Maximizing MESSENGER's Science Return with Technologies and Innovation

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ABSTRACT

The great success of the MErcury Surface, Space ENvironment, GEochemistry, and Ranging (MESSENGER) mission was possible only through a series of technological advances and a number of innovative uses of existing technologies. A Mercury orbital mission had been studied for 30 years before MESSENGER and was believed to require a multi-billion-dollar effort. However, the innovations developed by the MESSENGER team enabled the mission to be accomplished within NASA's low-cost planetary program, known as Discovery. Not only were key enabling developments put into practice before launch, but also a number of innovations were implemented after launch, and in-depth planning for critical events greatly enhanced the scientific return from the mission. These postlaunch improvements also simplified mission operations and saved enough propellant to permit extending the mission beyond the originally planned 1 Earth year at Mercury. The communications practices were optimized to ensure that even more science data could be returned to Earth, and the orbital period was lowered to give 50% more low-altitude coverage. When part of the Gamma-Ray Spectrometer had exceeded its useful life, the rest of the instrument was repurposed to give new insight into the rapidly varying magnetosphere.

INTRODUCTION

A Mercury orbital mission is very challenging. One concern is the difficulty of getting there with sufficiently low speed relative to the planet to enable capture into orbit about Mercury with the available onboard propulsion. Another is that once the spacecraft is in orbit, the Mercury thermal environment is exceedingly harsh. Mercury's orbit has a perihelion (point of closest distance to the Sun) of ~0.3 AU from the Sun. The solar intensity there is ~11 times greater than at Earth. Spacecraft surfaces can reach temperatures >300°C from solar illumination. However, the most severe thermal environment occurs when the spacecraft is between the planet and the Sun. The equatorial subsolar temperature on Mercury's surface is ~450°C, so the spacecraft has to cope with the extreme heating from the Sun on one face and strong infrared heating from the planetary surface on its opposite face. These peak heating intervals sandwiched between Mercury and the Sun can be adjacent to long eclipse periods that expose the spacecraft to severe cold and strain its battery reserves.

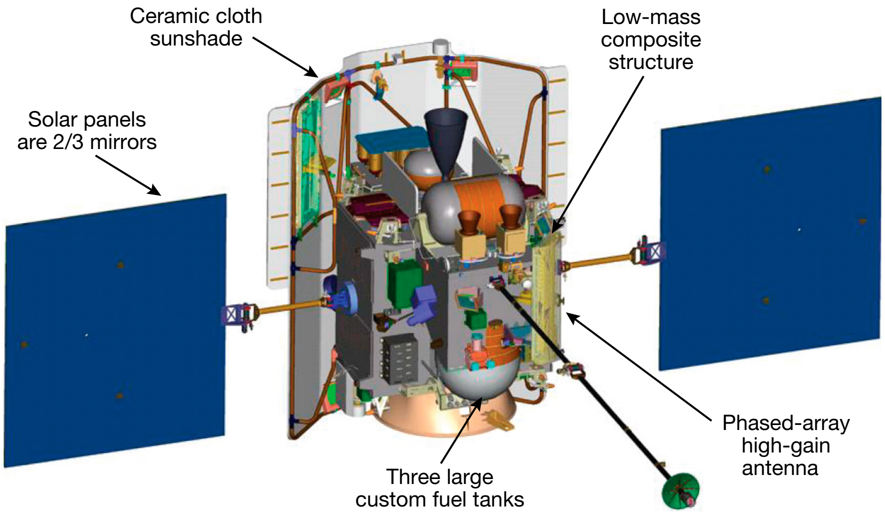

Figure 1. Key enabling technologies for MESSENGER.

MISSION-ENABLING TECHNOLOGIES

The MErcury Surface, Space ENvironment, GEochemistry, and Ranging (MESSENGER) mission relied on a few key technologies to enable it to get into orbit about Mercury and function in that extreme thermal environment.[1] Most of the technological innovations were related to the high thermal input in orbit, while others followed from the need for a great deal of onboard propulsion and the challenge of communicating back to Earth over a strongly variable distance from a position often close in the sky to the Sun when viewed from Earth (Fig. 1). A few of these enabling technologies are described briefly below, and more detailed descriptions can be found in the special MESSENGER volume of *Space Science Reviews* (Volume 131, 2007). The post-launch innovations are described here in greater detail.

Sunshade and Spacecraft Thermal Design

At Mercury's perihelion, the sunward face of MESSENGER receives >14 kWm^{-2} of solar radiation, which would quickly overheat and char any ordinary spacecraft. However, the MESSENGER design team wanted to use "ordinary" space components and electronics designed to operate in the temperature range 0°C to 30°C. The solution was a sunshade covered with a ceramic cloth that could withstand >1000°C on its sunward face. The sunshade was very effective. Even when its sunward face could be well above 300°C, the spacecraft behind it remained near room temperature.[2]

Because the sunshade must face the Sun at all times and its direction relative to the hot planet continually changes over the full range of azimuths and elevations, there was no single place to mount a radiator to expel excess spacecraft heat and keep the spacecraft body near room temperature. Radiator panels were placed on both sides of the spacecraft perpendicular to the spacecraft–Sun line, and they were connected to the interior of the spacecraft by diode heat pipes. So when the radiator on one side was facing the hot planet, that radiator effectively disconnected while the radiator on the opposite side was facing deep space and was fully functional.

High-Temperature Solar Panels

Spacecraft solar panels generally operate in the temperature range 0°C to 100°C. The efficiency of the solar cells rapidly diminishes at higher temperatures. More importantly, the stack-up of substrate panels, the adhesive to mount the cells, the cells themselves, the clear adhesive above them, and, finally, the protective cover glass must have sufficient robustness to survive the wide and rapid temperature extremes of being very hot in full sunlight and suddenly going into the extreme cold of eclipse shadows. This requirement is even more extreme for a Mercury mission, where the panel temperature could exceed 300°C when fully exposed to the Sun at 0.3 AU. This temperature would destroy the solar panels of other spacecraft. MESSENGER addressed the problem by surrounding each solar cell with optical solar reflectors that radiate much of the incident solar energy back into space to keep the peak temperature of the panels at acceptable limits.[3] A unique selection of materials in the layers of the completed panel was also required in order to survive the thermal stresses of extreme temperature swings.

The solar panels were sized for the most extreme case, which for MESSENGER was at 1 AU (or distance between the Sun and the Earth), right after launch. The power generated increased as the spacecraft moved sunward from Earth's orbit. To help keep the requirement for electrical power low near 1 AU, the spacecraft was rotated so that the Sun shone on the rear of the spacecraft (i.e., the side without a sunshade), to warm the spacecraft and reduce the need for power to heat it. This attitude at large solar distance allowed the total panel size to be minimized. More than adequate power was generated by the solar panels at 0.3 AU. Figure 2 illustrates how the panels could be tilted to point obliquely toward the Sun and reduce the thermal input to the panels.

Communications System and Its Phased-Array Antenna

Normally, deep-space missions use a dish antenna to maximize the rate of data transmission back to Earth.

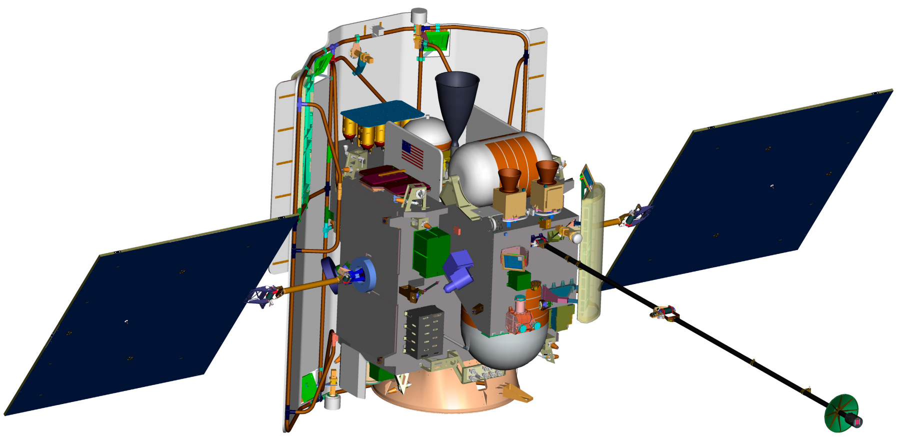

Figure 2. The solar panels on MESSENGER could be tilted inward of 0.7 AU from the Sun to reduce the thermal input while still producing sufficient power for the spacecraft.

However, the orbits of Mercury and Earth are asynchronous. So the direction back to Earth from Mercury covers all azimuths. A dish would need to be articulated, but doing so would regularly expose the dish, its pointing mechanism with its lubricants, and the cables or waveguides and feed materials to full solar heating. The dish might also reflect significant heat onto the spacecraft. To avoid these difficulties and the expense of developing a high-temperature pointing system, a more robust solution was developed.

