# THE MESSENGER SPACECRAFT GUIDANCE AND CONTROL SYSTEM[*]

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**ABSTRACT**

The MESSENGER – MErcury Surface, Space ENvironment, GEochemistry, and Ranging – spacecraft has successfully completed its first year of flight operations following launch on August 3, 2004. As part of NASA's Discovery Program, the spacecraft will observe Mercury during flybys in 2008 and 2009 and from orbit for one Earth-year beginning in March of 2011. The guidance and control system combines extensive flight software with various sensors and actuators to maintain a 3-axis stabilized spacecraft and to implement desired velocity changes. This paper presents the overall system architecture and describes the various software functions.

**INTRODUCTION**

The MESSENGER (MErcury Surface, Space ENvironment, GEochemsitry, and Ranging) spacecraft was successfully launched from Kennedy Space Center in Florida on August 3, 2004. As part of NASA's Discovery Program, MESSENGER will be the first spacecraft to closely observe the planet Mercury since the Mariner 10 flybys of the mid-1970s. MESSENGER has completed its single Earth flyby and will make two flybys of Venus and three of Mercury prior to orbiting the planet for one Earth-year beginning in March 2011. The planetary flybys are interspersed with five large deterministic deep-space maneuvers (DSMs) that target the spacecraft for its Mercury orbit insertion (MOI) maneuver in 2011. The Mercury flybys will assist in developing the focused science gathering of the year-long orbit phase of the mission (ref. 1).

Figure 1 shows the MESSENGER spacecraft configuration and the locations of some of the main engineering components and science instruments. The primary factors driving the spacecraft design were the high temperatures and radiation doses to be encountered at Mercury (ref. 2). Protection from this environment is accomplished with a large sunshade, which shields the spacecraft components from direct exposure to the Sun. This shade must be kept between the main body and the Sun at all times when the spacecraft is within 0.85 AU of the Sun. The shade has been sized to allow small deviations from direct Sun pointing when needed for science observations or engineering activities. Power generation is handled with solar panels 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 eclipse periods when in orbit about Mercury. The spacecraft carries high-, medium-, and low-gain antenna sets for X-band communication with Earth. Two electronically steerable high-gain phased-array antennas are mounted on the sunshade and on the back of the spacecraft. 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 are mounted on the spacecraft to provide coverage in all directions. MESSENGER has a sophisticated propulsion system featuring both bi-propellant and mono-propellant thrusters that can operate in either blow-down or fixed-pressure modes. This system is described in more detail in a later section.

MESSENGER carries a diverse suite of miniaturized science instruments to globally characterize the planet (ref. 3).  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.
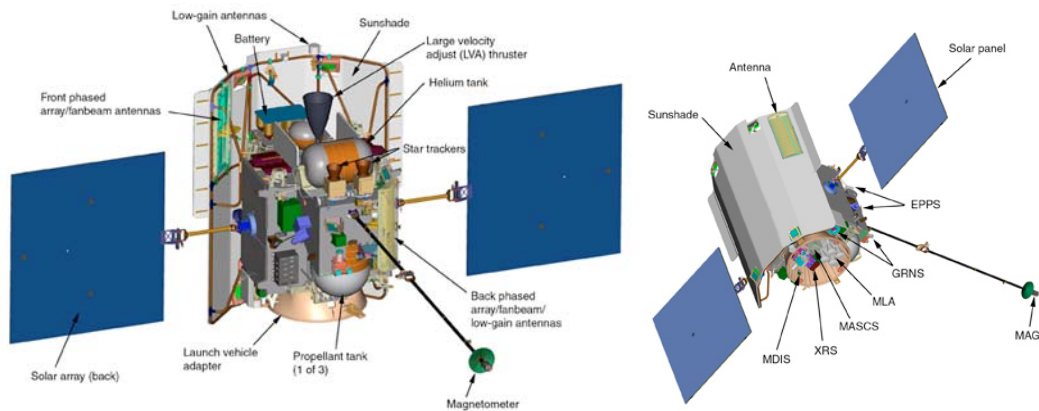
**Figure 1.  MESSENGER Spacecraft Components and Science Instruments**

This paper presents an overview of the guidance and control (G&C) system for MESSENGER.  The hardware components include the flight computers, a sensor suite of star trackers, an inertial measurement unit (IMU), and Sun sensors, and an actuator suite of reaction wheels and thrusters. The interfaces between these components are presented in the first section on system architecture. The primary software functions coordinating the interaction of the hardware components are attitude determination, guidance, and attitude control. Implementation details and descriptions of other supporting software functions are presented in the second  section on software architecture.

**GUIDANCE AND CONTROL (G&C) SYSTEM ARCHITECTURE**

The primary functions of the MESSENGER guidance and control subsystem are to maintain spacecraft attitude and to execute propulsive maneuvers for spacecraft trajectory control. Software algorithms run in the main processor (MP) to coordinate data processing and commanding of sensors and actuators to maintain a 3-axis stabilized spacecraft and to implement desired velocity changes. The software also controls the orientation of the two solar panels, electronic steering for the two high-gain phased-array antennas, and, optionally, pivot positioning for the MDIS cameras.  An additional interface with the MLA provides range and slant angle to the planet's surface used to configure the instrument but does not involve any active mechanical or electronic steering.

**Sensors and Actuators**

The sensor suite consists of star trackers, an IMU, and Sun sensors as shown in Figure 2, along with the coordinate axes defining the spacecraft body frame. Inertial attitude reference is provided by two ASTR star trackers from Galileo Avionica, both of which are mounted on the top deck looking out along the –Z axis. Typically only a single tracker is powered, with the other acting as a cold spare. Spacecraft rotation rates and translational accelerations are provided by a Northrop-Grumman S-SIRU (Scalable Space Inertial Reference Unit) IMU with four hemispherical resonance gyroscopes (HRGs) and Honeywell QA3000 accelerometers. The IMU has two power supply and processor boards providing internal redundancy with the second board acting as a cold spare. One board and all four gyros are powered at all times, while the four accelerometers are powered only when performing a trajectory correction maneuver (TCM). 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. There are two separate Sun sensor systems consisting 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 Sun sensors are always powered, providing two independent Sun direction readings at all times.

