Design and Test of a Solid State Charged Particle Detector for Cubesat

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Abstract

A solid state boron-ion implanted silicon Charged Particle Detector (CPD) was designed, built, and tested as one of the payloads for a Stanford University/Lockheed Martin Cubesat (10cm cube, 1 Kg) project intended for a low earth orbit. Design drivers to be discussed will include cost, size, mass and schedule. Two detectors were utilized with shielding to allow for two separate energy ranges to be detected. Stanford Research Institute facilities were used for testing. Design considerations will be discussed relating to tuning of the electronics for various low earth orbit altitudes, along with matching voltage requirements for the electronics with a low power, 3.7 Volt spacecraft bus (1 watt). Other payloads include a developmental sun sensor and Honeywell 3 axis solid state magnetometer. Overall cubesat development will also be discussed, including structure, communications, power, and processor. Testing techniques and current results will be shown for this ongoing project.

Introduction

A Lockheed Martin/Stanford University joint project was established in 2001 to become the core curriculum of a Certified Spacecraft Design Engineer program. Twenty-three Lockheed Martin engineers were the students, divided into 4 teams. Professor Robert Twiggs of the Department of Aeronautics and Astronautics, Space Systems Development Laboratory is the professor of this currently ongoing project. Two teams are flying an Aerospace, Corp. payload of an accordion style deployable solar array. The other team is flying an extremely sensitive magnetometer in order to detect extremely low frequency magnetic fields which are released prior to an earthquake.

For our project, called Hypercube (Team 1), the original design drivers included cost ($5K for one engineering unit and one flight unit), schedule (two academic quarters) and size (10cm cube, 1 Kg). Much later in the development phase these dimensions were increased to 4” by 5” by 5” and 1.25 Kg in order to accommodate the Aerospace Corp-developed launcher for a possible shuttle launch. This indicated a non-developmental, commercial off-the-shelf (COTS) component selection. Three payloads were also chosen; a magnetometer, developmental sunsensor, and a charged particle detector (CPD).

The mission of the Hypercube project is both scientific and technology development. Scientific data will be collected with the magnetometer and charged particle detectors, while the sunsensor and structure will provide the main technology development mission: both are new and have not flown yet. Many of the other COTS components will also provide a technology demonstration mission; the Li-Ion battery, radio and many of the other electronics.

Subsystem Design

The power subsystem for Hypercube consists of 3 components: solar cells provided by Lockheed Martin, a Lithium-Ion rechargeable battery, and power grid. The solar cells are 22% efficient multijuction GaAs, with 2.43 volts and 370mA per 4X6cm cell. There will be 3 cells on the 4"X5" sides (4 sides, 7.2 volts before back bias diode) and 2 cells on the two 4"X4" sides (4.8 volts). These lower voltage sides will have a boost regulator to get them up to a charging voltage ability. The secondary battery is a Panasonic Lithium Ion 3.7 volt 1500 mAhr cell. It includes some protection circuitry of 4.3 volt overvoltage cutoff, undervoltage cutoff of 2.3 volts, and overcurrent cutoff of 3.5 amps. Back-bias diodes will be placed in series with each side's solar array, and there will be MOSFET switches controlling power to an antenna release mechanism, the radio and terminal node controller (TNC), and the payloads. The onboard computer will be on all the time there is power. Voltage regulator chips will be used to achieve a 5 volt subarrary (CPD preamplfier and discriminator, sunsensor and TNC), 15 volt subarray (CPD discriminator and magnetometer), and 50 volt supply (CPD detector bias). A 2.5 volt reference from the magnetometer will be used for a variable voltage on the CPD discriminator, and the onboard computer and radio can work straight off the battery. Power will be cycled for conservation reasons; a nominal 90 second scenario would have the onboard computer on all the time (as it draws only 11-24 milliamps), cycle through the payloads for 10 seconds and the TNC for 20 seconds (with transmission for 4 seconds and reception for 16 seconds). This can be over-ridden by a data dump command, and there is enough power to allow for 10 minutes of transmit/receive.

