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IMPROVEMENTS IN TRAJECTORY OPTIMIZATION FOR MESSENGER: THE FIRST MERCURY ORBITER MISSION

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MESSENGER (MErcury Surface, Space, ENvironment, GEochemistry, and Ranging), the seventh NASA Discovery Program mission, will utilize a carefully planned sequence of Venus and Mercury gravity assist flybys to deliver the 3-axis-stabilized, dual-mode-propulsion spacecraft into Mercury orbit. During spring 2000 a methodical search of newly identified Venus-Venus transfer trajectories yielded two new minimum-ΔV ballistic trajectories to Mercury within MESSENGER's highly-constrained mission requirements. Both minimum-ΔV launch opportunities in March and May 2004 not only satisfy a wide array of science goals, but also meet engineering, operational, programmatic, and cost constraints. These new baseline and backup trajectories provide substantial improvements for several mission-critical aspects of MESSENGER's five-year journey to Mercury over the lowest-ΔV trajectories known at the start of Phase B studies. Some of these performance improvements were noted as key factors in NASA's June 2001 confirmation for Phase C/D funding.

INTRODUCTION

MESSENGER (MErcury Surface, Space ENvironment, GEochemistry, and Ranging), the seventh NASA Discovery Program mission[1], will use a carefully planned sequence of propulsive maneuvers and Venus and Mercury gravity assists to transform a low-energy launch trajectory into a science-rich Mercury orbit. Designed and operated by The Johns Hopkins University Applied Physics Laboratory (JHU/APL) in Laurel, Maryland, MESSENGER draws leadership from the Carnegie Institution of Washington and key contributions from NASA's Jet Propulsion Laboratory (JPL), Goddard Space Flight and Kennedy Space Centers, numerous universities, and various subcontractors.

This paper will describe all guidelines and constraints that affect the trajectory optimization and maneuver design for MESSENGER's journey to Mercury. These factors include science goals, engineering limitations for subsystems, operational simplicity, risk mitigation, and program cost. In addition, a brief historical perspective will establish how recent advances in trajectory optimization yielded MESSENGER's reference mission plans for Discovery Program selection (Phase A) and late Phase B mission confirmation by NASA.

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After defining MESSENGER's Mercury orbiter trajectory design, attention shifts to describing details of key trajectory characteristics in the context of mission guidelines and constraints. Upon completion of the detailed trajectory and maneuver highlights, the final focus is on a comparison between the old and new baseline and backup launch opportunities, with emphasis on mission-enabling features of the current mission plan.

TRAJECTORY AND MANEUVER DESIGN GUIDELINES AND CONSTRAINTS

Any endeavor to design a low-risk Mercury orbiter mission having a comprehensive science payload, yet remaining within cost and programmatic constraints of NASA's Discovery Program, will constrain trajectory and maneuver design in a way that leaves few viable options. Factors affecting trajectory optimization and mission design are as follows: Discovery Program directives, science payload, spacecraft operational capability, and programmatic.

The 1998 Discovery Program Announcement of Opportunity (AO)[2] imposed restrictions on launch dates, available launch vehicles, and total mission cost. Thermal protection, propellant mass fraction, and a comprehensive science payload left one choice – the Delta 2925H-9.5 (formerly 7925H-9.5), which is the highest-performance (see Figure 1) expendable launch vehicle allowed within Discovery. The Delta 2925H-9.5 launches from Cape Canaveral Air Force Station, thereby limiting declination of launch asymptote (DLA) to $\pm 28.5^\circ$ without performance degradation. The AO guideline that "a mission launch every 12 to 24 months," the AO requirement "that launch can take place by September 30, 2004," and a Discovery mission (CONTOUR) launch set for July 1, 2002, all constrain any MESSENGER baseline and backup launch opportunities to begin and end between July 1, 2003, and September 30, 2004. Although the AO does not specify maximum mission duration, it does set a total mission cost limit at FY 1999 $299 million. For the many cost considerations of a Mercury orbiter mission, the MESSENGER project could fund up to a six-year heliocentric transfer followed by a one-year orbit at Mercury.

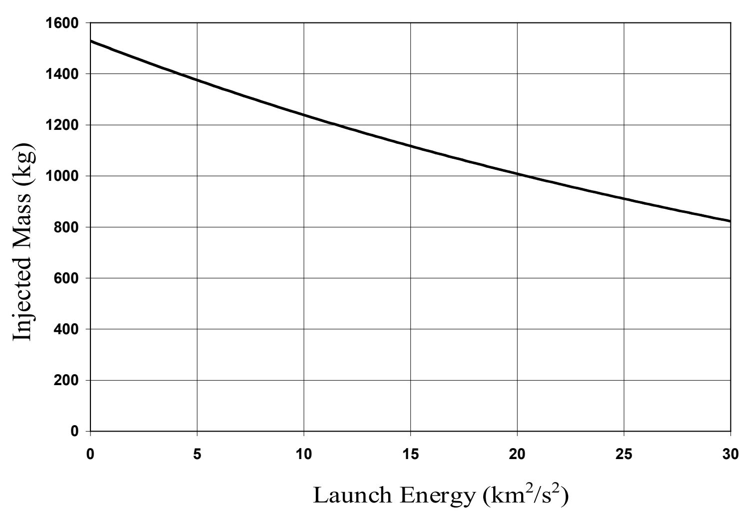

Figure 1 Delta 2925H-9.5 Performance for 99.0% Probability of Commanded Shutdown

The suite of seven science instruments comprising MESSENGER's science payload[3] imposes additional requirements on the trajectory and propulsive maneuver frequency. The science instruments address six scientific objectives.[4] The Mercury orbit trajectory must enable stereo monochrome imaging at 250 meters average resolution and color imaging at 2 kilometers average resolution of > 90% of Mercury's surface. MESSENGER's 80º-inclination, 200-km periherm altitude by 12-hour (± 1 minute) period orbit with 60ºN initial periherm latitude will provide stereo monochrome imaging of over 98% of Mercury's surface within one Earth year of Mercury orbit insertion. A near-polar (initial orbit inclination of 78º-82º) Mercury orbit requiring infrequent orbit adjustment (two closely-spaced maneuvers every three months) will not only facilitate measurement of planetary magnetic dipole strength and direction, but also enable libration amplitude measurement (solid vs. liquid core determination) and gravity field determination. Low-altitude north polar passes will offer opportunity for accurate determination of surface topography and elemental composition of radar-detected volatile deposits.

