SpinSat Mission Overview

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ABSTRACT

The SpinSat flight is a small satellite mission proposed by the Naval Research Laboratory and Digital Solid State Propulsion (DSSP) LLC to demonstrate and characterize the on-orbit performance of electrically controlled solid propellant technology in space. This is an enabling technology for the small satellite community that will allow small satellites to perform maneuvers. The mission consists of a spherical spacecraft fitted with Electrically Controlled Solid Propellant thrusters and retro-reflectors for satellite laser ranging (SLR). The spacecraft will be deployed from the International Space Station. This paper presents a mission overview and emphasis will be placed on the design, thruster design, optical layout, and unique SSA observation opportunities of the mission.

1. MISSION CONCEPT

The Naval Research Laboratory is in a work for outside parties agreement with Digital Solid State Propulsion (DSSP) LLC, to perform and spaceflight demonstration of an advanced rocket/projectile thruster technology that employs a special new class of energetic but non-pyrotechnic materials known as Electrically-Controlled Solid Propellants (ESPs). NRL is designing the spacecraft, and DSSP is designing the electrically controlled solid propulsion system. The spacecraft, known as SpinSat, is based on the Atmospheric Neutral Density Experiment (ANDE) design[1], will provide a test platform to demonstrate and characterize the on-orbit performance of the thruster technology.

There are two primary goals of the SpinSat mission. The first goal is to characterize the performance of the ESP thrusters on orbit. The second goal of the mission is to provide a calibrated drag experiment at higher solar activity than the ANDERR and ANDE2 missions to provide a monitor of total neutral atmospheric density.

The SpinSat spacecraft is a 22"-diameter aluminum sphere, Fig. 1, with the ESP thrusters physically arranged on the exterior of the satellite to provide two basic maneuvers as spin-up (de-spin) maneuver and a normal thrust maneuver**.** For the spin-up maneuver, pairs of thrusters will be co-aligned 180 degrees apart, will provide a tangential component force on the exterior; for de-spin, a 2nd pair of thrusters will provide the opposite force. For the normal thrust maneuver, thrusters will be oriented perpendicular to the exterior of the satellite to provide force in the normal direction. Another set of thrusters placed at the opposite pole will provide normal force in the opposite direction.

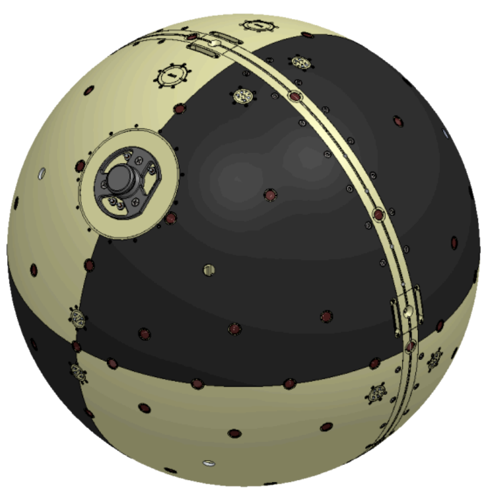

Fig. 1. SpinSat spherical spacecraft.

The spacecraft itself acts as the primary sensor for the third experiment; with a well-determined and characterized ballistic coefficient the routine collection of radar tracking and satellite laser ranging data will provide a high-resolution atmospheric drag data set used to derive thermospheric density.
SpinSat has manifested by the DoD Space Test Program (STP) to be deployed from the International Space Station (ISS) in April of 2014.

2. THRUSTER DESIGN

2.1 The First-generation Microthruster

The SpinSat propulsion system is based on DSSP's first-generation microthruster design. A microthruster consists of two coaxial electrodes separated by approximately 1 mm of electric solid propellant (ESP). The center electrode is a stainless steel rod and the outer electrode is a hollow aluminum case. The total length is 13 mm, which amounts to 0.1 grams of propellant.

