APPENDIX B

The NASA-Provided Medium Explorer Spacecraft

1. OVERVIEW

The intent of this Appendix is to define an envelope of NASA-provided spacecraft consistent with the allocation of budgets provided in this Announcement of Opportunity. The information provided is intended to simplify the task of the proposal team in developing a spacecraft concept for use in their proposal and to ensure that the proposed investigation lies within the bounds of the MIDEX program. The capabilities here apply to NASA-provided spacecraft designed, built, integrated, and tested at GSFC as well as those procured by NASA. It does not apply to spacecraft provided by the PI in the PI-Mode.

The detailed information presented here is based heavily on the component heritage developed in the Small Explorer (SMEX) program. It is intended as a common reference basis for the proposers and evaluators. The proposed spacecraft concept should be used to demonstrate mission feasibility, but should not be regarded as a final design. Should the proposer desire, it is acceptable to substitute a different set of components such as those in an existing industry spacecraft. In order to establish the credibility of a concept derived in this manner, the heritage and reference basis of the concept must be described in sufficient detail to allow GSFC to determine the probable cost of (and the cost risk associated with) the proposed spacecraft concept.

The proposal requirement is to determine the basic spacecraft concept and overall system architecture that meet the scientific requirements and to show mission feasibility. This is accomplished by choosing the spacecraft resources from the data in Section 3 (or a suitably documented other source), performing a conceptual design and addressing the technical information requested in Appendix F. **Requests for enhanced performance beyond the spacecraft baseline can usually be accommodated but must be included in the cost of the instrument (capped at $40M) for proposal purposes.** During mission definition, system trades will be performed to determine the most cost effective means of implementing the selected investigation. This may result in a reallocation of funding between the instrument and spacecraft.

It should be recognized that a spacecraft concept generated with either the NASA in-house component set or that from industry does not represent a decision binding on NASA to implement the mission in that manner. GSFC will determine the spacecraft development mode within one month after selection.

Experience has shown that launch-vehicle constraints, fiscal constraints, and the wide variety of science objectives make a common multimission spacecraft impractical. Structures and attitude control systems are designed for the unique needs of each mission. Major subsystem elements, however, can be taken from an inventory of spacecraft subsystem designs that have been developed for other missions. These items include standard batteries, transponders, power systems, data handling systems, and instrument interfaces.

The baseline MIDEX spacecraft system architecture, described in Section 2 below, is versatile and high-performing with heritage from the Submillimeter Wave Astronomy Satellite (SWAS) mission, which is part of the Small Explorer (SMEX) program. The architecture is single-string and nonredundant with some degraded modes of operation possible due to the versatility and robustness of the system design. It is the intent of GSFC to conform to industry standards whenever possible in the development of MIDEX spacecraft.

Through the SMEX program, GSFC, in partnership with the investigator teams, has developed or is in the process of developing a variety of missions using the key elements of the baseline architecture. These missions represent the present thrust and hands-on knowledge base of the Center's small spacecraft missions. The Solar, Anomalous, and Magnetospheric Particle Explorer (SAMPEX, in orbit since 1992), Submillimeter Wave Astronomy Satellite (SWAS, in flight integration and test phase), Transition Region and Coronal Explorer (TRACE, in detailed design phase) and Wide-Field Infrared Explorer (WIRE, in extended definition phase) missions are examples of these. The Fast Auroral Snapshot Explorer (FAST, fully qualified and validated for launch) mission required an alternative architecture driven by the limited power and mass resources available to the spacecraft, and by the harsh radiation environment of the FAST orbit.

Section 7 provides a description of the SWAS and FAST missions, while Table B-3 summarizes key characteristics of all five SMEX missions. This information is provided for reference and to illustrate the flexibility of the baseline spacecraft architecture and associated components.

2. BASELINE MEDIUM EXPLORER SPACECRAFT ARCHITECTURE

The baseline spacecraft architecture is illustrated in Figure B-1. Considered to be radiation tolerant, this spacecraft can be flown in all low Earth orbit environments. All systems use unregulated 28 volt power from the Power System Electronics (PSE). All systems communicate through a MIL-STD 1553 or 1773 data bus.