The actuator suite consists of reaction wheels and thrusters as shown in Figure 3. 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 Aerojet 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 propulsion system has a large bi-propellant engine and two sets of mono-propellant thrusters - twelve 4-N thrusters and four 22-N thrusters. Eight of the 4-N thrusters, designated A1-4 and B1-4, are used for attitude control. The remaining four 4-N thrusters, designated S1&2 and P1&2, are used for small velocity changes, while the four 22-N thrusters, designated C1-4, are used for larger velocity changes. The LVA (large velocity adjust) main engine is used for very large velocity changes, such as the five DSMs and MOI. Fuel is carried in two of the three main tanks and in the smaller auxiliary tank; the third main tank contains oxidizer for the LVA burns. A small helium tank regulates pressures of the main tanks, while the auxiliary tank is unregulated. 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-N and 22-N thrusters, and a flange heater is used for the LVA.
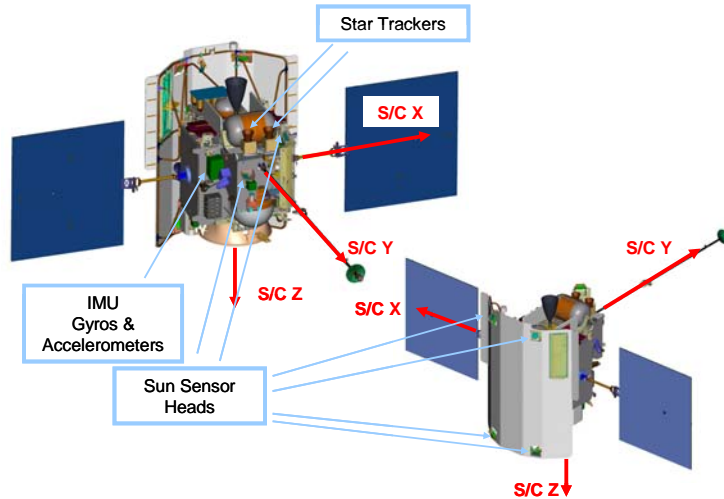

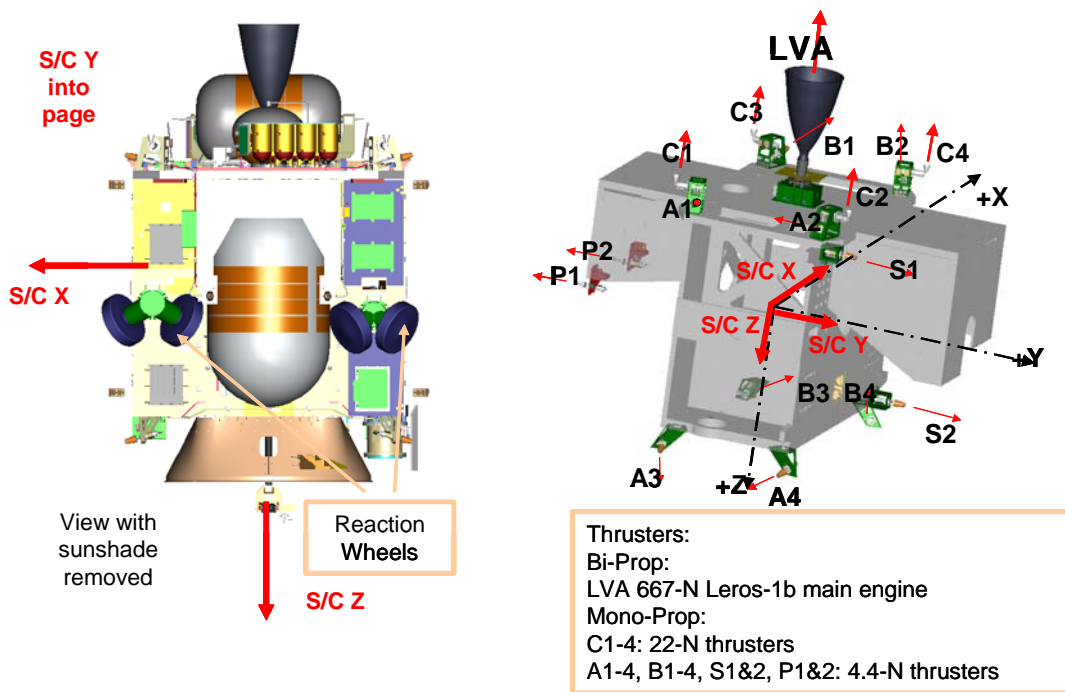
**Figure 2.  MESSENGER G&C Sensors**

**Figure 3. MESSENGER G&C Actuators**

The G&C system also interfaces with actuators for three other spacecraft components. Solar panel rotation is performed using two MOOG solar array drive assemblies (SADAs). The drives can rotate in two directions about the X axis through an angular arc of 228° in the YZ plane centered at the –Z axis. Panels are rotated in steps of 0.02° at a constant rotation rate of 2°/s (100 steps per second). The beam width (or field-of-view) of the two phased-array antennas is 12° in the XY plane and 3° normal to it. The boresight, centered in this beam, can be steered through an angular arc of 120°. The antennas are mounted with boresights centered in the +X, +Y and –X, –Y quadrants. Full 360° coverage of Earth direction in the XY plane is obtained by rotating the spacecraft about the Y axis when necessary. The MDIS 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).

**Flight Computers and Hardware Interfaces**

MESSENGER is equipped with two sets of flight computers, each of which contains one main processor (MP) and one fault protection processor (FPP). The MPs and FPPs are RAD6000 processors operating at 25 MHz and 10 MHz, respectively. Each pair of MP and FPP is packaged with other cards providing command, uplink, and downlink functions in an integrated electronics module (IEM). A single MP performs all nominal spacecraft functions while the two FPPs monitor spacecraft health and safety. Both the G&C and C&DH (command and data handling) software run in the MP. 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."

The block diagram in Figure 4 shows the communication links between the flight computers and the G&C hardware components. The star trackers and the IMU interface with the primary MP directly via the 1553 data bus. The primary MP interfaces to a redundant Power Distribution Unit (PDU), two Data Processing Units (DPUs), and the two FPPs via the 1553 data bus. The PDU provides the interface to the Sun sensors, reaction wheels, and the propulsion system. The DPUs provide the interface to all science instrument processors. G&C software in the MP passes attitude data to the imager and laser altimeter instruments and pivot positions for the MDIS cameras via the DPU. Both FPPs also access Sun sensor data

from the PDU via the 1553 data bus; the FPPs use software provided by the G&C team to perform an independent Sun direction computation using these data.

