Low Cost Environmental Sensors for Spaceflight: NMP Space Environmental Monitor (SEM) Requirements

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An outstanding problem in spaceflight is the lack of adequate sensors for monitoring the space environment and its effects on engineering systems. By adequate, we mean low cost in terms of mission impact (e.g., low price, low mass/size, low power, low data rate, and low design impact). The New Millennium Program (NMP) is investigating the development of such a low-cost Space Environmental Monitor (SEM) package for inclusion on its technology validation flights. This effort follows from the need by NMP to characterize the space environment during testing so that potential users can extrapolate the test results to end-use conditions. The immediate objective of this effort is to develop a small diagnostic sensor package that could be obtained from commercial sources. Environments being considered are: contamination, atomic oxygen, ionizing radiation, cosmic radiation, EMI, and temperature. This talk describes the requirements and rational for selecting these environments and reviews a preliminary design that includes a micro-controller data logger with data storage and interfaces to the sensors and spacecraft. If successful, such a sensor package could be the basis of a unique, long term program for monitoring the effects of the space environment on spacecraft systems.

I. Introduction

IN 1994 the National Aeronautics and Space Administration (NASA) created the New Millennium Program (NMP). The objective of this program is to conduct spaceflight validation of breakthrough technologies that will significantly benefit future space- and Earth-science missions. Selected technologies are focused on mitigating risks for first time users and enable the insertion of new technologies into NASA missions. As will be discussed here, examples of such potential technologies to be validated by NMP are those to be tested on ST8 and ST9. In addition to testing these technologies, NMP is exploring the possibility of making a space environmental monitor available for inclusion on validation flights. The reason for this follows from the need to characterize the validation-flight environment so that the test results can be extrapolated to the end-users' environments. This requirement translates into a short-term objective of developing a "hockey-puck" size Space Environmental Monitor (SEM) to be flown with future NMP technology validation flights. The long-term goal is to include an inexpensive SEM with every NMP validation flight. This paper presents on overview of representative NMP technologies and then uses them to determine the sensors to be included in a SEM. The notional SEM requirements are presented along with suggested sensors and a data acquisition architecture.

II. NMP Technologies

A NASA NRA for the ST8 technologies was recently issued calling for the development of the following four subsystem technologies:

- Deployment of Ultra Lightweight Booms,
- Deployment of Lightweight Solar Array,
- Thermal Management Subsystem for Small Spacecraft,
- Commercial-Off-The-Shelf (COTS)-Based High Performance Computing for Space.

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In this section, these technologies and their flight validation objectives are described briefly to provide the technology context for the SEM.

A. ST8-1: Deployment of Ultra-Lightweight Booms

The first technology, ultra lightweight deployable structures, represents a fundamental technology upon which a myriad of future space applications depend. It is an enabling technology for large membrane structures such as solar sails and telescope sunshades, solar array assemblies, large aperture optics, instrument booms, and antennas by offering significant mass savings and compact volumes for easy packaging for launch. The ST8 flight objectives are:

- Validation of boom deployment, including the dynamics and uniformity of the deployment action and the completeness with which the boom secures into its final state of deployment;
- Characterization of the structural mechanics and dynamics of the deployed booms; and
- Validation of design approach and predictive methods for deploying ultra lightweight booms by correlating flight measurements with analytical models developed through ground testing.

B. ST8-2: High-Performance Solar Array.

The second technology, lightweight solar arrays at the multi-kilowatt level, promises greater than a factor of three increase in the power per unit mass of spacecraft power systems over those currently available. However, the flimsiness of these structures and the uncertainties in deployment mechanisms and dynamics when in space preclude ground validation of their deployment characteristics. A space validation experiment is required to verify the deployment technology and to characterize the effects of the space environment on the structural dynamics and power generating performance of these ultra lightweight arrays. The flight validation objectives are:

- Characterization of the deployment, controllability, and structural dynamics of a lightweight solar array assembly;
- Verification of the predicted structural and photovoltaic performance of the deployed solar array, including the behavior and durability of the photovoltaics, any supplemental optics, and panel materials in the space environment;
- Verification of secure deployment after the solar array is deployed;
- Verification that the deployed solar array is dynamically stable;
- Validation of photovoltaic cell, blanket, and solar array technology that is capable of being qualified for future NASA missions; and
- Validation of all structural and electrical performance models used to scale up to 7 kW (if flight demonstration is subscale and/or not fully power producing).

