmosphere would be examined by radio occultation experiments observed from the Earth. The expected mission lifetime in Mars orbit is about two years, limited by attitude control fuel for directing the body mounted communications dish at Earth.

## SPACECRAFT DESCRIPTION

Nozomi, whose name means "Hope" in Japanese, was know as Planet-B prior to launch. The spacecraft, shown in Figure 2, is very small by interplanetary standards. It had an injected mass of 540 kg, including 280 kg of propellant. The spacecraft bus measures only 1.6 m by 0.6 m and supports two fixed solar panels, which span 6.2 m from tip-to-tip. The spacecraft is spin stabilized, with attitude control maintained by ten mono-propellant 2.3 N thrusters. Large maneuvers are performed by a single bi-propellant 500 N engine.

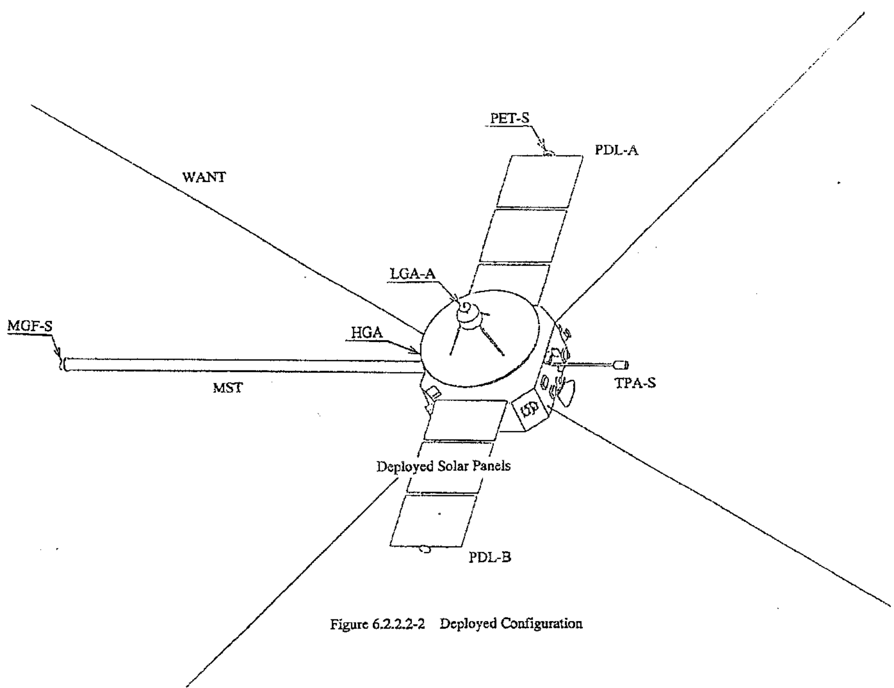


**Figure 2 The Nominal Spacecraft**

Communications is via two low gain antennae (LGA-A and LGA-B) which provide uplink and downlink at S-band frequencies, and a high gain antenna (HGA) which supports S-band uplink and both S-band and X-band downlink. LGA-A is located on the spacecraft spin axis, mounted on top of the antenna feed of the 1.6 m parabolic HGA mesh dish. LGA-B is mounted on the opposite side of Nozomi and is offset 1 m from the spin axis.

The scientific payload has a mass of 36 kg. Instruments include a visible light camera, ultraviolet spectrometer, neutral and ion mass spectrometers, dust counter and a suite of fields and particle sensors. NASA provided the Neutral

Mass Spectrometer for the science payload which was build by the Goddard Space Flight Center with Dr. H. Neumann as Principal Investigator.

## COOPERATIVE OPERATIONS

The primary tracking station for Nozomi is the 64 m antenna at the Usuda Deep Space Center. This station supports telemetry, command and navigation data collection functions. One limitation of this situation is that ISAS would only be in contact with the spacecraft about 10 hours per day. Another NASA contribution was to provide supplemental tracking for navigation data while Nozomi was out of view of the Usuda station. This support is provided by JPL's Deep Space Network (DSN) tracking stations at Goldstone and Madrid. Note that telemetry and command functions are not provided by the DSN stations.

