ORBITAL ACROBATICS IN THE SUN-EARTH-MOON SYSTEM

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ABSTRACT

This paper presents an overview of some unconventional trajectory techniques for space missions in the Sun-Earth-Moon system. Topics covered include libration-point orbits, gravity-assist maneuvers, and Earth-return trajectories. The ISEE-3/ICE flight experience is used to illustrate the utility of a special class of libration-point orbits called "halo-orbits." Five lunar gravity-assist maneuvers used by the ISEE-3/ICE spacecraft are also discussed. Of particular interest is the final lunar swingby that sent the spacecraft into a heliocentric trajectory that would eventually intercept Comet Giacobini-Zinner. As an example of the Earth-return trajectory concept, a proposed mission that includes flybys of three comets and two asteroids is described.

Keywords: Libration-Point Orbit, Gravity-Assist Maneuver, Earth-Return Trajectory

1. INTRODUCTION

The motion of a spacecraft in the complex gravity field of the Sun-Earth-Moon system is a subject that has attracted a great deal of interest. Celestial mechanicians have shown that a large variety of unusual orbits exist in this rather special restricted four-body problem. Of greater importance, however, is the fact that the gravitational fields of the Sun, Earth, and Moon can be used to form trajectories that have considerable practical value. For the past eight years, a spacecraft originally named the International Sun-Earth Explorer-3 (ISEE-3) has used several of these trajectories to accomplish a number of important firsts in space science. These scientific achievements have included continuous monitoring of solar wind conditions upstream from the Earth, exploration of the distant geomagnetic tail, and an encounter with a comet.

Some of the more important orbital concepts from the standpoint of mission applications are described in this paper. The discussion highlights flight results from the ISEE-3 mission, but future mission possibilities are also reviewed.

2. LIBRATION-POINT ORBITS

In 1772, the French mathematician J. Lagrange demonstrated that there are five positions of equilibrium in a rotating two-body gravity field. Three of these "libration points" are situated on a line joining the two attracting bodies, and the other two form equilateral triangles with these bodies. As shown in Figure 1, a total of seven libration points are located in the Earth's neighborhood. Five of them are members of the Earth-Moon system and two belong to the Sun-Earth system. In the reference frame used here, the Sun-Earth line is fixed and the Earth-Moon configuration rotates around the Earth. Although the collinear points are unstable, very little propulsion is needed to keep a spacecraft at or near one of these points for an extended period of time. A variety of stationkeeping techniques for spacecraft in the vicinity of a collinear libration point are discussed in Reference 1.

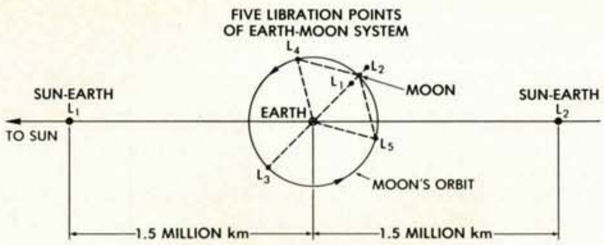

Figure 1. Libration Points in the Vicinity of the Earth

The L_1 and L_2 points of the Earth-Moon and Sun-Earth systems are noteworthy because their unique positions are advantageous for several important applications in astronautics (References 1, 2). For instance, the Sun-Earth L_1 point is an ideal location to continuously monitor the interplanetary environment upstream from the Earth. A suitably instrumented spacecraft placed in the vicinity of this point can provide data on the solar wind about an hour before it reaches the Earth's magnetosphere. However, the spacecraft cannot be stationed too close to the L_1 point because the Sun is directly behind this point when viewed from the Earth. This alignment is a problem because the intense solar noise background will severely degrade downlink

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telemetry. To avoid the zone of solar interference, a "halo orbit" is utilized (References 3, 4, 5). This orbit has a period of approximately six months and passes slightly above and below the ecliptic plane, as shown in Figures 2 and 3. It is inherently unstable, and occasional stationkeeping maneuvers are required to keep the spacecraft close to the halo path. Fortunately, these maneuvers are only needed about once every three months and their associated ΔV costs are quite small ($\sim$ 10 m/sec per year).