C. ST8-3: Thermal Management

The need for mass savings becomes ever more critical as spacecraft sizes shrink to accommodate smaller and more efficient payloads, and advances in thermal control technologies are an integral part in meeting this requirement. There is a critical need for advanced thermal control technology that would allow the low mass, low power, and compactness necessary for future spacecraft. This new technology would not only save mass but it would also enable design flexibility in component placement (i.e., free of thermal constraints) and minimize–if not eliminate–the need for supplemental electrical heaters. The flight objectives are:

- Validation of the performance of a thermal control subsystem designed specifically for a small (< 150 kg) spacecraft having a total power generation of $\leq$250 W and corresponding power dissipation of $\leq$200 W
- Validation of analytically predicted savings in spacecraft mass, power, and volume of thermal control system designed for small spacecraft when compared with conventional thermal control techniques; and
- Validation of analytical models used to predict thermal performance of optimized component locations enabled by new thermal control system.

D. ST8-4: COTS Based High Performance Computing

Onboard high performance, low power computing for science and autonomy data is required on many future NASA space science missions. It is envisioned that these high performance computing systems will be used as an adjunct to a radiation hardened ultra-reliable spacecraft control computer and associated avionics, acting as computer servers or as instrument processors. Specific usage will ultimately depend on the specific mission requirements. The flight objectives are:

- Validation of the radiation fault models, system models, laboratory testing procedures, design tools and fault tolerance techniques with respect to system level predicted fault rates and representative locations in *natural space radiation environments*;

Table 1. NMP Technology Orbital Parameters

PARAMETER	UNIT	VALUE
Orbit: Apogee	km	5400
Orbit: Perigee	km	250
Mission Life	Months	6
Temp.: Solar Array	deg C	-105 to +110
Temp.: Sun Sensor	deg C	-30 to +50
Temp.: Magnetometer	deg C	-80 to +80
Rad Max: (14 mil Al at 5400 km):	Mrad/yr	5
Rad Max: (116 mil Al at 5400 km)	krad/yr	50
Atomic Oxygen (250 km)	Atoms/cm^2-yr	~1e22

- Validation that low cost fault tolerance techniques can provide predictable and acceptable levels of reliability for space based COTS onboard data processors while maintaining orders of magnitude performance improvement over state of the art radiation hardened systems in a minimal overhead, scalable architecture.

III. NMP Technology Environments and Monitoring Needs

A summary of the orbital parameters for ST8 are listed in Table 1. To accommodate the technologies mentioned above, orbits between 250 and 5400 km altitude are being considered. These orbits lead to the radiation exposures behind thin and thick shields listed in Table 1. The temperature variations for various exposed components are also given in the table and it is seen that the variations exceed 100 °C[1]. Finally, as will be discussed in a later section, atomic oxygen exposures are greatest at low altitudes.

The physics describing the space environment and its interactions with spacecraft are described by Hastings and Garrett[3]. The requirements for an environmental monitor to characterize these environments and their interactions are depicted in Table 2. In the first column, environmental parameters are listed ranging from contamination and radiation to mechanical environments. The ST8 technologies are listed at the head of the columns along with proposed ST9 technologies. (the ST9 technologies are described in a just-released NASA NRA). The intersections of the environments with the technologies are marked in Table 2. From this table, the key environments were selected for monitoring and are shown in gray in column 1. These key environments and their associated sensors are listed below—the sensors will be described in the next section:

- Contamination: QCM
- Atomic Oxygen (AO): Actinometer
- Ionizing Radiation: TID Radiometer
- Cosmic Radiation: Charged Particle LET Detector
- EMI: Magnetometer
- Temperature: Thermal Measurement Unit

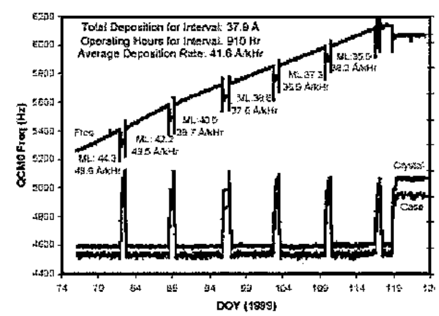

Figure 1. Line-of-sight QCM located in view of the ion engine showing the deposition of the molybdenum-propellant.

