Return to Mercury: The MESSENGER Spacecraft and Mission

Robin M. Vaughan, James C. Leary, Richard F. Conde, George Dakermanji, Carl J. Ercol, Karl B. Fielhauer, David G. Grant, Theodore J. Hartka, T. Adrian Hill, Steven E. Jaskulek, James V. McAdams, M. Annette Mirantes, David F. Persons, and Dipak K. Srinivasan
The Johns Hopkins University
Applied Physics Laboratory
11100 Johns Hopkins Road
Laurel, MD 20723
443-778-5006
robin.vaughan@jhuapl.edu

Abstract—NASA's MESSENGER mission, part of its Discovery Program, is the first mission to return to the planet Mercury since the Mariner 10 flybys in 1974 and 1975. The spacecraft incorporates many innovative features, including a sunshade made of ceramic cloth for protection from the Sun, a pair of electronically steerable phased-array antennas, and specially hardened solar panels. A suite of seven miniaturized science instruments, along with the antennas, will globally characterize the planet's composition, structure, atmosphere, and charged particle environment. MESSENGER was launched on August 3, 2004, and performed its single Earth flyby on August 2, 2005. The spacecraft will make two flybys of Venus and three of Mercury prior to orbiting the planet for one Earth-year beginning in March 2011. Highlights of a busy first year of flight operations include initial testing of all spacecraft systems and instruments, execution of six trajectory control maneuvers, and instrument observations of the Earth and Moon surrounding the August flyby. [1,2]

TABLE OF CONTENTS

1. INTRODUCTION

The MErcury Surface, Space ENvironment, GEochemistry, and Ranging (MESSENGER) mission will characterize Mercury in detail by observing the planet from orbit for one Earth year. Although it had long been desired to supplement the initial flybys with an orbital mission, early studies had deemed this infeasible, or at least prohibitively expensive, due to mass and thermal constraints. MESSENGER utilizes a new trajectory first discovered by analysts at the Jet Propulsion Laboratory (JPL) [1,2], and later refined by analysts at the Johns Hopkins University Applied Physics Laboratory (APL), to achieve orbit with an overall spacecraft mass of 1107 kg (including propellant). Flybys of Earth, Venus, and Mercury itself are interspersed with five large deterministic deep-space maneuvers (DSMs) that target the spacecraft for its Mercury orbit insertion (MOI) maneuver in 2011. A composite structure that is tightly integrated with the propulsion system further reduced the required dry mass for the spacecraft.

The high temperatures and radiation doses to be encountered at Mercury were a key engineering challenge in the spacecraft design [3]. Protection from this environment is accomplished with a large sunshade, which shields the spacecraft components from direct exposure to the Sun and allows them to operate at conditions typical of other interplanetary spacecraft. The geometry of the orbit about Mercury was also chosen to limit the worst-case exposure conditions. The solar panels are necessarily exposed to the Sun throughout the mission and were specially designed to handle the temperatures and solar input flux expected at 0.3 AU [4].

MESSENGER is a collaboration between the Carnegie Institution of Washington and APL and was selected as the seventh Discovery mission with formal project start in January 2000. The spacecraft engineering and science instrument design evolved over the 3-year period from January 2000 to the spring of 2003. Assembly and integration of the spacecraft began in February 2003 and continued up to launch in August 2004. Flight operations are now supported from the mission operations center at APL with communications through NASA's Deep Space Network (DSN) antennas. Mission updates can be found at the project web site http://messenger.jhuapl.edu.

[1] 0-7803-9546-8/06/$20.00© 2006 IEEE
[2] IEEEAC paper #1562, version 3, 31 January 2006

2. MISSION DESIGN

Application of Mercury gravity-assist flybys to lower the spacecraft orbit period prior to Mercury orbit insertion was first described by C. L. Yen in the 1980s [1]. Most of the planet-to-planet transfers closely follow planet:spacecraft orbital resonances, whereby there is an integer ratio between the orbit periods of the targeted planet and the spacecraft. There were three opportunities for MESSENGER launch in 2004, with varying combinations of Venus, Mercury, and Earth flybys as shown in Table 1. These trajectories had flyby altitudes at Venus and Mercury above the minimum limits of 300 and 200 km, respectively, Sun-spacecraft-ΔV angles for DSMs that would not expose sensitive parts of the spacecraft to direct sunlight, and no maneuvers needed during solar conjunctions (periods where the Sun-Earth-spacecraft angle < 3°). At the beginning of development the March launch opportunity was the mission baseline trajectory. Schedule delays and additions to spacecraft testing plans forced the project to shift to the last opportunity as the baseline trajectory, moving the orbit insertion date from 2009 to 2011.

Table 1 2004 Launch Opportunities

Month	March	May	August
Launch dates	10-29	11-22	30 Jul-13 Aug
Launch period (days)	20	12	15
Launch energy (km^2/s^2)	≤ 15.700	≤ 17.472	≤ 16.887
Earth flybys	0	0	1
Venus flybys	2	3	2
Mercury flybys	2	2	3
Deterministic ΔV (m/s)	≤ 2026	≤ 2074	≤ 1991
Total ΔV (m/s)	2300	2276	2277
Orbit insertion date	6 Apr 2009	2 Jul 2009	18 Mar 2011

Having launched in August 2004, the spacecraft is following a 6.6-year ballistic trajectory to Mercury, shown in Figure 1. This trajectory has one Earth gravity assist, two Venus gravity assists, three Mercury gravity assists, and five deep-space maneuvers (planned bi-propellant course-correction maneuvers before Mercury orbit). The longest solar conjunction, lasting about 1.5 months, will begin several days before the first Venus flyby, where a minimum altitude over 3100 km decreases the risk associated with potential loss of communication with the spacecraft. From launch to MOI in March 2011, the spacecraft will complete more than 15 orbits of the Sun and travel nearly 8 billion kilometers relative to the Sun. The Earth-Venus and Venus-Mercury phasing required a DSM for each of these legs. Additional DSMs near the first aphelion after each Mercury flyby move the upcoming Mercury encounter closer to Mercury's perihelion, where the spacecraft's orbit period decreases. The orbital resonances include 1:1 for Earth and Venus, and 2:3, 3:4, and 5:6 for Mercury. The ΔV allocation for MESSENGER, listed by category in Table 2, provides an ample contingency ΔV for recovery from anomalies.

On the third day after entering Mercury orbit in mid-March 2011, an MOI clean-up maneuver will place the spacecraft into the initial primary science orbit. The MESSENGER spacecraft's initial science orbit will have an 80º (±2º) orbit inclination relative to Mercury's equator, 200-km (±25 km) periapsis altitude, 12-hour (±1 minute) orbit period, 118.4º argument of periapsis (60º N periapsis latitude, with 56º N to 62º N acceptable), and a 348º (169º to 354º) longitude of ascending node. These requirements, expressed in Mercury-centered inertial coordinates of epoch January 1.5, 2000, place the spacecraft in an orbit capable of meeting the mission's primary science and engineering requirements. Solar gravity, solar radiation pressure, and subtle spatial variations in Mercury's gravity will alter the spacecraft orbit by moving periapsis north, increasing orbit inclination, and rotating the low-altitude descending node away from the Sun direction (for Mercury at perihelion). With science goals requiring infrequent orbit-phase trajectory adjustments, pairs of orbit-correction maneuvers occur at about the same time every Mercury year, or every 88 Earth days. While the first orbit-correction maneuver (OCM) of each OCM pair increases the orbit period slightly, the second OCM of the pair lowers the periapsis altitude to 200 km and returns the spacecraft orbit period to about 12 hours. By placing the even-numbered OCMs (2, 4, 6) at apoapsis 2.5 orbits after the odd-numbered OCMs (1, 3, 5), there will be time to determine the performance of the first OCM in an OCM pair, and time to upload an adjustment to the second OCM's ΔV target. The nominal mission plan ends science data collection one year after MOI, when the spacecraft's orbit will have a periapsis altitude near 500 km, a periapsis latitude approaching 72° N, and an orbit inclination near 82°.

