NASA-GSFC Nano-Satellite Technology Development

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Abstract. The scientific understanding of key physical processes between the Sun and Earth require simultaneous measurements from many vantage points in space. Nano-satellite technologies will enable a class of constellation missions for the NASA Space Science Sun-Earth Connections theme. These technologies will also be of great benefit to other NASA science enterprises.

Each nano-satellite will weigh a maximum of 10 kg including the propellant mass. Provisions for orbital maneuvers as well as attitude control, multiple sensors and instruments, and full autonomy will yield a highly capable miniaturized satellite. All onboard electronics will survive a total radiation dose rate of 100 krad/s over a two year mission lifetime. Nano-satellites developed for in-situ measurements will be spin-stabilized, and carry a complement of particles and fields instruments. Nano-satellites developed for remote measurements will be three-axis-stabilized, and carry a complement of imaging and radio wave instruments. Autonomy both onboard the nano-satellites and at the ground stations will minimize the mission operational costs for tracking and managing a constellation. Partnerships with private industry and academic institutions will be utilized for the development, manufacturing, and testing of the nano-satellites.

Key technologies under development will be described, which include: advanced, miniaturized chemical propulsion; miniaturized sensors; highly integrated, compact electronics; autonomous onboard and ground operations; miniaturized onboard methods of orbit determination; onboard RF communications capable of transmitting data to the ground from far distances; lightweight, efficient solar array panels; lightweight, high output battery cells; a miniaturized heat transport system; lightweight yet strong composite materials for the nano-satellite and deployer-ship structures; and simple, reusable ground systems.

Introduction

The primary objective of this development effort is to enable flying tens to hundreds of nano-satellites in a constellation to make multiple remote and in-situ measurements in space. This will revolutionize the scientific investigations of key physical processes explored by the Space Science and Earth Science communities. For the first time scientifically, simultaneous measurements in both space and time will be resolved. To remain consistent with present day Space Science mission cost caps, it is necessary to miniaturize the satellites to maximize the number delivered to space with one launch vehicle. To accomplish this, we plan to develop advanced technology components to make these future spacecraft and their onboard instruments compact, lightweight, low power, low cost, and survivable in a space radiation environment over a two year mission lifetime. These technology components will be matured to a level from which they can be readily adapted to specific science and mission objectives in the onset of the next century. We plan to manufacture and test each nano-satellite for a recurring cost not to exceed $500K. By producing a large quantity of nano-satellites for a given mission, the per unit cost will be reduced to a fraction with respect to traditional procurements of one or two spacecraft per mission. Deployer ships will be developed to carry the nano-satellites in space from launch vehicle separation, and to release them at the appropriate time to achieve their desired orbits. Mission opera-
tional costs will be minimized by the incorporation of both onboard and ground autonomy and use of heuristic systems.

**Mission Overview**

**Baseline Mission**

The first Sun-Earth Connections venture to employ a nano-satellite constellation is the Magnetospheric Constellation mission of the Solar Terrestrial Probes (STP) line. This mission will deploy up to one hundred autonomous nano-satellites in order to perform both in-situ and remote measurements of the magnetosphere. The Magnetospheric Constellation mission is targeted for a FY 2007 launch. Technology readiness is expected by FY 2004, which means that all critical technologies will be at a maturity level to allow easy infusion into the mission's implementation phase with a minimum of risk. NASA currently mandates the mission cost to be less than $120M for the implementation and operations phases. As a result, the manufacturing and testing of one hundred nano-satellites are targeted to remain within a total cost of $50M.

By definition, a nano-satellite weighs 10 kg or less, including the propellant mass. The conceptual nano-satellite for the STP Constellation missions is depicted in figure 1. It is a cylindrical spacecraft with a 30 cm diameter and a height of 10 cm. It is spin-stabilized with its spin axis normal to the ecliptic plane. This configuration will maximize sunlight exposure on its solar cells, which are mounted around the circumference of the cylinder as shown. At least 5 watts of power is expected to be generated by multi-junction solar cells, and batteries will keep spacecraft operations alive during eclipse periods. The nano-satellite will be designed to carry a complement of miniaturized instruments, primarily measuring particles and fields for the STP missions. Figure 2 illustrates a deployer-ship concept for the nano-satellite constellation. Multiple deployer-ships can be used for large nano-satellites constellations. A trade study will be performed to determine the optimum number of nano-satellites per deployer-ship.

As of today, a formal science definition team has not assembled and agreed to the specifics of the Magnetospheric Constellation mission. As a result, a baseline mission was developed as a typical mission scenario in order to identify the technology drivers. In this mission scenario, nano-satellites are placed in varying highly elliptical orbits. Figure 3 presents the baseline mission orbital concept. Every orbit shares the same perigee radius of 3 Re. Apogees vary from 12 Re to 42 Re in 3 Re increments. Eleven unique orbits (3 x 12, 3 x 15, ..., 3 x 42) constitute a 'swarm'. Each swarm contains 22 spacecraft, with each orbit containing two nano-satellites. Figure 4 demonstrates the nano-satellite swarm concept.

Initially the two nano-satellites per orbit will be simultaneously deployed in opposite directions. This will aid in deployer-ship inertia considerations from the angular momentum generated as a result of the deployment. A constellation will require simultaneous operation of
multiple swarms of spacecraft. Swarms all lie in the same orbital plane initially, but have their lines of apses separated by $30^\circ$. Perturbations from the moon, Earth, sun, and other celestial bodies will eventually cause the nano-satellites to become randomly distributed in space.

![Figure 3. Orbit Concept](image1)

The deployer ship will eject the nano-satellites at 3 Re with a minimum spin rate of 20 rpm to ensure sufficient stabilization. Each nano-satellite will boost itself to its particular elliptical orbit by firing its orbital insertion thruster when its spin axis is aligned with the velocity vector. Pulsing of miniature nutation thrusters will then be used to reorient the spin axis of the nano-satellite for optimum sunlight and communication effectiveness.

![Figure 4. Nano-satellite Swarm Concept](image2)

The low power available on the spacecraft for the communications subsystem has created the requirement to send data to Earth only during the portion of each nano-satellite orbit near perigee. This amounts to about 4.1 hours in duration for the 12 Re apogee orbit, and 4.3 hours for the 42 Re apogee orbit. As a consequence, the onboard memory must be sufficient to hold a full orbit's worth of data. The largest orbit satellites must be treated with the highest communications priority as they pass close to Earth. Satellites in the smaller orbits can hold several orbits of data for the same onboard memory size, and thus need to download to Earth less often.