Navigation and mission design for Nozomi is performed by ISAS. But as part of the DSN support, orbit determination is conducted in parallel with ISAS by the Multi-Mission Navigation (MMNAV) team at JPL, with each group generating solutions based on data collected from their respective tracking networks. This provides several distinct advantages to the project. First, the MMNAV solutions contain data from widely spaced DSN tracking stations which increases the navigation information content, especially for short data arcs. Second, spacecraft events, such as command uplinks and maneuvers, were scheduled during passes at the Usuda tracking station in Japan. As a result, maneuver design and initial assessment was often derived from MMNAV solutions based on DSN tracking data explicitly scheduled to bracket these critical activities. This required close coordination between ISAS and MMNAV regarding the timing of tracking passes and the scheduling of product deliveries. This was examplified during several mission redesigns, prompted by injection errors and the resulting reoptimization of the trajectory. Lastly, the fact that ISAS and MMNAV used different models, data sets and calibrations, allowed an independent verification of orbit determination solutions. Even though ISAS did not levy explicit navigation accuracy requirements on MMNAV, the favorable comparison of orbit determination estimates greatly increased confidence in the navigation process.

## ORBIT DETERMINATION

MMNAV orbit determination for Nozomi was performed by estimating the spacecraft trajectory and related physical parameters using data from DSN tracking stations. Dynamic models employed in the estimation process include

Newtonian and relativistic gravitational accelerations, oblateness, tides, solar radiation pressure (simple bus) and maneuvers. In addition, atmospheric drag was modeled for the launch phase and Earth periapsis passages.

Estimated parameters included the spacecraft state, solar pressure, maneuvers, drag (where appropriate), Doppler spin biases and range biases. Data models included corrections for Earth orientation parameters and calibrations for troposphere and ionosphere delay. Changes to this basic estimation strategy, incorporated as necessary in response to operational considerations, are discussed below. A summary of those differences is shown in Table 2.

| | **Prior to First Lunar Swingby** | **After the First Lunar Swingby** |
|---|---|---|
| Solar Pressure | Simple Bus Model | Oriented 3D Structure |
| Solve for Doppler Spin Bias | Yes | Yes |
| Remove Doppler Spin Signature | No | Yes |
| Solve for Range Bias | No | Yes |
| Doppler Data Weight | 2.5 mm/sec | 0.2 mm/sec |
| Range Data Weight | 10 meters | 3 meters |
| Typical Arc Length | 7 days | 21 days |
| Mapped Velocity Uncertainty ($3\sigma$) | 3.4 mm/sec | 1.1 mm/sec |
| Mapped Position Uncertainty ($3\sigma$) | 1.48 km | 0.56 km |

**Table 2. Summary of Estimation Strategy Differences By Mission Phase**

Tracking passes typically four hours in duration were scheduled to bracket maneuvers such that pre and post burn estimates could be delivered in accordance with a detailed operations plan provided by ISAS. The data used included S-band two way coherent Doppler, S-band two way range and angles.

For the majority of the cis-lunar phase, tracking of Nozomi was via LGA-B. This induced a sinusoidal spin signature which depends on spin rate, aspect angle and antenna offset. Due to this effect, Doppler data were usually weighted at 10 mm/sec, well below its inherent precision level. The spin also induces a Doppler bias which is proportional to the spin rate (approximately 10 rpm). For periods when LGA-A was used, the spin signature was absent and the Doppler data were weighted at 0.2 mm/sec.

Range data was also effected by the spacecraft spin, but this was usually ignored because the range data were weighted at 10 m and the spin signature amplitude was only 1 m. Biases in the range data due to station calibration errors were also accounted for and were on the order of 15 m.

Angle data were only used when the spacecraft was close to the Earth. This low precision, yet critical data type, is used to constrain early trajectory estimates when short arcs of Doppler have minimal information content and range biases can lead to erroneous solutions. Angle data, available only on the DSN 26m sub net, were weighted at 0.02 degrees.