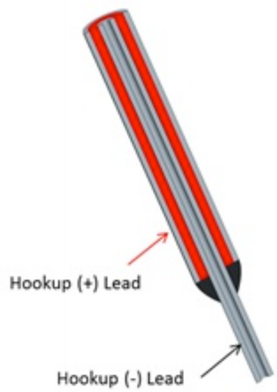

Fig. 2. DSSP's "Gen-1" Microthruster

The key feature of the microthruster is that the center electrode is wrapped with an insulating plastic over the majority of its length. The plastic stops near the thruster opening, leaving a small conductive path on the face of the thruster. When current sufficient for ignition is applied, the plastic will burn along with propellant, creating a subsequent layer of conductive ESP. This patented "burn-away" method is designed to ensure even and flat propellant ignition from the face of the thruster down to its base.

2.2 UMS Thruster Cluster

For the SpinSat mission, DSSP's microthrusters were adapted to the satellite's existing Universal Mounting System (UMS) plugs. This simplified satellite structural design and allowed for a modular propulsion solution. On the SpinSat, each UMS plug houses six microthrusters. There are twelve of these UMS "clusters" or 72 total thrusters per satellite. The UMS clusters are identical with the exception of a flow straightener plate, which guides the propellant exhaust gases at an angle relative to the normal direction. Angled plates are installed on UMS clusters that are tasked with rotational maneuvers. Straight plates are added to UMS clusters tasked with translational maneuvers. A total of twelve UMS clusters are installed on the satellite. Eight are angled and four are straight.

Fig. 3 shows a cross section of a UMS cluster and illustrates key design features. Rather than having separate outer cases, each thruster case is formed by machining a single piece of aluminum. Thus, the aluminum is an electrode common to all thrusters. The six center electrodes protrude from the rear of the UMS, where they mate directly with the cable harness.

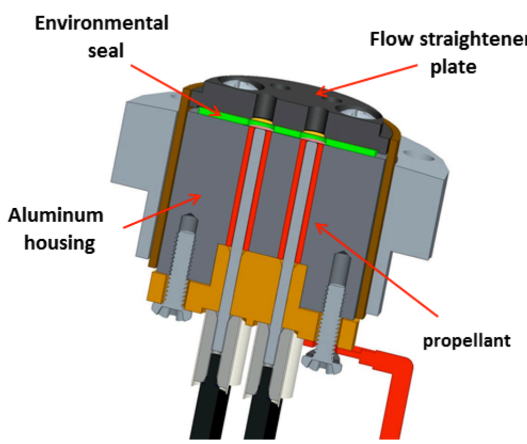

Fig. 3: Cross-section of a UMS Cluster.

In order to protect the propellant from contaminants, the UMS is designed with an environmental barrier that sits between the flow straightener plate and the electrodes. Multiple layers of Mylar, Teflon, and Viton serve this purpose. An image of a UMS cluster prototype appears in Fig. 4.

Fig. 4: UMS Cluster Prototype

3. THRUSTER ELECTRONICS DESIGN

3.1 <u>Ignition Power Control</u>

ESP ignites when sufficient power density is achieved in the propellant. However, designing a power delivery system is challenging because propellant conductivity is highly variable. It varies with applied electric field, temperature, pressure, ignition history, and several other parameters. A particularly interesting characteristic is that ESP conductivity increases rapidly in the presence of strong electric fields. In fact, for small electrode gaps such as that of the microthruster, the load will appear as a short circuit at critical voltage levels. Consequently, the propulsion controller must not only regulate power to the load but also tolerate intermittent short-circuit conditions.

DSSP's solution is a switching power converter that operates in current-mode. Specifically, the circuit is a non-inverting buck-boost converter that has very little output filter capacitance. Current delivered from a DC source is stored in an inductor and then released into the load. The switch timing is such that all of the inductor current, and thus energy, is allowed to dissipate in the load before the next charge cycle begins. In this way, since frequency and inductor current are programmable, average power is controlled. Typical values for energy and frequency on the SpinSat are 0.002 Joules and 100kHz, respectively, which amounts to 200 Watts of ignition power including losses. Since current is limited on a per-cycle basis, intermittent short-circuits are tolerated.

3.2 Propulsion Control Module

The power converter is the heart of the Propulsion Control Module (PCM). The PCM is a single-sided PCB assembly that contains the converter power electronics, the microcontroller and logic devices, communications hardware, and an array of MOSFET switches that perform individual thruster addressing, Shown in Fig. 5. Two PCMs exist on the SpinSat – each tasked with controlling 6 UMS clusters or 36 thruster elements.

