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A RESILIENT MISSION DESIGN FOR THE MESSENGER 2004 MERCURY ORBITER

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ABSTRACT

A resilient mission design minimizes implementation risk by maximizing the number and duration of opportunities for recovery from launch and maneuver errors or delays. MESSENGER is a small, 3-axis stabilized spacecraft designed to obtain comprehensive scientific observations of Mercury during two flybys and one year in orbit. NASA's eighth Discovery mission, MESSENGER demonstrates a resilient mission design by offering launch opportunities in 2004 and 2005, a ballistic trajectory with several deep space maneuvers that target propulsion-free planetary gravity assists, and multiple recovery options from maneuver delays. Defining contingency recovery options prior to detailed spacecraft design may affect selection of design parameters such as thermal shade size or propellant margin (and therein, propellant tank size). Having optimal recovery options in place prior to implementation of mission-critical events maximizes accomplishment of mission objectives. Similar prudent planning recently saved NASA's first Discovery Program mission, the Near Earth Asteroid Rendezvous (NEAR). When the NEAR mission's largest maneuver aborted on December 20, 1998, the existing contingency recovery plan was refined and implemented to minimize propellant usage and preserve 100% of the mission goals.

INTRODUCTION

The MESSENGER mission design demonstrates prudent planning by offering four launch periods, two Mercury arrival dates, and plans for recovery from delays for all major propulsive maneuvers. MESSENGER's 5.5-year heliocentric trajectory uses gravity assists from Earth (one), Venus (two), and Mercury (two), as well as five planned maneuvers en route to one year of orbiting Mercury. Launch periods of 14-15 days are identified in March-April 2004, August 2004, and August 2005. Launch energy never exceeds 16.0 km^2/s^2. Initial heliocentric trajectories for each launch period began with 1.35-year and 1.00-year Earth-Earth transfers for the 2004 launch periods and a direct Earth-Venus transfer for launch in 2005.

Single maneuver recovery options for delays of four deep space maneuvers and Mercury orbit insertion (MOI) provided full constraint compliance for delays up to 7, 10.5, 3, 3, and 3.5 days, respectively. Corresponding ΔV increases of up to 12, 14, 1, 4, and 61 m/s come from re-optimizing ΔV through MOI. The third and fourth maneuvers offer second recovery opportunities about 100 days later, after one heliocentric orbit, for comparable ΔV increases. Similarly, MOI has second and third recovery opportunities at subsequent perihelion passages 106 and 214 days after nominal MOI. In addition, insertion directly into the "prime science" 12-hour period Mercury orbit allows recovery from significant orbit capture maneuver shortfalls. These results helped establish an adequate margin for ΔV not dedicated to a 99%-confidence correction of planned maneuvers and planetary flybys.

Delays in execution of Mercury orbit correction maneuvers are either short-term or long-term, and are limited by thrust-vector orientation constraints. Short-term delays of 12, 24, and 36 hours cost less than 0.5 m/sec. Long-term delays occur in multiples of 44 days, with uncorrected periapsis altitude upward drift giving rise to lower resolution science observations for some sub-spacecraft latitudes.

Current Discovery guidelines[1] (MESSENGER selection in parentheses) include launch vehicle no larger than a Delta 7925H (Delta 7925H), phase C/D development less than three years (34 months) through launch + 30 days, and total mission cost not to exceed $299 million ($286 million) in Fiscal Year 1999$.