Figures 5 and 6 show the target mass and power allocations for the nano-satellite subsystems. Note that the dry/inert mass of the propulsion system is included in the structures allocation. Also note that figure 5 represents the propellant mass requirement for the 3x42 Re nano-satellites, the most demanding of the baseline mission orbits. This is because it takes more fuel to perform higher apogee orbital maneuvers. The propellant requirements reduce along with the orbit size. This signifies an increasing mass margin (not shown) for the smaller orbit nano-satellites.

![Figure 5. Target Mass Budget](image3)

**Other Applications**
A variety of scientific missions can be enhanced or enabled by the use of nano-satellite constellations. The nano-satellite technologies reduce the cost per satellite and the cost of launching the constellation. This enables missions that require a large number of components. Nano-satellite constellations are modular and robust, capable of accommodating individual failures without compromising the accomplishment of the mission goals.
Spin-stabilized nano-satellites, or ‘nano-spinners’, can provide in-situ measurements for atmospheric, magnetospheric, near interplanetary space, and near Earth radiation research. Three-axis-stabilized nano-satellites, or ‘nano-pointers’, can provide remote measurements for Earth science, solar science, atmospheric, magnetospheric, astrophysics, and interplanetary space research. More specifically, astronomy missions can use nano-satellite constellations to provide affordable very long baseline arrays. A sparsely populated array separated by large distances could provide high resolution information on astronomical objects. These constellations can also provide stereo viewing of the Earth or other celestial objects by using the identical sensors from different angles. This provides a new technique for observing the atmosphere and surface of the Earth. Future constellations may be composed of a combination of nano-spinners and nano-pointers. Additionally, these satellites will fly in formations that are controlled and maintained through inter-satellite communications.

Technologies

Overview
Nano-satellites require technologies that radically reduce the mass and power of components without compromising performance. In addition to miniaturizing components, the technology efforts are looking at methods to integrate similar functions across subsystems. For example, all subsystem electronics, including instruments, could be integrated within the C&DH subsystem. Multifunctional solutions also offer significant savings over traditional approaches. Technology investments are required to develop or adapt components to accommodate the expected radiation environment. Simple, effective methods of thermal control are essential to keep the nano-satellite operational during extreme temperature variations. Autonomy is a critical technology that impacts every subsystem. Constellations with tens to hundreds of spacecraft must be highly autonomous to be practical. The nano-satellite ground system must be kept inexpensive, simple, and made interoperable with other missions.

Propulsion
Many unique challenges exist in developing a propulsion system for a nano-satellite. Present chemical propulsion technologies, which rarely dominate the power system resources on large spacecraft, cannot fit within the power constraints of a spacecraft this small. Also, as propulsion systems decrease in size, the increasing mass/volume ratio of the propellant tanks and the fixed masses of the other components combine to decrease the propellant mass fraction. Furthermore, the standard cost of today’s thrusters and other components is prohibitive when multiplied across tens or hundreds of spacecraft.

Our research to date has led us to focus on chemical propulsion technologies. While certain electric propulsion (EP) technologies (e.g. pulsed plasma and field emission EP) can be made to operate at 1 W input power, they would provide only extremely small impulse bits (on the order of 10-7 to 10-6 N-s). This makes EP far less versatile, as they would not be applicable to spin-stabilized nano-satellites, nor to three-axis-stabilized nano-satellites with significant reorientation requirements. However, developments in ultra low power EP will be followed for possible applicability to a limited number of three-axis-stabilized missions.

Our initial propulsion priorities have been identified by examining the baseline mission. In this mission, propulsion is needed onboard each nano-satellite for two distinct functions. First, each nano-satellite must raise its orbit apogee to the appropriate radius (from 12 to 42 Re). Then, it must reorient the axis of the spinning nano-satellite from the velocity direction (within the orbit plane) to its science attitude (perpendicular to the ecliptic plane). These maneuvers present fairly challenging delta-velocity (ΔV) and attitude-control (ACS) requirements. Table 1 shows the derived ΔV thruster requirements, and table 2 shows the derived ACS thruster requirements. In addition to these requirements, the propellant mass shall be a minimum of 80% of the total propulsion subsystem mass.

<table>
<thead>
<tr>
<th>Table 1. ΔV Thruster Requirements</th>
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<tr>
<td><strong>Total Impulse</strong></td>
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<tr>
<td><strong>Thrust</strong></td>
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<tr>
<td><strong>Input Power (during burn)</strong></td>
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<tr>
<td><strong>Specific Impulse</strong></td>
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<table>
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<th>Table 2. ACS Thruster Requirements</th>
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<tbody>
<tr>
<td><strong>Total Impulse</strong></td>
</tr>
<tr>
<td><strong>Minimum Impulse Bit</strong></td>
</tr>
<tr>
<td><strong>Response Time</strong></td>
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<tr>
<td><strong>Pulse Rate</strong></td>
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The ΔV and ACS thrusters can have independent or shared feed systems, depending on whether a single type of propellant can be used for both applications.
Obviously a liquid propellant, such as hydrazine or an advanced monopropellant, might allow a shared feed system. However, the rather tight thermal control requirement for liquid systems (and the associated heater power) presents a challenge. Alternately, a solid-propellant motor can be used for the ΔV, after which a cold gas micro-thruster would be used to reorient the nano-satellite. Although this option requires separate subsystems, it may actually be easier to implement. The cold gas subsystem could store the propellant either in gaseous or liquid form. To use a liquid-stored propellant, either a gas generator would have to be developed, or use of a liquid with a very low vapor pressure is required. Another approach to meeting the ACS requirements is to use miniature solid propellant gas generators as thrusters. Using micro-machining techniques, dozens of micro-sized “thrusters on a chip” can be manufactured simultaneously. When mounted around the circumference of the spinning nano-satellite, such thrusters could be used for reorientations.

The following products are most desirable for our applications, and are actively being pursued for development: miniaturized, solid propellant ΔV motors with a low cost/mass ignition system; miniaturized liquid propellant thrusters (hydrazine or advanced monopropellant); ultra low power cold gas micro-thrusters; low cost tanks and other feed system components; low power gas generators for liquid-storage cold gas feed systems; and micro-machined solid propellant motors for attitude control firings.

A solid propellant motor is an attractive option to provide the necessary ΔV for injection into the final mission orbit. Because the initial mission apogees of the nano-satellites are not tightly constrained (a good distribution is more important than exact placement), the small ΔV errors typical of a solid motor are acceptable. Also, the spin-stabilized control scheme for the baseline mission ensures that no active attitude control is necessary during the ΔV burn. Therefore a solid motor should not be difficult to integrate into the nano-satellite. However, many challenges remain in the development of an acceptable motor. The motor must be able to accommodate a wide range of ΔV requirements without incurring costly changes to the nano-satellite’s mechanical interface. The safe/arm system, typically a mechanical device on larger motors, must be downsized radically, and use of a non-mechanical switch would require the concurrence of range safety personnel. Also, the thrust level must be limited to ensure that the baseline spin rate of 20 rpm is adequate to maintain the nano-satellite’s attitude. Furthermore, low cost fabrication techniques must be employed to ensure that the purchase of one motor for every nano-satellite is not prohibitively expensive. Finally, an acceptable thermal design must be devised to limit heat input to the nano-satellite from the burned-out motor.