The G&C software is implemented in Simulink$^{TM}$ and converted to C code using the RTW$^{TM}$ (Real-Time Workshop) tools, both provided as part of Matlab$^{TM}$. This code is integrated with the rest of the flight software, also implemented in C, that operates under the VxWorks 5.3.1 real time operating system. The G&C MP software functions are split into two main tasks that run at 1 Hz and 50 Hz, respectively. The 50-Hz task contains only those functions necessary for immediate attitude and trajectory control and is streamlined to run as efficiently as possible. All other functions are contained in the 1-Hz task. Interfaces with the various sensors and actuators are split between the tasks as dictated by their functional content. The rates, sources, and destinations of data collected from and commands sent to the sensors and actuators are shown in Table 1.
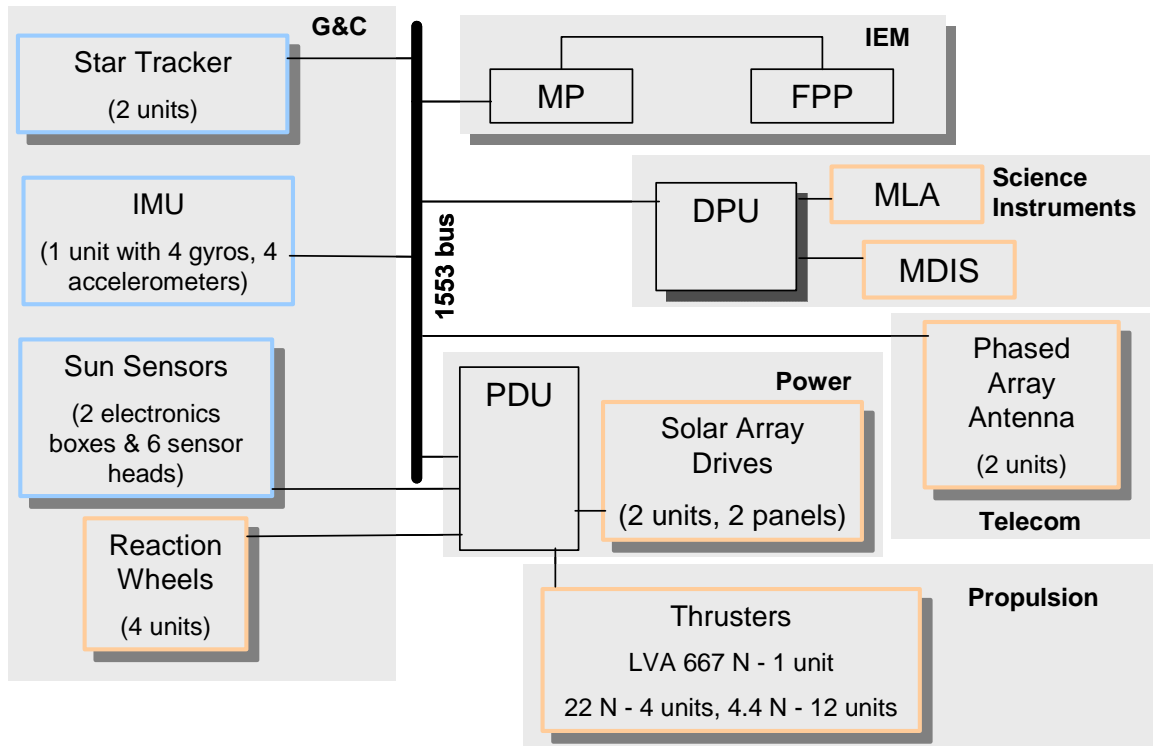


**Figure 4. MESSENGER G&C System Block Diagram**

## G&C FLIGHT SOFTWARE ARCHITECTURE

The G&C MP software contains the following major functional blocks: attitude determination, attitude control, guidance, momentum management, propulsion operations, and solar panel control. Maintaining or changing spacecraft attitude is coordinated by the attitude determination, guidance, and control functions. The attitude determination function runs entirely at the 1-Hz rate, and the control functions run at 50 Hz. Guidance functions are split between these two tasks, as are propulsion system operations and momentum management. Solar panel control, phased-array antenna steering, and the MDIS and MLA interfaces run at 1 Hz.

### Attitude Determination

The attitude determination block processes star tracker and gyro rate data to generate an estimated spacecraft attitude and rotation rate. "Sanity" checks on the sensor data are performed before solving for attitude and rate; individual measurements can be rejected if they differ too much from preceding values or

if they are too noisy. In the original launch formulation, currently in use on the spacecraft, an extended Kalman filter algorithm uses the accepted gyro measurements and the most recent valid star tracker data to solve for the three attitude error states, three body rate states, four gyro biases, and three relative alignment parameters for the two star trackers (when attitude solutions from both are available). The estimator and attitude propagation algorithms are designed to compute valid estimates when any three or all four of the gyros are providing valid readings. Star tracker rate data are substituted for gyro data if fewer than three valid readings are available, even for very short time periods. Several of the Kalman filter parameters are changed when rate data sources are switched, effectively restarting the filter and resetting the gyro bias estimation. When no valid tracker attitude solutions are available, the filter is not run and attitude is propagated forward in time from the last valid attitude estimate using any valid gyro rate readings. The estimator provides a quaternion correction and gyro biases to the 50-Hz control task used to update its knowledge of spacecraft attitude.