LAUNCH

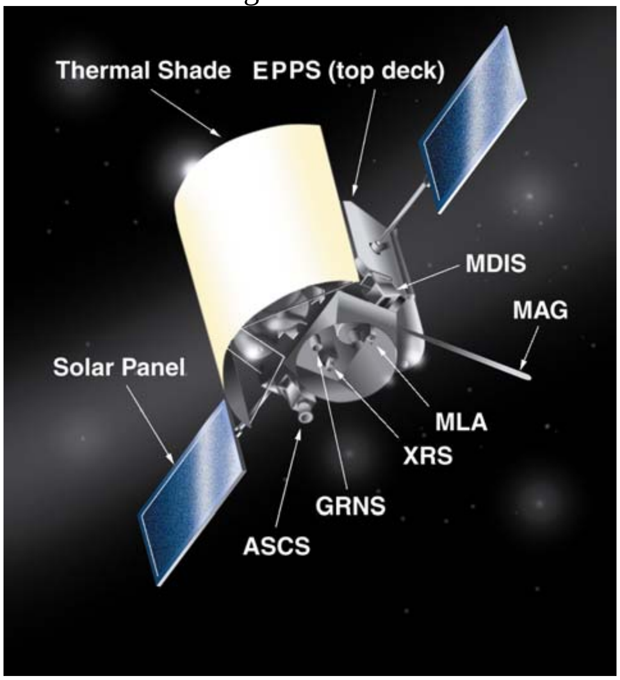
Payload Characteristics and Constraints

Since 1996 Mercury orbiter trade studies led by Carnegie Institute of Washington (CIW) and The Johns Hopkins University Applied Physics Laboratory (JHU/APL) have produced a spacecraft concept that blends low-risk, reliable operation, maximum science, and trajectory optimization. Power for the 3-axis stabilized spacecraft comes from batteries (planned for use during early launch phase and solar occultation passage) and high-temperature tolerant, dual-sided, rotatable solar arrays. Thermal control is achieved by passive means using an opaque ceramic-cloth thermal shade between the spacecraft and the Sun. Communication with Earth-tracking antennas is possible at all times except solar conjunction through low- and medium-gain antennas and two phased-array antennas. For current Delta 7925H performance the 1066 kg MESSENGER launch mass includes 636 kg propellant with a 2700 m/sec ΔV capability.

A 645 N bipropellant thruster for all maneuvers larger than 20 m/sec and coupled sets of lower thrust monopropellant thrusters comprise the spacecraft's dual mode propulsion system. The large thruster provides about 40% more thrust than NEAR's counterpart[2], thereby reducing gravity loss during the long MOI maneuver at Mercury arrival. All deterministic maneuvers performed closer to the Sun than Venus (~0.7 AU) are designed well within ± 12.7 º pitch and ± 15 º yaw constraints that keep sunlight away from the spacecraft bus. Some maneuver contingency scenarios allow sunlight on the primary thruster during burn attitude, but never allow sunlight closer than one degree from deck-mounted instruments. Except for part of the 24-minute MOI burn, all propulsive maneuvers are directly observable from Earth-based tracking stations. Applied to trajectory optimization, this translates into a minimum 2.0° Sun-Earth-probe (SEP) angle for nominal or delayed maneuvers to ensure reliable command transmission without solar interference. Flyby minimum altitudes are 300 km at Earth and Venus (compare to Galileo's 305-km altitude[3] Earth flyby on December 8, 1992 and Cassini's 284-km altitude[4] Venus flyby on June 24, 1999), and 200 km at Mercury.

The science payload, shown in Figure 1, is designed to answer many key questions about Mercury's past and present. These questions include: 1) What is the origin of Mercury's high density?, 2) What are the composition and structure of its crust?, 3) Has Mercury experienced volcanism?, 4) What are the nature and dynamics of its thin atmosphere and Earth-like magnetosphere?, 5) What is the nature of its mysterious polar caps?, and 6) Is a liquid outer core responsible for generating its magnetic field? One notable science goal is to obtain global stereo imagery of Mercury's surface at 250 meters/pixel resolution. All science data and spacecraft health data will be stored on two 8-Gbit solid state recorders.

Integration of MESSENGER's Mercury orbit (assuming an average Sun-facing spacecraft area) documented orbital parameter fluctuation and spacecraft attitude during propulsive maneuvers. The size, orientation, and observed variation in the spacecraft's orbit at Mercury rest within limitations set for the science payload, power (e.g., eclipse duration), and thermal control. Additional constraints have markedly less effect on trajectory optimization and maneuver design.



Mercury Dual Imaging System (MDIS)
Gamma-Ray and Neutron Spectrometer (GRNS)
Magnetometer (MAG)
Mercury Laser Altimeter (MLA)
Atmospheric and Surface Composition Spectrometer (ASCS)
Energetic Particle and Plasma Spectrometer (EPPS)
X-Ray Spectrometer (XRS)
Radio Science – uses telecommunication system

Figure 1 – MESSENGER Spacecraft with Science Instrument Locations

Launch Opportunity Prime and Alternates

Utilization of Earth-Earth transfer trajectory design techniques similar to those employed by Farquhar and Dunham for the CONTOUR (Comet Nucleus Tour) project[5] lead to two launch opportunities in six months, followed by a launch opportunity one year after the second opportunity. In 1998 McAdams[6] summarized the next ten years of ballistic trajectories to Mercury having post-launch (through MOI) deterministic ΔV below 2500 m/sec and utilizing Yen's method[7] of multiple Venus and Mercury gravity assists to reduce Mercury arrival velocity. Of all these opportunities (launches in 2002, 2004, 2005, and 2007), only the 2004 launch with 2005 Earth swingby offers both a backup launch window and fewer than three Mercury swingbys, thereby reducing flight time and enabling a three-Mercury swingby backup. Since the proximity and number of launch periods eliminates the need for longer (20-30 day) windows, a shorter (14-15 day) window is chosen to maximize spacecraft mass.

Launch Opportunity Prime

The implementation strategy for defining a 15-day launch window yields the maximum initial spacecraft mass for a constraint-adjusted, near-minimum total ΔV trajectory to Mercury orbit. Spacecraft dry mass, launch energy, and post-launch ΔV requirements dictate launch aboard a 3-stage Delta 7925H, the largest launch vehicle allowed within NASA's Discovery Program. The fixed launch energy (C_3 in Table 1), declination of launch asymptote (DLA in Table 2), and a 99.0% probability of commanded shutdown, define the maximum spacecraft mass delivered to the heliocentric transfer orbit. Figure 2 shows the trajectory profile for a March 23, 2004 launch, day one of the prime launch window.

