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Spacecraft Environment and Effects Monitoring Instrumentation for Small Satellites

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Future satellite systems will be required to survive and function in the space environment for much longer durations (10–15 years) than their present counterparts to achieve greater cost effectiveness. Therefore, the characterization of orbital space environment and its effects on spacecraft systems have received considerable attention in the recent years. Instrumentation is described for long-term measurement of key physical parameters characterizing the low-Earth-orbit environment and its effects on degradation of spacecraft materials and solar arrays. These measurements enable active, real-time monitoring of the spacecraft environment and the health of spacecraft and payload systems. The instrumentation, called Space Active Modular Materials Experiments, is designed to be compact, low power, and lightweight. It is ideally suited for applications to small satellites, especially those designed for long-term, autonomous operation.

Introduction

THE major components of a low-Earth-orbit (LEO) environment are atomic oxygen, ionizing radiation, direct and return-flux contamination, near-vacuum, solar radiation, thermal cycling during orbital motion, micrometeoroids, and debris.

Atomic oxygen (AO) predominates at low altitudes in LEO and can erode thermal control and optical materials, as well as polymeric and composite materials. This erosion can result in the degradation of performance of these materials. As shown in Fig. 1, the AO fluence in LEO is a strong function of altitude and solar activity (as predicted by the Environmental Workbench code). Figure 2 shows the erosion of carbon fiber-reinforced plastic by 8 km/s oxygen atoms at a fluence of 3×10^{20} atoms/cm^2 (Ref. 1). Onboard monitoring of atomic oxygen fluence at specific locations, and its effects on materials used on exposed (especially ram-facing) spacecraft and payload surfaces, will provide diagnostic information useful in assessing systems' health and performance.

Ionizing radiation consists of trapped protons and electrons in the geomagnetic field and solar and galactic cosmic rays. It is well known that over a period of time particulate radiation degrades the crystalline structure of semiconductor devices affecting their proper functioning. Energetic protons, whether trapped or generated by solar flares, and energetic heavy nuclei in cosmic rays, can deposit their energies into the silicon used in semiconductors. This energy can far exceed the critical charge thresholds typical of modern integrated circuits and can cause single event upsets (SEUs) or, worse, permanent damage due to single event latchups in complementary metal oxide semiconductor circuitry. SEUs can cause system software and instrument data corruption and can lead to system malfunction. These effects are expected to become more significant for future spacecraft with gigabits of memory and >10-year lifetime.

The radiation dose effects for a 1700-km orbit (as predicted by the AE8 and AP8 modules of the Environmental Work Bench code) are shown in Fig. 3. According to Fig. 3, commercial electronic components, which are typically rated to withstand total dose of 5–10 krad (silicon), will have a life of 6–12 months at 1700 km, assuming a 0.64-cm-thick aluminum housing. Military- or space-rated components are designed normally to withstand much higher total doses, typically 100 krad to 1 Mrad.

Radiation also damages the crystalline structure of photovoltaic cells, thus degrading the power generating capacity of solar arrays over the long term. Figure 4 shows the degradation in the performance data of the global positioning system (GPS) solar array due to synergistic effects of radiation and molecular contamination.[2,3]

In addition to the effects on electronic components and photovoltaic cells, radiation can also damage polymeric and other organic compounds, including thermal paints that use organic binders. With large doses over time, radiation can adversely affect the strength of structural components made out of composite materials.[4] In situ measurement of the radiation environment in terms of cumulative dose vs energy, as well as particle energy spectrometry, is important for characterizing the orbital environment, identifying instrument data corruption, and monitoring power system performance.

In the near-vacuum of space, most materials used in spacecraft and instrument construction outgas, at least to some extent and for some time duration, releasing contaminants, which can condense on cold optical surfaces, solar arrays, and thermal coatings.[5] The effects of contamination on degradation of GPS solar array output is shown in Fig. 4. The phenomenon of contamination is highly sensitive to the specific details of spacecraft/instrument materials, design and construction (vent paths, seals, etc.), and relative spatial orientation of outgassing sources and receiving surfaces. Clearly, the measurement of contamination at specific locations is essential for interpreting data from optical instruments and assessing power system performance.

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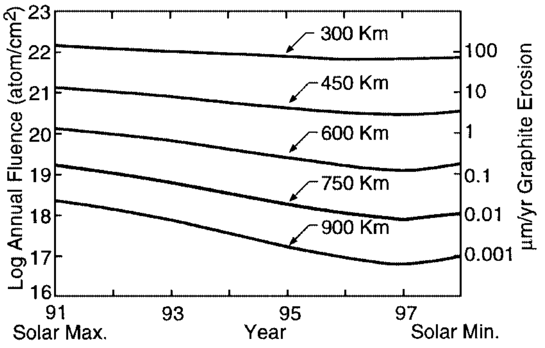


Fig. 1 Annual AO fluence vs year for altitudes between 300 and 900 km.

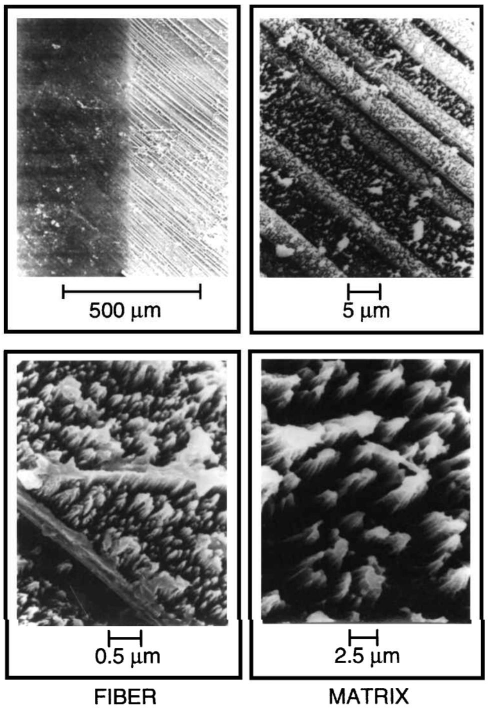


Fig. 2 Scanning electron micrography analysis of carbon fiber-reinforced plastic irradiated by $\sim 3 \times 10^{20}$ cm^2 5-eV oxygen atoms. Top left contrasts virgin and irradiated materials; remaining views emphasize erosion patterns at increased magnification as shown.