**Table 1 G&C SENSOR AND ACTUATOR INTERFACE RATES (TO/FROM MP)**

| Device | Basic Interface Rate | Measurement or Command | Software Source or Destination | Rate |
|---|---|---|---|---|
| IMU | 100 Hz | Gyro integrated angular rate | 1-Hz task<br>50-Hz task | 100 samples<br>2 samples |
| | | Accelerometer integrated linear velocity | 50-Hz task | 2 samples |
| | | Diagnostic data | 1-Hz task | 1 sample |
| Star trackers (2) | 10 Hz | Quaternion and rate<br>rate only<br>diagnostic data | 1-Hz task<br>50-Hz task<br>1-Hz task | 10 samples<br>latest sample<br>1 sample |
| Sun sensors (2) | 1 Hz | Head ID and Sun aspect angles | 1-Hz task | 1 sample |
| SADAs (2) | 1 Hz | Reference and potentiometer voltages | 1-Hz task | 1 sample |
| | | Step commands | 1-Hz task | 1 command of up to 100 steps each second as needed |
| Reaction wheels (4) | 50 Hz | Tachometer time pulse information | 50-Hz task | 1 sample |
| | | Wheel torque commands | 50-Hz task | 1 command |
| Thrusters (17) | 50 Hz | On/off commands | 50-Hz task | 1 command to each thruster |
| Latch valves (9), heaters | 1 Hz | Open/close commands, on/off commands | 1-Hz task | 1 valve or heater command per second if needed when thrusters are firing |
| Phased-array antennas (2) | 1 Hz | Steering commands | 1-Hz task | 1 command to each transponder |
| MDIS | 1 Hz | Pivot position | 1-Hz task | 1 value |
| MLA | 1 Hz | range and slant angle values | 1-Hz task | 1 set of values |

This current estimator implementation assumed a near-perfect synchronization between the IMU and MP clocks and that periods with no gyro data would be of reasonably long duration due to some failure in the IMU hardware or spacecraft interface with it. A larger than expected drift between the MP and IMU

clocks was noticed prior to launch and has been the source of some problems with the attitude estimation in flight. The IMU outputs its data message every 10 ms as measured by its internal clock. These messages 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 gyro and accelerometer measurements are not always separated by exactly 10 ms. The IMU data pattern seen in flight (and in ground tests) has "bursts" of irregularly spaced measurements that recur approximately every 200 s. These bursts include repeated measurements where the actual time change between measurements reported by the IMU is 0 and skips in the measurements where the actual time change is 20 ms. These inconsistencies in the time interval between IMU messages (as read by the bus) are ignored by the software which assumes that the IMU messages are always separated by exactly 10 ms This resulted in frequent switching between rate sources during the burst periods and the attendant filter reconfiguration caused the estimator to attribute actual rotation about the Y axis to a gyro bias during the first few days after launch. During this time, the spacecraft was commanded to point the +Y axis at the Sun and maintain a rotation rate of 0.0005 rad/s about the +Y axis. However, the actual rotation rate (derived from differencing the estimated attitude quaternions) was slowly decreasing, while the estimated gyro biases were continually increasing (ref. 4). Because the gyro bias estimate from the Kalman filter was growing, the control law attributed increasing amounts of the IMU sensed rate to bias, and slowed the actual rotation rate of the spacecraft to compensate.  So while the control law believed the rotation rate was being maintained at the commanded rate, the spacecraft was actually slowing down. There was never a long enough period of time between the bursts with a regular pattern of IMU data for the filter to correct this error. This condition was ameliorated by setting filter parameters to turn off gyro bias estimation. Attitude estimation has been functioning satisfactorily since this reconfiguration, including long periods at near-fixed attitudes with no commanded rotation rate and shorter periods with intentional rates about different body axes. However, the high amount of rejected IMU gyro samples during the burst periods still causes a periodic drop in quality of the attitude estimate reported by the estimator.

An extensive analysis of the effects of the MP/IMU clock drift given the existing software design has led to a decision to proceed with a flight software update that includes improved handling of the actual data pattern as read from the IMU. The IMU data interface is being altered to provide time tags as generated by the IMU to both G&C software tasks. Spacecraft body rate will now be computed from two non-consecutive gyro readings surrounding short data gaps. Both the estimator and controller will retain memory of a longer sequence of IMU data samples and will use values from the last valid samples for some time before switching over to another source of rate information. The Kalman filter will no longer be run with star tracker rate substituted for gyro rates. Instead, the estimator will switch to a "tracker-only mode" where it simply reports star tracker estimated attitude and rate directly if there is a long-duration loss of gyro rate data. Star tracker relative alignment has also been removed as a filter state; a simple low-pass filter runs outside of the Kalman filter to estimate these alignment parameters when data from both trackers are available. Many of these changes have already been incorporated in the G&C software for the New Horizons mission, but they were recommended too late in the development cycle to be incorporated in the MESSENGER software prior to launch. The new software is scheduled to be uploaded to the spacecraft and tested in-flight in the fall of 2005.

The checks done on star tracker and gyro data in preparation for attitude estimation are also used to detect a "data drop out" condition for the IMU and each tracker. Each of the 100 gyro measurements and the 10 attitude solutions from each tracker has an associated validity flag when input to the G&C software that is constructed from flags provided by the devices and also status checks on their respective 1553 bus communications. A measurement is rejected if this flag indicates it is invalid. Additional checks are made on both data sets to determine the final subset of rejected measurements. Star tracker solutions are rejected if the reported RMS covariance is above a preset magnitude, if the rate change between two consecutive samples is above a preset maximum value, or if there are too many consecutive readings reporting the same rate magnitude.  The gyro rate checks are similar: measurements are rejected if the magnitude of rate change between two consecutive readings exceeds another preset maximum value, if the number of consecutive readings with the same rate magnitude exceeds a preset maximum, if the time difference between two "consecutive" readings exceeds a preset duration, or if the difference between the measurement time and spacecraft clock time exceeds a preset duration. Statistics on the number of valid

and rejected measurements are output with each execution of this 1-Hz block. When a period of consecutive rejected gyro rate or tracker attitude measurements extends beyond a preset duration (over multiple executions of the block), the "data loss" flag for the IMU or a star tracker is set.

The attitude determination block also monitors the health of star tracker and IMU hardware. A set of hardware diagnostic flags from the self-checks performed by each device is input to the block. Selected states from these internal hardware diagnostic flags are combined with other checks performed in this block to generate an overall "hardware problem" condition for the IMU and each star tracker at each execution of the block. When a "hardware problem" persists longer than a preset duration (over multiple executions of this 1-Hz block), a "hardware problem" flag is set. The additional hardware condition monitored for the star trackers is mode cycling. For normal operation, a tracker should maintain its fine tracking mode. Flags are set if the tracker appears to be cycling between its coarse and fine tracking modes too frequently or spending too long in its coarse tracking mode. The additional hardware condition monitored for the IMU is a transition from force-to-rebalance (FTR) to whole-angle mode (WAM). The gyro's internal control loop will automatically switch to WAM mode when the rotation rate exceeds a preset magnitude, but it will not return to FTR mode once the rate returns below this threshold. Since the rate data produced in WAM mode is considerably noisier and of lower quality, the flight software monitors gyro mode flags and the estimated rate to determine when a reset back to FTR mode is needed.

