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Trajectories and Orbital Maneuvers for the First Libration-Point Satellite

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On August 12, 1978, a scientific spacecraft called International Sun-Earth Explorer-3 (ISEE-3) was launched towards the interior sun-Earth libration point, L1. The spacecraft was placed into a "halo orbit" around the L1 point on November 20, 1978, thus becoming the first libration-point satellite. During its 100-day transfer trajectory, ISEE-3 lingered in a region where the gravitational effects of the sun and the Earth are comparable, leading to some interesting tradeoffs concerning the maneuver strategy for halo-orbit insertion. Following orbit insertion, stationkeeping maneuvers were required to maintain the delicate equilibrium in the halo orbit. Details are presented for all of the ΔV maneuvers that were executed prior to the completion of the first halo orbit on May 14, 1979. Orbit selection, trajectory design, and the scientific objectives of the ISEE-3 mission are also discussed.

Introduction

MOTION in the vicinity of the libration points of the restricted three-body problem has been a favorite topic for researchers in celestial mechanics ever since the existence of these points was discovered by Lagrange in 1772. The subject received a considerable boost in popularity during the early years of the Space Age, and several ideas concerning the utilization of libration-point orbits were introduced.[1,2] In 1972, some 200 years after Lagrange's discovery, a decision was made to incorporate a libration-point orbit in a NASA flight project. Finally, on November 20, 1978, a spacecraft called International Sun-Earth Explorer-3 (ISEE-3) was inserted into a "halo orbit" around the interior sun-Earth libration point, L1. This was the first time that an artificial satellite had been placed into a libration-point orbit.

This paper describes the trajectories and orbital maneuvers involved in the ISEE-3 mission. The flight history from launch in August 1978 to the completion of the first halo orbit in May 1979 is documented. Pertinent aspects of the pre-flight mission analysis are also given.

Mission Objectives

ISEE-3 is a scientific satellite that has been designed to carry out a number of particles and fields investigations in the interplanetary medium. It has been stationed in an orbit around the sun-Earth L1 point to aid studies of solar-terrestrial interactions. Because the sun-Earth L1 point is located between the sun and the Earth about 1.5 million km (~0.01 AU) from the Earth, ISEE-3 is able to continuously monitor the characteristics of the solar wind and other solar-induced phenomena such as solar flares about an hour before they disturb the space environment near the Earth. While ISEE-3 is monitoring conditions in the interplanetary medium upstream from the Earth, two other spacecraft of the ISEE Program, ISEE-1 and 2, are analyzing dynamical processes inside the Earth's magnetosphere. Thus, ISEE-3 provides the important "input function" for the measurements of ISEE-1 and 2. Additional ISEE-3 objectives include the study of cosmic rays, solar radio bursts, and a wide variety of interplanetary plasma phenomena. One-AU baseline support for experiments on deep-space probes such as Voyager is also provided. Detailed descriptions of the scientific objectives and experiments for the ISEE Program can be found in Refs. 3 and 4.

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Spacecraft Description

Figure 1 depicts the ISEE-3 spacecraft in its orbital configuration with booms and antennas deployed. The drum-shaped spacecraft is spin stabilized and has a nominal spin rate of approximately 20 rpm. To satisfy a science constraint, the spin axis is oriented perpendicular to the ecliptic plane with an allowable deviation of ±1 deg. Attitude determination is accomplished with a redundant pair of high-resolution sun sensors (measurement accuracy ~0.1 deg).

A panoramic attitude sensor (measurement accuracy ~1 deg) is also available. It will usually take at least two weeks to accurately determine the spacecraft attitude with the sun sensors. The panoramic attitude sensor was needed for real-time attitude operations early in the mission. However, this sensor is not very useful in the halo orbit due to the poor sun-spacecraft-Earth geometry in this region.