Table 2 Current ΔV allocation

ΔV Budget Category	ΔV (m/s)
Deep space maneuvers	**1009**
Launch vehicle, navigation errors (99%)	**121**
Mercury orbit insertion	**867**
Orbit correction maneuvers	**85**
Contingency	**169**
Total	**2251**

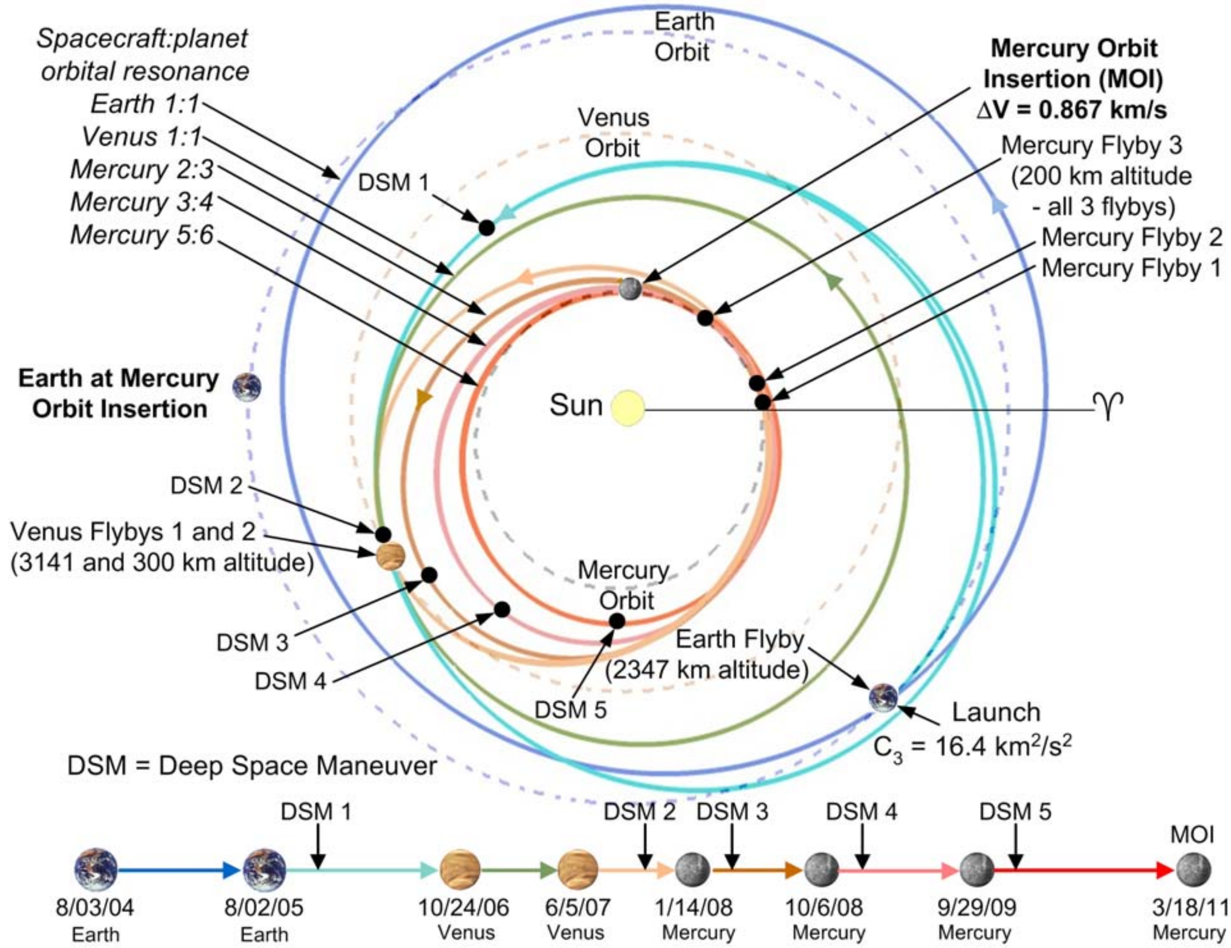
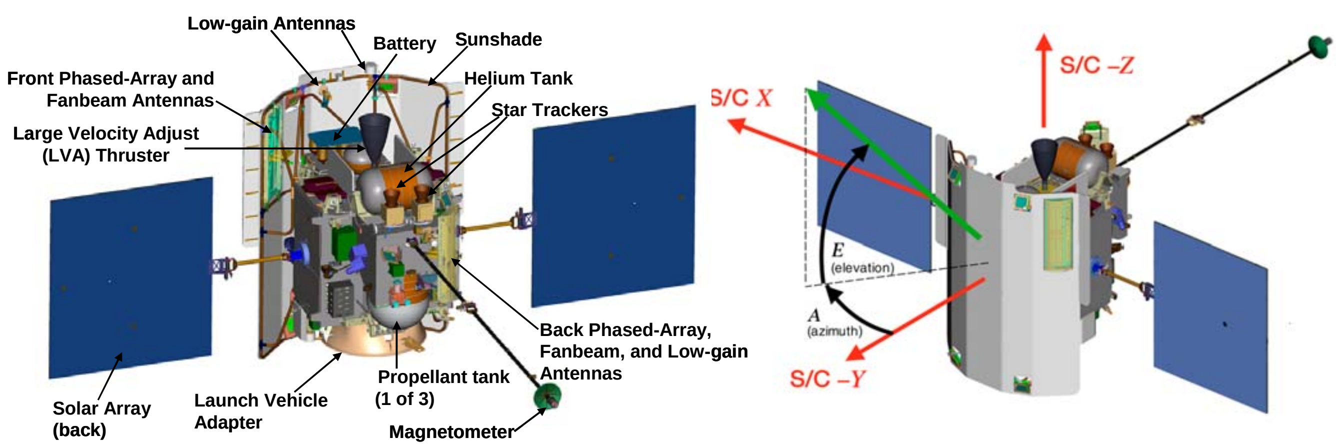


Figure 1. MESSENGER's Heliocentric Trajectory (viewed from north of Earth's orbit plane)



a. Spacecraft Components

b. Spacecraft coordinate axes

Figure 2. The MESSENGER Spacecraft and its Coordinate System

3. SPACECRAFT DESIGN

The MESSENGER spacecraft is designed to operate for a total mission lifetime of 8 years in environments that differ widely from start to end of the mission [5]. The spacecraft is comprised of the traditional systems of structures, propulsion, thermal, power, telecommunications, avionics, flight software, guidance and control (G&C), and instrument payload. Key elements of the engineering systems are shown in Figure 2, along with the spacecraft body coordinate system. (Instruments are shown in a later section). While the harsh solar environment at Mercury most directly influenced thermal and power system design, significant challenges were faced in developing each of the other systems.

Structure and Mechanisms

The primary spacecraft structures are the core, the adapter ring, the sunshade, the solar panels, and the magnetometer boom. The core of the spacecraft tightly integrates support panels with the propulsion system [6]. The support panels are made of graphite-cyanate ester (GrCE) and were supplied by Composite Optics, Inc. (COI). They are arranged in a "double H" configuration and provide mounting surfaces for most of the engineering components. The main propellant tanks are embedded in three compartments in the double H core as shown in Figure 3. The struts and other supports for the tanks transfer lateral loads to the corners of the structure, allowing the composite panels to be thin relative to their size. A thin, copper conductive ground plane is placed over the composite panels to improve conductivity for grounding.

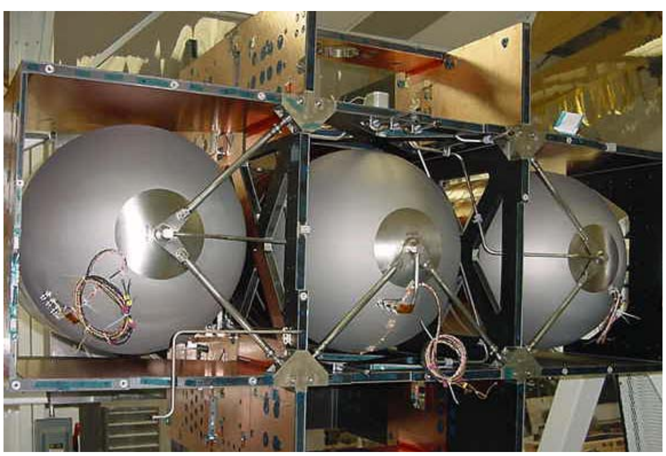


Figure 3. MESSENGER Integrated Structure and Propulsion System, Main Tank Bottom Views

With the composite structure designed to channel all loads into the center column, a square-to-round adapter was necessary to match up with the launch vehicle's separation clamp band interface. The solution chosen was to machine an aluminum forging, carefully tailored to distribute the structural loads evenly from the corners of the center column to the round vehicle interface. The forward adapter flange was slotted between each bolt to accommodate the thermal expansion mismatch between the aluminum adapter and the stiff composite structure.

Three mechanical assemblies were required to deploy: the two solar panels and the 3.6-m magnetometer boom. Each of these has two sets of "saloon-door" type hinges with locking pins. The solar array hinges are located at each end of the arms connecting the panels to the core structure. The panels were released first and allowed to over-travel and settle; then, the arms were released and settled into position. The magnetometer boom is separated into two segments with one hinge between the spacecraft structure and the first segment and the other between the two segments. The magnetometer boom deployed similarly to the panels with separately commanded release and settle events for each hinge line. After confirmation of a full deployment, all six hinge lines were pinned in place to prevent hinge rotation during maneuvers using the large thruster.

Telecommunications

The radio frequency (RF) telecommunications system consists of redundant General Dynamics Small Deep Space Transponders (SDST), redundant Solid-State Power Amplifiers (SSPA), two Phased-Array Antennas, and two Medium-Gain and four Low-Gain Antennas. The two electronically steerable high-gain phased-array antennas are mounted on the sunshade and on the back of the spacecraft. The two medium-gain fanbeam antennas are co-located with the phased arrays. Each of these antenna sets nominally provides coverage in diametrically opposite quadrants of the plane normal to the sunshade; full 360° coverage in this plane is accomplished by rotating the spacecraft to follow the changing Sun and Earth positions. Four hemispherical low-gain antennas (LGAs) are mounted on the spacecraft to provide coverage in all directions. The SSPAs and all antennas were manufactured, assembled, and tested by APL.