Spacecraft operational capability encompasses issues such as subsystem component masses and technology readiness. For example, reflected solar radiation from Mercury's surface onto the thermally-vulnerable spacecraft bus and science instruments limits the sub-solar time spent by the spacecraft at low altitude. This requirement also sets a minimum Mercury-to-Sun distance (0.38 AU) during low-altitude, sub-solar fly over by restricting the spacecraft orbit's longitude of ascending node to be between 73º and 248º in the Mercury equator and equinox of epoch reference frame. Figure 2 shows a

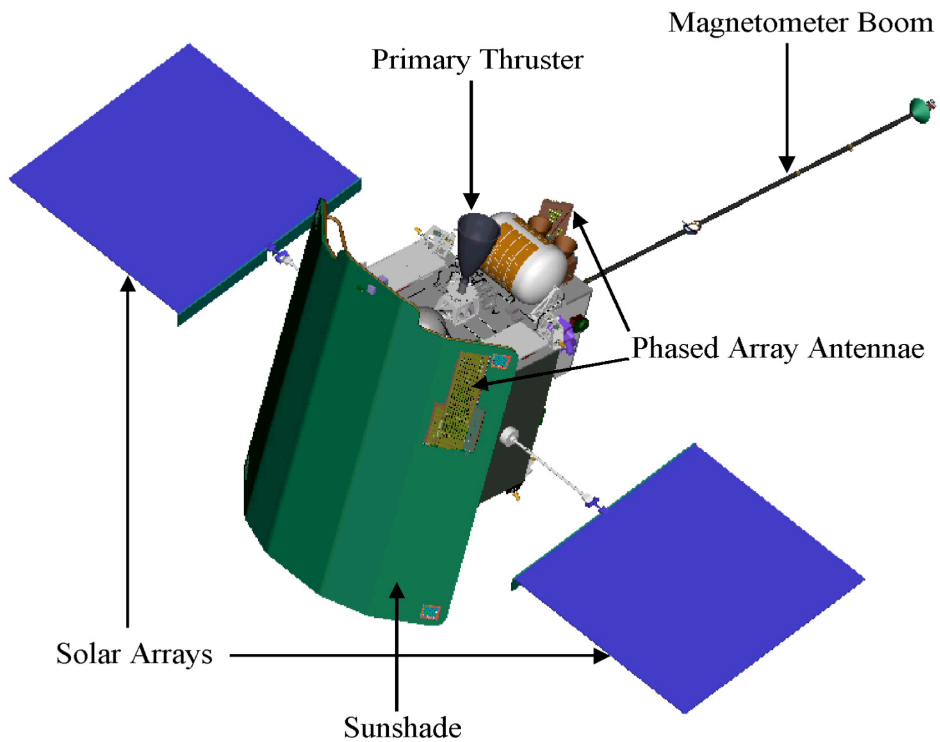

Figure 2 Oblique View of Current MESSENGER Spacecraft Configuration

view of the spacecraft with components identified that most affect the trajectory and maneuver design requirements. Onboard ΔV capability for the dual-mode (monopropellant/bipropellant) chemical propulsion system was 2700 m/sec for the 1999 Concept Study[5] and 2300 m/sec at the May 2001 Preliminary Design Review (PDR). In order to protect the primary spacecraft bus from overheating due to direct sunlight during bipropellant propulsive maneuvers closer than 0.8 AU from the Sun, the ΔV-spacecraft-Sun angle must be from 78˚ to 102˚. For even greater safety margin, this angle should be between 80˚ to 100˚ during the Mercury orbit insertion (MOI) maneuver. In addition, sufficient link margin for maintaining reliable downlink telemetry necessitates that the Sun-Earth-spacecraft angle exceeds 2.5˚ during all propulsive maneuvers. In order to prevent a significant increase in battery mass, the maximum-duration solar eclipse passage is 65 minutes. A 60-minute eclipse occurs about ten weeks after MOI. Long-term (primary science orbit) spacecraft orbit period is set at 12 hours for several reasons. These reasons include: (1) periherm altitude drifts upward more quickly for longer orbit periods (45% faster for 13.5 hours versus 12.0 hours, resulting in significant orbit correction ΔV increase and degraded science return), and (2) ease of scheduling deep space tracking resources and Mission Operations staff schedules.

Programmatic guidelines focus on providing prudent operational margins. For example, minimum altitudes for planetary gravity assists of 300 km for Venus and 200 km for Mercury decrease planetary impact probability well below NASA planetary protection requirements[6] when considering navigation and orbit determination techniques utilized by MESSENGER. Heliocentric deterministic (having defined requirements prior to launch) maneuvers shall be performed during the first orbit after a planetary flyby when multiple heliocentric orbits exist between flybys. For < 10 m/sec extra ΔV, risk decreases by extending allowable ΔV delays from a few days to a few months. Both a requirement for zero-propulsive-assist (unpowered) planetary flybys and a guideline for minimum total number of planetary flybys and deterministic ΔVs contribute to lowering risk and total mission cost. The minimum baseline launch window duration is 15 days.