Communication for the cubesat centers around a COTS Yaesu VX-1R radio with 150 mA receive current and 400 mA transmit current at 500 mW. A W2FS KISS TNC will also be used, which requires 270 mW of power. Four elements of 50cm steel measuring tape will be used as an antenna in a turnstyle deployment. A burn cord will hold the antenna folded around the body of the spacecraft until power is supplied from the battery to a burn wire. The data transfer will be at 1200 bits per second AFSK using AX.25 APRS protocol to allow amateur HAM operators to download data at 145.825 MHz.

Figure 1

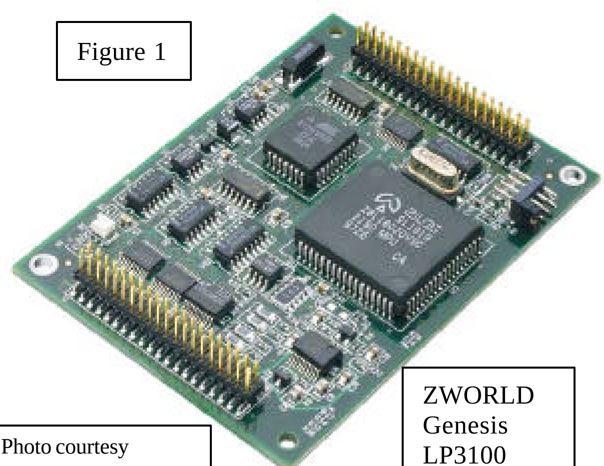


Photo courtesy ZWORLD webpage

ZWORLD Genesis LP3100 Processor

For command and data handling, a ZWORLD Genesis LP3100 (shown in figure 1) was selected as the onboard processor, due to its size (64X89X13 mm), weight and low power consumption (11 to 24 mA). It has 8 designated digital outputs, 8 configurable I/O's, and 4 12-bit A/D input channels. It also has a very high operating temperature range (-40 °C to +70 °C) and programs in Dynamic C. For memory, 512K flash EPROM is used for data storage and 512K static RAM is used for programs and variable space. A typical data cycle would be operation of the payloads for 10 seconds, generating 400 bytes of data. The LP3100 receives this data, stores it, and packages it in telemetry frames. The transmitter is then turned on for 4 seconds to allow for data transfer; 512 bytes of data can be transferred in 3.4 seconds at 1200 bps, plus the APRS frame is 128 bytes with 20% overhead. This means that 4 telemetry frames are required to transfer all the data from one cycle. Alternatively, a 10 minute data dump can be commanded, which will allow transmission of 90K bytes or 180 cycles (each EPROM holds 1000 cycles).

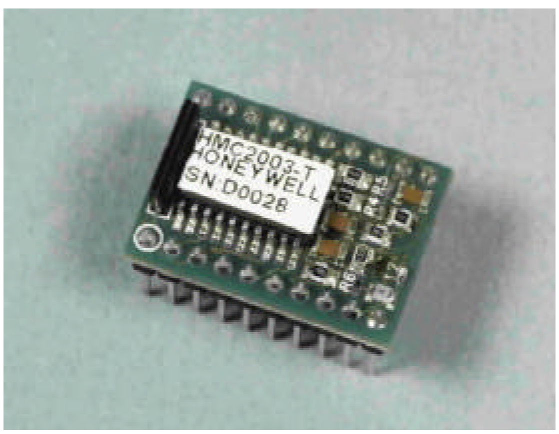


Figure 2: Honeywell HMC 2003 3 axis hybrid magnetometer Photo courtesy Honeywell SSEC webpage

Payloads

A magnetometer (figure 2) was selected as a payload to work in concert with the sunsensor and solar array current monitors to determine spacecraft attitude. We chose the Honeywell HMC 2003 3 axis unit because of its cost, weight, size (1" X .75"), and low power requirements (20mA current). It has a sensitivity of 40 μgauss, with a field range (saturation) of $\pm$ 2 gauss. This sensitivity will allow scientific data collection as well as attitude determination, and will be particularly useful when measuring charged particle fluxes around the spacecraft. It is a solid state device and can withstand 100 g shock and 2.2 g rms vibration with an operating temperature range of –40 °C to +85 °C. The analog output of magnetic field detection on all three axes (.5 to 4.5 volts) will be interfaced directly to the A/D input channels on the microprocessor, so that the only additional circuitry needed will be a set/reset circuit.

