QB50
Critical Design Review
Data Package Document 1

Ex-Alta 1 Design Overview Report

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Ex-Alta 1 Design Overview, QB50 Critical Design Review

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Acronyms:

The following acronyms are used throughout this document and are presented here to allow the reader easy reference.

4S1P	**4** Series **1** **P**arallel
ADC	**A**nalog-**D**igital **C**onverter
ADCS	**A**ttitude **D**etermination and **C**ontrol **S**ubsystem
CARISMA	**C**anadian **A**rray for **R**ealtime **I**nvestigations of **M**agnetic **A**ctivity
CCSDS	**C**onsultative **C**ommittee **for** **S**pace **D**ata **S**ystems
CGSM	**C**anadian **G**eospace **M**onitoring
CMOS	**C**omplementary **M**etal **O**xide **S**emiconductor
COCOM	**C**oodrinating **C**ommittee for **M**ultilateral **E**xport **C**ontrols
COTS	**C**ommercial **O**ff **T**he **S**helf
CSA	**C**anadian **S**pace **A**gency
CSP	**C**ubeSat **S**pace **P**rotocol
DAC	**D**igital **A**nalog **C**onverter
DFGM	**D**igital **F**lux**G**ate **M**agnetometer
DLR	German Aerospace Centre
EPS	**E**lectrical **P**ower **S**ystem
Ex-Alta 1	**Ex**perimental **Alberta**n #**1** Satellite
FEA	**F**inite **E**lement **A**nalysis
FEC	**F**orward **E**rror **C**orrection
FM	**F**requency **M**odulation
FPP	**F**light **P**reparation **P**anel
GPS	**G**lobal **P**ositioning **S**ystem
ICD	**I**nterface **C**ontrol **D**ocument
IF	**I**ntermediate **F**requencies
ISIS	**I**nnovative **S**olutions **I**n **S**pace

LNA	**L**ow **N**oise **A**mplifier
MEMS	**M**icro **E**lectro**m**echanical **S**ystem
MNLP	**M**ulti **N**eedle **L**angmuir **P**robe
OBC	**O**n **B**oard **C**omputer
OBDH	**O**n **B**oard **D**ata **H**andling
PCB	**P**rinted **C**ircuit **B**oard
RBF	**R**emove **B**efore **F**light
RF	**R**adio **F**requency
SRF	**S**atellite **R**eference **F**rame
SRP	**S**atellite **R**eference **P**oint
SSC	**S**urrey **S**pace **C**entre
STS	**S**tructural **S**ubsystem
TRL	**T**echnology **R**eadiness **L**evel
UHF	**U**ltra **H**igh **F**requency
U of **A**	**U**niversity of **A**lberta
UTC	**C**oordinated **U**niversal **T**ime
VOC	**V**oltage **C**ontrolled **O**scillator
WOD	**W**hole **O**rbit **D**ata

References:

Ref. No	Document Name	Document Info
[R-01]	NanoMind Datasheet	Gomspace datasheet for NanoMind - http://gomspace.com/documents/GS-DS-NM712C-1.1.pdf
[R-02]	NanoHub Datasheet	Gomspace datasheet for NanoHub - http://gomspace.com/documents/GS-DS-NANOHUB-1.3.pdf
[R-03]	NanoCom Datasheet	Gomspace datasheet for NanoCom - http://gomspace.com/documents/GS-DS-U482C-5.0.pdf
[R-04]	NanoPower Datasheet	Gomspace datasheet for NanoPower - http://gomspace.com/documents/gs-ds-nanopower-p31u-8.0.pdf
[R-05]	Surrey ADCS Datasheet	QB50 datasheet for Surrey ADCS - https://www.qb50.eu/index.php/tech-docs/category/2-adcs
[R-06]	Telemetry Definition List	Available as a companion PDF – Telemetry_Definition_List.pdf

1. System Overview

1.1. General Description

The Experimental Albertan #1 (Ex-Alta 1) Cube Satellite, being prepared for participation in the QB50 mission, is a 3U CubeSat flying the Multi Needle Langmuir Probe (MNLP) payload from QB50 as well as a Digital Fluxgate Magnetometer (DFGM) designed by the University of Alberta (U of A) space physics department. The design borrows heavily from the standard 2U GomSpace QB50 package adapted to a 3U structure to provide an additional unit for the DFGM payload and deployable boom. All components except the two payloads and magnetometer boom are commercial off the shelf components (COTS), and GomSpace provides a tested, flight proven, and fully integrated platform with which to fly the two payload packages seamlessly and safely.

1.2. Subsystem Overview

Every subsystem, aside from the payloads, on Ex-Alta 1 are composed of COTS hardware with space heritage. Since GomSpace provides a complete satellite core, every subsystem is already well suited to interface with the rest of the satellite and interface issues are kept to a minimum. Every GomSpace system will communicate with a robust I2C protocol utilizing the CubeSat Space Protocol (CSP) to provide fast, simple, and dependable command and data transfer. This section briefly summarizes each subsystem and the hardware maturity of their components.

1.2.1 Structure

The structure of Ex-Alta 1 is a 3U CubeSat Structural Subsystem (STS) manufactured by Innovative Solutions In Space (ISIS). The STS is composed of two separate frames connected together with trusses (called ribs by ISIS). The ribs connect the two frames and are placed at the edges of each defined unit. The ribs also provide a structure onto which the electronics boards, which use the CubeSat PC104 form factor, may be mounted in a stack. Between each unit, the ribs also provide mounting for interstage panels which hold electric knives for antenna/probe release and instrumentation for the Attitude Determination and Control Subsystem (ADCS). The STS parts are made from Aluminum (AL) 6082 and the side frames are black anodized as per the CubeSat specification. The anodization of the side frames prevents the rails from cold welding to the deployment pod. The threaded holes in the frames, the ribs and the side shear panels are all alodyned in order to maintain conductive pathways to ensure the satellites ground charge is constant.

1.2.2 Power

The power system includes a P31uS NanoPower electrical power system (EPS) with latch-up protected switches. The battery pack (GomSpace BP4) is composed of 4 Lithium Ion (Li-ion) 18650 space proven batteries in a 4 series 1 parallel (4S1P) formation with 39 WH capacity. The solar panels are equipped with Azurspace triple junction Gallium Arsenide (GaAs) 30% efficiency cells. All power system hardware elements are space proven COTS manufactured by Gomspace.

1.2.3 On Board Computer (OBC) and On Board Data Handling (OBDH)

Ex-Alta 1 uses the NanoMind A721D as an OBC, which uses an ARM7 processor with two I2C busses, one 4Mbyte Data Flash and a 2 GByte SD Card. A dedicated board named the NanoHub is the flight preparation panel (FPP) and burn wire release controller for the satellite. The NanoMind-NanoHub combination provides one of the best solutions for cube satellite command and data handling due to the hardware having space heritage, in addition to a space proven software library and real time operating system supplied standard with the NanoMind.

1.2.4 Attitude Determination and Control System

Ex-Alta 1's ADCS is entirely contained within the Surrey Space Centre (SSC) ADCS that was offered to teams participating in the QB50 mission. The system utilizes a pair of complementary metal oxide semiconductor (CMOS) cameras operating as Sun/Nadir sensors, in conjunction with rate gyros, coarse photodiode sun sensors, and a magnetometer, in order to determine the spacecraft's relative orientation in its orbit. To provide further control, the system features a single reaction wheel, acting as a momentum wheel, and 3 magnetorquers oriented in three orthogonal directions aligned with the CubeSat reference frame.

1.2.5 Communication

Communication is based on NanoCom U482C Half duplex Ultra High Frequency (UHF) transceiver outputting ~1.0 W radiated power. The transceiver supports up to 9600 baud, with Consultative Committee for Space Data Systems (CCSDS) framing and forward error correction (FEC). The antenna is a four-rod omnidirectional turnstile antenna that is secured with burn wire against Ex-Alta 1 prior to deployment and deploys via torsion springs.

1.2.6 Thermal Control

A 2-D nodal thermal model was developed to calculate the thermal gradients that will be encountered while in orbit. By employing a 2-D nodal analysis, each surface of the satellite in its respective 10x10x10cm unit forms a node for which the surface temperature is obtained and the various layers between the outer surface and the stack of boards are converted into resistance values for the thermal circuit.

The results of our analysis show that the extreme high and low temperatures of the satellite will remain within the operational temperatures of the temperature critical component, the batteries. The model will continue to be refined and we will nonetheless explore the best strategies for thermal control (passive), such as adding insulation or heat pipe, or using thermal coatings to keep the temperature of the components within operational temperatures, including a margin of at least ±5 °C.

