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MESSENGER MISSION OVERVIEW AND TRAJECTORY DESIGN

James V. McAdams*

MESSENGER (MErcury Surface, Space ENvironment, GEochemistry, and Ranging) will be the first spacecraft to orbit the planet Mercury when this NASA Discovery Program mission begins its one-year Mercury orbit phase in 2009. The spacecraft will utilize two or three Venus flybys (two for the March 2004 baseline launch, three for the May 2004 backup launch) and two Mercury flybys during its 5-year ballistic trajectory to Mercury. The baseline and backup launch opportunities have similar designs for launch, planetary flybys, and propulsive maneuvers. Planned trajectory-correction maneuvers include two during the heliocentric orbit, two for Mercury orbit insertion, and six for periherm and orbit-period adjustment during the Mercury orbit phase. The initial primary science orbit has 80° orbit inclination, 200-km periherm altitude, 60°N sub-spacecraft periherm latitude, and 12-hour orbit period.

BACKGROUND

Except for Mercury, each of the six planets closest to the Sun have been explored by more than one spacecraft. Mercury, the closest planet to the Sun, has a 3:2 spin-orbit resonance and the greatest diurnal range in surface temperature. Earth-based radar observations yielded data suggesting ice deposits in permanently shadowed craters near Mercury's north pole. During its three flybys of Mercury between March 1974 and March 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.[1,2] Solid support for in-depth exploration of the planet Mercury is documented in NASA's Space Science Strategic Plan[3] and planetary scientists' contributions to the National Research Council's Solar System Exploration Decadal Survey.[4]

Recent improvements in ballistic trajectory design and spacecraft technology have dramatically lowered both the cost and energy requirements for Mercury orbiter missions. While mission studies conducted in the 1970s lowered Mercury orbit-insertion propellant requirements by utilizing Venus gravity assists,[5, 6, 7, 8] studies during the 1980s lowered total ΔV required by using reverse ΔV-gravity assists with Mercury flybys[9,10,11]. The most recent of these studies, concluded in 1990 by the Jet Propulsion Laboratory (JPL), used two 800-kg dry mass orbiters each having 3.2-km/s onboard ΔV capability using a Titan IV/Centaur launch vehicle. The MESSENGER (MErcury Surface, Space ENvironment, GEochemistry, and Ranging) mission will use one 500-kg dry mass orbiter with 2.3-km/s onboard ΔV capability using a Delta II 7925H launch vehicle. The JPL study's best opportunity in 2004 using two Venus and two Mercury flybys required 28.4-km^2/s^2 launch energy for a one-day launch period and a 3.8-year transit from Earth to Mercury orbit insertion. MESSENGER's best opportunity in 2004 also uses two Venus and two Mercury flybys, yet requires only 15.7-km^2/s^2 launch energy for a 20-day launch period with a longer 5.1-year transit from Earth to Mercury orbit insertion.

* MESSENGER Mission Design Lead, The Johns Hopkins University Applied Physics Laboratory, 11100 Johns Hopkins Road, Laurel, MD 20723-6099

Since 1990 Mercury orbiter mission studies continued the trend of reduced propulsive requirements. These studies include JPL and TRW's Discovery-class Hermes Orbiter[12] in 1994, European Space Agency's 1994 Mercury orbiter[13] study, and the 1998 ISAS (Japan's Institute of Space and Astronautical Science) Mercury orbiter[14] study. The ISAS Mercury orbiter (August 2005 launch, 16.0 km^2/s^2 launch energy, 2.9-km/s onboard ΔV) used two Venus and two Mercury flybys as part of the 4.2-year heliocentric transfer to Mercury orbit insertion. The most dramatic improvements in ballistic trajectory optimization were made in 1999 by Langevin,[15] who further reduced launch energy and onboard propellant requirements by using up to five Mercury flybys, and in 2001 by Yen,[16] who improved Venus-to-Venus transfer phasing to optimize spacecraft trajectory shaping from Venus gravity assists. Langevin's 2005 launch, five-Mercury-flyby trajectory required a 6.0-year heliocentric transfer and only 1.9-km/s onboard ΔV (assuming 0.6 km/s for navigation, Mercury orbit corrections, and margin). Unfortunately, spacecraft thermal heating, Mercury orbit arrival inclination, and extended flight time all violated MESSENGER mission constraints. Serious overheating of the MESSENGER spacecraft would have occurred during the fourth Mercury flyby, which required a low-relative-velocity, low-altitude encounter over the sub-solar point when Mercury was approaching its perihelion. The Mercury arrival orbit inclination would have to be changed after orbit insertion from 90º to 80º, thereby requiring additional ΔV, risk, and operational complexity. MESSENGER's heliocentric trajectory design is based on Yen's work in 2001.

In 1997 and 1999, competitive Mercury orbiter mission proposals were prepared by The Johns Hopkins University Applied Physics Laboratory (JHU/APL) and by JPL for consideration by NASA's Discovery Program of low-cost planetary exploration missions. Although neither proposal made the final selection in 1997, NASA awarded primary management responsibility to the JHU/APL team in July of 1999. The 1999 reference trajectory[17] with which MESSENGER began Phase B in early 2000 required 16.0-km^2/s^2 launch energy for a 15-day launch period with a longer 5.5-year transit from Earth to Mercury orbit insertion and a 2.7-km/s onboard ΔV capability. MESSENGER, NASA's seventh Discovery Program mission, draws leadership from the Carnegie Institution of Washington and JHU/APL. A consortium including various NASA centers, numerous industry partners, and many educational and research institutions completes the MESSENGER Team.

SCIENCE OBJECTIVES AND INSTRUMENT SUITE

The MESSENGER mission will address six important scientific questions[18] on Mercury's composition and field structure. Answers to these questions, which will offer insights far 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.

Recent "comprehensive" Mercury orbiter mission studies often utilize advanced propulsion, two orbiters, and sometimes a surface vehicle. These studies have typically encountered numerous obstacles that prevented or delayed full funding from space agencies. In order to provide significant mission cost reduction along with the maximum science return, the MESSENGER study team elected to use one Mercury orbiter spacecraft in a low-maintenance orbit without surface landers. MESSENGER's seven-instrument science payload[19] and primary science orbit at Mercury were designed to maximize science return directly related to the six scientific objectives using a single spacecraft. Complementary to the 50-kg, 84-W payload is the X-band transponder, the primary spacecraft component for radio science. Table 1 provides each science instrument's name, acronym, and primary purpose(s) or measurement objectives. Four of the seven science instruments are mounted on the +z-axis deck within the launch vehicle adaptor.

Table 1

MESSENGER SCIENCE PAYLOAD

Science Instrument Name	Acronym	Major Objectives
Mercury Dual Imaging System	MDIS	map Mercury's surface in visible to near-infrared wavelengths
Gamma-Ray and Neutron Spectrometer - Gamma-Ray Spectrometer - Neutron Spectrometer	GRNS GRS NS	measure surface elemental abundances; characterize polar deposits
X-Ray Spectrometer	XRS	determine element composition of Mercury's surface by solar-induced X-ray fluorescence
Magnetometer	MAG	measure magnetic field of Mercury
Mercury Laser Altimeter	MLA	measure libration of Mercury and topography of northern hemisphere
Mercury Atmospheric and Surface Composition Spectrometer - Ultraviolet-Visible Spectrometer - Visible-Infrared Spectrograph	MASCS UVVS VIRS	measure surface reflectance and exospheric particle emissions during Mercury limb scans
Energetic Particle and Plasma Spectrometer - Fast Imaging Plasma Spectrometer - Energetic Particle Spectrometer	EPPS FIPS EPS	examine ion species in and around Mercury's magnetosphere
Radio Science - X-band Transponder	RS	measure Mercury's gravity field; support laser altimetry

MESSENGER's imagers[19] (MDIS) include the 10.5º field-of-view (FOV) wide angle (WA) imager and the 1.5º FOV narrow angle (NA) imager. Mounted on opposite sides of a pivoting platform, each imager can point from 50º toward the Sun to nadir, where they are coaligned with the other instruments, to 40º anti-sunward. The NA imager spatial resolution ranges from 10 m at periherm to 390 m at apoherm of the 200-km altitude by 12-hour period primary science orbit. The WA imager spatial resolution ranges from 72 m to 5.4 km.

SPACECRAFT DESCRIPTION

Operational reliability, simplicity, and low cost were the main drivers for the design, construction, and testing of the MESSENGER spacecraft. Operational reliability was achieved via a high level of functional redundancy, whereby nearly every critical function of the spacecraft has backup capability in the event of component failure. Operational reliability was the primary factor behind the design of spacecraft autonomy, or equipping the spacecraft with the ability to detect serious problems and subsequently configure its attitude and power consumption to prevent or minimize damage while awaiting command instructions from Mission Operations. In a harsh thermal environment where the Sun's energy at Mercury's perihelion is 11 times greater than at launch, the spacecraft must be fully capable of detecting major problems and placing itself in a safe mode of operation. Simplicity is demonstrated by the fact that every component of the spacecraft, excluding the solar arrays and MDIS instrument, is fixed and body-mounted to the spacecraft structure. Low cost for the spacecraft is achieved by combining proven technology with new technology as necessary to minimize risk and maximize science return.

