Space Sci Rev (2008) 141: 117–152
DOI 10.1007/s11214-008-9416-2

The THEMIS Constellation

P. Harvey · E. Taylor · R. Sterling · M. Cully

Received: 17 March 2008 / Accepted: 8 July 2008 / Published online: 23 September 2008

Abstract The Time History of Events and Macroscale Interactions during Substorms (THEMIS) mission is the fifth NASA Medium-class Explorer (MIDEX), launched on February 17, 2007 to determine the trigger and large-scale evolution of substorms. The mission employs five identical micro-probes (termed "probes"), which have orbit periods of one, two and four days. Each of the Probes carries five instruments to measure electric and magnetic fields as well as ions and electrons. Each probe weighs 134 kg including 49 kg of hydrazine fuel and measures approximately $0.8 \times 0.8 \times 1.0$ meters ($L \times W \times H$) and operates on an average power budget of 40 watts. For launch, the Probes were integrated to a Probe Carrier and separated via a launch vehicle provided pyrotechnic signal. Attitude data are obtained from a sun sensor, inertial reference unit and the instrument Fluxgate Magnetometer. Orbit and attitude control use a RCS system having two radial and two axial thrusters for roll and thrust maneuvers. Its two fuel tanks and pressurant system yield 960 meters/sec of delta-V, sufficient to allow Probe replacement strategies. Command and telemetry communications use an S-band 5 watt transponder through a cylindrical omni antenna with a toroidal gain pattern. This paper provides the key requirements of the probe, an overview of the probe design and how they were integrated and tested. It includes considerations and lessons learned from the experience of building NASA's largest constellation.

Keywords THEMIS · Microsatellite · Probe · Constellation

PACS 94.30.-d · 94.30.Cl · 94.30.Cb · 94.30.Ch · 94.30.Cj · 94.30.C- · 94.30.Cp · 94.30.Lr · 94.30.Va · 94.30.Xy · 96.50.Fm

P. Harvey (✉) · E. Taylor · R. Sterling
Space Sciences Laboratory, University of California at Berkeley, Berkeley, CA, USA
e-mail: prh@ssl.berkeley.edu

M. Cully
ATK Space Division, Beltsville, MD, USA

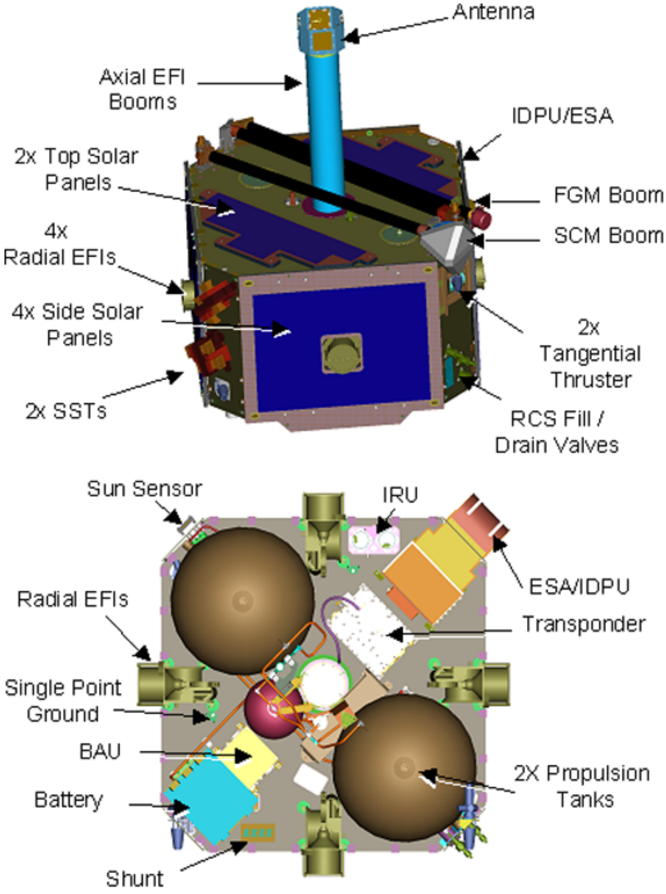

Fig. 1 Probe bus and instrument subsystems

1 Introduction

The THEMIS mission employs five simple, identical probes, shown in Fig. 1 that fly independent and synchronized orbits around earth. The orbit periods are designed to produce a combined measurement set resulting from the apogee region conjunctions due to the natural evolution of the orbits. The probes communicate independently with the operations center that operates each probe in a round-robin serial fashion during normal operations. While the probes are highly autonomous, attitude and orbit determination is maintained by the ground operations center with all orbit and attitude maneuvers nominally taking place during ground contacts. Thorough discussions of the mission design and instruments are presented in Angelopoulos et al. (2008) and the operations are described in Bester et al. (2008). The ground-based observatories measuring the aurora are described in Harris et al. (2008).

All five THEMIS probes were launched together on a Delta II 2925-10 ELV from the Eastern Range into a stable $1.07 \times 15.4\ R_e$ orbit. This orbit was near the planned orbits of the inner probes, called P3, P4 and P5. From this orbit, probes P1 and P2 would need to accelerate to their final orbits, while P3 through P5 would decelerate into their final orbits. The Probe Carrier (PC), a simple mechanical fixture bolted to the 3rd stage, dispensed the probes, directly into this common initial orbit spin-stabilized at 16 ± RPM. An on-board hydrazine propulsion Reaction Control System (RCS) performed the final placement of each probe into its final orbit with minor trimming prior to the prime science tail season.

The spinning probes are passively stable, even under worst-case scenarios. The single-string design is simplified by a minimal hardware complement, inherent functional redundancy, with the instruments and the bus designed for graceful degradation. Analyses showed that probes in the P3 or P4 orbits would have sufficient fuel to accelerate to the P1 or P2 orbits, or decelerate to P5, allowing a replacement of a failed probe. Since four probes define the minimum mission, THEMIS benefits from "constellation redundancy" with a reliability of 80% for the nominal 2-year life and 93% for the minimum 1-year life (Frey et al. 2008).

The five flight instruments include a Electro-Static Analyzer (ESA), Solid State Telescope (SST), Fluxgate Magnetometer (FGM), Search Coil Magnetometer and Electric Field Instrument (EFI) (Angelopoulos et al. 2008). All instruments are identical on all five probes and were built using production methods. The instruments have adjustable data rates to suit different orbit profiles and utilize heritage burst-data collection strategies incorporated in the Instrument Data Processing Unit (IDPU), which has the single electrical interface to the bus (Taylor et al. 2008).

2 Systems Engineering

The engineering of a constellation of probe, rather than a single probe, influenced nearly every discipline involved in the mission, from the number of probes, their level of redundancy, their power and mass, their magnetic and surface charging cleanliness, to how to integrate and test them, and even how to implement the NASA review process. From the project start, engineers and managers understood they needed a Probe design which was simple to build, test and operate. To implement "standard" spacecraft features on multiple probes would likely exceed the schedule and cost caps of the program.

Redundancy. Figure 2 illustrates the electrical block diagram for the Probe. Probes use a single-string design, taking advantage of inherent redundancy and having added redundancy only when mission critical and practical. Probes have only one Bus Avionics Unit (BAU) and one IDPU since it would be impractical to make these redundant. On the other hand, probes have redundant heater circuits since this was practical to implement. Spin plane booms are inherently redundant with one another every quarter rotation, and this is sufficient to meet the E-Field timing requirements. The axial boom and magnetometer boom have redundant firing circuits because of mission criticality and minimal increase in complexity.

Robust Design. Probes are both simple and robust as possible. Solar Arrays cover all six sides and the BAU simply shed power loads automatically if the battery gets too low, assuring a positive power configuration in any attitude. Probes, therefore, have no on-board maneuver capability and their RCS "cat bed" heaters are nominally off. All maneuvers are calculated in detail by Mission Operations, simulated on the ground and then uploaded to each Probe as needed (Bester et al. 2008). In addition, Probe attitude stability during on-orbit deployments was extensively analyzed and was one fault tolerant in most cases (i.e. Probe deployment and EFI deployment). This architecture simplified the design, implementation and cost of the Probes while keeping flight operators in complete control.

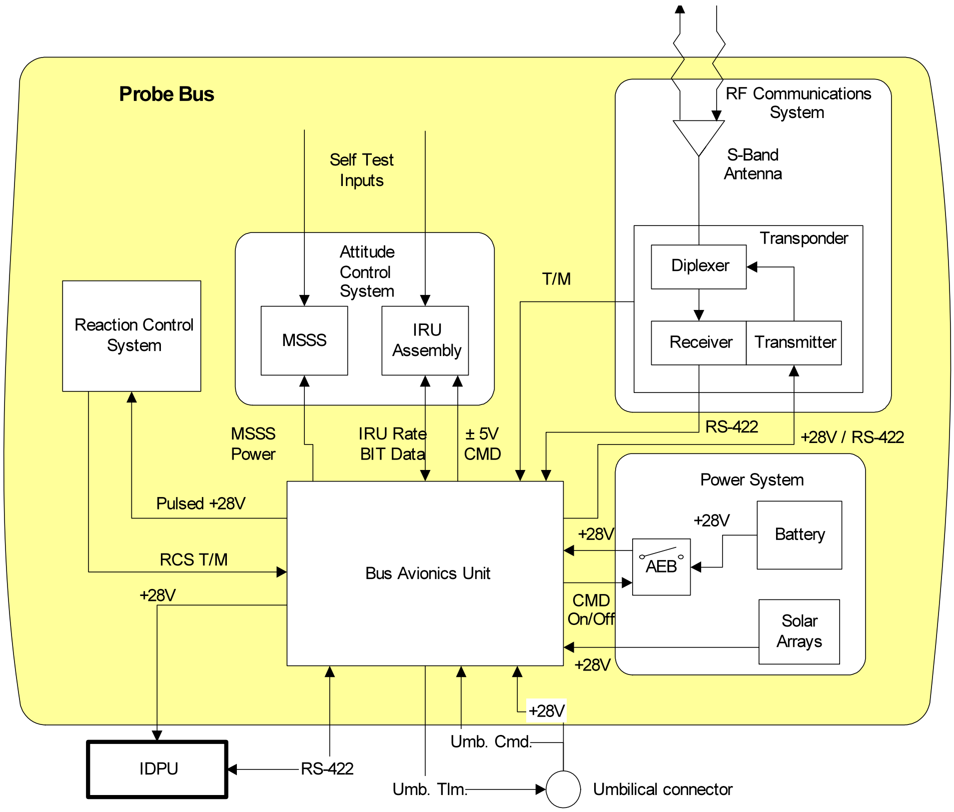

Fig. 2 Probe electrical block diagram

Fault Detection. Probes incorporate on-board fault detection and correction (FDC) sequences to keep the Probes both thermally safe and power positive at all times. Real Time Sequences (RTSs) react to table driven Limit Monitors (LMs) to turn on heaters when components get too cold and turn off loads in response to under-voltage or over-temperature conditions. Solar Arrays cover all six sides and the BAU steps down power loads as the battery drops lower in voltage, assuring a positive power configuration in any attitude. Probes, therefore, have no need for on-board maneuver capability to stay power positive and their RCS "cat bed" heaters are nominally off. Table 1 shows the load shedding sequence, the voltage and the depth of discharge the load shedding begins at, and the loads that are shed. All non-essential loads (e.g. instrument payload, pressure transducer and inertial reference unit (IRU)) are shed first, in Loadshed 1. Primary heaters for the Instrument and then the Bus are turned off next if the voltage continues to drop. Secondary heaters are set to come on ~ 5 degrees lower than the primary ones, saving some power by letting components get a little colder. The primary circuits of critical heaters such as the RCS heaters are never turned off.

