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MESSENGER Mission Design and Navigation

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Abstract Nearly three decades after the Mariner 10 spacecraft's third and final targeted Mercury flyby, the 3 August 2004 launch of the MESSENGER (MErcury Surface, Space ENvironment, GEochemistry, and Ranging) spacecraft began a new phase of exploration of the closest planet to our Sun. In order to ensure that the spacecraft had sufficient time for pre-launch testing, the NASA Discovery Program mission to orbit Mercury experienced launch delays that required utilization of the most complex of three possible mission profiles in 2004. During the 7.6-year mission, the spacecraft's trajectory will include six planetary flybys (including three of Mercury between January 2008 and September 2009), dozens of trajectory-correction maneuvers (TCMs), and a year in orbit around Mercury. Members of the mission design and navigation teams optimize the spacecraft's trajectory, specify TCM requirements, and predict and reconstruct the spacecraft's orbit. These primary mission design and navigation responsibilities are closely coordinated with spacecraft design limitations, operational constraints, availability of ground-based tracking stations, and science objectives. A few days after the spacecraft enters Mercury orbit in mid-March 2011, the orbit will have an 80° inclination relative to Mercury's equator, a 200-km minimum altitude over 60°N latitude, and a 12-hour period. In order to accommodate science goals that require long durations during Mercury orbit without trajectory adjustments, pairs of orbit-correction maneuvers are scheduled every 88 days (once per Mercury year).

Keywords Gravity assist · MESSENGER · Mercury · Mission design · Navigation

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1 Introduction

The launch of the MESSENGER (MErcury Surface, Space ENvironment, GEochemistry, and Ranging) spacecraft on 3 August 2004 began a new phase of exploration for the closest planet to our Sun. The 7.6-year trajectory the MESSENGER spacecraft will follow includes one Earth flyby, two Venus flybys, three Mercury flybys, and a one-year Mercury orbit phase. The three Mercury flybys will offer the opportunity to obtain images of most of the 55% of Mercury's surface not imaged by Mariner 10 in 1974 and 1975. As the exploration phase transitions from "flyby" to "orbit", MESSENGER will use a suite of seven science instruments to acquire data that will help answer six fundamental questions regarding the formation, composition, and field structure of Mercury and the region of space influenced by Mercury (Solomon et al. 2001).

Engineers at The Johns Hopkins University Applied Physics Laboratory (JHU/APL), numerous subcontractors, and universities worked closely with the mission's Principal Investigator, located at the Carnegie Institution of Washington, to design and assemble a spacecraft and science payload that would meet the mission's science objectives. The spacecraft's design includes features that ensure redundancy of most mission-critical functions, provide autonomy during time-critical activities, and offer reliable operation during a mission that places the probe between 30% and 108% of Earth's average distance from the Sun. Some of these features include a fixed sunshade to protect delicate spacecraft parts from direct sunlight, a dual-mode (fuel only or a fuel/oxidizer mix) propulsion system for course-correction maneuvers, two high-temperature-tolerant articulating solar arrays and a battery for power, state-of-the-art attitude control and telecommunications subsystems, and seven science instruments for data collection. While characteristics of the spacecraft's trajectory and propulsive maneuvers affected the design of many of these spacecraft components, spacecraft and ground-station functional limitations impose many operational constraints and guidelines on the spacecraft.

In order to fulfill all mission objectives, the spacecraft must follow a complex trajectory with many planned trajectory-correction maneuvers (TCMs) and planetary flybys. The mission design team at JHU/APL and the navigation team at KinetX, Inc., work together to design the most fuel-efficient, lowest-risk trajectory and TCMs possible. The remaining trajectory is re-optimized after each propulsive maneuver and planetary flyby to accommodate execution errors and trajectory uncertainties. In addition, each portion of the completed trajectory is determined at the highest possible precision. Much effort is required to ensure success for events that have the greatest change of the trajectory (i.e., planetary flybys and Mercury orbit insertion). The navigation team uses observed changes in the spacecraft's orbit to improve the accuracy of models (e.g., solar radiation pressure and Mercury gravity) for orbit perturbations in order to improve trajectory predictions.

The early operations activity of the mission design and navigation teams has been a key part of the early mission success. The design and implementation of TCMs, along with trajectory optimization and orbit determination, have both corrected errors introduced at launch and targeted the first two planetary flybys (of Earth and Venus). Highlights of these activities and the resulting TCM performance are provided.

2 Historical Background

A direct trajectory to Mercury requires substantial launch energy (i.e., a minimum C_3 of about 50 km^2/s^2). However, by using a Venus gravity-assist maneuver, the launch energy required is about 70% less than that needed for a direct transfer. This important result was originally obtained by Michael Minovitch (Minovitch 1963; Dowling et al. 1997). The Venus gravity-assist concept was developed further by Sturms and Cutting (1966) and eventually led to a proposal for a Mariner spacecraft mission to Mercury in 1973 (Bourke and Beerer 1971). The original flight plan called for a launch in November 1973, a Venus flyby in February 1974, and one Mercury flyby in March 1974. However, as suggested by Guiseppe Colombo in 1970 (Murray 1989), the Mercury flyby aim point can be chosen so that the spacecraft's heliocentric orbital period is exactly twice Mercury's orbital period, allowing the spacecraft to return to Mercury 176 days after the first encounter. Used twice during the flight of Mariner 10, this technique led to three successful Mercury flybys (Dunne and Burgess 1978). These flybys, which provided images of 45% of Mercury's surface, occurred on 29 March 1974, 21 September 1974, and 16 March 1975 at altitudes of 703 km, 48,069 km, and 327 km, respectively.

The spectacular success of the Mariner 10 mission generated considerable interest in further exploration of Mercury. An orbiter-class mission was the next logical step. However, it was soon realized that the energy requirements to place a meaningful payload into orbit around Mercury were rather formidable. A variety of ballistic mission modes utilizing near-perihelion propulsive maneuvers, powered Venus flybys, and multiple Venus flybys were studied in some detail (Hollenbeck et al. 1973). Because these results were less than satisfactory, Friedlander and Feingold (1977) concluded that a Mercury orbiter mission would require the use of a low-thrust delivery system such as solar-electric propulsion. This situation prevailed until 1985 when Chen-wan Yen (1989) showed that a "reverse delta-VEGA (ΔV Earth Gravity Assist) process" using multiple Venus and Mercury flybys along with selected propulsive maneuvers could produce energy-efficient ballistic trajectories for a Mercury orbiter.

Since 1990 Mercury orbiter mission studies continued this trend of lower propulsive requirements. These studies include a Hermes Orbiter study by Jet Propulsion Laboratory (JPL) and TRW (Cruz and Bell 1995), one by the European Space Agency (Grard et al. 1994), one by McAdams et al. (1998), and another by Yamakawa et al. (2000) from Japan's Institute of Space and Astronautical Science (ISAS). The 1998 and 1999 Mercury orbiter studies described an August 2005 launch with 16.0 km^2/s^2 launch energy that required two Venus and two Mercury flybys as part of the 4.2-year heliocentric transfer to Mercury orbit insertion. Preceding this trajectory with a one-year Earth–Earth transfer, and following it with a major propulsive maneuver and third Mercury flyby, forms the basis for the trajectory followed by the MESSENGER mission. More recent studies of ballistic mission modes for a Mercury orbiter have shown that ΔV costs for this mission could be reduced even further (Langevin 2000; Yen 2001; McAdams et al. 2002). The BepiColombo mission, planned by the European Space Agency to launch as early as 2013, will rely on high-thrust propulsion and multiple flybys of Venus and Mercury during its multi-year heliocentric transfer to Mercury.

3 Science and Engineering Background

The science requirements, spacecraft configuration, and launch vehicle performance influence the selection of the Sun-centered (cruise phase) and Mercury-centered (orbit phase) portions of the MESSENGER mission's trajectory. Other factors such as funding limitations and the allowable range of launch dates, while also important, are beyond the scope of this article. A complex interdependency among launch vehicle performance (e.g., the maximum spacecraft weight that can be delivered to a particular Earth escape speed and direction), available technology for the spacecraft subsystems and science instruments, propulsive requirements for various trajectories, and space environment conditions must be addressed early in the spacecraft design process. Early in the mission's conceptual design it became clear that the most capable launch vehicle allowed by NASA's Discovery Program would be required for MESSENGER. This requirement led to the selection of the Boeing Corporation's Delta II 7925H-9.5 launch vehicle.

Characteristics of the trajectory, specifications for TCMs, and requirements for navigation all led to spacecraft design features with inherent operational constraints. Examples of these design features include the type, number, and placement of propulsive thrusters, as well as placement and field-of-view of the spacecraft's visible-light-wavelength cameras. Mission design and navigation team members work with mission operations and science team members during the mission's cruise and orbit phases to ensure that all spacecraft operational constraints are met with adequate margin for all planned and contingency events that alter the spacecraft's trajectory.

3.1 Science Requirements

Mercury is the only planet in the inner solar system that has yet to be orbited by a spacecraft. During its three Mercury flybys in 1974 and 1975, Mariner 10 imaged about 45% of the surface at an average resolution of about 1 km and $<1\%$ of the surface at better than 500-m resolution (Murray 1975a, 1975b). Mercury has the highest known uncompressed density of any planet or satellite in the solar system. This observation, coupled with the determination of surface composition from analysis of MESSENGER science data, could reveal clues that would help to understand the processes by which planetesimals in the primitive solar nebula accreted to form planets.

The spacecraft's seven science instruments will acquire data in order to address six important questions on the nature and evolution of Mercury (Solomon et al. 2001). 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 planet's magnetic field structure.
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), Gamma-Ray and Neutron Spectrometer (GRNS), X-Ray Spectrometer (XRS), Magnetometer (MAG), Mercury Laser Altimeter (MLA), Mercury Atmospheric and Surface Composition Spectrometer (MASCS), Energetic Particle and Plasma Spectrometer (EPPS), and an X-band transponder for the Radio Science (RS) experiment. Table 1 shows the role each instrument has in linking science objectives to the orbit at Mercury. Other articles in this issue offer a more comprehensive examination of the structure and function of all science instruments (Anderson et al. 2007; Andrews et al. 2007; Cavanaugh et al. 2007; Goldsten et al. 2007; Hawkins 2007; McClintock and Lankton 2007; Schlemm et al. 2007).

