Project Spartnik: Microsatellite Design, Construction, Testing, and Operation by Undergraduate Students at San José State University

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Spartnik is a microsatellite under construction at San José State University. Three distinct payloads are being designed, fabricated, tested, and integrated by students solely at the undergraduate level. These payloads consist of a digital color camera, a Micro-Meteorite Impact Detector, and a fairly sophisticated communications package that transmits and receives data on the amateur radio frequency band. As part of a year-long senior design project, students are distributed into seven separate subsystems in which they learn about the design and construction of their specific components and how they interact in a satellite as a whole. Significant progress has been made in all subsystem areas, with project completion slated for late 1998.

Introduction

Microsatellite Project Spartnik was first initiated in 1993 with the express intent of giving students practical, hands-on experience in the design, construction, testing, and operation of actual spacecraft hardware. Previous senior design projects consisted of theoretical paper and mock-up designs. As part of their capstone course of their senior year, undergraduate students from Aerospace, Mechanical, Computer, and Electrical Engineering disciplines, as well as Computer Science students, with help from local industry mentors have significantly advanced the state of the art by producing a small, robust, highly integrated microsatellite.

This will be the first satellite built by San José State University (SJSU). In honor of Sputnik, the Russian built satellite that was the first to orbit the Earth successfully, and in honor of the SJSU nickname, the Spartan, our design team agreed to designate our satellite as “Spartnik”.

Mission Statement

The primary objective of this project is to educate San José State University students on how to design and manufacture a satellite in a real-world setting. Local industry mentors from Lockheed Martin, Space Systems/Loral, and many other companies who are donating engineering guidance and/or hardware are assisting this team.

The experimental payload consists of a communications package (which will transmit on the amateur radio frequencies), a color digital camera, and a Micro-Meteorite Impact Detector (MMID). All the payloads were selected to demonstrate the usefulness of microsatellites as economical platforms for space related experiments.

Mission Objectives

As discussed in the mission statement, the primary objectives were the education of students in spacecraft design and manufacturing and to demonstrate the usefulness of constructing an inexpensive microsatellite. Along with the primary objective, the design team has outlined other mission objectives to be accomplished by the Spartnik project:

- To manufacture and secure the launch of a reliable, low-cost satellite, which may be used for both scientific research and educational purposes;
- To provide practical design experience in a team environment for senior students in preparation of integration into the industry work force;
- To show the feasibility of a cooperative effort between academia and industrial sectors in space vehicle and mission design;
- To promote Aerospace studies within the Silicon Valley youth community while recruiting and
exposing prospective students to the SJSU Aerospace Engineering Program;
• To demonstrate the ease and low-cost potential of satellite imaging technology and how it can be used in our everyday lives;
• To educate young people about the importance of space exploration and the necessity of space-based research;
• To provide a mobile and versatile platform on which to demonstrate scientific techniques and theory and to encourage students and others to become involved in aerospace education and research;
• To provide hands-on opportunities for undergraduate students in the field of satellite communication, tracking, maintenance, manufacturing and related technologies.

Mission Requirements and Constraints

The requirements that have been placed on the Spartnik design team are:
• Meet the secondary payload constraints for as many launch vehicles as possible;
• Design to multiple orbit configurations, ranging from 450 to 900 km altitude and at an inclination greater than 40°;
• Support all payloads, maximizing the operational life of each in a space environment;
• Overall spacecraft operational life should be a minimum of two years;
• Ease of assembly and disassembly for manufacturing and maintenance;
• Use as many acceptable, non-space rated parts as possible;
• Design spacecraft so that a propulsion system is not needed;
• Use of passive and autonomous systems where possible;
• Communications system should use HAM frequencies for uplink and downlink.

Early identification of the design drivers of the satellite helped the students focus on the priority issues of the design from the start. This enabled a design process with clear direction and saved time by requiring minimal design iterations of the spacecraft.

Due to the limited budget of the Spartnik project, the satellite will be placed into a Low Earth Orbit (LEO) as a secondary payload through a donated launch. Since we do not yet know which launch vehicle Spartnik will be utilizing, we need to meet as many launch vehicle secondary payload constraints as possible. By meeting these constraints, we will increase our chances of receiving a donated launch.

