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The Johns Hopkins University Applied Physics Laboratory
Solomon, S. C.
Department of Terrestrial Magnetism, Carnegie Institution of Washington

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The Johns Hopkins University Applied Physics Laboratory; Laurel, MD 20723;
and Solomon, S. C.; Department of Terrestrial Magnetism, Carnegie Institution of
Washington, Washington, D.C. 20015
andrew.santo@jhuapl.edu / Fax: (240) 228-6635

ABSTRACT

MESSENGER is the MErcury Surface, Space ENvironment, GEochemistry, and Ranging mission to orbit and scientifically explore Mercury. The mission plan integrates launch in March 2004, two flybys of both Venus and Mercury, and Mercury arrival in April 2009. MESSENGER will orbit Mercury and carry out science measurements for one Earth-year. This paper describes the MESSENGER mission, focusing on the status of the spacecraft development. The MESSENGER team overcomes the obvious challenges with a low-cost, robust design enabling high quality science return and low technical risk.

INTRODUCTION

Mercury is the least explored inner planet and, as such, is a planet of many mysteries. Most of what is known about Mercury comes from the three Mariner 10 flybys in 1974 and 1975. In the decade following Mariner 10, it was generally thought that a spacecraft could not be inserted into orbit around Mercury by a conventional propulsion system, a view that colored the priority placed on further exploration of the planet. With the recognition that gravity-assisted trajectories(1), using the launch systems available in the Discovery Program, now permit the insertion of a spacecraft into Mercury orbit, the need for a scientific exploration of the innermost planet has again been recognized.

To implement this Discovery-Class mission to Mercury, the Carnegie Institution of Washington and The Johns Hopkins University Applied Physics Laboratory head a consortium to provide the spacecraft and scientific instrumentation. The consortium team members include Composite Optics, Inc., providing the lightweight spacecraft structure, GenCorp Aerojet, providing the spacecraft propulsion system, and Goddard Space Flight Center, the University of Colorado, and the University of Michigan providing instruments. Co-engineered with planetary-science specialists from twelve institutions, MESSENGER has been designed to accommodate the severe environment, supply the required large spacecraft velocity change, and facilitate all science observations.

The MESSENGER spacecraft is presently in fabrication, having successfully completed its Critical Design Review in March 2002. After moving through a one-year integration and test period (starting January 2003), the spacecraft is scheduled for launch in March 2004 aboard a Delta II 7925H-9.5 from Cape Canaveral, Florida. It will enter Mercury orbit in April 2009 following two flybys of Venus and two flybys of Mercury. Orbital data collection concludes in April 2010. Science data processing, archiving, and analysis are funded through September 2011. Further information on the MESSENGER mission may be found in earlier publications(2-4).

SCIENCE AND PAYLOAD OVERVIEW

The MESSENGER scientific mission(2) is focused on answering six key questions that will allow us to understand Mercury as a planet:

1. Why is Mercury so dense?
2. What is the geologic history of Mercury?
3. What is the structure of Mercury's core?
4. What is the nature of Mercury's magnetic field?
5. What are the materials at Mercury's poles?
6. What volatiles are important at Mercury?

The MESSENGER science payload(3) was carefully chosen to answer these questions. The Data Processing Unit (DPU) accomplishes much of the collective processing for the payload and provides the primary interface between the spacecraft and the payload. Most of the instruments are fixed-mounted, so coverage of Mercury is obtained by spacecraft motion over the planet. The instruments and their role in addressing the science questions follow:

Mercury Dual Imaging System (MDIS)

The MDIS narrow-angle imager and wide-angle multispectral imager map landforms, surface spectral variations, and topographic relief from stereo imaging. A single-degree-of-freedom pivot platform assists in pointing.

Gamma-Ray and Neutron Spectrometer (GRNS)

The gamma-ray sensor (GRS) measures emissions from radioactive elements and cosmic-ray-stimulated gamma-ray fluorescence and maps elemental abundances in crustal materials. The neutron sensor (NS) provides hydrogen sensitivity to ice at Mercury's poles.

X-Ray Spectrometer (XRS)

XRS is designed to measure the fluorescence in low-energy X-rays stimulated by solar gamma rays and high-energy X-rays and map elemental abundances in crustal materials.

Magnetometer (MAG)

The magnetometer, attached to a 3.6-m boom, maps the detailed structure and dynamics of Mercury's magnetic field and searches for regions of magnetized crustal rocks.

Mercury Laser Altimeter (MLA)

MLA couples an infrared laser transmitter with a receiver that measures the round-trip time of a laser burst reflected off Mercury's surface, yielding a distance measurement. It produces highly accurate measurements of topography and measures Mercury's libration.

Mercury Atmospheric and Surface Composition Spectrometer (MASCS)

The ultraviolet-visible spectrometer (UVVS) measures abundances of atmospheric species, and the visible-infrared spectrometer (VIRS) detects minerals in surface materials.

Energetic Particle and Plasma Spectrometer (EPPS)

EPPS measures the composition, spatial distribution, energy, and time-variability of charged particles within and surrounding Mercury's magnetosphere.