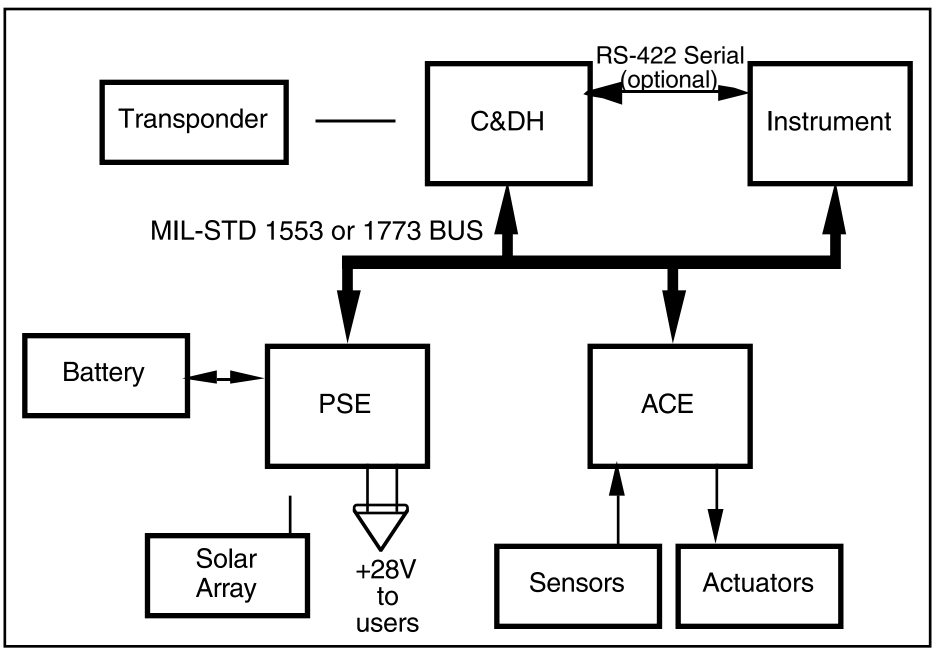

FIGURE B-1 BLOCK DIAGRAM OF THE BASELINE MIDEX SPACECRAFT ARCHITECTURE

A. Command and Data Handling System

The Command and Data Handling (C&DH) system controls all spacecraft operations, supports attitude control system computations, collects and stores telemetry, maintains time, processes commands, initiates deployment, and interfaces to the instrument.

The C&DH is based on a 386 class computer. The C&DH is capable of ingesting ground commands at 2.0 kilobits per second and of outputting telemetry at selectable rates up to 1.8 megabits per second. All telemetry is encoded to reduce errors. After accounting for the overhead of encoding, Consultative Committee on Space Data Systems (CCSDS), ground/space link communications protocol, and the bandwidth allocated to spacecraft telemetry, approximately 1.4 Mbps is available to the instrument.

Bulk data storage is accomplished using volatile, solid state random-access-memory devices with error detection and correction. The bulk memory is uniquely sized for each mission in order to limit size, mass, and cost. The baseline capacity for instrument data storage is approximately 80 Megabytes, limited by what can be downlinked in a ground station pass. Typically, only 60% of the pass duration can be allocated for telemetry downlink.

The C&DH operates with a time-division-multiplexed protocol. Each piece of instrument data read by the C&DH is marked with the time, with a resolution of 4.0 milliseconds prior to storage. More accurate timing resolution may be negotiated on a case by case basis during mission definition. Spacecraft time is maintained to better than 3 seconds absolute time.

B. Transponder

The Medium Explorer spacecraft uses a Ground Network (GN) compatible transponder. It operates in S-band and can support all C&DH data output rates. If required for orbit determination, the transponder can perform coherent Doppler reflection and range-tone broadcast. Tracking and Data Relay Satellite System (TDRSS) compatibility is not included in the baseline. However, mission trade studies will be performed during the definition phase to determine the technical feasibility and cost effectiveness of utilizing TDRSS.

C. Power System

The spacecraft power system design can support orbital average loads up to 240 watts at the end of two years. A NiCd type battery is used to power the spacecraft during the eclipse period of the orbit, with both 9A-Hr and 21A-Hr batteries available. Peak loads are limited to 1/2 of the selected battery rating, i.e., 4.5A for the 9A-Hr battery and 10.5A for the 21A-Hr battery. Unregulated 28V (+/-7) power is distributed to the subsystems. The baseline solar arrays are fixed flat wings with an area of ~6 m^2 and a temperature range of -80 to +90 degrees C. The arrays can deliver up to 240 watts orbital average (EOL) to the bus in a typical low Earth orbit, assuming 2/3 Sun and

1/3 eclipse, when the normal to the plane of the arrays is within 5 degrees of the Sun line. Missions with longer eclipse periods or less favorable Sun angles would achieve a proportionally lower power output from the arrays. The spacecraft power allocation is 175 watts orbital average. Other batteries, alternate solar array configurations and arrays with higher watt densities are available. However, the cost of these enhancements must be included in the instrument budget.

D. Attitude Control and Determination

Attitude control is a mission unique system. The C&DH provides for the hosting of a digital attitude controller, which typically is incorporated for science pointing mode operations. This controller collects sensor data and issues actuator commands through the Attitude Control Electronics (ACE) box. The ACE serves as the hardware interface for the attitude control system (ACS) and is also used to provide a C&DH-independent safehold controller. The safehold controller is used for initial attitude acquisition and on-orbit emergencies. The baseline safehold will safe the spacecraft and maintain a power positive attitude, but will not provide any special instrument protection. Such protection must be accounted for in the instrument budget. The baseline system incorporates onboard attitude determination and orbit propagation. For a three-axis-stabilized platform, the baseline spacecraft is capable of providing sunline pointing accuracy of 2 arc-min per axis and 2 deg accuracy rotation about the sunline for most low Earth orbits. With the addition of attitude sensors beyond the baseline, such as a star tracker and an upgrade to an inertial grade gyro, pointing accuracy of 30 arc-secs, 3 axes is achievable. With the provision of a suitable fine error signal from the instrument, sub arc-second boresight accuracy is achievable. Precision sensors beyond the baseline capability must be costed in the instrument budget. Simpler missions, such as flown on SAMPEX, can be accommodated with a subset of the baseline sensors. ACS performance, including accuracy, stability, and knowledge, depends entirely on the hardware devices that are flown and the specific spacecraft and instrument characteristics. Therefore, ACS sensors and actuators should be chosen to support the specific requirements of each mission. The TRACE mission described in Table B-3 utilized a representative set of baseline sensors and actuators.

E. Thermal

Thermal designs are unique to each mission. Power limitations require primarily passive designs that rely on fixed radiators, blankets, thermostatically controlled heaters, and selective coatings. Instrument thermal control is an instrument function.

The baseline architecture assumes an instrument that is thermally isolated from the spacecraft as much as is practical.