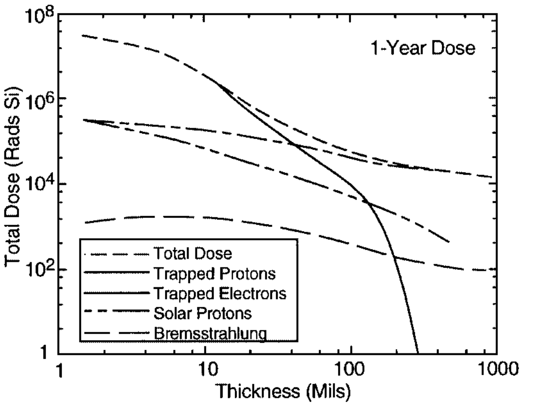


Fig. 3 Annual total doses at 1700-km, 28.5-deg inclination circular orbit.

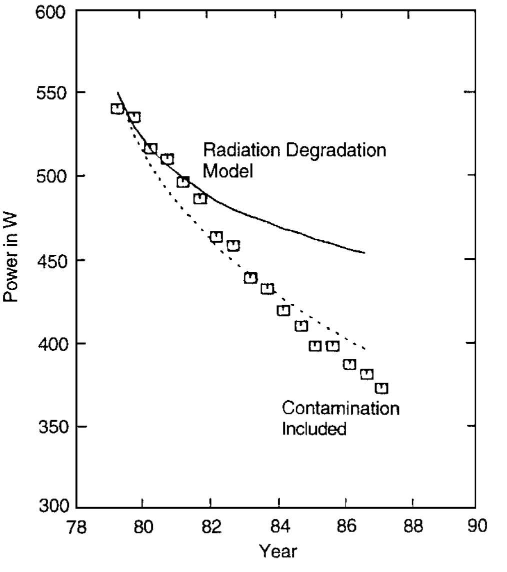


Fig. 4 GPS solar array performance.

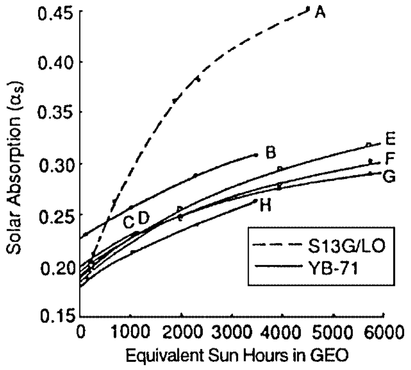
The uv component of solar radiation causes polymerization of materials, in particular, outgassed organics after they condense on optical surfaces, resulting in degraded optical characteristics of instruments. Polymerization due to solar uv also causes discoloration (darkening) of thermal paints, resulting in increased solar absorptivity and, thus, adversely affecting thermal performance.[6,7]

Earth-orbiting spacecraft can experience temperature variations from 200 K while in shadow to 350 K in direct sunlight. The highest temperature is also influenced by internal power dissipated by the system. The temperature extremes are usually moderated by providing appropriate conduction paths through the spacecraft structure. Still, the spacecraft and instrument systems are subjected to thousands of cycles of temperature variations over the life of the mission. Materials, such as thermal coating, can crack due to repeated thermal expansions and contractions. Then, other processes, such as radiation and atomic oxygen can cause further degradation.

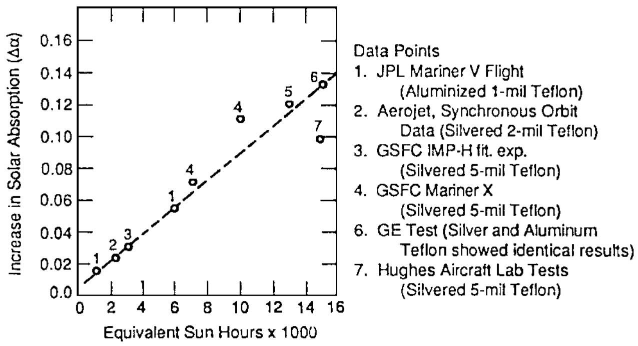
The effects of various components of the space environment on materials were treated separately in the preceding discussion. Of course, in practice, these effects are occurring simultaneously and synergistically. Figures 5a and 5b show flight data documenting the increase in solar absorptivity of two thermal paints (YB-71 and S13G/LO) and a thermal blanket [aluminized fluorinated ethylene propylene (FEP) (Teflon®)]. A significant increase in absorptivity is noted for the silicone-based S13G/LO paint and the aluminized Teflon blanket. The silicate-based YB-71 paint exhibits a moderate increase in absorptivity.[8]

Micrometeoroids and debris, respectively, represent natural and manufactured objects in orbit around the Earth. The orbital debris, resulting from remnants of upper stages of rockets, spacecraft operations, etc., are far more abundant than micrometeoroids. The density of debris is the highest in the altitude range of 600–1100 km, tapering off at altitudes below 200 and above 1100 km. Micrometeoroids are extremely small (typically a few micrograms) compared to debris (typically tenths to a few milligrams), with highest density at lower altitudes, decaying monotonically with increasing altitude. Erosion of spacecraft and optical instrumentation surfaces due to impacts of micrometeoroids and debris can adversely affect the system performance, especially for long-duration spacecraft.

The preceding discussion demonstrates the importance of measuring the spacecraft environment and its effects on materials and solar cells. Such measurements provide the databases necessary for 1) characterizing orbital environment including molecular and particulate contamination environment, 2) monitoring the health of spacecraft/instrumentation systems and scheduling their operations, 3) the development of materials and designs of future spacecraft/instrument systems, and 4) refinement of mission operations to minimize induced deleterious effects. Obtaining accurate measurements as a function of time permits isolation of the influence of various spacecraft and instrument operations, e.g., orientation with respect to ram, solar angle, covers on/off, etc. The Spacecraft Active Modular Materials Experiments (SAMMES) instrumentation[9] performs accurate, time-resolved measurements of the spacecraft environment and actively evaluates the performance of materials and solar cells in this environment. It is a compact, lightweight, low-power



a) YB-71 (silicate-based) and S13G/LO (silicone-based) paints



b) Aluminized FEP thermal blanket

Fig. 5 Increase in solar absorptivity.

system that can perform experiments autonomously via preprogrammed sequences or can be reprogrammed by ground command. The current SAMMES instrumentation contains a suit of sensors to measure all relevant components of the LEO environment, and their synergistic effects on materials and solar cell arrays, with the exception of micrometeoroid/debris and SEUs. The latter measurements will be incorporated into the next generation of SAMMES, which is currently in the design stage.