Miniaturized liquid propellant thrusters are another promising technology. Liquid propellants offer storage density and performance comparable to solid propellants, but with the additional capability to restart the engines for multiple burns. It appears that the performance of hydrazine is inadequate for the ΔV portion of the baseline mission. However, hydrazine could be used for missions with lower ΔV requirements, or for attitude control on spin-stabilized or three-axis-stabilized nanosatellites. To develop miniature hydrazine thrusters also entails many challenges. The power required to operate valves must be reduced by an order of magnitude. For three-axis-stabilized applications, the thrust level must be reduced by two or three orders of magnitude. Additionally, smaller thrusters will require novel thermal design approaches to prevent flow choking or premature combustion.

Advanced monopropellants, such as those based on hydroxylammonium nitrate (HAN) and other chemicals, offer all the advantages of hydrazine with several additional benefits, including higher specific impulse, higher density, non-toxicity, and lower freezing point. They may even be usable for the ΔV impulse of cold gas thrusters, they cannot be used for any substantial ΔV on a nano-satellite. However, because the compatibility, stability, and performance of advanced monopropellants have only begun to be characterized, they will be more difficult to develop for nano-satellites than hydrazine. Advanced monopropellants for nano-satellites are definitely a long-term but very promising technology.

One potentially near-term technology is the ultra low power cold gas thruster. Because of the low specific impulse of cold gas thrusters, they cannot be used for any substantial ΔV on a nano-satellite. However, their simplicity and multiple-pulse capability make them a good choice for attitude control. The primary challenge, however, is decreasing the required input power. As with the monopropellant thruster, the power required to open the thruster valve must decrease by an order of magnitude. One likely subsystem configuration is a blowdown feed system with a high pressure gas tank feeding the thruster directly, thereby eliminating any need for a pressure regulator. This configuration enhances the challenge by increasing the force required to lift the poppet off the seat. Initial assessments show that something other than a traditional, normally-closed solenoid valve must be used in order to ensure a reasonable mass for the thruster. There are many possibilities, but none has yet to be demonstrated superior.
Of course, a storage tank is also required to store gas or liquid propellants. Again, because of the quantity of nano-satellites proposed in the baseline mission, the per-unit cost must be minimized. Therefore low cost fabrication techniques are required, whether the tank is metallic or composite. But the design must not be so crude that the tank mass becomes a significant percentage of the total nano-satellite mass. Ideally, the tank can be integrated into the satellite proposed in the baseline mission, the per-unit cost must be minimized. Therefore low cost fabrication mass becomes a significant percentage of the total nano-techiques are required, whether the tank is metallic or composite. This choice, although simple, presents overall problems: the evaporation rate would be highly dependent on temperature; the low thruster inlet pressure would result in poor performance; and the exhaust could possibly condense on cold spacecraft surfaces. Alternately, a gas generator could be used, although this would require some power input.

Finally, small, solid propellant gas generators could be used as ACS thrusters. Forty-eight 50 mN-sec pulses are required to reorient the nano-satellite after it achieves the required altitude. Although this could be achieved either by a monopropellant or cold gas thruster, it could also be achieved using an array of gas generators. Such gas generators are currently under development at NASA’s Lewis Research Center. By incorporating micro-electromechanical systems (MEMS) techniques, the devices can be produced relatively inexpensively. Propellant selection, low-power ignition, and thruster array packaging are some of the challenges ahead for this technology.

Guidance, Navigation & Control
The Guidance Navigation and Control (GN&C) subsystem key technologies and concepts have been identified to enable the successful performance of spin-stabilized and three-axis-stabilized nano-satellites for future missions. Described here are the key technologies for both spin-stabilized and three-axis-stabilized applications.

The initial GN&C requirements are based upon the spin-stabilized baseline mission. Technologies include miniaturization of a sun sensor and horizon crossing indicator. The miniature precision ‘fan’ sun sensor will pinpoint the sun virtually anywhere in the entire celestial sphere with every satellite rotation. The sun sensor will be required to weigh less than 0.25 kg, draw less than 0.1 W, operate on no greater than a 3.3 V bus, and meet a 0.1° resolution requirement. The miniature horizon crossing indicator has a small bore-sight FOV that is mounted at an angle off the spin axis. As the spacecraft rotates, a cone of coverage is formed. The sensor must be capable of detecting Earth from 3 to 5 Re with a pointing accuracy of 0.05°. Total horizon crossing indicator weight and power will be less than 0.2 kg and 0.1 W, respectively.

To precess the spacecraft spin axis from the orbit plane to the ecliptic normal requires a nutation damper in conjunction with thrusters. The damper will reduce a 15° nutation angle in under a few hours. Past experience has shown that this low amount of damping energy may become difficult to measure. Therefore, a trade study will be completed and a prototype damper developed with the proper testing apparatus.

In the future, there will be nano-satellite missions flying in Low Earth Orbit (LEO) which will be three-axis-stabilized. These missions will require a new type of very lightweight, low power GN&C components. Technologies for these applications include a miniaturized gyroscope (micro-gyro), a wide angle earth sensor, and a miniature star tracker. Future efforts will also include a miniature reaction wheel, a miniature three-axis magnetometer, electro-magnets, a two-axis sun sensor, and a two-axis earth sensor.

Of particular interest to LEO missions is the incorporation of GPS onboard the nano-satellites, to eliminate ground-based ephemeris generation. This allows for increased autonomy and simpler, more accurate time resolution onboard the spacecraft. For GPS to fit within the constraints of a nano-satellite, the receiver electronics need to be miniaturized into a layer within the C&DH module.

The three-axis-stabilized LEO missions are further classified into two sets of pointing requirements, coarse and fine. The coarse pointing missions will require the development of miniature wide angle and miniature two-axis earth sensors. Development goals include a factor of four weight reduction while maintaining 0.1° resolution. Fine pointing missions will require a miniature star tracker and a miniature two-axis sun sensor. The initial design iteration will focus on minimizing weight by sacrificing resolution to the order of an arc-minute. The second design iteration will be to bring the resolution to the order of an arc-second.