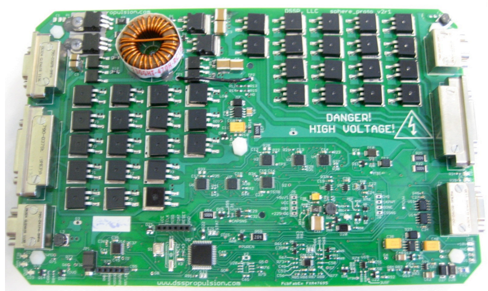

Fig. 5. PCM Circuit Assembly

The power supply for the PCM is a regulated 5VDC, 0.2A bus provided by the avionics system. Since this is not enough power to drive the thrusters directly, immediate thruster power comes from a bank of capacitors. The PCM uses four wet tantalum capacitors, wired in series and charged to 190V. The caps are housed in the PCM enclosure but are installed separately in a rigid plastic dish. It takes approximately three minutes to charge the capacitors prior to each ignition pulse.

Each PCM has a dedicated RS-422 serial communications interface with the avionics host. The host configures the PCM prior to issuing a fire command. Configuration commands include thruster selection (any subset of 36 thrusters can be addressed), power level setting, and pulse width setting (adjustable over 50, 100, and 200 milliseconds). Each PCM interfaces with its UMS clusters using cable harnesses.

4. THRUSTER TESTING

DSSP has tested their millipound-scale microthrusters on a thrust stand and verified that their performance is adequate to perform the mission of positioning SpinSat in all directions. This performance has been verified both with bare thrusters in vacuum as well as with integrated, flight-weight and flight-design Universal Mounting System (UMS) sets of 6 microthrusters; it has also been verified after being subjected to the critical tests outlined in the NASA GSFC-STD-7000 standards for environmental verification. Due to its survival of launch-equivalent vibration, humidity soak, thermal vacuum, and ability to fire at high and low temperatures more stringent than those seen on a prior equivalent spherical satellite mission, and due to thrusters' ability to still provide useful thrust after those tests, DSSP's electric solid propellant-using thrusters have passed environmental qualification and are ready for launch and on-orbit operation.

4.1 Testing Hardware

All performance testing has been conducted at DSSP inside a vacuum chamber pumped to a vacuum pressure equal to or less than 0.1 millitorr to simulate near-Earth environments. All thrusters were placed on DSSP's microthruster thrust stand, capable of measuring impulse bits between 0.00001 and 0.01 pound-seconds with +/- 10% accuracy (+/- 5% from 0.0001 to 0.001 pound-seconds, where most thruster performance resides). Fig. 6 shows the thrust stand with a test article mounted; the thrust stand behaves as a pendulum, with a laser finding the magnitude over time of the pendulum's deflection when subjected to a force. The resulting decaying exponential can be reduced to a force vs. time curve with high accuracy.

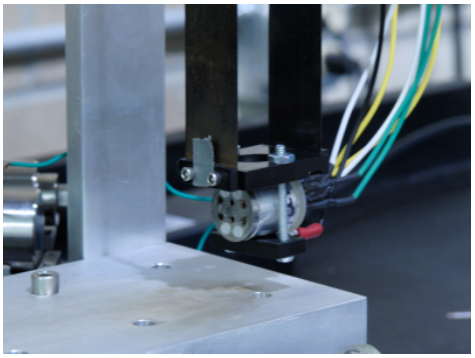

Fig. 6: Six Thrusters Mounted to DSSP Microthruster
Thrust Stand

Environmental testing was done with a blend of in-house and external hardware. Humidity testing was done by placing test hardware in a DSSP designed plastic chamber with constant high humidity for two weeks. Vibration testing was done to Delta IV standards as a representative value at Martin Test Labs in Sacramento, CA; thermal vacuum was done to a standard of +60 to -30 C at vacuum, with the early testing done at Martin Test Labs but final qualification thermal vacuum cycling performed in-house at DSSP.

