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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 spacecraft was launched on 3 August 2004. The 6.6-year ballistic trajectory to Mercury will utilize six gravity-assist flybys of Earth (one), Venus (two), and Mercury (three). With three trajectory correction maneuvers completed by mid-December 2005, many more maneuvers will be necessary during the journey to Mercury and the subsequent 1-year duration Mercury orbit phase. The spacecraft's design and operational capability will enable real-time monitoring of every course-correction maneuver. 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 an 80-deg orbit inclination relative to Mercury's equator, 200-km periapsis altitude, 60°N subspacecraft periapsis latitude, and a 12-h 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.

I. Introduction

MORE than 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 3 August 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 six trajectory correction maneuvers (TCMs) by mid-December 2005, the spacecraft will need many more TCMs: four large deterministic maneuvers and 20 to 30 statistical TCMs during the heliocentric orbit, two maneuvers for Mercury orbit insertion, and six maneuvers during the 1-year duration Mercury orbit phase.

With funding and mission oversight coming from the NASA Discovery Program, the MESSENGER mission relied on 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. Six years later, the spacecraft is approaching two years of successful operation in space.

The spacecraft's seven science instruments will acquire data 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 following 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), the magnetometer (MAG), the 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.

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. Whereas navigation team members at KinetX, Inc., and JHU/APL share responsibility for trajectory optimization and maneuver design, Yen[4] of the Jet Propulsion Laboratory, California Institute of Technology, and McAdams et al.[5] discovered each

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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 and 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-deg inclination orbit.
Determine the structure of Mercury's magnetic field.	Minimize periapsis altitude; maximize altitude-range coverage (MAG).	
Simplify orbital mission operations to minimize cost and complexity.	Choose orbit period of 8, 12, or 24 h.	Mercury orbit periapsis altitude from 200–500 km, apoapsis altitude near 15,200 km for 12-h orbit period.
Map the elemental and mineralogical composition of Mercury's surface.	Maximize time at low altitudes (GRNS and XRS).	
Measure libration amplitude and gravitational field structure.	Minimize orbit-phase thrusting events (RS and MLA). Orbit inclination of 80 deg, latitude of periapsis near 60°N (MLA and RS).	Orbital inclination of 80 deg, periapsis latitude drifts from 60°N to 72°N, primarily passive momentum management, two orbit-correction ΔV (31 h apart) every 88 days.
Determine the composition of radar-reflective materials at Mercury's poles.	Orbit inclination of 80 deg achieved; latitude of periapsis maintained near 60°N (GRNS, MLA, MASCS, and EPPS).	
Characterize exosphere neutrals and accelerated magnetosphere ions.	Wide-altitude range covered and visibility of atmosphere at all lighting conditions.	Extensive coverage of magnetosphere, orbit cuts bow shock, magnetopause, and upstream solar wind.

Table 2 MESSENGER launch options for 2004

	Month		
Month	March	May	Aug. (final)
Launch dates	10–29	11–22	30 July–13 Aug.[a] 3 Aug.
Launch period, days	20	12	15
Launch energy, km^2/s^2	$\leq$15.700	$\leq$17.472	$\leq$16.887 (16.388)
Earth flybys	0	0	1
Venus flybys	2	3	2
Mercury flybys	2	2	3
Deterministic ΔV, m/s	$\leq$2026	$\leq$2074	$\leq$1991 (1966)
Total ΔV, m/s	2300	2276	2277 (2251)[b]
Orbit insertion date	6 April 2009	2 July 2009	18 March 2011

[a]Final launch period started on 2 August because of delays in availability of launch facility.
[b]Lower total ΔV reflects reduced propellant load required to meet spacecraft launch weight limit.

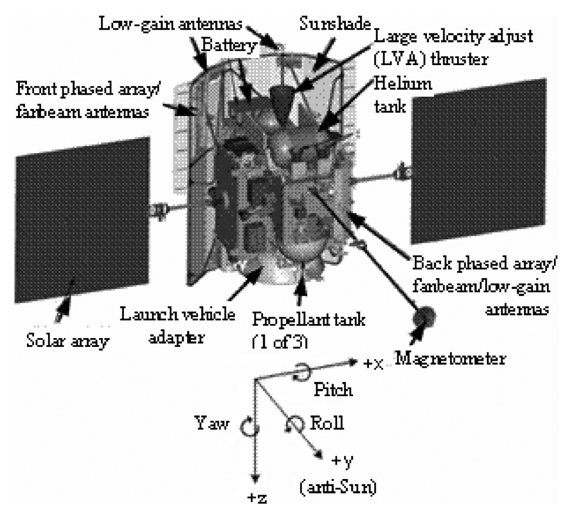

Fig. 1 Flight configuration of MESSENGER spacecraft (excludes thermal blanket covering).

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. A MESSENGER mission design paper[7] in 1999 depicted 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 et al.[8] in 2001. The first detailed account of MESSENGER's Mercury orbit phase, which appeared in early 2003 (Ref. 9), mapped 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.