F. Instrument Data Interface and Control

It is strongly recommended that the instrument contain its own controller to allow a decoupled interface with the spacecraft and facilitate testing and integration. The instrument/spacecraft data-system interface is via the MIL-STD 1553 or 1773 bus, which can accept science payload data rates of at least 50 kbps. Higher data rates, up to 900 kbps, can be accommodated using the high-speed RS-422 serial port to the C&DH.

3. RESOURCES OF THE NASA MEDIUM EXPLORER SPACECRAFT SYSTEMS

Proposers should use the spacecraft resources provided in this section to estimate the requirements of the Medium Explorer spacecraft systems as they apply to their specific proposals. Proposers must address each topic covered in paragraphs A through G, below. The data in this section have been derived from experience on previous missions. Proposers are required to discuss the rationale for deviations from these guidelines. Design margin is not included in these numbers. Unless otherwise noted, power is expressed as orbital average load.

	Mass Basis
A. Structural/Mechanical (percentage of maximum observatory mass to orbit)	20%
• This includes the following: primary structure (aluminum, traditional construction), miscellaneous bracketry and fasteners, balance weights (nonspinning spacecraft)	
B. Solar Array and Shunts	
• Solar arrays (includes shunt assemblies and harness)	7.3 kg/m^2
	Mass Basis
• Hinge/damper assemblies (deployed array)	1 kg/m^2
C. Battery	
• 9-amp-hour battery (11 cm x 18 cm x 30 cm)	11.3 kg

• 21-amp-hour battery (13 cm x 23 cm x 38 cm)		23.1 kg
D. Electrical		
• Electrical harness (percentage of observatory dry mass)		6%
• Spacecraft umbilical assemblies		0.45 kg
E. Thermal		
• Typical heater orbital average power consumption (percentage of total spacecraft orbital average power)		5%
• Battery radiator (15 cm x 25 cm)		1.4 kg
• Transponder heat sink		1.1 kg
• Thermal blankets (percentage of spacecraft dry mass)		1.5%
F. Avionics	Mass	Power
• Command and Data Handling — The size, mass, and power consumption of this assembly depends on the mission requirements. The basic box, which can support the ACS requirements of most missions, contains 80 Megabytes of bulk data storage for instrument telemetry. Storage for spacecraft telemetry and software has already been allocated in addition to these numbers. (30 cm x 28 cm x 23 cm)	12.0 kg	13.0 W
• Power System Electronics — Includes shunt drivers, if necessary (32 cm x 26 cm x 24 cm)	14.0 kg	10.0 W
• Transponder-5 W RF output (30 cm x 18 cm x 12 cm)	4.1 kg	
- Standby		4.5 W
- Transmitting		33.0 W
	Mass	Power
• 2 Fixed S-Band omni-directional antennas	1.5 kg	n/a

- Initiator box — This is only flown if initiators are required (i.e., to deploy solar arrays, booms, etc.). Each module, which operates six devices, occupies approximately 410 cm^3 and weighs approximately 0.68 kg.

G. Attitude Control System

The ACS support varies considerably from mission to mission. The following resource estimates are typical for hardware used in the first five SMEX missions.

- Attitude Control Electronics Box
  The basic box interfaces to the MIL-STD 1553 or 1773 bus, provides a hardware interface to the ACS sensors and actuators, drives electromagnetic torquers, and contains safehold circuitry.

- ACE box (28.6 cm x 25.5 cm x 10 cm)	12.0 kg	6.0 W

- Resources required by baseline ACS devices are listed in Table B-1.

4. DESIGN MARGIN REQUIREMENTS

All proposals must contain margins of at least 20% mass and 20% orbital average power for the observatory. Larger mass margins are suggested for instruments that require significant development. The mass margin should be calculated as a fraction of the maximum observatory wet mass to orbit. The power margin should be calculated relative to the EOL (2 year) orbital average power generation capability.

5. INTEGRATION AND TEST AND GROUND SUPPORT EQUIPMENT

Depending on the implementation mode, the science instruments will be mechanically, electrically, and functionally integrated to the spacecraft using GSFC

TABLE B-1 BASELINE ACS DEVICE RESOURCES

Device	Component requirements		
	Size (cm)	Mass (kg)	Orbital Avg Pwr (W)
Reaction Wheel (per wheel)	(dia x ht)		
• 0.14 N-m, 2.80 N-m-s	20 x 10	4.3	10.0
• 0.175 N-m, 20 N-m-s	25.4 x 9	6.5	20.0
• Electronics box for three wheels	3.1 x 16.5 x 16.5	2.3	included in above power
Electromagnetic Torquer (per bar)	(dia x ht)		
• 30 kilopole-cm	2.3 x 49.5	0.73	5.5
• 60 kilopole-cm	2.8 x 66.0	1.8	5.5
• 130 kilopole-cm	2.3 x 85	2.1	3.7
• 170 kilopole-cm	3.9 x 94	5.0	1.42
Gyro (3 axis)	28 x 17 x 17	4.9	24.0
Digital Sun Sensor (1 arc-min accuracy)			
• Sensor	11 x 8.5 x 3	0.32	in elect.
• Electronics	14.5 x 12.5 x 8	1.13	2.0
Coarse Sun Sensor	Negligible	0.05	N/A
3-Axis Magnetometer (0.5 deg accuracy)			
• Sensor	11.5 x 11.5 x 11.5	0.3	in elect.
• Electronics	7.6 x 17.5 x 17.5	1.8	1.7

facilities or contractor provided facilities. Following this and prior to launch, the combined spacecraft/instrument system will be thoroughly tested to verify its performance. Instrument operational, safe-to-mate and test procedures are the responsibility of the investigator. During integration and testing, a local area network (LAN) to the spacecraft integration and test ground support equipment (GSE) will be the interface to the instrument GSE. The instrument GSE must recognize and process instrument data over the LAN using the LAN's communications interface protocol. GSFC will provide a test conductor to control spacecraft operations during integration and testing activities.

