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Utilization of Multi-Body Trajectories in the Sun-Earth-Moon System

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ABSTRACT

An overview of three uncommon trajectory concepts for space missions in the Sun-Earth-Moon System is presented. One concept uses a special class of libration-point orbits called "halo orbits." It is shown that members of this orbit family are advantageous for monitoring the solar wind input to the Earth's magnetosphere, and could also be used to establish a continuous communications link between the Earth and the far side of the Moon. The second concept employs pretzel-like trajectories to explore the Earth's geomagnetic tail. These trajectories are formed by using the Moon to carry out a prescribed sequence of gravity-assist maneuvers. Finally, there is the "boomerang" trajectory technique for multiple-encounter missions to comets and asteroids. In this plan, Earth-swingby maneuvers are used to retarget the original spacecraft trajectory. The boomerang method could be used to produce a triple-encounter sequence which includes flybys of comets Halley and Tempel-2 as well as the asteroid Geographos.

UTILIZATION OF MULTI-BODY TRAJECTORIES IN THE SUN-EARTH-MOON SYSTEM

INTRODUCTION

The motion of a spacecraft in the complex gravity field of the Sun-Earth-Moon System is a topic 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 realization that the Sun, Earth, and Moon can be used to form trajectories that have considerable practical value. The recognition of this potential has led to the development of some extremely useful multi-body trajectory concepts for space exploration. Three of these concepts are described in this paper.

The three trajectory techniques are quite different. One idea involves the use of periodic orbits around certain equilibrium points in the Sun-Earth-Moon System. Another scheme uses multiple swingbys of the Moon for orbital control. The third method employs Earth-swingby maneuvers to produce a desired flight profile.

Important mission applications are associated with all of the aforementioned trajectory concepts. One of these missions was initiated in 1978 and is still active. It is expected that other missions will be implemented in the near future, perhaps in the 1980's. Details of some notable mission applications are included in the discussion that follows.

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, while the other two form equilateral triangles with these bodies. Although the three collinear points are unstable and the two triangular points are only quasi-stable, very little propulsion is needed to keep a spacecraft at or near one of these points for an extended period of time.

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.

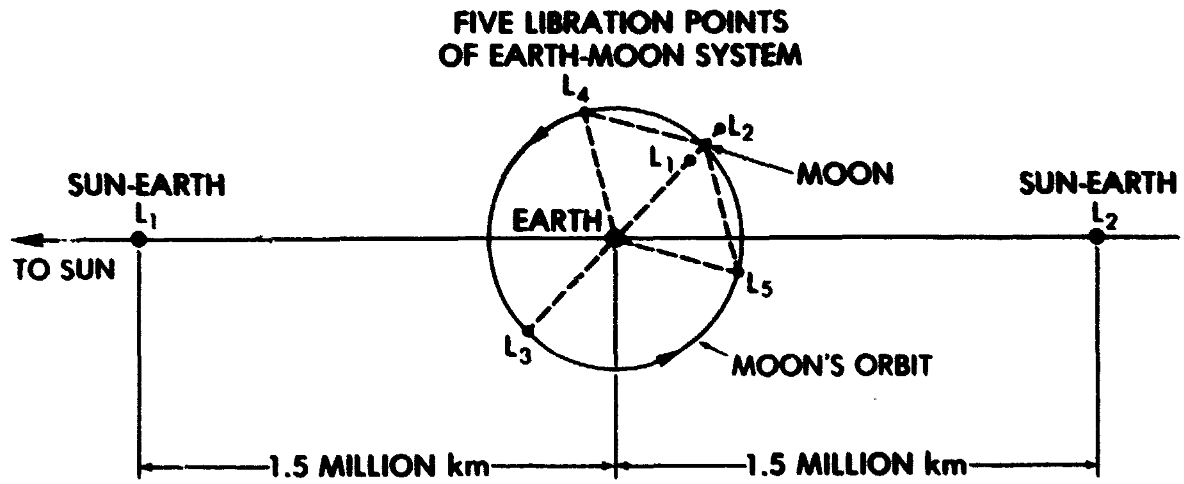


Figure 1. Libration Points in the Vicinity of the Earth

The Sun-Earth L_1 point is especially significant because it is an ideal location for a scientific spacecraft. At this site, it is possible to continuously monitor the state of the interplanetary medium upstream from the Earth. This input function is very important for many magnetospheric experiments. It should be noted that the characteristics of the solar wind and other solar-induced phenomena would be available about an hour before they reach the near-Earth space environment.

However, there is a formidable practical difficulty associated with the placement of a spacecraft at the upstream libration point. Because the Sun is directly behind this point when viewed from the Earth, downlink telemetry from the L_1 region would be swamped by the intense solar noise background. Although the Sun is a fairly small object as seen from the Earth (angular size ~0.5 degrees), the zone of solar interference subtends an angle of almost six degrees. Fortunately, it is possible to overcome this problem by using a special periodic orbit known as a "halo Orbit." This orbit is illustrated in Figures 2 and 3. Notice that the halo orbit avoids the interference zone by passing slightly above and below the Ecliptic plane.

