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The *.Sat CubeSat Bus: When Three Cubes Meet

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ABSTRACT: A collaborative effort between Stanford University and the National Aeronautics and Space Administration (NASA) begun in 2003 continues today with the primary objective of supporting a NASA Ames Research Center (ARC) biological payload scheduled for launch into orbit aboard a satellite conforming to the International CubeSat Specification authored in 2001. The Stanford portion of the program, named “StarDotSat” (*.Sat), seeks to put forth a standardized spacecraft bus adhering to the CubeSat Specification that provides all necessary services to one or more attached payload modules.

The CubeSat specification was the result of a joint effort by the California Polytechnic University, San Luis Obispo (CalPoly SLO) and Stanford University to provide the basis for low-cost, rapid-turnaround access to space geared primarily towards the University audience. Leading to development of a standardized launcher known as the Poly-Picosatellite Orbital Deployer (P-POD) and subsequent launch service partnerships by the team at CalPoly, the CubeSat standard has grown to include over 70 universities worldwide.

Based upon the CubeSat Specification, the *.Sat design implements a standardized set of mechanical, electrical, and software interfaces that allow the Bus Module, itself a single-sized CubeSat, to provide services such as electrical power, command and data handling, and communications to one or more Payload Modules. By defining the demarcation point between Bus and Payload(s), the *.Sat program allowed the interface itself to become the focal point for independent development of the Bus and Payload elements prior to integration.

The initial mission, encompassing the use of the Stanford *.Sat Bus with a NASA ARC-designed biological payload has been named “GeneSat-1” (GS-1). The overall mission is a cooperative effort between NASA and various universities partnered at the Space Technology Center, managed by San Jose State University, including Santa Clara University, CalPoly, and Stanford.

This paper will present the development and implementation of the *.Sat Bus from the perspective of the Stanford University development team, highlighting the successful cooperation between academia and industry expertise.

INTRODUCTION

In the field of satellite design, there is often discussion of adopting standards to accelerate the design, integration, test, and flight operations of complete satellite systems by reducing the degree to which common subsystems need to be iterated upon during the design-cycle. To this end, critical subsystems comprised within satellite “bus” systems are a common focal point of efforts to standardize both the hardware and software interfaces to a payload component. In so doing, the end goal is to free mission designers from the need to re-design critical systems and interfaces for every mission, allowing them the freedom to concentrate upon the mission requirements.

However, the systems engineering trade with standardization must balance the upper bounds of a standard capability within an envelope that will satisfy the greatest number of mission requirements possible. A standard that limits the mission requirements can be construed as counter-productive in providing a robust design for a wide variety of space missions.

In the academic setting, such as at Stanford University, adoption of a “standardized” bus infrastructure with a robust set of capabilities is a potential asset, as it provides relief for the limited-duration schedules of academic projects, allowing focus on mission design rather than design-iteration for critical subsystems required on every space vehicle (S/V). In an effort to minimize time from design to flight, *.Sat establishes a robust set of standardized interfaces applicable to academic missions, which can serve scientific and commercial missions as well. As an added benefit, adoption of standardized infrastructure between universities allows easier collaboration both in spacecraft and mission development, as well as integration, test, and launch operation phases - traditionally very expensive, both in cost and duration.

As the foundation of the StarDotSat (*.Sat) program, the CubeSat specification was the result of a joint effort in 2001 by the California Polytechnic University, San Luis Obispo (Cal Poly) and Stanford University to provide the basis for low-cost, rapid-turnaround access to space geared primarily towards the University audience. Leading to the development of a standardized launcher known as the Poly-Picosatellite Orbital Deployer (P-POD) and subsequent launch service partnerships fostered by the team at CalPoly, the CubeSat standard has grown to include over 70 universities worldwide.

Based upon the CubeSat Specification, the *.Sat design implements a standardized set of mechanical, electrical, and software interfaces that allow the Bus Module (BM), itself a single-sized CubeSat, to provide services such as electrical power, command and data handling, and communications, to one or more Payload Modules (PM). By defining the demarcation point between Bus and Payload(s), the *.Sat program allowed the interfaces shown in Figure 1 to become the focal point for independent development of the Bus and Payload elements prior to integration.

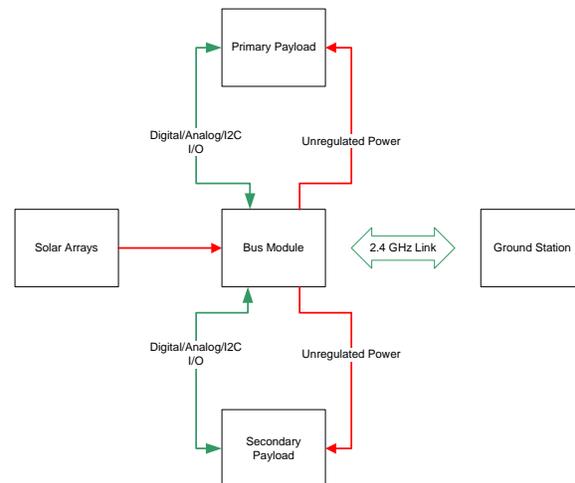


Figure 1: Key *.Sat Data and Power Interfaces

The initial mission, encompassing the use of the Stanford *.Sat Bus with a NASA Astrobiology-designed biological payload has been named “GeneSat-1” (GS-1). The overall mission is a cooperative effort between NASA and various universities partnered at the Space Technology Center (STC) located at the Ames Research Center, managed by San Jose State University (SJSU), including Santa Clara University (SCU), Cal Poly, and Stanford.

SYSTEM REQUIREMENTS

To support the initial design for the *.Sat Bus, the overall design requirements can be categorized as either academic or science-related. Drawing from heritage Stanford University picosatellite programs including QuakeSat and NemaSat, the *.Sat team had an established mission to support a NASA biological payload. Thus, there were aspects incorporated into the design to accommodate this particular mission while keeping in mind the need to keep the interfaces and related technologies as robust as possible.

Driving the academic side of the project was the overarching need to fit within the realm of the academic schedule. One of the key aspects of the CubeSat

Specification in general is to provide rapid turnaround times by limiting the physical size and mass of the vehicles. In order to stay within the overall CubeSat Spec, the design of the *.Sat bus system was constrained to remain within the parameters of a single-unit CubeSat, taking a modular approach to leverage up to the full triple-unit CubeSat accommodation in the P-POD launcher.

Inherent to the constraints on vehicle size, the upper limits of power and communications capability also provide grounds for a standardized envelope in this form-factor. As a secondary payload subject to the needs and requirements of the primary customer on the Launch Vehicle (L/V), the need to support a variety of possible orbits was another key requirement of the system. To support this need, a variety of simulation and design tools were implemented to provide rapid analysis of the effect of orbital parameters on power, communications, and attitude control/determination.

In support of the GS-1 biological payload, the requirements were very specific in the areas of thermal interfaces/protection as well as in the micro-G environment allowed for the S/V in orbit. Due to limited shelf life of the biology, the test and integration requirements of this payload provide additional challenges for the system design. In many respects, these particular requirements are rather unique for S/V in the picosatellite class. They are difficult to meet in traditional missions involving biology.

As with any mission involving science, the typical end product that all customers seek is simply the experimental data and knowledge of the conditions under which they were collected. To this end, a robust capability to collect telemetry both for S/V health and status, as well as experimental conditions was a core requirement on the data handling and communications subsystems.

IMPLEMENTATION

Space Vehicle Overview

Figure 2 and Figure 3 show the *.Sat S/V in the configuration that will support the GS-1 mission: a one-cube Bus Module (BM) and a Payload Module (PM) occupying two cubes. This configuration utilizes the full three-cube capacity of the P-POD. The entire spacecraft is covered with full-length aluminum body panels, which provide both thermal radiation surface as well as a substrate for the solar arrays. The entire S/V is 339 mm long, as per the CubeSat Specification, with a total mass not to exceed 3 kg.

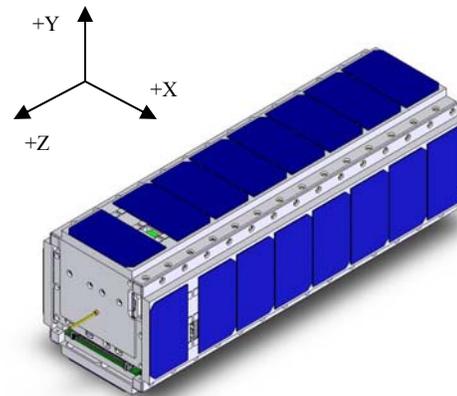


Figure 2: 1.0 Bus/2.0 Payload Configuration
(Body panels attached, S/V global coordinate frame as shown)

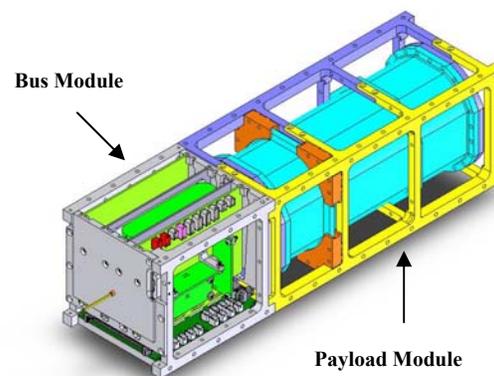


Figure 3: 1.0 Bus/2.0 Payload Configuration
(Body panels removed)

Payload Overview

The GS-1 payload is a fully autonomous platform for performing genetic experiments on E.Coli bacteria. The payload consists of a 12-well fluidics plate and uses 12 custom optical units for fluorescence (GFP) and growth-rate measurement.