IV. Environmental Monitor Sensors

In this section the requirements for the key environmental sensors are given along with examples of their usage on previous missions (note: these examples are provided to illustrate possible solutions and are not intended to imply that they are the sensors of choice). The sensor examples have been flown and provide insight into the expected results and into potential development costs for the SEM.

A. Contamination: QCM

Spacecraft contamination requires a contamination source, a transport mechanism, and sink. The source of contamination comes from thrusters, material outgassing, and microorganisms. The transport mechanisms are: electrostatic return (ESR), diffusion, and line-of-sight motion of molecules while contamination sinks are solar arrays, thermal radiators, optical components, and planetary surfaces.

A commonly used sensor for characterizing contamination deposition is the Quartz Crystal Microbalance, QCM. These sensors have been flown for over 20 years to detect contamination on spacecraft. A recent example of QCM usage in space is the Remote Sensor Unit, RSU, found on the DS1 spacecraft and used to characterize molybdenum contamination from the xenon ion engine[4]. The RSU was mounted on the spacecraft and had two QCMs where one had a line-of-sight view of the ion engine and the other did not. Sample data from the line-of-sight QCM showing the deposition of molybdenum propellant are presented in Fig. 1. In the figure, the change in QCM frequency is proportional to the deposited mass (~42 Å/kHr).

Table 2. Environment-Technology Intersections.

Environment	ST8-1 Ultra-Light-weight Booms	ST8-2 Light-weight Solar Arrays	ST8-3 Thermal Manage-Ment	ST8-4 COTS Computing	ST9-1 Solar Sail	ST9-2 Aero-capture	ST9-3 Large Space Telescope	ST9-4 Precision Landing	ST9-5 Precision Formation Flying
Contamination									
Outgassing	X	X	X		X	X	X	X	
Thruster		X	X		X	X	X	X	X
Radiation									
Electrons	X	X		X	X		X		X
Protons	X	X		X	X		X		X
Cosmic Rays				X					X
UV	X	X		X	X		X		
Plasmas	X	X			X	X			
Neutrons				X					
Magnetic Field	X	X		X	X		X		
Particulate									
Orbital Debris		X	X		X		X		X
Meteoroids	X	X			X		X		X
Atomic Oxygen	X	X			X		X		X
Temperature									
High T	X	X	X	X	X	X	X	X	X
Low T	X	X	X	X	X	X	X	X	
Transient T	X	X	X	X	X	X	X	X	
Atmospherics									
Aero. Drag						X		X	X
Atm. Heating						X		X	
Pressure						X		X	
Mechanical									
Deploy. Motion	X	X			X		X		
Launch Vib.	X	X	X		X	X	X	X	X
Gravity Gradient	X				X				

B. Atomic Oxygen: Actinometer

Atomic oxygen can erode spacecraft surfaces, alter materials through chemical transformations, and de-link polymers weakening their mechanical properties. The effect depends on the altitude of the spacecraft and the orientation of the exposed material relative to the atmospheric oxygen flux. The fluxes are highest in the spacecraft RAM direction. As an example, the erosion rate for Kapton was found to be ~2.8 μm when exposed to 10^{20} oxygen atoms/cm^2.[2,5] The dependence of atomic oxygen fluence on spacecraft altitude is shown in Fig. 2. As seen in the figure, the fluence decreases approximately exponentially with altitude and is dependent on the solar cycle. In addition the AO erosion depends on the orientation of the exposed material with respect to the spacecraft motion.

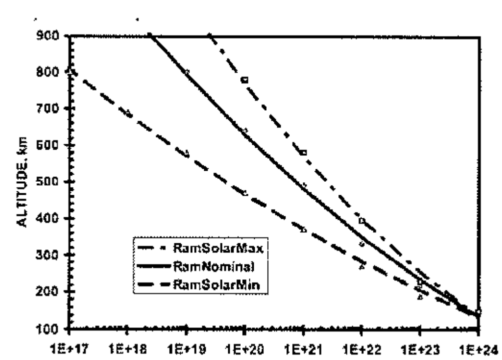

Figure 2. Atomic oxygen fluence dependence on spacecraft altitude.[2]

A device called the actinometer, designed to measure the effect of atomic oxygen on various insulators and