Finally, Radio Science (RS) uses the Doppler effect (the shift in the frequency of the spacecraft radio signal with changes in the spacecraft velocity relative to Earth) to measure Mercury's mass distribution, including spatial variations in crustal thickness. The MESSENGER spacecraft provides ample mass and power (Table 1) to the payload to help alleviate design complexity.

Table 1: Science Payload Characteristics

Payload	**Mass (kg)**	**Power (W)**
DPU (2)	3.3	4.2
MDIS	6.8	6.7
GRNS	13.4	23.6
XRS	3.4	5.4
MAG & boom	3.7	5.3
MLA	6.9	25.6
MASCS	3.1	5.9
EPPS	2.6	6.4
Misc. (harness, reserve, etc.)	6.8	1.3
RS	*	*
Total	**50.0**	**84.4**

* RS is included as part of engineering subsystems

MISSION DESIGN

The MESSENGER trajectory begins with launch on March 10, 2004, continues with gravity assists at Venus (twice) and Mercury (twice), and arrives at Mercury on April 6, 2009. The cruise phase offers opportunities for early science during Venus and Mercury flybys. The Mercury orbit phase lasts one Earth-year.

Trajectory optimization for a Delta II 7925H launch vehicle with 9.5-foot-diameter fairing yields two launch periods in 2004. The first launch window is 20 days in duration (March 10-29, 2004). Starting on May 14, 2004, the second launch window duration is only 7 days.

Figure 1 displays the ecliptic plane projection of the spacecraft heliocentric trajectory, along with a key-event timeline. The heliocentric trajectory begins with a transfer orbit to Venus. Between the June 24, 2004, and March 16, 2006, Venus flybys, the spacecraft orbits the Sun 3.7 times, while Venus completes only 2.8 orbits. Mercury flybys include 200-km altitude, dark-side close approaches on July 21, 2007, and April 11, 2008. Deep-space maneuvers follow each Mercury flyby; both target the spacecraft for the next Mercury encounter. At orbit insertion, which occurs from April 6-8, 2009, the spacecraft enters a 12-hour, 200-km by 15,195-km-altitude orbit inclined 80° to Mercury's equator (Figure 2). This near-polar orbit inclination maximizes science return and minimizes the total time that Mercury interferes with communications. The maximum solar eclipse time during orbital operations is just over 60 minutes. Over the orbital lifetime, solar perturbation forces raise the periherm (orbit minimum) altitude. Therefore, every 88 days (one Mercury year), a two-burn sequence is executed to reduce the periherm altitude to 200 km and adjust the period to 12 hours.

