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TRAJECTORY DESIGN AND MANEUVER STRATEGY FOR THE MESSENGER MISSION TO MERCURY

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Destined to become the first spacecraft to orbit the planet Mercury, the MESSENGER (MErcury Surface, Space ENvironment, GEochemistry, and Ranging) spacecraft was launched on August 3, 2004. The 6.6-year ballistic trajectory to Mercury will utilize six gravity-assist flybys of Earth (one), Venus (two), and Mercury (three). Having completed three trajectory correction maneuvers (TCMs) by mid-November 2004, many more maneuvers will be necessary during the journey to Mercury and the subsequent one-year-duration Mercury orbit phase. The spacecraft's design and operational capability will enable real-time monitoring of every TCM. A complex mission plan will provide multiple opportunities to obtain observational data that will help fulfill the mission's scientific objectives. Soon after entering Mercury orbit in mid-March 2011, the initial primary science orbit will have 80° orbit inclination relative to Mercury's equator, 200-km periapsis altitude, 60°N sub-spacecraft periapsis latitude, and a 12-hour orbit period. With science goals requiring infrequent orbit-phase trajectory adjustments, pairs of orbit correction maneuvers occur at about the same time every Mercury year, or every 88 days. For the first time, the spacecraft's orbit design at Mercury accounts for the best available Mercury gravity model, small solar pressure perturbations due to changes in the solar array tilt angle, and an improved strategy for performing orbit correction maneuvers.

INTRODUCTION

Nearly three decades after the world marveled at the Mariner 10 spacecraft's detailed images of Mercury, the MESSENGER (MErcury Surface, Space ENvironment, GEochemistry, and Ranging) spacecraft was launched on August 3, 2004. With delays forcing MESSENGER to the third launch opportunity of 2004, the spacecraft trajectory will utilize one Earth flyby, two Venus flybys, and three Mercury flybys during its 6.6-year ballistic trajectory to Mercury. Having successfully completed three trajectory correction maneuvers (TCMs) by mid-November 2004, the spacecraft will need many more TCMs: five large deterministic maneuvers and 20 to 30 statistical TCMs during the heliocentric orbit, plus two maneuvers for Mercury orbit insertion, and six maneuvers during the one-year-duration Mercury orbit phase.

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With funding and mission oversight coming from NASA's (National Aeronautics and Space Administration) Discovery Program, the MESSENGER mission relied on The Johns Hopkins University Applied Physics Laboratory (JHU/APL) and the Carnegie Institution of Washington for leadership in the design, development, testing, and operation of the spacecraft. The spacecraft, which has a fixed sunshade for protection from sunlight, has a dual-mode (fuel only or a fuel/oxidizer mix) propulsion system for course corrections, two solar arrays and a battery for power, and a suite of science instruments for data collection. With formal NASA approval announced in July 1999, the MESSENGER mission's detailed design began in January 2000. Five years later, the spacecraft is approaching six months of successful operation in space.

The spacecraft's seven science instruments will acquire data in order to address six important questions[1] on Mercury's nature and evolution. Answers to these questions, which will offer insights well beyond increased knowledge of the planet Mercury, are the basis for the science objectives:

1. Map the elemental and mineralogical composition of Mercury's surface.
2. Image globally the surface at a resolution of hundreds of meters or better.
3. Determine the structure of the planet's magnetic field.
4. Measure the libration amplitude and gravitational field structure.
5. Determine the composition of radar-reflective materials at Mercury's poles.
6. Characterize exosphere neutrals and accelerated magnetosphere ions.

The science instruments include the wide-angle and narrow-angle field-of-view imagers of the Mercury Dual Imaging System (MDIS), the Gamma-Ray and Neutron Spectrometer (GRNS), the X-Ray Spectrometer (XRS), Magnetometer (MAG), Mercury Laser Altimeter (MLA), the Mercury Atmospheric and Surface Composition Spectrometer (MASCS), the Energetic Particle and Plasma Spectrometer (EPPS), and an X-band transponder for the Radio Science (RS) experiment. Table 1 shows how these instruments link science objectives to the spacecraft orbit at Mercury. Recent publications[2,3] offer a more comprehensive examination of the structure and function of all seven science instruments.

Table 1. Mapping science objectives into Mercury orbit design

Mission Objectives	Mission Design Requirements	Mission Design Features
Globally image surface at 250-m resolution	Provide two Mercury solar days at two geometries for stereo imaging of entire surface; near-polar orbit for full coverage (MDIS)	Orbital phase of one Earth year (13 days longer than two Mercury solar days) with periapsis altitude controlled to 200-500 km; 80° - inclination orbit
Determine the structure of Mercury's magnetic field	Minimize periapsis altitude; maximize altitude-range coverage (MAG)	Mercury orbit periapsis altitude from 200-500 km, apoapsis altitude near 15,200 km for 12-hour orbital period
Simplify orbital mission operations to minimize cost and complexity	Choose orbit with period of 8, 12, or 24 hours	
Map the elemental and mineralogical composition of Mercury's surface	Maximize time at low altitudes (GRNS, XRS)	
Measure the libration amplitude and gravitational field structure	Minimize orbital-phase thrusting events (RS, MLA)	Orbital inclination 80°; periapsis latitude drifts from 60°N to 72°N; primarily passive momentum management; two orbit correction ΔVs (30 hours apart) every 88 days
	Orbital inclination 80°; latitude of periapsis near 60°N (MLA, RS)	
Determine the composition of radar-reflective materials at Mercury's poles	Orbital inclination 80°; latitude of periapsis maintained near 60°N (GRNS, MLA, MASCS, EPPS)	
Characterize exosphere neutrals and accelerated magnetosphere ions	Wide altitude range coverage; visibility of atmosphere at all lighting conditions	Extensive coverage of magnetosphere; orbit cuts bow shock, magnetopause, and upstream solar wind

When the need arose (twice during the year before launch) to increase spacecraft testing and enhance redundancy, launch was delayed to the next available launch opportunity. Table 2 shows how each launch delay affected selected mission performance parameters. While KinetX and JHU/APL share responsibility for trajectory optimization and maneuver design, C. L. Yen of the Jet Propulsion Laboratory discovered[4,5] each innovative heliocentric transfer trajectory used as launch opportunities for MESSENGER in 2004. Additional studies by Langevin[6] yielded ballistic trajectory options that were useful for MESSENGER contingency plans. In 1999 the next MESSENGER mission design paper[7] depicts the current mission plan (including an Earth flyby one year after launch, but excluding the third Mercury flyby) as a backup option. The March and May launch opportunities during 2004, which each had two Mercury flybys and heliocentric transfer times just over five years, were described by Yen[4] and McAdams[8] in 2001. The first detailed account of MESSENGER's Mercury orbit phase, which appeared in early 2003[9], maps science requirements and spacecraft operational constraints to the trajectory design and propulsive maneuver strategy. The following material is the first account of MESSENGER's current full-mission trajectory and maneuver plan.

Table 2. MESSENGER launch options for 2004

Month	March	May	August	August (launch day)
Launch dates (mm/dd)	3/10-29	5/11-22	7/30-8/13*	**8/3**
Launch period (days)	20	12	15	**-**
Launch energy (km^2/s^2)	≤ 15.700	≤ 17.472	≤ 16.887	**16.388**
Earth flybys	0	0	1	**1**
Venus flybys	2	3	2	**2**
Mercury flybys	2	2	3	**3**
Deterministic ΔV (m/s)	≤ 2026	≤ 2074	≤ 1991	**1966**
Total ΔV (m/s)	2300	2276	2277	**2251****
Orbit insertion date (mm/dd/yy)	4/6/09	7/2/09	3/18/11	**3/18/11**

* The final launch period started August 2 due to delays in the availability of the launch facility.

** Lower total ΔV reflects a reduced propellant load required to meet the spacecraft launch weight limit.

SPACECRAFT DESIGN AND MANEUVER CONSTRAINTS

A description of the spacecraft configuration and operational limitations illuminates MESSENGER's trajectory design and maneuver strategy. The spacecraft has a robust thermal protection subsystem that enables routine execution of propulsive maneuvers at Sun-spacecraft distances from 1.07 AU to near Mercury's perihelion at 0.31 AU, marking a 12-fold difference in the Sun's energy on the spacecraft. Major components of the three-axis-stabilized MESSENGER spacecraft (Figure 1) include two movable solar arrays with 5.3-m^2 maximum Sun-facing area, a 2.5-m by 2.1-m ceramic cloth sunshade, the spacecraft bus, and a magnetometer boom. As the spacecraft reaches Mercury's distance from the Sun (0.30 to 0.46 AU),

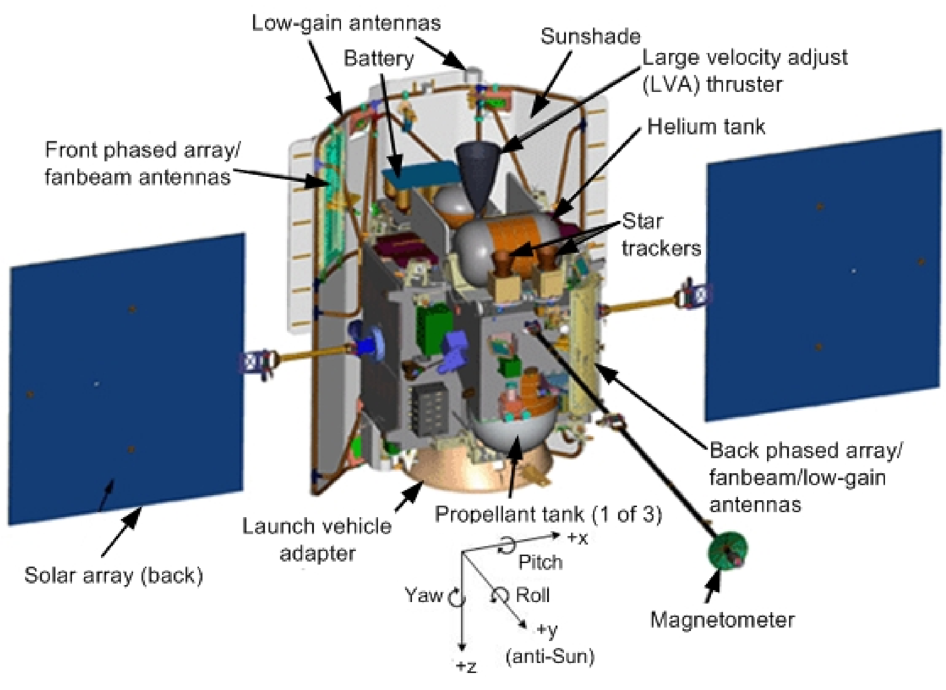

Figure 1. Flight configuration of the MESSENGER spacecraft (excludes thermal blanket covering)

the arrays tilt up to 65º away from the Sun in order to keep the Sun-facing solar array surface temperature below 150ºC. Knowledge of solar array orientation is required for accurate determination of the solar pressure perturbation on the spacecraft's trajectory. A 23-ampere-hour NiH_2 battery provides power during solar eclipse. Battery capacity was a primary consideration for the selection of the lowest sub-spacecraft periapsis latitude. Given the 200-km periapsis altitude, 12-hour orbit period, 60º N initial sub-spacecraft periapsis latitude, and the corresponding predicted northward drift rate, the 61.5-minute maximum solar eclipse duration is well below the 65-minute (conservative limit set during the early design phase) upper bound. The telecommunications subsystem has X-band transponders, four low-gain antennas (required for real-time monitoring of course correction maneuvers), two medium-gain fanbeam antennas, and two high-gain phased-array antennas. The requirement to monitor maneuvers explains why no maneuver may occur during solar conjunction, i.e., periods where the Sun-Earth-spacecraft angle is less than 3º. In addition to keeping maneuvers away from solar conjunction, Earth's orientation relative to the spacecraft body frame must provide a link margin > 3 dB using one of the four low-gain antennas.

