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Low Cost Rapid Response Spacecraft, (LCRRS)
A Research Project in Low Cost Spacecraft Design and Fabrication in a Rapid Prototyping Environment

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Abstract

The Low Cost Rapid Response Spacecraft (LCRRS) is an ongoing research development project at NASA Ames Research Center (ARC), Moffett Field, California. The prototype spacecraft, called Cost Optimized Test for Spacecraft Avionics and Technologies (COTSAT) is the first of what could potentially be a series of rapidly produced low-cost satellites. COTSAT has a target launch date of March 2009 on a SpaceX Falcon 9 launch vehicle. The LCRRS research system design incorporates use of COTS (Commercial Off The Shelf), MOTS (Modified Off The Shelf), and GOTS (Government Off The Shelf) hardware for a remote sensing satellite. The design concept was baselined to support a 0.5 meter Ritchey-Chretien telescope payload. This telescope and camera system is expected to achieve 1.5 meter/pixel resolution. The COTSAT team is investigating the possibility of building a fully functional spacecraft for $500,000 parts and $2,000,000 labor. Cost is dramatically reduced by using a sealed container, housing the bus and payload subsystems. Some electrical and RF designs were improved/upgraded from GeneSat-1 heritage systems. The project began in January 2007 and has yielded two functional test platforms. It is expected that a flight-qualified unit will be finished in December 2008. Flight quality controls are in place on the parts and materials used in this development with the aim of using them to finish a proto-flight satellite. For LEO missions the team is targeting a mission class requiring a minimum of six months lifetime or more. The system architecture incorporates several design features required by high reliability missions. This allows for a true skunk works environment to rapidly progress toward a flight design. Engineering and fabrication is primarily done in-house at NASA Ames with flight certifications on materials.

The team currently employs seven Full Time Equivalent employees. The success of COTSATs small team in this effort can be attributed to highly cross trained engineering team. The engineers on the team are capable of functioning in two to three engineering disciplines which allows highly efficient interdisciplinary engineering collaboration. NASA Ames is actively proposing mission concepts to use the COTSAT platform to accomplish science. If the COTSAT team validates this approach, it will allow the possibility for remote sensing missions to produce a high science yield for minimal cost and reduced schedule. Another aim of this approach is to yield an accelerated pathway from a Phase A study to mission launch. Leaders in the aerospace industry have shown interest in this methodology. Several visits and tours have been given in the lab. Although the concept of low-cost development is initially met with skepticism from some within the prohibitive aerospace industry, the project's efforts have been highly praised for the accomplishments met within a limited time and budget. Overall the development has progressed tremendously well and the team is answering critical questions for current and future low-cost small satellite developments. COTSAT subsystems are not limited to a specific weight class and could be adapted to produce smaller platforms and to fit various launch vehicles.

1 Introduction

COTSAT is a $\sim$ 400kg small satellite. The system is baselined to accommodate a 0.5 m Ritchey-Chretien telescope as its primary payload. In the system's current architecture the telescope has been replaced by a payload camera that is sensitive to extended red emission (ERE) wavelengths. The avionics for the system are housed in an single-atmosphere artificial environment. Figure 1 shows a CAD model of the interior of the sealed container, Figure 2 shows the actual prototype. The container component of the project is the key parameter that allows for a critical path on cost and schedule to become a reality. The concept has been proven by the former Soviet Union with the first satellite Sputnik ([9]). Several other Soviet pressurized spacecraft have been designed and flown successfully. The heritage design attracted the interest of ARC because of the great cost reduction possibilities. For example the stabilization of a CCD at a condition of $\Delta T \simeq 0$ is a reduced risk for minimal design cost for the primary camera and star tracker systems. The thermal transfer with an air medium in the container allows for several reduced cost design options. The design includes three-axis control with an estimated accuracy of 35 arcsec pointing capability. ARC designed and developed GOTS systems including reaction wheels, star tracker and electrical power systems. Each system was developed in-house at ARC. COTS parts are frequently utilized to save on development cost, including for example a PC/104 onboard computer. There are three radio devices on the system. A low data rate UHF beacon, flown on GeneSat-1, is a COTS radio that broadcasts a 64 character message. The second one is the MHX-2400/2420 2.4 GHz MicroHard radio transceiver that also has spaceflight heritage on the GeneSat-1 flight. The MicroHard radio will be used for spacecraft commanding. The third radio is a SpaceQuest transmitter capable of a 1.4 Mbps data rate. The government established a partnership with Santa Clara University to operate all three of these radio systems. The avionics has several redundancy features added to minimize the risk characteristics of this flight system. The overall scope for a mission with a COTSAT is to have a rapid deployment of a remote sensing system with reliable performance for at least six months. The spacecraft objective is to accommodate low cost access to space for variable remote sensing payloads. The architecture allows for future expansion for possible biological payloads. Thermal attributes are also helped in this approach. Since lab air at standard atmospheric pressure with a desiccant will be encapsulated in the container prior to launch integration, the thermal characteristics will be manageable for the avionics. Two fans move the air as a convective coolant promoting an isothermal gaseous environment for electronics extending overall spacecraft subsystem lifetime characteristics. The project uses industry data interfaces standards such as USB 2.0 and Ethernet, from which the project can leverage off existing software device drivers already written for COTS items. In some cases a modification to the software drivers are necessary, however, overall this approach is effective and has proven significant cost reductions.

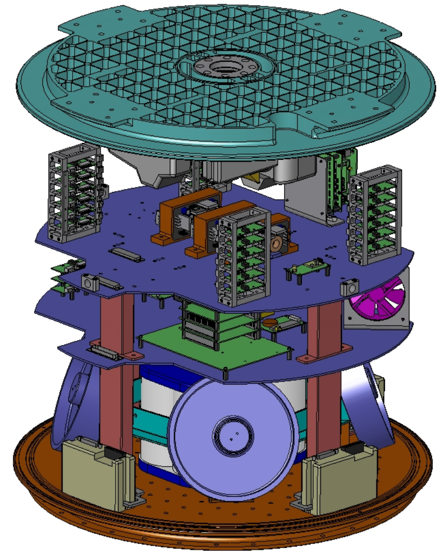


Figure 1: Avionics of BUS internal/sealed container shell removed

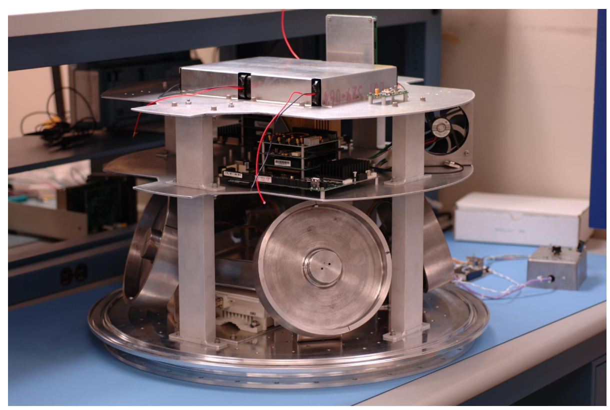


Figure 2: Bus Avionics in build phase

2 Structure

Throughout the design of COTSAT, heritage spacecraft design techniques utilizing inexpensive aluminum sheet metal and extrusions are applied to minimize fabrication costs while ensuring NASA flight quality. COTSAT benefits from a highly skilled sheet metal fabrication shop employed on-site at NASA Ames Research Center.

To further reduce cost, avionics components are sealed in a container to prevent exposure to vacuum, extreme temperatures, and other characteristics of the space environment. This sealed container is the main structure of the spacecraft, providing the interface to the launch vehicle as well as mounting point for the solar array structure. The solar array structure has been designed as a standalone frame of thin aluminum sheet shear panels and extruded aluminum supports that provide mounting points for each array. To minimize load transfer to the solar arrays, the solar structure will "float" during launch on twelve separate vibration isolators.

Several members of the COTSAT team have attended training courses (provided by various NASA centers) specific to critical hardware handling and spacecraft mechanical assembly to ensure structural and component-level assembly quality is maintained. Certified technicians will be responsible for applying torque, per specification, to all fasteners.

2.1 Sealed Container

The sealed container, as shown in Figure 3, is filled with one atmosphere standard laboratory air during integration. The sealed container is comprised of an aluminum cylinder with light-weighted monolithic aluminum endplates at each end. Each endplate features a redundant o-ring and Marmon clamp interface system. Manufacturing tolerances have been specified to maintain seal integrity and proper alignment of spacecraft components. The cylindrical portion of the sealed container is primarily fabricated from rolled-and-welded sheet aluminum and features external, longitudinal ribs and billet flanges. The external ribs help prevent structural failure in modes including axial loading, acoustics, and buckling; additionally, these ribs provide attachment points for an otherwise void round surface.

The aft endplate is designed to mate with the launch vehicle separation system while the forward endplate is designed to mate with a number of payload systems, including optical systems. Both endplates feature a waffle grid pattern on one side to provide a stiff yet lightweight structure capable of heat dissipation for any thermally sensitive components. On the reverse side, a grid of #10-32 blind threaded holes on a 1.500" pitch is provided to allow for non-specific, modular avionics equipment buildup within the sealed container. When the system is serviced, the aft endplate and avionics are removed as a single unit and cabling can be disconnected via connectors located at the pass-through port. This design will allow access to the avionics in approximately one hour.