A flight-configuration view of the MESSENGER spacecraft (Figure 1), shows the side of the spacecraft usually protected from direct sunlight by the sunshade. The three-axis stabilized spacecraft has four reaction wheels for pointing control. Two solar arrays with 5.3 m^2 maximum Sun-facing area, populated with 2/3 optical solar reflectors and 1/3 multi-junction cells, provide 400-W power at

MESSENGER's maximum solar distance of 1.0 AU. Once the spacecraft's solar distance matches Mercury's distance from the Sun (0.30 to 0.46 AU), the solar arrays maintain an operational temperature below 150ºC by tilting up to 65º away from the Sun. A 23-ampere-hour NiH_2 battery provides power during solar eclipse. The telecommunications subsystem employs X-band transponders, four low-gain antennas, two medium-gain fanbeam antennas, and two high-gain phased-array antennas to provide the capability for a communication link for all mission-critical events. Downlink data rates can be as high as 104 kbps when using the 70-meter diameter Deep Space Network (DSN) antennas. When the spacecraft is < 0.8 AU from the Sun, a sunshade allows at least 12º-off-Sun pointing in any direction and protects the spacecraft (excluding solar arrays) from direct sunlight exposure. MESSENGER can store up to 6.61×10^9 bits of data on its 8.00×10^9 bit capacity solid-state recorder. The dual-mode propulsion system uses a fuel/oxidizer hydrazine/nitrogen-tetroxide mix for the 660-N large-velocity-adjust (LVA) thruster (specific impulse of 316 seconds) and 16 smaller monopropellant thrusters (specific impulse of 200 to 230 seconds). The monopropellant thrusters include four 22-N LVA-TVC (thrust vector control) on the same deck as the LVA thruster and twelve 4.4-N thrusters used for attitude control and trajectory correction maneuvers (in any direction) requiring < 3 m/s ΔV.

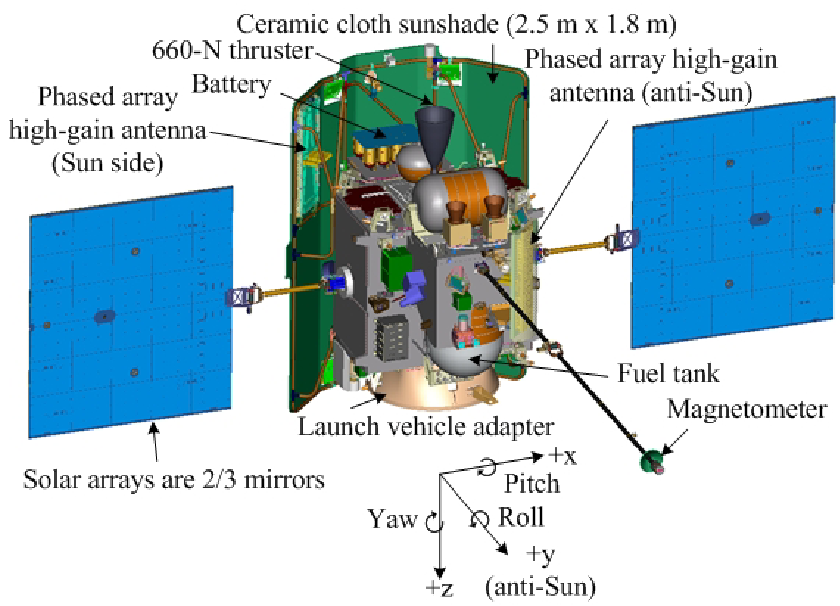

Figure 1 Flight Configuration of the MESSENGER Spacecraft

HELIOCENTRIC TRAJECTORY

The spacecraft will utilize two Venus gravity assists, two Mercury gravity assists, and two deep space maneuvers during its 5.1-year ballistic trajectory to Mercury (Figure 2). Although a third Mercury gravity assist offers significant reduction in spacecraft propulsive requirements (e.g., up to 683 m/s or nearly 30% of the onboard propellant if all the ΔV was expended in bipropellant maneuvers), the resulting 24% increase in mission duration limits this option to recovery from a major anomaly. The MESSENGER spacecraft's 12.7 orbits of the Sun include nine orbits that appear to rest on top of each other. This trajectory design technique, known as phasing orbits, maximizes the trajectory shaping available from the Venus and Mercury flybys for a given launch date and trip duration limits between launch and Mercury orbit insertion. A 2.8:3.8 Venus:spacecraft orbit resonance between the Venus flybys indicates 2.8 Venus orbits of the Sun while the spacecraft completes 3.8 orbits. Similarly, a 2:3 Mercury:spacecraft orbit resonance occurs between Mercury flybys, and a 3:4 Mercury:spacecraft orbit resonance occurs between Mercury flyby 2 and Mercury orbit insertion. Although the Planetary Flybys section will explain the primary trajectory shaping from each Venus and Mercury flyby, the goal is to use as little onboard propellant as possible to lower the spacecraft's velocity relative to Mercury's velocity at orbit insertion.

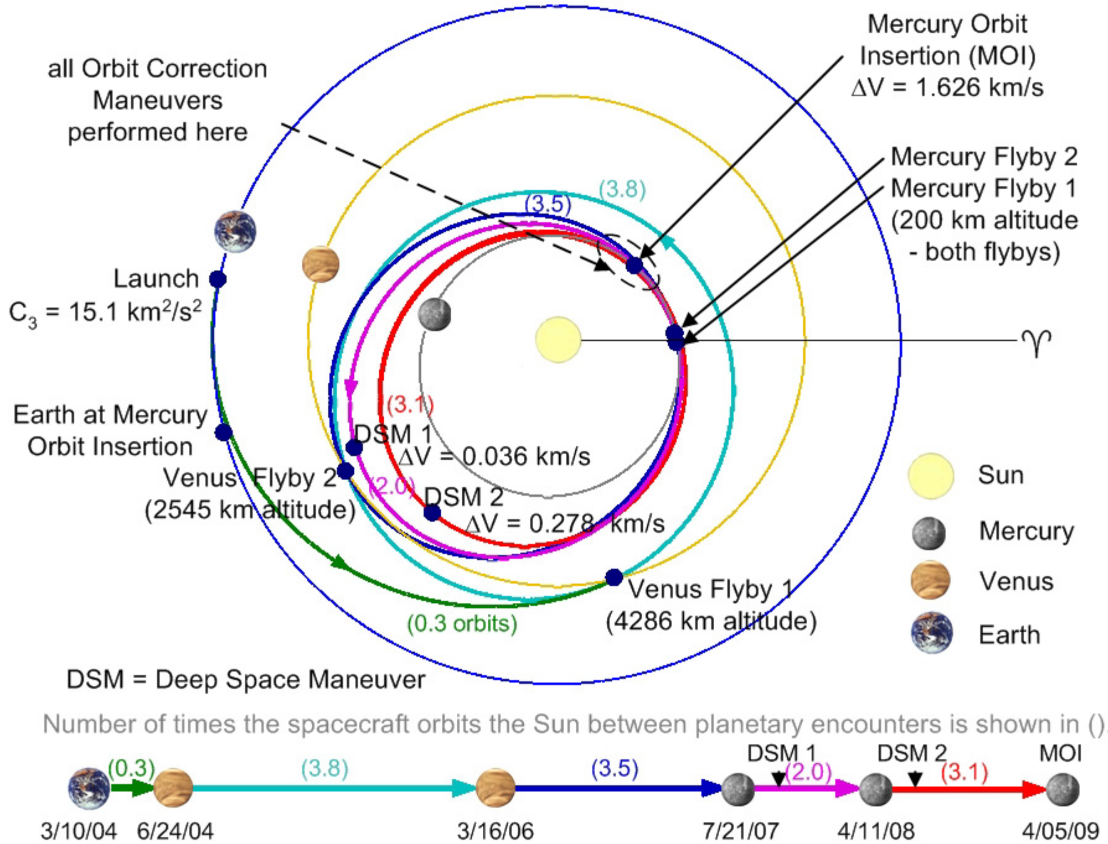

Figure 2 North Ecliptic Pole View of Baseline Heliocentric Trajectory

An important aspect of the heliocentric trajectory, solar conjunctions, must be factored into the spacecraft's functional design and trajectory navigation strategy. Solar conjunctions for spacecraft that are always closer to the Sun than Earth, as MESSENGER is, are defined by a Sun-Earth-spacecraft angle < 3º. During a solar conjunction, the Sun either degrades or prevents communication with the spacecraft – whether the spacecraft is between Earth and the Sun (inferior conjunction) or on the opposite side of the Sun (superior conjunction). Special attention is given by mission planners to the longest solar conjunctions and to solar conjunctions near important spacecraft activities. For MESSENGER the longest solar conjunction, lasting just under 20 days and not occurring near an important spacecraft activity, begins about three months after the first Venus flyby. During MESSENGER's heliocentric trajectory only two solar conjunctions, lasting about five days each, occur near mission-critical events. The first solar conjunction begins about two days after the second Mercury flyby and the second ends about three days before Mercury orbit insertion. The longest solar conjunction during Mercury orbit phase lasts about 9.5 days, of which 6.3 days have the Sun-Earth-spacecraft angle < 2º.

LAUNCH WINDOW AND ΔV ALLOCATION

During March of 2004 a Delta II 7925H launch vehicle will place the MESSENGER spacecraft into the heliocentric transfer trajectory targeted to the first Venus flyby. MESSENGER will be only the third spacecraft to be launched with this Delta configuration. Given a probability of command shutdown exceeding 99% for the first 19 days of the 20-day (10-29 March) launch window, the Delta rocket can deliver a spacecraft weight of 1130 kg to the required launch energy (C_3) of 15.7 km^2/s^2. Daily launch window duration is 12 seconds.

MESSENGER’s launch trajectory from lift off to first DSN contact is shown in Figure 3. The parking orbit coast period of about 26 minutes is the shortest of two parking orbit coast periods for the given launch time. This coast period ends at SECO2, shortly after the launch vehicle exits Earth’s shadow. All launch sequence mission-critical events are monitored with ground-based antennas. Figure 3 shows the location of a 34-m-diameter antenna (Deep Space Station 45 or DSS-45) at Canberra, Australia, that is one of the first DSN antennas able to contact the separated spacecraft.