Table 1 -
Mercury Orbiter Prime Launch Summary

Launch dates	Mar 23- Apr 6, 2004 (15 days)
Launch energy	C_3 = 16.0 km^2/sec^2
Launch vehicle	Delta-7925H-9.5
Initial launch mass	1066 kg

Launch window definition began with March 26, 2004 as the minimum total ΔV heliocentric trajectory from launch through MOI, given a 2.5° minimum SEP angle constraint for the second deterministic ΔV (ΔV_2). This SEP angle constraint ensures a minimum of five days to upload and verify a command to execute a delayed ΔV_2. By adding and subtracting days to the March 26 launch date, 13 "constrained-minimum total ΔV trajectories were generated with the first and last launch dates having launch energy close to 16 km^2/sec^2 (March 24 – April 5). By increasing post-launch ΔV on March 23 and April 6, launch energy was constrained to 16.0 km^2/s^2. The result is two days added to the launch window with no reduction of initial spacecraft launch mass, but at the loss of extra ΔV margin for the first and last days of the launch window.

mid August 2004, and early-mid August 2005. Each launch opportunity offers the convenience of equal launch energy, post-launch ΔV, and comparable planetary flyby geometries.

The first backup launch period, summarized in Table 3, opens four months after the prime launch period closes. Tables 2 and 3 show how small differences are between MESSENGER

Table 2 – MESSENGER Launch Window for 2004 Prime Opportunity

Launch Date	DLA (deg)	Flyby Min. Altitude (km) Earth	Venus 1	ΔV_1 (m/s)	ΔV_2 (m/s)	ΔV_3 (m/s)	ΔV_4 (m/s)	ΔV_{MOI} (m/s)	ΔV_{MARGIN} (m/s)
03/23/04	4.3	5536	3544	291.4	249.4	72.7	239.0	1555.1	100.0
03/24/04	3.9	5773	3553	277.8	251.5	72.8	238.6	1554.8	112.1
03/25/04	3.5	6314	3447	282.6	242.2	72.7	238.9	1555.0	116.1
03/26/04	3.1	7101	3293	294.5	228.7	72.6	239.7	1555.6	116.5
03/27/04	2.7	7968	3166	303.1	219.8	72.5	240.3	1556.0	115.9
03/28/04	2.4	8776	3092	305.0	218.4	72.5	240.4	1556.0	115.2
03/29/04	2.0	9536	3054	302.1	222.2	72.6	240.1	1555.9	114.8
03/30/04	1.6	10229	3046	295.5	230.3	72.6	239.6	1555.5	114.2
03/31/04	1.2	10875	3057	286.0	241.4	72.7	238.9	1555.0	113.5
04/01/04	0.8	11483	3083	274.7	254.9	73.0	238.2	1554.5	112.3
04/02/04	0.4	12064	3120	261.7	270.4	73.3	237.5	1553.9	110.8
04/03/04	0.0	12626	3163	247.5	287.4	73.9	236.8	1553.3	108.7
04/04/04	-0.4	13173	3212	232.1	305.9	74.6	236.2	1552.8	106.1
04/05/04	-0.8	13712	3264	215.8	325.7	75.4	235.7	1552.2	102.8
04/06/04	-1.2	14246	3318	198.5	346.7	76.4	235.3	1551.7	99.0

Constants: C_3 = 16 km^2/sec^2; flyby altitude for Venus 2 (300 km) & Mercury 1 & 2 (200 km)

Launch from the Eastern Test Range's Launch Complex 17 delivers MESSENGER into a 185-km altitude, 28.5º inclination parking orbit before injection into the heliocentric transfer orbit. Trajectory integration for a March 23, 2004 launch set lift-off at 10:44 am UTC (5:44 am EST), parking orbit insertion 9-10 minutes later, and transfer trajectory injection at 11:18 am UTC (see Figure 3). After parking orbit insertion the spacecraft receives full sunlight during Earth departure, including DSN acquisition of signal less than 50 minutes after launch.

Launch Opportunity Alternates

Additional launch opportunities are in early-

prime and backup launch periods. The main differences between prime and backup launches are DLA and Earth-to-Earth transfer. While the prime launch opportunity requires DLA near zero, the second half of the backup launch opportunity incurs small launch vehicle performance losses from DLA values between –28.5° and –33.5°. The Earth-to-Earth transfers shift from type III (transfer angle between 360° and 540°) in the ecliptic plane, to a 360° transfer inclined at 5.5° with respect to the ecliptic plane.

A second backup launch period occurs exactly one year after the August 2004 launch period.

While the number of days and DLA required are identical for both backup launch periods, the August 2005 launch eliminates the need for an Earth gravity assist. All launch periods have nearly identical requirements for total onboard ΔV and all trajectory-altering events.