Design Philosophy and Concept

The design philosophy of SAMMES requires it to be the least intrusive to a variety of spacecraft buses and adaptable to a number of communications interfaces. These features enable SAMMES to integrate easily with spacecraft, to take advantage of quick launch opportunities, and to shorten the lead time to obtain orbital environment and materials data for designers' use. SAMMES possesses distributed modular architecture, so that its sensors, their processing electronics, and data acquisition are programmable and reconfigurable by ground command. SAMMES also carries carefully calibrated sensor systems to ensure well-characterized data of high quality. Two versions have been designed to meet these requirements. SAMMES-1 is designed primarily for LEO altitudes between 250 and 750 km for three years. This design is robust to 1000 km, and the hardware can be operated at higher altitudes with reduced lifetimes. SAMMES-1 was designed for the U.S. Air Force Space Test Experiment Platform mission 3 (STEP-3), which was launched in June 1995 but was aborted due to booster malfunction. SAMMES-2 is currently being built for a radiation-dominated elliptical orbit (460×1800 km) with a design life of one year. SAMMES-2 is designed for the Space Test Research Vehicle (STRV-2) mission.

Figure 6 shows the SAMMES system configuration. It features distributed architecture consisting of a multipayload controller (MPC) and several sensor modules. Each sensor module itself incorporates distributed architecture, consisting of several sensors, which can be located either on the module surface or placed at specific locations of interest, remote from the module. The MPC accepts power and commands from the host spacecraft and sends data to the host for telemetry to ground stations. A special feature of SAMMES, which makes it adaptable to a large class of spacecraft buses, is the choice of command/data interfaces for communication with the host

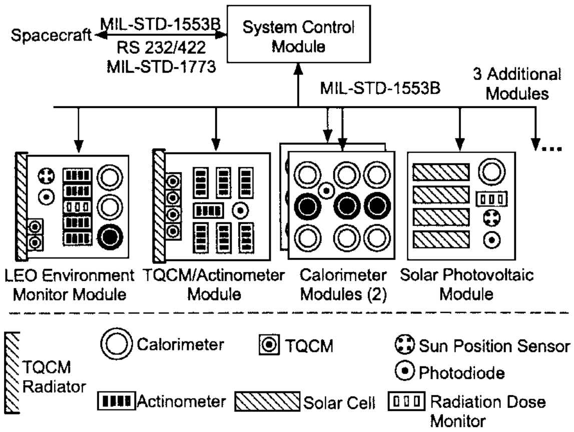


Fig. 6 SAMMES system configuration.

spacecraft, namely, MIL-STD-1553B, RS232C/422, or an optional MIL-STD-1773 fiber-optic interface.

The MPC provides power distribution and control to the sensor modules. It also controls the command/data communication with the sensor modules via a MIL-STD-1553 data bus with the MPC as the bus controller (BC) and the sensor modules as remote terminals (RTs). The MPC provides programmable command sequences for performing spacecraft environment measurements and materials/solar array monitoring and performs preprocessing of sensor data, if required. The raw and/or processed data are stored in the MPC until the spacecraft computer is ready to retrieve it for transmission to ground. This feature allows SAMMES to operate essentially autonomously, i.e., spacecraft command resources are not tied up in running SAMMES experiments. The architecture of Fig. 6 also allows the sensor modules to be plugged directly into the spacecraft computer, if the latter accepts 1553 communication. In this scenario, the spacecraft processor must assume relevant functions of the MPC. As shown in Fig. 6, the SAMMES controller can drive several modules. The current hardware is capable of servicing eight modules.

Instrumentation Description

The general design features of the SAMMES hardware are listed in Table 1. The controller and the sensor modules represent instrumentation that is very compatible with small satellite systems in terms of weight, size, and power consumption. Each sensor module weighs only 2.5 kg and measures approximately $15 \times 15 \times 15$ cm. The controller weighs 4.3 kg and measures $15 \times 20 \times 20$ cm. Each module is designed to operate in two modes: a normal operating mode and a power-conserving quiescent mode. The power consumption of SAMMES is modest: for the controller, less than 10 W (12 W for SAMMES-2) in the operating mode and less than 4 W in the quiescent mode and for the sensor modules, less than 4–9 W (depending on the particular module) in the operating mode and less than 2–6 W (depending on the particular module) in the quiescent mode. The total power consumption of the system can be optimized by maintaining it in the quiescent mode most of the time and bringing the sensor modules up sequentially into the operating mode for data collection.

MPC

The MPC supplies conditioned 28 V of dc power to the sensor modules. The current design incorporates solid-state switches to control power to up to eight modules. The MPC continuously monitors the current drawn by each module and turns it off in the case of excessive current draw. The MPC unique characteristic is the use of two Intel 87C196 microcontrollers: a host spacecraft microcontroller and a sensor module microcontroller to separate host spacecraft-related functions, e.g., communications, interpretation of commands/data for setting particular experiment configuration, etc., from the instrument control functions, e.g., sensor temperature settings, data sampling intervals, communications with sensor modules and among modules, etc. This feature makes the MPC truly