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Table 1 THEMIS load shed sequence

LM	Description	Limit exceeded	RTS	RTS description
LM 02	Loadshed 1	Battery voltage < 29.0 V (30% state of charge)	RTS04	IDPU, Catbed Htrs, ISO valve, IRU, Press Transducer are powered off
LM 03	Loadshed 2	Battery voltage < 28.0 V (25% state of charge)	RTS05	Instrument primary heaters and transmitter are powered off
LM 04	Loadshed 3	Battery voltage < 27.0 V (20% state of charge)	RTS06	Bus primary heater is powered off

design, implementation and cost of the Probes while keeping flight operators in complete control.

Low-Power Design. Fitting multiple probes within the constraints of the launch vehicle fairing limited the overall size of the probe solar arrays. This restricted available power to almost all subsystems, except ones with relatively low duty cycle such as the transmitter. Chief power users such as the flight computers have their clock frequencies as low as possible to keep power to a minimum. The Catalyst Beds and Inertial Reference Units are powered off until maneuvering. In order to survive earth eclipse of up to 3 hours, the surface materials passively bias all temperatures up several degrees so that heater power is minimized in eclipse.

Power Loads by Mode. The IDPU has a number of independent configurations, which mainly affect Instrument data storage rate. Three basic modes, as described below, effect power consumption and dissipation. A fourth mode, engineering mode, affects the IDPU data rate only:

- SAFE POWER MODE—IDPU Power-On State. Core Systems (LVPS, PCB, and DCB) are powered on, all instruments off. Mode is entered on reset (power-on), by ground command, or in response to flag in Probe status field (power-down imminent) in preparation for IDPU load shed. Saves power and the contents of SRR.
- LOW POWER MODE—IDPU Core Systems (LVPS, PCB, and DCB) and FGM are powered on, all other instruments off. Mode is entered by ground command in preparation for maneuvers (FGM data for attitude determination) or in response to flag in Probe status field (low-power) in case of low power condition.
- SCIENCE MODE (Nominal)—Normal operating state, full science data collection. IDPU Core Systems, instrument sensors and associated electronics are powered on. Mode is entered by ground command (instruments are powered on one at a time during early operations).
- ENGINEERING MODE—Higher engineering rate and additional telemetry points telemetered. Operational only, typically during ground contact. Mode is typically entered by ground command in preparation for early operations (instrument health and safety diagnostics) and special case instrument operations (boom deploy, high voltage turn-on).

Table 2 provides the measured power consumption by instrument modes.

Separation Design. The design of the separation sequence and carrier changed several times during the project, driven by both engineering and safety concerns. Initially, Probes would be commanded autonomously by the individual Probe Bus Avionics Unit computer

Table 2 Instrument power consumption by mode

Instrument mode	IDPU	EFI	FGM	SCM	ESA	SST	Power
SAFE MODE	ON	OFF	OFF	OFF	OFF	OFF	6.8 W
LOW POWER MODE	ON	OFF	ON	OFF	OFF	OFF	7.7 W
SCIENCE MODE	ON	ON	ON	ON	ON	ON	15.5 W

to release from the carrier. A second concept incorporated a separation system timer with a dedicated battery hosted on the PC to release the Probes. Both concepts were identified as high-risk developments due to the nature of controlling explosives at the range and the critical aspect of requiring the near simultaneous firing of the four lower Probes from the Probe Carrier to avoid collisions. NASA management at GSFC and KSC recommended, and then implemented, the extension of the launch vehicle third stage separation event onto the PC. The extension lines were provided by the launch vehicle provider and qualified as if part of the vehicle itself. Finally, at GSFC recommendations (based on previous mission lessons learned), a Probe separation status system was added by ATK Space to diagnose if any probe had not separated. This data was relayed to the ground through the launch vehicle telemetry system and this, in fact, verified that all systems separated within 1 millisecond of the commanded time.

Reviews. THEMIS conducted a thorough review program with a NASA-provided review team composed of GSFC and HQ-selected members called the Integrated Independent Review Team (IIRT). A total of 39 reviews were conducted at the system and subsystem levels, 27 of which formally run by the IIRT. These reviews resulted in 269 Requests for Action (RFA) and all actions were formally documented and closed by the IIRT prior to launch.

THEMIS benefited greatly by the experience and insights provided by the IIRT. Given that the constellation presented new and unique challenges for the THEMIS team, the bilateral discussions proved effective in improving the design and implementation of the constellation.

Spares. Based upon past experience in the Cluster I&II programs, in which most of the instruments relied upon their spares at one time or another, THEMIS chose to build at least one spare of each instrument. This decision proved wise as three of the five instruments swapped out sensors for the spare unit. Spare bus components included a battery, side solar panel and top/bottom solar array although none of these were actually used.

Radiation. THEMIS has a two-year design life, mainly driven by its radiation environment and 100% total ionization dose margin. Originally planned for launch in Fall 2006 and having two winter campaigns in 2007 and 2008, the launch vehicle was delayed until just past the winter campaign of 2007. To ensure that baseline objectives remain intact and avoid a radical mission redesign late into the program, a 5-month coast-phase was inserted into the mission giving it a total duration of 29 months, and cutting the radiation margin to about 65%.

Figure 3 shows the approximate annual radiation dose encountered by the P3/P4 electronics inside increasing amounts of Aluminum shielding. Calculations were made by Innovative Concepts. The P1 and P2 orbits have less radiation exposure since they spend less time in the electron belts, while P5 has a slightly higher dose rate based upon more time in these belts. Based upon these simulations, THEMIS enveloped these radiation requirements and

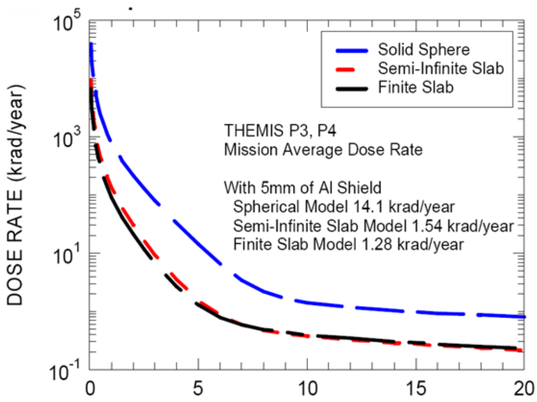

Fig. 3 The dose depth curve for probe 3 and 4 orbits in millimeters of Al

baselined using 5 mm of aluminum (or equivalent) and 66 krad tolerant electronic parts. Together these requirements were found to provide an achievable balance between parts costs and Probe mass.

Single Event Upsets. Parts were selected to be SEL-immune to a LET of >37 MeV cm^2/mg, or else protected against damage by protection circuitry. Both the bus and instrument systems implemented SEU recovery schemes. In the bus, flight software kept three copies of all data in memory and continuously scrubbed out bit errors. The instrument data processor implemented a three-second watchdog timer, which resets the processor if it does not "pet" the Watchdog in that amount of time. The instrument processor also includes a hardware Error Detection and Correction (EDAC) circuit and scrubber subsystem, which uses the upper sections of the instrument Solid State Recorder to store the error correction codes. The scrubber operates on data 4 bytes at a time, generating a byte of check-bits in the ECC segment for every 32 bits in the data section. All FPGAs in the instrument used a Triple-Modular-Redundancy scheme to avoid any single SEU causing an error.

3 Payload

The THEMIS science payload combined the science instruments into a single package with shared data processing and storage capabilities. The instrument suite was designed, built, tested and delivered as a single item for integration with each probe. While providing greater scientific capabilities in on-board power and logic sharing, the approach also provided a single electrical interface to the probe, allowed completely parallel instrument and bus development schedules, while greatly simplifying probe I&T. The only exception to this rule was the axial EFIs, which were contained in cylindrical composite tube that was integrated with the Bus structure at ATK following test verification at UCB.

3.1 The Instrument Complement

The instrument complement consists of five instruments: the Fluxgate Magnetometer (FGM), the Search Coil Magnetometer (SCM), the Electrostatic Analyzer (ESA), the Solid State Telescope (SST) and the Electric Field Instrument (EFI). These sensors are controlled by, and data is returned through the Instrument Data Processing Unit (IDPU), which has the

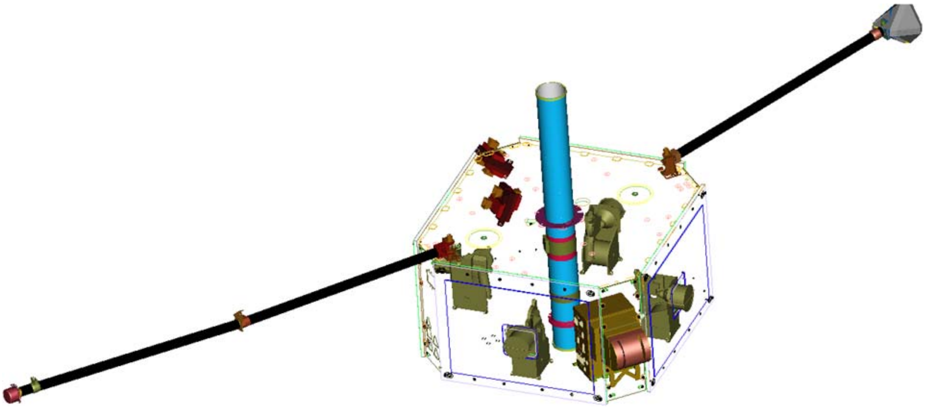

Fig. 4 Payload accommodation showing FGM and SCM deployed

electrical interface to the Bus Avionics Unit. All harnessing from the sensors to the IDPU was built and tested prior to Probe delivery, further simplifying subsequent I&T.

An overview of the instrument performance characteristics and mission requirements is given in (Angelopoulos et al. 2008). Details of each instrument and IDPU are provided in a series of companion papers by Auster et al. (2008), Bonnell et al. (2008), McFadden et al. (2008), Larson et al. (2008), Roux et al. (2008), and Taylor et al. (2008).

