

FLIGHT SYSTEM TESTBED VERIFICATION OF THE MARS PATHFINDER ATTITUDE CONTROL SYSTEM

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Pathfinder is the first in a series of Mars exploration projects that are fundamental elements of the Discovery Program missions funded by the National Aeronautics and Space Administration (NASA). These projects differ significantly from recent ones in that the development life cycles are limited to three years and are cost capped at \$150 million (in Fiscal Year 1992 dollars). Pathfinder's primary objectives are to demonstrate a low cost lander delivery system to the Martian surface, and to deploy and operate four science instruments and experiments, including a Microver. Launch is currently scheduled for 021 December 1996 and arrival on 04 July 1997.

To validate proper spacecraft function and performance prior to actual launch and in-flight operations, the spacecraft must be subjected to a battery of tests at the assembly, subsystem, and system levels. One facility where much of this testing is performed is the Flight System Testbed for Pathfinder (FST/P). The FST/P is comprised of flight hardware test stations, flight software test stations, and a main integration and test station. It is here that approximately 80% of all spacecraft electrical interfaces were verified and 85% of the defined functional tests to date have been dry-run prior to formal system level integration and test.

Considerable effort was undertaken to provide the necessary environment for validation of the Pathfinder Attitude Control System (ACS) at each level. Ground support equipment hardware controllers and load simulators provide the mechanisms for verifying proper operations of the flight sensors and actuator (thruster) driver interface unit. Verifying function and performance of spacecraft attitude determination and control algorithms and flight software code is performed through the use of ground support equipment real-time software dynamics algorithms and hardware models. Integration of all these components in the FST/P and execution of numerous defined test cases provide for complete validation of this subset of essential spacecraft capabilities required for a successful formal system level integration and test program, and eventually, a safe arrival at Mars.

Although performance testing still lies ahead, the FST/P has already proved to be a major benefit in the development and validation of a significant number of spacecraft functionalities, such as the ACS. Early interface verification and functional test execution help to insure that formal system level integration and test milestones are met on time and within budget, therefore, making the Pathfinder experience a positive one.

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INTRODUCTION

Pathfinder is the first in a series of Mars exploration projects that are fundamental elements of the Discovery Program missions funded by the National Aeronautics and Space Administration (NASA). These projects differ significantly from recent ones in that the development life cycles are limited to three years and are cost capped at \$150 million (in Fiscal Year 1992 dollars). Pathfinder's primary objectives are to demonstrate a low cost lander delivery system to the Martian surface, and to deploy and operate four science instruments and experiments, including a Micro-Over.

The Pathfinder Mission

Launch of the Pathfinder spacecraft is currently scheduled for 02 December 1996. Injection will result in an Earth-Mars type 1 (ballistic) transfer trajectory with a nominal transit time of approximately seven months. Four primary statistical trajectory correction maneuvers are planned during cruise. Arrival is scheduled for 04 July 1997, at which time the Cruise Stage is jettisoned from the remainder of the spacecraft. Following Mars atmospheric entry and early descent, the parachute is deployed. During terminal descent, the Heat Shield is released, bridle deployed, airbags inflated, and Backshell mounted rocket assist deceleration motors ignited prior to surface impact. Following Lander roll stop, the airbags are deflated and retracted, the side petals opened, and the Lander and Micro-Over configured for operations. Key events then include the return of critical entry, descent, and landing engineering data; return of imaging, meteorology, and spectroscopy science data; and operation of the Micro-Over to conduct science and technology experiments.

The Pathfinder Spacecraft

The launch system selected for the Pathfinder spacecraft is the two-stage McDonnell Douglas Delta 117925 launch vehicle. The third, or upper stage, is a PAM-D (Payload Assist Module) booster which will spin stabilize the spacecraft while providing the final velocity required to place it on a trajectory to Mars. Lift-off will occur at Launch Complex 17A, Eastern Test Range, Cape Canaveral Air Station.

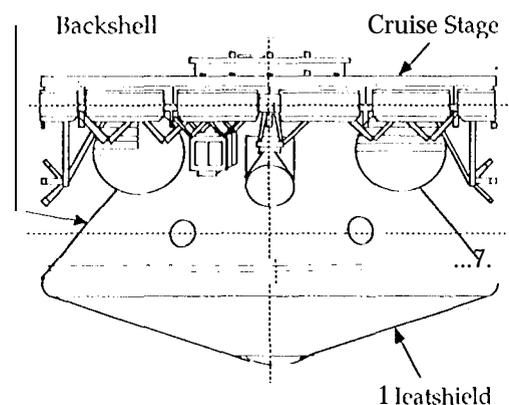


Figure 1 Pathfinder Spacecraft in the Nominal Cruise Configuration

The Pathfinder spacecraft is comprised of five major components: Cruise Stage, Backshell, Heat Shield, Lander, and Micro-Over. Figure 1 shows the spacecraft in the nominal cruise configuration, with only the first three elements being visible.

The Cruise Stage contains the necessary equipment essential for interplanetary flight, providing for spacecraft power, a telecommunications link with ground operators, attitude determination functions, translational maneuver capability, and precession and spin rate control. The top portion of the Cruise Stage is almost entirely covered with Gallium Arsenide solar cells, allowing solar energy to be harnessed and provide power to the spacecraft subsystems. Telecommunications link is provided via an X-band medium gain antenna. Sensors used for

attitude determination are the dual redundant Ball Star Scanner Assembly and the Adcole Digital Sun Sensor Assembly. The latter is comprised of an electronics unit and five sensor heads mounted in such an orientation that 4π steradian coverage is nearly achieved. The propulsion subsystem is comprised of four non-regulated (blow-down) hydrazine propellant tanks connected to a total of eight 4.45N (1lbf) thrusters. These thrusters are arranged into two clusters, and provide the mechanisms for performing translation maneuvers, and precession and spin rate control.

The Backshell, covered with a SLA-561 V ablative material, will help to protect the Lander from aerodynamic heating recirculation flow as it enters the Martian atmosphere. Mounted onto the Backshell is a parachute canister and three solid-fuel rocket assist deceleration motors. There is also a Low Gain Antenna (LGA) which will be used for telecommunications during entry and early parachute descent.

The Heat Shield, also covered with the same SLA-561 V ablative material, will provide the primary Lander protection under direct, extreme aerodynamic heating resulting from atmospheric entry.

An exploded view of the Pathfinder spacecraft is shown in Figure 2, exposing the Lander component. The purpose of the Lander component is to provide support to the Microver and three science instruments during Martian surface operations. A simple tetrahedron design has been incorporated to limit the possible landing orientations. Airbags, mounted to each of the petals, will be used to minimize landing shock. Actuators will be used to retract the airbags once the Lander has rolled to a complete stop. The Lander can right itself on the base petal if it happens to land on any of the three side petals by driving additional actuators connected to these petals. Once open, Gallium Arsenide solar cells mounted on the inside of each petal will provide spacecraft power for surface operations. All thermally sensitive electronics will be enclosed in an insulated Integrated Support Assembly (ISA) located within the tetrahedron. This enclosure will provide a controlled environment to minimize the effects of extreme temperature variations on the Martian surface. The Lander High Gain Antenna (HGA), LGA, and the imaging camera will be mounted on top of the ISA. The Microver will be mounted to one of the three side petals. An illustration of the Lander with all three side petals open is shown in Figure 3.

