

CubeSat: Developing a Standard Bus for Picosatellites

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Abstract

The purpose of the CubeSat development is to define a standard bus that can be used by anyone needing a simple picosatellite. Defining a standard bus, developing standard hardware components using commercial off the shelf components and a standard spacecraft frame will simplify the development of picosatellites. The CubeSat development will provide a standard spacecraft frame, a spacecraft controller, radio transceiver, attitude determination and control, solar cells, batteries, and an interface for a payload. The developer needs only to concentrate on the payload.

Keywords: picosatellite, standard bus, CubeSat

1. Introduction

The StenSat Group is a team of engineers and radio amateurs who came together to design satellites in their spare time, simply because they thought it was cool. Kevin Doherty, Hank Heidt, Jim McGuire, and Dave Niemi made up the original group, which designed and built the first StenSat amateur radio satellite. Joining the team for the CubeSat effort are Ivan Galysh and Gil Dutchover from the Naval Research Laboratory. Each member brings unique skills to the group, and we're all committed to designing and launching a successful satellite.

The group is attempting to develop a standardized generic satellite bus. A solid, general-purpose vehicle design will allow users to concentrate on their unique payloads, and free them repeated satellite system design efforts. Our CubeSat design will attempt to provide standard structures and services to allow simple yet flexible control of a variety of payloads, while constraining those payloads as little as possible.

The test vehicle being designed (tentatively named StenSat-2), will comply with the interface requirements of the CubeSat program being administered by Prof. Bob Twiggs of Stanford University.

2. CubeSat Requirements

The CubeSat interface control document specifies the picosatellite launcher interface to be eight 8.5 mm wide by 10 cm long mating surfaces of 7075 aluminum, which when taken in pairs define four parallel edges of a 10 cm cube. The 10 cm cube has a total mass of 1 kg. No further constraints on the mechanical interface are defined at this time.

The most limiting constraint is that the total mass of the satellite to not exceed 1 kg. This constraint is extremely severe, not because it is too small, but because it is too small in relation to the defined 10 cubic centimeter volume. The satellite density must be no greater than one gram per cubic centimeter; CubeSats have an implied requirement that they float in water. The low density has required compromises in the chassis design, which will be described later. The current CubeSat design has a mass of approximately 800 grams (less payload), of which circa 500 grams is structure enclosing empty space.

(The StenSat group has recommended that the CubeSat dimensions be reduced so as to keep the mass below 1 kg, in consideration of the density of the materials that will be used to construct it.)

In addition to the program requirements, our team has selected the following design guidelines:

- ◆ The material cost shall be less than one thousand dollars.
- ◆ The design shall be simple and use off-the-shelf components wherever possible
- ◆ The system shall be fault tolerant. Critical systems shall be 1+1 redundant.
- ◆ Wherever possible, multiple functions shall be integrated into a single assembly
 - Example: solar panels integrated with magnetorquer coils
 - Example: internal circuit boards serve as structural members

3. Current Implementation

StenSat-2 is a cubic structure of aluminum with a stack of circuit boards inside as shown in figure one. Each face of the satellite is covered with solar cells as shown in figure two. The center of the satellite has rechargeable batteries and a power distribution system. The batteries divide the satellite into two halves. One half of the satellite contains the satellite computer, communications electronics, and attitude control system. The other half is available for a mission payload.

Preliminary estimates indicated that over half of the spacecraft mass budget would be consumed by the mechanical structure. These estimates include consideration for a full-enclosure radiation shield comprised of 1.5mm (0.059") thick aluminum. The system mass allocation is as follows:

1 kg total:	500 gm structure
	200 gm systems
	100 gm batteries
	<200 gm Mission Payload

The mechanical structure aims to be as simple and functional as possible. The aluminum chassis is composed of three basic parts: a side panel, a top panel, and a corner rail. The complete chassis requires two tops, four sides, and four corners. The individual elements are secured using a thermally conductive epoxy. The circuit boards in the interior of the spacecraft serve as torsional strength members. Final assembly of the spacecraft requires some simple fixtures that guarantee mechanical alignment.