The spacecraft is equipped with a hydrazine propulsion system for orbit and attitude control.[5] This system has a total of twelve thrusters made up of four radial, four spin-change, two upper-axial, the two lower-axial jets. The fully redundant thruster configuration permits the spacecraft to thrust in any direction without changing its attitude. The hydrazine fuel is stored in eight conospherical tanks. These tanks contained 89 kg of usable propellant at launch which corresponds to a ΔV capability of ~430 m/sec.

The tower structure supports the medium-gain S-band antenna as well as some scientific instrumentation. The medium-gain antenna has a flat, disk-like ("pancake") pattern that is perpendicular to the spin axis and has an effective beamwidth of 12 deg. Its gain is roughly 7 dB over an isotropic antenna. At the libration-point distance, the S-band telemetry system can sustain a continuous bit rate of 2000 bits/s.

Mission Design

In this section, the principal factors affecting the orbit selection and trajectory design for the ISEE-3 mission are reviewed. The purpose here is to summarize the most important results of the pre-flight mission design studies. A more-complete discussion of these studies is given in Ref. 6.

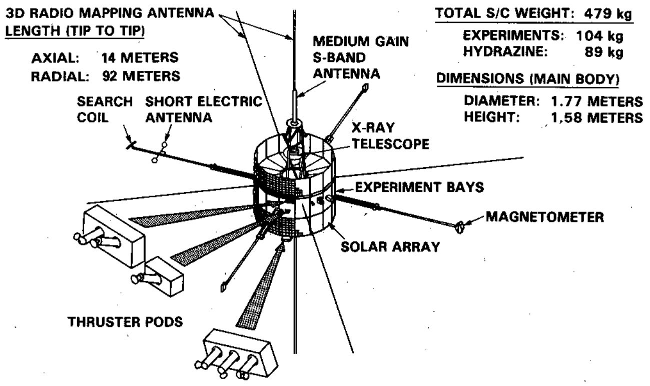

Fig. 1 ISEE-3 spacecraft.

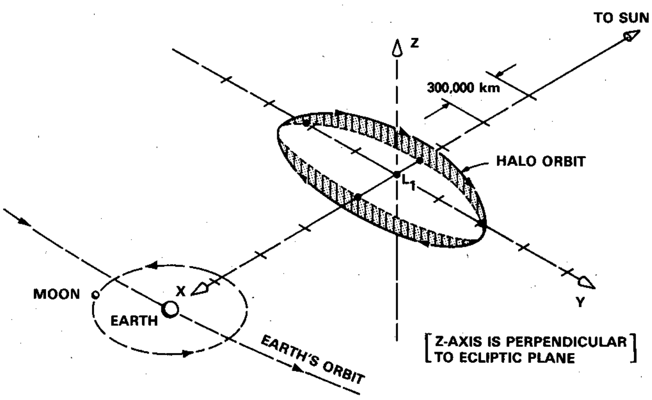

Fig. 2 ISEE-3 halo orbit around the sun-Earth L1 libration point.

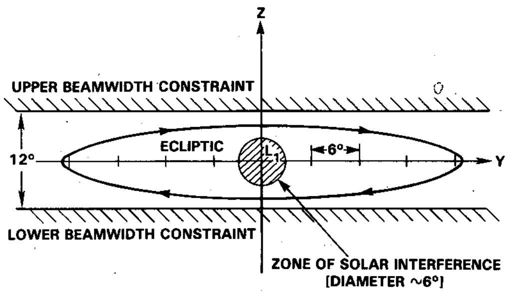

Fig. 3 Halo orbit as seen from Earth.

The ISEE-3 spacecraft could not be located too close to the sun-Earth L1 point because the sun is directly behind this point when viewed from the Earth, and downlink telemetry would have been overwhelmed by the intense solar noise background. Although the RF sun at S-band subtends an angle of only 0.75 deg, solar noise inputs to the sidelobes of ground antennas will substantially increase the extent of the blackout zone. For a 26-m antenna at S-band, the interference zone subtends an angle of about 6 deg. To avoid this region, the ISEE-3 mission orbit was designed to pass slightly above and below the ecliptic plane as shown in Figs. 2 and 3. This orbit is known as a class-1 halo orbit.[7] It has a period of approximately six months and is inherently unstable. Notice that the narrow beamwidth of the spacecraft's medium-gain antenna places an upper bound on the out-of-plane amplitude of the halo orbit.