Table 3. QCM Specification

PARAMETER	UNIT	VALUE
Diameter	mm	3.6
Sensor range	g/cm^2	> 10^{-4}
Temp. Sensitivity	Hz/°C	<2.5
Temp. Range	°C	-43 to +80
Temp. Accuracy	°C	<±1
Temp. Precision	°C	<±0.2
Sensor Output	bps	1
Sensitivity	ng/cm^2	> 4.4

metals, was flown on SAMMES (Space Active Modular Materials Experiment) in 2000 on STRV-2. [6] The device consists of three thin metal resistive films (silver, carbon, and a reference silver sample), a heater, and a thermometer. The test coatings can be paralyned to protect and prevent oxidation of the thin metal layers while on the ground. Materials like paralyne eroded quickly in space thus exposing the two test surface to the space environment. Alternatively, the test coating can be Kapton in which case its erosion is determined when the underlying silver film begins to change resistance. The reference surface is coated with alumina which prevents its erosion all together. A comparison of the surface resistances of the two samples with the reference resistor values allows the elimination of temperature effects.

The actinometer specifications are listed in Table 4. As indicated above, the actinometer can be operated in several erosion modes and so can be used to determine the erosion of resistive and insulating materials. The characterization modes are: (a) metal erosion and (b) insulator erosion modes. In addition the heater can be used to adjust the film temperature from -50 to +80 °C to allow the study of temperature effects. The temperature modes are: (a) constant temperature and (b) variable temperature. The data rate for these measurement is low and estimated to be 10 bps.

C. Radiation: Total Ionizing Dose Radiometer

The total dose radiation environment behind a 2.5-mm (100-mil) aluminum spherical shield (2π) is shown in Fig. 3 for various orbital inclinations and altitudes. For altitudes below 1000 km the total-dose radiation is less than 10 rad(Si)/day. A particularly compact radiation monitoring package was flown on the Clementine mission in 1994. The electronics box, shown in Fig. 4, was called the RRELAX (Radiation and RELiability Assurance Experiment). This experiment contained a number of dosimeters termed RADMONs facing in the x-, y-, and z-directions on the box. The RADMONs labeled Z1 to Z1 on the upper surface had different shields that ranged from a 25-μm Kapton layer to a 4-mm Kovar lid allowing the characterization of the radiation energy spectrum.

Table 4. Atomic Oxygen Specifications.

PARAMETER	UNIT	VALUE
Range	Atoms/cm^2-year	10^{16} to 10^{22}
Test Coating	NA	TBD
Test 1 Resistor-Silver	Ohms	1 to 100
Test 2 Resistor-Carbon	Ohms	1k to 100k
Temperature: Range	°C	-50 to +80
Temperature: Accuracy	°C	±1
Sensor Output	bps	10

Total dose is measured by the RADMONs using a p-FET (p-type Field Effect Transistor). These measure dose by following the shift in the threshold voltage which is sensitive to radiation charge accumulated in the gate oxide. The threshold voltage is temperature sensitive and so it must be operated at a current where the current-voltage characteristics are independent of temperature.

Results from the RRELAX dosimeters relevant to the SEM, shown in Fig 5, reflect the journey of the spacecraft as it passed through the Earth's radiation belts and encountered a solar flare as it orbited the Moon. Also apparent is the sag or fade in the data due to loss of radiation induced charge from the gate oxide. Two critical parameters for the p-FET dosimeters are its temperature sensitivity and dose fading. These and other specifications are listed Table 5.

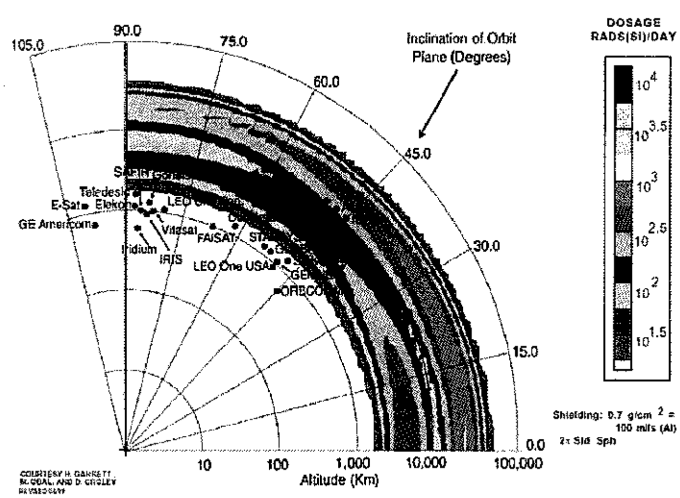

Figure 3. Daily total ionizing radiation dose behind a 2π 2.5-mm (100-mil) thick aluminum solid spherical shield