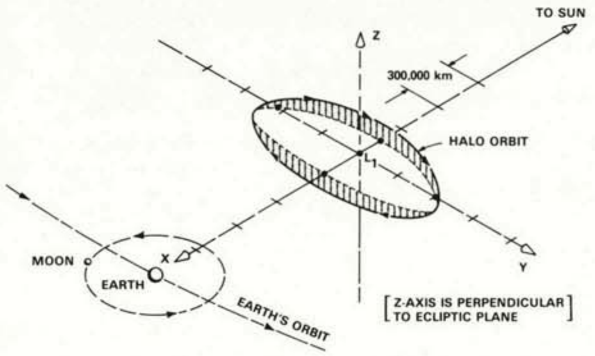


Figure 2. Halo Orbit Around the Sun-Earth L_1 Libration Point

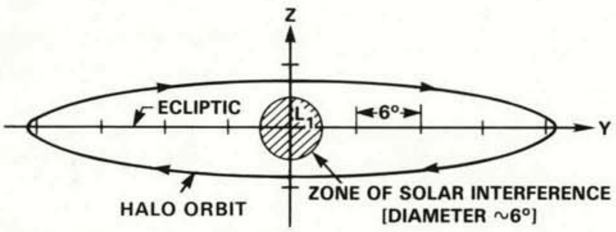


Figure 3. Halo Orbit As Seen from Earth

On August 12, 1978, the ISEE-3 spacecraft was launched toward the Sun-Earth L_1 point. One hundred days later, on November 20, 1978, ISEE-3 was inserted into a halo orbit around the L_1 point, thus becoming the first libration-point satellite. The unusual transfer trajectory used to reach the halo orbit is illustrated in Figure 4. Three orbit maneuvers with a total ΔV expenditure of 54 m/sec were needed to achieve the desired mission orbit.

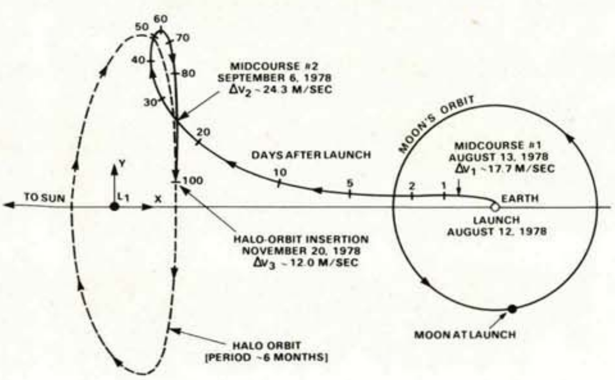


Figure 4. ISEE-3 Transfer Trajectory to Halo Orbit

During the halo-orbit phase (November 1978 to June 1982), fifteen stationkeeping maneuvers with a ΔV cost of 30 m/sec were required for orbit maintenance. Further details of the ISEE-3 flight performance from 1978 to 1982 can be found in References 6 and 7.

The success of the ISEE-3 libration-point mission has spawned a number of follow-on proposals. As part of its contribution to the International Solar Terrestrial Physics (ISTP) Program, the European Space Agency (ESA) is planning to station a solar observatory called SOHO in an ISEE-3 type halo orbit in 1994 (Reference 8). A libration-point orbit around the Sun-Earth L_2 point has been mentioned as a possible location for a Prognoz mission in the early 1990s. Finally, it is expected that another ISTP spacecraft called Wind (to be provided by NASA) will occupy a Sun-Earth L_1 halo orbit where it will serve as a replacement for ISEE-3.

Halo orbits may also play an important role in future lunar operations. In a concept originally proposed twenty years ago (References 9, 10), it was shown that a data-relay satellite located in a halo orbit around the Earth-Moon L_2 point could provide an uninterrupted communications link between the Earth and the far side of the Moon (see Figure 5). This capability will be essential for a manned base on the Moon's farside. In the more immediate future, a farside communications satellite could facilitate the navigation and control of unmanned lunar vehicles in scientifically interesting areas such as the Tsiolkovsky crater.