The maximum y-axis amplitude of the ISEE-3 halo orbit is almost 700,000 km. A smaller-amplitude periodic orbit around the libration point could have been used, but recurring out-of-plane maneuvers (ΔV cost ~ 20 m/s per year) would have been needed to satisfy the communications constraints. More importantly, a large-amplitude halo orbit was preferred because the ΔV requirement for orbit insertion is much smaller for a large-amplitude orbit. The total saving amounts to about 150 m/s.

A further reduction in the ΔV cost was obtained by using a looping transfer trajectory to the halo orbit as shown in Fig. 4. Depending on the launch date, the insertion ΔV for this so-called "slow" transfer is somewhere in the range of 40-60 m/s. This is about 100 m/s less than comparable costs for optimized "fast" transfers (flight times ~ 50 days). The launch energy required for transfer trajectories to the ISEE-3 halo orbit is only slightly greater than the energy needed to reach lunar distance. This energy requirement ($C_3 \sim -0.6$ km^2/s^2) is virtually independent of launch date.

The launch window for the ISEE-3 mission was dictated primarily by the moon's orbital position. This situation came about because the moon was needed for attitude determination during the early transfer phase. (The rapid alignment of the sun, spacecraft, and Earth limited the usefulness of the sun-Earth combination for attitude determination. However, the sun-spacecraft-Earth geometry was acceptable for the first four hours after injection, i.e., until $I+4$ h.) To satisfy the attitude accuracy requirement and to ensure that the moon's brightness would be great enough to trigger the panoramic attitude sensor, it was desirable to maintain a Sun-spacecraft-moon angle between 90 and 150 deg for a two-day interval following launch. Inspection of the orbital geometry in Fig. 4 shows that this constraint bounds the launch window to roughly four days for a waning moon and only two days for waxing moon.

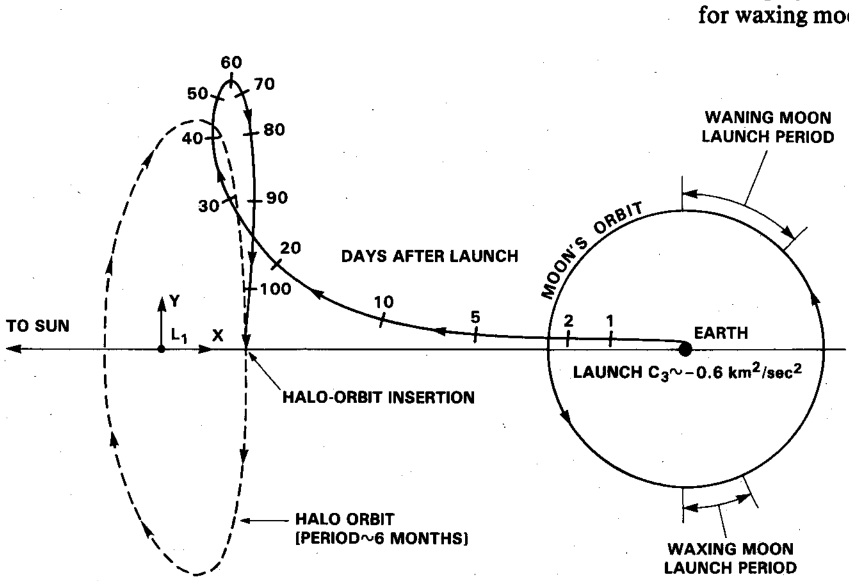

Fig. 4 Transfer trajectory to halo orbit.