A developmental sunsensor was donated for our use by QinetiQ Ltd (figure 3). Its extremely low weight and small size along with its minimal power requirements (20mW) were important factors in our decision to

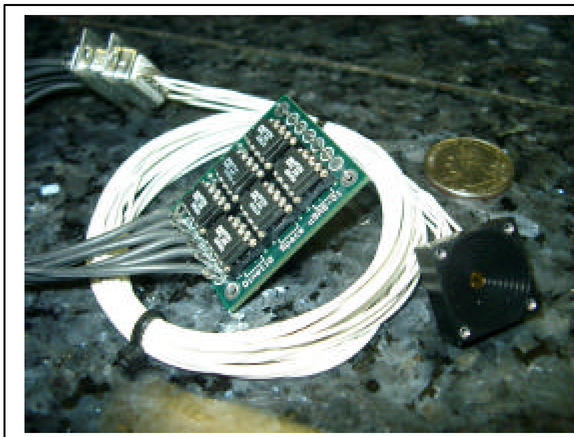


Figure 3: QinetiQ, Ltd. Prototype sunsensor with electronics board

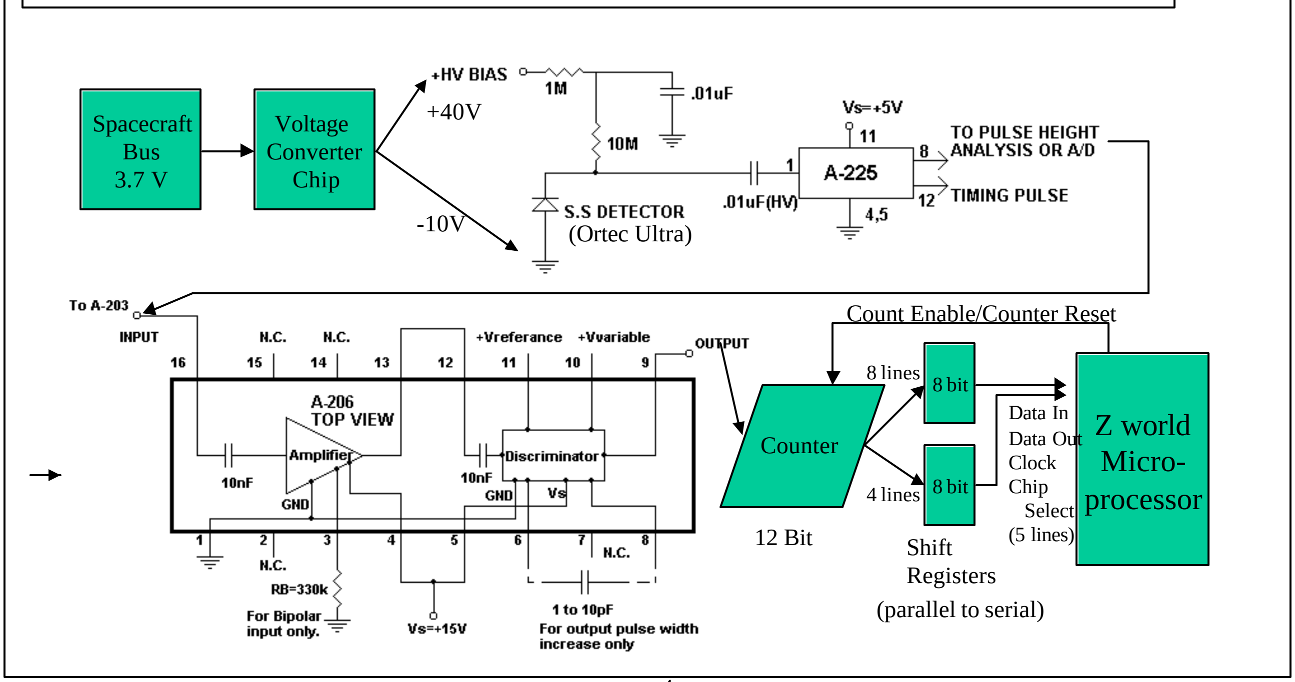
include it as a payload. The total unit consists of an electronics board (40X35 mm) with a temperature sensor and two solar detector units (20mm square). Its operating voltage is 5 volts, and its analog output varies between 0 and 5 volts. By comparing the relative differences between the 4 analog voltages from the photodetector quad cell in each detector head, one can determine the angle of the sun relative to the detector mounting plane. The field of view for each detector is $\pm$ 60 degrees, so a minimum of two separate detector heads allow for solar direction determination. QinetiQ claims a resolution of better than .2°, however we probably will be unable to verify this accuracy with our test equipment and are expecting a 1° nominal resolution.