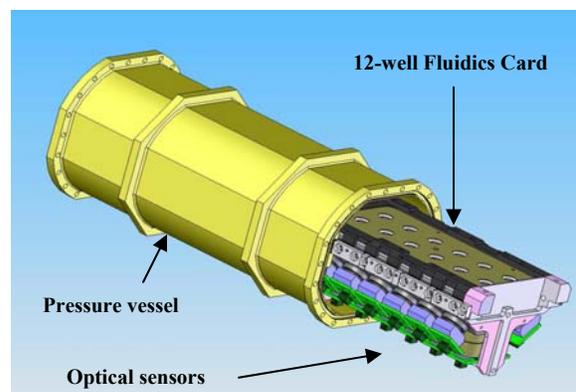


Figure 4: GS-1 Payload Module Internals
(Payload frame is not shown.)

The internal payload environment is maintained at atmospheric pressure, 90% relative humidity (RH), and an active-phase temperature of $35\pm 0.5^\circ\text{C}$.

Bus Module Overview

The core of the *.Sat system is a modular approach to housing the various components of the integrated S/V. To provide the maximum flexibility in final mission configuration, the Bus Module (BM) was designed as a single-unit CubeSat that, for all intents and purposes, could fly as a standalone CubeSat. The primary mechanical interface at both ends of the BM allows either a configuration with dual payload modules (PM) of varying sizes, or a single dual-unit CubeSat payload, such as GS-1 at one end as shown in Figure 5.

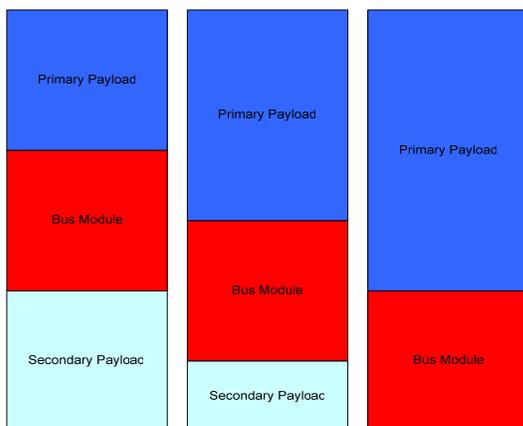


Figure 5: *.Sat Modular Configurations
(L-R: 1.0/1.0/1.0, 0.5/1.0/1.5, 1.0/2.0 configurations)

The BM provides all essential services to the PM(s) including Command and Data Handling (C&DH), Electrical Power (EPS), Communications, Attitude Determination and Control (ADCS), Thermal Control (TCS), and primary Structural/Mechanical support.

The BM internal design houses the electronics printed-circuit boards (PCB) within the BM itself in a modular configuration consisting of a backplane/routing board and a series of connected daughter-cards for Command and Data Handling (C&DH), Electrical Power (EPS), Communications, and Attitude Determination (ADS) subsystems. The outer body panels were designed to accommodate both the solar arrays as well as components of the Attitude Control Subsystem (ACS).

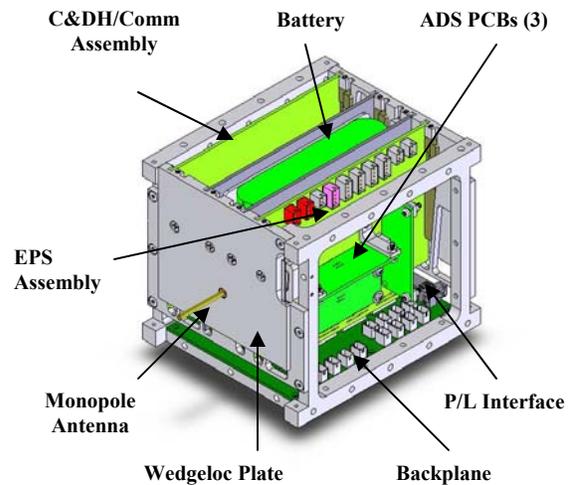


Figure 6: The *.Sat Bus Module Internals

Structural/Mechanical Subsystem

Physically, the structure is organized around the standard CubeSat module size of 100 mm cubed, plus a 6.5 mm long interstitial spacing at both ends of each module. The total length of a triple-unit S/V is 339 mm. The *.Sat design does not include any deployable elements to preserve mechanical reliability and simplicity during on-orbit deployment and operations.

The primary structural frames are shown in Figure 7. 7075 Aluminum square tube-stock was machined in an Electrical Discharge Machining (EDM) process into a series of CubeSat-sized frames treated in an alodine process (chromate conversion) to maintain electrostatic conductivity. A pattern of orthogonal cross-drilled bolted interfaces, using M2.5x0.45 Helicoil inserts, between the frames and body panels allows shear transfer to carry loads during the launch phase.

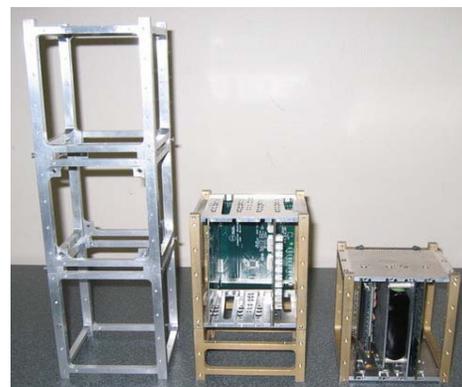


Figure 7: *.Sat Structural Prototypes
(Left frame is 1.0/1.0/1.0 configuration, bare aluminum;
Center frame is 0.5/1.0 configuration, alodined finish;
Right is BM prototype with full card-stack and battery)

The internal electronic systems are mounted onto a backplane mounting board and individual components can be removed as necessary for testing and servicing. All major interfaces into and out of the BM (payload interfaces and umbilical) are located physically on this backplane board in addition to the connections between the EPS, Communication, and C&DH subsystems.

The mechanical interface between the BM and the PM(s) utilized a system of interlocking “feet” dimensioned to conform to the 6.5mm sizing of the interstitial spacers outlined in the CubeSat Specification. A machined and threaded M2.5 hexagonal set-screw provides the initial alignment of the BM and PM(s) prior to final integration with the body panels, which are the primary load-bearing structure for the S/V.

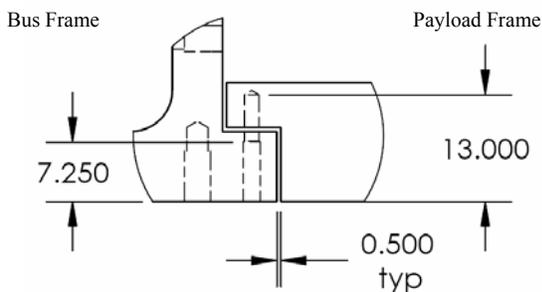


Figure 8: BM to PM Mechanical Interface

(Note “Foot” on Bus Frame is 6.5mm long, as per the CubeSat Spec, with the allowance for 0.5mm float between the modules to facilitate final alignment to the body panel bolt pattern)

The internal electronic PCBs are mounted onto the structure via a set of machined end plates utilizing the wedge-lock mechanism often seen in larger aircraft and spacecraft applications. In addition to providing a robust mechanical interface, the wedge-lock allows for direct thermal conduction between the structure and the PCBs.

Bus Electronics

The internal *.Sat bus electronics components are laid out on six custom PCBs connected with standard JST/JED-type board-to-board connectors. The bulk of the system functionalities reside on two boards; an EPS board for power conditioning and fault mitigation circuitry, and a C&DH board for the main system processor and the associated data acquisition and storage circuitry. As shown in Figure 6 and Figure 9, these two boards slide vertically into a signal routing, or backplane to fit the CubeSat form factor. This method of packaging was selected over a stacked configuration for ease of board replacement or modification.

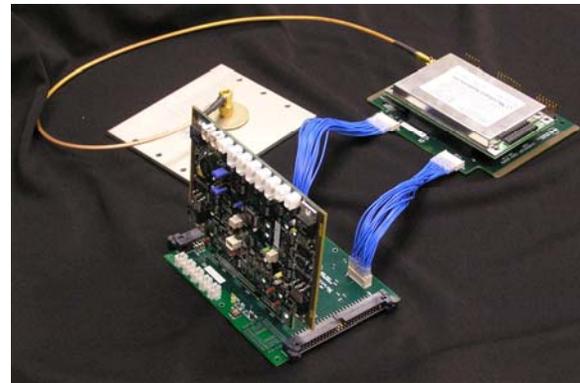


Figure 9: Electronic PCBs in Bench-Testing Configuration

(The backplane and EPS board are in the foreground and the C&DH board with attached transceiver are in the upper right. The series of blue wires are extender cables that duplicate the board-to-board connections between the C&DH and backplane, allowing easy access to all surface mount circuitry. In flight configuration the C&DH board with attached transceiver sits parallel to EPS board.)

Other components, including the communications transceiver, and the Attitude and Determination System (ADS) are mounted within the core PCB configuration. The transceiver is mounted on the C&DH board, as shown in Figure 10 with its serial interface and control signals exposed to the processor. The ADS, which is housed on a series of three small boards, is attached to the EPS board with board-to-board connectors. The ADS boards are mounted orthogonally to each other enabling the measurement of linear acceleration and spin rate in each of the spacecraft’s three axes.