To minimize propellant, a new class of very lightweight reaction wheels and miniature magnetic coils will be developed. Initial studies have begun to clarify the LEO applications wheel torque and momentum storage requirements. Clarification of these studies will evolve
into an industry wide search into the current state-of-the-art in space and terrestrial applications. A development plan will be based upon the results of these studies.

The GN&C subsystem has identified advanced navigation concepts to meet our constellation objectives. The goal is to develop a set of 10 km resolution navigation concepts that require low power, weight and volume. The level of development will vary as each concept's advantages and disadvantages are identified. A systems level trade study will select the optimal concept for our first nano-satellite constellation mission. This concept will be developed for flight. Currently, there are four concepts that require further study: Navigation using Magnetometer Data, TDRSS Onboard Navigation System (TONS), Navigation using Ground Stations, and Navigation using GPS.

Navigation using Magnetometer Data assumes the spacecraft attitude is known. As the spacecraft passes through a low altitude region of the orbit, the magnetometer data can be compared to an onboard magnetic field model. This information is processed through a Kalman filter to produce an onboard ephemeris solution. During higher altitudes, the ephemeris is propagated.

TONS was successfully completed on the NASA Extreme Ultraviolet Explorer (EUVE) mission. This system uses the doppler shift of the communications signal from TDRSS to generate onboard navigation solutions. This concept will be expanded to cover the highly elliptical orbit requirements of the baseline mission. A study will show the impact of implementing TONS for nano-satellite missions.

After the successful TONS experiment, the system was modified to use the communications doppler shift from ground stations. The Ground Onboard Navigation System (GONS) is currently being developed [1]. As with the TONS, the GONS program will be evaluated for potential use in nano-satellite missions.

### Command & Data Handling

Developing the Command and Data Handling (C&DH) subsystem for a nano-satellite presents some unique challenges, with low mass (0.25 kg) and low power (0.5 W) requirements the biggest drivers. Advanced microelectronic solutions must be developed to meet these challenges as well as to support the diverging requirements of multiple missions. The microelectronics developed must be modular and of scalable packaging to solve the problem of developing a solution to both reduce cost and meet the requirements of various missions. This development will utilize the most cost effective approach, whether infusing commercially driven semiconductor devices into spacecraft applications or partnering with industry to design and develop low cost, low power, low mass, and high capacity data processing devices. The major technologies that will be covered in this section include: lightweight, low power electronics packaging; radiation hard, low power processing platforms; high capacity, low power memory systems; and radiation hard, reconfigurable, field programmable gate arrays (RHrFPGA). Other requirements of a nano-satellite C&DH subsystem are included in table 3.

<table>
<thead>
<tr>
<th>Item</th>
<th>Requirement</th>
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<tbody>
<tr>
<td>Power</td>
<td>0.5 W</td>
</tr>
<tr>
<td>Weight</td>
<td>0.25 kg</td>
</tr>
<tr>
<td>Input Data Rate</td>
<td>2 kbits/s</td>
</tr>
<tr>
<td>Output Data Rate</td>
<td>100 kbits/s</td>
</tr>
<tr>
<td>Data Storage</td>
<td>2 Gbits</td>
</tr>
<tr>
<td>Encoding</td>
<td>Advanced Convolutional; Reed Solomon</td>
</tr>
<tr>
<td>Processing Speed</td>
<td>12 MIPS</td>
</tr>
<tr>
<td>Radiation Tolerance</td>
<td>100 krads total dose</td>
</tr>
</tbody>
</table>

In order to meet the challenge of developing a low mass C&DH, a lightweight and low power electronics packaging method must be used. The packaging method that will be chosen must have a small volume and small footprint (6 cm x 6 cm x variable height). The packaging technique must provide data on programmable substrates to accelerate the process of "prototype to flight" with less cost. The packaging technique must also provide data on compliant interconnects for space use. Figure 7 illustrates one such electronics package, a multi-chip module (MCM) made by Pico Systems Inc. [2].

![MCM Concept](image)
This stackable MCM technique enables modularity and scalability for flexibility in design to meet the needs of multiple missions. The approach shown in figure 7 allows for rapid custom designs, fast design iterations and moderate design costs, while allowing high performance working over required temperature ranges with radiation tolerance.

A combined effort in the reduction of mass, power, size, and cost is underway to produce optimal electronics. The CMOS Ultra Low Power Radiation Tolerant (CULPRIT) system on a chip, and "C&DH in your Palm" are technologies that will enable the power reduction required for nano-satellites. The goals of these technologies are a 20:1 power reduction over current 5 V technology, foundry independence of die production, and radiation tolerance.

Every three years memory technology enables a doubling of memory capacity and a halving of silicon area. Memory trends starting in 1996 are toward a 3.3 V core and a 3.3 V I/O, reducing by 1/3 the power for Gbit size solid state recorders. Trends in packaging technology are enabling denser 3-D stacking in smaller volume packages for multi-bit stacks in the next three to five years. This will be accomplished by incorporating Chip Scale Packaging technology where the package is less than 1.2 times the area of the silicon. DRAM memory will be at the 128 Mbit per die level within the next three years. With these current trends, it appears promising that an off-the-shelf solution is viable for the C&DH subsystem of a nano-satellite.

Another technology enabling a decrease in volume is the RHrFPGA. The RHrFPGA reduces volume by replacing many logic functions/circuits with one die. The RHrFPGA also allows concurrent design by decoupling the logic design from the module, shortens the design schedule, lowers the part count, and eases rework.

To reduce nano-satellite power and mass, the technologies described can be applied to multiple subsystems as well as instrument electronics. This allows for higher levels of electronic integration, effectively combining spacecraft subsystem electronics and instrument electronics into the smallest possible mass, power and volume.

Power Systems
Total spacecraft power is limited by the small satellite size. The sun's power density is on the order of 1.35 kW/m². Assuming 15% conversion efficiency for a 0.3 m x 0.1 m disk shaped spacecraft (cross section of 0.03 m²), with a 67% area coverage, this results in a total electric power of only 4.0 W. Lightweight, efficient solar array panels that minimize the effective array mounting area are needed. Dual or triple junction GaAs solar cells that give a 18% conversion efficiency at end of life (EOL), and assuming a more optimistic area factor of 85%, will result in only 6.2 W at EOL. Small satellites that do not have extended solar panels simply do not intercept a large solar power density and must use the available power very efficiently.

Technology development for GaAs solar cells and LiIon battery cells already exist, however, our desire to improve low power efficiency requires that we experiment with a small satellite model using a 3.3 V power bus. For a small spinning satellite, it is expected that three solar cells will be connected in series along the spin axis, and groups of three will be connected in parallel around the circumference. Each section will generate 3.3 V and rotate into and out of sunlight as a unit. Voltage drops at 3.3 V, bus regulation, circuit protection (e.g. fuse or circuit breaker) and LiIon battery discharge characteristics are being explored.