Additional unknown parameters, dependent upon the launch vehicle, were the orbital elements. We therefore designed the satellite to meet all possible orbits. The Spartnik team has designed to 12 candidate orbits which fall within the range of 450 to 900 km altitude and orbit inclinations greater than 40°. These candidate orbits are 450, 500, and 700-km altitudes at 40, 45, 60, and 90° inclinations each. These orbits, along with the analysis of each subsystem, give approximate satellite performance in any LEO.

An overall requirement on all subsystems was to support and maintain the payloads. Their goal was to maximize the operating life of the payloads, thus obtaining the maximum amount of data from each payload.

Along with maximizing the life of the payloads, the satellite was designed to have a minimum of two years operational life on orbit. The design team was required to identify all components that will be affected by this time constraint and either modify or design them so they can meet or exceed the minimum operational life.

The spacecraft bus was designed for ease of assembly and disassembly. This improves manufacturing time, ease of maintenance and final assembly before attaching to the launch vehicle. In addition, it facilitates the rapid replacement of any components, should the need arise.

The next requirement placed on the project was to use non-space rated components successfully. Although this minimizes costs, it may decrease the reliability of some of the systems. As a result, tradeoffs were examined which maximized the reliability of the systems and still maintained our low-cost goal. A major factor in minimizing cost was the types of components donated to the project by industry. The baseline design takes into consideration the use of non-space rated parts where donations of space-rated components were not attained.

To reduce the complexity of the spacecraft design, the attitude control system was designed be exclusively passive with no on-board propulsion systems. Therefore the satellite will not be able to perform station keeping or attitude adjustment burns.

To reduce the power requirements for the satellite, most of the systems have been designed as
passive where possible, such as thermal and attitude determination and control. These subsystems used this type of philosophy in their design parameters. The satellite has also been designed with autonomous systems on board, which enable the spacecraft to maintain itself with minimal communication from the ground station. This will help in emergency situations, as the spacecraft will be able to identify and perform the necessary tasks autonomously and correct or minimize any problems, should they occur.

The communications subsystem will be able to transmit and receive voice communications on amateur radio frequencies. This ability will help the Spartnik project obtain FCC licensing for the ground station and the satellite.

The various spacecraft subsystems are discussed in detail in the following sections.

Launch Vehicle / Orbit

At this time, Spartnik does not have a commitment from an organization for a donated launch vehicle. Research is currently ongoing about the possibilities of using the following launch systems: Orbital Sciences Corporation, NASA H2, Russian Kosmos, NASA Expendable Launch Vehicle Systems, Ariane IV, Conestoga, Pegasus, Atlas and Lockheed Martin Athena Launch Vehicle.

The satellite will be classified as a secondary payload by the launch vehicle company due to the size and weight of the spacecraft (roughly 50 cm$^3$ and 37 kg).

To increase the versatility of the satellite in adapting to possible launch vehicles, the team has designed a secondary payload adapter system that can be modified to bolt to any launch vehicle. Modification time of the adapter system is estimated to be no more than one week, once the launch vehicle is known. Designing the system to meet various launch vehicles will increase the possibility of receiving a donated or discounted launch as a secondary payload.

The orbital elements, dependent upon the launch vehicle, are critical design parameters, which were carefully examined. The design team used 12 baseline candidate orbits. These selections enabled the team to predict how the satellite will perform in almost any orbit. The 12 candidate orbits are LEOs with altitudes of 450, 500, and 700 km at inclinations of 40°, 45°, 60°, and 90° with an eccentricity of zero. The altitude ranges were specified from the definition of a LEO; the design team has planned for an orbit altitude between 700 and 900 km to give the satellite a longer orbit life. They included the range of 300 to 700 km to account for the possibility of launching on the STS, which has nominal altitudes in this range. The latitude of the Spartnik ground station, located in San Jose, California, set the minimum inclination at 40°. For an orbit altitude of 450 km, the ground station would rarely “see” the satellite if its inclination were below 40°. POHOP, a program developed by JPL, was used to construct simulations for each of these candidate orbits. Due to perturbation effects, the design team determined that the minimum altitude for Spartnik should be 450 km to maintain the two-year operational lifetime. POHOP simulations showed excess orbit decay for orbit altitudes of less than 450 km.