Additional features of the spacecraft affect the orientation of the spacecraft during mission-critical events such as course correction maneuvers. After the spacecraft's –y axis is re-oriented to point toward the Sun (planned to begin on March 30, 2005), the sunshade will protect the spacecraft bus from direct sunlight exposure as long as the sunshade center panel's surface normal stays within 12º of the direction to the Sun. Spacecraft rotations in yaw of ±15º and +13.5º to –12.4º in pitch define the operational zone where direct sunlight never impinges on any part of the spacecraft protected by the sunshade. This 12º off-Sun tilt limitation for most of the mission leads to a constraint for large deterministic maneuvers that the Sun-spacecraft-ΔV angle must be between 78º and 102º. This constraint on spacecraft attitude during propulsive maneuvers limits the opportunities for performing efficient orbit correction maneuvers (OCMs) to twice per 88-day Mercury year during Mercury orbit phase. These two opportunities for performing OCMs occur when the spacecraft orbit plane (dark diagonal lines through Mercury in Figure 4) is nearly perpendicular to the Sun-Mercury line. Neglecting solar gravity and small solar radiation pressure perturbation effects, these OCM opportunities arise near Mercury orbit true anomaly angles of 12º (where Mercury orbit insertion occurs) and 192º. Furthermore, since science objectives require highly accurate knowledge of the spacecraft's orbit, the time between OCMs must be maximized. About one Mercury year after the spacecraft's periapsis altitude is 200 km, the periapsis altitude nears the 500-km upper limit expressed in Table 2. For these reasons all OCM pairs occur once every 88 days, when Mercury is near 12º true anomaly. Additional details of the spacecraft design are available from a source[10] prepared a few months prior to the start of spacecraft assembly at JHU/APL.

Detailed monitoring of the propulsion system throughout the mission will provide spacecraft operators with information helpful for maximizing the efficiency of propulsive maneuvers. The dual-mode propulsion system uses hydrazine fuel and nitrogen-tetroxide oxidizer for the 672-N large-velocity-adjust (LVA) thruster (specific impulse of 316 s) and 16 smaller monopropellant thrusters (specific impulse of 200 to 235 s). The monopropellant thrusters include four 26-N LVA-TVC (thrust vector control) on the same deck as the LVA thruster and twelve 4-N thrusters that are used for attitude control and small trajectory correction maneuvers in any direction. Each propulsive maneuver design accounts for the current propulsion system performance to minimize both execution error and propellant consumed for attitude control (not in the direction of the ΔV). The propulsion system's final design and pre-launch test results appear in a paper[11] prepared just over a year before launch.

The most complex maneuver type (bipropellant), which uses the LVA thruster, has four segments that impart ΔV: fuel settle, auxiliary tank refill, main burn, and ΔV trim. For the large deterministic ΔVs during heliocentric transfer, called deep space maneuvers (DSMs), the 15-s-duration fuel-settle burn forces the fuel to the end of the fuel tank needed for the subsequent thruster activity. The settle burn contributes to the overall DSM ΔV by using four of the twelve 4-N hydrazine thrusters. The second burn segment diverts fuel into the auxiliary fuel tank (as needed depending on the fuel already in the auxiliary tank) while continuing 4-N thruster activity. The vast majority of the LVA ΔV achieves up to 672-N thrust with the combustion products of fuel and oxidizer exiting through the LVA thruster. The fourth segment, ΔV trim, provides a precision clean up of the overall ΔV target using all four 26-N LVA-TVC thrusters.

HELIOCENTRIC TRAJECTORY

The spacecraft will utilize one Earth gravity assist, two Venus gravity assists, three Mercury gravity assists, and five major course-correction maneuvers or DSMs during its 6.6-year ballistic trajectory to Mercury (Figure 2). As of Mercury orbit insertion (MOI), the spacecraft will have completed more than 15 orbits of the Sun and traveled 7.9 billion kilometers relative to the Sun. Unlike both earlier launch opportunities in 2004, less ideal Earth-Venus and Venus-Mercury phasing requires addition of a DSM for each of these legs. Allocation of propellant for these first two DSMs was offset by adding a third Mercury gravity assist and subsequent DSM to achieve a 47% reduction in MOI ΔV. A 1:1 Venus:spacecraft orbit resonance between the Venus flybys indicates one orbit of the Sun for both Venus and the spacecraft. Similarly, a 2:3 Mercury:spacecraft orbit resonance occurs between Mercury flybys 1 and 2, a 3:4 Mercury:spacecraft orbit resonance occurs between Mercury flybys 2 and 3, and a 5:6 Mercury:spacecraft orbit resonance occurs between Mercury flyby 3 and MOI. Although the Planetary Flybys section will explain the trajectory shaping achieved by each flyby, the goal is to minimize propellant usage while decreasing the relative velocity difference between the spacecraft and Mercury at orbit insertion.

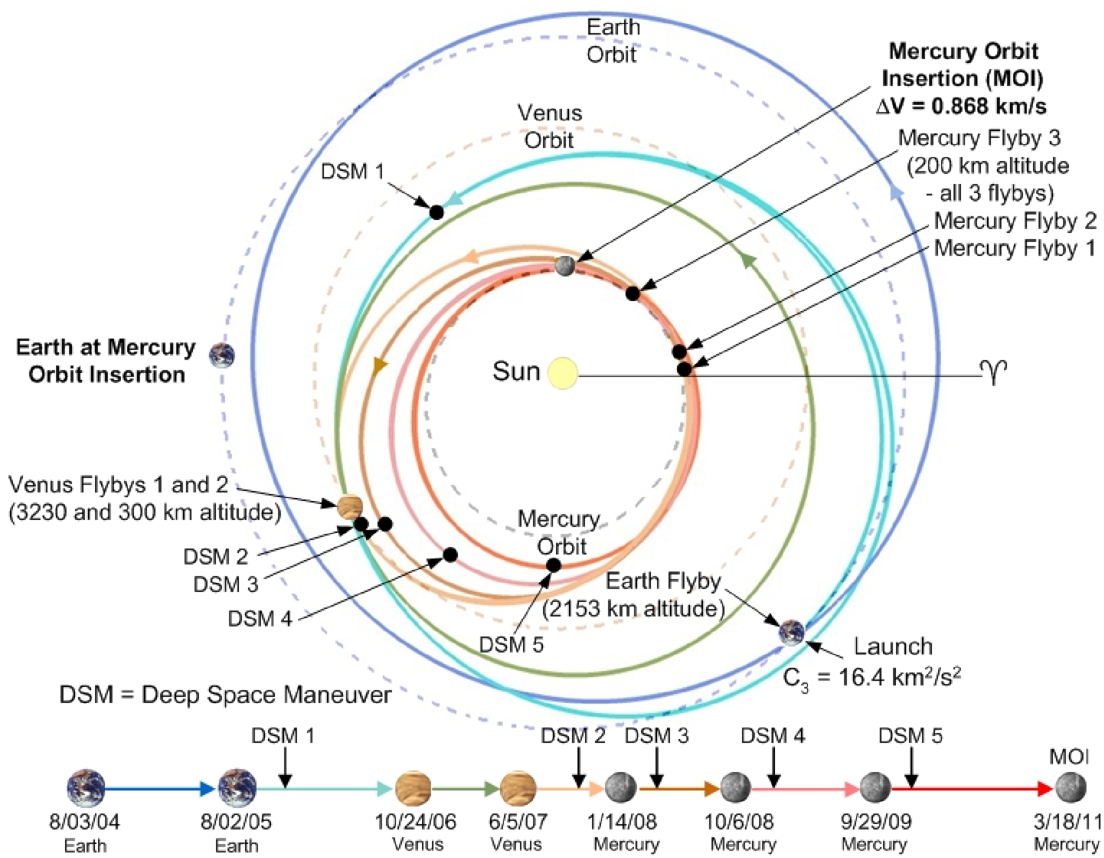

Figure 2. North ecliptic pole view of MESSENGER's heliocentric trajectory

The spacecraft's functional design, operations plan, and trajectory navigation strategy account for a number of solar conjunctions. Solar conjunctions for spacecraft that usually remain closer to the Sun than Earth, as MESSENGER does, are regions where Sun-Earth-spacecraft angle < 3º. During solar conjunction, the Sun either degrades or prevents spacecraft communication with ground stations – whether the spacecraft is between Earth and the Sun (inferior conjunction) or on the opposite side of the Sun (superior conjunction). The spacecraft will function without operator intervention throughout each solar conjunction, even those near important spacecraft activities. For MESSENGER the longest solar conjunction, lasting about 1.5 months, begins several days before the first Venus flyby. A Venus close-approach altitude of over 3200 km, along with the absence of major mission-critical events for more than seven months, greatly reduces the risk associated with this solar conjunction. During MESSENGER's heliocentric trajectory another lengthy solar conjunction, lasting about five weeks, occurs several days after DSM 2. The Mercury Orbit section addresses the significance of solar conjunctions during the spacecraft's first year at Mercury.

LAUNCH

On August 3, 2004, at 06:15:56.5 UTC, MESSENGER became the second planetary mission launched aboard a Delta II 7925H launch vehicle from Cape Canaveral Air Force Station. With the launch service provider (Boeing) working closely with NASA Kennedy Space Center and JHU/APL, the 1107.25-kg spacecraft departed Earth orbit with a 16.388 km^2/s^2 launch energy at a -32.66° declination of launch asymptote (DLA) relative to the Earth mean equator at the standard J2000 epoch. The large DLA was chosen to limit the mission's aphelion distance to 1.077 AU, thereby meeting minimum power margin requirements. While the first hour after launch (flight path appears in Figure 3) closely followed the planned trajectory, the larger-than-average 2.0-σ under burn required the Navigation Team to quickly provide Deep Space Network tracking stations with early orbit determination solutions to improve their antenna pointing to the spacecraft. Before the spacecraft separated from the third stage, Boeing's third-stage despin slowed the spacecraft from ~58 revolutions per minute (rpm) to 0.015 rpm, far below the 2 rpm tolerance. After separation the solar panels deployed and the spacecraft pointed its sunshade away from the Sun, thereby saving heater power by using sunlight to warm the spacecraft bus when the spacecraft-Sun distance is near 1 AU.