1.2.7 GPS

A NovAtel OEM615 global positioning system (GPS) is used as a daughter board of the Surrey ADCS system with the Taoglas AP10.F GPS antenna mounted on one of the interstage panels. The GPS is based on space proven hardware with Coordinating Committee for Multilateral Export Controls (COCOM) limit removal.

1.2.8 Hardware Summary

The following list summarizes all hardware elements on Ex-Alta 1, their manufacturer, the Technology Readiness Level (TRL) and any notes concerning their space heritage.

Table 1-1: Hardware Overview Table

Element	*Hardware*	*Manufacturer*	*TRL*	*Notes*
Structure	3U STS	ISIS	9	Space Heritage in 2013
Structure	Interstage	Gomspace	9	Space Heritage on GOMX-1
Structure	FPP	Gomspace	9	Space Heritage on GOMX-1
Structure	Deployable Boom	U of A/DLR	4	Deployment tested in a parabolic flight
OBC	NanoMind A712D	Gomspace	9	Space Heritage on PWSAT, ArduSat-1, ArduSat-X, CubeBug-1, Triton-1, GOMX-1
OBDH	NanoHub	Gomspace	9	Space Heritage on ArduSat-1, ArduSat-X, GOMX-1
EPS	NanoPower P31us	Gomspace	9	Space Heritage on FUNcube, ArduSat-1, ArduSat-X, Triton-1, GOMX-1
EPS	BP4	Gomspace	9	Space Heritage on ArduSat-1, ArduSat-X, GOMX-1
EPS	Solar Panels	Gomspace	9	Space Heritage on ArduSat-1, ArduSat-X, GOMX-1
Comm	NanoCom U482C	Gomspace	9	Space Heritage on ArduSat-1, ArduSat-X, Xatcobeo, Humsat-D, GOMX-1
Comm	ANT430	Gomspace	9	Space Heritage on ArduSat-1, ArduSat-X, GOMX-1
GPS	NovAtel OEM615	NovAtel	9	Based on NovAtel OEM-V with space heritage on RAX-2
GPS Antenna	Taoglas AP.10F	Taoglas	9	Recommended by Surrey Space
ADCS	Surrey Space QB50 ADCS	Surrey Space	9	Heritage expected before launch date
Payload	MNLP	UiO	6	Space heritage expected Q2/2014
Payload	DFGM	U of Alberta	6	TRL 8 expected 10/2014

1.3. Satellite Reference Point (SRP) and Frame (SRF)

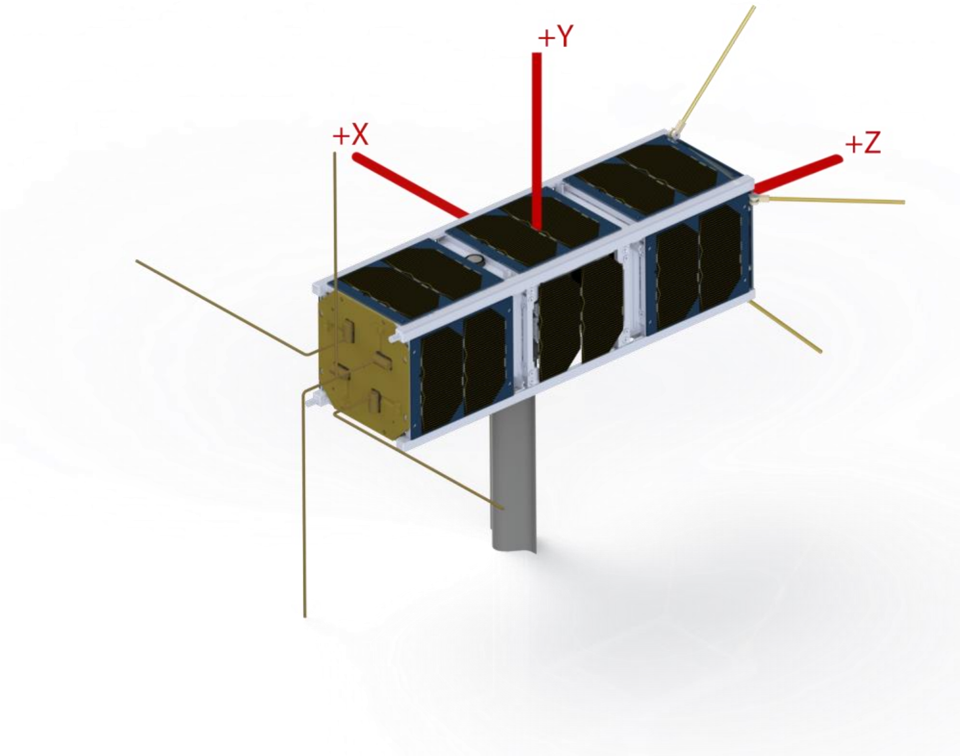


Figure 1-1: Satellite Reference Frame

1.4. System Diagrams

1.4.1 *Electrical Architecture Diagram*

The following diagram represents the Ex-Alta 1 Satellite as a schematic summarizing the digital and electrical interfaces between the elements.

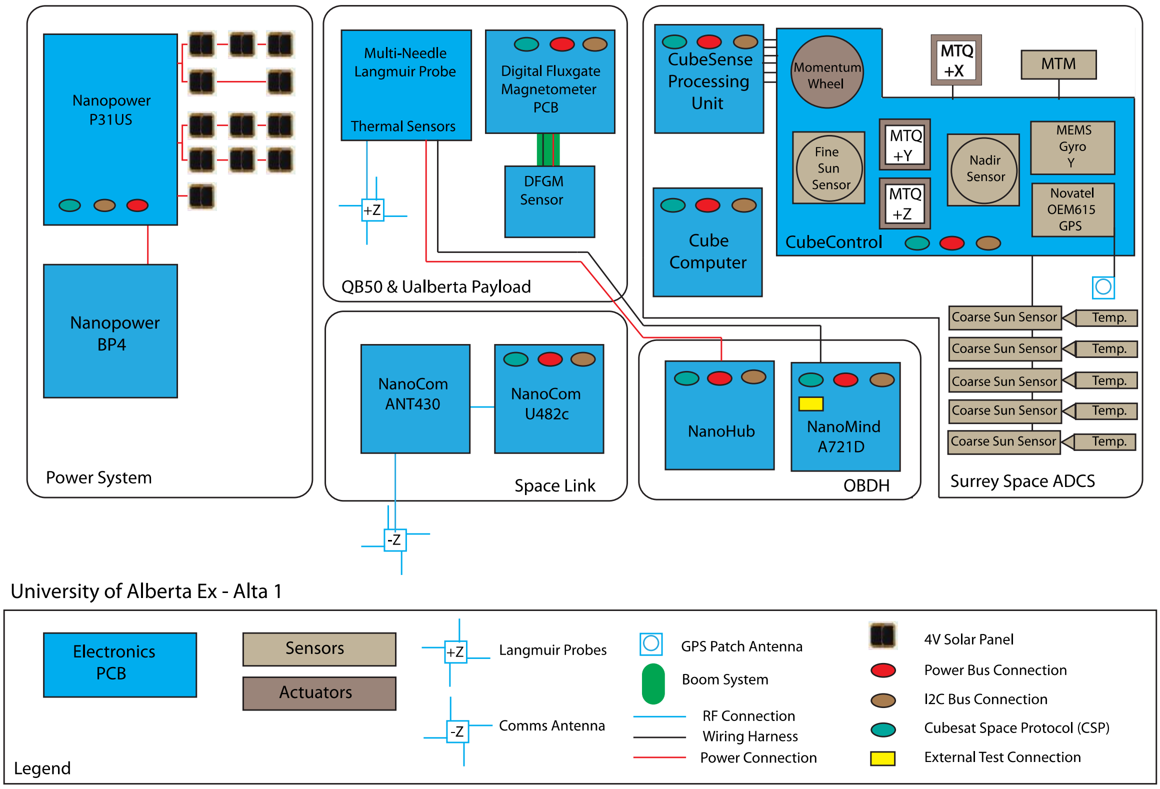


Figure 1-2: Electrical Schematic of Ex-Alta 1

1.4.2 *PC104 Stack Layout*

The following images detail the PCB stacks in the ram and anti-ram sections of the satellite.