CONCEPT STUDY BASELINE AND BACKUP TRAJECTORIES

The MESSENGER Concept Study baseline and backup heliocentric trajectories were primarily derived from Yen's 1985 initial documentation[7] of a method combining Venus and Mercury gravity assists. The first improvement to the 1985 lowest total-ΔV case for the 2005 launch opportunity, a ΔV reduction of 20-30 m/sec, came via Yen, Horsewood, and McAdams[8] in 1998. Modification in 1998-1999 of this improved trajectory, using two types of Earth-Earth transfers, resulted in MESSENGER's Concept Study baseline and backup launch opportunity trajectories[5]. On July 7, 1999, NASA selected a team headed by Carnegie Institution of Washington and JHU/APL to begin Phase B study of this mission plan as NASA's seventh Discovery mission, the first time a spacecraft would explore the closest planet to the Sun in over three decades. Due to ongoing updates in Delta 2925H-9.5 projected performance and MESSENGER spacecraft design, mass data will not be presented here.

Concept Study Baseline Trajectory

The baseline, or primary, launch opportunity in the MESSENGER Concept Study utilized a type III launch-to-Earth flyby transfer to graft into the 2005 launch opportunity without paying the small launch performance penalty associated with high launch declination in August 2005. Determining launch window open and close dates, given a minimum 15-day guideline, began with the minimum total ΔV (March 26, 2004) launch-to-MOI trajectory. The 2.5˚ Sun-Earth-spacecraft (SEP) constraint required a shift earlier from the minimum-ΔV location for the second deterministic ΔV (ΔV_2). Additional details concerning formulation of the 15-day March 23 – April 6, 2004, baseline launch window are found in Ref. 5. The 2700 m/sec total ΔV budget includes 2386 m/sec deterministic, 114 m/sec for navigation and attitude control, and a 100 to 117 m/sec margin. A Delta II 3-stage rocket launched from Cape Canaveral Launch Complex 17 would deliver the spacecraft into the short-coast option, 185-km altitude parking orbit.

The heliocentric transfer trajectory (see Figure 3) required one Earth flyby, two Venus flybys (one at minimum altitude), two Mercury flybys, and four deterministic ΔVs, a total of nine operation-intensive events, prior to MOI. After launch the spacecraft follows a type III (transfer angle between 360˚ and 540˚) Earth-to-Earth transfer in the ecliptic plane. After a medium-altitude Earth gravity assist, a near-perihelion maneuver (ΔV_1 in Figure 3) adjusts the non-optimal, type IV Earth-to-Venus transfer phasing. The longest solar conjunction (35 days) begins five days after ΔV_2.

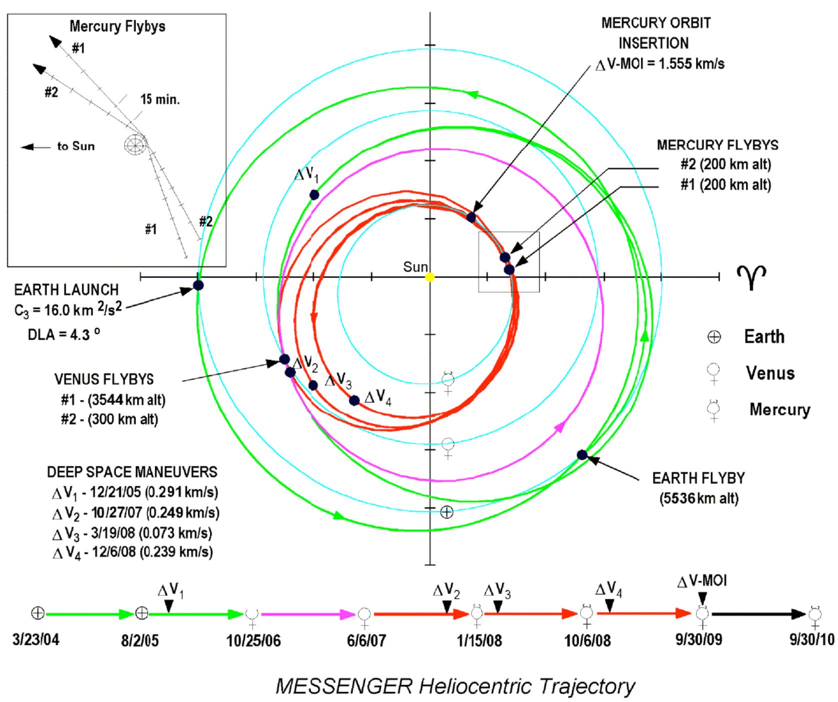

Figure 3 North Ecliptic Pole View of Concept Study Baseline Trajectory

Both unpowered Venus flybys make significant contributions toward shaping the spacecraft trajectory closer to Mercury's orbit. The first Venus flyby, with medium-range close approach altitudes just over half a Venus radius, reduces the spacecraft orbit's perihelion and aphelion and increases orbit inclination. An approach phase angle of 13˚ to 17˚ (varies depending on launch day) indicates a spacecraft view of a brightly sunlit Venus. However, pre-Venus flyby targeting accuracy and post-flyby orbit reconstruction accuracy and timeliness would suffer somewhat due to the timing of solar conjunction. For the first Venus flyby, solar conjunction (SEP angle < 2˚) occurs beginning 3-4 days before and ending two weeks after close approach. One Venus orbit later the spacecraft returns for a 300-km altitude second Venus flyby, a flyby that moves spacecraft aphelion to Venus' orbit and perihelion near Mercury's perihelion. An approach phase angle of 21˚ to 22˚ indicates a spacecraft view of a brightly sunlit Venus. After the second Venus flyby, a near-aphelion maneuver (ΔV_2 in Figure 3) adjusts the non-optimal, type III Venus-to-Mercury transfer phasing. Any delay in implementation of ΔV_2 would violate the guideline for performing maneuvers at > 2.5˚ SEP angle.