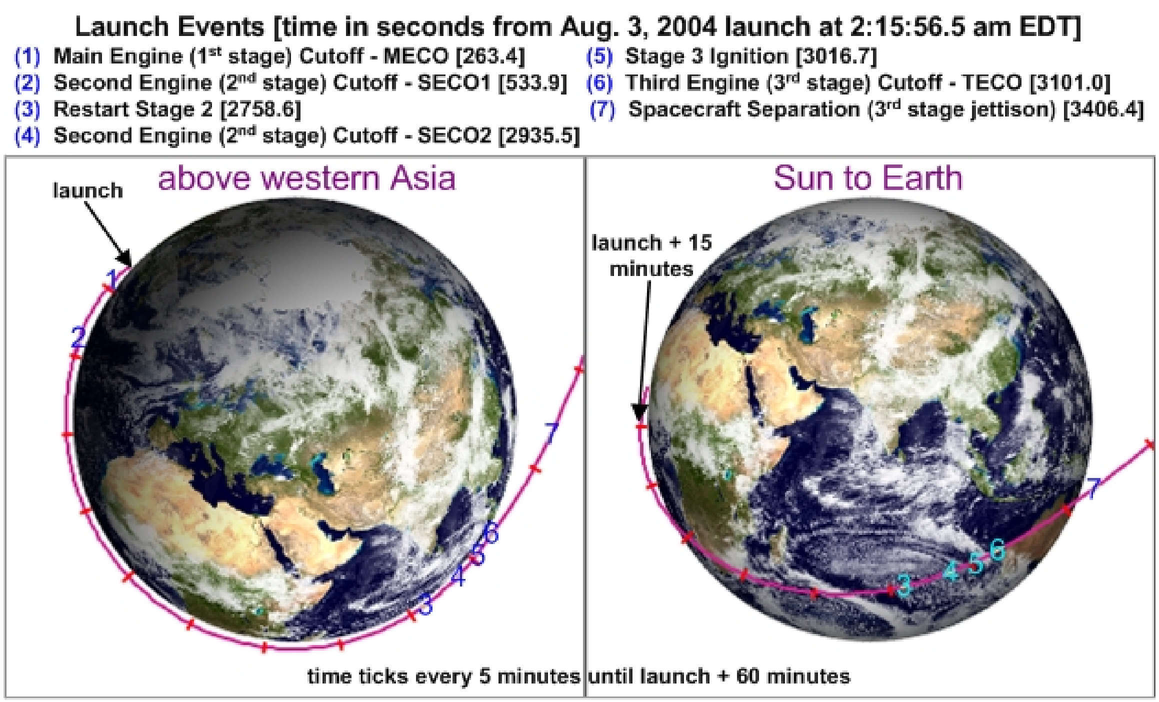

Figure 3. Launch trajectory for the first hour after launch

MANEUVER STRATEGY FOR THE HELIOCENTRIC TRANSFER

The MESSENGER spacecraft's heliocentric trajectory requires five deterministic maneuvers with ΔV magnitude > 50 m/s (DSMs), two smaller deterministic maneuvers, and close to 30 statistical TCMs for correcting maneuver execution errors and planetary flyby target errors. The more efficient LVA bipropellant thruster will be the primary thruster for each DSM. Most maneuvers requiring ΔVs of 3 m/s to 20 m/s between 78º and 102º away from the Sun-to-spacecraft direction will be performed using the four 26-N LVA-TVC thrusters. These four thrusters also serve as the primary attitude control thrusters during DSMs. Maneuvers with ΔV direction within 12º of the Sun-to-spacecraft direction will be performed using a pair of 4-N thrusters mounted on the sunward or anti-Sun sides of the spacecraft. Maneuvers with ΔV directions outside the above restrictions will be performed less efficiently using vector components of a combination of two of the above maneuver types.

The ΔV allocation for MESSENGER is listed by category in Table 3. The minimum ΔV for contingencies, and the corresponding nominal ΔV for deterministic maneuvers, are also shown. With nearly 40% of the total ΔV allocated to Mercury orbit insertion, extensive studies have been and will be conducted to ensure a safe and efficient MOI with well-formulated contingency plans. Similarly, contingency plans have been developed and will be expanded for each DSM. The ΔV budget for launch vehicle and navigation errors is a 99th percentile value determined by KinetX using Monte Carlo analyses. Since this ΔV budget applies only to the heliocentric transfer phase, the "contingency" category includes variations in expected ΔV for Mercury orbit correction maneuvers. The 1966 m/s deterministic ΔV from Table 1 includes DSMs, 6 m/s deterministic ΔV from the "navigation errors" category, MOI, and Mercury OCMs.

Table 3. ΔV allocation after the fourth post-launch trajectory reoptimization

	ΔV (m/s)
Deep space maneuvers	1008
Launch vehicle, navigation errors (99%)	121
Mercury orbit insertion	868
Mercury orbit correction maneuvers	85
Contingency	169
Total	**2251**

Maneuvers planned for the heliocentric transfer will exercise propulsion modes with all three thruster types (4-N, 26-N, and 672-N) enabled as primary thrusters. While the first three TCMs utilized the 26-N thrusters, TCM-4 (planned for May 2005) will only use 4-N thrusters, and every DSM will exercise the 672-N bipropellant thruster. Both completed and planned heliocentric course correction maneuvers, excluding the statistical maneuvers with unknown ΔV, are summarized in Table 4. With ΔV magnitude error averaging 0.3% and ΔV direction error averaging 0.3º for the first three TCMs, the MESSENGER mission has successfully removed launch dispersions and completed preliminary Earth flyby targeting.

MESSENGER's DSMs serve two primary purposes: helping with Earth-Venus and Venus-Mercury phasing, and moving the next Mercury encounter closer to Mercury's location at MOI. As the only DSM near perihelion, DSM-1 increases the spacecraft's speed relative to the Sun, thereby raising aphelion and establishing the spacecraft's Venus flyby 1 arrival date. The first DSM near aphelion, DSM-2, also reduces the spacecraft's Sun-relative speed; this maneuver lowers perihelion enough to set up the first Mercury flyby. From the Mercury encounter locations on Figure 2 and the DSM ΔV magnitudes from Table 4, note how DSM-3 to -5 magnitude is directly proportional to the change in Mercury's position between the previous and next Mercury encounters. The final three DSMs shift the following Mercury encounter position counterclockwise (as viewed from north of the ecliptic plane) by slightly increasing the spacecraft's Sun-relative speed. Since DSM-3 to -5 occur near aphelion, each maneuver raises perihelion slightly. If a spacecraft anomaly or ground station outage causes a delay for either of the first two DSMs, the DSM could be delayed up to 11 days (until reaching the sunshade orientation constraint on the Sun-spacecraft-ΔV angle). More flexibility exists for DSM-3 to -5, since each DSM can occur a few months later during the next heliocentric orbit.

Table 4. Deterministic maneuvers during heliocentric transfer

Name	Status	Maneuver Date	Earth Range (AU)	Sun Range (AU)	Sun-S/C-ΔV (deg)	Sun-Earth -S/C (deg)	ΔV (m/s)
Requirement (DSMs only) →					**(78° to 102°)**	**(> 3°)**	
TCM-1	complete	2004 Aug 24	0.000	1.015	93.2	124.2	17.90
TCM-2	complete	2004 Sep 24	0.051	1.067	92.8	118.0	4.59
TCM-3	complete	2004 Nov 18	0.124	1.071	88.6	103.2	3.25
TCM-4	planned	2005 May 5	0.212	0.924	TBD	60.9	0.20
DSM-1	**planned**	**2005 Dec 13**	**0.716**	**0.603**	**89.8**	**37.5**	**325.68**
TCM-15	planned	2007 Jan 23	1.846	0.871	86.0	5.2	2.90
DSM-2	**planned**	**2007 Oct 22**	**1.694**	**0.703**	**87.4**	**3.4**	**189.80**
DSM-3	**planned**	**2008 Mar 17**	**0.678**	**0.685**	**87.5**	**43.3**	**74.18**
DSM-4	**planned**	**2008 Dec 06**	**1.596**	**0.626**	**89.1**	**6.3**	**241.61**
DSM-5	**planned**	**2009 Nov 29**	**1.529**	**0.565**	**89.4**	**7.4**	**176.59**

PLANETARY FLYBYS

In order to minimize the fuel needed from launch to MOI, the spacecraft's trajectory derives most of the required modification from one Earth flyby, two Venus flybys, and three Mercury flybys. The Earth flyby and second Venus flyby will provide the project's scientists with unique opportunities for in-flight instrument calibration and the mission operators with opportunities to practice command and observation sequences needed for the Mercury flybys. Since the first Venus flyby will occur during solar conjunction, these opportunities will not be possible there. Due to the higher encounter altitudes and low approach phase angles at the Venus flybys, optical navigation (OpNav) images are not required to ensure accurate flyby targeting. OpNav image-taking sequences will be practiced prior to the low-altitude Mercury flybys, where OpNavs are mission-critical for accurate and risk-minimal flyby targeting. Another technique for reducing orbit uncertainty, using Deep Space Network delta-differential one-way ranging[12], will be used twice each week for about a month starting five to six weeks before each of the Venus and Mercury flybys.

Table 5 shows how each major trajectory adjustment (primarily the planetary gravity assists) contributes toward the goal of reducing the spacecraft's velocity relative to Mercury at orbit insertion. The table lists orbital parameters that most affect the velocity difference that the spacecraft's propulsion system must correct in order to enter into orbit around Mercury. By minimizing the difference in these parameters

Table 5. Effect of planetary gravity assists and DSMs on the heliocentric transfer orbit

Event Name	Longitude of Perihelion(LP)	LP to goal	Orbit (OI) Inclination	OI to goal	Perihelion Dist. (PD)	PD to goal (AU)	Aphelion Dist.(AD)	AD to goal (AU)
Launch	**194º**	117º	**6.5º**	0.5º	**0.923 AU**	0.615	**1.077 AU**	0.610
Earth Flyby	**133º**	56º	**2.6º**	4.4º	**0.603 AU**	0.295	**1.015 AU**	0.648
DSM-1	**-**	-	**-**	-	**-**	-	**1.054 AU**	0.687
Venus Flyby 1	**105º**	28º	**7.9º**	0.9º	**0.546 AU**	0.238	**0.901 AU**	0.434
Venus Flyby 2	**47º**	30º	**6.7º**	0.3º	**0.332 AU**	0.024	**0.745 AU**	0.278
DSM-2	**-**	-	**-**	-	**0.325 AU**	0.017	**-**	-
Mercury Flyby 1	**56º**	21º	**7.0º**	0.0º	**0.313 AU**	0.005	**0.700 AU**	0.233
DSM-3	**-**	-	**-**	-	**0.315 AU**	0.007	**-**	-
Mercury Flyby 2	**68º**	9º	**7.0º**	0.0º	**0.302 AU**	0.006	**0.630 AU**	0.163
DSM-4	**-**	-	**-**	-	**0.309 AU**	0.001	**-**	-
Mercury Flyby 3	**81º**	4º	**7.0º**	0.0º	**0.303 AU**	0.005	**0.567 AU**	0.100
DSM-5	**-**	-	**-**	-	**0.308 AU**	0.000	**-**	-
Mercury orbit (goal)	**77º**	**-**	**7.0º**	**-**	**0.308 AU**	**-**	**0.467 AU**	-

between the spacecraft's orbit and Mercury's orbit, the velocity change required for the spacecraft at orbit insertion is reduced. Because the spacecraft and Mercury are in orbits tilted less than 8º relative to Earth's orbit, the longitude of perihelion is approximately the angle from the Sun-Earth direction at the autumnal equinox (increasing counterclockwise) to the Sun-object direction when the object is closest to the Sun (perihelion). The exact definition of longitude of perihelion is the sum of the longitude of the ascending node and the argument of perihelion. Unlike missions that utilize planetary gravity assists to reach the outer planets, each of MESSENGER's gravity-assist flybys decelerates the spacecraft relative to its motion around the Sun. This immediate effect at the point where the gravity assist occurs should not be confused with the overall effect of the gravity assists, which decreases the spacecraft's heliocentric energy and thus increases the spacecraft's Sun-relative average orbital speed by nearly 60% (average orbital speeds relative to the Sun are 29.8 km/s for Earth and 47.9 km/s for Mercury). Since DSMs 2 to 5 occur near aphelion, Table 5 only shows the resulting perihelion adjustment. Note also that the time required for each spacecraft orbit about the Sun (orbit period) decreases from 365 to 266 days after the Earth flyby, to 225 days after Venus flyby 1, to 144 days after Venus flyby 2, to 132 days after Mercury flyby 1, to 116 days after Mercury flyby 2, and to 105 days after Mercury flyby 3. Mercury orbits the Sun every 88 days.