Having three completely separate methods of determining spacecraft attitude is an important characterization ability. Our roughest form of attitude determination will be monitoring solar cell current from each side. Since we have solar cells on all 6 sides of our spacecraft, this should give us a good idea of where we are pointing relative to direct sunlight. This can be checked with our magnetometer, which is sensitive enough to give us a feel for geographical location as well as good resolution for spacecraft attitude. Finally, the sunsensor will give us very fine attitude determination, but since this is a developmental unit we can check these results with the magnetometer. We should then be able to report on all three techniques for attitude determination and verification and their relative resolution, which should be of great interest to future small satellite developers.

Charged Particle Detector

The charged particle detector (CPD) was designed around the Ultra boron ion implanted silicon solid state detector produced by Ortec in Oak Ridge, TN. Due to our expected orbit, we chose the 100 µm depth series, which with an active detector area of 25 mm^2 allows an electron energy detection range between about 4 and 100 KeV. Two detectors will be used to detect two different energy ranges; the first will have only an opaque cover to block sunlight but allow all energies of charged particles to interact with the detector. The second will have a thickness of aluminum over it which will block the lower energy electrons but allow 100 KeV and higher to interact with the detector. Since we expect to encounter more electrons (~10^5 e- per cm^2 sec steradian)[1] than the detector can handle during our expected 3 month operational life (max of 10^{13} e- per cm^2 total number), the field of

Figure 4: Electronic layout of charged particle detector (some elements courtesy AMPTEK webpage)



view will be limited by retracting the detector face by several millimeters from the outside edge of the spacecraft, and the entrance aperture on the spacecraft surface will be smaller than the detector area.

This low amount of noise necessitated an extremely low noise preamp. After several discussions with AMPTEK and Ortec it was decided to use the AMPTEK A225 preamp (which has a noise of 2.5 KeV FWHM in silicon, compared to ~4 KeV for the detector) and A206 discriminator. The diagram in figure 4 depicts AMPTEK's suggested wiring diagram. Note that the Ultra detector requires a 50 volt bias. The preamp and discriminator were mounted on the test board supplied by AMPTEK, and this was secured to a larger board which contained a protoboard with a 555 timer circuit used to generate pulses to test the electronics along with the 4 required voltage sources (50, 5, 15 and 2.5 volts)—see figure 5. It was seen in early testing that the preamp circuit was extremely sensitive and required battery voltage sources in order to avoid being swamped with the noise coming from power sources (see figure 6). Best results so far have been achieved by isolating the preamp board inside an aluminum box or by placing the entire large board with batteries in a large metal container.