The Microver Flight Experiment (MFEX) is a NASA sponsored experiment of which the primary objective is to determine autonomous mobile vehicle performance in the poorly understood Martian terrain. The Microver is a six-wheel rocker bogie design vehicle.

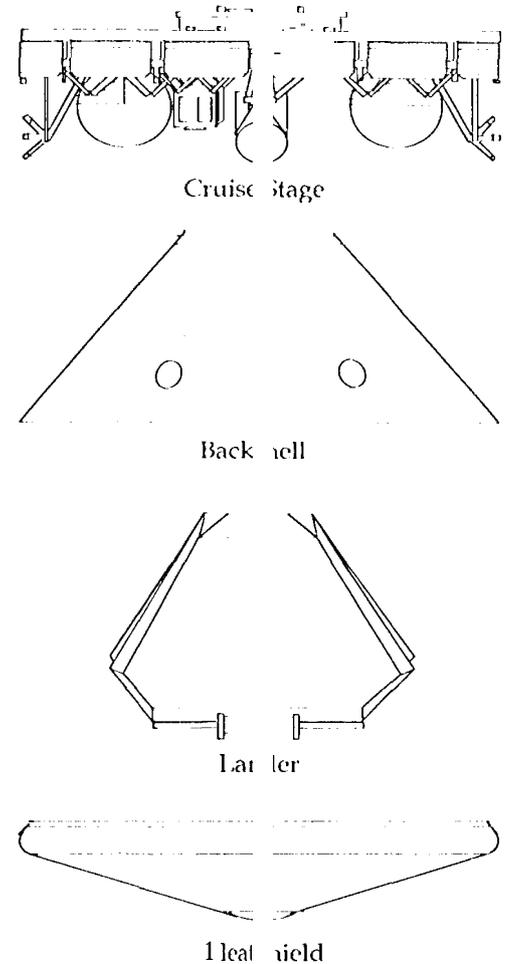
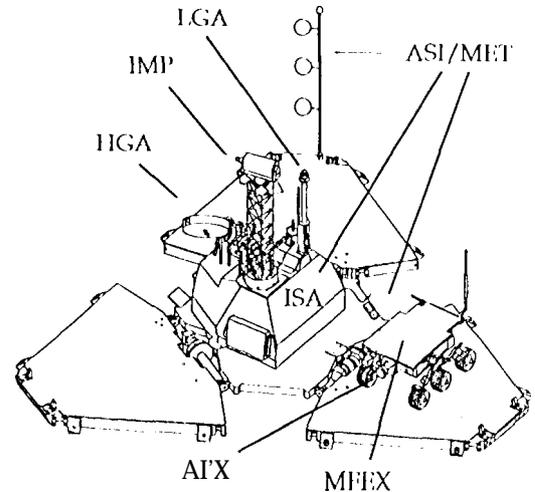


Figure 2 Exploded View of the Pathfinder Spacecraft

Each of the six wheels is independently actuated and geared to provide the performance necessary to operate in soft sand. The front and rear wheels are independently steerable, providing the capability to turn in place. Maximum vehicle speed is 0.4 meters per minute. Similar to the Cruise Stage and Lander, power for the Microover will nominally be supplied by Gallium Arsenide solar cells mounted on top of the vehicle. All thermally sensitive electronics are contained in an insulated Warm Electronics Box (WEB). Control is provided by an 80C85 computer which performs I/O operations to approximately seventy sensor channels and services cm-board devices such as the cameras, modem, motors, and experiments.



The three Pathfinder spacecraft science instruments consists of the Imager for Mars Pathfinder (IMP), the Atmospheric Structure Instrument / Meteorology Package, and the Alpha-Proton X-ray Spectrometer. The Imager for Mars Pathfinder is a color stereoscopic imaging system with both azimuth and elevation control. The CCD camera system itself is mounted on top of a deployable 1.0 meter mast. The Atmospheric Structure Instrument / Meteorology Package will allow for reconstruction of acceleration, atmospheric density, temperature, and pressure profiles during entry and descent. Temperature and pressure transducers which will provide some of this data will also be used to measure variations for post-landed conditions. Finally, the Alpha-Proton X-ray Spectrometer is an elemental composition instrument which will be used to determine the chemistry of surface materials.

Figure 3 Lander with Side Petals Open

ATTITUDE AND INFORMATION MANAGEMENT SUBSYSTEM

The Attitude and Information Management (AIM) subsystem represents an integration of traditionally separate attitude determination and control and command and data handling functions onto a single processor platform, and within a single subsystem. For the Jet Propulsion Laboratory (JPL), this represents a significant departure from past in-house planetary projects, and exemplifies the institution's commitment to building smaller and more cost-effective spacecraft on shorter development life cycles. The AIM provides for three-axis inertial attitude determination and control of the spacecraft following separation from the Delta II/PAM-D launch system through the Mars entry turn; determines the peak deceleration level during entry, descent, and landing; determines the "tilt" of the Lander shelf with respect to the local gravity vector following landing; and allows for HGA articulation control during surface operations. In addition, the AIM receives demodulated input signals from and outputs encoded data transfer frames to the telecommunications subsystem, performs on-board computations, executed both real-time and stored sequence commanding, provides precision clock information, and maintains the remote interface with the Microover. The AIM components and interfaces are illustrated in Figure 4.

Subsystem Description

All control functions are provided through the Mars Pathfinder Flight Computer (MFC), a single on-board 32-bit RAD6000SC (single chip) processor possessing 128 Megabytes of random access memory and operating on a standard VMEbus. Nominal flight software execution is from this memory. Selectable computer speeds range from 2.5 to 20 Mhz. Non-volatile memory storage for the AIM flight software is provided by the PROM (Programmable Read Only Memory) board. Single bit error correction is provided by the PROM, and in addition, the PROM board actually consists of electrically erasable PROM, allowing for required updates during in-flight operations.

The "uplink" board consists of the Hardware Command Decoder (HCD), Critical Relay Controller (CRC), and Bus Controller (BC) functions. The HCD is the uplink data entry point to the AIM from the telecommunications subsystem. The CRC portion can be further broken down to five sub-functions: inputs select logic, sequence start detector and polarity ambiguity resolver, error detection and correction, command buffer memory, and critical command logic. The BC controls the modified 1553 bus (non-transformer coupled for reduced power) to which the Remote Engineering Units (REUs) are interfaced. The "downlink" board consists of the Reed-Solomon Downlink (RSD) and Timing Unit (TU) functions. The RSD is the downlink data exit point from the AIM to the telecommunications subsystem. It appends a 32-bit synchronization pattern to the front of the 8800-bit transfer frame and a 1280-bit Reed-Solomon pattern to the end of the transfer frame. The TU provides basic timing functions for the AIM.