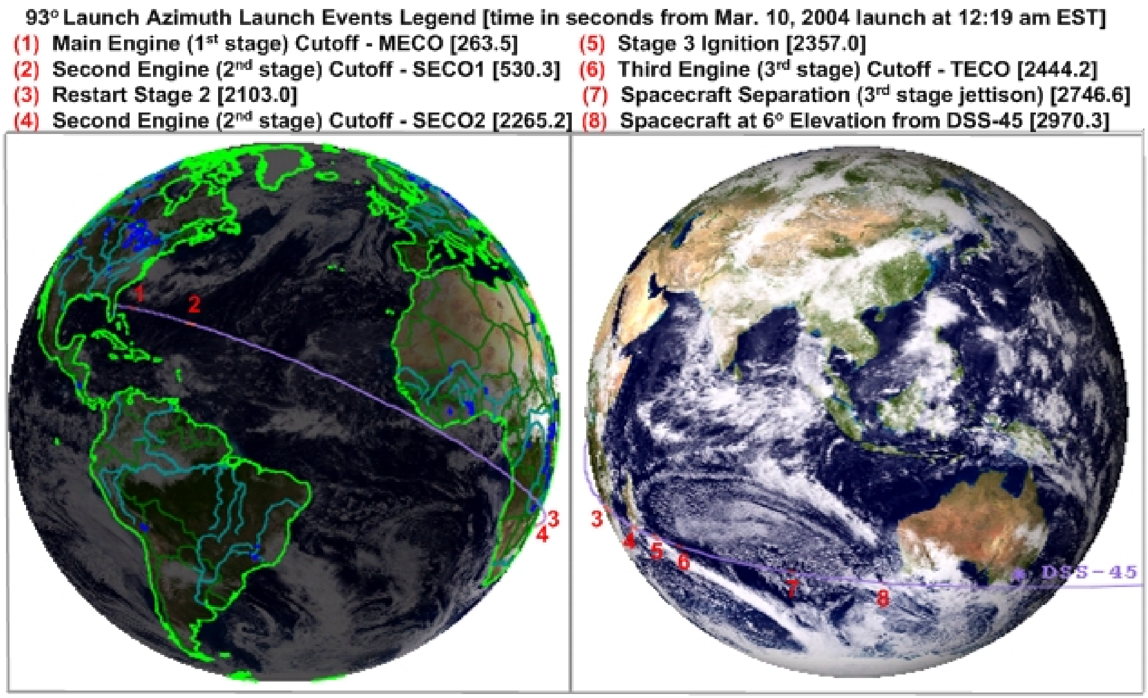

Figure 3 Key Launch Trajectory Events for the 10 March 2004 Launch

Details of MESSENGER’s 20-day launch window are provided in Table 2. The launch energy (C_3) and declination of launch asymptote (DLA) are fixed for selected dates in order to reduce the number of launch vehicle ascent-to-orbit trajectories. The pairing of dates for this process yielded < 5 m/s extra ΔV cost (compared to minimum total ΔV) to the spacecraft. The maximum launch energy occurs at the end of the launch window. The two-part Mercury orbit insertion ΔV (ΔV_{MOI}) is constant for consecutive launch dates throughout the launch window to meet navigation constraints for Mercury approach. Using a bright star in Mercury approach optical navigation images will improve orbit determination and reduce target uncertainties for the spacecraft’s primary Mercury orbit insertion maneuver.

Table 2

BASELINE LAUNCH WINDOW (INTEGRATED TRAJECTORIES)

Launch Date (2004)	C_3 (km^2/s^2)	DLA (deg)	Venus Flyby Altitude (km)		Deep Space Maneuvers (m/s)		ΔV_{MOI} (m/s)	ΔV_{MARGIN} (m/s)
			1st	2nd	ΔV_1	ΔV_2		
March 10	15.109	11.71	2545	4286	36	278	1626	94
March 11	15.027	11.70	2572	4294	39	272	1626	97
March 12	14.963	11.70	2598	4298	40	274	1626	94
March 13	14.912	11.70	2624	4302	42	272	1626	94
March 14	14.912	11.70	2647	4325	43	268	1625	97
March 15	14.912	11.70	2670	4328	44	269	1625	95
March 16	14.799	11.65	2691	4361	45	262	1620	106
March 17	14.799	11.65	2713	4364	47	262	1620	105
March 18	14.799	11.65	2733	4366	50	262	1620	103
March 19	14.799	11.65	2751	4397	50	256	1618	110
March 20	14.829	11.66	2769	4399	51	255	1618	109
March 21	14.869	11.67	2786	4399	52	256	1618	107
March 22	14.891	11.62	2799	4444	54	254	1625	100
March 23	14.955	11.62	2813	4447	55	249	1625	104
March 24	15.035	11.61	2826	4448	56	247	1625	106
March 25	15.132	11.56	2837	4449	57	247	1625	105
March 26	15.238	11.45	2847	4448	59	247	1625	103
March 27	15.321	11.42	2851	4489	62	244	1632	96
March 28	15.475	11.56	2857	4489	62	240	1632	99
March 29	15.661	11.65	2861	4491	63	239	1632	100

The ΔV allocation for MESSENGER is listed by category in Table 3. The minimum ΔV for contingencies, and the corresponding maximum ΔV for deterministic maneuvers, are also shown. With over 70% of the total ΔV allocated to Mercury orbit insertion (MOI), extensive studies have been and will be conducted to ensure safe and efficient MOI with well-formulated contingency plans. The ΔV budget for launch vehicle and navigation errors is a 99th percentile value determined using Monte Carlo analyses. Since this ΔV budget does not apply to the Mercury orbit phase, the "contingencies" category covers variations in expected ΔV for Mercury orbit correction maneuvers.

Table 3

ΔV ALLOCATION

	ΔV (m/s)
Deep space maneuvers	314
Launch vehicle, navigation errors (99%)	180
Mercury orbit insertion	1625
Mercury orbit correction maneuvers	87
Contingency	94
Total	**2300**

DEEP SPACE MANEUVERS

The MESSENGER spacecraft's heliocentric trajectory requires only two deterministic maneuvers, known as deep space maneuvers (DSMs). Since the ΔV magnitude of each DSM is much greater than 10 m/s, the more efficient LVA bipropellant thruster will be used. Constraints for LVA thruster usage include a sunshade-protection requirement for the Sun-spacecraft-ΔV angle to be between 78º and 102º when Sun-spacecraft distance is < 0.8 AU and a solar-conjunction avoidance requirement that the Sun-Earth-spacecraft angle exceed 3º. Along with the solar conjunction avoidance requirement, Earth's orientation relative to the spacecraft body frame must provide a link margin > 3 dB using one of the four low-gain antennas. The lowest link margin for any MESSENGER DSM, including analysis of different launch dates and contingency options, is 3.2 dB.

Each LVA ΔV has four segments – fuel settle, auxiliary tank refill, main burn, and ΔV trim. For the 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 two of the four 22-N LVA-TVC thrusters. The second burn segment channels fuel into the auxiliary fuel tank (as needed depending on the fuel already in the auxiliary tank) and continues thruster activity using two of the four 22-N LVA-TVC thrusters. The vast majority of the LVA ΔV achieves up to 672-N thrust with the combustion products of fuel and oxidizer exiting through the LVA thruster. The fourth segment, ΔV trim, provides a precision clean up of the overall ΔV target using all four 22-N LVA-TVC thrusters. During each burn segment a combination of secondary 22-N LVA-TVC and 4.4-N thrusters provide attitude control within a specified operational dead band.

MESSENGER's DSMs, positioned near the first aphelion after each Mercury flyby, help move the next Mercury encounter closer to Mercury's orbit perihelion. From Figure 2 notice that the much larger ΔV for DSM-2 provides far greater movement (from Mercury flyby 2 to Mercury orbit insertion) of the next Mercury encounter than DSM-1 does between Mercury flybys 1 and 2. If a spacecraft anomaly or ground station outage caused a delay for either DSM, the DSM could be delayed either a few days (until reaching the sunshade orientation constraint on the Sun-spacecraft-ΔV angle) or a few months to the next heliocentric orbit. This fast recovery costs < 1 m/s ΔV through MOI for DSM-1 delays, and < 12 m/s ΔV through MOI for DSM-2 delays. For the longer delay, a near-minimum ΔV is possible with the spacecraft's –y-axis pointed at the Sun (zero off-Sun pointing for the sunshade at Sun-spacecraft-ΔV angle = 90º). This slow recovery costs < 20 m/s ΔV through MOI for DSM-1 delays and < 86 m/s ΔV through MOI for DSM-2 delays. In each case the ΔV cost is less than the contingency ΔV allocation in Table 3. Table 4 shows how DSM-1 and DSM-2 meet LVA ΔV constraints for nominal and contingency cases at the beginning and end of the launch window.

Table 4

DEEP SPACE MANEUVERS WITHIN CONSTRAINTS

Maneuver Name	Maneuver Status	Maneuver Date	Earth Range (AU)	Sun Range (AU)	Sun-S/C-ΔV (deg)	Sun-Earth-S/C (deg)
Requirement →					(78° to 102°)	(> 3°)
2004 March 10 Launch						
DSM-1	**nominal**	**2007 Sep 19**	**1.600**	**0.672**	**96.9**	**14.6**
DSM-1	short delay	2007 Sep 22	1.601	0.679	78.1	14.7
DSM-1	long delay	2007 Dec 31	0.818	0.472	90.1	28.5
DSM-2	**nominal**	**2008 Jun 12**	**0.537**	**0.625**	**94.1**	**31.8**
DSM-2	short delay	2008 Jun 15	0.517	0.628	78.0	30.6
DSM-2	long delay	2008 Sep 22	1.525	0.575	90.0	11.2
2004 March 29 Launch						
DSM-1	**nominal**	**2007 Sep 19**	**1.591**	**0.672**	**92.6**	**14.9**
DSM-2	**nominal**	**2008 Jun 12**	**0.529**	**0.624**	**91.0**	**30.9**

PLANETARY FLYBYS

In order to minimize the fuel required from launch to MOI, the MESSENGER spacecraft will utilize two Venus gravity assists and two Mercury gravity assists. The Venus flybys will serve mission operators as test beds for command and observation sequences needed for the Mercury flybys. Due to the higher encounter altitudes and higher approach/departure phase angles at the Venus flybys, optical navigation (OpNav) images are not required to ensure accurate flyby targeting. OpNav image-taking sequences will be practiced prior to the low-altitude Mercury flybys, where OpNavs are mission-critical for accurate and risk-minimal flyby targeting.