Electronically Steerable Phased-Array Antenna

The Johns Hopkins University Applied Physics Laboratory (APL) had developed a simple slotted waveguide antenna for a missile system. The antenna consisted of a rectangular aluminum waveguide tube with some slots cut into one side. It was very rugged and could withstand temperatures >300°C. For MESSENGER, several waveguides were set side by side and fed with phase offsets. By adjusting the relative phases of the antenna "sticks," the radiation pattern formed a beam. Making those phases programmable transformed the set of waveguides into an electronically steered, phased-array antenna (PAA) that could be steered to cover 120° in azimuth (Fig. 3). One PAA on the front of the spacecraft and one on the back, along with the ability to rotate the spacecraft about the spacecraft–Sun line, enabled coverage of the entire 360° azimuth range. An additional medium-gain fan-beam antenna ensured good communications during emergency situations.[4]

One concern with slotted waveguide antennas is their linear polarization. The NASA Deep Space Network (DSN) antennas are circularly polarized, so connecting to a linearly polarized spacecraft antenna would have the effect of reducing the data rate for the given amount of spacecraft transmitted power. APL antenna engineers studied options to convert the polarization of the slotted waveguide antenna from linear to circular and devised a modification that placed small parasitic radiator elements near each slot that convert the polarization of the transmitted waves to circular. With this modification, the spacecraft downlink data rate was doubled.

Trajectory Design

Orbital Mechanics

Arriving at Mercury with a sufficiently low relative speed that would enable the spacecraft to be captured into orbit about Mercury would require a velocity change of ~16 km s^{-1}, many times what is possible for an onboard chemical propulsion system. The key to keeping the onboard propulsion requirements within an achievable range was using multiple planetary flybys to successively lose orbital energy at each encounter. The MESSENGER trajectory used a total of six planetary flybys on its interplanetary cruise. There were one Earth, two Venus, and three Mercury flybys, with maneuvers between encounters, to lose enough speed relative to Mercury that the spacecraft could burn propellant to become captured into an orbit about Mercury. This process took ~6.6 years

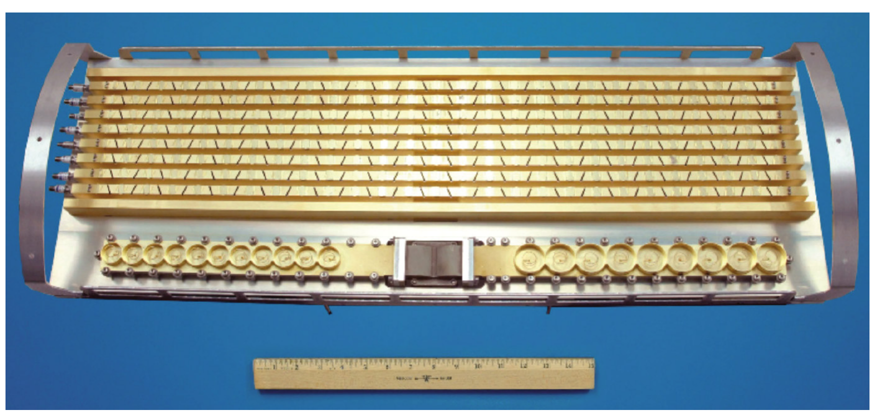

Figure 3. MESSENGER phased-array antenna. The upper portion shows the eight slotted waveguides. The lower portion is the medium-gain fan-beam antenna for emergency operations.

and was the most complex cruise phase of any planetary mission attempted until that time.[5]

The initial orbit at Mercury was near polar and highly eccentric, with a 12-h period. It had a periapsis, or point closest to the planet, that varied from 200 to 500 km and a ~15,000-km apoapsis, or point farthest from the planet. The large apoapsis provided several hours of cooldown time away from the hot planet between periapsis passes to help keep onboard temperatures under control. The highly eccentric orbit also meant that MESSENGER had to concentrate its high-resolution imaging and other periapsis studies on a single hemisphere. For MESSENGER, this was the northern hemisphere.

Propulsion System

Even with six planetary flybys, MESSENGER needed >2200 m s^{-1} of onboard propulsion. The spacecraft used a dual-mode system with hydrazine fuel and nitrogen tetroxide oxidizer to supply a large bipropellant engine for major velocity changes and several smaller monopropellant hydrazine thrusters for attitude control and smaller maneuvers. The high onboard velocity change capability meant that the spacecraft mass was >50% propellant at liftoff, leaving little remaining capacity for all of the spacecraft subsystems and components. MESSENGER was therefore a very mass-constrained design, and it could not support the high mass of any of the then-existing propellant tank designs. Extremely lightweight, customized titanium tanks were developed specifically for this mission. Each of the three tanks weighed only ~9 kg but could hold ~270 kg of propellant.[6]

Very-Low-Mass Design

The high propulsion requirement and the need to fit on the relatively small launch vehicles available to the Discovery Program meant that the rest of the spacecraft components had to be very-low-mass designs. In addition, the strong thermal gradients on the spacecraft body as it went from the intense heat of simultaneous full Sun on one face and the hot planet on the other to the cold of eclipses behind the planet could distort the spacecraft structure and destroy the precise pointing control and knowledge needed by several of the science instruments. To meet both the challenge of low structural mass while minimizing thermal distortions, the MESSENGER primary structure was constructed from carbon fiber composite materials specially modified to withstand higher temperatures.

Payload Technologies

In addition to the innovations that enabled the spacecraft to reach and survive in orbit about Mercury, there were several unique instrument developments that enabled the collection of key science data. Most of these innovations were required to overcome the extreme thermal environment that the instruments experienced in orbit at Mercury.

Gamma-Ray Spectrometer

The Gamma-Ray Spectrometer was one of the sensors on MESSENGER for determining the elemental composition of the surface of Mercury. It measured gamma rays emitted by the elements in the planetary surface when they were excited by nuclear collisions from the cosmic rays that continually bombard the planet. An unwanted background to gamma rays being measured from the planet is contributed by gamma rays locally generated in the spacecraft by cosmic rays. To resolve the planetary gamma rays in the high-background environment, MESSENGER relied on the very high spectral resolution of a high-purity germanium detector. However, these detectors must operate at cryogenic temperatures. MESSENGER had the first actively cryocooled gamma-ray spectrometer on an interplanetary space mission. Cooling the detector to temperatures of ~80 K while the spacecraft's sunshade was exposed to temperatures of ~600 K was quite challenging. The spacecraft was designed to maintain the sensor in extreme thermal isolation, and a small, lightweight, tactical cryocooler adapted from military missiles kept the instrument mass in check.[7] The cryocooler was a limited-life item with an expected lifetime of 8,000–12,000 h.

Mercury Dual Imaging System

Charge-coupled device image sensors need to be cool while operating to maintain their electronic noise at a low level and produce clean images. Imaging the dayside of Mercury, which can reach 450°C, is not readily compatible with a charge-coupled device camera that works best in the temperature range –10°C to –40°C. The innovative solution for MESSENGER was to incorporate an extensive passive thermal control system that used reservoirs of phase-change material in thermal contact with the focal plane detectors. The thermal reservoirs would cool down during the 8 h of each orbit when the spacecraft was far from the planet and could radiate to cold space. These phase-change reservoirs were connected to high-thermal-heat-capacity radiators by diode heat pipes, which effectively disconnected when the radiators faced the hot planet. These elements all worked together to keep the charge-coupled device image sensor in its desired temperature range.[8]

Magnetometer and Mercury Laser Altimeter

Mariner 10, the only other mission to have visited Mercury—it flew by the planet three times in 1974 to 1975—made groundbreaking observations, but the MESSENGER Magnetometer was designed to resolve the remaining ambiguity of the internal planetary magnetic field. Mercury's magnetic field is much weaker than Earth's, so a very low spacecraft magnetic background was required. The MESSENGER Magnetometer was placed on a 3.6-m boom to help ensure the needed magnetically quiet local environment. The Magnetometer was the only boom-mounted instrument on the spacecraft. The boom extended out the anti-sunward face of the spacecraft and would be in shadow most of the time. However, at the extremes of allowable spacecraft off-pointing from the Sun, the instrument at the end of the boom would stick out beyond the sunshade shadow and be directly exposed to the Sun. Therefore, the Magnetometer was given its own local sunshade to prevent overheating.