Mechanical. Though none of the instruments have critical pointing requirements, integration of the boom-mounted magnetometers nevertheless required precision measurements both for balance calculations and for attitude determination Pankow et al. (2008). The FGM and SCM booms are one-shot deployment mechanisms responsible for holding the sensors still with respect to the probe chassis. Thus, after the booms were mounted to the top deck, precise measurements were made using a portable coordinate measuring machine (CMM). Instrument accommodation is shown in Fig. 4.

The EFI provides 3D coverage once its Spin-Plane and Axial booms are deployed. These boom systems are located on the Probe Center of Gravity (CG) so that the deployed wires are orthogonal. The fuel tanks composite CG was aligned with EFI so that the wires will stay orthogonal even as fuel is depleted. The boom deployment sequence had Spin Plane Booms deploy first, followed by the Axial Booms in order to maintain spin stabilization. Details of this sequence are given in Pankow et al. (2008).

The ESA and SST sensors poke through the corner panels in mid span. While the SST heads were light enough to mount to the panel, the ESA was internally mounted to the IDPU chassis for support at that elevation. Table 3 lists properties of the instruments.

Electrical. The IDPU-to-Probe electrical interfaces consist of a low speed bi-directional serial interface for commands, housekeeping, and status information exchanged between the Probe and instrument, as well as a high-speed serial Clock and Data lines for science telemetry. An 8 MHz clock and 1 Hz tick line combined with a Probe UTC message provides synchronization of the two systems. The 8 MHz clock was used to synchronously sample science quantities in the IDPU.

The BAU provides instrument commands, time and probe status to the IDPU every second using the serial interface. A buffered sun-sensor pulse is used by the IDPU for spin-sectoring the SST and ESA data.

Table 3 Instrument mass, power and on-orbit temperature predictions

Instrument	Mass (kg)	Avg power (W)	Min temp (C)	Max temp (C)
IDPU	4.25	8.00	−37.3	49.2
EFI w/6 Booms	12.22	0.24	−32.5	40.7
FGM w/Boom	1.33	0.85	−54.9	15.1
SCM w/Boom	1.76	0.09	−57.1	14.6
ESA	2.87	1.70	−37.3	45.8
SST	1.35	1.20	−20.0	11.6
TOTAL	23.78	12.08		

Table 4 THEMIS contamination requirements

Sensor	Key requirement on probe
FGM	Mag < 1 nT at 2 m
SCM	Mag low AC fields
EFI	ESC < 10e–5 ohms/cm^2
ESA	Molecular < 0.01 μg/cm^2
SST	Molecular < 0.1 μg/cm^2

The IDPU provides instrument housekeeping packets to the BAU, which is combined with its data into CCSDS frames for downlink. Stored science data is transmitted over the high-speed link when commanded to do so.

Power. The Probe provides the IDPU a Main power service and an Actuator service. The Actuator service is used for deployments.

Thermal. Since each probe is very small, body mounted instruments were expected to experience larger thermal extremes than in previous missions. Temperatures for the instrument components were set at very wide ranges of −50°C to +65°C survival and −50°C to +50°C operational. Predicted on-orbit extremes are shown in Table 3 and are based upon probe-level thermal balance testing.

Contamination. While several THEMIS sensors are sensitive to contamination, they were designed for easy handling and simple integration to the probe. The key contamination requirements imposed on the Probe are provided in Table 4.

The ESA and SST sensors are sensitive to molecular and particulate contamination at the sub-micron level. Both have covers and an external purge provided by the instrument for integration and test.

The EFI sensors are sensitive to handling issues; i.e. asymmetries in the reflective properties of the sphere, which would generate a spin-period photo-emission. Deployment testing at I&T required a clean room environment and handling with gloves. The EFI also required that all Probe surfaces be electro-statically clean to 10^{-8} ohms/cm^2, which equated to a requirement of limiting the total Probe exposed surface to have less than 1 cm square of insulating surface. All exterior surfaces and apertures, which are eclipsed had to meet this requirement.

Typical sources of contamination on the Probe were mitigated to a satisfactory level for THEMIS instruments. The solar panels required particular attention to ensure they were both electrostatically and magnetically clean. To be electrostatically clean, the panels were designed such that: all cover glasses were conductively coated; areas between cells and all perimeter areas were covered by conductive layers; cell interconnects were covered by conductive shielding; no exposed insulators were present; and each panel substrate was directly connected to the spacecraft single point. To be magnetically clean, cell laydown and wiring of each panel met the derived requirement that the resultant magnetic field be less than 24 pT as measured at SCM, with the predicted field strength having a peak of 9 pT. This was accomplished by orienting the cells, strings and back wiring so their magnetic fields cancel and by configuring the power and return wires in twisted pairs to minimize the stray magnetic fields generated by the current flow. For molecular contamination, wire harnesses, solar array panels, thermal blankets and heaters were baked out prior to instrument I&T. The Thermal Vacuum chamber was baked prior to probe insertion, its contamination level monitored during the test, and the chamber backfilled with dry nitrogen at the end of the test.

3.2 Fabrication and Test

Parts. Parts selection required GSFC-311-INST-001 and GSFC PPL-22, Appendix B derating practices. In general, we used grade 3 parts with some up-screening to grade 2 for key items in critical sub-systems such as the Bus Avionics Unit. We organized the "Common Buy" program, purchasing parts for all instruments both for the raw cost efficiency as well as to limit the number of different part types and purchasing lots needed. By limiting part types and lots, we lowered the probability that we would have to open the Probes and replace parts due to a NASA Alert.

Manufacturing. The sheer number of subsystems drove manufacturing decisions towards automated circuit board fabrication as well as internal test functions for each subsystem. For example, the field instruments can stimulate all sensors and the particle instruments can simulate counts and energies. We arranged for manual work to be performed by the same technician for all units of the same design. This included hand soldering, harnessing, thermal blankets and thermal taping. These actions yielded remarkably uniform performance and substantially accelerated the flight test program.

Flight Software. UCB developed the IDPU software using PC-based assembler and linker products developed in-house. The 208 software requirements, specification and test verification were actively reviewed by NASA Independent Verification and Validation (IV&V), and their recommendations were very helpful. For the most part, FSW performance analyses and data products used Excel spreadsheets, and the modular software was tested on the IDPU engineering model. See Taylor et al. (2008).

3.3 Instrument Suite Integration

Instrument Integration and Test (II&T) was a two step process, in which sensors were tested at the box level for unique functions, then integrated to the IDPU and flight harness forming the Instrument Suite. This maximized the instrument level test time while minimizing personnel and facility resources.

For II&T, we used the UCB/SSL E125 clean room. Instrument harnessing was built using mockups of the Probe deck, and Multi-Layer Insulation (MLI) blankets were made at GSFC using instrument mockups. All harnessing and MLI were baked out in UCB/SSL vacuum chambers.

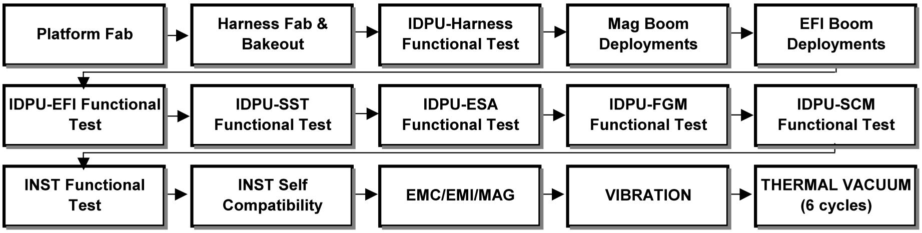

Fig. 5 Fabrication/flow for engineering and flight suites

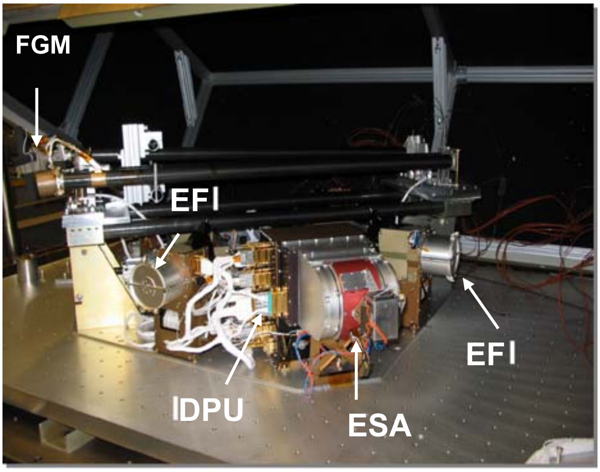

Fig. 6 Instrument suite in TV preparation

As expected, testing the first flight model (FM1) was pivotal in the maturation of procedures for subsequent models. We tested subsequent models in pairs of instrument suites subsequently; i.e. FM2, FM3 and FM4, FM5. The Instrument Ground Support Equipment (IGSE) used a language nearly identical to the ITOS used by the probe bus. Thus procedures which were developed at II&T flowed with only minor modifications into Probe I&T. This flow is shown in Fig. 5 and a photo of a suite in Thermal Vacuum is shown in Fig. 6.

4 Probe Bus and Carrier

The THEMIS Probe is a highly optimized system that met the extreme challenges posed by the mission. As summarized in Table 5, each probe consists of the bus subsystems and the instrument suite, consisting of four EFI radial instruments, two EFI axial instruments, one ESA, one pair of SSTs, one SCM, one FGM, and an IDPU. The bus subsystems include Structural/Mechanical, Thermal, Reaction Control Subsystem, Attitude Control Subsystem, Power, Communications, and Avionics.