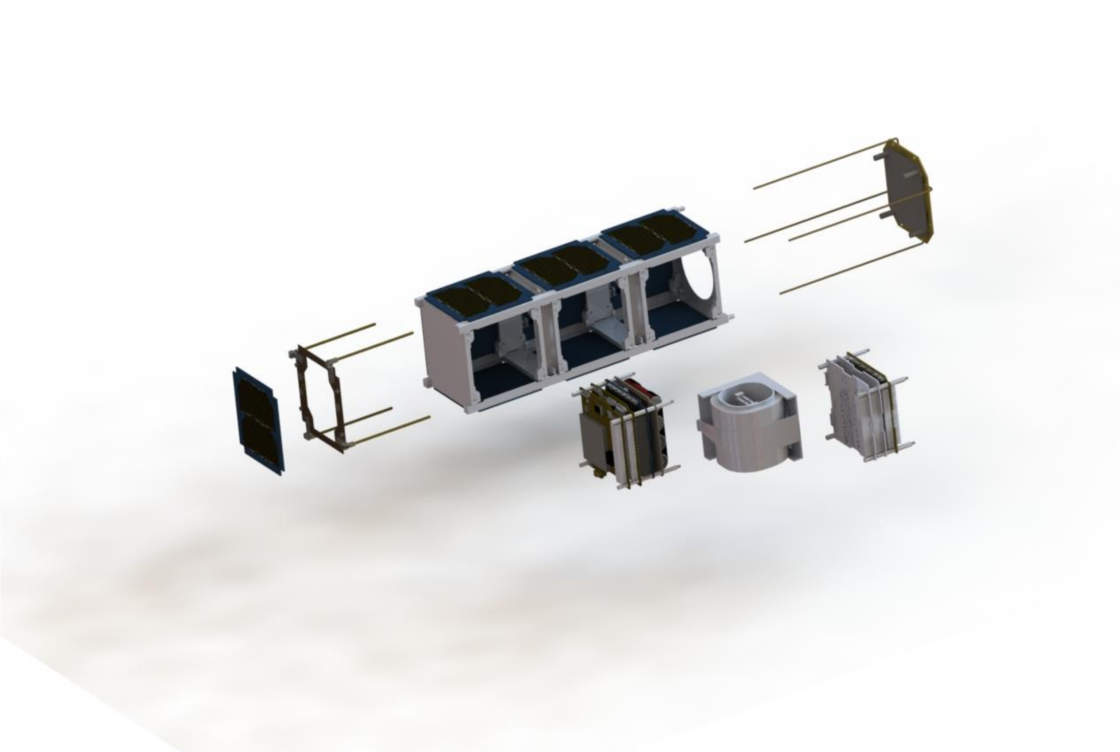


Figure 1-3: Exploded diagram of satellite units.

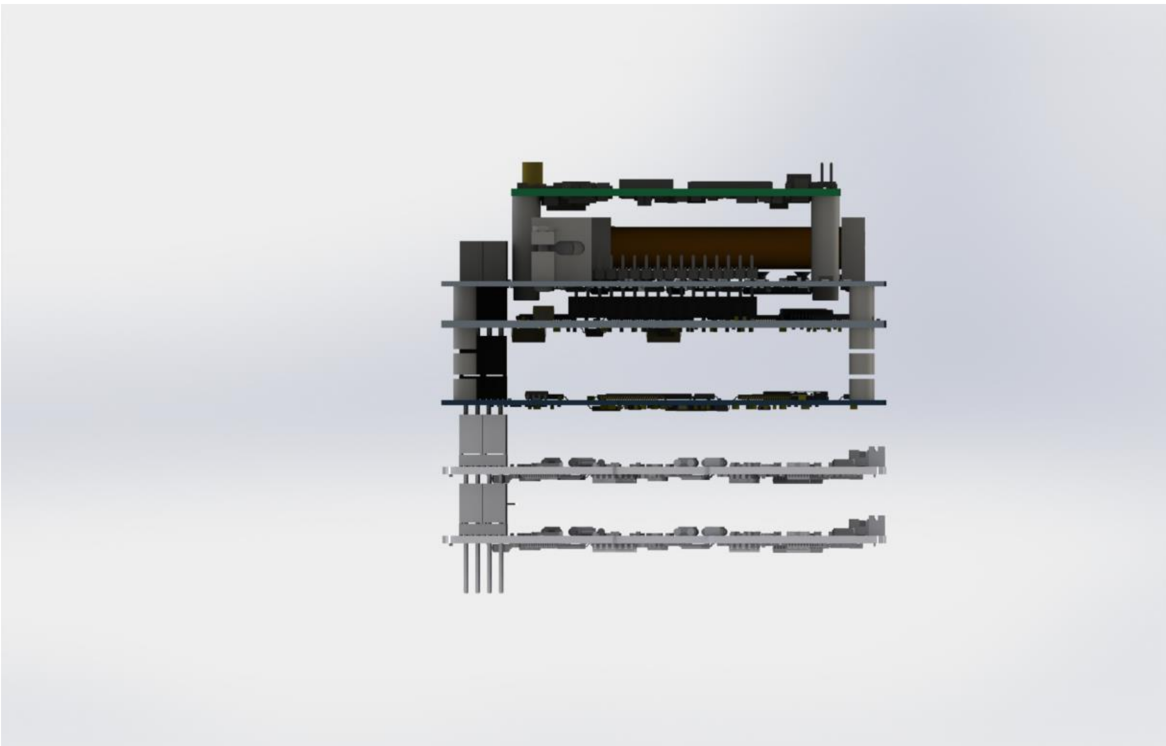


Figure 1-4: Ram unit PCB stack.

Figure 1-5: Anti-ram unit PCB stack.

1.4.3 PC104 Pin Assignment

The following section describes the pin assignment on the PC104 bus throughout the system.

Final					
	H1 Odd	H1 Even		H2 Odd	H2 Even
1,2	CAN				
3,4	CAN				
5,6					
7,8					
9,10					
11,12					
13,14					
15,16					
17,18					
19,20					CubeSense
21,22	ADCS I2C				
23,24	ADCS I2C				
25,26				5V0	5V0
27,28				3V3	3V3
29,30				Gnd	Gnd
31,32		EPS Chrg		AnaGnd	Gnd
33,34	RX OBC				
35,36	TX OBC				EPS RX
37,38					EPS TX
39,40					
41,42	Sys I2C				
43,44	Sys I2C				
45,46				Vbatt	Vbatt
47,48	5V0	3V3			
49,50	5V0	3V3		5V0	
51,52	5V0	3V3		3V3	

Table 1-2: CubeSat Bus Pin Out

Green	Anti-Latch Configured by NanoHub
Red	Only Used on One Board, Not Used
Blue	Anti-Latch Configured by NanoPower
Grey	Standard CSK - Not Used
Yellow	Standard CSK - Used
Pink	Surrey ADCS Pins

Table 1-3: Pin Out Table Legend

	Power Allocations for Blue Cells			Power Allocations for Green Cells	
H1-47,48	ADCS	ADCS			
H1-49,50	-	GPS, Mind, Hub	H2-49	DFGM, mNLP	
H1-51,52	BP4	Com	H2-51	DFGM, mNLP	

Table 1-4: Power Allocation

Power Switching For Each Board	
Board	**Power Source**
NanoHub	H1-50 for 3V3, H2-29,30,32 for Gnd, 2 switched outputs
NanoMind	H1-50 for 3V3, H2-29,30 for Gnd
Magnetometer	Uses H2-49 for 5V0, H2-51 for 3V3, H2-29,30 for Gnd
QB50 m-NLP	H2-49 for 5V0, H2-51 for 3V3, H2-29,30,32 for Gnd
NanoCom	H1-52 for 3V3, H2-46 for V_TX, H2-29,30,32 for Gnd
NanoPower	Supplies Power - 6 switched, 3 not switched
BP4 Battery	Heater uses H1-51 for 5V0
Surrey ADCS	H1-47 for 5V0, H1-48 for 3V3, H2-29,30,32 for Gnd
GPS	H1-50 for 3V3, H2-29,30,32 for Gnd

Table 1-5: Power Switching

1.5. Concept of Operations

The concept of operations for the satellite can be summarized as a flow chart as shown in Figure 1-3: Satellite Concept of Operations Flow Chart.