Table 1 Mapping of 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
Determine the structure of Mercury's magnetic field	Minimize periapsis altitude; maximize altitude-range coverage (MAG)	Mercury orbit periapsis altitude of 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°; latitude of periapsis near 60°N (MLA, RS)	Orbital inclination drifts from 80° to 82°; periapsis latitude drifts[1] from 60°N to 72°N; primarily passive momentum management; two orbit-correction ΔVs (30 hours apart) every 88 days
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

[1]Solar and Mercury gravity perturb the spacecraft orbit away from some science requirements. Orbit-correction ΔVs can correct periapsis latitude drift

3.2 Spacecraft Configuration

The spacecraft's physical characteristics (Fig. 1) and software configuration were designed to accommodate all aspects (both extremes and routine activity) of MESSENGER's heliocentric cruise and Mercury orbit phases. Only those features that equip the spacecraft to function during trajectory-altering events (e.g., planetary flybys or TCMs) or trajectory-related environmental extremes (e.g., solar eclipse) will be discussed. The MESSENGER spacecraft's design emphasizes simple, proven techniques and functional redundancy, along with a combination of carefully selected advanced technologies and minimal moving parts. Key design features include a fixed ceramic-cloth sunshade, a dual-mode (bipropellant-

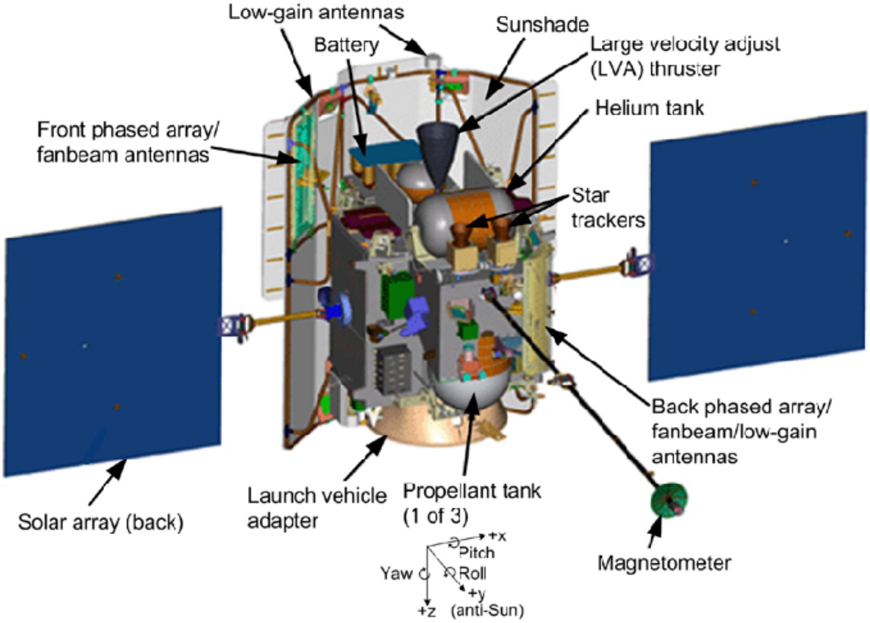

Fig. 1 Fully deployed, false-color view of the MESSENGER spacecraft. This view shows spacecraft details that are hidden by layers of thermal blankets or heat-resistant, Nextel ceramic cloth. This fully-deployed configuration and Sun-relative orientation is typical during much of the mission after late-March 2005

monopropellant) propulsion system, two rotating solar arrays, three-axis stabilization, and a state-of-the-art telecommunications system (Leary et al. 2007).

The propulsion system includes 17 thrusters (Fig. 2) arranged to allow small TCMs (<20 m/s ΔV) in any direction and small-to-large TCMs for ΔV in the $+z$ direction using either the four 26-N "C" thrusters or the 672-N large-velocity-adjust (LVA) thruster. The telecommunications subsystem includes four low-gain antennas that were mounted on the spacecraft such that planned and contingency course-correction maneuvers would be able to be monitored from Earth-based tracking stations with sufficient link margin. In addition, a 23-Ah NiH_2 battery will provide survival power for over an hour during eclipses. Software features include spacecraft autonomy that enables the spacecraft to operate independently (without commands from ground-based operators) for weeks at a time, a feature that will be used during multiple, long-duration solar conjunctions.

3.3 Operational Requirements and Constraints

After identifying those spacecraft subsystems with significant design criteria for ensuring safe spacecraft function throughout the mission, it is useful to describe the corresponding operational requirements and constraints. The trajectory and propulsive maneuvers designed by the mission design team and reconstructed by the navigation team must demonstrate compliance with every requirement and constraint defined in the JHU/APL-produced System Requirements Document. A brief description of these requirements and constraints will increase understanding of the complexity inherent in MESSENGER's mission design. It

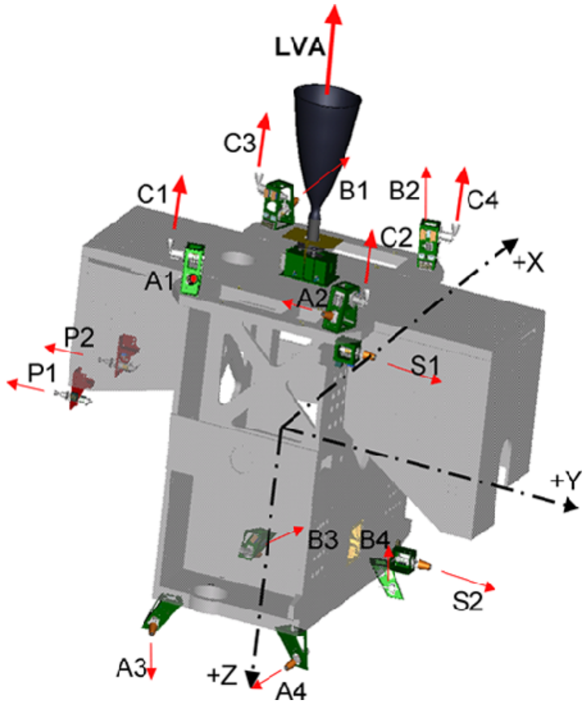

Fig. 2 View showing thruster orientation for the MESSENGER spacecraft. There are 12 4-N thrusters (P1 and P2 that poke through the sunshade, S1 and S2 on the opposite side of the spacecraft, and four thrusters A1 to A4 and B1 to B4 on each side of the spacecraft parallel to the Y–Z plane), four 26-N thrusters (C1 to C4), and one 672-N thruster (LVA)

may also become apparent that some orbit or maneuver options described in other Mercury orbiter studies would violate MESSENGER requirements or constraints.

With a severe thermal environment during Mercury orbit phase, the spacecraft's thermal subsystem had the most issues directly related to the spacecraft trajectory and propulsive maneuver design. Even though a circular orbit around Mercury would offer the opportunity for uniform, global images of Mercury's surface, Nelson et al. (1995) note that such an orbit would subject the spacecraft to an unmanageable thermal load. The 200-km-minimum-altitude by 12-hour orbit followed by MESSENGER at Mercury is stable (periapsis altitude and latitude increase, but orbit period is almost constant) and thermally manageable (Ercol and Santo 1999). Thermal analysis of this orbit at Mercury revealed acceptable worst-case temperatures for the sunshade, spacecraft bus, solar arrays, and science instruments. This thermal analysis related a spacecraft orbit orientation angle to the maximum acceptable spacecraft internal temperature. The 3 August 2004 launch brings the spacecraft's orbit to within 6° of the upper limit for this orbit orientation angle. This constraint effectively places the spacecraft orbit periapsis near the day/night terminator when Mercury is closest to the Sun.

Since the sunshade must protect the spacecraft bus from direct sunlight exposure during propulsive maneuvers that are performed <0.7 AU (Astronomical Unit, which equals the average Earth–Sun distance) from the Sun, a spacecraft maneuver attitude constraint is necessary. For TCMs that require either the bi-propellant LVA thruster and/or the C thrusters mounted on the same deck as the LVA thruster, the Sun–spacecraft–ΔV angle must be between 78° and 102°. This is equivalent to requiring the spacecraft–Sun direction to be <12° from the $-y$-axis direction. This spacecraft attitude constraint during propulsive maneuvers provides two opportunities for performing orbit-correction maneuvers (OCMs) per 88-day

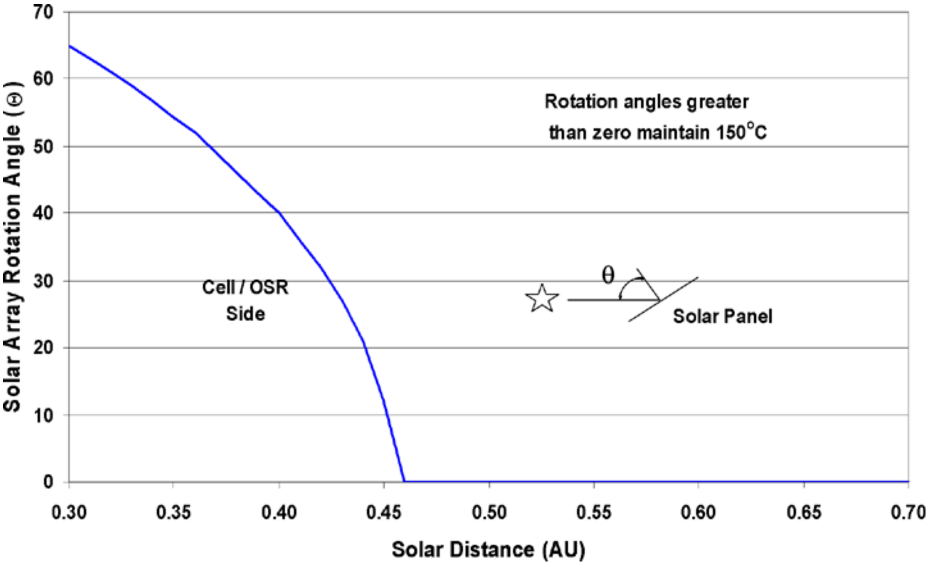

Fig. 3 Solar array angle constraint for preventing damage due to high temperature. Values of 0.30 AU and 0.46 AU, the range of solar distance over which the solar array must be tilted, correspond closely to MESSENGER's minimum and maximum solar distance in Mercury orbit

Mercury year. These two opportunities occur when the spacecraft orbit plane and Sun–Mercury line are nearly perpendicular. These OCM opportunities arise shortly after Mercury's minimum distance from the Sun (where Mercury orbit insertion occurs) and one-half Mercury orbit later. Because science objectives require long intervals between large spacecraft orbit adjustments, the time between OCMs will be maximized. Since the spacecraft's periapsis altitude nears the 500-km upper limit shown in Table 1 about one Mercury year (88 days) after Mercury orbit insertion (MOI), all OCM pairs will occur once every 88 days, soon after Mercury is closest to the Sun.

In order to maintain solar array temperature below an upper limit, the tilt angle of the solar array surface normal relative to the Sun direction must be carefully controlled. Mission design and navigation software requires knowledge of solar array orientation to predict solar pressure perturbations on the spacecraft's orbit. The thermal requirement for solar array rotation, shown in Fig. 3, is to keep the solar array surface, populated with 30% solar cells and 70% optical surface reflectors (OSRs), below 150°C. The size of the solar arrays was chosen to provide sufficient power to perform a TCM at a solar distance of 1.1 AU.

