ORBIT DETERMINATION OF HITEN FOR INSERTION INTO LUNAR ORBIT

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Between 15 February 1992 and 10 April 1993 the Japanese HITEN spacecraft carried on data gathering in orbit about the Moon. These 14 months represented the first time since the Apollo-17 mission in December 1972 that there was communication with any active spacecraft in orbit about our nearest celestial neighbor. This paper describes 1) the JPL discovery of an integrated trajectory which revealed the opportunity for a near zero impulsive ΔV transfer of HITEN from geocentric to selenocentric orbit ancl 2) the tracking and orbit determination support provided by the JPL Deep Space Network and Multimission Navigation Team in support of the final targeting for lunar insertion.

INTRODUCTION

Three planetary missions have had the unique property that each involved an encounter with a target body which was not included in the nominal mission profile at launch, The first two were Mariner Venus Mercury and the International Sun Earth Explorer-3 (ISEE-3) in the 1970's. The latest is MUSES-A (HITEN) which was launched in January 1990.

The NASA/JPL Mariner 1973 was launched on a trajectory for a flyby of Venus and encounter of Mercury. While in transit, the aim point at Mercury was selected so that the spacecraft's trajectory was able to achieve subsequent multiple flybys of Mercury. The second example, the NASA/GSFC ISEE-3 was launched in 1978 into a halo orbit about the Earth-Sun libration point at a geocentric distance of 1.5 million kilometers. After an extended period in halo orbit, the spacecraft was subjected to a small maneuver which altered the trajectory so that it led to a series of close approaches to the Moon. The final lunar swingby on 22 December 1983 at an altitude of 119 kilometers caused a transfer from geocentric to heliocentric orbit. This metamorphosis from ISEE-3 into the International Cometary Explorer (1CE)

led to Man's first encounter with a comet when the spacecraft flew through the tail of Comet Giacobini-Zinner in September 1985.]'-[2]

This brings us to the most recent example. In January 1990, the Japanese ISAS MUSES-A mission launched the HITEN/HAGAROMO mother-daughter spacecraft. An illustration of the spacecraft is provided in Figure 1. Trajectory design for this mission made use of many of the lunar gravity assist concepts employed in the ICE mission. The subsatellite, HAGAROMO, was released and presumably went into selenocentric orbit at the initial lunar flyby. It was intended to serve as a beacon, but failed prior to release. After a brief sojourn in the Earth's geotail region, HITEN was then targeted for 2 low altitude aerobraking passes in the Earth's atmosphere. It then made an excursion through cis-lunar space which brought HITEN into the vicinity of the two Earth-Moon L_4 and L_5 stable Lagrangian points. The aerobraking and libration point excursion phases where secondary mission objectives. After launch, the opportunity for a 1992 capture into lunar orbit was discovered and implemented.

The focus of this paper is to 1) provide a description of the process which led to the discovery of an integrated trajectory which revealed the existence of the opportunity for a near zero impulsive ΔV transfer of HITEN into lunar orbit and 2) to document the tracking and orbit determination support provided by the JPL Deep Space Network (DSN) and Multimission Navigation Team in support of the final lunar targeting.
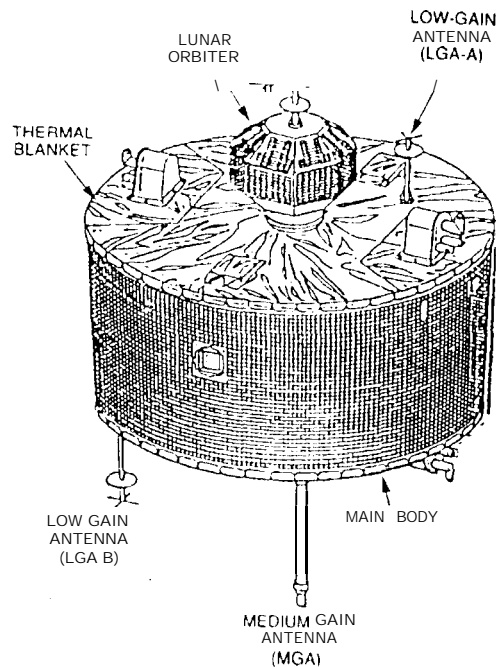