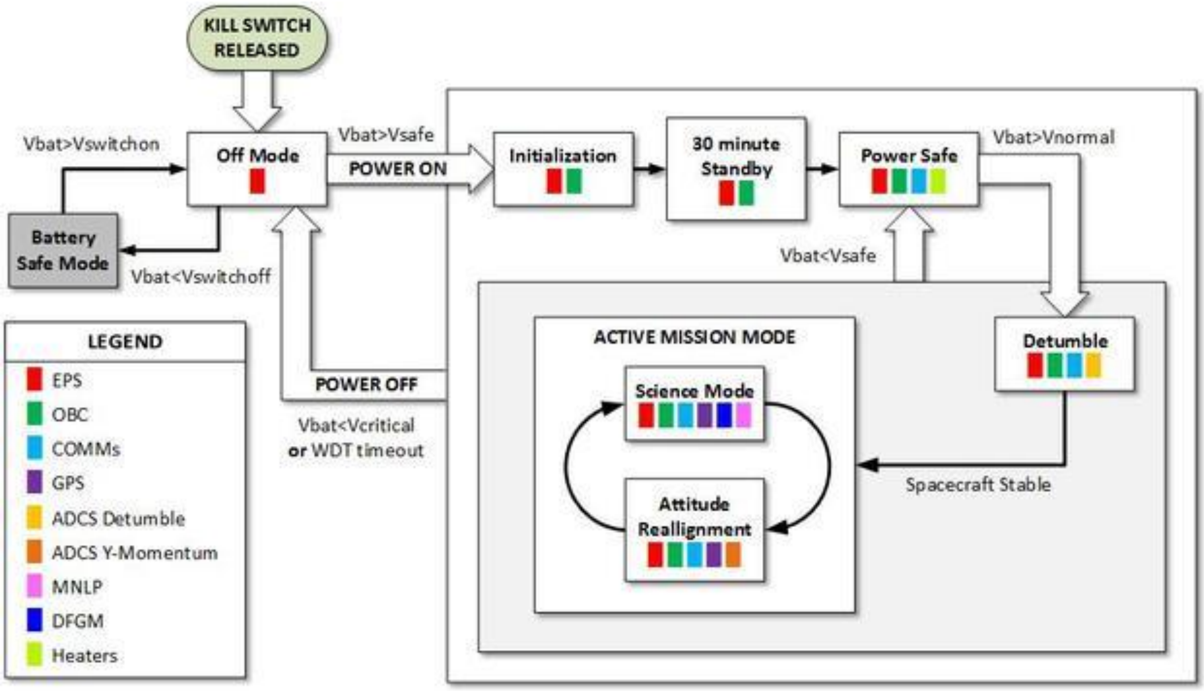


Figure 1-3: Satellite Concept of Operations Flow Chart

The values of Vnormal, Vsafe, and Vcritical are defined such that Vcritical is less than Vsafe, and Vsafe is less than Vnormal. Vswitchon and Vswitchoff are binary on and off voltages marking a connection or disconnection of the battery from the power system. These values are configurable and described later on in the power section. Note that deployment of antennas and probes occurs only after the 30 minute stand-by period.

1.6. Spacecraft Mode Description

While the spacecraft is space-borne, it will operate in one of 6 modes. Note that standby mode only occurs during the initial phase of the mission, and initialization is a short mode performed between off and standby to check for errors. Off, Power Safe, Detumbling, and Active Mission have defined power budgets. Each of these modes is briefly described below in terms of what must be accomplished from a software point of view:

1.6.1 Off

In off mode, all systems are off except for the EPS and the EPS will attempt to restart the system if the battery voltage is more than Vsafe. Restarting the system means giving power to the flight computer and switching to the initialization state. The system will turn off as long as a kill signal is received from the FPP, whenever the battery voltage falls below Vcritical, or after a heartbeat timeout from the flight computer. If the battery voltage falls below Vswitchoff, the EPS will disable all outputs and most electronics on the EPS itself leaving only the ability to charge the battery from the solar cells until the battery voltage rises above Vswitchon.

1.6.2 Initialization

When the initialization mode is evoked, the flight computer and peripherals are recovering from a period of inactivity and diagnostics will be performed to assess the health of the spacecraft. Data corruption detection will be performed where possible and housekeeping data will be obtained and stored for later transmission. This will operate serially and once complete, the spacecraft shall exit initialization mode and enter standby mode.

1.6.3 Standby

When initialization is complete, the spacecraft shall enter standby mode. In this mode, the spacecraft will wait for at least 30 minutes. This provides the 30-minute dormant period following ejection from the launch vehicle required by QB50. Timing shall be kept by the known clock-rate of the flight computer, this is because the GPS has not yet been activated to synchronize with Coordinated Universal Time (UTC). At the end of the 30-minute standby period, the GPS will be pinged to obtain telemetry and synchronize to UTC and the spacecraft will enter detumbling mode.

1.6.4 Detumbling

After the spacecraft has completed the 30 minute standby period, the spacecraft will initiate detumbling mode. In this mode, the communication systems and ADCS will be activated. The spacecraft will begin transmitting whole orbit data (WOD) and diagnostic information recorded during initialization will be transmitted relative to Coordinated Universal Time (UTC). The ADCS will work to correct the alignment of the spacecraft and stop any spinning or tumbling (spinning or tumbling does not significantly affect the spacecraft's ability to communicate and function but it will interfere with payload measurements). Once the spacecraft is stable, it will exit detumbling mode and enter power safe mode.

1.6.5 Power Safe

Power safe mode is evoked when the spacecraft exits detumbling mode or has insufficient power to remain in active mission mode where insufficient power is indicated by a battery voltage below Vsafe. While in power safe mode, the spacecraft will continue to transmit and receive WOD and battery heaters will be activated. If the battery voltage falls below Vcritical, the spacecraft will exit power safe mode and enter off mode. If the battery voltage goes above Vnormal, the spacecraft will exit power safe mode and enter active mission mode.

1.6.6 Active Mission

While in active mission mode, the spacecraft will collect housekeeping parameters and cycle between collecting payload measurements (science mode) and realigning its attitude (attitude realignment mode) so that the MNLP payload points along the ram direction and the magnetometer boom points towards the center of the Earth. Collected information will be stored on the MicroSD card within the flight computer along with a time stamp. When a communication opportunity is reached, the communication system will retrieve information from the flight computer and attempt to transmit it to a ground station. Once the flight computer receives confirmation that a data packet has been received by a ground station, the data will be erased from the MicroSD card to clear room for the next round of science data. If the battery voltage falls below Vsafe, the spacecraft will exit active mission mode and enter power safe mode.

1.7. Spacecraft Mode Determination

Figure 1-4 describes the hierarchy for determining the mode of the spacecraft. Three levels are present, with the EPS controlling the highest level and the OBC controlling the lowest level. This arrangement is necessary to ensure the health of the satellite and validity of scientific data.

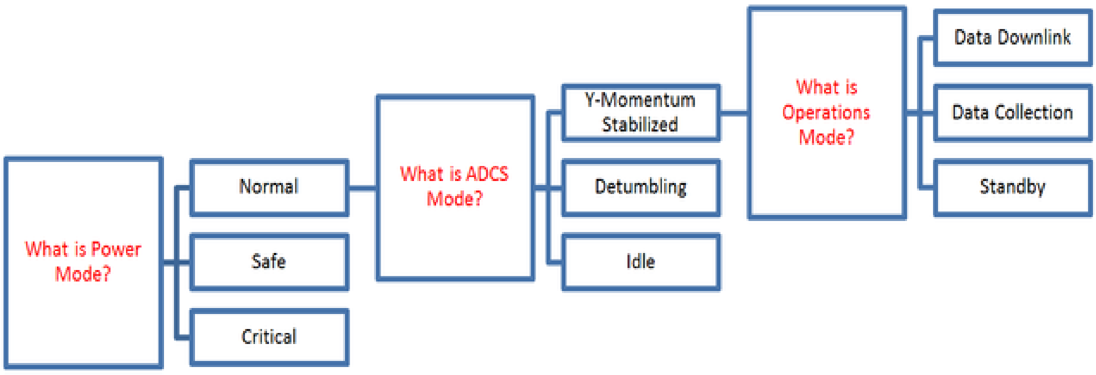


Figure 1-4: Satellite Mode Determination Hierarchy

2. Payload Design

The chosen payload for Ex-Alta 1 is the MNLP, developed by University Of Oslo, which gives high time resolution measurements (up to 10 kHz sampling rate) of absolute electron density and spacecraft floating potential.

Current measurement range of the MNLP uses 3 decades (i.e. 1 nA to 1 μA), but adjustable by in-flight automatic gain control, the electron density ranges of 108 m^{-3} to 1012 m^{-3} (adjustable to match mission requirements).

The accuracy is based on 24 bit raw data, but down-sampled to 10, 12 or 16 bit data product. Sampling rates up to 10 kHz, but fully adjustable.