The sealed container accommodates five optical ports (four star tracker ports and one payload port) and two electrical pass-through plates. The electrical pass-through ports contain off-the-shelf, hermetically sealed connectors welded into removable 304L stainless steel plates. Each electrical pass-through provides enough pins and connectors for solar arrays, a magnetometer, a ground service umbilical, a number of radios and antennas, a separation switch, and any connections needed for an external payload; including (but not limited to) servo motor connections and non-explosive actuator (NEA) connections.

A weld-test chamber was fabricated onsite by the Ames Research Center's Light Metal Fabrication Group in May 2007 to verify conformance to American Welding Society's standard D1.2. The test chamber was filled with helium and has maintained a gauge pressure for seven months. It is assumed nearly all pressure loss is due to the number of pressure gauge fittings sealed only with PTFE tape. Ames Research Center is currently modifying a vacuum chamber to confirm the integrity of the sealed container.

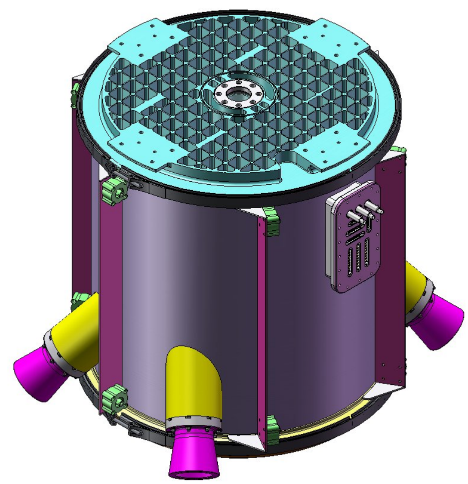


Figure 3: External view of Sealed Container

2.2 Spacecraft Bus

The COTSAT bus/avionics structure is designed in an effort to maintain modularity between various subsystems. The interior of the sealed container is built in three modular tiered compartments, stacked on top of each other: the aft endplate and two pedestals. The reaction wheels are mounted to the aft endplate to enhance its structural strength and to minimize jitter by removing any possible resonance with the reaction wheels' support structure. The middle pedestal is the center point for Command and Data Handling (C&DH) and the Attitude Determination and Control System (ADACS), providing mounting points for components such as the PC/104 stack, inertial measurement unit (IMU), global position system (GPS) receiver, and torque coil h-bridges. The forward-most pedestal provides the location for the primary components of the electrical control system (ECS). This modular approach allows for testing of individual subsystems and components during testing, servicing, and integration without the need for a completed spacecraft assembly.

3 Electrical Power Subsystem

The COTSAT Electrical Power System (EPS) architecture utilizes a distributed power approach and self-monitors system health. The time needed to design a power architecture, and build and qualify it are all very costly. COTSAT's approach to using a configurable and generalized design standardizes the system architecture and components. This standardization allows for quick production, ease in reproducibility, and a reduction in cost due to a resulting reduction in design effort and implementation. The electrical power subsystem architecture consists of a central power controller (CPC) and up to 32 remote power controllers (RPC). This architecture allows a flexible means to design and build small spacecraft in a cost effective manner. Both of these designs leverage off of the circuit design methodology and components flown and proven on the GeneSat-1 mission [3]. The incorporation of these designs into this system provides for high fidelity and an overall increased confidence in reliability. The GeneSat bus capability to control three subsystem power paths has been incorporated into the CPCs design and further expanded to be able to control 32 of these power subsystems. Each electrical power subsystem is outfitted with active and passive controls to detect and mitigate Single Event Upsets (SEUs) or latch-ups experienced by on-orbit radiation. The

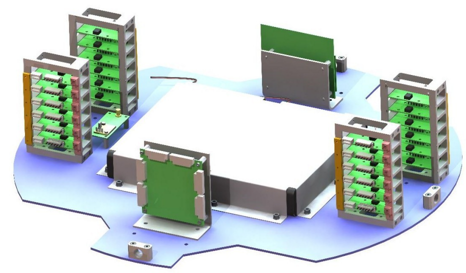


Figure 4: Electrical Power System

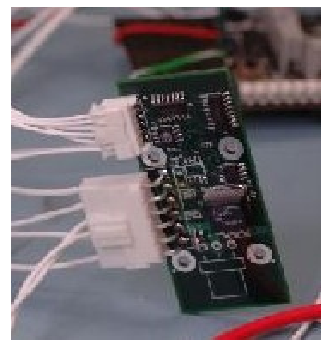


Figure 5: Remote Power Controller

systems also measure the system health by monitoring temperature of subsystems as well as current and voltage readings of all power delivery devices. If an anomaly is detected, software on a PIC microcontroller will indentify the problem and will shut down, using a watchdog timer for full system shutdown similar to the GeneSat-1 flight configuration. A model of the electrical power system is shown in Figure 4.

3.1 Remote Power Controller

The RPC is a small form factor (0.5"x1"x2") module that provides clean and down converted DC power to a system by regulating voltage and safely limiting current from an unregulated power source. The small form factor makes this method extremely cost effective to implement. Please refer to Figure 5.

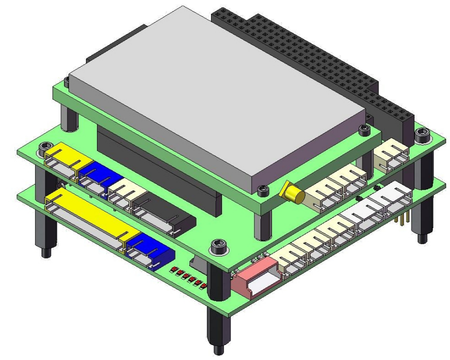
The CPC regulated power is configurable. Jumper settings further allow unregulated power to be passed through should that be a desired setting while still implementing all of the other RPC suite features described below. One such example includes using an RPC configured to monitor solar panels aboard a spacecraft. This configurability allows for easy expansion or contraction of spacecraft design features. For example, should the power requirement of a payload change from needing one supply, at 5V, to needing two supplies, one at 10V and one at 3V, this can easily be accommodated by removing the 5V RPC and dropping in two newly configured RPCs. In this example a new layout or qualification testing was not required thus saving on time and cost. These designs allow for up to 7A maximum current. Current monitoring circuitry will shut off the regulator aboard the RPC should there be a short circuit, latch-up incident or some other form of failure. After a preconfigured timeout occurs, the circuitry will reset the device and attempt to provide power to the once failed device. This feature is particularly important, because a power cycle is needed to mitigate a failed state in many cases. The RPC also outputs voltage and current sensing signals allowing for a centralized controller to monitor its power profile. The RPC's enable-line allows for a given controller to switch the RPC ON (regulate power) or OFF (disallow sourcing power). The enable line and voltage/current sensing lines allow for software monitoring, thus giving feedback to a user, and allow for latch-up and failure mitigation via smart software.

3.2 Central Power Controller

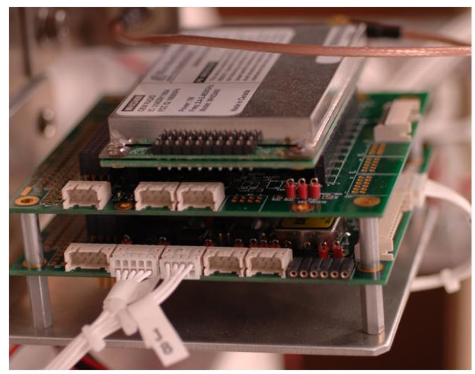
Given the capabilities of the RPC, the need for a centralized controller, and the need for a spacecraft bus, the CPC was conceived. Please refer to Figure 6.

The CPC consists of a PC/104 form factor stand alone primary board, with the option of an added expansion board with expanded capabilities. The small, low-power, and standard form factor allows for use in many applications. It also comes with the standard 8-bit ISA connector as a pass-through allowing for it to be stacked onto an existing PC/104 platform. The PC/104 form factor also allows for easy integration into cubesat projects, robotics projects, military applications, and many other products utilizing PC/104 computers or projects that are limited in size and mass. The system utilizes the low power electronics that are heritage from GeneSat-1 mission.

The primary board is configured to manage 20 RPCs, including eight RPCs with an added load control line that can be used for multiplexing, switching, polarity setting, or any other digital signal associated with the end system. It carries 6.4 MB of flash memory, a real time clock and watchdog timer, four channels of a multiplexed RS232 com port, an onboard pressure sensor, an onboard relative humidity sensor, and input lines for 16 precision AD590 temperature sensors. The RS232 com channels may be used to send data through a transceiver thus establishing a radio link. The expansion board expands the CPC's capacity, by simply stacking it on to the primary board. It increases the CPC's ability to manage 32 RPCs, and utilizes input lines that may be used for six sun sensors. It has two onboard RPCs established for a flight heritage beacon (flown on GeneSat-1) as well as a MHX2400/2420 Microhard radio (also flown on GeneSat-1). Connections are provided on the expansion board for installation of these two radios. The expansion board is designed such that the Microhard radio neatly stacks onto this configuration, leading to an overall compact, low-mass, high-fidelity, low-power design. This compact design fits in a volume smaller than that of a single cubesat.