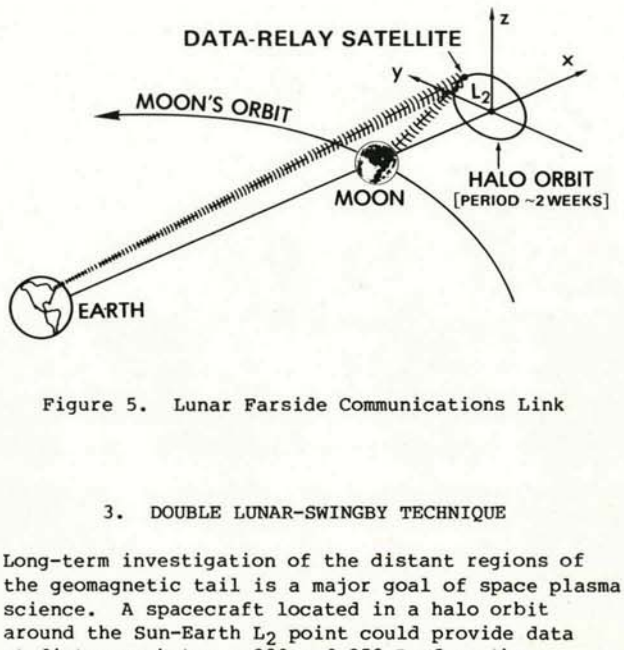


Figure 5. Lunar Farside Communications Link

3. DOUBLE LUNAR-SWINGBY TECHNIQUE

Long-term investigation of the distant regions of the geomagnetic tail is a major goal of space plasma science. A spacecraft located in a halo orbit around the Sun-Earth L_2 point could provide data at distances between 220 and 250 R_E from the Earth. However, a trajectory that allows repeated longitudinal scans of the magnetotail between 60 and 250 R_E is preferred. Cross-sectional coverage at various downstream distances is also favored. Conventional Earth orbits will not satisfy these specifications because the apsidal line of an elliptical Earth orbit is essentially fixed in inertial space and appears to rotate about one degree per day with respect to the Sun-Earth line. Therefore, to obtain the desired coverage, it will be necessary to control the rotation of the apsidal line to maintain the apogee segment in the tail region. (The geomagnetic-tail axis is always aligned within a few degrees of the Sun-Earth line.) The required orbital rotation could be achieved by employing propulsive maneuvers, but the ΔV cost would be about 400 m/sec per month! This reality led to the development of a technique that uses a series of lunar gravity-assist maneuvers for orbital control.

The basic procedure is depicted in the top section of Figure 6. Assume that a spacecraft is initially located near the apogee of the smaller orbit at A_1. After its next perigee passage, the natural orbital precession will position the spacecraft for a trailing-edge swingby of the Moon at S_1. The swingby maneuver at S_1 will then rotate the line of apsides back to the Sun-Earth line and will also raise the apogee distance to A_2. A leading-edge lunar swingby at S_2, after the Moon has completed one full orbit plus the S_1S_2 segment, will return the spacecraft to its original orbit. This sequence of orbit pairs could be repeated indefinitely or, by slightly changing the swingby conditions at S_1 and S_2, the spacecraft could be placed into different periodic orbits as shown in the other sections of Figure 6. The three classes of orbits illustrated in Figure 6 represent just a few of the many solutions that can be formed with the double lunar-swingby technique. Additional solutions can be obtained by increasing the time interval as well as the number of orbital loops in the inner trajectory segment ($S_2A_1S_1$). Four- and five-month outer loops ($S_1A_2S_2$) are also possible. Details of these solutions are given in References 11 and 12.

An interesting property of the trajectories in Figure 6 is that they are doubly periodic. This fact is exhibited in Figure 7 where the one-month class orbit is plotted with respect to a fixed Earth-Moon line. Notice that the Moon does not occult the trajectory from an observer on the Earth.

The periodic orbits in Figures 6 and 7 have been generated with a simplified patched-conic dynamical model. When a more realistic model that includes the effects of solar perturbations and the Moon's orbital eccentricity is used, the symmetrical shapes are distorted and the apogee distances are changed. Nevertheless, the average apsidal rotation can be kept at the required Sun-synchronous rate.

