The MESSENGER Spacecraft

The MErcury Surface, Space ENvironment, GEochemistry, and Ranging (MESSENGER) spacecraft was launched from the Cape Canaveral Air Station on 2004-08-03, on an approximately 8 year mission to become the first probe to orbit the planet Mercury. The initial mission included one year orbit of Mercury which was subsequently extended to a total of four years.

Most of the science data collected by the MESSENGER mission originates from instruments on the spacecraft and are relayed via the telemetry system to stations of the NASA Deep Space Network (DSN). Radio Science (RS) experiments (Doppler velocity and ranging as well as radio occultations) require that DSN hardware also participate in the data acquisition. The following sections provide an overview first of the spacecraft, then of the DSN ground system, and the spacecraft clock reset and use of MET partitions.

For more information on the spacecraft see Leary et al., 2007. For more information on the DSN see Asmar and Renzetti, 1993.

1. Instrument Host Overview - Spacecraft

The MESSENGER spacecraft was built by the Johns Hopkins Applied Physics Laboratory (JHUAPL) to withstand the harsh environments encountered in achieving and operating in a Mercury orbit. It can be divided into eight subsystems: structures and mechanisms; propulsion; thermal; power; avionics; software; guidance and control; radio frequency (RF) telecommunications, and payload.

a. Structures and Mechanisms

The spacecraft's structures and mechanisms consist of the composite core structure, the aluminum launch vehicle adapter, and its deployables.

The core of the spacecraft is the integrated structure/propulsion system. The structural load paths are optimized by using lightweight titanium fuel tanks designed specifically for the structural configuration chosen. The tank struts transfer lateral loads to the corners of the structure, allowing the use of composite panels that are thin relative to their size. The composite structure is designed to channel all loads into a center column, which necessitates a square-to-round launch vehicle adapter. This was a machined aluminum forging, carefully tailored to distribute evenly the structural loads from the corners of the center column to the round vehicle interface.

The solar panel design and development effort proved to be challenging from a material engineering prospective. Extensive, ply-by-ply structural analysis was required for the solar array substrate due to the large cantilever in the stowed configuration and the use of high-conductivity, but relatively low-strength, graphite-cyanate ester (GrCE) materials in the sandwich face sheets.

The sunshade support structure is welded titanium tubing construction. The tubing supports five antennas, the solar monitoring sensor for the X-Ray Spectrometer (XRS), four digital Sun sensors, and the sunshade. The final shape of the sunshade projected area was tailored to bring the center of solar pressure as close to the measured center of mass of the spacecraft as possible. By utilizing solar pressure, tilting the spacecraft relative to the Sun can unload the momentum wheels without expending fuel.

Three mechanical assemblies required deployment: the two solar panels and the 3.6-m Magnetometer boom. The solar array panels were released first and allowed to settle; the arms were then released. The Magnetometer (MAG) boom deployment used the same sequence. After confirmation of full deployment, all six hinge-lines are pinned in place to prevent hinge rotation during high-thrust maneuvers.

b. Propulsion

The spacecraft's propulsion subsystem consists of its state-of-the-art titanium fuel tanks, the thruster modules, and the associated plumbing.

The subsystem is a pressurized bipropellant, dual-mode system using hydrazine (N_2H_4) and nitrogen tetroxide (N_2O_4) in the bipropellant mode and N_2H_4 in the monopropellant mode. Three main propellant tanks, a refillable auxiliary fuel tank, and a helium pressurant tank provide propellant and pressurant storage. 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 new lightweight main propellant tank was developed and qualified specifically for MESSENGER. The tank configuration is an all-titanium, hazardous-leak-before-burst design. A small 6Al-4V titanium auxiliary tank is a hazardous-leak-before-burst design. It has an internal diaphragm to allow positive expulsion of propellant for use in small burns.

The propulsion subsystem includes a total of 17 thrusters of three types. The large velocity adjustment (LVA) thruster is a flight-proven Leros-1b. Four 22-N, monopropellant LVA-TVC thrusters provide thrust vector steering forces during main thrust burns and primary propulsion for most of the smaller velocity adjustment (delta-V) maneuvers. Twelve monopropellant thrusters provide 4.4 N of thrust for fine attitude control burns, small delta-V burns, and momentum management.