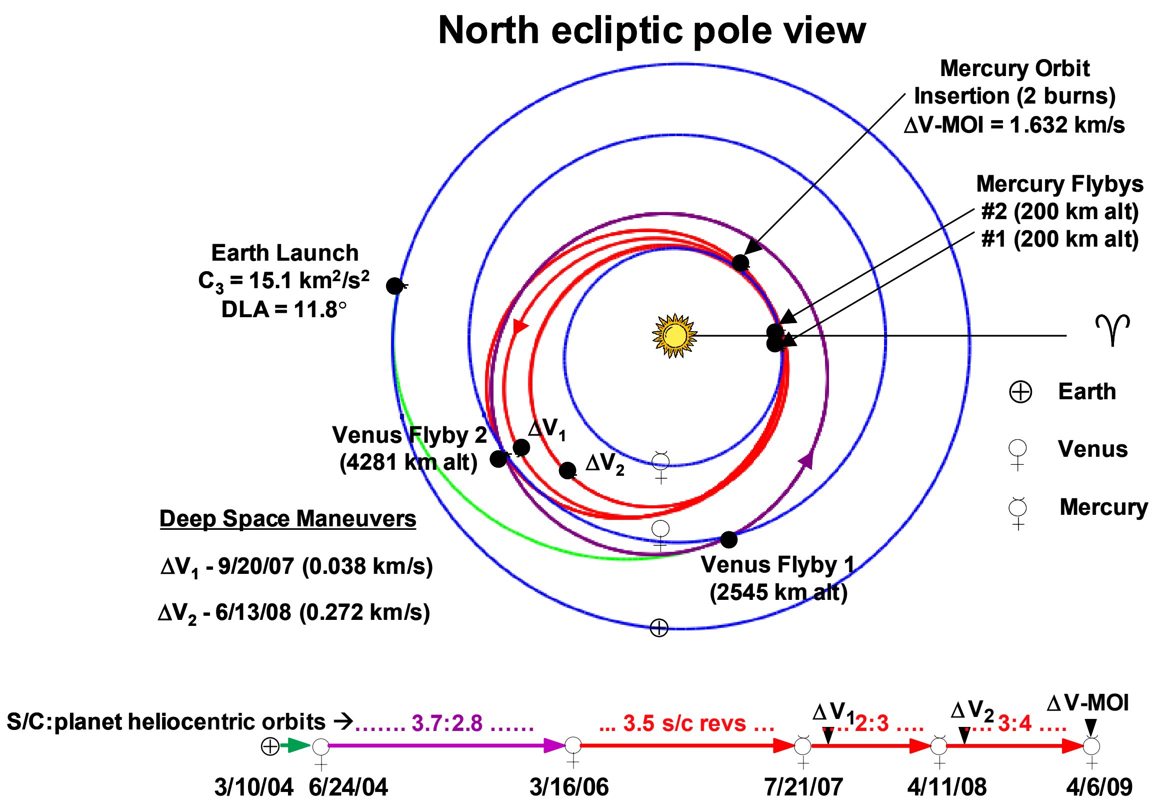

Figure 1: Baseline Mission Trajectory

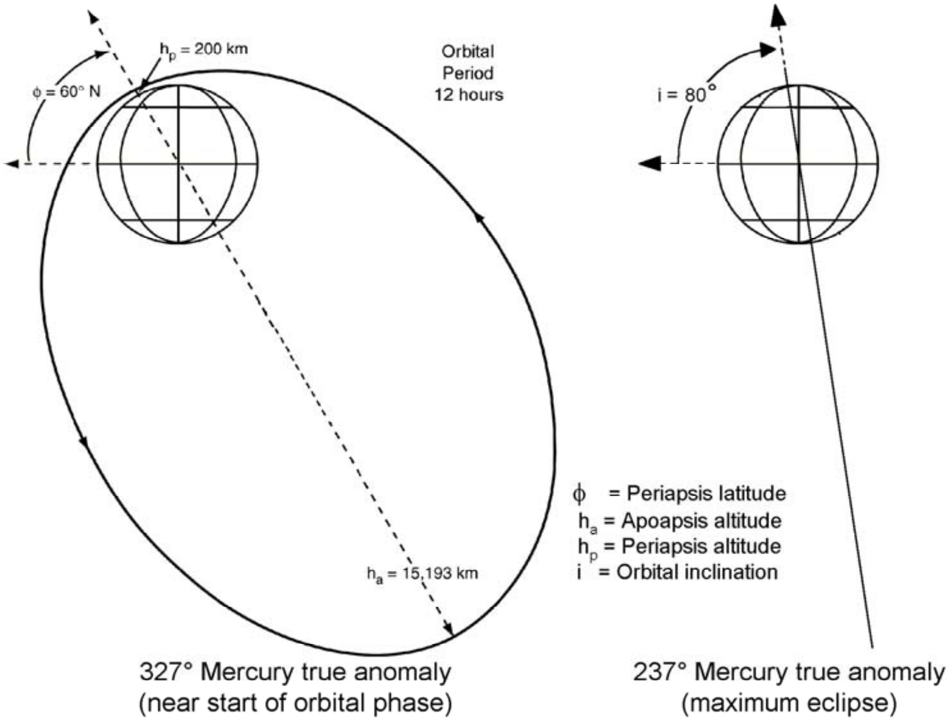

Figure 2: Mercury Orbit

OPERATIONS CONCEPT

Approximately 45 minutes after launch, the spacecraft separates from the launch vehicle third stage. At third-stage separation, the responsibility for spacecraft attitude control passes from the launch vehicle to the spacecraft guidance and control subsystem. The spacecraft detumbles using thrusters, and the solar panels are deployed and pointed at the Sun. From launch until separation, the spacecraft is battery powered. The forward and aft hemispherical low-gain antennas are connected to the redundant transponders to yield effective omni-directional coverage. The first scheduled ground contact is 50 minutes after launch. Twenty-four-hour communication coverage is scheduled for the first five days of operations.

Cruise phase is divided into two sections. During the outer-cruise phase, while the Sun distance is greater than 0.85 AU, the sunshade is pointed away from the Sun to reduce heater power requirements. During the inner-cruise phase, inside a Sun distance of 0.85 AU, the sunshade is pointed toward the Sun to protect the spacecraft from direct solar illumination. Cruise phase is designed to minimize operations support except for short periods of intense activity around the flybys. During the first few weeks of cruise, a series of component functional tests check the health of the spacecraft and the payload. Also during this time low-level burns are performed to correct for any trajectory errors, and the magnetometer boom is deployed. After this initial checkout period, the spacecraft maintains a low level of activity and communicates through the medium-gain antennas. Ground contacts are scheduled for an eight-hour duration, twice per week. During ground contacts the current spacecraft health is analyzed, the next week's series of command sequences are uploaded, and the recorded telemetry is dumped.

Communication with the spacecraft is accomplished using the NASA Deep Space Network (DSN). The 34-meter diameter antennas are used for routine uplink and downlink coverage. The 70-meter diameter antennas are used for emergency communications and short durations surrounding the flybys and critical propulsive maneuvers.

During the orbital phase, one eight-hour 34-meter DSN pass is scheduled daily, with 1.5 hours of each pass allocated for DSN and tracking overhead. Doppler tracking and ranging are used to satisfy navigation and science requirements. Throughout the orbital phase, both of the 11-W on-board transmitters are used simultaneously to boost downlink signal levels and maximize science return. During each pass, the mission operations center monitors spacecraft health and downlinks recorded data captured from the previous day.

SPACECRAFT DESCRIPTION

The MESSENGER system[4] is built around off-the-shelf components and features a distributed architecture. This design approach allows parallel subsystem development, test, and integration so that a compressed spacecraft development and integration schedule can be accommodated. The use of the standard 1553 bus as a data interface among processors gives the system built-in redundancy and fault tolerance, flexible software-defined interfaces, compatibility with many commercially available components, and a reduction of interconnecting harness. The spacecraft in the deployed flight configuration is shown in Figure 3.