The "hardware problem" and "data loss" flags for the IMU and star trackers are sent to the FPP where they are monitored by certain autonomy rules. These rules can reset a tracker, switch from tracker 1 to tracker 2, or switch to the redundant processor in the IMU in an attempt to maintain normal hardware operation and keep attitude data flowing to the G&C software.

**Guidance Functions for Spacecraft Pointing**

The primary function of the guidance block is to compute the desired (or commanded) spacecraft attitude and rate based on a specified pointing option possibly combined with a scan pattern. Orientation relative to the inertial frame and rotation rates that follow the commanded pointing profile are computed in the 1-Hz task. A companion block in the 50-Hz task propagates the 1-Hz commanded attitude through the intervening 50-Hz intervals between each 1-Hz update. Ten pointing options are available to point antennas at the Earth, point instruments at or near various celestial bodies, or align thrusters with a target ΔV direction (ref. 5). Pointing targets include directions in the EME2000 inertial frame specified as vectors or right ascension and declination angles, directions from the spacecraft to the Sun, Earth, or a target planet, directions in the target planet body-fixed frame specified as vectors or as latitudes, longitudes, and heights, directions in a local vertical, local horizontal (LVLH) frame given as azimuth and elevation angles, or points on the target planet that optimize illumination geometry. Scan patterns combining periods of fixed-rate rotations about specified axes with pauses can be added to the base pointing option. These are used to design mosaics or continuous scans that enable target motion in an instrument field-of-view. Motions can be rotations about axes in the spacecraft body frame, the inertial frame, or the LVLH frame or translations along inertial axes. Each axis may have a different combination of rates, pauses, and motion reversals. The guidance system enforces certain compatibility restrictions between the scan frame and the base pointing option.

The "downlink" pointing option is used to set the default spacecraft attitude in the absence of any science or special engineering pointing targets. This "downlink" attitude aligns a specified body axis, normally the + or −Y axis, with the Sun line and places the Earth line in one of the quadrants of the XY plane covered by one of the two antenna sets. Small offsets of the Sun line from the + or −Y axis are allowed to achieve desired observation geometry or to dump momentum passively by altering the solar torque acting on the spacecraft. A fixed offset can be manually commanded, or a variable offset can be computed automatically based on estimated system momentum. The downlink pointing option can be configured for automatic determination of a spacecraft-Sun line offset from the +Y or −Y axis that can achieve a desired target momentum on the X and Z axes while maintaining the spacecraft-Earth line in one of the antenna fields-of-view (ref. 6) The Y momentum can be adjusted by applying a small rotation of

equal magnitude but opposite direction to each of the solar panels. Attitude offsets from + or –Y to Sun and panel position adjustments will be used in orbit to minimize the frequency of propulsive momentum dumps, which generate small perturbations to the spacecraft trajectory from thruster firings.

The guidance block continuously interpolates on-board ephemeris models to obtain the position and velocity of the Sun, the Earth, a target planet, and the spacecraft, all referenced to the solar system barycenter. The target planet will be Earth, Venus, or Mercury, depending on the mission phase. Celestial body positions and velocities and models for the shape, size, and rotation of the target planet are used when needed to formulate the commanded attitude. Each body has both a precise and coarse ephemeris model. The precision models are used under normal spacecraft science gathering operations and are necessary to meet the pointing requirements of the science team. The coarse models are lower precision models used as a contingency for spacecraft safing operations. They are valid over long time spans and require only occasional updates by ground controllers. The precision models use Chebyshev polynomials and a simple linear conversion of spacecraft time to TDT (terrestrial dynamic time) to extract the necessary positions and velocities for the spacecraft or the celestial body.  The precise ephemeris spans are valid over shorter durations (hours or days) and must be updated by ground controllers at regular intervals as the mission progresses. The guidance block performs a magnitude check on the TDT value, magnitude checks on the computed position vectors for each body, and checks the available time ranges covered by the polynomials. Flags are set if the algorithm is unable to obtain a valid TDT time, obtains an invalid position, or detects that the remaining time covered by any ephemeris model is less than a preset duration. Demotion to a safe mode is requested if insufficient time or ephemeris information is available to reliably compute the desired spacecraft attitude.

Two separate Sun directions in the spacecraft body frame are computed every second from the Sun sensor data. Validity flags provided by the DSS hardware are checked to determine a "data drop out" condition for each DSS system. If the "data drop out" condition persists for longer than a preset number of seconds, a "data loss" flag is set. This flag is provided for information only, and there is no associated autonomy action taken by the FPP. Sun direction derived from the on-board ephemeris models and estimated attitude is compared with the direction derived from the Sun sensor data as a check on the ephemeris data.  If the ephemeris data are not consistent with the Sun sensor readings, the precise models are declared invalid, the spacecraft begins to use the coarse models as a backup, and demotion to a safe mode is requested.

The guidance block enforces two attitude safety constraints, both of which arise from the need to protect various spacecraft components from the extreme heat and radiation near Mercury. The first constraint is a Sun keep-in (SKI) zone that is intended to keep the sunshade between the spacecraft bus and the Sun. The Sun line must be kept within specified bounds around the –Y axis in order for the shade to shield the spacecraft components. Although intended to be used when near the Sun, the SKI zone bounds can be defined around any spacecraft body axis. For operational convenience, a SKI zone around the +Y axis is enforced for portions of the mission where Sun range is greater than 0.85 AU. The second constraint is a planet "hot pole" keep-out (HPKO) zone. This constraint keeps the top deck pointing away from the surface of Mercury once the spacecraft is in orbit to protect the battery and other components from radiation reflected off the planet's surface. The constraint is applied only for certain portions of Mercury's orbit when it is nearest the Sun and for the portion of the spacecraft orbit when it is nearest the sunlit northern hemisphere. This constraint is not enabled during cruise. When enabled, the G&C system monitors estimated spacecraft attitude for violations of either constraint. If a violation is detected, the system automatically overrides the commanded attitude and performs a turn back to a safe attitude. Demotion to a safe mode is requested.

