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. NASA/GSFC nano-satellite technologies will enable a class of constellation missions for the Office of Space Sciences Sun-Earth Connections theme. 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 krads 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 sensing 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. In this paper we describe key technologies that are currently under development to support constellation class missions.

1. Introduction

The primary objective of the NASA/GSFC development effort described herein is to enable flying tens to hundreds of nano-satellites that will perform remote and coordinated in-situ measurements in space. This will revolutionize the scientific investigations of key physical processes explored by the Space Science and Earth Science communities. To remain consistent with present day Solar Terrestrial Probe Line cost caps, it is necessary to miniaturize the satellites so as 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 able to survive their radiation environment over a two year mission lifetime. These technology components will be readily adaptable to any Constellation-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 small fraction of satellite procurements for traditional missions. 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 operation costs will be minimized by the incorporation of both onboard and ground autonomy and use of heuristic systems.

2. Mission Overview

2.1 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 2008 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 minimum 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 is 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 will be generated by multi-junction solar cells, and batteries will keep spacecraft operations alive during eclipse periods. The nano-satellite will carry a complement of miniature instruments, measuring particles and fields. Figure 2 illustrates a deployer-ship concept for the nano-satellite constellation. Multiple deployer-ships can be used for large nano-satellite constellations. A trade study will be performed to determine the optimum number of nano-satellites per deployer-ship.

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 several highly elliptical orbits. Figure 3 presents the baseline mission orbital concept. Every orbit shares the same perigee radius of 3 R_E. Apogees vary from 12 R_E to 42 R_E in 3 R_E increments. 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. Perturbations from the moon, Earth and Sun, and other celestial bodies will eventually cause the nano-satellites to become randomly distributed in space.

The deployer ship will eject the nano-satellites at 3 R_E 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 thrusters will then be used to reorient the spin axis of the nano-satellite for optimum sunlight and communication effectiveness.

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 R_E apogee orbit, and 4.3 hours for the 42 R_E 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 when 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 4 and 5 show the target mass and power allocations for the

Science Closure and Enabling Technologies for Constellation Class Missions, edited by V. Angelopoulos and P. V. Panetta, pp. 115-122 , UC Berkeley, Calif., 1998.

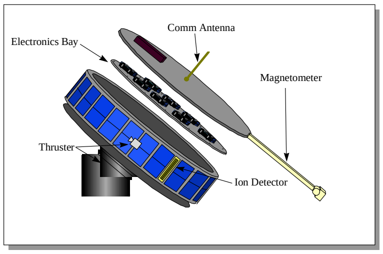

Figure 1. STP Nano-satellite Concept

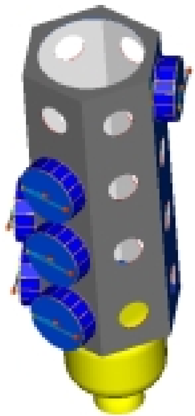

Figure 2. Deployer-ship Concept

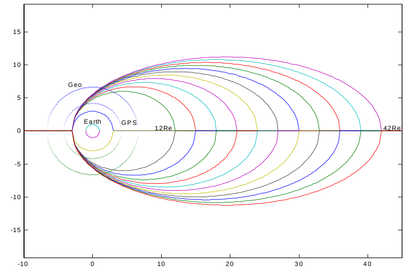

Figure 3. Orbit Concept

nano-satellite subsystems. The dry mass of the propulsion subsystem is included in the structures allocation. Also note that Figure 5 represents the propellant mass requirement for the 3x42 R_E nano-satellites, the most demanding of the baseline mission orbits.

3. Technologies

3.1 Overview

Nano-satellites require technologies that radically reduce the mass and power of components without compromising performance. In addition to miniaturizing components, we 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 inter-operable with other missions.

3.2 Propulsion

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 provide extremely small impulse bits (on the order of 10^{-7} to 10^{-6} N-s) making them currently far less versatile than chemical propulsion. 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 R_E). 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 velocity change (ΔV) and attitude-control (ACS) requirements. Table 1 shows the derived ΔV thruster requirements, and Table 2 shows the derived ACS thruster requirements.

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.