Figure 1   HITEN/HAGAROMO Spacecraft, Prior to Subsatellite Release

MISSION DESCRIPTION

HITEN was launched on January 24, 1990 by Japan's Institute for Space and Astronautical Science (I SAS). HITEN and it's small accompanying subsatellite HAGAROMO were launched together as MUSES-.4. One of HITEN's primary objectives was to demonstrate attitude and mbit control technology of a spinning spacecraft for control of the line of apsides as a precursor to the current GEOTAIL mission.[3] This control made regular use of gravity assist lunar swingbys. Accurate targeting of lunar swingbys early in the mission was important with respect to the total AV budget for the entire mission. The DSN was requested to provide tracking and orbit determination support during the launch and early orbit phase of the mission as well as during other phases of the mission where accurate OD was required. High accuracy was more readily achievable with tracking data from a worldwide network rather than from the Japanese domestic stations which offered limited viewing geometry relative to the DSN.

JPL's Multimission Navigation Team provided post injection OD which assisted the ISAS recovery from a non nominal injection. Maneuvers based on JPL orbit solutions were performed which pumped the en-bit out from the actual 6 clay period initial orbit with an apogee at approximately 3 x 105 km to the desired 12 day period orbit with an apogee beyond the lunar orbit. At this point the nominal mission was resumed.

The mission scenario continued with the release of HAGAROMO at the first lunar swingby and firing of its onboard thruster for injection into lunar orbit. HAGAROMO's exact fate is unknown as the subsatellite suffered a loss of its electrical system prior to separation. The intended lunar beacon is assumed to be a mute voyager in possible eternal orbit about the Moon.

HITEN continued its journey on a geocentric trajectory inducting a series of lunar swingbys in conjunction with propulsive trajectory change maneuvers (TCMs) which maintained the distant apogees in the anti-Sun direction, i.e. the geotail direction. This phase of the mission continued through 1990. In March 1991, HITEN was targeted for 2 aerobraking passes through the Earth's atmosphere. The first was at an altitude of 125km and the second at 120km.[4-6]

After surviving these two brief escapades, HITEN spent the next six months in distant Earth orbit and passed in the vicinity of the two stable Earth-Moon libration points L_4 and L_5. From HAGAROMO's release to the completion of the libration point excursion, HITEN fulfilled the prelaunch mission plan.

The road to lunar capture began immediately after the second aerobraking pass in March 1991. The orbit profile from aerobraking to lunar capture is illustrated in Figure 2. The trajectory is displayed in two coordinate systems labeled A and B. In the illustration on the left labeled A the coordinate system is geocentric inertial as viewed from above the ecliptic plane. The direction to the vernal equinox is indicated. The plot begins with the final aerobraking orbit revolution and indicates the subsequent close swingby of the Moon (S_9) which greatly increased the apoapsis distance and started the spacecraft on a trajectory leading to lunar capture. Also indicated is the final lunar insertion point reached in February 1992.

3

The illustration on the right labelled B is the same orbit as viewed in a non-inertial rotating frame in which the Earth-Moon direction is fixed. The plot is again viewed from above the ecliptic, but begins sometime after the date of the lunar swingby (S_9). This plot also ends with the point of lunar orbit insertion in February 1992.
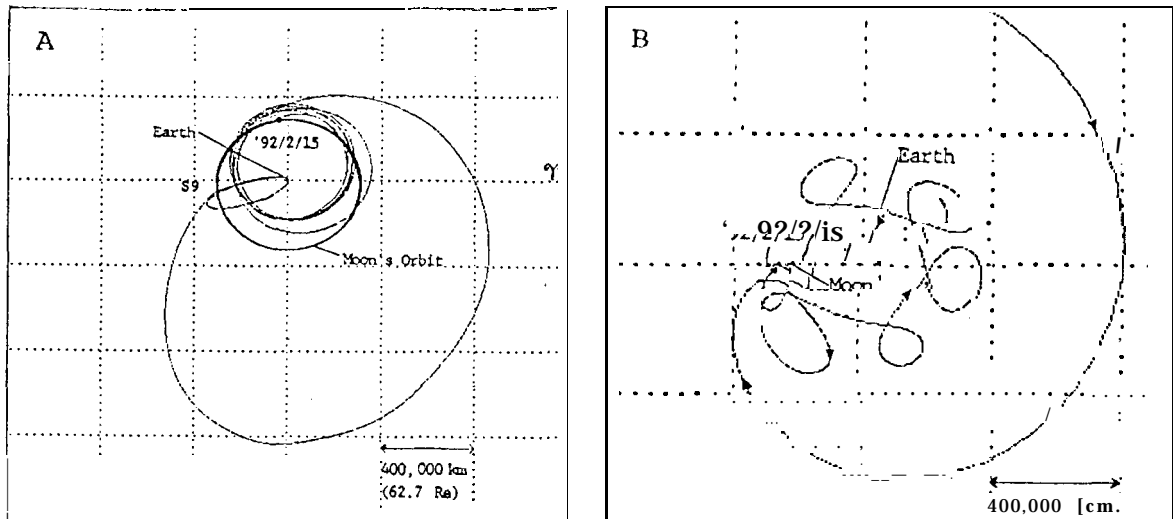