II. 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 astronomical units (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 (Fig. 1) include two movable solar arrays with 5.3-m^2 maximum sun-facing area, a 2.5×2.1 m ceramic cloth sunshade, the spacecraft bus, and a magnetometer boom. As the spacecraft reaches Mercury's distance from the sun (0.30–0.46 AU), the arrays tilt up to 65 deg away from the sun to keep the sun-facing solar array surface temperature below 150°C. Knowledge of solar array orientation is required for accurate determination of the solar pressure perturbation on the spacecraft's trajectory. A 23-A $\cdot$ h NiH_2 battery provides power during solar eclipse. Battery capacity was a primary consideration for the selection of the lowest subspacecraft periapsis latitude. Given the 200-km periapsis altitude, 12-h orbit period, 60°N initial subspacecraft periapsis latitude, and the corresponding predicted northward drift rate, the 61.5-min maximum solar eclipse duration is well below the 65-min upper bound (a conservative limit set during the early design phase). 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, that is, periods where the sun–Earth–spacecraft angle is less than 3 deg. 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 was reoriented to point toward the sun on 8 March 2005, the sunshade protected the spacecraft bus from direct sunlight exposure as long as the surface normal of the sunshade center panel stays within 12 deg of the direction to the sun. Spacecraft rotations in yaw of ± 15 deg and from $+13.8$ to -12.4 deg in pitch define the operational zone where direct sunlight never impinges on any part of the spacecraft protected by the sunshade. This 12-deg maximum angle (sunshade surface normal to the spacecraft–sun direction) for most of the mission leads to a constraint for large deterministic maneuvers that the sun–spacecraft–ΔV angle must be between 78 and 102 deg. 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 the Mercury orbit phase. These two opportunities for performing OCMs occur when the spacecraft orbit plane is nearly perpendicular to the sun–Mercury line. When solar gravity and small solar radiation pressure perturbation effects are neglected, these OCM opportunities arise near Mercury orbit true anomaly angles of 12 (where Mercury orbit insertion occurs) and 192 deg. Furthermore, because 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 listed in Table 2. For these reasons, all OCM pairs occur once every 88 days, when Mercury is near 12-deg true anomaly. Additional details of the spacecraft design are available from a source[10] prepared a few months before 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 from 200 to 235 s). The monopropellant thrusters include four 26-N LVA thrust vector control (TVC) thrusters on the same deck as the LVA thruster and 12 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 prelaunch 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 ΔV 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 4 of the 12 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 cleanup of the overall ΔV target using all four 26-N LVA–TVC thrusters.

III. Heliocentric Trajectory

The spacecraft will utilize one Earth gravity assist, two Venus gravity assists, three Mercury gravity assists, and five major (DSMs) during its 6.6-year ballistic trajectory to Mercury (Fig. 2). As of Mercury orbit insertion (MOI), the spacecraft will have completed more than 15 orbits of the sun and traveled 7.9 billion km relative to the sun. Unlike both earlier launch opportunities in 2004, less ideal Earth–Venus and Venus–Mercury phasing requires the 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.

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 <3 deg. During solar conjunction, the sun either degrades or prevents spacecraft communication with ground stations,

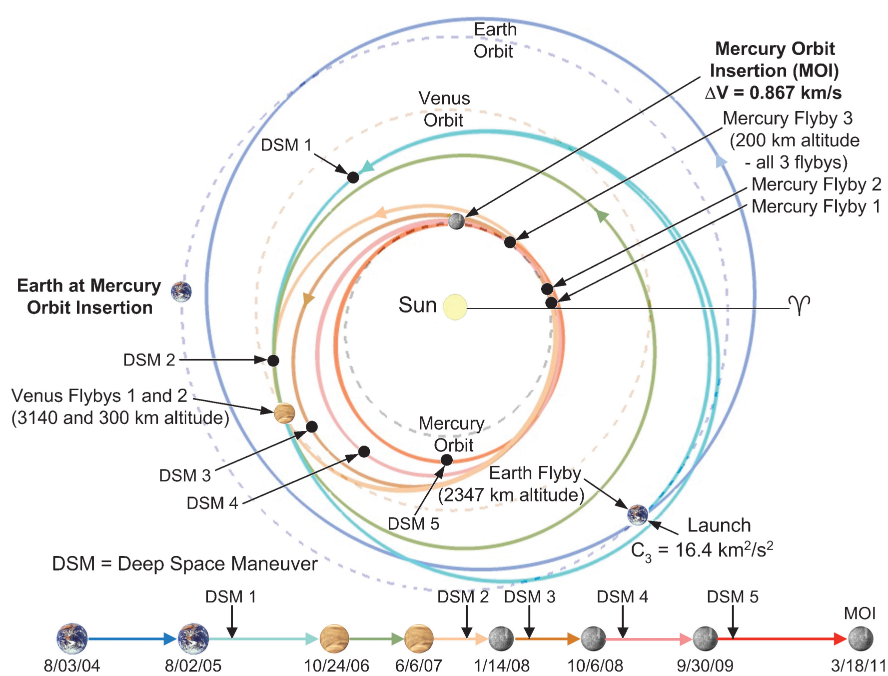

Fig. 2 North ecliptic pole view of MESSENGER's heliocentric trajectory.

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. During MESSENGER's heliocentric trajectory the first solar conjunction, lasting about 5 weeks, begins several days before Venus flyby 1. For MESSENGER, the longest solar conjunction, lasting about 1.5 months, begins one year after the first Venus flyby. A Venus close-approach altitude of over 3100 km, along with the absence of major mission-critical events for more than 7 months, greatly reduces the risk associated with this solar conjunction. The "Mercury Orbit" section addresses the significance of solar conjunctions during the spacecraft's year at Mercury.

