AIAA-98-4283

DISCOVERY-CLASS MERCURY ORBITER TRAJECTORY DESIGN FOR THE 2005 LAUNCH OPPORTUNITY

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Abstract

The mid-1980s discovery by C. L. Yen of a ballistic trajectory technique utilizing multiple Venus and ΔV-Mercury gravity assists offers a low-risk approach to maximizing payload delivery into Mercury orbit. Recent studies have demonstrated the viability of a Mercury orbiter mission, utilizing this type of trajectory, within NASA Discovery Program guidelines. Application of a detailed spacecraft design to the lowest-risk near-term mission opportunity enabled a more rigorous analysis of key trajectory design aspects. This opportunity, which requires launch in 2005, has trip times of 4.2 and 5.6 years for trajectories having two and three Mercury swingbys, respectively. Trajectory optimization software improvements and realistic spacecraft operational constraints contributed to a lower-ΔV solution than previously published, tempered with operational constraints that affect the timing of deterministic ΔVs. For example, solar conjunction (where solar interference disrupts spacecraft-Earth tracking station communications) and the desire for real-time monitoring of all ΔVs, led to moving a planned ΔV earlier than the minimum ΔV date. Other sources of additional ΔV come from gravity loss due to a non-impulsive maneuver at Mercury orbit insertion.

Introduction

Of the six planets closest to the Sun, only Mercury has no comprehensive orbiter mission en route, in progress, or concluded. A Mercury orbiter mission has long been part of NASA's core program of solar system exploration[1]. After a brief review of Mercury orbiter studies, this paper will focus on selected heliocentric and hermicentric trajectory analyses at a typical NASA phase A/B level of detail for the 2005 launch opportunity. These analyses have the greatest impact on post-launch ΔV and launch energy, parameters needed for determining maximum initial spacecraft mass. Summary heliocentric transfer trajectory data for other launch opportunities will demonstrate the uniqueness and preference associated with launching in 2005. The ballistic trajectories offering the largest payload and lowest risk to Mercury orbit utilize two Venus gravity assists (swingbys), two or three Mercury swingbys, and a few strategically placed propulsive maneuvers.

Current Discovery guidelines include launch vehicle no larger than a Delta 7925H, phase C/D development less than three years through launch + 30 days, and total mission cost not to exceed $299 Million in Fiscal Year $1999.

Mercury Orbiter Historical Overview

Since the 1974-75 Mariner 10 Mercury flybys, a Mercury orbiter mission has appeared too risky or expensive for serious consideration. However, recent technological advances are working to reverse this trend in the U.S.A., Europe, and Japan. Yen[2] has documented Mercury orbiter mission studies[3,4,5] prior to 1985.

In 1985 Yen[2] described a new method with improved performance for ballistic Mercury orbiter missions. This method offers the lowest launch energy and post-launch ΔV requirements by utilizing two Venus gravity assists followed by up to three Mercury gravity assists with subsequent ΔV near aphelion. The Venus swingbys lower the spacecraft orbit's perihelion and aphelion as well as perform much of the 7° plane change from Earth orbit to Mercury orbit.

The Mercury swingby-ΔV pairs lower the spacecraft orbit's aphelion and rotate the orbit line of apsides towards Mercury's line of apsides, thereby reducing the ΔV required for Mercury orbit insertion. Each Mercury swingby and subsequent ΔV place the spacecraft into an orbit with period nearly equal to an integer number of Mercury orbit periods. This spacecraft:Mercury orbit resonance increases from 2:3 to 3:4 to 5:6 with one, two, and three Mercury swingbys. In 1997 McAdams determined that a contingency fourth Mercury swingby using a 7:8 resonance and very small aphelion ΔV (to prevent moving Mercury orbit insertion away from Mercury's perihelion) required nearly 200 m/sec less ΔV and 23 months longer trip time than the trajectory with three Mercury swingbys.

Since 1985 several Mercury orbiter studies have applied Yen's ballistic trajectory method or chosen a low-thrust (solar electric propulsion-SEP or solar sail) trajectory. Ballistic Mercury orbiter mission studies include a dual orbiter[6] and the first Discovery-class approach[7] by JPL, as well as a 2002 launch Discovery-proposed mission by Carnegie Institute of Washington /Johns Hopkins University Applied Physics Laboratory, and an ESA-proposed mission that would launch in 2009[8]. The Japanese government is currently investigating a ballistic approach Mercury orbiter mission using a more preliminary version of the 2005 launch trajectory presented here and a spin-stabilized spacecraft[9]. Kluever and Abu-Saymeh[10] examined optimal SEP Mercury orbiter trajectories that utilize one Venus swingby and New Millenium DS-1 propulsion technology.