Figure 2 HITEN Lunar Transfer Trajectory

A brief historical review of the activity at JPL which made it possible for ISAS to modify the HITEN mission profile to redirect the spacecraft for lunar capture is provided in the following section.

On 15 February 1992 HITEN went into selenocentric orbit. What had begun as a Earth orbiter mission 25 months earlier became the first live spacecraft in lunar orbit since the Apollo missions of the 1970's. This return to the Moon marked the first resumption of extended communication with any spacecraft or instruments in orbit or on the surface of our closest celestial neighbor since the Apollo-17 mission in December 1972 and the deactivation of the ALSEP instrumentation in the late 1970's.

The initial orbit had a period of approximately 3.5 days with aperilune altitude of approximately 420 km. Trajectory change maneuvers were executed to prevent periods of extended duration solar shadowing which would otherwise have occurred in mid 1992. Third body effects had a significant effect on the orbit. It appeared that these would eventually lead to an escape from the Moon and return to geocentric orbit. However, the execution of occasional small maneuvers maintained the selenocentric orbit. Eventually, concerns in regard to the duration of upcoming solar shadow passages led to the implementation of a maneuver which targeted HITEN for a frontside impact on 10 April 1993.

4

HITEN LUNAR TRANSFER TRAJECTORY

Early in the spring of 1990, the Hiten spacecraft was in an orbit about the Earth that extended to the vicinity of the Moon's orbit and beyond. As a result of the premature demise of HAGAROMO, ISAS investigated the possibility of sending HITEN into lunar orbit. The amount of propellant remaining (about 250 m/s ΔV) was sufficient to perform the planned aerobraking experiment but achieving lunar orbit seemed doubtful and certainly both of these mission objectives could not be accompli shed using conventional orbit transfer techniques.

The classical method for design of lunar transfer trajectories is the Hohmann transfer orbit. A spacecraft is injected into an elliptic transfer orbit with perigee near the Earth and intersects the Moon's orbit near apogee. An orbit transfer maneuver is performed at lunar periapsis that places the spacecraft in a closed orbit about the Moon. This type of trajectory was used by the Apollo mission and other missions to the Moon.

Since the amount of propellant required for the Hohmann transfer was close to the capability of the Hiten spacecraft, it was speculated that some additional savings could be obtained by first modifying the spacecraft orbit about the Earth with a Moon gravity assist. After some initial tries, it became evident that this method alone would not suffice. At this time, it was suggested[7] that the trajectory segment approaching the Moon may be a capture orbit. It was speculated that an escape trajectory via gravity assist from the Earth-Moon system could be connected to a capture orbit via a bicliptic transfer maneuver far from the Earth-Moon system and thus achieve some savings over the Hohmann transfer orbit.

During the last weekend of May 1990, an attempt was made to design a bicliptic transfer orbit from HITEN's initial orbit about tile Earth to lunar capture. This work was being pursued in a casual manner because the low expectation of success did not seem to justify expenditure of any additional effort. The method used was to initialize the orbit at the Moon and integrate backward until an escape trajectory was found. Hitens orbit was then modified with two small propulsive maneuvers to phase the orbit for lunar flyby and escape. The escape trajectory was then targeted by a simple shooting method to intercept the capture orbit and a propulsive maneuver calculated to reverse the trajectory from escape to capture. Initial results indicated that this maneuver would be very large (about 400 m/s). With the bicliptic transfer method it is not possible to perform the transfer far enough away from the Earth-Moon system to achieve smaller ΔV because the Sun's gravity accelerates the spacecraft away from the Earth-Moon system at about 1.5 million kilometers from the Earth.