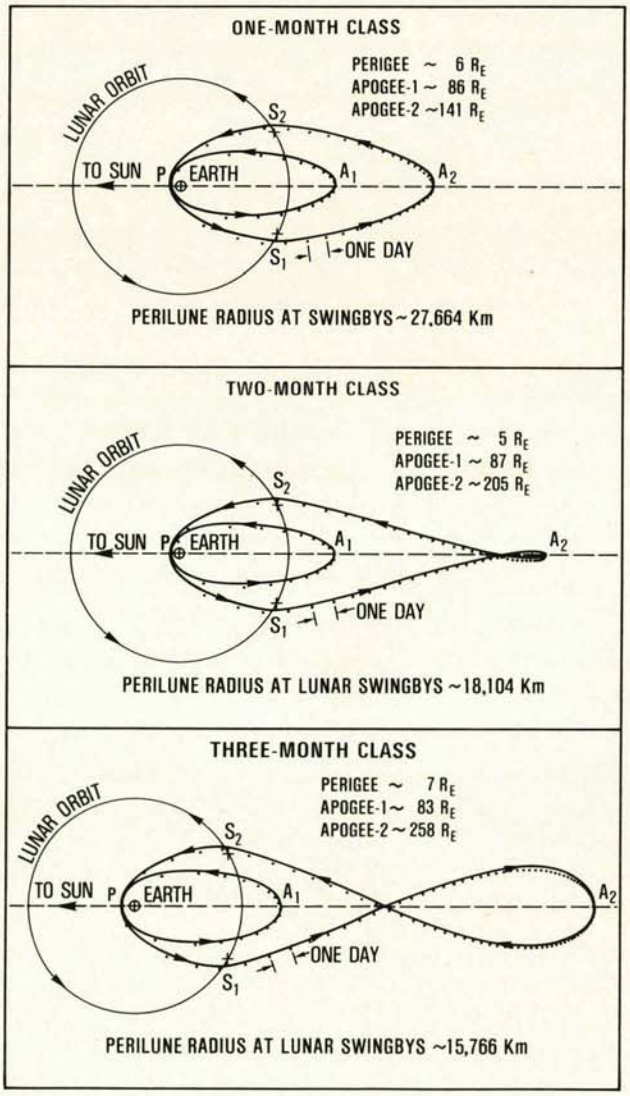


Figure 6. Sun-Synchronous Periodic Orbits Using Double Lunar-Swingby Technique

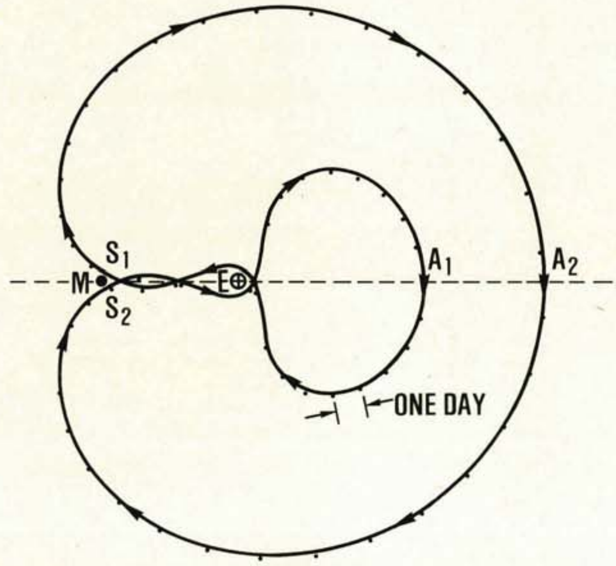


Figure 7. Double Lunar-Swingby Trajectory in Earth-Moon Reference Frame (One-Month Class)

The first flight application of the double lunar-swingby concept came in 1983 when it was used by the ISEE-3 spacecraft (Reference 13). Figures 8 and 9 show the ISEE-3 trajectory from June 1982 to September 1983. A ΔV maneuver of only 4.5 m/sec was used to initiate the transfer from the halo orbit to the geotail. Except for small navigational corrections, this maneuver was sufficient to complete the flight path shown in Figure 8. However, a large out-of-plane maneuver was needed at A_1 to change the inclination of the ISEE-3 trajectory and set up the lunar encounter at S_1. As shown in Figure 9, swingby maneuvers at S_1 and S_2 produced a five-month geotail traverse that extended out to an apogee distance of 237 R_E. The propulsive maneuver on June 1, 1983 was needed to satisfy targeting and phasing constraints for a later gravity-assist maneuver in December 1983 (discussed in the following section).