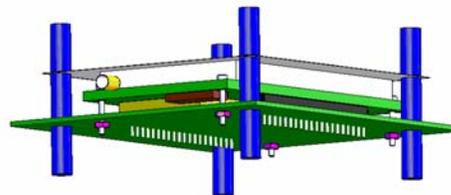


Figure 10: Communication transceiver / C&DH board Configuration

Command & Data Handling (C&DH) Subsystem

The C&DH board is designed to perform several spacecraft functions including command validation, command processing, health and status gathering, and storage of collected data. Additionally, since the *.Sat bus supports up to two attached payload modules, the C&DH is also able to control and monitor those payloads, as well as provide coarse temperature control for them.

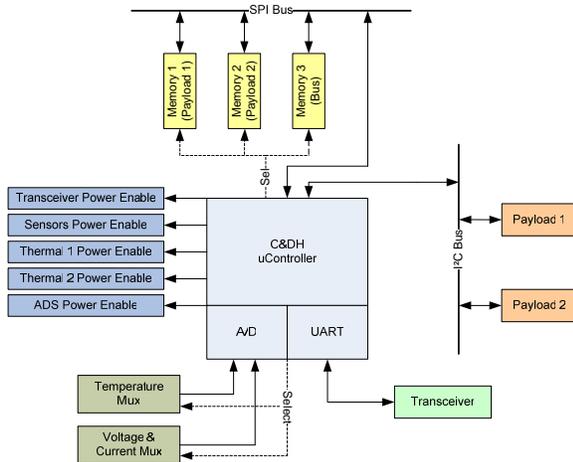


Figure 11: C&DH Block Diagram

Microcontroller

To implement these functions adequately, the C&DH board contains a microcontroller unit (MCU) capable of performing data and logic operations. Because of limited power, the CubeSat as a platform for electronics is more akin to that of today's PDAs or digital cameras than to that of today's personal computers. Hence, to limit C&DH power consumption a low-power version of the Microchip PIC18 was selected for the *.Sat bus. Apart from being a low-power device itself, because it aggregates a variety of devices into one package, the PIC18 further reduces consumption simply by reducing the overall chip count. The PIC18 has on-chip flash memory to store program code, on-chip RAM, and an integrated A/D converter for data acquisition. These features along with the several integrated communication devices make the PIC18 a single device solution for many of the software and data handling functions. The processor's key information is listed in Table 1.

Table 1: C&DH MCU Specifications

Processor	Microchip PIC18F6720LF
Instruction Size	8 bit
Power @ 3.3 V	0.1375 W
Clock Speed/MIPS ⁽¹⁾	16 Mhz/4
Data Memory (RAM)	3840 Bytes
Program Memory (Flash)	128 kBytes
USARTs ⁽²⁾	2
A/D Converter	10 bit resolution

- (1) MIPS = Millions of Instructions Per Second,
 (2) USART = Universal Synchronous Asynchronous Receiver Transmitter

Data Protocol

The industry-standard Inter-Integrated Circuit (I²C) protocol was chosen as the data-transport protocol to move data between the bus processor and any attached payloads. At the hardware level, I²C specifies a two-

wire bus with one line for the clock signal and the other for serial data transmission. The standard clock rate is 100 kHz, but a "high-speed" mode of 400 kHz is also defined. Aside from the benefit of I²C's small bus size, which reduces pin count, the protocol is also attractive due to its provision for supporting a network of interconnected devices. Similar to the Internet Protocol (IP), I²C inserts device address information into the data stream at the start of every communication. Seven and ten bit addressing is available which allows for networks of 128 or 1024 devices respectively. The *.Sat bus only requires support for three devices (the bus itself, plus two payloads), but the selection of I²C does allow for more attached devices without extensive redesign (from a data perspective). Another driver in the choice of I²C was the large number of devices, especially microcontroller/microprocessors that have on-board I²C modules. Because of this, payload designers will have a large selection of compatible components to choose from to perform bus communication.

The same I²C bus connects to the PIC18's I²C module and the two payload connectors. While the protocol does allow for a multiple master mode, where different devices can negotiate for control of the bus, the *.Sat standard operating procedure specifies that the bus processor is always the master, and will initiate all communications. As represented in the diagram below each payload module will be assigned a fixed address to be used at all times.

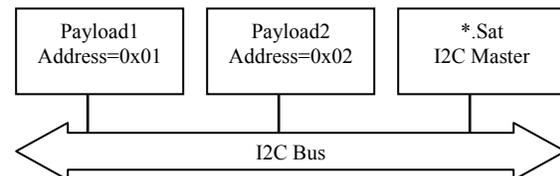


Figure 12: I²C Bus Configuration
 (The I²C bus has up to three attached devices: the *.Sat processor, which serves as the I²C master, and two payload devices, which are I²C slaves. I²C addresses are shown.)

Sub-system Control

As the "brains" of the system, the C&DH board is able to enable and disable all major sub-circuits on the *.Sat bus via digital logic lines contained in the CD&H/EPS interface. By sending binary outputs to the EPS, the C&DH has the ability to control the switching regulators that power the communications transceiver, the attitude determination system, and the various sensor networks collecting bus telemetry. The C&DH system also has control over MOSFET networks supplying unregulated power to both the payloads and to the safe-mode payload heaters. Having this ability gives the C&DH fine-grained control over the total

system power consumption. Subsystems can be powered on and off in two ways, in response to power anomalies (latch-up conditions) or can be duty cycled to maintain a software prioritized certain power profile.

Data Acquisition

The C&DH is also responsible for acquiring data from all of the onboard sensors and assessing the health of the spacecraft based on their output. Two main classes of sensors are built into the *.Sat bus for health monitoring purposes. Analog Devices AD590 proportional-to-absolute-temperature (PTAT) devices are used to monitor satellite temperatures, and the PIC18's analog-to-digital converter module coupled with Maxim MAX4372H current sensors are used to measure voltage and current on critical current pathways. All data is digitized using the MCU 10-bit analog-to-digital converter module and remains in engineering units (A/D output code, 0-1024) when stored, and subsequently transmitted to the ground. The responsibility of interpreting the data code as a floating-point value is levied on the ground software.

A total of thirty-two PTAT sensors are placed throughout the S/V in contact with thermally critical items in every subsystem of the satellite including the communications transceiver, system battery, and solar panels and payloads. Temperature data assists in determining the health of these components and which subsystems can be safely powered. In measuring system temperatures, the need to perform simultaneous or high-rate (>1000 samples per second) sampling of these sensors is obviated by the slow changing nature of the signal. Because of this, all thirty-two PTAT outputs are connected to a 32-channel multiplexer, whose output is then sent to a single A/D converter on the PIC18. This design gives the C&DH the ability to acquire the temperature data in a random access fashion, by selecting different multiplexer channels, but still conserves microcontroller pins. Temperatures are sampled at a nominal rate of 1/60 Hz (1 sample per minute).

The C&DH also monitors a variety of system voltages and currents. Solar panel and battery pack outputs are acquired continuously, as are the current draws of the payloads, payload heaters and communication system. By gathering S/V state information, the C&DH is able to switch to the appropriate power consumption mode, and preempt a variety of power crises. For example, in a situation where the payload experiment is running, but battery power is low and solar panel output is at a minimum, the C&DH will enter a "low-power" mode where the system exchanges payload power in favor of communication power. While a single communication pass may be lost, the experiment remains viable. In a

much more serious situation the C&DH is able to reduce power by shutting down sub-circuits in a prioritized manner in an effort to maintain critical satellite functionality.

Similar to the PTAT outputs, the voltage and current signals pass through a 32-channel multiplexer before reaching a second A/D input on the PIC18. While voltages and currents can change much more rapidly than temperatures, recognizing anomalies such as latch-ups can be achieved by comparing the current values to known critical minima or maxima. The high sampling rate needed to capture the characteristic rise or fall is not needed, and so the multiplexer design is justified. Therefore, current on critical circuits will still be sampled at a much higher rate than the temperatures, nominally 10 Hz.

Electrical Power Subsystem (EPS)

The Electrical Power System is the power generation, storage, and delivery system for the satellite bus. The power scheme implemented on *.Sat is quite similar to that of a laptop computer, where the battery of the laptop is analogous to the Lithium Ion battery on the *.Sat bus, and the DC power supply is analogous to the solar cells. Just as a laptop can run unplugged from the wall outlet, the satellite can run in the eclipse phase of the orbit without solar cell power generation. Likewise, just as a laptop can be operated and charged simultaneously when plugged into the wall, during non-eclipse phases of the orbit when the solar panels are illuminated, the satellite is powered directly from the solar cells, with excess power used for battery charging.

The EPS provides on-orbit power for the bus and up to two payloads. The bus generates power with Spectrolab UTJ solar cells mounted on 4 solar panels (unregulated power). The EPS regulates power for the C&DH, the bus sensors, and the communications system. A set of secondary Lithium-Ion batteries gives the system the ability to deliver stored energy when the energy from the solar panels alone is inadequate. The EPS also contains several onboard circuits to help protect the satellite from failing due to LEO power anomalies. Figure 13 shows a simplified representation of the EPS power components and their connections.