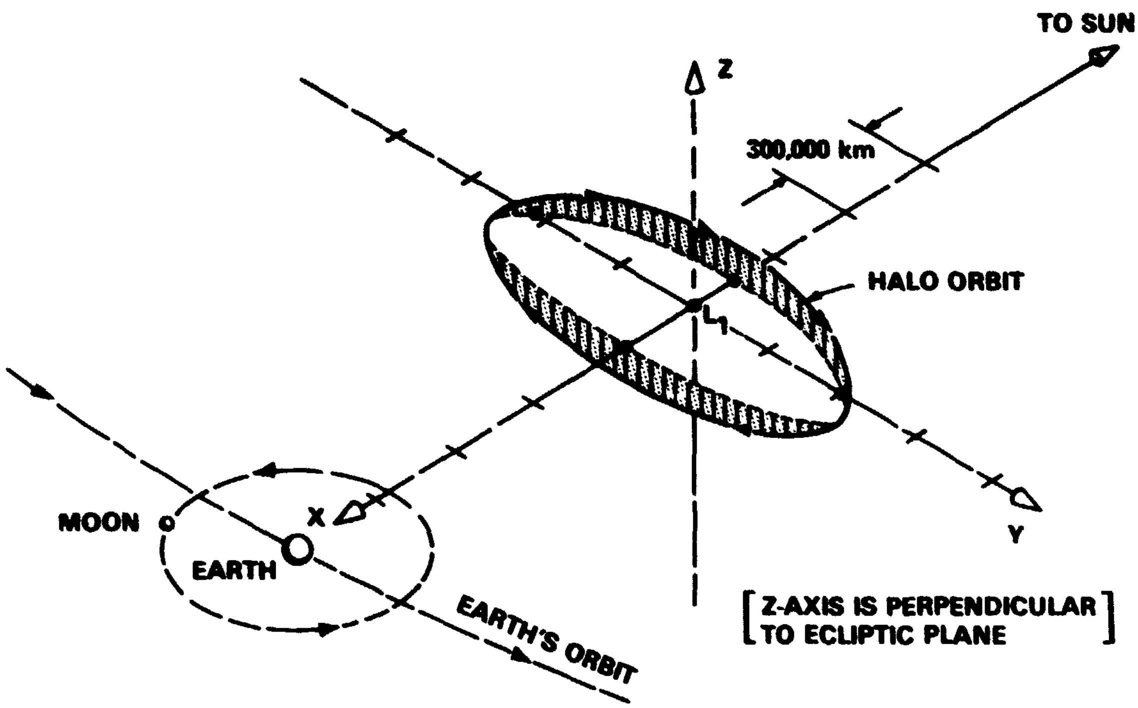


Figure 2. Halo Orbit Around the Sun-Earth L_1 Libration Point

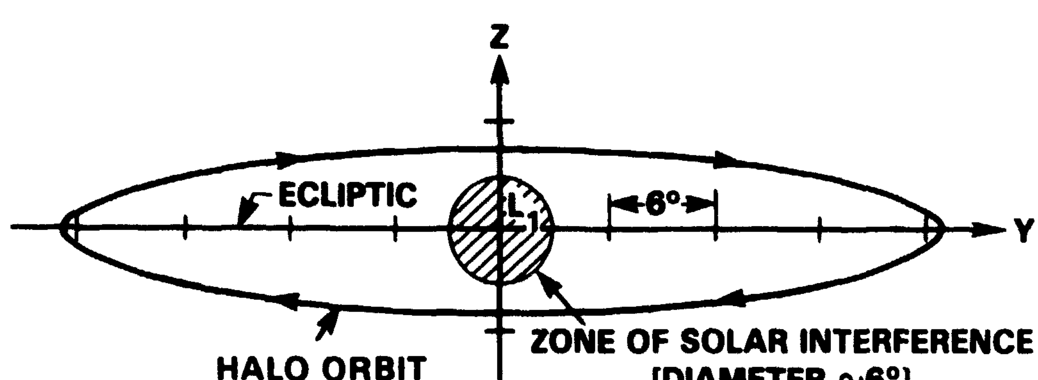


Figure 3. Halo Orbit as Seen From Earth

In 1972, NASA decided to incorporate the halo-orbit concept in a flight project. Subsequently, on August 12, 1978, a Delta rocket launched a spacecraft called International Sun-Earth Explorer-3 (ISEE-3) towards the L_1 point. The ISEE-3 spacecraft entered the halo orbit on November 20, 1978, thus becoming the first man-made libration-point satellite. Figure 4 shows the looping transfer trajectory that was used to arrive at the insertion point. During its 100-day transfer, ISEE-3 lingered in a region where the gravitational effects of the Sun and the Earth are comparable, which brought about some interesting tradeoffs concerning the maneuver strategy for halo insertion. Details of the pre-flight definition and early flight history for the ISEE-3 mission can be found in References 1 and 2.

Because of the inherent instability of the halo orbit, stationkeeping maneuvers are required for orbital maintenance of ISEE-3. Nominally, these maneuvers are needed about once every three months, and the average ΔV cost is under 15 m/sec per year. With its present ΔV capacity, ISEE-3 could stay in the halo orbit beyond the year 2000.