6. ORBIT SELECTION GUIDELINES

Orbit selection is driven by many factors including science requirements, orbit lifetime, ground station coverage, and radiation concerns, including sensitivity to trapped radiation in the South Atlantic Anomaly.

A. Ground station coverage

In order to reduce mission operations costs, the spacecraft must operate autonomously for long periods of time. It must store the science data onboard, and quickly transmit the data to the ground station when a station pass occurs. The average pass duration for a low Earth orbit is only 8-10 minutes, depending on orbit altitude and inclination. Typically, only 60% of the pass duration can be allocated for telemetry downlink. High inclination orbits are recommended because the longest period between passes is reduced, minimizing the bulk memory requirements and the required downlink time.

B. Attitude controllability

Missions that require three-axis stabilization cannot adequately manage momentum build-up with only magnetic torquers in near equatorial orbits. To avoid costly gas-reaction control systems, mid- to high-inclination orbits are recommended for missions that require three-axis pointing, if compatible with science objectives.

C. Radiation

Normal design practices, such as those used by GSFC on the spacecraft described in this Appendix, typically result in a minimum effective aluminum shielding thickness of 2.5 mm (0.1 inches), and will, therefore, accommodate the radiation tolerance capabilities outlined in Table B-2. The FAST implementation, in addition to satisfying low weight and power limitations, was designed to operate in a harsher radiation environment than SWAS and the other SMEX missions.

TABLE B-2 RADIATION TOLERANCE

	Dose Capability at the Part-Level (krads Si)	**Orbit Capability (2 yr. mission life)**		
		$\lvert i \rvert < 60$	$\lvert i \rvert > 60$	Other
Baseline Architecture	5	O.K.	Analysis Required*	Analysis Required*
FAST Implementation	30	O.K.	O.K.	Analysis Required*

* Analysis required to consider effective shielding, mission duration, specific orbit environment, solar cycle, etc. Proposers with radiation questions about a specific orbit may use the questionnaire in Figure B-13 to obtain information.

7. SMALL EXPLORER MISSION IMPLEMENTATIONS

The MIDEX spacecraft capability is based on SMEX heritage. The SMEX mission designs have distinct configurations, which were uniquely tailored (particularly the FAST mission) to meet the requirements of the missions. The key characteristics of all five SMEX missions are summarized in Table B-3. The SWAS and FAST spacecraft designs are briefly described in the following paragraphs.

A. SWAS Spacecraft Design

Schematics of the SWAS instrument and spacecraft payload, both assembled and as subsystems, are shown in Figures B-2 and B-3.

The SWAS is a three-axis-stabilized, stellar-pointed spacecraft with a pointing accuracy of 38 arc-sec and jitter less than 19 arc-sec. The spacecraft will, typically, point the science instrument at 3-5 targets per orbit. Target selection is constrained such that the solar arrays always face within ±15 degrees of the Sun, except during eclipse. The Principal Investigator will generate the observation timeline and select the guide stars. The spacecraft will "nod" from an on-source target position to an off-source target position up to three degrees away in less than 15 seconds. This nodding occurs approximately every 40 seconds. Attitude control, including pointing and nodding, are accomplished by using three magnetic-torquer coils, one digital sun sensor, five coarse sun sensors, three momentum wheels, one magnetometer, three inertial gyros, and a high accuracy Charge Coupled Device (CCD) star tracker. The ACS uses the spacecraft computer to perform closed loop attitude determination and control.

TABLE B-3 CHARACTERISTICS OF SMEX MISSIONS

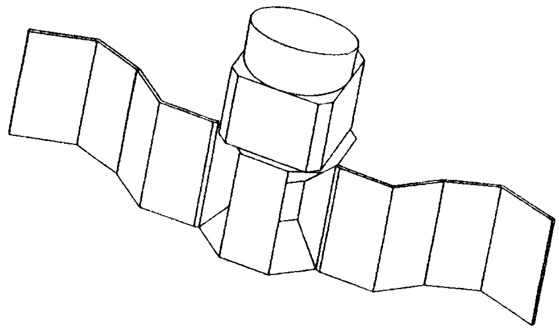

FIGURE B-2 THE SWAS MISSION.