Highly elliptical orbits in the ecliptic plane where the apogee velocity is very low will cause a several hour eclipse during part of the year. Spacecraft batteries to cover this eclipse period presents a significant mass impact. With only a 10° orbit plane inclination relative to the ecliptic, a satellite in a highly elliptical orbit will always be exposed to the sun whenever it is at least 6° from Earth. For such orbits the maximum eclipse period will be about one hour. Inclusion of spacecraft batteries is then justified. Passive thermal control will be used to keep the spacecraft electronics within 10° C of ambient temperature, and hence not require electric power for heating. Using such a scenario, a battery requirement of about 2 amp-hours at 3.3 V will allow full spacecraft functionality during an eclipse. Twelve AA size LiIon batteries meet the requirement and only weigh 480 grams. For missions where the spacecraft mass impact due to batteries is unacceptable, power for instrument data collection and communication to Earth may be traded off. Structurally integrated battery systems are a longer term solution, and are discussed in the Mechanical section of this paper.

Circuits that have short, high current demands, such as thruster solenoids and fuses, need to be augmented with components that have a lower power density than batteries, but also have lower internal resistance. Ultra capacitors are a candidate for this application.

Miniaturization of the power system electronics (PSE) to meet the weight and size requirements of the nanosatellites is a considerable challenge. The ideal approach is to eliminate the PSE completely, by having a fixed electrical load and batteries provide the needed...
bus regulation. This yields a simplified system consisting of the solar cells, batteries, and minimal circuitry. A more immediate approach to miniaturization is to produce hybrid modules that measure approximately 2" x 1.25" x 0.5" and weigh about 100 grams for each PSE component, namely the solar array regulator, battery regulator, and low voltage power converter. Beyond this, the combination of these three components into one module will reduce the size and weight another order of magnitude.

Thermal
Since missions will involve a constellation of nanosatellites in elliptical orbits with progressively larger apogees, earth shadows lasting several hours can result. Referring to the baseline mission described above, each nanosatellite is a spinning, cylindrical spacecraft with the spin axis normal to the ecliptic plane, with body mounted solar arrays. A study was conducted to investigate several thermal control strategies, including design robustness and the effect of the long earth shadow on each design. The results described indicate basic features of each design strategy and help to guide a thermal design, but do not reflect individual component temperatures due to the simplified nature of the model.

Three thermal configurations were considered: (1) top and bottom of the spacecraft are insulated, the inside of the cylindrical solar array is not insulated allowing internal heat transfer between the internal equipment and the solar array; (2) the entire spacecraft is insulated, top and bottom as well as inside the solar arrays, except for a radiator on top, sized to radiate the internal electrical dissipation; and (3) the internal equipment is thermally isolated as well as possible from an "outside shell" with a controllable two-phase heat transport device, which can be "shut off" during earth shadows, serving as the only thermal coupling between the equipment and a radiator on the outside surface.

For configuration (1), where only the top and bottom are insulated with multilayer insulation (MLI), the basic insun temperatures are set by the thermal optical properties (solar absorptance and emittance) of the solar cells and adjusted, if necessary, to near room temperature with a relatively small radiator area on the sides or end panels of the spacecraft. The key advantage of this configuration is its reliability, or robustness. Since the temperature of the spacecraft is set by a high energy balance (heat in = heat out) dominated by the absorbed solar energy, the operational temperature of the spacecraft is relatively insensitive to top & bottom MLI properties, or, largely, to internal heat dissipation. However, the feature that yields the operational reliability, i.e., the high energy balance, also results in a rapid drop in temperature when the solar load disappears during the earth shadow. During the maximum 475 minute eclipse used for study purposes, internal temperatures dropped by about 60° C, which would result in internal temperatures in the range of -30 to -40° C. At the same time, the solar arrays dropped to a temperature of about -60° C. Since specific components with specific temperature limits have not been identified to date, the feasibility of these end-of-eclipse temperatures must be judged on the basis of general spacecraft equipment. Based on past experience, these end-of-eclipse temperatures may be reasonable, at least as survival temperatures, for at least some spacecraft electronics. Certain other components may have a problem with these temperatures. Also, the potential cold temperature problems will be reduced if further orbital analyses and mission planning determine that actual eclipse periods are shorter than assumed. End-of-eclipse solar array temperatures are not a problem.

Configuration (2) is fully insulated except for a passive radiator sized to yield operational (in-sun) internal temperatures near room temperature. Because of the insulated nature of the design, which has a much smaller overall energy balance than configuration (1), this configuration is much more sensitive to MLI properties, and to internal power dissipation than configuration (1). However, eclipse performance improves. During the ~8 hour eclipse, internal temperatures drop by only about 20° C, a marked improvement, with end-of-eclipse temperatures well within the range of most spacecraft components. It should be noted that the solar arrays, since they are now isolated from the body of the spacecraft, drop to temperatures of about -110° C. Even these solar array temperatures should not pose a problem since the solar arrays of many geosynchronous satellites drop routinely to temperatures of about -150° C during the 72-minute eclipse experienced by these spacecraft at each equinox season. So, if warmer end-of-eclipse equipment temperatures are required, they can be achieved, at the expense of some reliability, with this "passive" thermal design.

The key feature of configuration (3) is that the internal equipment is completely isolated, both radiatively and conductively, from the outside "shell", and the equipment is coupled to an external radiator with a two-phase heat transport device, such as a capillary pumped loop (CPL) or loop heat pipe (LHP). Operational temperatures are again maintained to temperatures of about 20° C nominal with a properly sized radiator. However, the temperature is also totally dependent on the proper operation of the two-phase "loop". The two-phase heat
transport device can be made redundant by the addition of a second loop if single fault tolerance is desired. Note that redundancy is not a consideration for the other two configurations studied. During the ~8 hour eclipse, a further improvement is realized, with internal temperatures dropping by as little as 6°C if the internal payload is well insulated from the exterior of the spacecraft. As in configuration (2), the internal temperatures are sensitive to the MLI effectiveness and whether the internal equipment is conductively coupled or isolated from the bottom of the spacecraft. As in configuration (2), the solar array temperatures drop to about -110°C. For certain equipment or science instruments, the temperature control afforded by this type of "active" design may be necessary.

A moderate amount of technology development will be necessary to enable a two-phase heat transport system for use in a nano-satellite. First, the small size and low heat transport requirements of the nano-satellite will necessitate significant downsizing of today's flight qualified two-phase systems that are ready to fly on EOS AM. This reduction will be accomplished by leveraging recent successful tests of a small, cryogenic two-phase CPL. This miniaturized two-phase technology also has terrestrial applications in medical and recreational equipment. The second area of development will pursue the best method by which to "shut down" the system during nano-satellite eclipses. Currently, a certain amount of heater power is used to "shut down" these systems, but in the nano-satellite application, the goal would be to reduce or eliminate this need for control power during the eclipse.