Two unpowered 200-km minimum altitude Mercury flybys followed by post-aphelion maneuvers complete spacecraft orbit apsidal rotation and lower aphelion enough to enable Mercury orbit insertion in late September 2009. This third Mercury encounter is the first with arrival velocity low enough to use onboard propellant for MOI. Each ΔV occurs during the first post-flyby orbit according to the previously stated programmatic guideline. The Mercury gravity assists and subsequent ΔVs produce successive spacecraft: Mercury orbital resonance of about 2:3 and 3:4. The Mercury flybys (see Figure 3 upper left) have approach phase angles of 112˚ and 121˚ on January 15, 2008, and October 6, 2008. The spacecraft hyperbolic excess velocity relative to Mercury is 5.79 km/sec, 5.15 km/sec, and 3.37 km/sec at Mercury flybys 1, 2, and MOI, respectively.

On September 30, 2009, the heliocentric transfer orbit culminates with Mercury orbit insertion into a 125-km by 12-hour, 80˚-inclination orbit. Choice of the lowest allowed initial periherm altitude resulted in some improvement in science return (higher resolution near periherm) and savings of several m/sec in orbit correction ΔV. The 1.55 km/sec MOI maneuver lasts nearly 25 minutes, includes about 70 m/sec gravity loss ΔV, and rotates periherm latitude from 63.4˚ to the required 60.0˚N. Although insertion into an orbit with much higher eccentricity would decrease gravity loss ΔV, there are a number of real and potential disadvantages to this strategy. Real disadvantages include: (1) solar gravity perturbations and solar pressure on the spacecraft in highly eccentric Mercury orbits tend to quickly increase periherm altitude beyond limits set by science requirements, thereby introducing the need for an extra maneuver near apoherm before the predicted orbit, gravity field, or perturbations are well understood, (2) the ΔV-Sun orientation constraint may be violated for the large apoherm-lower ΔV at periherm (needed to establish the 12-hour orbit period), and (3) risk of not achieving the ± 1 minute orbit period tolerance increases for longer thrust times needed for the orbit period adjust maneuver after MOI. A potential disadvantage is that MOI premature cutoff is more likely to result in failure to enter Mercury orbit for a higher eccentricity initial orbit.

Figure 4 shows the 12-hour primary science orbit for MESSENGER's 12-month Mercury orbital phase.

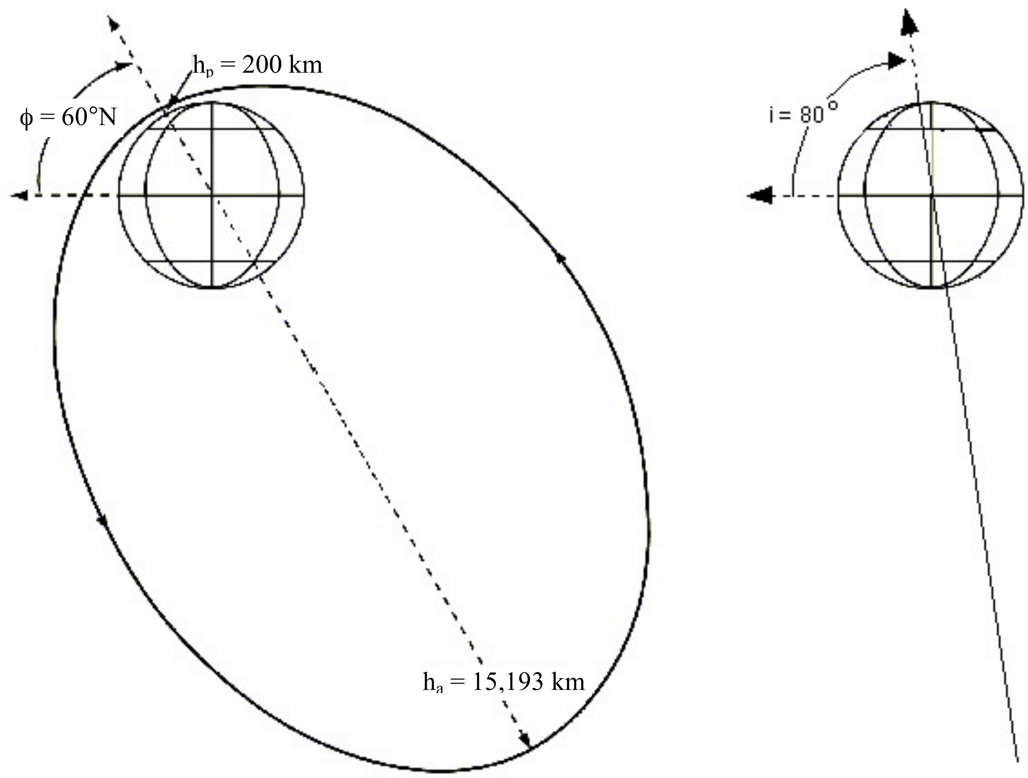

Figure 4 MESSENGER Primary Science Orbit at Mercury

Solar pressure and solar gravity perturbations on the spacecraft orbit increase periherm altitude 80-90 km/month thereby requiring periodic propulsive correction. In addition, orbit inclination increases at an average of 0.15˚/month, periherm latitude shifts northward at an average of 1.0˚/month, and other orbital parameters vary at rates within mission guidelines and requirements. Although the spacecraft orbit orientation relative to the Sun allows orbit correction maneuvers to remain within the ΔV-Sun orientation constraint for 2-3 days every half Mercury year (44 days), science instrumentation requires maximizing the time between propulsive orbit corrections. Therefore, since the maximum allowable periherm altitude is approached within 100 days, two orbit correction maneuvers (OCM's) will occur once per Mercury year. As shown in Table 1, the first OCM of each pair lowers periherm to 200 km and, 18 hours later, the second OCM adjusts orbit period to 12 hours. Expected periherm altitudes prior to periherm lowering ΔVs are 411, 475, and 460 km, with a 502-km periherm altitude one year after MOI.