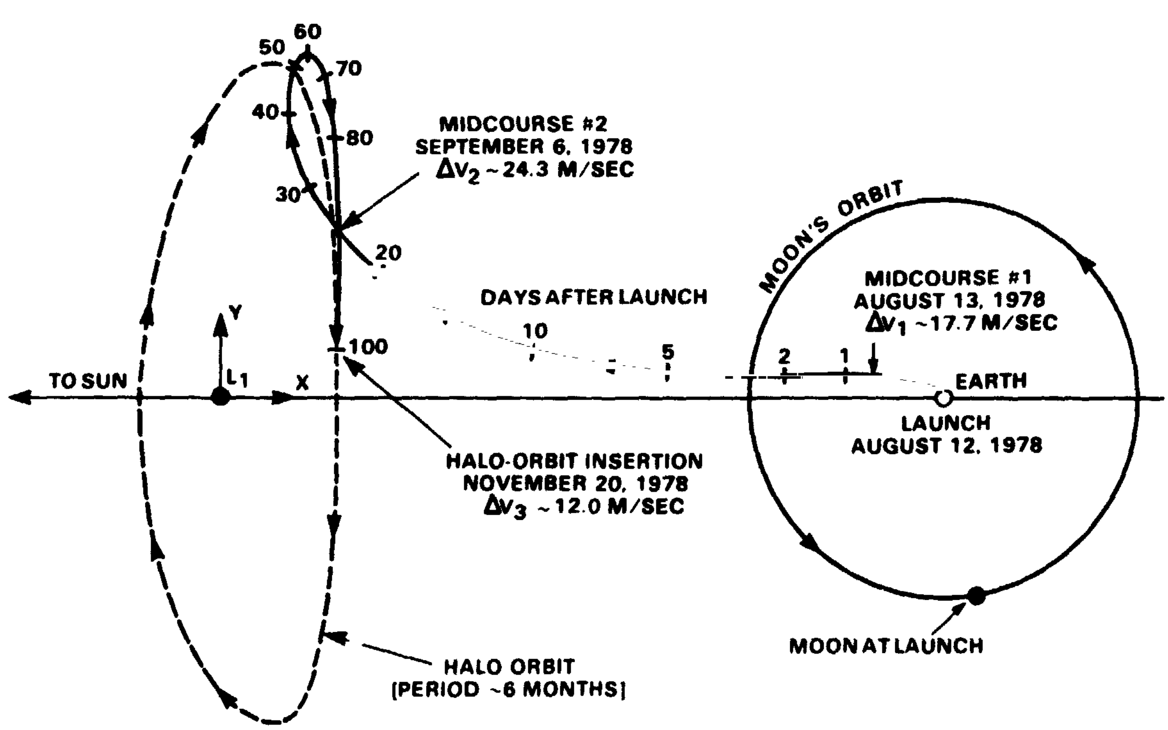


Figure 4. ISEE-3 Transfer Trajectory to Halo Orbit

Halo orbits may also play an important role in future lunar operations (Reference 3). For instance, 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 would greatly facilitate the navigation and control of an unmannered "Lunokhod" vehicle on the Moon's far side. Although Soviet plans for future lunar exploration are not clear, the growing number of Russian publications concerning halo orbits in the Earth-Moon System is rather intriguing (e.g., see Reference 4).

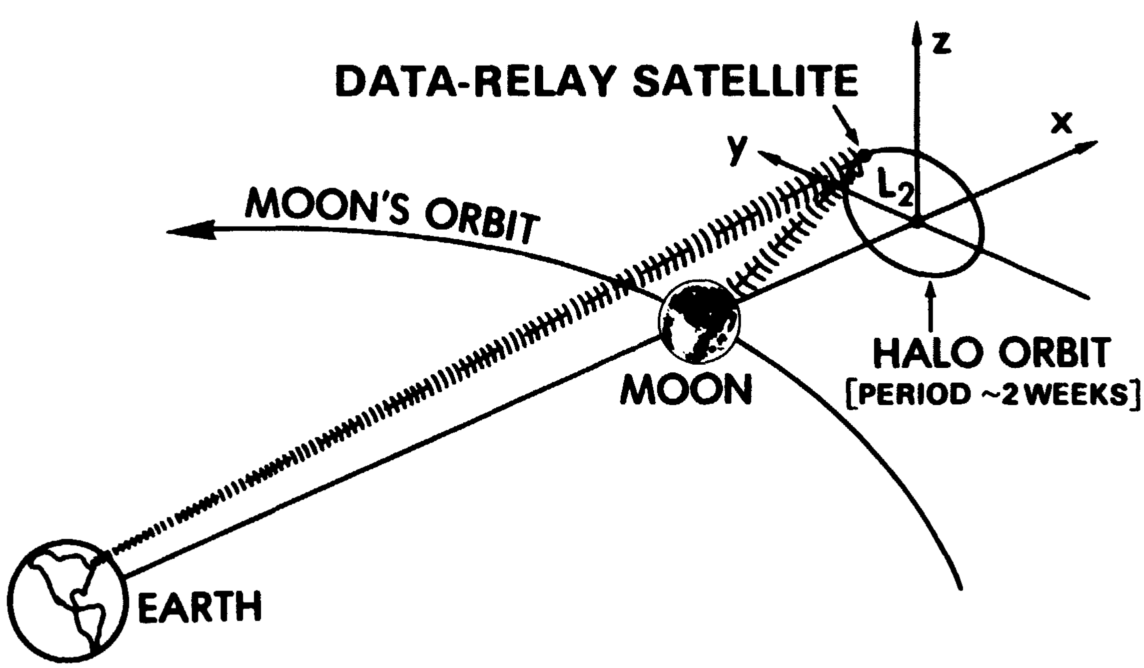


Figure 5. Lunar Farside Data Link Using Halo Comsat

DOUBLE LUNAR-SWINGBY TECHNIQUE

As mentioned earlier, a spacecraft stationed in a halo orbit around the Sun-Earth L_1 point is able to monitor upstream interplanetary conditions on a continuous basis. However, to obtain a better understanding of the cause and effect relationships between time-dependent magnetospheric processes, simultaneous measurements in the distant geomagnetic tail will also be needed. The main region of interest begins at the Moon's orbit (~60 R_E) and extends as far as the L_2 libration point (see Figure 6). This is mostly unexplored territory. Earth-orbiting spacecraft have probed the mag-

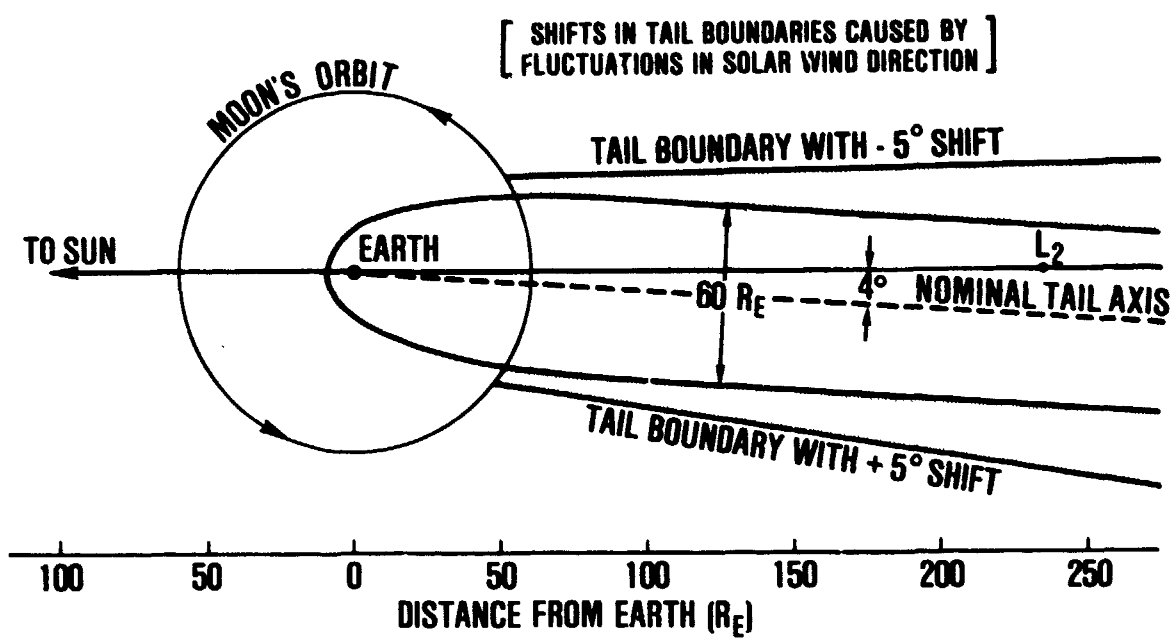


Figure 6. Geomagnetic Tail in Ecliptic Plane

netotail out to distances of about 80 R_E, but the only measurements beyond this point have been obtained from single traverses by Pioneer-8 at 500 R_E and Pioneer-7 at 1000 R_E.