The Mercury Laser Altimeter was designed to measure the distance to the surface of the planet with <30-cm resolution out to ranges of 1800 km. The long-range capability required a large receiving aperture for the reflected laser light. However, the traditional Cassegrain reflective telescope style of receiver might, at times, be partially illuminated by the hot planet and suffer sufficient transient thermal distortion to severely reduce its optical quality and temporarily destroy its long-range sensitivity. The innovative solution for Mercury Laser Altimeter was to break the receiver aperture into four separate refractive telescopes, each with a sapphire lens and a beryllium housing. The four signals were then combined into the receiving detector, providing a thermally robust laser receiver.[9]

X-Ray Spectrometer

The MESSENGER X-Ray Spectrometer was an X-ray fluorescence instrument to measure the atomic composition of the planetary surface. One part of the instrument was a solar monitor to measure the incident solar X-ray spectrum at Mercury. Gas proportional counters mounted to the body of the spacecraft measured the resultant X-ray atomic fluorescence lines emitted by planetary surface material. Proportional counters are relatively unaffected by modest thermal variations. The solar monitor, however, was a silicon PIN diode pointed directly at the Sun and required thermal protection. A beryllium X-ray-transparent window was placed in front of the solar monitor and reached temperatures of ~1000°C. With careful thermal design, the silicon PIN diode, which was only a few millimeters behind the beryllium window, operated between –10°C and –30°C.[10]

Summary of Prelaunch Innovations

MESSENGER was a unique planetary mission that comprehensively explored Mercury and did so as part of NASA's Discovery Program, a series of low-cost space missions. It had to endure an extreme thermal environment and other challenges. It was enabled by the key technologies of the thermal protection, power, communication, propulsion, and mechanical systems described briefly above. Several of the scientific instruments were also enabled by unique innovations to ensure that they could make high-quality observations of Mercury and fulfill the promise of understanding this neighboring planet in the inner solar system. However, even after launch, the MESSENGER engineering team found ways to improve the mission further.

POSTLAUNCH INNOVATIONS

Solar Sailing

As described by Bedini (this issue), MESSENGER's path to Mercury relied heavily on six planetary gravity assists prior to Mercury orbit insertion (MOI). Each successive gravity assist ensured that MESSENGER's trajectory remained on course. A substantial deviation at any of these flybys would likely have resulted in a mission failure, because the spacecraft did not carry sufficient reserve propellant to correct for a large error in the flyby targeting.

Initially, MESSENGER adopted the proven strategy of using propulsive maneuvers to ensure that planetary flybys were executed as planned. A series of small rocket burns were used to keep the spacecraft on the proper trajectory. These burns remove perturbations (largely due to errors in modeling small forces such as solar radiation pressure, or SRP) and errors (from uncertainty in the orbit solution) in the trajectory. Typically, as time passed, orbit knowledge would improve, and the propagation arc to the desired flyby would shrink; this knowledge helped refine the estimated flyby location and allowed propulsive maneuvers to remove much of the targeting error. The orbit knowledge, the execution of the maneuvers, and the prediction of future trajectory perturbations were never perfect, however, and these errors would lead to inaccuracies in the flyby location. These errors shrank as the time to the encounter dropped, but the penalty in spacecraft velocity change, or ΔV, for correcting these errors grew substantially. As a result, a series of contingency maneuvers were incorporated into the timeline so the team was ready to design and execute any that would be required. Typically, only one or two such maneuvers would be required, although conservative planning practices usually allowed for four opportunities. Once the flyby was completed, the orbit determination uncertainties were markedly reduced, and a final cleanup maneuver could be executed (as necessary) to complete any missing ΔV from an imprecise trajectory. Such a departure maneuver needed to be executed as close as possible to the flyby because temporal delays in the execution drive up the ΔV (and propellant) cost substantially.

This propulsive maneuver paradigm was used successfully for the first three of MESSENGER's six planetary flybys. Table 1 summarizes the number of propulsive

Table 1. Comparison of the first three propulsively controlled flybys and the solar-sailing-controlled fourth flyby

Flyby	Planetary Body	Approach Maneuvers (No.)	Departure Maneuvers (No.)	Total Flyby Penalty (m/s)
1	Earth	1.3	0.0	1.7
2	Venus	2.8	35.7	40.0
3	Venus	0.8	0.0	1.0
4	Mercury	0.9	0	2.4

maneuvers required and the approximate propellant cost to navigate each of these flybys.

The first flyby of Mercury also began with this propulsive approach, as an approach maneuver was executed ~1 month before the flyby. The initial approach maneuver left nontrivial flyby errors, which would have resulted in a 5 m s^{-1} cost to the mission if these errors were corrected with a departure maneuver after the flyby. It was substantially less costly to correct these errors with a maneuver before the flyby, and several team members advocated that this correction be made in advance of the encounter to conserve propellant, particularly given the propellant reserve consumption required at the second Venus flyby. However, the next propulsive correction maneuver opportunity was only 4 days before MESSENGER's first encounter with Mercury. Anomalous execution of such a maneuver could jeopardize the spacecraft's first opportunity to collect science observations of Mercury, so there was pressure to delay the maneuver until after the flyby. At that point, the team recognized that a simple adjustment to the solar-array orientation would change the force due to SRP enough to correct the bulk of the flyby errors without introducing a risk to the flyby science data collection by executing an additional maneuver. The successful demonstration of correcting flyby-targeting errors with SRP (solar sailing) prompted the team to refine the technique for the second and third Mercury flybys as well as for the approach to Mercury for the orbit-insertion maneuver. This paradigm shift eliminated all planned flyby targeting and post-flyby cleanup maneuvers, reducing the flyby cost and decreasing the workload on spacecraft operators. Further, by using solar sailing to correct errors at the flybys, the flyby accuracy was maintained and in some cases improved because solar sailing offers greater precision than the conventional targeting with small-scale trajectory-correction maneuvers.

With the techniques previously described, managing the angular momentum stored in the spacecraft attitude-control momentum wheels became a straightforward, albeit time-consuming, operational activity. However, when the control of the angular momentum was combined with the control of the trajectory, the problem became substantially more complicated. MESSENGER's mission constraints did not allow decoupling trajectory control from momentum control, so these problems had to be solved simultaneously, chiefly because of the inability to align the center of pressure of the spacecraft with the center of mass within the spacecraft attitude constraints.[11]

The MESSENGER team then faced the problem of minimizing the peak momentum buildup in the wheels while simultaneously minimizing the flyby arrival condition targeting error. The control authority to solve this problem was derived from the temporal history of the spacecraft attitude and the solar-array orientation, both of which were subject to direct constraints. By manipulating the spacecraft attitude and array orientation, the resultant SRP forces and torques could be steered to achieve the necessary objectives. The flyby targeting was developed in a coordinate frame attached to the flyby body[12] for convenience, which allows linearization of the targeting portion of the problem. Despite this simplification, the problem remained difficult to solve because of the nonlinearity of the angular momentum growth due to SRP and the minimax nature of the problem. Further, the dual objectives of the problem are somewhat disjointed because the momentum and trajectory objectives are expressed in different units and do not lend themselves to easy combination into one single objective. These objectives are also sometimes conflicting because decreases in the angular momentum may lead to increases in the targeting error and vice versa. Many techniques have been proposed to solve multi-objective parameter optimization problems of this type.[13] As is typical for such problems, in general there is no global optimal solution, and for MESSENGER, it was not necessary to pay the (usually high) computational cost to identify the complete set of possible solutions because the real objective was to satisfy the mission constraints, and many solutions exist that would meet this goal. For this reason, the objectives were combined into a weighted, scalar metric that determined an overall solution quality.

Choosing the relative weighting between the momentum and trajectory objectives was subjective and required some engineering judgment. Although the control of the trajectory was useful and the overall aim of the solution, ultimately the momentum management was a notably higher priority because if the momentum limits were violated, the spacecraft would autonomously execute a self-protecting propulsive momentum dump. These autonomous momentum dumps have severe penalties because they carry all of the risks of operating the propulsion system, they perturb the trajectory, and they result in an operating mode demotion of the spacecraft. These activities raise mission risk and consume propellant, which are contrary to the objectives of the solution. However, there was considerable flexibility in the momentum constraint, making this objective easier to achieve. So although minimizing the peak momentum is desirable, a more complete statement of the objective is to reduce the peak momentum to below a prescribed threshold while simultaneously minimizing the flyby targeting error. The weights of the two objectives were then tuned so that the momentum would remain below the desired limit and then the targeting would be more heavily weighted.

It was not possible to develop an attitude and solar-array orientation plan all the way out to the ensuing planetary encounter. This situation arose primarily because the modeling lacked sufficient accuracy to predict the momentum over long time periods. Furthermore, science and engineering activities were often not planned >5 weeks in advance, so these unplanned activities introduced perturbations to both the trajectory and momentum that had to be managed. As a result, the process for planning and implementing adjustments to the attitude was on a 4- to 5-week design cycle. The process would begin by taking the most recent orbit solution from the navigators and solving the above optimization problem over a 2- to 3-week interval, allowing for any planned science or engineering attitude activities during that time frame. Although solving the optimization problem took only a few hours, the process to generate and test the necessary command sequences required 7–10 days. Once the sequence was loaded to the spacecraft and executed, the ensuing orbit determination would trigger the process again. This cycle introduced substantial lead time (~5 weeks) to an ability to make adjustments to the trajectory. This process proved insufficient during a planetary approach when the situation was more dynamic. During the time period of a flyby approach, the feedback loop was shortened by reducing the duration of spacecraft command loads from 2–3 weeks to 1 week and by the elimination of any unplanned engineering and science activities. These changes helped reduce the design cycle to ~15–20 days, which allowed sufficient control of the trajectory.