Heliocentric Transfer

Spacecraft Description and Constraints

In order to transition between the preliminary pre-phase A and the more-detailed phase A/B analyses, a realistic spacecraft design, operational constraints, and science requirements must be defined. The assumed spacecraft receives power from solar arrays and batteries (launch phase and solar occultation). The dual-mode (bipropellant/monopropellant) propulsion system uses a 660N primary thruster for all maneuvers larger than 20 m/sec. This thruster provides about 40% more thrust than the NEAR spacecraft's large thruster in order to reduce Mercury orbit insertion (MOI) gravity-loss ΔV. All propulsive maneuvers performed less than 0.7 AU from the Sun are designed within tilt-angle constraints in pitch and yaw. Propulsive maneuvers must accommodate real-time monitoring from Earth, with the exception of part of the 24-minute MOI burn. A DE405-based Mercury orbit integration based upon a nominal Sun-facing spacecraft area provides a detailed assessment of orbital parameter fluctuation and maneuver attitude. The size, orientation, and allowed variation in the spacecraft's orbit at Mercury are within limitations set for the science payload, power, and thermal control. Constraints not mentioned above have lesser effects on the spacecraft trajectory and maneuver design.

Rationale for Launch in 2005

Before entering detailed discussion on the trajectory to Mercury, it is appropriate to focus on rationale for selecting the 2005 launch opportunity. Key mission parameters for upcoming Mercury orbiter launch opportunities are listed in Table 1. Post-launch ΔV does not include navigation or margin. Post-launch ΔV and mission duration in Table 1 exclude time after Mercury orbit insertion. Although the 2002 launch opportunity offers the largest initial spacecraft mass (1080 kg vs. 1060 kg for 2005), the 2005 launch provides a shorter flight time with a ΔV contingency option of a third Mercury swingby. A large penalty affecting launch mass arises for the 2002 launch opportunity. Note also that high launch energy requirements for the 2004 and 2007 launches result in much lower payload masses.

Table 1
Near-Term Launch Opportunity Comparison*

Launch Date	Aug-Sep 2002	Jun-Jul 2004	Jul-Aug 2005	Jul 2007
ΔV_{PL} (km/s)	2379	2010	2420	2338
Duration (years)	4.5	5.3	4.2	5.2
C3 (km^2/s^2)	12.7	28.9	16.3	21.8
Extreme DLA (°)	-44.0	-18.4	-33.3	?
Mercury Swingbys	3	3	2	3
Concerns	1 km/s ΔV at Venus	Lower payload; backup	No schedule backup	Lowest payload

* Entries scaled to account for 20-day launch window.

Launch

The strategy implemented for defining a 20-day launch window yields the maximum initial spacecraft mass for a constraint-adjusted, near-minimum total ΔV trajectory to Mercury orbit. Spacecraft dry mass and post-launch ΔV requirements dictate launch aboard a 3-stage Delta 7925H, the largest launch vehicle allowed within NASA's Discovery Program. The maximum launch energy and extreme DLA (see Table 2), together with a

99.0% probability of commanded shutdown (PCS - standard launch vehicle parameter), define the maximum spacecraft mass delivered to the heliocentric transfer orbit. Details of the launch trajectory prior to first contact with a Deep Space Network (DSN) tracking antenna are not pertinent here.