Table 4 – Multiple Launch Options Summary

Open Date	Close Date	Duration (years)	# Flybys Earth	# Flybys Mercury	Comment
3/23/04	4/06/04	6.5	1	2	prime
8/02/04	8/15/04	6.2	1	2	1st backup
8/02/05	8/15/05	5.2	0	2	2nd backup

Table 3 - MESSENGER Launch Window for August 2004 Backup

Launch Date	DLA (deg)	Flyby Min. Altitude (km) Earth	Flyby Min. Altitude (km) Venus 1	ΔV_1 (m/s)	ΔV_2 (m/s)	ΔV_3 (m/s)	ΔV_4 (m/s)	ΔV_{MOI} (m/s)	ΔV_{MARGIN} (m/s)
08/02/04	-26.3	7216	3571	286.0	255.4	72.7	238.7	1554.8	100.0
08/03/04	-26.6	7536	3561	275.1	255.2	72.8	238.4	1554.6	111.4
08/04/04	-26.9	8106	3444	281.4	244.6	72.7	238.8	1554.9	115.1
08/05/04	-27.2	8842	3296	292.7	231.9	72.6	239.5	1555.4	115.5
08/06/04	-27.5	9722	3180	299.8	224.5	72.5	240.0	1555.8	115.0
08/07/04	-27.8	10286	3111	301.0	223.9	72.5	240.0	1555.8	114.4
08/08/04	-28.1	11004	3081	297.3	228.8	72.6	239.7	1555.6	113.7
08/09/04	-28.4	11459	3078	290.1	237.6	72.6	239.2	1555.2	112.9
08/10/04	-28.9	12029	3093	280.4	249.3	72.8	238.5	1554.7	111.8
08/11/04	-29.5	12071	3122	268.9	263.2	73.1	237.8	1554.2	110.3
08/12/04	-29.9	12941	3161	255.8	279.0	73.5	237.1	1553.6	108.4
08/13/04	-30.3	13212	3207	241.5	296.4	74.1	236.5	1553.1	105.9
08/14/04	-30.8	13748	3258	226.2	315.3	74.9	236.0	1552.5	102.8
08/15/04	-33.5	14232	3312	209.8	335.4	75.8	235.5	1552.0	99.0

Constants: $C_3 = 16\ km^2/sec^2$; flyby altitude for Venus 2 (300 km) & Mercury 1 & 2 (200 km)

HELIOCENTRIC TRANSFER

In addition to providing maximum launch resiliency, MESSENGER's heliocentric trajectory blends operational constraints, low-risk propulsion-free Venus and Mercury gravity assists, and maximum recovery from maneuver delays. Figure 4 displays the heliocentric (launch to MOI) trajectory in fixed Sun-Earth coordinates. Unlike Figure 2, which shows one orbit between Mercury encounters, this representation shows each heliocentric orbit. The resonance between spacecraft and Mercury orbits is shown in Figure 4 as two (purple) orbits between the Mercury flybys, and three (green) orbits between the second Mercury flyby and Mercury arrival. Neglecting a ± 0.017 AU variation in Sun-Earth distance, Figure 4 provides a way of directly measuring not only the component of the SEP angle projected into the ecliptic (Earth orbit) plane, but also the spacecraft distance from the Earth. The importance of the SEP angle was mentioned in the "Payload Characteristics and Constraints" section. Spacecraft distance from Earth is significant for telecommunications system design parameters as well as for determining data transmission rates, link margins, and approximate one-way light time.

Venus Flybys

Two unpowered Venus flybys offer significant reduction in onboard propulsion requirements by utilizing Venus' gravity field to shape MESSENGER's trajectory closer to Mercury's orbit. Throughout all prime and backup launch windows Venus flyby dates vary less than two days, a stability resulting from the best Venus flyby position being roughly opposite the location of the Mercury flybys. The Venus flybys remove energy from the heliocentric trajectory and rotate the spacecraft trajectory plane nearer to Mercury's orbit plane. By splitting the effect over two encounters, the first flyby produces a spacecraft orbit period exactly equal to Venus' orbit period. This assures that the spacecraft and planet will meet again one Venus period later.

The first Venus flyby targets close approach altitudes of 3046 to 3553 km over the launch period (Tables 2 and 3), achieving reduction in perihelion and aphelion and increase in orbit inclination. The low pre-encounter phase angle (13° to 17°) indicates that the spacecraft views a mostly sunlit Venus during its approach trajectory. However, solar conjunction (spacecraft passes within 2° of the Sun as viewed from the Earth) occurs between 3-4 days before and two weeks after this Venus flyby. During this time Earth-based antennas will track the spacecraft, even though command transmission and data downlink will be degraded. Therefore, greater emphasis is planned for pre-encounter orbit determination and targeting ΔVs. In addition, high close approach altitude reduces the effect of target errors during reduced communication capability, an important risk mitigation point.