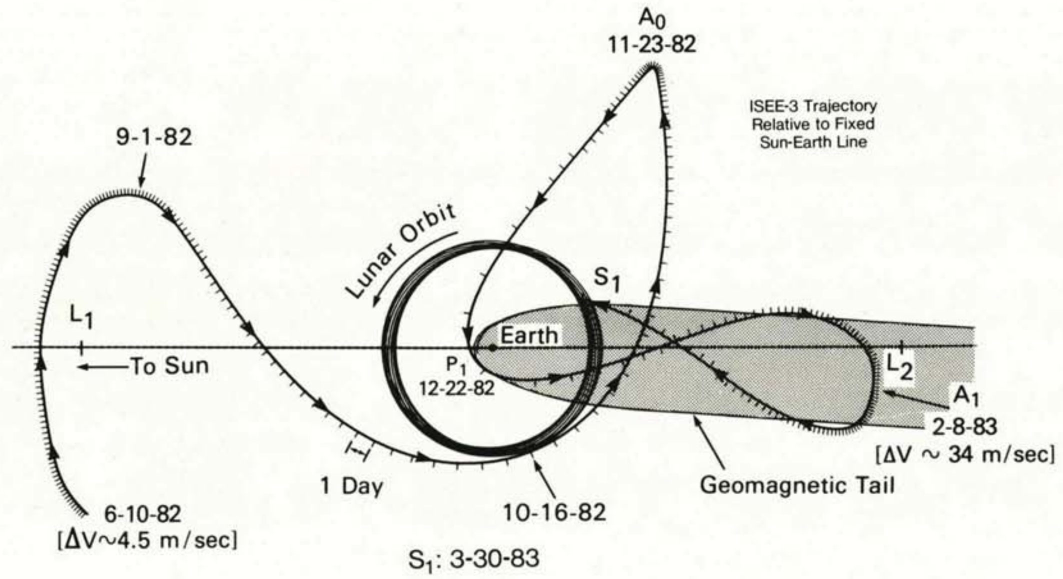


Figure 8. ISEE-3 Transfer from L_1 Halo Orbit to Geomagnetic Tail

Three projects scheduled for implementation in the 1990s are planning to use double lunar-swingby trajectory profiles. One of them is a Japanese mission called MUSES-A that will be launched sometime in 1990 (Reference 14). A second Japanese spacecraft called Geotail will follow in 1992. The third example, NASA's Wind spacecraft (mentioned in the previous section), will reside in a sunward double lunar-swingby orbit for about two years before it is subsequently placed into a Sun-Earth L_1 halo orbit.

4. UTILIZATION OF LUNAR GRAVITY-ASSIST MANEUVERS FOR EARTH ORBITAL ESCAPE

In addition to their effectiveness for orbital control, lunar-swingby maneuvers can be used to supply the energy increase needed to escape from the Earth-Moon system and enter a heliocentric orbit (References 15, 16). As early as January 1959, a lunar flyby was used to catapult the Soviet space probe Luna-1 into a solar orbit. However, using the Moon to send a spacecraft toward a specific target in heliocentric orbit is a more sophisticated task. At the lunar swingby, the aim point in the impact plane, the relative velocity vector between the spacecraft and the Moon, and the timing of the lunar encounter are precisely related. This type of lunar-swingby maneuver was performed for the first time on December 22, 1983, when the ISEE-3 spacecraft was targeted for an encounter with Comet Giacobini-Zinner.

The ISEE-3 escape trajectory is shown in Figure 10. This is a continuation of the profile given in Figures 8 and 9. A total of five lunar swingbys and fifteen propulsive maneuvers were needed to carry out the transfer from the halo orbit to the escape trajectory. The four planned maneuvers (shown in Figures 8, 9, and 10) accounted for 72 m/sec of the total ΔV cost of 77 m/sec. Maneuver execution errors and orbit determination uncertainties were responsible for the remaining 5 m/sec. Flyby distances for swingbys S_1 to S_4 were on the order of 20,000 km from the Moon. The S_5 swingby was much closer, passing the lunar surface at an altitude of only 120 km. Immediately following the successful S_5 escape maneuver, NASA announced that it had decided to change the name of ISEE-3 to the International Cometary Explorer (ICE).

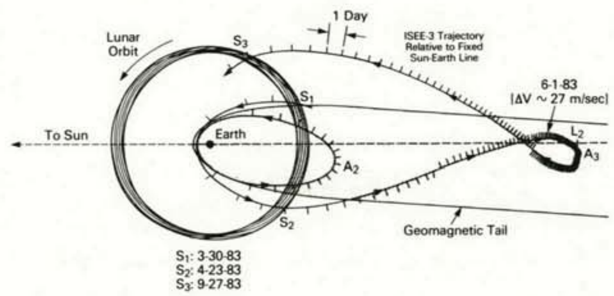


Figure 9. ISEE-3 Five-Month Geotail Excursion

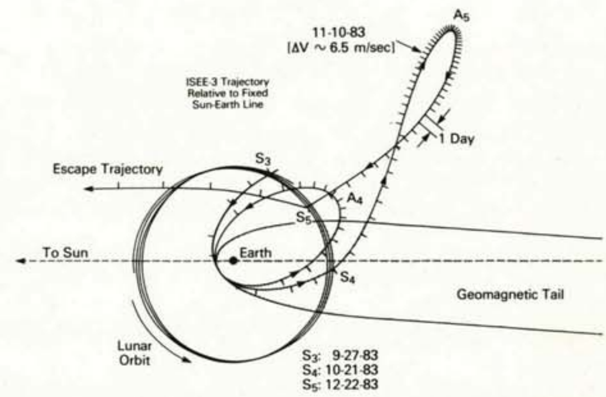
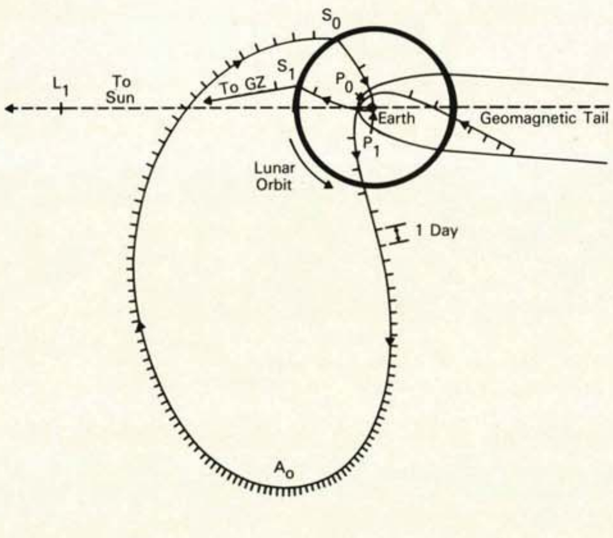