Table 1 System design features

Module	Weight,[a] kg	Size, cm	Power,[b] W Quiescent	Power,[b] W Normal operation	Command/data interface	Data storage	Onboard data processing	Upload programmability
MPC SAMMES-1	4.3	15 × 20 × 20 high	2.7	8.6	MIL-STD-1553 RS 232 RS 422 Option: MIL-STD-1773 fiber optic	7 Mbyte RAM (battery-backed) 1 Mbyte EEPROM	Total 8 kbytes or eight 1-kbyte blocks	Yes
MPC SAMMES-2	5.9	15 × 20 × 20 high	3.5	12.7	MIL-STD-1553 RS 232 RS 422 Option: MIL-STD-1773 fiber optic	7 Mbyte RAM (battery-backed) 1 Mbyte EEPROM	Total 8 kbytes or eight 1-kbyte blocks	Yes
Sensor modules SAMMES-1	2.4–2.6	15 × 16.5 × 14 high	1.9–4.9	3.9–8.4	MIL-STD-1553 Option: RS 232/422	16×10^3 RAM 16×10^3 EPROM Option: 640 kbyte RAM	Optional modif.	Optional modif.
Sensor modules SAMMES-2	3.3–4.1	15 × 16.5 × 14 high	1.9–4.9	3.9–8.4	MIL-STD-1553 Option: RS 232/422	16×10^3 RAM 16×10^3 EPROM Option: 640 kbyte RAM	Optional modif.	Optional modif.

[a]Including sensor weight, 50-mil magnesium alloy housing for SAMMES-1, 100-mil aluminum housing for SAMMES-2.
[b]Excluding sensor heaters: QCM heater ∼ 1.6 W each, actinometer heater ∼ 0.5 W each, and calorimeter heater ∼ 0.5 W each.

adaptable to different spacecraft in that the same experiment can be flown on a different spacecraft with changes in the software of only the host microcontroller. The two microcontrollers share data via a dual-port RAM. The sensor module microcontroller has access to the mass memories [RAM and electrically erasable programmable ROM (EEPROM)] for storing sensor data. Each microcontroller is provided with an external EEPROM, which stores a duplicate copy of the main code in the microcontroller's erasable programmable ROM (EPROM). This feature is built in to protect against loss of functionality if the microcontroller's main program is corrupted due to single event upsets.

The design of the digital section of SAMMES-2 MPC differs substantially from SAMMES-1 MPC due to the high-radiation hardness requirement for SAMMES-2. A higher MPC reliability is required for the STRV-2 mission, where its function is to control not only the modules shown in Fig. 6 but also other high-priority sensors such as a midwave infrared (MWIR) sensor and a laser communications experiment. The SAMMES-2 MPC digital electronics employs components with a minimum radiation hardness of 60 krad (silicon). The two microcontrollers in SAMMES-1 MPC are replaced by two Thompson/Motorola 68T020 microprocessors and other peripherals. The peripherals include programmable ROMs (PROMs) for baseline program storage, RAMs for storing uploaded program modifications, and field programmable gate arrays (FPGAs) for watchdog timer and other controls. The Data Device Corporation (DDC) 1553 hybrids in SAMMES-1 MPC are replaced by radiation-hardened and more monolithic DDC Advanced Communication Engine (ACE) 1553 chips. The two microprocessors exchange data via radiation-hardened dual-port RAMs as in SAMMES-1. The design of the main data storage memory card, which incorporates Densepak RAMs and EEPROMs, is the same for the two MPCs, except that SAMMES-2 MPC employs tungsten spot shielding for individual memory modules.

One other design change from SAMMES-1 MPC to SAMMES-2 MPC is in the power section. The latter uses a higher capacity (70-W) filter on the heater power line to meet the keep-alive heater requirements of additional sensor modules of the STRV-2 mission.

SAMMES-2 sensor module design is essentially the same as SAMMES-1, but with one significant difference in the design philosophy. To meet the higher radiation-hardness requirement, without incurring high costs of replacing components with radiation-hardened parts, it was decided to spot-shield critical components in the modules. The shielded components include microprocessors and A/D converters, and 1553 communication integrated circuits (ICs). The reason behind assuming higher risk with the sensor modules and not with the MPC is understood from Fig. 6. The distributed architecture of SAMMES ensures that in the event of failure of a sensor module the rest of the system will continue to function normally. On the other hand, the functionality of the MPC is central to the functionality of the entire system, and this demands substantially greater reliability of the MPC.

Fig. 7 SAMMES-1 flight modules.

The SAMMES MPC incorporates separate communications interfaces for host spacecraft and for the sensor modules/instruments. The latter interface in the current design is MIL-STD-1553B. To make the MPC adaptable to a wide class of spacecraft buses, three basic and one optional interfaces are provided for host spacecraft communications. The built-in, basic communications are MIL-STD-1553, RS232, and RS422. The MPC is designed to accommodate an optional MIL-STD-1773 fiber-optic interface, if desired.

To make the instrumentation as independent of the spacecraft systems as possible, the MPC has the capability to process sensor data using preprogrammed or uploaded algorithms. It incorporates mass storage to store raw or processed data. The mass storage consists of 7 Mbytes of RAM and 1 Mbyte of nonvolatile EEPROM for preserving critical data when the MPC is not powered. The SAMMES design recognizes the importance of early on-orbit phenomena, when materials begin outgassing in the near-vacuum of space and provides the opportunity to obtain critical contamination measurements. However, power may be only intermittently available immediately after orbit insertion, when solar arrays are just deployed and spacecraft systems are being checked out. Therefore, the RAM is backed by a battery to keep the memory powered and preserve crucial data for about 2 weeks. Of course, as long as it is available, the battery power can also be used to save data in the RAM anytime during the mission when power to SAMMES is turned off. The battery-backed RAM augments the EEPROM for storage of critical data when power is not available.