Earth

One year after launch an Earth flyby (Figure 4) provides significant trajectory shaping by lowering perihelion to 0.6 AU from the Sun and by moving perihelion direction more than 60° closer to that of Mercury. The Earth flyby enables the launch vehicle to lift a slightly heavier spacecraft (more propellant for a greater ΔV margin), because maximum DLA for the August 2005 launch period would be slightly higher than that used in August 2004. With TCM-1 dedicated to correcting launch dispersions, launch energy shortfalls for the 2004 launch create a situation where TCM-1 ΔV decreases after launch (versus increases

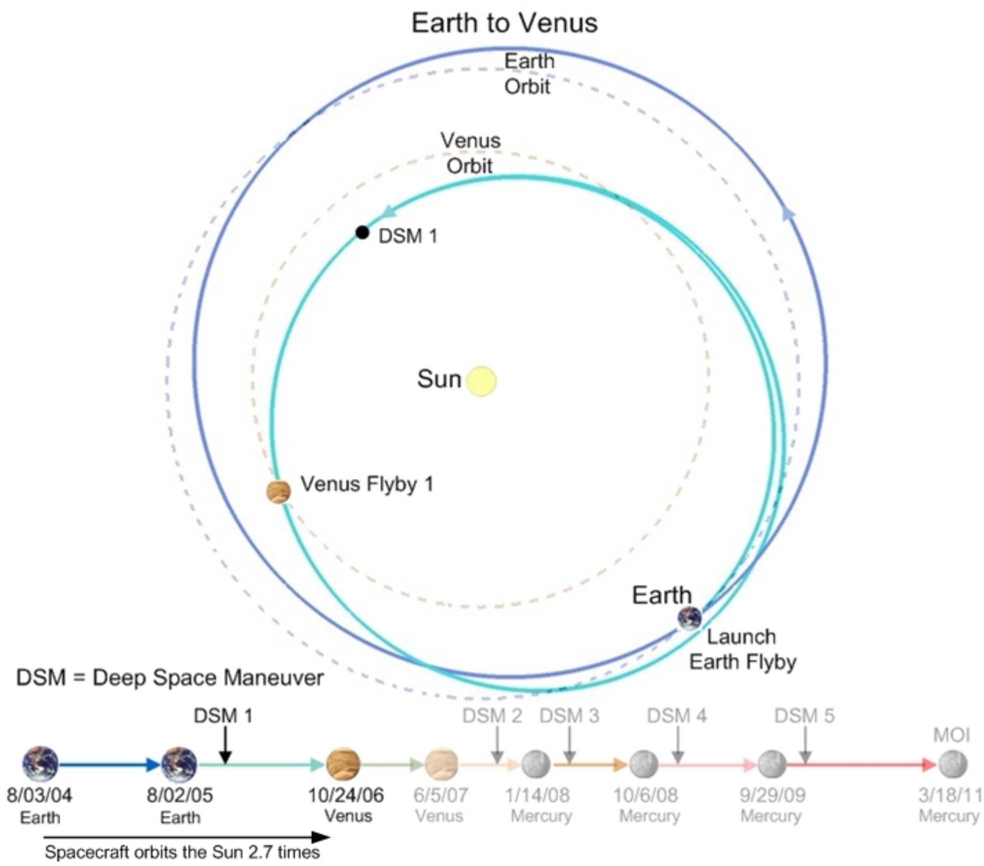

Figure 4. North ecliptic pole view of the Earth to Venus flyby 1 trajectory

for an August 2005 launch), thereby allowing a more complete checkout of spacecraft health before TCM-1. The Earth flyby creates opportunities for calibrating certain science instruments using the Moon, thereby removing science observations from the early post-launch operations schedule. Close approach for Earth (Figure 5) will occur 2155 km over central Mongolia, high enough to avoid solar eclipse.

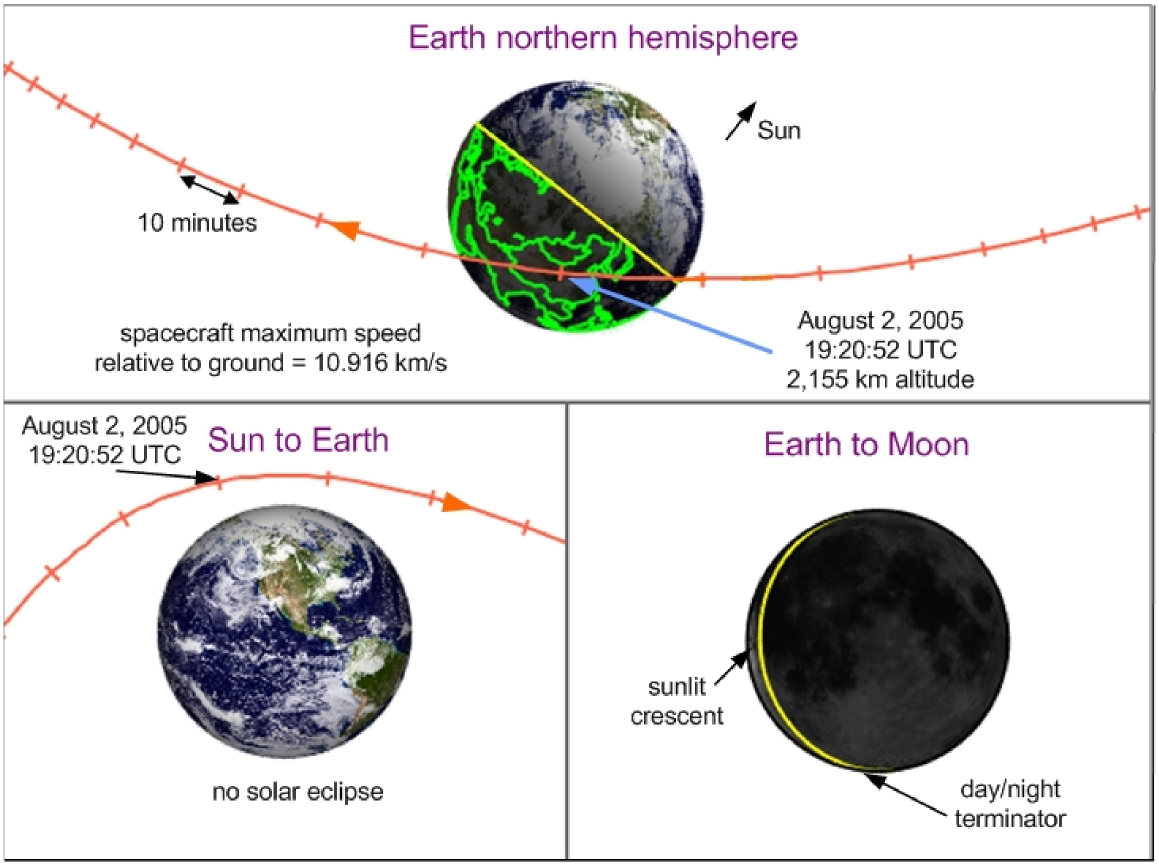

Figure 5. Views of MESSENGER's Earth flyby trajectory (at close approach time)

Venus

The primary purposes of the first Venus flyby (Table 5, Figures 6 and 7) are to increase the spacecraft orbit's inclination and to reduce the spacecraft's orbit period. The spacecraft's orbit inclination must increase to 7.9º (beyond the desired 7.0º) to position the second Venus flyby at the same point in Venus' orbit one Venus year later. Although the spacecraft will approach a brightly illuminated Venus, a 1.4° Sun-Earth-spacecraft angle will almost certainly prevent any reliable transmission of data to or from the spacecraft near close approach. Careful planning by mission operators will ensure that the spacecraft is prepared not only for the 56-minute solar eclipse, but also for not "hearing" from flight controllers for another month (at the end of solar conjunction). The time between each Venus flyby, and the heliocentric location of Venus relative to where Mercury must be for the first Mercury flyby, are carefully optimized to minimize the spacecraft's propellant requirement for course corrections and Mercury orbit insertion.

The second Venus flyby (Figure 8) is the first to lower perihelion enough to permit a Mercury flyby. Both Venus flybys move the spacecraft's aphelion and perihelion significantly closer to Mercury's perihelion and aphelion. Although the spacecraft's hyperbolic excess velocity relative to Venus (9.07 km/s) differs by only about 3 m/s for Venus flybys 1 and 2, the much lower minimum altitude for Venus flyby 2 increases the flyby speed relative to the Venutian surface by about 10% (from 12.337 to 13.585 km/s).