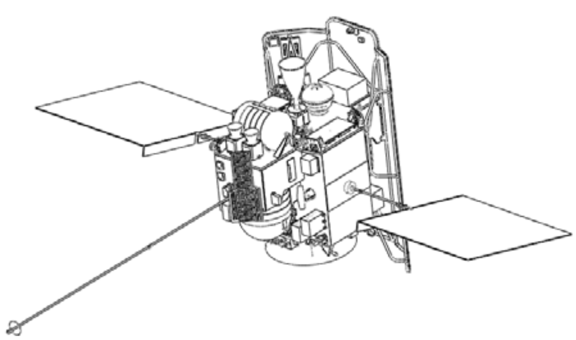

Figure 3: Flight Configuration

A dual-mode chemical propulsion system is integrated into the spacecraft structure to make economical use of mass. The large side panels, two interior vertical panels, and the bottom deck are used to mount the propulsion hardware and the spacecraft electronics. The locations of the propellant tanks are selected to maintain the spacecraft center-of-mass as the propellant is consumed. Attached to the exterior of the bottom deck is the launch adapter. With the exception of the launch adapter and the radiators, which are made of aluminum, the structure is entirely made of graphite composite.

Thermally, MESSENGER employs a simple design utilizing a large ceramic-cloth sunshade attached to one side of the spacecraft. The sunshade is sized to protect the spacecraft from the thermal energy of the Sun at Mercury, while allowing up to 15° off-Sun pointing in yaw and pitch. The two small side panels adjacent to the solar arrays are used as thermal radiator surfaces. On the top deck, there are two smaller radiator panels used to dissipate the heat from the solid-state power amplifiers. Two large solar panels (5.3 m^2 total area) are designed to provide spacecraft power from 1.01 AU to 0.30 AU and are supplemented with a 23 A-hr NiH_2 battery.

The “brains” of the spacecraft consist of redundant integrated electronics modules (IEMs) that house two processors (Main Processor at 25 MHz and Fault Protection Processor at 5 MHz) each. Attitude determination is performed using redundant star cameras and an Inertial Measurement Unit (IMU) containing four gyros and four accelerometers. Six Digital Solar Sensors (DSSs) provide backup. Attitude control is primarily accomplished using four reaction wheels, with thruster control when necessary.

The primary means of spacecraft communication is via circularly polarized X-band phased array antennas. Two electronically steerable phased arrays, one looking through the sunshade and the other on the opposite side of the spacecraft, are used to replace a gimbaled high-gain antenna. The phased arrays are simple, passive waveguide-based antennas fed by solid-state power amplifiers that are located inside the spacecraft body. Together, the phased arrays have a field-of-regard that allows high-gain communication over the entire orbital phase while the spacecraft maintains sunshade pointing for thermal protection.

Most of the miniaturized instruments are located on a science deck facing Mercury. All of the instruments are fixed-mounted to the outside of the spacecraft, except the magnetometer, which is boom-mounted. Except for the EPPS and GRNS, body-mounted instruments are fixed to the bottom deck and point in a common direction, outward and perpendicular from the bottom deck. The

EPPS is mounted on the top deck and the neutron portion of the GRNS is mounted on the side of the "hatbox" covering the tank on the back of the spacecraft.

A block diagram of MESSENGER showing component redundancy and heritage is illustrated in Figure 4. All spacecraft-critical components that have a credible failure mode are block redundant or degrade gracefully. All devices with a limited lifetime, e.g., array drives, gyros, and reaction wheels, are carefully selected to exceed the maximum 8-year lifetime requirement.

Table 2 shows the mass breakdown of the spacecraft at the major milestones of development and the current best estimate (CBE). The maximum launch mass of the spacecraft is approximately 1115 kg for the baseline mission.

Table 2: Subsystem Mass Breakdown

	Concept Study Mass w/ Reserve (kg)	CoDR/SRR Mass (kg)	PDR Mass (kg)	CDR Mass (kg)	CBE Mass (kg)
Payload	**37.10**	**36.50**	**47.50**	**46.31**	**47.23**
Spacecraft					
IEM	**14.90**	**11.88**	**11.84**	**12.06**	**12.06**
Power System	**62.10**	**65.11**	**71.29**	**87.72**	**89.94**
RF	**23.00**	**22.81**	**23.04**	**24.62**	**24.62**
Attitude Control	**29.20**	**23.94**	**29.60**	**32.81**	**33.86**
Thermal	**28.90**	**28.43**	**31.23**	**34.36**	**35.95**
Propulsion	**63.90**	**72.87**	**81.96**	**83.92**	**81.73**
Structure	**76.20**	**70.24**	**81.53**	**110.59**	**118.88**
Harness	**24.00**	**25.75**	**24.95**	**21.55**	**21.55**
Total Dry Mass	**359.30**	**362.76**	**407.39**	**453.94**	**465.82**
Total S/C Mass	**995.30**	**1029.69**	**1011.07**	**1061.42**	**1073.30**
Max Launch Mass	**1066.00**	**1097.07**	**1097.07**	**1115.20**	**1115.20**
Mass Margin	**16.44%**	**15.73%**	**17.43%**	**10.59%**	**8.25%**