The spacecraft's battery must be able to supply the spacecraft's reduced power level requirement when the solar arrays do not receive sunlight during solar eclipse. Mass margin concerns early in the development phase limited the battery size to meet spacecraft power requirements for up to 65 minutes. With the planned initial orbit size and orientation, the spacecraft should experience a 61.5-minute maximum-duration eclipse in late-May 2011. Another requirement is that no ΔV may be performed within two hours of a solar eclipse.

Trajectory optimization and maneuver design must meet telecommunications subsystem requirements. These requirements include the goal (not a strict requirement) to monitor 100% of every planned propulsive maneuver, the responsibility to determine start and stop times for data transmission during Mercury orbit phase, and the need to define times when interference from the Sun prevents reliable spacecraft communication. The mission design team scheduled all OCMs to avoid Earth occultation, when Mercury blocks the spacecraft–Earth line of sight. As the Sun–Earth–spacecraft angle drops below 3°, the spacecraft enters solar conjunction—a region where solar interference degrades spacecraft communication with Earth ground stations. Time for data downlink is scheduled for eight hours on every other orbit and five additional 2- to 8-hour tracks per week. Knowledge of the spacecraft attitude, including solar array orientation, during these data downlink times is useful for precise modeling of the solar radiation pressure acting on the spacecraft and thus for reducing trajectory estimation and propagation error.

Navigation requirements and the maneuver design process time place more constraints on Mercury orbit insertion and on the spacing between OCMs. In order to lower risk by improving Mercury approach orbit determination, the navigation team must verify that a bright star is visible in the MDIS narrow-angle field-of-view for optical navigation images of Mercury. In addition, the quick-look maneuver performance assessment, subsequent maneuver update planning, and spacecraft upload time needed between OCMs 1 and 2, 3 and 4, and 5 and 6 must be accomplished within about 30 hours (2.5 orbits).

Analyses performed by the navigation team confirmed that the planetary flyby minimum-altitude constraints are conservative. Planetary protection requirements that establish a maximum allowable probability of impact (1×10^{-6}) for the Venus flybys are met with significant margin with MESSENGER's 300-km minimum-altitude constraint. Using planned Mercury approach navigation techniques, a 200-km minimum-altitude constraint for Mercury flybys provides a minute probability of impact.

4 Interplanetary Cruise Phase

The 6.6-year trip from launch to Mercury orbit insertion is one of the longest interplanetary cruise phase options considered for MESSENGER. The spacecraft system design lifetime accounted for a seven-year journey to Mercury followed by a one-year Mercury orbit phase. During interplanetary cruise phase the primary gravitational body changes from Earth (launch and flyby), to the Sun (between planetary flybys), to Venus (flybys), and to Mercury (flybys and arrival). As many as 40 TCMs, including five large deep-space maneuvers (DSMs), may be required during this mission phase.

Orbit determination (OD) during the cruise phase relies primarily on the Deep Space Network (DSN) Doppler and ranging tracking data. On approach to the Venus and Mercury flybys, these data will be augmented with both DSN Delta Differential One-Way Ranging (ΔDOR) and optical navigation measurements. After each OD solution, the navigation team will perform a mapping of the trajectory and its uncertainties to the aim point in the b-plane at the next planet flyby. The b-plane is the plane normal to the incoming asymptote of the hyperbolic flyby trajectory that passes through the center of the target body (Earth in the case of the Earth flyby). The 'S-axis' is in the direction of the incoming asymptote and hence is normal to the b-plane. The 'T-axis' is parallel to the line of intersection between the b-plane and the Earth Mean Ecliptic plane of J2000 (and is positive in the direction of decreasing right ascension). The 'R-axis' (positive toward the South Ecliptic Pole) completes the mutually orthogonal, right-handed Cartesian coordinate axes with origin at the center of the Earth such that $\hat{T} \times \hat{R} = \hat{S}$, where $\hat{T}$, $\hat{R}$, and $\hat{S}$ are unit vectors in each axis direction. The ideal aim point is determined by optimizing the trajectory over the remaining mission from the current epoch up to Mercury orbit insertion and on into the orbit about Mercury. The goal of maneuver design for TCMs is to stay on or near the optimum trajectory while minimizing the use of propulsive fuel.

A single ΔDOR measurement determines the angular position of the spacecraft relative to a baseline between two DSN sites, and hence it contains unique information compared with the line-of-sight measurements of Doppler and ranging from a single station. The baselines typically used are the Goldstone–Canberra and the Goldstone–Madrid station pairs. Optical navigation (opnav) provides a direct measure of the spacecraft position relative to the target body by using MDIS to take images of the target body against a field of background stars.

The DSN long-range planning schedule currently includes periods for obtaining MESSENGER ΔDOR tracking that includes measurements from two different baselines each week (for four weeks total) starting five weeks prior to each Venus and Mercury flyby. ΔDOR and opnav tracking taken before the first Venus flyby, which occurs when MESSENGER is at superior conjunction, will be used for after-the-fact data flow and processing tests since the aim point is over 3,000 km altitude. At this first Venus flyby, Doppler and ranging data types were sufficient for navigation and the enhanced accuracy available from ΔDOR and opnav was not required. This allowed the navigation team to validate the ΔDOR and opnav processing on the first Venus flyby in a parallel, non-critical process without affecting the regular navigation orbit determination and delivery process. Subsequent Venus and Mercury flybys will use ΔDOR during the approach phase to determine the targeting maneuvers and estimate the resulting flyby trajectory. For the Mercury flybys, orbit solutions including opnav images will be used as independent data types to resolve any apparent conflicts between the solutions based on radiometric tracking.

4.1 Launch

MESSENGER had three launch opportunities in 2004, but schedule delays and additions to spacecraft testing plans led to the choice of launching in the latest opportunity. Table 2 shows selected mission performance parameters for all three launch opportunities during 2004. Chen-wan Yen of JPL discovered all three heliocentric transfer trajectories used as launch options for MESSENGER in 2004, with engineers at KinetX and JHU/APL performing trajectory optimization, maneuver design, and navigation.

On 3 August 2004, at 06:15:56.537 UT, the MESSENGER spacecraft left Earth aboard a Delta II 7925H launch vehicle from Cape Canaveral Air Force Station in east central Florida. With launch services provided by Boeing and NASA Kennedy Space Center, the JHU/APL-built 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 (1 January 2000 at 12:00:00 ET) epoch. While there was little deviation from the planned trajectory during the first hour after launch (Fig. 4), the larger-than-expected 2.0-σ under burn required that the navigation team provide early orbit determination solutions to DSN tracking stations to improve their antenna pointing to the spacecraft. Prior to stage 3-spacecraft separation, Boeing's nearly perfect third-stage despin slowed the spacecraft from about 58 revolutions per minute (rpm) to 0.015 rpm. After separation the solar panels deployed and the spacecraft pointed its sunshade away from the Sun, thereby saving heater power by warming the spacecraft bus with solar energy when the spacecraft–Sun distance is near 1 AU.

4.2 Heliocentric Trajectory

Between launch and Mercury orbit insertion the spacecraft flies by Earth once, Venus twice, and Mercury three times. Five additional major course-correction maneuvers or DSMs are

Table 2 MESSENGER launch options for 2004

Month	March	May	August[1]	August (launch day)
Launch dates	10–29	11–22	30 Jul–13 Aug	3 Aug 2004
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	1965[2]
Total ΔV (m/s)	2300	2276	2277	2250
Orbit insertion date	6 Apr 2009	2 Jul 2009	18 Mar 2011	18 Mar 2011

[1]The final launch period started on 2 August due to delays in the launch facility's availability

[2]Lower total ΔV on launch day reflects a reduced propellant load required to meet the spacecraft launch weight limit

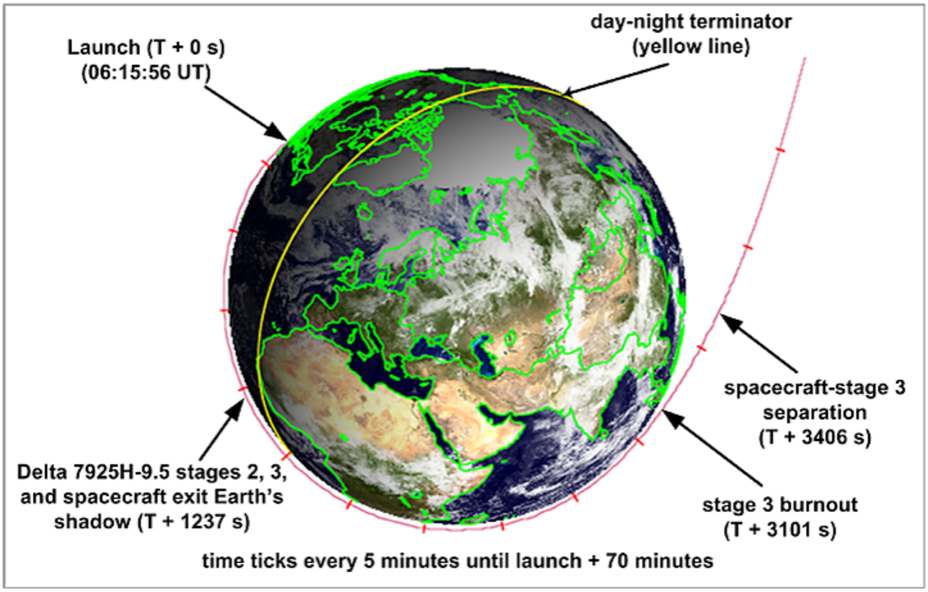

Fig. 4 Launch trajectory with final confirmed times for selected events. All confirmed launch event times were <10 s from the times predicted two weeks before launch

needed during the 6.6-year ballistic trajectory to Mercury. Figure 5 shows the Earth-to-Venus flyby 1 transfer orbit, which includes an Earth flyby one year after launch and the largest DSM, which is just before the spacecraft reaches the following perihelion. A month-long solar conjunction will begin shortly before Venus flyby 1.