The results of this study help to establish the direction the baseline thermal design will take as further details of the nano-satellite system design evolve.

RF Communications

The nano-satellite onboard RF communications must be capable of transmitting data to the ground and receiving ground commands. These requirements are complicated by the small size, low mass, and low power available. Table 4 lists the subsystem specifications. The tracking system should be coupled with this communications subsystem, to maximize efficiency in mass and power. The communications subsystem is further complicated by constellations requiring spin-stabilized nano-satellites. A spinning nano-satellite cannot easily point an antenna toward Earth. Therefore, a low gain omni antenna is assumed and communication to the ground must be done near perigee, when the range is 3-5 Earth radii. A large ground antenna and aggressive communications coding are required to achieve reasonable data rates with minimum power. This places an additional burden on the ground stations for both sensitive receivers/bit synchronizers and advanced decoders. These same considerations limit the data rate for satellite-to-satellite communication.

Table 4. RF Communications Specifications

<table>
<thead>
<tr>
<th>Item</th>
<th>Requirement</th>
</tr>
</thead>
<tbody>
<tr>
<td>Mass</td>
<td>0.5 kg</td>
</tr>
<tr>
<td>Power Consumption</td>
<td>0.5 W</td>
</tr>
<tr>
<td>Transmission Data Rate</td>
<td>Up to 100 kbits/s</td>
</tr>
<tr>
<td>Command Reception Data Rate</td>
<td>1 kbits/s</td>
</tr>
<tr>
<td>Range</td>
<td>3-5 Re</td>
</tr>
<tr>
<td>Channel Type</td>
<td>BPSK</td>
</tr>
<tr>
<td>EIRP</td>
<td>0.15 W (-8.2 dBW)</td>
</tr>
<tr>
<td>Carrier Frequency</td>
<td>8470 MHz</td>
</tr>
</tbody>
</table>

Nano-satellites for STP missions are too high above Earth to simultaneously receive the minimum signal requirement from 4 GPS satellites, even when they are at their perigee of 3 Re. The nano-satellites will use ground station tracking data for orbit determination. Since nano-satellites cannot afford the weight of transponders, they will use a small stable oscillator and one-way, return only Doppler tracking. Highly stable oscillators that exhibit low power and low voltage are being developed for the nano-satellite RF system.

Although the inclusion of an onboard command receiver is highly desired, it puts an additional strain on an already challenging nano-satellite mass and power budget. For this reason, the concept of a totally autonomous, receiverless nano-satellite design appears most attractive. However, "receiver on a chip" technology is advancing quickly enough that including a receiver onboard looks feasible. The biggest disadvantage of a receiver now becomes the ground personnel and software needed to support the ability to command the nano-satellite. Command actions taken onboard will of course be limited to basic functions such as "transmit data" because of the lack of redundancy and mechanical functions. Although scenarios have been defined to allow the nano-satellites to autonomously determine when to transmit their stored data, utilizing a receiver to control the telemetry downlink from the ground still has value. This will allow the economy of fewer ground stations than the fully autonomous mode would allow. The capability of uploading flight software changes, as well
as sending a master reset if necessary, exists with an onboard command receiver.

**Mechanical/Structures**

The structural and mechanical technologies promise to provide lower cost, lower weight and improved performance for future spacecraft. The nano-satellite structural/mechanical system encompasses both the nano-satellite itself and the deployer-ship which carries and deploys the nano-satellites in orbit. Although the mechanical/structural aspects of developing a nano-satellite remain quite challenging, the deployer-ship presents its own set of technological challenges.

The nano-satellite mechanical system is planned to be low-cost, lightweight, modular, easily designed and fabricated. To accomplish this, the system must be kept as simple as possible. The ideal nano-satellite mechanical design would consist of a one-piece structure to which all other components are mounted. There would be no moving parts and the structure would also serve as a thermal conductor and radiator, and as the substrate for electronics boards and solar cells. We are striving to reach this ideal through the development of the following enabling technologies.

Multifunctional structures can provide thermal control, serve as substrates for printed circuit boards, and integrate other functions directly into the structural components. Examples of these include diamond facesheet honeycomb panels, and structures with an embedded energy storage system (structural battery system). The diamond facesheet honeycomb panels can serve as a structure, thermal conductor and radiator, and printed circuit board substrate. The diamond facesheet provides 10 times greater thermal conductivity than aluminum and can dissipate heat from high power density electronics modules with a low mass comparable to carbon fiber composites. The structural battery system consists of a honeycomb panel where the cells of a nickel-hydrogen battery (or other high power density cell technology) are formed into a honeycomb core. This approach provides a structural panel with an embedded energy storage system for significantly less mass and volume than if the panel and batteries were integrated separately. Structural materials which can inherently provide some radiation shielding for electronics are also desirable.

Concurrent engineering and fabrication techniques will be used to create a single computer model for the design, analysis (structural, thermal, and dynamic), and fabrication of the nano-satellite and its components. Dynamic modeling capabilities to simulate nano-satellite deployments will provide faster designs and a reduction in the amount of deployment testing required. This approach will significantly lower development costs by reducing duplication of effort, chances for errors, the number of drawings and paperwork required.

Mass production techniques not traditionally used for spaceflight hardware will be used, such as investment casting and injection molding. The quantity of nano-satellites required and mission cost goals restrict the use of high cost, traditional spacecraft fabrication techniques. Options being considered for the nano-satellite structure material are: cast aluminum; cast aluminum-beryllium alloy; injection molded plastic; fiber reinforced plastic; and flat stock composite construction. The material will be selected based on mass, cost, manufacturability, ease of assembly and integration, and suitability of the material to the space environment.

Streamlined testing techniques are needed for up to 100 nano-satellites per mission. Performing a complete test program on each unit would be prohibitively expensive and time consuming. We need to reduce the quantity of testing required while assuring product quality to meet program cost and schedule goals. We plan to develop lot testing and statistical quality control methods to verify quality and structural performance by testing a small subset of the total number of the nano-satellites.

The deployer-ship will carry the nano-satellites and deploy them into their proper transfer orbits. The deployer-ship will consist of a fairly conventional spin-stabilized or three-axis stabilized spacecraft. The deployer-ship release system will impart the required minimum spin rate of 20 rpm to the nano-satellites. The following innovative deployer-ship designs, nano-satellite packaging, and deployment techniques will help accomplish these goals. These options range from simple, low cost designs to complex, higher cost designs. The higher cost options will improve the accuracy of the deployment direction and spin axis orientation.