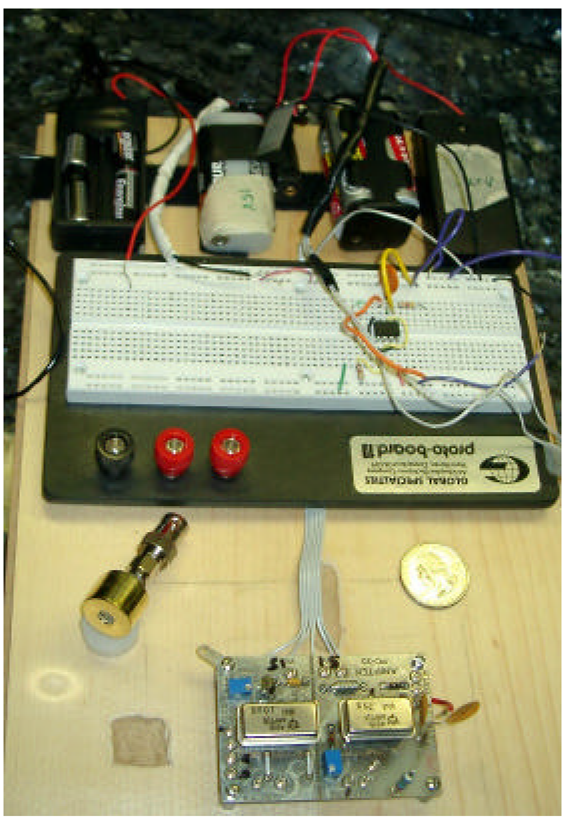


Figure 5: CPD test board: 4 voltage sources across top (batteries): left to right: 50V, 15V, 5V, 2.5V. Protoboard with 555 timer circuit, AMPTEK test board with A225 preamp and A206 discriminator, Ortec Ultra detector to left (gold can)

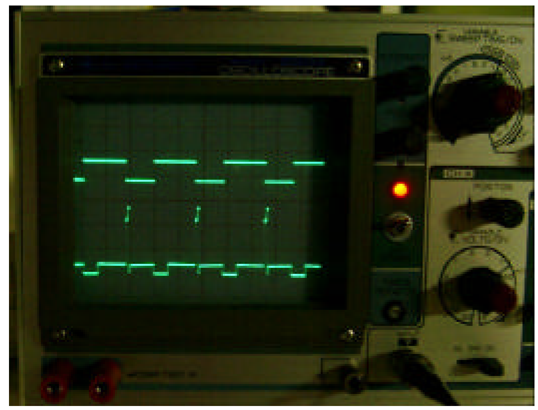


Figure 6: typical preamp output—top trace is 555 pulse timer output, input to detector electronics. Preamp output is lower trace, note shaped spike at negative pulse from 555

Figure 7: electron beam test chamber at SRI
Photo courtesy SRI

Testing of the detector and electronics with a known electron source will be accomplished utilizing at the SRI-developed Spacecraft Charging test facility at Stanford Research Institute's Menlo Park campus (figure 7). This was developed for measurement and test of material electrostatic discharges and for exposing samples and instruments to wide-area beams of electrons between 5 keV and 30 keV. The chamber can be evacuated to a pressure of approximately 5 x 10^{-5} Torr, and a multipactor electron source of approximately 30 cm diameter is used to generate the wide-area electron beam to expose the test samples at the prescribed energies and current densities. Beam uniformity measurements using a swept faraday-cup have previously indicated beam intensity variations of +/- 25% across the sample area. Beam current densities from approximately 0.5 nanoAmps/cm^2 to greater than 20 nanoAmps/cm^2 can be generated. Since this is a much higher count rate than our current electronics allow (the rise time of the preamp is 2.4 μsec, suggesting that our maximum electron count rate would be about 300,000 e- per second). Prior measurements, in which the test sample area was covered with an aluminum plate, have also been used to determine the average beam current density at the sample position.

Structure

The design of our cubesat is shown in figures 8 and 9. Figure 8 shows the 4 internal electronic boards: processor, payload/ADC, power and TNC. The radio is not visible, but the two blue dots are where the charged particle detectors will be. The 3 solar cells which fit on the larger sides can also be seen. Figure 9 shows the top part which fits on the main structure almost like a 'lid'. This unit houses the antenna which will be wrapped around the groove during stowage and launch. Cabling will take up the interior part of this structure. Standoff springs will be used between the cubesat and launcher, and also between the cubesat and any other cubesats which will be in the same launcher assembly.