Table 2
Mercury Orbiter Launch Summary

Launch dates	Jul 31- Aug 19, 2005 (20 days)
Launch energy	C_3 = 16.3 km^2/sec^2
Launch vehicle	Delta-7925H-9.5
Initial launch mass	1060 kg (max for 99% PCS)

"launch energy for post-launch ΔV exchange" was performed for launch dates July 31 and August 1. Launch energy was reduced by 0.3 km^2/s^2 at a cost of 39.7 m/s (increasing the maximum post-launch ΔV by only 4.5 m/s since the July 31 unconstrained case required 35.2 m/s less post-launch ΔV than the August 14 maximum post-launch ΔV). The result is two days added to the launch window with no reduction of initial spacecraft launch mass, but at the loss of extra ΔV margin for the first two days of the launch window. To achieve final launch window, the second deep space maneuver (DSM) was moved earlier by 9-12 days at a penalty of 9-15 m/s ΔV to comply with the "monitoring during all maneuvers" constraint. This DSM shift moved the maneuver to a Sun-Earth-spacecraft angle of

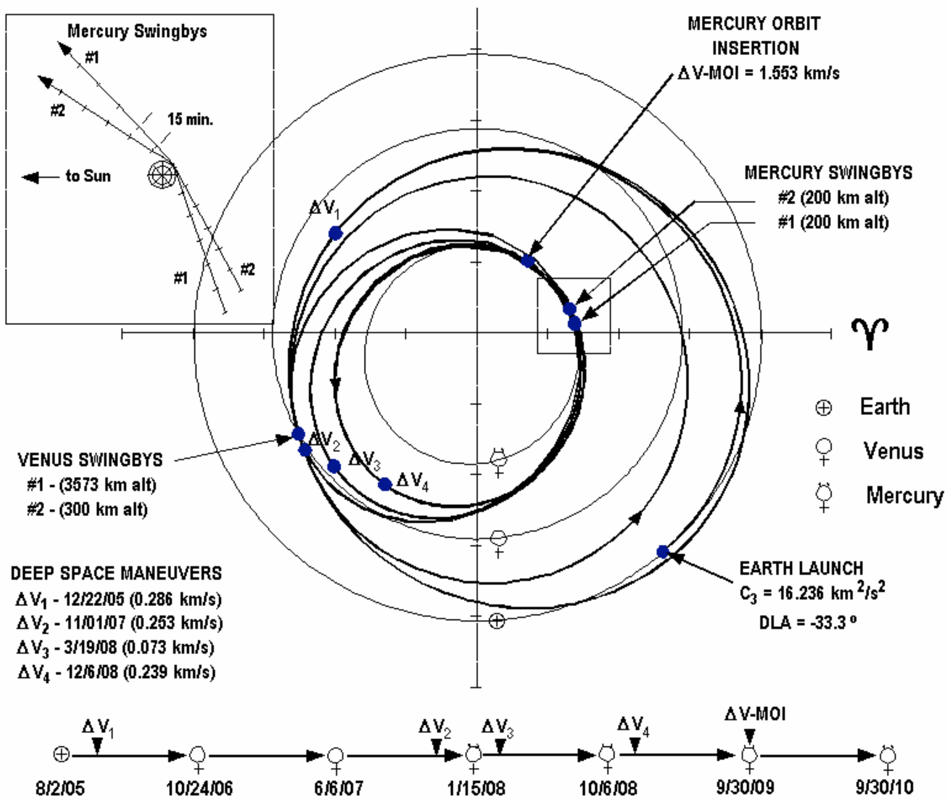

Figure 1 – Ecliptic Plane Projection of Mercury Orbiter Trajectory

Launch window definition begins with determining the minimum total ΔV case (August 7, 2005 launch). This was performed and verified with two independent software tools. By adding and subtracting days to the August 7 launch date, 18 "minimum total ΔV" trajectories were generated with the first and last launch dates having nearly equal launch energy (August 2-19). Since maximum post-launch ΔV occurs on August 14, a 2°, where solar interference would not prevent spacecraft communication. Figure 1 shows a sample trajectory profile for an August 2, 2005 launch.

Venus Swingbys

The Mercury orbiter mission employs two Venus swingbys and two DSMs prior to the first Mercury encounter. Throughout the 20-day launch window, the Venus swingby dates remain somewhat stable, varying less than two days. This stability occurs because the best position for Venus at swingby is roughly opposite the location of the Mercury encounters. For an orbiter mission, the Mercury encounters need to move toward Mercury's perihelion, because the orbit insertion ΔV is least when applied while Mercury is closest to the Sun. Even though it requires more ΔV to lower the spacecraft orbit to perihelion than to aphelion, the saving achieved in performing orbit insertion near Mercury's perihelion more than compensates.

