ABSTRACT

The number of spacecraft designed and built over the next century will grow exponentially as communication satellite networks proliferate and NASA continues to push towards the development of miniaturized microspacecraft to replace its traditional "grand tour" crafts. Since costs in the space industry are measured in terms of weight (dollars per pound launched) and reliability, unit costs pale in comparison to launch costs and the cost of replacing an entire vehicle in the case of a catastrophic failure. Space systems present unique application for microelectromechanical systems (MEMS) technology. MEMS technology can be applied to miniaturize many of the components in a space vehicle, and can improve overall reliability.

This paper will identify potential applications of MEMS in a space system, describe space environment factors, and review efforts to develop appropriate packaging and space qualification methodologies. Finally, a flight experiment for testing the performance of typical MEMS devices and packages in the space environment will be described.

INTRODUCTION

Many foresees of MEMS devices have been proposed for application to space systems in order to realize reduction in size, weight, and power consumption at the component level [1, 2]. At the same time, significant issues remain to be examined that have critical influence upon the viability of MEMS devices for space. As a new technology, each individual MEMS device will need to go through some kind of a space qualification process to ensure its reliability and compatibility with the space environment. It is not clear, however, that the traditional qualification methods are applicable to MEMS, and in general these methods are perceived as far too costly. New approaches need to be developed for devices and packages.

The availability of miniature components has caused a shift in the ways spacecraft are architected and designed. The traditional architecture involves spacecraft bus connecting subsystems which themselves consist of individual packaged components. The new architecture aims at eliminating at least one of the levels of integration by packaging entire subsystems as highly-integrated components, eliminating the spacecraft bus entirely in favor of a distributed architecture containing active components with structural elements.

Ensuring the reduction of life-cycle (i.e., development, qualification, integration, launch, and operation) cost is critical for justification of using MEMS on spacecraft. Incremental reduction in total system mass to reduce launch costs is not sufficient reason. There are conventional technologies that are miniature and space qualified (for example, the 25g1iton G2000 and the 7/g AlliedSignal Minimatic provide a 2 axis rate sensing in a hermetic, military-qualified package [3]). The advantage of MEMS lies in the ability to package many devices and their support electronics on a common substrate, significantly reducing mass, power and thermal control requirements of an entire subsytem.

Space qualification processes of such integrated modules will rely heavily on processes developed for microelectronics and multi-chip modules (MCMs). However, MEMS have a different set of failure modes requiring unique analyses and tests. Early flight demonstration will be critical for the reduction of qualification costs.

SPACE APPLICATIONS

MEMS sensors and actuators are most commonly considered in the context of either science instruments or inertial guidance, although they can potentially impact spacecraft subsystems such as propulsion, state-of-health monitoring, or active mechanical and thermal control structures.

Payload Sensors. Science sensors typically fall into one of two categories: remote sensors (for planetary or astrophysical observation at a distance) and in-situ sensors (which measure the physical environment surrounding the sensor, including particles and fields). MEMS-based remote sensor elements might include infrared focal planes (bolometers, thermopiles, or
Golay cells), spectrometers (1-ary-Perot interferometers or gratings), shutters and filters (including those wave filters and resonators), pointing and steering devices, or adaptive optical elements. 'In situ' remote sensing platform requires a large aperture, precision pointing and guidance, and the ability to process and transmit large amounts of data. In the foreseeable future, it is unlikely that such platforms will weigh less than .5 kg independent of the Mi:MS contribution and the case of the thrust ofvalve unit will only be based primarily on performance.

In situ sensors, on the other hand, are often deployed in remote environments under severe mass and power constraints. Since the sensors are typically not deployed in a very local environment, there is typically an advantage to deploying large number of sensors. Incorporation of Mi:MS is usually justifiable simply on the basis of size, mass, power consumption, and cost. These sensors cover the entire scope of physical and chemical phenomena involved in the analysis of liquids, solids, gases, plasmas, or fields in free space. A comprehensive review is well beyond the scope of this paper, except to point out that small (<1 kg) autonomous scientific instruments with communication, data handling, power, and rudimentary mobility (the ability to be remotely deployed and to acquire samples) can revolutionize the study of planetary surfaces. Several small instruments are currently being developed, including miniature penetrators deployed directly from space, and tiny robotic rovers.

Guidance Sensors. Spacecraft attitude, (orientation in free space) and position can be determined initially, optically, or by using known fields. A combination of these methods may be needed depending upon the various mission phases. Inertial sensors include accelerometers with performance requirements of p g f r(1.3)(117 and gyroscopes with better than 1°/h. Usually, attitude can be determined using star trackers with pitch/yaw accuracy better than 1 arc sec and roll better than 10 arc sec, and sun and horizon detectors will better than 1° accuracy. Finally, in satellites orbiting the earth, attitude can be determined using magnetometers with a 1.5-3° accuracy [4]. These goals were to integrated these into a single electronic package similar to the packages used for the spacecraft, with all the necessary power supplies and computers, that can then be integrated into the main computer.