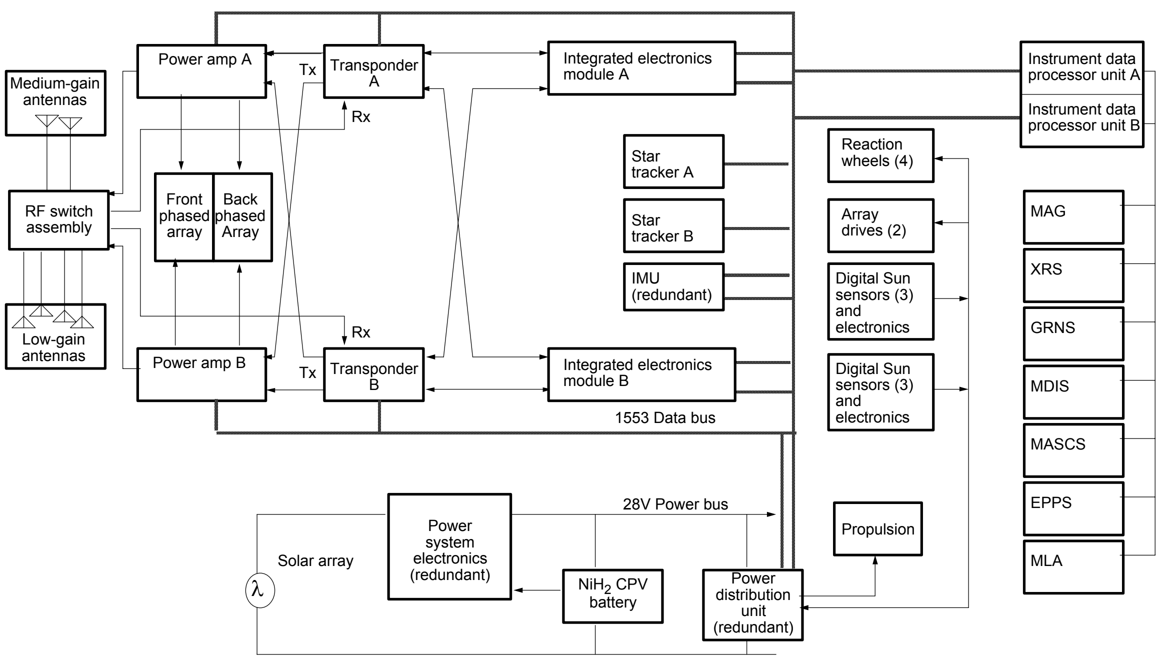

Figure 4: Schematic Block Diagram

<u>Propulsion Subsystem</u>

The state-of-the-art, regulated, dual-mode propulsion subsystem, shown in Figure 5, is required to provide > 2300 m/s of velocity change (ΔV) for the spacecraft over the mission. Three identical, low-mass (~9 kg each) titanium main propellant tanks, two fuel and one oxidizer, feed the thruster complement through filters, activation-surge limiting venturis, and flow-sequencing latch valves. By using identical tanks, tank-qualification, manufacturing, and tooling costs are minimized. A small auxiliary diaphragm fuel tank is used in a blow-down mode to provide propellant for settling burns, attitude control, momentum dumping, and small ΔV maneuvers.

A composite overwrap pressure-vessel helium tank feeding two parallel series-redundant regulators through filters and high-pressure latch valves provides pressurization for the propulsion subsystem. Pyrotechnic isolation valves (normally open) are in place to cross-strap the pressurization system in the event of a regulator or high-pressure latch valve failure. Regulated pressure feeds the propellant tanks through redundant check valves and, on the oxidizer side, an additional isolating latch valve. The pressurization system is similar in design philosophy to the Near Earth Asteroid Rendezvous (NEAR) configuration built by team member Aerojet.

The main thruster is a 660-N, 317-s I_{sp} LEROS-1b bipropellant unit. Four 22-N, 290-s I_{sp} bipropellant thrusters provide steering forces during main thruster burns and primary propulsion for most small ΔV maneuvers. Two are used for settling burns prior to bipropellant burns. Twelve 4.4-N, 220-s I_{sp} monopropellant thrusters are arranged in two double-canted sets of four for attitude control, two for anti-sunward thrusting, two for toward-sun thrusting. The 4.4-N thrusters provide redundant three-axis attitude control, pure-couple momentum-dumping torques, propellant settling forces, and fine ΔV control in all three axes.