The phased-array antennas were developed specifically for the challenges of reliable scientific data return from Mercury orbit and are a mission-enabling technology [7]. The phased arrays use slotted waveguide technology, with novel parasitic monopoles attached to the slots to achieve right-hand circular polarization. These electronically scanned antennas have no mechanical components that could fail in the challenging thermal environment of the Mercury. The phased-arrays are designed to work over a 350°C range in temperature. The effective isotropic radiative power (EIRP) of the MESSENGER downlink is 42.9 dBW when using one of the phased-array antennas.

The RF system provides a reliable spacecraft command capability, a high quality and quantity of spacecraft housekeeping telemetry and scientific data, and highly accurate Doppler and range data for navigation. This system is also integral to the scientific payload, namely the

Mercury Laser Altimeter experiment and radio science experiments. Precise Doppler and range data will enable the detection of a molten core within Mercury that is believed to exist. The MESSENGER spacecraft communicates in X-band with DSN stations located in Goldstone, California; Madrid, Spain; and Canberra, Australia, for spacecraft command and telemetry reception. MESSENGER is the first spacecraft to utilize Turbo coding for downlink. This coding results in an extra 0.9-dB margin in the RF downlink, which corresponds to nearly a 25% increase in data return from the planet.

Power

The power system is designed to support about 390 W of load power near Earth and 640 W during Mercury orbit. Power is primarily provided by two solar panels that are mounted on small booms that extend beyond the sunshade and are capable of rotating to track the Sun. These are supplemented with a battery to provide power during launch and eclipse periods when in orbit about Mercury. Power distribution to spacecraft components is handled by the Power System Electronics (PSE), the Power Distribution Unit (PDU) [8], and the Solar Array Junction Box. A peak-power-tracker topology is used that is based on the APL-designed Thermosphere-Ionosphere-Mesosphere Energetics and Dynamics (TIMED) spacecraft power system [9]. This architecture isolates the battery and the power bus from the variations of the solar-array voltage and current characteristics, and optimizes the solar-array power output over the highly varied solar-array operating conditions of the mission.

Each 1.54 x 1.75 m solar panel contains alternating rows of triple-junction solar cells and Optical Solar Reflector (OSR) mirrors. The solar cells are 0.14 mm thick and have a minimum efficiency of 28%. The solar cell strings are placed between the OSRs with a cell-to-OSR ratio of 1:2 to reduce the panel absorbance. The normal operational solar-array maximum-power-point voltage is expected to vary between 45 and 95 v, but this figure does not include the higher transient voltages expected on the cold solar arrays at the exit from eclipses. Thermal control is performed by tilting the panels away from normal incidence with increased solar intensity. All material and processes used in the solar panels are designed to survive the worst case predicted temperature of 270°C and an estimated total radiation dosage of 4×10^{14} equivalent 1-Mev/cm^2 electrons.

The loads are connected directly to the 22-cell, 23-Ahr NiH_2 battery. Each battery pressure vessel contains two battery cells. The nominal bus voltage is 28 v and can vary between 20 and 35 v depending on the state of the battery. The primary battery charge control is ampere-hour integration; charge-to-discharge (C/D) ratio control performed by the flight software. The battery is charged at a high rate with available solar array power and gradually ramped down to a trickle charge state as full charge is reached. The software also monitors the battery pressure. If the battery pressure reaches a predetermined level indicating full state of charge, the battery charge is commanded to trickle rate. The battery voltage is controlled to preset safe levels with temperature-compensated voltage limits that are implemented in hardware. Whenever the battery voltage reaches a limit, the battery charge current will taper. The battery charge-control technique used reduces the battery overcharge and its associated heat dissipation and extends the battery life.

Propulsion

The MESSENGER propulsion system, provided by Aerojet, is a pressurized bipropellant, dual-mode system using hydrazine and nitrogen tetroxide (N_2H_4) in the bipropellant mode and hydrazine in the monopropellant mode. Three main propellant tanks, a refillable auxiliary fuel tank, and a helium pressurant tank provide propellant and pressurant storage. These tanks provide propellant storage for approximately 368 kg of fuel and 231 kg of oxidizer [10].

A new lightweight main propellant tank was developed and qualified for MESSENGER [11]. The tank configuration is an all-titanium, hazardous-leak-before-burst design with a measured mass less than 9.1 kg, including all attachment features. The design includes two titanium baffles used for spacecraft nutation control. A titanium vortex suppressor is provided at each tank outlet to delay vortex formation. The main propellant tanks are symmetrically positioned about the spacecraft centerline to maintain center-of-mass control during propellant expulsion in flight. Two fuel tanks flank the center oxidizer tank.

The helium tank is a titanium-lined composite over-wrapped leak-before-burst pressure vessel (COPV) based on the flight-proven A2100 helium tank. A second outlet was added to the existing helium tank to provide a dual pressurization capability for the fuel and oxidizer systems. A small titanium auxiliary tank is a hazardous-leak-before-burst design. It has an internal diaphragm to allow positive expulsion of propellant. The tank operates in blow-down mode between 2.07×10^6 and 7.58×10^5 Pa (20.7 and 7.58 bar or 300 and 110 psia) and is recharged in flight.

MESSENGER carries seventeen thrusters. Three thruster types, arranged in five different thruster module configurations, provide the required spacecraft forces and torques. The large velocity adjustment (LVA) thruster is a flight-proven Leros-1b that will provide a minimum of 667 N of thrust. Four 22-N, monopropellant LVA-thrust-vector-control (TVC) thrusters, designated C1-4, provide thrust vector steering forces during LVA burns and primary propulsion for intermediate ΔV maneuvers. They are fed with hydrazine in both the pressurized and blow-down modes. Twelve monopropellant thrusters each provide 4.4 N of thrust and specific impulse of 220 s for fine-attitude-control burns, small ΔV burns, and momentum management. These thrusters are also fed with N_2H_4 in both the pressurized and blow-down modes. Eight 4.4-N thrusters, designated A1-4 and B1-4, are arranged in double canted sets of four for redundant three-axis attitude control. Two 4.4-N thrusters, designated S1&2 provide velocity changes in the sunward direction. The final two 4.4-N thrusters, designated P1&2, provide velocity changes directed away from the normal Sun direction.

A set of nine latch valves controls helium, fuel, and oxidizer flow between the tanks and to the thrusters. Catalyst bed heaters are provided for the 4.4-N and 22-N thrusters, and a flange heater is used for the LVA. When enabled, all catalyst bed heaters are time controlled, while the LVA flange heater is controlled with mechanical thermostats. Primary and secondary thruster heaters can total 149 W. Heaters are also installed on the propellant and pressurant tanks, thruster valves, valve panel, fill-and-drain valve bracket, and various propellant manifolds. Primary helium and main propellant tank heaters are controlled by spacecraft software. Primary and secondary tank heaters can total 156 W.

Thermal Design

MESSENGER has a low-risk passive thermal control design that is dominated by the sunshade. The shade is made of aluminized Nextel 312-AF 10 ceramic cloth and multi-layer aluminized kapton. The shade will be heated to a maximum temperature of 350°C at Mercury but creates a benign thermal environment for the main spacecraft bus, allowing the use of essentially standard electronics, components, and thermal blanketing materials.

Digital Sun sensors, one of the phased-array antennas, the X-ray solar monitor, and two thrusters are mounted on Sun-facing side of the sunshade and will experience as much as 11 times the solar environment near Earth. The Sun sensor materials and housing have been modified for better thermal radiation, and special filters are placed over them to attenuate the incoming sunlight. A Nextel radome protects the phased-array antenna. The three low-gain antennas on the sunshade also have Nextel radomes. Thermal control for the solar panels is provided by the chosen ratio of reflectors to power cells and appropriate rotation of the panels relative to the Sun line, as described in the power section. All of these Sun–illuminated components are thermally isolated from themselves and the rest of the spacecraft.

During certain orbits about Mercury, the spacecraft will be between the Sun and the hot planet for approximately 30 minutes. During this period, the unshaded back of the spacecraft will get a direct view of the hot Mercury surface. Components such as the battery and star trackers are positioned such that the spacecraft body blocks a substantial portion of the planet view, minimizing direct radiation from the thermal environment. Planet-viewing instruments such as the imager required a very specialized thermal design to allow full operation during this hot transient period. Diode heat pipes are employed in both the spacecraft and imager thermal designs to protect attached components when radiator surfaces are exposed to the thermal radiation emitted by Mercury. The diode heat pipes effectively stop conducting when the radiator surface begins to get hot, and return to conduction when the radiator surface cools, restoring normal thermal control.