IV. Maneuver Strategy for Heliocentric Transfer

The MESSENGER spacecraft's heliocentric trajectory requires 5 deterministic maneuvers with ΔV magnitude >50 m/s (DSMs), 2 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 from 3 to 20 m/s between 78 and 102 deg 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 12 deg of the sun-to-spacecraft direction will be performed using a pair of 4-N thrusters mounted on the sunward or antisun sides of the spacecraft. Maneuvers with ΔV directions outside the cited restrictions will be performed by either a turn and burn approach (if no thermal safety margin is violated) or a less efficient vector components method using a combination of two of the aforementioned 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 MOI, 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, Inc., by using Monte Carlo analyses. Because this ΔV budget applies only to the heliocentric transfer phase, the contingency category includes variations in expected ΔV for Mercury OCMs. 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 Current ΔV allocation

ΔV budget category	ΔV, m/s
DSMs	1008
Launch vehicle, navigation errors (99%)	121
MOI	867
OCMs	85
Contingency	169
Total	2250

Maneuvers planned for the heliocentric transfer will exercise propulsion modes with all three thruster types (4, 26, and 672 N) enabled as primary thrusters. Whereas the first three TCMs used the 26-N thrusters, TCMs 5 and 6 used only 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 <0.1 m/s and ΔV direction error <0.4 deg for the first six TCMs, the MESSENGER mission has removed launch dispersions, targeted Earth flyby, and initiated Venus 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, reduces the spacecraft's sun-relative speed; this maneuver lowers perihelion enough to set up the first Mercury flyby. From the Mercury encounter locations in Fig. 2 and the DSM ΔV magnitudes from Table 4, note how the magnitude of DSMs 3–5 is directly proportional to the change in Mercury's position between the preceding 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. Because DSMs 3–5 occur near aphelion, each maneuver raises perihelion slightly. If a spacecraft anomaly or ground station outage causes a delay for DSM 2, 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 DSMs 3–5 because each DSM can occur later during the next heliocentric orbit.

V. Launch

On 3 August 2004, at 06:15:56.5 Universal Time, Coordinated, 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 deg 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. Although the first hour after launch (flight path appears in Fig. 3) closely followed the planned trajectory, the larger-than-average 2.0-σ underburn 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 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

Table 4 Deterministic maneuvers during heliocentric transfer (TCM 4 not needed)

Name	Status	Maneuver date	Earth range, AU	Sun range, AU	Sun–spacecraft–ΔV, deg	Sun–Earth–spacecraft, deg	ΔV, m/s
Requirement (DSMs only) →					(78–102)	(>3)	
TCM 1	Complete	24 Aug. 2004	0.000	1.015	93.2	124.2	17.90
TCM 2	Complete	24 Sept. 2004	0.051	1.067	92.8	118.0	4.59
TCM 3	Complete	18 Nov. 2004	0.124	1.071	88.6	103.2	3.25
TCM 5	Complete	23 June 2005	0.096	0.962	42.7	53.4	1.10
TCM 6	Complete	21 July 2005	0.029	0.999	124.2	52.4	0.15
DSM 1	Complete	12 Dec. 2005	0.688	0.604	92.4	37.3	315.63
TCM 15	Planned	23 Jan. 2007	1.846	0.871	88.3	5.2	2.87
DSM 2	Planned	22 Oct. 2007	1.694	0.703	87.4	3.4	201.32
DSM 3	Planned	17 March 2008	0.678	0.685	87.5	43.3	73.79
DSM 4	Planned	06 Dec. 2008	1.596	0.626	89.3	6.3	240.48
DSM 5	Planned	29 Nov. 2009	1.529	0.565	89.6	7.4	176.55

the sun, thereby saving heater power by using sunlight to warm the spacecraft bus when the spacecraft–sun distance is near 1 AU.

VI. Planetary Flybys

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. Because the first Venus flyby will occur during solar conjunction, these opportunities will not be possible there. Because of 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 before the low-altitude Mercury flybys, where OpNavs are 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 5–6 weeks before each of the Venus and Mercury flybys.

