NEAR SPACECRAFT AND INSTRUMENTATION[1]

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The NEAR spacecraft design is mechanically simple, and geared toward a short development and test time. Except for the initial deployment of the solar panels and protective instrument covers, the spacecraft has only one moveable mechanism. Its' distributed architecture allows parallel development and test of each subsystem, yielding an unusually short spacecraft integration and test period. Several innovative features of the NEAR design include the first use of an x-band solid state power amplifier for an interplanetary mission, the first use of a hemispherical resonator gyroscope in space, and extremely high-accuracy high voltage power supply control. The spacecraft design enables exciting, first-rate science return with low cost, quick turn-around, and low technical risk.

INTRODUCTION

The NEAR spacecraft is being developed at The Johns Hopkins University Applied Physics Laboratory (JHU/APL) as a project under the NASA Discovery Program, a series of small-scale, low-cost, quick turn-around spacecraft projects. NEAR has a 29-month development time, and will be launched in February, 1996, by a Delta II booster. The spacecraft will enter deep space and after approximately three years of transit time, rendezvous with the near-earth asteroid Eros. The spacecraft will remain in orbit around Eros for approximately one year, during which time scientific data from on-board instruments will be collected and transmitted to Earth. The top-level system requirements for the spacecraft are summarized in Fig. 1.

SYSTEM DESCRIPTION

Figure 1 shows the spacecraft in the deployed flight configuration. A 1.5 m antenna and four solar panels are mounted on the outside of the forward deck. The solar panels are folded down along the spacecraft sides during launch and are deployed shortly after separation from the launch vehicle. Most electronics are mounted on the forward and aft decks. The science instruments, except for the magnetometer, are hard-mounted on the outside of the aft deck with co-aligned fields-of-view. The magnetometer is mounted on the High Gain Antenna (HGA) feed. The interior of the spacecraft contains the propulsion module.

The spacecraft design was selected for its' mechanical simplicity. The solar panels, the HGA, and the instruments are all fixed. The solar panels and HGA can be fixed because throughout most of the mission the Sun-spacecraft-Earth angle is less than 40 degrees[1]. While the HGA is pointed to the Earth, the Sun-solar panel angle is small and the energy output from the resultant solar illumination incident on the panels is sufficient. During the first two months after launch and for a short time during the Earth flyby, the Sun-spacecraft-Earth angle is greater than 40 degrees. At these times, the telecommunication link is sufficient to allow communications with the Earth through the medium

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Top Level Requirements

- 3-axis attitude control
  - 1.7 millirad pointing stability
  - 50 microradians pointing knowledge
- 1420 m/s ΔV
  - ΔV magnitude accuracy 0.2%
  - ΔV pointing accuracy 6 millirads
- 3 kbps downlink during asteroid operations
- 100 Mbits of on-board data storage
- On-board time knowledge to 20 milliseconds
- Survive unattended for up to 12 days

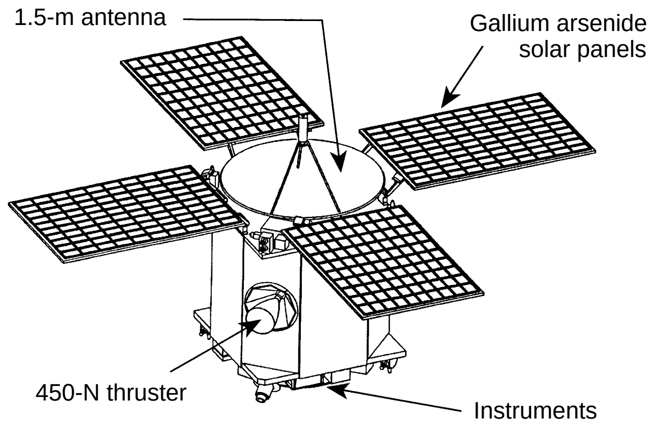

Fig. 1. NEAR Spacecraft Summary

or low gain antennas while keeping the solar panels pointed within 40 degrees of the Sun. During asteroid operations, the geometry allows the spacecraft to be oriented as needed for scientific data-taking, while maintaining the required sun-solar panel angle.

While mechanically simple and reliable, hard-mounting the HGA, solar panels, and instruments drives other areas of spacecraft and mission design. The resultant spacecraft moments-of-inertia are such that closed-loop control of the vehicle pointing must be maintained throughout the mission. Because the power system is designed for 100% sunlight operation, the 450 N thruster location had to be chosen such that the panels could be oriented towards the Sun during all large ΔV maneuvers. Finally, scientific operations and high speed downlink to Earth cannot always be carried out simultaneously.

Table 1 shows the mass and power breakdown of the spacecraft. The launch weight of the spacecraft is fixed at 805 kg. At the launch site, any unused dry mass will be added to the fuel weight allotment. The power system is sized to meet the system requirements during transit at aphelion, where the minimum power margin occurs. Meeting this requirement provided more than enough power for operation of instruments during the scientific part of the mission.

The system level block diagram is shown in Fig. 2. The spacecraft is designed with a distributed architecture where subsystems do not share common hardware. This design approach allows parallel subsystem development, test, and integration so that the compressed spacecraft development and integration schedule can be met. Distributed among the subsystems are seven processors: one in the Command and Data Handling subsystem, two in the Guidance and Control subsystem, and four that control the instruments. These processors allow inflight reprogrammability of functionality. All processors are Harris RTX2010's except the G&C subsystem Flight Computer which is a Honeywell 1750A. The use of the MIL-STD-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 cables.

MECHANICAL AND THERMAL SUBSYSTEM

The spacecraft is to be launched on a Delta 7925-8. Figure 3 shows the spacecraft in its launch configuration. In the launch configuration, the four solar panels are restrained around the body of

Table 1. Mass and Power Summary

Component	Mass (kg)	Power (W)
Instruments		
Multi-Spectral Imager (MSI)	7.8	13.9
NEAR Imaging Spectrograph (NIS)	14.2	20.0
X-Ray/Gamma-Ray Spectrograph (XGRS)	27.3	31.3
Magnetometer	1.6	1.5
Laser Rangefinder	5.1	26.8
Propulsion		
Propulsion Structure	33.1	
Propulsion System	85.1	
Propellant and Pressurant	319.7	
Power		
Solar Panels	46.1	
Battery	12.2	4.3*
Power System Electronics	6.1	2.5*
Telecommunication		
High Gain Antenna	6.5	
Medium/Low Gain Antennas	0.7	
Solid State Amplifiers (2)	4.1	38.7*
Transponders (2)	8.2	18.1*
Command Detector Units (2)	0.7	
Telemetry Conditioner Units (2)	1.7	3.8*
RF Switches, Coaxial cables	3.0	
Guidance and Control		
Reaction Wheels (4)	12.9	20.0*
Star Tracker	2.7	9.9*
Inertial Measurement Unit	5.3	21.4*
Digital Sun Sensors (5)	1.9	0.3*
Attitude Interface Unit	6.4	10.8*
Flight Computers	4.7	8.0*
Command and Data Handling		
Command and Telemetry Processors (2)	9.8	18.2*
Solid State Recorders (2)	3.0	6.4*
Power Switching Unit	5.9	0.7*
Mechanical		
Spacecraft Primary Structure	78.0	
Spacecraft Secondary Structure	18.1	
Despin Mass and Balance Mass	6.1	
Thermal		
Thermal Blankets, Heaters, Thermostats	11.0	
Propulsion Survival Heaters		75.8*
Spacecraft and Instrument Survival Heaters		71.0*
Instrument Operations Heaters		40.2
Harness		
Harness and Terminal Boards	38.8	4.5*
Totals	787.8	314.4*