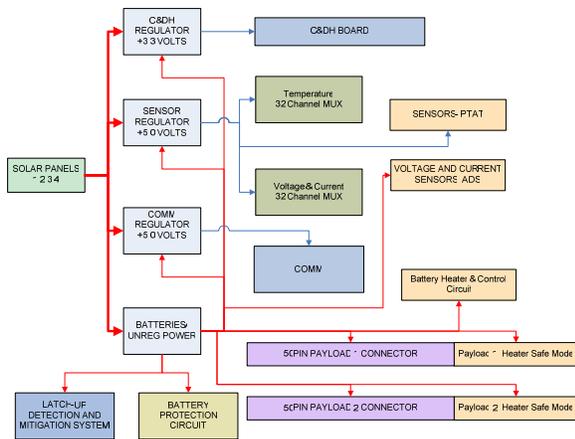


Figure 13: EPS Block Diagram

Power Generation

Solar cells were chosen for on-orbit power generation, as alternate power sources such as radioisotope-thermoelectric generators (RTGs) and primary (non-rechargeable) batteries were not suitable for the mission requirements. The limiting factors with this option of power generation are solar cell surface area availability, solar cell degradation, and orbital attitude parameters regarding satellite inclination, sun illumination angle, and eclipse duration.

The solar cells selected for *.Sat are 28.3% efficient Ultra Triple Junction (UTJ) Solar Cells from SpectroLab. Each of the four main satellite panels has two strings of solar cells connected in parallel, with each string containing four cells wired in series, yielding a total cell area of 224 cm² on one satellite face (each solar cell is 28cm²). Each solar cell string on the panel is individually Schotky diode protected, and is expected to generate a V-loaded output of 9.24 Volts when in direct sunlight. After passing through the diode protection this voltage is minimized to 8.9 Volts. Each string of solar cells will have a current production capability of 0.461 A or 0.922A per panel in direct orthogonal sun illumination. Power production is a function of the sun-angle on the solar cells. The orbit and attitude of the satellite are critical to determine the exact power generation/consumption profile for a specific mission. Power production is described by a cosine function of the angle of incidence from the orthogonal illumination point. In the orthogonal orientation to the sun a solar panel should produce 8.206 watts.

Batteries

Based on a detailed trade study that evaluated charge and discharge capabilities, thermal characteristics and vacuum qualification, Panasonic CGR18650C Lithium

Ion rechargeable batteries were selected as secondary batteries. The nominal output voltage of these batteries is 3.6V, with a beginning of life (BOL) charge capacity of 2150mAh. *.Sat incorporates a battery pack wired with two banks in parallel of two-cells each as shown in Figure 14. The composite battery pack is configured for an output voltage of 7.2 Volts nominal and a standard charge capacity of 4300mAh. An external heater was added to the battery pack to assure that its temperature does not fall below 0°C during charging and -20°C during discharge — the point at which terminal plating starts to occur. Any amount of plating increases terminal resistance, and if allowed to progress can ultimately lead to a complete open circuit, effectively disconnecting the battery from the EPS and rendering it useless.

Electronics Power Draw

The spacecraft bus Subsystem power draw varies with the timing protocol of a satellite subsystem “ON” states. The *.Sat subsystem power categories and calculated power draws are:

Component	Power (W)
Comm power standby mode	1.438
Comm power transmit mode (20 min every 24hrs)	4.375
Thermal power	0.670
ADS power	0.520
EPS, C&DH, Backplane	0.252

Table 2: Subsystem power draws

(The EPS, C&DH, and Backplane power consumption of 252mW are based on continuous operation. The other subsystems are duty-cycled as needed to support the payload(s).)

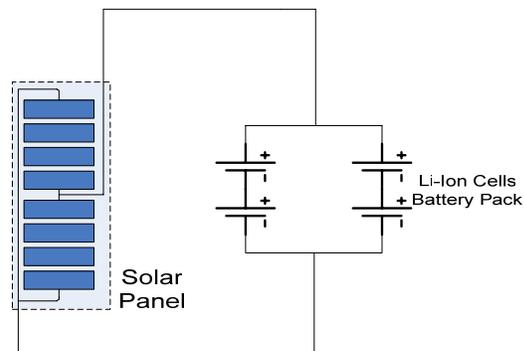


Figure 14: Solar panel and battery arrangement

(Each panel contains two strings of four solar cells. The battery pack contains two redundant sets of two cells wired in series.)

Latch-up Protection

The satellite must have the ability to detect and transition to an electrical “safe mode” if an on-orbit anomaly should cause a system latch-up. The EPS is

the primary defense against these conditions with the C&DH providing a redundant layer of protection.

Since *.Sat is built with Commercial Off-the-Shelf (COTS) components, the satellite is more vulnerable to a radiation event causing circuitry to latch, possibly causing a catastrophic failure or permanent loss of satellite function. Few parts were available in radiation-hardened packaging to protect from radiation events such as solar flare, coronal mass ejections, and random cosmic radiation. Hence, voltage and current sensing circuitry is placed on the front end of all power delivery lines and monitored by both EPS and C&DH.

During operation, if an abnormal current draw is detected, the EPS and C&DH subsystems have the ability to shutdown power to any specific line. Critical regulated power pathways for Comm, sensors, and C&DH have redundant, independent latch-up mitigation capability. The C&DH shutdown mechanism is a MOSFET-enabled network controlled by the Flight Software (FSW). The FSW continually evaluates and monitors safe power draw levels and will shutdown a “problem circuit” if unsafe levels are detected. After a complete power-cycle, the FSW will assess the “problem circuit” and will bring it back online if it detects a safe start up voltage and current draw.

The regulated power circuits also have a passive shutdown (faster response time than FSW) implemented to achieve a redundant shutdown function if power shutdown cannot be accomplished within a pre-determined critical time period. The passive network has the capability of checking the health of latched circuitry and will not allow the power to flow to the node again until normal power draw is detected.

Battery Protection

During the eclipse portions of the orbit, Comm power is supplied only by the onboard battery pack. Failure or degradation of this battery could jeopardize the ability to maintain power to the Comm Subsystem during all phases of the orbit. Therefore, *.Sat is required to have protection circuitry to prevent a catastrophic failure of the Battery due to over-charging or excessive discharge.

The battery is charged directly from the solar cell output power bus. The battery protection circuitry based on a TI UCC3911 2-Cell Lithium-Ion Battery Protection IC insures that the battery is maintained between a maximum of 8.4 volts and a minimum of 6.0 volts. The device’s primary function is to protect Li-Ion cells in a two-cell battery pack from being either overcharged (over-voltage) or over-discharged (under-voltage). It employs a precision band gap voltage reference that is used to detect when either cell is

approaching an over-voltage or under-voltage state. When on-board logic detects either condition, the series FET switch opens to protect the cells.

A negative feedback loop controls the FET switch when the battery pack is in either the over-voltage or under-voltage state. In the over-voltage state the action of the feedback loop is to allow only discharge current to pass through the FET switch. In the under-voltage state, only charging current is allowed to flow. The operational amplifier that drives the loop is powered only when in one of these two states. In the under-voltage state the chip enters sleep mode until it senses that the pack is being charged.

As a precaution to insure that *.Sat batteries can recover from an excessive discharge condition, the EPS also has the capability to shutdown all systems and recover nominal battery capacity if the battery voltage falls to 5.6 Volts or below. The satellite solar cells can then recover the batteries to full charge or a minimum voltage of 7.7 volts before allowing satellite to power-up again. This prevents power-up current spikes from driving the circuit into a repeated cycle of dipping below the 5.6V threshold.

Grounding

The bus has several separate ground paths from various subsystems that join together at a common point. The EPS board has a single ground plane layer connected to the chassis and to the negative terminal of the satellite batteries. All grounds in the system join at this common point. Grounding the system in this way limits power system noise caused by digital and analog ground crosstalk.

Both payload modules have identical grounding schemes. Each sends four separate analog grounding lines to the common point. Both PM 1 and PM 2 have two separated digital grounding traces going to the common point ground.

The common point ground plane is attached to a turret on the EPS board, which is then attached directly to the satellite chassis with a low gauge wire. This chassis ground needs to be electrically attached to the alodined chassis of each payload cube to prevent any possibility of spacecraft charging and arcing. Also, each payload must have a way of making sure they are attached to the common point ground electrically. This is a satellite restriction/standard the payload must strictly follow for successful bus operation.

Standard Payload Interface

*.Sat provides up to two identical interfaces that can support independent Payloads simultaneously. This standard interface provides consistent mechanical, electrical and software services to both payloads regardless of configuration in the overall S/V.

Hardware Interface

The bus provides power, data communication, digital, and analog signal lines through a 50-pin connector. The payload connector is a 50 Pin Samtec EHT-125-01-S-D-RA-01 connector located on the *.Sat backplane board. Both *.Sat payload connectors have identical pin assignments, with each pin rated at 1 A.

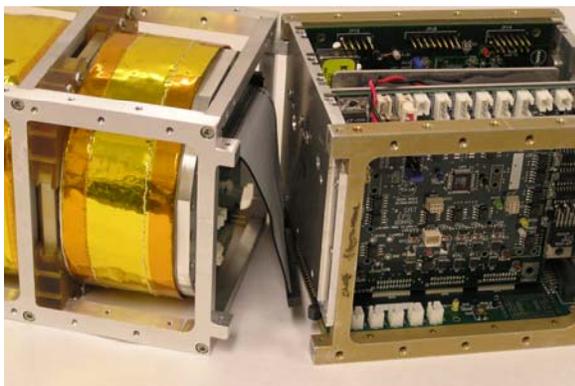


Figure 15: BM to PM Interface

The 50 PIN interface connector has four unregulated payload power lines. Additionally, the connector supports four isolated analog grounds and two isolated digital grounds that connect to the EPS common point ground, which is electrically connected to a single point on the chassis. Twenty of the connector's pins support 10 PTAT sensors used to monitor the temperature of the payload structure and solar cells. The connector also has one binary enable line to control a payload heater in safe mode, and a single binary enable to reset the payload microprocessor if needed. An isolated safe mode payload heater power line is also included, which will only be needed to maintain temperature if the four unregulated power lines are turned off due to a power crisis. The standard I²C clock and data lines along with an I²C reset line, which is used by the bus to indicate to all payloads that an I²C bus reset/re-synchronization is needed, are included and comprise the entire payload data communication interface. The remaining 12 of the 50 pins are unpopulated and available for future expansion needs.