Propulsion. The attitude control of deep space vehicles and the orbit maintenance of satellites is often accomplished by propulsion systems. There are two types of propulsion systems, electric and chemical. The simplest system would be a cold gas propellant, yet the implementation of Mi:MS in such a system does not seem like a reasonable goal at this point due to the high leakage rates in microvalves. Assuming that no more than 10% of the propellant can be lost during a 5-year mission, the required leakage rate cannot exceed 10⁻⁶ sec⁻¹/sec for gas stored at greater than 1000 psi and working against vacuum [5]. Fluid flow controllers are still needed for propellants stored as liquids or solids, even though issues remain regarding propulsive efficiency of the system due to the dominance of the boundary layer and device survivability during thruster burns when flame temperatures are in the 1000°C range. Several micropropulsion systems concepts for ion and chemical thrusters are currently being developed [5, 6, 1], however, major advantages over conventional technologies (such as Moog's 7g cold gas thruster valve unit) will only be realized when many Mi:MS devices are arrayed and entire systems (tank, valves, thrusters) are integrated in a single compact unit with minimal transfer of fluid from storage to space.

SPACE ENVIRONMENT

In many respects the space environment is similar to the environment under the hood of an automobile. Unlike cars, spacecraft also experience severe shock and radiation environments. Moreover, the conditions greatly vary in different types of missions and different phases within a single mission. Table 1 presents a comparison of the automotive and space environmental parameters. The space parameters given are for the earth orbiting Space Technology Research Vehicle (STRV-2) mission which will host the MAPLI-2 experiment described later in this paper.

Table 1: The Automotive vs. the Space Environments

<table>
<thead>
<tr>
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<tbody>
<tr>
<td>Operating Temperature</td>
<td>-40°C to +22°C</td>
<td>&gt;5°C to +25°C</td>
</tr>
<tr>
<td>Thermal Cycling</td>
<td>&gt; 1,000</td>
<td>-25°C to +10°C</td>
</tr>
<tr>
<td>Humidity</td>
<td>100%</td>
<td>35% to 60%</td>
</tr>
<tr>
<td>Vibration</td>
<td>15 g, 10,000 Hz</td>
<td>Up to 0.4 g, 20-2000 Hz</td>
</tr>
<tr>
<td>L/M propulsion</td>
<td>Up to 200 V/m</td>
<td>Up to 70 V/m</td>
</tr>
<tr>
<td>Shock</td>
<td>N/A</td>
<td>20 g, 100[17]</td>
</tr>
<tr>
<td>Radiation</td>
<td>N/A</td>
<td>2090 g, 4-10 kHz</td>
</tr>
<tr>
<td>Depressurization</td>
<td>N/A</td>
<td>106 rads/year</td>
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</table>

1 atm to vacuum at 1 psi/sec
Delicate devices can be easily protected during the harsh launch phase by delaying power up until the end of the phase. Stops and other temporary means can also be used for additional protection. Such protection cannot be used during a pyrotechnics separation event usually used for deployment or separation of various on-board components. Traditionally, this shock has been attenuated through the various joints that separate the pyro device from other sensitive components, but as spacecraft shrink in size, the number of joints is reduced significantly increasing the impact of the shock experienced by microdevices. Each shock presents an opportunity for an interconnect to fail or for a microcrack to form in the inherently stressed atomic structure [9]. Such cracks can grow slowly and eventually lead to failure [10]. A typical shock response spectrum resulting from a small separation nut commonly used in space is shown in Figure 1.

Figure 1: Shock response spectrum 1 foot away from separation nut. Dashed line presents response parallel to the line of separation, solid line presents the response at 90° angle.

The severe micro-environment in space causes various effects on microelectronic systems at expected radiation levels for earth orbiters, measured in Mrem (1 rad = 6.25x10^−6 MeV), vary with altitude and can reach up to 104 rads per year [11]. The total radiation dose due to trapped radiation belts where electron- and protons are present, solar flares causing increased heavy charged particles and galactic cosmic rays carrying high energy charged particles. The radiation is such complicated by the fact that it is statistical in nature and that variation in dosage rate, along with total dose, cause different effects. The result of a recent paper showing enhanced damage at a lower dose rate raises concern with the traditional radiation testing methods bombarding devices with particle at a high rate to accelerate life. [12]. The various types of radiation effects may impact prospective MEMS devices into three ways:

1. On-board analog and digital microelectronics co-fabricated with the MI:MS devices;
2. At the transition points where MI:MS sensors convert a particular form of energy to electrical energy or MI:MS actuators convert electrical energy to another form; and
3. Within the MI:MS device itself.