In order to implement the concept of "constellation redundancy," each of the five probes is identical in design and capable of being placed in any of the THEMIS orbits. The probe design was driven by a number of requirements including

- All five had to be small enough and light enough to be launched on a single launch vehicle;
- Assuming small solar arrays, each probe had to be power efficient;

Table 5 Observatory facts

Number of probes	Five
Mass	Probe bus dry mass: 51 kg Instrument mass: 26 kg Probe dry mass: 77 kg Propellant: 49 kg Probe wet mass: 126 kg Allowable mass: 134 kg
Power	Probe bus power: 11 W Instrument power: 15 W Heater power (EOL/24 hr orbit/3 hr eclipse): 11 W Probe power: 37 W Available power: 40.5 W Battery capacity (BOL): 12 AHr
Communications	S band EIRP: 2.4 dB W Two-way Doppler tracking Uplink command rate: 1 kbps Downlink telemetry rates: 1 kbps to 1.024 Mbps
C&DH	Command and telemetry format: CCSDS Version 1 Engineering data storage: 64 MB, 5 days worth Timing: 8 MHz, 1 Hz and UTC distribution
ACS	Spin rate (Science): 20 rpm Spin axis orientation: $< 1°$ (knowledge), $< 3°$ (control) Spin phase knowledge: $< 0.1°$ Ground based attitude determination
Propulsion	Monopropellant hydrazine system Number of thrusters: 4 (4.4N ea.) Total ΔV: 940 m/s Propellant: 49 kg
Probe carrier	Probe carrier mass: 147 kg Total payload mass: 777 kg Mass to orbit capability: 829 kg
Science instruments	Instruments Flux gate magnetometer Search coil magnetometer Electrostatic analyzer Solid state telescope ($\times 2$) Electric field instrument radial ($\times 4$) and axial ($\times 2$) Booms 5-m axial booms ($\times 2$) 20-m radial booms ($\times 4$) 1-m SCM boom 2-m FGM boom Instrument data processing unit
Science Data Volume	Data volume: ~ 400 Mbits per day 5 days worth of storage
Radiation Environment	Total dose: 66 krads (2 years, 5 mm Al shielding, RDM of 2)
Reliability	Observatory Ps = 0.91 (2 years) Mission Ps = 0.94 (4 of 5 s/c required for mission success)

- Given the low mass, each probe had to use radiation-hardened electronics;
- To implement the orbits, the design had to maximize its fuel carrying capacity;
- To avoid contaminating the magnetic measurements, the design and components had to be non-magnetic and non-permeable;
- To reduce surface charges that would impact EFI, the exterior materials were conductive and grounded;
- Given the 3-hour shadows while operating the instruments, the design included considerable thermal blanketing, thermostatically controlled heaters, and careful selection of surface materials;
- In order to operate in any attitude/orientation, the structure design had to tolerate extreme temperature swings from $-115°C$ to $+105°C$.

The major subsystem designs and how these subsystem designs achieve the mission objectives are described in the following sections.

4.1 Structure and Mechanical Subsystem

The THEMIS Probe Bus structure provides mechanical support for all other subsystems and consists of ultra lightweight panels constructed of composite graphite epoxy face sheets and an aluminum honeycomb core. All panels have embedded fittings of either titanium and/or aluminum that have been machined to minimize mass. The sandwich panels have M55J/RS-3 facesheets and Aluminum 5056 honeycomb core. The core/facesheet bond is unsupported FM-73. Aluminum and titanium inserts are bonded into the sandwich to provide component interfaces and mating patterns for edge joints. The panels are joined at their edges using threaded fasteners and shear pins. High strength A286 fasteners in locking Phosphor Bronze helicoils are used. The probe is rectangular in shape with overall dimensions of approximately $82 \times 82 \times 45$ cm in order to provide simplicity and minimizing costs in the solar arrays. The structure is divided into a lower deck, an upper deck, four corner and side panels. The lower deck is the primary mounting surface for the instruments, propulsion system and Probe components. It also interfaces to the probe separation system. The side panels double as substrates for the solar cells. The exterior surface of the upper deck provides inserts for mounting the two magnetometer booms. The probe structure is designed so that all four side panels could be removed during I&T to allow access to the internal components.

The instrument and Probe components are mounted to the lower deck simplifying the load-bearing structure design and facilitating integration. The lower deck and separation adapter fitting is the probe primary structure, carrying the load from all the internal components, side panels and upper deck into the probe separation fitting and ultimately through to the Probe Carrier and launch vehicle. Strength margins were assessed for all structural and thermal design load cases using safety factors of 1.25 on yield and 1.4 on ultimate. Design limit loads of 10.2 G lateral and 6.03 G lateral, applied simultaneously, were used. A detailed structural FEA model was created and verified. The model was used to assess normal modes and strength.

The primary structure must also withstand the extreme temperature swings during early orbit operations and eclipses. The design employed low conductance composite structure and isolated solar panels in order to minimize internal thermal swings between full-sun and shadow operations. Extensive analysis and development testing was performed on the new composite elements of the structure. These environments are simulated via vibration testing and panel level thermal cycling at the subsystem level prior to delivery of the probe structure to Integration and Test (I&T). The mass of the entire structure and mechanical subsystem

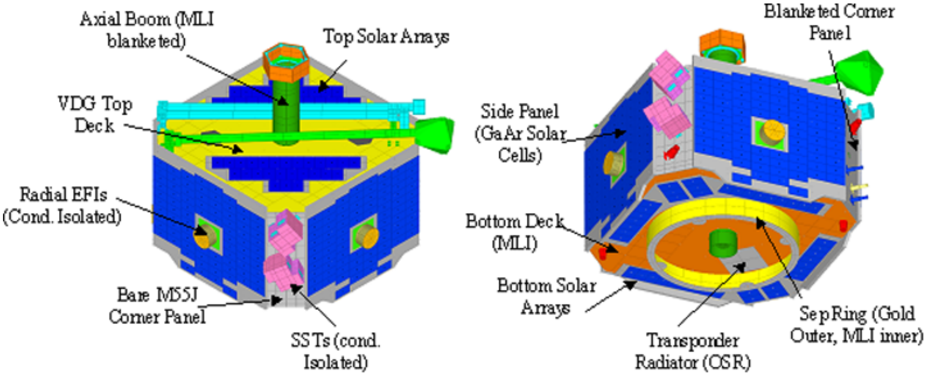

Fig. 7 Probe thermal features

including mounting hardware is 15 kg and represents approximately 19.5% of the Probe dry mass.

A proper grounding scheme was essential to minimize generation of conducted and radiated noise and to ensure predictable system level performance. The probes used a "modified" Single Point Grounding (SPG) system. The SPG is located within the BAU. It ties primary power returns and signal grounds to its chassis ground at one point. This SPG is then connected with a thick braid wire to the separation ring, which is used to terminate chassis grounds from all components. The separation ring was used instead of the probe structure because it provided a lower impedance connection than the composite structure.

4.2 Thermal Control Subsystem

The Probe thermal design was a challenge given the 3-hour eclipses, the need for maneuverability and the probe's low mass. Its thermal subsystem employs a hot-biased design using solar heat to bias component temperatures upward so the probe can survive long eclipses with minimal heater usage (less than 12 watts orbit average). Additionally, the design allows the probe to be thermally safe in nearly all sun aspect angles. With the exception of the transponder, all components either radiated directly to space or were coupled by a standard bolted interface to the spacecraft structure. Electronics boxes with significant power dissipation were painted black to radiatively coupled to the spacecraft interior as well. The transponder was similarly mounted and coated but its base was also covered with Optical Solar Radiators (OSRs) that had a direct view to space through an opening in the spacecraft deck. Components are blanketed with Multi-Layer Insulation (MLI) and have simple thermo-statically controlled film heaters. Thermistors are used for temperature monitoring. High-efficiency MLI blankets minimize heat loss from the hydrazine Reaction Control System, which must always remain above 5°C to keep the fuel from freezing in the lines. The probe includes external coatings with high solar absorptance-to-emittance ratios, such as Vapor Deposited Gold (VDG). In order to reject the transponder heat, Optical Solar Reflectors (OSR's) are used on the bottom of the probe. See Fig. 7.

4.3 Reaction Control Subsystem (Propulsion)

Each THEMIS Probe includes a Reaction Control Subsystem (RCS) to correct launch vehicle dispersion errors, inject each probe into its respective mission orbit, maintain the orbits,

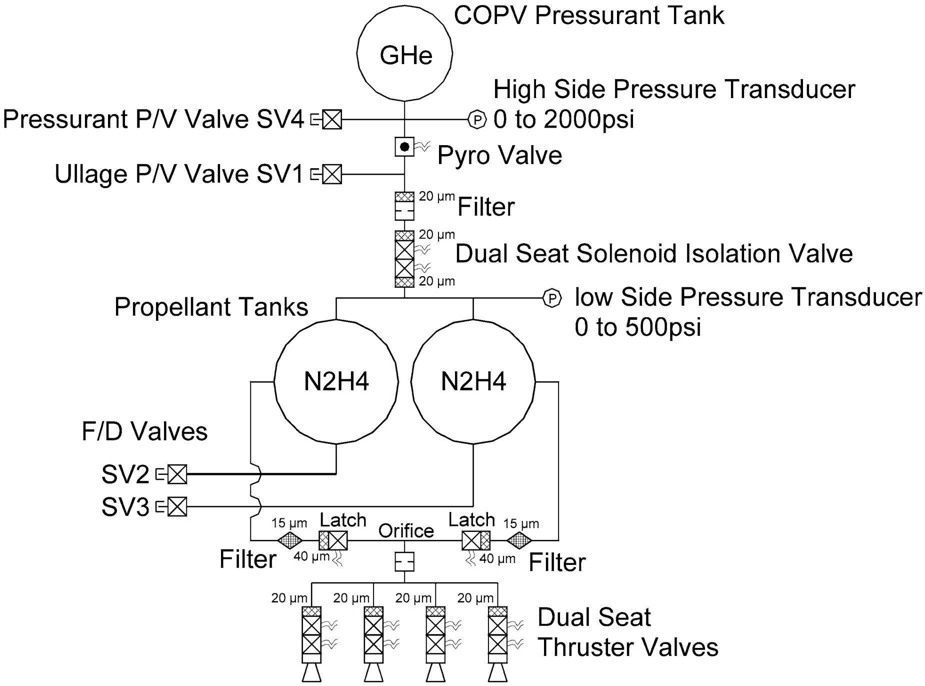

Fig. 8 RCS block diagram

adjust spin-axis pointing and maintain a nominal spin rate. The fundamental robustness of the mission design is due to the capability of probes 3 or 4 (P3, P4) to fully replace any probe, should it fail. Thus, the RCS has been sized for a nominal mission profile plus the worst-case contingency of replacing the P1 probe.

The probe is capable of both axial and side thrusting for orbit maneuvers with minimal efficiency loss allowing for operational flexibility. The tangential thrusters also act individually for spin rate adjustment. As shown in Fig. 8, the system consists of two fuel tanks, four 4.4 newton thrusters, a pressurant tank, latch valves, pyro valves, and miscellaneous hardware.

The two lightweight fuel tanks hold up to 49 kg of hydrazine (in total) and were specially designed and qualified for the THEMIS program. The tanks are made of high strength alloy (inconel) and are supported by the bottom and top panels via integral polar fittings. Tanks were verified non-magnetic by testing at UCLA. A high-pressure Carbon Over-wrapped Pressure Vessel (COPV) tank and pyrotechnic actuated valve dramatically enhance the systems capability. Once the fuel in the tanks has been depleted by approximately 25%, ground personnel command the pyrotechnic valve to open, which connects the high pressure tank to the fuel tanks. The resulting increase in pressure provides significantly more delta-V, totaling 960 meters per second.

The two axial engines provide 4.4 Newtons of thrust allowing for major orbit changes of the probe. In addition, two tangential engines of the same size provide either spin control or lateral thrust to the probe.