Structures

The current configuration of the Spartnik satellite is an octagon with a height of 12.38 in. with the antennas in the stowed position. The satellite has a diameter of 17.11 in. and a mass of approximately 37 kg once fully assembled. Figure 1-1 shows these dimensions and basic exterior layout of the satellite.

To conform to the requirement of modularity and ease of assembly, subsystem components were mounted onto three trays. Figure 1-3 shows an exploded view of Spartnik.

The structural members have been machined out of 1/2 inch 6061-T6 Aluminum honeycomb panels. The spacers and electrical component boxes were made from solid 6061-T6 Aluminum. The internal trays have been equally spaced with 2.9-inch distance between them. Four cylindrical spacers support each tray with the exception of the third tray, which is being supported by the aluminum battery boxes. The trays were divided between the subsystems. The first tray (bottom panel) holds the computer and communications system, the second houses the power subsystem components, and third has been allocated to payload.

The eight side panels and the top panel will be assembled into one piece, which is called the shell. The three trays will be assembled and stacked on top of each other, with all internal wire connections completed. The shell is then slipped over the internal trays and bolted to the first tray at each of the eight vertices. Once the satellite has been bolted together, four threaded steel rods will be inserted through the structure and bolted on the top and
Figure 1-1 Spartan Dimensions
Dimensions are in inches

bottom panels to aid in retaining structural integrity. Figure 1-3 shows this in an exploded view.

The eight spacers, through bolts, and satellite base plate serve a dual purpose. The spacers and through bolts are used for structural rigidity and as thermal conduction posts. The satellite base-plate, which will stay with the satellite after booster separation, acts as a primary radiator surface. The solar arrays mounted to the exterior surface of the satellite provide energy with which to power the subsystems and will provide attitude determination via current versus Sun incidence angle curves. Using components, which serve dual purposes, reduces the number of final components required to assemble the satellite, making the design less complex.

Along with meeting the above requirements the structure has been designed to withstand the launch environment. This environment can prove difficult to design to when the mass and volume constraints are extremely strict. There are various contributors to the launch loads. The smaller, but sustained loads are generated from the thrust of the vehicle and wind buffeting during launch. These loads can range from 2-12 Gs and are not always predictable. Stronger loads are generated from shock and acoustic vibrations and quasi-static loads. These loads and their frequencies can not be changed, rather the satellite has been designed to not have natural frequencies equal to the frequencies at which these loads occur. Table 1-1 shows peak shock loads and their related frequencies for several of the possible launch vehicles.

<table>
<thead>
<tr>
<th>Launcher</th>
<th>Frequency(Hz)</th>
<th>Peak Shock</th>
</tr>
</thead>
<tbody>
<tr>
<td>STS</td>
<td>31</td>
<td>50</td>
</tr>
<tr>
<td>Athena</td>
<td>1500</td>
<td>15</td>
</tr>
<tr>
<td>Pegasus</td>
<td>1000 and above</td>
<td>200</td>
</tr>
<tr>
<td>Ariane IV</td>
<td>1500 and above</td>
<td>2000</td>
</tr>
</tbody>
</table>

Table 1-1 Launch Vehicle Shock Loads

To aid in the design, computer modeling of the structure has been done using Pro/MECHANICA, ABACUS, and COSMOS. The results provide preliminary responses of the structure when subjected to environmental loads. To qualify the computer modeling, environmental shake tests have been conducted on the structure. Shake table tests were conducted at United Technologies Corporation, Chemical Systems Division. The testing was comprised of a Shock Test, Sine Sweep, and Random Vibration Test. These tests correlated well with the computer modeling.