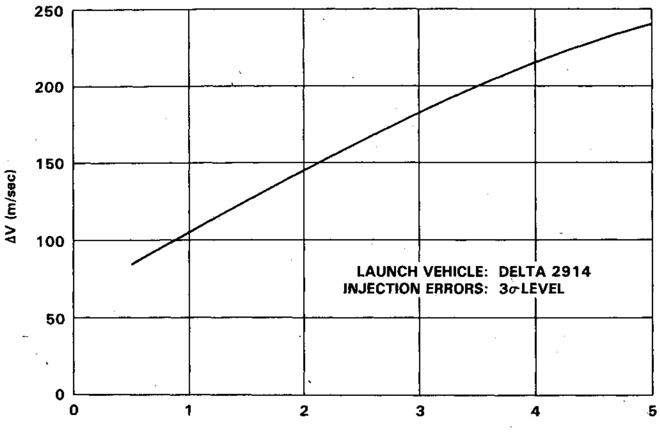

Fig. 5 ΔV requirement for first midcourse maneuver (prelaunch estimate).

A three-stage Delta-2914 rocket was selected for the ISEE-3 launch. This vehicle is relatively inexpensive and very reliable, but potential trajectory dispersions associated with the Delta-2914 are rather high. To correct these dispersions, an early midcourse maneuver was planned. Figure 5 shows the effect of 3σ-level errors on the magnitude of this maneuver. The maneuver was scheduled to begin about 18 hours after injection into the transfer trajectory ($I+18$ h). This scheme reduces the ΔV cost while allowing sufficient time for orbit determination, accurate attitude estimates, and maneuver computation.

For planning purposes, it was assumed that ISEE-3 would have a total ΔV capability of 400 m/s (as noted earlier, the actual capability at launch was ~430 m/s). Out of this total, 150 m/s was allocated for maneuvers in the halo orbit, leaving 250 m/s for midcourse corrections and the halo-orbit insertion maneuver. These allocations are not quite as generous as they appear to be, because spacecraft attitude constraints generally will not permit ISEE-3 to assume an optimal orientation for ΔV maneuvers (i.e., the required thrust direction is achieved by using combinations of radial and axial thrusters). Nevertheless, the ΔV budget was large enough to enable the ISEE-3 mission to recover from some rather serious underburns of the Delta rocket. An elaborate contingency strategy was devised for this circumstance and is fully described in Ref. 8.

Flight History

This section contains a summary of the ISEE-3 mission from launch to the completion of the first orbit around the sun-Earth L1 point. During this time, five orbit-adjust maneuvers were performed. Key factors involved in the planning and execution of these maneuvers are outlined here.

Early Transfer Phase

The transfer trajectory was targeted to the point on the halo orbit that is closest to Earth [i.e., below the ecliptic at $y=0$ (see Figs. 2 and 4)]. By varying the flight time to this point, it

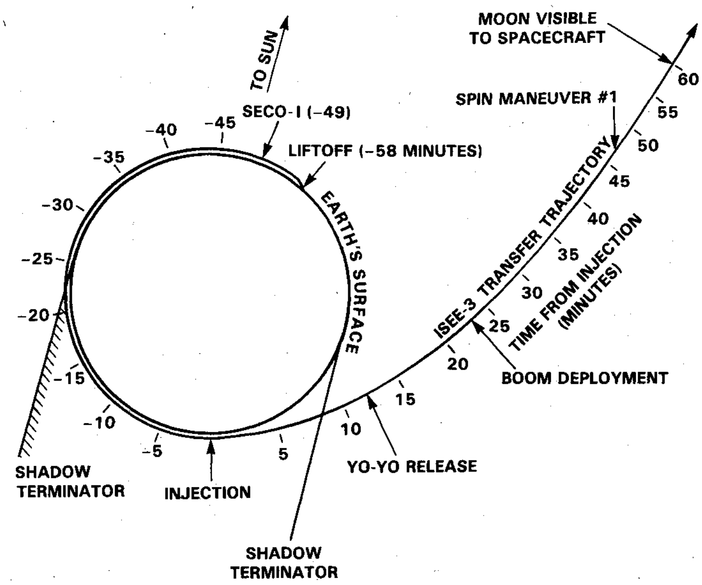

Fig. 6 Parking orbit and early transfer phase.

was possible to find a particular time that essentially eliminated the out-of-plane component of the insertion maneuvers, while the total ΔV cost for this maneuver stayed close to its minimum value. Some parameters of the planned ISEE-3 transfer trajectory are listed in Table 1.