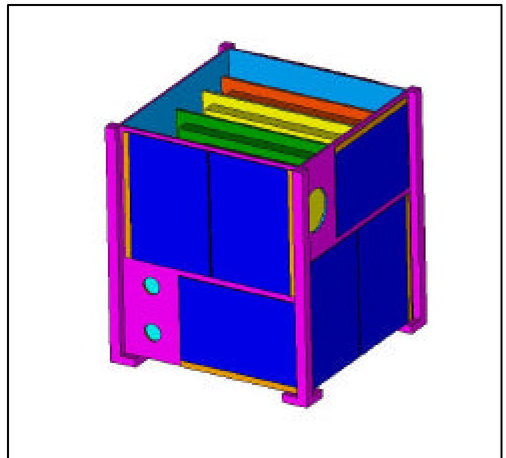


Figure 8: Hypercube structure design showing CPDs, solar cells, and electronic boards

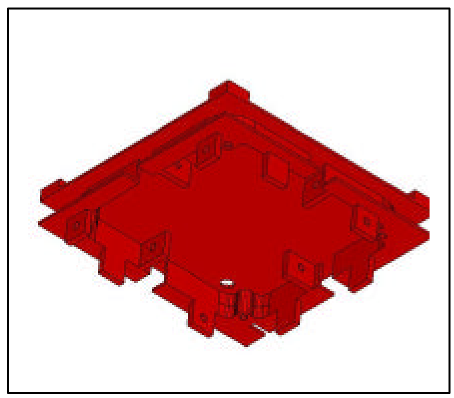


Figure 9: Hypercube top structure design, including antenna grooves

Future Techniques

A technology worth discussing at this point is a new material deposition technique being developed by a group at Lockheed Martin Aeronautics in Fort Worth, TX. This technique uses a laser to fuse powdered metal into complex shapes. The metal is transported to the fusion point by 4 gas channels. Original prototypes have been made in 316 stainless steel, but since this technique does not work with aluminum due to its high reflectivity in its molten state, final flight versions might be made of titanium.

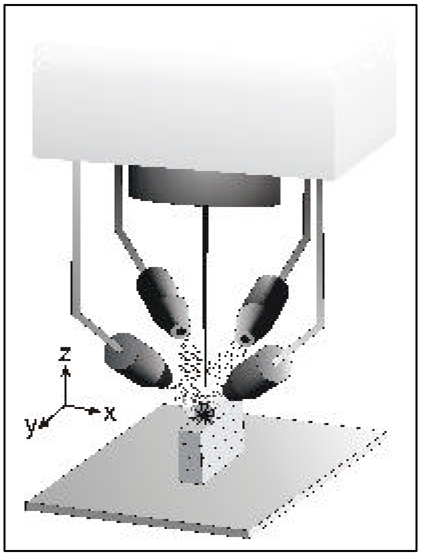


Figure 10

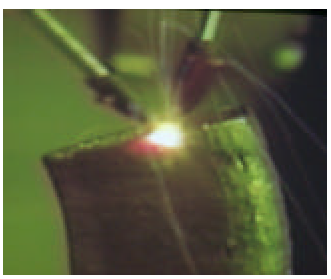


Figure 11

Figure 12

To bring the powdered metal to the desired location in space, 4 gas channels are used with variable flow (figure 10). Consistent particle size and purity are critical to an even distribution of material. A doubled Nd:YAG laser is used to fuse the powdered metal in the exact location needed to construct the material (figure 11). An inert gas such as Argon is used to transport the powdered metal and fill the chamber where the fusion takes place in order to avoid oxidation. Heat dissipation can be controlled by the flow of the inert chamber gas. Currently with this technique, boxes with walls on the order of 1mm wide are possible. Bracing is generally required to keep the walls of the structure from buckling, but this might be accomplished by using channels to support electronic boards. Figure 12 shows an example of a type of structure in stainless steel.

References

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AMPTEK data sheets for A206 and A225 electronic components
http://www.amptek.com/a225.html

Ortec-online web page and data sheets: http://www.ortec-online.com/detectors/chargedparticle/ultra.htm

Maxim web page and data sheets: http://www.maxim-ic.com/

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QinetiQ, Ltd. Data sheets and ICD for prototype Mark 0 sunsensor