**Attitude Control**

The attitude control block in the 50-Hz task monitors the difference between actual and desired spacecraft attitude and rate and attempts to drive the error between them to zero by issuing appropriate actuator commands. It provides a choice between two wheel control algorithms and a single thruster control

algorithm. MESSENGER is the first mission of the Applied Physics Laboratory (and probably also the first interplanetary mission) to use a nonlinear control law to compute wheel torques (ref. 7). A more traditional time-optimal slew-PID (proportional-integral-derivative) control law is also available as a backup; this is similar to the algorithm used for the Near Earth Asteroid Rendezvous mission. The nonlinear algorithm has been chosen as the default for wheel control for reasons discussed in the next paragraph. For either algorithm, when four wheels are available, the controller adds biases to the commanded wheel torques to keep the wheel speeds away from zero; these biases sum to a net zero torque applied to the spacecraft and thus do not alter the control action. These biases are not applied when only three wheels are available, but control is still maintained using the normal logic for the selected algorithm.

With the slew-PID algorithm, a time-optimal slew is achieved by rotating the spacecraft about an eigen-axis with maximum torque until a control switch line is reached. The switch line is then followed by alternating acceleration and deceleration to minimize possible overshoot of the target attitude. Control finally switches to the PID logic once the attitude error falls below a specified small threshold. "Chattering" of the torque values is often seen for some period of time during turns due to the alternating sign of the torque commands when following the switch line. Chattering may add unnecessary stress on the wheel hardware and possibly excite flexible modes if the wheels are capable of following these rapid transitions. The nonlinear algorithm minimizes this chattering of the commanded torques by adjusting their values based on actual attitude error and slew rate, with no explicit sign changes, instead of following a switch line. It is formulated similarly to the PID control logic but has a self-adjusting limiter placed on the proportional and integral attitude error terms. For smaller error values, the computed torques are identical to those of the PID formulation. The nonlinear algorithm can achieve an eigen-axis slew but often deviates slightly from a pure eigen-axis turn, especially when the turn has components along all three body axes. Pre-launch simulations showed no significant difference in total turn time using these two algorithms and elimination of any chattering of the torque commands for most cases using the nonlinear algorithm. The only problem found with the nonlinear algorithm was that the deviation from a pure eigen-axis turn occasionally resulted in SKI violations for turns moving from a point near the zone boundary back towards the center of the zone. The algorithm was modified slightly to limit the size of the deviation from an eigen-axis turn by minimizing the non-eigen axis portion of the slew rate to force the slew axis to remain as close to the eigen axis as possible. The system meets all performance requirements when using this new algorithm, and it produces a more stable torque command profile.

The thruster attitude controller is a standard phase-plane controller. Thrusters are selected to be fired based on the location of switch lines in a phase-plane. A "line of action" is defined for each thruster as the direction of the torque it applies to the spacecraft. The phase plane axes are the dot products of the angle and rate errors with these lines of action. The switch line is chosen such that a thruster is commanded on if it decreases either the angle or rate error. The thruster controller is used during momentum dumps and TCMs and would also be used if fewer than three wheels are available to the controller.

Wheel angular speeds (magnitudes and directions) are computed in the 50-Hz task using timing information supplied by tachometers on the wheels themselves. The algorithm implements summing logic to compute wheel speeds from the time required to complete a single revolution. The inputs to the algorithm are time durations and a set of flags indicating when a full revolution has been detected. In each 50-Hz cycle, the duration represents either a full revolution or an increment to be added to an accumulating time. The flags indicate whether the duration represents a full revolution, the final increment to be added to obtain the time for one full revolution, or another increment to be added to accumulating time for one revolution that has not yet completed. The algorithm checks for missing tachometer readings from each wheel and for certain inconsistencies in the computed speeds such as speeds over the maximum physically attainable values, too long a time with no speed update because the time for a single revolution is still accumulating, or too many gaps in the tachometer readings to compute a valid speed. Other wheel health checks include a comparison of the change in wheel speed to the expected change based on wheel torque commands and a check on wheel speed persistently at or exceeding a preset maximum value. Flags indicating missing tachometer data, lack of response to torque commands, or persistent overspeed condition for each wheel are sent to the FPP for autonomy rule action if needed.

**Momentum Management**

System momentum is computed in the 50-Hz task from the angular speeds of the four reaction wheels and the estimated spacecraft rotation rate. A simple low-pass filter is applied to the estimates to reduce noise on the values. The ground team monitors the momentum magnitude and will command thrusters to be used to dump momentum from the reactions wheels when necessary. As a fail-safe measure, the 50-Hz task also contains logic for triggering an autonomous momentum dump. Flags are sent to the 1-Hz propulsion operations block to initiate a momentum dump using thrusters when momentum magnitude exceeds limits that could compromise controllability using the wheels. There are two types of autonomous momentum dumps, each triggered by different momentum magnitude levels. The "red" or "leisurely" momentum dump is initiated at a lower momentum level and performs a 1-hour warm up of thrusters by the catbed heaters before firing the thrusters. The "white" or "immediate" momentum dump is initiated at the higher threshold and fires thrusters with no catbed heater warm-up using a special "cold start" pulsing profile. The system maintains the current attitude using thrusters during the momentum dump and transitions back to wheel control when momentum falls within a specified tolerance of the target magnitude or after a preset maximum time has elapsed.  Only the 4.4-N thrusters are used for commanded or autonomous momentum dumps. Demotion to a safe mode is requested when an autonomous momentum dump is executed. Spacecraft solar torque will also be used as a passive means of managing system momentum by altering spacecraft attitude and solar panel positions as described in the preceding guidance section.

**Guidance Functions for TCMs**

TCMs are executed by ground command only. One of the ten pointing options is used to specify the $\Delta V$ profile and cutoff tolerances in addition to the spacecraft orientation needed to align the selected thrusters with the $\Delta V$ direction. A separate enable command is used to initiate thruster firing once at the burn attitude. The 1-Hz guidance block contains the steering logic for achieving the target $\Delta V$. Both fixed inertial vector and time-varying $\Delta V$ profiles are supported; the time-varying $\Delta V$ is primarily intended for use in the powered turn scenario for MOI. There are three choices for burn termination criteria. The first is a simple open-loop time option in which the burn is terminated a specified duration after it begins. The second is a closed-loop option in which the burn is terminated when the magnitude of the estimated $\Delta V$ is within a specified tolerance of the target magnitude. The third and most accurate option has the same termination criteria as the second option but applies corrections to the commanded attitude as the burn progresses to achieve a closer match to the desired burn direction.