ISEE-3 was originally scheduled to be launched during a waning moon period (July 23-26), but conflicts with other projects slipped the launch date to a waxing moon interval (Aug. 12-13). The spacecraft was launched by Delta rocket #144 at 11:12 local time on Aug. 12, 1978. The parking orbit and the first hour of the transfer trajectory are shown in Fig. 6. A long parking-orbit coast was employed because, for launches during the summer months, this choice leads to lower ΔV costs for halo-orbit insertion.[6] Separation from the launch vehicle's solid third stage occurred about 72 s after injection. At this time, the spacecraft was spinning at 46 rpm and its spin axis was located in the orbital plane of the transfer trajectory. After the release of the yo-yo despin mechanism, boom deployment, and a small spin adjustment using the hydrazine thrusters, the spacecraft was ready for its first attitude maneuver.

Because it was very likely that the ΔV vector for the first midcourse maneuver would be almost entirely in the transfer plane, the flight plan called for a 90-deg reorientation of the spin axis to orbit normal. The low inclination of the transfer plane to the ecliptic made it possible to carry out this attitude change without violating the spin-axis sun angle constraint of 90 ± 10 deg. Following the reorientation, the spacecraft was spun up to 38.7 rpm. This maneuver was needed to offset the large decrease in spin rate that would accompany the deployment of the spacecraft's radial antennae at $I+3$ days. The high spin rate was also expected to minimize attitude perturbations during the upcoming midcourse maneuver. A complete documentation of all ISEE-3 attitude and spin operations is presented in Ref. 9.

Early tracking estimates of the ISEE-3 trajectory revealed that the launch-vehicle performance was very close to nominal. The largest error occurred in the magnitude of the velocity at injection and was only 3 m/s below the desired value. As shown in Table 2, the size of the midcourse maneuver needed to correct the launch-vehicle errors was well under the 3σ prelaunch estimate. Notice that practically all of the ΔV correction was supplied by the radial thrusters. A radial ΔV of 16.98 m/s was planned for this maneuver, but subsequent tracking data showed that the actual ΔV input was about 4.5% high. Details of the implementation of ISEE-3 ΔV maneuvers and analyses of the ensuing execution errors can be found in Ref. 10.

Table 1 Nominal trajectory parameters

LAUNCH TIME (GMT):	15:12 AUG. 12, 1978
INJECTION TIME (GMT):	16:10 AUG. 12, 1978
INJECTION PARAMETERS:	
RADIUS:	6564.1 km
VELOCITY:	10.990 km/sec
SPIN-AXIS SUN ANGLE:	82.61°
FLIGHT TIME TO HALO ORBIT:	104.75 DAYS
ΔV REQUIREMENT FOR HALO-ORBIT INSERTION:	
IN-PLANE MAGNITUDE:	36.95 m/sec
OUT-OF-PLANE MAGNITUDE:	0.03 m/sec

Table 2 Maneuver summary (injection to $I+48$ h)

MANEUVER	START TIME	PURPOSE
SPIN #1	I+0.8 HOURS	SPIN UP TO 9.1 rpm
ATTITUDE #1	I+2.5 HOURS	ORIENT SPIN AXIS TO ORBIT NORMAL
SPIN #2	I+4.3 HOURS	SPIN UP TO 38.7 rpm
MIDCOURSE #1	I+18.1 HOURS	CORRECT LAUNCH-VEHICLE ERRORS RADIAL ΔV = 17.74 m/sec AXIAL ΔV = 0.11 m/sec
ATTITUDE #2	I+42.0 HOURS	ORIENT SPIN AXIS TO NORTH ECLIPTIC POLE