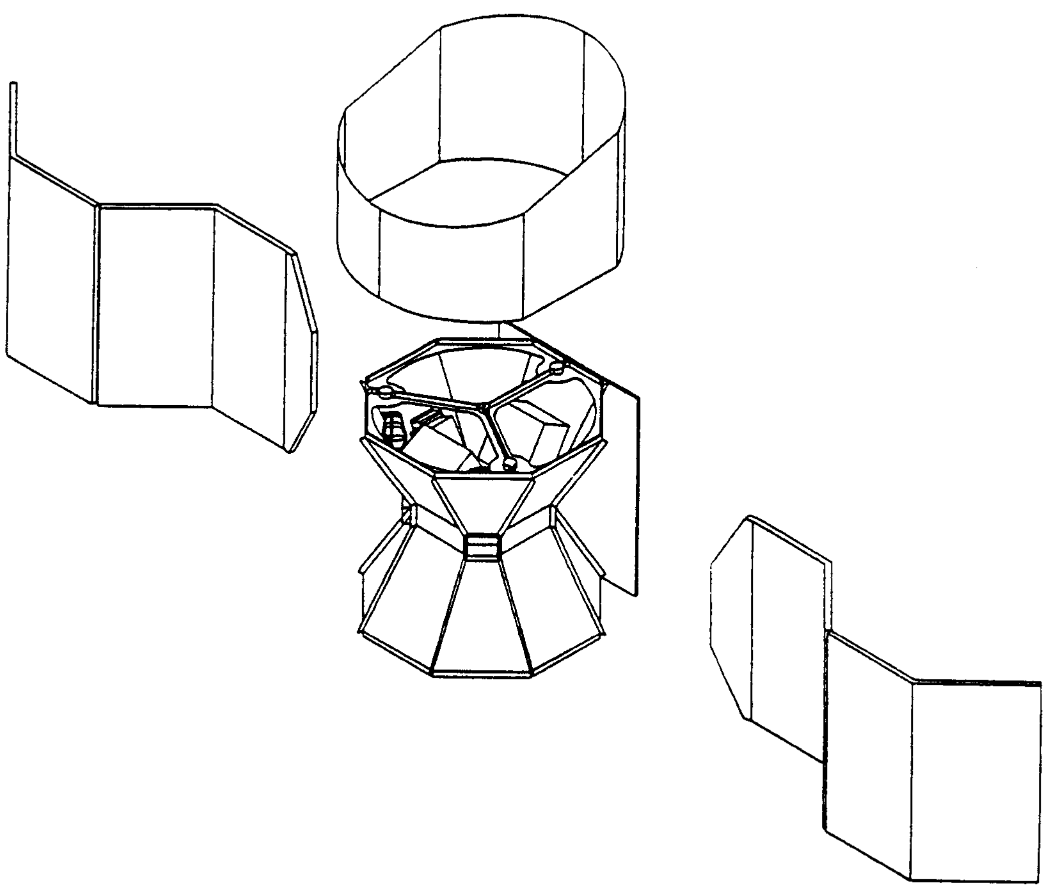

Figure B-3 An exploded view of the SWAS instrument and spacecraft.

Four deployable, nonarticulated solar panels and one body-mounted panel contain gallium arsenide solar cells totaling 3.4 square meters and provide 230 watts of orbital average power that is distributed to the spacecraft and instrument via an unregulated 28 volt bus. A 21 amp-hour nickel-cadmium battery is included in the spacecraft to provide power during eclipse and when load demands exceed the solar array power output. The orbital average power consumption of the spacecraft hardware is 133 watts. The instrument consumes 59 watts.

Temperatures of the spacecraft subsystems are maintained passively between -15 Celsius and +50 Celsius by using surface finishes, regulated conduction paths, blankets, and thermostatically controlled heaters.

The C&DH system for the SWAS mission includes dual 32-bit processors and 70 megabytes of bulk memory. All subsystems and the instrument interface to the C&DH via a MIL-STD 1553 data bus. Instrument data are collected at approximately 12 kbps average rate. A NASA standard S-band transponder and a five-element, micro-patch antenna completes the C&DH system. The stored data are transferred to the ground at 1.8 Mbps data rate. Commands are uplinked at 2 kbps transmission rate.

The instrument weighs 71 kg and attaches to the top of the spacecraft structure as a single module. The total observatory mass is 252 kg. The SWAS mission is to be operated in a 600 km circular orbit with a 65 degree inclination.

B. FAST Spacecraft Design

The FAST spacecraft is a 12 rpm, spin stabilized spacecraft with its spin axis oriented parallel to the orbit axis. The instrument electronics include a 32-bit data processing unit which performs the science data processing and recording in a one-gigabit, solid-state memory. The stored data are transferred to the ground at one of three selectable high data rates of 900 kbps, 1.5 Mbps, and, for certain ground stations, 2.25 Mbps. The instruments weigh 51 kg; the total observatory mass is 180 kg. The FAST mission requires 1 year of operation in a 350 X 4,200 km, 83 degree inclination orbit. Schematics of the FAST instrument and spacecraft payload, both assembled and as subsystems, are shown in Figures B-4 and B-5. A block diagram of the spacecraft architecture is shown below in Figure B-6.

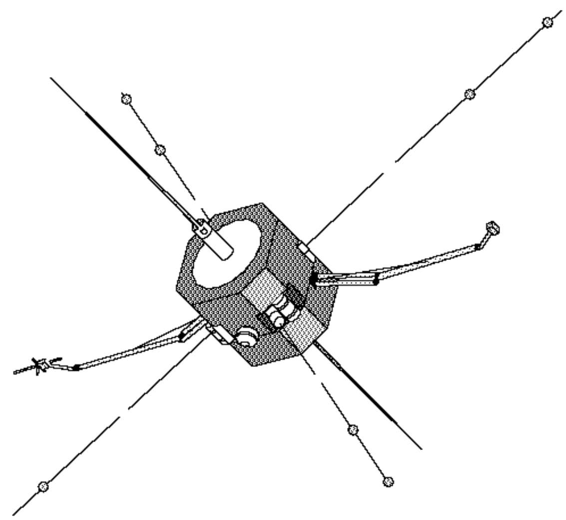

FIGURE B-4 THE FAST MISSION.