However, in the process of attempting to target a biclipitc transfer, it was noticed that the partial derivative of ΔV with respect to variations of orbital eccentricity at the Moon was negative. The orbital eccentricity at the Moon was used as a parameter to control the capture orbit and very small changes in this parameter produced huge changes in the ΔV required for t ransfer. If too large a change in eccentricity was attempted the spacecraft would escape into orbit about the sun or fall into closc orbit about the Earth indicating a highly unstable orbit. This

behavior suggested that a minimum ΔV solution existed somewhere between these extremes. The eccentricity at the Moon was Varied and HITEN's orbit modified in an attempt to follow the decrease in ΔV Until the minimum was found. Very small changes in eccentricity at the Moon were required starting in the third decimal place and eventually in the fifth decimal place, After about 100 tries a trajectory was found that required about 30 m/s of AV.

This initial trajectory was adapted to the expected time period when this procedure might reasonably be attempted and transmitted to ISAS in early June of 1990. Since the phasing of the escape and capture orbits were not favorable, a deep space maneuver of over 100 m/s was required to phase the Earth escape trajectory with the lunar capture trajectory. After some revision in the mission, ISAS adopted a variation of this approach and the resulting trajectory is shown in Figure 2.

COVARIANCE ANALYSIS

Covariance analysis was performed at J PI. to assess the lunar orbit insertion support. The then current trajectory was used. For the purpose of analysis, there was no particular requirement on target location or on time of flight. Hence there was only a goal to establish the uncertainty in predicting the lunar encounter aim point.

Bar charts in Figure 3 show the view periods for the DSN 26 meter diameter DSS 16 at Goldstone, California and DSS 66 near Madrid, Spain which were to track the spacecraft prior to lunar insertion. Two-way coherent range and range-rate data were simulated for these 26 meter sites at a sample rate of 1 measurement per 60 seconds. Data arcs extended from 3 clays prior to perilune until 1.5 days prior to perilune ancl were assumed to be free of any maneuver. This tracking schedule allowed for orbit solution delivery in sufficient time for analysis and implementation of any targeting trim maneuver (if required) during the final pre-perilune view period from the ISAS tracking site.

DSS 16		■	■	
DSS 66	•|	■		
EVENT			★ JPL sol. to ISAS	★ Lunar Clos.Ap.
DAY	12 FEB	13 FEB	14 FEB	15 FEB

Figure 3   Tracking Schedule and Events

Table 1 indicates the data weights for range and Doppler data and the apriori one sigma uncertainties associated with all parameters of interest. Uncertainties in

6

estimated parameters were mapped to lunar closest approach and the target plane uncertainties determined. This will be discussed in more detail later.

Table 1
ORBIT DETERMINATION ERROR ASSUMPTIONS

Data Weights

DATA	WEIGHTS
Range	1011"1
Range Rate	1.5 cm/s

Apriori Parameter Uncertainty

PARAMETER	UNCERTAINTY
S/C position	10^5 km
S/C velocity	102 km/s
S/C solar reflectivity coefficient	0.4
Range data biases	50 m

ORBIT DETERMINATION

Data circuits were not available between JPL and ISAS. Each agency therefore performed independent orbit determination prior to the orbit insertion using radio metric tracking data from their respective tracking network. Due to the long baseline between DSN stations, JPL enjoyed a clear advantage of station geometry in its orbit determination effort. JPL orbit solutions were sent to ISAS per an established schedule.

HITEN is a spin stabilized spacecraft with its spin axis aligned along the centerline of the cylindrical bus. There was a circularly polarized low gain antenna (LGA) at each end of the bus, but link margin analysis mandated the use of the medium gain antenna (MGA) during this phase of the mission. The MGA is a linearly polarized antenna located at one end of the cylinder. Location of these antennas is illustrated in Figure 1. The orientation was parallel to, but offset from the spin axis by just over 0.6 meters. This induced a sinusoidal signature in the range rate data having an amplitude which is a function of the antenna spin axis offset distance, spin rate and aspect angle as viewed from the ground station. Alias frequencies may also be present when the sample rate is less than the spin rate.