The Venus flyby 1-to-Mercury flyby 1 transfer trajectory (Fig. 6) includes a second Venus flyby at almost exactly the same point in Venus' orbit and a DSM one orbit later. The second

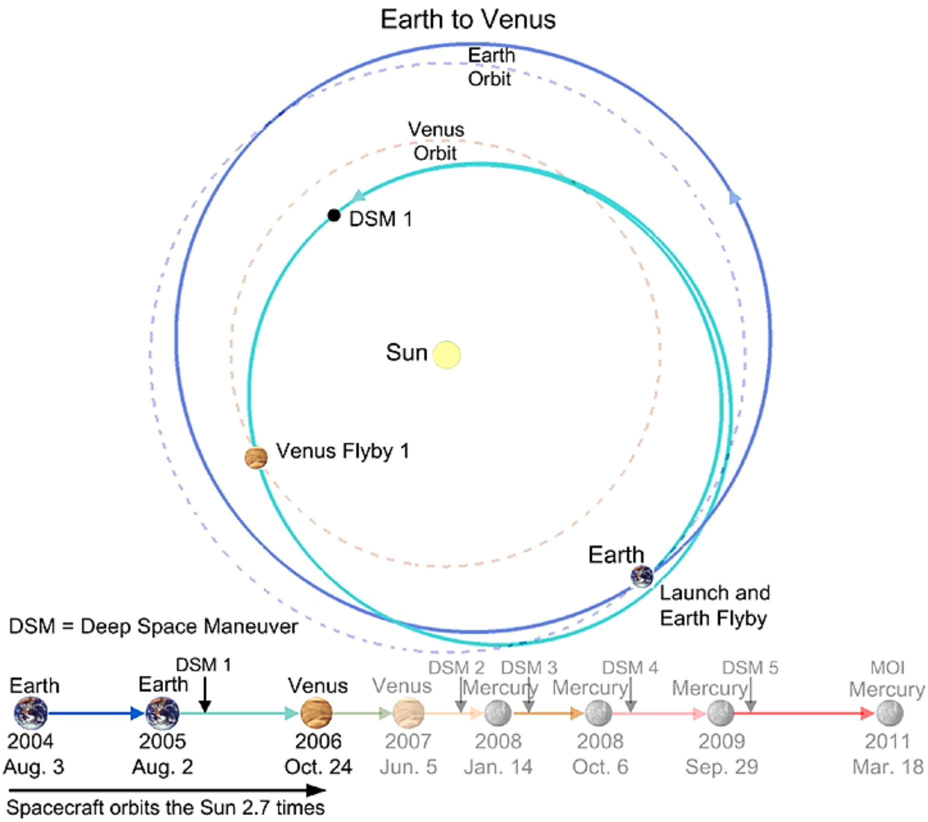

Fig. 5 North ecliptic pole view of the Earth-to-Venus flyby 1 trajectory. *Dashed lines* depict the orbits of Earth and Venus. *Timeline* fading helps emphasize primary events

Venus flyby and DSM-2 bring the spacecraft to Mercury at the right date and velocity to begin the "reverse delta-VEGA" process described in Sect. 2. The DSM-2 date was shifted earlier to place the maneuver before the mission's longest solar conjunction.

The Mercury flyby 1-to-Mercury orbit insertion transfer trajectory, shown in Fig. 7, includes three Mercury flyby-DSM segments that lower the spacecraft speed relative to Mercury while lowering spacecraft orbit period closer to Mercury's heliocentric orbit period. During this portion of the heliocentric transfer the longest solar conjunctions each last about two weeks. The spacecraft completes more than ten orbits of the Sun during this mission phase, or 2/3 of the 15.3 orbits of the Sun between launch and MOI.

4.3 Planetary Flybys

If successful, MESSENGER will be the first spacecraft to use more than five planetary gravity-assist flybys. A detailed description of the trajectory-shaping contribution of each planetary flyby is given by McAdams et al. (2005). Without each planetary flyby, the spacecraft would require far too much propellant and a larger launch vehicle. Although impractical for a ballistic-trajectory Mercury orbiter, a direct Earth–Mercury transfer would take four to five months.

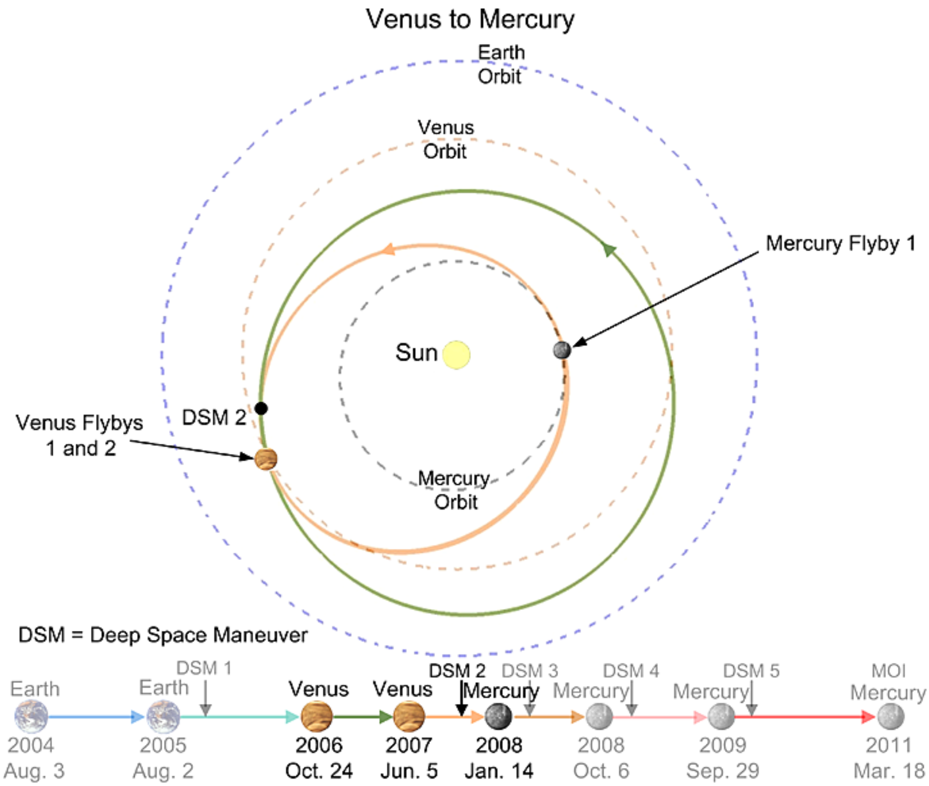

Fig. 6 North ecliptic pole view of the Venus flyby 1-to-Mercury flyby 1 trajectory. *Dashed lines* depict the orbits of Earth, Venus, and Mercury

One year after launch an Earth flyby (Fig. 8) lowered the spacecraft orbit's perihelion to 0.6 AU from the Sun and moved the perihelion direction more than 60° closer to Mercury's perihelion direction. Since the maximum DLA for the August 2005 launch period would be slightly higher than for August 2004, and since launch energy is nearly identical for the August 2004 and August 2005 launch periods, the Earth flyby enables the Delta II 7925H-9.5 to launch a slightly heavier spacecraft. The Earth flyby provides science instrument calibration opportunities using the Moon, thereby removing science observations from the early post-launch operations schedule. Close approach for Earth occurred about 2348 km over central Mongolia, high enough to avoid solar eclipse.

The first Venus flyby (Fig. 9) increased the spacecraft orbit's inclination and reduced the spacecraft's orbit period to almost exactly the heliocentric orbit period of Venus. The second Venus flyby will then occur at the same point in Venus' orbit 225 days later. As the spacecraft approached a brightly illuminated Venus, a 1.4° Sun–Earth–spacecraft angle limited the reliability of data transmission to or from the spacecraft near close approach. The spacecraft not only relied on battery power during a 56-minute solar eclipse just after the flyby but also received commands from flight controllers for only one day after the eclipse (until near the end of solar conjunction). The second Venus flyby (Fig. 9) is the first to lower perihelion enough to enable a Mercury flyby. However, use of a 313-km minimum altitude for the second Venus flyby leads to the need for a subsequent maneuver (DSM-2) to target the next planetary flyby (Mercury flyby 1). Each Venus flyby moves the spacecraft's

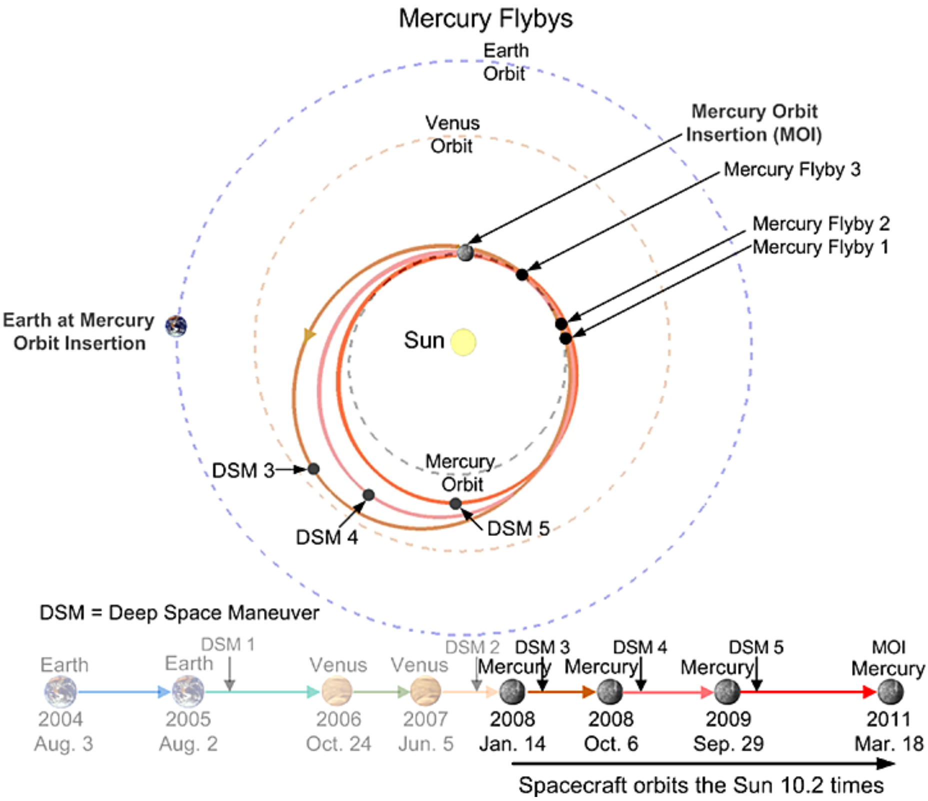

Fig. 7 North ecliptic pole view of the Mercury flyby 1-to-Mercury orbit insertion trajectory. *Dashed lines* depict the orbits of Earth, Venus, and Mercury

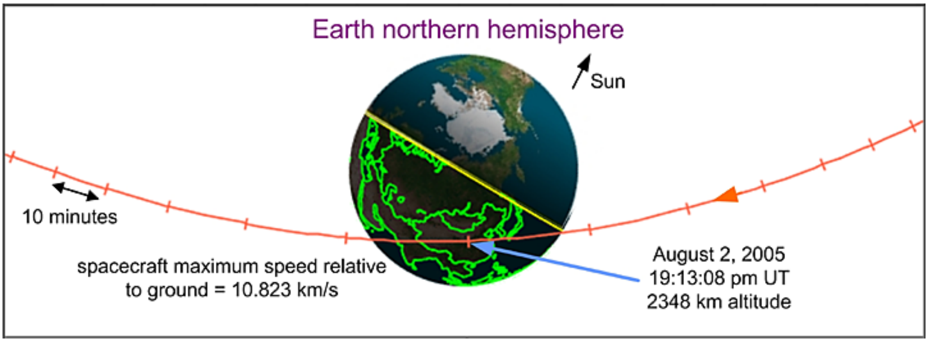

Fig. 8 View of the Earth flyby trajectory from above northern Asia. Major country borders are outlined with *green lines* on Earth's night side. The *yellow line* marks the position of the day/night or dawn/dusk terminator

aphelion and perihelion much closer to Mercury's perihelion and aphelion, thereby reducing the velocity change required for MOI.