Table 1
CONCEPT STUDY 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 Jun 2010	**Periherm lower**	**23.7**
23 Jun 2010	**Period adjust**	**3.7**
Total		**79.0**

Concept Study Backup Trajectory

Because of minimal differences between the Concept Study baseline and backup trajectories, little needs to be said about this backup launch opportunity. Instead of a type III launch-to-Earth flyby transfer, this backup mission uses a 5.5˚-inclination (relative to the ecliptic plane) one-year transfer beginning in early August 2004. Total ΔV and launch energy do not change between baseline and backup missions. A daily launch window summary like Table 1 in this paper appears in Ref. 5 for the August 2-15, 2004, backup launch opportunity. The only noteworthy difference between baseline and backup is a declination of launch asymptote between –28.5˚ and –33.5˚ (guideline violation resulting in slightly lower launch vehicle performance) for the last six days of the 14-day window.

NEW BASELINE AND BACKUP TRAJECTORIES

The new MESSENGER baseline and backup trajectories arose from research conducted by Yen[9] during winter and spring of 2000. The discovery of new Venus-Venus and Mercury-Mercury transfer techniques[10] by Yves Langevin (Institut d'Astrophysique Spatiale, Orsay, France) in 1999 prompted JHU/APL to channel NASA funds to JPL (Yen) in early 2000 to search for minimum-ΔV Mercury orbiter trajectories. Yen discovered new methods for incorporating Venus gravity assists into ballistic trajectories to Mercury. By applying two new methods, Yen found two trajectories with spring 2004 launch dates, nearly optimal Earth-Venus and Venus-Mercury phasing, and spacecraft: Venus orbit resonance of 3.8:2.8 (two Venus flybys for the baseline) and 1.3:1.3 followed by 2.9:1.9 (three Venus flybys for the backup). This near-perfect timing between planetary flybys not only reduced launch energy, deterministic ΔV, and total trip time, but also increased the minimum Venus flyby altitudes and eliminated the need for two propulsive maneuvers present in the previous baseline and backup trajectories.

New Baseline Trajectory

The new baseline launch opportunity utilizes a type I launch-to-Venus flyby transfer to maximize the potential offered by excellent Earth-Venus launch-arrival phasing. McAdams discovered that it was possible to establish a 20-day launch window, using only 2300 m/sec total ΔV, with each day having launch energy below the Concept Study's fixed value of 16 km^2/sec^2. A Delta II 3-stage rocket launched from Cape Canaveral Launch Complex 17 would deliver the spacecraft into the short-coast option, 185-km altitude parking orbit. Preliminary trajectory optimization for launch on March 10, 2004, placed lift off at 5:31 A.M. UTC, with stage III ignition (start of injection onto the heliocentric transfer trajectory) only 39 minutes later and initial contact with Canberra's DSS-34 antenna as early as 49 minutes after lift off. The spacecraft receives full sunlight about 15 minutes before injection. Preliminary timing of lift off and injection came from mission analysts at NASA Kennedy Space Center.

The heliocentric transfer trajectory (see Figure 5) requires two Venus flybys (neither at minimum altitude), two Mercury flybys, and two deterministic ΔVs, a total of six operations-intensive events, prior to MOI. Each of these events will have multiple trajectory correction ΔVs, requiring less intensive mission operations activity, in order to fine-tune targeting for the next major event. After launch the spacecraft follows a type I (transfer angle < 180˚) Earth-to-Venus transfer. The longest solar conjunction passage lasts 11 days and does not occur near any significant operation-intensive event.