The software for the two MPCs is to a large degree mission dependent. The SAMMES-1 MPC, designed for the STEP-3 mission, controlled five RT sensor modules, as shown in Fig. 6. The SAMMES-1 test modules are shown in Fig. 7. The MPC and

Table 2 SAMMES sensor complements

Sensor	Measurement	Operating modes	Range	Precision (accuracy)[a]
Actinometer	Erosion due to AO and synergistic effects	Float Constant temp	10^{16}–10^{22} cm^{-2} 1–10 kΩ[b] 1–100 Ω, 10k Ω–1 MΩ[c] -20–80°C	±0.2°C (±0.5°C)
Calorimeter	Changes in α/ϵ due to erosion and contaminant deposition	Float Constant temp	$\alpha/\epsilon = 0$–1 -50–80°C	$\Delta\alpha/\epsilon = \pm 0.0006$ (± 0.004) ±0.2°C (±0.5°C)
Quartz crystal microbalance	Contaminant accretion, coating erosion	Float Constant temp Ramp temp Track temp with calorimeter[b]	-50–80°C	±0.2°C (±1.0°C)
TRDM	Cumulative dose of ionizing radiation	Float Constant temp	1–1000 krad	
Sun position sensor	Evaluate shadowing effects, verify spacecraft attitude	——	35–60 deg from module zenith	±3 deg
Sun sensor	Local solar irradiance, shadowing effects	——	35–60 deg from module zenith	±3 deg
Solar cell strings	Current vs voltage characteristics	——	0–6.25, 25, 100 mA 0–125, 500, 2000 mA 0–2.2, 8.8, 35 V	Greater of 0.2% of measured value or 1 mV/1 mA

[a]Values for temperature control. [b]Space Environment Monitor only. [c]TQCM Material Test Module.

sensor modules exchanged data, but the modules themselves did not communicate with each other. Software for the MPC's sensor module microcontroller was developed to accomplish this single mode of communication. (The host spacecraft microcontroller software was developed to meet STEP-3 spacecraft communication protocol.) By contrast, the sensor modules for the STRV-2 mission not only communicated with the MPC, but also sent data from one sensor module to another (RT–RT transfers). Additionally, the MPC was required to transfer experiment data from the sensor modules to the spacecraft BC and upload code from the spacecraft to the sensor modules at high speeds through the MPC (RT–BC and BC–RT bent-pipe transfers). The software for these four modes of communication was much more complicated than for the SAMMES-1 MPC. An interrupt-driven, real-time program was developed for this purpose incorporating a specialized, compact ($\sim$2 kbytes) operating system. A highly sophisticated Windows-based simulator of the SAMMES-2 MPC was developed to aid in the development of sensor module software and to train the experimenters in the use of the MPC for conducting on-orbit experiments.

The sensor modules are preprogrammed, standalone devices with software that is mission independent to a large degree. The microcontrollers in the modules for the STEP-3 mission were required to configure the sensors according to the parameters sent by the MPC, acquire sensor data at a fixed frequency (maximum 1 Hz), and send the data to the MPC when commanded to do so. Apart from these functions, the sensor modules for the STRV-2 mission are required to perform additional data and code transfer functions mentioned in the preceding paragraph.

The present MPC design provides limited capability for onboard processing of raw sensor data. A total of 8 kbytes in eight blocks 1 kbyte each is available for this purpose. Standard, simple algorithms for computing running averages, picking maxima/minima, etc., can either be preprogrammed or uploaded. Onboard processing allows more efficient use of MPC's mass storage by storing only processed data, rather than all of the raw data, a feature especially useful for high-data-rate experiments.

The MPC is equipped with an analog temperature sensor, which can be monitored by the host spacecraft for assessing the controller's health. There is also a survival heater within the MPC, which is thermostatically activated when the temperature of the electronics is below -25°C.

The SAMMES-1 MPC is packaged in a 0.13-cm-thick magnesium enclosure with mounting points on two sides. The radiation-hardened SAMMES-2 MPC enclosure has identical external dimensions with 0.25-cm-thick aluminum walls.

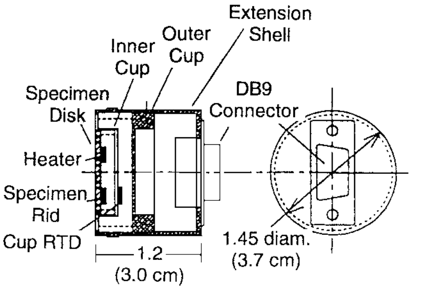


Fig. 8 Calorimeter sensor.

Sensors

There are four types of sensor modules in the current SAMMES suite. They are space environment monitor module, calorimeter material test module, temperature-controlled quartz crystal microbalance (TQCM) material test module, and solar photovoltaic evaluation module. Each module incorporates a variety of sensors, which are described next.

Table 2 lists the SAMMES sensors, their measurement quantities, parameter ranges, and accuracies. Each of the modules just mentioned carries a complement of various sensors. The first four entries in Table 3 represent unique sensors and will be described in some detail.

The calorimeter, shown in Fig. 8, is a device that measures very small changes in the ratio of solar absorptivity α to infrared emissivity ϵ of a material due to exposure to the space environment and contamination. Based on the original concept of Reichard and Triolo,[10] the calorimeter consists of a thermally isolated sample disk and an inner cup surrounding the disk, which are exposed to the environment. The inner cup sits thermally isolated within an outer cup, which is fastened to the module cover with good conductive contact. The exposed area of the disk is equal to the exposed area of the annulus of the inner cup. The inner cup is thermally isolated from the outer cup. A heater and a precision platinum resistance temperature (PRT) sensor are attached to the underside of the sample disk. The inner cup also carries a PRT on its lower surface. The sample disk and the exposed annulus of the inner disk are coated with the material to be tested for effects of space environment. The disk can float to attain an equilibrium temperature with respect to its surroundings, or it can be heated to maintain the temperature of the material at a constant value. A thermal model of the calorimeter was developed that relates the changes in α/ϵ to

Table 3 SAMMES complement of sensors

Test module	Sensors[a]
Space Environment Monitor module	1 Mirrored calorimeter, contamination sensor 1 Kapton/aluminum calorimeter, AO monitor 1 Black calorimeter, irradiance monitor 2 Kapton/silver actinometers[b] 1 Carbon and 1 Kapton/carbon actinometer[b] 1 Carbon-coated TQCM, AO monitor 1 Bare aluminum TQCM, contamination monitor 1 Sun sensor 1 Sun position sensor 1 Radiation monitor
Calorimeter materials test module	8 Test material-coated calorimeters 1 Mirrored calorimeter, contamination monitor 1 Sun sensor
TQCM materials test evaluation module	4 Test material-coated TQCMs 7 Test material-coated actinometers 1 Sun sensor
Solar photovoltaic evaluation module	Up to 5 solar cells (strings), on-module or remote 1 Radiation sensor 1 Mirrored calorimeter, contamination monitor 1 Sun sensor 1 Sun position sensor

[a]This sensor configuration is tailored to STEP-3 mission.
[b]Actinometer units may be replaced by radiation sensors.