* indicates configuration at minimum power point

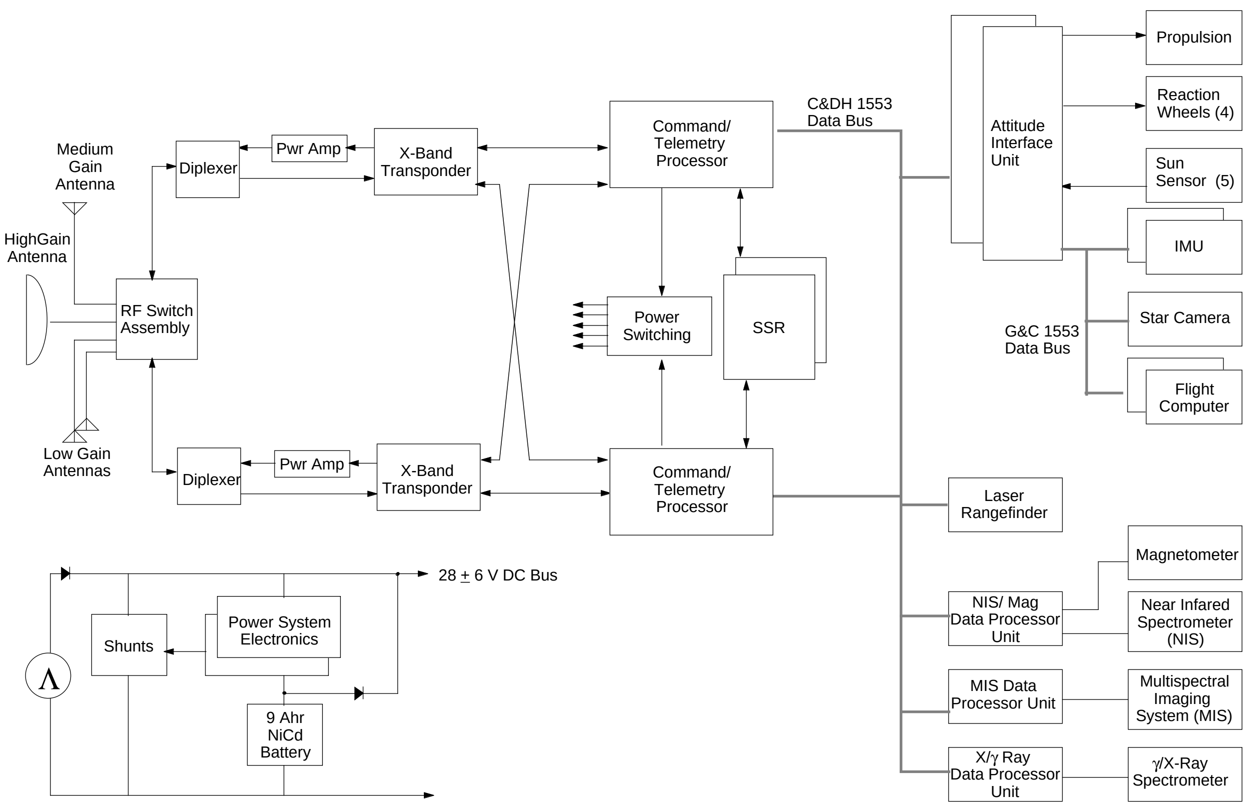

Fig. 2. System Block Diagram

Fig. 3. Spacecraft in Launch Configuration During Vibration Testing

the spacecraft. After separation from the third stage of the Delta, a Yo-Yo despin mechanism on the spacecraft simultaneously releases the solar panels and despins the spacecraft from a maximum of 69 revolutions per minute (rpm) to a nominal spin rate of 0 rpm. Once they are released, spring loaded hinges deploy the solar panels to the on-orbit configuration. The use of a single mechanism to both release the solar panels and remove the spacecraft angular momentum allows a reduction of system mass. This technique has extensive APL heritage.

The spacecraft is designed with two independent structures: the propulsion system and the spacecraft. APL designed and fabricated the spacecraft structure. Aerojet, the propulsion system vendor, built the propulsion system structure (described in the Propulsion System description below). These systems are coupled at the aft deck. While this design exacted a penalty in weight, it allowed a clean interface definition, and independent design and test capability for the propulsion subsystem.

Figure 4 is an exploded view of spacecraft structure. The spacecraft structure is composed of the spacecraft adapter, two decks, and eight side panels. The spacecraft adapter is a one-piece aluminum forging. It conforms to the standard McDonnell Douglas 3712-C clamp-band interface. The top of the payload adapter connects directly to the aft deck. The aft deck is 1.47 x 1.47 x 0.057 m aluminum

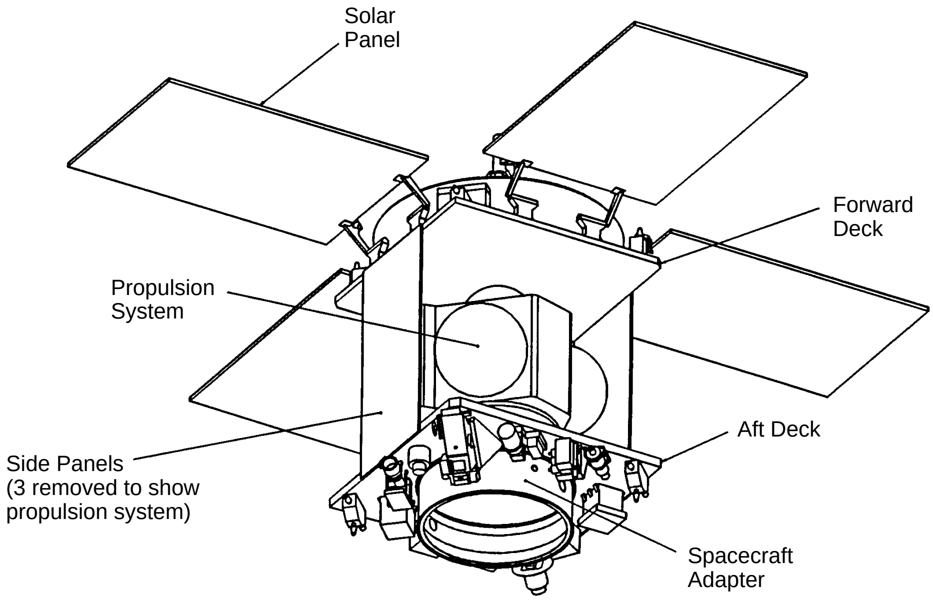

Fig. 4. Spacecraft Structure

honeycomb core with 0.64 mm 2024-T81 aluminum facesheets. The inserts and edge members are magnesium. The instruments are mounted on the aft side of the aft deck, while components of the Guidance and Control subsystem are mounted on the forward side. The propulsion subsystem mounts to solid aluminum inserts in the aft deck creating a direct load path through the spacecraft adapter to the Delta third stage.

The eight side panels carry the loads from the aft deck to the forward deck. The side panels are 12.7 mm aluminum honeycomb with 0.13 mm aluminum face sheets. Four of the panels are removable for integration of units mounted in the spacecraft interior. The panels are bolted to each other, and to the forward and aft decks. To minimize structure mass, there is no external framework to which the sidepanels are bolted, rather edge members in the side panels make up the frame.

The thermally-benign mission geometry allows a simple thermal design using passive radiators and thermostatically controlled heaters to keep the spacecraft components within their operating temperature limits. The spacecraft is designed to run near its' cold operating limit during the transit to the asteroid, when the spacecraft is in a low power consumption mode, so that when all instruments are powered during asteroid operations, the spacecraft will still operate below its' upper operating temperature limit.

Multi-layer insulation covers the entire spacecraft except for areas designated as radiators. The thermal blankets are composed of a Kapton outer layer and 15 aluminized mylar inner layers separated by a Dacron mesh. The propulsion system, the HGA, some instrument sensors, and the battery are thermally isolated from the spacecraft using multi-layer insulation and low conductivity mounting hardware. The HGA is painted with S13/ZGLOP-1 to keep the reflector temperature below 90 degrees centigrade at 1 AU, and to prevent solar focusing on the feed. The embedded 450 N bi-propellant thruster is surrounded by a gold-plated heat shield to protect the spacecraft during burns. The radiator surfaces are areas of the spacecraft side panels covered with 0.0127 mm silver Teflon.

The spacecraft heaters are all redundant and are controlled by mechanical thermostats. The setpoints of the primary and secondary heater thermostats are offset so that the secondary heaters will never be energized unless a primary heater fails. In addition to the thermostatic control, the heaters are grouped on relays based on the thermal sensitivity of the component. Instrument heaters are placed on their own bus, because of their sensitivity. The thermal inertial of the propellant tanks make them relatively insensitive to heater power cycles. This feature is used during emergency low power situations when the tank heaters can be turned off for up to 12 hours to allow for power system recovery. For additional thermal margin, up to 44 W of excess power from the solar panels can be selectively directed to forward and aft deck heaters, rather than to the dissipative analog shunts on the backs of the panels.

The largest, highest watt density thermal load comes from the Solid State Power Amplifiers (SSPA). To allow steady-state operation of one of the SSPAs, they are mounted directly to the inside of a side panel on a radiator doubler. The doubler is 0.129 m^2 of aluminum-beryllium alloy 3 mm thick. The power amplifiers are radiatively coupled to the spacecraft interior. If both SSPA's are powered off, a 26 W heater will maintain the SSPA's within temperature limits.

PROPULSION SUBSYSTEM

The Aerojet-supplied propulsion system is shown is Fig. 5[2]. It is composed of one 450 N N_2H_4/NTO bi-propellant large velocity adjustment thruster (LVA), four N_2H_4 monopropellant 21 N thrusters, and seven N_2H_4 monopropellant 3.5 N thrusters. The monopropellant thrusters are arranged in six thruster modules mounted to the forward and aft spacecraft decks. The four 21 N thrusters point in the same direction as the LVA thruster. They are used for thrust vector control during the bi-propellant burns. The 21 N thrusters supply enough torque to compensate for any initial misalignment of the LVA thruster with respect to the spacecraft center-of-mass and the expected movement of the spacecraft center-of-mass as propellant is expended. The 3.5 N thrusters are used for momentum dumping and orbit maintenance around the asteroid. The locations of the monopropellant thrusters were chosen so that the loss of any one thruster does not affect performance.

The specific impulse of the LVA thruster is 313 seconds. The specific impulse of the monopropellant thrusters ranges from 206 to 234 seconds. A minimum ΔV increment of 10 mm/s is achievable in all directions. An impulse bit of 0.2 Ns is achievable using the 3.5 N thrusters, 0.7 Ns for the 21 N thrusters. A total ΔV of 1450 m/s is achievable.

The propulsion structure is composed of two pieces; an interface support structure (ISS), and a core propulsion system structure (PSS). The ISS is a circular section with forward and aft flanges. The forward flange connects to the PSS, the aft flange connects to the spacecraft structure. In addition to supporting the propulsion system structure, the ISS provides thermal isolation between the spacecraft deck and the core propulsion system. The entire ISS is a graphite epoxy construction. The PSS consists of an aluminum honeycomb core bonded to graphite epoxy face sheets.

The tank mounting inserts and the PSS to ISS interface spools are installed into the honeycomb before the interior layer of composite is installed. This results in a tightly dimensional controlled monocoque hexagon shaped structure with integral tank mounting inserts. The tanks and fluid components are connected using 3AL-2.5 V titanium tubing. The system manifolds are completely welded. All of the 12 system thrusters have CRES inlets lines and are connected to their respective titanium manifolds with the use of inertially welded bi-metallic tube joints.