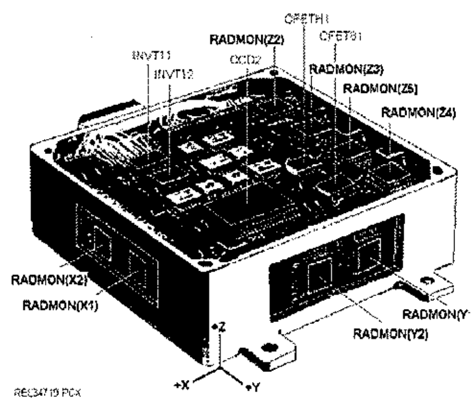

Figure 4. RRELAX is a 624 g, 2.4 watt, 10.2 cm x 10.2 cm x 3.8 cm box used to characterize 166 test devices and the electron, proton, solar flare environment.

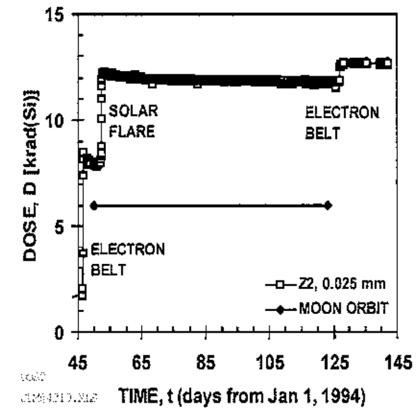

Figure 5. Total dose determined from the RRELAX p-FET dosimeter.

D. Radiation: Charged Particle LET Detector

Single event upsets affect microcircuit memory elements such as registers and memories by changing the state of the stored information. The effect is caused by high energy particles such as Cosmic Rays or protons. The effect is generally non-destructive when latch-up is not involved. The memory bit upsets can be used to detect the flux of high energy particles. Two types of devices can be used: DRAM (Dynamic Random Access Memory) and SRAM (Static Random Access Memory). The DRAM is susceptible to total dose radiation that not only shifts its threshold voltage but also increases the leakage current. The increased leakage current may be the failure mechanism. [7] Some basic DRAM requirements are listed in Table 6.

SRAMs can also be used as particle detectors. A custom SRAM was designed for the RRELAX that had an upset threshold that could be adjusted by an applied voltage. [8] This allows the chip to be operated in a spectrometer mode capable of determining the LET (Linear Energy Transfer) of the incoming particle.

Results from the Clementine mission are shown in Fig. 6. The results were measured when the spacecraft was orbiting the Moon. The SRAM upsets reached almost 100,000 upsets in a 30 minute period during the solar flare event of 12 Feb 1994. The specifications for the SRAM are listed in Table 7.

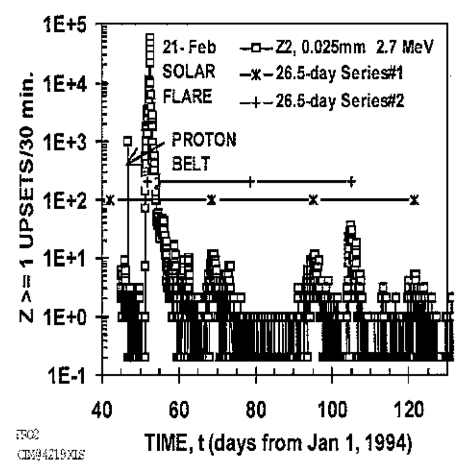

Figure 6. Single-event upsets determined from 4-kbit upsetable SRAM.

E. EMI: Magnetometer

An example of fluctuations in the Earth's magnetic field, seen on the GOES-7 in 1989, is shown in Fig. 7. The graph shows an extreme case where the normal geomagnetic field at geosynchronous orbit (upper data) was replaced by the solar wind field as the Earth's field was compressed for a few hours by a coronal mass ejection. Note that the time scale is about half a day and the data rate is about a data point per minute.