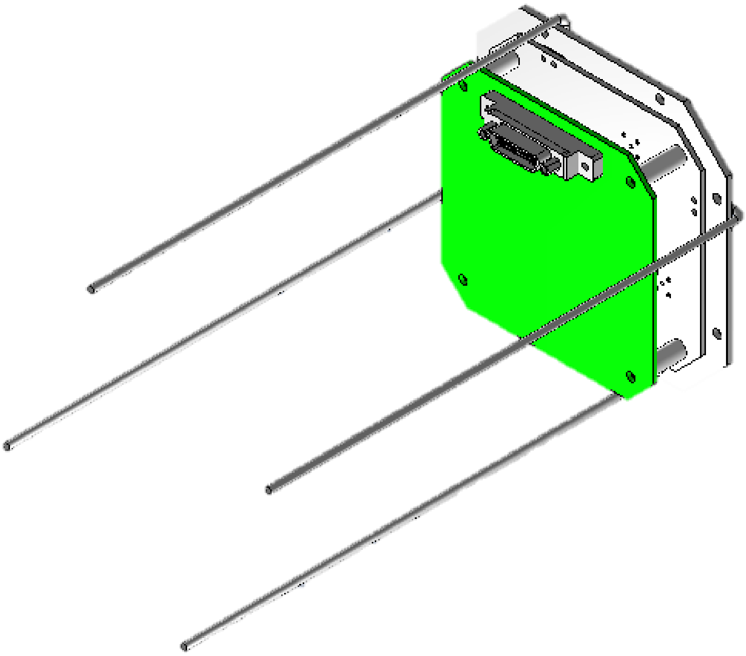


Figure 2-1: MNLP Payload (electronics and needles)

2.1. MNLP boom system

The MNLP boom system consists of four separate booms, mounted on the common top plate of the STS. Each of the booms has an individual deployment mechanism, operated by the MNLP printed circuit board (PCB) on command from the OBC.

The probes will be held down to the side panels using the interstage panels, in the same manner as the antenna rods. The probes will be released with the antenna rods using burn wires in addition to the release mounted on the MNLP payload face.

2.2. MNLP Electron Emitter

An electron emitter shall be implemented in the center of the MNLP boom system Figure 2-2, to get rid of collected electrons that drive the CubeSat floating potential below functional limits when the satellite is in eclipse where photoelectron emission is not possible. The design of this Electron emitter is currently in progress, and preliminary results show very promising results with emission currents as high as 20 µA - >5 times higher than the maximum amount of collected current by positively biased MNLP probes. The emitter will be connected to the chassis via ground to provide a solid conducting connection to remove excess negative charge from the satellite body.

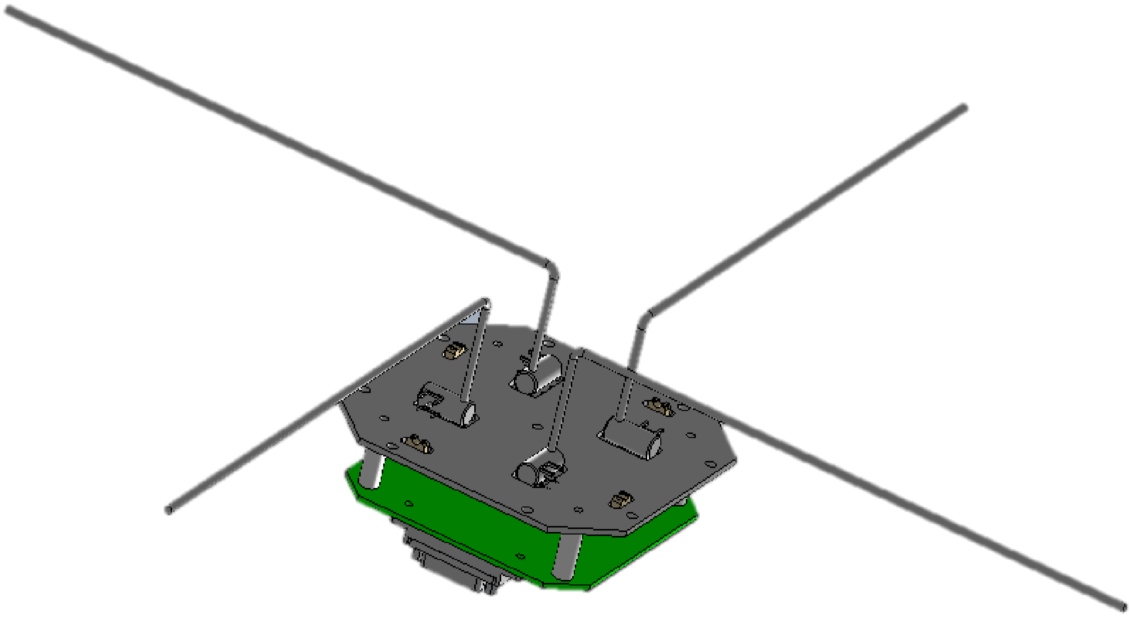


Figure 2-2: MNLP Science Unit Boom System

2.3. Digital Fluxgate Magnetometer (DFGM)

The new U of A DFGM is a compact, temperature compensated, modern science instrument which can make low-noise measurements of the magnetic signatures of space weather current systems. The instrument uses a novel approach to extend the sensitivity from the typical 1H to the kHz range to study higher frequency waves.

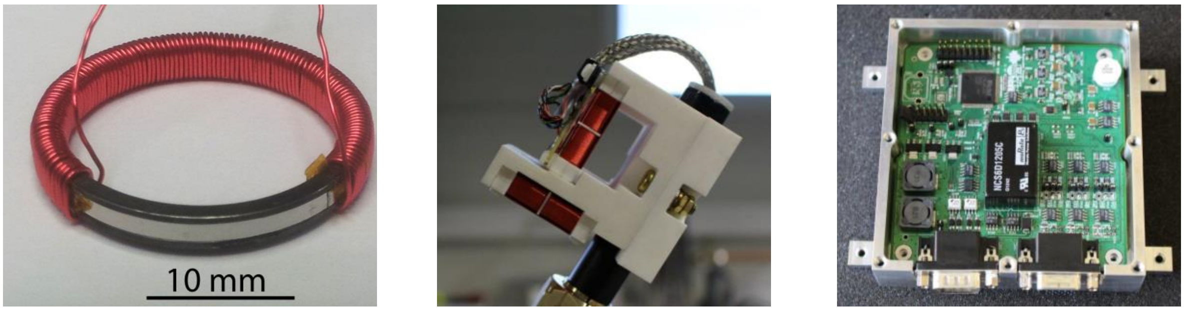


Figure 2-3: DFGM sensing element (left), boom mounted sensor (middle), and electronics (right).

The DFGM will be based on a magnetometer developed by U of A PhD student David Miles which is scheduled to fly on the Norwegian ICI-4 sounding rocket program in October 2014. The DFGM will use a miniaturized version of the ICI-4 sensing element (**Figure 2-3** left) to create a volume optimized sensor based on that developed for ICI-4 (**Figure 2-3** middle). An optimized version of the ICI-4 magnetometer electronics package (**Figure 2-3** right) is being created for the cube-satellite and is expected to occupy a single PC-104 card. The DFGM sensor will be deployed away from the cube-satellite by an ultra-high expansion coiled boom developed and provided by the German Aerospace Centre (DLR).

The DFGM is based originally on the Narod Geophysics Ltd. instrument deployed extensively by the U of A in the Canadian Array for Realtime Investigations of Magnetic Activity (CARISMA) ground network as part of the Canadian Space Agency (CSA) Canadian Geospace Monitoring (CGSM) program. The DFGM is based on a design developed for the ICI-4 sounding rocket which incorporates updates from the Cassiope/ePOP satellite MGF payload, during Phase A and A2 of the ORBITALS satellite mission, and during Phase A of the PRIMO secondary science payload on the CSA PCW mission. Collectively this CSA investment led to the current digital instrument with high frequency (kHz) sensitivity and a radiation hard upgrade path. This design is also the basis of four new Canadian stations as part of the recently funded CSA Geospace Observatory (GO) Canada program. The DFGM will be optimized for accommodation on the Ex-Alta 1 cube satellite to maximize the science return. The flight of the DGFM will achieve *"System prototype demonstration in a space environment"* (TRL-7) providing significant credibility and risk-reduction for future satellite applications.

Significantly, the miniaturized DFGM brings the science magnetometer along the TRL development path towards a commercial instrument which can be flown and marketed to the rapidly expanding cube satellite market. Specifically, the performance of the DFGM in the laboratory greatly exceeds that of commercial sensors - a space qualified miniaturized sensor of this type would enable future cube satellite missions to use the sensor for not only the standard cube satellite attitude determination - but also to complete science quality measurements on cube satellite platforms. For example, the attitude determination and control system being developed by the Surrey Space Centre (SSC) for QB50 includes a boom mounted magnetometer and which Surrey intend to develop for the expanding cube satellite market. Developing a Canadian miniaturized magnetometer offers the basis entering this market place with a sensor whose sensitivity and frequency range exceed other instruments which are currently available in the market (e.g., from ZARM Teknic).

The magnetometer is mounted on a deployable coil boom that will serve two important purposes:

1. To remove the very sensitive magnetometer from the electric noise within the satellite body and allow for full use of the measurement resolution.
2. To provide a passive attitude stabilization via an increased moment of inertia.

The boom is stored within the middle cube U of the satellite, and is released only when all antennas and booms elsewhere have been successfully deployed and the satellite has detumbled. The magnetometer boom is discussed later in the document.