Lander

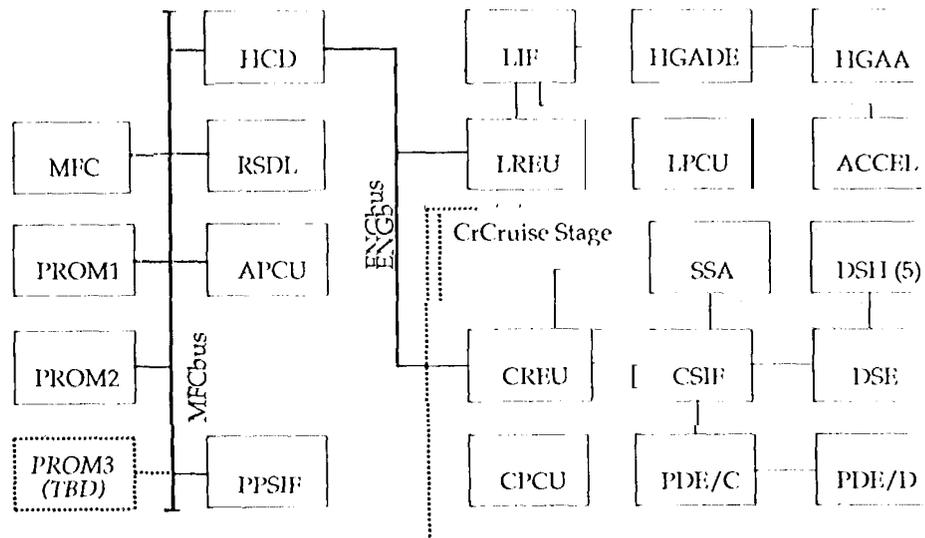


Figure 4 Block Diagram - Attitude and information Management Subsystem

The AIM Power Converter Unit (APCU) accepts input from the 28VDC power bus and provides conditioned power for all assemblies on the VMEbus (-5v) and provides additional voltage levels for the MFC (+3.3v), the HCD (-12v), and the VMEbus (+3v). The Lander PCU

(LPCU) accepts input from the 28VDC power bus and provides conditioned power for all non-VMEbus assemblies in the Lander Module: LREU (-15v, ±12v), LIF (-15v, ±12v), and the Accelerometer (+5v, ±15v). Finally, the Cruise Stage PCU (CPCU) accepts input from the 28VDC power bus and provides conditioned power for all assemblies in the Cruise Stage Module: CREU (+5v, ±12v), CSIF (+5v, ±12v), and the Propulsion Drive Electronics (+5v, ±12v).

The Power and Pyro Switching Interface (PPSIF) board accepts commands from the MIC and provides the relay drive signals to the power distribution unit, shunt regulator, and pyro switching unit of the PPS subsystem. Power distribution is inhibited via a triple relay to the main battery. Pyro switching is inhibited via a double series relay. The first latching relay allows for the system to be armed, and it can only be armed from the ground. The second latching relay allows for the system to be enabled, and it can only be enabled following spacecraft separation from the TAM-D booster. This configuration prevents inadvertent single-fault induced pym events from occurring during launch pad operations.

The Remote Engineering Units (REUs) accept temperature data, analog signals, and digital data from various locations on the spacecraft. The REUs digitize temperature and analog data and provides this information on request via the modified 1553 bus. All digital input and output signals pass through the appropriate interface unit [Lander Interface Unit (LIF) or Cruise Stage Interface Unit (CSIF)], and discrete status signals from the Heat Shield will be passed directly to the REU.

The all IUS (Inertial Upper Stage) CS-203 Star Scanner Assembly (SSA) [used before on the Magellan spacecraft and IUS booster rockets] is the source for acquisition of spacecraft stellar reference. Modifications to the unit have been made to accommodate the higher Pathfinder spacecraft spin rate of 1.9 rpm. The "V" slit type scanner will detect star pulses and provide magnitude and spacecraft relative elevation information. The data will then be compared to an on-board full-sky star catalog. From this and data obtained from the Sun Sensor, three-axis spacecraft attitude determination can be performed. The Star Scanner electronics is the only AIM component that is dual string redundant (channels A and B). The Adcole 27530 Digital Sun Sensor (DSA) previously flown on the Mars Observer spacecraft is the sun sensor of choice for the Mars Pathfinder spacecraft. The design of the DSA has been modified to accommodate the expected power bus voltage variation (27 - 36VDC). Recall that the DSA is comprised of five detector heads and a single electronics unit. The five heads will allow for full-sky coverage, except for a 20 degree full cone obscuration due to the Heat Shield and Back Shell.

The Propulsion Drive Electronics (PDE) provides control and drive functions for the propulsion subsystem devices. Commands from the MIC pass through the CSIF and CREU to the PDE control unit (PDE/C) and the driver interface (1111/1~) to drive the thrusters, drive the isolation latch valves, drive the catalyst bed heaters, and collect latch valve status. The general design was derived from the Cassini spacecraft's Valve Drive Electronics (VDE).

A dual-axis actuator package (gimbal) provides for elevation and azimuth control of the HGA. They also represent the only articulating component of the AIM. A position loop for each degree of rotational freedom is provided through readout of the motor commutation signal and calculation of the on/off duration. A pointing accuracy of 0.2 degree per axis is required. The maximum slew rate for each axis is 0.3 degree/second.

The Accelerometer (ACCEL) actually consists of two three-axis sensor packages and a set of electronics. The first sensor package will be used by AIM during entry, descent, and landing to detect the peak deceleration level and to determine the "tilt" of the lander shelf during surface

operations. The nominal operating range for this package is 0-40 g. The second sensor package will be used by Science to measure the atmospheric density profile and viability. The nominal operating range for this package is 0-0.1 g.¹

General AIM Test Program

The main objective of the AIM test program is to subject the subsystem to a battery of tests at the assembly and subsystem levels prior to formal system level integration and test.

Flight Hardware. Due to a considerable amount of Cassini Project design inheritance, the breadboard development stage was bypassed altogether for many of the AIM electronics boards such that initial fabrication and assembly was performed at the engineering model level. Once design validation was complete, fabrication and assembly of the actual flight units, and some flight spare units, was authorized to commence. To verify functionality, test specifications and associated test software were developed for each board and later utilized as part of the approved validation procedure. Additional testing, such as voltage margin, temperature margin, and vibration, would obviously be required. However, for efficiency these tests would not be performed at the assembly level, but rather subsystem level integration and test.

Flight Software. An incremental development approach was followed such that preliminary versions of flight software would be generated and tested before the actual flight version was officially delivered. For each version, individual objects or modules would be first be unit tested. Following a successful round of testing, these objects would then be integrated and then tested in this configuration. Once the flight software met all success criteria at this level, it was made available for subsystem integration and test.