By achieving progressively lower Mercury encounter velocities, the three Mercury flybys (Fig. 10) and subsequent DSMs are mission enabling. For direct transfers (no gravity-assist

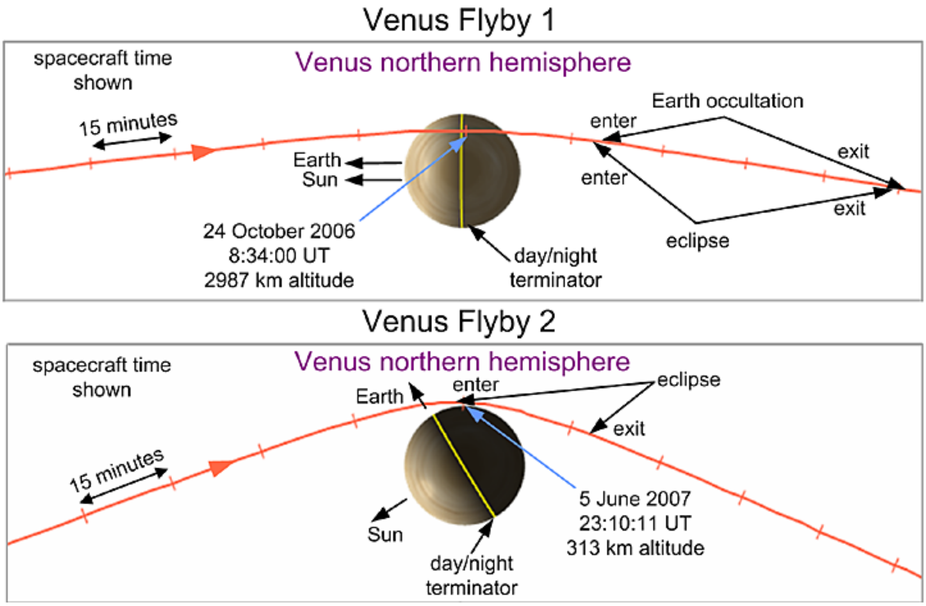

Fig. 9 View of both Venus flyby trajectories from above Venus' north pole

Table 3 Mercury encounter summary

Event	Phase, C/A[1] – 1 day (deg)	Phase, C/A + 1 day (deg)	V_∞ (km/s)	Sun–Earth–S/C (deg)	Earth range (AU)
Mercury 1	117	51	5.817	16.5	1.157
Mercury 2	127	36	5.177	2.3	0.659
Mercury 3	104	40	3.381	15.4	0.798
MOI	94	–	2.201	17.4	1.029

[1] Phase is the Sun–Mercury center-spacecraft angle. Close approach (C/A) denotes the minimum spacecraft–Mercury altitude

flybys) from Earth to Mercury at a minimum launch energy near 50 km^2/s^2, the MOI ΔV is about 10 km/s. Accounting for trajectory shaping by both Venus flybys, the MOI ΔV required for zero, one, two, and three Mercury flybys is at least 3.13 km/s, 2.40 km/s, 1.55 km/s, and 0.86 km/s, respectively. Trajectories utilizing fewer than two Mercury gravity assists would overheat the spacecraft during MOI and the following orbit phase.

Three 200-km minimum-altitude Mercury flybys, each followed by DSMs near the first aphelion after each flyby, are needed to reduce the Mercury arrival velocity enough to enable MOI in March 2011. The Mercury flybys and subsequent DSMs will yield successive orbits having spacecraft : Mercury orbital resonances of about 2 : 3, 3 : 4, and 5 : 6 (i.e., the spacecraft orbits the Sun five times while Mercury completes six orbits). Table 3 shows how this strategy reduces spacecraft hyperbolic excess velocity relative to Mercury, also known as V_∞.

Figure 10 shows how the spacecraft can observe opposite sunlit sides of Mercury's never-before-imaged hemisphere soon after close approach. With 1.5 Mercury solar days (176

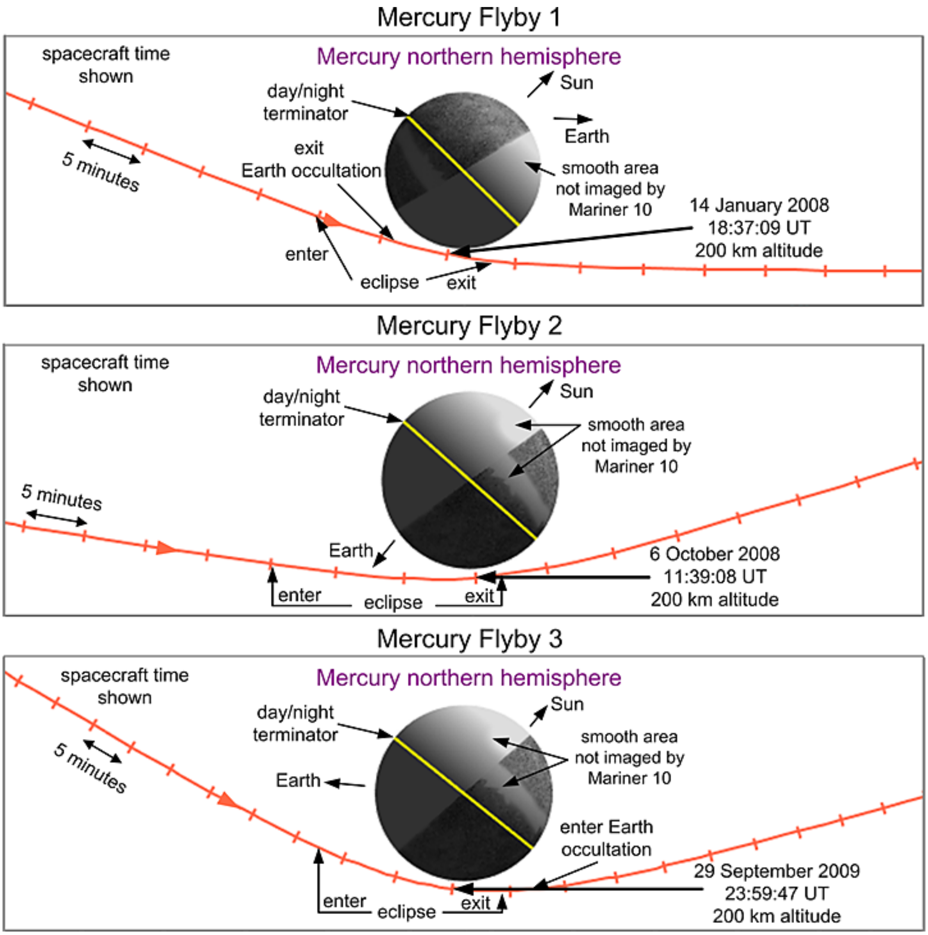

Fig. 10 View of all Mercury flyby trajectories from above Mercury's north pole

Earth days per Mercury solar day) between the first two Mercury flybys, the spacecraft will view opposite sunlit hemispheres. With two Mercury solar days between the second and third Mercury flybys, the same hemisphere is sunlit during these flybys. Figure 11 shows the timing and geometry for the Mercury flybys with Earth occultations.

4.4 Trajectory Correction Maneuvers

In order to remain on course for MOI, the MESSENGER spacecraft will need to perform five deterministic (i.e., characteristics are known with reasonable uncertainty prior to launch) maneuvers with ΔV magnitude >40 m/s (DSMs), two much smaller deterministic maneuvers, and numerous statistical TCMs. The statistical TCMs will correct errors associated with maneuver execution, planetary flyby aim point, and trajectory perturbation force models. The more efficient LVA bipropellant thruster will be the primary thruster for each DSM. Most maneuvers requiring ΔVs from 3 to 20 m/s with Sun–spacecraft–ΔV angle between 78° and 102° will use the four 26-N C thrusters as primary thrusters. These four thrusters also serve as attitude control thrusters during DSMs. A pair of 4-N thrusters mounted on the

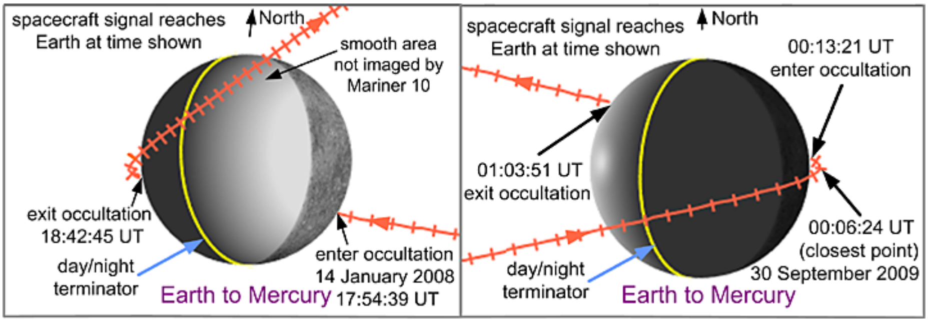

Fig. 11 Earth view of Mercury flyby trajectories occulted by Mercury

Table 4 ΔV budget after the first 35% of the heliocentric cruise phase

Maneuver category	ΔV (m/s)
Deep-space maneuvers	1032
Launch vehicle errors, navigation (99%)	115
Mercury orbit insertion	862
Mercury orbit-correction maneuvers	84
Contingency	136
Total	2229

spacecraft's sunward or anti-Sun side will provide the needed course correction when ΔV direction is within 12° of the Sun-to-spacecraft direction. Maneuvers with ΔV directions outside the above guidelines will require vector components from a combination of two of the above maneuver types.

The ΔV budget for MESSENGER is shown in Table 4. The 21 m/s reduction in ΔV capability (2,250 m/s in Table 2) since launch is attributable to increases in corrections after the first Venus flyby, propulsion system performance updates, and a six-day shift in DSM-2. While it is ideal to direct the spacecraft along a "minimum ΔV" trajectory, the MESSENGER mission will perform contingency plans and risk mitigation studies for each major propulsive maneuver. The ΔV budget for launch vehicle errors and navigation is a 99th percentile value derived from Monte Carlo analyses conducted by the navigation team. Since this ΔV budget category does not include the Mercury orbit phase, the "contingency" category includes variations in ΔV for MOI and OCMs.