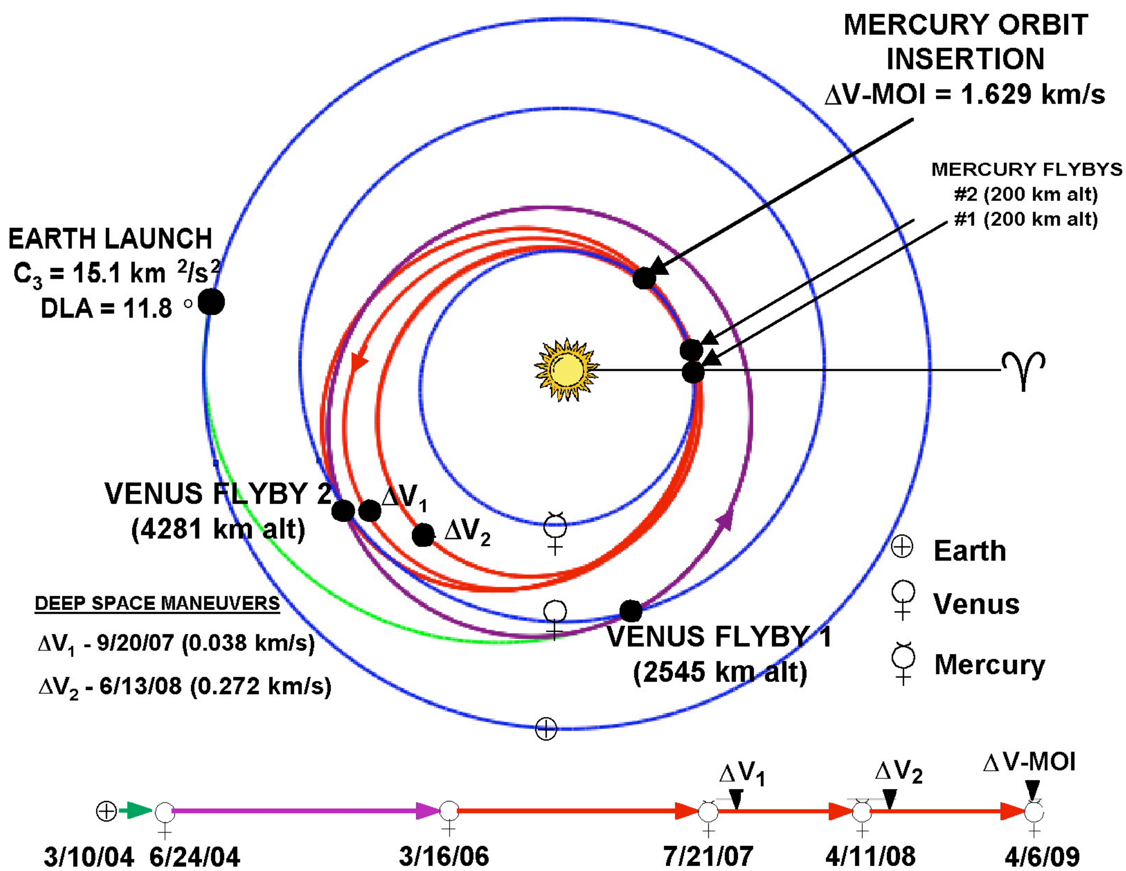

Figure 5 North Ecliptic Pole View of New Baseline Trajectory

Both unpowered Venus flybys provide significant resizing and shaping of the spacecraft trajectory closer to Mercury's orbit. The first Venus flyby, with medium-range close approach altitude just over 0.4 Venus radius, reduces the spacecraft orbit's perihelion and aphelion and increases orbit inclination. An approach phase angle of 162° to 165° indicates a spacecraft view of a thin sunlit crescent Venus. Even though this high phase angle at approach will limit the effectiveness of optical navigation images for approach orbit determination, small Venus and spacecraft ephemeris uncertainties and the medium-altitude close approach 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 moves spacecraft aphelion near Venus' orbit and spacecraft perihelion near Mercury's perihelion. An approach phase angle of 33° indicates a spacecraft view of a brightly sunlit Venus. After the second Venus flyby, the spacecraft begins a type VII zero-ΔV Venus-to-Mercury transfer.

Two unpowered 200-km minimum-altitude Mercury flybys followed by post-aphelion maneuvers complete spacecraft orbit apsidal rotation and lower aphelion enough to enable Mercury orbit insertion in early April 2009. Each ΔV occurs during the first post-flyby orbit according to the previously stated programmatic guideline. The Mercury gravity assists and subsequent ΔVs produce successive spacecraft: Mercury orbital resonance of about 2:3 and 3:4. The Mercury flybys have approach phase angles of 110° and 126° on July 21, 2007, and April 11, 2008. Figure 6 clearly shows how the spacecraft can view opposite sides of the never-before-imaged (bright surface) hemisphere of Mercury soon after minimum altitude passage. Also note that post-flyby orbit

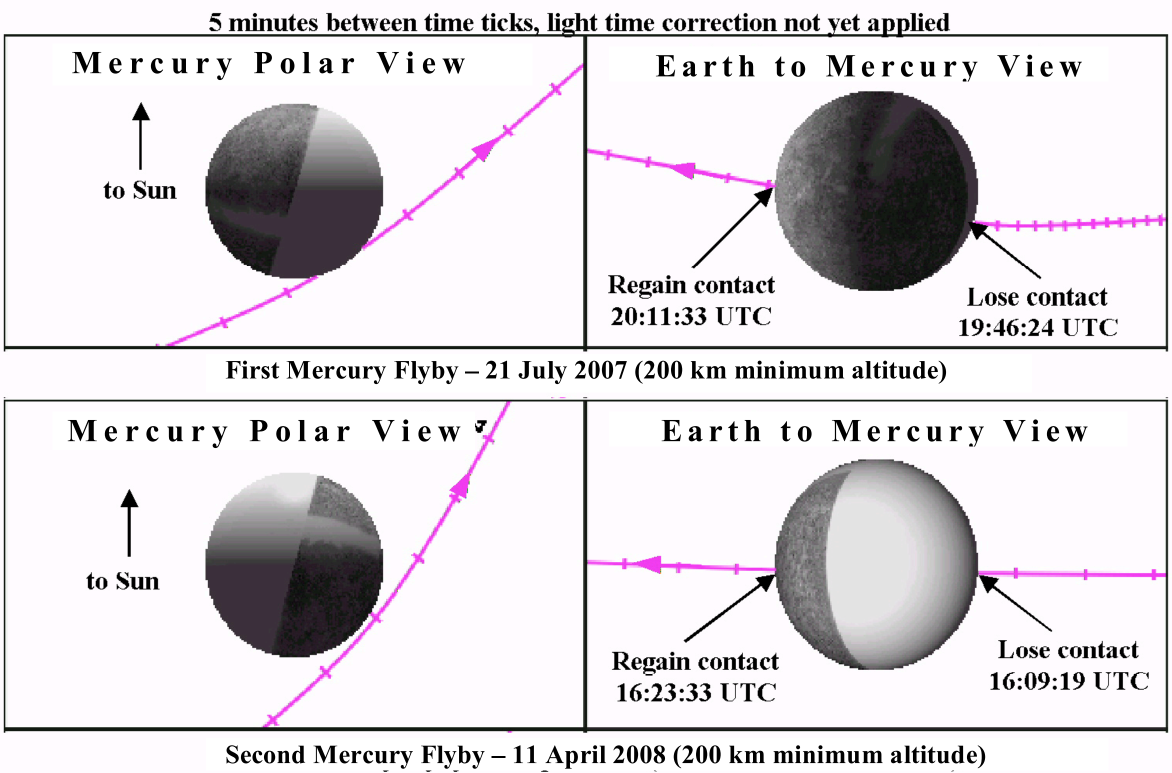

Figure 6 New Baseline Trajectory Mercury Flyby Views

reconstruction will be partially incomplete for both Mercury flybys due to the spacecraft's loss of a communications link near close approach and during solar conjunction from three to six days after the second flyby. The spacecraft hyperbolic excess velocity relative to Mercury decreases from 5.76 km/sec to 5.45 km/sec to 3.41 km/sec at Mercury flybys 1, 2, and MOI, respectively. The actual spacecraft velocity relative to Mercury at close approach is 7.06 km/sec and 6.81 km/sec for Mercury flybys 1 and 2.