In order to maintain mass balance throughout the mission life, the two tanks were mechanically arranged to allow for symmetric fuel depletion. The pressurant and propellant sides of the RCS are interconnected to provide both symmetry and added probe reliabil-

Fig. 9 Miniature sun sensor

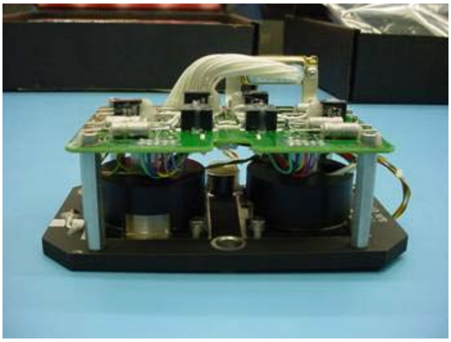

Fig. 10 IRU assembly

ity. Latch valves are located strategically to prevent adverse propellant migration during the launch phase of the mission. Following launch, both latch valves were opened to take advantage of *stabilizing* propellant migration inherent in this configuration.

Tank, line and thruster heaters are thermostatically controlled to maintain the hydrazine propellant comfortably above its freezing point. Thruster catalyst bed heaters are controlled by the BAU. The Flight Operations Team preheats the catalyst bed 30 minutes prior to firing in order to prevent cold-start degradation. The entire RCS weighs only 12 kg without fuel and is approximately 15% of the Probe dry mass.

4.4 Attitude Control Subsystem

The Attitude Control Subsystem (ACS) measures sun pulses and vehicle motions needed to support maneuvers, spin rate control and science data analysis. The ACS components are the Miniature Spinning Sun Sensor (MSSS) and the Inertial Reference Unit (IRU). The MSSS provides the sun elevation once per spin and assists in the calculation of spin rate. Using multiple spin pulses, flight software is able to determine the spin rate. The IRU is a solid-state assembly which measures angular rate of motion of the probe in X and Y axes. While these devices provide probe-relative data, the near Earth FGM data are used to verify probe absolute attitude once per orbit.

ACS telemetry is transmitted to the ground where it is processed into physical coordinates. If maneuvers are required, ground systems calculate the necessary commands to be sent to the probe, and these commands are verified on a ground-based avionics simulator prior to application in space. The ACS Bus components together weigh only 0.6 kg. Figures 9 and 10 show the Miniature Sun Sensor and IRU assembly.

Probes are stable spinners by design, as long as the radial booms are deployed before the axial booms. Even if severely perturbed, probes naturally return to their spin stable position without intervention.

The ACS design depends completely upon both the spin stability of the probes throughout all mission phases and the knowledge of the FGM sensor with respect to the probe body. To achieve spin stabilization, the probes are configured to have their center of mass closely aligned to the geometric axis. This alignment is accomplished through painstaking placement of components and by adjusting balance masses prior to launch. Careful design and measurements of the FGM boom, its repeatability and stiffness in thermal extremes were essential in providing accurate attitude knowledge to mission operations.

4.5 Power Subsystem

The Power subsystem is designed to provide all of the necessary power for the bus and instrument subsystems for the life of the mission in both sunlight and during eclipse. The power system is a Direct Energy Transfer (DET) system with the battery and solar array connected directly to the power bus. The solar array power control and battery charging are performed using linear and sequential switching shunts.

Each probe has eight solar arrays that provide power generation in any orientation. There are two arrays mounted on the bottom and top decks and there are four side panels. The arrays use high efficiency cells that are bonded to the composite substrates. The side panels are also primary structure that adds to their design complexity since they have to transfer loads between the top and bottom decks. In order to reduce surface charging, all the cover glass is electrically grounded to a common ground on each panel. This is accomplished by bonding a highly conductive grid onto the panels following cell placement.

The probe is highly efficient in power usage with approximately 36.85 watts required in full science mode for a 24-hour orbit, which includes a 3-hour eclipse and a 30-minute transmitter turn-on. The capability for that orbit at the mission End of Life (EOL) is 40.35 W. The top and bottom panels are intended to provide approximately 20 W at EOL, which is sufficient power to enable the probe bus components to survive anomalous attitudes in a low power condition.

High efficiency triple-junction Gallium Arsenide solar cells are used. The cells are approximately 4×6 cm, and they have an average BOL efficiency of 27% at room temperature. Each side panel has four strings and each top and bottom panel has two strings. Each string consists of 20 series cells with integral bypass diodes. Strings are carefully arranged on the panels to cancel the magnetic field generated by cells. Cover glasses are 8 mm thick with UV reflective coating. The cover glasses also have ITO coating to provide electrical conductivity and electrostatic cleanliness. The cover glasses are inter-connected by conductive epoxy to provide a bleed-path for surface charges to chassis.

Power is stored onboard by a lithium-ion battery that maintains full probe power during eclipse. The battery is lightweight and has a 12.0 A h capacity. The major power subsystem components weigh approximately 10.3 kg and represent approximately 13% of the total Probe bus dry mass.

4.6 Communication Subsystem

The Communication subsystem consists of an S-Band transponder, diplexer and circularly polarized antenna mounted to the center boom structure.

The antenna consists of six receiver/transmit stack patch antennas and a power divider. These antennas are extremely lightweight and must have a conductive surface in order not to

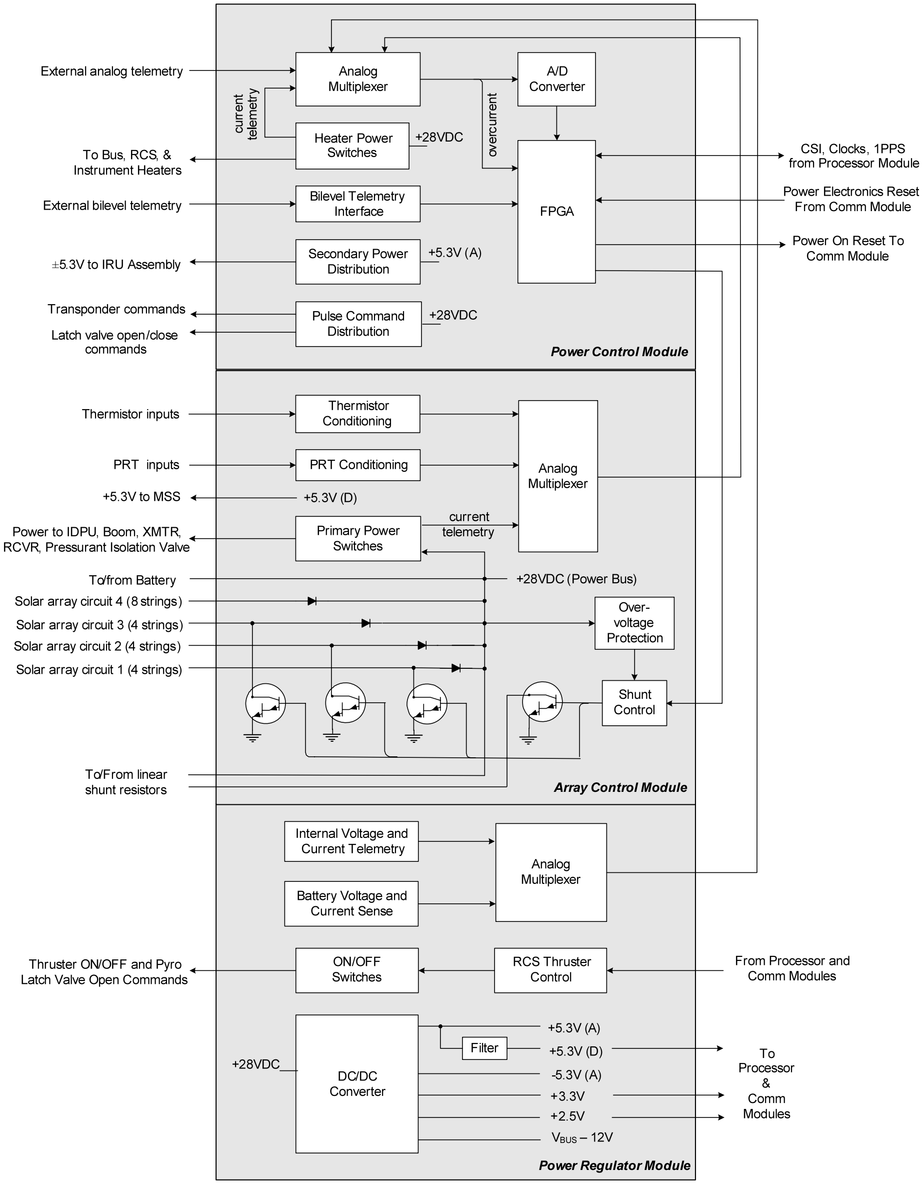

Fig. 11 Electrical power system block diagram

build up surface charge. For high data rate communication, they provide increased gain in a ±45° band about the plane perpendicular to the spin axis. Although reliable communication is possible outside this region, there is a small null region about the antenna boresight, and if the probe orientation is such that the line-of-sight to ground falls within this region, communication is not possible. However, since the probe is inertially pointed, communication outages of this kind would last for only small fraction of an orbit.

Fig. 12 S-band transponder

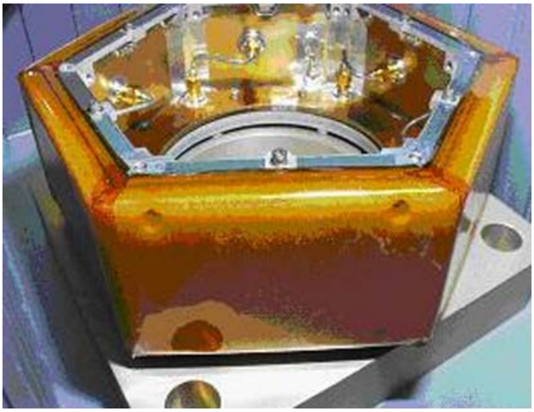

Fig. 13 S-band antenna

The antenna is connected to the transponder via the diplexer. The CXS-610 transponder contains a receiver and a 5 W transmitter. The antenna is always coupled to the receiver with no switches in the receive path, and the receiver is always powered. It demodulates command signals and outputs both data and timing to the BAU. For telemetry, the BAU provides baseband signals to the transmitter which phase modulates them onto the carrier. The transponder can be operated in a coherent mode that provides turn-around ranging capability.

Robust link margins exist for the uplink, even for the case of Probe 1 at apogee. For the downlink, multiple telemetry rates ranging from 1 kbps to 1024 kbps are provided. The total mass of the communication subsystem is 3.2 kg and represents 4% of the Probe dry mass. Figure 12 shows the transponder. The S-Band flight antenna is shown in Fig. 13.

4.7 Avionics

The Bus Avionics Unit (BAU) provides numerous functions for the probe bus and contains the flight computer. The BAU provides the communication interface, instrument electrical interface, data processing and power control for the probe bus. It contains a set of stacked modules with a total weight of 3.0 kg and average power of 7 watts. The electrical block diagram is shown in Fig. 14 and a photo is provided in Fig. 15.