A spacecraft located in a halo orbit around the Sun-Earth L_2 point could provide continuous data between 220 and 250 R_E. 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 desired. These goals are not easily attained. Because the geomagnetic-tail axis is always aligned within a few degrees of the Sun-Earth line, it will be necessary to control the rotation of the apsidal line of an Earth-orbiting spacecraft to maintain the spacecraft's apogee segment in the tail region. The required orbital rotation of approximately one degree per day could be achieved by using propulsive maneuvers, but the ΔV penalty would be about 400 m/sec per month. A more-practical way to accomplish this task is to employ a series of lunar gravity-assist maneuvers.

The basic procedure is shown in the top section of Figure 7. 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 with respect to the Sun-Earth line will position the spacecraft for a trailing-edge

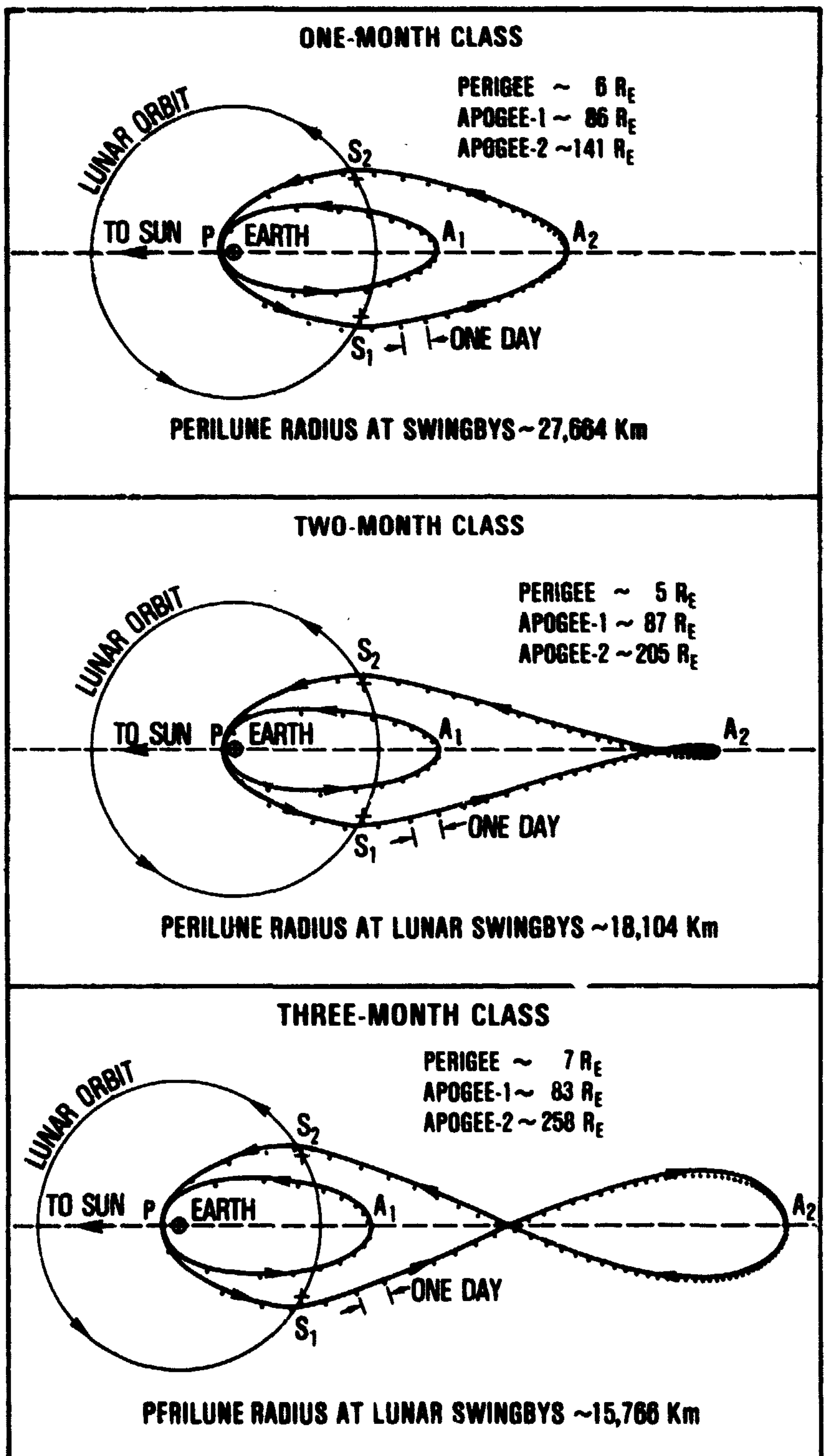


Figure 7. Sun-Synchronous Periodic Orbit; Using Double Lunar Swingby

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 to A_2. A leading-edge lunar swingby at S_2, after the Moon has completed one full orbit plus the $S_1 S_2$ segment will return the spacecraft to its original orbit. This sequence of orbit pairs could be repeated indefinitely or, by slightly changing the swin... conditions at S_1 and S_2, the spacecraft could be placed into different periodic orbits as shown in the other sections of Figure 7. The twisting nature of these orbits has led to the descriptive terminology, "pretzel orbits."

The three classes of orbits illustrated in Figure 7 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_2 A_1 S_1). Details of these solutions are given in Reference 5.

An interesting property of the gravity-assist trajectories is exhibited in Figure 8. Here the one-month class orbit of Figure 7 is plotted in a reference frame where the Earth-Moon line is fixed. Notice that the combination of lunar swingbys and Sun synchronization gives rise to a special type of orbit that is doubly periodic.

The periodic orbits in Figures 7 and 8 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. A typical example of a realistic trajectory simulation is shown in Figure 9. This case begins with both the spacecraft and the Moon near M_1 and ends with the Moon in position for another swingby at M_2. In order to provide better coverage of the magnetotail, a canted orbit was selected for the initial three-month outer loop (M_1 A_1 S_1).