Each face on the Stensat-2 cube is outfitted with solar cells. While the vehicle is expected to have an attitude control system, the design does not prohibit the vehicle from being operated in a free-flight "tumbling" mode. Also, if the attitude control system fails, solar power is guaranteed regardless of spacecraft orientation. System power calculations are shown below:

1.25 W minimum @ solar cells in sun
(assumes conservative 15% solar cell efficiency)
500 mW mean @ power bus
(assumes 60% power converter efficiency, 30% time in eclipse)
250 mW target bus dissipation (less payload)
(assumes 10% transmitter duty cycle, 50% transmitter efficiency)

Each side panel contains an array of eight 20x40 mm solar cells. The cells are connected as two strings of four cells each. The cells are mounted to a 0.5mm (0.020") thick PCB, which is epoxied to the aluminum surface plate. A small charge-pump voltage converter is provided on each side. If the panel has sufficient illumination to operate the charge-pump, the array will connect itself to the main spacecraft power bus through a low resistance FET. Failed solar arrays, open or short, are electrically disconnected from the main power bus.

A pair of switching regulators provides system power. Each regulator is capable of operating the system should the other fail. The regulators convert a nominal 1.6V from the solar panels to a more usable 4.0V. The regulated bus is also connected to a Lithium-Ion battery capable of operating over 2.8V-4.2V. The battery provides power to the spacecraft during periods of eclipse, and provides peak power beyond the 250mW mean dissipation target. Both the 1.6V unregulated and 4.0V regulated busses are available to all boards within the spacecraft.

The communications system in Stensat-2 is the largest power consuming section. The system is designed to operate as a Mode J repeater in the Amateur Radio frequency bands. Since the satellite will operate within the amateur radio bands, all information regarding the frequencies, telemetry, and operating procedures shall be published and available to the amateur radio community. Both halves of the communications system were designed to be compatible with typical amateur radio satellite ground stations. Figure three shows the configuration of the communications system.

Stensat-2 uses a 2-meter uplink at 145.84MHz. The satellite receiver is designed around a Motorola MC13136 with preselector and preamplifier. Receiver sensitivity is expected to be approximately 1uV at the antenna feed point. A sub-resonant center-fed dipole antenna is mounted on the exterior of the spacecraft, and will deploy when the spacecraft exits the launch tube. The receiver is capable of detecting voice audio in a 4kHz bandwidth, or DTMF tones at 100 bits/second or less.

The Stensat-2 downlink operates in the 70cm band at 436.625MHz. The transmitter, a Motorola MC13176 with Class-C final, delivers +30dBm into a resonant center-fed dipole antenna. The transmitter has a 4kHz analog bandwidth for its voice-repeater operation, and is capable of 9600 baud operation using a standard amateur-radio FSK modem.

The satellite controller contains a single chip computer. The processor selected is the PIC16C77 microcontroller from Microchip. The controller accepts commands from a DTMF decoder using a dual tone sequence through its digital interface. The controller generates amateur radio AX.25 telemetry packets that are compatible with the amateur radio Automatic Packet Reporting System (APRS) protocol. The telemetry packet contains spacecraft temperature data, bus voltage data, power current data, payload sensor data, and operation status collected with the PIC's analog to digital converters and digital ports. The PIC also provides analog sensor signals and digital signals to the payload. Figure four shows the configuration of the satellite controller.

The satellite processor is protected with a latchup detection circuit. Since the PIC is susceptible to single event upsets and latchup conditions, some external circuits are added to protect the PIC. A high-side current switch with a current limiting sensor is used to detect a latchup condition that cause the PIC's current to increase significantly. An external watchdog timer is included to handle single event upsets and latchups that are not detected by the current switch. The internal PIC watchdog timer is also used in the event the external watchdog timer latches up or upsets. Monitoring small current increases with a current sensor and software is used to detect latchups that do not cause the current switch to cycle power to the controller board. Using all of the methods improves the reliability of the PIC surviving in space until it fails from total dose radiation exposure. Operating the PIC at a lower voltage increases the life of the PIC.