Flyby Results

With the solar-sailing techniques, MESSENGER was able to maintain flyby accuracy with a reduction in mission risk. Table 2 compares the MESSENGER flyby accuracy for planetary encounters controlled by solar sailing with those controlled by propulsive maneuvers. The results show that controlling the trajectory by passive means resulted in flybys that were on par with, or better than, those using a conventional propulsive trajectory control, for relatively small-scale adjustments. So, while it could be argued that solar sailing improved the planetary flyby accuracy, the real benefit was the reduction in mission risk by eliminating the flyby approach and departure maneuvers. Not only did the sailing approach eliminate the cost and risk of planning and executing maneuvers, but it did so at a time when the programmatic risk of executing these maneuvers was high, as the flybys offered unique opportunities for science observations and instrument calibration.

The MOI approach did have higher errors than prior flybys that utilized the solar-sailing approach. Several complicating factors and a bit of bad luck caused this circumstance. First, the sailing problem was additionally constrained in arrival epoch (instead of simply the B-plane, or plane perpendicular to the inbound hyperbolic asymptote, intercept) because the MOI burn design was predicated on a specific epoch to ensure the correct Mercury-relative orbit. Second, as the time to MOI decreased, there was reluctance to make specific modifications because of the volatility of the B-plane solution. In addition, a superior solar conjunction occurred 1–2 weeks in advance of MOI, but its effect was mitigated largely by the addition of four tracking passes per week of delta-differential one-way ranging during this period. Nevertheless, convergence of the various fit arcs occurred too late to make a definitive determination of further solar-sailing adjustments before MOI. Fortunately, the effect of radial error, as well as error in time of arrival itself, in achieving the ideal B-plane target could be mitigated somewhat by shifting the start time of MOI execution. By shifting the execution time 5 s earlier, the targeted post-MOI orbit period could be more closely achieved with an acceptable increase in achievable periapsis altitude of only a few kilometers. After the effects of both B-plane delivery errors and MOI execution errors were reconstructed, the resultant spacecraft orbit achieved a 206.8-km altitude at the first post-MOI periapsis and an orbit period of ~43,195 s, determined from the time between the first and second post-MOI

Table 2. The accuracy and propulsive penalty of the solar-sailing flybys of Mercury were comparable to or better than the propulsively controlled flybys of Earth and Venus

Flyby	Approach Maneuver Cost (m s^{-1})	Departure Maneuver Cost (m s^{-1})	Total Flyby Penalty (m s^{-1})	B-Plane Target Miss Distance (km)	Periapsis Altitude Offset (km)
Earth	1.3	0.0	1.7	22.1	+1.0
Venus flyby 1	2.8	35.7	40.0	36.0	–52.8
Venus flyby 2	0.8	0.0	1.0	5.7	+1.4
Mercury flyby 1	0.9	0	2.4	10.4	+1.4
Mercury flyby 2	0	0	–0.7	2.6	–0.8
Mercury flyby 3	0	0	–0.5	3.5	–0.5
MOI approach	0	N/A	N/A	8.0	+6.0

periapses. These were ~6.8 km and ~261 s longer than planned, respectively, but well within MOI requirements.

Increased Data Throughput

The RF Subsystem

The RF telecommunications subsystem on MESSENGER had three major functional requirements: (*i*) spacecraft command capability, (*ii*) the highest-possible quality and quantity of spacecraft telemetry and science data return, and (*iii*) highly accurate Doppler and range tracking data to determine precisely the spacecraft velocity and position. The system operated in the X-band frequency range: 7.2 GHz for uplink from ground stations and 8.4 GHz for downlink from the spacecraft. Communications were accomplished via the 34-m and 70-m antennas of NASA's DSN stations in Goldstone, California; Madrid, Spain; and Canberra, Australia. Standard margin policies require that each link be designed with a 2-dB margin to protect against data loss caused by any short-term fades.

Because of the inner-planet trajectory of MESSENGER, the Earth could be in any azimuthal direction viewed from the spacecraft. This geometric constraint presented a significant RF design requirement in that high-gain coverage had to be achievable in all directions. The antenna configuration shown in Fig. 4 accomplished this requirement. Two diametrically opposite-facing phased-array antennas provided the high-gain downlink signal to meet the mission data requirements (boresight gain >28 dBic) and still survive the extreme thermal environment (operational from –100°C to +300°C). Each PAA was capable of electronically steering ±60° in the *X*–*Y* plane of the spacecraft coordinates from the directions indicated in Fig. 4. Spacecraft rotation about the *Y* axis in conjunction with the electronic scanning of the antenna beam provided the omnidirectional high-gain coverage.[4] Two fan-beam antennas provided medium-gain uplink and downlink capabilities, and four low-gain antennas completed MESSENGER's antenna suite. The novel APL-designed PAA is shown in Fig. 3.[14]

During a typical 8-h DSN tracking pass, the ground-station elevation angle varied from 5° to >60° as the spacecraft rose on the horizon, passed through its zenith, and subsequently set. This movement markedly changed the link capacity during each pass, because both the antenna gain and noise temperature improve at higher elevation angles, as shown in Fig. 5.[15] Typically, a single downlink data rate for the entire 8-h pass is determined by calculating the worst-case link capability (typically at the start or end of a pass, when the ground-station elevation angle is low) and applying a 3-dB link margin to that link condition; this rate would be used throughout

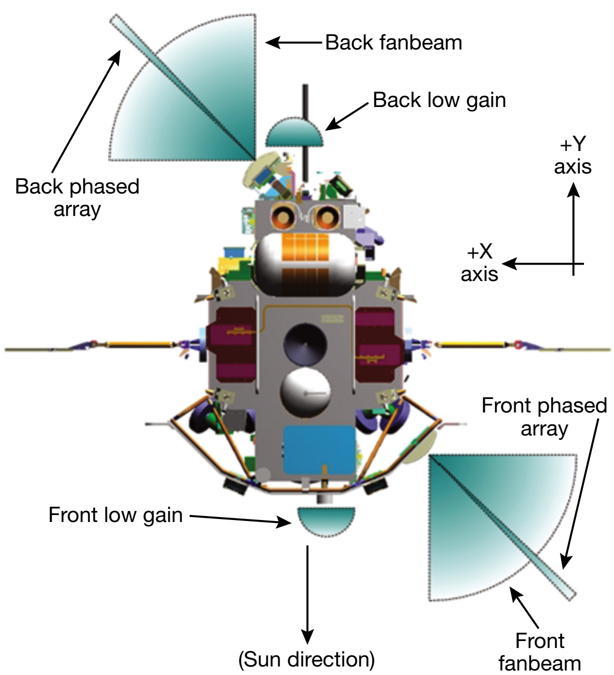

Figure 4. The MESSENGER spacecraft antenna suite. The PAA electronically scans ±60° from its boresight in the plane of the figure.

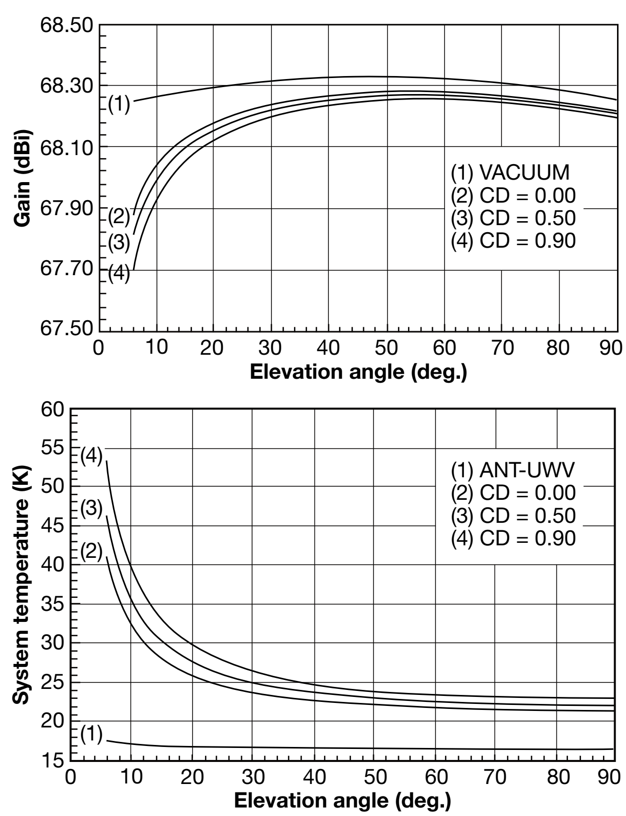

Figure 5. DSN ground station performance as a function of elevation angle. ANT-UWV is the antenna performance without weather and the CD traces are the cumulative distribution of effects of 0%, 50%, and 90% weather. (Reprinted with permission from Ref. 12, © 2015, NASA Jet Propulsion Laboratory.)

the pass. This procedure was part of the assumed concept of operations before launch of the spacecraft. This operational paradigm did not realize the full downlink capacity of a pass because the increased performance of the ground station at higher elevations was not realized and the 2-dB margin remained as untapped potential. To realize this extra capacity, an innovative low-risk method of data-rate stepping was developed; this method leveraged a new software protocol to ensure that no science data would be lost.