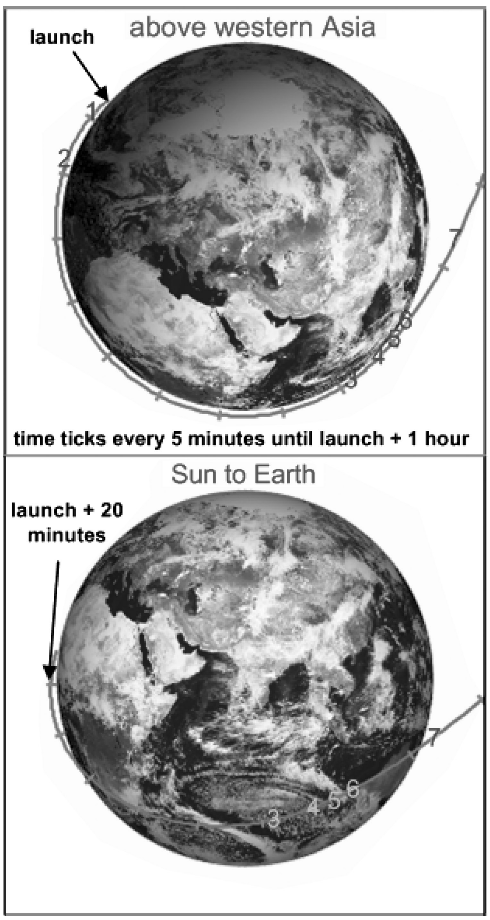

Fig. 3 Launch trajectory for first hour after launch. Launch events (seconds after liftoff): 1) main engine (first stage) cutoff (MECO) (263.4), 2) second engine (second stage) cutoff (SECO1) (533.9), 3) restart stage 2 (2758.6), 4) second engine (second stage) cutoff (SECO2) (2935.5), 5) stage 3 ignition (3016.7), 6) third engine (third stage) cutoff (TECO) (3101.0), and 7) spacecraft separation (third stage jettison) (3406.4).

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. Table 5 lists orbital parameters that most affect the velocity difference that the spacecraft's propulsion system must correct to enter into orbit around Mercury. As the size and orientation of the spacecraft's orbit approach the size and orientation of 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 8 deg 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.) Because DSMs 2–5 occur near aphelion, Table 5 shows only 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.

A. Earth

One year after launch, an Earth flyby (Fig. 4) provides significant trajectory shaping by lowering perihelion to 0.6 AU from the sun and by moving perihelion direction more than 60 deg 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

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

Event name	Longitude of perihelion (LP),°	LP to goal,°	Orbit inclination (OI), deg	OI to goal, deg	Perihelion distance (PD), AU	PD to goal, AU	Aphelion distance (AD), AU	AD to goal, AU
Launch	194	117	6.5	0.5	0.923	0.615	1.077	0.610
Earth flyby	133	56	2.6	4.4	0.603	0.295	1.015	0.548
DSM 1	——	——	——	——	——	——	1.054	0.587
Venus flyby 1	105	28	7.9	0.9	0.546	0.238	0.901	0.434
Venus flyby 2	47	30	6.7	0.3	0.332	0.024	0.745	0.278
DSM 2	——	——	——	——	0.325	0.017	——	——
Mercury flyby 1	56	21	7.0	0.0	0.313	0.005	0.700	0.233
DSM 3	——	——	——	——	0.315	0.007	——	——
Mercury flyby 2	68	9	7.0	0.0	0.302	0.006	0.630	0.163
DSM 4	——	——	——	——	0.309	0.001	——	——
Mercury flyby 3	81	4	7.0	0.0	0.303	0.005	0.567	0.100
DSM 5	——	——	——	——	0.308	0.000	——	——
Mercury orbit (goal)	77	——	7.0	——	0.308	——	0.467	——

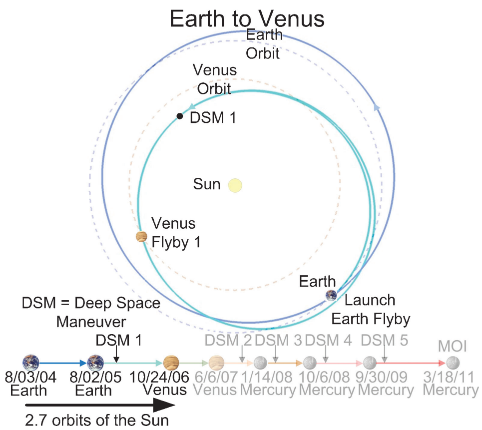

Fig. 4 North ecliptic pole view of Earth-to-Venus flyby 1 trajectory.

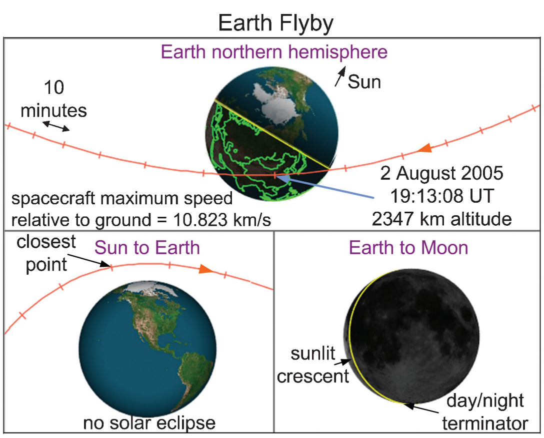

Fig. 5 MESSENGER's Earth flyby trajectory.

decreases after launch (vs increases for an August 2005 launch), thereby allowing a more complete checkout of spacecraft health before TCM 1. The Earth flyby creates opportunities for calibrating several science instruments using the moon, thereby removing science observations from the early postlaunch operations schedule. Close approach for Earth (Fig. 5) occurred 2347 km over central Mongolia, high enough to avoid solar eclipse.

B. Venus

The primary purposes of the first Venus flyby (Table 5; Figs. 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 7.9 deg (beyond the desired 7.0 deg) to position the second Venus flyby at the same point in Venus's orbit one Venus year later. Although the spacecraft will approach a brightly illuminated Venus, a 1.4-deg 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-min solar eclipse, but also for not hearing from flight controllers for another month (at the end of solar conjunction). The times 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 MOI.

The second Venus flyby (Fig. 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 Venusian surface by about 10% (from 12.337 to 13.585 km/s).