(a) CAD Model

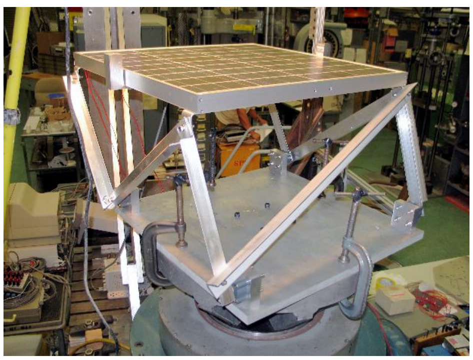


(b) Actual Hardware

Figure 6: Central Power Controller

The CPC will handle all safe mode operations of the satellite and will handle and execute commands and uplink mission control data commands. When the PC/104 ADACS control systems are off the CPC will execute a $\dot{B}$ control function pointing the spacecraft, with the torque coils, that will park the satellite in an optimized power generation attitude, antenna pointing and thermal characteristics.

(a) Solar Arrays



(b) Mounting Structure

Figure 7: COTSAT Solar Arrays and Mounting Structure

3.3 Solar Arrays

COTSAT is using Kyocera KC40T-1 panels, as shown in Figure 7(a), that were intended for home power generation. With a few modifications the system is achieving a reliable solar panel fit for space application for the cost of \$10/watt compared to the industry standard of \$500/watt to \$1000/watt. The solar panel assembly consists of eighteen 40 watt panels in a hexagon configuration around the sealed avionics container. The panels are mounted in this position by structural support angles connected in turn to the longerons through vibration isolation mounts.

Single 40 and 50 watt panels were subjected to 19 g of vibration in both the longitudinal and vertical axes. Both the glass substrate and the electrical connections survived. Later the panels were tested in a thermal vacuum chamber, and have been shown to keep the same level of performance between -70^o C and +100^o C. Part of the solar array mounting structure is shown in Figure 7(b).

Figure 8: Optima gel-cell battery pack

The arrays were modified to relieve outgassing by drilling approximately 12000 0.01” diameter holes through the sealing membrane. In addition, the plastic J-Box was removed. Six sun sensors mounted in three orthogonal directions were also added to the solar array assembly.

3.4 Battery

COTSAT power is provided by a commercial deep cycle marine/RV sealed lead acid (SLA) model D34M purchased from Optima, shown in Figure 8. This battery provides 13.2 VDC fully charged with a capacity of 55 Ah or 726 watt-hours of power. The cost of the battery was approximate \$220.

A battery monitoring board has been designed which incorporates system power kill, battery voltage monitoring, charge/discharge current monitoring, timed overcurrent shutdown and an external battery charger connection circuitry that will allow for optimal battery life and cycling.

4 Communications

The communications subsystem is divided in three main data paths: a command and telemetry bidirectional link, a high speed downlink for payload data transmissions and a beacon in the amateur band for outreach purposes.

4.1 Command and Telemetry

The main command and telemetry link utilizes the MHX-2420 transceiver as shown in Figure 9. The MXH-2420 is an off-the-shelf OEM radio-modem from Microhard Systems Inc. and is backwards compatible to the discontinued MHX-2400, which was flown successfully on GeneSat-1 [6] and is still operational after more than

Figure 9: MicroHard Radio, Low data rate

one and a half years in orbit. The MHX-2420 is a high-performance embedded wireless data transceiver. Operating in the 2.4 to 2.48 GHz ISM band, this frequency hopping spread-spectrum radio-modem is capable of providing reliable wireless data transfer using a simple asynchronous serial interface. Major advantages over other options were its 1-watt RF output power, its comparatively good sensitivity, a relatively slow frequency hop time interval, its compact size, and its operational flexibility. Built-in features of this module include addressing, retransmission protocols, encryption and forward error correction. The over-the-air data rate is fixed at 172 kbps providing a theoretical maximum throughput of 83 kbps. The signal is Gaussian Frequency Shift Keying (GFSK) modulated. It has 20 pseudo-random, user selectable frequency hopping patterns. The transceiver is a full radio-modem that performs packetization, modulation and demodulation. The MHX-2420 electrical interface is particularly simple, which promotes rapid integration. It accepts regulated 5 volts from the electrical power subsystem and communicates with the Command and Data Handling (C&DH) processor through a serial port with selectable data rates of up to 115.2 kbaud. Hardware flow control is used to manage data flow. MHX-2420 power consumptions in its different operating modes are shown in Table 1. The onboard antenna for this transceiver is the RooTenna 2 COTS patch antenna from Laird Technologies, with a gain of 15dBi. The antenna gain pattern is shown in figure 10.

4.2 High-speed downlink

The large volume of data generated by the payload is transmitted to the ground by a dedicated S-band transmitter, the TX-2400 from SpaceQuest, as shown in Figure 11. This module delivers 2.5 Watts with a maximum data rate of 1.4 Mbps using GMSK modula-

Mode	Mean Power (W)	Peak Power (W)
Standby/Receiving	1.15	1.15
Receiving (w/ACK)	1.45	5.51
Transmitting	4.38	5.51

Table 1: MHX-2420 power consumption

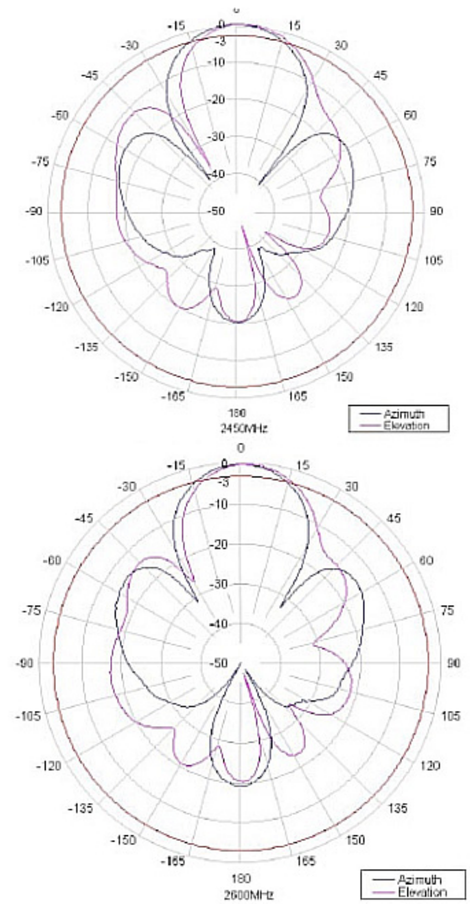


Figure 10: R2T24 Antenna Pattern

tion. SpaceQuest has conducted qualification testing to verify the transmitter will withstand the shock, vibration, temperature extremes and vacuum of space with minimized outgassing into the host spacecraft. The transmitter is contained in a compact package of 2.7 x 1.4 x 0.38 inches excluding connectors and has a mass of less than 4 oz. The transmitter is also qualified to withstand a 1000 g shock and 100 g acceleration while operating. Table 2 summarizes the TX-2400 transmitter environmental specifications.

This radio was successfully flown on several missions to include Bigelow Aerospace Genesis I and II and US Air Force Academy FalconSat III among oth-

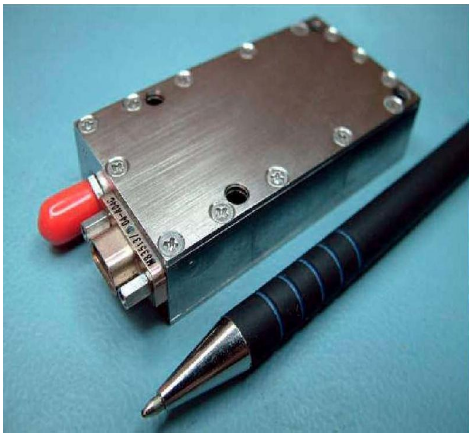


Figure 11: Space Quest Radio, High data rate

Specification	Value
Operating Temperature:	-20^{o}C to +70^{o}C
Random Vibration:	20 G, 20-2,000 Hz, 3-Axes
Shock:	1,000 G
Acceleration:	100 G
Altitude:	Unlimited
Humidity:	To 95%

Table 2: TX-2400 transmitter environmental specifications

ers. A dedicated 17.5dBi COTS patch antenna from Wireless Edge Ltd. radiates the signal to the ground. The antenna gain pattern for this antenna is shown in Figure 12.