The DSMs before and after the Venus swingbys vary much in magnitude and date. The first DSM, which occurs near the first of two perihelions en route to Venus, varies about two weeks in order to account for Earth-Venus phasing, early launch window launch energy reduction, and shifts in the second DSM to avoid solar conjunction. The second DSM, which occurs just over one orbit into the type III Venus-to-Mercury transfer, is affected by the constraint for real-time observation of the maneuver, and is near aphelion to efficiently target the first Mercury swingby. Strategy for final placement of this DSM was discussed in the earlier "Launch" section.

Two Venus swingbys allow greater advantage of Venus' gravity field to reduce propulsion requirements. The overall effect of using Venus is to remove energy from the heliocentric transfer and rotate the spacecraft trajectory plane nearer to Mercury's orbit plane. A consequence of splitting the effect over two encounters is that the first swingby must produce a spacecraft orbit period that exactly matches Venus' orbit period. This assures that the spacecraft and planet will meet again one Venus period later. Refer to Table 3 for the effect each swingby has on the spacecraft orbit inclination and perihelion/aphelion distances. The calculations that establish the required orbit optimize the post-encounter flight path angle, the change in inclination, and the passage altitude to accommodate the finite hyperbolic excess velocities before and after the encounter.

The passage distance for the first encounter optimizes at altitudes ranging from 3102 to 3573 km over the launch window. The low phase angle (13° to 17°) during the first encounter indicates that Venus will be nearly fully illuminated by sunlight as seen by the spacecraft on its pre-encounter trajectory. However, solar conjunction (spacecraft passes behind the Sun as viewed from the Earth) about a week after this encounter. The communication link is uncertain 2-3 days before and 14-18 days after this Venus swingby. Therefore, greater emphasis is needed on final pre-encounter orbit determination and targeting ΔV(s).

Table 3
Planetary Swingby Effect on Heliocentric Orbit Inclination and Apsidal Distance

Body Name	Inclination (deg)	Perihelion (AU)	Aphelion (AU)
Earth	2.4	0.60	1.02
Venus	8.0	0.55	0.90
Venus	6.7	0.33	0.75
Mercury	7.0	0.31	0.70
Mercury	7.0	0.30	0.63

The second Venus swingby establishes an orbit with aphelion near Venus' orbit radius and perihelion near Mercury's perihelion distance. In order to satisfy NASA planetary protection requirements[11], Venus passage altitude should not be less than 300 km. This minimum altitude may decrease with generation of a high-precision integrated trajectory and a detailed plan for pre-encounter orbit determination. Even though the optimum close approach altitude is below 300 km, no propulsive ΔV is required during this encounter. The 21°-22° approach phase angle indicates that, like the first Venus swingby, Venus will be brightly illuminated by the Sun as seen from the spacecraft.

Mercury Swingbys

A pair of unpowered (zero propulsive ΔV) 200-km altitude Mercury swingbys followed by near-aphelion DSMs act to slow the spacecraft enough to enable Mercury orbit insertion at the third Mercury encounter. This method is similar to the well-known ΔV-Earth gravity assist technique used by Near Earth Asteroid Rendezvous and planned by STARDUST. However, here the Mercury gravity assists lower aphelion and work with the DSMs to rotate the spacecraft orbit line of apsides closer to the Mercury orbit line of apsides. The Mercury-Mercury transfer orbits have successive spacecraft:Mercury orbital resonance of nearly 2:3 and 3:4. Each heliocentric orbit during this phase is indistinguishable in a conventional trajectory profile such as Figure 1. The bipolar plot with fixed Sun-Earth line (Figure 2) clearly shows each heliocentric orbit and preserves spacecraft distances and orientation with respect to the Earth and Sun. The inset of Figure 1 shows both Mercury swingbys on Mercury's dark side.

The Mercury swingbys occur in 2008 on January 15 and October 6 with no more than six hours variation throughout the 20-day launch window. The approach phase angles of 112° and 121° indicate that the spacecraft can better view the sunlit portion of Mercury after close approach. The relative velocities at encounter decrease from 5.79 km/s to 5.15 km/s to 3.37 km/s at Mercury orbit insertion (MOI). Any imaging during the Mercury swingbys necessitates an analysis characterizing the encounter viewing geometry. A sample of data for this type analysis is seen in Table 4 (1 R_M = 2439.7 km). Sub-solar latitude is 0° since Mercury's equator and heliocentric orbit plane are nearly identical.