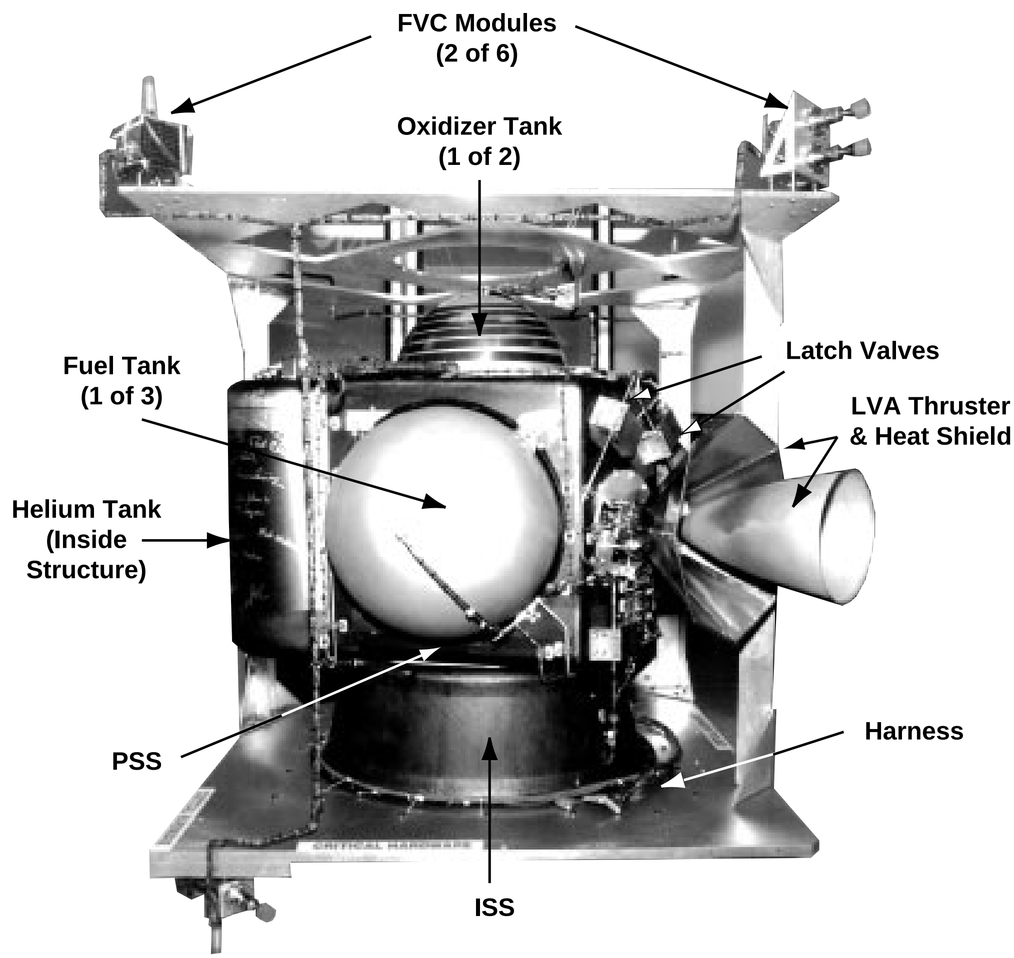

Fig. 5. Propulsion System

The propulsion system carries 209 kg of hydrazine and 109 kg of NTO oxidizer in two oxidizer and three fuel tanks. The 55.1 liter oxidizer tanks are located along the launch vehicle spin axis equidistant from the spacecraft center-of-mass. The oxidizer tanks have in-line propulsion management devices and use the forces from a settling burn for delivery of gas-free propellant. The 91.0 liter fuel tanks are arranged 120 degrees apart in the LVA thruster plane. They use elastomeric diaphragms for positive gas-free expulsion. The location of the propulsion tanks is selected to maintain the spacecraft center-of-mass along the thrust vector of the 450 N thruster throughout the mission as the propellant is depleted. Center of mass control is achieved by timing the opening and closing appropriate propellant tank latch valves during the large burns. This timing is pre-programmed from the ground.

The propulsion system hydraulic system schematic is shown in Fig. 6. Pressurization is by a common regulated helium system. A 43.4 liter tank contains 1.7 kg of helium. The high pressure helium is isolated from the oxidizer and fuel by a high pressure latching valve and a series redundant regulator. At launch the helium pressure is 21380 Kpa. The regulated pressure to the propellant tanks is 1655 Kpa. Propellant vapor isolation is achieved by five check valves and an oxidizer pressurant latch valve. Additionally, the common pressurant line is maintained at a higher temperature than the tanks to minimize oxidizer condensation in the common pressurant system. Four pressure transducers monitor the downstream regulator and tank pressures.

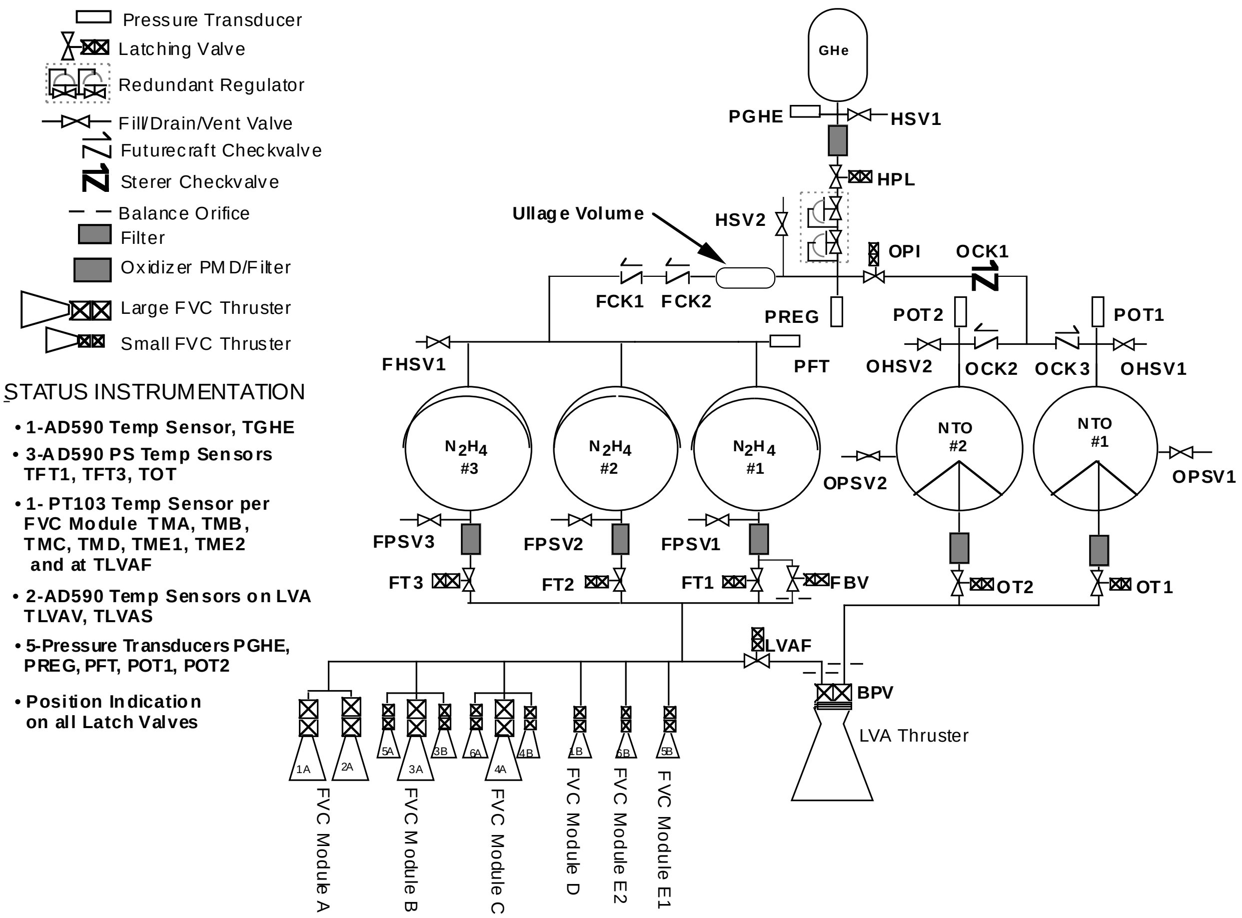

Fig. 6. Propulsion System Hydraulic Schematic

Fuel flows from the tanks through individual filters and the tank latch valves to the inlets of the monopropellant thrusters, or through the LVA latch valve to the inlet of the bi-propellant thruster. Oxidizer flows from the tanks through individual filter/propellant pressure management device assemblies, through the tank latch valves and into the inlet of the bi-propellant thruster. Fill, drain, and vent valves are located throughout the system to facilitate pressurant and propellant servicing, as well as functional and leak testing. An N_2H_4 latching bleed valve is provided to control the manifold fill rate, and minimize the pressure surge prior to the opening of a larger fuel tank latch valve.

THE TELECOMMUNICATION SUBSYSTEM

The telecommunication subsystem is an X-band system capable of simultaneously transmitting telemetry data, receiving spacecraft commands and providing doppler and ranging tracking. Figure 7 is a subsystem block diagram[3]. Redundant, unswitched transponder/command detector unit (CDU) pairs are connected to the antennas through a coaxial switching network, allowing two separate command reception paths at independent data rates. The redundant telemetry conditioning units (TCUs) are cross-strapped to the transponder exciters.

The redundant Motorola transponders are the core of the telecommunication system. The transponders are a modified Cassini design that provides a wider input bus voltage, different frequency selections, and a coaxial receiver input instead of waveguide. The use of coaxial cabling reduces mass and minimizes the complexity of the mechanical assembly.

Uplink data is demodulated by a command detector unit (CDU) designed for the Cassini mission and provided by the Jet Propulsion Laboratory (JPL). The NEAR CDU is identical to the Cassini design, however for simplicity the command/telemetry processors are hardwired to the CDUs so that only the 7.8 bps and the 125 bps data rates can be selected. The 7.8 bps rate is used for emergency communications; the 125 bps rate is for normal communications.

The downlink hardware, developed by APL, uses a solid state power amplifier with an output level of 5 watts. This represents the first use of solid state X-band amplification for a deep space mission. Two convolutional codes are supported: rate 1/2, k=7 code, and rate 1/6, k=15 code. The rate 1/2, k=7 code is used at launch and during cruise until the Deep Space Network (DSN) can support the rate 1/6, k=15 code. Rate 1/6 is baseline for the entire asteroid rendezvous phase. For either rate code the data is concatenated with a Reed-Solomon 8 bit (255,233) block code.