4.2 <u>Thruster Performance Results</u>

Testing proceeded in multiple stages. In the first stage, baseline performance data was gathered with thrusters under no environmental stressors save vacuum, which represented the best-case of performance. In the second stage, a variety of tests (vibration, thermal cycling, vacuum soak, etc.) were performed individually and thrusters fired to measure each separate test's effect on performance. The final phase, full qualification, saw every test done to the thrusters collectively, and then thrusters fired to measure final expected-on-orbit performance. Though the thrust data from that final series of tests is not yet available, there is no indication that thrusters will not continue to operate after the full battery of qualification tests. Instead, test data for the first two phases will be shown.

To explain the primary performance metrics, the thrust curves of a typical set of thrusters must be examined. These thrusters are multipulse, and each pulse delivers a measurable amount of thrust for a software-controlled duration. Fig. 7 shows a typical UMS's results, with each thruster being a separate set of points.

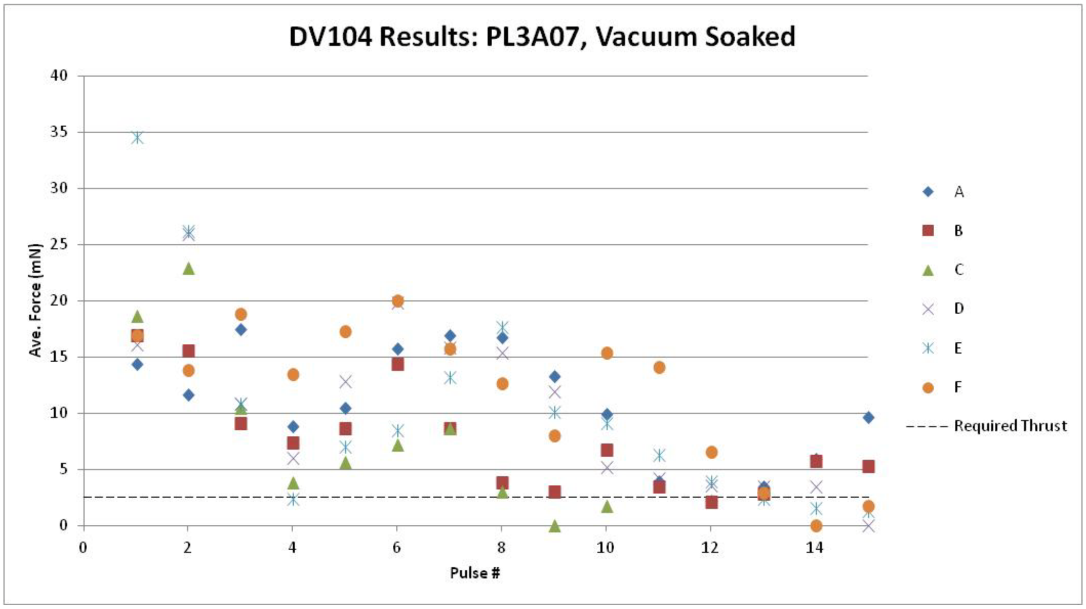

Fig. 7. Typical Set of Microthruster Performance Data

From this data, a clear and common 'shape' to how thrust degrades with thruster lifetime emerges, and the primary metrics of performance can be explained. The first is 'minimum thrust'; pulses of a thruster that do not achieve 2.5 mN of thrust are counted as failures. Thruster lifetime, in seconds, is another metric. This data was taken with each pulse lasting 0.1 seconds, while other test series examined 0.05 and 0.2-second pulses. Since there were not show stopping differences, to unify the performance metric thruster lifetime is counted in seconds of operation instead of pulses. A maximum of 1.5 seconds of operational time (15 pulses above, at 0.1 seconds pulse width) was set as where testing would cease, as a maximum; 1.0 seconds of operational time was desired. Finally and most importantly the total impulse of each thruster in millinewton-seconds was calculated, with a minimum goal of 5 mN*s; this is the total impulse delivered over the entire thruster lifetime, and serves as a measure of how much rotational or translational velocity each separate thruster can impart to a satellite.