## LAUNCH AND EARLY OPERATIONS

The Nozomi was launched on July 3, 1998 from Uchinoura, Japan. The spacecraft was injected directly into a highly elliptical Earth orbit and was quickly out of view of Japanese tracking stations. The Santiago, Chile 9m tracking station, acting under contract to the DSN because of its favorable location with respect the Nozomi ground track, acquired the spacecraft 35 minutes after launch. MMNAV was able to quickly determine that a significant overburn had occurred, using only the initial angle data, and transmitted this data to ISAS. At launch plus 3 hours, the DSN 26m station at Goldstone acquired the spacecraft, but a double hardware failure prevented the collection of Doppler and range data for the entire pass. Despite this difficulty, angle data from Goldstone used in concert with Doppler and range from Santiago confirmed that there had been a three sigma overburn.

MMNAV delivered state vectors from four solutions during the initial launch support. ISAS was able to use this data to update frequency and pointing predicts for the first Japanese tracking pass, which occurred eleven hours after launch. ISAS also used this data to design a 45 m/s injection correction maneuver, which was executed just three hours after the start of the initial Japanese tracking pass.

## PHASING ORBITS

It is worth noting that the phasing orbits, by their very nature, add operational flexibility to the mission plan. Many small maneuvers are required to deliver the spacecraft to the first lunar swingby aimpoint. The large number of phasing orbits present frequent opportunities for perform maneuvers at apoapsis and periapsis[3] (where the $\Delta V$ expenditures are minimized). They also allow time to characterize and verify spacecraft systems (maneuver, attitude control, thermal, communications, etc.) and ground systems prior to critical on-orbit events.

The large post injection maneuver, mentioned in the previous section, was not intended to remove all of the injection error. Its purpose was only to place Nozomi in an orbit from which addtional maneuvers could reestablish an viable trajectory without exceeding the spacecraft $\Delta V$ budget. The new mission plan included significant changes in the dates and sizes of maneuvers early in the phasing orbits. A 50 m/s burn was also added on August 16, 1999, after which the trajectory closely matched the prelaunch plan.

After implementing the new tracking schedule, a routine was quickly established in which numerous state vectors were exchanged. ISAS and MMNAV made daily deliveries for the three days leading up to each maneuver and for two days afterward (for design and verification purposes respectively). Vectors were also exchanged at least once a week between maneuvers.

Orbit determination during this phase was fairly routine. Tracking was provided by the DSN 26m sub net and the trajectory orientation was such that the inbound legs used LGA-A with no Doppler spin signature, and the outbound legs used LGA-B, with the spin signature. The highly elliptical orbit, combined with the frequent maneuvers and the large spin signature made the estimation process relatively insensitive to small error sources such as solar pressure and non-gravitational accelerations. Short data arcs, which did not span maneuvers, were sufficient to determine period and inclination and progressively refine targeting the first lunar swingby. Everything seemed under control.

## FIRST LUNAR SWINGBY

The first lunar swingby phase began with a final targeting maneuver (3 cm/sec), executed after apoapsis on September 16, 1998. Predicting the trajectory ahead to the lunar encounter was somewhat uncertain due to atmospheric drag effects during an Earth periapsis passage on September 20 at an altitude of 630 km. DSN 34m Standard subnet antenna supports were added to the tracking schedule. The typical Doppler spin signature was present for most of the outbound journey until the Nozomi orientation was changed to enable scientific observations of the Moon. The lunar flyby occurred on September 24, 1998 at an altitude of 4094 km providing return of the first Japanese pictures of the lunar surface.

The swingby altered the trajectory such that the outbound leg now used LGA-A and the Doppler could be weighted at its noise level. At this point large discontinuities in Doppler data were observed between successive DSN tracking passes and it became obvious that the MMNAV predictions of the lunar flyby, based on tracking from the Earth-Moon transfer, were in error by 1.1 km. Analysis eventually showed that Doppler data from several passes acquired at the 34m antenna at Goldstone, which appeared normal, were in fact corrupted and that the errors were masked by the spin signature and the loose data weight. The source of the problem was eventually identified in the data collection software at the station and procedures were put in place to flag the errors for correction prior to navigation processing.