4.3 Beacon

A UHF transmitter onboard the spacecraft broadcasts basic information about system health. This allows for

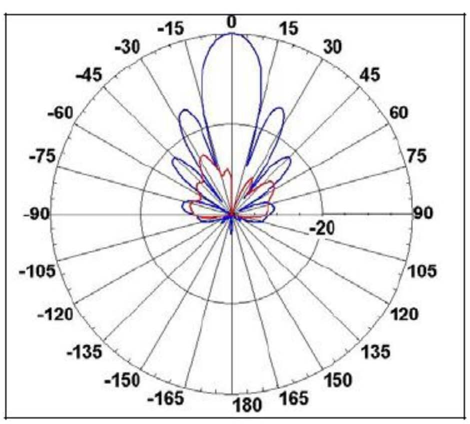


Figure 12: 140 Antenna Pattern

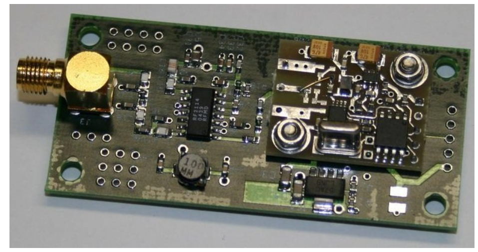


Figure 13: Beacon

the execution of a secondary outreach program, where universities and amateur radio operators around the world can directly receive spacecraft information and conduct basic analyses. This UHF transmitter was developed by the The StenSat Group LLC and used successfully in the GeneSat-1 Mission. It has a RF output power of 250mW and transmits AX.25 packets at 1200 bauds. It uses FSK modulation. Figure 13 shows the beacon board.

4.4 Communication Station

A single ground communication station will be used to support mission operations. Located at Santa Clara University, this station uses two separate 3-meter parabolic tracking antennae, one for ISM-band command and telemetry data and the other for high-speed S-Band payload data. Figure 14 shows the 3-meter dish assembly. In addition, a co-located OSCAR-Class amateur radio station is used to receive the UHF beacon signal in support of the missions Education and Public Outreach program. Apart from the antennae, additional ground station equipment includes data routing and configuration control workstations, transceivers and modems matching the on-board communication components, and antenna tracking equipment. Apart from modifications to support the high-speed payload link, the station design has been used for several years in support of a number of NASA, university and amateur radio missions [8], [5], [7].

4.5 Mission Control

Mission operations will be conducted by students in Santa Clara University's Robotics Systems Lab, the same team in charge of operations for multiple other NASA Ames missions such as GeneSat-1. The mission operations architecture uses a distributed, internet-based software system that allows command and telemetry

2

COTSAT Link Budget *

Item	**Symbol**	**Units**	**S-band Down**	**MHX Down**	**MHX Up**	**Beacon Down**
Orbit Altitude (km)		km	450	450	450	450
Elevation Angle		deg	10	10	10	10
Frequency	f	GHz	2.2	2.4	2.4	0.4371
Transmitter Power	P	Watts	2.5	1	1	0.25
Transmitter Power	P	dBW	4	0	0	-6
Transmitter Line Loss	L_l	dBW	-1	-1	-1	-2
Avg Transmit Antenna Gain	G_{pt}	dBi	17.5	3	15	-3
Transmit Total Gain	G_t	dB	16.5	2	14	-5
Eq. Isotropic Radiated Power	EIRP	dBW	20	2	14	-11
Propagation Path Length	S	km	1570	1570	1570	1570
Space Loss	L_s	dB	-163	-164	-164	-149
Propagation and Polarization Loss	L_a	dB	-3	-3	-3	-3
Receive Antenna Diameter	D	M	3	3	3	2
Receive Antenna Eff	η	0	0.6	0.6	0.6	0.6
Peak Receive Antenna Gain	G_{rp}	dBi	34.2	35.0	35.0	16.6
Receive Antenna Line Loss	L_r	dB	-0.5	-0.5	-0.5	-0.5
Receive Antenna Beamwidth **	θ	deg	3.2	2.9	2.9	24.0
Receive Antenna Pointing Error	E	deg	0.5	0.5	0.5	5
RX Antenna Pointing Error Loss	L_θ	dB	-0.3	-0.4	-0.4	-0.5
Receive Antenna Gain with pointing error	G_r	dB	33.4	34.1	34.1	15.6
System Noise Temperature ***	T_s	K	300	585	850	300
Data Rate	R	bps	1400000	86000	86000	1200
Eb/No	Eb/No	dB	30.0	20.7	31.1	25.5
Bit Error Rate	BER		10-5	10-5	10-5	10-5
Required Eb/No ****	Req Eb/No	dB-Hz	10	13.5	13.5	13.5
Implementation Loss		dB	-3	-2	-2	-2
Margin		dB	17.0	5.2	15.6	10.0

* Equations are from [10]. Spreadsheet assumes zenith pass of S/C.
** Assumes Ground antenna is parabolic
*** Rx noise temp=300K (Estimated)
**** GMSK:Req=10dB

Table 3: Link Budget for the S-band radio, the MHX2400/2420 and the Beacon

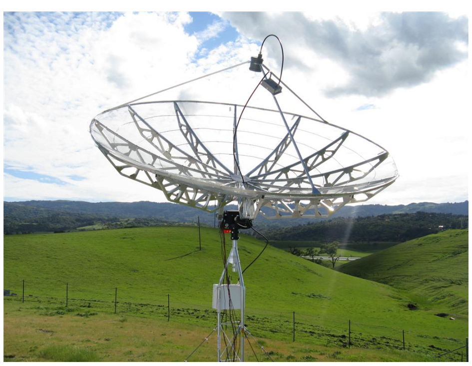


Figure 14: The ground station 3m dish

operations to be performed from any networked location; in practice, however, such operations will be limited to pre-approved locations such as at the ground station, from the NASA Ames Multi-Mission Operations Center and from the SCU robotic operations facility. Mission control hardware includes several workstations, communication equipment, and a variety of large format displays. Control software relies on a data streaming server that routes realtime data between different applications such as the command and control software, the mission database, and communication station applications. A variety of COTS and research-grade software tools support mission and contact planning, health and anomaly management, and payload data processing functions. All mission control activities are compliant with NASA requirements for information assurance and configuration control, and a formal training and certification process is used to admit students to the mission operations team.

5 Command and Data Handling

A critical portion of the satellite is the Command and Data Handling (C&DH) subsystem. The C&DH subsystem provides a number of critical capabilities, including spacecraft health and status monitoring, communication with ground stations, payload science data management and subsystem management. The C&DH subsystem, Figure 15, is a two-tiered system consisting of the CPC and a PC/104 computer.

As discussed previously, the CPC handles system health, mitigates problems, and communicates to the ground via a MHX 2400/2420 transceiver. The MHX-2420 transceiver is the only two-way communication on the satellite and is responsible for receiving uplinked spacecraft commands and relaying system health and status reports to the ground station. Because flight heritage design was incorporated into this design, it stands as a critical component to the reliability of the C&DH system. The CPC interfaces with a PC/104 computer via an RS232 communication link and utilizes this channel to relay information. This interface allows for commands to be sent, processed, and responded to.

The PC/104 computer handles complex tasks such as star tracker attitude estimation, attitude control system tasks, payload image processing, payload data storage, and high data rate down-link communication. Overall system control of the PC/104 computer is controlled by the CPC. When the overall spacecraft is in good health with adequate power, the PC/104 system will be activated to perform precision attitude control and payload science data processing.

A PC/104 avionics stack was chosen for the flight computer due to the heritage in rugged industrial applications and the flexibility in adding expansion peripheral hardware. The PC/104 avionics stack selected is an Advanced Digital Logic ADL855 PC/104 CPU board with PC/104 I/O expansion boards. The ADL855 consists of a 1.0 GHz Celeron-M Processor with 1 GByte of ECC DDR DDR-DRAM, 4xUSB 2.0, 2xRS232 COM ports, and 10/100MBit LAN. The ADL855 has been coupled with a heat pipe thermal solution, allowing direct thermal transfer to the satellite structure and a wide operating temperature range of -40 C to +85 C. For onboard storage, the ADL855PC is connected to an 8 GByte RAID 1 configuration utilizing two Sandisk Extreme III compact flash cards for added reliability.

Additional spacecraft input/output capabilities have been added to the PC/104 avionics stack via PC/104 expansion cards. A Sealevel 3514 ACB-104.Ultra high speed RS-422 serial interface, capable of 6 Mbs sustained transfer rate has been added to interface with the high data rate communications radio. A Diamond Ruby-1612XT, 16-channel, 12bit digital to analog output interface with 24 digital I/O lines is used for controlling the satellite reaction wheels.

5.1 Satellite Software

In order to utilize COTS hardware with maximum flexibility, an operating system with a high level of industry driver support is required. MS-Windows and GNU/Linux are two of the top widely supported operating systems with respect to hardware driver support. In the early phases of the project, MS-Windows was the clear choice for the satellite operating system based simply on the fact that MS-Windows has the most widely vendor supplied hardware driver support. This in turn would maximize the pool of available COTS hardware from which satellite hardware could be selected. As a result, cost would be reduced by not developing custom drivers, and by providing a range of compatible hardware with various performance options.