Launch Vehicle Adapter

The Launch Vehicle Adapter (LVA) is an interface between the satellite and the launch vehicle, as well as a release mechanism for the satellite. During launch the LVA will act closely to a rigid body so that all the loads will be transferred to the satellite without any damping or amplification. The design of the LVA is versatile so that it can be attached to a wide range of launch vehicles. This was accomplished by making a solid circular launch vehicle plate with the capability of the mounting holes being drilled once the launch vehicle is determined, making the LVA a universal design. The design has a release mechanism that releases the satellite with minimal stress. It uses a 28V DC signal that most launch vehicles provide. The G&H Technologies release bolt is a separation nut mechanism, where the 28 V DC signal activates a mechanism to separate a nut. The satellite will still launch if only one half of the nut separates from the bolt. Once released, a spring will push the satellite away from the launch vehicle. Most of the LVA will remain with the launch vehicle except for the satellite base plate and the bolt. The satellite plate will stay on the -Z panel and act as a thermal reflector to help keep the satellite in its required temperature range. Figure 1-2 shows the different parts of the LVA.
It was necessary to insure that the satellite and launch vehicle adapter fit into the allowable secondary payload fairings for all possible launch vehicles. The launch vehicle adapter is approximately 6.7 inches high by 11.5 inches in diameter. This brings the overall height of the satellite and LVA to 19.08 inches and 17.11 inches diameter.

If Spatnik launches on the STS-Hitchhiker System, the LVA, excluding the satellite base-plate, will return with the orbiter. Since the G&H mechanism is reusable, the LVA system can be reused after a base plate for a new satellite has been machined. This reduces the cost and manufacturing time for subsequent satellites and its release system. After separation, the remaining components of the LVA will be self-contained without any loose parts. This meets the STS Orbiter safety requirements for onboard systems to not pose any danger to the astronauts.
The only connection between Spartnik and the launch vehicle will be through a MIL-Specified four-pin connector, which is attached to the G&H release mechanism. This connector, along with the G&H release mechanism, will remain with the launch vehicle after separation of Spartnik.

**Attitude Determination and Control**

The attitude determination and control system is completely passive and uses spin stabilization and a controlled tumble to maintain an acceptable orientation. A spin will be induced by solar radiation pressure on the communications antennas. These antennas are coated with a highly reflective tape, Aluminum FOSR, on one side and black anodized on the other. Figure 1-5 shows the satellite from an edge-on view. It also shows the communications antennas, with the black anodized side on the right and the highly reflective coating on the left. If the Sun vector is perpendicular into the page, the solar-radiation pressure differential between the reflective and the anodized sides will cause the satellite to spin to the left, which is opposite the positive z-axis.

To prevent the spacecraft from continuously spinning up, four soft iron rods have been mounted perpendicular to the spacecraft's z-axis, as shown in Figure 1-6. As the iron rods, or hysteresis rods, rotate within Earth's magnetic field, the magnetic dipoles within the rods will realign themselves to run in the same direction as Earth's magnetic field lines. This realignment generates eddy currents, which in turn generate a torque, countering the spin of the satellite. At each half turn of the rods, the current flow will reverse and the poles will switch. This flipping of the poles and reversal of the current in the rods will create heat within the iron rods. The heat is the conversion of the spin energy of the satellite. Thus as heat is radiated out into space, the spin energy is also dissipated.

This control system is useful in keeping the spacecraft pointed in one direction. It was necessary to find a way to ensure proper pointing for the camera lens, which will be mounted on the +Z-face. Inducing a controlled tumble of the spacecraft's Z-axis did this. Bar magnets were placed inside the honeycomb shell, with the poles parallel with the satellites' Z-axis. These magnets will interact with the Earth's magnetic field and align the z-axis of the spacecraft with the local magnetic field lines. This is a controlled tumble because we can simulate the angle between the spacecraft Z-axis and an inertial frame. Figure 1-7 shows the approximate attitude of the spacecraft for a quarter of a polar orbit.

The controlled tumble will ensure that the camera lens will be pointing toward the Earth in the Northern Hemisphere only. The magnets will align the satellite so that the camera is pointing out to space in the Southern Hemisphere.