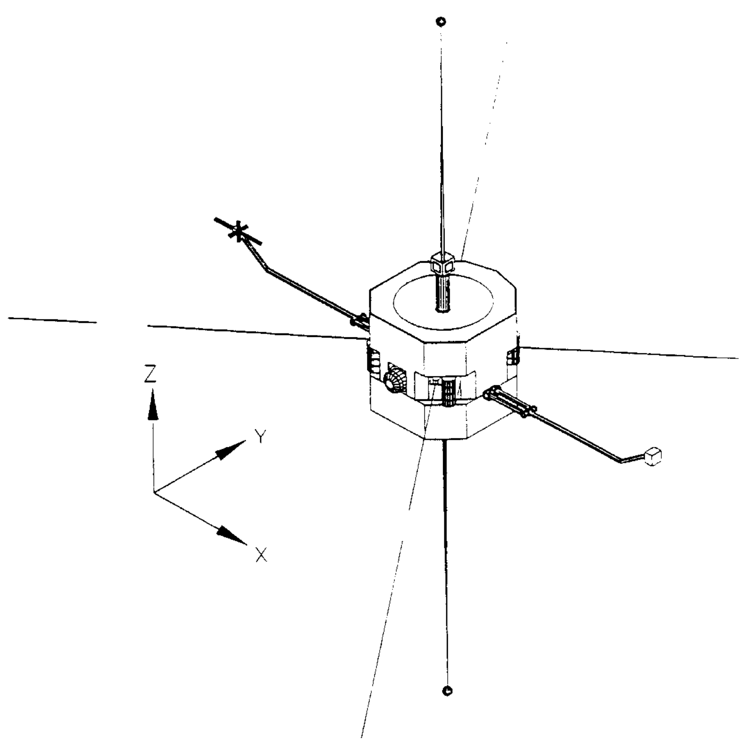

FIGURE B-5 AN EXPLODED VIEW OF THE FAST INSTRUMENT AND SPACECRAFT.

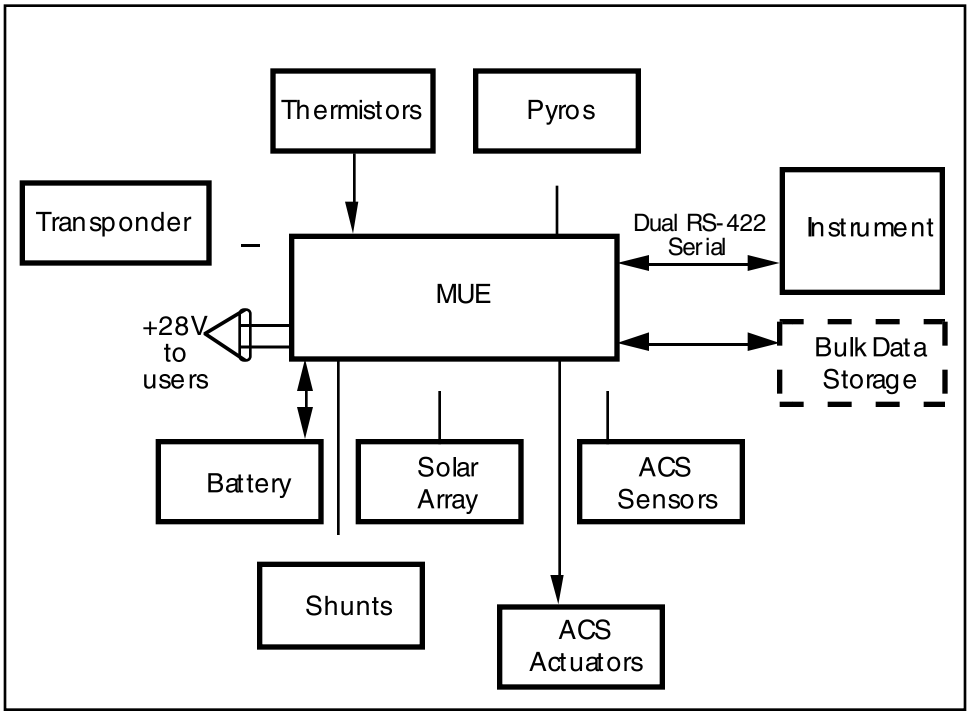

FIGURE B-6 BLOCK DIAGRAM OF THE FAST SPACECRAFT

The Mission Unique Electronics (MUE) perform all major spacecraft functions including command and data handling, attitude control, battery charge control, power distribution, firing pyrotechnic devices, spacecraft health and safety functions, and instrument and launch vehicle interfacing. The spacecraft computer performs power distribution, data encoding/decoding, and launch vehicle interface. Each functional capability has been minimized in order to conserve mass and power.

The MUE computer consists of dual, 8-bit, 80C85 microprocessors, a minimal analog ACS controller, and an analog battery charge controller. Ground commands are received at 2.0 kilobits per second. Health and safety telemetry is downlinked at 4.0 kilobits per second. The higher telemetry downlink rates (up to 2.25 Mbps for some stations) are driven by the instrument electronics. All telemetry is encoded to reduce errors. The dual 8-bit, 80C85 microprocessors, a NASA S-band transponder (same as the baseline architecture), and a four element high gain micropatch antenna comprise the command and data handling system for the FAST spacecraft. Spacecraft time is maintained to better than 3 seconds.

The MUE has no data storage capability except for a small housekeeping buffer. With some modifications the MUE can control a solid state bulk-data storage device (shown in broken lines in Figure B-6, but not required for the FAST mission), although it can only accept instrument data into the solid state recorder at rates less than 4.0 kbps. A MUE-controlled solid state bulk-data storage device can only be dumped for downlink at 4.0 kbps. In the FAST mission the MUE allows the instrument to downlink its own stored data at the higher rates.