Spinning deployer-ship, simple “Let-Go” deployment: In this case, the deployer-ship is spinning at 20 rpm with the nano-satellite spin axis aligned with the deployer-ship spin axis. The deployer-ship spin axis is then oriented in the desired direction and the nano-satellite is released by a simple mechanism which lets the nano-satellite go while imparting no additional spin. The nano-satellites are released in opposing pairs to maintain the balance of the deployer-ship.
Stabilized deployer-ship, "Frisbee" deployment: In this case, the deployer-ship is three-axis stabilized and placed in the desired orientation. The nano-satellites are deployed with their spin axis perpendicular to the direction of deployment like a frisbee. The deployment mechanism imparts spin during the deployment.

Stabilized deployer-ship, Axial deployment: This case also utilizes a three-axis stabilized deployer-ship but the spin axis of the nano-satellites are coincident with the direction of deployment. The nano-satellites are spun up to the desired spin rate on a spin platform and then deployed.

Actuator technologies which are small, reliable, and space qualified will be required for the deployment of nano-satellites. We are presently evaluating numerous types including: miniature pyrotechnic devices; paraffin actuators; non-explosive initiators; memory metal devices; and thermal knife release units. The selected actuators will be based on reliability, timing, cost, availability and ease of use.

**Instruments**

Instruments for in-situ and remote measurements must be miniaturized to fit within the mass and volume constraints of a nano-satellite. Power consumption must also be scaled down accordingly. Instrument sensitivities cannot be compromised in the process. Instrument electronics need to be combined with spacecraft subsystem electronics to achieve higher degrees of integration, yielding reduced mass and volume. Instrument software will be designed to evaluate the onboard data and adjust instrument data rates and modes to efficiently capture the data of highest priority.

Traditionally, the NASA Announcements of Opportunity (AO) process has been used to select the institutions providing instruments for a spacecraft mission. A set of guidelines for mass, power, and telemetry rates is usually given for the instruments sought. For the development of a nano-satellite, we see the need to competitively select the instrumenters early, to develop the instruments and spacecraft subsystems synergistically. The personnel from the selected institutions will also be part of the spacecraft design team. A highly integrated spacecraft will result, reducing both time and cost for final spacecraft integration and testing.

Nano-satellites for in-situ measurements, such as those baselined for the STP Constellation missions, will carry a low energy particle detector (electrons and ions) and a magnetic field instrument. These instruments are similar to those used on the NASA WIND satellite. The particle detector will explore the suprathermal (just above solar wind plasma energy to 1 MeV) particle population between the solar wind and the low energy cosmic rays. It will also study particle acceleration and transport, wave-particle interactions, and monitor Earth’s magnetosphere particle input and output [3][4].

The particle detector contains a semi-conductor telescope using ion implanted silicon detectors to measure electrons, and ions with energy above about 20 keV. A magnet is used to separate electrons and ions. An angular resolution of about $20^\circ \times 30^\circ$ will be covered over 4 pi steradians in one nano-satellite rotation. One of the challenges in reducing the mass and power consumption of the particle detector is the miniaturization of the high voltage power supply.

The magnetometer consists of a tri-axial fluxgate sensor and an electronics module. The instrument sensor is mounted on a deployable boom, while the electronics module is placed inside the spacecraft structure. The sensor can be made small enough today to be used on a nano-satellite. The challenge remains to reduce the electronics module to the size of the C&DH unit, or less, while maintaining the sensitivity and accuracy of present day, larger sized magnetometers.

It is scientifically desirable to fly a sensitive electric field instrument to complement the magnetoplasma measurements made onboard spin-stabilized nano-satellites. However, the development of a miniaturized electric field detector presents quite a challenge. Present day spacecraft electric field instruments require extremely long booms to make sensitive measurements. A long boom is not feasible for a nano-satellite since it creates a significant mass impact, deployment problems, and stability concerns for the spinning platform. Innovative techniques need to be developed to make the required electric field measurements without the need for a long boom.

Nano-satellites for remote sensing will be three-axis-stabilized, and carry a complement of instruments including multispectral imagers, various spectrometers, and altimeters. Present day instruments for Earth science missions are typically on the order of 50 kg and 50 W. Great strides are needed to reduce the mass and power of these types of instruments to the <1 kg, 1 W class. Miniaturized telescopes and interferometers are also desired for these applications. Common sensors flown on separate nano-satellites in a tight formation could create a large virtual instrument. Instruments could initiate communications across nano-satellites to coordinate observations and to communicate the identification of interesting scientific events.
Ground Systems

Figure 8 shows the ground system concept for a nano-satellite constellation. The large number of spacecraft in a constellation is a challenge to the ground system in getting all of the data to the users. In the baseline mission, there are times when up to nine spacecraft would be within communications range of a ground station at a single time. We have modeled the ground station contacts and can support the constellation with only two stations, located on opposite sides of the earth. The schedulers will prioritize the contacts, with the spacecraft in the higher period orbits getting priority. Spacecraft in the lower period orbits have more opportunities to dump their data, and therefore can have lower priority without risking any data loss.

Users on nano-satellite constellations will need new tools to manage the missions and the resulting science data. Users will use simulation tools to investigate changes to the constellation configuration for missions whose formation can be adjusted. Users will be able to explore options to improve the science after they have evaluated the initial results of the constellation. The large amounts of data that can be generated by large constellations presents challenges to users. Data fusion tools will be needed to integrate all of this data. Users will need browse tools to locate data, and visualization tools to use the data.

The large number of spacecraft is a configuration control challenge for the tracking data, the schedules, the command loads, the science data, and the engineering data. The ground system will use IDs, color coded user interfaces, and other techniques to ensure that the operators and users can keep track of the data associated with a particular satellite. Constellation concepts beyond the first STP mission will influence the ground system and operations. Constellations that fly in a close formation can benefit by the use of inter-satellite communications to reduce ground station contention. The data would flow from a single spacecraft to the ground, instead of coming from every spacecraft. Communications protocols for inter-satellite communications will be investigated in the future. Constellations in low Earth orbit can use GPS to monitor and control the formation, eliminating the requirement to perform these functions on the ground.

Users on nano-satellite constellations will need new tools to manage the missions and the resulting science data. Users will use simulation tools to investigate changes to the constellation configuration for missions whose formation can be adjusted. Users will be able to explore options to improve the science after they have evaluated the initial results of the constellation. The large amounts of data that can be generated by large constellations presents challenges to users. Data fusion tools will be needed to integrate all of this data. Users will need browse tools to locate data, and visualization tools to use the data.