Table 3 Alternative maneuver sequences for midcourse #2 and halo insertion

MANEUVER TIME (DAYS AFTER INJECTION)		ΔV REQUIREMENT (m/sec)				
		MIDCOURSE #2		HALO INSERTION		
MIDCOURSE #2	HALO INSERTION	RADIAL	AXIAL	RADIAL	AXIAL	TOTAL
TOTAL ΔV MINIMIZED						
20	104.8	4.6	1.0	34.4	0.3	40.3
25	104.5	7.7	0.2	31.4	0.1	39.4
30	104.4	9.6	-0.1	31.8	–	41.5
CONSTRAINED SOLUTION WITH GOLDSTONE COVERAGE						
25	103.1	23.3	1.2	17.1	3.1	44.7

Approximately one day after the completion of the midcourse correction, the spacecraft spin axis was reoriented to the North Ecliptic Pole. It was planned to keep the spacecraft at this attitude for the remainder of the mission.

Halo-Orbit Insertion

Because of the sensitivity of the transfer trajectory to the execution errors in the first midcourse maneuver, the ISEE-3 trajectory was now displaced from its nominal halo insertion point by more than 200,000 km. Of course another midcourse correction could easily remedy this situation. However, the "best" strategy for this correction and the following halo-orbit insertion maneuver was not at all obvious. The difficulty was caused by the need to optimize multi-impulse trajectories in the restricted three-body problem. Although this is an interesting research topic, the in-flight analysis of the maneuver sequence was necessarily limited to the investigation of a few possibilities.

Preflight simulations had suggested that the interval from $I+20$ days to $I+30$ days might be a good time to schedule the second midcourse maneuver. A few of the options with midcourse #2 applied during this period are given in Table 3. In the first three cases the flight time was varied to minimize the total ΔV requirement. A midcourse at $I+25$ days gives the smallest total ΔV, but there is very little difference in the three solutions. Notice that the total ΔV cost is just a little more than the nominal ΔV requirement for halo insertion that is listed in Table 1.

Table 4 Maneuver requirements for shifted insertion dates (midcourse #2 at $I+25$ days)

	ΔV REQUIREMENT (m/sec)				
	MIDCOURSE #2		HALO INSERTION		
INSERTION DATE	RADIAL	AXIAL	RADIAL	AXIAL	TOTAL
NOVEMBER 18	26.1	0.3	20.5	2.3	49.2
NOVEMBER 19	25.5	0.4	19.7	2.4	48.0
NOVEMBER 20	25.0	0.5	19.0	2.5	47.0
NOVEMBER 21	24.5	0.7	18.3	2.7	46.2
NOVEMBER 22	23.9	1.0	17.7	2.8	45.4
NOVEMBER 23	23.3	1.2	17.1	3.1	44.7
NOVEMBER 24	22.8	1.6	16.6	3.5	44.5
NOVEMBER 25	22.2	2.1	16.1	4.0	44.4
NOVEMBER 26	21.5	2.9	15.7	4.9	45.0
NOVEMBER 27	20.9	4.5	15.3	6.7	47.4
NOVEMBER 28	20.4	9.1	14.9	12.1	56.5

It is convenient to divide the ΔV costs into radial and axial components. The radial component lies in the spacecraft spin plane, and the axial component is perpendicular to this plane. Because the spacecraft spin axis points to the North Ecliptic Pole, the radial component is also parallel to the ecliptic plane.

The minimum ΔV case was an acceptable solution. However, because there was no shortage of hydrazine fuel at this time, other options were examined. A particularly attractive alternative surfaced when it was discovered that, by shortening the flight time, the radial component of the halo-insertion maneuver could be closely algined with the Earth-spacecraft line. This tactic would make it possible to obtain accurate doppler tracking estimates of the radial component while the maneuver was still in progress. These estimates could then be used to update the insertion maneuver and reduce any execution errors. The parameters for this special case, which also included maneuver coverage by the Goldstone Tracking Facility, are listed in the bottom line of Table 3. Comparison with the minimum-ΔV case shows that the ΔV penalty for optimal doppler coverage is only 5.3 m/s. Furthermore, the ΔV split between the midcourse and insertion maneuvers has changed considerably. As a general rule, it is preferable to have a smaller insertion maneuver.