To control nutation about the z-axis of the spacecraft, a nutation damper hoop was mounted parallel to the xy-plane of the spacecraft, which can be seen in Figure 1-6. The nutation damper was filled 40% with synthetic 10-40-weight grade oil. As the hoop rotates the fluid will lag behind, acting as a spin damper until it reaches equilibrium with the satellite. If the spacecraft starts to nutate in small angles, waves will be formed on the surface of the fluid. The waves will propagate around the hoop and dampen the small nutations. If the satellite
experiences large nutations, the viscous fluid will act as a slug to dampen out the unwanted nutations.

To verify the passive attitude control system, a computer simulation has been developed that will take into account Earth’s magnetic field and attitude control hardware. This software was used to select the number of magnets used.

In order to determine the orientation of the spacecraft, current measurements will be taken from the solar strings mounted on the exterior panels of the satellite. The angle between the solar vector and the solar string is directly proportional to the current going through the string and a current incidence curve will be created. The on-board CPU will compute the orientation of the spacecraft, relative to the local inertial coordinate system of the satellite, using these current readings. This will give the spacecraft’s attitude at any time on the orbit and will provide a means of predicting the attitude at any future time. This will allow for the planning of when a picture can be taken of Earth’s surface, as well as the Moon.

To aid the current sensor calculations in determining when the camera lens is nadir pointing, two 1μ infrared sensors will be mounted equidistant on either side of the camera lens. When both IR detectors are registering a reading then the surface of the Earth is in view of the camera lens.

Power

The power system is a battery-dominated configuration, where the batteries determine the bus voltage. The batteries are Sanyo rechargeable commercial grade Nickel Cadmium (NiCd) donated to Spartnik by Eagle Picher in order to qualify their testing procedures for a new process. Eagle Picher tested and matched these batteries using a new procedure, which they are developing for low Ampere-hour microsatellite batteries. Telemetry will be recorded on the health and performance of the batteries while on orbit, and will be given to Eagle Picher so that they may validate and improve their testing procedures.

A dual battery pack design was used to minimize the depth of discharge to approximately 10%, thus increasing the number of possible cycles and operational life of the batteries. Figure 1-8 is a simplified wiring diagram depicting the power system configuration.

The power budget for the on-board systems is summarized in Table 1-2. The satellite will have various operational modes where different systems will be operating. The power system is capable of handling all the systems at once for a limited time.
Table 1-2 System Power Requirements

<table>
<thead>
<tr>
<th>System</th>
<th>Load Requirement (W)</th>
</tr>
</thead>
<tbody>
<tr>
<td>CPU</td>
<td>1</td>
</tr>
<tr>
<td>Camera</td>
<td>0.001-6.1</td>
</tr>
<tr>
<td>MMI</td>
<td>1</td>
</tr>
<tr>
<td>Receiver</td>
<td>0.5-5.0</td>
</tr>
<tr>
<td>Transponder</td>
<td>0.5-5.0</td>
</tr>
<tr>
<td><strong>Total</strong></td>
<td><strong>13.2 max.</strong></td>
</tr>
</tbody>
</table>

Figure 1-8 Power Wiring Configuration

The transponder is a transient load with a maximum of 5W. This will allow for interference and corrupted signals from the satellite. The system will be able to boost its power to offset any anomalous interference.

The nominal bus voltage is 7.2 V from the battery packs. Each payload will have a Maxim step down DC-DC converter to bring the bus voltage to the systems' required voltage. The peak voltage of the batteries is 9.0 Volts DC. In order for the solar arrays to recharge the batteries to full capacity the array voltage needs to be higher than that of the batteries. The solar array design uses 16% efficient Gallium Arsenide 2x4 cm cells that were donated by Applied Solar Energy Corp. The solar array will provide a nominal voltage of 9.76 V and approximately 8 Watts, at end-of-life (3 years). This is acceptable for the recharging of the NiCd batteries, and supplying the required system load. Solar Power Corporation has manufactured the solar arrays, and Lockheed Martin Missiles and Space (LMMS) is currently adhering the solar arrays to the structure. Figure 1-9 shows the solar array surface layout for the side panels.