MESSENGER's DSMs help target the spacecraft during the Earth–Venus 1, Venus 2–Mercury 1, Mercury 1–Mercury 2, Mercury 2–Mercury 3, and Mercury 3–MOI legs. The first two DSMs complete the partial targeting from the previous planetary flyby to deliver the spacecraft to the required aim point at the next planetary flyby. The last three DSMs move the next Mercury encounter closer to Mercury's MOI location. The pre-perihelion DSM-1 increases the spacecraft's Sun-relative speed, thereby raising aphelion and setting the Venus flyby 1 arrival conditions. The first DSM near aphelion, DSM-2, reduces the spacecraft's Sun-relative speed, lowering perihelion enough to target the first Mercury flyby. Placement of DSM-2 7.5 days before solar conjunction entry provides opportunity for recovery from either delays or errors, while limiting direct sunlight exposure on the LVA thruster. The ΔV magnitude for DSM-3 to DSM-5 (Table 5) is directly proportional to the change in

Table 5 Deep-space maneuvers during heliocentric transfer

Name	Maneuver date	Earth range (AU)	Sun range (AU)	Sun–S/C–ΔV (deg)	Sun–Earth–S/C (deg)	ΔV (m/s)
Requirement →				(78° to 102°)	(>3°)	
DSM-1	Dec 2005	0.688	0.604	92.4	37.5	315.6
DSM-2	Oct 2007	1.670	0.680	73.6	4.1	223.7
DSM-3	Mar 2008	0.677	0.685	87.4	43.3	73.4
DSM-4	Dec 2008	1.596	0.626	89.1	6.3	241.8
DSM-5	Nov 2009	1.529	0.565	89.5	7.4	177.5

Mercury's position (Fig. 7) between the previous and next Mercury encounters. These final three DSMs shift the upcoming Mercury encounter position counterclockwise with small increases in the spacecraft's Sun-relative speed.

5 Mercury Orbit Phase

By the time MESSENGER completes final preparations for both the mission-critical Mercury orbit insertion and the Mercury orbit phase, the mission design and navigation teams will be ready to conduct maneuver design, trajectory optimization, orbit determination, landmark tracking, and Mercury gravity field determination activities. A carefully devised Mercury approach navigation and TCM plan will precede a two-part conservative plan for Mercury orbit insertion. After Mercury orbit insertion, many factors such as early orbit determination and spacecraft health assessments will offer clues concerning what, if any, adjustment may be necessary to the orbit phase flight plan.

5.1 Mercury Orbit Insertion

The MESSENGER spacecraft's initial primary science orbit is required to have an 80° (±2°) orbit inclination relative to Mercury's equator, 200-km (±25 km) periapsis altitude, 12-hour (±1 minute) orbit period, 118.4° argument of periapsis (60°N periapsis latitude with 56°N to 62°N acceptable), and a 348° (169° to 354° acceptable) longitude of ascending node. These requirements are expressed in Mercury-centered inertial coordinates of epoch January 1.5, 2000.

The MOI strategy is a low-risk approach for placing the spacecraft into the primary science orbit as soon as possible within mission planning process constraints. This strategy uses two "turn while burning" variable thrust-direction maneuvers (MOI-1 and MOI-2) with the bipropellant LVA thruster providing the vast majority of each ΔV. To allow sufficient time for a rapid MOI-1 performance assessment and some post-MOI-1 orbit determination, the spacecraft will complete four full 16-hour orbits of Mercury (Fig. 12) between MOI-1 and MOI-2. For each MOI maneuver the thrust vector is nearly opposite to the spacecraft velocity vector. The time of MOI-2 is chosen to place daily data downlink periods (beginning about 40 minutes after periapsis on every other 12-hour orbit) between 8:10 am and 4:10 pm EDT (Eastern Daylight Time, 12:10 to 20:10 UT). During the second half of each Mercury year (e.g., 45 to 88 days after MOI-2) the daily downlink period shifts to 10:50 pm to 6:50 am EDT (02:50 to 10:50 UT). Even with time zone changes twice per year in

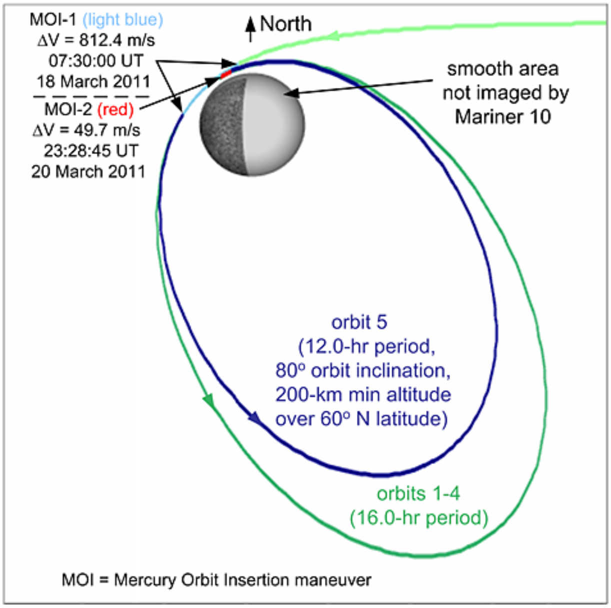

Fig. 12 View from the Sun of both Mercury orbit insertion maneuvers and the initial Mercury orbits. The longer-duration MOI-1 lasts nearly 14 minutes

the United States, and with time variations to the spacecraft's apoapsis introduced between OCM pairs, the daily downlink times remain close to standard daylight working shift hours for the JHU/APL-based (eastern United States) mission operations team about 50% of the time. A Sun–Earth–spacecraft angle $>17°$ ensures that solar interference will not disrupt spacecraft communications during the orbit insertion process. In addition, the spacecraft attitude enables a low-gain antenna to maintain contact between the spacecraft and Earth throughout both MOI maneuvers.

With MOI occurring when Mercury is near its perihelion, Mercury's rapid heliocentric orbital motion would, without a correction, quickly rotate the Sun-relative spacecraft orbit orientation (which moves far more slowly) until the sunshade is unable to shield the spacecraft bus at the required MOI-2 maneuver attitude. Although MOI-2 can occur less than three days after MOI-1, adding a small MOI-2 ΔV component normal to the orbit plane will meet the sunshade protection (Sun–spacecraft–ΔV angle range) constraint but will have little effect on orbit characteristics such as inclination.

5.2 Orbit Phase Navigation

A preliminary navigation accuracy analysis has shown that the planned DSN tracking coverage for MESSENGER's orbit provides sufficient navigation performance and margin throughout all mission phases, including the orbit phase. The preliminary navigation studies for the orbit phase used DSN Doppler and ranging that were conservatively weighted at 0.5 mm/s and 30 m, respectively, according to a DSN schedule of ten 8-hour tracks per week. Current DSN coverage includes 12 tracks per week. In addition, optical landmark tracking (planned during the orbit phase) was not included in the results below. Optical images of landmarks (e.g., craters) from MDIS enhance navigation performance by improving accuracy in the gravity field determination and by decreasing the time needed after OCMs or

Table 6 MESSENGER orbit uncertainties (1-σ) for a 10-day arc solution[1]

Orbit phase	Cross track (km)	Down track (km)	Out of plane (km)
Post Mercury orbit insertion	2	7	1
Mercury orbit (face-on viewing geometry)	17	100	5
Mercury orbit [2] (edge-on viewing geometry)	17	32	10

[1] Uncertainties include the effect of gravity uncertainties and gravity tuning, mapped to 10 days after the end of the arc

[2] Includes data outage due to spacecraft occultations by Mercury

momentum dumps for precise orbit determination. This navigation technique was first used successfully on the Near Earth Asteroid Rendezvous project (Miller et al. 2002), the first asteroid rendezvous mission.

Table 6 shows navigation orbit uncertainties in three orthogonal directions along the orbit: cross track (radial at periapsis and apoapsis), down track (in the direction of the orbit velocity vector), and out of plane (normal to the plane of the orbit). Table 6 includes uncertainties for three different orbit phases. The phases chosen for the study included the orbits immediately following MOI when the navigation team first tunes a model of the Mercury gravity field, the period where the orbits are nearly face-on when viewed from the Earth (near zero inclination to the plane-of-sky), and the period where the orbits are nearly edge-on when viewed from the Earth (near 90° inclination to the plane-of-sky).

The two viewing geometries represent near worst-case viewing angles for DSN Doppler tracking data, so these cases help bound the expected errors over the MESSENGER orbit. When viewing the orbit nearly face-on, the Doppler data are less sensitive to errors in the down-track directions because the line-of-sight to Earth is almost normal to the orbit velocity. This effect is seen in Table 6, where the down-track error for the face-on geometry is larger than that for the other two viewing angles. When viewing the orbit nearly edge-on, the Doppler data are less sensitive to errors in the out-of-plane direction, as the last column in Table 6 indicates. For other intermediate viewing angles to the plane-of-sky, experience has shown that the orbit errors are generally less than those at the bounding cases for face-on or edge-on geometry. Note that all three cases in Table 6 assume the same starting a priori gravity uncertainties. In reality the last two cases will occur some time after MOI; so some gravity tuning will have occurred and the errors should be less.

For the typical MESSENGER periapsis velocity of 3.8 km/s, the down-track errors in Table 6 imply a 1-σ (worst orbit viewing case) navigation timing uncertainty of about 26 s. A typical timing uncertainty will probably be less than 10 s (1-σ). Even when combined with maneuver execution error, the result is within the requirement for maintaining the MESSENGER orbit period to ±1 minute (2-σ). The cross-track uncertainties at periapsis indicate the uncertainty in periapsis altitude. Note that the cross-track uncertainties from Table 6 meet the requirement to control periapsis altitude between 125 km and 225 km (3-σ) after orbit insertion and after each maneuver to lower periapsis to within 60 km (3-σ) of the nominal target altitude of 200 km.

5.3 Gravity Tuning for Navigation

The navigation strategy for the orbit about Mercury includes gravity estimation and localized tuning of a twentieth degree and order spherical harmonic field in addition to the usual spacecraft dynamic parameters and orbit state parameters. Such a representation of the gravity field is at higher degree and order than can be determined from the available tracking so that high-frequency signatures in the Doppler data will not alias the estimates of the low degree and order harmonic coefficients. The uncertainties shown in Table 6 include the a priori gravity uncertainties derived from a scaled lunar field. Since sub-spacecraft periapsis longitude on Mercury moves 360° in 59 days, it will take at least this long to obtain measurements for a global gravity model; hence, determining a series of locally tuned gravity models will be part of navigation operations during the initial orbit phase.

The navigation strategy for orbit determination when MESSENGER is first in Mercury orbit is to perform incremental gravity field tuning as periapsis moves through 360° in longitude. A complete circulation of Mercury by the periapsis point takes about 118 orbits. During this time, the navigation team will estimate spacecraft state and other parameters along with a harmonic gravity field over 10-day-long arcs of Doppler and range data. The harmonic coefficient estimates from these fits will be highly correlated, but experience with other planetary orbiters has shown that this technique fits the short-period gravity perturbations with the least amount of aliasing (Christensen et al. 1979; Williams et al. 1983). The gravity field harmonic coefficients from each 10-day arc represent a local fit to the particular orbit data (especially periapsis) and are not valid globally.