Venus

Both unpowered (zero-ΔV) Venus flybys alter the size and shape of the spacecraft trajectory closer to Mercury's orbit. The first Venus flyby, with a moderate-range minimum altitude just over 0.4 Venus radii, not only reduces the spacecraft orbit's perihelion and aphelion, but also increases orbit inclination. Pre-encounter phase (Sun-spacecraft-Venus) angles of 163˚ to 166˚ 24 to 3 hours before the first Venus flyby indicate that the spacecraft will approach a thin, sunlit, crescent Venus. Small Venus and spacecraft ephemeris uncertainties and the medium-altitude close approach should assure sufficient targeting accuracy for the first Venus flyby. About 2.8 Venus orbits later and 3.8 spacecraft orbits later the spacecraft returns for a 0.7-Venus-radius altitude second Venus flyby, a flyby that lowers spacecraft aphelion near Venus' orbit and spacecraft perihelion close to Mercury's perihelion. An approach phase angle of 33˚ to 41˚ 24 to 3 hours before the second Venus flyby indicates a spacecraft view of a brightly sunlit Venus. After Venus flyby 2, the spacecraft begins a type VII, zero-ΔV, Venus-to-Mercury transfer. The longest solar eclipse for either Venus flyby is 20 minutes.

Mercury

Two unpowered 200-km minimum-altitude Mercury flybys, followed by deep space maneuvers just after the next aphelion, rotate the spacecraft orbit line of apsides and lower aphelion enough to enable Mercury orbit insertion in early April 2009. The Mercury gravity assists and subsequent DSMs lead to spacecraft:Mercury orbital resonances of about 2:3 and 3.1:4.1, respectively. The spacecraft's approach phase angles (24 hours before the flyby) are 115˚ and 131˚ for the first and second Mercury flybys. Figures 4 and 5 show how the spacecraft can view opposite sides of the never-before-imaged (bright surface) hemisphere of Mercury soon after close approach. The post-flyby orbit reconstruction for both Mercury flybys must account for the spacecraft's loss of communications (Earth occultation) near close approach and for the solar conjunction from two to seven days after the second flyby. The hyperbolic excess spacecraft velocity relative to Mercury decreases from 5.75 km/sec to 5.43 km/sec to 3.42 km/sec at Mercury flybys 1, 2, and MOI, respectively, for the 10 March 2004 launch. The actual spacecraft velocity relative to Mercury at close approach is 7.06 km/sec and 6.80 km/sec for Mercury flybys 1 and 2, respectively.

Tables 5 and 6 provide summaries of pre- and post-encounter visual details of Mercury as seen from the MESSENGER spacecraft for the 10 March 2004 launch date. An apparent optical illusion will occur about 5.7 hours before and after the first Mercury flyby, when the spacecraft's speed relative to Mercury's center equals the speed of the sub-spacecraft Mercury surface point relative to Mercury's center. In effect, the rotation of Mercury as seen from the spacecraft will appear to reverse direction. Phase angle is defined as the Sun-Mercury-spacecraft angle. The "Mercury surface % sunlit" is calculated from the square of the cosine of half the phase angle.