After assembling the first hardware/software integrated test platform, it was determined that MS-Windows could not provide the satellite with reliable task scheduling, task preemption, or adequate timing and multitasking as required by the satellite software. In fact, on the initial test platform it was found that the control system would go unstable when the controls task would become delayed by the star tracker attitude estimation process. Thus, in addition to the hardware compatibility requirement, a soft real time operating system with the ability to schedule and prioritize tasks is also required for the satellite software and control system. GNU/Linux with the 2.6 kernel has demonstrated the capability of performing sufficient controls and thread management in other industry and academic projects. As a result, GNU/Linux was selected for the satellite operating system. Cost reduction was also a byproduct of the switch to GNU/Linux. Open source software libraries and community supported software testing could then be leveraged to reduce development costs. The Debian GNU/Linux distribution was selected due to the vast number of available software packages and the ease of streamlining the system to a minimal set of installed components. In addition it was more important to have the rock-solid stability found in Debian rather than bleeding-edge software versions.

The Linux kernel is compiled with low-latency preemption options and high resolution timers turned on for soft real time performance. The majority of the

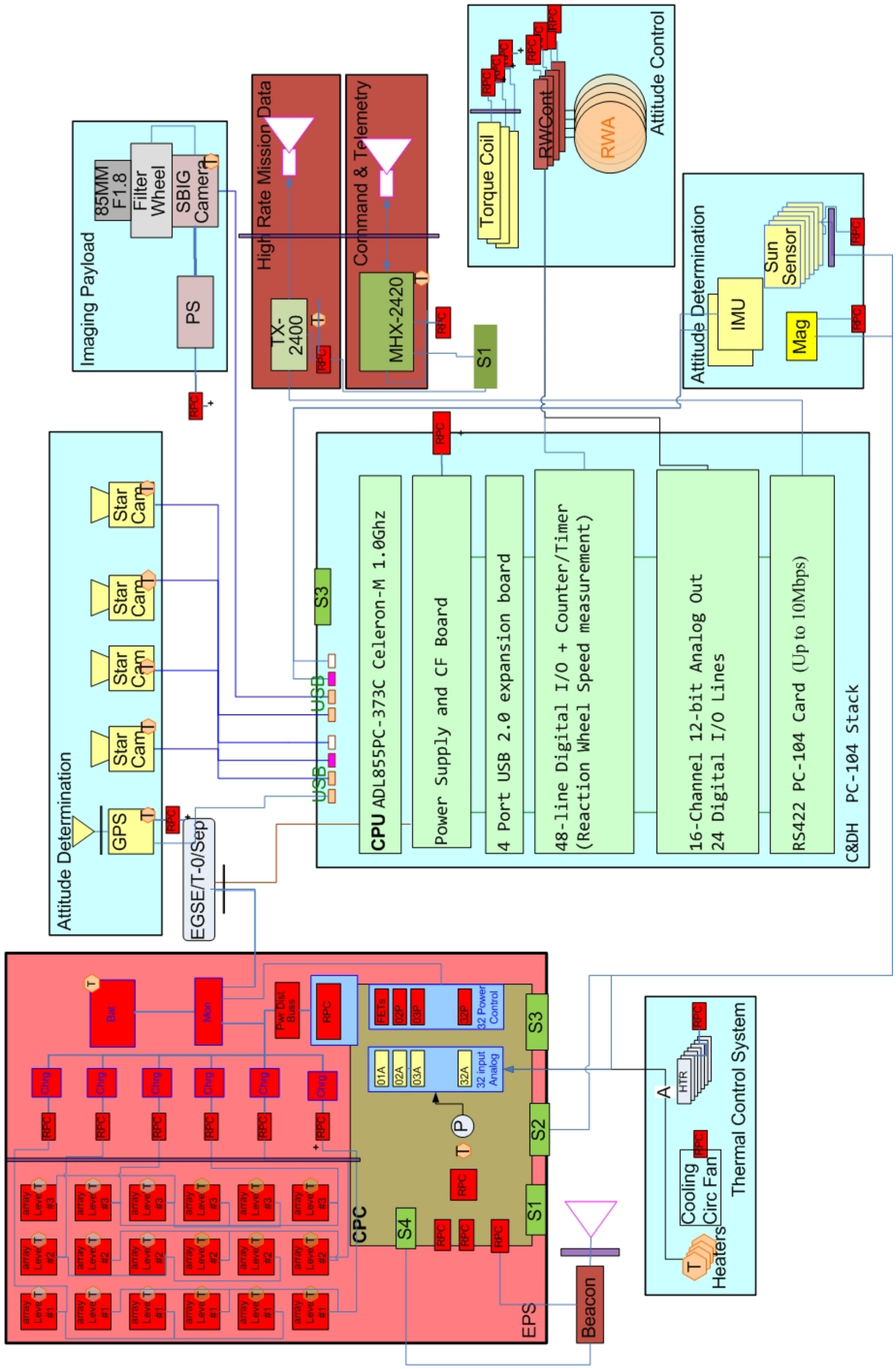


Figure 15: Functional Block Diagram

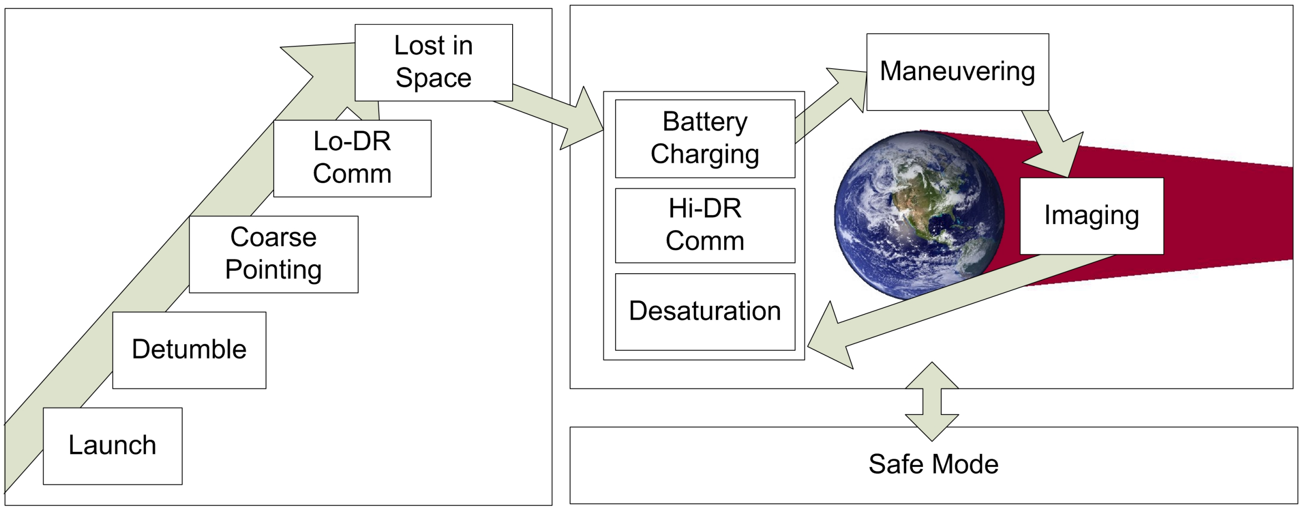


Figure 16: Overview of Operating Modes

real-time kernel preemption patches from Ingo Molnar are already in the kernel source tree from kernel.org. Task scheduling and prioritization is implemented using Posix Threads (PThreads), a standard threading library. PThreads is commonly used in Linux for a number of applications including kernel drivers. By using a POSIX compliant threading structure, the software will operate on any POSIX compliant operating system, preventing the software from becoming locked to a particular operating system or distribution. Additionally, the use of a standard library maximizes reusability and makes the generated code more readily understood by other software programmers, hence reducing training time and cost.

In a rapid prototyping environment with multiple software developers, the software must be developed in a manner which maximizes parallel development and still mitigates the risk of multiple software versions. By using the power of the Debian package management system, parallel rapid development and configuration management is achieved. Each module, for example the module to interface with hardware such as digital to analog input/output, is written and compiled into a dynamic shared object library. Each shared object library for the satellite software is then distributed as a Debian package, which ensures proper software version dependency resolution and proper previous software version removal.

6 Attitude Determination and Control

6.1 Control System

COTSAT will use a three-axis attitude determination and control system (ADACS) using four reaction wheels (0.02 Nms capacity, 0.065 Nm torque output) in a pyramidal configuration with the apex of the pyramid along the satellite roll axis. The reaction wheels are located as close to the center of mass of the spacecraft as possible to reduce wheel induced jitter. Three magnetic torque coils will be used for detumbling, coarse-pointing, and to desaturate the reaction wheels. Both the reaction wheels and the torque coils are developed in-house at NASA Ames. Three-axis attitude knowledge will be provided by two IMUs, four Star Trackers and four sun sensors. The Star Tracker algorithms are developed in-house, as described in section 6.6.