Downlink support assumes DSN 34-meter high efficiency and beamguide antennas, with 70 meter dish support for critical operations and emergencies. Eight downlink data rates are used: 9.9 bps, 39.4 bps, 1.1 kbps, 2.9 kbps, 4.4 kbps, 8.8 kbps, 17.6 kbps, and 26.5 kbps. The selected rate is a function of the downlink coding scheme, the DSN asset, the spacecraft to earth distance, and the Sun-spacecraft-Earth geometry.

The 9.9 bps rate is used for emergency recovery. The 39.4 bps rate is used when the spacecraft is within 0.5 AU of earth, allowing the spacecraft to fly in a conservative sun pointed attitude for as long as possible while maintaining an acceptable downlink data rate on a medium or a low gain antenna. At distances greater than 0.5 AU, the high gain antenna is Earth pointed, and a 1.1 kbps downlink rate is used. This configuration supports the aphelion burn and the Mathilde flyby[1].

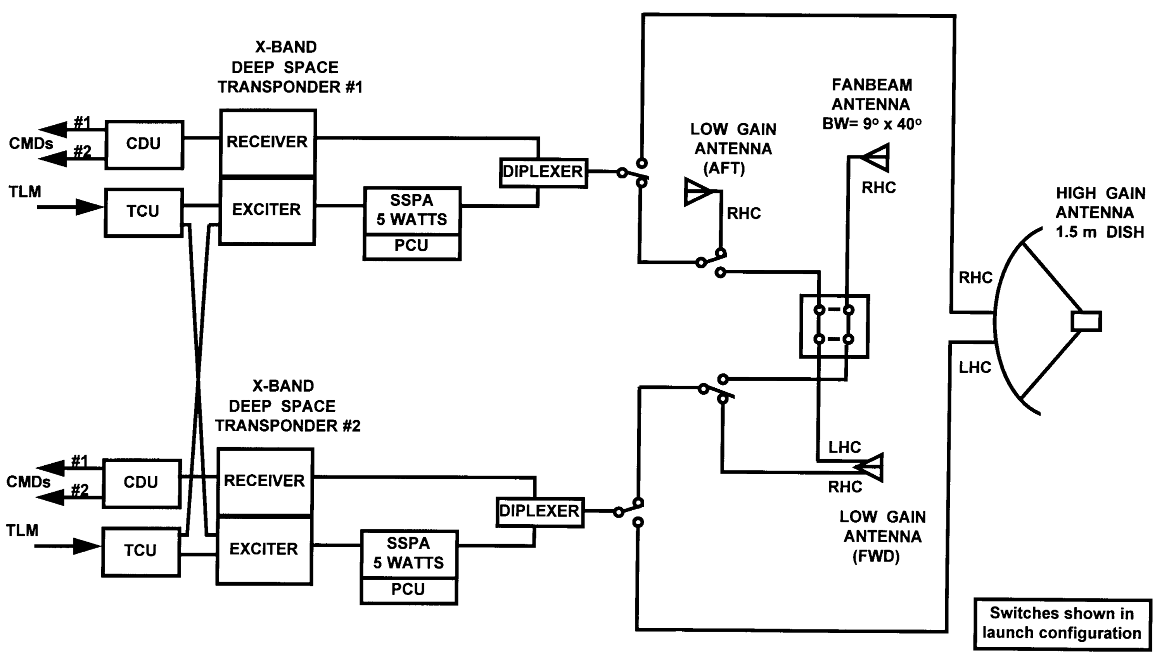

Fig. 7. Telecommunication Block Diagram

The remainder of the rates between 2.9 bps and 8.8 kbps are used for the asteroid rendezvous phase with the DSN 34 meter antenna. The occasional use of the 70-meter DSN antennas for science operations will permit data downlink at the 17.6 kbps and the 26.5 kbps data rates. The anticipated downlink return for the asteroid phase is shown in Fig. 8. The data rate increases over the rendezvous period as function of the decreasing spacecraft to earth range.

The primary antenna is a high gain 1.5 m paraboloid. The downlink gain is 40 dBic. Co-located on the feed assembly are a low gain antenna and the magnetometer. The presence of the magnetometer requires careful selection of materials for the feed assembly and the coaxial cable connectors.

The low-gain hemispherical antennas mount on the fore and aft ends of the spacecraft. When connected to the transponders through the switching network, the hemispherical antennas effectively provide omnidirectional coverage. The low gain antenna is a dual frequency microstrip patch that provides a peak downlink gain of 6 dBic. The medium-gain, fanbeam antenna is a dual frequency microstrip array. This antenna provides wide plane coverage from the boresight of the high gain antenna to 40 degrees off the boresight. The narrow plane 3 dB beamwidth is 8 degrees. The peak downlink gain is 10 degrees off the boresight of the high gain antenna and is 18.8 dBic. In addition to its near-Earth use, the medium gain antenna is used as the emergency antenna to locate the Earth and as a wide beamwidth antenna during asteroid operations to enable a relaxation of the Earth pointing requirement.

POWER SUBSYSTEM

The power system comprises four 1.83 x 12.2 m gallium arsenide solar panels, a 9 amp-hour super NiCad battery, and the power system electronics as shown in the block diagram (Fig. 9)[4]. The power system is a direct energy transfer system. The spacecraft bus is regulated at 33.5 +/-0.5V

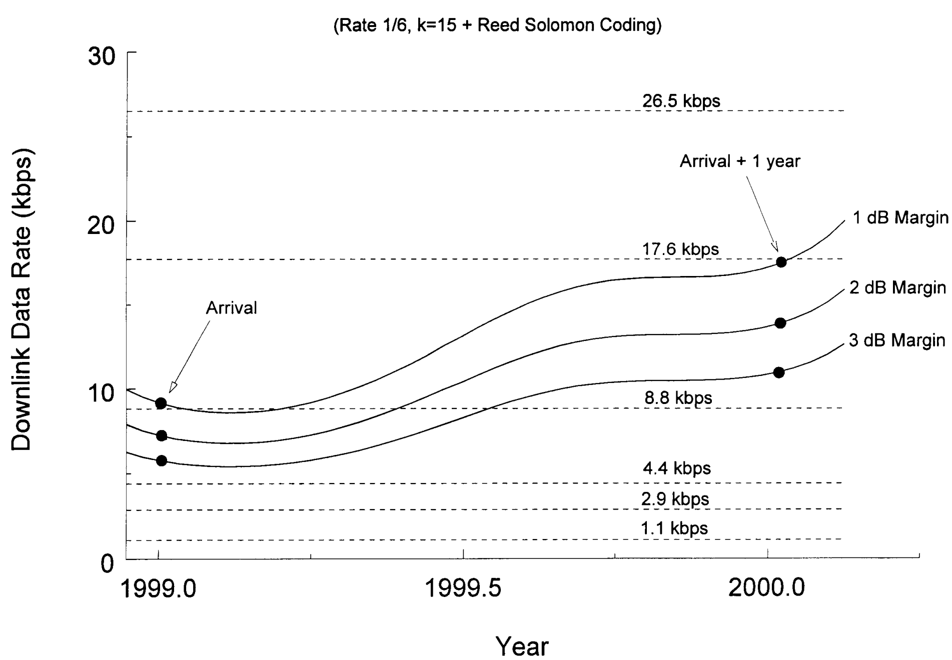

Fig. 8. Downlink Data Rate at Eros with DSN 34m Antenna

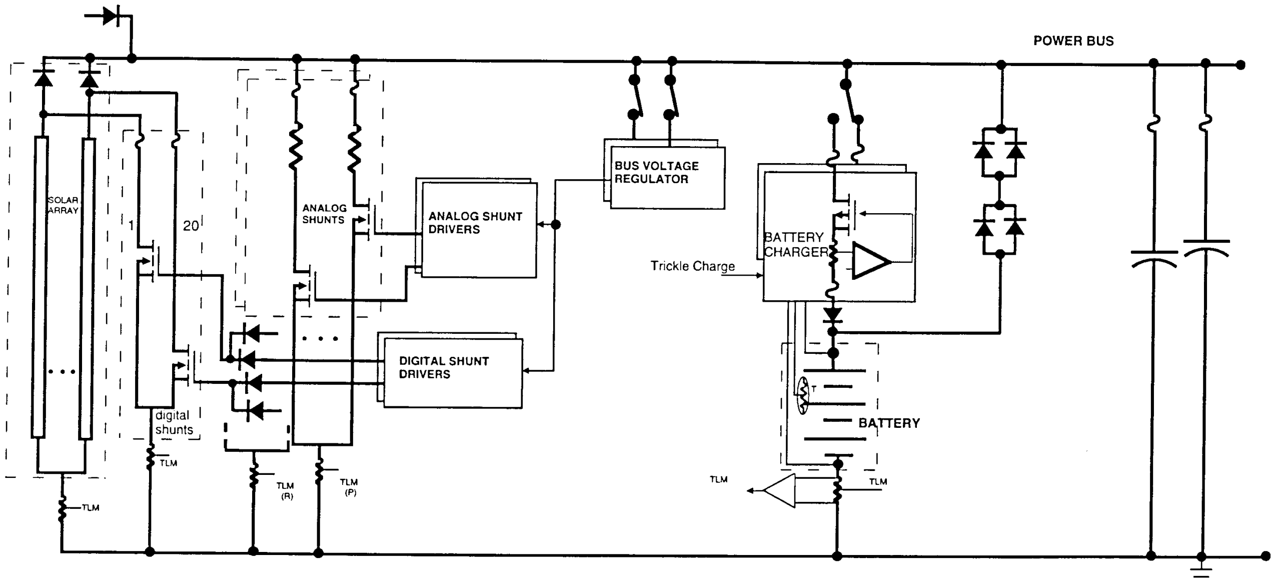

Fig. 9. Power System Block Diagram

when the solar array power is adequate to supply the load and battery charge power. The bus follows the battery voltage whenever the battery is in discharge.