2.3.1 Instrument Concept

The DFGM is based on the fluxgate payload developed for the ICI-4 sounding rocket mission. This design is in turn based on a concept developed for the Canadian Space Agency ORBITALS radiation belt satellite. A detailed description of the design (and the source of some of the figures) can be found in Miles et al., (2013). Fluxgate magnetometers measure the local static and low frequency magnetic field by modulating the field experienced by the sensor by driving a ferromagnetic core into and out of magnetic saturation. This modulated field creates EMF is a secondary "sense" winding which is determined by the background field. Figure 2-4 shows a block diagram of a single component of the fluxgate sensor.

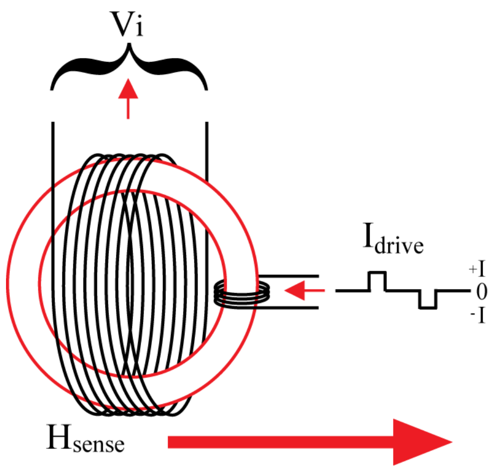


Figure 2-4: Current Idrive forces the ferromagnetic ring core into and out of saturation. This modulates the magnetic field experienced by the larger solendoidal sense winding creating a voltage Vi which measures the background magnetic field.

This sensor uses the established technique of double-winding a ring core to sense two components of the magnetic field using one ring-core. The electronics were designed such that the noise floor of the overall instrument was set by the intrinsic noise of the two ring-cores in the sensor. The X and Y components are derived from a single sense winding on each core while the Z component is derived from sense windings on each sensor core connected in series. Figure 2-5 shows the sensor design schematically. The two dual wound sensor bobbins are mounted on a common block (potentially of PEEK) to minimize temperature sensitivity.

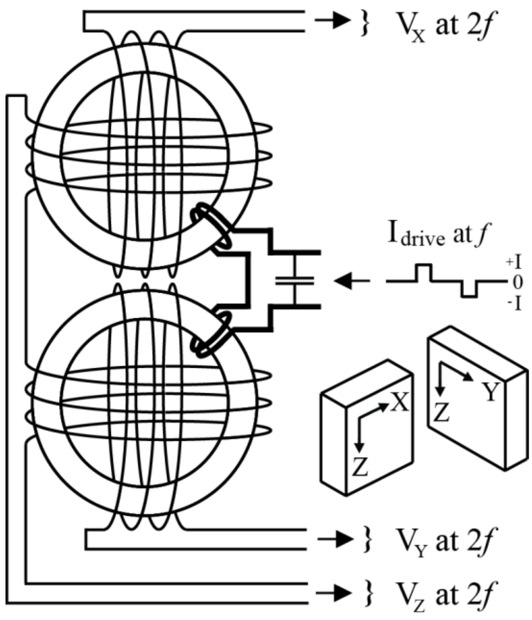


Figure 2-5: The X and Y components are derived from a single sense winding on each core while the Z component is derived from sense windings on each sensor core connected in series. The X, Y and Z windings are all mounted orthogonally to reconstruct the vector magnetic field.

Figure 2-6 shows a single component block diagram of the instrument. Magnetic feedback is used to null the majority of the magnetic field inside the sensor. The measurement of the ambient field is then the sum of the applied magnetic feedback and the measured small residual field in the sensor. An FPGA controller generates a ~28 800 Hz drive signal which is power amplified (PA) and sent into the drive winding to periodically saturate and unsaturated the ring cores. The direction of saturation is alternated to avoid magnetising the core. The modulated core permeability creates a fluxgate signal for each magnetometer component corresponding to the magnetic field strength inside the sensor in that axis. The current output from the sensor is converted to a voltage (I/V) and is sampled by the analog to digital converter (ADC) to become the input to a control loop implemented in the FPGA. The output of the control loop is sent to the digital to analog converter (DAC) and converted into a precise, temperature compensated current (V/I). This negative feedback drives each component of the magnetic field in the sensor head towards zero.

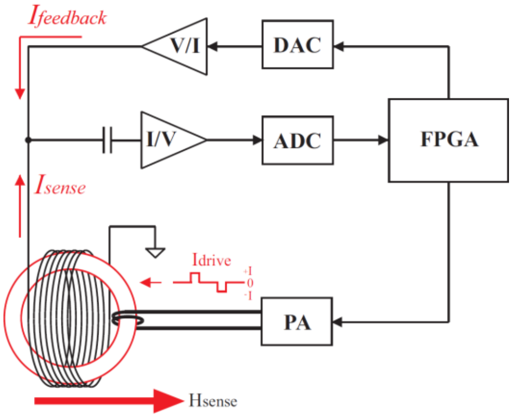


Figure 2-6: Schematic of one fluxgate component showing the major components. The FPGA controller generates a ~28 800 Hz drive signal that is power amplified (PA) and sent into the drive winding to periodically saturate and unsaturate the ring cores. The sense coil is conditioned by a current to voltage converter (I/V) and the captured synchronously by an ADC. The FPGA uses the fluxgate signal as the input to a control loop to drive the magnetic field in the sensor towards zero. The output of the control loop is converted to an analog signal (DAC) and then to a high-precision, temperature compensated current to provide magnetic feedback (V/I).

2.3.2 Instrument Resource and Interfaces

Table 2-1: DFGM specifications

Instrument Parameter	Specification	Comment
Supply Voltage	5V +/- 5%	
Supply Current	< 50 mA @ 5 V Average < 250 mA @ 5 V Peak	
Data/Housekeeping Interface	I2C	3.3 V CMOS, speed and pull-up resistance TBD
Data Product	Custom	100 samples per second per channel. Three channels. 2 x 16 bit words per sample.
Housekeeping Data	Custom	8 x 16 bit housekeeping words. Updated at 100 Hz but intended to be read by the spacecraft at ~ 1 Hz.
Timing Interface	1 PPS Input	3.3 V CMOS, synchronized to falling edge
Sensor Interface	Custom Analog	Connector and pinout TBD. Requires five twisted pairs: Drive +, Drive -, XSense+, XSense-, YSense+, YSense-, ZSense+, ZSense-, Thermo+, Thermo-
Sensor Dimensions	<50x50x50 mm	TBD
Electronics Dimensions	1 PC104 Card	Implements extra requirements imposed by the Cubesat Kit specifications
Sensor Mass	<100g	Exact mass TBD
Electronics Mass	< 125 g	
Harnessing Mass	<100 g	
Survival Temperatures	Sensor: -40 to +85 C Electronic: -40 to +85 C	
Operating Temperatures	Sensor: -40 to +85 C Electronic: -40 to +85 C	Sensor: 0 to +40 C recommended

2.3.3 Concept of Science Operations

The DFGM will continually generate three component 100 sps data whenever powered. The spacecraft is responsible to read this data over the I2C interface and process it into selected data products for transmission to the ground.

Within the expected telemetry constraints the telemetered data is expected to be made up of four constituents:

1) Continuous low cadence time series (1 or 10 Hz)
2) Low cadence averaged spectra
3) Filter banks
4) Bursts of full cadence time series

It is anticipated all the raw 100 sps data will be time stamped and placed in a rig buffer in local storage (SDCard). Low data rate products (1) and (2) will be routinely generated and downlinked. Based on (1) and (2) project scientists will select and priorities products (3) and (4) and select time intervals of interest which will be uploaded to the spacecraft. In the absence of requests for specific data intervals the spacecraft will fill whatever telemetry is available with data from the auroral oval.

3. Structural Subsystem

The structural subsystem for Ex-Alta 1 is the ISIS 3U STS with a Gomspace FPP on the +Z face and interstage panels with electric knives to hold down the MNLP booms and antennas during launch. The ISIS STS is developed as a generic, modular nano satellite structure that is compliant with the CubeSat standard. The ISIS STS is composed of two black anodized AL 6082 frames that are connected together with alodyned AL 6082 ribs. The ribs connect the frames together, support interstage panels between the units and provide a mounting platform for the electronics boards. Each threaded insert is alodyned to allow for conductive pathways throughout the structure to ensure a common ground state. The modular nature of the STS allows the assembly of electronic board stacks prior to structural assembly.

3.1. Interstage Panel

Interstage panels are designed to fit the mid-section of an ISIS structure as shown in figure 3-1. The interstage panels have integrated electric knives to tie down the MNLP booms and the antennas prior to deployment. As well, the interstage panels provide extra space for sensors.