Subsystem Integration and Test. Once an electronic board or set of boards had been functionally validated, it was made available for electrical integration and test. These set of tests were designed to ensure proper electrical connectivity between boards. Following successful electrical integration, flight software was then combined with the flight hardware. This extensive set of tests validated the functionality of the subsystem. At this point, environmental testing could be scheduled. The engineering model of the AIM subsystem was subjected to sine and random vibration loads to simulate launch, atmospheric entry, and landing conditions. It was also subjected to voltage and temperature margin tests using adjustable laboratory power supplies and a thermal chamber. Finally, electro-mechanical compatibility tests were conducted to ensure that actual conducted emissions and susceptibility characteristics were within defined specifications. The flight model of the AIM subsystem was also subjected to functional, voltage margin, and temperature margin tests. Electro-mechanical compatibility testing for radiated (as opposed to conducted) emissions and susceptibility and sine and random vibration tests would be deferred to formal system level integration and test for efficiency.

External AIM Interface Verification. Rather than delaying the validation of all external AIM interfaces, such as the Radar Altimeter, Airbag Retraction Actuators, Petal Actuators, etc... , until formal system level integration and test, a decision was made to perform as many early electrical and functional integrations as possible to mitigate schedule risk "downstream". A major subset of these type of tests are the End-to-End Information System (E:IS) demonstrations. Since project start, a number of demonstrations have been performed. In each successive test the system, consisting of the ground data system, ground support equipment, the AIM, and the Microrover became more mature. Software simulations were replaced with engineering model hardware, which in turn was replaced with actual flight hardware and ground components that would be used for in-flight operations.

System Integration and Test. At the System level, a superset of mechanical and electrical/functional activities must be performed in preparation for launch. The first phase of formal system level integration and test was validation of all electrical interfaces. Following successful completion of these tests, the Lander was subjected to random vibration and centrifuge tests to simulate the launch and landing environments, validating the workmanship of each flight article. After the post-test inspection had identified no anomalous conditions, the flight system was then subjected to a Suite of tests to confirm proper functionality. Following mechanical spacecraft 'stacking', the flight system would be subjected to another round of functional testing. After which time the formal system level environmental test program would commence. These tests will include acoustics, spin balance, solar thermal vacuum in both cruise and landed configurations. Finally, a pre and post ship test would be performed at the Jet Propulsion Laboratory and then at the Kennedy Space Center.

Flight System Testbed for Pathfinder

A number of different laboratory facilities have been utilized in support of AIM subsystem verification. These have included the shared Cassini-Mars Pathfinder Project avionics electronics laboratory, the inertial sensors laboratory, and even the Spacecraft Assembly Facility. However, none has been more utilized to date than the Flight System Testbed for Pathfinder (FST/P). The facility itself is an approximately 1150 square foot laboratory located in close proximity to most of the AIM subsystem development personnel. It is a certified class 100,000 ppm (parts per million) cleanroom where both temperature and humidity levels are monitored. Mechanisms are in place to ensure proper grounding of personnel and equipment and general cleanliness of the facility. The facility itself is divided into four separate work areas: hardware inspection and rework, flight hardware test stations, flight software test stations, and the main integration and test station. A class 100 ppm laminar flow bench and microscope are available for inspection and/or minor rework of electronic boards. Individual flight hardware test stations were constructed for standalone in assembly level test. A flight software station is available for developers to integrate and test the software. Finally, the main integration and test station is where each of the major AIM components is brought together. It is here that AIM subsystem integration and test and AIM external interface verification activities have and will continue to take place.

Although this particular facility is managed by the Mars Pathfinder Project, the JPL institution's FST organization was instrumental in its development. Electrical ground support equipment hardware, engineering, and technician support services were provided. In addition, the multi-mission spacecraft dynamics simulation software and engineering support for project specific adaptation were also provided. This assistance has benefited the Mars Pathfinder Project in both cost and schedule.

FST/P Highlights

There is a long list of completed tasks performed in the FST/P, tasks which assisted in mitigating schedule risk during formal system level integration and test. Most notable are the electrical integration and functional verification of most external AIM interfaces and the "dry run" of most system functional test procedures for System Functional Test No. 1 and 2. Of the ten other subsystem with which the AIM interfaces, six were partially or completely validated in the FST/P. These were:

- Mechanical Systems
- Entry, Descent, and Landing
- Radar Altimeter
- Atmospheric Structures Instrument/Meteorology
- Lander Mounted Microrover Equipment
- Imager for Mars Pathfinder

Three others were partially or completely simulated with electrical ground support equipment. These were:

- Telecommunications
- Power and Pyro Switching
- Propulsion

The only subsystem not integrated or at least simulated was the Aeroshell (Heat Shield and Back Shell) Instrumentation Package. A close look indicates that approximately 80% of the external AIM interfaces were validated prior to formal system level integration and test.

System Functional Test No. 1 and 2 were comprised of actually nine separate test procedures. Of these, eight were exercised, or dry run, in the FST/P prior to execution in as part of the formal system level integration and test program. These were:

- Microover Interface Verification
- AIM/Power Control
- Entry, Descent, and Landing 1
- Attitude Determination and Control
- Fault Protection
- JGA Actuator Control
- Launch and Cruise Sequences
- Entry, Descent, and Landing 11

The only test procedure not explicitly exercised in the FST/P was the Data Flow test, since most of the functionality had already been validated in early testing. With this as the only exception, approximately 85% of the test procedures were validated prior to System Integration and Test.

ATTITUDE CONTROL 'SYSTEM'

The Attitude Control Subsystem (ACS) for the purposes of this discussion pertain to those spacecraft capabilities required for an Earth-Mars type 1 transfer (ballistic) trajectory only. Attitude and articulation control functions required for Mars atmospheric entry, descent, and landing and eventually surface operations will not be addressed. The primary spacecraft hardware components are the DSA, SSA, IMU, and Thrusters. Specifics on each of these units were discussed in previous sections,

The primary flight software objects are the:

- ACS Mode Commander
- Inertial Vector Propagation Function
- Sun Function
- Star Identification Function
- Mass Properties Estimator Function
- Thruster Function
- Attitude Estimator Function
- Attitude Control Function
- Delta V Controller Function

ACS Mode Commander

The ACS Mode Commander essentially manages the Attitude Control System by determining which of the flight software objects are run during any given 125 ms RTI (Real Time Interrupt) time slice. A simplified diagram of the ACS Mode Commander is shown in Figure 5. Within SUNINIT, Sun Sensor data is used for spacecraft attitude determination with respect to sunline. Open-loop thruster control is possible while in SUNINIT. Following successful transition to SUNLINE, the spacecraft can then be commanded to a number of spin rate, precession, and translational control activities that do not require stellar reference for proper execution. Minimum requirements call for the spacecraft to be able to execute all functions

Earth-Mars type 1 transfer trajectory from within SUNLINE. However, the spacecraft can be commanded to CELESTIAL for operational efficiency. Sun Sensor and Star Scanner is used for three-axis spacecraft attitude determination. Once a successful transition to CELESTIAL is made, the spacecraft can be commanded through a full suite of spin rate, precession, and translational control activities.