At the beginning of life, the solar array produces 1880 W at 1 AU from the Sun. At aphelion – 2.2 AU from the Sun – the solar array power is estimated to be 400 W. The panels are 25.4 mm aluminum honeycomb core with 0.003 mm aluminum facesheets. On the front of each panel there are 960 solar cells connected with silver mesh. Cells are isolated from the aluminum substrate by Kapton. Spectrolab supplied and installed the GaAs/Ge solar cells with UVR-coated, CMX cover slides. These cells have an average beginning-of-life efficiency of 18.5%. On the back of each panel are three dissipative analog shunts composed of thin-film foil heaters encapsulated in Kapton. The backs of the panels and shunts are painted with A-276 thermal paint.

The solar cells are arranged in five groups per panel, with each group independently shunted through non-dissipative digital shunts. Coarse solar array power control and bus voltage regulation are performed by sequentially turning on and off the digital shunts in response to the available solar array power and spacecraft load demand. The fine solar array power control is performed by a six stage linear sequential full dissipative shunt. When the power changes are such that the linear shunt range is exceeded the digital shunts are activated. The shunting sequence is designed to minimize magnetic disturbance to the magnetometer.

The battery is a Hughes 9 ampere-hour, 22 cell super-NiCad battery. The battery cells were fabricated by Eagle Picher. The recommended storage mode for the super-NiCad battery is to maintain it fully charged at cold temperatures at a trickle charge rate, which is compatible with the mission profile. The battery is charged from one of the redundant battery chargers. Two levels of battery charging are selectable by command: a constant C/75 trickle charge, or a constant current charge tapering to a pre-selected temperature-compensated voltage limit. Eight V-T curves are available by command. The battery temperature is expected to remain between 0-10 degrees centigrade during flight. No inflight battery reconditioning is planned.

The power system electronics contains the electronics that are required to regulate the power bus, control the battery charging, control shunting of excess power, and interface command and telemetry with the spacecraft. All electronics are redundant.

COMMAND AND DATA HANDLING SUBSYSTEM DESCRIPTION

As shown in Fig. 2, the Command and Data Handling (C&DH) subsystem comprises redundant APL-built command and telemetry processors, redundant Solid State Recorders (SSR), a power switching unit to control spacecraft relays, and an interface to a redundant 1553 standard bus for communicating with other processor-controlled subsystems. The redundant components are cross-strapped among themselves, and among the redundant uplink chains of the telecommunications subsystem. The functions provided by the C&DH subsystem are command management, telemetry management, and autonomous operations.

The command function operates on cross-strapped inputs from the two CDU's at either of two rates: 125 bps (normal mode) or 7.8 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 C&DH subsystem. Four types of commands are supported: relay commands are directed to the power switching unit to change the state of the spacecraft relays; dedicated data commands are directed over specific serial interfaces to control SSR and telecommunications subsystem operation; 1553 data commands are directed to a specified remote terminal on the 1553 bus; and C&DH-specific commands are interpreted within the C&DH to control its own operation.

Some of the C&DH-specific commands provide facilities for storing commands for later execution, either at a specified Mission Elapsed Time (MET), or when the spacecraft conditions warrant autonomous action. A series of commands that perform a specific function can be stored as a 'command macro;' the entire series of commands can be invoked by a single 'macro execute' command. During normal operations, the C&DH will invoke command macros that have been scheduled for execution at a specific MET. In this way, operations are carried out when the spacecraft is out of ground contact. Command macros can also be invoked by the autonomy function of the C&DH, to place the spacecraft in a safe condition. Approximately 56 K bytes of memory, 4000 commands, is available for stored commands in each processor.

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 C&DH internal event history buffers. This data is packetized where necessary, and packed into CCSDS-compliant transfer frames. The transfer frames are directed to the SSR's, the downlink, or both. Data recorded on the SSR's is read back, packed into transfer frames and placed into the downlink on command. Recorder playback data can be interleaved with realtime data on the downlink, and data can be recorded on one of the redundant SSR's while the other recorder is read back.

SEAKR provides the SSR's. These recorders are constructed out of 16 Mbit IBM Luna-C DRAMs. One recorder has 0.67 Gbits of storage; the other recorder has 1.1 Gbits of storage because it contains an additional memory board which is designated as the flight spare, and will be used to replace either of the other memory boards in the event of a ground test failure.

The downlink data rate is selectable among eight rates ranging from 26.5 kbps to 9.9 bps to match the communication link capability throughout the mission. For all except the highest downlink rate, the recorder capacity exceeds the downlink capacity, so bandwidth is limited by the downlink. While the C&DH subsystem controls the rate of collection of realtime data to match the downlink rate, the rate at which data is placed on the recorder is under the control of the subsystems. Each remote terminal on the 1553 bus can request the C&DH to pick up and record up to 5336 bits of data per second. This feature allows the spacecraft operators complete flexibility with respect to the bandwidth used by each instrument.

The autonomy function of the C&DH is carried out completely through the use of rules that invoke command macros. These rules compare any of the housekeeping data—analog, relay telltale, digital from 1553 remote terminals—to limits or ranges that indicate improper operation. Separate comparisons can be made on up to two 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 exist at least as long as the rule timer before action is taken. Once a rule detects a condition that must be addressed, 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 C&DH 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.

Although the rule definitions are quite simple and predictable, a surprising degree of sophistication in autonomous response can be achieved. Several conditions (bus regulator failure, for example) require a simple algorithm to be followed to find and correct a failure. These can be constructed through the use of several interacting rules. The use of rules, rather than hard-coded autonomy, allowed the autonomy software to be coded and thoroughly checked out, while the appropriate autonomy rules and actions themselves were devised and modified until late in the development cycle. An additional benefit of the rule-based autonomy system is that the C&DH system needs only one mode of operation: during safing actions, the C&DH subsystem, in effect, sends commands to itself to take the appropriate safing actions. It does not need to 'know' whether it is in a safe or operational mode.

Most of the functionality of the C&DH subsystem is achieved through Forth-language software running on the command and telemetry processors, based on the Harris RTX2010. Because this subsystem is not reprogrammable in flight, it needs to be robust and well-tested, yet flexible. To achieve the goal of robustness and testability, the design was kept very simple. Each one-second major frame is divided into eight 125 ms minor frames. All tasks are allocated to one or more minor frames. For example, each minor frame, the software examines the possible command sources (realtime uplink, autonomy rules, timetagged command storage, for example) in priority order, and executes one command per minor frame from the highest priority source with a command pending (if any). Each remote terminal is assigned a minor frame during which the C&DH will pick up data from that source. Autonomy rules are evaluated during a particular minor frame. This time-slice design eliminates the unpredictable nature of realtime tasking systems. Further, as discussed above, the C&DH software can follow this same routine at all times—no separate paths are followed during different modes of operation.

Flexibility is achieved in two ways. First of all, commands to the G&C subsystem and the instrument subsystems appear to the C&DH to be simply data passed on to the appropriate destination for interpretation and action. Similarly, from the C&DH perspective, telemetry data are simply packets to be picked up and recorded at the subsystems' request. Since these other subsystems are reprogrammable, commands to modify their operation or changes to telemetry output can be made at will, without requiring any change to the C&DH. Thus, considerable flexibility in operating the spacecraft is available. Second, the autonomy rules and associated command macros provide a first-order infinity of possible behaviors that can be invoked on the C&DH. Because these rules and macros are reprogrammable (subject to default on reset), considerable flexibility in C&DH operation can also be achieved.

GUIDANCE AND CONTROL SUBSYSTEM DESCRIPTION

The Guidance and Control (G&C) subsystem is composed of a suite of sensors for attitude determination, actuators for attitude corrections, and processors to provide continuous, closed loop attitude control. Table 2 gives specifications for these items. In operational mode, the attitude is controlled to a commanded pointing scenario. In safe modes, the G&C maintains the solar panels pointed to the Sun for maximum power, and attempts to place the Earth within the medium-gain antenna pattern to establish ground communications. The G&C subsystem also controls the thrusters for ΔV maneuvers. Finally, the G&C subsystem recognizes many internal failure modes and initiate autonomous actions to correct them.

Table 2. Guidance and Control Component Descriptions

Item	Supplier	Characteristics
Inertial Measurement Unit	Delco	Gyros (4): 30 mm Hemispherical Resonator Gyros Rate bias < 0.01 deg/hr, over 16hr ARW < 0.001 deg/hr$^{1/2}$ Accelerometers (4): Sunstrand QA-2000 <100 µg RMS noise
Star Tracker	Ball	FOV: 20 x 20° Sensitivity: +0.1 to +4.5 M_i Coordinate Error (central 15°): 9 arcsec No. of stars tracked: 5 Output Rate: 5 Hz
Reaction Wheels	Ithaco	Brushless DC motor Momentum: 4 Nms (@ 5100 RPM) Torque: 0.025 Nm
Sun Sensors	Adcole	Quantization: 0.5° Accuracy: 0.25°
Attitude Interface Unit	JHU/APL	Clock: 6 MHz Memory (16 bit words): RAM: 64K EEPROM: 64K PROM: 2K Processor: RTX 2010
Flight computer	Honeywell	Clock: 9 MHz Memory (16 bit words): RAM: 512K EEPROM: 256K PROM: 16K Processor: MIL-STD-1750A

The G&C sensor suite shown in Fig. 2 comprises five Digital Solar Attitude Detectors (DSAD's), a Star Tracker, and an Inertial Measurement Unit (IMU). This IMU contains hemispherical resonator gyros for rate determination and accelerometers for measuring ΔV. This is the first use of hemispherical resonator gyros in space.