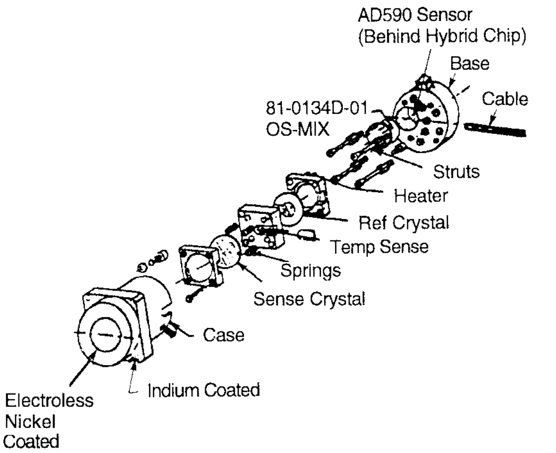


Fig. 9 Temperature-controlled quartz crystal microbalance.

the details of the calorimeter construction, optical properties of the coating, and the measured temperature of the specimen disk, the inner cup and the outer cup. The calorimeter is used for contamination measurements on the environment monitor and solar photovoltaic modules and for materials degradation evaluation on the calorimeter module.

The TQCM, Fig. 9, provides highly sensitive (nanogram) measurement of contaminant accretion or erosion of material from the sense crystal. The TQCM operates on the principal that the beat frequency between the sense crystal and an internally located, unexposed reference crystal is proportional to the mass deposited (or eroded) from the sense crystal. The latter can be actively or passively cooled to enhance the deposition of particular contaminants on it. In the current SAMMES design, the TQCMs are cooled passively with external radiators, rather than actively using thermoelectric coolers, to keep the power consumption low. The TQCMs are used both as contamination accretion and material erosion monitors. In the erosion applications, the sense crystal is coated with the material of interest, and the TQCM registers extremely small losses in mass of the material as a result of the erosion due to synergistic effects of various components of the space environment. The lowest temperatures to which the TQCMs can be cooled depends on several parameters. These include thermal design parameters, pointing of the radiator (facing deep space as much as possible), orbital characteristics, i.e., solar angle with respect to the radiator and the TQCMs, etc. Thermal design parameters include the conductive coupling between the TQCM flange and the module structure (should be very high), the thermal isolation between the radiator and the module structure (should be very high), thermal isolation of the sense/reference crystal pack from the TQCM housing, available radiator area, and so on. Physical Sciences, Inc. (PSI), has developed detailed thermal models for expected on-orbit crystal temperatures for the TQCMs on the space environment monitor and TQCM material test modules.

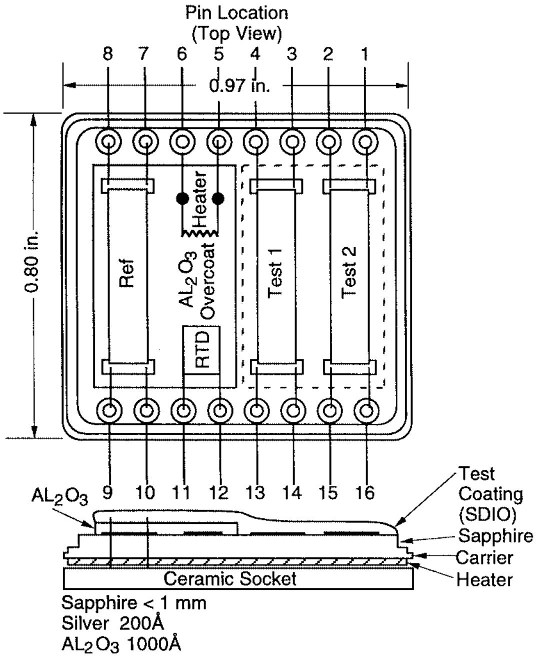


Fig. 10 Actinometer schematic.

Figure 10 shows the actinometer sensor, which is used to measure the fluence of atomic oxygen and its effects on materials. This sensor is essentially a resistance-measuring device. Its basic design incorporates three resistance strips called reference, test 1, and test 2; a temperature sensor; and a heater on a thin sapphire substrate rigidly bonded to a ceramic socket, which plugs into a receptacle on the module. The resistance strips can be of several types: silver coated with a test material such as Kapton®, pure carbon, carbon coated with Kapton, or some high-impedance material. Kapton-coated silver, pure carbon, and Kapton-coated carbon actinometers are used on the environment monitor module for continuous measurement of atomic oxygen fluence; silver for lower dynamic ranges (10^{16}–10^{19} O atoms/cm^2) and carbon for large flux exposures (10^{19}–10^{22} O atoms/cm^2). For the actinometers used for monitoring atomic oxygen effects on materials, the silver strips are coated with the material of interest. When the material has completely eroded due to atomic oxygen and other effects, the silver is exposed and its measured resistance changes rapidly (due to oxidation); this switch type actinometer is used on the TQCM material test module. As indicated by dashed lines in Fig. 10, the two test material strips are exposed to atomic oxygen through a cutout in the actinometer cover. The covered strip is coated with aluminum oxide, which is not reactive to atomic oxygen. This strip, however, is kept at the same temperature as the test strips when the heater is on, thus compensating for the dependence of the resistance on temperature.

The total radiation dose monitor (TRDM) is shown in Fig. 11. It is designed as a plug-in replacement for the actinometer and uses the same analog and processing electronics.

The four sensors just described are complemented by standard sun sensors on all modules to measure local solar irradiance. The space environment monitor and solar photovoltaic modules also incorporate sun position sensors to indicate the solar angle and the spacecraft attitude.

Sensor Modules

The sensors just described, and their processing electronics, are incorporated into four sensor modules, each with a specific function.