Figure 10. ISEE-3 Escape Trajectory

Even though ICE passed extremely close to the Moon, it could only achieve a hyperbolic excess velocity of 1.67 km/sec ($C_3 \sim 2.8$ km^2/sec^2). This value is close to the practical limit for a single escape maneuver. However, by using a special double lunar gravity-assist technique (Reference 17), it is possible to reach a higher energy level. The use of the double gravity-assist technique was considered in formulating a contingency plan for the ISEE-3/ICE departure. This alternative escape trajectory is shown in Figure 11. Notice that, in this example, a hyperbolic excess velocity of 2.12 km/sec ($C_3 \sim 4.5$ km^2/sec^2) is produced.

Recently, it has been shown that repeated energy-increasing lunar swingbys can be carried out by using one-year Earth-return trajectories between the swingby maneuvers (Reference 18). With this method it should be possible to obtain a hyperbolic excess velocity of 3 km/sec with four swingby maneuvers.



EVENT	1984 DATE	DISTANCE	C_3 (km/sec)^2
P_0	Sept. 5	9.4 R_e	
A_0	Oct. 24	307 R_e	
S_0	Dec. 17	2283 km	−0.46
P_1	Dec. 19	1.8 R_e	+1.20
S_1	Dec. 21	1800 km	+4.49

Figure 11. Double Lunar Gravity-Assist to Comet Giacobini-Zinner

5. EARTH-RETURN TRAJECTORIES

The use of an Earth-return trajectory in connection with a flyby mission to a comet can significantly augment the overall scientific value of the mission. By employing a series of Earth-swingby maneuvers, it is usually possible to construct a trajectory that will encounter additional comets and/or asteroids. The idea of using Earth-return trajectories for cometary intercept missions was originally proposed in the 1970s (References 19, 20, 21). Sometime later it was applied to the 1985-86 Halley opportunity (Reference 22). Figure 12, taken from Reference 22, depicts a five-year Earth-return trajectory with a Halley encounter in March 1986. A very similar flight path will be used by the Giotto spacecraft to return to the Earth in July 1990 when it will utilize an Earth gravity-assist maneuver to reach Comet Grigg-Skjellerup in July 1992 (Reference 23).*

Perhaps the most ambitious proposal thus far with regard to Earth-return trajectories is a mission that would carry out flybys of three comets and two asteroids over a 12-year span. A summary of the nominal flight profile for this "grand tour" of comets and asteroids is given in Table 1.

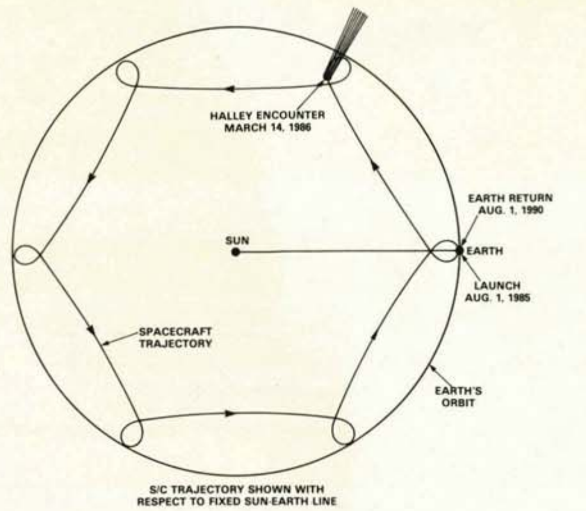


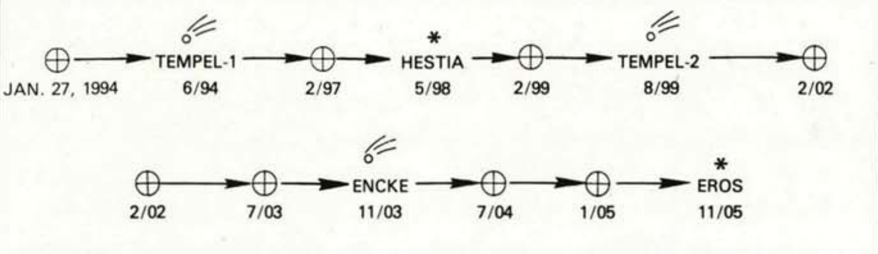
Figure 12. Halley Encounter with Five-Year Earth-Return Trajectory