4. Mission Payload

Almost half of Stensat-2's interior volume is available for use by mission payloads. The mission payload has physical access to one 'top' face of the spacecraft, and mounting locations for up to three dedicated circuit cards. Simple payloads can be administered through the resources provided by the system bus. Complex payloads may carry their own CPU and telemetry system, and may modify the external structure of the spacecraft. For example, an experiment that requires exposure to the space environment may require significant modifications to the top cover. Moderate modifications to the spacecraft structure will not compromise the system architecture.

Board-to-board connectors have been defined as part of the Stensat-2 system, and only one connector is required for compliance with the system specification. Additional payload-specific connectors may be added as necessary.

The mission payload is restricted in its power usage. The spacecraft aims to maintain 500 mW mean power consumption including the payload. The main PIC CPU is capable of measuring mission payload power as well as total system power. The mission payload may be disconnected from the main power bus in the event of an over current failure.

The main PIC CPU provides a minimal set of generic services on the system bus. A limited number of simple resources are provided as a baseline for simple mission payloads.

The main PIC CPU supervises the mission payload. The PIC requires periodic confirmation that the payload is operating

properly, and will reset the payload if confirmation is not received within a preprogrammed timeout period. The PIC may be programmed to automatically reset the payload on a predetermined schedule, or may be instructed to perform a reset through a DTMF uplink command.

Two 8-bit A/D converters monitor signals on the spacecraft bus. The A/D conversions occur on a predetermined schedule, and the conversion values are included in a standard downlink telemetry packet.

Four bi-directional digital I/O lines are provided for mission payload control and status functions. Each signal direction may be controlled independently. Output signals (inputs to the mission payload) may be controlled by DTMF uplink commands. Input signals (outputs from the mission payload) are monitored periodically and are included in a standard downlink telemetry packet.

Expansion resources are provided through the use of an I²C bus and direct telemetry access. The I²C bus uses an industry-standard 2-wire interface, and operates at a 400 kbps rate between the controller and the mission payload. DTMF uplink commands may be converted into I²C bus commands. For data-intensive mission payloads, access to the receiver demodulator output and the transmitter modulator input is provided. The mission payload may monitor the uplink continuously, and may directly modulate the transmitter as scheduled by the main PIC controller.

The first payload will be an attitude control experiment. The payload will consist of magnetic field sensors, magnetorquers, and a dedicated attitude control processor. Three orthogonal magnetic field sensors provide measurements for each axis. The attitude control processor will correlate the sensor data with keplerian elements uploaded from a ground station. Attitude control is achieved using three pairs of orthogonal magnetorquers mounted on the exterior of the spacecraft. Maintaining an optimal sun-pointing orientation may increase the available system power to 1 watt mean dissipation, and makes deployable solar arrays a viable reality for higher power experiments. Subsequent satellite designs will incorporate the attitude control system as a standard element.

5. Component Radiation Data

The MC13136 and MC13175 receiver and transmitter integrated circuits are manufactured using Motorola's MOSAIC process. The MOSAIC process produces radiation tolerant devices. The transmitter amplifier uses bipolar-junction technology and is also radiation tolerant. These devices have been used in previous commercial and military satellites.