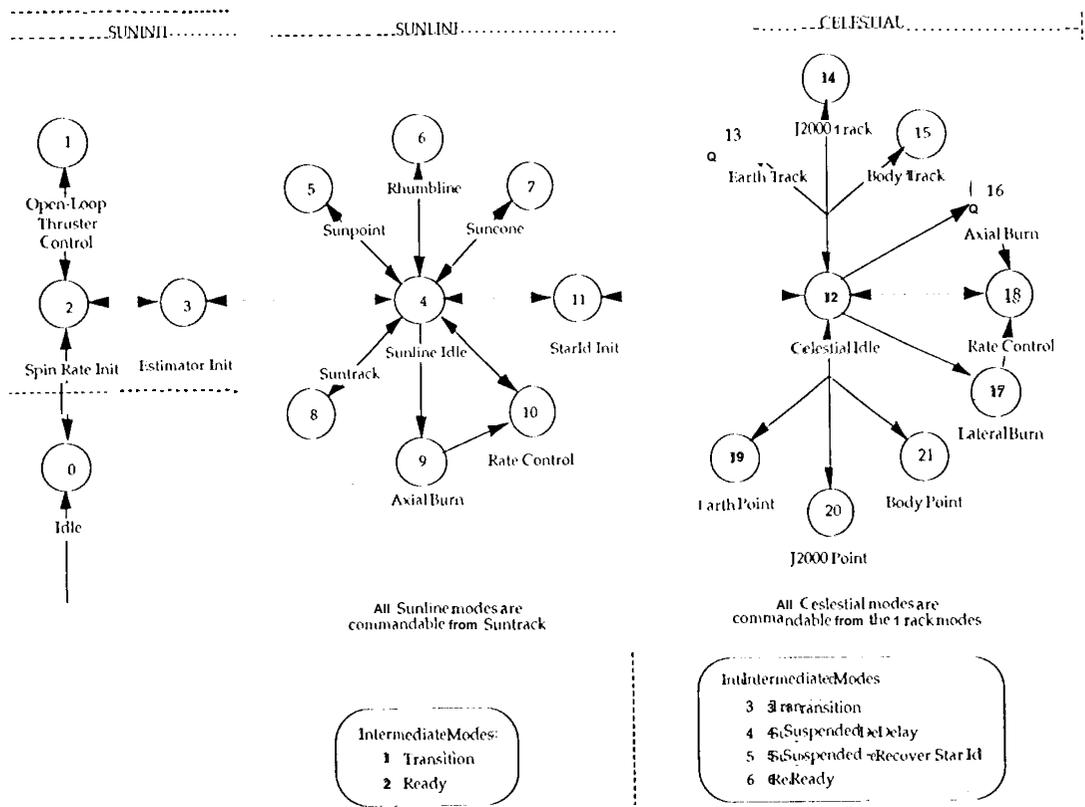


Figure 5 Attitude Control System Mode Commander Diagram

Inertial Vector Propagation Function

Inertial Vector Propagation provides to other flight software objects spacecraft-relative inertial vectors for celestial targets of interest in the J2000 coordinate frame. The algorithm produces time-varying vectors that can be connected head-to-tail to create a tree configuration where the convention used in it's construction follows the natural architecture of the solar system. Therefore, at times it will be necessary to perform a vector summation to produce a desired resultant vector. Since spacecraft-relative motion of most celestial targets of interest during Earth-Mars cruise can be mathematically expressed in first and second order terms, the motion can be adequately approximated through propagation of conic sections. Higher order polynomial propagation is not necessary. The algorithm consists of one foreground and two background processes. Vector requests from other objects such as the Attitude Estimator and

Attitude Controller functions are honored in the foreground task and actual vector propagation is processed via two separate and distinct background tasks: the "triple evaluation" (initialization) and "routine" processes.

Sun Function

In Sunline Initialization (SUNIN1'I), the Sunfunction is executed to produce the following outputs: (1) calculated sun vector in the spacecraft body coordinate frame, (2) computed residuals and gradients, and (3) coarse spin rate estimation. It should be noted that computed residuals and gradients require that an estimated attitude quaternion first be produced from the Attitude Estimator function. In addition to these outputs, the Sunfunction also provides for a qualitative analysis of the sun data. "Ibis spin rate quality index is a function of the quantity of 'data good' measurements. In order for a spin rate estimation measurement to be considered 'good', the sun presence flag must be set and the computed cone angle must satisfy the angular constraint defined by a flight software parameter resident in the Sunfunction. This index is then used by the ACS Mode Commander to enable the transition to SUNLINE Idle.

Star identification Function

A transition to CELESTIAL idle can only be achieved through the successful completion of the Star identification Algorithm. It is designed to be a robust function which relies on advanced back-end filters rather than highly-sensitive front-end filters whenever possible. This provides more data to be processed that otherwise would have been discarded. To minimize the performance impacts caused by spurious pulses resulting from noise or solar proton events, a mini-batch technique is used to eliminate pulses that do not produce pairs [recall that the Star Scanner is a 'V-Slit' type unit that nominally produces two events for a single star]. The Star Identification Algorithm is divided into two operational modes: (1) Attitude Initialization and (2) Star Tracking. The first part of each mode operates identically in what is termed Pulse Collect and initial Buffering, however, the second and final part of each mode are completely different.

Pulse Collection and Initial Buffering. On each RTI, the Star function returns the number of pulses detected during the last 125ms time slice and for each pulse produces a data record containing the following information:

- time (timetag)
- VisMag (visual magnitude)
- channel_ID (detector A or B)
- rate (three-axis spin rate)
- quat_ext (estimated quaternion)
- varXYZ (uncertainty in quaternion)
- deltaQ_accu (accumulated delta Q)

Each data record must then be compared with the selected visual magnitude threshold value. Only after successfully passing the criterion will the data record be placed in the queue and eventually into the pulse buffer. A mini-batch technique is then used to form valid pulse pairs which successfully pass the time delta, magnitude, and channel criteria while eliminating spurious pulses. Valid pulse pairs are finally placed into an array for use in the second and final part of either the Attitude Initialization or Star Tracking modes.

Attitude Initialization. Data records from the pulse pair array are placed into a pulse pair frame, the size of which corresponds to five spacecraft revolutions (default value). The data records contained within the pulse pair frame contain the following information:

- two timetags
- two visual magnitudes
- two estimated quaternions
- single channel

If the acquired frame flight software flag is set, the Sun vector in both the attitude knowledge acquisition (initial reference) and J2000 coordinate frames is determined via the inertial Vector Propagation function and then made available to the Star identification function. A copy of the on-board star catalogue is also made available to the Star Identification function and each star's angle from the Sun vector in the J2000 coordinate frame is calculated. Magnitude and geometry checks are then performed against the data records in the pulse pair frame. Stars which pass the stated criteria are placed in a temporary candidates catalogue, and then checked for ambiguity. If a single star passes this final check, then a final quaternion is produced. Multiple stars passing the final check require that at least two quaternions match to within a given error tolerance before a final quaternion is produced.