A variety of low cost magnetometer systems exist. [9] The primary reason for including a magnetometer is to characterize the varying magnetic fields which are produced by the operation of the spacecraft or by the

Table 5. Total Dose Radiation Specifications

PARAMETER	UNIT	VALUE
Range	krad	0.1 to 50
Sensitivity	MV/rad	0.2
Temp. Sensitivity	krad/°C	0.001
Sensor Output	bps	1

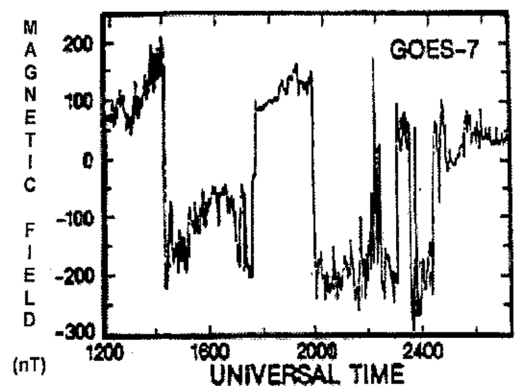

Figure 7. GOES-7 magnetic field measurements.

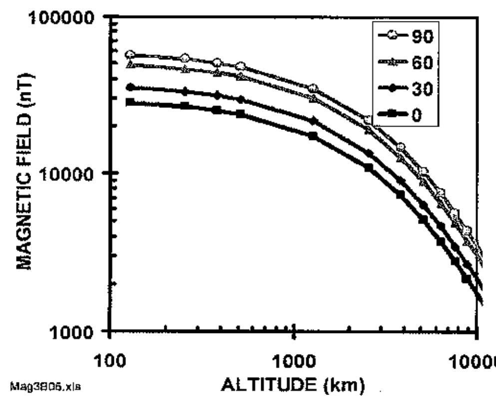

Figure 8. Magnitude of the Earth's magnetic field at different latitudes. [1]

ambient field. In addition the magnetometer can be used for attitude determination—knowledge required to interpret results from the AO and QCM sensors.

Including a magnetometer requires attention to magnetic cleanliness—both within the SEM and on the spacecraft. As the use of a boom is not considered for the SEM, the accuracy depends on uncertainties and variability in the intrinsic spacecraft magnetic moments. A plot of the Earth's magnetic field, shown in Fig. 8, indicates that the field must be known at least to the μT range as the range for a typical magnetometer for attitude control below 6,000 km. [1] Additional specifications for the magnetometer are listed in Table 8.

Table 6. DRAM Specifications

PARAMETER	UNIT	VALUE
Upset: Rate Range	Upsets/bit-s	<0.02
Upset: Threshold	MeV-cm^2/mg	>1
Radiation Hardness	krad	20
Sensor Output	kbps	0.01
Total Bits	Bits	3.4
Manufacturer	--	Micron, Xylinks

Table 7. SRAM Specifications

PARAMETER	UNIT	VALUE
Upset Rate Range	Upsets/bit-s	<0.02
Upset Threshold	MeV-cm2/mg	0.1 to 10
Radiation Hardness	krad	20
Sensor Output	kbps	0.01
Total Bits	Bits	4K
Manufacturer	--	JPL/MOSIS

Table 8. Three-Axis Magnetometer

PARAMETER	UNIT	VALUE
Magnetic Field Range	μT*	±50
Resolution	nT	<10
Sensor Output	bps	0.1
Attitude Accuracy	deg.	<3
*below 6000 km		

Table 9. QCM Specification

PARAMETER	UNIT	VALUE
Temp. Range	°C	-43 to +80
Temp. Accuracy	°C	<±1
Temp. Precision	°C	<±0.2
Sensor Output	bps	1

F. Temperature: Thermal Measurement Unit

The need for temperature measurements is ubiquitous. For example, it is often needed by sensors in order to remove their temperature sensitivity. The most temperature sensitive sensors in the SEM are the QCM, actinometer, and p-FET dosimeters. In addition to sensor specific temperature measurements, a temperature sensor needs to be included as a health check on the data logger electronics. Temperature specifications are listed in Table 9.

G. Conceptual Design

A conceptual layout for the SEM is shown in Fig. 9. It shows the sensors arranged on the top surface of a box that houses the electronics. This design indicates that sensor are located in the same unit and that there is no provision for connecting remotely to a sensor. The data-logger architecture is shown in Fig. 10. It calls for the use of 12-bit ADCs to measure the sensors and a temperature controller to stabilize the temperature of the sensors. In addition the temperature controller is used to heat the QCM's to desorb contaminants.