Much emphasis has been placed on the extreme heat and high temperatures associated with a spacecraft orbiting Mercury. At the beginning of the mission, after launch with the sunshade pointing towards the Sun, the spacecraft will use heater power to make up for the maximum solar and planet environments that are not present. In order to reduce heater power consumption and increase solar array power margin, MESSENGER is designed to be flipped such that the anti-Sun side can be illuminated during outer cruise (i.e.oriented with the sunshade pointed away from the Sun).

Avionics

MESSENGER is equipped with redundant integrated electronics modules (IEM) that provide uplink and downlink processing, all routine flight software functions, and fault protection. Each IEM is partitioned into five 6U, compact PCI-compatible daughter cards, a backplane, and a chassis [12]. Three of the five daughter cards communicate over a 32-bit-wide PCI bus operating at 25 MHz. Three cards, the Main Processor (MP), Fault Protection Processor (FPP), and Solid-State Recorder (SSR), were designed and manufactured by BAE Systems to APL specifications. The MP and FPP boards are nearly identical. The Interface Card, Converter Card, and backplane were designed and built by APL to capture the MESSENGER-unique requirements. An Oven Controlled Crystal Oscillator (OCXO) located outside the IEM chassis provides precision timing.

The MP and FPP boards utilize RAD6000 processors operating at 25 MHz and 10 MHz, respectively, to run normal and fault protection software applications. The MP is populated with 8 Mbytes of random access memory (RAM) and 4 Mbytes of electrically erasable programmable read-only memory (EEPROM); the FPP is populated with 4 Mbytes each of RAM and EEPROM. The MP includes a MIL-STD-1553 interface that is configured either as a Bus Controller (BC), if primary, or Remote Terminal (RT), if backup. The FPP includes a MIL-STD-1553 interface that is configured as a simultaneous RT and Bus Monitor (BM). The SSR implements 8 Gbits of user memory. The memory is implemented with upscreened commercial Synchronous Dynamic RAM (SDRAM). Memory contents are retained through an IEM reset but are lost if IEM power is turned off.

The Interface Card implements the unique functions required. A Critical Command Decoder responds to a small set of uplink commands directly in hardware. A mission elapsed time is generated and time tags data with a resolution of 1 μs and an accuracy of 1 ms. Downlink framing hardware has a highly adjustable bit rate that can be tuned to the capabilities of the RF link. An uplink buffer accepts serial command data from both transponders and implements sync detection and a serial-to-parallel conversion function in order to provide the uplink data to the MP. An image interface circuit receives 4 Mbits/s of image data that are transferred to the SSR.

Flight Software and Autonomy

C&DH (command and data handling) and G&C functions are combined in a single flight software application that runs in the MP. The MP flight code is implemented in C and operates under the VxWorks 5.3.1 real-time operating system. The G&C software is implemented in Simulink™ and converted to C code using the RTW™ (Real-Time Workshop) tools, both provided as part of Matlab™.

Only one MP is designated "active" or "primary" and executes the full MP flight software application. The other MP will typically remain unpowered and does not serve as a "hot spare." If powered, the backup MP remains in boot mode and supports rudimentary command processing and telemetry generation for the purpose of reporting the health status of that processor, and to support uploads of code and parameters to EEPROM. It operates as a RT on the 1553 data bus. The primary MP serves as 1553 bus controller and manages all communication with devices on that bus.

C&DH functionality in the primary MP includes uplink and downlink management using the Consultative Committee for Space Data Systems (CCSDS) protocol, command processing and dispatch to other spacecraft processors and components, support for stored commands (command macros) and time-tagged commands, management of the 8 Gbit SSR and file system, science data collection, image compression, telemetry generation, memory load and dump functions, and support for transmission of files from the SSR on the downlink using CCSDS File Delivery Protocol (CFDP). The uplink and downlink functions include control of two transponders via the 1553 bus. A number of C&DH functions interface to the spacecraft through the IEM Interface Card. For example, the uplink/downlink data buffers are on that card. The interface card also allows for critical hardware commands to be sent from ground or the FPP to force resets of spacecraft processors.

The SSR data management, storage, and playback design is new for MESSENGER. A file system is used to store engineering telemetry and science data; files can be selectively located on the SSR for downlink to optimize science return [13]. Contingency files can be stored and downlinked only if needed. Raw images are stored on the SSR and can be compressed using the integer wavelet compression algorithm or reduced using subframing techniques. Downlinking of files to the ground is accomplished using the CFDP; this protocol provides guaranteed delivery of all file data. MESSENGER is the first NASA mission to fly CFDP and the first APL mission to use an SSR file system.

The MP software also implements a number of engineering functions. Analog temperature data are collected from Temperature Remote Input Output (TRIO) sensors, via a 1553 interface to the PDU. A peak-power-tracking algorithm optimizes charging of the spacecraft battery via the PDU interface. A heater control algorithm manages the temperatures of the main propellant tanks and the pressurant tank. Motor step commands for the solar-array drives and transponder commands for phased-array antenna steering are computed based on desired positions provided by the G&C software or ground input. The main G&C functions included in the MP software are described in the next section.

Fault protection is centralized in the two FPPs. Each FPP independently collects spacecraft health information over the MIL-STD-1553 bus and over dedicated links to the PDU. The two FPPs are always powered. The FPP corrects most faults by sending commands to the MP. Each FPP can reset the MP in its own IEM or select which of two stored flight applications the MP loads and executes. The FPP can also reconfigure the MPs using special commands sent directly to the PDU. The PDU command interface allows the FPPs to swap the bus controller functionality between MPs or power on and switch to the redundant MP and declare it primary. The IEM interface board includes hardware limits to prevent a failed FPP from continuously sending commands that would disrupt spacecraft operation.

Each FPP executes an identical flight code application that supports a command and telemetry interface to the MPs via the 1553 data bus. The code application includes an autonomy rule engine, which accepts uploadable health and safety rules that can operate on data collected from the 1553 data bus or a state message transmitted by the primary MP. The action of each rule can dispatch a command (or a series of commands from a stored FPP macro) to the primary MP for subsequent execution by the MP to correct faults. Fault correction can include actions such as switching to redundant components, demotion to one of two safe modes, or shedding power loads [14]. The same rule engine is available in the MP software. MP autonomy rules are used to implement some routine functions such as RF reconfiguration, as well as additional fault responses not covered by FPP rules.

Guidance and Control

The MESSENGER guidance and control subsystem maintains spacecraft attitude and executes propulsive maneuvers for spacecraft trajectory control. Software algorithms run in the MP to coordinate data processing and commanding of sensors and actuators to maintain a 3-axis stabilized spacecraft and to implement desired velocity changes. Multiple options are available for pointing the spacecraft sunshade, antennas, thrusters, and science instruments at designated targets. The orientation of the two solar panels is also software controlled to maintain a Sun offset angle that provides sufficient power at moderate panel temperatures. The system enforces two attitude safety constraints. The most important of these is the Sun keep-in (SKI) constraint that keeps the sunshade pointed towards the Sun to protect the spacecraft bus from extreme heat and radiation. The hot-pole keep-out constraint protects components on the top deck of the spacecraft from additional thermal extremes due to re-radiation of sunlight from the surface of the planet once in orbit [15, 16].

The sensor suite consists of star trackers, an inertial measurement unit (IMU), and Sun sensors. Inertial attitude reference is provided by two autonomous star trackers (ASTRs) from Galileo Avionica, both of which are mounted on the top deck looking out along the –Z axis. Spacecraft rotation rates and translational accelerations are provided by a Northrop-Grumman Scalable Space Inertial Reference Unit (S-SIRU) IMU with four hemispherical resonance gyroscopes (HRGs) and Honeywell QA3000 accelerometers. The IMU has two power supply and processor boards providing internal redundancy. MESSENGER also carries a set of Adcole digital Sun sensors (DSSs) to provide Sun-relative attitude knowledge if there is a failure in the primary attitude sensors. Two separate Sun sensor systems consist of a DSS electronics (DSSE) box connected to three DSS heads (DSSHs), two of which are located on opposite corners of the sunshade and one on the back of the spacecraft.

The actuator suite consists of reaction wheels and thrusters. The primary actuators for maintaining attitude control are four Teldix RSI 7-75/601 reaction wheels. All four wheels are always operating; each can provide a maximum torque of 0.075 Nm, with maximum momentum storage of 7.5 Nms. Thrusters in the propulsion system are used for attitude control during TCMs and momentum dumps, and may also be used as a backup system for attitude control in the event of multiple wheel failures. Eight of the 4.4-N thrusters (A1-4 and B1-4) are used for attitude control. The remaining four 4.4-N thrusters (S1 and S2 or P1 and P2) are used for small velocity changes, while the four 22-N thrusters (C1-4) are used for larger velocity changes. The LVA main engine is used for very large velocity changes, such as the five DSMs and MOI. In addition to thruster on/off commands, the flight software operates latch valves and heaters while thrusters are firing.