The propulsion thermal system employs heaters to maintain acceptable system temperatures. Heaters are used during the cruise phase to maintain propellant temperatures and in the operational phases to pre-heat thrusters in preparation for operation.

c. Thermal

The thermal subsystem consists of the spacecraft's ceramic-cloth sunshade, its heaters, and its radiators.

The primary element of the thermal design is the ceramic-cloth sunshade, which protects the vehicle from the intense solar environment encountered when inside of Venus orbit. Creating a benign thermal environment behind the sunshade allowed for the use of essentially standard electronics, components, and thermal blanketing materials. Non-standard thermal designs were required for the solar arrays, sunshade, digital Sun sensors, three of the seven instruments, and the phased-array antennas. These components have been designed to operate at Mercury perihelion (Mercury closest to the Sun) and also during orbits that cross over one of Mercury's hot poles.

d. Power

The power subsystem consists of the spacecraft's solar arrays, battery, and the controlling electronics.

The power subsystem utilizes a peak power tracker architecture that 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 operating conditions of the mission. The power system is designed to support about 390 W of load power near Earth and 640 W during Mercury orbit.

Because of the mission's large solar distance variations, the requirements placed on the solar array design are rather severe. The maximum solar array voltage during normal operations is expected to vary between 45 and 95 V, but this range does not include the higher transient voltages expected on the cold solar arrays at exits from eclipses. Triple junction solar cells are used. The solar cell strings are placed between Optical Solar Reflector (OSR) mirrors with a cell-to-OSR ratio of 1:2 to reduce the panel absorbance. Thermal control is performed by tilting the panels away from normal incidence with increased solar intensity. In case of an attitude control anomaly near Mercury, the solar array temperature may reach 270 degrees C. All material and processes used in the solar panels are designed to survive the worst- case predicted temperatures.

e. Avionics

The avionics subsystem consists of the spacecraft's processors, its solid-state data recorder, and the data handling electronics. These components are packaged as an Integrated Electronics Module, or IEM.

The IEM implements command and data handling (C&DH), guidance and control, and fault protection functions. Its design utilizes five daughter cards: the Main Processor (MP), the Fault Protection Processor (FPP), the 8-Gbit Solid-State Recorder (SSR), the Interface Card and the Converter Card. These cards are tied together by a backplane, and a chassis utilizing the 6U compact peripheral component interconnect (cPCI) standard. Three of the cards (MP, FPP and SSR) implement fairly generic functions while the Interface and Converter Card are much more mission-specific.

A primary driver of the IEM architecture was to simplify spacecraft fault protection. The FPP independently collects spacecraft health information which are continuously evaluated by a rule-based autonomy system. The FPP corrects faults by sending commands to the MP and other components. The IEM Interface board includes hardware limits to prevent a failed FPP from continuously sending commands that would disrupt the spacecraft operation.

f. Software

The software subsystem consists of the spacecraft's processor-supported code that performs command and data handling (C&DH), and spacecraft guidance and control (G&C). It consists of two applications, the MP and the FPP, implemented as C code under the VxWorks 5.3.1 real-time operating system.

The MP software implements all C&DH and G&C functionality in a single flight code application running on the MP card.

The C&DH functionality includes uplink and downlink management, command processing and dispatch, support for stored and time-tagged commands, management of the 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). It also collects analog temperatures and implements a peak power tracking algorithm to optimize charging of the spacecraft battery via a power subsystem interface.

The G&C functionality maintains spacecraft attitude, manages spacecraft momentum, executes deep-space propulsive maneuvers, controls the solar arrays for optimized pointing to the Sun, manages spacecraft thermal environment by ensuring the sunshade always faces the Sun and enables a host of pointing options and instrument pointing control in support of science operations.

The MP also contains a boot mode which supports rudimentary command processing and telemetry generation for reporting health status and to support uploads of code and parameters to EEPROM.