The FAST power system consists of body-mounted gallium arsenide solar cells totaling 5.6 square meters which typically generate 50 watts of orbit-average power. Power is distributed to the spacecraft and instruments via an unregulated 28-volt bus. A 9 amp-hour nickel cadmium battery is included in the spacecraft to provide power during eclipses and when load demands exceed the solar array power output. The orbit average power consumption of the spacecraft hardware is 33 watts. The instruments consume 15 watts orbit-average power, 31 watts when operating. Instruments are frequently powered off in order to maintain a positive energy balance. The power system electronics in the MUE is similar to the baseline power system, but with a cruder battery charging control allowed by the shorter (1 year) mission life requirement. The system capacity is approximately 100 watts, the power bus characteristics are identical to the baseline power system, and peak loads are limited to 1/2 of the selected battery rating (4.5 amp-hour).

The ACS capabilities of the FAST spacecraft are limited to simple designs that can be implemented with analog controllers, with limited support provided by the MUE microprocessor. The FAST mission ACS uses a 3-axis magnetometer, a spinning sun sensor, a horizon crossing indicator, and two air-coil magnetic torquers. Spin rate and spin-axis orientation are maintained by the magnetic torquer coils. The ACS provides closed-loop spin-rate control. Spin-axis precession is performed open loop and is closed via ground commands. After computation on the ground, attitude knowledge is accurate to within one degree. Definitive attitude and orbit determination must be done on the ground.

The FAST thermal system design follows the same philosophy as the baseline spacecraft architecture. Temperatures of the spacecraft subsystems are maintained passively between -15 Celsius and +50 Celsius by using surface finishes, limited conduction paths, blankets, and thermostatically controlled heaters.

C. SAMPEX, WIRE, and TRACE Spacecraft Designs

Schematics of the SAMPEX, WIRE, and TRACE instrument and spacecraft payloads, both assembled and as subsystems, are shown in Figures B-7 thru B-12.

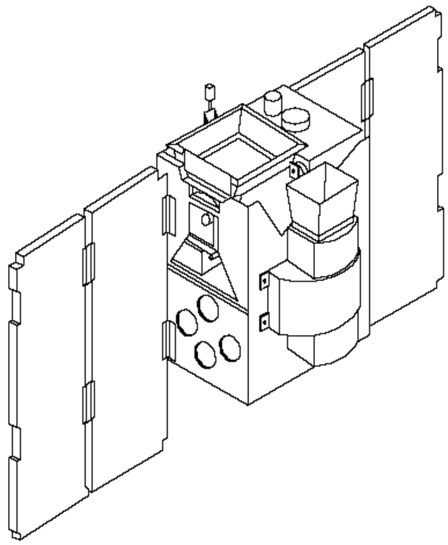

FIGURE B-7 THE SAMPEX MISSION.