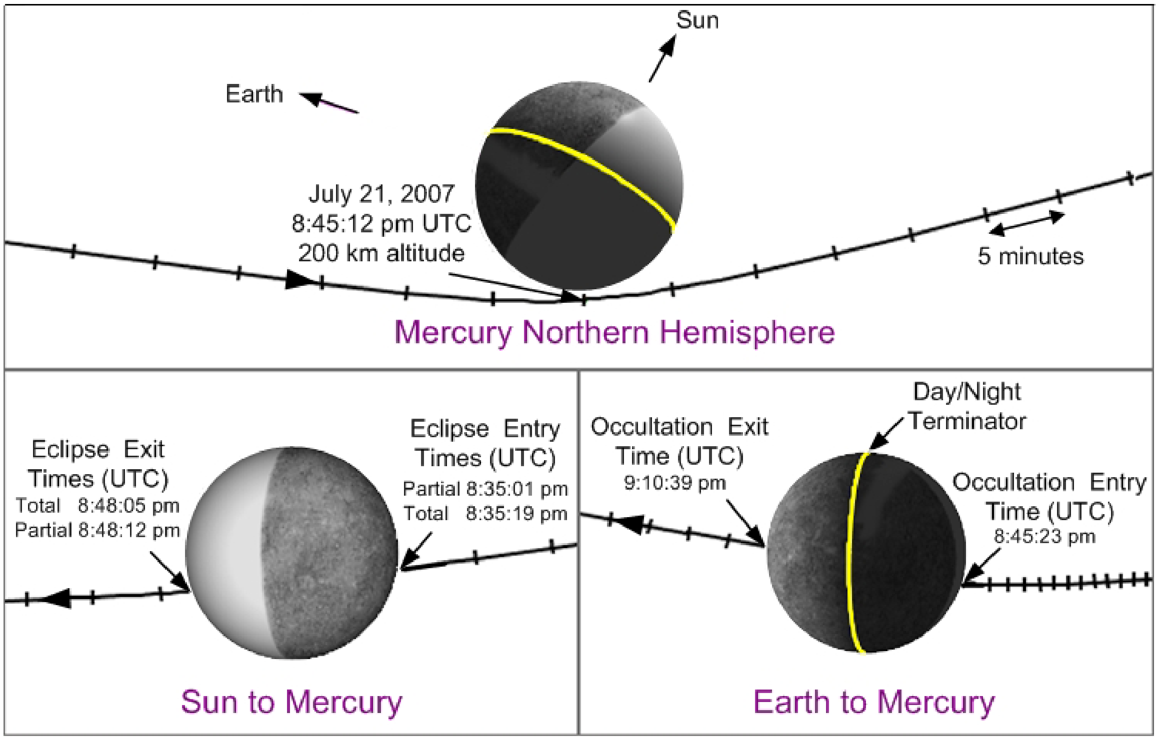

Figure 4 Mercury Flyby 1 Trajectory for 10 March 2004 Launch

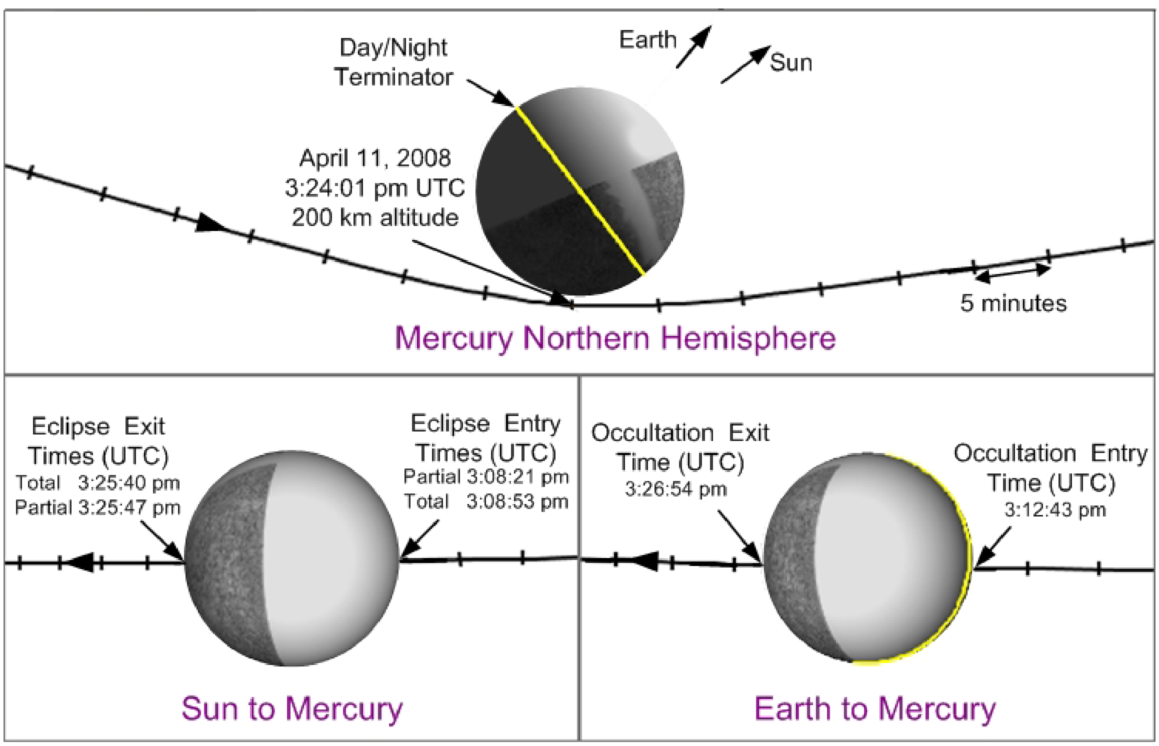

Figure 5 Mercury Flyby 2 Trajectory for 10 March 2004 Launch

Table 5

ENCOUNTER PARAMETERS FOR MERCURY FLYBY 1

Time From Closest Approach	Mercury As Seen From Spacecraft					
	Altitude (km)	Phase Angle (deg)	Mercury Surface % Sunlit	Angular Diameter (deg)	Sub-S/C Latitude (deg)	Sub-S/C Longitude (deg, West)
-3.0 days	1,486,767	123.7	22.2	0.188	7.96 N	44.73
-2.5 days	1,239,923	121.4	23.9	0.225	7.97 N	47.97
-2.0 days	992,778	119.1	25.7	0.281	7.97 N	51.16
-1.5 days	745,232	116.9	27.4	0.374	7.96 N	54.30
-1.0 days	497,189	114.7	29.1	0.560	7.93 N	57.33
-0.5 days	248,610	112.7	30.7	1.114	7.84 N	60.09
-3 hours	61,517	113.1	30.4	4.372	7.25 N	60.32
-1 hour	19,577	118.0	26.5	12.724	5.72 N	55.58
-15 minutes	3,822	138.1	12.8	45.865	0.54 S	35.64
0	200	146.2	8.4	135.110	16.88 S	324.11
+15 minutes	3,822	76.3	61.8	45.866	9.65 S	249.94
+1 hour	19,576	56.0	78.0	12.724	3.79 S	229.76
+3 hours	61,517	51.1	81.4	4.372	2.17 S	225.06
+0.5 days	248,663	50.8	81.6	1.113	1.53 S	225.29
+1.0 days	497,586	52.8	80.2	0.559	1.42 S	227.94
+1.5 days	746,397	55.0	78.7	0.373	1.38 S	230.77
+2.0 days	995,091	57.3	77.0	0.280	1.35 S	233.62
+2.5 days	1,243,487	59.6	75.3	0.224	1.34 S	236.44
+3.0 days	1,491,276	61.8	73.6	0.187	1.33 S	239.23

Table 6

ENCOUNTER PARAMETERS FOR MERCURY FLYBY 2

Time From Closest Approach	Mercury As Seen From Spacecraft					
	Altitude (km)	Phase Angle (deg)	Mercury Surface % Sunlit	Angular Diameter (deg)	Sub-S/C Latitude (deg)	Sub-S/C Longitude (deg, West)
-3.0 days	1,417,794	141.1	11.1	0.197	1.56 N	209.25
-2.5 days	1,179,702	138.7	12.4	0.236	1.57 N	212.52
-2.0 days	942,834	136.3	13.9	0.296	1.57 N	215.75
-1.5 days	706,764	133.9	15.3	0.394	1.57 N	218.91
-1.0 days	471,120	131.6	16.8	0.590	1.56 N	221.96
-0.5 days	235,527	129.6	18.1	1.175	1.54 N	224.71
-3 hours	58,332	130.1	17.8	4.606	1.38 N	224.73
-1 hour	18,560	135.4	14.4	13.343	0.99 N	219.47
-15 minutes	3,598	157.3	3.9	47.664	0.59 S	197.63
0	200	130.9	17.3	135.110	4.05 S	125.96
+15 minutes	3,598	59.2	75.6	47.664	1.95 S	54.20
+1 hour	18,560	37.3	89.8	13.343	0.45 S	32.35
+3 hours	58,332	32.0	92.4	4.602	0.03 S	27.09
+0.5 days	235,596	31.5	92.6	1.175	0.13 N	27.07
+1.0 days	471,655	33.5	91.7	0.590	0.16 N	29.68
+1.5 days	708,429	35.7	90.6	0.393	0.17 N	32.50
+2.0 days	946,387	38.1	89.4	0.295	0.17 N	35.33
+2.5 days	1,185,826	40.4	88.1	0.235	0.17 N	38.13
+3.0 days	1,426,922	42.7	86.7	0.196	0.17 N	40.88

MERCURY ORBIT INSERTION

Orbital parameter requirements for the MESSENGER spacecraft's initial primary science orbit include an 80º (±2º) orbit inclination, 200-km (±25 km) periherm altitude, 12-hour (±1 minute) orbit period, 118.4º argument of periherm (60ºN periherm latitude with 56ºN to 62ºN acceptable), and a 308º (248º to 73º) longitude of ascending node. These requirements, expressed in the Mercury-centered equator and equinox of the January 1.5, 2000, frame, are derived from a combination of science and engineering requirements along with characteristics of the Mercury arrival geometry. For the 10 March 2004 launch date, the optimal heliocentric trajectory yields an initial periherm latitude of 56ºN. The MOI strategy for MESSENGER sets MOI-1 start time and thrust direction to obtain the required 4ºN latitude rotation of the line of apsides.

The Mercury orbit-insertion strategy focuses on maximizing the likelihood of successfully placing the spacecraft into the primary science orbit in the minimum time possible within mission planning process constraints. This strategy uses two powered-turn maneuvers (MOI-1 and MOI-2) using the LVA thruster operating at 672-N thrust and 316.1-s specific impulse, such that the spacecraft completes four full orbits between MOI-1 and MOI-2. Since each maneuver slows the spacecraft's Mercury-relative velocity, the direction of each maneuver's thrust vector is nearly opposite of the spacecraft velocity vector. In order to improve Mercury approach orbit determination accuracy, MOI-1 start time is shifted slightly (hours) from the minimum-ΔV Mercury arrival time. This time shift is determined by the placement of a bright star within the MDIS imager's field-of-view that includes at least a portion of Mercury. Figure 6 shows the location of MOI-1 and MOI-2 and depicts the shape, size, and orientation of the first five orbits.

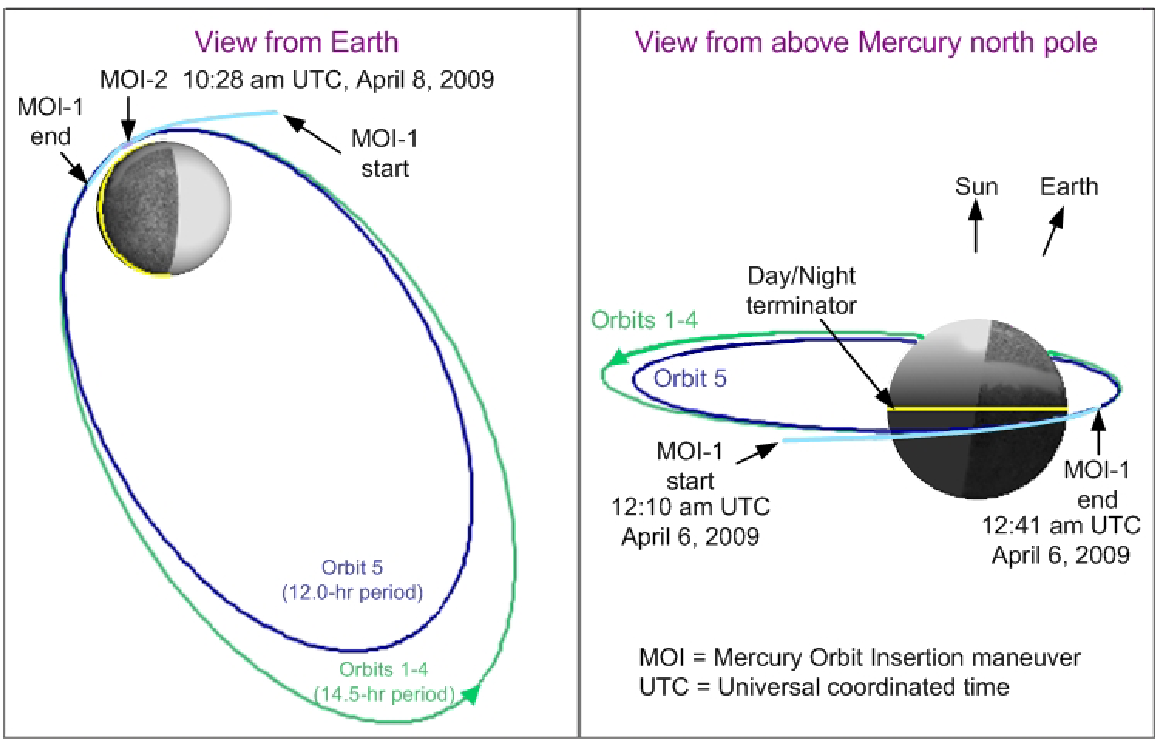

Figure 6 Mercury Orbit Insertion Trajectory for 10 March 2004 Launch

The initial orbit period of the spacecraft with respect to Mercury between MOI-1 and MOI-2 is between 12.8 and 16.0 hours, a range that demonstrates the stability needed for reliable post-MOI-1 timing updates for MOI-2. Orbits with higher orbit periods produce unacceptable drift rates in both orbit period and periherm altitude. The choice of four phasing orbits between MOI-1 and MOI-2 gives at least 51 hours for preliminary MOI-1 maneuver performance assessment, orbit determination, MOI-2 maneuver design, simulation, and maneuver tweak upload and verification. The 12.8-hour lower limit also ensures that too long of a burn duration for MOI-1 will not produce an initial orbit period less than the 12.0-hour target for the primary science orbit. Although Mercury arrival dates vary by two days across the 20-day March 2004 launch window, the initial phasing orbit period range also ensures that daily science and spacecraft health data downlink periods (beginning four hours before apoherm on every other 12-hour orbit) may be conveniently placed near 8:30 am to 4:30 pm EST. A Sun-Earth-spacecraft angle > 6º ensures that no solar interference will corrupt communications with the spacecraft at this time.