Except for the Encke encounter, the flyby speeds are quite low. Notice that six Earth-swingby maneuvers and four propulsive maneuvers (total $\Delta V \sim 1.05$ km/sec) are needed to complete the trajectory sequence. The propulsive maneuvers are used to adjust the dates of the Earth swingbys and to increase the orbital energy level. At the final swingby in January 2005, the C_3 value is about 37 km^2/sec^2, which is considerably higher than the launch C_3 of 15 km^2/sec^2.

There are essentially two stages of the small-body tour: a primary phase to be completed in 1999 with the Tempel-2 intercept and an extended mission phase that adds the Encke and Eros encounters. The primary phase is relatively straightforward and requires only two Earth-swingby maneuvers.

Trajectory plots for the more complicated extended phase are shown in Figures 13 and 14. Figure 13 begins with a kidney-shaped, 1-1/2-year loop that is located in the ecliptic plane. A powered swingby in July 2003 leads to a one-year return trajectory that includes the Encke flyby. This is followed by an Earth swingby that transforms the trajectory into an Earth-like heliocentric orbit ($a = 1$ AU, $e = 0.017$) with an inclination of 11.6 degrees. Relative to a fixed Sun-Earth line, this orbit appears to oscillate about the Earth in a line perpendicular to the ecliptic. A final Earth swingby in January 2005 targets the trajectory to an encounter with Eros in November 2005. The fortuitous alignment of the Earth and Eros in 2005 results in a flyby speed at Eros of only 1.2 km/sec.

*On September 17, 1985, the Giotto project scientist, R. Reinhard, asked R. Farquhar to investigate post-encounter options for Giotto. In response to this request, it was suggested that Giotto should use a ΔV maneuver after the Halley encounter to place it into a five-year Earth-return trajectory. The authors are pleased that the Giotto project has decided to follow this recommendation.



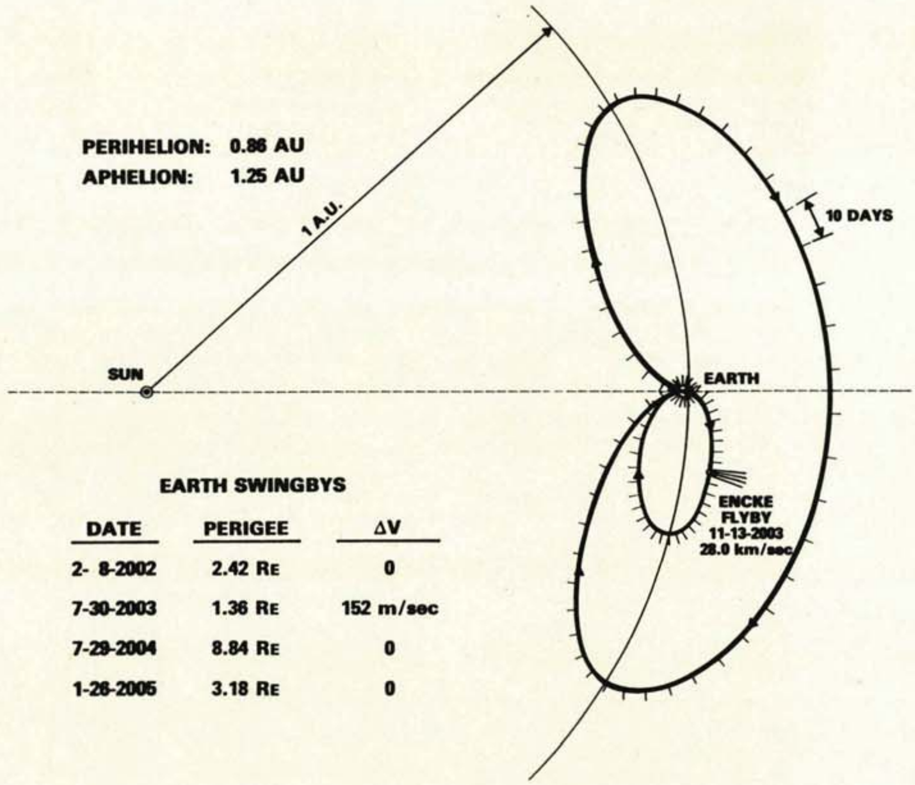
EARTH-SWINGBY MANEUVERS			
SWINGBY DATE	PERIGEE [EARTH RADII]	BEND ANGLE [DEGREES]	HELIOCENTRIC INCLINATION AFTER SWINGBY [DEGREES]
FEB. 13, 1997	2.48	58.4	2.0
FEB. 8, 1999	4.19	41.6	5.6
FEB. 8, 2002	2.42	53.5	0.0
JULY 29, 2003	1.36	71.1	10.9
JULY 29, 2004	8.84	19.3	11.6
JAN. 26, 2005	3.18	40.4	8.5