Although all of these elements are not block redundant, robustness is achieved both by internal redundancy and overlapping functionality. The IMU—a key component of the G&C subsystem—contains four gyros and four accelerometers aligned such that any three are sufficient for three-axis rate/acceleration measurement. Thus a single failure of a gyro or accelerometer can be accommodated. Additionally, the IMU contains redundant power supplies and processing units. Four of the five DSAD heads provide overlapping coverage of the spacecraft hemisphere facing the Sun (the fifth provides partial coverage in the opposite hemisphere to aid in recovery from an inadvertent tumble). Both the Star Tracker and the DSAD's provide an absolute inertial reference. Since the DSAD's provide only one inertial reference vector with poor resolution, they cannot substitute for the Star Tracker in an operational scenario, but they can be used with the gyro rate information to safe the spacecraft. The on-board visible imaging instrument provides a functional backup to the Star Tracker via ground processing of star images in operational modes.

The actuator complement contains four reaction wheels, 11 small, monopropellant thrusters, and one large bipropellant thruster. All normal attitude control is 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 results in no loss in functionality. External torques—such as solar radiation torque—cause system momentum to build up, eventually beyond the point where the wheels can be used to control the vehicle. The monopropellant thrusters are used to dump system momentum before this occurs. The thrusters are also used for attitude control during ΔV maneuvers since the torques produced by the thrusters overwhelm the reaction wheels.

Figure 2 shows the two processors used to drive the G&C hardware, the Attitude Interface Unit and the Flight Computer. Both processors are completely redundant and fully cross-strapped. Normally, only one of each type of processor is powered at a time. Redundant units may be powered for checkout, or reprogramming while the primary units maintain control.

The Flight Computer is used to perform the computationally intensive processing. Once each second, the outputs of the attitude sensor suite are processed to provide a filtered estimate of the spacecraft state (attitude and body rates) accurate to 50 microradians. A desired state is computed from the commanded pointing scenario—which may be specified in inertial or asteroid-fixed coordinate systems—using uploaded ephemerides of the Earth, the asteroid, and the spacecraft. These 1 Hz outputs are propagated forward in 40 ms steps until the next update. Control outputs are generated every 40 ms to null the difference between the commanded state and the observed state with a pointing accuracy of 1.7 milliradians. The results of the Flight Computer's calculations are passed over an internal G&C 1553 bus to the second G&C processor, the Attitude Interface Unit (AIU).

The AIU interfaces to all the G&C subsystem sensors and actuators. It performs the remote terminal functions on the main spacecraft 1553 bus to receive commands from and send telemetry to the C&DH. It performs all the bus controller functions on the internal G&C 1553 bus. The AIU collects data from the sensor suite at an appropriate rate and sends it to the Flight Computer over the G&C 1553 bus. It accepts the Flight Computer control output and sends it to the actuators at the 25 Hz rate. In addition, the AIU uses the Flight Computer's inputs to compute its' own control outputs. If the AIU does not receive a message from the Flight Computer in any 40 ms interval, it will send its' own computed control outputs to the actuators to maintain control of the vehicle. The AIU can propagate the observed state forward for some time using the gyro rate information, producing control output to maintain the last desired state. However, if the Flight Computer does not resume communications, the AIU is capable of using the IMU gyro and DSAD sensor data to perform the Sun Safe mode on its' own.

Because the spacecraft has no passive stabilization, the G&C actively controls the actuators at all times to maintain the desired attitude. Potential failures of sensors, actuators, and software must all be detected and tolerated to keep the spacecraft safe. Both the Flight Computer and the AIU perform checks on data and state, and change their mode of operation to accommodate detected anomalies. Some of the more important checks are the Sun Keep-in and Momentum Keep-in. The Sun Keep-in zone is a cone about the axis perpendicular to the solar panels; the Sun vector must stay within the keep-in zone, or a safing action will be triggered. To maintain control of the vehicle using the reaction wheels, the system momentum must be kept beneath a certain limit. If the G&C detects that system momentum has exceeded this limit, it initiates an autonomous momentum dump, requesting the C&DH to give it access to the thrusters. The C&DH monitors this thruster use, and restricts it to preset limits to prevent G&C from causing mission failure. The AIU performs certain checks on the Flight Computer input and, as noted above, can safe the spacecraft if the Flight Computer fails to communicate or to pass simple checks on its' operation. The C&DH subsystem uses autonomy rules to monitor the operation of the AIU software, and take action if an anomaly is detected.

The AIU is a RTX 2010 processor. It uses a foreground/background processing scheme programmed in C. The foreground processing is tied to the 25 Hz control loop rate. In each 40 ms time slice, the foreground processor performs routine tasks (e.g., control outputs) and some specific tasks assigned to that time slice. Then it returns to the background processor, which handles longer term, aperiodic tasks, such as command processing. The Flight Computer is a Honeywell 1750A programmed in Ada. It uses the standard Ada multi-tasking features to handle the timing of 25 Hz, 1 Hz, and longer term tasks.

INSTRUMENT SUBSYSTEMS

The facility instruments carried on the spacecraft are a visible-light imager, an IR spectrograph, a 3-axis magnetometer, an x-ray/gamma-ray spectrometer, and a laser rangefinder. The telecommunications subsystem full coherent transponder will also be used for radio science by measuring the two-way Doppler from normal spacecraft telemetry.

The Multi-Spectral Imager (MSI) designed and built by JHU/APL, Fig. 10, will provide visible images of the asteroid surface. It incorporates *f*/3.4 rad-hard refractive optics and an eight-position filter wheel covering the range from 450 to 1100 nm. The filter wheel is designed primarily to discriminate iron-containing silicate minerals, with one broadband filter for low-light imaging and optical navigation. The field-of-view (FOV) of 50.6 mrad by 39.3 mrad is divided into 537 x 244 pixels (161 x 95 mrad), giving a 10 x 16 m resolution at 100 km. The CCD is a frame transfer unit with electronic shuttering (10 ms to 1 sec), and anti-blooming control. At 12-bits per pixel, an uncompressed image is 1.6 Mbits. Using a dedicated high speed link to the Solid State Recorder's, an image rate of 1 Hz can be sustained. Various forms of data compression are selectable by command,

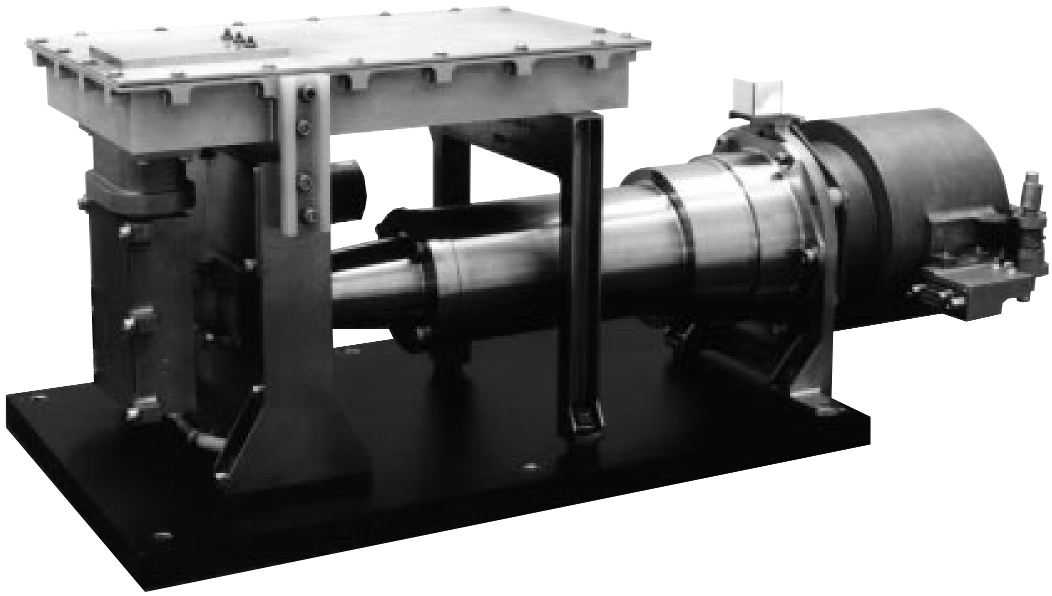

Fig. 10. Multi-Spectral Imager

as well as automatic exposure control. Using these latter options, or changing the filter between images, will lower the image rate.

The NEAR IR Spectrograph (NIS), Fig. 11, covers a 0.8 to 2.6 micrometer spectral range in 62 bins. This spectral range is achieved by dispersing the spectrum onto passively-cooled Ge and InGaAs line array detectors. The NIS uses a 1 second integration time, and can produce spectra at a 1 Hz rate. A 1-D scan mirror allows the NIS's FOV to be boresighted with the MSI, or scanned more than 1.57 rad away in 3.49 mrad increments. Mirror scanning combined with spacecraft motion will be used to build up spectral images. A three-position slit mechanism allows two FOV's: 13.3 x 13.3 mrad and 6.63 x 13.3 mrad. These provide spot sizes of 0.65 x 1.3 km or 1.3 x 1.3 km from a 100 km distance. The entrance slit can be closed altogether for dark count measurements. The NIS also carries a diffuse gold inflight calibration target that can reflect sunlight into the spectrograph by correctly positioning the spacecraft and the scan mirror. Sensor Systems Group supplies the NIS optics and mechanisms.

A three-axis fluxgate magnetometer can measure the asteroid's magnetic field from DC to 10 Hz. The sensor head is mounted on the high gain antenna feed, while the electronics are mounted on the spacecraft top deck. The sensor, supplied by NASA/Goddard Space Flight Center, has eight selectable sensitivity levels with full scale outputs ranging from 4 nT to 65,536 nT. Internally, the field is sampled at 20 Hz and converted to a 16-bit digital output. The signal can be digitally low pass filtered and resampled to lower the output bandwidth. The output of any single axis can be bandpassed between 1 and 10 Hz and reported in telemetry. The magnetometer has an internal calibration coil that can be exercised by command.