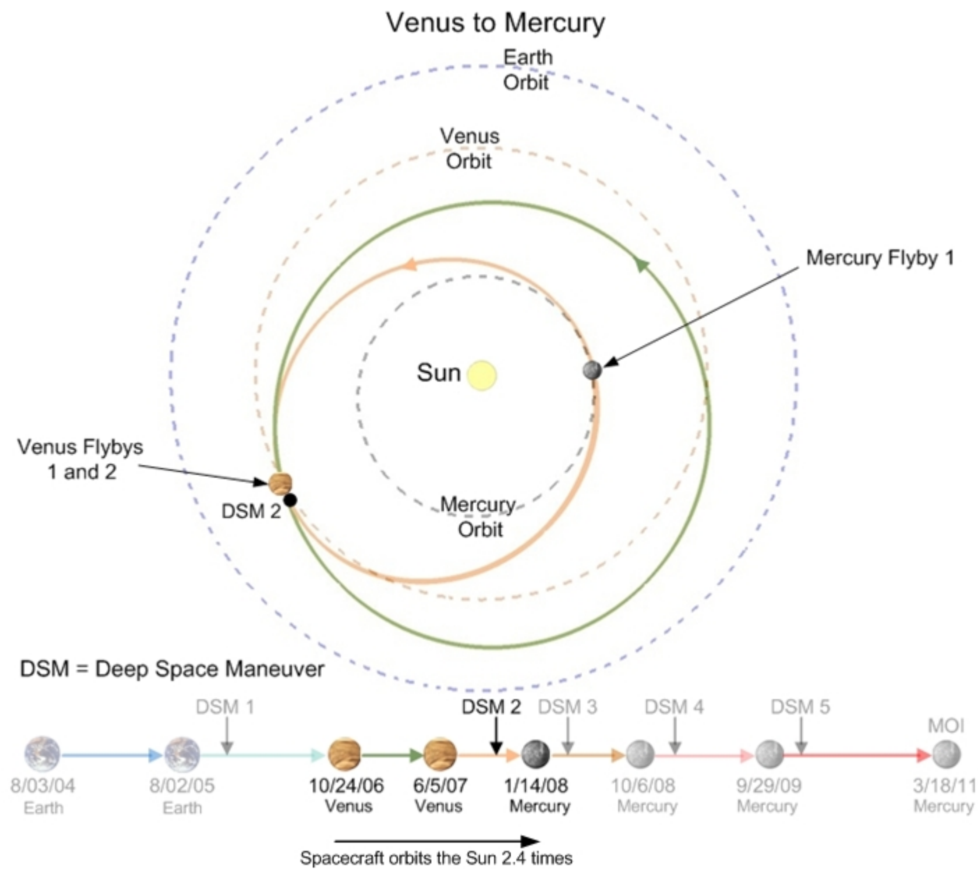

Figure 6. North ecliptic pole view of the Venus flyby 1 to Mercury flyby 1 trajectory

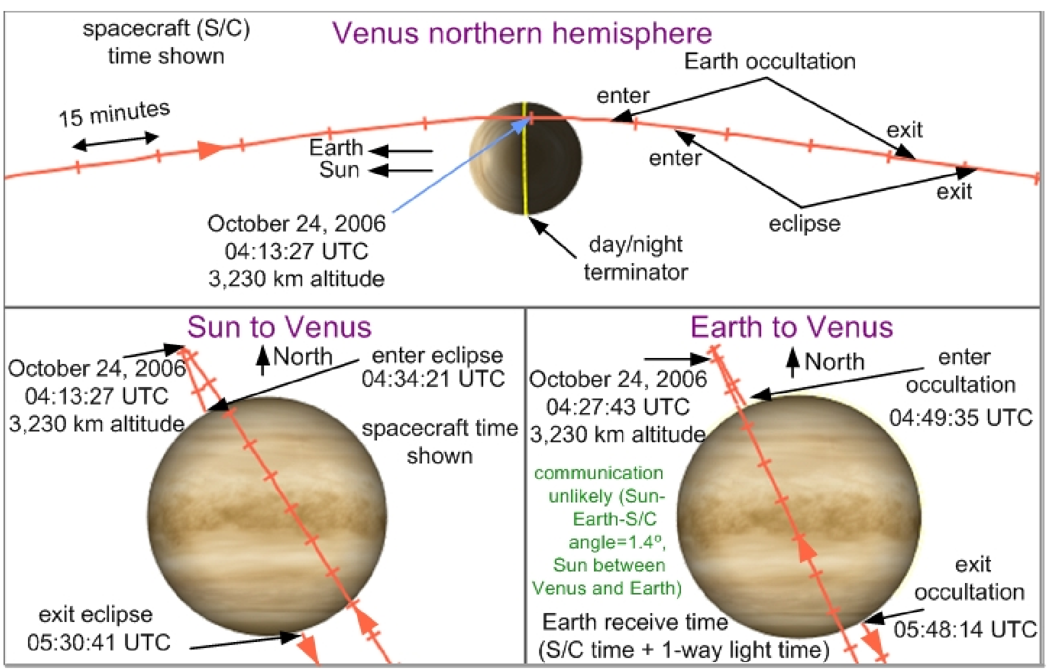

Figure 7. Views of MESSENGER's first Venus flyby trajectory

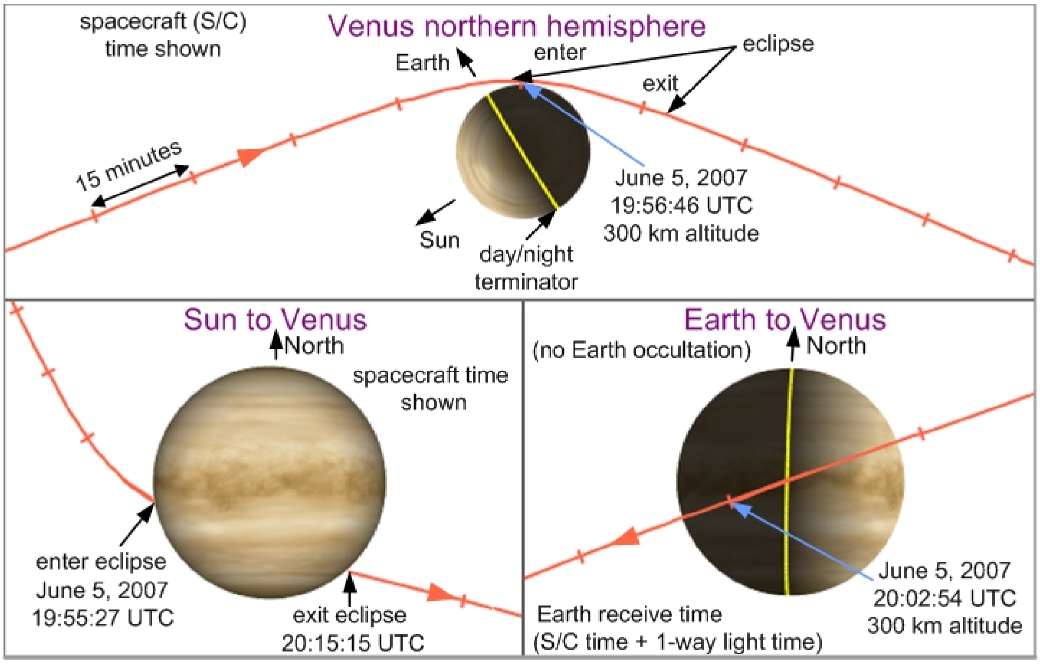

Figure 8. Views of MESSENGER's second Venus flyby trajectory

Mercury

Although the MESSENGER spacecraft's journey to Mercury orbit appears excessively long, the Mercury flybys (Figure 9) are mission enabling. For direct transfers (without gravity assists) from Earth to Mercury at the lowest possible launch energy (more than three times higher than the MESSENGER spacecraft's launch energy), the ΔV required for MOI is close to 10 km/s. The MOI ΔV required for zero, one, two, and three Mercury gravity assists is at least 3.13 km/s, 2.40 km/s, 1.55 km/s, and 0.86 km/s, respectively. A more realistic estimate of the MOI ΔV required to achieve an orbit compatible with MESSENGER science goals is greater than all but the last (which corresponds to the MESSENGER spacecraft's nominal flight path) of these values. Trajectories with fewer than two Mercury gravity assists would lead to catastrophic spacecraft overheating during MOI and the following orbit phase.

Three 200-km altitude Mercury flybys (Figures 10-12), followed two months later by DSMs, will complete the spacecraft orbit rotation and change the orbit's size closer to Mercury's orbit, thereby enabling MOI in March 2011. The Mercury flybys and subsequent DSMs will produce successive spacecraft: Mercury orbital resonance of about 2:3, 3:4, and 5:6 (i.e., the spacecraft orbits the Sun five times while Mercury completes six orbits). This sequence of orbital resonances reduces spacecraft-Mercury hyperbolic excess velocity - V_∞ (Table 6). "Phase" in Table 6 is the Sun-Mercury center-S/C angle. Although the third Mercury flyby rotates the spacecraft's longitude of perihelion 4° past Mercury's longitude of perihelion (Table 5), this extra orbit rotation must occur to achieve the 5:6 spacecraft:Mercury orbital resonance.

Table 6. Mercury encounter summary

Event	Phase, C/A* - 1 day (deg)	Phase, C/A + 1 day (deg)	Speed relative to surface (km/s)	V_∞ (km/s)	Sun-Earth-S/C (deg)	Earth Range (AU)
Mercury 1	117	51	7.113	5.822	16.5	1.157
Mercury 2	127	36	6.594	5.175	2.3	0.659
Mercury 3	104	40	5.303	3.380	15.4	0.798
MOI	94	-	4.363**	2.200	17.4	1.029

* C/A = close approach ** occurs at 288-km altitude after completing the first 30% of MOI-1

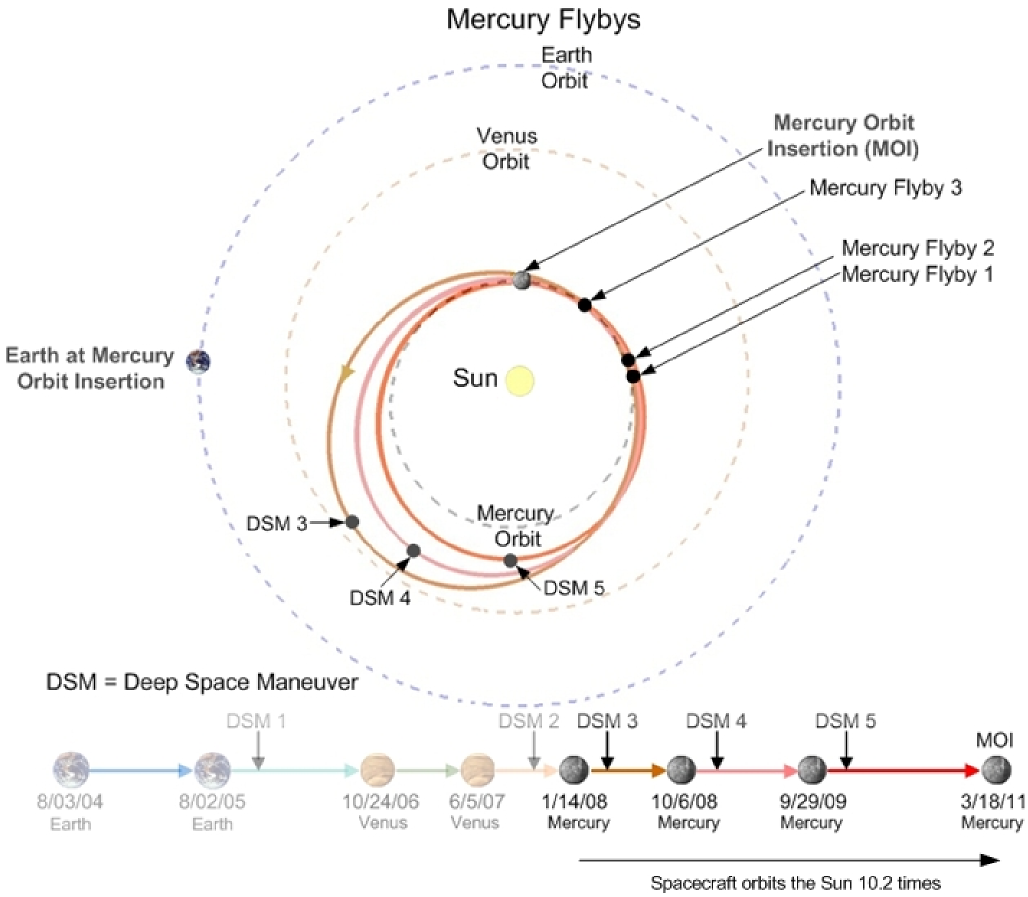

Figure 9. North ecliptic pole view of the Mercury flyby 1 to Mercury orbit insertion trajectory