Star Tracking Mode. Before any processing is performed a time check is performed to ensure that the data contained in the pulse pair array is not too old (a user defined parameter). If so, Star Tracking is re-initialized at Pulse Collection and Initial Buffering. Data in the pulse pair array are then buffered for ambiguity checks. Again, a mini-batch technique, this time to eliminate pulses that are used for more than one pair. A copy of the on-board star catalogue is made available to the Star Identification function. Magnitude and geometry checks are performed against the buffered data records. For each pulse produced, a data record containing the following information is produced:

- time
- residual. uc
- gradient
- deltaQ. accu

A maximum number of ten such records is produced for each iteration of the Star Identification function, or per RTI under nominal operations, and provided to the Attitude Estimator function.

Mass Properties Estimator Function

The propulsion system utilizes four hydrazine mono-propellant fuel tanks containing elastomeric modulation devices. It is a blow-down system with Helium as the pressurant. The Mass Properties Estimator function is based on the ideal gas law as applied to the Helium pressurant. Knowledge of the initial fuel fill conditions and rate of mass loss assuming an isothermal (or constant temperature) process allows the diagonal elements of the inertia matrix and center of mass to be determined during cruise phase of the mission.

Thruster Function

Feed pressure, equal to the propellant tank pressure minus the pressure loss in the fuel lines, is assumed to be equally applied to each of the eight thrusters. Thruster Force and Specific Impulse are simply a function of this feed pressure. Mass flow rate data, which is simply the ratio of the former to the latter, is provided to the Mass Properties Estimator function. Thruster force and moment about the spacecraft center of mass are then input to the Attitude Estimator function.

Attitude Estimator Function

The Attitude Estimator function uses a dynamic spacecraft state propagation model driven by feedforward torques from thruster commands and Sun Sensor and Star Scanner measurements to perform the following calculations:

- Spacecraft State Vector
- Angular Momentum Vector
- Inertia Matrix: off-diagonal components

A functions] block diagram is shown in Figure 6. The primary elements consists of the dynamic state propagation model, prefilter, and Extended Kalman Filter.

Dynamic State Propagation Model. Recall that the initial attitude and rate information is provided to the Attitude Estimator function via the Sun and/or Star Identification function. Thruster commands are translated to applied force and moment about the spacecraft center of mass. Based cm simulation results, it has been determined that a 3rd order approximation of the attitude quaternion and a 2nd order approximation of the attitude rate performed every RTI yield sufficiently accurate results.

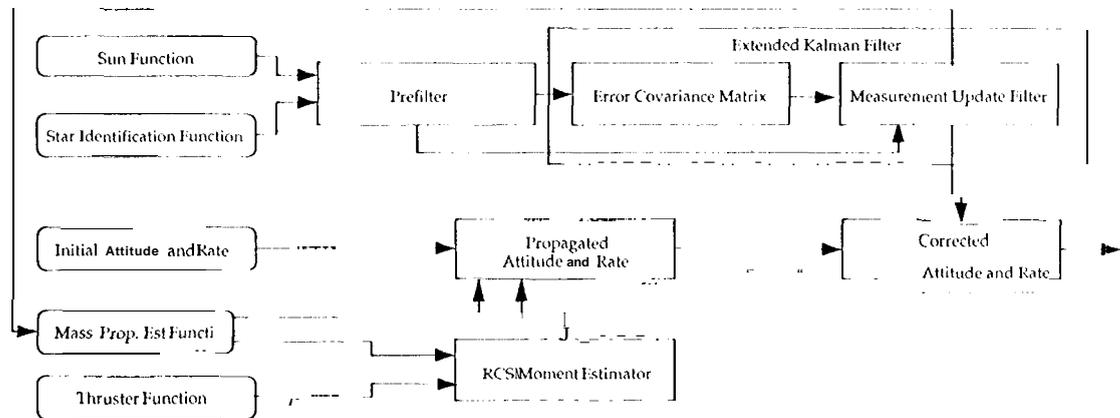


Figure 6 Attitude Estimator Function Block Diagram

Prefilter. The purpose of this filter is to combine Sun Sensor and Star Scanner measurements within a given time interval into a single three-axis attitude pseudo measurement vector with its associated uncertainty.

Extended Kalman Filter. Is composed of two routines, Error Covariance Matrix Propagation and Measurement Update Filter. The Error Covariance Matrix can be propagated through a first order approximation as long as the time interval is small. The Measurement Update Filter generates the best estimate of the error state vector and the Error Covariance Matrix based on the Prefilter information and the propagated Error Covariance Matrix.

Delta V Controller Function

Translational spacecraft motion can be imparted in one or a combination of both maneuver methods: axial delta v and lateral delta v.

Axial Delta V Maneuver. An axial delta v maneuver can be commanded in either the positive or negative spacecraft Z-axis direction through definition of the appropriate command argument. Once the correct thrusters have been autonomously selected for a given maneuver the algorithm simply commands these thrusters on continuously for the specified time duration. Spin rate control is active during the burn, however, precession control is not. Thruster misalignment, thrust variation, timing, and mass property uncertainties may produce an excursion during the burn. An error budget exists to account for these uncertainties such that the maneuver execution error can be bounded. The simplicity of this algorithm allows for execution in both the CELESTIAL and SUN LINE major modes once the spacecraft attitude is initialized.

Lateral Delta V Maneuver. Unlike the other maneuver method, a lateral delta v maneuver is constrained to be executed from the CELESTIAL major mode only, since three-axis spacecraft attitude must be known *a priori*. A specified delta V vector in the J2000 coordinate frame determines the 'clock' angle orientations for the center of each thruster burn arc (thruster actuations are such that two pairs are fired cm-half a revolution from one another). A specified time duration is also used here, however, the total on time must be decomposed to the maximum allowable burn arc autonomously calculated. Spin rate or precession control is not active during the burn. Again thruster misalignment, thrust variation, timing, and mass property uncertainties may produce an attitude excursion during the burn. An error budget exists to account for these uncertainties such that the maneuver execution error can be bounded.²

FST/P TEST PLAN

Functional Verification

The first stage of development involved creation of fundamental building blocks. Recall that both the DSA and the SSA were standard procured items which required only slight modifications to accommodate the operational characteristics of the Pathfinder spacecraft. In contrast, the PDI was a custom design derived from that of the Cassini VDE. These components were to be subjected to stand-alone or assembly level tests at facilities other than the FST/P. Flight software development was planned to begin with individual algorithm development by control system analysts. Breaking from JPL tradition, these individual would then utilized "C" programming language autocode generation to assist in considerably shorting the development life cycle by eliminating the usually time-consuming task of manual translation. Individual flight software objects would then validated (unit tested) through the "PATHSIM" group of software models, again in-facility other than the FST/P.