Because MOI occurs as Mercury nears its perihelion, Mercury's accelerating heliocentric angular motion will rotate the Sun-relative spacecraft orbit orientation to the point of preventing sunshade protection of part of the spacecraft. This can occur 1.5–3.5 days (shorter time applies for late-March 2004 launches) after MOI-1, thereby adding risk to the choice of longer phasing orbits. An LVA cutoff 2.6 minutes (157 s) early will place the spacecraft into an initial 3.5-day period phasing orbit. MOI-1 and MOI-2 ΔVs are 1417.7 m/s and 203.9 m/s, respectively. The maximum Sun-S/C-ΔV angle of 103.1º (at the end of MOI-2) brings the edge of the launch vehicle adapter within 0.4º of direct illumination by sunlight. With no apoherm ΔV to lower periherm altitude, the initial primary science orbit will have an 81.9º-orbit inclination and 221-km periherm altitude.

The quantitative Mercury orbit insertion summary in Table 7 shows compliance of MOI-1 and MOI-2 with angular constraints. Analysis using a 15-s increment for each ΔV further verified constraint compliance. Reduction in MOI ΔV magnitude across the 20-day launch window is due primarily to a lower spacecraft-Mercury relative approach velocity. After MOI-1, the initial orbit periods are 14.5 hours for both the 10 March and 29 March launch dates. The minimum link margin during MOI is 4.6 dB at the end of MOI-1 for the 29 March launch.

Table 7

MERCURY ORBIT INSERTION MEETS OPERATIONAL REQUIREMENTS

Orbit Insertion Maneuver Segment	Maneuver Date Year Month Day	Maneuver Time (UTC) hh:mm:ss	ΔV (m/s)	Earth-S/C Range (AU)	Sun-S/C Range (AU)	Sun-S/C-ΔV Angle (deg)	Sun-Earth -S/C Angle (deg)
Requirement →						**(78° to 102°)**	**(> 3°)**
Baseline 3/10/04 Launch (Day 1 of 20)							
MOI-1 settle/refill	2009 Apr 06	00:10:14	2.3	1.292	0.317	95.6	6.3
MOI-1 LVA start	2009 Apr 06	00:10:59	1587.3	1.292	0.317	95.6	6.3
MOI-1 trim end	2009 Apr 06	00:41:31	2.5	1.292	0.317	98.5	6.3
MOI-2 settle/refill	2009 Apr 08	10:28:10	6.5	1.258	0.311	84.9	8.9
MOI-2 LVA start	2009 Apr 08	10:29:27	24.6	1.258	0.311	85.0	8.9
MOI-2 trim end	2009 Apr 08	10:30:03	2.5	1.258	0.311	86.2	8.9
	orbit insertion total ΔV =		**1625.7**				
Baseline 3/29/04 Launch (Day 20 of 20)							
MOI-1 settle/refill	2009 Apr 07	23:29:25	2.3	1.265	0.312	83.8	8.4
MOI-1 LVA start	2009 Apr 07	23:30:10	1590.3	1.265	0.312	83.9	8.4
MOI-1 trim end	2009 Apr 08	00:00:52	2.5	1.265	0.312	95.4	8.5
MOI-2 settle/refill	2009 Apr 10	10:30:00	6.5	1.224	0.308	85.2	11.0
MOI-2 LVA start	2009 Apr 10	10:31:17	28.1	1.224	0.308	86.0	11.0
MOI-2 trim end	2009 Apr 10	10:31:56	2.5	1.224	0.308	88.3	11.0
	orbit insertion total ΔV =		**1632.2**				

Unlike high-thrust trajectory orbit-insertion scenarios for most planets, MESSENGER's robust orbit insertion strategy offers credible contingency scenarios that enable recovery to the required primary science orbit. For instance, if a decision were made months or years before the nominal MOI date to delay

Mercury orbit insertion, a small ΔV, often < 10 m/s, could transform the old Mercury approach trajectory into a third gravity-assist flyby that returns the spacecraft to Mercury 1.5-years later at a significantly lower velocity relative to Mercury. Near the first aphelion after the third Mercury flyby, a 183 m/s ΔV would set up a contingency MOI date of 22 September 2010. This contingency strategy uses the third Mercury flyby to adjust the spacecraft:Mercury resonant period from 3:4 to 5:6 and would save as much as 684 m/s ΔV (if all the ΔV gained is used with the LVA thruster) compared with the nominal ΔV budget.

MERCURY ORBITAL OPERATIONS

After completing Mercury orbit-insertion, the spacecraft begins a more than 12-week long coast phase without orbit correction maneuvers (OCMs) that will provide sufficient time to refine Mercury's gravity model and update perturbing force models. Occasional thruster firings for adjusting spacecraft angular momentum will perturb the trajectory by mm/s of unintentional ΔV.

Each pair of OCMs will return the spacecraft to the original size and shape of the primary science orbit. Solar radiation pressure, solar gravity, and subtle spatial variations in Mercury's gravity will alter orbit orientation by moving periherm north, increasing orbit inclination, and rotating the low-altitude descending node in the anti-Sun direction (for Mercury at perihelion). The first OCM of the pair will impart a ΔV opposite to the spacecraft velocity direction at apoherm in order to lower periherm altitude to 200 km. Since this ΔV decreases the spacecraft orbit period by nearly 15 minutes, the next OCM will occur at periherm close to the velocity direction to return orbit period to 12 hours. To give enough time to assess the first OCM's performance, update the orbit and next maneuver design, and upload the ΔV update, the second OCM of the pair must occur 2.5 orbits after the first OCM. To keep periherm altitude under 500 km while meeting sunshade orientation constraints, OCM pairs must occur about one 88-day Mercury year apart. This interval marks the time when the spacecraft orbit plane is nearly perpendicular to the spacecraft-Sun line.

Table 8

MERCURY ORBIT CORRECTION MANEUVERS SATISFY REQUIREMENTS

Orbit Correction Maneuver Segment	Maneuver Date Year Month Day	Maneuver Time (UTC) (hh:mm:ss)	ΔV (m/s)	Earth-S/C Range (AU)	Sun-S/C Range (AU)	Sun-S/C-ΔV Angle (deg)	Sun-Earth -S/C Angle (deg)
Requirement →						**(78° to 102°)**	**(> 3°)**
Baseline 3/10/04	Launch						
OCM-1 start	2009 Jul 03	15:51:23	26.3	1.225	0.315	85.7	12.1
OCM-1 end	2009 Jul 03	15:53:08	-	1.225	0.315	85.6	12.1
OCM-2 start	2009 Jul 04	21:12:14	4.1	1.245	0.312	92.7	10.8
OCM-2 end	2009 Jul 04	21:13:02	-	1.245	0.312	91.9	10.8
OCM-3 start	2009 Sep 29	14:35:51	25.2	0.788	0.315	87.0	15.0
OCM-3 end	2009 Sep 29	14:37:37	-	0.788	0.315	86.9	15.0
OCM-4 start	2009 Sep 30	19:58:29	3.9	0.819	0.312	94.1	16.0
OCM-4 end	2009 Sep 30	19:59:14	-	0.819	0.312	93.4	16.0
OCM-5 start	2009 Dec 26	13:21:21	23.7	0.808	0.315	88.4	16.9
OCM-5 end	2009 Dec 26	13:23:05	-	0.808	0.315	88.2	16.9
OCM-6 start	2009 Dec 27	18:46:11	3.7	0.780	0.312	95.6	15.5
OCM-6 end	2009 Dec 27	18:46:53	-	0.780	0.312	95.2	15.5
DeterministicΔV	**for orbit phase**	**=**	**86.9**	**m/s**			

During the Mercury orbital phase 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, and Mercury surface albedo must be carefully coordinated with the spacecraft's complex attitude profile. The effects of solar pressure on the MESSENGER spacecraft trajectory have been calculated.[20] These effects include periherm altitude drift to 450–500 km prior to OCM-1, -3, and, -5; northward periherm latitude drift (decreasing the argument of periherm) of at least 9º; orbit inclination increasing by ~ 2º; and longitude of ascending node decreasing 5º. Maximum duration between OCMs follows from a requirement for long periods in a stable orbit to determine the planetary gravity field and to carry out orbit determination to the accuracy needed to recover the physical libration amplitude from altimetry.

Science observation sequences provide the dominant influence on spacecraft attitude during Mercury orbit phase. The instrument deck points toward Mercury during most science observation sequences (up to 100% of the orbit when the orbit plane "faces" the Sun as shown on the left side of Figure 7). The right side of Figure 7 shows a noon-midnight orbit, where, in order to maintain the sunshade toward the Sun, the spacecraft must rotate 180º twice per orbit. In this orientation, the orbit has zones where Mercury surface science observations are not possible. One key science goal, global stereo image coverage of Mercury's surface with MDIS, requires two 176-day Mercury solar days. After the three-day orbit insertion process and 352-day allocation for global stereo imaging, ten days of orbital operations may be allocated exclusively for targeted science observations or to fill gaps in data acquired. These are the three primary segments of the 365-day nominal orbit phase.