In the event of in-flight ΔV usage exceeding allowed margins, the spacecraft could delay Mercury orbit insertion by about 1.5 years by targeting a third Mercury swingby on the same day at about the same time of the previously planned MOI. A subsequent aphelion propulsive maneuver on November 29, 2009 will place the spacecraft into a 5:6 orbital resonance with Mercury, culminating in MOI on March 17-18, 2011. Accounting for lower gravity loss at MOI, total ΔV saved with three versus two Mercury swingbys approaches 500 m/s.

Table 4
Mercury Swingby #2 (October 6, 2008) Viewing Summary

Milestone Description	Sub-spacecraft latitude (deg)	Sub-spacecraft W longitude (deg)	Sub-solar W longitude (deg)	S/C-Mercury-Sun angle (deg)	UTC (hh:mm:ss)
8 R_M alt before	0.49	227.43	357.29	129.82	20:13:00
1 R_M alt before	-0.30	196.96	357.33	160.28	21:07:14
5 minutes before	-0.93	170.26	357.33	172.71	21:14:00
close approach	-1.45	132.56	357.34	135.21	21:19:00
5 minutes after	-1.36	94.84	357.34	97.51	21:24:00
1 R_M alt after	-0.95	68.10	357.34	70.79	21:30:46
8 R_M alt after	-0.22	37.62	357.38	40.29	22:25:00