Table 2 presents the expected physical impact of each type of damage caused by radiation [13]. Proper packaging to protect devices from these effects may prove to be much heavier than the device itself. However, proper design of devices and choice of materials can mitigate the risk of failure associated with these effects.

Table 2: Potential Impact on MI:MS Due to Radiation

<table>
<thead>
<tr>
<th>Radiation Effect</th>
<th>Physical Impact</th>
</tr>
</thead>
<tbody>
<tr>
<td>Ionization damage (electrons, protons)</td>
<td>Charge trapping, interface state growth at oxide-silicon interfaces</td>
</tr>
<tr>
<td>Single Event Upset (protons, galactic cosmic rays)</td>
<td>Deposition of electron-hole pairs by single particle -- current pulse</td>
</tr>
<tr>
<td>Single EventLatchup (protons, galactic cosmic rays)</td>
<td>Localized, self-sustaining high current condition in semiconductor materials</td>
</tr>
<tr>
<td>Single Event1 hard 1 MeV (protons, galactic cosmic rays)</td>
<td>Localized charge deposition results in permanent effect -- &quot;microdose&quot;, rapture</td>
</tr>
<tr>
<td>Single Event Burnout (galactic cosmic rays)</td>
<td>Single particle induces collapse of large voltage across thin oxide</td>
</tr>
<tr>
<td>Displacement Damage (neutrons, protons)</td>
<td>1 displacement of lattice atoms; minority carrier lifetime, doping level effects</td>
</tr>
</tbody>
</table>

PACKAGING AND QUALIFICATION

Packaging techniques for MI:MS borrow heavily from those developed for microelectronics. Similarities include hermeticity and chip-level integration techniques such as MCMs; differences include a unique set of failure modes due to the mechanical nature of MI: MS that are still not very well understood. It is difficult to develop cost-effective qualification techniques when dedicated modeling and simulation tools are not yet available and no complete understanding of when and how macro-scale laws break down.

Packaging for Reliability. Most packaging systems for space parts are hermetic due to a perceived increase in reliability, and to minimize the potential outgassing of materials otherwise encased in a hermetic enclosure. The latter concern is more relevant for contemporary
spacecraft, since outgassing products that might evolve from polymeric materials could redeposit on out-of-desired surfaces, such as optics, solar cells, and instruments. By employing more advanced two and three-dimensional packaging approaches, a systematic reduction in the surface area that might contribute to outgassing can be achieved, since 10-100 times volume reductions in the total ensemble of circuit components and their associated connectors, boards, and harnesses may be realized. The latter (non-component-related packaging elements) create a large outgassing product control problem. Connectors and the associated cables, each of which contain polymeric material, can be reduced through advanced packaging. As such, through outgassing from level one packaging may be more than compensated by the systematic reduction throughout the packaging hierarchy.

That from making a case for abandoning hematicility, the balance of trade-offs suggest a careful consideration of factors. Simple edicts such as “no plastic packaging allowed” seem expedient, but may be shortsighted as a component selection strategy, as they may preclude the most promising options. New packaging approaches, of course, and new technologies required may require a certain level of understanding regarding system level reliability.

Space Qualification. Verifying reliability usually involves understanding failure modes in a hierarchical sense and establishing a means of systematically eliminating their occurrence in particular assemblies or desired product and mission life. The understanding of failure modes can be very involved, usually based on a previous understanding of similar assemblies or elements. As such, new technologies create stress, as they require considerations beyond those associated with more conventional assemblies. Once the most important failure modes are identified, they are reconciled against the likely environmental conditions associated with the mission or product. If such a reconciliation cannot be satisfied, other performed, then the technology must be improved, or qualified, or its expected lifetime environmental conditions must be adjusted. From this point, an attempt is usually made to establish test methods that would aggravate any real failure modes if possible by subjecting assemblies or elements to simulated conditions, usually in some “accelerated” or more intense manner. These screens may then be applied to each assembly or element.