The X-ray/Gamma-Ray Spectrometer (XGRS) is really two instruments. The x-ray spectrometer, Fig. 12, measures x-ray florescence on the asteroid excited by solar flare x-rays using three gas-filled proportional counters: one bare, one with a Mg filter and one with an Al filter. Two solar monitors are mounted on the top deck of the spacecraft, facing away from the asteroid surface. One

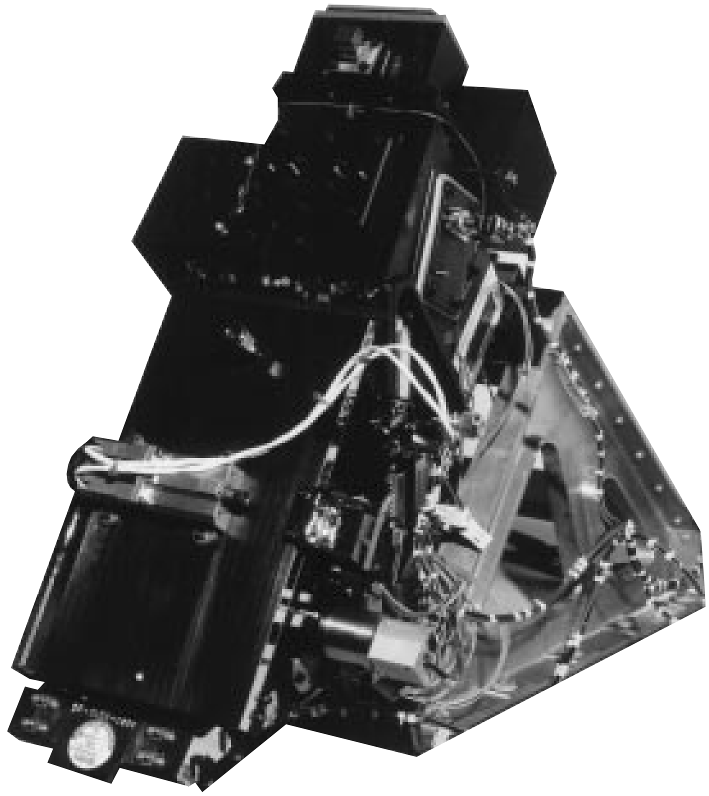

Fig. 11. NEAR IR Spectrograph

solar monitor is another gas-filled proportional counter. The second solar monitor is a new technology development: a high resolution, solid state Si detector with intrinsic thermo-electric cooler. The gas-filled proportional counters, collimated to an 87.3 mrad FOV, have an energy resolution of about 1 keV Full Width at Half Maximum (FWHM). The high resolution solid state solar monitor has about twice that resolution. Risetime discrimination is used to filter out cosmic rays, energetic particles, and other sources of background noise. Throughput is at least 10 kHz. An Fe-55 calibration source can be used for inflight calibration of this instrument.

The gamma-ray spectrometer is a NaI scintillator with an active BGO shield. The scintillator and shield use separate photomultiplier tubes. The instrument measures the energy range 0.3 to 10 MeV in 10 keV steps. Scintillator resolution is 8.5% FWHM at 662 keV. Active anti-coincidence is used to reduce the high energy gamma-ray background, with separate accumulation of the first and second escape peaks. A unique feature of both the x-ray and gamma-ray spectrometers is the high accuracy high voltage control electronics that maintains about 0.5 V stability of the high voltage (1500 V) power supplies over the full temperature range.

The laser rangefinder (NLR), Fig. 13, is a direct-detection single-pulse rangefinder. It uses a diode-pumped Nd:YAG laser transmitter, supplied by McDonnell Douglas, at a wavelength of 1.06 mm. The laser delivers a 15 mJ pulse of 12 ns duration. It has a maximum divergence of 300 mrad at the instrument's maximum range of 50 km. The resolution measured in test is better than 0.5 m. Repetition rates are selectable among 1/8 Hz, 1 Hz, 2 Hz, and 8 Hz (for a 2-second burst). An internal fiber optic calibration path is used for inflight calibration. The laser rangefinder is boresighted with the MSI.

The instruments are controlled by four Data Processing Units (DPU) based around the RTX 2010RH microprocessor programmed in Forth. Three of these DPU's have a common processor board, 1553 bus interface, analog housekeeping interface, power supply, and chassis design. Basic timing is provided by a 1 Hz interrupt generated by the C&DH subsystem 1 Hz clock. The common elements are augmented by sensor-specific interface boards. The common DPU's run a core of common software. This software provides the boot PROM functions, 1553 interface, analog housekeeping hardware interface, realtime tasking environment, and timer and time management functions. A telemetry utility accepts housekeeping and science telemetry records, packetizes them as required, and manages the handshaking with the C&DH subsystem over the 1553 bus for telemetry collection. A command utility pulls command messages off the 1553 bus and performs checking (checksums, number of words, etc.). The common software implements some commands that are common among the instruments, such as memory uploads, command macro management, telemetry configuration. Instrument-specific commands are passed to the instrument-specific command processor. Other instrument-specific software uses the scheduling facilities of the common software to accomplish its realtime and background tasks. One of these common DPU's controls the MSI, the NIS and Magnetometer share a common DPU, and the x-ray and gamma-ray spectrometers share the third.

The NLR DPU has a smaller and different memory configuration from the common DPU's, and doesn't use the C&DH 1 Hz interrupt for timing. The common core software was modified for the NLR DPU memory configuration, and certain features—such as command macro capability and multi-tasking—were deleted to fit the smaller memory size. The NLR software uses a time-slice

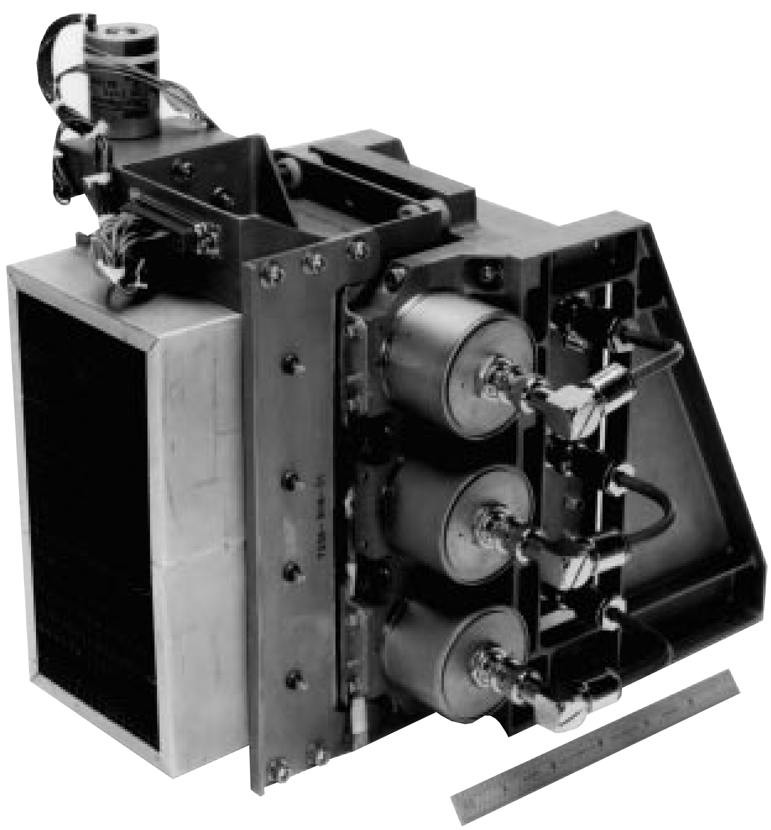

Fig. 12. X-ray Spectrometer

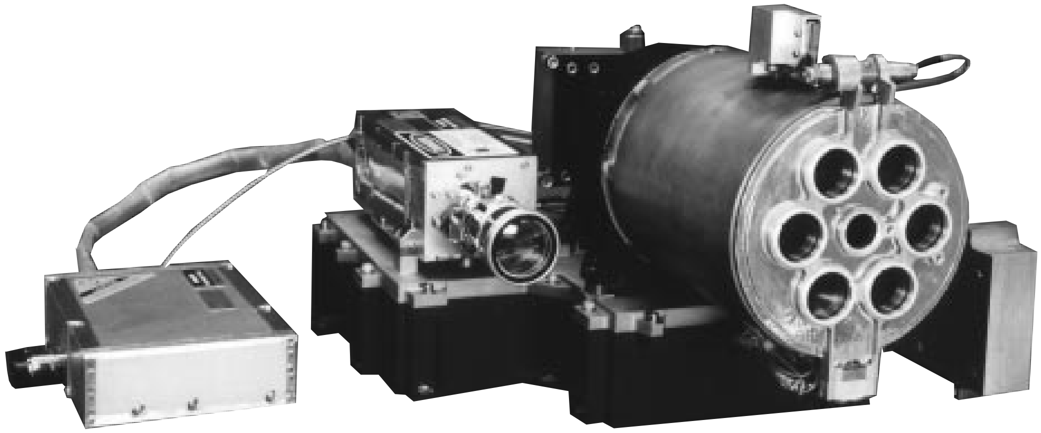

Fig. 13. NEAR Laser Rangefinder

design based on a single interrupt generated by an internal timer at 8 Hz. Synchronization with the C&DH subsystem is maintained by using 1553 bus sync messages to adjust the NLR DPU timer once each second.

SAFING DESCRIPTION

When operational, all spacecraft subsystems are under ground control. Even when out of ground contact, the spacecraft reacts to commands stored in processor memories and activated at a pre-programmed Mission Elapsed Time (MET). Due to the long periods of time out of ground contact, and because of the speed-of-light limitations on realtime telemetry and commanding, the spacecraft must be capable of autonomous action to respond to detected faults. The goal of the safing design is to detect health-threatening faults, and to keep the spacecraft safe until the next ground contact. Fault diagnosis and correction (beyond that needed to keep the spacecraft safe) is a Mission Operations function.