NASA is planning to use the double lunar-swingby concept in a four-spacecraft program called Origin of Plasmas in the Earth's Neighborhood (OPEN) that is scheduled to begin in the mid-1980's (Reference 6). A major goal of the OPEN program is to improve our understanding of the plasma processes that are important in controlling the Earth's nearby space environment. Extensive coverage of the distant geomagnetic tail is an essential component of this program.

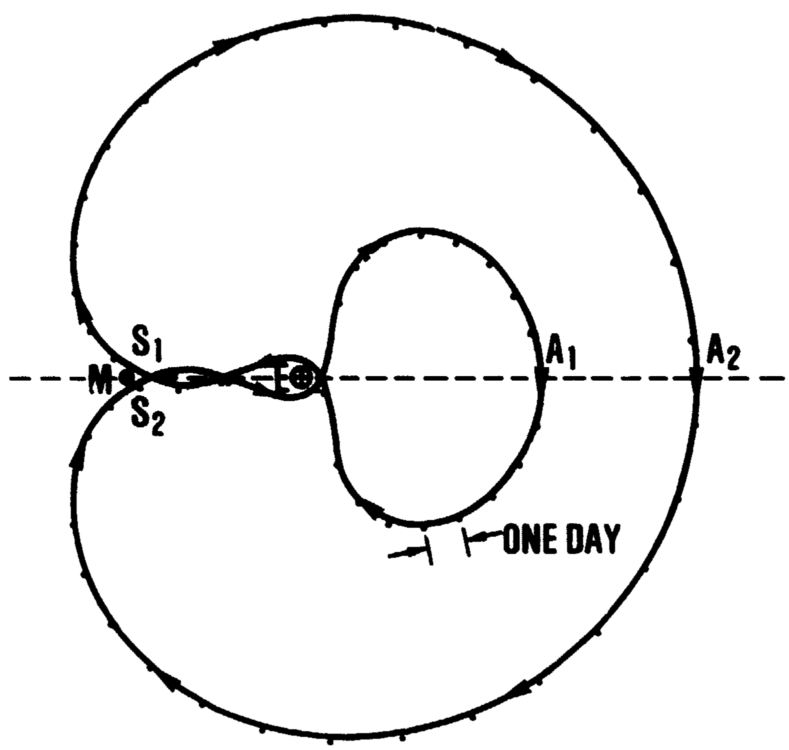


Figure 8. One-Month Class Periodic Orbit in Earth-Moon Reference Frame

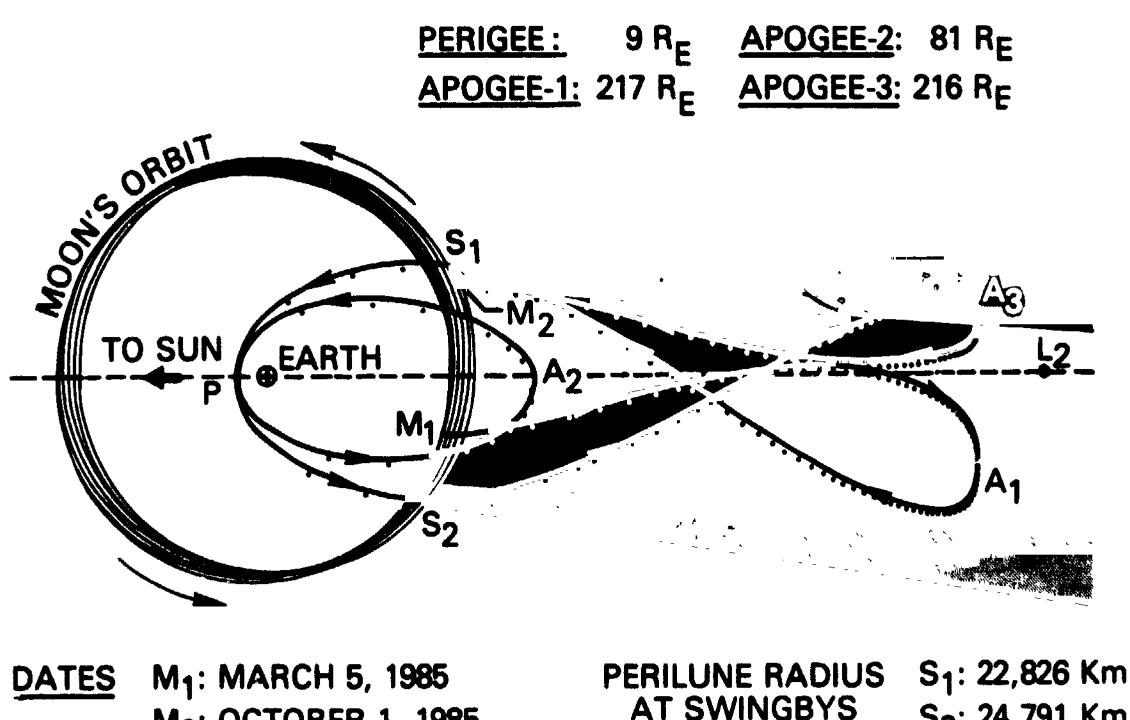


Figure 9. Realistic Simulation of Double Lunar-Swingby Trajectory

EARTH-SWINGBY TRAJECTORIES

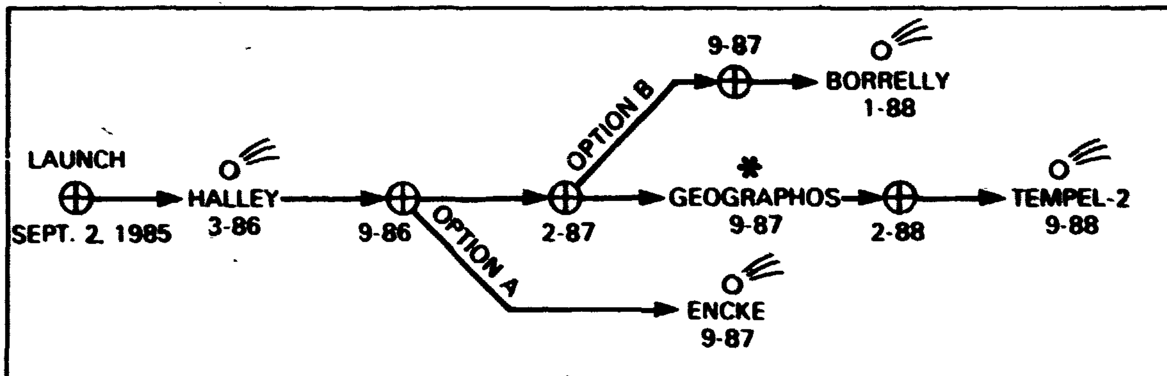
The use of the Earth's gravity field to control the path of a spacecraft in heliocentric space is an extremely useful trajectory concept for a specialized class of missions in the Sun-Earth System. By using a sequence of Earth-swingby maneuvers, it is sometimes possible to construct a trajectory that will intercept two or more of the comets and asteroids that pass through the inner solar system. The added scientific value of a mission profile that contains multiple encounters of these interesting bodies is fairly evident. An outstanding example of this ssion category is described here.

The leading candidate for a near-term cometary mission is Halley's comet which is scheduled to return in 1985-86 (Reference 7). Recently, it has been concluded that the most sensible way to carry out a reconnaissance of Halley would be to use a ballistic fast-flyby technique. As shown in Figure 10, it is possible to place a spacecraft into a trajectory that first intercepts Halley and then returns to the Earth's vicinity one year after launch (Reference 8). This "boomerang" trajectory scheme makes it possible to retarget the spacecraft to another comet after the Halley flyby. Of course, the spacecraft must be able to overcome the hazard of passing through Halley's dust cloud at 59 km/sec! Happily, studies performed at the Jet Propulsion Laboratory have shown that a survival probability of greater than 95% can be attained by using a relatively simple dust shield.