The G&C system interfaces with actuators for three other spacecraft components: the solar panels, the phased-array antennas, and the imaging cameras. Solar panel rotation is performed using two MOOG solar-array drive assemblies (SADAs). The drives provide an operational range of travel of 228° in the YZ plane centered around the –Z axis and bounded at 66° from the +Z axis towards the sunshade (–Y) and towards the back of the spacecraft (+Y). The boresight of each phased-array antenna is electronically steerable as described above. The cameras are mounted on a pivot platform with a rotary drive that provides an operational range of travel of 90° in the YZ plane, 40° from the +Z axis towards the sunshade (–Y), and 50° towards the back of the spacecraft (+Y).

Instruments

MESSENGER carries a diverse suite of miniaturized science instruments, shown in Figure 4, to characterize the planet globally [17,18]. Four of the science instruments are co-boresighted and mounted inside the launch vehicle adapter ring: two imaging cameras (Mercury Dual Imaging System – MDIS), a laser altimeter (Mercury Laser Altimeter – MLA), UV and IR spectrometers (Mercury Atmosphere and Surface Composition Spectrometer - MASCS), and an X-Ray Spectrometer (XRS). The two cameras are mounted on a pivoted platform that extends their observing range for flybys and in orbit. Other instruments located outside the adapter ring are a Gamma-Ray and Neutron Spectrometer (GRNS), an Energetic Particle and Plasma Spectrometer (EPPS), and a Magnetometer (MAG). The antennas are also used for radio science.

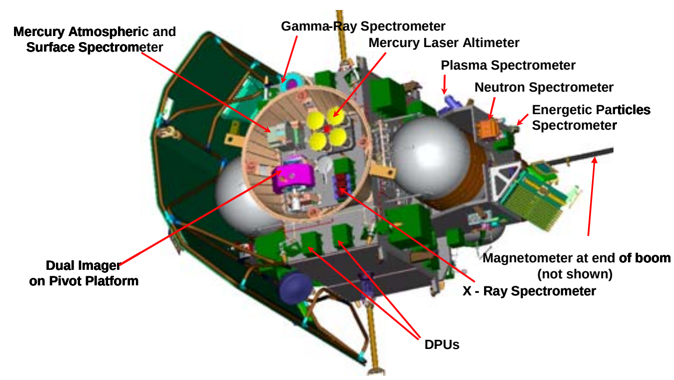


Figure 4. MESSENGER Science Instruments

This payload was very ambitious both because of its breadth of capabilities and in the limited amount of mass allocated to build it (50 kg for seven instruments). The payload power requirements were also limited, but not in a typical manner for science missions. The most limited power period is during early cruise, when the solar arrays are generating their lowest power; this limit restricted the size of instrument heaters that could be used. In contrast, during the orbital phase of the mission, the solar arrays generate ample power, but during eclipse the battery power is still very limited. The instrument designs were limited by their ability to dissipate heat to the spacecraft deck or to the space environment. This difficult thermal design (stay warm enough in cruise and during eclipse periods, but cold enough on orbit) was a significant driver throughout the payload development period.

Redundant data processing units (DPUs) buffer all data interfaces between the payload elements and the spacecraft;

one DPU is powered on whenever a payload element is active, while the other DPU is maintained off as a cold spare. The DPUs communicate with the spacecraft processors via the spacecraft 1553 busses but communicate with the instruments via separate dedicated RS-422 Universal Asynchronous Receiver Transmitter (UART) interfaces. The DPUs greatly simplify the spacecraft-to-payload interface.

All but one of the instruments use common power supply boards and processor boards that facilitate the use of identical power and data interfaces for every instrument in the payload. This arrangement also allows common software modules to be used to handle a number of routine tasks.

4. LAUNCH TO EARTH FLYBY

Launch and Spacecraft Checkout

On August 3, 2004, at 06:15:56.5 UTC, MESSENGER launched from Cape Canaveral Air Force Station aboard a Delta II 7925H launch vehicle. The spacecraft departed Earth orbit with 16.388 km^2/s^2 launch energy and -32.66° declination of launch asymptote (DLA) relative to Earth's mean equator at the standard J2000 epoch. Minimum power margin requirements (limited by a 1.077 AU maximum spacecraft-Sun distance and by a short-duration coast between each Delta II second stage engine firing) forced selection of this DLA. A 2.5-σ under-burn of the Star 48B solid rocket motor was the primary contributor toward a 2.0- σ overall deficit measured at a reference point shortly after spacecraft separation from the launch vehicle's third stage.

Following separation from the launch vehicle, each of the two solar panels was consecutively deployed by firing pyros to release their two hinges. The A and B sets of 4.4-N thrusters were fired to null the 1.72 °/sec rotation rate and turn the back of the spacecraft (+Y axis) towards the Sun. Attitude control then transitioned to the reaction wheels, and a slow rotation was induced about the Y-axis. The solar panels were rotated to place the cell side facing the Sun. Communication with the spacecraft used the front and back LGAs.

The spacecraft intentionally maintained its lowest safing mode for the first six days. Extensive testing of the RF system exercised all four LGAs and nearly all the switches and other components. Successful operation was verified using a variety of uplink and downlink rates. Attitude control tests varied the rotation rate about the Sun line and verified operation of the Sun sensors and star trackers.

The spacecraft was promoted to its normal operational mode on August 9, 2004, establishing a true 3-axis inertial attitude that kept the Sun on the +Y axis and placed the Earth in the back antenna (+X, +Y) quadrant. Once this downlink attitude was achieved, communications were transferred to the back fanbeam antenna. The back phased-array antenna was first used on August 31, 2004. A subset of the full scan range was tested using both four and eight phased-array elements. The antenna was scanned 15º off boresight to point in the true direction of Earth and then off-pointed from that scan direction over a subset of its full range where the link could be maintained. The values predicted for downlink received power matched the values reported by the DSN block V receiver (BVR). The trajectory geometry precluded testing of the front fanbeam and phased-array antennas until February 2005. The front phased-array antenna was first used for routine communications between March 8 and 14, 2005.

Other tests conducted during the first few weeks after launch verified solar panel rotation control modes and wheel control algorithms. Spacecraft temperatures were all within the predicted ranges, indicating nominal performance for thermal control in the back-to-Sun orientation. The solar arrays were producing the expected power output with ample margin while operating the standard complement of components. The actual maximum output power from the arrays was determined in peak power tests performed in September and December 2004. These tests turned on additional spacecraft loads until the battery began to discharge, indicating that maximum panel power was being consumed.

Trajectory Control Maneuvers

MESSENGER executed five TCMs between August 2004 and July 2005. The dates and the target and achieved ΔV magnitudes for these are shown in Table 3. TCMs 1, 2, and 3 used the four 22-N C thrusters, drawing fuel from the main tanks. TCM 5 used the two 4.4-N S thrusters, and TCM 6 used the two 4.4-N P thrusters, drawing fuel from the auxiliary tank. All of these trajectory corrections were performed as "turn-and-burn" maneuvers, where the spacecraft attitude was changed to align the thrusters with the target ΔV direction in the inertial frame. The spacecraft was in a back-to-Sun orientation for TCMs 1, 2, 3, and 5 but was flipped to a shade-to-Sun orientation for TCM 6. (TCM 4 was not needed and was cancelled.)

TCMs 1 and 2 were designed to correct the somewhat larger than expected launch injection error. TCM 3 was a small deterministic correction required for the August launch trajectory. TCMs 5 and 6 refined the targeting for the Earth flyby. These maneuvers proved to be sufficiently accurate that TCMs 7 and 8, scheduled 5 days before and 10 days after Earth closest approach, were not needed.

The LVA main engine was successfully fired for the first time for DSM 1 (TCM 9) on December 12, 2005. This maneuver is the largest of the five DSMs, with a target magnitude of over 300 m/s, as shown in Table 3. With execution of DSM 1, the project has now exercised all of the thrusters and each of the three maneuver modes.

Table 3. MESSENGER TCMs

TCM #	Date	Target ΔV (m/s)	Actual ΔV (m/s)
1	8/24/04	18.0	17.90
2	9/24/04	4.59	4.59
3	11/18/04	3.24	3.25
5	6/23/05	1.14	1.10
6	7/21/05	0.147	0.150
9 (DSM 1)	12/12/05	315.72	315.63

Flips and Flops: Shade- Versus Back- to-Sun Orientation

Pre-launch plans called for turning to a shade-to-Sun orientation in April 2005. In the first two months of 2005, the neutron spectrometer (NS) sensor on the GRNS, located on the back side of the spacecraft, began experiencing temperatures above the normal level. In March, the star tracker also reached high temperatures due to the decreasing spacecraft distance from the Sun. The turn to a shade-to-Sun attitude, termed a "flip," was moved up to March 8, 2005. The magnetometer boom deployment was performed directly after this flip when temperature conditions for one of the hinges were optimum. The turn and the boom deployment were successful, reducing the temperature of both the NS sensor and the star trackers, while maintaining positive power margins.