All processors are capable of taking autonomous action. The C&DH is responsible for performing checks on subsystem housekeeping telemetry, and initiating autonomous action to correct anomalous situations. It monitors and maintains the aspects of the G&C subsystem configuration that are under direct C&DH subsystem control. It also monitors the status of the G&C software, and takes autonomous action to correct abnormal operation. The G&C subsystem is responsible for performing checks for the correct operation of its own sensors, actuators, and the Flight Computer software. The G&C subsystem will ignore bad sensor data, and maintain attitude control in the best possible mode consistent with the available good sensor data. The G&C will also monitor for unsafe spacecraft states, and take corrective action. The instrument processors can take advantage of the C&DH housekeeping checks, and/or perform internal checks on their own operation.

The critical aspects of spacecraft safety are power and communications. Because the spacecraft can operate on battery power only for a short period (less than 2 hours), keeping the fixed solar panels pointed towards the sun and the spacecraft load below the solar panel output is the first priority of the safing design. Because only the directional antennas (HGA and medium gain) can be used to communicate with Earth throughout the mission, pointing the medium gain antenna at the Earth is the second priority of the safing design.

Figure 14 is the operating mode state transition diagram. Correctable faults are handled in Operational Mode. For example, a self-detected instrument fault, passed to the C&DH through an error code in housekeeping data, is such a correctable fault; the error code causes the C&DH to turn off the affected instrument, but otherwise does not affect spacecraft operations. Any serious fault affecting a critical spacecraft subsystem results in remedial action (such as turning off a subsystem and bringing a backup on-line) and entry into one of the two safe modes.

In Earth Safe Mode, the spacecraft is placed in a power safe configuration with an emergency mode downlink. The solar panels are pointed directly at the Sun, and the medium gain antenna is placed on the Earth. Either the C&DH or the G&C is capable of initiating Earth Safe Mode in response to detected conditions. Any power system component anomaly except low bus voltage (LVS) results in Earth Safe mode. A C&DH or G&C subsystem processor reset (except for the Last Resort Timer C&DH reset) result in Earth Safe Mode. A violation of the Sun Keep-in one or an autonomous momentum dump also triggers Earth Safe Mode. The underlying assumption in Earth Safe Mode is that the G&C subsystem retains knowledge of the location of the Earth, and that the telecommunications subsystem is operational.

Knowledge of the mission geometry allows a simple algorithm for autonomous re-acquisition of Earth in the event of an attitude anomaly. In Sun Safe Mode, the spacecraft is placed in a power safe configuration with an unmodulated carrier downlink. The solar panels are pointed directly at the Sun, and the spacecraft is rotated to sweep fan beam antenna through the hemisphere facing the Sun and Earth. Every 10 hours, redundant components of the telecommunications subsystem are exercised. Either the C&DH or the G&C are capable of initiating Sun Safe Mode in response to detected conditions. The G&C subsystem initiates Sun Safe Mode if it loses knowledge of the location of the Earth (some sensor failures/loss of ephemeris or UT). The C&DH subsystem sets a timer each time it receives a good command transfer frame. If the time between good command messages exceeds a commandable limit (default: 7 days), the C&DH subsystem initiates Sun Safe Mode.

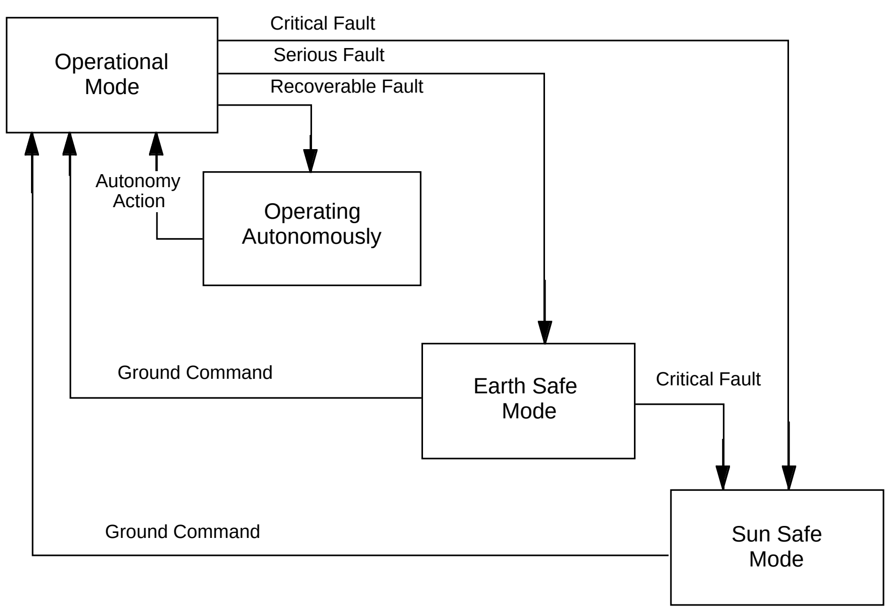

Fig. 14. Mode Transition State Diagram

If the spacecraft does not communicate at a scheduled ground contact, the ground station waits to observe the unmodulated carrier sweep past. This may not occur for some time, especially if an telecommunications subsystem component failure or LVS caused entry into Sun Safe Mode. Once the carrier is observed, the time that the Earth will again be in the medium gain antenna pattern can be calculated from the Sun Safe rotation rate. A command to stop rotation is sent, timed to intercept the spacecraft as the antenna pattern next sweeps the Earth. After confirmation (via the carrier signal) that the spacecraft has stopped rotation, the process of commanding diagnostic telemetry and recovering from Sun Safe can begin.

For redundancy and reliability, discrete lines are available to provide limited communications between the G&C subsystem and the C&DH subsystem without the use of the 1553 bus. All the discrete lines are fully cross-strapped among the redundant processors. Safe modes are coordinated between the G&C and C&DH subsystems using four of these discretes. A 2-bit code indicates the current subsystem mode: Operational, Earth Safe, Sun Safe/Rotate, or Sun Safe/Hold. If either the C&DH or the G&C subsystem enters a safe mode, the other will enter the same mode.

LVS is a special case of the Sun Safe Mode. Three levels of low bus voltage are detected: 26 V, 25 V, and 23 V. The 26 V level can be disabled and the 25 V level enabled to allow for component aging. Both of these levels trigger the same safing algorithm. Once the bus voltage reaches the 25 to 26 V range, more severe reductions in power use beyond those taken in Sun Safe mode are needed before recovery can begin. All systems that are not immediately critical (for example, spacecraft heaters and the solid state power amplifiers) are turned off. Current limit checks on most powered subsystems are enabled, and any over-current system is turned off. After the bus voltage recovers to at least 28 V, heaters are turned back on. The solid state power amplifiers are not turned on for 24 hours after LVS, to allow sufficient time for the battery to recharge. At that time, the normal, Sun Safe Mode unmodulated carrier transmission will begin.

If the bus voltage does not recover and continues to fall to the 23 V level, a last ditch effort to save the spacecraft is initiated. The two Flight Computers and the Star Tracker are turned off, both to save more power and to force the G&C subsystem into its most basic safing state using only the AIU, IMU, and DSAD’s. The IMU and the backup AIU are powered cycled to force reinitialization, and the backup AIU is given control of the actuators. Many of the other, normal autonomy actions on the spacecraft are disabled, on the chance that some error in those algorithms is causing the spacecraft problem. Similar actions are taken if the Last Resort Timer resets the C&DH processor. This 12.1 day timer is implemented in hardware and can only be reset by a special, coded command from the ground. If this timer expires, then all previous autonomous efforts to restore communications with Earth have failed, and drastic measures are required.

The proper operation of the C&DH and G&C subsystems is critical to the safety of the spacecraft. The C&DH subsystem is responsible for maintaining a minimal complement of G&C subsystem components (for example, at least three wheels). It also uses two of the discrete lines described above to monitor the operation of the AIU software. One of these lines, called ‘Active’ indicates that the AIU has control of the G&C actuators and that the spacecraft responds to its control outputs. The other discrete line is called ‘Heartbeat’ and is toggled every other second when certain internal checks on the AIU software operation are passed. This Heartbeat signal is toggled in a different routine, and in response to different criteria, than the hardware watchdog timer. Together, the Heartbeat and watchdog timer cover a wider class of errors than each would separately. If either the

Active or the Heartbeat discrete indicate a problem with the AIU software, the C&DH subsystem will bring the backup AIU online. As long as the C&DH subsystem can find and bring online a 'good' AIU (that is, with good discrete indications), it will continue to monitor and switch systems as necessary. If the C&DH cannot get a good indication from either AIU, it will power both and stop checking.

The last discrete line is used to coordinate an autonomous momentum dump, should one ever be required. The AIU requests use of the thrusters via discrete. The C&DH will allow the G&C subsystem control of the thrusters for up to five minutes, but then disables them regardless of the state of the discrete. An attempt to use the thrusters autonomously for longer than five minutes will also cause the backup AIU to be brought online.

The C&DH processor itself is completely redundant. Both redundant processors are continuously powered, and either can be commanded from the ground and carry out critical safing actions. There are two differences between the 'primary' and the 'backup' C&DH subsystem: only one processor can control the 1553 bus, and only one can use the downlink. Only the 'primary' (bus controller) processor can use 1553 data for safing decisions or send 1553 bus commands. Because critical safing is carried by discretes, this does not limit the ability of the backup to initiate and respond to safe modes. Designation of the 'primary' and 'backup' C&DH is under ground control, and is never switched autonomously. The only link between the processors is an exchange of mode over the 1553 bus, so that safe modes can be coordinated even with a discrete line failure from an AIU.

ACKNOWLEDGEMENTS

The research described in this paper was carried out under contract with NASA. Special acknowledgements are given to the following Johns Hopkins University/Applied Physics Laboratory personnel who contributed: Theodore Hartka/David Persons–Mechanical, Carl Ercol–Thermal, Larry Mosher–Propulsion, Jason Jenkins-–Power, David Stott–Command and Data Handling, J. C. Ray–Guidance and Control, Robert Bokulic–RF.

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