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. However, many challenges remain in the development of an acceptable motor. The motor must be able to accommodate a 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. 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. Low cost fabrication techniques must be revised. Finally, a thermal design that limits 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 if the autonomous nanosatellite weight is to be kept below 10 kg. However, hydrazine could be used for missions with lower ΔV requirements, or for attitude control on spin-stabilized or three-axis-stabilized nano-satellites. 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

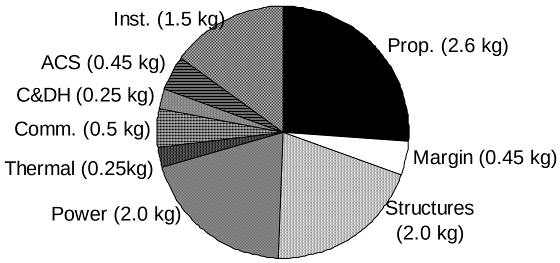

Figure 4. Target Mass Budget

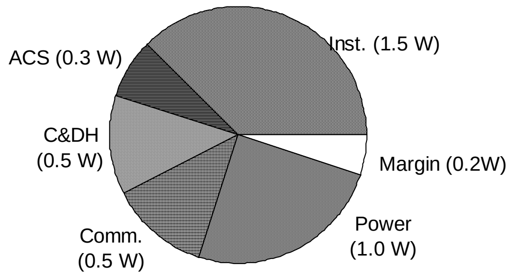

Figure 5. Target Power Budget

Total Impulse	3000 - 7000 N-sec
Thrust	445 N maximum
Input Power (during burn)	<1 W
Specific Impulse	280 sec

Table 1. ΔV Thruster Requirements

Total Impulse	2.4 N-sec
Minimum Impulse Bit	0.044 N-sec
Response Time	<0.005 sec
Pulse Rate	1 Hz

Table 2. ACS Thruster Requirements

advantages of hydrazine with several additional benefits, including higher specific impulse, higher density, non-toxicity, and lower freezing point. They represent 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 but their simplicity and multiple-pulse capability make them a good choice for attitude control. The primary challenge is decreasing the required input power for the thruster valve 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. Initial assessments a non-traditional, normally-closed solenoid valve must be used in order to ensure a reasonable mass for the thruster.

As mentioned above, the propellant in a cold gas subsystem could also be stored as a liquid. The gas might be generated by choosing a liquid with a very high vapor pressure. This choice, although simple, presents several 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 a 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.

3.3 Guidance, Navigation & Control

Guidance Navigation and Control (GN&C) subsystem key technologies and concepts have been identified to enable the successful altitude determination of spin-stabilized and three-axis-stabilized nano-satellites for future missions. They 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 watts, operate on no greater than a 3.3 volt 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 R_E with a pointing accuracy of 0.05°. Total horizon crossing indicator weight and power will be less than 0.2 kg and 0.1 watt, 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.

We have 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.

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 [*Gramling,* 1996]. As with the TONS, the GONS program will be evaluated for potential use in nano-satellite missions.

Item	Requirement
Power	0.5 Watts
Weight	0.25 kg
Input Data Rate	2 kbits/s
Output Data Rate	100 kbits/s
Data Storage	2 Gbits
Encoding	Advanced Convolutional; Reed Solomon
Processing Speed	12 MIPS
Radiation Tolerance	100 krads total dose

Table 3. C&DH Requirements

Of particular interest to Constellation 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.

3.4 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 being the biggest drivers. Advanced microelectronic solutions must be developed to meet these challenges. The microelectronics developed must be modular and of scalable packaging 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 in the design and development of 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). The requirements of a nano-satellite C&DH subsystem are included in Table 3.

In order to develop 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 "proto-type to flight" with less cost. The packaging technique must also provide data on compliant interconnects for space use. Figure 6 illustrates one such electronics package, a multi-chip module (MCM) made by Pico Systems Inc. [*Banker et al.*, 1998].

This stackable MCM technique enables modularity and scalability for flexibility in design to meet the needs of multiple missions. The approach shown in Figure 6 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

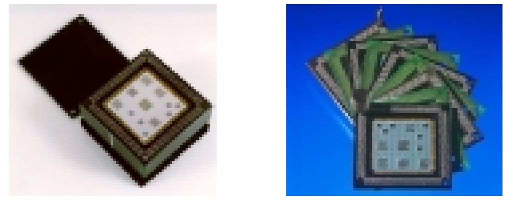

Figure 6. MCM Concept

for nano-satellites. The goals of these technologies are a 20:1 power reduction over current 5 Volt technology, foundry independence of die production, and radiation tolerance.

Every three years memory technology advances enable 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 radiation hard, reconfigurable, field programmable gate array. 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.

The above technologies allow for higher levels of electronic integration, effectively combining spacecraft subsystem electronics and instrument electronics into the smallest possible mass, power and volume.

3.5 Power Systems

Total spacecraft power is limited by the small satellite size. The Sun's power density is 1.35 kW/m^2. Assuming 15% conversion efficiency for a 0.3m x 0.1m disk shaped spacecraft (cross section of 0.03m^2), with a 67% area coverage, this results in a total electric power of only 4.0 watts. Lightweight, efficient solar array panels that minimize the effective array mounting area are needed. Dual or triple junction GaAs solar cells that give 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. 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 volts, bus regulation, circuit protection (e.g. fuse or circuit breaker) and LiIon battery discharge characteristics are being studied.