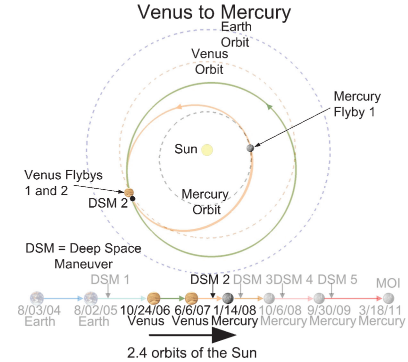

Fig. 6 North ecliptic pole view of Venus flyby 1 to Mercury flyby 1 trajectory.

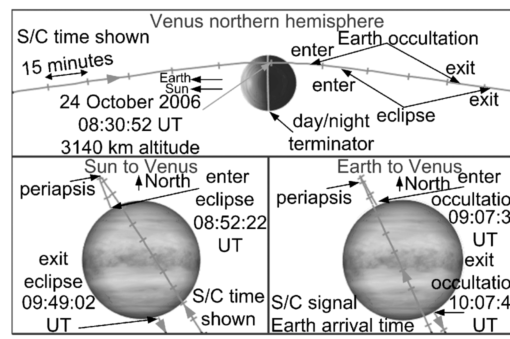

Fig. 7 MESSENGER's first Venus flyby trajectory.

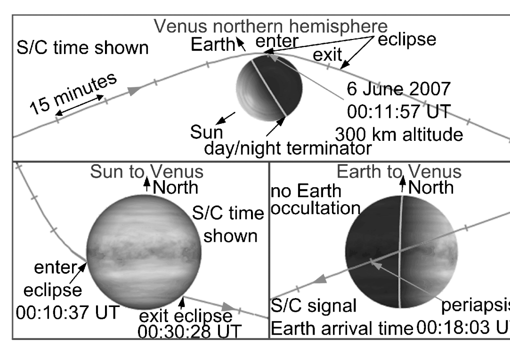

Fig. 8 MESSENGER's second Venus flyby trajectory.

C. Mercury

Although the MESSENGER spacecraft's journey to Mercury orbit appears long, the Mercury flybys (Fig. 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, 2.40, 1.55, 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 (Figs. 10–12), followed two months later by DSMs, will complete the spacecraft orbit rotation and change the orbital dimensions to be closer to those of 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, that is, the spacecraft orbits the sun five times while Mercury completes six

Table 6 Mercury encounter summary

Event	Phase, C/A[a]− 1 day, deg	Phase, C/A+ 1 day, deg	Speed relative to surface, km/s	V_∞, km/s	Sun–Earth–spacecraft, deg	Earth range, AU
Mercury 1	117	51	7.103	5.811	16.5	1.155
Mercury 2	127	36	6.586	5.166	2.2	0.660
Mercury 3	103	40	5.302	3.379	15.4	0.799
MOI	94	——	4.363[b]	2.200	17.4	1.029

[a]Close approach.
[b]Occurs at 288-km altitude after completing first 30% of MOI.

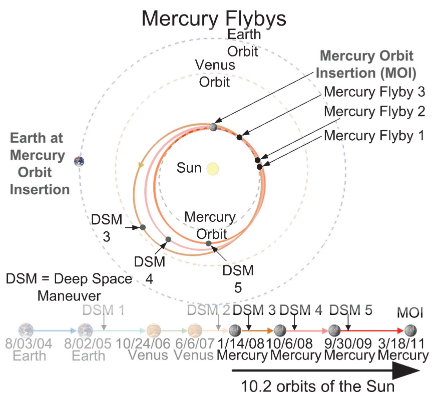

Fig. 9 North ecliptic pole view of Mercury flyby 1 to MOI trajectory.

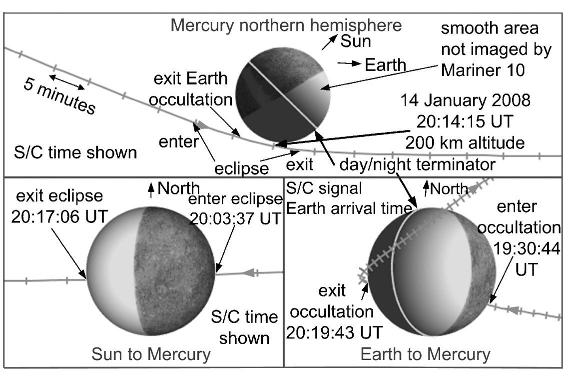

Fig. 10 MESSENGER's first Mercury flyby trajectory.

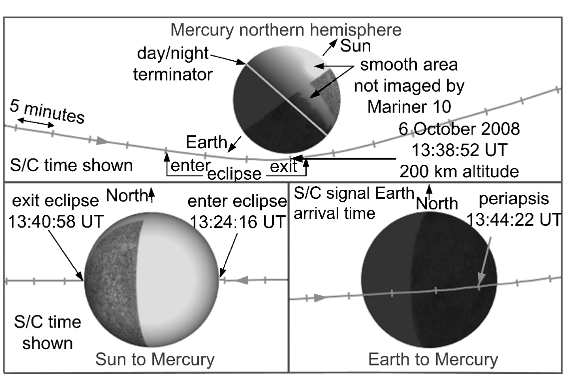

Fig. 11 MESSENGER's second Mercury flyby trajectory.

orbits. This sequence of orbital resonances reduces the spacecraft–Mercury hyperbolic excess velocity V_∞ (Table 6). Phase in Table 6 is the sun–Mercury–spacecraft angle. Although the second Mercury flyby will occur during a brief solar conjunction (minimum sun–Earth–spacecraft angle about 2 deg), approach OpNav images will be transmitted to Earth before solar conjunction entry. Although the third Mercury flyby rotates the spacecraft's longitude of perihelion 4 deg past Mercury's longitude of perihelion (Table 5), this extra orbit rotation must occur to achieve the 5:6 spacecraft:Mercury orbital resonance.