The PIC microcontroller underwent radiation testing at the Naval Research Laboratory. Latchup and SEU testing were performed by exposing the die to a laser. The laser is used to approximate high energy particles. Latchup occur at about 17 MeV. When exposing certain parts of the die, the processor continued to operate during a latchup event. The maximum current measured during a latchup was about 600 milliamperes. The PIC controller was never damaged by a latchup. The SEUs occurred at about 2 to 3 MeV. It took little energy to flip bits. Exposing the EEPROM to the laser actually induced program execution errors. None of the EEPROM bits could be flipped by laser exposure. The maximum energy level exposed was about 300 MeV. Even at this high energy level, no damage was induced. When the watchdog timer was exposed to the laser, it stopped working. It was not able to reboot the processor. For total dose testing, the PIC controller was exposed to an x-ray source. The x-ray source was calibrated so that the results could be correlated to a cobalt-60 radiation source. The PIC controller failed after being exposed to 20 Krads. The failure was due to the EEPROM being set to all zeros. The PIC controller was able to be reprogrammed. For a second exposure, the PIC controller was exposed to an additional 15 Krads before complete failure. The PIC controller couldn't be reprogrammed after the second exposure. Current CMOS devices typically have a radiation tolerance around 20 Krads.

6. Satellite Structure Evolution

When the group started designing the CubeSat, the group went through a few design evolutions. The initial design specification early in the program was for a 3.5-inch cube. Figure five shows the structure of the 3.5-inch CubeSat. The design was highly modular, using only two different types of panels. All of the side panels are identical and are made of 1/8-inch aluminum. The center area is machined to allow for recessed mounting of a solar panel and magnetorquer coil. The inner side of the panels has rails machined to secure circuit boards. The top and bottom covers are identical and are designed to deploy

antennas. The antennas are constructed of piano wire and wrap around a racetrack machined into each top and bottom cover. Solar panels can be mounted on the exterior surfaces of both top and bottom covers. The exterior structure also provides shielding for the RF electronics. The RF circuit board mounts in a well machined into the “top” cover. The RF components are oriented into the panel cavity and are largely isolated from the remainder of the spacecraft. The remaining circuit boards are stacked and can slide into a partially assembled structure using the rails on the side panels.

Assembly is simple. The RF circuit board has connectors for each face. The connectors are used to connect the solar cells, magnetorquers, and sensors to the bus. Each side panel contains the mating connector. Three of the four side panels are attached to both the top and bottom covers, leaving one side open. The connectors help in alignment and provide the snap together assembly. The circuit board stack with battery compartment slides into the structure through the open side. The last side panel is then snapped in and all screws can be inserted.

The second structure design was based on the 3.5-inch cube. The main difference is increasing the size to a 10-centimeter cube and adding rub rails. The 10 cm cube meets the CubeSat structural requirements. Panels on each of the six faces were 3mm (0.118”) thick aluminum. The structure alone had a mass greater than 1.5 kilograms, far exceeding the 1.0 kg CubeSat program limit.

The latest design uses minimal aluminum. Four slotted posts are connected together with thin sheets of aluminum. The assembly requires epoxy. No screws are used. The circuit boards contribute to the structural strength. The battery is not mounted in an enclosure, but is epoxied between two circuit boards. Even with this minimal approach to the chassis, the mass of the structural elements still exceeds half of the 1 kg budget.