Coarse-pointing of the spacecraft is achieved using the magnetic torque coils, with attitude knowledge provided by the IMU. The control algorithm in this mode runs entirely on the CPC; the PC/104 can be switched off in this mode to save power. The fine pointing control, (when imaging with the science instrument or when communicating through the high-datarate radio) nominally uses three of the four reaction wheels. The attitude knowledge will be provided by the Star Trackers, with the control algorithm running on the PC/104 stack. The control algorithm uses a quaternion error controller. The fine pointing control achieves an overall accuracy of 35 arcseconds, with jitter less than 15 arcsec/sec.

6.2 Operating Modes

During *launch* all subsystems are switched off. *Detumbling* and *coarse pointing* of the spacecraft is done using the torque coils, with the control algorithm running on the CPC. The PC/104 is switched off in these states. During *call home*, the CPC uses the magnetometer and the torque coils to keep the microhard radio pointed to the ground station. The PC/104 is switched on in the *lost in space* mode, to allow the startrackers to acquire attitude information. During *Maneuvering*, *imaging*, *comm* and *desaturation* the PC/104 controls the reaction wheels. In *standby* mode, the Spacecraft is in low-power mode; the torque coils keep the solar arrays pointed at the sun to charge the batteries. In *safe* mode, only the bare essential subsystems are functioning. Thermal or low-power conditions may cause the spacecraft to enter this mode. Figure 16 summarizes the different modes of the spacecraft avionics system.

6.3 Reaction Wheels

The reaction wheel system (RWS), as shown in Figure 17, uses four wheels in a 4/3 redundant configuration for fine pointing control. Three-axis control software has been developed at NASA Ames in Matlab for use on the Linux-PC/104 platform. The reaction wheel system is built in house at NASA Ames. By housing the RWS within the COTSAT sealed container this design can make wide use of COTS technology.

The wheel design itself is an evolution from the Personal Satellite Assistant reaction wheel designed at Ames making use of off the shelf kit motors and widely available bearings. The wheel is sized based on the torque and angular momentum required to rotate the Spacecraft with the baseline optical payload 180° in 90 seconds and to track an object on the earth (such as a ground station or coral reef) from LEO with margin. The mounting angle is selected to a balance pointing

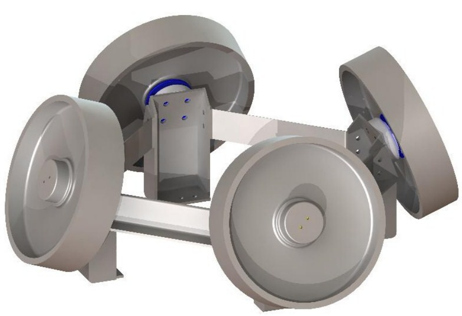


Figure 17: Reaction Wheel Assembly

Figure 18: Magnetic Torque Coils

performance in each of the three-axis of control. The wheel is manufactured in NASA Ames machine shops from a water knifed billet, CNC machined and balanced in NASA Ames test labs.

The servo drive electronics are procured as a COTS item with NASA Ames COTSAT team making minor modifications to the internal electronics and developing interface circuitry between the OEM servo controller and the PC/104 & CPC electronics. NASA Ames electrical design provides power conditioning for the servo motor drive electronics and synchronization between the power conditioners to reduce electronic noise.

The driver for this development activity is to produce a RWS at near one 10th the lowest cost available in the industry of a system with equivalent control authority, acceptable jitter and one year mission life.

6.4 Torque Coils

The torque coils serve three purposes: detumbling, coarse three-axis control of the spacecraft and momentum unloading of the reaction wheels. The coarse pointing is achieved through a $\dot{\vec{B}}$ control law using the well-understood magnetic field of the earth and magnetometer readings. There are three torque coils on the spacecraft, two of which are rectangular and another which is circular. The rectangular coils are mounted on the underside of the 50-watt solar panels and the circular coil is mounted on the exterior of the sealed avionics container. Two fixtures were designed and built to wind the torque coils with a minimum of effort. Construction of a torque coil is shown in Figure 18.

The design of the torque coils was based on the size constraints of their mounting locations and the assumption of using approximately 3 W of power. The diameter of the circular torque coil is 24 inches, as is the

Figure 19: MicroStrain 3DM-GX2 IMU

side length of the rectangular coils. Each coil has 200 turns of 26-gauge copper wire and produces 0.43 N-mm of torque and 16.3 Wb of magnetic flux. The rectangular coils draw approximately 3.2 W of power, while the circular coil draws approximately 4 W due to the smaller loop area.

6.5 Inertial Measurement Unit

The attitude control system utilizes a USB Microstrain 3DM-GX2 IMU for satellite orientation information. The GX2 IMU, Figure 19, is a MEMS based sensor, which incorporates triaxial gyros, accelerometers, magnetometers and temperature sensors into a compact 40 gram package. Although the Microstrain does not outperform more traditional space-rated inertial sensors, the unit is quite attractive due to its low mass, power (90 mA), and cost (less than $2k). The USB interface for the device also allows for data rates up to 250 Hz and easy system integration without additional interface hardware. Although the GX2 unit contains an integrated magnetometer for absolute orientation determination, magnetic interference from satellite components such as the reaction wheels is expected to interfere with magnetometer readings. As a result, the attitude control system only utilizes the rate information from the IMU. Star tracker information is then used at regular intervals to correct for gyro drift and to obtain absolute orientation information.

6.6 Star Tracker

There are four star trackers on the satellite, each of which is comprised of a Lumenera LW230 monochrome machine-vision camera and a FUJINON HF35SA-1 35 mm lens. The star tracker cameras are all connected to and powered by the PC/104 stack via USB 2.0 ports. The camera has resolution of 1616x1216 pixels, with each pixel represented as an 8-bit value (0 being black and 255 being white). The system provides an approximately 11-degree field of view in the horizontal dimension and 8-degree field of view in the vertical dimension. Image processing and attitude determination is performed on the PC/104 stack.

The entire star tracker software package was initially designed, written, and tested in MATLAB and Simulink. This allowed for rapid algorithm development and rapid deployment to test systems. However, execution speed was slow because of the complexity of integrating MATLAB code into C++ software. For this reason, most of the code was rewritten in native C++ except for the image processing, for which the code was auto-generated from Simulink and Real-Time Workshop. In order to identify stars in images, the software contains a star database derived from the 118,218-star Hipparcos catalog [1]. The database contains a list of every star pair within the camera field of view and the angular distance between those pairs. It also contains the inertial position information for each individual star directly from the Hipparcos catalog. In order to keep the star database size small, only stars of magnitude 6.5 or brighter were included[1]. This resulted in a star database containing 8,789 stars and 211,255 star pairs within the camera field of view.

The star tracking process begins when image data is retrieved by the software from the data buffers in the camera. The image is translated into a binary image via a threshold brightness value so that 'on' (bright) pixels are represented by 1s and 'off' (dark) pixels are represented by 0s. The binary image is then searched for 'blobs,' which are just connected groups of 'on' pixels. These blobs represent unidentified stars or other objects such as planets, deep sky objects, other satellites, or noise. The centroids of the blob locations are computed, and a unique pattern recognition algorithm is applied to identify which, if any, stars are represented. During this process, false stars are effectively removed and only repeatedly and uniquely identifiable stars are stored. After stars are identified, another algorithm is applied on their position information to determine the attitude of the satellite. The attitude is computed as a set of Euler angles: right ascension (RA), declination (Dec), and roll. The first two Euler angles are computed by using a linear system that is derived from vector algebra and the information of two identified stars in the image. The roll angle is computed using an iterative method that relies on the information of a single star and the first two Euler angles.

The star tracker system has undergone several tests

[1]Software settings and the camera aperture are adjustable to effectively eliminate dimmer stars from image processing

for robustness and performance. It has been tested in a lab environment on a dynamic one-dimensional demonstration setup (see section 7.4), where it successfully determined attitude accurately from an artificial star field. It has also been tested on image sequences taken from a location with a pollution free night sky, when it was determined experimentally that the precision of the star tracker is approximately 15 arcsec in RA and Dec measurements, and approximately 30 arcsec in roll measurements. Furthermore, in additional testing it was determined that the update rate of a single star tracker is approximately 3 Hz. Since up to three star trackers will be in use simultaneously, the precision and update rates of a single star tracker will be improved.

By using a commercial-off-the-shelf imaging device, removing software processing from the star tracker itself, developing software in-house, cost was greatly reduced. The hardware (camera and lens) costs approximately \$5k , while the labor for software development, integration, and testing costs approximately \$50k, with most of the labor cost stemming from the fixed cost of software development. Since little or no further software development is required for future production, the cost per unit will remain on the order of \$10k, which is approximately 2 orders of magnitude less expensive than commercial star trackers with comparable performance characteristics.