## WEAK STABILITY BOUNDARY

Following the second lunar swingby, Nozomi was placed in an even more eccentric Earth orbit with a period of three months and an apogee distance of 1.7 million km. Three maneuvers were performed during this phase to target the second lunar swingby. But a number of factors contributed to the general degradation of orbit determination accuracy during this phase. These included the extreme apogee range and the fact that the spacecraft was moving at less than 1 km/sec perpendicular to the line of sight. Nozomi was also close to zero degrees declination where there are known limitations on Doppler information content. In addition, the spacecraft was in the vicinity of the weak stability boundary, where its motion was no longer dominated by the Earth gravitational potential. As a result, the trajectory was very sensitive to solar pressure due variation due to small attitude perturbations. Lastly, the spacecraft was reoriented such that tracking data was again collected via the LGA-B antenna and Doppler data degraded by the presence of a spin signature.

This situation required a complete reworking of the orbit determination strategy. It was now necessary to employ longer data arcs and fit through maneuvers to tie the trajectory down in geocentric space. The solar pressure bus model was abandoned in favor of a three dimensional model based on the Nozomi physical structure. This model included a cylinder and a flat plate to represent the spacecraft body and solar panels respectively. Daily updates to the spacecraft orientation were introduced to reduce solar pressure mismodeling. Finally, the Doppler data were preprocessed, on a pass by pass basis, to remove the spin signature so the data could be weighted at its inherent noise level.

Navigational tracking data were also exchanged between ISAS and MMNAV late in this phase so orbit determination could be performed on joint data sets in support of the critical second lunar swingby targeting.

## SECOND LUNAR SWINGBY

As a result of the previously mentioned effort, the second lunar swingby proceeded with little of the excitement of the first swingby. The final 30 cm/sec targeting maneuver was delayed from November 4 to December 16 to assure that solar pressure midmodeling would not effect the flyby. The inbound lunar swingby occurred on December 18, 1998 at an altitude of 2810 km Delivery accuracy was on the order of 500 m. Once again, images of the lunar surface were collected by Nozomi.

## TRANS MARS INSERTION

The final event of the cis-lunar phase was the 420 m/s Trans Mars Insertion (TMI) burn, which placed the spacecraft into heliocentric orbit. The maneuver occurred at Earth periapsis on December 20, 1999, at an altitude of 995 km (just 48 hours after the second lunar swingby). This maneuver was critical because each m/s of error in TMI execution would cost 3.5 m/s during a TMI correction (TMIc) burn 12 hours after periapsis[4].

TMI occurred out of contact with ground stations. ISAS uplinked the final burn parameters just prior to the end of Usuda visibility. The Goldstone tracking complex had the first pass following the burn. The DSN 26m station was employed because of the high slew rates necessary for this support and the importance of angle data in determining the initial post burn trajectory. The DSN 26m station at Madrid was also configured in case anticipated high winds forced closure of the Goldstone. Tracking was further complicated by the 25 rpm spin necessary for attitude stability, antenna switching, spacecraft reorientation and the rapid change in aspect angle during the support.

Weather conditions improved at Goldstone and the DSN 26m station acquired Nozomi on schedule. MMNAV had the primary responsibility to make a rapid assessment of the maneuver performance. Using just one hour of data, it was determined that a 100 m/s under burn had occurred and MMNAV promptly informed ISAS via voice lines. ISAS immediately began preparations for the correction burn, which had to be performed during the next Usuda pass. The Goldstone 26m pass was extended to allow further refinement of the trajectory.

The near real time assessment by MMNAV provided accurate antenna frequency and pointing predicts updates for the critical command uplink at Usuda. Close coordination between the two agencies enabled the design and successful execution of the 340 m/s trans Mars orbit insertion correction maneuver. But the TMIc left Nozomi with insufficient propellant to carry out its science objectives if it entered Mars orbit in October, 1999.

This resulted in another mission redesign. The new plan adds 3 full solar orbits, two Earth swingbys and one lunar swingby to the Mars transfer trajectory. Arrival at Mars is now scheduled to occur in January 2004. The final Mars orbit will still enable the mission to achieve all of its science objectives.

## ACKNOWLEDGMENTS

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