On April 6, 2009, the heliocentric transfer orbit concludes with Mercury orbit insertion into a 200-km by 12.8-hour, 80˚-inclination orbit. Choice of the nominal initial periherm altitude cost only 4 m/sec extra orbit correction ΔV compared with the Concept Study plan. The 1.617 km/sec initial MOI maneuver lasts nearly 31 minutes, includes about 115 m/sec ΔV for gravity loss, the cost to arrive at a higher (200 km vs. 125 km for the Concept Study) post-MOI periherm, and a periherm latitude rotation from 62.3˚ to the required 60.0˚N. An additional 12 m/sec clean-up ΔV, scheduled three to four periherms after MOI, provides a highly accurate (± 1 minute) orbit period adjustment to 12.0 hours. This two-burn MOI strategy effectively eliminates the probability of wasting propellant to *increase* the orbit period to 12 hours in the event of an overextend-ed burn duration. Figure 4, unchanged from the Concept Study, shows the 12-hour prima-ry science orbit for MESSENGER's 1-year Mercury orbital phase. Both parts of the MOI occur using a brief propellant-settling burn with four 22-N thrusters followed by a 660-N bipropellant thruster firing oriented near the instantaneous spacecraft velocity direction. The inertial spacecraft orbit orientation relative to the Sun provides a cooler (by ~ 4˚C) maximum spacecraft bus internal temperature during low-altitude sub-solar fly over.

The orbit correction maneuver strategy remains the same as for the Concept Study. However, the change in initial periherm altitude from 125 km to 200 km changes expected periherm altitudes prior to periherm lowering ΔVs (see Table 2) to 482, 461, and 433 km, with a 457-km periherm altitude one year after MOI. In addition, no Earth occultation occurs for nearly three months after MOI (compared with Earth occultation for the first 38 orbits after the Concept Study MOI).

Table 2
NEW BASELINE/BACKUP ORBIT PHASE DETERMINISTIC ΔV

Date	Event	ΔV (m/sec)
04 Jul 2009	**Periherm lower**	**26.0**
04 Jul 2009	**Period adjust**	**4.0**
01 Oct 2009	**Periherm lower**	**24.2**
01 Oct 2009	**Period adjust**	**3.8**
27 Dec 2009	**Periherm lower**	**21.7**
28 Dec 2009	**Period adjust**	**3.4**
Total		**83.1**

New Backup Trajectory

The new May 2004 backup launch opportunity begins within two months after the new baseline launch window. Launch window analysis for the new baseline and backup trajectories by McAdams demonstrated equal maximum spacecraft dry mass for both the 20-day baseline launch window and a 2-day backup launch window. MESSENGER mission management's choice of a 12-day minimum backup launch window duration requires a reduction in the maximum spacecraft dry mass offered by the 20-day baseline and 2-day backup launch windows. Preliminary trajectory optimization for launch on May 23, 2004, placed lift off at 5:29 A.M. UTC, with stage III ignition (start of injection onto the heliocentric transfer trajectory) only 31 minutes later. The spacecraft receives full sunlight 5-10 minutes before injection. Preliminary timing of lift off and injection timing came from mission analysts at NASA Kennedy Space Center.

The heliocentric transfer trajectory (see Figure 7) requires three Venus flybys (all with altitude > 1 Venus radius), two Mercury flybys, and two deterministic ΔVs, a total of seven operations-intensive events, prior to MOI. After launch the spacecraft follows a type II Earth-to-Venus transfer. The longest solar conjunction passage lasts 32 days starting two days prior to the third Venus flyby. Fortunately, Venus is mostly sunlit on approach (47˚ phase angle), minimum altitude is a distant 2.4 Venus radii, and the spacecraft will orbit the Sun 2.4 times before the next major mission event (first Mercury flyby). With no ability to correct targeting errors for at least five weeks after the third Venus flyby, accurate pre-encounter targeting and multiple heliocentric revolutions before the next flyby (Mercury flyby 1) make the risk of the third Venus flyby manageable.