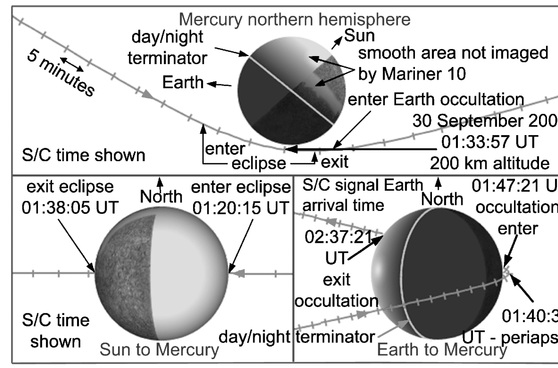

Fig. 12 MESSENGER's third Mercury flyby trajectory.

Approach and departure views of Mercury for the three Mercury flybys (Figs. 10–12) offer glimpses 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 Figs. 10–12) hemisphere of Mercury soon after close approach. Because 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.

VII. MOI

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

The MOI strategy maximizes the probability of successfully delivering the spacecraft into the primary science orbit in the minimum time possible within mission planning constraints. This strategy uses two turn-while-burning variable thrust-direction maneuvers (MOI 1 and 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 2. With each maneuver slowing the spacecraft's Mercury-relative velocity, the thrust vector is almost opposite to the spacecraft velocity vector. 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 2 and shows 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.

The initial orbit period of the spacecraft at Mercury between MOI 1 and 2 is between 12.8 and 16.0 h, 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 2 gives 51–64 h for preliminary MOI 1 maneuver performance assessment, orbit determination, MOI 2 maneuver design, simulation, and maneuver update upload and verification. The 12.8-h 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-h target for the primary science orbit. The initial phasing orbit period range also ensures that daily data downlink periods (beginning 4 h before apoapsis on every other 12-h orbit) start from near 0830 to 1630 hrs EST. A sun–Earth–spacecraft angle >17 deg 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 2–3 min of the 14.5-min 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 to 75 s from MOI 1 to 2 due to the longer settling time (partly due to the large MOI 1 ΔV expelling much of the spacecraft's mass).

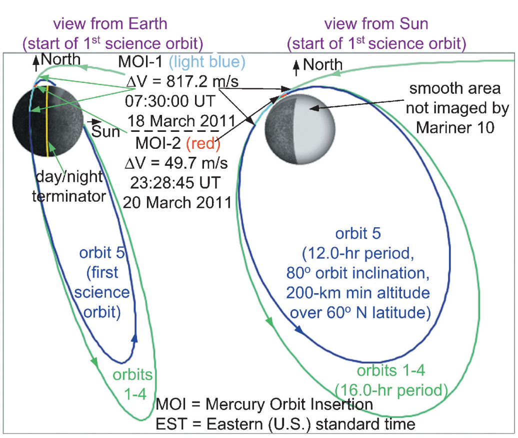

Fig. 13 MOI trajectory and maneuvers.

VIII. Mercury Orbital Operations

After completing MOI, the spacecraft begins a more than 12-week-long coast phase without OCMs. This will be enough time to refine Mercury's gravity model and update perturbing force models to further minimize trajectory propagation errors. Occasional thruster firings for adjusting spacecraft angular momentum will perturb the trajectory by at most a few millimeters per second 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 be used to 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 (Fig. 14) with apoapsis altitude matching that of the 200-km periapsis altitude by 12-h orbit. Two-and-one-half orbits (30.6 h) 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 h. 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 OCMs 1 and 2 and OCMs

Table 7 Mercury orbit insertion meets operational requirements

Maneuver segment[a]	Date	Time, hr	ΔV,[b] m/s	Earth–spacecraft range, AU	Sun–spacecraft range, AU	Sun–spacecraft–ΔV, deg	Sun–Earth–spacecraft, deg
Requirement→						(78–102)	(>3)
MOI 1 settle/refill	18 March 2011	0230:00	0.2	1.024	0.309	94.6	17.4
MOI 1 LVA start	18 March 2011	0230:45	815.6	1.024	0.309	94.8	17.4
MOI 1 LVA mid	18 March 2011	0237:30	——	1.024	0.309	95.6	17.4
MOI 1 trim end	18 March 2011	0244:26	2.6	1.024	0.309	95.6	17.4
MOI 2 settle/refill	20 March 2011	2328:45	0.5	0.956	0.314	78.8	18.4
MOI 2 LVA start	20 March 2011	2330:00	46.4	0.956	0.314	79.9	18.4
MOI 2 trim end	20 March 2011	2330:55	2.7	0.956	0.314	80.9	18.4

[a]See "Spacecraft Design and Maneuver Constraints" section (final paragraph) for maneuver segment definitions.
[b]Orbit insertion total $\Delta V = 868.1$.