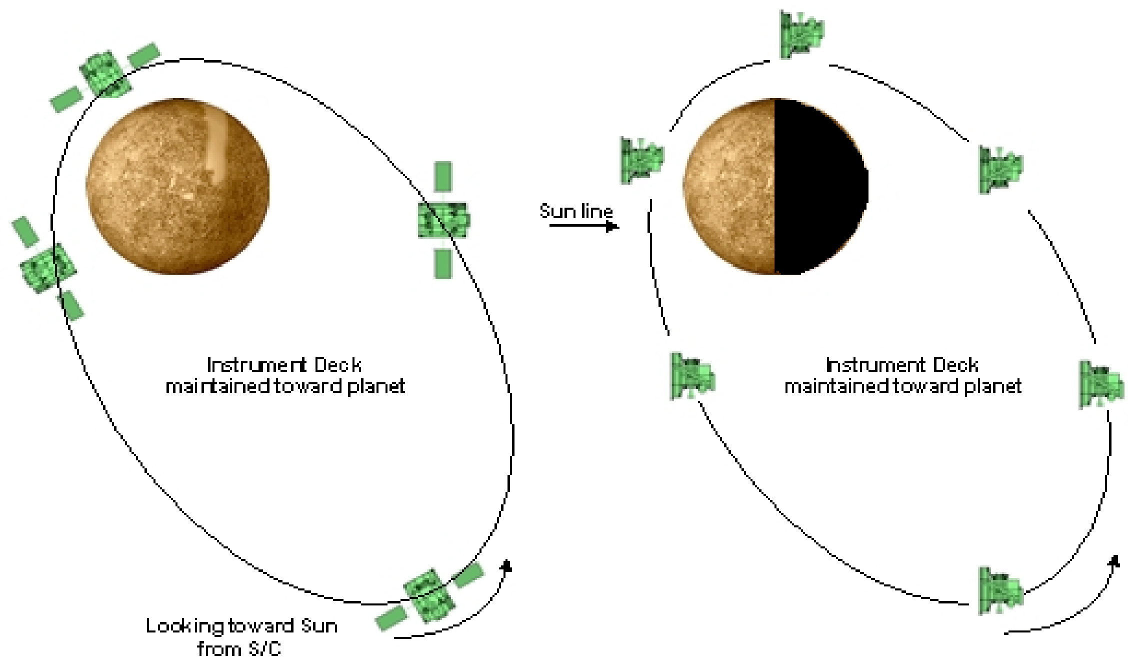

Figure 7 Nominal Spacecraft Orbit Phase Attitude (Excluding Data Downlink)

Data downlink will occur during about 6.5 hours of scheduled DSN ground-station tracks that begin four hours before apoherm and end four hours after apoherm. In this attitude the spacecraft rolls, keeping the sunshade oriented toward the Sun, to maximize data rate using a phased-array antenna. If Earth is on the Sun side of the spacecraft, the front phased array is active. If Earth is on the anti-Sun side of the spacecraft, the back phased array is active. Figure 8 depicts the timing of the six shifts between front- and back-side antenna clusters during the Mercury orbit phase. Also shown in Figure 8 are the downlink data rates using 34-m DSN antennas. If 70-m antennas are utilized during the highest data rate (lowest Earth-spacecraft range) periods, downlink rates could exceed 100 kbits per second.

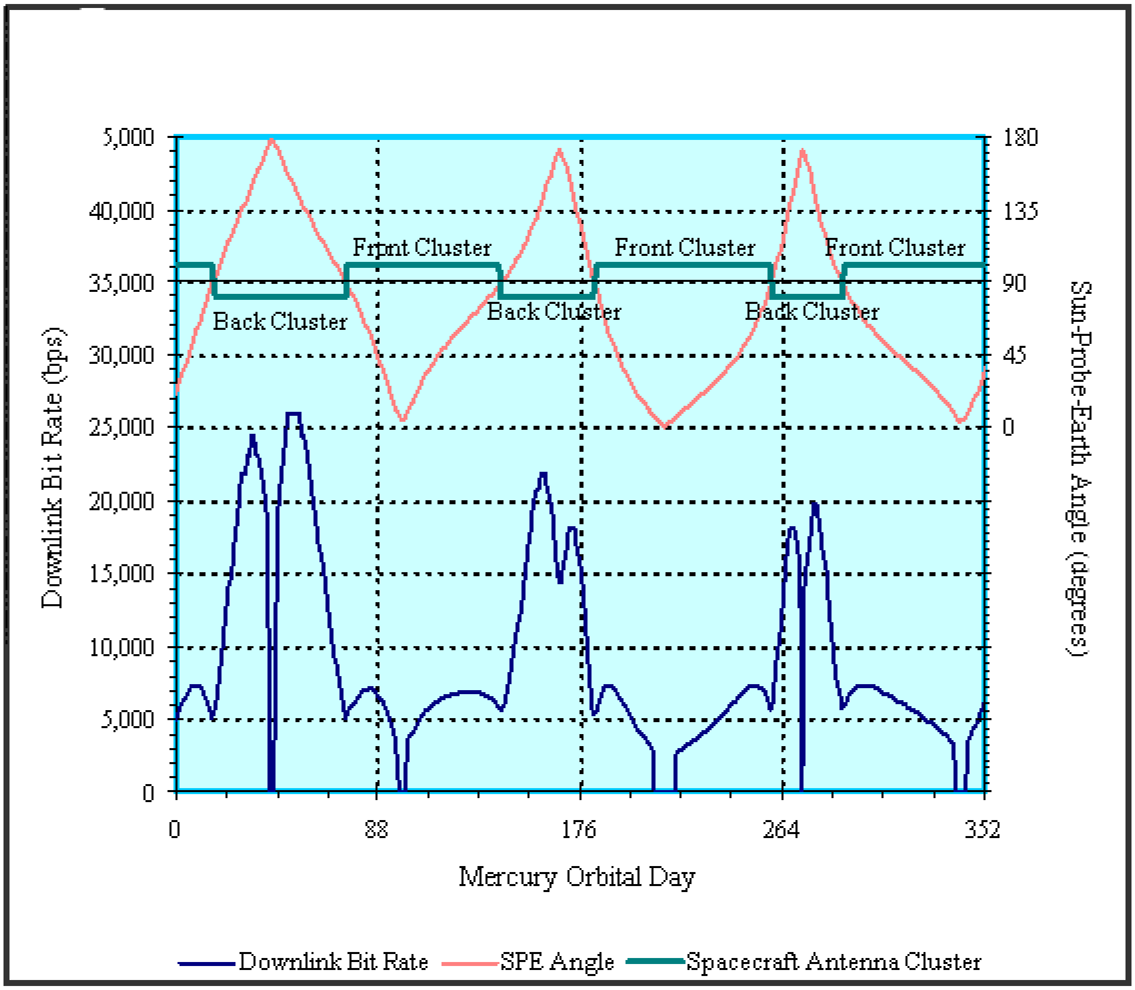

Figure 8 Mercury Orbit Phase Data Transmission Parameters

CONCLUSIONS

With a successful launch aboard a Delta II 7925H launch vehicle in March 2004, the MESSENGER mission will become the first spacecraft to orbit Mercury in April 2009. A capable suite of seven scientific instruments will achieve the carefully formulated science objectives during two Mercury flybys and a one-year Mercury orbit phase. A simple, highly redundant spacecraft design with a dual-mode propulsion system will protect the spacecraft bus and instruments from the extreme thermal and radiation environment of space near the planet Mercury. A robust trajectory and maneuver design strategy accounts for recovery from a wide range of anomalies. Deterministic maneuvers, which include two deep space maneuvers, a two-part Mercury orbit insertion, and six orbit correction maneuvers, all comply with a restrictive set of operational constraints.

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APPENDIX

Backup Launch Opportunity

In the event that MESSENGER is not launched in March 2004, a 10-day backup launch opportunity will be ready in May 2004. There are very few notable differences between the two trajectories and deterministic maneuvers. These differences include one additional Venus flyby, one additional month for the heliocentric transfer, and a 12% higher maximum launch energy (C_3) requirement. A slight reduction in the onboard propellant requirement helps offset the C_3 increase. Figure A1 shows the heliocentric trajectory for the first day of the backup launch window. Figures A2 and A3 depict the Mercury flybys, each occurring one Mercury orbit later than for the baseline March 2004 launch. The final 25% of the main Mercury orbit insertion maneuver (Figure A4) is obscured from Earth view by Mercury. All other deterministic ΔVs are fully observable.