The second Venus flyby establishes an orbit with aphelion near Venus' orbit radius and perihelion near Mercury's perihelion distance. The 300-km minimum encounter altitude, adopted to satisfy NASA planetary protection requirements[8], may decrease upon generation of both a high-precision integrated trajectory and a detailed pre-encounter orbit determination plan. Even though the optimum close approach altitude is below 300 km, this flyby requires no propulsive ΔV. A 21°-22° approach phase angle indicates that, like the first Venus flyby, the approaching spacecraft will view a mostly sunlit Venus. Figure 5 shows views of the second Venus flyby from the Sun, where a 20-minute solar occultation forces reliance on battery power, and Earth, from which the flyby is shown to be clearly visible.

Deep Space Maneuvers and MOI

To achieve the lowest possible total ΔV the MESSENGER spacecraft must perform four propulsive maneuvers and MOI within discrete windows of opportunity. These windows of opportunity close (maximum maneuver delay) when a constraint is reached. These constraints are: 1) large thruster orientation such that thermal shade blocks sunlight from instruments or opposite end of the spacecraft bus, 2) SEP angle greater than 2° to guarantee maneuver observability from Earth without Sun interference, and 3) ΔV penalty from maneuver recovery through MOI is less than 85 m/sec. The number of windows of opportunity for each maneuver increases from one for DSMs 1 and 2, to two for DSMs 3 and 4, to three for MOI.

The method for defining the cost of delays in DSM execution or MOI uses a conservative single ΔV recovery strategy. For some delays an optimal recovery strategy may require more than one ΔV. For example, NEAR completed two maneuvers on January 20 and August 12, 1999 to correct execution errors from a 932 m/sec anomaly recovery maneuver[9] performed on January 3, 1999. For MESSENGER's DSMs the pre-DSM trajectory was propagated beyond the planned DSM time in 1-day delay increments. Each recovery DSM complies with the constraints listed in the previous paragraph. Subsequent maneuvers also comply with each constraint, and no planetary flyby requires a propulsive maneuver. Total ΔV was minimized from the recovery DSM through MOI. Figure 6 charts "ΔV cost," which is the increase in total ΔV for all propulsive maneuvers through MOI for short-term delays in DSMs and MOI. Figure 6 also shows which constraint defines the maximum delay for each DSM. Recovery from MOI delay utilizes a near-perihelion ΔV and results in a 1.5-year delay in MOI.

Delays in DSM 2 offer a special case because the SEP angle equals 2° for the longest-allowable delay. Since the minimum total ΔV for the heliocentric trajectory requires an SEP angle less than 1°, the nominal case moves DSM 2 five days before the spacecraft reaches the 2° SEP angle constraint, and re-optimizes the entire heliocentric trajectory. Therefore, short delays in DSM 2 actually *reduce* the magnitude of the recovery DSM 2 and the total ΔV. The ΔV cost for DSM 2 in Figure 6 results from net increase in total deterministic ΔV for moving the nominal DSM 2 date earlier. For an extra 12 m/sec propellant or corresponding reduction in ΔV margin, DSM 2 may be delayed up to ten days. At this point the required large thruster alignment for DSM 2 moves the thermal shade close to its limit for protecting science instruments from sunlight.

While Figure 6 shows the first window of opportunity for maneuver recovery, DSMs 3, 4, and MOI have additional recovery opportunities on subsequent heliocentric orbits. DSM 3 occurs near the first of two aphelions between the two Mercury flybys. Similarly, DSM 4 occurs near the first of three aphelions between the second Mercury flyby and MOI. A second window of opportunity for DSM recovery occurs near the second aphelion after each Mercury flyby. This costs 33 m/sec more total ΔV for a 103-day DSM 3 delay and 69 m/sec for a 98-day DSM 4 delay. An extraordinary condition occurs in the event of a missed MOI maneuver – Mercury's gravity field sends the spacecraft very close to a 6:5 Mercury:spacecraft orbit resonance. The result is that a near-perihelion maneuver 107 days after the missed MOI enables a 1.5-year delayed MOI costing only an extra 67 m/sec total ΔV. An extra 85