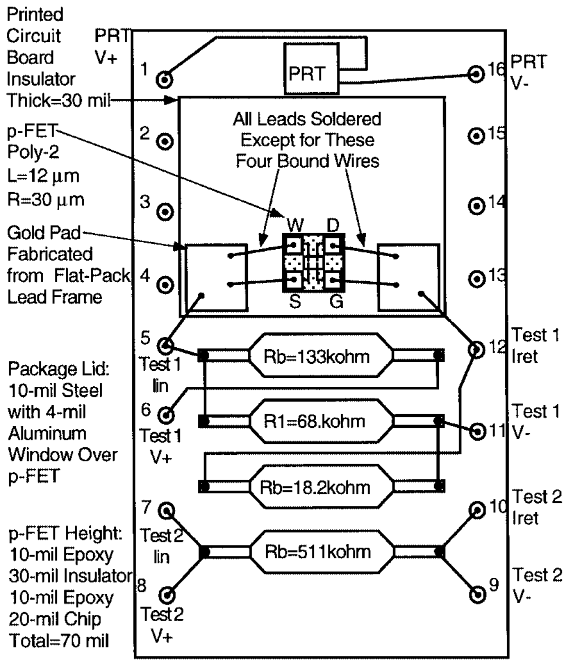


Fig. 11 Total dose radiation monitor.

Table 3 lists the sensors on each module. The space environment monitor module contains at least one sensor of each type, including logarithmic amplifier-based actinometers for continuous AO fluence monitoring, to measure the local environmental parameters. The calorimeter material test module incorporates calorimeters and a sun sensor. The TQCM material test module incorporates TQCMs, switch-type low- and high-impedance actinometers, and a sun sensor. The solar photovoltaic evaluation module incorporates up to five solar cells/strings, one calorimeter as a contamination sensor, a total dose radiation monitor, a sun sensor, and a sun position sensor. An important design feature of these modules is that any or all of the sensors, and the solar cells, can be placed at specific locations, e.g., where contamination or other effects are to be assessed, away from the processing electronics in the modules themselves, including the TQCMs and their radiators. Another equally important feature is the modular design of sensor/module interface. All of the sensors and solar cells plug into sockets on the top faces of the modules, thus allowing easy replacement or substitution prior to launch. This capability is useful if it becomes necessary to replace or repair a failed sensor or if new material/solar cell technologies become available for on-orbit testing during the several months of spacecraft integration and launch processing.

The four SAMMES test modules are shown in Fig. 7. All of the SAMMES test modules are identical in mechanical design and construction in terms of size, housing, external connectors, and mounting. This permits standardization for the production of modules. In addition, the digital electronics and power boards are identical for all modules. Only the analog electronics and sensor interface boards and the backplane change from module to module as the sensor complement changes. The space environment monitor module and the TQCM material test module incorporate TQCMs and use radiators mounted on the sides of the modules for passive cooling of the TQCM sense crystal. These modules require the radiators to be pointed toward deep space and away from other spacecraft hardware, as much as possible. The presence of radiators also means that three faces are available for mounting these modules. For mounting the calorimeter or solar photovoltaic modules, four faces are available.

Figure 6 shows the sensor arrangements on the four modules, which were configured for the SAMMES mission on the STEP-3 satellite. The sensors are held in place by the top cover with cutouts so that all sensors are flush with the upper surface of the cover. The TQCM radiator is thermally isolated from the module and is attached by very low-conductance standoffs to one face of the module. The radiator plate is bent around the module on two sides to provide short extension tabs. The tabs reduce the degradation in radiator performance when its main surface is directly exposed to solar illumination. The radiator face is covered with very low α/ϵ aluminized Teflon. This radiator design allows the TQCM sense crystal to be cooled to $-40°C$, or to much lower temperatures, depending on the particular orbital/solar angle conditions. All sensor modules are equipped with analog temperature monitors, which the host spacecraft can use for instrument health assessment, and survival heaters, which are activated when the electronics temperature is below $-25°C$.

The architecture of the sensor module electronics is common (other than the power board) for all modules. The heart of the electronics is an Intel 87C51 microcontroller, which controls the sensor excitations via D/A converters and collects sensor data from analog multiplexers via A/D converters. The microcontroller has 16 kbytes of EPROM for storing software for module operation. It also controls the communication from the DDC 1553 hybrid IC.

Operational Features

The SAMMES MPC and sensor modules provide a great degree of operational flexibility to the spacecraft operators and experimenters for monitoring on-orbit environment and its effects on spacecraft/instrument systems, materials, and solar arrays. The SAMMES system is autonomous and can be run with minimal intervention on part of the spacecraft processor. For example, the sensor modules, individual sensors, and their operating modes, e.g., temperatures and heating rates, can be run by preprogrammed command schedules stored onboard or can be reprogrammed from the ground. Because the MPC and sensor modules have three modes, off, quiescent, and operating, the modules (and sensor heaters) can be multiplexed to control peak power consumption. An important characteristic of the sensor module design is the ability to keep the sensor heaters powered when the module is in a quiescent mode. This temperature control feature allows material samples on the sensors to be maintained at constant high temperatures when the module is in a power conserving, quiescent mode. Moreover, the design allows the sensor modules to be powered directly by the spacecraft, independently of the controller, so that the integrity of the samples and sensors themselves, e.g., calorimeters, can be assured when only very limited power is available, as is the case just after shroud separation prior to orbit insertion.

The MPC software permits the selection of individual sensor modules, sensor modes (float, constant temperature, or temperature ramp), sensor data channels, data rates and durations, and the timing for execution of various experiment/data takes segments. It is organized in terms of operational configurations (OCs) and experiment configurations (ECs). The EC tables specify sensor modules and sensor parameter values. The OC concatenates the EC with sensor data rates, housekeeping data taking specifications, and channel-making instructions. There are 64 OCs and 64 ECs in the present version of the MPC software, with eight (8-bit) words per OC and 96 words per EC. Together, the ECs and OCs are organized in command schedules, which can be preprogrammed or uploaded. These schedules are executed using as a cue the time provided by the spacecraft processor. In addition to two hardwired signals for quiescent and thermostat overrides, 21 software commands are needed to run the MPC.