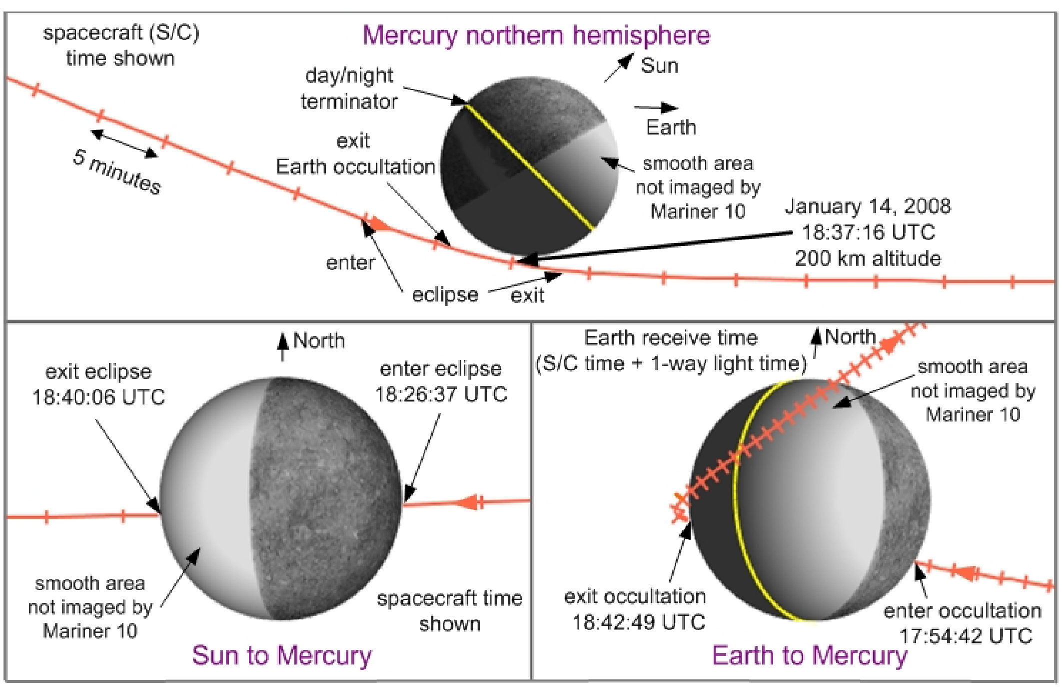

Figure 10. Views of MESSENGER's first Mercury flyby trajectory

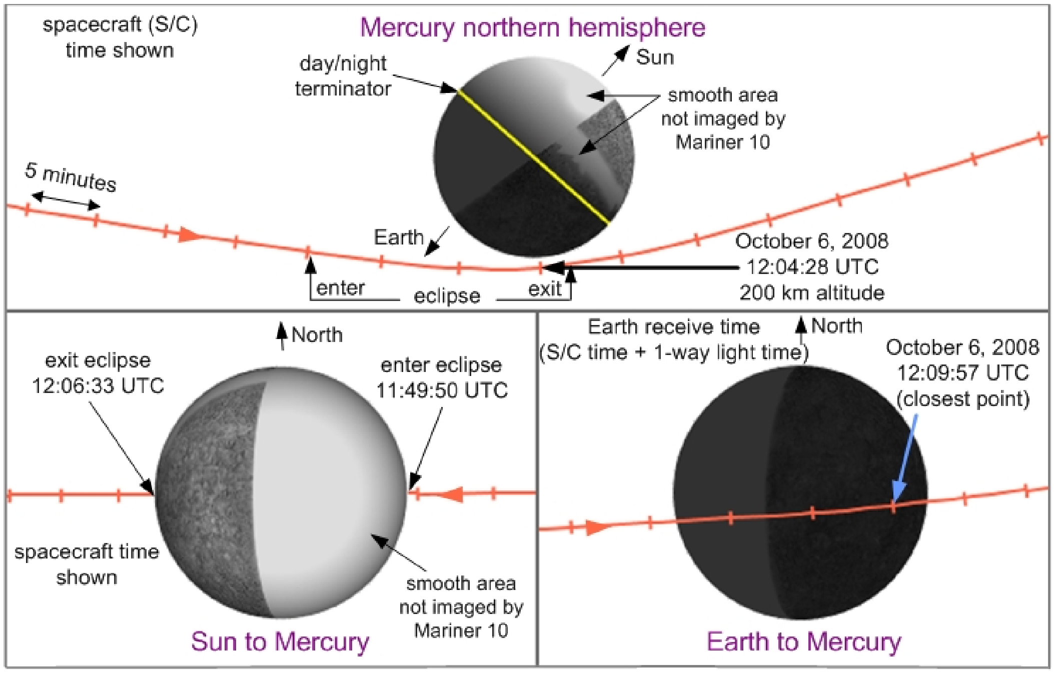

Figure 11. Views of MESSENGER's second Mercury flyby trajectory

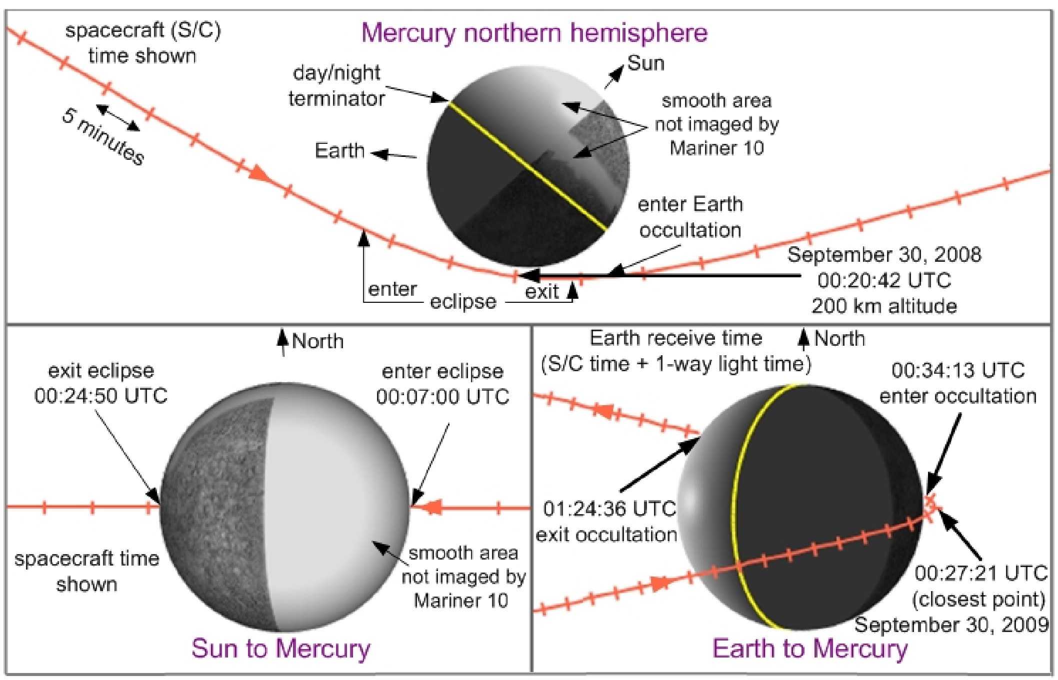

Figure 12. Views of MESSENGER's third Mercury flyby trajectory

Approach and departure views of Mercury for the three Mercury flybys (Figures 10-12) offer a glimpse of the regions of Mercury's surface that are observable in favorable solar illumination conditions. Figures 10 and 11 show how the spacecraft will observe opposite sides of the never-before-imaged (bright, featureless surface in Figures 10-12) hemisphere of Mercury soon after close approach. Since about 1.5 Mercury solar days (176 Earth days per Mercury solar day) elapses between Mercury 1 and 2, the spacecraft will view opposite hemispheres in sunlight. With two Mercury solar days between Mercury 2 and 3, the same hemisphere is sunlit.

MERCURY ORBIT INSERTION

The MESSENGER spacecraft's initial primary science orbit is required to have an 80º (±2º) orbit inclination relative to Mercury's equator, 200-km (±25 km) periapsis altitude, 12-hour (±1 minute) orbit period, 118.4º argument of periapsis (60º N periapsis latitude with 56º N to 62º N acceptable), and a 348º (169º to 354º) longitude of ascending node. These requirements, expressed in Mercury-centered inertial coordinates of epoch January 1.5, 2000, are derived from science and engineering requirements along with characteristics of the Mercury arrival geometry. While the optimal heliocentric trajectory provides 49º N initial periapsis latitude, MOI-1 start time and thrust direction are adjusted to obtain the remaining 11º N rotation of the line of apsides to end at 60º N.

The Mercury orbit-insertion strategy maximizes the probability of successfully delivering the spacecraft into the primary science orbit in the minimum time possible within mission planning process constraints. This strategy uses two "turn while burning" variable thrust-direction maneuvers (MOI-1 and MOI-2) using the LVA thruster operating at 672-N thrust and 316.1-s specific impulse, such that the spacecraft completes four full orbits of Mercury from MOI-1 to MOI-2. With each maneuver slowing the spacecraft's Mercury-relative velocity, the thrust vector is almost opposite to the spacecraft velocity vector. In order to minimize Mercury approach orbit determination uncertainty, the approach trajectory is designed to insure the availability of bright stars in the best orientation near Mercury's limb for approach Opnav images. Figure 13 shows the location of MOI-1 and MOI-2 and depicts the shape, size, and orientation of the first five orbits. The maneuver start times, durations, and ΔV magnitudes will be updated periodically to account for small changes in expected spacecraft mass, approach velocity, and Mercury orbit daily downlink time.