The FPP application runs on the FPP card and implements an autonomy rule engine, which accepts uploadable health and safety rules that can operate on data collected from the various spacecraft subsystems via several interfaces, including an interface to the power subsystem. The action of each rule can dispatch commands to the MP or to the power subsystem to correct faults.

Fault correction can include actions such as switching to redundant components, demotion to lower spacecraft modes (Safe Hold or Earth Acquisition), or shedding power loads. The FPP can swap the bus controller functionality between MPs, power on and switch to the redundant MP, select which of two stored applications the MP loads and can reset the MP.

The spacecraft has two safing modes. During safing, all time-tagged command execution is halted and the spacecraft is taken to a pre-defined simple state. Safe Hold is the first level of safing and assumes knowledge of ephemeris time, orbit, and attitude. Earth Acquisition, the lowest level of safing responding to the most critical faults (e.g., battery at low state of charge) and no knowledge of ephemeris time, orbit, or attitude (with respect to the inertial reference frame) is assumed. The spacecraft is put into a slow rotation (one revolution every 3.5 hours) allowing the antenna suite to sweep past the Earth periodically regardless of location.

g. Guidance and Control

The primary functions of the guidance and control subsystem are to maintain spacecraft attitude and to execute propulsive maneuvers for spacecraft trajectory control. It consists of the spacecraft's attitude sensors including star cameras and Sun sensors integrated with controllers including reaction wheels.

The system enforces two attitude safety constraints: the Sun Keep-In constraint that keeps the sunshade pointed towards the Sun to protect the spacecraft bus from extreme heat and the hot-pole keep-out constraint that protects components on the top deck from re- radiation of sunlight from the surface of Mercury.

The sensor suite consists of star trackers, an IMU, and Sun sensors. The primary actuators for maintaining attitude control are four reaction wheels, each of which provides a maximum torque of 0.075 Nm and can store up to 7.5 Nms of momentum. 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.

The G&C system also interfaces with actuators for three other spacecraft components to position them properly based on knowledge of the Sun, Earth, and target planet directions relative to the spacecraft. These are the two solar array drive assemblies; the phased array antenna; and the pivot platform for the Mercury Dual Imaging System (MDIS) instrument. In addition, an interface to the Mercury Laser Altimeter (MLA) instrument provides a range and ''slant angle'' used to set the instrument's internal configuration parameters for surface observations.

h. Radio Frequency (RF) Telecommunications

The RF telecommunications subsystem consists of redundant General Dynamics small deep space transponders, solid-state power amplifiers, phased-array antennas, and medium- and low-gain antennas. The phased-array antennas have no mechanical components that could fail in the extreme thermal environment of Mercury. They are designed to work at the 350 degrees C temperatures to be encountered. The spacecraft is the first to utilize turbo coding for downlink, resulting in an extra 0.9 dB margin, corresponding to nearly a 25% increased in data return.

i. Payload

The MESSENGER payload consists of seven instruments: the Mercury Dual Imaging System (MDIS), the Gamma-Ray and Neutron Spectrometer (GRNS), the X-Ray Spectrometer (XRS), the Magnetometer (MAG), the Mercury Laser Altimeter (MLA), the Mercury Atmospheric and Surface Composition Spectrometer (MASCS), and the Energetic Particle and Plasma Spectrometer (EPPS). They are described in Solomon et al., 2007.

Instrument design was constrained along several dimensions. The payload mass was limited 50 kg for the seven instruments. The demanding thermal requirements to stay warm enough during cruise and eclipse periods, but cold enough on orbit, were significant constraints. Although the spacecraft solar arrays generate ample power during the orbital phase of the mission, power is much lower during the early cruise phase, restricting the size of instrument heaters that could be used. Power is also limited to the battery during eclipse.

The over-all spacecraft architecture specified distributed power and data processing for the instruments; each instrument had its own power supply and microprocessor. Redundant Data

Processing Units (DPU)buffer all data interfaces between the payload elements and the spacecraft. Common, flight-ready power supply and processor boards, including basic software functions, are used by all but one instrument, allowing development of a common set of ground support equipment hardware and software. This system architecture allowed payload development and testing to proceed separately from the rest of the spacecraft.