Baseline data gives a reasonable best-case scenario for all three metrics, shown in Table 1. As can be seen, the baseline data indicates that each thruster will surpass its planned requirement handily in both total impulse and thruster lifetime; the requirement on the first was 5 mN*s, and thrusters range from two to three times that, while the thruster lifetime was required to be 1.0 seconds minimum and thrusters are well past that. The requirements that are not soundly met is the minimum thrust requirement, as well as (not shown) variability; both of these are interrelated, since the primary cause of both is thrusters that have currently unexplained 'failed pulses' that result in minimal thrust. When this happens, that pulse is counted as a failure by our metrics, and the variability of that pulse when compared inside a population of thrusters is increased. Operationally, however, it will be apparent when a thruster fails to fire, and ground control can simply command the next pulse of that thruster without serious consequences to the mission; for this reason, these were considered acceptable.

Table 1: Baseline Performance Results

Metric	Value	Variability
Total Impulse (mN*s)	**12.3**	**+/- 2.5**
Thruster Lifetime (s)	**1.39**	**+/- 0.13**
% of pulses meeting minimum thrust (2.5 mN)	**85%**	**N/A**

The next phase of testing involved exposing sets of thrusters to a single off-nominal condition at a time, and studying their performance after that condition was imposed. The full list of stressors applied: thermal cycling to +60/-30 C in atmosphere, vibration to full launch loads, vacuum storage for 2 weeks, firing multiple thrusters simultaneously, firing at +50 and -20 C in vacuum, and humidity soak for 2 weeks.

For each set of tests, a recheck was done of performing the same performance tests, and compared to the baseline with a 90% confidence level. The only two tests to give measurable and significant shortfalls in performance were the thermal cycling test and the humidity soak. In the latter case, after examining the thrusters, DSSP discovered that the initial humidity seal was faulty; after replacing it, another set of thrusters passed a retest. That leaves thermal cycling, which had a statistically significant decrease in the number of pulses meeting the minimum thrust requirement (76% instead of 85%) and in the thruster lifetime (1.18 s instead of 1.39 s). The thruster lifetime still meets requirements, but the minimum thrust issue- thermally cycled UMSs are more likely to have 'failed' pulses- remained throughout this phase. As before, since this is something that can be corrected via operational firing schemes, it was not considered a show-stopping defect; that aside, thrusters continued to perform at baseline levels for all environmental stressors.

The final set of testing was full flight qualification. A set of UMSs, each with 6 thrusters, went through a serial set of tests before being put on the thrust stand to measure thrust. First, a flight-like inspection process was implemented, where UMSs were studied for all measurable parameters (propellant weight, electrode concentricity, etc) and the closest to nominal were selected to continue. Those UMSs then went through a two week humidity soak, then sets of launch-level vibration on all three axes (as well as the flight electronics), followed by ten days of thermal cycling in a vacuum. Once all these tests were complete, UMSs would be placed on the thrust stand and fired at all durations and at high and low temperature to validate thruster operation at the full scale of the operational envelope. While thrust testing is not yet complete, results will be available for presentation at the conference, as the conclusion of testing is planned for mid-June. With the conclusion of these tests, microthrusters will be flight-validated and ready to provide SpinSat with orientation capability on-orbit.

5. SPACECRAFT DESIGN

5.1 Mechanical

The 57 kg, 22” diameter spherical spacecraft for the SpinSat mission is shown in exploded view shown as Fig. 8. Thrusters are located in four pairs to induce a spin (de-spin) of the spacecraft along latitudinal bands located at ±15º. Two pairs of thrusters are aligned normal to the sphere surface located in latitudinal bands at ±82º.