For the reasons cited above, the maneuver sequence with the favorable doppler coverage was adopted, and the second

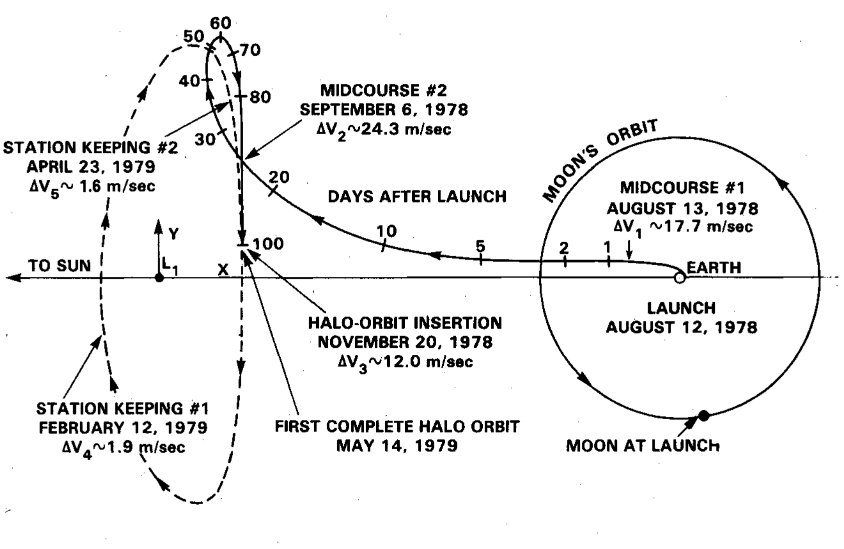

Fig. 7 ISEE-3 flight profile.

Table 5 Summary of ΔV maneuvers

MANEUVER	RADIAL (IN-PLANE) COMPONENT PLANNED (m/sec)	REALIZED (m/sec)	EXECUTION ERROR	AXIAL (OUT-OF-PLANE) COMPONENT (m/sec)
MIDCOURSE #1	16.98	17.74	4.5%	0.11
MIDCOURSE #2	25.02	24.29	2.9%	0.54
HALO INSERTION	11.73	11.73	–	2.30
STATIONKEEPING #1	2.01	1.94	3.5%	–
STATIONKEEPING #2	1.52	1.58	4.0%	–

midcourse maneuver was scheduled for $I+25$ days. However, before this maneuver was implemented, it was observed that the halo insertion would occur on Thanksgiving Day, Nov. 23. Fortunately it was found that the insertion date could be changed while retaining the prescribed doppler geometry. This was done by shifting the insertion point along the halo orbit by roughly one-day intervals. The ΔV requirements for insertion dates before and after the Thanksgiving Day date are listed in Table 4. To avoid weekends and holidays, it was decided to schedule the halo insertion maneuver for Monday, Nov. 20. This decision added another 2.3 m/s to the ΔV cost, but decreased the total flight time to ~ 100 days.

As shown in Fig. 7, the second midcourse maneuver took place at $I+25$ days on Sept. 6, 1978. The actual ΔV input of midcourse #2 was about 3% lower than planned, which caused the post-maneuver trajectory to drift away from the nominal halo insertion point by approximately 6000 km along the positive x-axis (i.e., $\Delta x \sim 6000$ km). Instead of correcting the error with another midcourse maneuver, it was decided to carry out the insertion maneuver as scheduled by placing the spacecraft into a neighboring libration-point orbit. Fortuitously, it was found that this scheme reduced the insertion ΔV cost by about 7 m/s.

The halo-orbit insertion maneuver was performed on Nov. 20, 1978. Real-time doppler tracking revealed a 3% overburn situation, so the maneuver was truncated to give the desired performance. Sometime later, ΔV estimates based on several weeks of tracking data showed that the execution error for the insertion maneuver was less than 1 cm/s. The final parameters for the insertion maneuver are listed in Table 5.