To ensure that the batteries recharge at the proper rate and at a rate slow enough to maximize life, without having the system continually being in a state of charge, the solar arrays were split into two systems. The two solar arrays can be switched between the two battery packs. The system can use both arrays to recharge the batteries, or split the arrays between the two packs, or remove one or all arrays from the system until needed. The batteries are being tested by LMMS to determine the voltage-temperature (VT) curves that will characterize when the charge controller should put the system into a state of charge.

To ensure maximum battery life, each battery pack will be reconditioned every two months.

Payloads

The Spartnik satellite project hopes to prove that an undergraduate senior design class can build a successful microsatellite that may be used for space-based research by both universities and industry. To demonstrate this concept there are three payloads onboard: a color digital camera, a Micro-Meteorite Impact Detector (MMID), and a communications package. The digital camera and MMID are discussed here; the communication package is discussed in a subsequent section.

The first experimental payload is the color digital camera, the Kodak DC40. It has four megabytes of Flash RAM for image storage and uses a charge-coupled device (CCD) developed by Eastman Kodak. Modifications to the camera in preparation for the harsh environment of space include the removal of the flash, the liquid crystal display, plastic components, and the conformal coating of the electronics boards.

The camera is mounted to the inside top face of the satellite, as shown in Figure 1-10, and will be nearly nadir pointing over the Northern Hemisphere due to the nature of the attitude determination and control system. One concern for the camera is the effects of infrared radiation on the Flash RAM. It is unknown if repeated exposure to high radiation levels would adversely affect the quality of the digital images. Flying this camera not only shows that surface imaging from a microsatellite can be done, but it will also qualify Flash RAM operation in the space environment.

The second payload is the MMID, similar to the one used on the Webersat satellite. An impact on the piezo-electric sensor will cause a hardware interrupt.
attraction. Since Webersat is in a particular orbit and Spartnik may be launched into a different orbit, the number of impacts may vary. The data collected from the MMID experiment can then be compared with the data from Webersat. Figure 1-10 shows the piezo-electric sensor mounted to the outside of the +z face. The sensor is connected directly to the onboard computer.

**Computer Hardware / Software**

The on-board computer main processor, the 80C188EC, is based on Intel 386 CMOS technology that was tested to be reliable in radiation high environments. The main CPU has one 256k EPROM chip that will hold the boot-up code. Once the computer completes the boot-up sequence, the operational program will be transferred to main memory. All sensors are routed through the analog to digital converter that consists of two A/D chips, allowing a total of 128 sensors. There will be four megabytes of static ram (RAMDISK) available to store data. By having the operational code stored in main memory mission control will have the ability to upload new code at anytime.

To ensure the reliability from radiation single event upsets and transient upsets several contingencies have been implemented. An Error Detection and Correction (EDAC) chip is implemented between main memory and the processor. A checksum algorithm is used in the communications packet software.

The function of the onboard CPU is to operate, maintain, and troubleshoot the onboard systems. This will be done through an autonomous software program. Housekeeping telemetry and regular self-tests will be conducted and recorded to ensure optimum operating performance. The onboard computer will manage the communications payload (AX.25) and the transfer of telemetry to the ground station and the uplink of commands. This will be done either by the student ground personnel or autonomously through pre-programmed commands and automatic tracking.

This system has been designed to use no more than one watt of power and use a minimal printed circuit board size and will have the capability to uplink at 9,600 BPS and downlink (power permitted) at 57,600 BPS.
Thermal

Some of the past microsatellites and amateur satellites have regarded thermal control as an unimportant topic. Because of the lack of data in this area and a difference of opinion between microsatellite builders and industry consultants, the Spatnik team elected to form a separate Thermal Control Systems (TCS) team to eliminate the microsatellite thermal operations uncertainty.

Calculations were carried out to characterize Spatnik's orbital thermal environment loading for a range of possible mission orbits. Operating temperatures were then calculated, estimating the thermal properties of the components onboard the satellite. This information was used to evaluate the thermal functionality of Spatnik's design.