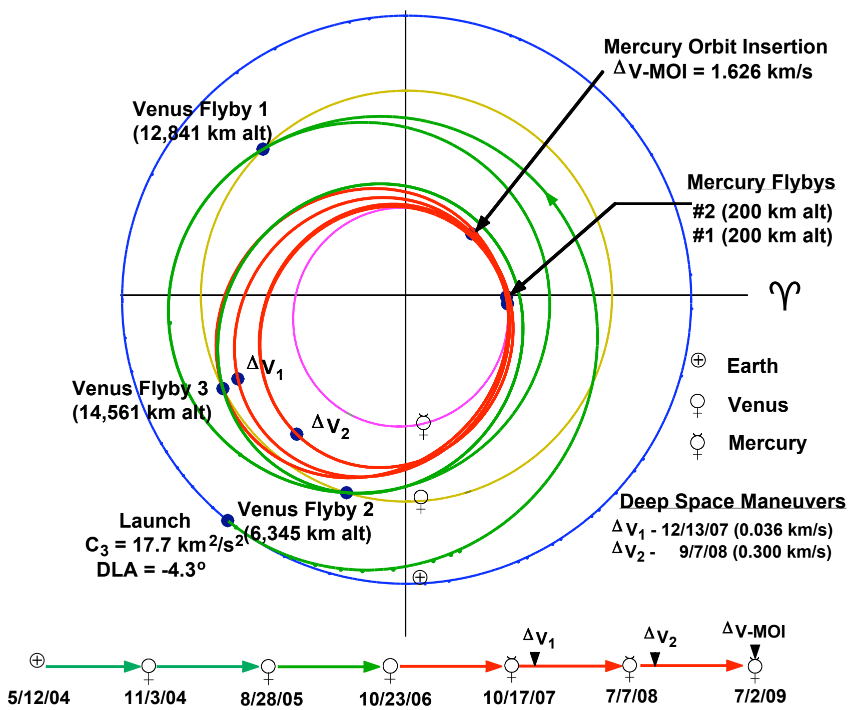

Figure 7 North Ecliptic Pole View of New Backup Trajectory

All three unpowered Venus flybys help shape the spacecraft trajectory closer to Mercury's orbit. The first Venus flyby, with medium-range close approach altitude just over 2.1 Venus radii, reduces the spacecraft orbit's perihelion and aphelion and increases orbit inclination slightly. An approach phase angle of 8˚ to 6˚ indicates a spacecraft view of a mostly sunlit Venus, which leads to a post-flyby solar eclipse of about one hour. Spacecraft subsystem engineers are designing tests to ensure that Sun-facing solar panel surfaces handle the corresponding cooling/heating cycle. About 1.3 Venus orbits later and 1.5 spacecraft orbits later the spacecraft returns for a 1.0 Venus-radius altitude second Venus flyby, a flyby that moves spacecraft aphelion near Venus' orbit and perihelion near Mercury's perihelion. An approach phase angle of 164˚ indicates a spacecraft view of a thin sunlit crescent Venus. About 1.9 Venus orbits and 2.9 spacecraft orbits later, the third Venus flyby occurs (see last paragraph for details). After the third Venus flyby, the spacecraft begins a type V zero-ΔV Venus-to-Mercury transfer.

Two unpowered 200-km minimum altitude Mercury flybys followed by post-aphelion maneuvers complete spacecraft orbit apsidal rotation and lower aphelion enough to enable Mercury orbit insertion in early July 2009. Each ΔV occurs during the first post-flyby orbit according to the previously stated programmatic guideline. The Mercury gravity assists and subsequent ΔVs produce successive spacecraft:Mercury orbital resonance of about 2:3 and 3:4. The Mercury flybys have approach phase angles of about 112˚ and 129˚ on October 17, 2007, and July 7, 2008. Once again the spacecraft can view opposite sides of the never-before-imaged (bright surface) hemisphere of Mercury soon after minimum altitude passage. Also note that post-flyby orbit reconstruction will be partially incomplete for the first Mercury flyby due to the spacecraft's loss of communications during solar conjunction 5-6 days after the flyby. The spacecraft hyperbolic excess velocity relative to Mercury decreases from 5.77 km/sec to 5.31 km/sec to 3.40 km/sec at Mercury flybys 1, 2, and MOI, respectively. On July 2, 2009 the heliocentric transfer orbit ends with Mercury orbit insertion into a 200-km by 12.8-hour, 80˚-inclination orbit. The MOI clean-up and orbit correction strategy match the baseline.

COMPARING CONCEPT STUDY AND NEW MISSION PLANS

A detailed comparison of baseline and backup trajectories between Concept Study and new trajectories favors the new trajectories for nearly every point. This comparison is presented in Table 3 in priority order, highest to lowest. Major reduction in post-launch ΔV and a slight decrease in maximum launch energy resulted in a mission-enabling additional 60 kg spacecraft payload mass.

CONCLUSIONS

The latest MESSENGER baseline and backup launch opportunities represent a mission-enabling trajectory optimization breakthrough. Without the new mission plan, the nominal science payload and numerous risk-mitigating, hardware redundancy and backup systems would likely not be in place. Mission confirmation by NASA would have been much more difficult to achieve.

Table 3
COMPARISON OF CONCEPT STUDY AND NEW MISSION PLANS

Evaluation Point	Superior Option	Noteworthy Differences, Reason for Superiority
Post-launch ΔV/Launch energy	N	N: 400 m/sec less ΔV, 60 kg extra payload mass
Total flight time/Cost	N	N baseline (backup): > 6 (2) months less than CS
Number of operations-intensive events	N	N: 6 baseline, 7 backup; CS: 9 baseline, 9 backup
Timing of initial Mercury science data	N	N: baseline 7/21/07, backup 10/17/07; CS: 1/15/08
Timing of Mercury orbit insertion	N	N: baseline 6 months earlier, backup 3 months earlier than CS
First Venus flyby conditions	N	CS: flyby is 1/2 week into 2.5-week solar conjunction
Timing of heliocentric ΔV maneuvers	N	CS: ΔV_2 is 5 days before 5-week solar conjunction
Min. statistical ΔV (navigation & margin)	N	N: 317 m/sec baseline, 245 m/sec backup; CS: 213 m/sec
Mercury orbit insertion Earth observability	N	N baseline: 100%; CS baseline: first 75% only
Timing of initial science return (Venus)	N	N: 6/24/04 baseline, 11/3/04 backup; CS: 10/25/06
Baseline launch window duration	N	N: 20 days; CS: 15 days
Maximum s/c-Sun range (solar panel size)	N	N: 1.01 AU; CS: 1.11 AU
Earth flyby - extra requirements & reviews	N	N: none; CS: one Earth flyby
Second Mercury flyby conditions	CS	N: approach phase 5^o > CS, solar conjunction 3-6 days later
Development, Integration, Test schedule	CS	CS: 13 days greater schedule margin for baseline launch
Maximum spacecraft bus temperature	N	N: 4^o C cooler than CS for Mercury orbit limiting case
Backup launch window	N	N: >2 months closer to baseline than CS, duration 2 days less
First Mercury flyby conditions	N	CS: Earth occultation 7 times longer than N's Earth occultation
Second Venus flyby conditions	N	CS: 300-km minimum altitude; N: >4,200-km minimum altitude
Early Mercury orbit phase Earth tracking	N	CS: Earth occultation 1st 38 orbits; N: no occult. for ~ 3 months

Note: N = New, CS = Concept Study

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