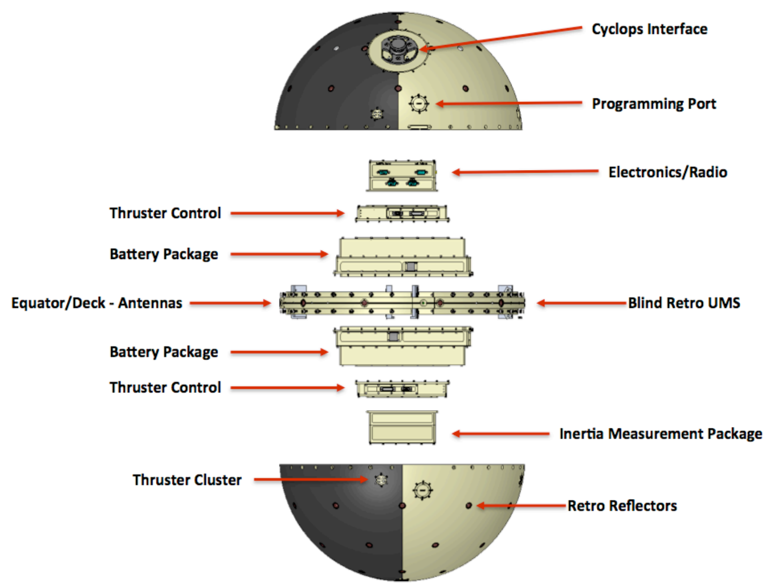

Fig. 8: Exploded View of SpinSat.

It is fitted with a set of sixty-eight 12.7 mm diameter optical retro reflectors for satellite laser ranging (SLR) located along five latitudinal bands at ±90º, ±67.5º, ±45º, ±22.5º, and 0º with one, six, eight, ten, and ten retros per band respectively. The last eight retros are located along a longitudinal band of 0° with latitudes of ±11.25º, ±33.75º, ±56.25º, and ±78.75º.

An interface bracket to the Cyclops deployment system is located at 45° latitude and 90° longitude. This bracket provides the sole interface to the Cyclops clamp mechanism, provides a push-off plane for the Cyclops separation springs, and houses the four separation switches (ITW series 65 model 401000 over travel switch) that inhibit SpinSat from powering up until it is deployed from the Cyclops. These switches/springs comprise a non-coplanar, parallel force system whose moments are balanced to minimize dip off during deployment.

Four 0.012” diameter Nitinol antennae are stowed in a shallow groove along the equator of SpinSat. Each antenna is 7.5” long from the tip to the surface of SpinSat and is loped and crimped at the free end. Three independent nylon ties that are severed by a burn wire after SpinSat is deployed from Cyclops restrain each antenna.

Fabrication of the hemispheres was accomplished by using a technique known as centrifugal casting. This process was chosen over three other possible ways of fabrication, which were as follows: metal spinning, investment/sand casting, and machining. The decision to use centrifugal casting was based previous proven performance and cost.

Centrifugal casting is a manufacturing process pioneered by Johnson Brass and Machine Company of Milwaukee, Wisconsin. For parts to be fabricated in this manner they must have cylindrical symmetry or be a section of a larger parent object with cylindrical symmetry. In this process a mold of the part to be cast is placed on a turntable spinning at high speeds. Molten alloys are then poured into the spinning cast and the spinning action continues while the cast cools. The rate of spin for centrifugal casting is a controlled variable dependent upon the part geometry and alloy but is a minimum of 60 G's. The centrifugal force functions in several ways to form a superior cast for certain types of parts; it forces the molten metal to uniformly take the shape of the cast, it reduces the amount of material needed for certain hollow geometries, increases the uniformity of cooling rates throughout the part, and for certain alloys it separates out impurities due to differences in their density. This centrifuge action of separating out impurities helps to significantly reduce the number of voids and bubbles as compared to traditional sand casting. For very critical uniformity requirements the casts undergo the process known as Hot Isostatic Pressing, also know as hipping, before they are finished and machined. In the hipping process the casts are placed in an inert argon atmosphere and then heated to just under their melting point while the pressure of the atmosphere is raised anywhere from 20,000 to 30,000 psi depending on the alloy for industrial purposes to over 100,000 psi for research purposes. By undergoing the hipping process voids are closed down to microscopic levels. It must also be mentioned that the centrifugal casting process works better as the size of the part increases. As can be seen the centrifugal casting process was ideally suited for the SpinSat hemispheres. This match helped to reduce the per hemisphere cost by on order of magnitude compared with previously mentioned methods of manufacture. The surface of each spacecraft is polished after machining to 32 micro inches. Further degrees of smoothness are achieved by the application of surface treatments. The exterior of each spacecraft has on it a beach ball type pattern consisting of gold irridite and black anodize type 2 class 2 coating.