Stationkeeping Maneuvers

Orbits in the neighborhood of a collinear libration point are unstable. Therefore, stationkeeping maneuvers are needed to keep the ISEE-3 spacecraft in its halo orbit around the sun-Earth L1 point. These maneuvers are designed to drive the diverging ISEE-3 trajectory back to the vicinity of a precomputed reference halo orbit. The reference orbit was determined by using a semianalytical technique, and closely approximates the true halo orbit.[11] A precise reference orbit is not necessary, however, because the spacecraft is loosely controlled about the approximate halo path. The loose fit reduces the total ΔV cost of the stationkeeping corrections.[12]

As shown in Fig. 7, two stationkeeping maneuvers were used to complete the first halo orbit. The first maneuver took place 84 days after orbit insertion, and the second occurred after another 70 days had elapsed. Performance results for these maneuvers are contained in Table 5. Thus far, an axial maneuver has not been required for halo-orbit maintenance.

In planning the stationkeeping maneuvers, appropriate ΔV corrections were calculated for several points along the most recent ISEE-3 orbit. The selection of a maneuver date was based on a number of factors including fuel cost, duration of coast periods, and the tightness of the control about the reference orbit. Preliminary computations for planning maneuvers 1, 2, and 3 are given in Table 6. Due to the high accuracy of the halo insertion maneuver, the first stationkeeping correction could have been delayed until the 112-day point. Although the ISEE-3 trajectory was beginning to diverge from the reference orbit at this point, the displacement was still very small.

If the present trend continues, the ΔV requirement for stationkeeping operations should be somewhere in the range of 10 to 15 m/s per year. With its present ΔV capability, the ISEE-3 spacecraft could remain in the halo orbit beyond the year 2000.

Concluding Remarks

The first complete orbit around the sun-Earth L1 point was accomplished by the ISEE-3 spacecraft on May 14, 1979. Present plans call for ISEE-3 to remain in its halo-orbit location until the mid-1980's. This will enable ISEE-3 to monitor variations in upstream solar-wind conditions over a significant portion of the solar cycle.

A replacement spacecraft for ISEE-3 has been identified in the NASA Five-Year Plan. This spacecraft is called the Interplanetary Physics Laboratory (IPL) and is currently scheduled for a 1985 launch. After IPL has been placed in orbit around the L1 point, it is quite likely that ISEE-3 will be transferred to the vicinity of the sun-Earth L2 libration point which is located in the distant geomagnetic tail. Both spacecraft will participate in a broader NASA program that involves multiple spacecraft investigations of solar-terrestrial relationships.

Table 6 ΔV requirements for stationkeeping maneuvers (planning data based on preliminary post-maneuver orbit estimate)

	STATIONKEEPING #1		STATIONKEEPING #2		STATIONKEEPING #3	
DAYS SINCE LAST MANEUVER	DEVIATION FROM REFERENCE ORBIT (km)	ΔV COST OF STATIONKEEPING MANEUVER (m/sec)	DEVIATION FROM REFERENCE ORBIT (km)	ΔV COST OF STATIONKEEPING MANEUVER (m/sec)	DEVIATION FROM REFERENCE ORBIT (km)	ΔV COST OF STATIONKEEPING MANEUVER (m/sec)
28	980	2.6	1810	0.7	2870	1.0
42	5090	7.2	2710	3.0	3680	1.2
56	3620	4.2	3330	2.1	3890	2.1
70	1200	2.6	870	1.5	4500	3.2
84	630	1.9	1520	2.5	4730	4.9
98	1670	2.4	4560	9.5	6400	6.3
112	3420	3.3	10130	18.3	11130	10.9

Other near-term proposals for libration-point satellites include a pinhole-camera X-ray telescope and a lunar farside data-relay satellite. The pinhole satellite concept is being evaluated by the Marshall Space Flight Center, and the farside comsat has been mentioned in connection with Soviet plans for a lunar exploration program. Mission analysis studies for both projects will be aided by the ISEE-3 flight experience.

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