Autonomy

Support costs are high if the mission operations and data analysis paradigm for a single satellite mission is scaled to support a constellation mission. Autonomy onboard the spacecraft and on the ground is therefore required to ensure that science objectives are efficiently and inexpensively met.
Nano-satellite autonomy will make use of onboard and ground-based remote agents, with the overarching goal of maximizing the scientific return from each satellite during the mission lifetime. The remote agents achieve this goal by monitoring and appropriately controlling spacecraft subsystems. Additionally, the onboard agent monitors the full complement of spacecraft sensors and instruments to heuristically separate scientific events of interest from background events, thereby intelligently fitting the science data within allocated spacecraft storage resources.

Nano-satellites with distant orbits are out of communications range of a ground station for nearly a week. Spacecraft subsystems could be compromised if faults occurring during this blackout period were not readily addressed. An unacceptable loss of scientific data could also occur. Therefore, the onboard agent will incorporate the capability to detect, diagnose and recover from faults. Corrective actions will be consistent with a maximum science goal: the actions taken will attempt to maximize scientific return, not necessarily maximally protect the spacecraft.

Certain failure scenarios may not be correctable by the onboard agent. These faults will be deferred to the ground agent for handling. Each spacecraft will include data in its telemetry stream on the health and status of each subsystem and a history of commands autonomously issued since the last ground contact. The ground system will then attempt to diagnose problems based on this data. Since the ground agent will have greater processing power than the onboard agent, the probability of resolving faults is higher. Additionally, collective knowledge of actions taken by all satellites in the constellation will reside within the ground system by virtue of the data dumps made during each contact. From this data the agent can detect trends and systematic conditions not otherwise observable onboard the spacecraft. Once corrective actions are determined, they can be implemented by uplinking commands or by modifying the onboard agent. Scenarios that cannot be not addressed by the autonomous systems will be deferred to a human operator.

Data acquisition rates for Earth-orbiting missions are projected to exceed terabits/sec. The resources required to store and transmit the resulting data sets would increase the size and mass of onboard systems. Furthermore, scientific return could be compromised when only a fraction of the massive volumes of data generated by instruments are analyzed because of limited data analysis budgets. To address this issue, the onboard remote agent will use heuristic techniques to classify events. Instead of viewing data from the spacecraft sensors and instruments as independent streams, the heuristic system will analyze the full complement of inputs to assign degrees of interest to the science data. High interest data will be stored and low interest data rejected. This intelligent filtering technique will thus reduce storage, RF and ground analysis requirements.

These highly autonomous systems will present a unique set of challenges not only to the system designers, but also to those involved in spacecraft testing. Careful consideration must be given to the design of the test program to ensure that the state-space of the remote agents is validated and verified. It is equally important to implement this program in a cost-effective manner. However, we could likely justify exerting considerable resources to address this issue since the methods developed to solve these challenges can be applied to numerous missions.

**Technology Transfer / Spinoffs**

In addition to enabling a wide array of scientific space missions, nano-satellite technology will have applications to a variety of industries. Such technology transfers, or “spinoffs,” have been and will remain an important link between NASA and other organizations. Several examples follow.

Two of the more versatile propulsion technologies are miniaturized ignition systems and ultra low power control valves. The former will increase the efficiency of many gas-generating or explosive devices, from air bags to pneumatic hand tools. The latter will enable the incorporation of precise and reliable fluid control into an ever-increasing number of medical devices, automotive systems, and aircraft systems.

The C&DH subsystem requires rugged, radiation tolerant, low power, and lightweight electronics. Once developed, this technology can improve many types of remote and mobile devices. Portable medical devices, advanced aircraft systems, and mobile communications equipment all can benefit from the C&DH characteristics.

The miniaturized two-phase heat transfer technology described in the thermal section has several potential terrestrial and commercial applications. A patent has been awarded for a “bio-CPL”, which can be applied to utilize excess body heat to warm appendages such as hands and feet in medical applications as well as for recreational equipment. Additional commercial possibilities exist in energy management for a variety of process and equipment applications.
Some of the more aggressive communications coding techniques, those with gains similar to turbo code, will become more routinely accepted and incorporated into commercial ground stations. Along with these coding techniques that allow "all" errors to be corrected in very weak signals, improvement in the quality of bit synchronizers is expected, which convert noisy analog inputs into clean, digital outputs. These advances will improve ground-to-ground as well as space-to-ground communications. Another benefit to ground-to-ground communications may be found in the new protocols developed in support of those nano-satellite missions that require inter-spacecraft communications.

The goal of the autonomy R&D effort is to use electronic and mechanical systems to meet mission objectives efficiently and effectively. This is done for nano-satellites by largely removing human involvement from the control loop. Manpower would be costly and ineffective since the spacecraft are frequently out of ground contact. This reasoning can be applied to any system, whether it be a commercial communications satellite, a ground-based robot, or a production line. Of course each effort will be implemented differently.

**Conclusion**

The nano-satellite technology initiative is well underway at NASA's Goddard Space Flight Center, in conjunction with private industry and academic institutions, to enable a class of constellation missions for the NASA Space Science Sun-Earth Connections theme as well as other NASA science enterprises.

Each nano-satellite will be an autonomous, highly capable miniaturized satellite with a maximum mass of 10 kg, and designed for a two year mission life. Provisions for orbital maneuvers, attitude control, onboard orbit determination, and command and data handling will be included. Fully capable power and thermal systems, RF communications, multiple sensors, and scientific instruments will be integrated on an efficient structure. Nano-satellites developed for in-situ measurements will be spin-stabilized, and those developed for remote measurements will be three-axis-stabilized. Autonomy both onboard the nano-satellites and at the ground stations will minimize the mission operational costs for tracking and managing a constellation.

Key technologies being actively pursued include miniaturized propulsion systems, sensors, electronics, heat transport systems, tracking techniques for orbit determination, autonomy, lightweight batteries, higher efficiency solar arrays, and advanced structural materials. Deployer-ships will carry and deploy a constellation of up to 100 nano-satellites, delivered to space by one launch vehicle. This initiative is scheduled to produce the first generation of mature technologies by 2004, with the launch of the first nano-satellite constellation in 2007.

Partnerships with other NASA centers, other government agencies, private industry, universities, and foreign institutions are in the process of being established. Areas we can benefit from partnering include the manufacturing and testing of up to 100 nano-satellites per mission, the development of multifunctional structures, and integration of instrument sensors and electronics with the spacecraft subsystems. As the nano-satellite initiative matures, we will be encouraging more partnerships. As a result, we expect the number of partnerships formed to increase with time. Partnering with other talented organizations will facilitate the successful technology development required for enabling multi-point science measurements in space by nano-satellite constellations.

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References


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