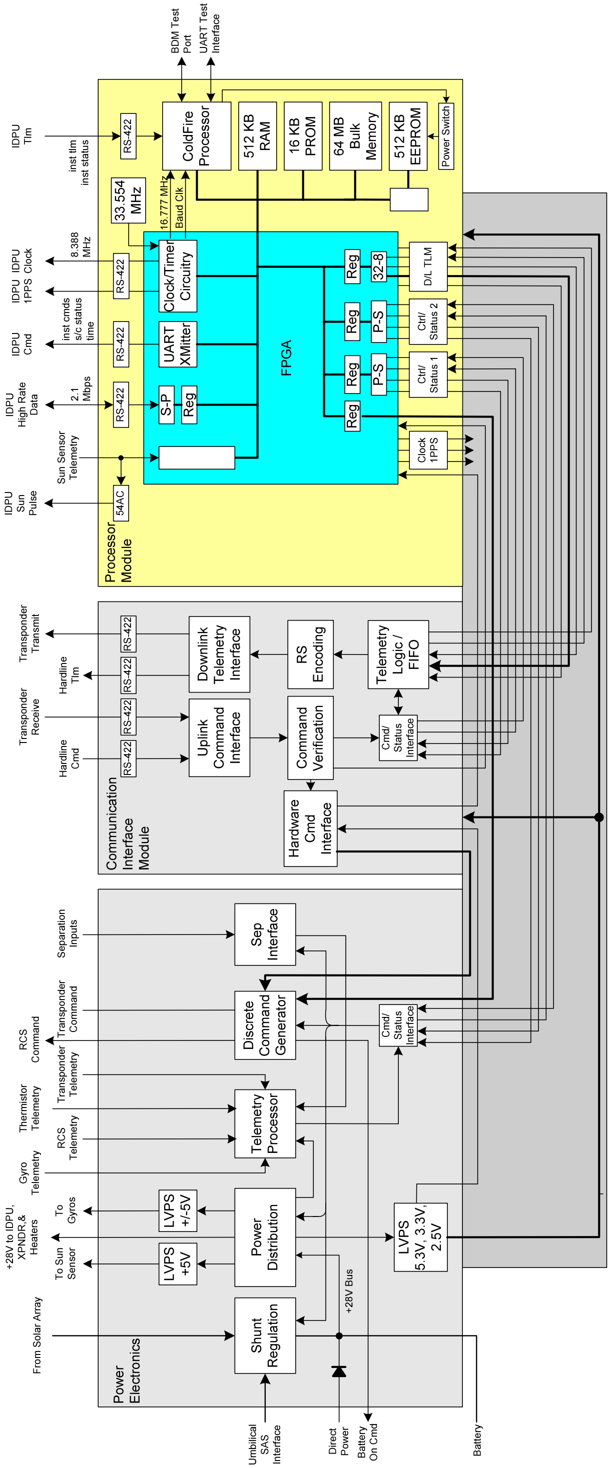

Fig. 14 BAU electrical block diagram

Fig. 15 Flight BAU

The Data Processor Module (DPM) contains a radiation-hardened computer featuring a Cold Fire processor operating at 16 MHz. This module performs all the onboard processing and data handling using 64 MB of bulk memory and a 2.1 Mbps data interface with the instrument electronics. The BAU hosts the RTEMS real-time operating system and the application control and data handling software for the probe bus. Instrument and bus housekeeping data is stored in the local bus memory with science data stored in the IDPU. During a ground pass the housekeeping data is transmitted directly by the processor, while science data is copied out of the IDPU to the transmitter.

The Communications Interface Module (CIM) receives the command bit stream from the receiver and provides CCSDS blocks to the processor. The card also processes a limited number of hardware commands that may be received from the ground and executed without the intervention of the processor. The card provides downlink telemetry data to the transmitter for transmission to the ground. The data may be real-time engineering, playback engineering data, and playback science data from the IDPU. Multiple telemetry rates are provided, ranging from 1 kbps to 1024 kbps. All data are encoded with rate 1/2 convolutional and Reed-Solomon encoding. The communications card also provides a hardline telemetry data stream for ground testing.

The Power Control Electronics (PCE) has 3 modules needed to control the solar array shunts, regulate battery charge, generate and distribute secondary voltages, generate and distribute discrete commands, and monitor separation signals from the launch vehicle and initiate probe separation from the probe carrier. The PCE also contains circuitry needed to condition and digitize most of the analog signals on the probe including IRU rate signals and temperature sensors.

The battery is charged at a fixed rate until the battery voltage reaches a commandable limit, at which point the charge current switches to trickle charge. The upper voltage limit can be selected conservatively so that no cell balancing is required. The BAU has the ability to autonomously remove power from the IDPU in case of over-current or battery under-voltage.

The BAU receives uplink commands at a fixed rate of 1000 bps. Commands are received using CCSDS protocols that guarantee correct, in-sequence delivery of variable-length command packets. All command transfer frames undergo several authentication checks. Invalid frames are rejected and the rejection is reported in telemetry. Commands sent to the probe will either be executed in real time or stored for later execution. Two kinds of stored com-

Fig. 16 Probe carrier ready for probes

mands are provided: Absolute Time Sequence (ATS) commands and Relative Time Sequence (RTS) commands. ATS commands have time tags expressed in UTC times, with a resolution of 1 second, specifying an absolute time of day. RTSs are command sequences that include relative delays between commands.

Since THEMIS orbits have long periods between contacts as well as radiation belt exposures, the BAU provides 64 MB of engineering data storage with error detection and correction (EDAC). Playback data stored in bulk memory is formatted into multiple segments, called virtual recorders, which allows for easy segregation into different types of data such as bus engineering data, instrument engineering data, event files, etc. The size of the virtual recorders is adjustable, allowing the memory to be remapped in the event of failed locations. The integrity of the data stored in bulk memory is maintained by memory scrubbing software, which uses the EDAC to correct single bit errors. Operating at a low priority, the memory scrub task cycles through all the data stored in bulk memory once per orbit.

The BAU maintains a precision onboard clock and distributes time to the IDPU in Universal Time Code (UTC). Time synchronization between the bus and the payload is achieved by the use of synchronous 8 MHz and 1 Hz clock signals sent to the IDPU. The IDPU uses the 8 MHz to collect science data. Once per second, the BAU sends the IDPU the UTC that will be valid at the next 1-Hz pulse. Together these actions allow the bus and instrument to properly time-tag all science and engineering data.

The BAU provides several autonomous functions that insure the health and safety of the probe while out of ground contact. A watchdog timer is provided which continuously monitors processor operations, and should a processor malfunction be detected, restarts the processor automatically. A checksum routine runs at low priority, checking static areas of memory. A telemetry and statistics monitoring function is provided which performs "limit check" operations on the data and which maintains telemetry statistics. If pre-specified conditions occur, it can initiate the execution of a stored command sequence.

The BAU utilizes system tables to implement operational controls and to ease ground system operations. System table operations constitute the primary ground interface for probe control functions such as stored command operations and modifications of on-board parameters. The BAU also has the ability to build "memory dwell" packets to monitor any memory location for diagnostic support.

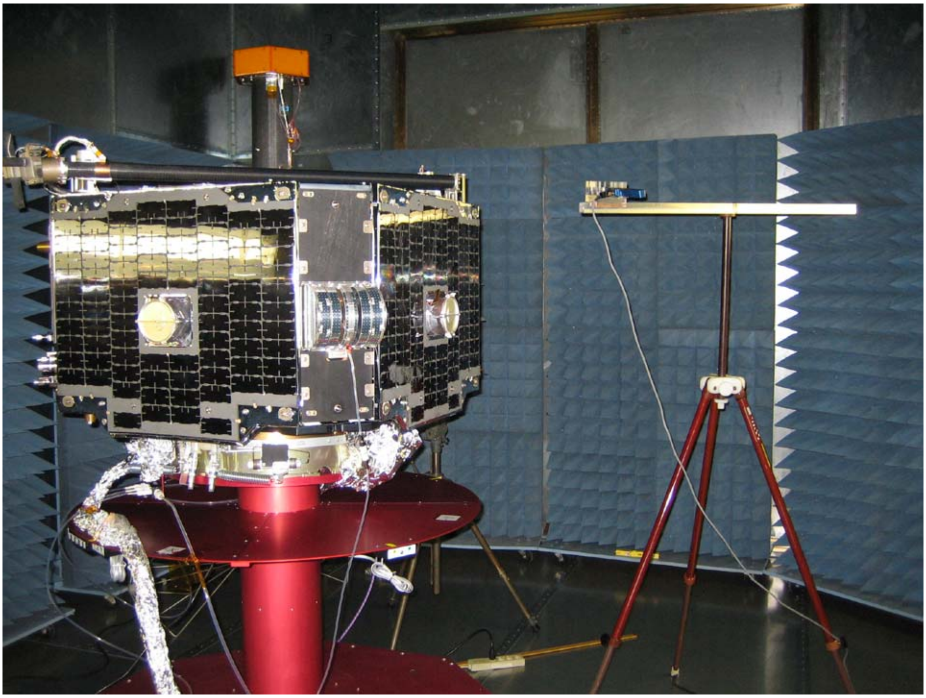

Fig. 17 Probe F2 in EMC

4.8 Probe Carrier and Separation System

The THEMIS Probe Carrier and Separation Systems met the challenge to launch all five probes on a single Delta II, modified to accommodate access to five payloads and provide five redundant separation signals. The payload challenges included imparting an initial stabilizing spin rate to each probe, maintaining a positive separation between probes and doing so even if one probe failed to separate.

The launch vehicle design had the Probe Carrier bolted to the third stage of the Delta II. At the end of the launch sequence, the third stage de-spun to 16 RPM and initiated separation pyrotechnics. As designed, the top probe deployed first and the lower four probes deployed simultaneously three seconds later. Launch vehicle analog telemetry confirmed the correct release profile and probe telemetry confirmed expected probe motions. Initial spin rates for all five were between 16 and 17 RPM while sun angles were within 6 degrees of each other.

There were a number of significant challenges in the separation system design. First, since the Probe Carrier was spinning at release, the four lower separation systems had to operate reliably with a side load. Second, in order to avoid collisions between probes and/or the carrier, the separation system had to move the probe quickly away from the separation plane while imparting a low tip off rate.

The Probe Carrier is predominately aluminum alloy, is weight optimized, and includes a patch panel that manifolds all of the umbilical electrical and control circuit cabling from the probes to the launch vehicle. The separation system was extensively analyzed and tested to properly characterize its performance and to verify all of the mechanical parameters

Fig. 18 Probe F2 in magnetics

that drive the overall Probe and Probe Carrier system clearance verification analysis. Figure 16 shows the Probe Carrier at Astrotech Space Organization (ASO) launch site payload processing facility.