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. However, only a 10° orbit plane inclination relative to the ecliptic, will reduce the maximum eclipse period to 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 will not require electric power for heating. Using such a scenario, a battery requirement of about 2 amphours at 3.3 volts will allow full spacecraft functionality during an eclipse. Twelve AA size LiIon batteries meet the requirement and only weigh 480 grams.

Circuits that have 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 nano-satellites 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. The combination of these three components into one module will reduce the size and weight another order of magnitude.

3.6 Thermal

Although an inclination change by 10° renders maximum shadows below 2 hours, we hereby study the case of a maximum 8 hour shadow for the purpose of generality. We investigated several thermal control strategies from the viewpoint of design robustness and the effect of the long earth shadow on each design.

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.

The key advantage of configuration (1) 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 and bottom multilayer insulation (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 8 hour 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. Based on past experience, these end-of-eclipse temperatures are reasonable.

Because configuration (2) has a much smaller overall energy balance than configuration (1), it is much more sensitive to MLI properties and to internal power dissipation. 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. For example, 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.

The key feature of configuration (3) is that the equipment is coupled to an external radiator only 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, 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 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. The small size and low heat transport requirements of the nano-satellite will necessitate significant downsizing of today's flight qualified two-phase systems. This reduction will be accomplished by leveraging recent successful tests of a small, cryogenic two-phase CPL.

3.7 RF Communications

The onboard RF subsystem must be small, light, and low power. 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 communications must take place near perigee, when the range is 3-5 Earth radii. A large ground antenna and high compression must be used 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.

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

Item	Specification
Mass	0.5 kg
Power Consumption	0.5 W
Transmission Data Rate	Up to 100 kbits/s
Command Reception Data Rate	1 kbit/s
Range	3-5 Re
Channel Type	BPSK
EIRP	0.15 W (-8.2 dbW)
Carrier Frequency	8470 MHz

Table 4. RF Communications Specifications

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. The capability of uploading flight software changes, as well as sending a master reset if necessary, would also exist with such an onboard command receiver.

3.8 Mechanical/Structures

The nano-satellite mechanical system will be kept as simple as possible. The ideal nano-satellite mechanical design should consist of a one-piece structure to which all other components are mounted.

Multifunctional structures can provide thermal control, shielding and serve as substrates for printed circuit boards. For example, diamond facesheet honeycomb panels can serve as a structure, thermal conductor and radiator, and printed circuit board substrates. 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. Another example is the structural battery system. It consists of a honeycomb panel whose core is filled with the cells of a nickel-hydrogen battery (or other flight qualified cell technology).

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 casting and injection molding. 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 for the space environment.

Streamlined testing is 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 be a 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 help accomplish these goals.

A spinning deployer-ship with 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.

A stabilized deployer-ship with "Frisbee" deployment: In this case, 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.

A stabilized deployer-ship with axial deployment: 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.

Actuators 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.

3.9 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 Announcement 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, and 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 [*Lin et al.*, 1995];[*Lepping et al.*, 1995].

One of the targets for reducing the mass of the particle detectors 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 for magnetometers as well as particle nanosatellite instruments remains to reduce the electronics modules to a fraction of the C&DH unit, while maintaining the sensitivity and accuracy of present day, larger-size designs.

3.10 Ground Systems

Figure 7 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

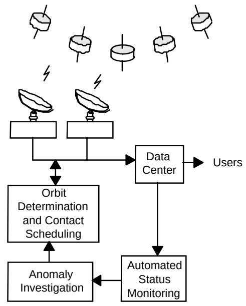

Figure 7. Nano-satellite Ground System Concept

have lower priority without risking any data loss.

Since the nano-satellites are autonomous, the operations concept for a mission requires only a few operators to determine the orbits of the spacecraft, schedule the ground stations, and to investigate anomalies on the spacecraft. Automated systems will monitor the housekeeping data from the spacecraft and they will flag problems for the spacecraft engineers to investigate. The large number of spacecraft allows the risk management to be different for this mission than for single spacecraft missions.

Except for commands to initiate the data downlink, the ground system will not command the nano-satellites for normal operations. The only commands that the ground system sends would be program loads to resolve or work around problems and failures.

The large number of spacecraft is a configuration control challenge for the data tracking, 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. 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.

3.11 Autonomy

Support costs are high if single-satellite mission operations and data analysis practices are scaled to 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.

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. 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.

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.

4. 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.

5. 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 2008.

Partnerships with other NASA centers, other government agencies, private industry, universities, and foreign institutions are currently being established in the areas of 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.

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