Orbit Determinations using range; and range rate radio metric tracking data prior to lunar orbit insertion solved for the spacecraft state (position ant] velocity), spacecraft coefficient of solar reflectivity and station range biases. Apriori uncertainties for these parameters arc consistent with those used for the covariance analysis (Table 1).

Computer turn around times posed no constraints. Hence the Newtonian force model included all the bodies available on the JPL Planetary Ephemeris file. Relativistic effects were computed for the Sun only. A 12x12 gravity harmonics field for the Earth and a 7x7 model[8] for the Moon were used.

Data weights used in the orbit determination were the same as used in the covariance analysis (Table 1). The total data arc from the two DSN sites spanned 32 hours on 12-14 February. The 2-way Doppler data from DSN stations were typically generated at a rate of 1 per 60 seconds. Some data recorded at 1 per 30 seconds were also used. Range data had a sample rate of 1 per 60 seconds throughout .

Figure 4 illustrates the post fit 2-way coherent Doppler and range observation residuals from D SN sites at G oldstone and Madrid. Both measurement dat a t ypes are fit with a near zero mean.
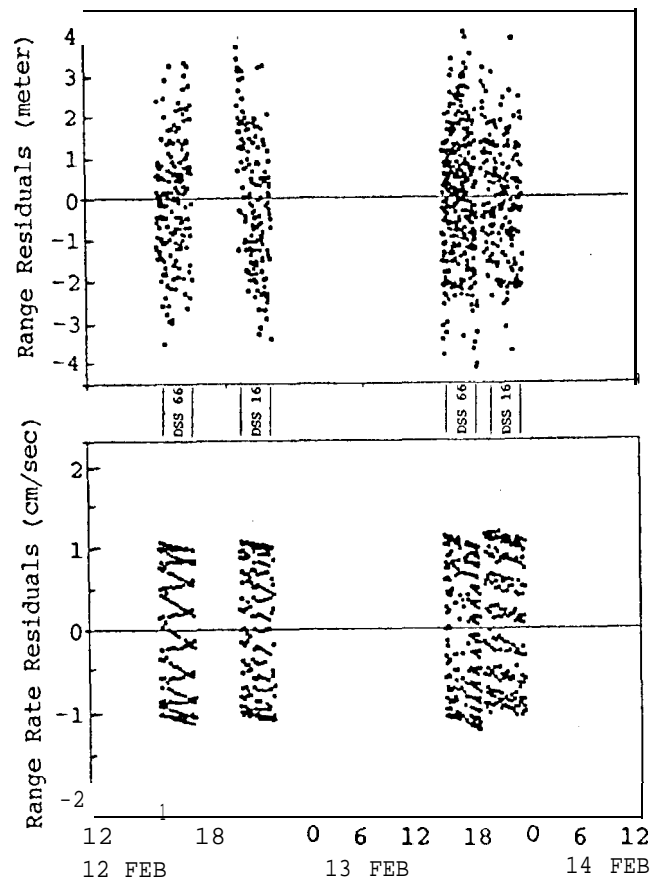
Figure 4 Post **Fit Range** and Range Rate Residuals

'The post fit Doppler residuals had a 1-sigma RSS of about 7.5 mm/see. It is noted that this does not reflect the noise in the data, but rather the unmodeled spin induced signature. Range data residuals were about 4.5 meters peak to peak with a l-sigma RSS of about 1.3 meters. Some of this is also due to the unmodeled antenna spin axis offset but here the effect is masked by the peak to peak range data noise.

DISCUSSION OF RESULTS

The converged orbit solution and associated uncertainties were propagated to the lunar B-plane as defined in Figure 5. Dispersion ellipses resulting from both the orbit determination and covariance analysis are superimposed in Figure 6. In reality the exact target point and the time of fright differed in these two cases. This was because the trajectory propagation used in the covariance analysis was maneuver free, while the actual trajectory includes maneuvers. This was assumed to have only a second order effect on the delivery accuracy statistics. As expected, the results indicate both magnitude and orientation compatibility of the covariance analyses and the actual orbit determination dispersion ellipses. The target plane error ellipse indicated a semi-major axis of 3.3 km oriented nearly parallel to the B·R direction.