Table 8 Mercury orbit correction maneuvers meet operational requirements

Maneuver segment	Date	Time, hr	ΔV,[a] m/s	Earth–spacecraft range, AU	Sun–spacecraft range, AU	Sun–spacecraft –ΔV, deg	Sun–Earth–spacecraft, deg
Requirement→						(78–102)	(>3)
OCM 1 start	15 June 2011	1129:20	4.22	1.319	0.311	96.1	3.3
OCM 2 start	16 June 2011	1806:18	26.35	1.314	0.313	99.6	4.8
OCM 3 start	09 Sept. 2011	1207:57	3.92	1.097	0.308	86.0	16.1
OCM 4 start	10 Sept. 2011	1842:20	24.18	1.129	0.309	93.8	15.3
OCM 5 start	05 Dec. 2011	1244:25	3.59	0.683	0.308	82.1	3.2
OCM 6 start	06 Dec. 2011	1916:13	22.22	0.693	0.309	89.9	6.0

[a]Orbit phase total $\Delta V = 84.48$.

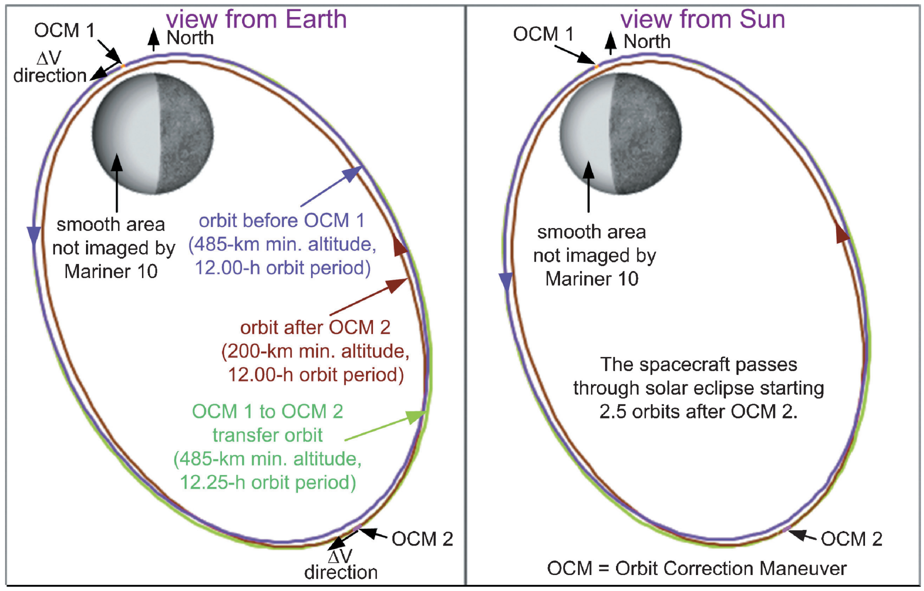

Fig. 14 Mercury orbit correction maneuver strategy.

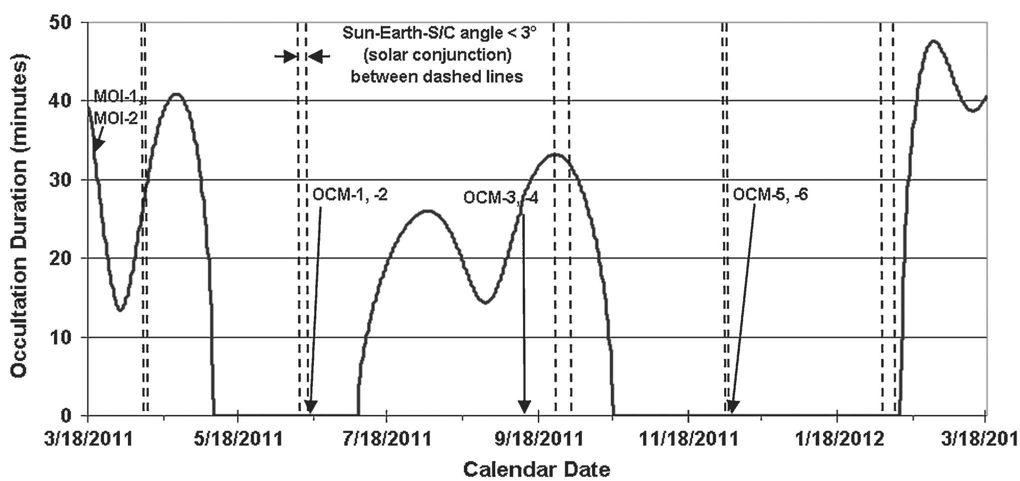

Fig. 15 Earth occultations and solar conjunctions during Mercury orbit.