7 Assembly Integration & Test

7.1 Quality Assurance

The approach taken to Quality Assurance (QA), is a key component in the success of COTSAT. This project is a skunk works type development activity where quality is integral to the project development rather than a separate function. Development is made by rapid prototyping and proof through testing while seeking minimal approval from a separate QA organization. Prior to part design or selection the relevant personnel gather and refine the requirements of the hardware and brainstorm solutions that are consistent with the project approach of low-cost, off-the-shelf solutions. The concept is further refined and brought back to the team for comments within minutes or days. Working drawings, software, and document development is captured by a project wide Subversion (SVN) server. Once drawings or engineering documents are of high-fidelity, they are posted in a common lab area for anyone on the project to review. The document control process utilizes a standard Ames Research Center (ARC) numbering convention, requiring project and engineering signatures for approval. The engineering signature is the Independent Technical Authority (ITA) on this project, traceable to the Chief Engineer. Despite minimal involvement from the ARC quality organization, manufactured parts are still fabricated to flight standards and tracked to maintain flight integrity. When purchasing materials, Manufacturer Test Reports (MTRs) and certificates of conformance are required and maintained within the project archive. Materials are kept in bonded stores or in a project area with restricted access. During both assembly and storage, Electrostatic Discharge (ESD) compliance is critical and enforced in the lab, per ARC Management System [2] process requirements. Assembly of critical components are supervised by a qualified engineer with critical tasks being tracked by written assembly and test procedures. Any material or subsystems leaving the lab (e.g. for environmental testing) are tracked closely by project personnel. Any non-conformance is handled immediately and removed from the lab area when appropriate. Articles tested to failure are retained for study.

Verification and validation of components is done on-site at Ames in labs utilizing equipment required and suitable to conduct these tests. Vibration, thermal, vacuum and hydrostatic tests have been performed on components. Additionally, spacecraft component testing has occurred at Air Force Research Lab (AFRL) in New Mexico.

7.2 Ames Manufacturing / Machining

The core philosophy of COTSAT is to build an inexpensive spacecraft using COTS wherever and whenever possible to minimize the number of fabricated parts from raw material. Strict delivery time and budget constraints are a constant concern. Coordination and communication between the project manager, the Systems Engineer (SE), the lead engineer, and the manufacturing team (down to the shop floor personnel) is critical. In response to this, the manufacturing team utilizes, whenever possible, production techniques used in prototype environments. This begins with CAD, where manufacturing decisions are made and incorporated into component design. An example of this is the clamping design found at the *Marmon clamp* interface on each endplate which relies upon a specific undercut typically requiring the part to be removed from one machine and setup on another to create this single feature. By altering the design, these parts can be manufactured on a single machine with reduced setup. The manufacturing team has made a point of involving

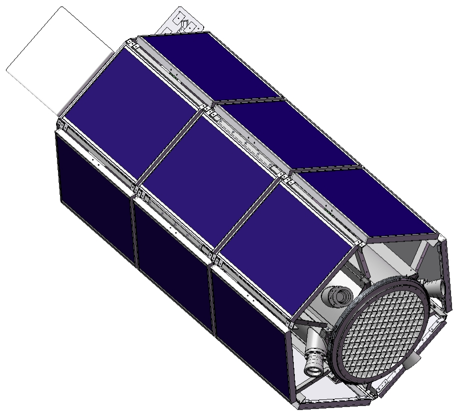


Figure 20: COTSAT Solid Works model

industry partners in the areas of raw material preparation, tooling, and metal removal techniques. Certified materials and standard aerospace manufacturing procedures are used throughout the manufacturing process. COTSAT uses an in-house production traveler and tracking system for hardware being fabricated by the NASA Ames Space Sciences Airborne Instrument Development Lab, the NASA Ames Machine Shop and the Light Metals Fabrication Shop.

7.3 Testing

The COTSAT philosophy to development is heavy on testing but still is to be fast and responsive. COTSAT is implementing an approach to test early and often with the objective being "test as you fly"/"fly as you test" by the end of the project. The focus is on rapidly building/buying hardware, writing software, and testing it as early as possible. Early in the project prototypes and test rigs were built for the reaction wheels, the sealed container, the C&DH system, the star trackers, solar arrays and the electrical power system. This focus is heavy on flat-sat laboratory bench top development tests where the various components are integrated and operated with flight software at various levels of development. This concurrent development process has allowed the team to find bugs early and develop performance profiles of the system as it moves though the development process. Specific tests required to demonstrate performance of higher risk components have been identified and are focused on as early in the development flow as possible which allows any shortcoming to be addressed early in the design cycle. This approach has lead to many interesting findings in developing the solar array configurations, software development, reaction wheel bearing selection and control system design.

COTSAT utilizes a streamlined environmental test flow with two basic phases: Environmental Qualification for newly selected items such as the solar arrays, and full system testing to qualify the system as a whole. The test flow through the assembly and test process as shown in Figure 21. Environmental test criteria are established for COTSAT using the approach outlined in [4]. The Qualification environment is based on an envelope of Falcon-1 and Minotaur Launch environments and the expected orbital environment at our 350-400 km altitude.

7.4 Test Platform / Robot

To date in the development process, the test platform has gone through two generations of design. This first test platform, as shown in Figure 22, allowed testing of a one degree of freedom control system (robot) driven off our input from the IMU and magnetometer outputs to the PC/104 stacks with a single reaction wheel. The findings and lessons learned from the first test platform, enabled the design of the second test platform, which is an optimized protoflight unit. The second test platform, as shown in Figure 23, has an integrated CPC/RPC electrical power subsystem and allows testing of the four reaction wheel control algorithm and the startracker algorithm in a one degree of freedom test setup with a simulated 360 degree starfield. Once course pointing is accomplished using the reaction wheel to turn the platform hung on a string in the lab the systems must do fine pointing with a star tracker. Currently the system uses a Lumenera Camera that does fine pointing by acquiring a star field image in the lab. The camera output is downloaded to the PC/104 stack where optimized Matlab program does analysis on the geometry of the field and determines magnitude as well. Absolute orientation from the star tracker is provided to the control system and is additionally used to correct for gyro drift. The custom Star Tracker code developed at Ames includes the bright star catalog of about 20,000 stars. The integrated control system including reaction wheel, magnetometer, IMU, star trackers is currently thought be capable of 30 arc second pointing capability. The CheapSat Star Tracker system cost to build to date is $17K per system compared to industry units valued at about $1 million.

Figure 24 is an example of the graphical display for the control system as implemented in the first test platform. In the display, a 3D reference graphical model mimics the test platform realtime from a remote wireless connection to the vehicle through MHX-2400 uHard