CFDP Description

Flight software contained a data-playback manager that automated the transmission of data files on the downlink using the CCSDS (Consultative Committee for Space Data Systems) File Downlink Protocol (CFDP). MESSENGER was the first spacecraft to implement this handshaking protocol, which was repeated on the Van Allen Probes mission and is planned for use on the Parker Solar Probe (formerly Solar Probe Plus) and Europa missions. With CFDP, each file is divided into small fragments known as protocol data units (PDUs). Should any individual PDU be dropped in transmission, the ground software produces negative acknowledgments that request a retransmission of the missed PDU, offering a more efficient way of retransmitting only the missing data as opposed to an entire file. Once a file is fully received on the ground, a "finished" directive from

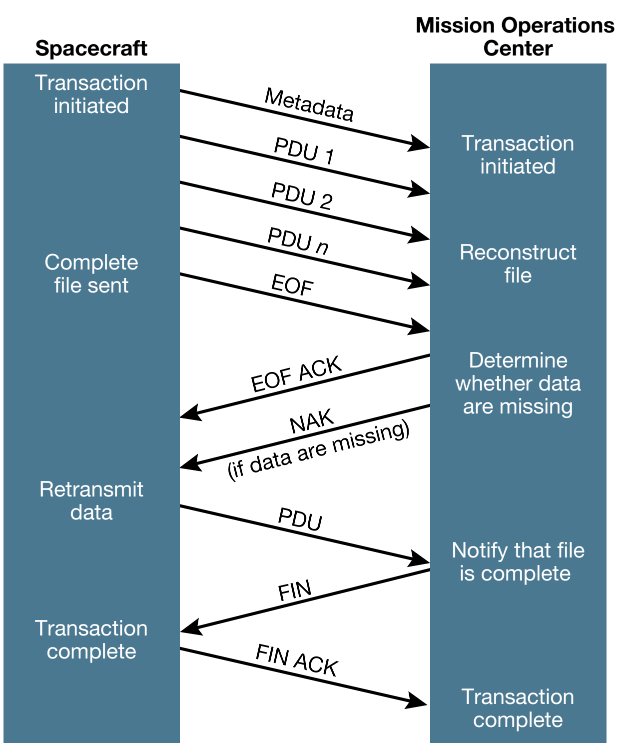

Figure 6. An example downlink file transfer with CFDP. ACK, acknowledgment; EOF, end of file; FIN, finished directive; NAK, negative acknowledgment.

the ground enables the flight software to delete the file from the spacecraft recorder. This handshake process is designed to span multiple tracking passes. This automation ensures that all data files from MESSENGER are recovered intact on the ground while minimizing needless retransmission of data that have already been received. Figure 6 illustrates this protocol.

Data-Rate Stepping with MESSENGER

To realize more data volume capacity, two changes were implemented: (*i*) changing data rates during a downlink pass to exploit the additional DSN ground station capability as the spacecraft's elevation increased and (*ii*) reducing the margin requirement to 0 dB. A challenge with performing mid-pass data-rate changes was ensuring that the configured bit rate of the spacecraft and the ground station were carefully coordinated because any bits "in transit" during such a data-rate change would be lost; sifting through and managing the data transactions and data storage was a complication for the mission operations team. The second change (reducing the downlink margin) increased the risk of data loss due to transient weather phenomena. CFDP enabled both of these actions by providing a "security blanket" to ensure that no downlink information was lost: any PDUs lost during a data-rate change or from intervals when the link margin dipped to <0 dB were retransmitted via the automated protocol, and the net data-rate gain far exceeded the lost bandwidth for retransmission errors.

MESSENGER 2013 Mission-Elapsed Time Clock Rollover Critical Event Activity

The MESSENGER spacecraft was launched 3 August 2004, with a 6.6-year cruise phase followed by orbit insertion and 1 Earth year of operations for the primary orbital mission phase. Time was maintained on the spacecraft by use of a mission-elapsed time (MET) that continuously incremented in integer seconds starting just before launch. The 28 bits allocated for the integer seconds portion of the MET were sufficient for the entire mission as originally planned and included margin for mission extension. However, a change in launch date resulted in an extended cruise phase. Even with the revised August launch date and longer cruise phase, the allocation was sufficient through the primary orbital mission, but it would roll over during an extended mission. This rollover would have occurred on 4 February 2013 at an MET of 268,435,455 s (2^{28}-1) unless intentionally rolled over in a controlled fashion before that time. Once a first mission extension was approved in 2012, 6 months of planning began in earnest with full participation from mission operations, systems engineering, project management, ground systems, science planning, and Science Operations Center (SOC) team members. During the kickoff meeting, ideas, options, and documented lessons learned over the years were collected, and the groundwork was laid for the practical considerations of both the flight and ground segments. This meeting was followed by a preliminary design review and then the critical design review, which demonstrated full readiness to support the activity on the flight and ground segments.

At the preliminary design review, it was agreed to manually force a clock rollover. There were several advantages to explicitly selecting the epoch, such as setting the post-rollover MET to a non-zero value (1000 s), to reduce the risk of unexpected untestable issues that might be triggered by a zero-value MET, and picking a date that minimized risk from a geometric and environmental factor perspective. For example, a time was selected that avoided bright objects in the star tracker field of view. Testing confirmed that an operational mode demotion to the first level of autonomous safe mode would occur if the rollover happened without valid star tracker data in the loop. The team also sought a time frame without any RF-signal Earth occultations or solar eclipses, and with benign thermal conditions with no solar-array off-pointing requirements and where protection from the worst planetary heat input, known as hot-pole keep-out protection, could be disabled. Such an interval would expedite any potential demotion recovery efforts. The team also preferred a date that was beyond the New Years' holiday but before the next superior solar conjunction and that afforded a second manual opportunity before the automatic rollover.

In choosing the rollover epoch, the team also wanted to avoid a Sun–probe–Earth angle of 90° that could affect communications in the event of a spacecraft mode demotion to the most severe level, which would put the spacecraft into a rotation mode called Earth-acquisition that sought to reestablish contact with Earth. Finally, the team preferred a Tuesday in order to take advantage of predefined science command blackouts that minimized interruption to science data collection and would be sufficiently far from a weekend should an anomaly occur. Amazingly, the team was able to find a date and time that accommodated all of these factors, and 20:30 UTC on Tuesday, 8 January 2013, was selected as the rollover epoch, with 29 January identified as the backup opportunity. The only real drawback was the fact that the Earth distance was relatively large at 1.4 AU, resulting in lower uplink and downlink rates and longer communication delays. The rollover activity was scheduled as a Level-3 DSN event, meaning that it was allocated additional communication briefings and real-time dedicated DSN engineering support.

Several flight configuration changes were identified to safely support the MET rollover, all of which were reviewed and tested in extensive ground-based simulations before implementation. Among them, an updated Terrestrial Dynamic Time-to-MET correlation parameter block was preloaded to the spacecraft's electrically erasable programmable read-only memory to guard against an unrelated processor reset, and it was sequenced to be committed into random access memory at the appropriate time; the latest spacecraft ephemeris was also preloaded and then activated after the rollover to ensure proper synchronization. Earth downlink pointing was extended in the sequence to be longer than a typical orbit, and the cadence of momentum desaturation dumps of the spacecraft reaction wheels was modified in the weeks leading up to the rollover to ensure an optimal control situation on the day of the rollover.

From the command and data handling subsystem, the solid-state recorder playback was halted early so that open transactions could be canceled before the rollover to ensure a clean state. The weeklong command sequence would have some commands set to execute before and some after the rollover, so time-tags to trigger beyond the rollover, written with small METs under the assumption of a successful reset to 1000 s, were disabled when preloaded. Those time-tags would be enabled only after confirmation of a successful rollover. That week's time-tag bias offset was also set to zero before the rollover and then reinstated afterward. All files were intentionally closed before and reopened after the rollover, resulting in a very small agreed-upon permanent gap in recorded data of just 16 min. Lastly, the redundant main processor MET was manually set to 1000 s on 4 January in advance of the primary activity, to protect against an unrelated processor swap condition, and then all relevant main processor parameters were finalized after the primary rollover in a two-step process.

The spacecraft's two fault-protection processors each implemented a countdown command-loss timer (CLT) used to autonomously demote the spacecraft to Earth-acquisition mode should no commands properly arrive from the ground within a configurable period of days. The rules that handled CLT expiration were disabled shortly before the rollover to prevent a false trigger, and a one-time-use autonomy rule was introduced to detect the rollover with an MET < 5000 s premise. It executed a CLT refresh after the rollover and subsequently re-enabled the CLT rules. An independent safety-net rule was also put in place to re-enable the CLT functionality after the scheduled rollover time regardless of whether the rollover actually happened, so as not to leave the spacecraft in a vulnerable state. On the payload side, the states of three instruments were modified before the rollover as a precaution: the Magnetometer was powered off, the Energetic Particle and Plasma Spectrometer's Fast Imaging Plasma Spectrometer sensor high voltage was ramped down, and the Mercury Dual Imaging System camera pivot was commanded to its stowed position.