Safe-Mode Heater Control

A critical need for any biological or temperature critical payload is a continuous, regulated thermal environment

while on orbit. While it is recognized that most payloads with such a characteristic will have their own payload-driven temperature control system, it is also recognized that overall power needs of the S/V may necessitate the powering down of a payload at some point during flight. However, in order to preserve a temperature-critical experiment the *.Sat bus is designed to provide heater power to maintain a “keep-alive” temperature at all times. This is achieved by independent power circuits for a payload’s electrical system and heating system, with the C&DH having enable/disable control over both. In this scheme, if the payload power is shutdown, heater power can be maintained even in any S/V safe mode. The payload heater power will only be shutdown in an emergency condition such as a battery or solar panel failure.

Software Interface

A software interface is also provided to standardize communication between the *.Sat software and payload systems. Beyond the choice of the I²C protocol for transport of data, an “application” level protocol was designed to govern the format and content of such communication. While I²C defines what happens at the bit and byte levels, the *.Sat application protocol gives specific meaning to the bytes placed on the I²C bus, and to the response of the devices attached to it. The protocol was modeled after the classic C-style function invocation, with the *.Sat bus acting as the caller and the payload providing the function logic. In this model the bus sends a function/command code to the payload along with the required set of parameters for that function. The payload will process the command and respond with a code and any required return value. The protocol does not define how the payload should approach command processing, only the formats of the call and of the response that it must adhere to. The sequence diagram below illustrates the protocol.

Each function (or command) is identified by a command code. The 256 possible functions are split into two major groups; Standard Interface Functions and User Functions.

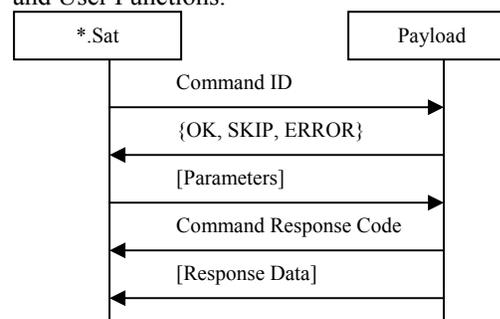


Figure 16: System Sequence Diagram
(Showing order of data flow between bus and payload during a command invocation.)

The Standard Interface is a set of 16 functions, to which each attached payload must be able to respond. Functions such as those for retrieving data packets, resetting the payload, and other functions common to most payloads are included in the Standard Interface. This “standard” section of the application protocol covers three of the four situations in which data exchange between the bus and payload is necessary: experimental data transfer, payload monitoring and telemetry services. The remaining 240 slots for User Functions satisfy the fourth: commanding of the payload by the payload operator. These codes are available for the payload designer to customize the interface for the mission at hand. For example, functions for enabling or disabling certain specialized circuits or subsystems within the payload module would be User Functions, since they are specific to that particular payload. A truncated command list from the GeneSat payload is given below to illustrate the concept of Standard versus User commands.

Table 3: Subset of GS-1 Command List.

Command Name	Type
Reset	Standard Interface
Ping	Standard Interface
GetDataPacket	Standard Interface
DownlinkVerify	Standard Interface
EnableValves	User
EnableFeeding	User

This clear separation between general and specific payload functionalities is key to enhancing *.Sat’s reusability, and in putting forth a “standard” that other payloads can adhere to. During the *.Sat and GeneSat development cycle this clearly defined standard interface allowed parallel development on the two software systems, with little worry about integration.

Payload Configuration

As part of the *.Sat command protocol a simple “Plug-n-Play” interface was established for attached payloads. In order to be compliant with the interface, a payload must provide a set of its configuration parameters to the bus when the ‘GetConfiguration’ interface command is invoked. The payload may choose to store a static version of its configuration in a file, or it may serve a dynamic version based on its current state. The bus issues the ‘GetConfiguration’ command upon power-up or when instructed by the ground station. The content of a payload’s configuration provides the bus with key information on how to interact with it. Information such as optimal I²C clock speed and payload packet sizes is included, as well as declarations of which bus telemetry the payload would like to receive. Instructions on how a payload would like its

temperature controlled (if applicable) is also available to the bus via this mechanism.

Payload Commanding

The *.Sat bus provides three methods of issuing commands to a payload. The first method, real-time commanding, can be performed only during a communication pass. In this method commands are sent from the ground software, received by the *.Sat bus command processor, and then routed to the payload. Every *.Sat command contains a “target” identifier, indicating to the processor the device for which the command is issued. Payload responses are then packetized and relayed immediately back to the ground.

If commands need to be executed at a specific time regardless of whether the satellite is in ground communication or not, passive commanding can be used. To activate passive commanding, commands are uploaded from the ground in the same fashion as in real-time commanding except the ‘real-time’ flag in the 2nd byte is cleared. When the bus command processor encounters this it automatically interprets the first four bytes of the parameter space as the mission time value at which to execute the command. The command is then stored in a priority queue based on its execution time. Command time resolution is guaranteed to 2 seconds. The command queue is redundantly stored in flash memory to protect the execution of critical events from unexpected system resets. That is, power cycling the system will not erase the command queue. Instead several ground to bus commands are available to delete or edit entries in the queue in the event that commands need to be changed. Of course these commands are only effective before the stored command is executed. Unlike real-time commanding, any output from stored commands is packetized then stored in flash memory (on the bus). The output is then available for downlink on subsequent communication passes and does not require further interaction with the payload.

Table 4: *.Sat Command Format
(Target field indicates which device to route command to. Type field distinguishes between real-time and stored commands.)

Field	Size (bits)	Description
CommandID	8	Identifier
Type	1	Real-time or Stored
Source	3	Ground asset asserting command
Target	4	Target device (bus, payload1, payload2, etc.)
Byte Count (n)	8	Number of bytes in command
Parameters	8*n	Command parameters
Checksum	8	8-bit checksum byte for error detection
Delimiter	16	Separates commands within stream

The *.Sat bus also supports a third method of commanding called ‘periodic’, in which the bus calls certain Standard Interface functions at a given rate to support the payload monitoring and telemetry dissemination functions. The rate at which periodic functions are called is fully configurable via ground control and may be modified as needed. Depending on the payload and the payload operator’s needs, this type of commanding may or may not be desired, and a payload may always choose to ignore commands that are issued in this fashion. The specific functions involved in periodic commanding are mentioned in subsequent sections.

Payload Experimental Data

It would be impossible to determine or predict the format in which a specific payload will store its experiment data. Beyond that, some of the goals of the *.Sat bus are to reduce coupling between the bus and payload development as much as possible, and to provide common services, such as ground communication. Therefore, no restriction is placed on the format or size of the data that the payload collects, and in general the bus can be considered simply a buffer between the ground and the payload. In this scheme the payload then collects its data in the format that is most convenient, and the ‘GetDataPacket’ Standard Interface function is used to copy a section of those data into a memory block on the bus for downlink buffering.

As indicated in the command list, the ‘GetDataPacket’ command does require an integer parameter to identify which section of memory to retrieve. The payload designer is responsible for creating a scheme for mapping the numbers into data blocks if the ‘GetDataPacket’ function is to be used. No restriction is placed on this mapping. Other than when called, the payload responds with a packet of data the size of which is specified in the configuration file.

As described above, the data packets retrieved from the payload are stored in the bus’s buffer memory until they are downlinked. Following the successful downlink of the packet(s), the ground software will send verification of the packet’s arrival back to the bus software. When this verification arrives from the ground, the bus will invoke the ‘DownlinkVerify’ command on the payload with the packet identifier as a parameter. The invocation of this command indicates to the payload that the given packet has indeed been received and can be marked as such. Depending on memory constraints a payload may be able to use this verification as a flag to free sections of storage for reuse. It is of note that data verification is provided for convenience only, and the protocol doesn’t enforce the

deletion behavior. In some cases the payload may choose to ignore ‘DownlinkVerify’ notifications altogether.

Payload Monitoring and Telemetry Dissemination

A certain level of payload health monitoring is provided by the *.Sat bus interface. The bus constantly monitors the payload’s voltage and current draw and uses these measurements for latch-up detection and recovery. Additionally, the payload’s temperature is monitored. However, actions taken based on this measurement are dependent on the current state of the spacecraft and payload configuration. For example, the GeneSat payload has a critical “keep-alive” temperature below which the biological payload would be compromised. Therefore, temperature control must be implemented at all times, whether or not the payload, with its own temperature controller, is powered. Thus, when the payload is not powered, the *.Sat bus uses the payload’s temperature measurement to control its internal heater. After the payload is powered, it takes control of the heater and the bus assumes the role of passive monitor.