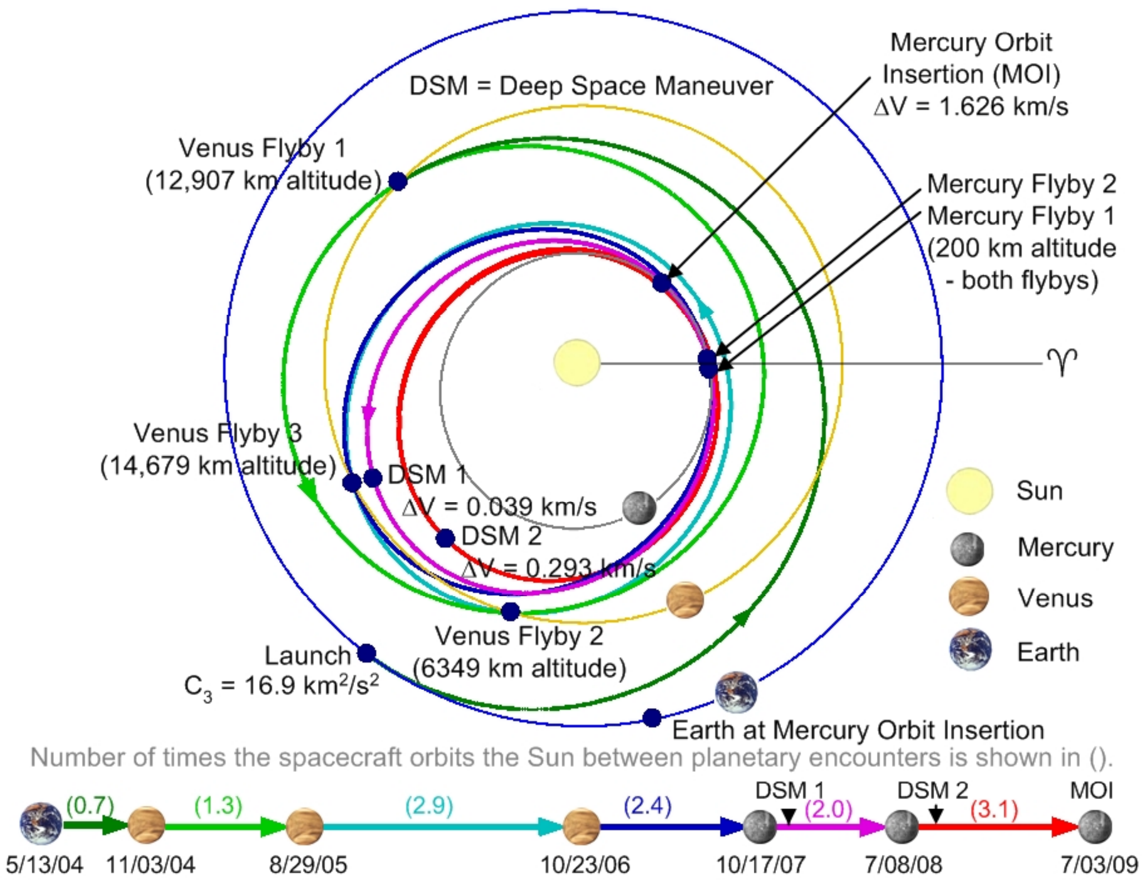

Figure A1 North Ecliptic Pole View of Backup Heliocentric Trajectory

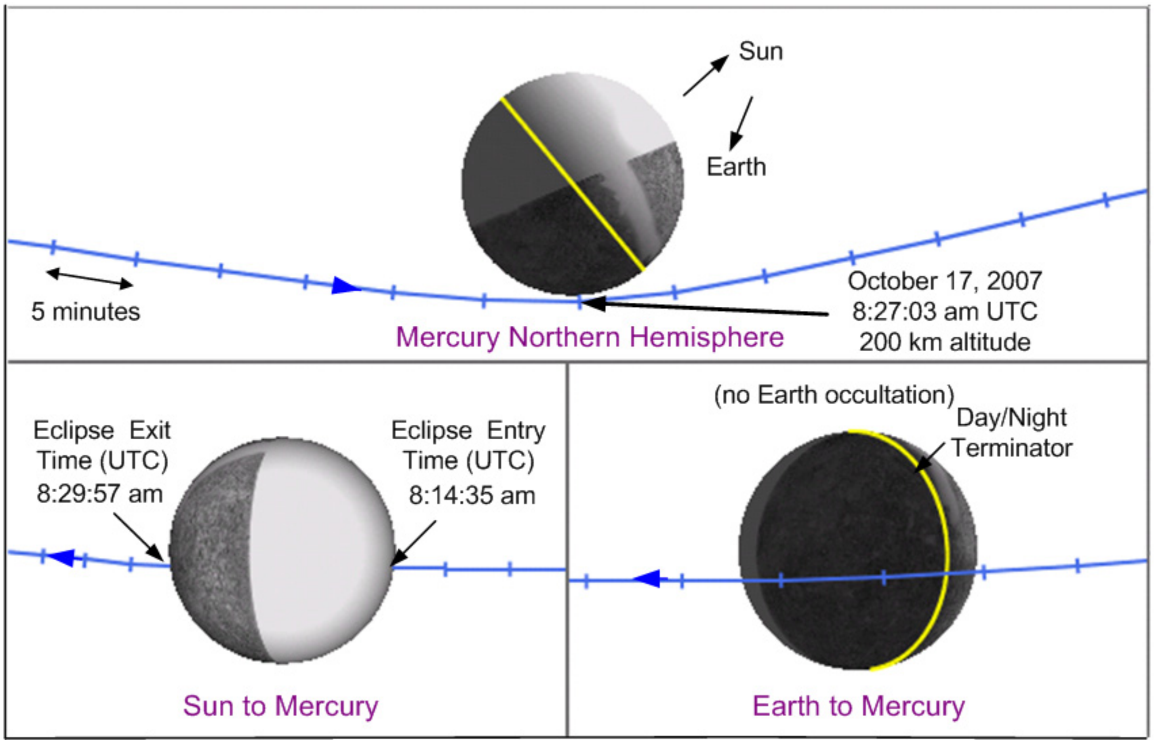

Figure A2 Mercury Flyby 1 Trajectory for 13 May 2004 Launch

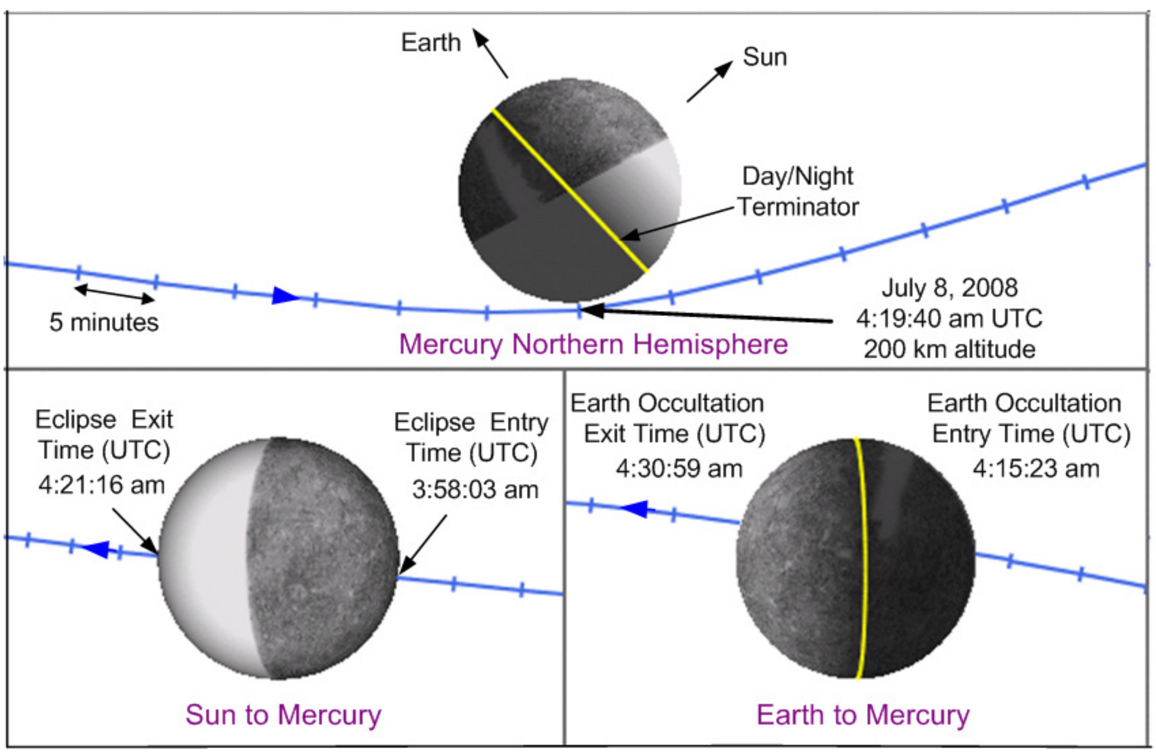

Figure A3 Mercury Flyby 2 Trajectory for 13 May 2004 Launch

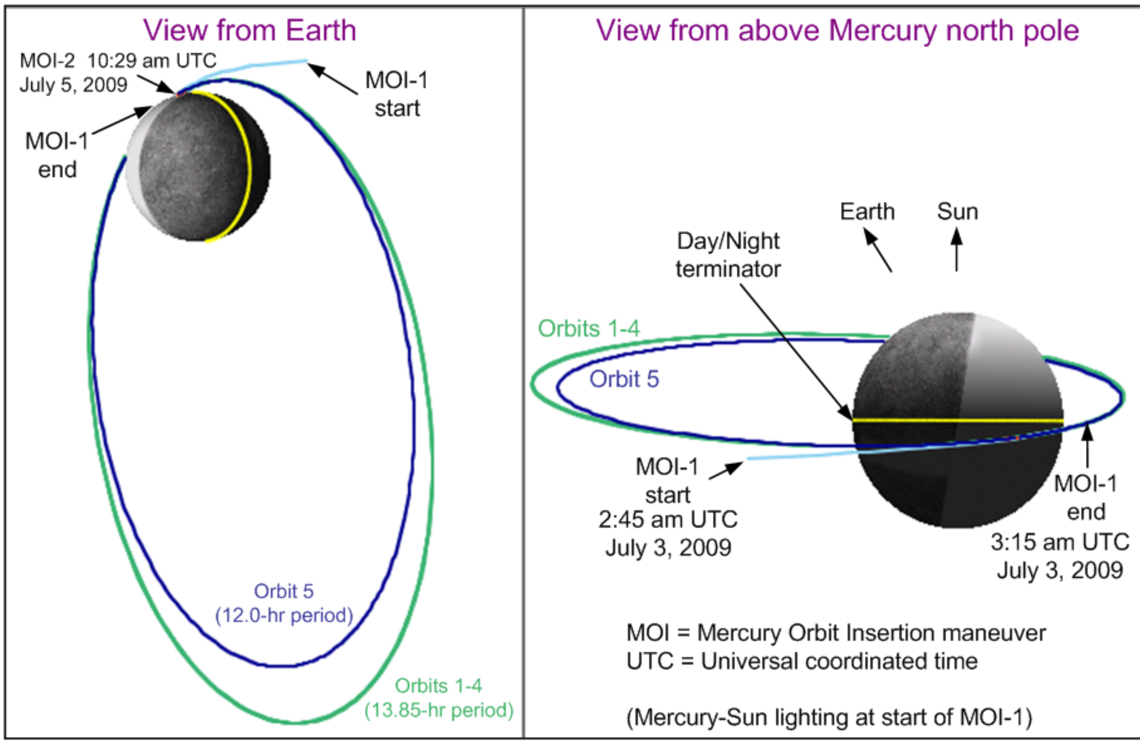

Figure A4 Mercury Orbit Insertion Trajectory for 13 May 2004 Launch