The operating modes of the sensors have yet to be specified but several system characteristics can be defined. For instance, the temperature of the more sensitive sensors may be elevated to minimize the effects of spacecraft temperature variations. These modes will need to be controlled by a micro-controller or FPGA which will pole each of the sensors and time tag the data resolved to at least 0.1 s. Since the spacecraft will likely be in an elliptical orbit, it is important to know where in the orbit measurement occurred—at an orbital velocity of 8 km/s for example, data will need to be known to within 1 km.

Local data storage will be essential. The amount of storage will depend on the sensors' data rates and the intervals at which the spacecraft requests data. The memory should be non volatile in case of a power outage. The interrogation of the SEM by the spacecraft will likely be variable. The SEM will need to be fault protected against causing a spacecraft fault and from receiving faults from the spacecraft. In addition, the integrity of the data should be protected but this will not be difficult as the sensor data are highly redundant.

The power and data interface to the spacecraft will need to be flexible because of their application specific nature and because data interfacing standards tend to change. At this time, the data interface is specified as being wired rather than wireless. A summary of the system level specification are listed in Table 10. The circuitry, in Fig. 10, shows provision for spare or additional sensors so that other instruments can be accommodated in the future.

Table 10. SEM System-Level Specifications

PARAMETER	UNIT	VALUE
Mass	g	250
Volume	cm^3	250
Power Operating	W	5
Power Quiescent	W	0.1
Temperature Rise	°C	20
Thermal Heat Sink	TBD	TBD
Output	bps	1000
Data Storage	Mbits	1
Fault Protection	TBD	TBD
Time Tag	s	0.1

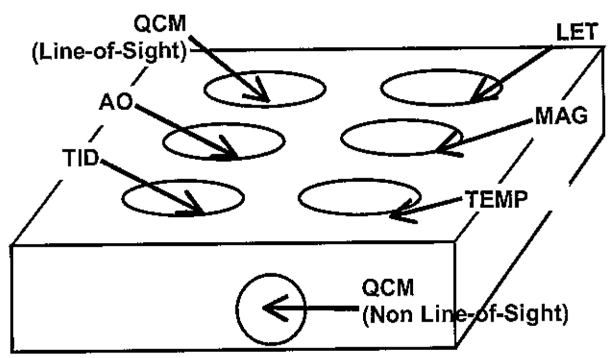

Figure 9. Conceptual layout.

V. Conclusions

This analysis indicates that the SEM will be an important adjunct to and provide key data on the space environment during the flight of the NMP technologies. This will significantly enhance the NMP experimenters' ability to interpret their results and extrapolate or scale them to other NASA Earth and Space Science missions. In addition the SEM will provide a bridge between NMP flights occurring over a number of solar cycles and under a variety of environmental conditions.

Six environments were selected using a best estimate of the environments that are most needed. They are: Contamination, Atomic

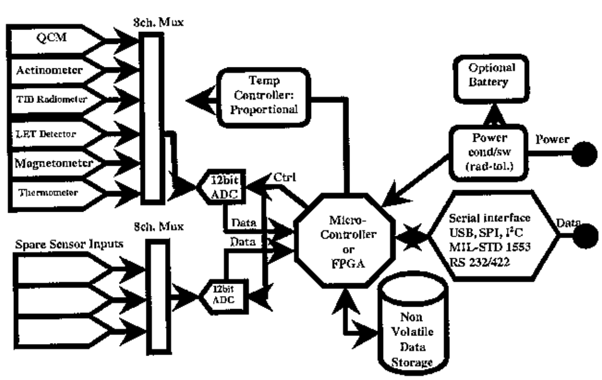

Figure 10. Conceptual SEM circuit.

Oxygen, Ionizing Radiation, Cosmic Radiation, EMI, and Temperature. It is anticipated that these environments can be measured by a suite of sensors fabricated into a small, hockey-puck size package. For now we are limiting our scope to the six environments. If successful, additional environments will be added in the future in a plug-and play manner allowing the SEM to be tailored to the environment of interest. If this effort is approved, we will seek an integrator who can obtain the sensors, package and test them, and supply the SEM on an as needed basis for NMP flights. Hopefully other parts of NASA, DoD and commercial satellite builders will find the SEM useful. By encouraging greater use of the SEM, we hope to lower the unit cost.

Acknowledgments

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