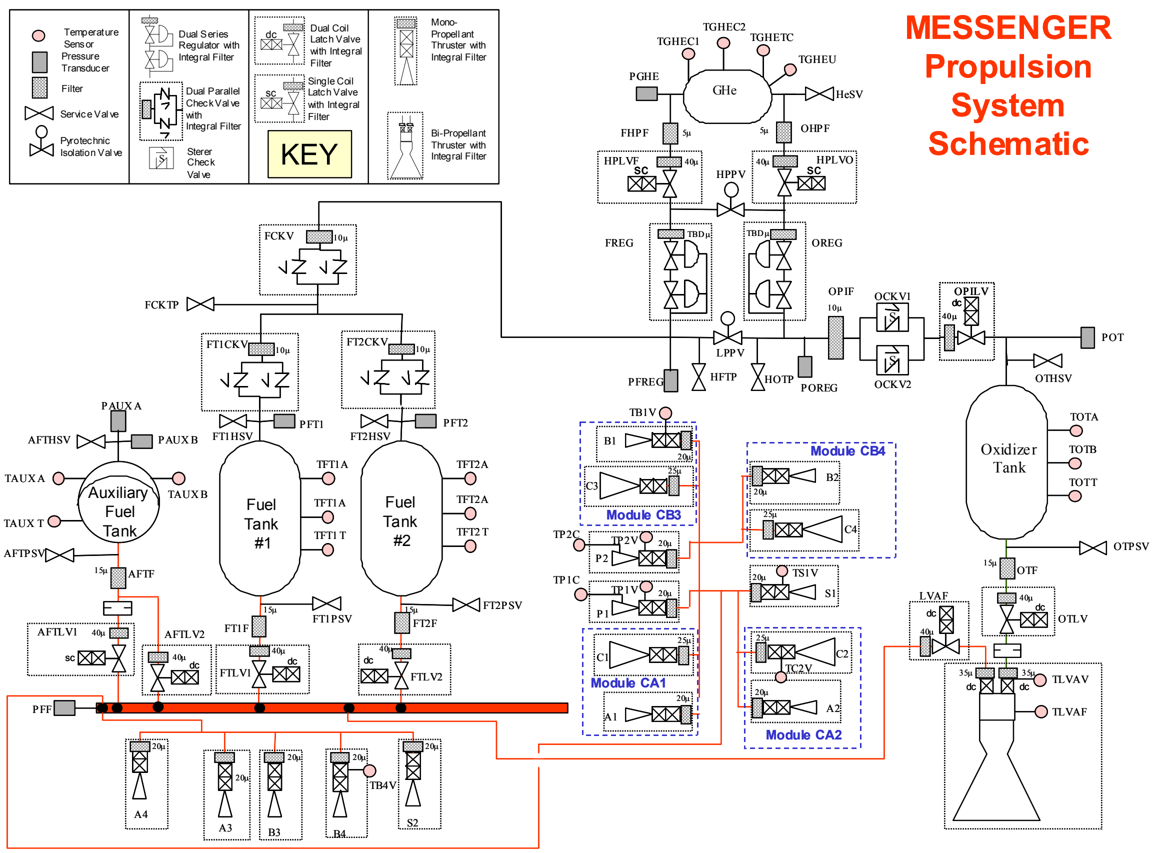

Figure 5: Propulsion Schematic

Thermal Subsystem

The thermal design[5] requires no louvers or other mechanisms, uses readily available materials, and needs little heater power. The general approach for thermal management is to reduce the heat load on the Sun side of the spacecraft via the sunshade to the point where radiating panels on the top and sides of the structure can remove any excess heat. The radiators are optimized to maximize heat rejection while minimizing worst-case infrared (IR) heat absorption from the hot side of Mercury.

The radiators have sufficient total area to handle the worst-case heat load. Diode heat pipes are utilized under components with high watt-densities to help distribute heat to the radiators. The solar arrays are placed 0.76 m from the side radiator panels to eliminate radiative coupling.

A detailed time-stepped computer model was developed to compute the spacecraft orbit-average and transient temperatures over a Mercury year. The model was checked by comparing temperature predictions for the solar array, thermal shade, and spacecraft core against published results from Mariner 10 when simulating similar environmental conditions.

The sunshade is constructed from layers of 3M Nextel ceramic cloth (aluminized and non-aluminized), rated to 1000°C, and conventional all-Kapton multi-layer insulation (MLI), rated in excess of 400°C. The predicted worst-case temperature for the front of the sunshade is 350°C, and the sunshade attenuates that to < 100°C on the spacecraft side.

Guidance and Control Subsystem

The Guidance and Control (G&C) subsystem is required to provide 3-axis knowledge to within 350 μrad, 3-axis pointing to 1.7 mrad, and instrument line-of-sight stability to within 25 μrad over an integration time of 0.1 seconds. It is composed of a suite of sensors for attitude determination, actuators for attitude corrections, and software within the core avionics to provide continuous, closed-loop attitude control and thruster control. In operational mode, the attitude is controlled to a commanded pointing scenario. In safe modes, the G&C maintains the sunshade pointing to protect the spacecraft from overheating, the rotation of the solar array to limit panel temperatures, and the position of the antennas with respect to Earth to establish ground communications. Finally, the G&C subsystem must recognize internal failure modes and initiate autonomous actions to correct them.

The sensor suite contains redundant digital Sun sensor assemblies, star trackers, and inertial measurement units (IMUs). The star tracker data and 3-axis gyro data provide on-board attitude determination, while the coarse Sun (angle) sensor data are used for fault protection. These robust, simple sensors provide a reliable measurement of the spacecraft attitude with respect to the Sun. If a ground or spacecraft failure causes the spacecraft attitude to go outside of the sunshade boundary, a safe mode is entered in which backup systems, including the backup main computer and a backup attitude control algorithm that is very simple and robust, are selected.