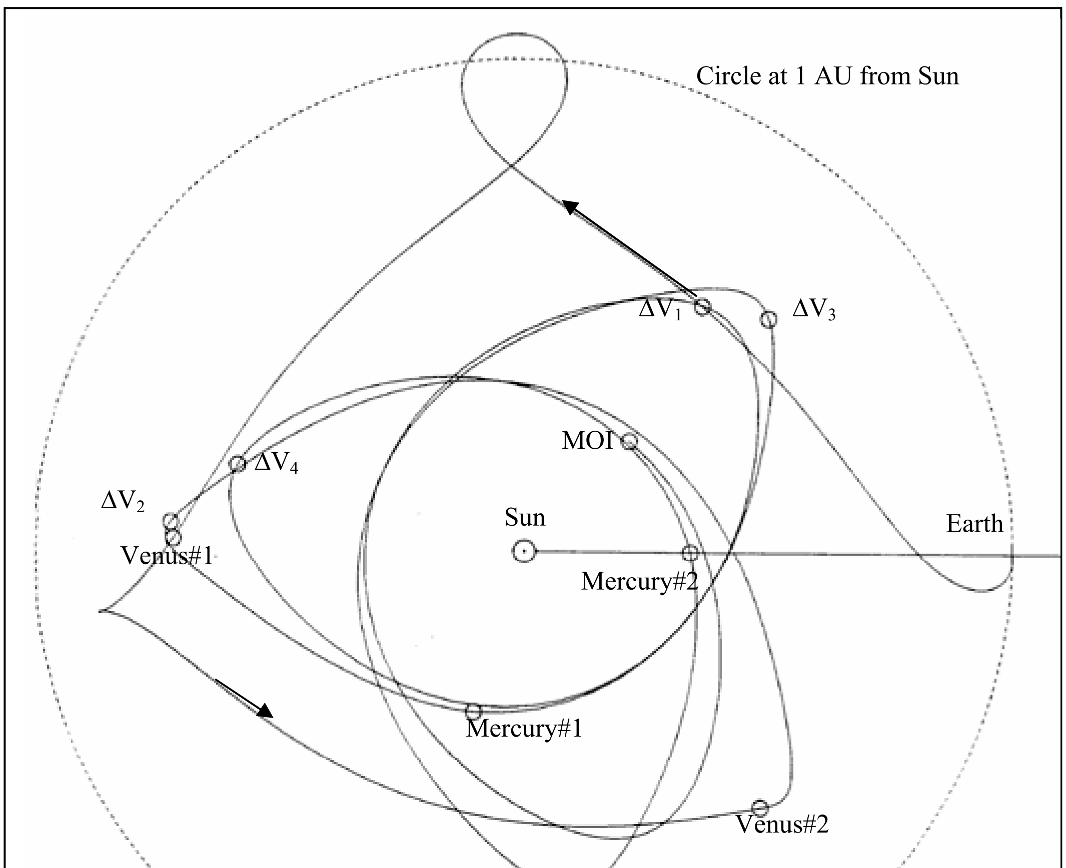

Figure 2 – 2005 Launch Mercury Orbiter Bipolar (Fixed Sun-Earth line) Trajectory

Mercury Orbit

Orbit Insertion

The heliocentric transfer ends at Mercury orbit insertion on September 30, 2009. The initial, near-polar, posigrade orbit has a 12-hour period and periherm and apoherm altitudes of 200 km and 15,193 km, respectively. The 24.4-minute, 1.553 km/s orbit insertion maneuver includes a gravity loss ΔV due to a finite burn penalty in Mercury's gravity field. The magnitude and direction of the arrival hyperbolic excess velocity will place the spacecraft into an orbit with one of two initial periherm latitudes. Figure 3 shows the solution that first flies over the north polar region. Although not considered here, actual periherm latitude selection depends on apsidal rotation ΔV, orbit maximum eclipse time, and science goals. During MOI the minimum altitude is 200 km, but the effect of the portion of the maneuver performed after the minimum altitude point is to drive the following periherm to 147-km altitude. A 7 m/s apoherm ΔV then raises the first post-MOI periherm to 200-km altitude. The difference between the two-phase 1.560 km/s MOI and impulsive 1.485 km/s MOI is 75 m/s. Orbit insertion into an intermediate initial orbit with longer period could reduce the finite burn penalty, but would likely require one or more maneuvers with the spacecraft attitude leaving sensitive components unprotected from direct sunlight. More investigation is needed in this area.

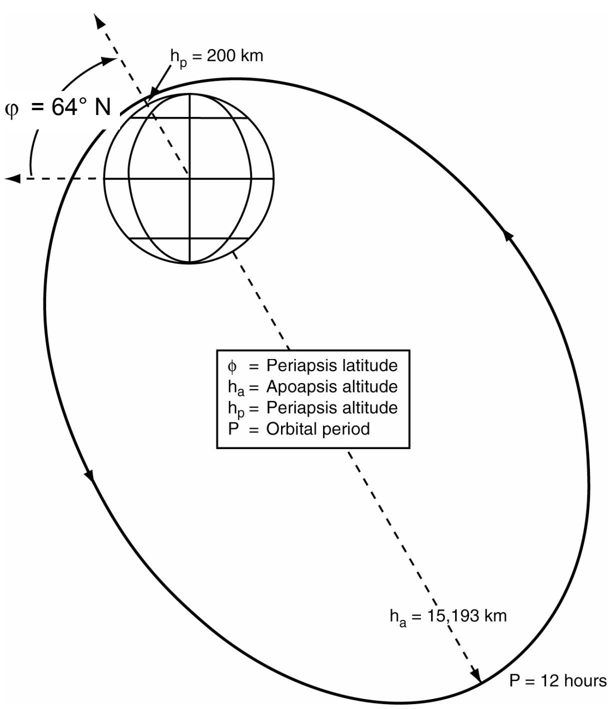

Figure 3-Typical Mercury Orbit Configuration

Orbit Maintenance

For the nominal 12 months at Mercury, a DE405-based integrated trajectory incorporated three pairs of ΔVs, one pair every time Mercury completes a solar orbit. This ΔV pair consists of an apoherm ΔV of 26.4 – 23.7 m/s for lowering periherm from 488 – 458 km down to 200 km, and a periherm ΔV of 4.1 – 3.7 m/s for adjusting orbital period from about 11.75 to 12 hours. Examples of orbital parameter variation include an 8°-9° northerly drift of periapsis latitude and a 4°-5° drift of the line of nodes pushing the low-altitude descending node farther into Mercury's dark side when Mercury is near perihelion. The line of nodes produces a sub-solar, low-latitude, fly over when Mercury is nearly 0.39 AU from the Sun.

Conclusion

Application of a spacecraft design, operational constraints, and science requirements for the 2005 launch Mercury orbiter added over 180 m/s ΔV to the minimum total ΔV patched conic solution. When combined with a navigation analysis of all deterministic ΔVs and planetary swingbys, these results lead to a lower, reliable estimate of non-deterministic (navigation/margin) ΔV. In addition, software improvements enabled discovery of a lower total ΔV Mercury orbiter trajectory than identified in previous research.

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