Accumulated $\Delta V$ is estimated by the 50-Hz guidance block using a simple low-pass filter applied to the integrated linear velocity readings provided by the accelerometers in the IMU.  The current estimated attitude is used to transform the $\Delta V$ to the inertial frame. This value is compared to the cutoff criteria to determine the time of final firing of any thruster selected to impart the $\Delta V$, providing a finer resolution than could be achieved by doing these computations in the 1-Hz task. An off-pulsing capability is available for the selected $\Delta V$ thrusters. These thrusters can be set to fire continuously until the TCM cutoff criteria are satisfied or to be cycled off and on in coordination with the attitude control thrusters. Although the off modulation was added to handle the altered thrust alignments of the 22-N thrusters due to plume impingement on top deck components, it can also be applied for TCMs using the 4.4-N thrusters.

The irregular sample pattern seen during the IMU "bursts" resulting from the MP/IMU clock drift also affects accelerometer data processing in the current software implementation. Skipped samples have the most serious effect for the processing done in the 50-Hz task. Two accelerometer data readings are input at each cycle for $\Delta V$ estimation. No time tags are passed in and the time change between the samples is assumed to be 0.01 s. When the accelerometer counts reflect the velocity change over a time period different from 0.01 s, the software computes the acceleration using the erroneous assumption for the time increment between the two samples. The worst case occurs for a skipped measurement where the time

change between the two samples is 0.02 s. The resulting "hiccups" or small, temporary jumps in estimated body acceleration during thruster firing have affected the execution accuracy for three of the TCMs performed to date. This problem has been corrected in the new flight software scheduled for delivery in fall 2005.

The two guidance blocks incorporate checks on the availability of accelerometer data and the status of attitude estimation before and during a burn. Thruster firing will not be initiated if insufficient accelerometer data are available, if the estimated attitude is too far from the expected burn attitude, if the attitude estimate is of low accuracy, or if the ΔV thrusters are not aligned closely enough to the target ΔV direction. Thruster firing will be terminated prematurely if accelerometer data are lost, if attitude estimate accuracy decreases, or if attitude and rate errors become too large. The accumulating ΔV profile is also monitored during the burn. Thruster firing will be terminated if the ΔV deviates too far from the target profile or after a preset maximum duration ("time out"). Parameter blocks can be set for each burn to enable or disable these checks and change the various thresholds defining acceptable deviations from the nominal conditions.

**Propulsion Operations**

The target ΔV magnitude drives the choice of the thruster type – 4.4-N, 22-N, or LVA - used for a TCM. This choice dictates a certain sequence of propulsion system configurations involving latch valve opening and closing and heater power status needed to initiate, sustain, and clean up after the desired thruster firings. These configuration sequences are called "propulsion modes." For a stand-alone momentum dump or a small ΔV, mode 1 is used to fire a subset of the 4-N thrusters, drawing fuel from the auxiliary tank. For an intermediate ΔV, mode 2 is used to fire the 22-N C thrusters, drawing fuel alternately from the two main fuel tanks. For a large ΔV, mode 3 is used to fire the LVA, drawing fuel and oxidizer from the main tanks. A settling burn using the A1, A2, B1, and B2 4-N thrusters is performed to move the fuel to the outlet end of the main tanks before firing the larger 22-N thrusters or the LVA for a mode 2 or 3 burn. For a mode 3 burn, there is also a trim segment where only the C thrusters are firing after the LVA is shut down. Initial preparation for a TCM is performed by ground command outside the G&C software prior to thruster firing. This preparation can also be triggered by the G&C software prior to firing thrusters for an autonomous momentum dump. The 1-Hz propulsion operations block manages the propulsion system configuration and monitors its status while thrusters are firing. Final clean up is performed in response to a signal from the G&C software after all thruster firing is terminated.

For mode 2 and 3 TCMs, the propulsion operations block completes main tank pressurization before thruster firing by opening regulator valves between the pressurant tank and the main tanks. It then manages the transition between the auxiliary fuel tank and the main fuel tanks, performing a complete or partial refill of the auxiliary tank. After the settling burn and any refill are complete, the fuel source for the thrusters is alternated by opening and closing the two main fuel tank valves following a pattern defined by ground controllers to maintain nearly equal amounts of fuel in each tank as the mission progresses. For mode 3 TCMs, the propulsion operations block pressurizes the oxidizer tank and opens its access valve before firing the LVA and closes the valves when LVA firing is complete. In addition to valve opening and closing, this block also turns off catbed heaters for the C thrusters or the LVA flange heater after the final firing of the associated thrusters.

Two types of checks are performed by the propulsion operations block to insure proper hardware configuration and operation. "Burn initiation" checks are done just before firing any thrusters to determine that conditions are correct to sustain thruster firing. If any of these checks is violated, the burn is not performed and the G&C software sets flags requesting propulsion system shutdown. "Burn abort" checks are performed continuously while thrusters are firing to determine if any changes have occurred that warrant immediate shutdown of the propulsion system. If any of these checks are violated, the block directs the transition to a state where only the 4.4-N thrusters are firing for attitude control while valves are closed for the main tanks and oxidizer tank, if necessary. After a duration sufficient to complete this reconfiguration, the G&C software sets flags requesting the remaining shutdown be performed followed by

demotion to a safe mode. The majority of these checks use various pressure readings from the tanks and fuel lines in the propulsion system, although one check uses a temperature reading from the LVA flange. The checks were specified by the propulsion vendor, Aerojet, based on system design and testing prior to launch. Examples are a check on the auxiliary tank pressure to insure sufficient fuel remains to complete another settling burn and checks on the main tank and regulator pressures to maintain the proper mixture ratio of fuel to oxidizer and proper operating pressure when firing the LVA. There are separate sets of checks for each of the three propulsion modes, each having a nominal time in each of the different segments of the mode where they should be enabled. Parameter blocks are available to change the enable state, alter the timing or pressure and temperature thresholds of each if desired.

**Solar Panel Control**

The solar panel control block supports three different control modes for the panels. There are separate and independent control paths for each panel, although both are commanded to the same orientations for most of the mission. The simplest control mode moves the panel to a specified angle fixed in the body frame; Sun direction is ignored in this mode. The other two control modes orient the panels relative to Sun direction as seen from the spacecraft.