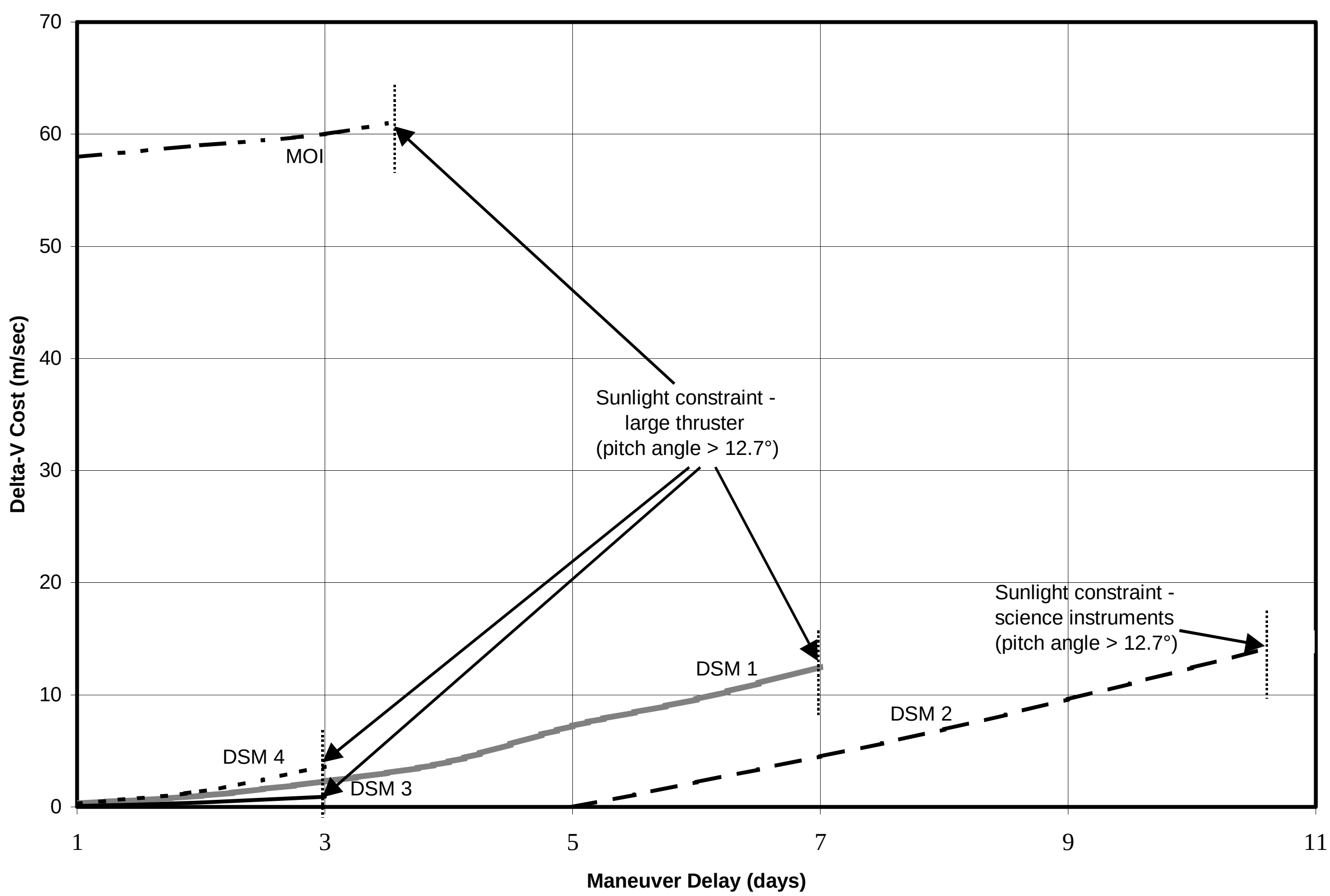


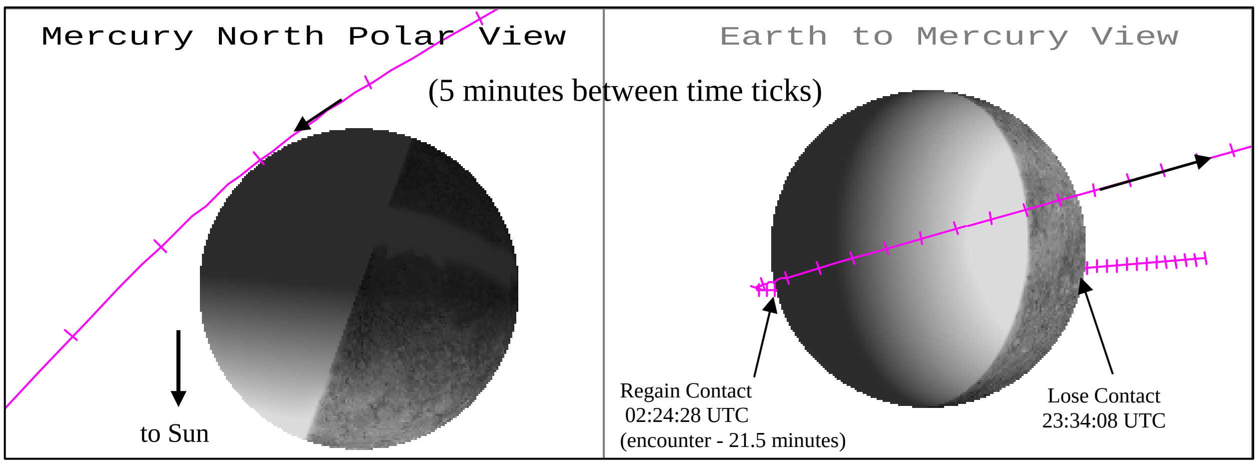
Figure 6 - Impact of Delay on Major Propulsive Events Prior to Mercury Orbit Phase

m/sec total ΔV is required for a MOI recovery maneuver two perihelions after the September 30, 2009 nominal MOI.