5 Integration and Test

5.1 Probe Integration at UCB

Integration of the Probe buses and Instrument Suites was performed at UCB/SSL in a class 10000 clean room. Each instrument suite, complete with flight harnessing, was rolled up to a Probe and electrically connected via extension cable. Following interface verification, all instrument components were mechanically integrated and alignments verified. Due to the proximity of the Berkeley Ground Station (BGS) and Mission Operations Center (MOC), end to end verification of RF communications with BGS and MOC were verified at this point.

5.2 Environmental Verification Testing

Environmental testing of the THEMIS probes was conducted at the Jet Propulsion Laboratory (JPL) in Pasadena, California, in two stages: first with a single "pathfinder" Probe F2 and later with all five Probes. While other probe buses and instrument suites were still in subsystem test, F2 proceeded to JPL and through environmental testing. The purpose was to uncover problems before fully integrating the other Probes. See Figs. 17–19.

Fig. 19 Probe F2 in vibration

Pathfinder Environmental Tests. The Pathfinder schedule and test sequence is shown in Table 6. The test team included staff from UCB, ATK Space and JPL. Daily teleconferences were held to coordinate staff and plans, and to discuss and resolves questions and issues as they arose. During the pathfinder test, we verified the mechanical ground support equipment used for lifting, rotating and manipulating the probe. We also tested and verified the electrical ground support equipment used to monitor and command the probe.

The magnetics survey took place late at night to avoid interference from vehicles passing outside the building. Three axis magnetic field measurements were taken: (1) all instruments off; (2) all instruments turned on; and (3) after 15 gauss de-perm. The Probe had measurements taken in both vertical and horizontal positions, with several rotations in each position. Measurements showed the Probe to be well within the specified requirement of 5 nT at 2.5 meters from the center.

Full Payload Environmental Tests. While the pathfinder was in environmental test at JPL the other four probe buses and instrument suites were in assembly and test at ATK Space and UCB. The final buses were delivered in May and June, enabling completion of Probes 3, 4, 1 and 5, a combined Pre-Environmental Review and return to JPL by the end of June.

Table 7 summarizes the activities of the five Probes and carrier in environmental testing. After arriving at JPL, the Probe Carrier was tested with the probe mass dummies, proving

Table 6 Probe F2 environmental test sequence

DATE	ACTIVITY
Mar 20–21	Arrival, offload and set up
Mar 22	Magnetics testing
Mar 23–24	Move to vibration facility and set up
Mar 27–30	Vibration tests X, Y & Z axes
Mar 31	Separation shock test
Apr 3	Move to EMC facility and set up
Apr 4–6	EMC tests
Apr 7	Move to thermal vacuum facility and set up
Apr 10–11	Thermal vacuum closeouts and blanketing
Apr 12	Install probe in TV chamber
Apr 13–16	Thermal balance
April 17–21	Thermal vacuum cycles (4×)
Apr 24	Move to magnetics facility, final mag test
Apr 25	Transport to UCB/SSL

Table 7 Full payload environmental test sequence

Week	Major Tests and Activities	
July 10	Post-ship probe functional Tests	
	Probe tank pressurization	
	PCA integration	
	PCA X and Y axis Vibration	
July 17	PCA Z axis vibration	
	PCA acoustics test	
	Probe separation-shock (5×)	
	Probe tank depressurization	
July 24	F3, F4 thermal vacuum set up	F1, F5 magnetics survey
	F3, F4 thermal balance	F1, F5 spin balance
	F3, F4 thermal vacuum cycles (4×)	F1, F5 alignment measurements
July 31	F2, F3, F4 magnetics survey	F1, F5 thermal vacuum set up
		F1, F5 thermal balance
Aug 7	F2, F3, F4 spin balance	F1, F5 thermal vacuum cycles (4×)
	F2, F3, F4 alignment measurements	F1, F5 move to bldg 179
Aug 14	F2, F3, F4 functional tests	F1, F5 spin balance (repeat)
		F1, F5 functional tests

that the carrier was capable of sustaining the worst-case masses of the fueled probes. After probe post-ship functional tests, the Probes were integrated to the Carrier while still on the vibration table (see Fig. 20). Environmental testing of the five probes was completed on schedule with no major anomalies. See Figs. 21 and 22.

Fig. 20 Integrating the probe carrier assembly on vibration table

5.3 Launch Site Activities

Following a four-month delay due to launch vehicle issues, the project received the green light to proceed to launch in December. Initial pre-launch activities took place at Astrotech Space Organization (ASO) located very near KSC. The flow of activities is shown in the chart below (Fig. 23).

As illustrated in Figs. 24 to 28, important activities were as follows:

- The five probes and carrier arrived at Astrotech on December 11, 2006.
- Functional tests, pressurization tests, and bolt cutter installation were carried out in the Payload Processing Facility (ASO1);
- The Probe Carrier was electrically checked in a "side-by-side" test with the 3rd stage, and integrated with its Separation System pyrotechnic lines in the Hazardous Processing Facility (ASO2);
- Following the holiday break, Probes were moved to ASO2. Probes were weighed individually, fueled all at once, then re-weighed individually;
- The probes were integrated to the PCA, electrically checked and spin-balanced;
- On January 29, the PCA was bagged, lifted and mated with the Boeing 3rd stage;
- On February 3, the containerized 3rd stage and payload were transported to the pad;
- At the pad, the THEMIS team tested and charged probe batteries and practiced launch sequences while the Boeing Delta II rocket was being prepared for launch;
- After delays due to lightning and high altitude winds, THEMIS was launched on February 17, 2007.

Fig. 21 Probe carrier assembly ready for Z-axis vibration

6 Lessons Learned

The development of the Constellation proved demanding on several levels. The following are conclusions from the experience:

- THEMIS validated the effectiveness of the "pathfinder" approach, and showed dramatic improvement in performance, schedule, and cost of subsequent units;
- Keeping the Probes identical, despite the fact that the mission required the Probes to be in different orbits, greatly benefited the test sequence;
- Using the same technician for similar tasks across all Probes proved effective in maintaining similar Probe performance;
- Requiring the instruments be designed with internal test features limited the need for drag-on test equipment through I&T;
- Vibrating the full PCA rather than individual probes (or pairs of probes) was a significant improvement. It made the test more flight like, saved schedule and provided useful experience with the full PCA before heading to the launch site;
- Following the Pathfinder Probe#2, performing the thermal vacuum test of the subsequent probes in sets of two worked well and saved considerable schedule time and costs;

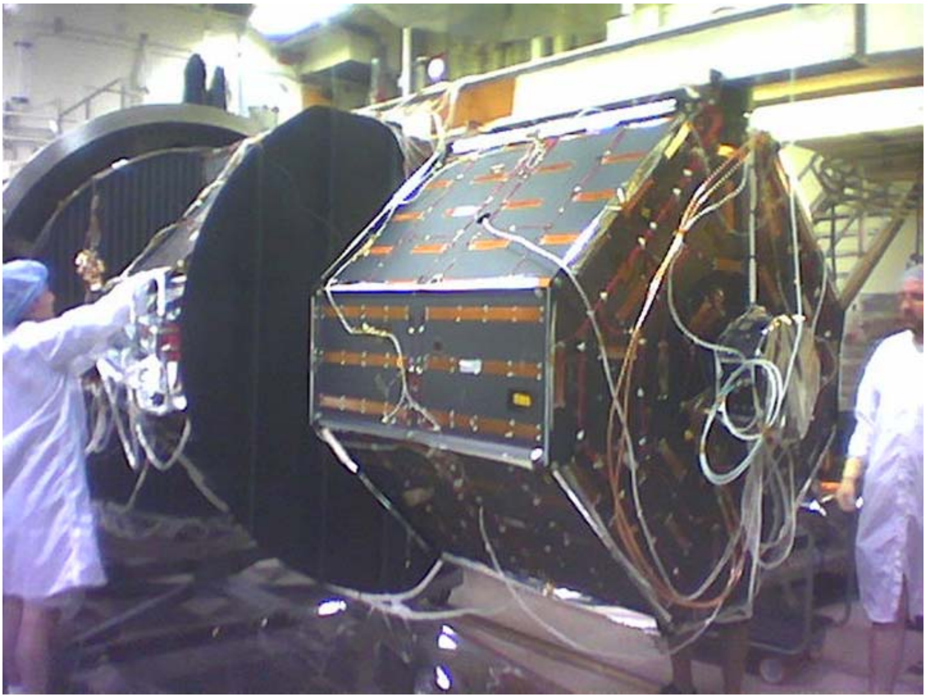

Fig. 22 Probes 3 and 4 (enclosed) at thermal vacuum

- Enclosing the probes within individually controlled thermal enclosures added substantial work and complexity. Despite this, the conclusion of the thermal engineers was that it resulted in a significantly better test and resultant thermal modifications.
- Environmentally testing the pathfinder was very useful and resulted in numerous improvements, especially in thermal design. Thermal blanket modifications to the transmitter radiator were suggested by the pathfinder tests. These modifications significantly improved transmit durations and were verified in subsequent thermal vacuum testing on other probes.
- Probe spin balance resulted in all five being balanced within specification with approximately 1.6 kg total balance mass. Substantial benefits were realized with the number of probes. The balancing of the first two probes was time consuming and initially indicated need for substantially more balance mass than expected (over 3 kg). Corrections made with the later probes showed that balance could be improved and the balance mass reduced. The two probes that were the first to be balanced were then re-balanced. The five probes were shown to have very high degree of uniformity and almost identical balance mass.
- The four sets of electrical GSE proved to be uniform and we could operate any Probe from any GSE. This fact greatly facilitated the schedule as the GSE often stayed in one spot while the Probes moved through the facilities and were attached to any GSE test set.