This short-arc technique of gravity field estimation uses the Square Root Information Factorization (SRIF) method to separate arc-dependent parameter information from the gravity field information. This technique was developed and used on early planetary orbiters when the planet gravity field was mostly unknown and the sampling was sparse due to high-eccentricity orbits. Examples are Vikings I and II at Mars (eccentricity ∼0.8), Pioneer Venus (eccentricity ∼0.8), and Magellan (eccentricity ∼0.4). These early planetary orbiter missions had relatively large orbit eccentricities, in most cases even larger than that of MESSENGER (eccentricity ∼0.7).

Once solutions are available from several 10-day arcs, that information will be combined with subsequent 10-day arcs to "bootstrap" a gravity solution that is valid for MESSENGER over an increasing range of longitudes. This technique extracts both short-period information from the velocity change at periapsis and longer-period orbit evolution information over the 10-day arc. As more data are accumulated, longer or shorter arc fits will be evaluated and included if appropriate to obtain the best navigation gravity model on an ongoing basis. After the periapsis has completed a circulation of Mercury, a tuned gravity solution will be obtained by combining tracking data from the first 118 orbits. This first circulation solution produces a locally tuned field for navigation that will be best determined for a band of latitudes around the mean periapsis latitude for that circulation. Gravity tuning will continue on subsequent data arcs and subsequent circulations of Mercury by the orbit periapsis. Each subsequent gravity tuning will use the a priori gravity information from the previous "global" field solution to constrain the final gravity solution.

A covariance analysis was also performed on a typical 10-day arc in the orbit phase by assuming a twentieth degree and order lunar field scaled to Mercury by the ratio of the radii. The resulting gravity field harmonic coefficients were assumed to have an a priori uncertainty of 50%. After processing 10 days of Doppler and range data, the low degree and order terms (up to degree and order 3) improved by factors of 2 to 3, while the higher degree and order terms did not improve much ($<1\%$) and were highly correlated. The results should improve as data arcs are combined and a more globally valid gravity field estimate emerges.

5.4 Orbit Correction Maneuvers

After entering the initial primary science orbit, the spacecraft coasts for more than 12 weeks without OCMs. This is sufficient time to refine Mercury's gravity model and update perturbing force models in order to reduce 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 spatial variations in Mercury's gravity field will move periapsis north, increase orbit inclination, and rotate the low-altitude descending node away from the Sun (relative to the orbit orientation at Mercury's perihelion). The first OCM of each pair will impart a ΔV parallel to the spacecraft velocity direction at periapsis, placing the spacecraft on a transfer orbit (Fig. 13) with apoapsis altitude matching the apoapsis altitude 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 and returns orbit period to 12 hours with a much larger (primary thruster is the LVA) ΔV opposite the spacecraft velocity direction. Selecting 2.5 orbits between the OCMs provides enough time to assess the first OCM's performance, determine the new orbit, design the next maneuver, and test, upload, verify, and enable the ΔV update. Keeping periapsis altitude below 500 km while meeting the sunshade orientation requirement and science constraints requires that OCM pairs occur once per 88-day Mercury year. Each OCM pair occurs about 0.31 AU from the Sun, when the spacecraft orbit's line of nodes (connecting the spacecraft orbit's southern and northern equatorial plane crossings) is nearly perpendicular to the spacecraft–Sun direction. Table 7 provides date, ΔV, and spacecraft orientation data for all six planned OCMs.

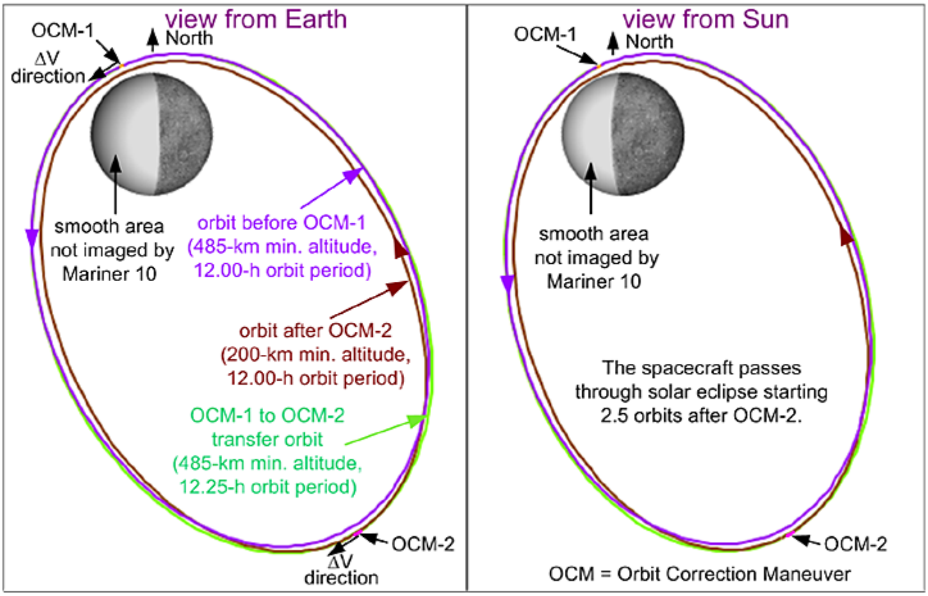

Fig. 13 View from the Earth and Sun of the first two orbit correction maneuvers, along with the preceding and subsequent spacecraft orbits

Table 7 Orbit correction maneuvers meet operational constraints

Orbit correction maneuver segment	Maneuver date (year month day)	ΔV (m/s)	Earth–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	4.22	1.319	96.1	3.3
OCM-2 start	2011 Jun 16	26.35	1.314	99.6	4.8
OCM-3 start	2011 Sep 09	3.92	1.097	86.0	16.1
OCM-4 start	2011 Sep 10	24.18	1.129	93.8	15.3
OCM-5 start	2011 Dec 05	3.59	0.683	82.1	3.2
OCM-6 start	2011 Dec 06	22.22	0.693	89.9	6.0

Deterministic ΔV for orbit phase = 84.48 m/s

5.5 Orbit Evolution

During the Mercury orbital phase the mission design and navigation teams will incorporate knowledge of past spacecraft attitude and predictions of spacecraft attitude to provide highly accurate orbit propagation and design of future OCMs. Trajectory perturbations caused by solar pressure, Mercury gravity field variation, solar gravity, end-of-life sunshade surface reflectance, and sunlight reflected off Mercury's surface (albedo effect) will be carefully coordinated with the spacecraft's planned attitude profile. All of these factors (except for Mercury albedo) are accounted for during Mercury orbit-phase propagations.

Solar gravity and Mercury oblateness, J_2, are major contributors to periapsis altitude increasing to 485, 467, and 444 km before OCM-1, -3, and, -5, respectively; periapsis latitude drifting north by $\sim$12°; orbit inclination increasing by about 2°; and longitude of ascending node (orbit plane-to-spacecraft–Sun line orientation) decreasing 6°. Figures 14 and 15 show the variation of solar eclipse duration, periapsis altitude, and periapsis latitude during the nominal one-year Mercury orbit phase.

Current orbit phase design uses a propagator that assumes a Mercury gravity model with normalized coefficients $C_{20} = -2.7 \times 10^{-5}$ and $C_{22} = 1.6 \times 10^{-5}$ (Anderson et al. 1987). Spacecraft attitude rules assume 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) each day, and solar array tilt varying as a function of solar distance (Fig. 3).

6 Mission Status

For the first 26 months after launch, Mission Design and Navigation have not encountered any significant change from the initial schedule of post-launch activities. The excellent functional health of the spacecraft has enabled the accomplishment of nominal pre-launch planned activities such as maneuver design, trajectory optimization, orbit determination, and solar radiation pressure model adjustment. Careful planning and testing of the maneuver prior to implementation produced six highly successful TCMs with average ΔV errors of 1.2% in magnitude and 1.0° in direction. In addition to the navigation and mission design teams, the guidance and control and mission operations teams have significant responsibility in the maneuver design process.

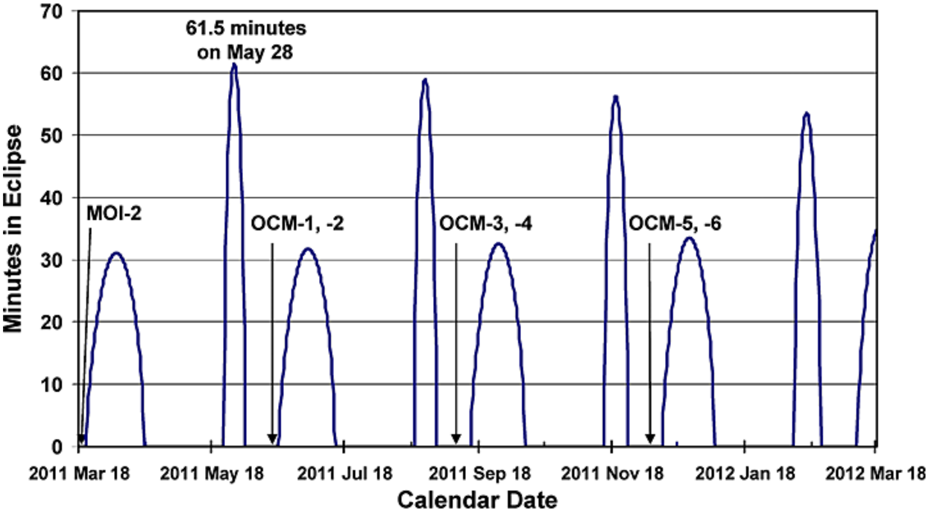

Fig. 14 Solar eclipse duration (short eclipses include periapsis) during Mercury orbit. Maximum eclipse duration decreases as periapsis latitude moves north

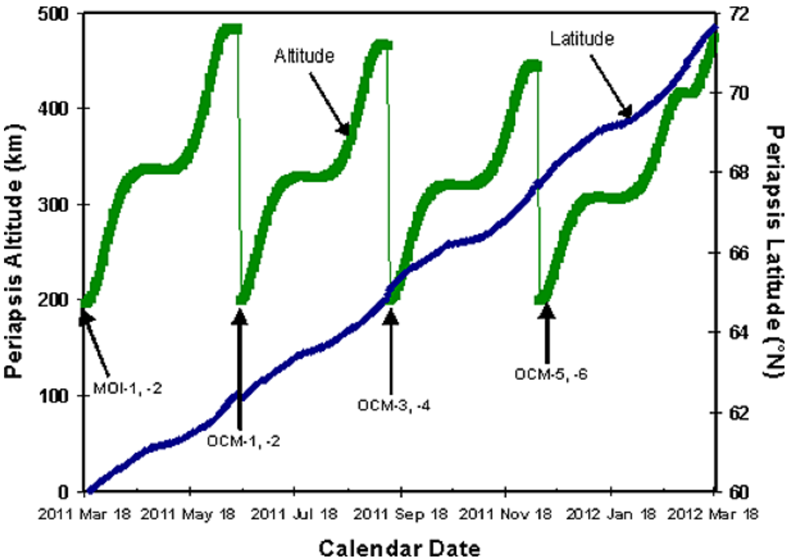

Fig. 15 Periapsis evolution during Mercury orbit phase

6.1 Launch and Early Operations

After launch, the goal of the navigation team has been to establish that the MESSENGER spacecraft is on the proper trajectory to return to the optimal Earth flyby condition. OD analysis is performed to calibrate the non-gravitational models for the spacecraft, and trajectory maneuvers are designed, executed and reconstructed to ensure that the predicted trajectory reaches the target. Prelaunch statistical analysis determined that the 99th-percentile launch error could be as large as 38 m/s, but after the first couple of days of DSN tracking were processed, the total correction required was only 21 m/s.