The fixed Sun offset mode maintains a constant angle between the Sun line and the vector normal to the panel pointing out from the side covered with solar power cells. (Technically this mode maintains a constant angle between the projection of the Sun line in the YZ plane and the panel normal vector. Since the panels can rotate only about the X axis, offsets of the Sun direction normal to the YZ plane cannot be tracked by panel rotation.) The panel rotates as needed to maintain this offset when the spacecraft attitude changes and the Sun direction moves. The Sun offset angles can be slightly different for each panel when using solar torque for passive control of spacecraft Y momentum.

The third control mode also maintains a desired Sun offset angle but can apply incremental changes to the offset to keep the panel's temperature within a specified range. Median temperature for each panel is computed from up to three valid sensor readings. The algorithm compares the median temperature to specified maximum and minimum values at fixed intervals. An increment in the appropriate direction is applied to the current Sun offset angle if the temperature is outside one of the bounding values. The increment moves the panel normal closer to the Sun direction if the panel temperature is below the minimum value or farther from the Sun direction if the temperature is above the maximum value.

The solar panel control block also enforces restrictions on the absolute panel positions and on the Sun offset angle and inhibits panel motion in two different conditions. In any control mode, the commanded panel position will not exceed preset maximum and minimum values on absolute panel position. The bounding positions are normally set just inside the hardware stops of the SADA drives. For the two Sun-relative control modes, the commanded panel position will not make the angle between the panel normal and the Sun direction larger than a preset maximum value which is nominally set to just inside the physical limit of 90°. The first inhibit condition will prevent panel rotation when the spacecraft rotation rate is above a preset magnitude. The second inhibit condition will prevent panel rotation while thrusters are firing. Both of these checks can be enabled and disabled by setting parameter values. The rotation is inhibited by setting the commanded panel position to the actual panel position.

The solar panel control block computes the actual panel position from potentiometer reference and position voltages and computes the difference between the two panel angles which is used by FPP autonomy rules. Median panel temperatures are also provided to the FPP. The low-level commanding of the SADA stepper motor is done outside the G&C Simulink-derived software using the commanded and actual panel positions computed in this block. Panel orientation commands or commands to inhibit panel motion can be sent directly to this motor control software, bypassing the G&C software's commanded positions. This software also checks the progress of each commanded rotation and reports a "slew failed" condition to the FPP when the sensed panel position does not reach the commanded position.

**Antenna and Instrument Interfaces**

Each phased-array antenna has 121 separate command words specifying the desired boresight direction in 1° increments over the 120° scan range. Two look-up tables are used to store these command words for the front and back antennas. The G&C software determines the look-up table and index in that table once each second based on the spacecraft to Earth direction derived from estimated attitude and ephemeris models. The Earth azimuth angle is the angle in the *XY* plane measured clockwise from the –*Y* axis to the projection in the *XY* plane of the unit vector from the spacecraft to the Earth. The look-up table index is set to that of the closest scan angle when Earth azimuth is within one of the antenna's scan ranges. It is set to the edge of the nearest region when Earth azimuth is outside both antenna scan ranges (index = 0 or 120). The two phased array steering parameters are passed to the phased-array controller task at a 1-Hz rate, regardless of the actual spacecraft attitude. For successful phased-array communications, the Earth elevation angle (angle of the Earth direction above or below the XY plane) must be near zero and azimuth must be in one of the antenna scan ranges. The low-level interface to the transponders that provides electronic steering commands for the antennas is done outside the G&C Simulink-derived software using the look-up table information output by the 1-Hz guidance block. This includes a "bit flip" of the transponder command word depending on which transponder is active. Antenna steering commands can also be sent directly to the transponders, bypassing the G&C software's steering information.

Two of the pointing options use the MDIS pivot range of motion as an additional degree of freedom. One of these options is designed to point the MDIS boresight directly at a specified target. In computing the commanded attitude, the software attempts to maintain the desired Sun direction in the body frame by using a non-zero MDIS pivot position. If the required pivot position is beyond the allowable range, the software then changes the Sun direction as much as possible within the active SKI bounds to get the MDIS boresight as close as possible to the target direction. This action effectively changes the spacecraft attitude. Another pointing option is designed to achieve a "double target" by first pointing a boresight fixed in the body frame at a primary target and then using the MDIS pivot range to point the MDIS boresight at a secondary target. The primary targeting for the fixed boresight is given precedence and determines spacecraft attitude. An MDIS pivot angle is computed that moves the camera boresight as close as possible to the secondary target without compromising the primary pointing. The guidance block outputs an MDIS pivot position to the DPU every second regardless of the active pointing option. The appropriate position within the operational range is sent when one of the MDIS pointing options is in use; otherwise a default position of 0° is sent. The low-level commanding of the MDIS pivot motor is done in the DPU software. Since this software supports manual commanding of the pivot that bypasses the G&C software, it must be commanded to use the G&C-provided position when one of the MDIS pointing options is in use. There is no feedback of actual pivot position to the G&C software.

The guidance block provides a range and "slant angle" to the MLA. These quantities are computed by solving for the intersection of the MLA boresight direction with the target planet's surface given the current estimated spacecraft attitude. The range is simply the distance from the spacecraft to the surface along the MLA boresight direction. The slant angle is the angle between the tangent to the planet's surface at the point where the boresight intersects it and the boresight direction. Since the MLA instrument can sense the surface reflections of the laser pulses only below a certain maximum altitude, these computations are only done when the spacecraft's distance from the target planet's surface is below this threshold. When above the maximum altitude or when the attitude is such that the MLA boresight does not intersect the planet's surface, the range and slant angles are set to maximum values indicating that no observation is possible. Like the antenna and MDIS interfaces, the MLA can be manually configured, bypassing the range and slant angle information provided by the G&C software. A command must be sent to instruct the instrument to use the G&C-provided information in setting its configuration.

**CONCLUSION**

Overall, the MESSENGER G&C system has been performing well since launch in August 2004. The spacecraft has responded correctly to commanded attitude and rate changes, and the desired attitudes have been maintained within the accuracy needed for current flight operations. Most of the pointing options and the scan pattern overlay have been exercised to achieve attitudes needed for spacecraft check out and science instrument calibration. Several tests of planned Mercury observations were carried out during the Earth flyby in August 2005. Five TCMs have been performed exercising all thrusters except the LVA. The only significant problem has been the impact of the time drift between IMU and MP clocks when commanding certain attitudes and executing TCMs. New flight software has been developed to alleviate these problems and is scheduled to be uploaded to the spacecraft and tested in-flight in fall of 2005.

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