In addition to monitoring “external” payload health, payload designers may choose to implement the ‘GetStatusPacket’ function that will be called on a periodic basis. The bus will then insert the results of these calls into the regular spacecraft telemetry stream. The intent of this function is to provide payload operators the facility to monitor critical payload parameters and track trends. Of course, there may be cases in which payload “status” data is very similar to experiment data. However, the *.Sat bus does not guarantee that there will be an opportunity to downlink experiment data every pass. For example, on some communication passes Payload 1 may take priority over Payload 2, and vice-versa on others. By including payload “status” with spacecraft status, a payload’s health can be monitored even when its experimental data does not make it to the ground.

In addition to monitoring, a given payload may require information about the overall spacecraft health or state (telemetry) in order to plan or execute an experiment. The *.Sat bus accommodates this need by providing access the system telemetry via a series of ‘Update’ commands. Because the bus processor is permanently in the master role, telemetry is “pushed” to the payloads by the bus, as opposed to a query-type system in which payloads could interrogate for data that they need. Real time clock values are pushed with the ‘UpdateTime’ command, system temperature values (32 channels max) are pushed with the ‘UpdateTemperature’ command, and voltage and current values (32 channels max) are pushed with the ‘UpdateSensor’ commands.

All three commands are called in the ‘periodic’ fashion described above.

Because receiving telemetry puts more demand on the payload processor, reducing the number of telemetry channels received to the minimum necessary is important. Payloads can provide the channels they need via the ‘SensorUpdate’ and ‘TemperatureUpdate’ configuration parameters. Based on these parameters the bus will only send the selected data to the payloads during ‘Update’ function calls. If a payload doesn’t require bus telemetry, indicated by <0> values for ‘SensorUpdate’ and ‘TemperatureUpdate’ in its configuration file, then the bus simply will not issue the periodic ‘Update’ commands and no effect on system performance will be incurred.

Attitude Determination & Control Subsystem (ADCS)

In small satellite systems typical of CubeSats, the pointing requirements are usually driven by the simple need to maintain a coarse nadir-pointing orientation for communication, as shown in Figure 17.

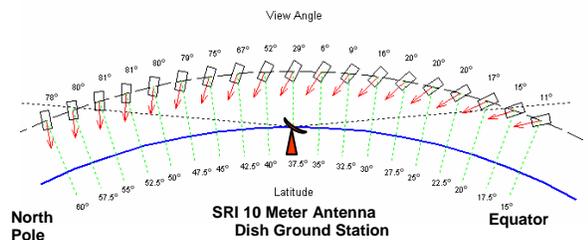


Figure 17: Orientation of *.Sat Over Stanford Ground Station

To provide the milli-G environment needed for GS-1 biological payload, the design requirements for S/V attitude rates on-orbit are to result in accelerations not greater than 10^{-3} G.

The *.Sat Bus Attitude Determination Subsystem (ADS) together with other *.Sat Bus subsystems is designed to provide sufficient telemetry data to allow ground personnel to establish the S/V attitude with respect to the sun vector.

Attitude Control Subsystem (ACS)

In the LEO environment in which most, if not all, CubeSats fly, the Earth’s magnetic field is sufficiently strong to provide a basis for a simple, passive Attitude Control System (ACS). As depicted in Figure 18, such a system was designed for use in the *.Sat Bus. It consists of a set permanent magnets and hysteresis rods placed on the inner faces of the satellite body panels.

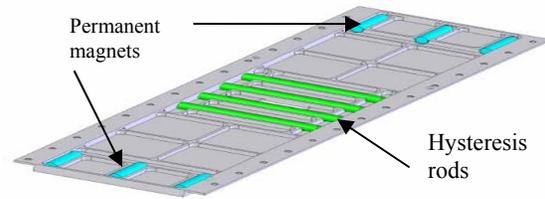


Figure 18: Magnet and Hysteresis Rod Placement

The main choices for magnets are Alnico-5, which has medium strength but high-temperature stability, and Neodymium Iron Boron (NdFeB), which has high strength but temperature stability only up to 150°C. NdFeB magnets were chosen, since the present S/V temperature is not expected to reach that high a level.

Three NdFeB Grade N40 permanent magnets (6.35mm diameter, 25mm long) are located at each end of the body panels. The South Pole of the magnets points to the +Z axis, so that the monopole antenna always points toward the Earth’s magnetic North Pole.

For the hysteresis rods, high-permeability soft iron material was selected because it has the highest energy loss per magnetization cycle. The rods (6.35mm in diameter and 71mm in length) are placed in groups of four on the inside of each body panel.

Software simulations based on CubeSim [5] and CubePowerSim [8] were used to predict the on-orbit dynamics of the spacecraft. Typical results are presented in Figure 19 below.

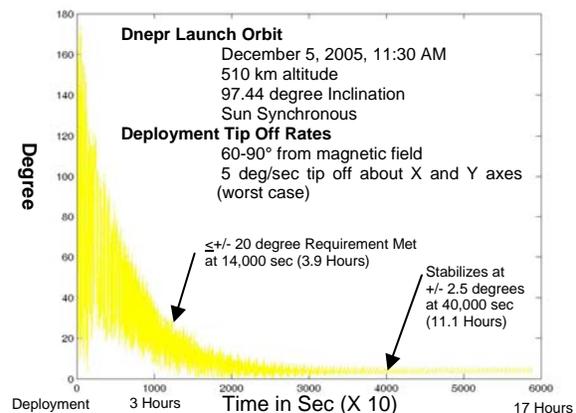


Figure 19: Stabilization Time from ADCS Simulation

Attitude Determination System (ADS)

In order to estimate the overall vehicle micro-G level, the *.Sat Bus uses a system of three-axis rate-gyros (Analog Devices, ADXRS150 MEMS) and three-axis accelerometers (Silicon Design, Inc. Model 1221, range ± 2 G, noise floor 2 μ -G RMS) in conjunction with a

crude sun-sensor based on the variation in current output from the solar panels.

Thermal Control Subsystem (TCS)

The biological payload survival temperatures and desired experimental conditions drive the requirements for the overall TCS. As a subsystem that depends heavily upon analysis and design of the entire integrated S/V, both Bus and Payload, the majority of design and analysis was conducted with models reflecting the entire S/V. During the experimental phase of the mission, the TCS is required to maintain the Payload temperature in a range from $+35\pm 0.5^\circ\text{C}$.

Analyses predict that the satellite wall temperatures are cold biased and range from $+20^\circ\text{C}$ to -10°C over a typical orbit. Because of the biology temperature requirement, the fluidics card is thermally isolated from the satellite to protect it from the extreme temperature excursions of the satellite. There are four key heat transfer mechanisms:

- Conduction from the structure to the payload enclosure
- Conduction from the payload enclosure to the fluidics card
- Radiation exchange between the payload enclosure surface and the solar panel substrates
- Radiation from the enclosure surface to the fluidics card.

Avoiding metal-to-metal contact can decrease conduction. Therefore, Ultem™, a structural plastic, with minimal contact area isolates the payload inside the structure. The optical/fluidics assembly has Ultem™ rails that slide into grooves in the pressure vessel. This reduces fluidics card heat loss to the enclosure. Temperature is not regulated independently at each well location; therefore it is imperative for all wells to be at approximately the same temperature to conduct a valid experiment. Available power is used to heat the fluidics card directly. Heaters are mounted to Aluminum plates that act as thermal dissipaters to minimize the thermal gradients over the card's surface.

Low-emissivity coatings are used to decrease heat transfer by radiation. The satellite panel surfaces are alodine-treated for electrostatic conductivity purposes. Multi-layered insulation (MLI) will be wrapped around the payload enclosure to decrease radiation from the enclosure to the walls. Also, the payload is nickel plated with $\epsilon=0.07$ so minimal heat is radiated from the card to the enclosure.

Thermal Analysis

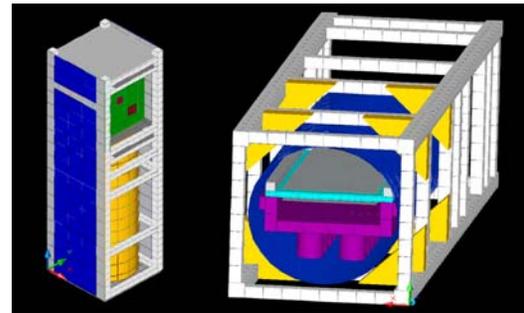


Figure 20: Thermal FDM Models
(The bus is located on the top cube. The cylinder represents the payload enclosure.)

A finite-difference method (FDM) analysis model of the integrated S/V was created using Thermal Desktop/SINDA with RadCAD.

The model has a total of 2400 nodes. Body panels, PCBs, payload enclosure, and heater plates were modeled as surfaces with unit thickness because of their aspect ratios. Solid bricks were used to model the fluidics plate and optics modules to observe cross sectional temperature gradients. Contact elements represented interfaces between mating surfaces. Most nodes are diffusion type. Boundary conditions for on-orbit simulations are defined by defining orbital parameters and specifying the number of positions per orbit to be analyzed. 14 positions were used in a typical analysis. Materials are assumed to have temperature-independent, isotropic properties. To account for power output by the solar cells, the solar absorptivity of the solar panels was calculated to be (α - % average solar cell efficiency).