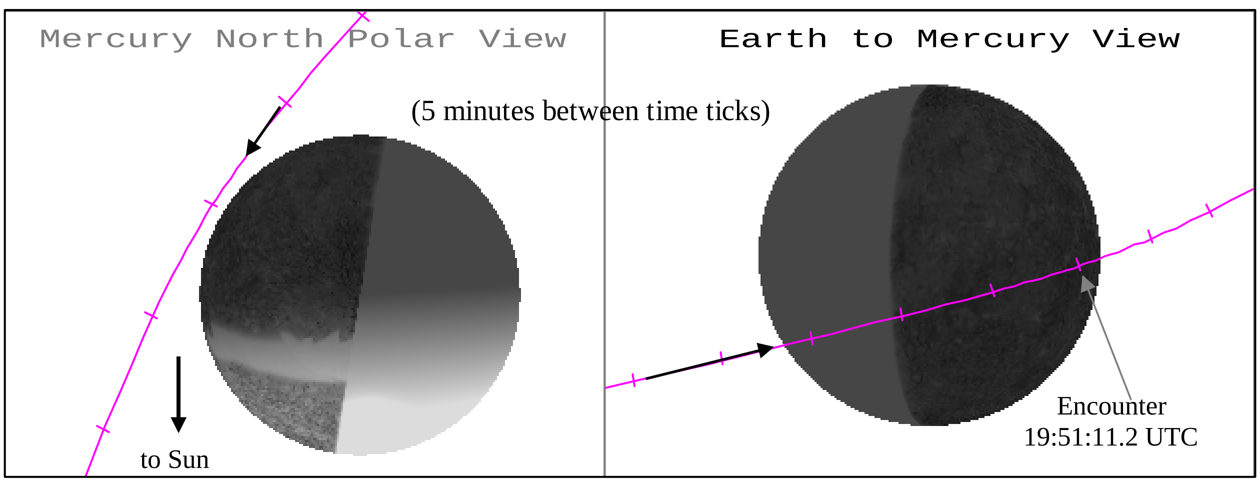
For post-launch problems that drive spacecraft ΔV reserves too low to complete the mission by September 30, 2010, MOI could become a third Mercury gravity assist. This third Mercury flyby leads to a near-aphelion maneuver and MOI delayed 1.5 years from September 2009 to March 2011. The plan allows mission recovery by lowering spacecraft propellant requirements by over 500 m/s ΔV, mostly due to a lower recovery ΔV that targets Mercury arrival 1.5 years after the original MOI.

Mercury Flybys

Two unpowered 200-km minimum altitude Mercury flybys followed by near-aphelion DSMs slow the spacecraft Mercury approach velocity enough to enable Mercury orbit insertion at the third Mercury encounter. This differs from the ΔV-Earth gravity assist technique used by NEAR and planned by STARDUST[10] in that the Mercury gravity assists lower aphelion and work with the DSMs



First Mercury Flyby - 15 January 2008



Second Mercury Flyby - 6 October 2008

Figure 7 – Views of Closest Approach for each Mercury Flyby

arrival velocity at MOI. In addition, a delayed MOI may serve as a nontargeted Mercury gravity assist followed by a small perihelion to rotate the spacecraft orbit line of apsides closer to the Mercury orbit line of apsides. The Mercury-to-Mercury transfer orbits have successive spacecraft:Mercury orbital resonance of nearly 2:3 and 3:4. The Mercury flybys occur on January 15 and October 6, 2008 at approach phase angles of 112° and 121°, so that the spacecraft will view a larger sunlit portion of Mercury after close approach. The spacecraft-Mercury relative velocity decreases from 5.79 km/s to 5.15 km/s to 3.37 km/s at Mercury flybys 1, 2, and MOI, respectively. Figure 7 displays views of each Mercury flyby from Mercury's north pole direction and from Earth. The north pole view gives a good perspective on how close and fast the flyby is, and what areas of Mercury are sunlit on approach and departure. Together, the flybys offer opportunity to observe most of the never-before-imaged areas of Mercury's surface. The Earth view clearly shows what portions of each encounter offer communication links with Earth, and what Mercury latitudes the spacecraft passes over. The first Mercury flyby enables Earth tracking from shortly before closest approach through the rest of the encounter, including all portions of the encounter with imaging of the never-before-imaged hemisphere. The second Mercury flyby offers opportunity for an Earth communications link throughout the encounter.

MERCURY ORBIT

Orbit Configuration

The heliocentric transfer ends at Mercury orbit insertion into a 125-km altitude by 12-hour, near-polar orbit on September 30, 2009. Orbit insertion requires a 24.4-minute, 1.553 km/s maneuver including 68 m/sec gravity loss ΔV and a periapsis rotation from 63.4° to 60.0° N. Insertion directly into a much more eccentric orbit with a period of 4-6 days would lower the gravity loss ΔV. However, the orbit variation from solar radiation pressure may lead to thrust vector orientations that leave science instruments exposed to direct sunlight.