Spacecraft temperatures were closely monitored after the flip, and it was soon apparent that more heaters were powered on than had been predicted from the pre-launch models. In particular, the secondary tank heaters were on almost continuously, along with the main propulsion tank heaters, to maintain the desired tank temperatures. One of the Sun sensor heads was running at a very low temperature, making powering of the back antenna heater desirable to keep it within its qualified temperature range. The excess heater power used much of the remaining capacity of the power system, leaving little or no power margin for instrument operation. This situation necessitated replanning of the overall back- versus shade-to-Sun transitions and the Earth flyby science activities.

Pre-launch plans had called for only one flip from back-to-Sun to shade-to-Sun orientation in the spring of 2005. But the spacecraft was now known to be thermally stable, with ample power margin, with its back to the Sun at distances greater than 0.95 AU. A turn back to the original back-to-Sun attitude – termed a "flop" – was performed on June 14, 2005. The spacecraft was left in this orientation until September 7, 2005, when the spacecraft-Sun distance once again fell below 0.95 AU. The shade will remain pointed at the Sun until March 2006. A flop to back-to-Sun orientation will occur on March 8, 2006, followed by another flip on June 28, 2006. This date corresponds to the last remaining time when the spacecraft will be outside 0.95 AU from the Sun.

Continuous instrument observations from the shade-to-Sun orientation had been planned for several days around Earth closest approach. While some of these could be performed in the back-to-Sun orientation, a few of the highest priority observations required a shade-to-Sun orientation. MDIS imaging and MASCS spectra are degraded in the back-to-Sun orientation due to stray light entering the adapter ring. The solar wind observation by the fast imaging plasma spectrometer (FIPS) sensor on the EPPS instrument required a shade-to-Sun orientation due to the sensor's mounting location. Constraints on the rotation axis for the turns and on solar panel control response caused the panels to be off Sun and the battery to discharge for brief periods during each flip or flop. No more than eight shade-to-Sun sessions were allowed to preserve battery life. Six sessions were distributed through July and August to accommodate the science observations as shown in Table 4. Another shade-to-Sun period was added around TCM 6 because the turns to and from burn attitude resulted in significant Sun azimuth excursions.

Table 4. Spacecraft Flip-Flops for Earth Flyby

Date	Activities
6/28/05	Practice scans and imaging of Earth similar to highest priority lunar observations
7/24/05	MDIS optical navigation imaging
7/27/05	MDIS optical navigation imaging and Earth-Moon mosaic
7/30-31/05	MDIS optical navigation and lunar imaging; MASCS lunar scans
8/2/05	MDIS Earth imaging; EPPS and MAG observations of Earth magnetic and particle environment (closest approach)
8/6-7/05	FIPS solar wind observations

Instrument Check Out and Calibration

Initial check out of all the instruments was performed in the first five weeks after launch. These tests exercised communications and power paths to the payload and verified basic instrument functionality. The focus then turned to maintenance and in-flight calibration activities. Observations of stellar targets with known spectral characteristics were made by the MASCS and XRS instruments; these provided both boresight alignment data and sensor performance information. The MDIS cameras imaged their calibration target mounted inside the adapter ring and later imaged Sirius and the M7 star cluster to obtain alignment and point spread information. Rotations about each of the spacecraft body axes after the magnetometer boom was deployed provided full three-dimensional observation of the interplanetary magnetic field.

Instrument calibration also took advantage of the Earth flyby that became part of the trajectory when launch was delayed to August of 2004. The Earth and Moon are ideal calibration objects since there are many other observations to use as standards of comparison and the near-Earth environment is well characterized.

The Earth observation campaign began with MLA scans of Earth in late May and early June 2005. The MLA optics successfully sensed the bright Earth in the first scans, allowing refinement of the boresight alignment in the spacecraft body frame. The laser was fired while aiming at a telescope at Goddard Space Flight Center during the second set of scans. A ground laser was also fired at the spacecraft during these scans. These laser firings were detected both on the ground and at the spacecraft. The MLA team has been involved in similar simultaneous ground-spacecraft laser observations on previous missions, but this was the first successful test. During these MLA scans the MDIS camera obtained its first images of the Earth and Moon when the spacecraft was 2.96 $\times 10^7$ km from Earth.

Starting in late July, several sets of images of the Earth against the star background were taken with the MDIS cameras as a test of optical navigation techniques to be used for the Mercury flybys and on approach to MOI. Lunar observations by MASCS and MDIS were captured two days before Earth closest approach. In addition to images of the Moon's disk, infrared and ultraviolet spectra were gathered of the area around the Moon and over the Orientale basin region. Observations of Earth around closest approach included MDIS imaging of the South American Galapagos Islands and Amazon basin, and MAG and EPPS sensing of the field and charged particle variations through the Earth's radiation belts. The MDIS wide-angle camera captured a "departure" movie of one full Earth rotation in the 24 hours following closest approach. The FIPS sensor on the EPPS instrument captured solar wind particle measurements a few days later. The MASCS instrument performed seven scans of the region around the Earth and Moon between August 5 and September 2 to observe the Earth's hydrogen corona.

Safing Events and Anomalies

For the most part, spacecraft operations since launch have proceeded as expected, but there have been some surprises. A larger than expected drift between the MP and IMU clocks has caused problems with the attitude estimation. IMU data are read by the 1553 bus every 10 ms as measured by the spacecraft clock. Since the IMU and MP clocks are not synchronized, the IMU measurements are not always separated by exactly 10 ms. The irregularity in the gyro data samples caused the estimator to attribute actual rotation about the Y axis to a gyro bias during the first few days after launch. The actual rotation rate was slowly decreasing from the commanded rate, while the estimated gyro biases were continually increasing [14]. This condition was corrected by stopping gyro bias estimation. The irregular IMU data pattern can also cause acceleration due to thruster firing to be overestimated. TCM 1 was scaled back from its original 21 m/s target magnitude to minimize this effect. An extensive analysis of the effects of the MP/IMU clock drift has been used to modify the flight software to improve handling of the actual data pattern as read from the IMU. The spacecraft transitioned to this new software on October 24, 2005.

To date, there have been four autonomous demotions to a safe mode. The first of these occurred during turns for IMU calibration in September 2004. Oversubscription of a data dump buffer caused a command to be rejected which in turn triggered the mode demotion. This event uncovered an error in conversion parameters in some autonomy rules when the SSPAs failed to turn on when expected after the demotion.

The second two demotions occurred in June and July 2005 when the spacecraft attitude put the Sun line closer to the star tracker boresights (-Z axis). Due to a flaw in the tracker optics design, a stray light path allows sunlight to degrade the tracker's ability to maintain attitude lock when the Sun is within 65° of its boresight. The tracker stopped producing attitude data during a MASCS star scan, and the subsequent slow drift away from the desired attitude eventually caused a safe mode demotion. The tracker was expected to lose lock again around TCM 6 because the Sun elevation would be in the stray light region when pointing the P thrusters in the ΔV direction. The IMU was reconfigured while tracker data were missing, causing the gyro rate data to be temporarily lost as well. The attitude control logic refused to turn the spacecraft back to its downlink attitude after the burn due to this loss of attitude knowledge. This event resulted in a mode demotion because the shade-to-Sun burn attitude was not compatible with the expected back-to-Sun downlink attitude.

The most recent safe-mode demotion occurred during the attempt to begin executing the new flight software on October 12, 2005. The primary MP was reset by ground command and the new software ran for several minutes. The G&C attitude estimator processed at least one erroneous star tracker measurement after the reset, causing the controller to initiate an unnecessary turn in the wrong direction to correct the perceived attitude error. The turn continued long enough for autonomy to force another MP reset when the attitude moved outside the SKI zone. The spacecraft was demoted to its lowest safe mode with the MP running the old version of the software. No problems had previously occurred on many occasions when powering up the tracker while the MP was already running the flight software. Therefore, the primary star tracker was turned off shortly before the MP was reset by ground command for the second attempt to switch to new software on October 24. A special autonomy rule was used to turn on the tracker and promote it to its normal operational mode shortly after the new software had started running. There were no unexpected attitude deviations or autonomy actions. The spacecraft has been successfully operating under the new software since that time.

Problems have also occurred with operating the Sun sensors and some of the science instruments. Sun sensor head 1, located on the top of the sunshade, started producing erroneous Sun direction readings after the flip on March 8, 2005. Manual commands are being included in spacecraft sequence loads to bypass use of this sensor head. The MAG heater operational cycle was found to cause interference with instrument readings. Changes to heater operation are being made to minimize this interference. The energetic particle spectrometer (EPS) sensor on the EPPS instrument has experienced periods of overly high currents when attempting to ramp up to high-voltage operational state. This behavior has been duplicated in ground tests, and strategies for achieving normal high-voltage operations are being investigated. Some of the MASCS scans have taken longer to complete than predicted in ground models. This situation has caused gaps in some of the scans since a new scan will not start if the previous scan is still in progress. Extra time is being added to scans while ground analysis and tests using the engineering model are performed. The gamma ray spectrometer (GRS) sensor on the GRNS instrument reaches lower temperatures than expected when not operating, causing its survival heater to power on more frequently than anticipated. The instrument is being maintained in a low-level state where its operational heater can be used to maintain the desired temperature in order to limit the temperature cycles on the cryocooler. Most of these instrument issues are relatively minor, with easily implemented work-arounds. They will not impact the ability to meet the science requirements for the mission.