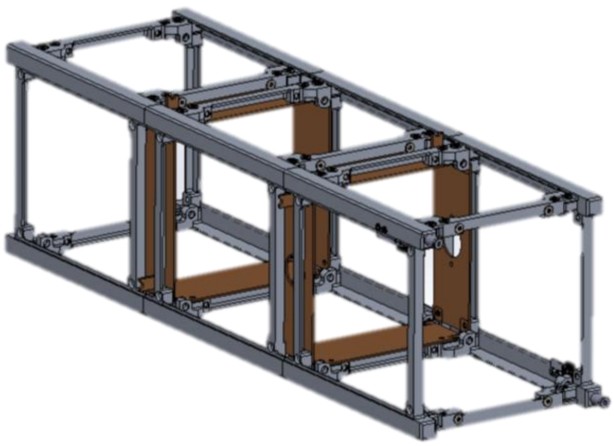


Figure 3-1: ISIS 3U STS with Interstage Panels

The interstage panels are equipped with a PicoBlade™ connectors used for connections to the antenna release mechanism.

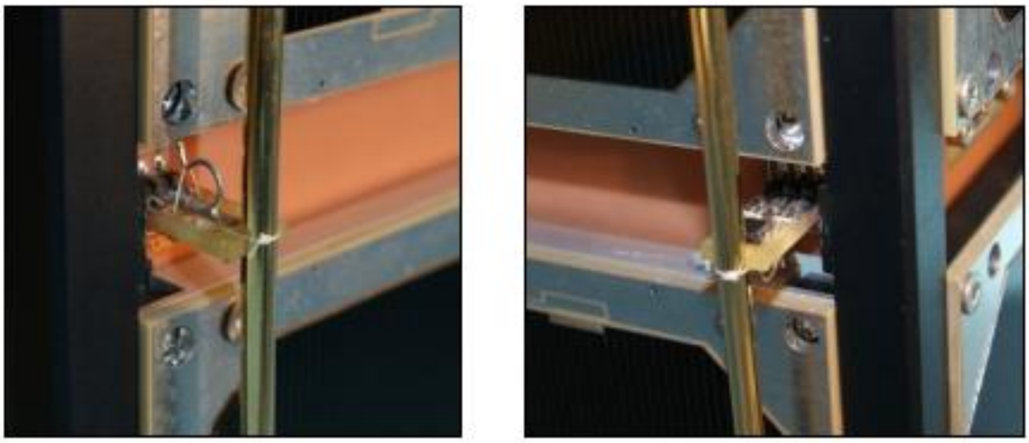


Figure 3-2: Interstage and Hold-down Mechanism

3.2. Flight Preparation Panel

The outside of the FPP features a connector (2.54mm header) and a USB connection (Mini-B). On the inside, a 12-pin horizontal PicoBlade connector allows direct interfacing to the NanoHub. On the NanoHub the connections are then distributed to the other systems as necessary like the kill switch to the NanoPower and the USB to the serial converter on NanoHub. This panel will be located on the +Z face of the satellite and fit within the specified access port dimensions. Three header slots are available for remove before flight (RBF) pins, one of which will be the MNLP disable bias, and two of which will be used for arming antenna and probe release. A slightly modified panel like in the photo below will be used.

Figure 3-3: Gomspace FPP Example

A preliminary rendering of the functional panel design from GomSpace for the QB50 mission requirement is displayed below.

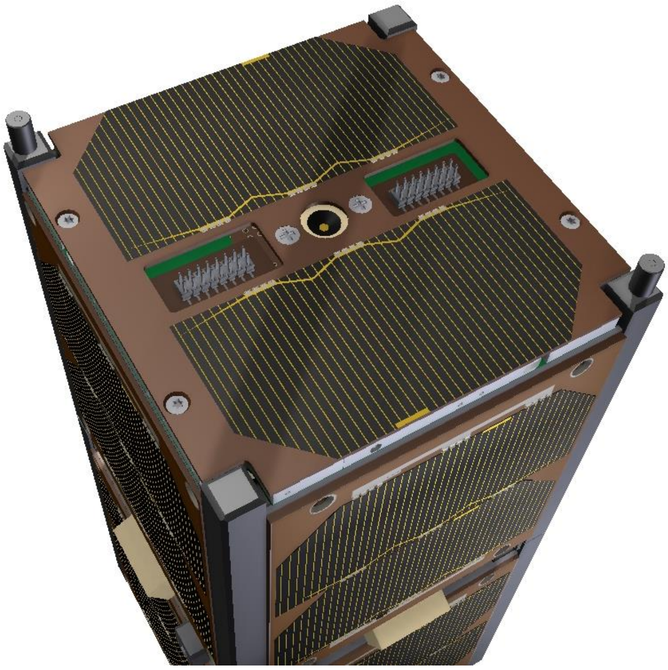


Figure 3-4: +Z Face FPP Rendering

3.3. Finite Element Analysis (FEA) Analysis

3.3.1 Simulation Goals

- Determine the natural frequency of the Ex-Alta 1satellite structure and boom
- Find the relationship between boom length and natural frequency
- Find the resultant stresses from 15 g quasi-static loading

3.3.2 Solver Information

Information was taken from Dassault Systems SolidWorks database.

Linear static analysis assumes the following:

- Materials are linear and comply with Hooke's Law
- Displacements are small and ignore geometric hardening
- Boundary conditions are constant
- Loads are applied infinitely slowly, therefore inertial and dampening forces are ignored.

SolidWorks FEA solver uses two methods, Direct Sparse and FFEPlus, to run simulations. Direct Sparse solver uses the direct method to solve the FEA stiffness matrix; this means on larger simulations it is not viable due to the solution time, and computation resources grow exponentially with the number of nodes. On larger simulations FFEPlus is the preferred solver because of its ability to approximate solutions by using iteration, and its ability to control the accuracy needed for a solution. Iteration in FFEPlus solver uses error in calculations to approach a threshold value. The only advantage of FFEPlus solver is a greatly reduced simulation time for large degree of freedom problems.

Frequency analysis and dynamic analysis in SolidWorks follow the same assumptions above except the loading condition. Frequency analysis does not require an excitation input to find resonant frequencies, but displacement values given will be useless due to no supports to stop motion. A "soft spring" can be added to a model to give displacement values relative to vibration but will stiffen the model an undetermined amount. When "soft spring" is used both cases should be run to determine the difference in resonant frequency.

3.3.3 Mesh Analysis

SolidWorks uses automatic mesh creation that uses either "solid" tetrahedral elements or shell triangular elements. Shell elements are only used in constant thickness geometry that has a thickness to width/length ratio respectively above 1:10. Tetrahedral is used for all other geometry. SolidWorks uses Jacobian points between nodes for both shell and solid elements, the number of which can be used to control accuracy in complex geometry.

The following characteristics are the limits for acceptable mesh:

- aspect ratio below 10:1, with good ratio being below 1:5, and ideal being 1:1
- nodal analysis (static simulation only) showing convergent values
- consistent location of maximum stress or deflection

3.3.4 Model Information

The following figure details the model used for the FEA analysis.

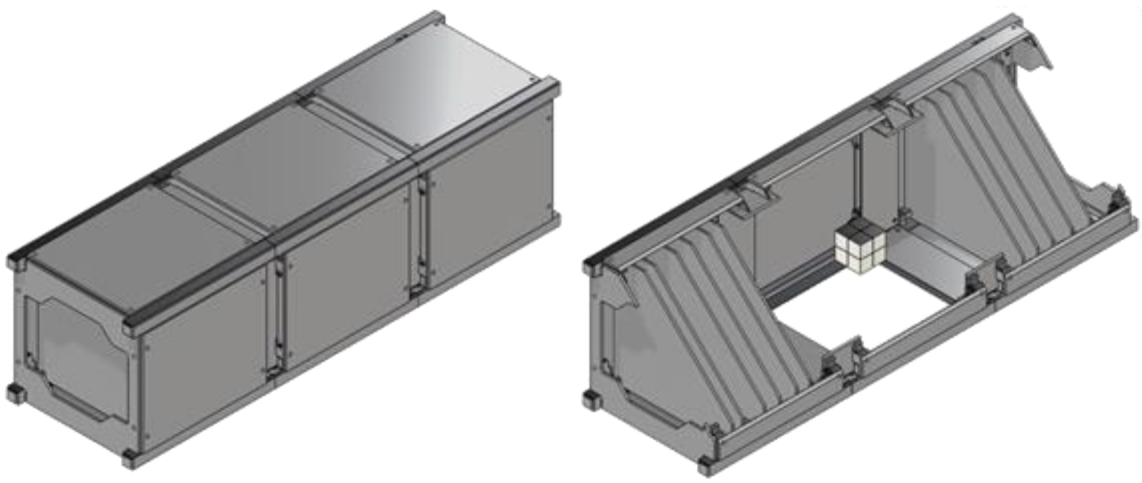


Figure 3-5: Diagram of structural model

A model was created to represent the structural components of the Ex-Alta 1 mechanical subsystem. Due to the complexity of the structure very few components were excluded from the analysis. Exclusions include kill switch assembly and magnetometer boom assembly. The kill switch assembly mass is only 2.35 grams and was left out of the assembly as an insignificant mass. The magnetometer boom assembly does however have a significant mass of 278 grams, which is represented by remote masses at the geometric center of its respective section. The sections in this report are referred to as Section 1 (containing power systems), Section 2 (containing magnetometer boom assembly), and Section 3 (containing ADCS and cameras). Four different circuit board types were created for simulation: Solar Panel, Antenna Board, Interstage Board, and Main Boards. Main boards vary in weight but were created out of the same material and have identical mounting geometry.