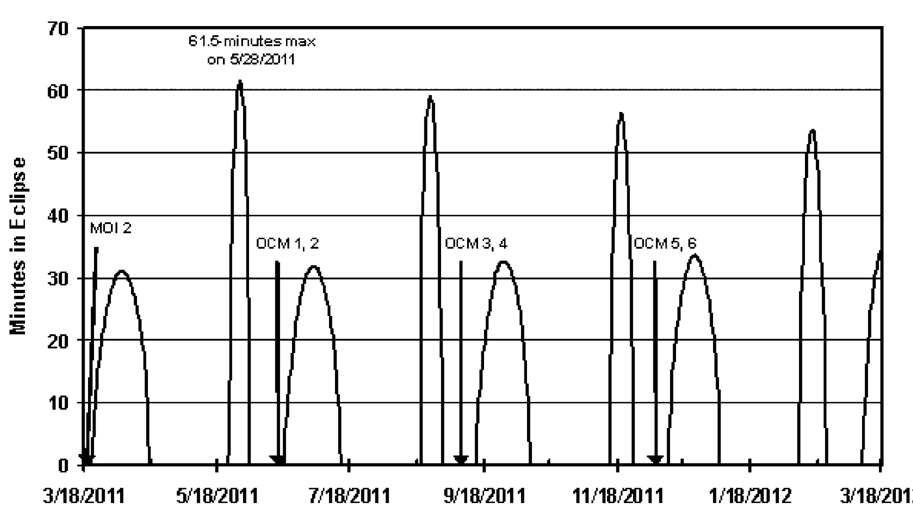

Fig. 16 Solar eclipses (short eclipses include periapsis) during Mercury orbit.

5 and 6 will occur soon after the spacecraft emerges from solar conjunction.

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. All of these factors (except for Mercury albedo) are accounted for during Mercury orbit phase propagations. These effects (where solar gravity and Mercury oblateness J_2 are dominant factors) cause periapsis altitude to increase to 485–444 km before OCM 1, 3, and 5; cause northward periapsis latitude drift (decreasing the argument of periapsis) of about 11.6 deg; cause an orbit inclination increase of about 2 deg; and cause a longitude of ascending node decrease of 6 deg. Figures 17 and 18 show the nonuniform variation of each of these orbital parameters. Including a Mercury gravity model[13] with normalized coefficients $C_{20} = -2.7 \times 10^{-5}$ and $C_{22} = 1.6 \times 10^{-5}$ 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-h downlink period, up to 16 h 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.

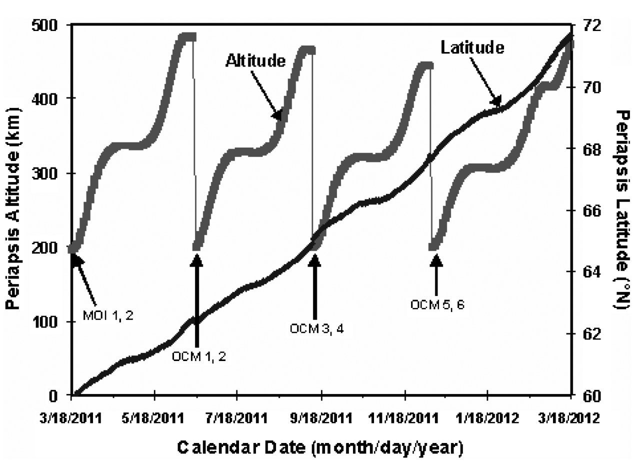

Fig. 17 Periapsis evolution during Mercury orbit.

IX. 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 2011. A suite of seven scientific instruments will achieve the carefully formulated science objectives during three Mercury flybys and a 1-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 reoptimize 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 DSMs, a two-part MOI, and six OCMs, comply with a restrictive set of operational constraints.

Acknowledgments

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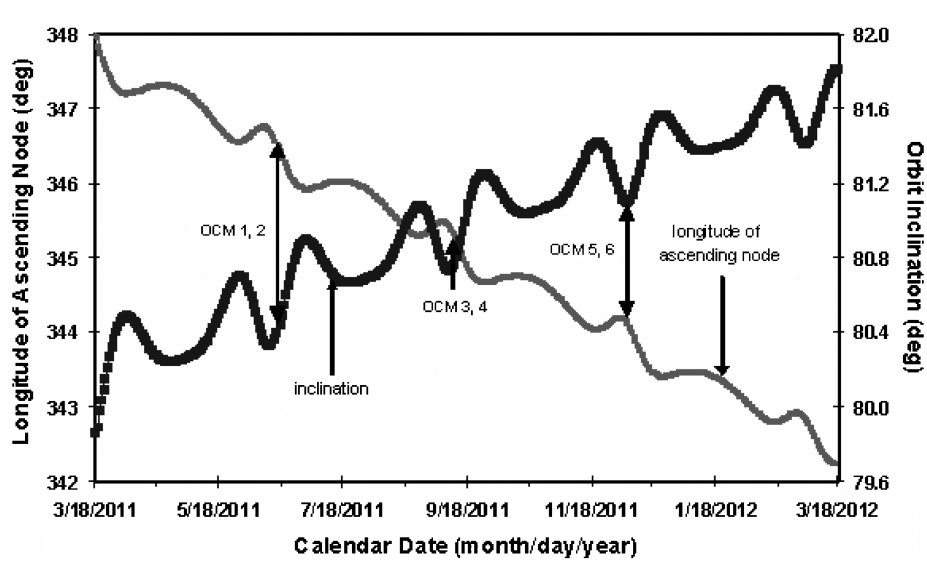

Fig. 18 Examples of orbit plane rotation during Mercury orbit.

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