Alternative trajectory profiles that can be achieved by using different combinations of Earth-swingby maneuvers are summarized in Table 1. In every case, the first Earth-swingby maneuver occurs at the end of the boomerang trajectory segment on September 2, 1986. The nominal plan then calls for a six-month Earth-to-Earth leg followed by another swingby maneuver on February 28, 1987, which sets the stage for a flyby of the asteroid Geographos on September 12, 1987. A third Earth-swingby maneuver on February 29, 1988 sends the spacecraft towards an intercept with comet Tempel-2 on September 26, 1988. It is worth noting that the flyby speeds at Geographos and Tempel-2 are only 12.7 and 13.0 km/sec, respectively.

A plot of the triple-encounter sequence is given in Figure 11. Viewing the spacecraft path with respect to a fixed Sun-Earth line clearly displays the boomerang-like features of the Halley and Geo-

Table 1
Alternative Encounter Sequences



EARTH SWINGBYS			
SWINGBY DATE	PERIGEE [EARTH RADII]	BEND ANGLE [DEGREES]	HELIOCENTRIC INCLINATION AFTER SWINGBY [DEGREES]
1 SEPT. 2. 1986	2.07	54.4	11.7
2 FEB. 28. 1987	5.43	27.3	10.4
3 FEB. 29. 1988	5.25	28.0	10.3
1A SEPT. 2. 1986	1.95[a]	56.0	11.2
2B FEB. 28. 1987	—	—	Standoff Encounter[b]
3B SEPT. 2. 1987	3.45	39.2	8.7

SMALL-BODY ENCOUNTERS				
ENCOUNTER DATE	SUN DISTANCE [AU]	EARTH DISTANCE [AU]	PHASE ANGLE [DEGREES]	FLYBY SPEED [Km/sec]
HALLEY: MAR. 28. 1986	1.11	0.61	113.0	58.9
ENCKE SEPT. 1. 1987	1.07	1.01	166.3	31.3
GEOGRAPHOS: SEPT. 12. 1987	1.01	0.37	170.0	12.7
BORRELLY JAN. 16. 1988	1.40	0.70	89.6	17.1
TEMPEL-2 SEPT. 26. 1988	1.39	1.01	94.8	13.0

[a] Powered Swingby Required [$\Delta V \sim 29$ m/sec].
[b] Trajectory Change is not Required Swingby is Targeted for Earth Return at Next Scheduled Date