The actuator complement contains four reaction wheels, twelve monopropellant thrusters, four bipropellant thrusters, and one large bipropellant thruster. All normal attitude control will be achieved using the reaction wheels alone. The four reaction wheels are aligned such that any three provide complete 3-axis control, so a single reaction wheel failure will result in no reduction in functionality. Redundant 3-axis accelerometers are included within the IMUs for closed-loop thruster control.

RF Telecommunication Subsystem

The RF telecommunications subsystem (Figure 6) incorporates redundant off-the-shelf General Dynamics small deep-space transponders (X-band) to provide command, telemetry, and high-precision tracking and ranging capability. The downlink hardware uses a solid-state power amplifier with an RF output level of 11 W. Two identical, circularly polarized, 15-cm x 76-cm phased arrays, mounted on opposite sides of the spacecraft, eliminate the need for a gimbaled high-gain antenna. These arrays, constructed from established aluminum waveguide technology, are rugged to withstand the high-temperature environment. Each array has an electronic scan angle of ± 45˚ in one dimension and is transmit-only. Each array incorporates a newly developed method for generating circularly polarized radiation from narrow-wall, slotted waveguide sticks. Controlling the beam direction in one axis and rotating the spacecraft about the spacecraft-Sun line provides the two degrees of freedom necessary to cover one entire hemisphere with each antenna. During the orbital phase, both solid-state power amplifiers are used simultaneously to return over 77 Gbits over the year. Two identical medium-gain, dual-frequency fanbeam antennas (one co-located with each phased array) provide for normal uplink, low-data-rate downlink, and emergency-mode communications. These medium-gain antennas each have 3 dB beamwidths of 90˚ x 7.5˚. There are also four dual-frequency, low-gain antennas on the spacecraft. Two of these antennas provide dual hemispherical antenna coverage during the near-Earth and emergency mode phases of the mission, while also providing communications during major engine burns. Rate 1/6 Turbo coding is used for all downlinks.

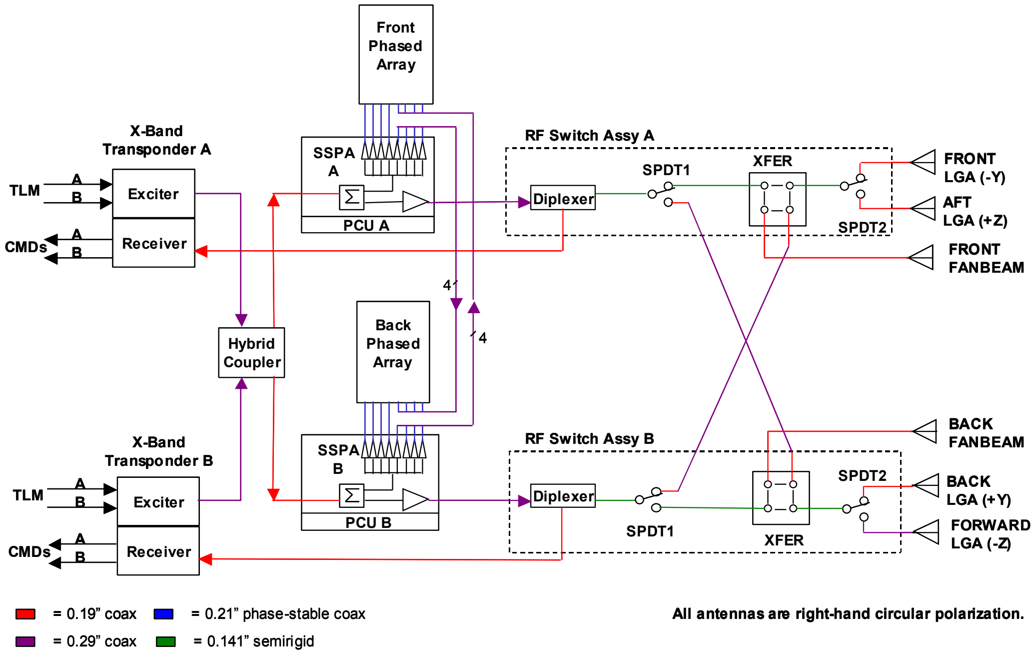

Figure 6: RF Subsystem Block Diagram

<u>Integrated Electronics Modules</u>

Within redundant integrated electronics modules (IEMs) there are two computers. One computer is dedicated for command management, telemetry management, and guidance and control. It is a RAD6000 running at 25 MHz and referenced as the main processor (MP). The second computer is dedicated for fault protection. It is also a RAD6000, but runs at only 5 MHz to conserve power. It is referenced as the fault protection processor (FPP). The MPs are on switched power, and normally only one is used at a time. The FPPs are on unswitched power, and each independently checks the health of the spacecraft throughout the mission. Each IEM also includes an oven-controlled crystal oscillator (OCXO) that provides after-the-fact timing accuracies on the order of ± 1 ms required for altimetry.

The command function operates on cross-strapped inputs from the two command receivers at one of four rates: 500 bps, 125 bps, 31.25 bps or 7.8125 bps (emergency rate). The format of the uplinked commands is Consultative Committee for Space Data Systems (CCSDS) compliant, with a separate virtual channel for each side of the redundant IEM. Commands can be executed in real time or can be stored for later execution. Execution of stored commands can be triggered by reaching a specific mission elapsed time, or by the detection of a spacecraft condition that must be corrected autonomously.