5.2 Payload Design

The payload consists of two inertial sensors, ADIS16385, chosen for their accelerometer sensitivity. The ADIS16385 will be used to measure the acceleration generated by the normal mounted thrusters and the rotational rate changes from the tangential thrusters. Since the sensors have digital interfaces, the interface circuitry is simple and handled by a custom microcontroller board using Arduino software.

5.3 Electrical Design

The electrical design consists of a stack of circuit boards. Starting at the bottom, the first board in the timer board. This board incorporates the NASA safety required timer circuits that operate independently and delay the powering of the satellite bus. The three independent timers receive power through the arm plug which connects two four separation switches which connect to the batteries. When the timers are activated, they will count for specified period and trigger latching relays. The latching relays turn the timers off and turn power on to the satellite bus.
The next board is the electrical power system (EPS). The EPS includes a 5-volt regulator for the Data Handling Unit, power switches for antenna release, power converters for the thrusters, and a discrete watchdog time that will cycle power to the satellite bus if not reset in time.

The third board is the data-handling unit (DHU) using an ATMEGA1280 microcontroller. It was chosen for its low power operation, multiple serial interfaces, and existing flight software from previous satellites. The DHU includes a micro-SD card for data storage, a real time clock to support scheduled operations, and interface circuitry to interface to the thruster controllers and sensors.

The fourth board is the communications board. It consists of a transmitter and receiver. The transmitter is the Stensat Radio beacon to be operated at 9600 bits/sec. It will output two watts of RF signal into a dipole antenna through a balun. The receiver is a Stensat transceiver design using only the receiver section since the frequency coordination will put the uplink at about 450 MHz and downlink at about 401 MHz. The transceiver cannot switch between the two frequencies due to the range of the VCO

5.4 Communications Design

The communications is performed with the above mentioned communications boards. There are four antennas installed around the equator of the satellite at equal distances. They are paired at opposite sides to make two dipoles, one for transmit and one for receive. The antenna material is made of nitinol, a flexible metal alloy that can maintain a straight shape and can be stowed with tight bends. The antennas will be stowed in a trough along the equator and secured with nylon cord. A resistor will be used to cut the cord by heating up and melting the cord. This has been done on previous satellites with great success.

6. MISSION OPERATIONS

6.1 Launch and Deployment

The launch and deployment of SpinSat into its 400km 51.6° inclination circular orbit is provided by the DoD Space Test Program. SpinSat will be launched to the ISS as part of the soft-stow cargo allotment on the SpaceX Dragon spacecraft launched by the SpaceX Falcon 9 two stage to orbit launch vehicle during the SPX-4 resupply mission to the ISS. The satellite will be transferred into the ISS. The ISS crew will remove SpinSat from the launch configuration. The safe plug will be removed and a test plug installed to verify that Spinsat is functioning properly and in a safe condition prior to installing it onto the Cyclops orbital insertion apparatus developed by NASA Johnson Space Center. Once installed on Cyclops, the crew will verify that all safety inhibits are functioning properly, remove the test plug, and install the arm plug. The Cyclops will then be placed in the Japanese airlock and the airlock cycled. The ISS team will robotically remove Cyclops/SpinSat and position it in the deploy orientation. Cyclops will then deploy SpinSat with a Δv of 0.5 m/s and be restowed into the Japanese airlock.

6.2 On-Orbit Operations

Once deployed SpinSat will power up a few minutes, final value pending concurrence from NASA Phase III Payload Safety Review after the plunger inhibit switches disengage from the Cyclops pusher plate. The spacecraft will go through early orbit checkout to verify command and control capabilities, reference instrumentation functionality, and thruster control module performance verification. The characterization of the ESD thruster technology will be performed by firing the ESD thrusters in pairs and measuring the changes to the spin rate via on-board rate instrumentation. Ground campaigns by the international laser ranging network will also provide a high resolution means to determine the spin rate of SpinSat from the ground. Small Δv firings normal to the surface of SpinSat will be characterized by on-board accelerometers. The lifetime of the satellite is expected to be >6 months and characterization should be complete within the first four months.

7. ACKNOWLEDGEMENTS

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8. REFERENCES

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