In the thermal system performance evaluation, surveys of component temperature ranges and maximizing satellite life were used. From manufacturer's specifications, the battery packs became the driving factor for the thermal design because they had the narrowest operational range. The batteries restrict Spatnik thermal target range as 0°C to 10°C for optimal efficiency, although the batteries will operate between -5°C to 25°C. Because of the power constraint, all thermal control must meet this temperature range with passive thermal control techniques.

It was found that Spatnik can be assumed to be isothermal for first order calculations. Subsequent thermal design modifications were made to better link components together in support this simplifying assumption. For example, most structural components were required to be aluminum for thermal conductivity benefits and joint fillers were used to minimize thermal resistance between components on main thermal conduction paths such as the through-posts. Solid aluminum blocks were prescribed to provide adequate thermal mass to "coast" battery temperatures through transient thermal loads, such as the Sun/shadow portions of orbits.

Studies of thermal system performance were done with several identified design parameters such as surface finish areas and orbit beta angle. Radiators of calculated size, made from of Flexible Optical Solar Reflector (FOSR) material on the top and bottom panels, will be used to balance Spatnik with its final orbital thermal environment. Figure 1-11 is a plot of steady-state bulk temperature in a 60° beta angle, 300 km orbit as a function of FOSR coverage.

Operationally, TCS will maintain an array of temperature sensors to collect battery and other thermal telemetry. Additionally, TCS will implement a Pseudo Active Thermal Control System (PATCS) software to provide the computer with thermally intelligent operational code.

TCS is currently engaged in final verification, detailing, and testing of Spatnik's design using an industry thermal modeling software known as SINDA as well as authoring the PATCS software. Additionally, TCS is expanding to review all environments Spatnik will encounter, from manufacturing to reentry.

![Figure 1-11 Percent FOSR vs. Temperature Range](image_url)

Communications

The onboard electronics are low power and frequency versatile. Telemetry data will be encoded into a packet. The system features nonstandard data throughputs of 9600 baud or 57.6 kbaud.

The onboard electronics operate on the principle of state of the art direct synthesis. The components are donations from Philips Semiconductors and are normally used in cellular telephones. Since the system has the ability to switch to a different frequency, the satellite can avoid interfering with another satellite. However, any frequency change will adhere to the recognized amateur bands of 144 MHz. for uplink and 435 MHz. for downlink.

Physically the communication electronics are a standard two-layer circuit board measuring approximately 5.98 inches in length by 4.01 inches.
in width. They are located on the bottom panel of the satellite and are connected to the four uplink and four downlink antennas. The uplink and downlink antennas are staggered on the top and bottom faces of the satellite with one antenna on every other vertex of the octagon shape. These placements can be seen in Figure 1-12. Both sets of antennas are made of a spring steel material similar to that found in venetian blinds. The orientation of the antennas with respect to the ground station receiving antenna is not pertinent to maintain a communication link.

![Figure 1-12 Communication Antennas Placement](image)

To simplify the licensing of the satellite, amateur radio bands are used. This will allow any licensed amateur radio operator to access the satellite. Unlike most amateur satellites which have a standard data throughput of 1200 baud, Spartnik will employ high speed data transmission rates of up to 57.6 kbaud to insure that all data is received by the ground station in a single pass of the satellite.

The packet protocol is currently under development. This routine will use the AX.25 communications software standards. When completed, the packet routine and software will become public domain to any university microsatellite project.

The commanding ground station is located in the San José State University Engineering Building and will be operated by HAM licensed students.

Conclusion

At this time the satellite proto-flight structure has been manufactured and assembled. Wiring diagrams of the electrical circuit boards have been finalized and the first revision of the printed circuit boards is complete. Battery testing is currently being completed and the solar strings have been manufactured. Environmental testing (vibration and shock tests) has been completed. These tests proved that the structure is robust. A COSMOS model has been constructed to verify these results. Manufacturing of the flight model began in August 1996 to be completed by late June 1998. Software programming of the onboard computer is ongoing to be completed by summer 1998. The Spartnik team has intensified their search for a launch donation for a launch date falling within spring 1999 - TBD.

Acknowledgement

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