The telemetry function collects engineering status and science data from the housekeeping interface, from dedicated serial interfaces, from the remote terminals on the 1553 bus, and from the internal MP event history buffers. These data are packetized where necessary, and then packed into CCSDS-compliant transfer frames for recording or real-time downlink. Each IEM has 8-Gbit of solid-state recorder memory for telemetry storage. The recorder is formatted with a simple DOS-like file system to assist with file prioritization and recorder management. Recorded data are read back, packed into compliant transfer frames, and placed into the downlink on command. Recorder playback data are interleaved with real-time data on the downlink. The interleave factor is programmable, but it is planned to have approximately 97% of the downlink bandwidth dedicated for recorder playback data. The downlink of recorded data is managed with the CCSDS File Delivery Protocol (CFDP). MESSENGER is one of the first missions to use high-reliability CFDP.

Since the mission is downlink-rate limited, the downlink data rate is selectable between 10 and 104,167 bps to match the capability throughout the mission. While the MP software controls the rate of collection of real-time data to match the real-time downlink rate, the rate at which data are placed on the recorder is under the control of the subsystems. Each remote terminal on the military-standard-1553b data bus can request the MP to pick up and record a variable amount of bits of data per second. This feature allows the spacecraft operators complete flexibility with respect to the bandwidth used by each instrument and spacecraft subsystem.

The fault protection function is carried out using rules that invoke command macros in the FPP. These rules compare any of the housekeeping data (e.g., analog, switch telltale, digital from 1553 remote terminals) to limits or ranges that indicate anomalous operation. Separate comparisons can be made on multiple housekeeping items per rule and the results and'ed or or'ed together. Each rule is associated with a timer, and the detected condition must persist long enough for rule-timeout before action is taken. Once a rule detects an anomalous condition, a command (usually a 'macro execute' command that invokes a series of commands) is executed to correct the condition and safe the spacecraft. While the FPP carries a set of default rules and command macros that will be loaded in case of processor reset, any of the rules can be deleted or changed, new rules can be added, and the command macros modified to meet changing conditions throughout the mission.

Power Subsystem

The power subsystem employs a peak-power tracking architecture. The spacecraft main bus is tied directly to the battery, maintaining the bus voltage at an unregulated 20-35V. A pulse-width modulated buck regulator is located between the array and the spacecraft bus. Pulse-width modulation (PWM) is controlled by the spacecraft main processor to optimize and regulate array power output.

The power system electronics contains the components required to regulate the power bus, control the battery charging, and interface command and telemetry with the spacecraft. The battery is a 23-Ah NiH_2 common pressure vessel (CPV) design now flying on several spacecraft. The CPV design offers significant mass and volume savings by combining two cells into each pressure vessel. The battery assembly consists of eleven vessels equipped with bypass circuitry in case of vessel failure. The bus voltage range reflects operation in the nominal and one-failed-vessel scenarios. Battery discharge is $\leq$ 60% for launch and orbital phases.

The solar array[(6)] consists of two panels totaling 5.3 m^2 of area. The solar array is packed on one side with two rows of optical solar reflectors (OSRs) for every row of high-efficiency, triple-junction cells. The backside is covered with aluminized Kapton to protect against planetary heating. The arrays extend beyond the sunshade and are connected to drive motors whose rotation axis is normal to the Sun-line to allow the panels to be rotated off solar normal to maintain an operational temperature under 150°C. Transient temperature spikes to 180°C, lasting under 15 minutes, occur during hot planet crossings when the panels are rotated edge-on to the Sun. The panels are qualified to withstand direct solar illumination at 0.3 AU, which raises the average panel temperature to 280°C.

The power provided by the solar array is a function of the spacecraft-Sun distance, the panel temperature, and the incident solar angle. At all times throughout the mission there is at least a margin of 10% on all nominal loads and a 10% margin on critical loads during emergency recovery. The panel size is driven by the power required immediately after launch, when the arrays generate their least amount of power. Table 3 details the power requirements over the various mission phases.

Table 3: Spacecraft Power Loads

	Outer Cruise @1.0115 AU	Inner Cruise <.85 AU	Worst Case Orbit
Science Instruments & Heaters	39.0	39.5	84.4
IEM	34.8	38.6	38.6
Power System	33.0	33.0	105.0
Telecommunications	76.6	76.6	128.6
Attitude Control	86.8	86.8	102.8
Thermal	30.0	30.0	30.0
Propulsion	52.0	144.0	118.0
Harness	5.6	7.0	9.7
Total Spacecraft Power (W)	357.8	455.5	617.1
Array Output -or- PSE Capability (W)	402	540.0	720.0
Power Margin (%)	11%	16%	15%

SUMMARY

MESSENGER is the MErcury Surface, Space ENvironment, GEochemistry, and Ranging mission to orbit Mercury for one Earth-year after completing two flybys of that planet and two of Venus. The orbital phase, guided by flyby data, will be a focused scientific investigation of this least-studied terrestrial planet. MESSENGER uses an advanced chemical propulsion system. The system design combines generous margins with appropriate technologies to define a mission that is able to address the key science questions at minimum cost and low risk.

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