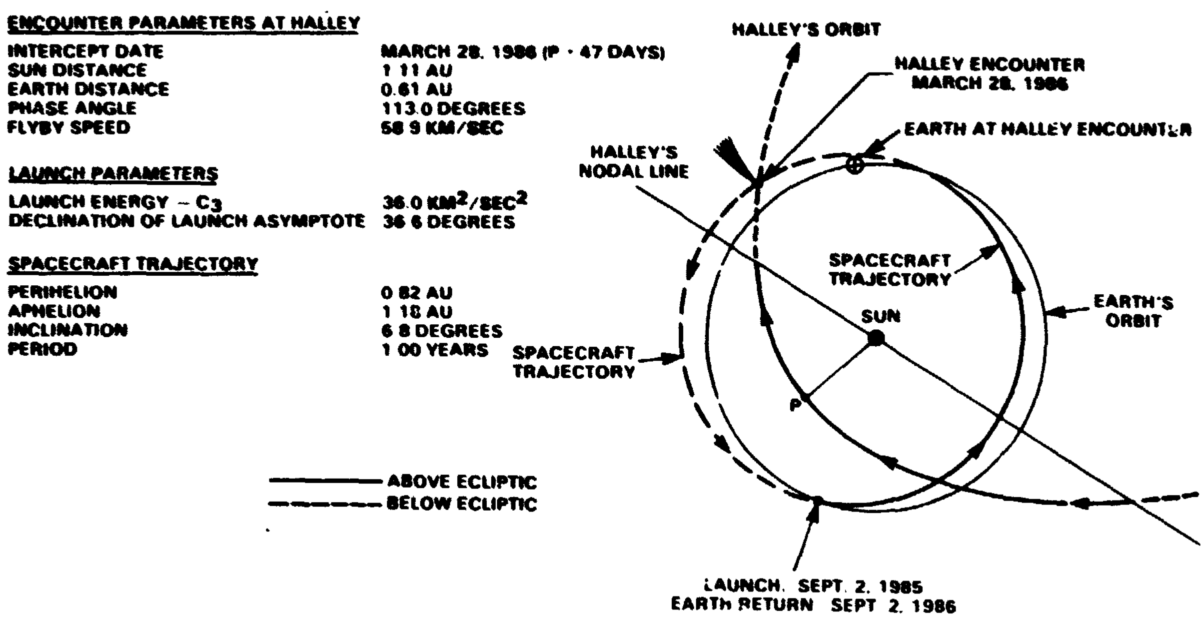


Figure 10. Halley Flyby Using Earth-Return Trajectory

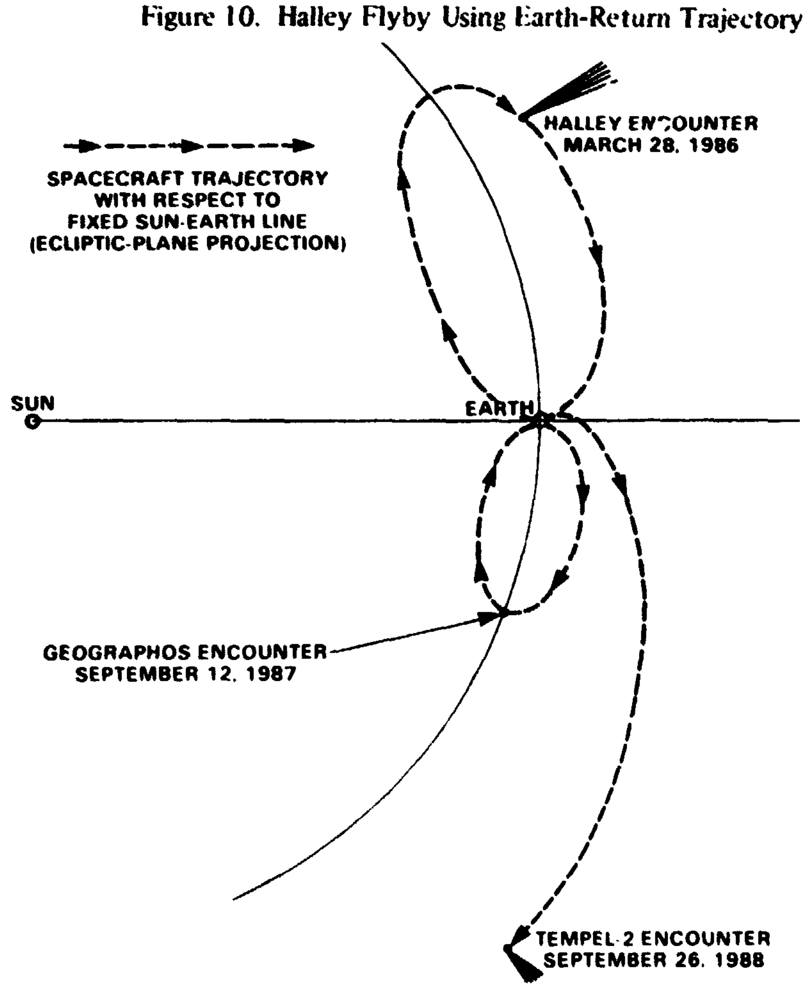


Figure 11. Trajectory Profile for Triple-Encounter Sequence

graphos trajectory segments. The six-month Earth-to-Earth segment appears as a point in this diagram because it oscillates about the Earth along an axis that is perpendicular to the Ecliptic plane.

Unfortunately, the launch window for the triple-encounter mission is very tight. As a matter of fact, sizeable propulsive maneuvers are required for all launch dates other than September 2, 1985. However, the ΔV penalty for these maneuvers can be minimized by applying one ΔV maneuver shortly after the Halley encounter and another one during the six-month Earth-to-Earth segment.* For a one-week launch window, the total ΔV cost is only 300 m/sec, a modest requirement.

A less-desirable way to circumvent the launch-window problem would be to delete the Geographos flyby from the mission profile. Propulsive maneuvers are not needed for the dual-encounter mission to Halley and Tempel-2, and the launch-energy requirement can be held below 40 km^2/sec^2 for a one-month launch window. Details of the launch window for this mission are given in the Appendix.

Trajectory profiles for the dual cometary missions Halley-Encke (option A) and Halley-Borrelly (option B) are also shown in Table 1. The launch window for these alternative missions is identical to the window for the Halley-Tempel-2 mission (see Appendix). However, propulsive maneuvers are required for the Halley-Encke option if the launch date is earlier than September 3, 1985. Even so, the ΔV requirement for the nominal launch date of September 2 is only 29 m/sec. The post-swingby trajectory for the Halley-Encke mission is illustrated in Figure 12.

The Earth-swingby trajectories identified in Table 1 could be used to carry out an exploratory survey of comets and asteroids. An ideal mission scenario (neglecting cost!) would use a single Shuttle/IUS launch to send a salvo of three spacecraft towards Halley (e.g., a primary spacecraft equipped with a good imaging system and two smaller probes). After the triple Halley encounter, the main spacecraft would go on to Geographos and Tempel-2 while the two "daughter" spacecraft performed flybys of Encke and Borrelly.

*Private communication from D. J. Ross of the Jet Propulsion Laboratory, May 1980.