PROPULSIVE MANEUVERS		
MANEUVER DATE	ΔV (m/sec)	
JULY 4, 1994	350	LAUNCH C_3: 15.2 Km^2/sec^2
AUG. 18, 1998	79	
AUG. 11, 2000	472	TOTAL ΔV: 1053 m/sec
JULY 29, 2003	152	

SMALL-BODY ENCOUNTERS				
ENCOUNTER DATE	SUN DISTANCE [AU]	EARTH DISTANCE [AU]	PHASE ANGLE [DEGREES]	FLYBY SPEED [Km/sec]
TEMPEL-1: 6-24-94	1.50	0.81	53.6	11.2
HESTIA: 5-10-98	2.10	2.68	115.9	6.5
TEMPEL-2: 8-27-99	1.49	0.77	63.6	12.5
ENCKE: 11-13-03	1.06	0.26	13.1	28.0
EROS: 11-3-05	1.78	2.23	89.7	1.2

Table 1. Small-Body Tour 1994-2005

The multibody trajectory described here is a key element of a proposed dual-spacecraft mission that would collect dust samples from Comets Tempel-1 and Tempel-2 in addition to performing the five small-body flybys. A general discussion of this mission is presented in Reference 24. Further details of the flight mechanics aspects are given in Reference 25.



DATE	PERIGEE	ΔV
2- 8-2002	2.42 R_E	0
7-30-2003	1.36 R_E	152 m/sec
7-29-2004	8.84 R_E	0
1-26-2005	3.18 R_E	0

Figure 13. Main Spacecraft Trajectory, 2002 to 2005, Relative to Fixed Sun-Earth Line Earth → Earth → Encke → Earth → Earth

6. CONCLUDING REMARKS

It has been shown that libration-point orbits, Earth and lunar gravity-assist maneuvers, and Earth-return trajectories can provide a large degree of orbital flexibility for spacecraft in the Sun-Earth-Moon system. The practicality of libration-point orbits and lunar-swingby maneuvers has been fully demonstrated by the flight of ISEE-3/ICE. Additional flight experience with these trajectory techniques will be gained from at least five more missions scheduled for launch in the 1990s.

It is anticipated that Earth-swingby maneuvers will also find wide application in the 1990s. The planned use by the Giotto spacecraft has already been mentioned. Other spacecraft that may use Earth-swingby maneuvers in this time frame include Galileo and the two Japanese comet probes, Suisei and Sakigake. Finally, it is hoped that a spacecraft for the proposed small-body grand tour will be available for launch in 1994.

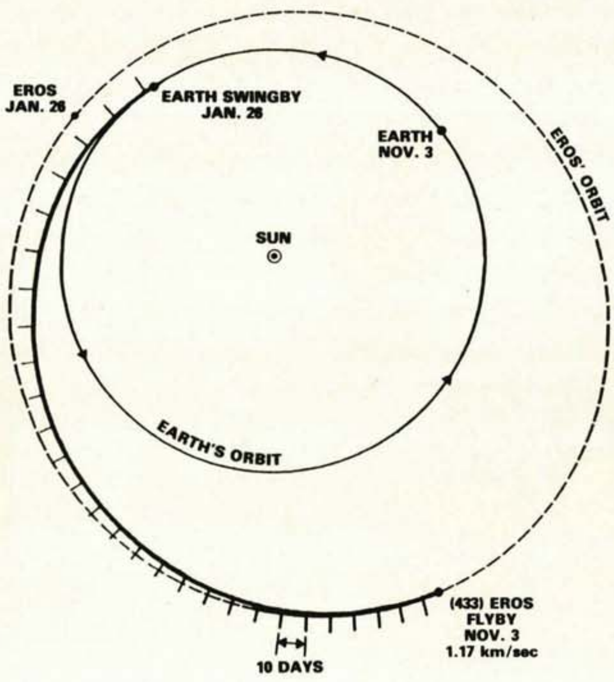


Figure 14. Earth to Eros in 2005

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