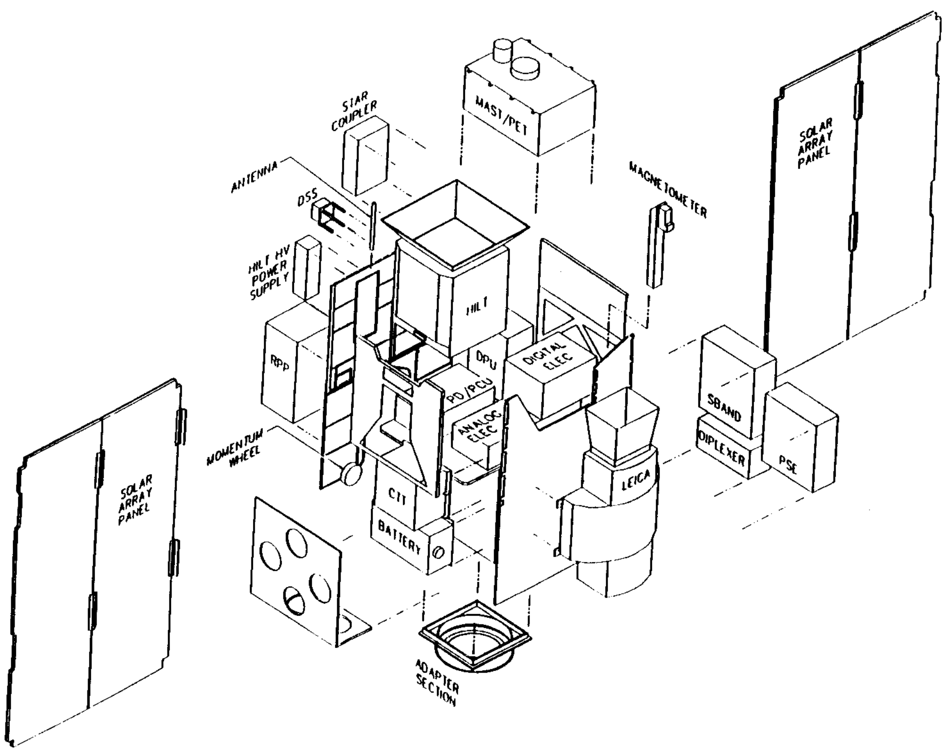

FIGURE B-8 AN EXPLODED VIEW OF THE SAMPEX INSTRUMENT AND SPACECRAFT

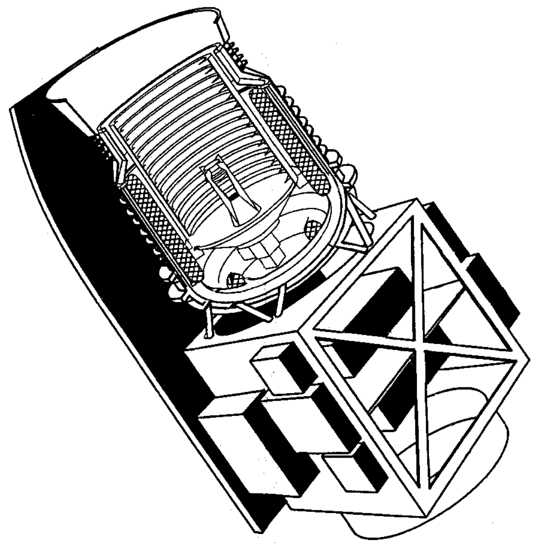

FIGURE B-9 THE WIRE MISSION

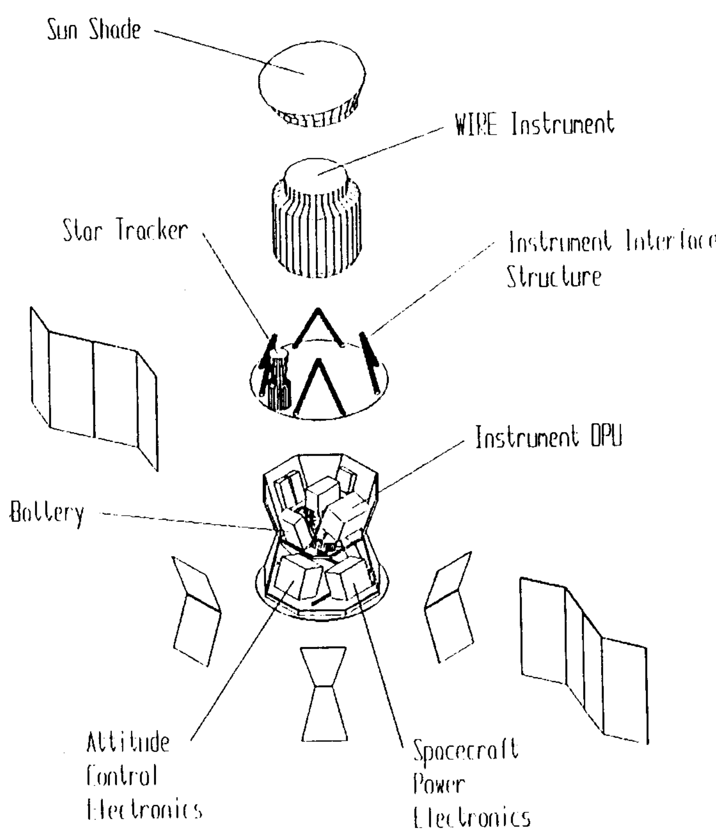

FIGURE B-10 AN EXPLODED VIEW OF THE WIRE INSTRUMENT AND SPACECRAFT

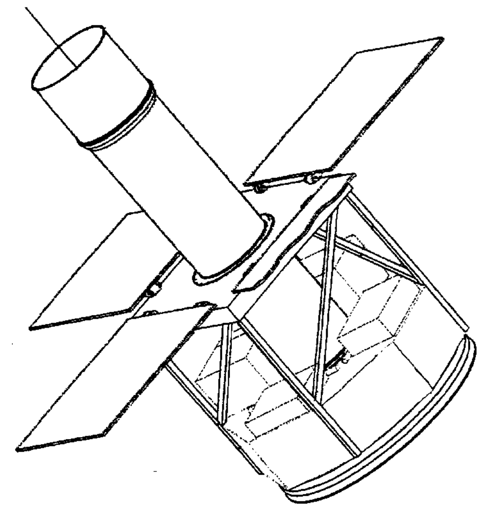

FIGURE B-11 THE TRACE MISSION

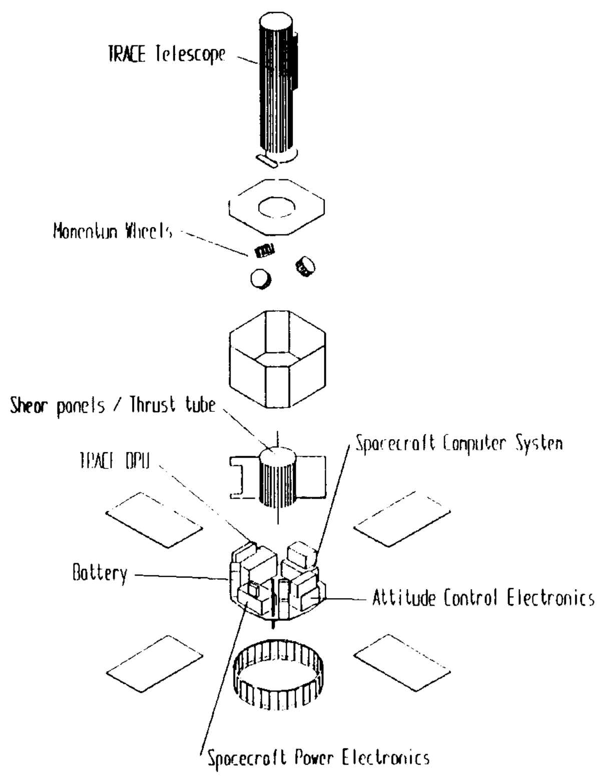

FIGURE B-12 AN EXPLODED VIEW OF THE TRACE INSTRUMENT AND SPACECRAFT

Request for general radiation information

Provide the following information to determine the radiation sensitivities of the mission orbit:

Apogee (km)	______________
Perigee (km)	______________
Inclination (deg)	______________
Argument of Perigee (deg)	______________
Lifetime	______________

Send the above information to the following:

Name: Mr. Ken LaBel

FAX: (301) 344-1719

E-Mail: Ken_LaBel@ccmail.gsfc.nasa.gov

FIGURE B-13 RADIATION QUESTIONNAIRE