In a typical on-orbit operational scenario, the MPC powers a sensor module (if off) or wakes it up (if quiescent), collects data from it, and puts it back into the quiescent mode. Then, the MPC goes sequentially to the other sensor modules to complete the data take cycle. Finally, per command schedule, the MPC may go into the quiescent mode itself for a prescribed period of time, and after that, go back into the operational mode to repeat the data take cycle or to execute a different command schedule.

Flight Heritage of the Instrumentation

The first flight of the SAMMES instrumentation was aboard the ill-fated U.S. Air Force STEP-3 mission in June 1995. The spacecraft did not make it to the intended orbit because the mission was aborted when the Pegasus XL launch vehicle malfunctioned. The mission objective was to test several advanced technologies including SAMMES-1 in a nominally 500-km, 77-deg inclination, circular

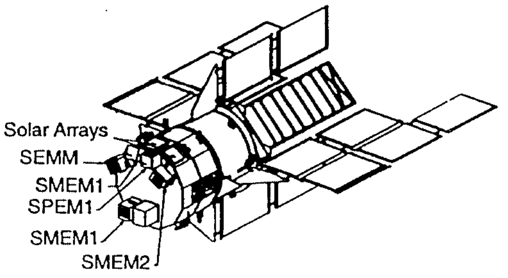


Fig. 12 SAMMES experiment configuration on STEP-3 spacecraft.

orbit for a period of one to three years. The purpose of SAMMES on this mission was to monitor the LEO space environment of the satellite and to evaluate the effects of this environment on several new materials and solar cell technologies.

Figure 12 shows the arrangement of various sensor modules on the STEP-3 satellite. The MPC was located inside the payload module on this mission. An environment monitor module, TQCM material test module, and one calorimeter material test module were mounted on the deployment plate of the satellite facing directly into ram to maximize the effects due to atomic oxygen. It is noted that the environment monitor and TQCM test modules were oriented such that their radiators are pointed essentially to deep space at all times. A second calorimeter material test module was mounted so that it would point into the wake of the satellite, where direct atomic oxygen flux effects would be minimal. Both calorimeter test modules carried calorimeters coated with advanced space materials, with some materials carried on both modules. The solar photovoltaic module on STEP-3 was mounted pointing essentially toward zenith. Five solar cell technologies were to be tested. Their representative arrays were incorporated into two panels, which were mounted to the spacecraft's structural panels.

A follow-on mission of SAMMES-2, STRV-2, in a radiation-dominated 460×1800 km elliptical orbit, is scheduled to launch in 1998. The objective of this mission will be to obtain space environment data including micrometeoroids and spacecraft contamination data, test performance of materials and electronic components in the radiation environment, and test an advanced MWIR sensor, active vibration suppression, and satellite-to-ground laser communications technologies.

Planned Future Capabilities

The SAMMES system architecture shown in Fig. 6 allows the expansion of the current suite of modules and complement of sensors to incorporate new sensors and instrumentation that have recently been added to SAMMES-2. For example, a micrometeoroid/debris monitor (developed by J. Kinard at NASA Langley Research Center) and an electronic testbed module have been added for the STRV-2 mission.

The present design of the sensor modules permits their autonomous operation, i.e., without the MPC, by direct 1553 communication link with the spacecraft processor. In this case, the spacecraft has to incorporate software to assume some or all functions of the MPC. The capabilities of autonomous sensor modules can be expanded significantly by adding a reasonable amount of mass memory (about 1 Mbyte).

Design concepts for incorporating additional sensors/instruments that can be controlled by the MPC have been developed. Examples include plasma probes, charged particle detectors, an ultraviolet/visible (UV/VIS) spectrometer, and a pressure gauge.

Studies have been conducted to significantly reduce the size and weight of our current designs. For example, using advanced electronics packaging technologies, the entire space environment monitor module can be reduced to a flat structural panel with embedded sensors and processing electronics. A typical size would be a $18 \times 18 \times 2.5$ cm thick panel weighing less than a kilogram and consuming less than 5 W. This instrument technology should be very attractive to microsatellites with limited real estate and power.

The MPC can be modified to separate its power and instrument control and communications functions from data processing functions. PSI has developed a concept for a separate neural net-based data analysis/processing module for instruments with very high data rates.

The instrumentation described for monitoring the space environment and its effects on materials and solar arrays is ideally suited for applications to long-duration monitoring of the health of small satellites systems and instruments. Apart from its compact size, lightweight design, and low-power consumption, the instrumentation will contribute toward autonomous operation of small satellites and their fleets. This will potentially reduce the costs of on-orbit operations of satellite fleets designed for long lifetimes. For example, data from the TQCM sensors can be used for onboard assessment of the accretion of molecular contaminants on optical sensors and materials. This information can then be used as a signal to deploy covers, or heat certain optical components, or adjust spacecraft attitude to point the instrument toward the sun, if appropriate. Data from total dose radiation monitors can be used for on-orbit self-maintenance of cryo-cooled detectors, i.e., when the dose reaches a certain predetermined level, the detector can be warmed up for annealing. Total dose radiation monitors can be embedded inside the instruments to assess the actual degradation of the electronics and to guide corrections to the instrument data. Internal SEU counters, which can also be built into the current SAMMES system, can be used as alarms to indicate possible data corruption or to trigger correction algorithms.

Examples of applications to spacecraft systems include the management of power and thermal systems. On-orbit monitoring of solar array degradation due to radiation and contamination effects will yield the rate of decrease in power output. This information can be used for scheduling of spacecraft systems/instrument operations and long-term mission planning. The degradation of thermal control and optical coating materials in the space environment can be monitored using the material test evaluation modules. The rates of increases in α/ϵ of radiator and mirror surfaces and thermal blankets can be measured and used for onboard predictions of the performance of systems using thermal models. This information can then be used to schedule instrument operations and long-term mission planning. An example would be loss of cooling capacity due to loss of radiation efficiency. Thus, the experiment that relies on availability of adequate cooling could be scheduled appropriately in the orbit. Finally, the space environment monitor or a combination of environment and solar photovoltaic monitors can be incorporated into small satellites, not as a payload, but as a standard spacecraft subsystem, for routine monitoring of the environment and spacecraft systems health.

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