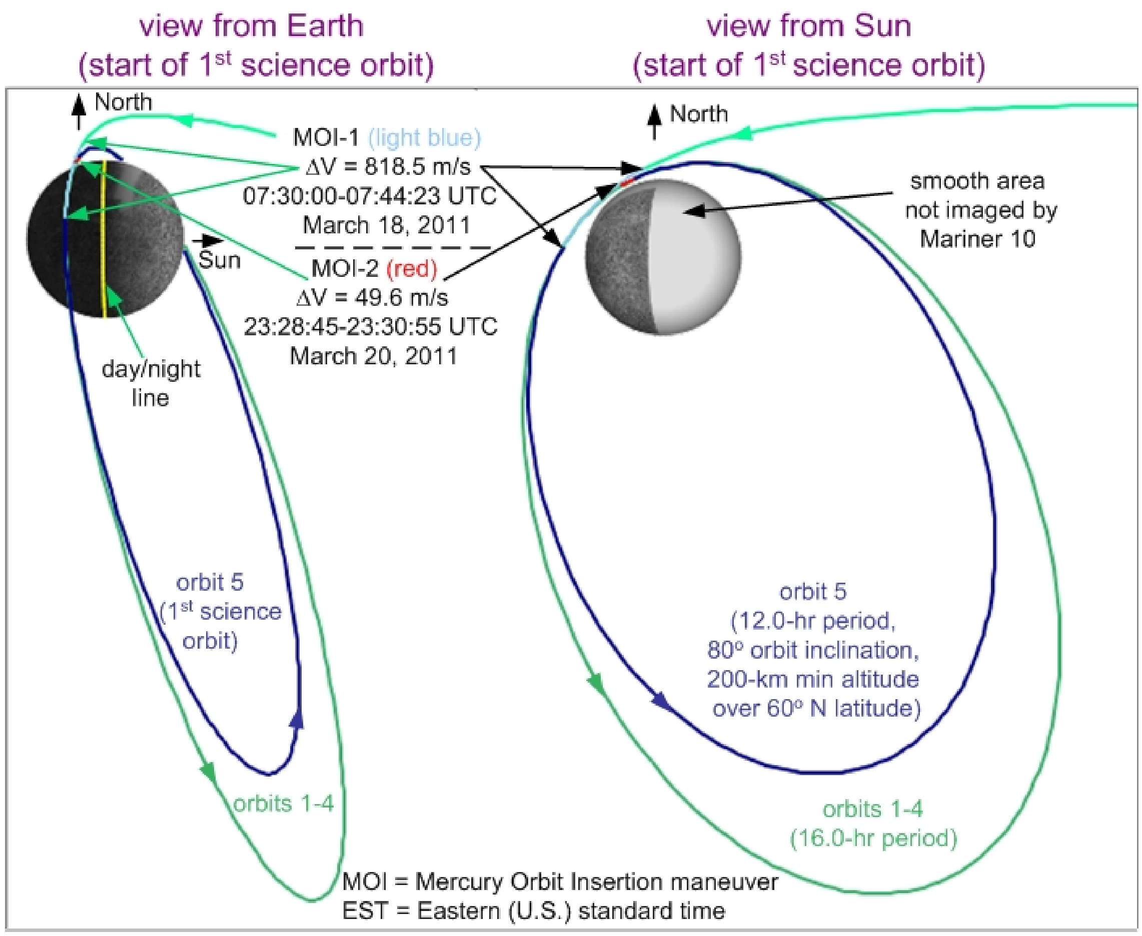

Figure 13. Mercury orbit insertion trajectory and maneuvers

The initial orbit period of the spacecraft at Mercury between MOI-1 and MOI-2 is between 12.8 and 16.0 hours, providing four phasing orbits stable enough to determine reliable post-MOI-1 timing updates for MOI-2. Orbits with longer periods produce unacceptable drift rates in both orbit period and periapsis altitude. The choice of four phasing orbits between MOI-1 and MOI-2 gives 51-64 hours for preliminary MOI-1 maneuver performance assessment, orbit determination, MOI-2 maneuver design, simulation, and maneuver update upload and verification. The 12.8-hour lower limit also ensures that too long of a burn duration for MOI-1 will not produce an initial orbit period less than the 12.0-hour target for the primary science orbit. The initial phasing orbit period range also ensures that daily data downlink periods (beginning four hours before apoapsis on every other 12-hour orbit) start near 8:30 am to 4:30 pm EST. A Sun-Earth-spacecraft angle > 17º ensures that solar interference will not corrupt communications with the spacecraft during the orbit insertion process.

Because MOI occurs as Mercury is near its perihelion, Mercury's high rate of heliocentric angular motion will quickly rotate the Sun-relative spacecraft orbit orientation until the sunshade is unable to protect the spacecraft bus at the required MOI-2 burn attitude. Although this can occur within three days after MOI-1, adding a ΔV component normal to the orbit plane will minimally alter orbit characteristics such as inclination, while enabling the sunshade protection (Sun-spacecraft-ΔV) constraint to be met. This option, required for OCM-2, costs less than 1 m/s more than the minimum in-plane ΔV.

The Mercury orbit insertion summary in Table 7 demonstrates compliance of each orbit insertion maneuver with constraints. Analysis using a 15-s increment for each ΔV further verified constraint compliance. With the current orbit insertion design, the minimum link margin drops below 3 dB (to no lower than 1.8 dB) during the last two to three minutes of the 14.5-minute duration MOI-1. No other planned MESSENGER course correction maneuver has a link margin below 3 dB. The propellant settle/auxiliary tank refill segment duration increases from 45 s to 75 s from MOI-1 to MOI-2 due to the longer settling time (partly due to the large MOI-1 ΔV expelling much of the spacecraft's mass).

Table 7. Mercury orbit insertion meets operational requirements

Orbit Insertion Maneuver Segment	Maneuver Date Year Month Day	Maneuver Time (UTC) hh:mm:ss	ΔV (m/s)	Earth-S/C Range (AU)	Sun-S/C Range (AU)	Sun-S/C-ΔV Angle (deg)	Sun-Earth -S/C Angle (deg)
Requirement →						**(78° to 102°)**	**(> 3°)**
MOI-1 settle/refill	2011 Mar 18	02:30:00	0.2	1.024	0.309	94.6	17.4
MOI-1 LVA start	2011 Mar 18	02:30:45	815.7	1.024	0.309	94.8	17.4
MOI-1 LVA middle	2011 Mar 18	02:37:30	---	1.024	0.309	95.6	17.4
MOI-1 trim end	2011 Mar 18	02:44:26	2.6	1.024	0.309	95.6	17.4
MOI-2 settle/refill	2011 Mar 20	23:28:45	0.5	0.956	0.314	78.8	18.4
MOI-2 LVA start	2011 Mar 20	23:30:00	46.4	0.956	0.314	79.9	18.4
MOI-2 trim end	2011 Mar 20	23:30:55	2.7	0.956	0.314	80.9	18.4
	orbit insertion total ΔV =		**868.1**				

MERCURY ORBITAL OPERATIONS

After completing Mercury orbit insertion, the spacecraft begins a more than 12-week long coast phase without orbit correction maneuvers (OCMs). This will be enough time to refine Mercury's gravity model and update perturbing force models in order to further minimize trajectory propagation errors. Occasional thruster firings for adjusting spacecraft angular momentum will perturb the trajectory by at most a few mm/s of unintentional ΔV.

Each pair of OCMs will return the spacecraft to the initial primary science orbit's size and shape. Solar gravity, solar radiation pressure, and subtle spatial variations in Mercury's gravity will alter orbit orientation by moving periapsis north, increasing orbit inclination, and rotating the low-altitude descending node in the anti-Sun direction (for Mercury at perihelion). For later refinements in the mission design, a

Mercury albedo perturbation will model Mercury-reflected sunlight onto the spacecraft. The first OCM of each pair will impart a ΔV in the spacecraft velocity direction at periapsis, placing the spacecraft on a transfer orbit (Figure 14) with apoapsis altitude matching that of the 200-km periapsis altitude by 12-hour orbit. Two-and-a-half orbits (30.6 hours) later, at apoapsis, the second OCM of each pair lowers periapsis altitude to 200-km by imparting a much larger (LVA thruster as primary) ΔV opposite the spacecraft velocity direction. This second OCM also will return the orbit period to 12 hours. This OCM strategy was changed from an apoapsis first/periapsis second strategy that worked well for other launch opportunities. The choice of 2.5 orbits between the OCMs provides the minimum time needed to assess the first OCM's performance, update the orbit determination and next maneuver design, and test and upload the ΔV update. To keep periapsis altitude under 500 km and meet sunshade orientation constraints, OCM pairs must occur once per 88-day Mercury year. Each OCM pair occurs when the spacecraft orbit's line of nodes is nearly perpendicular to the spacecraft-Sun direction. Table 8 lists timing, ΔV, and spacecraft orientation parameters for each OCM. Figures 15 and 16 show how OCM-1/-2 and OCM-5/-6 are situated between solar conjunctions and an eclipse season that includes periapsis of each orbit.

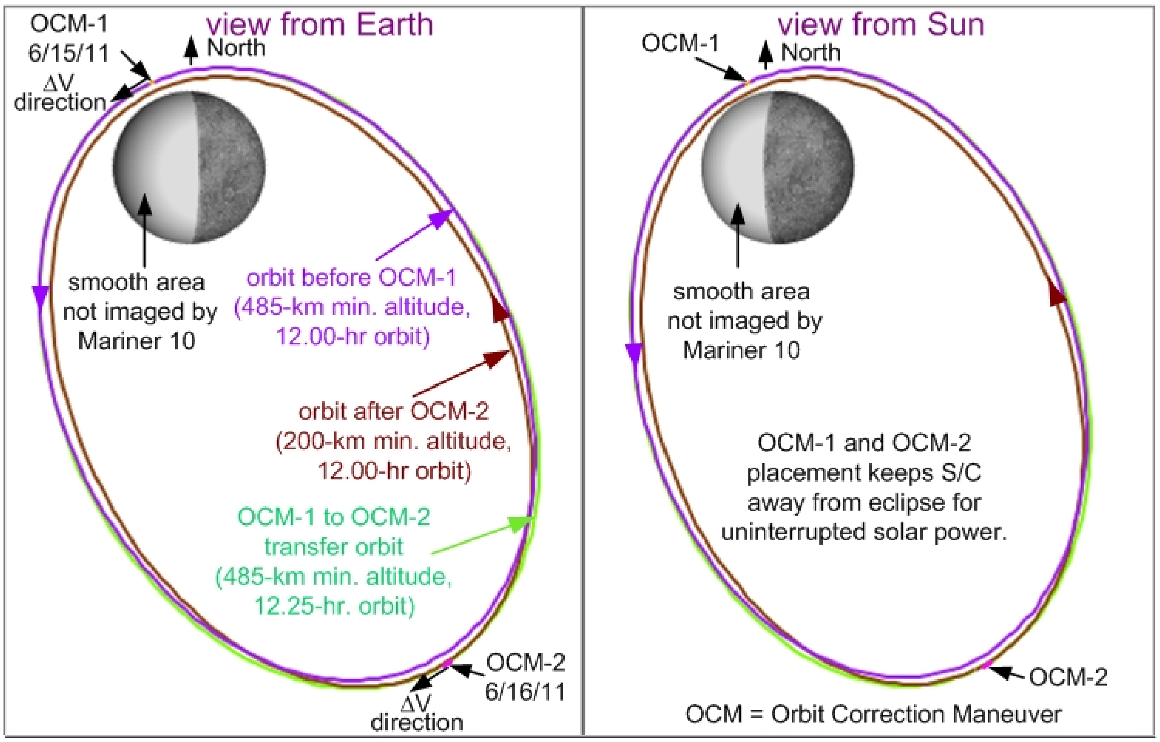

Figure 14. Mercury orbit correction maneuver strategy

Table 8. Mercury orbit correction maneuvers meet operational requirements

Orbit Correction Maneuver Segment	Maneuver Date Year Month Day	Maneuver Time (UTC) (hh:mm:ss)	ΔV (m/s)	Earth-S/C Range (AU)	Sun-S/C Range (AU)	Sun-S/C-ΔV Max. Angle (deg)	Sun-Earth -S/C Angle (deg)
Requirement →						**(78° to 102°)**	**(> 3°)**
OCM-1 start	2011 Jun 15	11:29:20	4.22	1.319	0.311	96.1	3.3
OCM-2 start	2011 Jun 16	18:06:18	26.35	1.314	0.313	99.6	4.8
OCM-3 start	2011 Sep 09	12:07:57	3.92	1.097	0.308	86.0	16.1
OCM-4 start	2011 Sep 10	18:42:20	24.18	1.129	0.309	93.8	15.3
OCM-5 start	2011 Dec 05	12:44:25	3.59	0.683	0.308	82.1	3.2
OCM-6 start	2011 Dec 06	19:16:13	22.22	0.693	0.309	89.9	6.0
DeterministicΔV	**for orbit phase**	**=**	**84.48**	**m/s**			