Figure 1, CubeSat Internal Structure

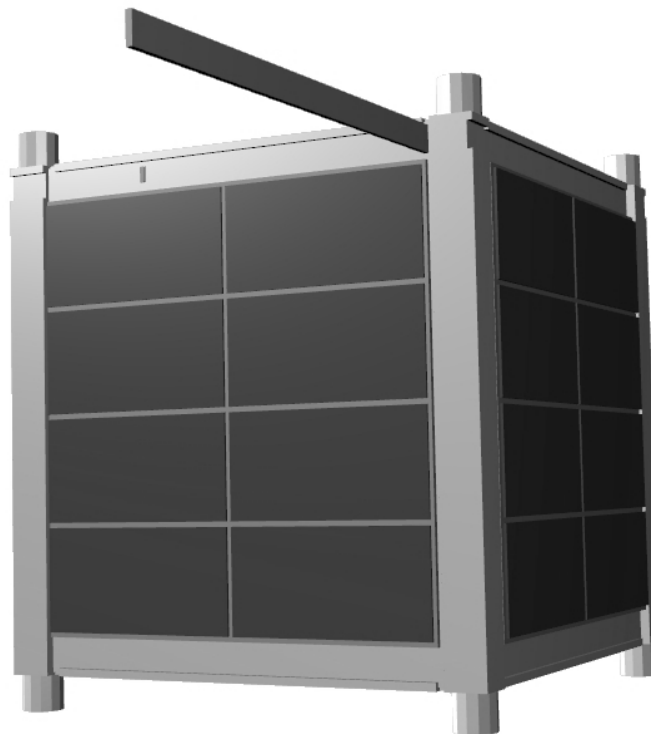


Figure 2, CubeSat External Structure

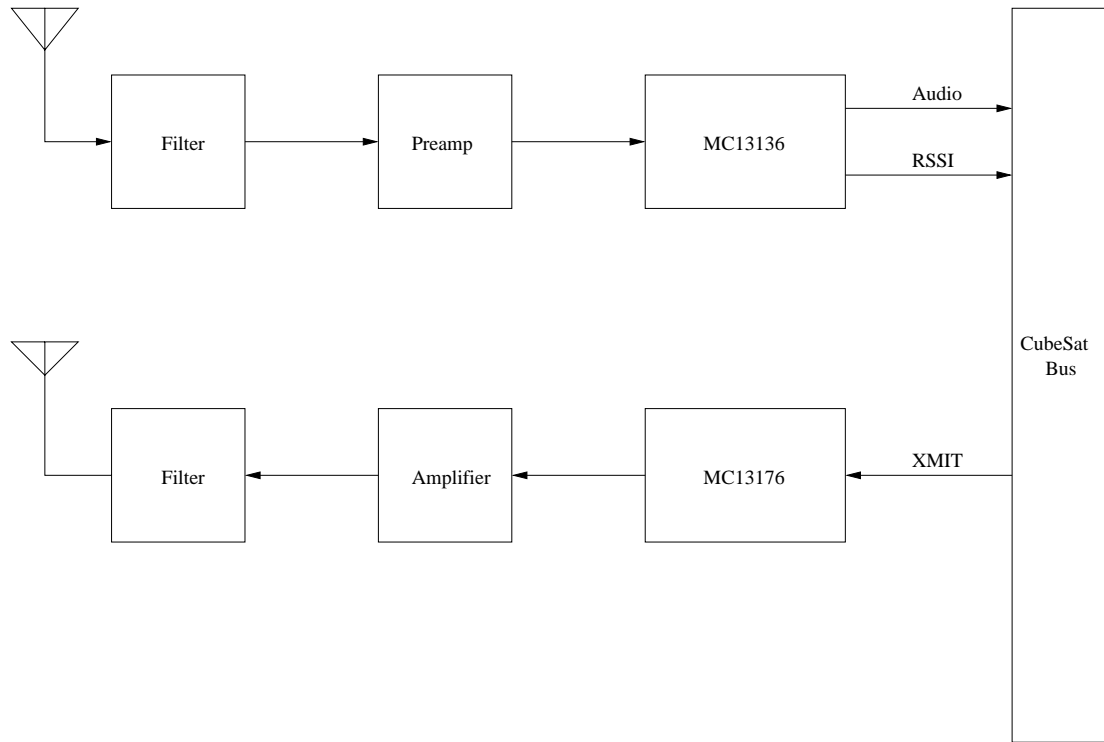


Figure 3, Communications System Configuration