Several changes were required within the ground system as well, both at the Mission Operations Center and the SOC, to support a successful rollover and seamless operations. For example, a second partition was introduced into the spacecraft telemetry time-ordered archive to facilitate processing of data because MET alone was no longer uniquely sufficient to order data by collection time. It was decided that all data would be played back from the solid-state recorder with a temporarily modified priority scheme that drove all pre-rollover data to come down first. Any pre-rollover data that came down after the rollover were manually entered into the first partition by the ground systems lead engineer. Mission operations acquired additional DSN antenna time leading up to this time frame to minimize the backlog telemetry awaiting downlink prior to the rollover so as to reduce the amount of manual processing. In addition, a time-conversion ground tool was modified to intentionally default within the graphical user interface to the new second partition when entering METs. Any time-conversion requests for pre-rollover times would require the user to proactively change the "2/" partition prefix to a "1/" in order to conduct the desired pre-rollover conversion calculations.

Another significant change to the Mission Operations Center involved modification of downlinked file names with a "1" prefix uniquely denoting post-rollover data. The MESSENGER file-naming convention consisted of a string of 10 digits representing the file open MET followed by a priority directory level and letter. To illustrate this change, a hypothetical pre-rollover file named 0000xxxxx_3_H was renamed by the ground system after downlink to 1000xxxxx_3_H after rollover. The change was tested and ready to be swapped into the production system as part of the timeline of planned activities. Post-rollover DSN Intermediate Data Record file names were also similarly modified upon receipt. The hardware simulator was extensively used for ground testing to provide realistic test products, such as the new file names, to the SOC and other end users for their own end-to-end testing. Lastly, the operations clock kernel file was modified and introduced well in advance to include the use of a partition parameter column. On the day of the rollover, the automatic timekeeping was disabled, a record was manually inserted into the new operations clock kernel, and daily values were assigned. Automated processing was re-enabled 4 days later once the clock drift-rate stabilized. Note that several corresponding changes were implemented at the SOC as well. Examples included partitions in product labels for pipeline processing and numerous tool changes along with temporary suspension of data transfers, all of which ensured a smooth transition.

In summary, 6 months of intensive planning and reviews involving all team elements of the project contributed to the highly successful in-flight rollover transition of the MESSENGER MET clock with minimal interruption to science data collection and processing. The mission continued uninterrupted for >2 additional years of Mercury orbit operations after this critical event, returning unprecedented data from the innermost planet.

Orbit Transition from 12 to 8 h

After completion of MESSENGER's primary mission on 18 March 2012, the team executed a maneuver to transition from an 11.6-h orbit period to an 8.0-h orbit period to lower the apoapsis (orbit location most distant from Mercury) from a 15,000-km altitude of the highly eccentric 12-h orbit to a 10,000-km altitude. Early analysis showed that this orbit change would have to be executed through a maneuver in the time range of 3–5 weeks after MOI plus 1 year, at an estimated cost of 98 m/s ΔV. Later analysis resulted in the plan that was eventually executed using less ΔV: less than 5 weeks after MOI plus 1 year, a transfer from an 11.6-h orbit to an 8-h orbit (at 277 km × 10,310 km altitude) was executed using ~85 m/s during two orbit-correction maneuvers (OCMs).

The greatest advantage for the 8-h orbit option was a 33% better surface image resolution of Mercury's south-polar region and 50% more orbits, thereby enhancing the science return. The increased number of orbits would greatly benefit low-altitude science observations of the far northern hemisphere, a region with unexplained magnetic variations and numerous permanently shadowed craters that were suspected of harboring water ice. The smaller orbit slowed the rate of change of the periapsis latitude slightly, thereby delaying the MESSENGER spacecraft's ultimate fate—high-speed impact with Mercury's surface. The strategy chosen for transitioning to the 8-h orbit offered an opportunity to more efficiently utilize the final accessible oxidizer in a single (versus multiple) bipropellant maneuver, yet another factor that further delayed Mercury impact. A lesser advantage was that, if further mission extensions were granted, in this shorter orbit the spacecraft would experience shorter eclipse durations, preventing dangerous battery depth of discharge during long eclipse seasons. The only known disadvantage of operating instead in an 8-h orbit was the reduction in time for spacecraft cooling (i.e., a shorter thermal recovery after a low-altitude subsolar crossing). This change primarily affected the Mercury Dual Imaging System wax phase-change thermal reservoir, which helped keep the sensor cool. The thermal capacity of the wax kept the sensor below the wax-melting temperature at times when sunlight reflected off Mercury and transferred heat into the imager. The phase-change reservoir relied on the long time away from periapsis to fully cool the wax before the next orbit.

Analysis of options for the best strategy to lower the spacecraft's orbit period to 8 h resulted in a plan with two OCMs, each centered at periapsis, where the spacecraft is closest to Mercury. To comply with Sun-relative pointing constraints, the two OCMs would need to be no more than 4.5 days apart. OCM-7 was designed to extract the maximum usable amount of oxidizer remaining in the oxidizer tank while lowering the orbit period to ~9.1 h. With significant uncertainty in the amount of usable oxidizer remaining, the flow of oxidizer was allowed to continue for as long as 29 s, with earlier termination of the bipropellant portion of OCM-7 occurring with detection of a dramatic drop in thrust via a custom autonomy rule. After detection of oxidizer depletion via sharply lower thrust or after reaching the 29-s bipropellant segment limit, the remainder of the maneuver would complete at ~85% lower thrust using the four largest monopropellant thrusters. This variable-duration, high-thrust plan for OCM-7 created a potential for >100% fluctuation in OCM-7 thrust duration. Relative to planning a conservative 14-s-duration bipropellant segment that was certain not to deplete oxidizer, the 29-s-duration bipropellant segment increased the remaining ΔV by ~12 m/s—enough to enable the final 6-week low-altitude hover mission extension (see below).

The second maneuver, OCM-8, was performed at periapsis 4.2 days after OCM-7, to complement OCM-7 in lowering the orbit period to 8 h. A plan to update the monopropellant source for OCM-8 was canceled because of the accurate implementation of OCM-7 and accurate orbit prediction between OCM-7 and OCM-8. Figure 7 depicts the placement of OCM-7 and OCM-8 along with the initial, post-OCM-7, and final orbit sizes and orientations.

A number of important lessons were learned from the process of lowering the spacecraft's orbit period to 8 h. First, the willingness to adapt from a less-complex, longer mixed propulsive mode OCM to two OCMs that fully utilized the available oxidizer yielded benefits realized in later mission extensions. Second, the willingness to depart from the nominal plan of continuing in an orbit with a period near 12 h yielded significant science breakthroughs that otherwise would have been impossible. The successful implementation of two maneuvers only 4.2 days apart, along with an option to update the second maneuver, provided team confidence in the ability to plan maneuvers close together—among the operations capabilities enabling the low-altitude hover campaign in March and April 2015. The team's careful modification of a number of proven procedures opened a new realm of possibilities for enhanced exploration of Mercury.

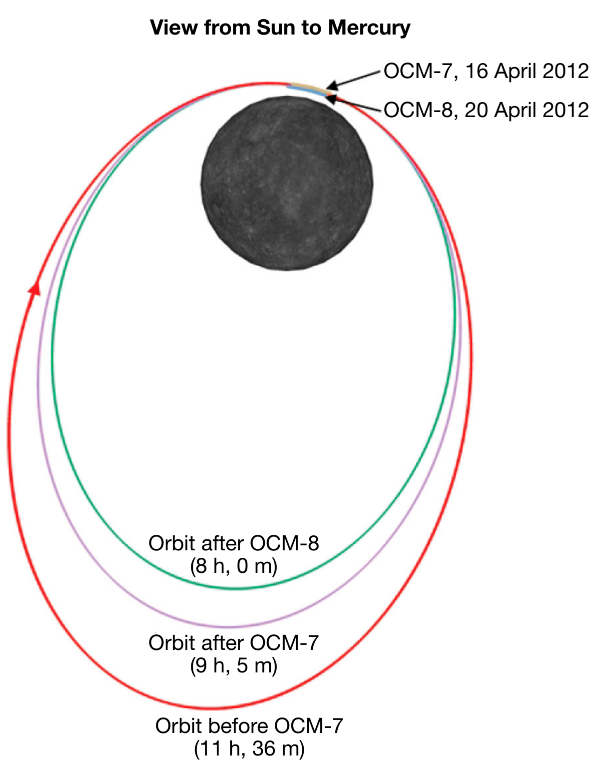

Figure 7. Maneuvers OCM-7 and OCM-8 lowered MESSENGER's orbital period from 12 to 8 h.