Temperature Predictions

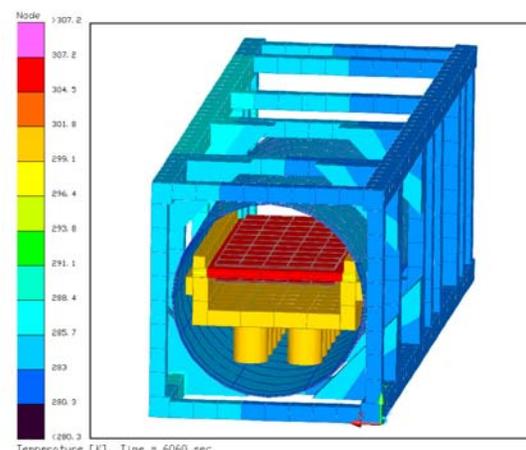


Figure 21: Post-Processed Thermal Model
(Body panels not shown, temperature scale in Kelvin)

Analysis on the proposed design shows that uniform heating can be achieved across the surface of the card. The uniform color on the fluidics card surface in Figure 21 indicates a uniform surface temperature. The results indicate a 2W peak heater power would provide adequate authority during the active control phase. Satellite face temperatures over a typical orbit are shown in Figure 22. During active operations, when the solar panels are outputting energy to bus and payload systems instead of heating the solar panels, satellite face temperatures range from +20°C to -11°C. The extremes occur during the portions of the orbit when the S/V passes over the Earth’s magnetic poles, where the S/V exhibits its highest angular rates. The other satellite faces cycle from +5°C to -11°C.

The cold-biased satellite allows for a heat-only temperature control scheme. Figure 22 shows the card temperature at 35±0.5°C suggesting sufficient isolation from the satellite and temperature control.

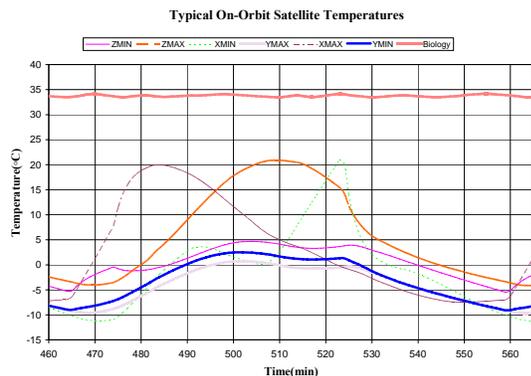


Figure 22: Typical On-Orbit Satellite Temperatures

Communications and Ground Station Subsystems

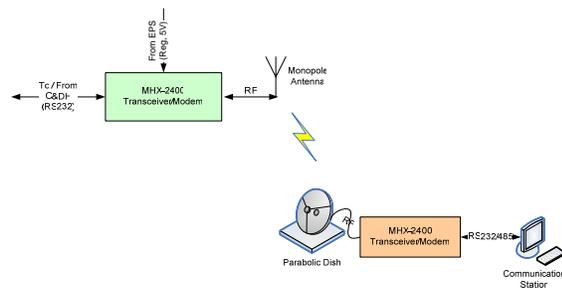


Figure 23: Communication System Block Diagram

The main function of the communications subsystem and the ground segment is to establish a link between the operator and the satellite in order to transfer information. The on-board communications subsystem must deliver spacecraft telemetry and science data to the operator on the ground within one year from measurement at a rate of at least 1 megabyte (MB) per

day (subject to orbital parameters) as well as receive 14.7 kilobytes (KB) of data from the ground station per day (subject to orbital parameters), when requested by the user or operator.

The Ground Station will archive and deliver upon request received Satellite telemetry and science data, providing a minimum of 10 gigabytes (GB) of digital storage and will also ensure the validity of commands sent to the spacecraft, as well as the integrity of telemetry and data received.

The Ground Station will track the satellite in its orbit when in line of sight. The Ground Station Antenna gain shall be such that, together with the tracking accuracy, maintains a link margin of more than 6 dB. The Ground Antenna must operate circularly polarized. In all instances, operation of the onboard communications and ground systems must comply with the Federal Communication Commission regulations.

Transceiver

The *.Sat transceiver is the MHX-2400, an off-the-shelf OEM radio-modem from Microhard Systems Inc. The MHX-2400 is a high-performance embedded wireless data transceiver. Operating in the 2.4000 to 2.4835 GHz ISM band, this frequency-hopping spread-spectrum radio-modem is capable of providing reliable wireless data transfer using an asynchronous serial interface.



Figure 24: Microhard MHX-2400

Its main characteristics are: one Watt of RF output power, serial interface at 115kbps, a sensitivity of -105 dBm, remote configuration, retransmission protocol and Forward Error Correction (FEC).

Space Vehicle Antenna

Two types of antennas were developed to be used with the *.Sat Bus: a microstrip patch antenna and a quarter-wave monopole antenna. Either antenna can be used on the BM with minimal changes, depending on the mission requirements.

For the GeneSat-1 Mission, a monopole antenna was chosen. Simulations have shown that using a passive attitude control system would meet the pointing requirements of both types of antenna, but since there is no flight data to validate the attitude simulation models, the more forgiving antenna is preferred for the first *.Sat mission.

Patch Antenna

A microstrip patch antenna was designed using the transmission line model. To achieve circular polarization, a slot was introduced in the patch together with a feed point 45 degrees offset. For the design, RT/Duroid 6002 from Rogers Corporation is used as a substrate.

The final dimensions, using a 50mm x 50mm Rogers RT/duroid 6002 substrate are the following: L = W = 32.7mm; Slot = 16mm x 0.5mm; Feed Point (@45°) = 12.6mm from edges.

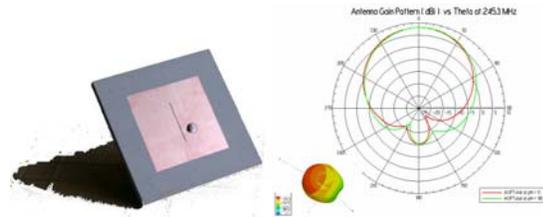


Figure 25: Circular Patch Antenna 3D Model and Radiation Pattern

The antenna bandwidth extends from 2.4 to 2.486GHz or 3.51% within 1.5:1 VSWR. VSWR is less than 1.5:1 throughout the entire ISM band.

The antenna gain is around +8 dBi and the beam-width at -6dB is ± 60 degrees. The slot inserted in the patch creates a 90 degree shift in the signals propagating in the two dimensions, creating a right-hand circular-polarization (RHCP).

Monopole Antenna

Because of the orientation given by the attitude control system, the monopole antenna will be located on the bus side (+Z face). The antenna was constructed with an SMB bulkhead connector and a 1.5mm diameter copper element.

The antenna can be housed between the P-POD Deployer back panel and pusher-plate in its operational configuration, so no deployment mechanism is needed.

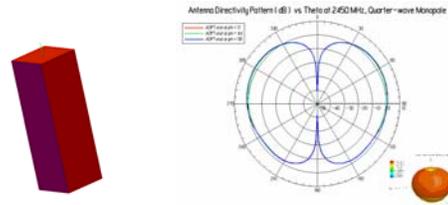


Figure 26: Monopole Antenna attached and Radiation Pattern

The antenna gain is around 2 dBi –more than 0 dBi between ± 30 -90 degrees. The signal radiated is linearly polarized.

Mounting Scheme

A mounting scheme for the Microhard MHX-2400 transceiver to the *.Sat structure was defined to maintain robustness in terms of vibration and thermal dissipation as the main objectives.

In order to achieve mechanical strength, the transceiver will be attached to the board with four screws, using the manufacturer-specified bolt-pattern. To dissipate heat generated in the RF-stage of the transceiver, an aluminum plate will be added as a heat sink, and thermal pads are placed between the plate and the components to create a thermal path from the heat-generating components to the metallic structure.

The transceiver performance was tested after the thermal pads were applied to make sure the RF circuitry was not altered. Figure 10 shows the final configuration for the C&DH board, with the transceiver, screws, aluminum plate and standoffs.

Doppler Shift and Path Delay

Doppler Shift cannot be compensated in the MHX-2400 transceiver. For that reason, the radio has to be able to handle the frequency shift for the mission orbit. The MHX-2400 was tested in the lab and the results showed that it can handle frequency shifts of 50 KHz, which is the maximum Doppler shift at 2.4GHz for a 510km orbit.

When radio waves travel a very long distance between transmitter and receiver, such as in a satellite application, the propagation delay could be very significant and could affect the data protocol in the modem.

Tests were conducted using only the digital stage of two radios, bypassing the RF stage, and connecting the TX and RX digital outputs and inputs through a digital delay circuit.

The maximum delay expected for a 510km is 10msec. Any data exchange should be completed before the frequency hop happens.

Setting the Hop Interval (MHX register S109) to 100msec and the maximum packet size to 255Bytes (MHX register S122) will allow sending any packet in both directions, finishing the data exchange within the same Hop Interval. The effective throughput of the transceiver will decrease approximately 20% in order to accommodate the propagation delay.

Thermal-Vacuum Characterization

The transceiver was tested in a thermal-vacuum chamber. Results showed an excellent performance in a simulated space environment. The transceiver operated successfully through cycles from -40°C to +60°C, exceeding the temperature range expected for the mission.

Ground Segment

The Ground Segment consists of the distributed Ground Segment facilities, components, and functionalities that have been designed and selected for meeting the needs of the GeneSat-1 mission. As depicted in Figure 28, the system consists of a single communication station, which is connected to a remote Mission Operations Center (MOC) via a secure Internet connection. Command and telemetry operations may be conducted both at the communications station as well as from the MOC.