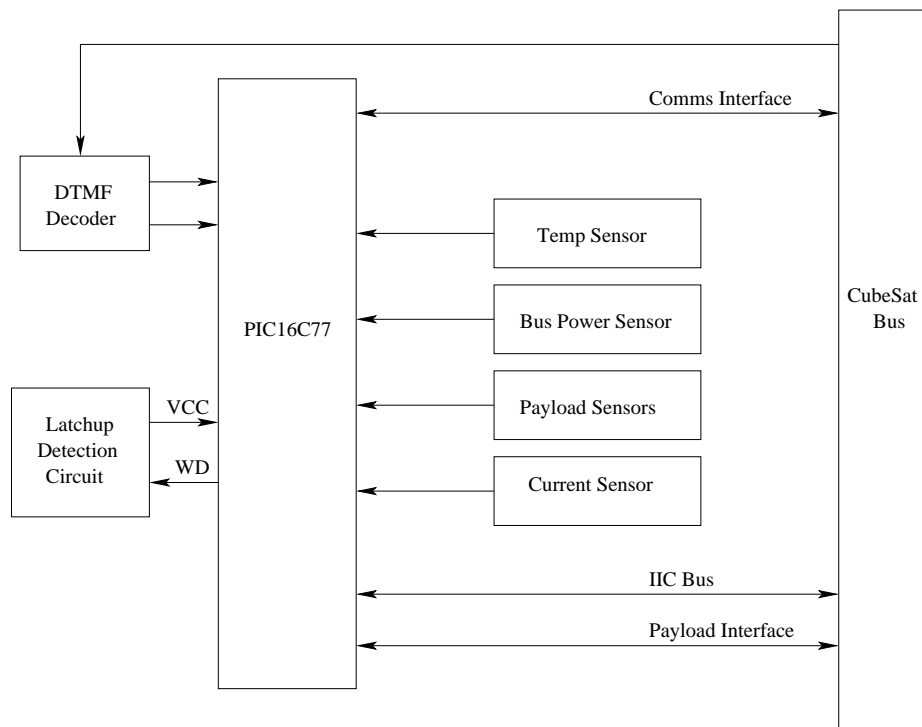


Figure 4, Satellite Controller Configuration

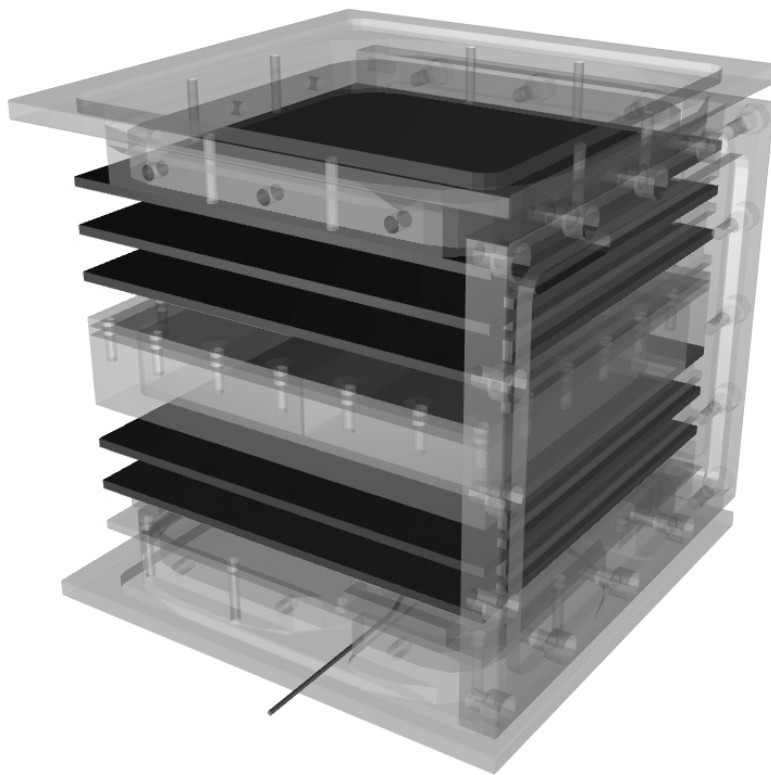


Figure 5, First CubeSat Implementation



Figure 6, Second CubeSat Implementation