2. Instrument Host Overview - DSN

The Deep Space Network is a telecommunications facility managed by the Jet Propulsion Laboratory of the California Institute of Technology for the U.S. National Aeronautics and Space Administration (NASA).

The primary function of the DSN is to provide two-way communications between the Earth and spacecraft exploring the solar system. To carry out this function it is equipped with high-power transmitters, low-noise amplifiers and receivers, and appropriate monitoring and control systems.

The DSN consists of three complexes situated at approximately equally spaced longitudinal intervals around the globe at Goldstone (near Barstow, California), Robledo (near Madrid, Spain), and Tidbinbilla (near Canberra, Australia). Two of the complexes are located in the northern hemisphere while the third is in the southern hemisphere.

Each complex includes several antennas, defined by their diameters, construction, or operational characteristics: 70-m diameter, standard 34-m diameter, high-efficiency 34-m diameter (HEF), and 34 m beam waveguide (BWG).

For more information see Asmar and Renzetti, 1993.

3. Instrument Host Overview - Spacecraft Clock Reset and Use of Clock Partitions

A planned reset of the on-board clock of the MESSENGER spacecraft occurred on January 8, 2013. This was commanded by Mission Operations and was done because the integer seconds part of the on-board mission-elapsed-time (MET) counter is not long enough to contain the larger MET values that would occur due to the extended mission. The MESSENGER team elected to command the clock reset and set MET to a small non-zero value to prevent disruptions in on-board timekeeping and other effects (that might have occurred if the clock were allowed to automatically rollover to 0 in early 2013) and to ensure that the MET counter would accommodate the remaining extended mission.

As a result of the spacecraft clock reset, a discontinuity was introduced and MET values are no longer guaranteed to be unique throughout the mission. This ambiguity is resolved in ground processing by the use of SPICE 'clock partitions' (partition 1 for pre-reset METs and partition 2 for post-reset METs) in the Spacecraft Clock (SCLK) kernel (which supports mapping MET to other time forms using SPICE routines as described below) and with MET values stored in PDS products, labels, and for some instruments, product file names. For MET values in products or labels, a '1/' or '2/' preceding MET indicates the partition.

When using SPICE routines, clock partition numbers should be included with MET input values. METs expressed without an explicit partition number are associated with clock partition 1 by default.

a. Spacecraft Clock (SCLK) SPICE Kernel

The SCLK SPICE Kernel provides information that correlates mission elapsed time (MET) as measured by the spacecraft's on-board clock with Terrestrial Dynamical Time (TDT) as defined by the International Astronomical Union (IAU). (TDT was later redefined by the IAU and renamed 'Terrestrial Time' with the acronym 'TT.' However, that acronym is ambiguous in the SPICE context.). A 'partition' is a segment of time when the MET count increments continuously. A single clock partition can continue for years. When a discontinuity in MET occurs, a new partition is defined. MET discontinuities can occur for a number of reasons and can result in either a jump forward in MET or a jump backwards. Spacecraft clock jumps or discontinuities can be either the result of anomalies or they can be deliberately commanded. When such a discontinuity occurs, the previous correlation of MET to TDT is not valid from that point forward. A new clock partition must be created in order to correctly associate MET with TDT. The SCLK contains a list of all partitions that have been defined and specifies the MET values at which each ends. SPICE takes the partitions into account when computing the encoded SCLK representations of MET that make up the first field in each SCLK record 'triplet.' Because of this, the encoded SCLK values in the kernel increment steadily regardless of partition changes.

Prior to the MESSENGER spacecraft clock reset, the MESSENGER SCLK kernel defined a single partition (partition 1). A second post-reset partition was introduced (partition 2) shortly before the reset. SCLK kernels from that time forward include both partitions.

The MET partition change is largely transparent to users of SPICE and the SCLK kernel (with the exception of users who are converting raw MET counts), since the MET values in the kernel are provided in encoded SCLK form.

4. References Cited

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