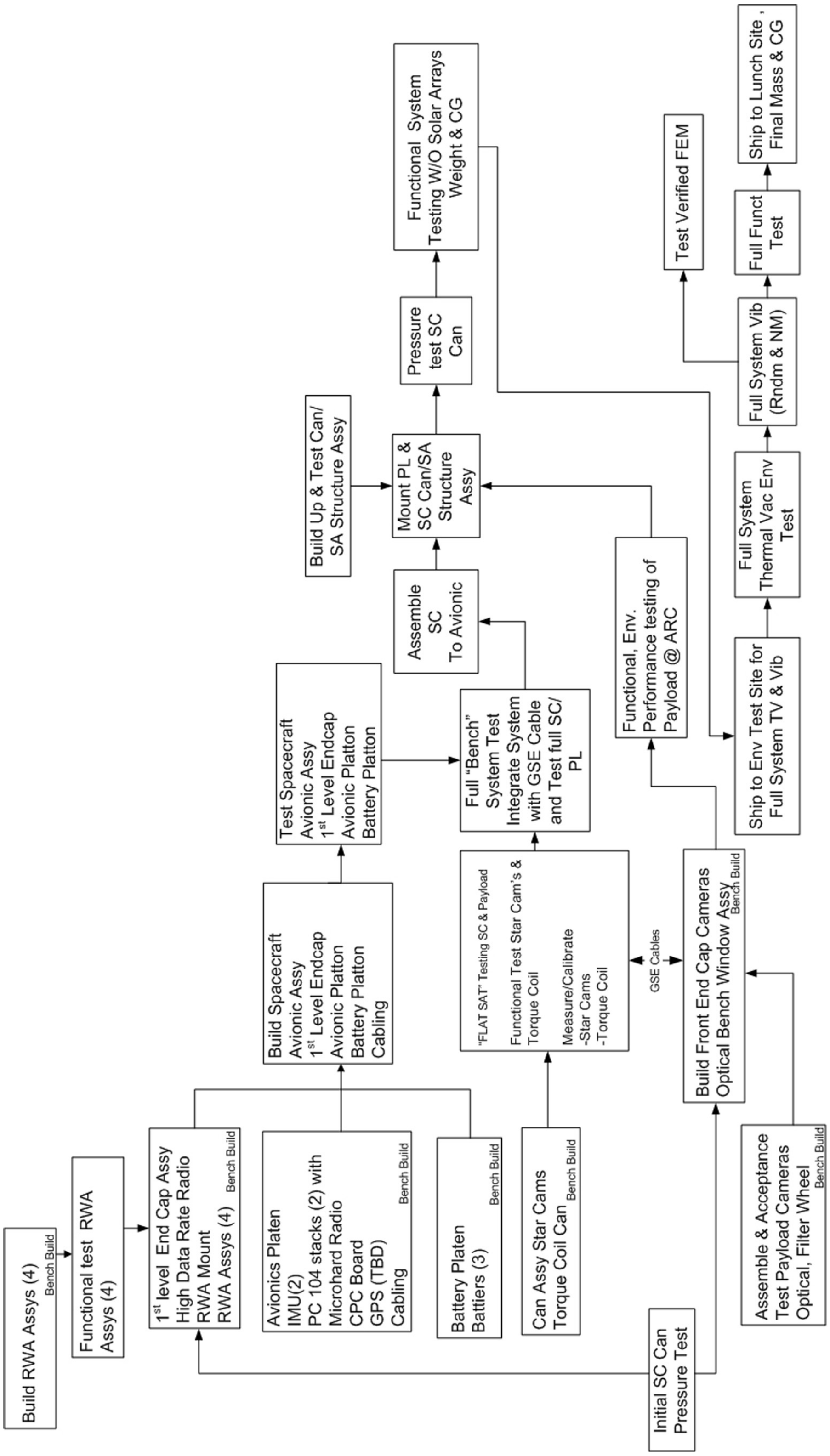


Figure 21: COTSAT Integration Flow

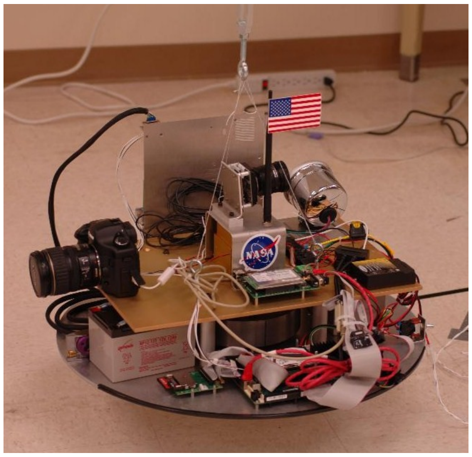


Figure 22: the first test platform

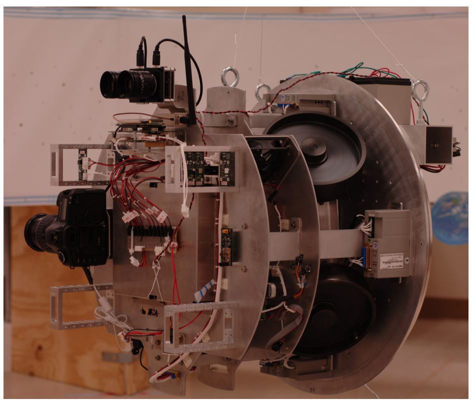


Figure 23: the second test platform

link talking to the ADL855 onboard PC/104 stack. Real time graphical output from the IMU, magnetometer and control system controller graphs is displayed to a remote user. This software development is the first implementation of ground station interface that we will continually develop until launch.

7.5 Mass Budget

The total system mass budget was developed starting with the candidate Launch Vehicle performance and orbit (Falcon-1, LEO) for 400 kg. Adding in a 25% margin (for the preliminary nature of the design) and utilizing 100 kg as the straw man payload allocation yields 200 kg available for the Spacecraft. Selecting masses from vendor data and appropriate assumptions for the structure (eg. .1" wall sealed container, reinforced bulkheads, band clamps, pass-through ports, solar array structure, etc.) yielded a final Mass budget as shown in Table 4.

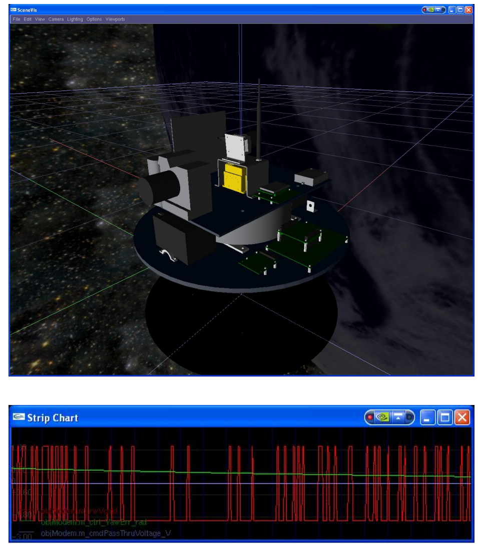


Figure 24: NASA Ames Software for CheapSat Control System

Subsystem	Mass (%)
Bus structure	15
Electrical Power System	38
C&DH	2
Guidance, Navigation	2
Attitude Control	7
Comm.	3
Launch Vehicle Interface	5
Payload Assy (baseline telescope and Focal Plane allocation)	28

Table 4: Mass Budget

8 Payload

8.1 Scientific Mission

The primary payload will be a COTS charge-coupled device (CCD) detector. This detector will be used to collect photometry data from several Extended Red Emission (ERE) targets within our galaxy. ERE is a faint photoluminescence emitted in wavelengths between 540 nm and 900 nm from many different types of astronomical objects, and likely originates from carbon-rich material. Collecting data from these targets will help discover the influence ERE has in determining the radiative balance in the diffuse interstellar medium (DISM), reveal the distribution of carbon in the DISM, and elucidate the role that extra-terrestrial organics may have played in the origin of life.

8.2 Payload Hardware

The camera chosen for this mission is an SBIG ST-9XE. This COTS camera is specifically designed for astronomical imaging, and as a result has very low read noise and dark current. The CCD is thermoelectrically cooled and maintained at a user defined temperature. These aspects, along with its large pixels (20μ) that are optimized for sensitivity and large FOV, make it ideal for the photometry requirements of the mission. The camera will be coupled with a Nikon 85 mm f/1.8 lens, and a CFW-9 filter wheel with four different filters, allowing for in and out-of-band measurements of the ERE features.

The camera will interface with the controller via USB connection. Almost all of the device drivers and software for the Linux environment are completely provided in either the package that came with the camera or from open source software. This drastically reduces the cost of development and simplifies integration. Power to the camera will be provided by an off the shelf converter (specifically made for SBIG cameras) that inputs 12V DC and outputs ±12V DC and +5V DC.

8.3 Scientific Mission Operations

To capture the necessary scientific data, COTSAT will employ five minute exposures with the payload camera. A sufficient signal to noise ratio will be achieved with the combination of 10 images per filter, which results in 40 images per target. This will take between 6 and 12 orbits to complete (less than a day), depending on the RA and Dec of the target. The CCD has a 512x512 pixel array and generates 16 bits per pixel, resulting in 168 Megabits (Mb) of data per target, not including dark frames. The C&DH system will be capable of co-adding these exposures, subtracting darks, and applying flat frames onboard. Performing these operations on orbit will minimize the amount of data to send to about 16 Mb. However, if all raw data is requested it will require three minutes of downlink time to transmit. Table 5 summarizes the data size calculation.

Because the field of view of the camera is so large (6.8 degrees), pointing requirements are minimal. The 30 arcsecond precision provided by the ADACS will be more than sufficient to capture the prescribed ERE target. The amount of jitter that is expected during pointing (≤ 15 arcseconds) may cause some smear in the image, however at an angular resolution of 48 arcseconds per pixel, this motion will not be detrimental to the data that is collected.

Item	**Description**	**Data (Mb)**
Exposure	512x512 image, 16 bits/pixel	4.2
Dark Frame	512x512 image, 16 bits/pixel	4.2
Totals per Target		
Raw	40 Exposures	168
Processed	4 Images (1 per filter)	16.8

Table 5: Payload data rate

9 Launch

A first prototype of COTSAT is proposed to launch in March 2009, on a SpaceX Falcon-9 launch vehicle. The vehicle will launch out of Kennedy Space Center (KSC), Cape Canaveral, into a 300km circular orbit with an inclination of 34.2°. A 4000 kg mass simulator will be used as a ballast for the first launch. The projected lifetime of the spacecraft at this altitude and inclination is about 50 days. Scheduling of the potential targets will be done on the ground and uploaded as commands to the spacecraft. Targets selection will be based on target availability and scientific interest. Future versions of COTSAT can be adapted to fit in a Falcon-1 fairing, thereby increasing the number of launch opportunities.

10 Conclusions

The aim of the Low Cost Rapid Response System (LCRRS) Project, COTSAT, is to develop a rapid prototype of an inexpensive spacecraft, using off-the-shelf equipment and heritage archtecture. The use of a single-atmosphere sealed container allows cost-effective implementation of the avionics bus and supporting subsystems. Through the in-house development (GOTS) of the single-atmosphere spacecraft structure, reaction wheels, star trackers, torque coils, electrical power system and software, the COTSAT team has achieved a cost reduction in excess of one order of magnitude relative to the industry standard. The modification of COTS hardware (MOTS), such as the solar arrays, servo motor drive electronics, battery charge control circuitry, and payload camera system, has greatly reduced the cost of the spacecraft subsystems. Other COTS subsystems have been implemented with no modification, such as the Microstrain IMU, SpaceQuest radio, MicroHard radio, PC/104 computer, and the magnetometer. The spacecraft objective is to accommodate low cost access to space for variable remote sensing payloads with a standard avionics interface to the payload. The architecture allows for future expansion including possible biological payloads. The first rendition of COTSAT has a targeted launch date of March, 2009 on a SpaceX Falcon-9 launch vehicle.

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