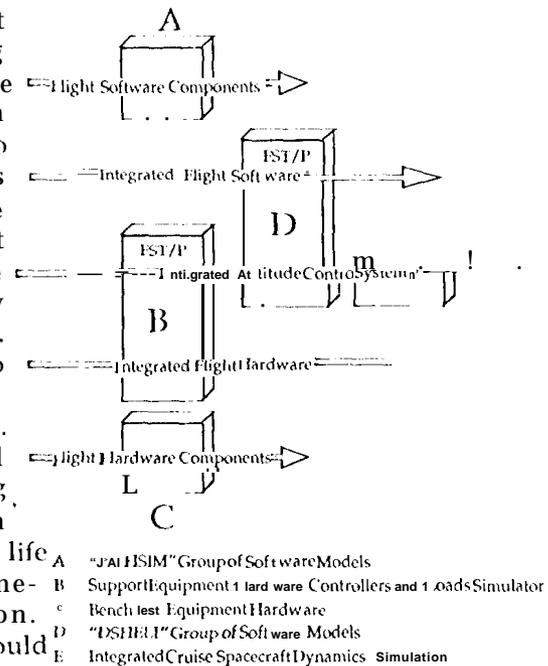


Figure 8 Functional Verification Diagram

The second stage involved separate flight hardware and software integration and verification activities. It was here that the capabilities of the FST/P were to be first utilized. Flight hardware electrical integration would be performed with electrical ground support equipment that would be used for both functional and performance verification activities. In regards to flight software, individual objects would first be integrated with one another and then code subjected to a higher fidelity, real-time spacecraft dynamics simulation. Similar to the electrical ground support equipment, this simulation will continue to be utilized for further testing. Although the processor for this simulation is not actually located within the FST/P, recall that the JPL institution's FST organization was instrumental in developing and adapting this multi-mission simulation for use on Mars Pathfinder.

The third stage was actual integration of both flight hardware and software components. To facilitate validation of all required functions, the ground support equipment used for hardware electrical integration and validation and the Dshell group of software models were integrated into a single system. This allows for "closed-loop" cruise spacecraft dynamics simulation with which the ACS can be verified. Figure 8 provides a high-level graphical summary of the test approach followed for the functional verification phase. It should be noted that the figure does not include a 'feedback' loop for rework and retest that is part of almost all development life cycles. Finally, each of the modes was exercised in a system-like configuration as part of the formal system level integration and test program. Although spacecraft power had been supplied by the actual Power and Pyro Switching (PPS) subsystem, the Propulsion Simulator still provided the inductive and resistive loads to the PDE during this test.

The fourth and final stage involved exercising select modes in a flight-like scenario where synchronization to spacecraft clock time and execution of activities via command sequence machine were important aspects. Following the validation of the spacecraft operations in the FST/P, the command sequences were executed as part of the formal system level integration and test program.

Performance Verification

Possibly the most crucial stage is actual performance verification of each specific mode under nominal and anomalous conditions. Previous tests helped to answer the question of does it work, while this stage of testing will help to determine how well it works. Both precision and accuracy of each attitude determination and control mode will be verified to ensure that performance characteristics are within the defined specifications. Test results will help to determine important parameter values. Testing will take place in the FST/P with the engineering model and flight spare hardware, since the actual flight units are being used as part of the formal system level integration and test program.

The final stage of testing will actually involve exercising the ACS as much as possible prior to launch on the actual flight units. Solar thermal vacuum, pre-ship test at JPL and the post-ship test at the Kennedy Space Center are some of the opportunities for performing these tests.

FST/P GROUND SUPPORT EQUIPMENT

As previously stated, essential flight hardware components to be used for ACS validation consists of the DSA, SSA, PDE, and the thrusters. To facilitate "closed-loop" dynamics simulation testing these components would have to be interfaced to special Electrical Ground Support Equipment (EGSE).

Ground Support Equipment Hardware

The DSA electronics box has a total of seven connectors. Five are obviously for the five DSA heads, one is for the CSIF, and the final one is designated as the test port. Although the ideal situation would have been to include all flight hardware as part of FST/P testing, a decision was made to bypass the DSA heads altogether and stimulate the DSA electronics directly through the test port. The advantages of which translated into both schedule and cost effectiveness, since considerable Cassini design inheritance of the EGSE was possible with this approach. The DSA Cent roller is essentially comprised of VMEbus-compatible Commercial-Off-The-Shelf (COTS) Digital to Analog Converter (DAC) cards, custom output amplifier cards, and a test panel.

Automatic threshold adjust and gray code signals originating from ground support equipment software pass through the controller and are fed into the IDSA electronics test port. Input data are either be stand-alone files used for electrical integration and test or real-time files used for closed-loop dynamics simulation.

Given no similar test port for the SSA, the only viable option for stimulation was through an LED hood which mechanically interfaced to the SSA baffle. Pulsing of the LED simulates star events visible to the SSA optics. The SSA Hood Controller consists of a COTS DAC card, custom output isolation card, and a test panel. Signals originating from the ground support equipment software pass through the controller directly to the LED. Again, input data are either stand-alone files used for electrical integration and test or real-time files used for closed-loop dynamics simulation.

It was understood early on that the propulsion subsystem would not be available for use in the FST/P due to scheduling incompatibility and the technical challenges it would present during actual operations. Therefore, the thrusters were bypassed altogether for the closed-loop dynamics simulation. Instead, PDE drive signals which would otherwise be sent to the propulsion subsystem would be sent to the electrical ground support equipment. To simulate the actual propulsion subsystem loads of two latch valves, eight thruster valves, and eight catalyst bed heaters, the Propulsion Simulator was designed, fabricated, and assembled. Since the general design of the Mars Pathfinder PDE was derived from that of the Cassini VIDE (Valve Drive Electronics), the general Mars Pathfinder Propulsion Simulator design was also derived from the Cassini Propulsion Simulator. It essentially consists of custom inductive and resistive loads cards, custom interface and receiver cards, and a COTS change of state card. Inductive load cards simulate the latch valve and thruster loads, while the resistive cards provide the catalyst bed heater loads. The interface card provides the link between the load cards and the receiver card and provides for either optical or relay isolation. The receiver card provides an interface to the change of state card by converting differential signals to single-ended TTL (Transistor Transistor Logic) signals.

Ground Support Equipment Software

Emulation of actual cruise spacecraft dynamics was accomplished through utilization of the DARTS Shell (Dshell) multi-mission simulation environment made available by JPL FST personnel. The Dshell integrates the DARTS (Dynamics Algorithms for Real-Time Simulation) computational engine with a library of actuator, sensor, and motor models. Although DARTS is capable of simulating tree-topology, flexible, multi-body systems, it was determined that sufficient accuracy could be obtained through simple rigid multi-body modeling of the Pathfinder spacecraft. Models used in this specific application consisted of those for the Sun Sensor, Star Scanner, and Thrusters. The Dshell is capable of operating from a number of computational platforms ranging from standalone workstations to embedded processors and could be run in real-time and non-real-time configurations. The Dshell would first be used for integrated flight software verification, however, it would be indispensable in the development of the integrated cruise spacecraft dynamics simulation described below.³

Integrated Cruise Spacecraft Dynamics Simulation

Integration of the hardware controllers and load simulators with that of the Dshell computational engine and models created the proper environment for validation of the flight hardware and software associated with the ACS. A high-level block diagram of the integrated cruise spacecraft dynamics simulation is shown in Figure 9.