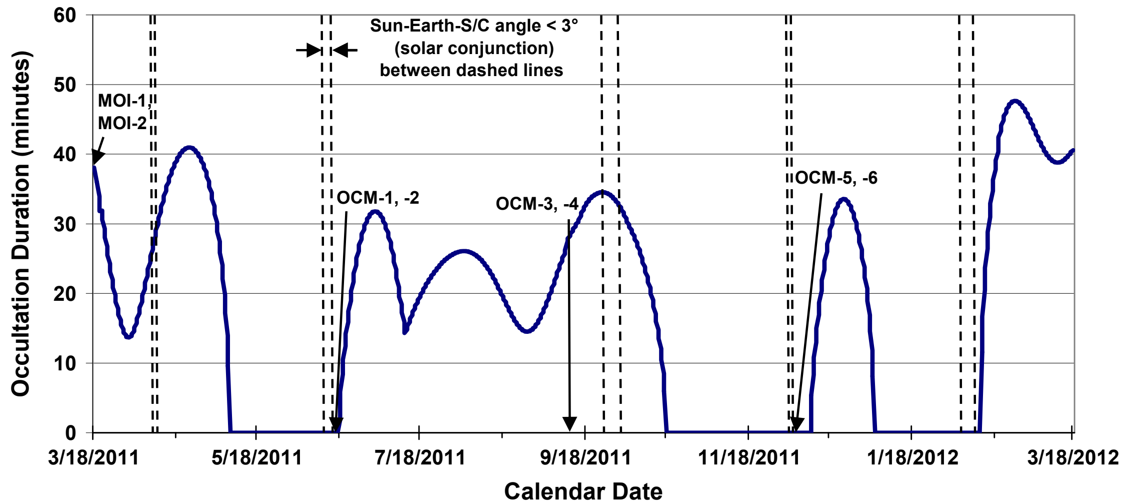

Figure 15. Earth occultations and solar conjunctions during Mercury orbit

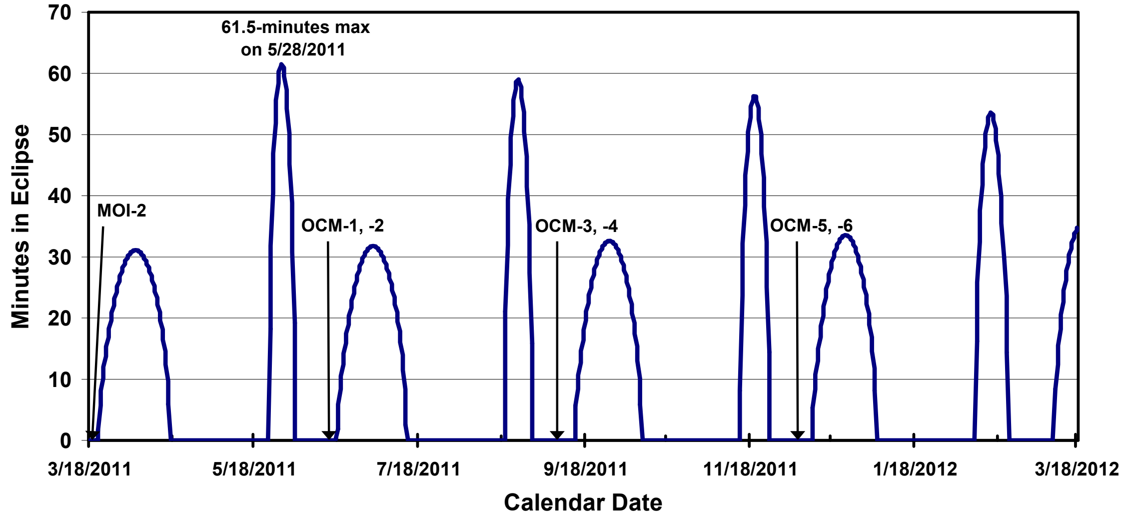

Figure 16. Solar eclipses (short eclipses include periapsis) during Mercury orbit

During the Mercury orbital phase knowledge of the predicted spacecraft attitude is vital for accurate orbit propagation and design of upcoming OCMs. Trajectory perturbations due to solar pressure, variations in Mercury's gravity, solar gravity, end-of-life sunshade surface reflectance, and Mercury surface albedo must be carefully coordinated with the spacecraft's complex attitude profile. For the first time, all these factors (except for Mercury albedo) are accounted for during Mercury orbit phase propagations. These effects (solar gravity and Mercury oblateness, J_2, are dominant factors) cause periapsis altitude to increase to 485–444 km prior to OCM-1, -3, and, -5; northward periapsis latitude drift (decreasing the argument of periapsis) of about 11.6º; orbit inclination increasing by about 2º; and longitude of ascending node decreasing 6º. Figures 17 and 18 show the non-uniform variation of each of these orbital parameters. Including a Mercury gravity model[13] with normalized coefficients C_{20} = -2.7x10^{-5} and C_{22} = 1.6x10^{-5} increases the northward drift of the periapsis point by nearly 30%, when compared with a point mass model for Mercury's gravity. The spacecraft attitude rules account for a daily 8-hour downlink period, up to 16 hours of science observation (sunshade toward the Sun with +z aligned with Mercury nadir when possible), and solar array tilt varying as a function of solar distance.

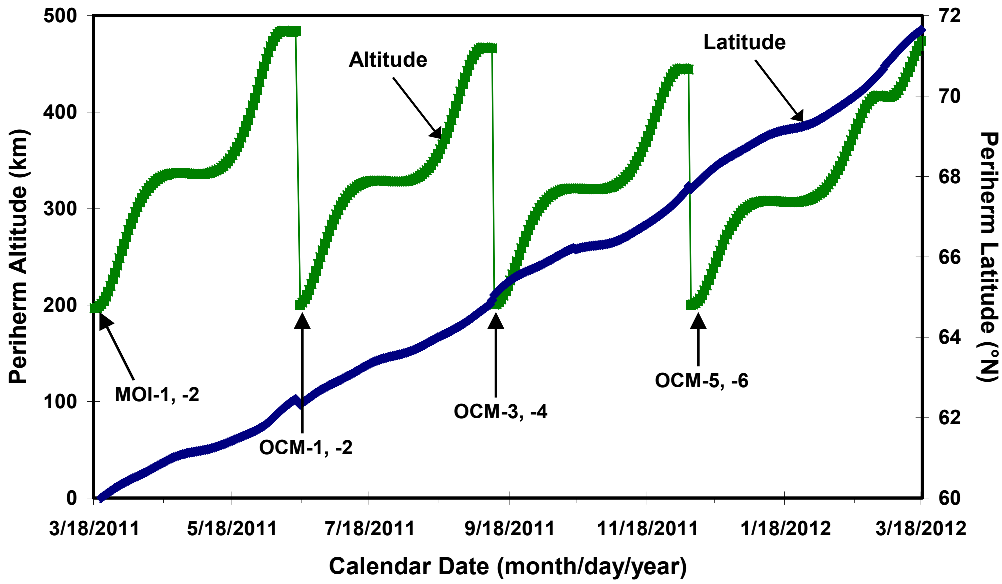

Figure 17. Periapsis evolution during Mercury orbit

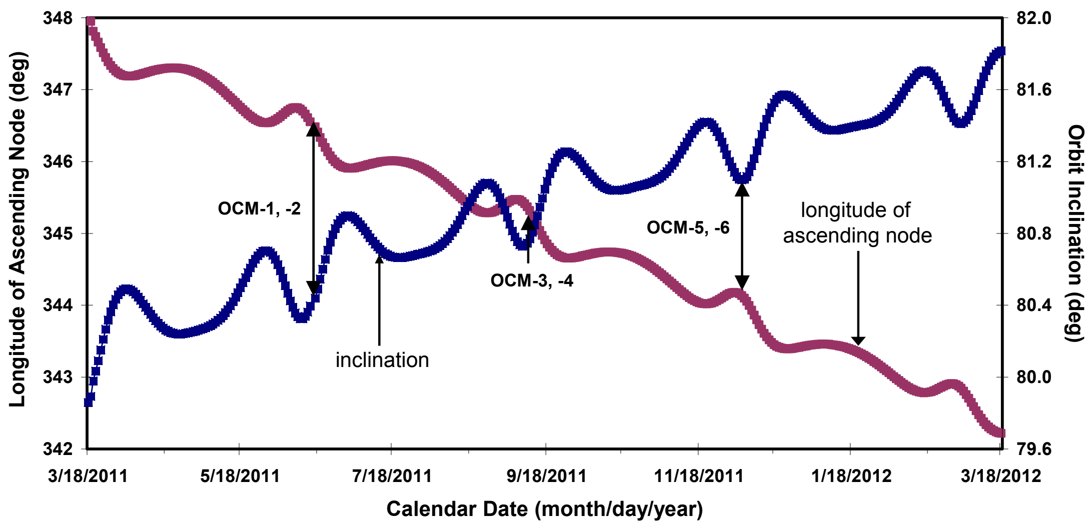

Figure 18. Examples of orbit plane rotation during Mercury orbit

CONCLUSIONS

With a successful launch aboard a Delta II 7925H expendable launch vehicle on 3 August 2004, and with nearly flawless execution of three course correction maneuvers, the MESSENGER mission is on track to place the first spacecraft into orbit around Mercury in March of 2011. A capable suite of seven scientific instruments will achieve the carefully formulated science objectives during three Mercury flybys and a one-year Mercury orbit phase. A highly redundant spacecraft design with a dual-mode propulsion system, sunshade, and articulating solar arrays will protect the spacecraft bus and instruments from the extreme thermal and radiation environment of the inner solar system. With launch delays forcing the mission to follow a long-duration trajectory to Mercury, mission operators have many opportunities to re-optimize the spacecraft's trajectory, refine nominal and contingency maneuver strategies, and explore opportunities to minimize risk. Launch delays also helped increase the ΔV budget for launch errors, navigation, and contingencies. All deterministic maneuvers, which include five deep space maneuvers, a two-part Mercury orbit insertion, and six orbit correction maneuvers, comply with a restrictive set of operational constraints.

ACKNOWLEDGEMENT

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