Pressurant as Propellant

Successful execution of OCM-7 and OCM-8 in April 2012 staved off planetary impact until August 2014, enabling a second extended mission, which commenced on 18 March 2013.[16] Subsequent OCMs further delayed impact through MESSENGER's final mission extension, which began on 18 March 2015, the same day that OCM-13 was executed. After OCM-13, only 1.4 kg of usable hydrazine (N_2H_4) remained in MESSENGER's auxiliary propellant tank (AUX), enough to remain aloft through 13 April 2015. To extend the low-altitude hover campaign to the end of April, the final four OCMs were performed by repurposing the propulsion system's pressurant as a propellant: 1.3 kg of gaseous helium (GHe) pressurant. No previous Earth-orbiting or interplanetary spacecraft had ever used pressurant in this manner.

The MESSENGER Propulsion System (MPS) was a lightweight, pressurized, dual-mode bipropellant system designed and built by Aerojet Rocketdyne. A more detailed description of the propulsion system was published previously.[17] Figure 8 shows the MPS schematic.

The MPS was equipped with 16 monopropellant hydrazine thrusters. Figure 9 identifies the thrusters and exhaust vectors. The C thrusters were Aerojet Rocketdyne 22-N (5 lb_f) MR-106Es, and the A, B, S, and P thrusters were Aerojet Rocketdyne 4.4-N (1 lb_f) MR-111Cs. To provide redundant three-axis attitude control, the A and B thrusters were arranged in double-canted sets of four. The S thrusters provided ΔV in the sunward direction, and the P thrusters provided ΔV in the anti-sunward direction. Aerojet Rocketdyne did not possess any analytical or test data on performance of the thrusters when used as cold gas engines, and because they were designed for liquid flow, orifices within the engines choked flow and limited GHe throughput.

Propellant was stored in three identical main tanks—two fuel (FT1/FT2) and one oxidizer—and the refillable AUX and GHe was stored in a single pressurant tank. AUX used an elastomeric diaphragm for positive propellant expulsion, and the propellant in the main tanks was expelled by using the helium pressurization system.

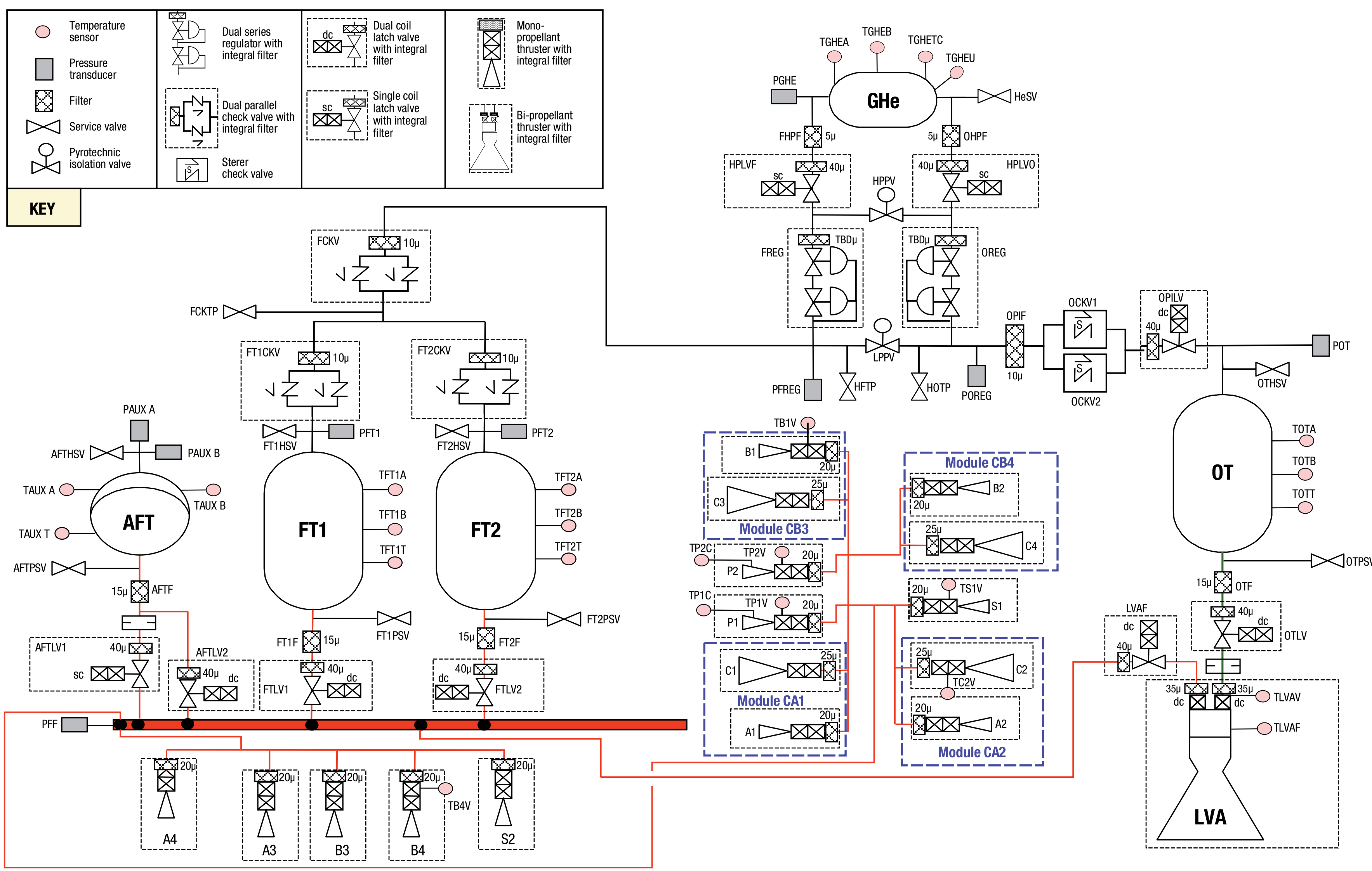

Figure 8. Schematic diagram of the MPS. Propellant valves on monoprop thrusters B1, B4, C2, S1, P1, and P2 were instrumented with thermocouples. Owing to their location on the spacecraft sunshade, thrusters P1 and P2 also baselined platinum resistance temperature detectors (RTDs) on their catalyst beds. AFT, auxiliary fuel tank; GHe, helium pressure tank; OT, oxidizer tank.

Remaining available GHe “propellant” was regulated at 280 psi maximum from the helium tank into FT1/FT2 and AUX, as all low-pressure latch valves were permanently opened. Helium consumption was computed via the ideal gas law (pressure–volume–temperature), taking into account unusable/trapped propellant vapor pressure and pressurant compressibility, by calculating tank loads before and after maneuvers, resulting in root-mean-square uncertainties between 5% and 15% depending on maneuver duration. Specific impulse for the engines using helium was calculated from the observed helium flow rates. All four helium maneuvers used the “4C” (C1/C2/C3/C4) thruster set for primary ΔV and the A and B thrusters for attitude control.

High-rate accelerometer data were collected during MESSENGER maneuvers in order to measure spacecraft acceleration and, ultimately, the ΔV imparted. All maneuvers of the low-altitude hover campaign were executed in closed-loop control such that maneuvers were terminated once the guidance and control software onboard indicated that imparted ΔV had reached a commanded value. On the ground, the high-rate accelerometer data were used to estimate thrust given an assumed spacecraft mass.

On 6 April 2015, MESSENGER’s OCM-15 was the first spacecraft maneuver ever to repurpose pressurant into a primary propellant source. OCM-15 commenced with a 225-s burn of the P1/P2 thruster pair penetrating MESSENGER’s heat shield using N_2H_4 from AUX. This was followed by 375 s of cold gas propulsion using pressurant GHe from FT1/FT2, which also pressurized AUX. The cold gas segment consumed 0.067 kg of GHe as calculated by the ideal gas law. Using GHe, the 4.4-N nominal MR-111C delivered an average of 3.3% of the equivalent N_2H_4 thrust predicted by Aerojet Rocketdyne’s nominal performance curves. Delivered specific impulse, calculated using measured thrust and GHe consumption, was 172 s, slightly under the theoretical GHe maximum of 179 s.[18]

A concern related to thruster health was whether the dual-seat thruster valves, powered without the benefit of cooling liquid hydrazine propellant, would heat above their nominal peak temperature of 150°C. Temperature data showed that flowing helium was able to cool the thruster valves, although not sufficiently to fully overcome ambient heating from the surface of MESSENGER’s sunshade.

OCM-15a took place on 8 April 2015 and consisted of a 303-s 4C cold gas “burn” off of FT1/FT2/AUX. Unlike during OCM-15, which had two smaller thrusters firing, the pressure regulator was not able to maintain sufficient flow rate to keep the GHe pressure constant for four comparatively large thrusters, so a steady decline in thrust was observed throughout the maneuver. OCM-15a consumed 0.494 kg of GHe. The 22-N nominal MR-106E delivered an average of 3.1% of the equivalent N_2H_4 thrust. Delivered specific impulse was 176 s.

As in OCM-15, flowing helium cooled the C thruster valves, although the higher flow rate for the MR-106Es was sufficient to overcome ambient heating.