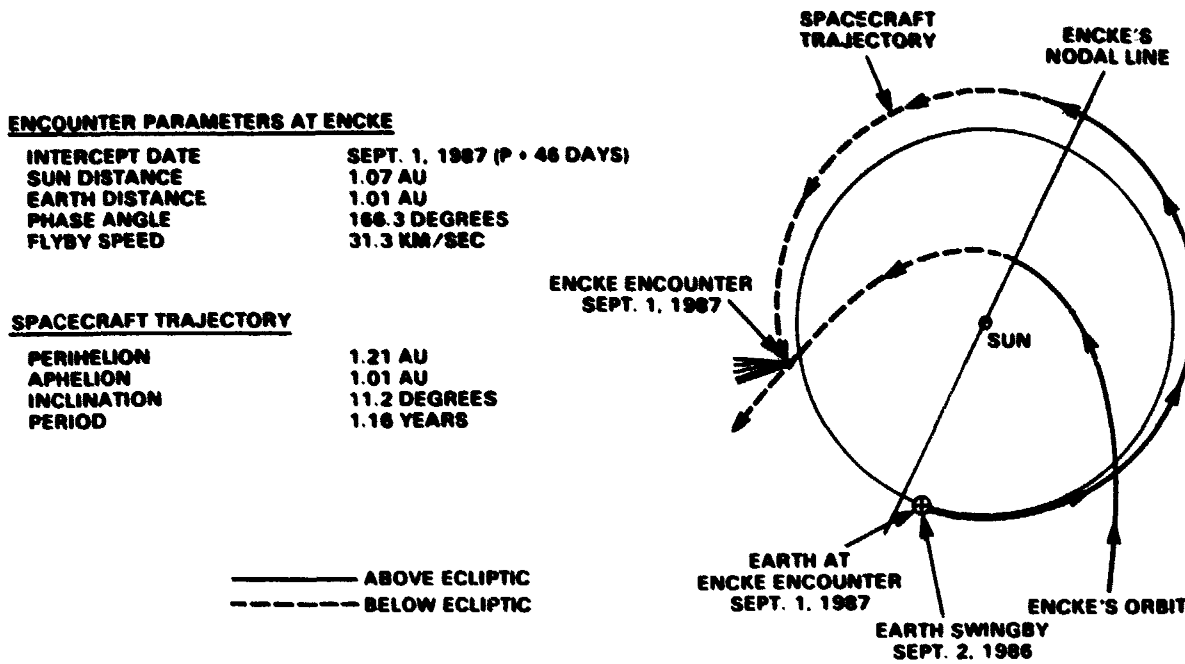


Figure 12. Trajectory Option for Encounter with Encke's Comet

CONCLUDING REMARKS

Three unorthodox, but very useful, trajectories have been described: the halo, the pretzel, and the boomerang. The mission-design flexibility provided by these novel trajectory concepts is truly remarkable. Pre-flight mission planning has been enhanced by the number and variety of available orbits. In-flight modification of a baseline trajectory profile is also possible. This expanded orbital capability has provided several new mission opportunities for scientific exploration in the Sun-Earth-Moon System.

ACKNOWLEDGMENT

The author would like to thank William H. Wooden II for his assistance in computing the Earth-swingby trajectories.

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APPENDIX

Launch-Window Variations for Dual Cometary Missions

Launch-window variations for a dual-encounter mission to Halley and Encke are given in Table A-1. A powered Earth-swingby is required for launch dates before September 3, 1985. Notice that the ΔV cost for this maneuver is quite large at N-14 days.

Parameter variations for the Borrelly and Tempel-2 trajectory options are listed in Table A-2. Multiple Earth-swingby maneuvers are required, but some of these swingbys are standoff encounters.

Orbital elements for the four comets and the asteroid Geographos are listed in Table A-3. These elements were used in the computations of the Earth-swingby trajectories.

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Table A-1
Launch-Window Variations for Halley-Encke Mission
[Nominal Launch Date: September 2, 1985]

	N - 14 Days	Nominal	N + 14 Days
Launch date	8-19-85	9-2-85	9-16-85
Launch energy-C_3 (km^2/sec^2)	35.5	36.0	39.2
Decl. of launch asymp. (deg.)	41.2	36.6	31.9
Halley intercept date	3-29-86	3-28-86	3-26-86
Sun distance (AU)	1.14	1.11	1.08
Earth distance (AU)	0.57	0.61	0.65
Phase angle (deg.)	114.3	113.0	111.7
Flyby speed (Km/sec)	58.2	58.9	59.9
Encke Option			
Earth swingby date	8-19-86	9-2-86	9-16-86
Perigee (Earth radii)	1.53	1.95	5.42
Bend Angle (deg.)	58.9	56.0	26.3
ΔV requirement (m/sec)	676.6	28.8	–
Encke intercept date	8-30-87	9-1-87	9-10-87
Sun distance (AU)	1.04	1.07	1.21
Earth distance (AU)	0.99	1.01	1.14
Phase angle (deg.)	165.8	166.3	167.4
Flyby speed (Km/sec)	31.1	31.3	31.0

Table A-2
Launch-Window Variations for Borrelly and Tempel-2 Options
[Nominal Launch Date: September 2, 1985]

Borrelly Option			
	N - 14 Days	Nominal	N + 14 Days
First swingby date	8-19-86	9-2-86	9-16-86
Perigee (Earth radii)	2.04	2.07	1.95
Bend angle (deg.)	55.3	54.4	53.4
Second swingby date[a]	2-15-87	2-28-87	3-14-87
Third swingby date	8-19-87	9-2-87	9-16-87
Perigee (Earth radii)	4.24	3.45	2.23
Bend angle (deg.)	34.2	39.2	49.3
Borrelly intercept date	1-15-88	1-16-88	1-15-88
Sun distance (AU)	1.40	1.40	1.40
Earth distance (AU)	0.69	0.70	0.69
Phase angle (deg.)	89.7	89.6	88.1
Flyby speed (Km/sec)	18.0	17.1	16.4
Tempel-2 Option			
First swingby date	8-19-86	9-2-86	9-16-86
Perigee (Earth radii)	2.04	2.07	1.95
Bend angle (deg.)	55.3	54.4	53.4
Second swingby date[a]	2-15-87	2-28-87	3-14-87
Third swingby date[a]	8-19-87	9-2-87	9-16-87
Fourth swingby date	2-15-88	2-29-88	3-13-88
Perigee (Earth radii)	7.08	6.37	5.19
Bend angle (deg.)	22.1	24.0	26.7
Tempel-2 intercept date	9-18-88	9-26-88	10-5-88
Sun distance (AU)	1.39	1.39	1.40
Earth distance (AU)	0.97	1.01	1.06
Phase angle (deg)	89.0	94.8	100.8
Flyby speed (Km/sec)	13.3	13.0	12.8

[a]Standoff encounter [Trajectory change is not required].
Swingby is targeted for Earth return at next scheduled date.

Table A-3
Orbital Elements
[Equinox 1950.0]

Body	T	q	e	Ω	ω	i
Halley 1986	Feb 9.661	0.5871	0.9673	58.153	111.853	162.238
Encke 1987	Jul 17.355	0.3317	0.8499	334.036	186.259	11.926
Geographos 1987	Jul 18.764	0.8192	0.3420	337.262	276.547	13.329
Borrelly 1987	Dec 18.268	1.3567	0.6242	75.268	353.333	30.325
Tempel-2 1988	Sep 16.718	1.3834	0.5444	119.118	191.041	12.432