Other advantages of MOI directly into a 12-hour orbit include maximum likelihood of orbit capture in case MOI stops prematurely. Orbit capture is possible for orbit apoherm distances approaching Mercury's sphere of influence of 70,000 miles (113,000 km). Backing off to account for solar radiation pressure would allow insertion into an orbit of 200-km periherm altitude by 6-day period. If analysis reveals a constraint-compliant plan for lowering apoherm of an intermediate orbit, with period between one and four days, then tens of meters/second of ΔV could be saved due to lower MOI gravity loss.

Figure 8 presents the 12-hour reference orbit for MESSENGER's 12-month orbital phase. This orbit period meets science requirements for data collection and data downlink and simplifies shift work for Mission Operations personnel. The orbit phase duration allows 352 days to acquire stereo mapping of Mercury's surface and two weeks for orbit establishment and margin.

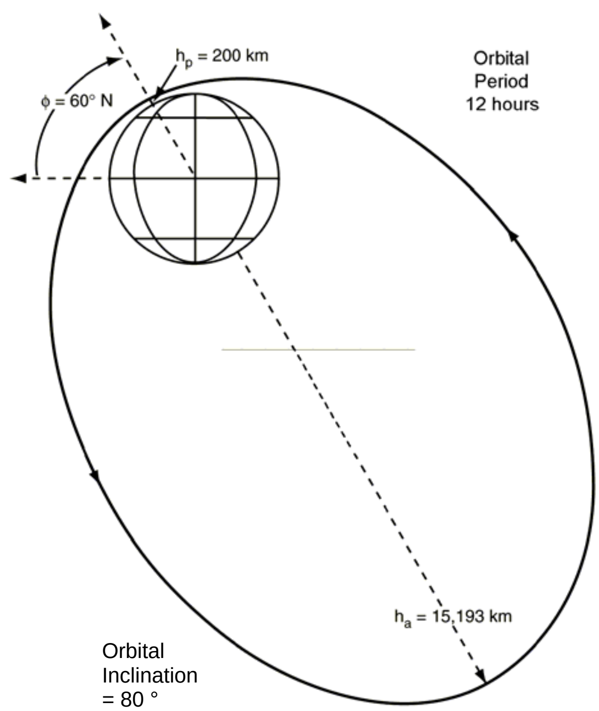


Figure 8 – Nominal Orbit at Mercury

Orbit Correction Strategy

Solar radiation pressure acting on the spacecraft increases periherm altitude, pushes periherm latitude northward, and alters other orbital parameters such that active correction is unnecessary. Plans for meeting a key science requirement, to minimize the frequency of propulsive corrections, are addressed by rolling the spacecraft each orbit for passive momentum control, and by placing two orbit correction ΔVs six hours apart once every 88 days. Since Mercury orbits the Sun every 88 days, the ideal large thruster alignment (one that maximizes spacecraft protection by the thermal shade) repeats every 44 days for apoherm ΔVs that lower periherm. The dates and magnitudes of these maneuvers, along with maneuvers that adjust orbit period to 12 hours, appear in Table 5. Periherm altitudes at the time of the periherm lowering ΔVs are 411, 475, and 460 km, respectively, with an end of mission periherm altitude of 502 km.

Table 5 – Orbit Phase Deterministic ΔV

Date	Event	ΔV (m/sec)
28 Dec 2009	Periherm lower	19.5
28 Dec 2009	Period adjust	3.0
27 Mar 2010	Periherm lower	25.2
27 Mar 2010	Period adjust	3.9
23 June 2010	Periherm lower	23.7
23 June 2010	Period adjust	3.7
Total		79.0

Delays in any of these orbit correction maneuvers may be corrected with minimal (<0.5 m/sec) extra ΔV at 12, 24, or 36 hours later. Slightly longer delays may be possible if the large thruster is not used, but ΔV components are applied with the less efficient smaller thrusters. Finally, if longer times are required to ensure safe thruster application, the next available recovery ΔV could occur 44 days later. This scenario may be problematic unless small (<< 1 m/sec) propulsive momentum dumps are able to continue every 4-5 days.

CONCLUSION

Scheduled for launch in 2004, NASA's MESSENGER spacecraft will be the first to orbit the planet Mercury. Engineers at The Johns Hopkins Applied Physics Laboratory and scientists throughout the United States of America have worked together to provide a simple, robust spacecraft design in an intense thermal/radiation environment with a comprehensive science instrument payload. Mission Design worked closely with these groups and project management to provide a resilient trajectory, one that offers multiple launch opportunities, provides direct Earth-based monitoring of planetary flybys and maneuvers, and enables recovery from maneuver delays of at least one week. Some maneuvers even offer recovery from delays of 14-15 weeks due to repeating heliocentric orbits between Mercury flybys. Maneuver delays in Mercury orbit also offer multiple recovery opportunities, although spacecraft angular momentum buildup may be a concern for long delays affecting all thrusters.

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