On the day following transmission of these results to ISAS, an orbit insertion burn of 81.8 m/s was executed based on the JPL orbit determination. Subsequent orbit determination at ISAS indicates that the achieved post insertion orbit is consistent with the final JPL pm-encounter orbit solution.
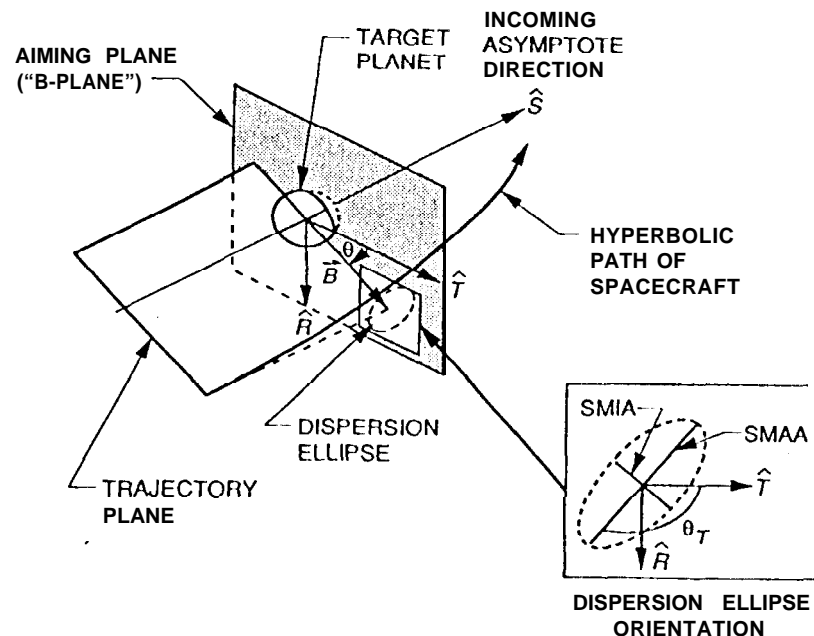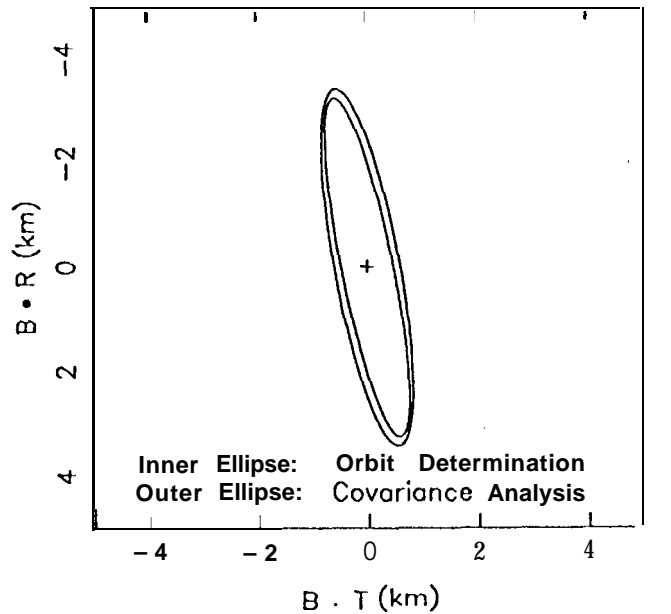

Figure 5 B-Plane Definition

9

Figure 6 Dispersion Error Ellipse

EPILOGUE

HITEN's Mission to the Moon came to an abrupt end with the termination of telemetry at the USUDA ground reception time of 15:03:25.7 UTC April 10, 1993. Tracking data collected during the final USUDA pass will be used in conjunction with a new 60th order and degree lunar harmonic gravity field recently developed at JPL.[9] The model is based on 1960s-1970s Lunar Orbiter and Apollo Missions tracking data. HITEN may provide the first opportunity to "test" the new model.

The new technique for transfer from Earth orbit to lunar orbit successfully demonstrated with HITEN is planned for use on the ISAS Lunar-A mission in 1997.

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