5. FLYBYS AND ORBITAL OPERATIONS

Science observations during the three Mercury flybys and in orbit are designed to achieve the goals shown in Table 5. Previously unseen regions of the planet will be mapped during the flybys. The remaining data are obtained over the four Mercury years (one Earth year) to be spent in orbit. The spacecraft orbital period is 12 hours, and at least one 8-hour DSN tracking pass is planned each day. Science data will be collected for 16 hours of each 24-hour period (two orbits) and then downlinked during the 8-hour tracking pass.

The orbit moves around the planet, resulting in differing viewing geometry and thermal conditions. During dawn-dusk orbits, the spacecraft flies near the terminator, avoiding exposure to the illuminated planet disk. During noon-midnight orbits, the spacecraft flies over the illuminated planet at periapsis and over the dark side near apoapsis. Because the sunshade must face the Sun at all times, there are zones in many of the orbits where the planet cannot be viewed by the instruments inside the adapter ring. The instruments themselves also have certain observing constraints, such as the minimum altitude of ~1200 km required for the MLA to sense the return of its laser signals. Significant pre-launch analysis has been performed to optimize the time periods devoted to different observation types so that all regions of the planet are adequately covered.

A major part of the science planning for orbital observations is tailoring the amount of science data stored on the SSR to the varying downlink rates that can be supported as Mercury moves relative to the Earth in its orbit about the Sun. The average daily expected bit rate is 17.9 kbps when using the phased-array antenna transmitting to a 34-m beam waveguide DSN antenna. The actual daily bit rate will vary from a minimum of 5000 bps to a maximum of over 35 kbps. Periods of high and low data rate are interspersed throughout the 1-year orbital mission, with the highest rates occurring during the first six months. The amount of data stored on the SSR will necessarily track the available data rate, with peak usage at 6.6 Gbits in periods of low data rate. The total science data return over the one year in Mercury orbit is estimated to be 135 Gbits.

The orbital geometry imposes other constraints on spacecraft operation. Chief among these is the need to carefully manage power consumption during eclipses. Periods of short (less than 30 minutes) and long (30 to 60 minutes) eclipses occur throughout the orbital phase. A maximum of 20 W is allocated to instrument operation during these periods. The spacecraft top deck must be pointed away from the planet's surface during the daytime periapses in the noon-midnight orbits. Solar panel orientation will be changed when passing through these periapses to minimize thermal effects. Small Sun offsets from the –Y axis during the downlink passes will be used as a passive momentum control mechanism to minimize thruster dumps that can degrade the accuracy of the spacecraft's orbit solution.

Table 5. MESSENGER Science Goals

Map the elemental and mineralogical composition of Mercury's surface	**MDIS, XRS, GRNS, MASCS**
Image globally the surface at a resolution of hundreds of meters or better	MDIS
Determine the structure of the planet's magnetic field	MAG, EPPS
Measure the libration amplitude and gravitational field structure	MLA, RS
Determine the composition of the radar-reflective materials at Mercury's poles	GRNS, EPPS
Characterize exosphere neutrals and accelerated magnetosphere ions	MASCS/UVVS, EPPS

6. CONCLUSIONS

MESSENGER has successfully completed the first leg of its long journey to the innermost planet. The spacecraft and mission design, along with an extensive pre-launch testing program, provided the team with confidence that all mission goals can be accomplished. The first year of flight has allowed the flight team to streamline routine operations and gain experience responding to anomalies through successful recovery from four safing incidents. The team is moving forward with guiding the spacecraft into the extreme environments through the Venus and Mercury encounters. Once at its final destination, MESSENGER will provide the first views of over half of Mercury. Observations by the ensemble of science instruments will help to answer key questions about the planet itself and about solar system formation.

ACKNOWLEDGEMENTS

The MESSENGER project is sponsored by the NASA Science Mission Directorate as part of its Discovery Program. The authors wish to thank the spacecraft development and mission operations teams who made this mission possible.

REFERENCES

[1] C. L. Yen, "Ballistic Mercury Orbiter Mission via Venus and Mercury Gravity Assist," *Journal of the Astronautical Sciences, 37*, No. 3, 417-432, 1989.

[2] Andrew G. Santo et al., "The MESSENGER Mission to Mercury: Spacecraft and Mission Design," *Planetary and Space Science*, 49, 1481-1500, 2001.

[3] C. J. Ercol and Andrew G. Santo, "Determination of Optimum Thermal Phase Angles at Mercury Perihelion for an Orbiting Spacecraft," 29th International Conference on Environmental Systems, Society of Automotive Engineers, Tech. Paper Ser., 1999-01-21123, Denver, CO, July 21-25, 1999.

[4] Paul D. Wienhold and David F. Persons, "The Development of High Temperature Composite Solar Array Substrate Panels for the MESSENGER Spacecraft," *SAMPE Journal, Vol 39, No 6, Nov/Dec 2003*.

[5] Andrew G. Santo et al., "MESSENGER: The Discovery-class Mission to Orbit Mercury," International Astronautical Congress, World Space Congress, American Institute of Aeronautics and Astronautics, Paper IAC-02-U.4.1.04, Houston, TX, October 10-19, 2002.

[6] David F. Persons, Larry E. Mosher, and Theodore J. Hartka, "The NEAR Shoemaker and MESSENGER Spacecraft: Two approaches to Structure and Propulsion Design," 41st Structures, Structural Dynamics and Materials Conference, American Institute of Aeronautics and Astronautics, Paper AIAA-00-1406, Atlanta, GA, April 3-6, 2000.

[7] Dipak K. Srinivasan, Robin M. Vaughan, Robert E. Wallis, M. Annette Mirantes, T. Adrian Hill, S. Sheng Cheng, Jonathan R. Bruzzi., and Karl B. Fielhauer, "Implementation of an X-Band Phased-Array Subsystem in a Deep Space Mission," 2005 IEEE Aerospace Conference, Big Sky, Montana, USA, March 5-12, 2005.

[8] B. Q. Le, Sharon X. Ling, Larry R. Kennedy, George Dakermanji, and Sean C. Laughery, "The MESSENGER Power Distribution Unit Packaging Design," 21st Digital Avionics Systems Conference (DASC), Paper 9B3, Irvine, CA, October 27-31, 2002.

[9] George Dakermanji, Clark Person, Jay Jenkins, Deanna Temkin, and Larry R. Kennedy, "The MESSENGER Spacecraft Power System Design and Early Mission Performance," Seventh European Space Power Conference, Stresa, Italy, May 9-13, 2005 (ESA SP-589, May 2005).

[10] Samuel R. Wiley, Katherine Dommer, and Larry E. Mosher, "Design and Development of the MESSENGER Propulsion System," AIAA-2003-5078, AIAA Joint Propulsion Conference, Huntsville, AL, July 2003.

[11] W. Tam, Samuel R. Wiley, Katherine Dommer, Larry E. Mosher, and David F. Persons, "Design and Manufacture of the MESSENGER Propellant Tank Assembly," AIAA-2002-4139, AIAA/ASME/SAE/ASEE Joint Propulsion Conference and Exhibit, Indianapolis, IN, July 9-10, 2002.

[12] Sharon X. Ling, Richard F. Conde, and B. Q. Le, "A Light Weight Integrated Electronics Module (IEM) Packaging Design for the MESSENGER Spacecraft," The Proceedings of the 21st Digital Avionics Systems Conference (DASC), 9B4, Irvine, California, October 27 - 31, 2002.

[13] Christopher J. Krupiarz, David A. Artis, Andrew B. Calloway, C. M. Frangos, Brian K. Heggestad, Doug B. Holland, and William C. Stratton, "File-based Data Processing on MESSENGER," Proceedings of the 5th IAA International Conference on Low Cost Planetary Missions, Noordwijk, the Netherlands, September 2003.

[14]Robert C. Moore, “Safing and fault protection for a mission to Mercury,” 21st Digital Avionics Systems Conference (DASC), Paper 9A4, Irvine, CA, October 27-31, 2002.

[15] Robin M. Vaughan, Daniel J. O’Shaughnessy, Hongxing S. Shapiro, and David R. Haley, “The MESSENGER Spacecraft Guidance and Control System,” Flight Mechanics Symposium, Goddard Space Flight Center, Greenbelt, MD, October 18-20, 2005.

[16] Daniel J. O’Shaughnessy and Robin M. Vaughan, “MESSENGER Spacecraft Pointing Options,” AAS paper 03-149, AAS/AIAA Spaceflight Mechanics Conference, San Juan, Puerto Rico, February 2003.