To account for the effects of solar radiation pressure (SRP), the navigation team uses a model made up of ten idealized flat plates that are oriented so as to approximate the SRP characteristics of the MESSENGER spacecraft bus (six plates) and the two articulating solar arrays (four plates for front and back of the independently pointed solar panels). During the OD process, scale factors for each of the plates in the SRP model are estimated along with the other spacecraft and observation model parameters. The SRP scale factor estimates depend on the spacecraft orientation model, which is driven by the attitude history and prediction file provided by the Mission Operations Center. The initial post-launch spacecraft attitude oriented the $+y$ spacecraft axis (opposite the sunshade) with the sunshade pointed away from the Sun. The spacecraft orientation has varied since launch, including an early slow spin about the Sun direction, and also subsequent pointing with small attitude rates about the Sun line to keep the appropriate antenna pointed toward Earth.

Solar radiation pressure scale factor estimates during the first few OD deliveries were about ninety percent of their a priori values. A scale factor of 1.0 indicates that the SRP model for that component is 100% effective. After a few weeks of data collection, however, the estimates began changing with data arc length and became sensitive to data weighting experiments. This result indicated a possible corruption of the solution due to the presence of un-modeled acceleration; so the navigation team began an analysis to check for early out-gassing accelerations that are common immediately after launch. Figure 16 shows the effect of delaying the beginning of the data arc on the SRP scale factor estimates for the parts of the SRP model illuminated during cruise with the spacecraft $+y$ axis oriented toward the Sun. These are flat plate Area6, representing the $+y$ projection of the spacecraft main body, and Area7 and Area8 representing the front side of the two solar panels. The solutions tend

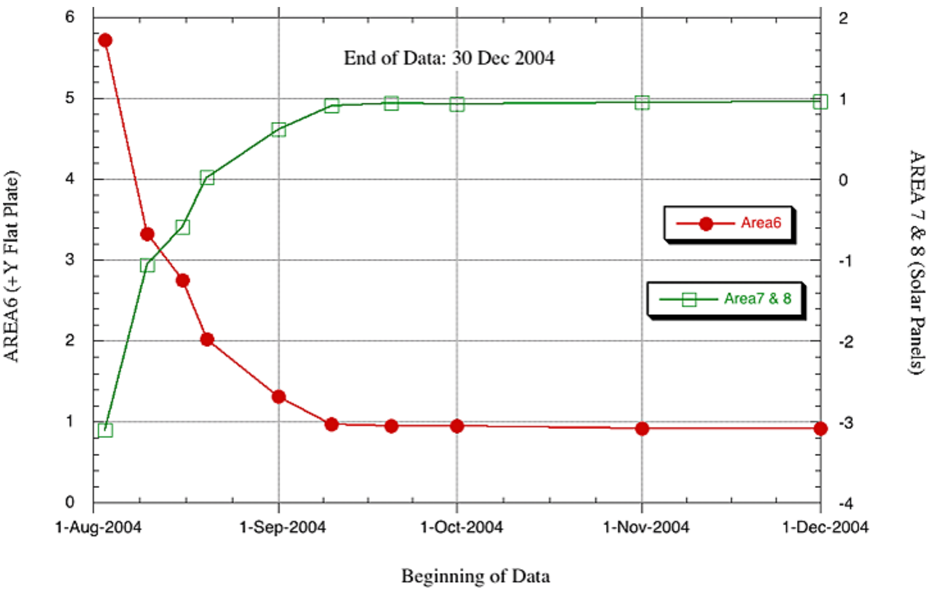

Fig. 16 Change in solar radiation pressure scale factor as a function of cutoff date for trajectory data. Effective area change prior to 10 September 2004 indicates that other non-gravitational accelerations (such as out-gassing) are present

to stabilize if the data arc excludes all data before about 10 September 2004, indicating that other un-modeled accelerations present before that time are aliasing into estimates of the scale factors. With such exclusion, the SRP acceleration estimates for the illuminated flat plates stabilized at between 90 and 96% of their pre-launch a priori values.

6.2 Maneuver Performance

Maneuver reconstruction results for the first seven TCMs appear in Table 8. The total ΔV expended thus far includes the 21 m/s required to correct launch injection errors and deterministic maneuvers in November 2004 and December 2005. By restricting the maximum size of the first burn to 18 m/s, more time became available for analysis and checkout of the

Table 8 Maneuver reconstruction for the first seven trajectory correction maneuvers

Parameter	Estimated	Planned	Reconstructed uncertainty (1-σ)	Difference from planned
TCM-1 at 21:00:07 UT, 24 August 2004, spacecraft event time				
ΔV (m/s)	17.9009	18.0000	0.0017	−0.099 (−0.55%)
RA (deg)	242.5414	242.8288	0.0032	−0.2874
Dec (deg)	−18.4960	−18.6426	0.0032	0.1466
TCM-2 at 18:00:00 UT, 24 September 2004, spacecraft event time				
ΔV (m/s)	4.5886	4.5899	0.0024	−0.001 (−0.03%)
RA (deg)	271.1789	271.3752	0.0096	0.1963
Dec (deg)	−24.1171	−24.3248	0.0171	0.2077
TCM-3 at 19:30:00 UT, 18 November 2004, spacecraft event time				
ΔV (m/s)	3.2473	3.2365	0.0006	0.011 (0.33%)
RA (deg)	315.7591	316.0507	0.0030	−0.2916
Dec (deg)	−15.2193	−15.4143	0.0152	0.1950
TCM-5 at 14:30:00 UT, 23 June 2005, spacecraft event time				
ΔV (m/s)	1.1033	1.1451	0.0011	−0.0418 (−3.65%)
RA (deg)	239.3472	239.6439	0.0438	−0.2967
Dec (deg)	3.7469	3.9750	0.0391	−0.2281
TCM-6 at 18:00:00 UT, 21 July 2005, spacecraft event time				
ΔV (m/s)	0.1505	0.0002	0.1468	0.0037 (2.51%)
RA (deg)	255.6900	252.1045	0.1548	3.5854
Dec (deg)	3.4969	6.3595	0.1503	−2.8626
TCM-9 at 11:30:00 UT, 12 December 2005, spacecraft event time				
ΔV (m/s)	315.6334	315.7200	0.0004	−0.0866 (−0.03%)
RA (deg)	217.2755	217.2570	0.0001	0.0185
Dec (deg)	−4.8828	−4.8652	0.0004	−0.0177
TCM-10 at 16:00:00 UT, 22 February 2006, spacecraft event time				
ΔV (m/s)	1.2807	1.4071	0.0005	−0.1264 (−8.98%)
RA (deg)	166.2108	168.8002	0.1273	−2.5894
Dec (deg)	9.2538	9.2222	0.3218	0.0316

[1] ΔV spherical angles, right ascension (RA), and declination (Dec) are in Earth mean equator of J2000 (1 Jan 2000 at 12:00:00 ET) coordinates

spacecraft systems prior to the completion of launch injection error correction. The design of the first maneuver, TCM-1, was split into two burns: the first part on 24 August 2004, and the remainder of about 4 m/s (renamed as TCM-2) on 24 September 2004. The first deterministic maneuver, re-optimized for execution on 18 November 2004, was renamed TCM-3. Contingency maneuvers before the Earth flyby (TCM-4 and TCM-7) and after the Earth flyby (TCM-8) were cancelled because it was more prudent to wait until the next planned TCM. TCM-5 and TCM-6 provided precise targeting for the 2 August 2005 Earth flyby. TCM-9 (also called DSM-1) included a 1-m/s over burn bias (compared with the optimal design) in order to maximize the potential for having a small clean-up maneuver (TCM-10) at a favorable spacecraft attitude. Additional maneuvers completed during the year 2006 include the 2.28-m/s TCM-11 (which used two components to maintain sunshade orientation) on September 12 and 0.50-m/s TCM-12 on October 5 for targeting the first Venus flyby. The final maneuver during 2006, the 36-m/s TCM-13 on December 2, corrected target offsets at the first Venus flyby and directed the spacecraft to a re-optimized aim point at the second Venus flyby. The co-location of solar conjunction ($<3°$ Sun–Earth–spacecraft angle) from 18 October to 17 November 2006 and a 2987-km-altitude Venus flyby on 24 October 2006 contributed to the need for a large correction at TCM-13.

The time shown for each TCM is the actual maneuver execution time on the spacecraft. Due to initialization and fuel settling burns, the onset of acceleration observed in the received Doppler tracking data is typically delayed by several seconds. The reconstruction accounted for these delays when modeling the TCM acceleration in the OD filter. Table 8 compares the estimated ΔV magnitude and direction, and their 1-σ uncertainty, to the planned design values.

For most maneuvers performed thus far, the ΔV magnitude is within 0.1 m/s of the goal and the direction is within a few tenths of a degree. Not shown in Table 8 is the total directional error for each TCM, which for TCM-1 was 0.309°, for TCM-2 was 0.274°, for TCM-3 was 0.342°, for TCM-5 was 0.374°, for TCM-6 was 4.577°, for TCM-9 was 0.026°, and for TCM-10 was 2.556°. The largest TCM performance variations (difference between the achieved and goal ΔV vectors) are associated with use of 4-N thrusters as primary thrusters.

7 Summary

A little over four decades after the first Mercury mission trajectory design studies, and about three decades after Mariner 10's three Mercury flybys, MESSENGER, the first Mercury orbiter mission, launched safely on 3 August 2004. While there are many innovative trajectory options for a Mercury orbiter, only a small subset are compatible with the MESSENGER spacecraft configuration, science objectives, and operational constraints. With launch delayed to the third launch opportunity of 2004, the spacecraft will achieve most of the required trajectory alteration using flybys of Earth (one), Venus (two), and Mercury (three). Mercury orbit insertion on 18 March 2011 will mark the start of a one-year Mercury orbit phase.

The launch and early operations phase has been successful for the MESSENGER mission. The mission design and navigation teams have performed the planned post-launch trajectory re-optimization and designed the first ten trajectory correction maneuvers to deliver the MESSENGER spacecraft to the October 2006 and June 2007 Venus flybys and then on to Mercury. The cost of correcting the launch injection error (21 m/s) was less than the

99th percentile, worst-case correction computed pre-launch (∼38 m/s). The early ΔV "savings" over what could have happened in the worst case, coupled with indications of precise maneuver capability, bode well for having adequate fuel for the remainder of the mission.

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