Problems in Integration and Test were concentrated on the first unit and fell off with each pair of subsystems tested. As the instrument suites and Probes were tested, Problem/Failure Reports (PFR) specified all units needing modification. For design errors or common fabrication errors, all units were modified, including units that had not yet been tested. Of the 171

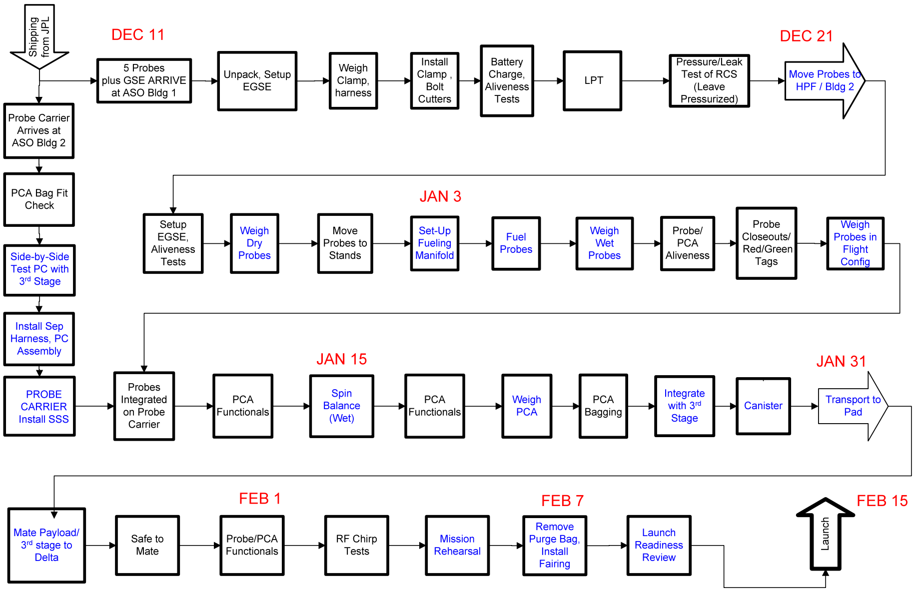

Fig. 23 Payload processing at ASO/KSC

Fig. 24 Fueling probe in Hazardous Operations Facility

Bus and Instrument I&T PFR's, 63 corrected multiple units providing down stream benefits. While modifications proceeded on the current unit, changes of future units occurred in parallel, effectively advancing the schedule of future units. As shown in Table 8, the first unit

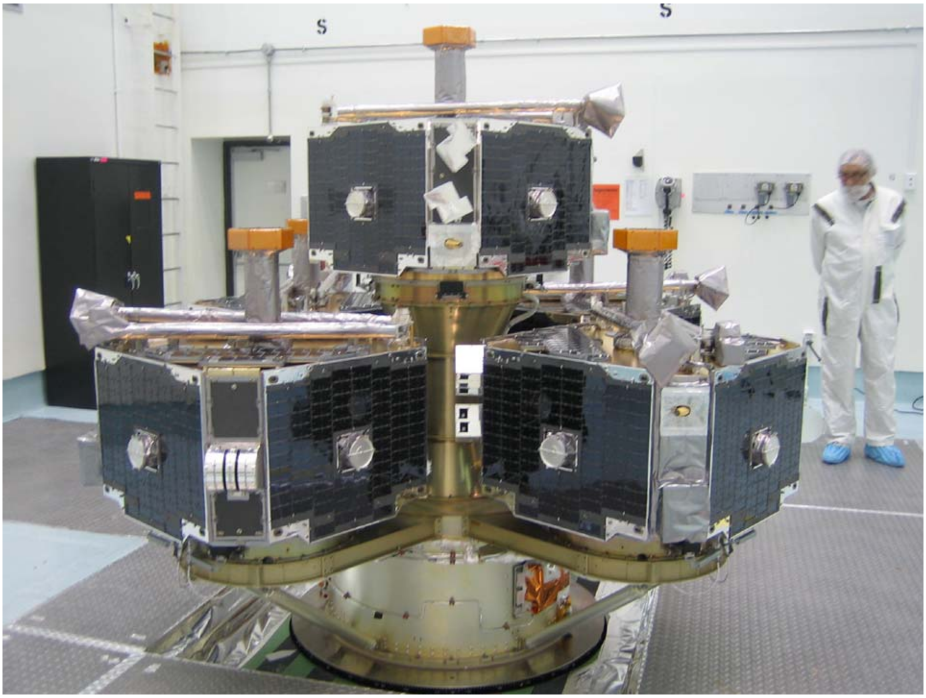

Fig. 25 Probe carrier assembly on spin balance machine

Table 8 Problems encountered at I&T (by unit order)

I&T new PFRs	1st	2nd	3rd	4th	5th	Total
Probe Bus in I&T	35	3	2	1	6	47
Inst Suite in I&T	9	3	3	3	3	21
Inst Suite	48	19	20	9	7	103
Total	92	25	25	13	16	171

found more than half the problems, followed by a rapid drop in problems in later units. Since the last unit was on the critical path, this had a real effect on the mission level schedule.

7 Project Performance

7.1 Technical Performance

The Probes and Carrier performed very well through integration and test, components accumulating an average of 810 hours overall and 250 hours in thermal vacuum conditions. Probes had an average of 350 failure free hours at launch. Of the 476 mission requirements, only 2 requirements were waived. These were due to minor variances on the EFI noise floor and clearances between Probes on the Carrier.

Final Probe dry masses were 4.7% below their 81.8 kg Not-to-Exceed limits and matched to less than 1%. Probes A–E measured 78.0, 77.6, 76.7, 76.7, and 78.1 kg, respectively. The

Fig. 26 PCA mate to the 3^{rd} stage

Probe Carrier Assembly weighed in at 777 kg with a 6.3% margin to the launch vehicle NTE of 829 kg. The final Probe power budget at 38.7 W has a 4.9% margin in surviving a 3-hour eclipse at the end of mission.

7.2 Schedule Performance

The successful scheduling of the THEMIS project required a mixture of optimism, dogged determination and endless coordination between UCB, ATK Space and GSFC management. From project funding to launch took 46 months, including a 4-month launch vehicle delay. The first Probe began integrating instruments at 31 months and all Probes completed all testing 9 months later.

The THEMIS top-level schedule is illustrated in Fig. 29. The master schedule linked together 23 instrument and 10 probe and carrier schedules, together totaling more than 5000 tasks. These were managed by schedulers at UCB and ATK Space and milestone-tracked by GSFC.

The development of instrument suites and probe buses used a 1, 2 and 2 approach to build the five Probes. The Environmental Verification Tests (EVT) used a different approach, verifying the first Probe then all five Probes and Carrier. The latter sequence provided "test-as-you-fly" configurations and shortened the overall project schedule.

Fig. 27 Delta II launch vehicle

All levels of integration, whether instrument or bus or probe, witnessed dramatic schedule improvements with each unit. These were principally due to (1) improved test procedures and (2) reduction in newly discovered problems.

7.3 Cost Performance

The THEMIS budgetary estimate at Confirmation Review was 158.3M (FY02) assuming a payload development and operations of $89.3M and a launch vehicle at $69M. Actual payload costs ended 4% high at $92.9M. Actual costs of the launch vehicle and the launch delay brought the total cost to $172.8M, yet still beneath the MIDEX cap of $180M. See Table 9.

The project experienced dramatic cost reductions as the units were fabricated. The design phase through CDR cost $20M, roughly one quarter the total, while costs of the first

Fig. 28 Launch, February 17, 2007

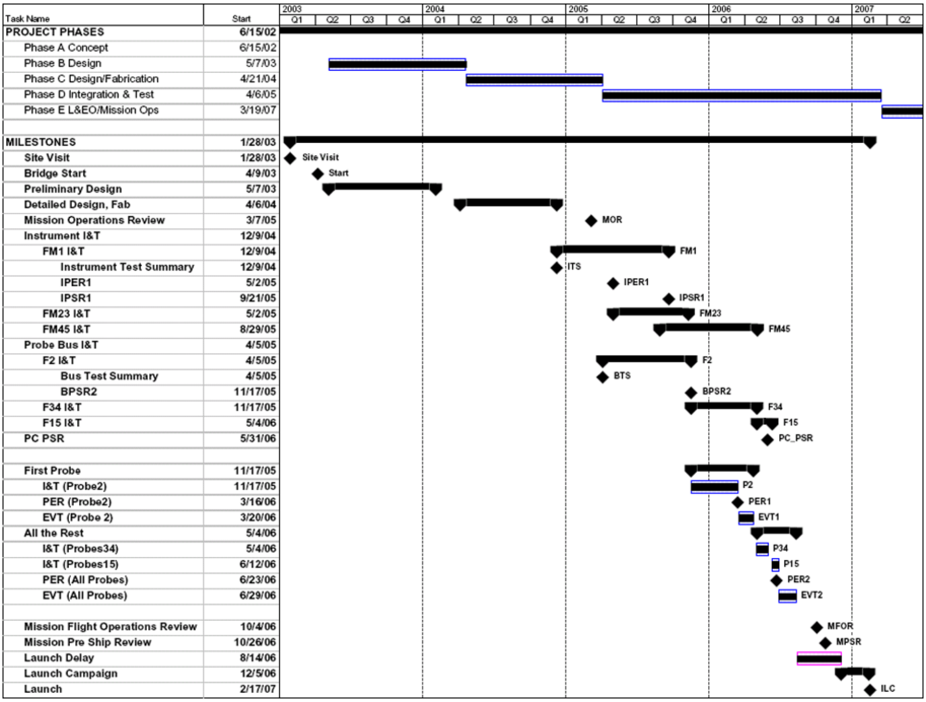

Fig. 29 THEMIS constellation schedule

Table 9 THEMIS cost in FY02 M$

THEMIS budget performance FY02 $M	Phase A–D	Phase E	Total
UCB/SAI Probe & carrier development	80.3	10.3	90.6
JPL environments and simulations	1.1	0.0	1.1
GSFC thermal and data analysis	0.4	0.9	1.2
Payload (w/o launch delay)	81.7	11.1	92.9
Launch vehicle (AO)	69.0		69.0
Launch vehicle additional costs	4.9		4.9
UCB/SAI impact of launch delay	3.1	3.0	6.0
Total	158.7	14.1	172.8

flight- Probe were another $20M. Remarkably, the four remaining Probes were completed at roughly half the cost of the first.

Acknowledgements The THEMIS mission is that rare combination of inspiration and imagination that challenges scientists, engineers, and managers alike. Without question, the success of the project is due to the indefatigable efforts and contagious optimism of the PI Vassilis Angelopoulos, who not only convinced all of us that it could be done, but that *we could do it*.

Dr. D. Pankow provided vehicle dynamics and flawlessly led the vibroacoustics and balance efforts. Dr. Auslander and UCB graduate students simulated vehicle dynamic behavior due to fuel slosh and wire booms. D. Curtis and S. Harris carefully verified each bus as C. Chen, H. Richard and M. Ludlam did the same for the instrument suites. Dr. M. Sholl verified the propulsion system throughout integration and led probe fueling, and M. Leeds provided RCS training and support. Dr. M. Bester verified communications with BGS and the Mission Ops Center.

Deputy Project Manager D. King and Project Scheduler D. Meilhan tracked an unbelievable number of tasks and kept it all under control as Financial Manager K. Harps kept costs in line. Mission Assurance Manager R. Jackson and quality personnel J. Fisher and C. Scholz managed to get all the parts, inspect all the components and track all the problems to closure.

ATK Space system engineers K. Brenneman and W. Chen, propulsion designer M. McCullough, separation system designer D. Jarosz and thermal engineer R. Zara were instrumental in the success of the probes and carrier. The Hammers company for excellent support of Bus flight software. ATK Space Vice President F. Hornbuckle committed the company to the project at a time when it was probably unpopular to do so.

Mission Manager F. Snow demonstrated great patience and support particularly while the project went through difficult times. The Explorers team of D. Lee, R. Miller, J. Thurber and D. Gates provided support where needed in government services, scheduling, RF expertise and communications. Finally, IIRT co-chairman Mark Goans and Brian Keegan provided valuable reviews, personal counsel and much-appreciated confidence in the THEMIS design and implementation.

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