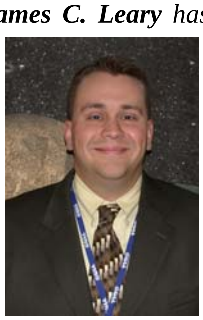
[17] Robert E. Gold, et al., “The MESSENGER Mission to Mercury: Scientific payload,” Planet. Space Sci., 49, 1467-1479, 2001.

[18] Robert E. Gold, Sean C. Solomon, Ralph L. McNutt, Jr., and Andrew G. Santo, “The MESSENGER Spacecraft and Payload,” International Astronautical Congress, World Space Congress, American Institute of Aeronautics and Astronautics, Paper IAC-02-Q.4.1.02, 9 pp., Houston, TX, October 10-19, 2002.

BIOGRAPHIES

Robin M. Vaughan *is a member of the Principal Professional Staff and a section supervisor in the Mission Design, Guidance and Control Group of the APL Space Department. She has been the MESSENGER guidance and control lead engineer since joining APL in 2000. Prior to joining APL, she worked on interplanetary navigation for the Voyager 2, Galileo, Cassini, and Mars Pathfinder missions at the Jet Propulsion Laboratory,California Institute of Technology in Pasadena, CA. She received a B.S. in mechanical engineering from Tulane University in New Orleans, LA, in 1981 and a master’s degree and Ph.D. in aeronautics and astronautics from MIT in Cambridge, MA, in 1983 and 1987, respectively.*

James C. Leary *has been a member of APL's professional staff since 1997. Currently, he is the Mission and Space Systems Engineering section supervisor in the Space Systems Applications Group of the APL Space Department. He served as Deputy Mission and Spacecraft System Engineer for MESSENGER until September 2003 when he was appointed to the Mission and Spacecraft System Engineer position. Currently, he serves as the Mission System Engineer during flight operations. James received a bachelor's degrees in physics and aerospace engineering from West Virginia University in 1995, followed in 1997 by a master's degree in Astronautics from George Washington University's Joint Institute for the Advancement of Flight Sciences at NASA Langley Research Center in Hampton, VA.*



Richard F. Conde *is is a member of the JHU/APL Principal Professional Staff and a section supervisor in the Electronics Systems Group in the Space Department. He received a B.S. degree in electrical engineering from Cornell University in 1981, an M.S. degree in electrical engineering from The Johns Hopkins University in 1985, and an Advanced Certificate for Post Master’s Study in computer science from The Johns Hopkins University in 1999. Since joining APL in 1981 he has led the development of a variety of critical space flight systems as a Lead Engineer, including the MESSENGER Integrated Electronics Module, the Advanced Composition Explorer Command & Data Handling System, the FEDSAT Adaptive Instrument Module, and the Midcourse Space Experiment Image Processor. He is currently Spacecraft Electrical System Engineer for the Radiation Belt Storm Probes project.*

Dr. George Dakermanji *heads the Power Systems Section in the Electronic Systems Group and is involved in all spacecraft power systems design and development activities at APL. In addition, he is the lead power systems engineer for the MESSENGER spacecraft. Prior to joining APL in 1993, he headed the power system group at Fairchild Space Company. His background and experience are in the areas of spacecraft power systems and power conditioning electronics.*

Carl J. Ercol *(Jack) is a member of the Principal Professionsl Staff in the Mechanical Systems group at APL. Mr. Ercol received a BSME from the University of Pittsburgh in 1982 and an MSME from the University of Maryland in 1985 where his graduate study was in heat transfer and thermodynamics. He has worked at APL since August, 1991, serving as the lead thermal control engineer for the Near Earth Asteroid Rendevous (NEAR) spacecraft and for the MESSENGER spacecraft. Before working at APL, Mr. Ercol was employed at the United States Naval Research Laboratory (NRL) were he began his career as a spacecraft thermal control engineer.*

Karl Fielhauer *is a senior engineer and supervisor in the RF Engineering Group of APL's Space Department. Mr. Fielhauer leads the Systems Engineering section within the APL's RF Engineering Group. He received a B.S. in 1985 from Lawrence Technological University in Southfield, Michigan, and an M.S.E.E. in 2002 from The Johns Hopkins University in Baltimore, Maryland. Before joining APL, Mr. Fielhauer worked for the Department of Defense and Litton's Amecom Division in College Park. Since joining APL's Space Department in 1997, he has focused primarily on the design and development of digital hardware for the TIMED and CONTOUR missions and their ground support equipment. Mr. Fielhauer was the Lead RF Communications Engineer on the CONTOUR Mission and has assumed the same role for the MESSENGER program. Mr. Fielhauer is a member of IEEE.*

David G. Grant *received a B.S.E.E. from Southeastern Massachusetts University in 1959 and an M.A. in applied mathematics from the University of Maryland in 1966. He joined APL in 1959 and has spent many years working in the Laboratory's Fleet Air Defense and SSBN Security programs. He became associated part-time with APL's Biomedical Engineering Program in 1967. In 1982 he was appointed a Program Manager in the APL Space Department. His major program management activities included the DoD Polar BEAR satellite program launched in 1986 and the NASA TIMED mission launched in 2001. He is currently the Program Manager of the MESSENGER mission launched in 2004. Mr. Grant was appointed to APL's Principal Professional Staff in 1970 and is a part-time Associate Professor of Biomedical Engineering in the Johns Hopkins School of Medicine.*

T. Adrian Hill *is a member of the Principal Professional Staff in the Embedded Applications Group at The Johns Hopkins University Applied Physics Laboratory. He was the Flight Software Lead for the MESSENGER mission. Before joining the Laboratory in 2000, he developed embedded software at Raytheon, GTE Spacenet, and Westinghouse Corp. He has previously led flight software development for satellites, including the Submillimeter Wave Astronomy Satellite (SWAS) and the Hubble Space Telescope (HST). He has a B.S.E.E. from The State University of New York at Buffalo and a M.S.C.S. from The Johns Hopkins University.*

Theodore J. Hartka *is a member of the Senior Professional Staff and a section supervisor in the Mechanical Systems Group of the APL Space Department where he has worked since 1988. He received a bachelor's degree in mechanical engineering from The Johns Hopkins University in 1987. He was the lead mechanical engineer for the MESSENGER spacecraft. Prior to MESSENGER, he was the lead mechanical engineer on the NEAR and APEX programs..*

Steve E. Jaskulek *is the MESSENGER Payload System Engineer. Mr. Jaskulek, at JHU/APL since 1981, heads the Enabling Technologies Section of the Lab's Space Instrumentation Group and is a leader for application specific integrated circuit (ASIC) development. He has extensive experience in the design, fabrication, testing, and integration of space instrumentation, and has been an instrument system engineer on a number of NASA planetary missions. On MESSENGER, Mr. Jaskulek is responsible for coordinating the technical and interface issues between the spacecraft and payload teams. He has a B.S.E.E. from Washington University in St. Louis and a M.S.C.S. from The Johns Hopkins University*

James V. McAdams *is a specialist in spacecraft trajectory optimization and maneuver design at The Johns Hopkins University Applied Physics Laboratory, where he has worked since 1994. He received B.S. and M.S. degrees in Aeronautical and Astronautical Engineering from Purdue University in the mid-1980s. During his eight years at Science Applications International Corporation in Schaumburg, Illinois, he participated in advanced mission studies to the Sun, asteroids, comets, and every planet in our solar system. His contributions to NEAR (Near Earth Asteroid Rendezvous) mission design continued from the formative phase of NASA's Discovery Program in 1989 to the February 2001 landing on asteroid 433 Eros. Mr. McAdams has served as the Mission Design Lead Engineer for MESSENGER since the project's inception.*

M. Annette Mirantes *is a member of the Senior Professional Staff in the Embedded Applications Group at The Johns Hopkins University Applied Physics Laboratory. She received a B.S. in electrical engineering from Purdue University and an M.S. in systems software engineering from George Mason University. She previously developed flight software for numerous missions at Orbital Sciences Corporation, including the ORBCOMM constellation, the Far Ultraviolet Spectroscopic Explorer (FUSE), and the Solar Radiation and Climate Experiment (SORCE). At APL, she has been the main processor flight software lead for the MESSENGER program.*

David F. Persons *is a section supervisor in the Mechanical Systems Group of the APL Space Department. He was the lead structural analysis and test engineer for the MESSENGER and NEAR spacecraft. Mr. Persons has served as a structural analyst for 23 years at JHU/APL. He received his B.S. in mechanical engineering from the University of Cincinnati in 1977, and his M.S in the same discipline from the University of Michigan in 1979..*

Dipak K. Srinivasan *is a member of the Senior Professional Staff in the RF Engineering Group of the APL Space Department. He received his B.S. and M.Eng. in electrical engineering in 1999 and 2000 in electrical engineering from Cornell University, and an M.S. in applied physics from The Johns Hopkins University in 2003. Mr. Srinivasan joined the APL Space Department in 2000, where he was the lead RF Integration and Test Engineer for the CONTOUR and MESSENGER spacecraft, and is currently an RF telecomm analyst for the MESSENGER project and lead system verification engineer for the New Horizons project.*