In practice, developers apply sets of test methods, chosen as though from a palette, with one of the largest “palettes” being documents, such as the military standard MIL-STD-883. New technologies create stress by forcing developers to consider a choice of the set of test methods or the “palette” itself. To avoid the real work implied by developing such a qualification approach, it has often been the habit of space systems designers to simply insist upon compliance to a pre-ordained set of test methods in blanket fashion, such as prescribed by documents like MIL-1,-11-38534 (general qualifications of hybrid microcircuits), which has recently been evolving to reflect more contemporary qualification approaches. Certainly the older compliance documents were reinforced by similarities across a great many monolithic integrated circuit or “chip-and-wire” hybrid processes. The advent of newer MCM, three-dimensional packaging, and MIL-MS technologies possess traits which do permit casual simplification of the reliability evaluation at assurance processes. Rather, more flexible compliance strategies are required, such as qualified manufacturers’ line (QML) approaches, which not only allow for board-mediated adjustment of reliability determinations to occur in a continuous manner, but focus on the quality of the process through which assemblies are created, instead of relying on the results of the screens alone. In some cases, QML permits screens to be eliminated partially or completely.

**FLIGHT EXPERIMENT**

The lack of flight heritage is a major impediment to the use of MCM devices in space. While ground testing of other qualification techniques can assure the reliability of a device or an assembly, its space “worthiness” is not proven until it has flown in space. An early flight demonstration of a MCM package is currently being developed cooperatively by Phillips Laboratory (11’-1) and the Jet Propulsion Laboratory (JPL). It is designed to fit in the electronics testbed on board the Space Technology Research Vehicle-2 (STRV-2, a joint U.K./U.S. mission) with its highly elliptical orbit, providing an unusual opportunity to explore the impact of harsh space environmental effects.

The experimental Microsystems and Packaging for Low Power Electronics II (MAPLE II), includes the integration of three Analog Devices ADX102, three Al 97040 microaccelerometers, and a single tunneling microaccelerometer developed at JPL [14]. The sensors are hosted by a low-power (<1.5 W) package that has a microcontroller-based data acquisition system. The experimental configuration provides in-situ monitoring capability for operating the accelerometers in a self-test mode, as projected satellite on orbit accelerations may fall below measurement resolution.
MA P/J-2's purpose is to evaluate the performance of commercial capacitive accelerometers and tunneling accelerometer in the harsh space environment. It is expected to experience mm-c than S.O(K) thermal cycles, each cooling-cycling with a 90-minute orbital period. Temperature extremes are expected to be -25°C and -40°C and maximum total radiation dose is projected at up to 15,000 rad.

The MAP/J-2 block diagram is shown in Figure 3. Power conversion of a single 28 VI X input to 5, ±15 V, and -11.5 V is performed with custom conversion circuitry. The sensed accelerations of each accelerometer are scaled and multiplexed to an Analog Devices AD7752 12-bit analog-to-digital converter (ADC), chosen based on recommendations by the manufacturer as a low-power ADC that had been found to meet custom tests to be tolerant to 15,000 rad total dose. The central processor in this case is a radiation-hardened 8051 "clone," which operates from the hardened 2 Kx8 Harris PROM and 8 Kx8 hardened SRAM (based on a defect 71 SIMOX process). Small-scale analog and digital components were judiciously selected based on availability in a tested format.

Since the STRV-2 acceleration environment is projected to be very high once the satellite is actually in orbit, an artificial acceleration stimulus is provided to the M1-MS devices through the onboard solenoid. The solenoid provides a convenient direct means of "injecting" known test impulses that allow performance degradations to be identified accurately. The experiment's timeline includes an initial 180-second warm-up period to permit initial diagnostics and to precondition the solenoid firing circuitry. Then a solenoid discharge occurs, and all seven accelerometers are monitored for 3 seconds. Next, two 90-second idle cycles occur, during which the solenoid circuit is charged and static environment readings are performed. MAP/J-2 then idles until the host controller requests a download of signature information. The completion download signifies the end of the MAP/J-2 experiment cycle, which is to be performed nominally 4 times per 90-min orbital period.

The packaging techniques used in integrating the 7 M1-MS devices and support electronics into the 5" by 6" printed wiring board include:
- (to be added by Jim)
- The following ground tests are planned for the MAP/J-2 experiment:
  - (to be added by Jim)
  - P1 also has the capability of independently assessing the radiation performance of components through its Cs161, flash x-ray, and low energy x-ray sources.

**SUMMARY**

Various M1-MS applications for space can potentially provide a significant reduction in size and mass of scientific, remote sensing, and communication space vehicles. While this will reduce the cost of the launch phase, it may not reduce the total life cycle cost. The introduction of a new technology such as M1-MS requires appropriate packaging and qualification to
ensure reliability in the harsh space environment. This is a costly process that can only be justified for technologies that significantly increase performance of enable new functions not possible with conventional technologies. MEMS can revolutionize the way spacecraft systems are designed and built by enabling new devices based on in situ sensors, distributed sensors, and actuators, and dense electronic packaging. To accelerate the space qualification process, we are conducting ground experiments and developing the first of many flight experiments.

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