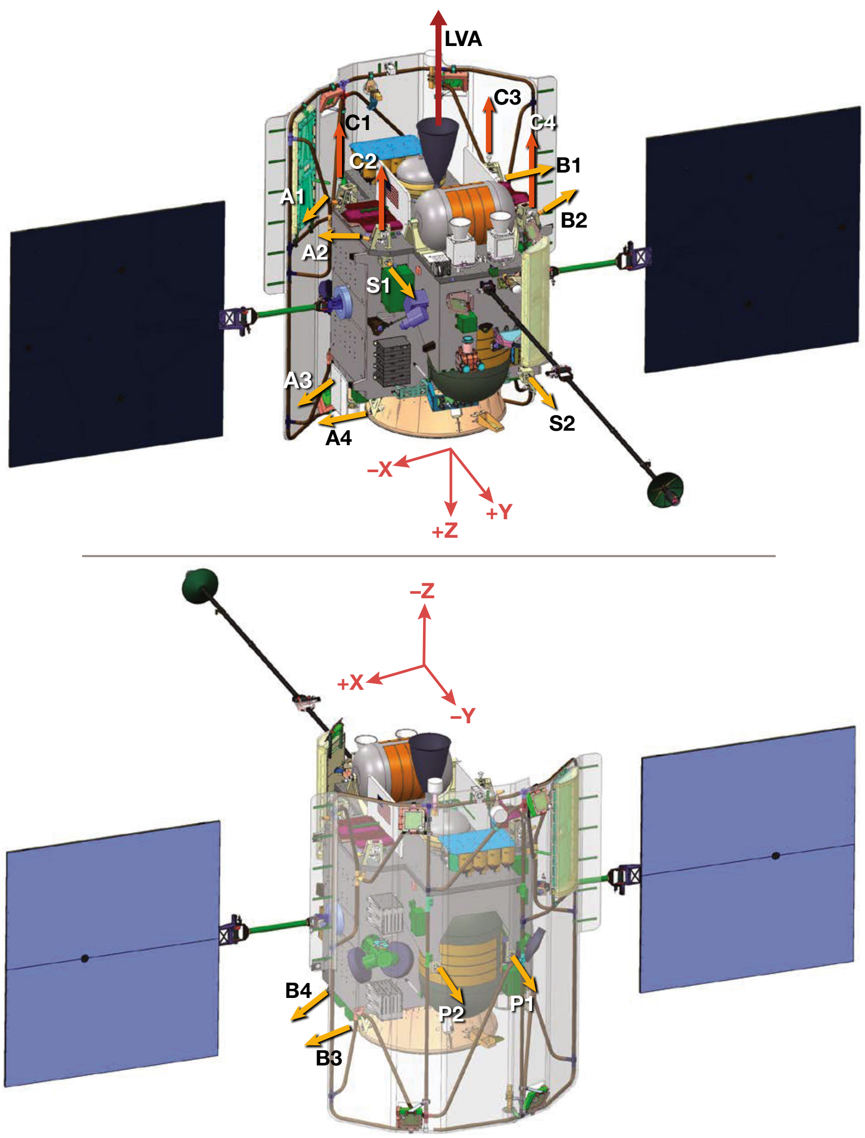

Figure 9. MESSENGER thruster locations and exhaust vectors.

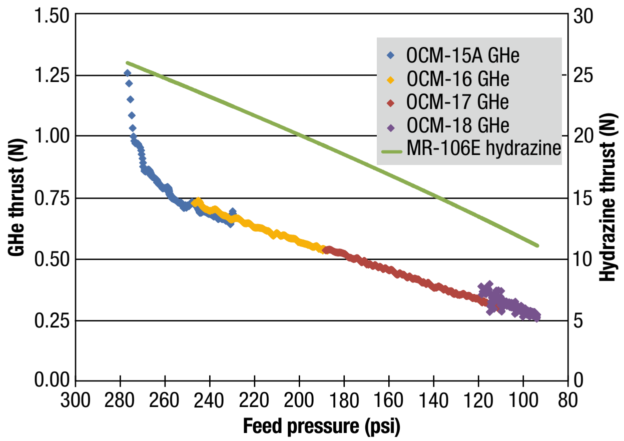

Figure 10. Thrust data from all four 22-N nominal MR-106E ΔV maneuvers using GHe propellant. Equivalent hydrazine thrust is plotted for comparison.

Three more 4C GHe maneuvers extended MESSENGER's orbit until impact on 30 April 2015. OCM-16 (14 April 2015) consumed 0.294 kg of GHe to raise periapsis altitude by 7 km. Thrust was 2.9% of the equivalent N_2H_4 thrust, and delivered specific impulse was 176 s. OCM-17 (24 April 2015) consumed 0.434 kg of GHe to raise periapsis altitude by 10 km. Thrust was 2.5% of the equivalent N_2H_4 thrust, and delivered specific impulse was 176 s. OCM-18 (28 April 2015) consumed 0.114 kg of GHe. Periapsis altitude was raised from 5.3 to 6.3 km, although on MESSENGER's subsequent orbit it was again below 5.3 km. Thrust was 2.3% of the equivalent N_2H_4 thrust, and delivered specific impulse was 176 s.

OCM-15a, OCM-16, OCM-17, and OCM-18 used a combined 1.3 kg of helium to successfully boost MESSENGER's periapsis altitude by a total of 28.8 km. Figure 10 shows GHe thrust as a function of feed pressure for the four MR-106E maneuvers. Figure 11 shows GHe

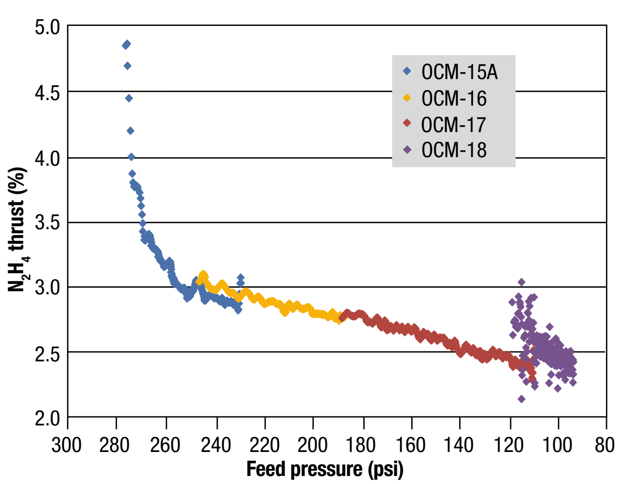

Figure 11. Delivered GHe thrust as a percentage of nominal MR-106E hydrazine thrust.

thrust as a percentage of equivalent hydrazine thrust. Although propellant is rarely the life-limiting item on a spacecraft, APL has demonstrated that available pressurant can be repurposed as propellant to extend missions. The maneuvers performed during MESSENGER's low-altitude hover campaign produced the only existing in-space cold gas helium performance data sets for two engines that can be used as is or correlated to predict the performance of similar thrusters.

CONCLUSIONS

NASA sponsored a variety of Mercury orbiter mission studies over a 30-year period, and the resulting missions were always considered too complex and expensive. Thus, Mercury essentially remained a mystery planet until MESSENGER showed a way to accomplish the mission within NASA's Discovery Program for low-cost planetary missions. The five primary enabling technologies made a Discovery mission to Mercury possible.

When MESSENGER was first proposed to NASA in 1996, it was stripped down to fit within the allowable cost cap. It was anticipated that the mission would return only ~1000 images and limited other science data, just enough to meet its primary science goals. It was very mass constrained by the limited capacity of the available launch vehicles and the necessarily high onboard propulsion demands.

However, the creative skills of the engineering team greatly expanded the science capabilities of the mission, and science data return far exceeded what was originally expected. MESSENGER ultimately returned >200,000 images that yielded a detailed view of Mercury's global surface. The mission overcame thermal limitations and switched to an 8-h orbit, which gave ~50% more close observations of the planet. In addition, solar sailing contributed to fuel savings that enabled a planned 1-year orbital mission to be extended to 4 years of orbital operations.

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Robert E. Gold is a physicist who has specialized in heliospheric and planetary research as well as the development of new instrumentation and mission concepts. He has extensive experience in mission development and operations. He is the Science Payload Manager for MESSENGER, the first spacecraft to orbit the planet Mercury, and was also the Payload Manager for the Near Earth Asteroid Rendezvous (NEAR) spacecraft, the first to orbit and land on an asteroid. He served as the Chief Technologist of APL's Space Department (now the Space Exploration Sector) for a decade and led the APL Space Instrumentation Group during the previous decade. He served as the APL Project Scientist for the NASA Advanced Composition Explorer (ACE) mission, which provides early warning of interplanetary disturbances before they reach Earth. He has written more than 75 scientific publications, and asteroid 4955 Gold was named for his work on the NEAR mission. His e-mail address is robert.gold@jhuapl.edu.

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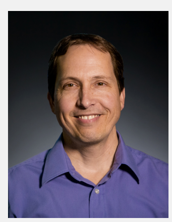

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