External dissemination of science and engineering data as well as mission status meta-data will be served from the MOC. In addition, a low-cost teleconferencing communications network will support communications, coordination and mission updates between Ground Segment facilities, the launch facilities, and external scientists/managers.

Communication Station

The Communication Station is a facility owned and operated by SRI International and located on land leased from Stanford University. Refurbishments to the station are underway to meet the needs of the GS-1 mission. The facility consists of 18 m parabolic antenna driven by a programmed track antenna pointing system. Additional equipment specific to the GeneSat-1 mission includes an antenna feed, a transceiver, a data processing workstations, and encryption equipment for the Internet connection.

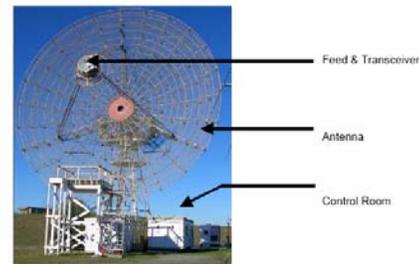


Figure 28: Ground Station Dish Antenna

Parabolic Antenna

The communication station antenna, shown in Figure 28, is a parabolic reflector 60 feet in diameter. The surface mesh is being conditioned to operate at 2.4GHz. The gain at 2.4GHz is approximately 50 dBi given estimates of an efficiency of 55%- and a beam width of 0.5 degrees. The antenna will only be operated at elevations higher than 10 degrees in order to minimize interference in the local region.

The parabolic reflector diameter is three-times larger than what is required to close the link with the S/V. This large diameter implies a narrow beam width, which is a concern given the expected antenna pointing errors. For these reasons, the parabolic antenna will be only illuminated in a 10 meter diameter area using a feed horn with a higher f/D ratio than appropriate for a 60 foot diameter antenna, thereby creating a wider beam. The resulting effect is a 10-meter-equivalent dish antenna with a gain of 45 dBi and a beam width of approximately 1 degree.

During contacts with the Satellite, the antenna will be positioned through a programmed track (slave) system. Programmed track control is an open-loop pointing control approach during contact which is susceptible to errors in satellite ephemeris and antenna pointing.

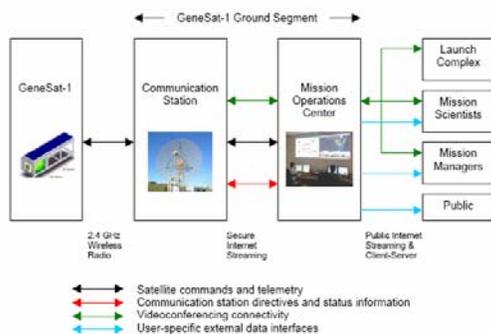


Figure 27: Ground Station Block Diagram

Feed Horn/Transceiver

The communication station will make use of a dual mode feed horn to obtain a narrow beam. The feed horn will be exited by two probes at 90 degrees from each other through a 3dB Hybrid Coupler to provide circular polarization.

A Microhard MHX-2400 transceiver, identical to the GeneSat transceiver, will be used for command transmission and data reception. The transceiver will be installed in a weather-pooof box and placed next to the feed horn on the antenna tripod. A RS485 signal will run from the transceiver at the feed horn to the ground station electronics room where it will be connected to the communication station data handling workstation.

The ground transceiver will be generating 1 watt of RF power (30 dBm) and the maximum antenna gain is 45 dBi. The Effective Radiated Power (ERP) is 30dBm +45 dBi = 75dBm. FCC Part 15 regulation limits the ERP to 36dBm. In order to operate the ground station a waiver to this rule is being filed at FCC. The basis for being beneficiary of this waiver is that the ground station antenna will only operate at elevations higher than 10 degrees and the antenna side lobes contain minimum power, so no interference will be affecting other ground receivers.

Link Budget

The link budget was calculated considering the Microhard MHX-2400 Transceiver at maximum power (1W), a spacecraft antenna pointing error of 45 degrees and a 10-meter parabolic dish antenna as a ground station. The following spreadsheet shows results for downlink and uplink for the monopole configuration.

		Down Link Budget		
Item	Symbol	Units	DL	DL
Orbit Altitude (km)		km	510	510
Elevation Angle		deg	10	90
Frequency	f	GHz	2.4	2.4
Transmitter Power	P	Watts	1	1
Transmitter Power	P	dBW	0	0
Transmitter Line Loss	Ll	dBW	-1	-1
Avg Transmit Ant. Gain	Gpt	dBi	0.0	0.0
Transmit Antenna Gain	Gt	dB	-1.0	-1.0
Eq. Isotropic Rad. Power	EIRP	dBW	-1.00	-1.00
Propagation Path Length	S	km	1719	510
Space Loss	Ls	dB	-164	-154
Prop. and Polariz. Loss	La	dB	-3	-3
RX Antenna Diameter	D	M	10	10
Receive Antenna Eff.	Eta		0.55	0.55
Peak Receive Ant. Gain*	Grp	dBi	45.42	45.42
RX Antenna Line Loss	Lr	dB	-0.5	-0.5
Receive Ant. Beam width*	Theta	deg	0.88	0.88
RX Ant. Pointing Error	E	deg	0.50	0.50
RX Ant. Pointing Err. Loss	Lθ	dB	-3.92	-3.92
RX Ant Gain w/ pointing	Gr	dB	41.0	41.0
System Noise Temp. **	Ts	K	585	585
Data Rate	R	bps	86000	86000
Eb/No	Eb/No	dB	23.8	34.4
Bit Error Rate	BER		10-5	10-5
Required Eb/No †	R.Eb/No	dBHz	13.5	13.5
Implementation Loss		dB	-2	-2
Margin		dB	8.3	18.9

Table 5 - 10-meter Dish Link Budget Using the Monopole Antenna

* Assumes Ground antenna is parabolic
 ** Rx noise temp=525K (Manufacturer data)
 † FSK:Req=13.5dB;
 All equations referenced are from SMAD III [10]
 Spreadsheet assumes zenith pass of S/C.

For a 510 km orbit, considering a minimum elevation contact of 10 degrees, the link margin for the downlink is better than 8.3 dB.

Access time

The *.Sat orbit is sun-synchronous with an altitude of 510 km. For this orbit in particular, a set of STK simulations was run reproducing the spacecraft attitude – antenna axis following the earth magnetic lines.

Using the patch antenna with a 120±60° beam width the average access time per day would be 9.29 minutes resulting in an average downlink capability of around 1.4 MB per day. Using a quarter-wave monopole antenna, the access time is longer. Simulations indicate that the average access time per day is 15.49 minutes. The communications subsystem downlink capability is around 2.3 MB per day.



Figure 29: 10-meter Dish Link Budget Using the Monopole Antenna

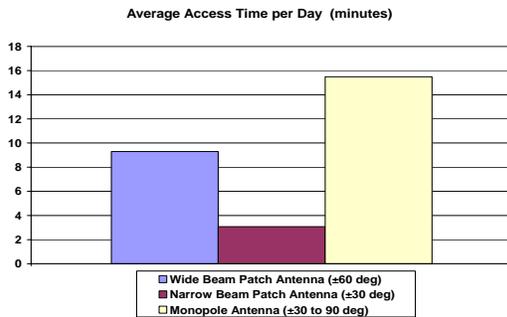


Figure 30: 510km orbit access time Statistics

CONCLUSION AND LESSONS-LEARNED

The development process for the *.Sat Bus and the GS-1 Payload was somewhat unique in that they were conducted in parallel. While the two design teams had independent schedules initially, keeping a dialogue to maintain the interface and capabilities of the fully-integrated system was a vital portion of the design process. As the designs matured, the primary difficulty came in maintaining the system capability while addressing the full gamut of specialized payload requirements.

Therein lies the challenge for any standardization effort – is it possible to put forth a set of capabilities robust enough to accommodate any number of mission parameters and requirements? As a counter-point, is it acceptable for a standardized S/V bus to dictate the limits of system capability to the mission payload designers?

The answer to these questions is difficult to put into absolute context. In the professional aerospace industry, cutting-edge capability is typically tailored to specific payload needs and not vice-versa due to prohibitions on the non-recurring engineering cost (NRE) of such development and the need for very specific and optimized fulfillment of the mission requirements. With the fast pace of technological progress, it is challenging to maintain a standardized system such that it can remain in synchronization with advances in the underlying technology and desired payloads.

That said, in the typical academic context, putting forth a standardized system with fixed capability may be especially useful in affording students a chance to focus solely upon mission design. With a constrained set of capabilities for the system, upper bounds on missions with academic experience as the primary goal can be accommodated much more quickly, avoiding the inevitable design iterations that must occur when the opposite approach is taken. A case in point would be

the success and relatively rapid adoption of the CubeSat Specification by so many universities worldwide. If the overall form-factor is acceptable, why should the community not accept a “standard” set of Bus services?

During the process of developing the interfaces for the *.Sat Bus, it rapidly became clear that the design drivers were not so much the technical issues specifically, as maintaining the dialogue necessary to keep the two design teams in lock-step at the interfaces themselves, be they data, power, communications, or mechanical.

While *.Sat is by no means proposing *the* standard for university or industry missions, its development offers a tantalizing glimpse that there is some viability in taking a “standards-based” approach for certain types of missions, whether academic or scientific. At the very least, to encourage dialogue within the Small Satellite community about putting forth the foundations of standards to further the experience of the next generation of satellite designers in the academic setting is a significant victory unto itself.

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