To help facilitate understanding of this system, a simple walkthrough of a commanded spacecraft turn to a given inertial attitude is as follows: the appropriate command is processed by the MFC and drive signals are sent through the HICD, CREU, and CSIF electronic boards to the PDE. The Propulsion Simulator COS card senses inductive load changes caused by the PDE drive signals and transfers this information to the Thruster Hardware Model. The model converts simulated thruster actuations to forces and torques about the spacecraft center of mass. The DARTS computational engine then determines the resultant spacecraft motion and transfers this information to both the Sun Sensor and Star Scanner Hardware Models. The Sun Sensor Model converts this information to appropriate ATA and GC signals that are sent across the Sun Sensor Controller's DAC and output amplifier cards directly into the DSA electronics. The Star Scanner converts the information to star events parameters that are sent through the Star Scanner Hood Controller's DAC card and is used to command the LED visible to the Star Scanner. All sensor data is then sent back to the MFC for processing and used for spacecraft attitude determination.

FST/P TEST RESULTS TO DATE

A concurrent engineering approach in the development of the closed-loop ground simulation and actual testing was necessary to facilitate the timely completion of the ACS Junctional verification task. The general simulation infrastructure and functionality related to the DSA and PDE interfaces, once deemed to be operational, were first utilized for initial verification of the modes categorized under SUNINIT and SUNLINI. Problems identified were related to a broad spectrum of classifications, citing ground and flight components as culprits. Some examples follow:

During one test session, flight telemetry and simulation output data indicated that the spacecraft spin rate had gradually and inexplicably increased from the nominal angular velocity level to that sufficient in causing anomalous flight software behavior. Analysis of the situation exposed an error in the FST/P electrical grounding configuration. A possible ground loop condition had been created when the +28VDC power supply of the spacecraft power bus simulator was found to have been tied to the defined spacecraft chassis (earth) ground without the required 10K Ohm resistance in series. Undesired transitory events in this "noisy" configuration may have been perceived by the Propulsion Simulator as rising and/or falling edges of square wave thruster pulses emanating from the PDE. It was believed that during this test session the occurrence of a single transient event was perceived as a spin thruster open command, misleading the Dshell into presuming that a continuous spin-up activity had been requested. This electrical grounding configuration problem has since been rectified. However, to further increase the robustness of the simulation a change to the Thruster Model was incorporated. Rather than determining thruster open/closed duration based on the time of rising and falling edges, an "area averaging" scheme is now being used to minimize the impacts of transient events. Since incorporating both modifications, the problem has not returned.

The critically of flight software timing, although always regarded as important, became more evident as ACS testing commenced. A number of test sessions resulted in suspended operations of BC scheduling, BC distribution, and ACS main flight software tasks due to apparent cycle slips. These unexpected occurrences required that the flight software be reset before resumption of test activities. To rectify the situation a number of modifications were implemented. Improvements were made to the VxWorks operating system that allowed for timing margin to significantly increase. The ACS main task was made more robust for synchronization to spacecraft clock time by increasing the timing tolerance range initially and then subsequently reducing and managing to the defined specifications.

Valid flight software command parameter ranges were confirmed for many activities, however, some commands contained smaller limits than expected. Rapid response and resolution by flight software engineering personnel minimized the impacts to the overall testing schedule.

Completion and integration of the SSA interface functionality to the closed-loop ground simulation allowed for the initial verification of the modes categorized under CELESTIAL. Again, initial integration of this set of capabilities identified problems related to both ground and flight components. Some examples of these follow:

Test results over several test sessions indicated that channel B of the actual SSA flight unit was excessively "noisy", such that stellar reference and three-axis attitude determination could not be accomplished. Detailed analysis led to the conclusion that the electrical test harness was not built correctly at the SSA interface in that a temperature sensor was expected internal to the SSA electronics, but was not present due to a subsequent change in requirements. Therefore, unterminated wires from the harness may have coupled with the preamplification circuitry and caused the anomalous condition. Although the problem is now well understood, modifications to the test configuration were not required. Testing continues without incident in the IST/P with the flight spare version of the SSA, which has the expected temperature sensor installed. It should be noted that the actual flight harness currently being used for formal system level integration and test is built correctly and that the actual SSA flight unit exhibits nominal noise levels for both channels in the flight configuration.

in at least two test sessions, it had been noted that the flight software estimated spacecraft quaternions returned through the nominal spacecraft telemetry Packets differed from that returned as part of a software "debug" telemetry packet and from that produced by the closed-loop ground simulation. Analysis later confirmed that the double-precision to floating-point data conversions were being performed incorrectly. Intermediate terms with negative values were defined as zeros. This has since been resolved.

Once testing of all ACS modes had concluded, a dry-run of the actual procedure to be exercised as part of the formal system level integration and test was performed. A number of liens had been identified as part of the dry-run, and several minor problem reports were actually written as part of the formal test. However, formal system level integration and test activities were completed in just over one-half of the allocated time and generally regarded as successful.

As problem reports continued to be resolved, preparations began for exercising select ACS modes in "flight-like" scenarios through complete synchronization to spacecraft clock time and command sequence activity execution. Although additional problems were identified during this phase of testing, the number of those attributed to ground support equipment hardware and/or software finally began to decrease. This was an important milestone, since every effort was being made in the IST/P to quickly resolve these problems and provide a high fidelity environment where the validity of test results would not be compromised through these types of errors. Other problems identified include the following:

Setting the spacecraft clock time to that required for a given command sequence to begin execution caused the suspended flight software task problem to manifest itself again. This time floating-point exceptions triggered these unwanted events. Modifications were incorporated that increased the robustness of the floating-point exception handling and set spacecraft clock handling functions. This problem has not re-occurred.

A discrepancy in the spacecraft-relative position of the Earth vector as calculated by the flight software IVJ function and that made available in the closed-loop ground simulation for a given spacecraft clock time provided an indication of a problem. Analysis showed that the flight software was using an incorrect spacecraft base for the given spacecraft clock time. Distinction from Earth based, Sun based, and Mars based definitions are required to account for third body influences near Earth and Mars. In this case, a Sun based definition should have been used instead of the Earth based. Modifications were made to correct this situation.

A final dry-run of the actual command sequences to be exercised as part of the formal system level integration and test was performed and completed without major incident. Actual execution of the formal test was also completed without incident, assuring all those involved that the ACS was functionally operational. Although performance verification awaits, the ACS functional development and test program was deemed a resounding success!

CONCLUSION

Given the task of validating the Mars Pathfinder ACS prior to launch and in-flight operations, a collaborative effort was undertaken by project and JPL FST personnel in developing a suitable environment for functional and performance verification. To help facilitate this endeavor, a closed-loop ground simulation consisting of hardware controllers and load simulators, and real-time software dynamics algorithms and models was designed and developed for the FST/P. Functional verification was successfully accomplished, allowing fundamental flight hardware and software problems to be identified and resolved prior to formal system level integration and test. Preliminary test planning for extensive performance verification has been completed and calls for the same closed-loop simulation to be utilized for testing under nominal and anomalous conditions. Expectations are such that any and all outstanding flight hardware and software problems adversely affecting accuracy and/or precision of commanded activities will be identified and resolved prior to formal system level integration and test.

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