The material used by Gomspace for these parts is Glass/Polyamide composite, which IPC 6012c class 3/A standard IPC-TM-650, Method 2.4.18.1 stating that the material tensile must be 248 MPa at a minimum deflection of 12%. For simplicity the material was assumed to isotropic for this study, which allowed for the calculation of Young's Modulus of 2.06 GPa. This acceptable because the stiffness is needed to find a solution to the study but the focus is on the structure and the circuit board have been proven through prior use. The masses had to be and assigned manually, as shown in Table 3-1: Component assigned masses

, because the surface components were removed to simplify the assembly. The resultant overall mass of the simulation structure and components is 4557 grams. By simplifying the circuit boards, shell mesh could be used in the study and greatly reduces the resources necessary for mesh creation.

Section	Component	mass (g)	Given Designation
1	Antenna Board	30	UHF Turnstile Antenna
	Main Board 1	45	Bus Board
	Main Board 2	50	Undetermined
	Main Board 3	50	
	Main Board 4	50	
	Main Board 5	50	
	Main Board 6	240	BP4 Power Board
2	Boom mass	278	Magnetometer Boom Assembly
3	Main Board 7	979	ACDS and Cameras
	Main Board 8	979	
	Main Board 9	979	
	Main Board 10	50	Digital Fluxgate Magnetometer
	Main Board 11	45	Bus Board
	Antenna Board	30	UHF Turnstile Antenna
1,2,3	Solar Panel	50	Solar Panel
1,2,3	Interstage Board	14	Interstage Board

Table 3-1: Component assigned masses

Remote masses representing the Magnetometer Boom Assembly were placed at the geometric center of Section 2 and connected to the same mounting area as circuit board supports. Eight masses were connected in Section 2 similarly to simulate a distributed mass. While the stiffness of the boom assembly is not accounted for, this is a conservative approximation of the model. The natural frequency is described by a simple spring relation, where increasing the mass "m" or decreasing the stiffness "k" will lower the natural frequency, justifying a model with increased mass to simulate a boom assembly attached to the interior of the structure.

$$f_n = \frac{1}{2\pi}\sqrt{\frac{k}{m}}$$

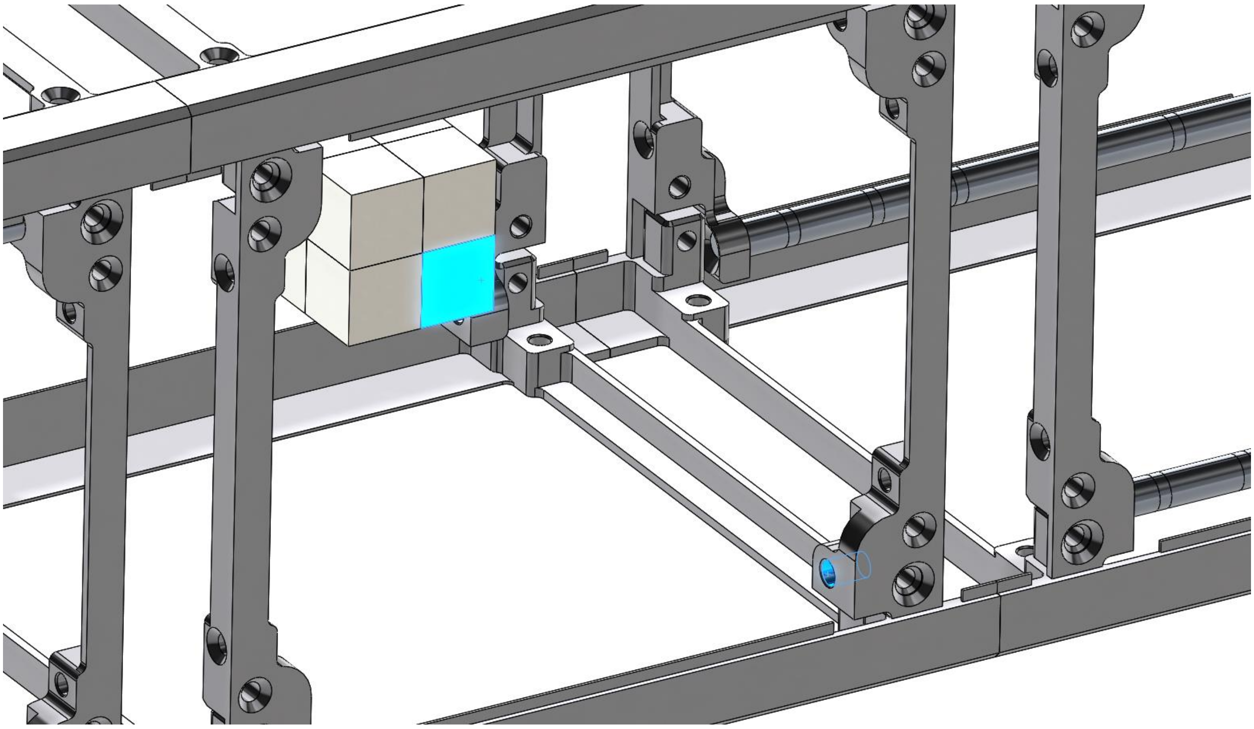
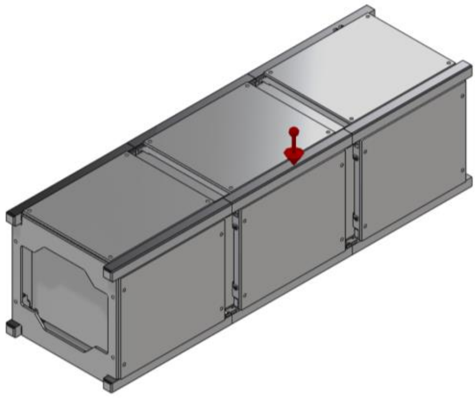


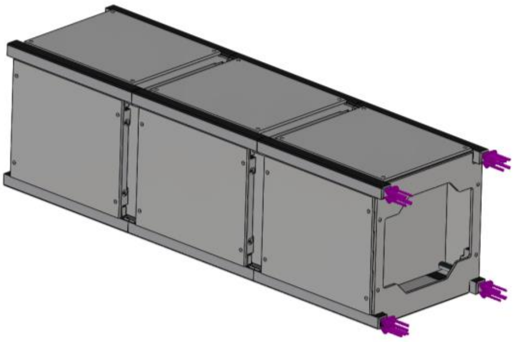
Figure 3-5: Boom remote mass connection geometry, highlighted mass (center) have a rigid connection to the **highlighted cylinder (bottom right).**

The 6082-T6 aluminum sub structure was an original imported model from the manufacturer and was left unmodified. Mesh dependency was determined for the sub structure on its own to determine what mesh refinement could be used in assembly studies and to help determine stability in assembly simulations. 4.5mm 6061-T6 aluminum rods were used to mount the Main Boards to the frame, the diameter is the thickness of the rod and spacers use in the assembly of each section.

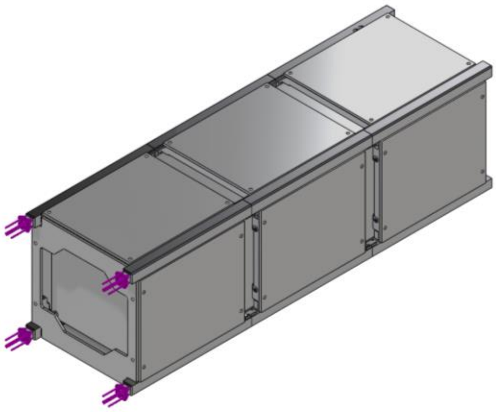
The boundary conditions for quasi-static were given by ISIS and are graphically depicted below.



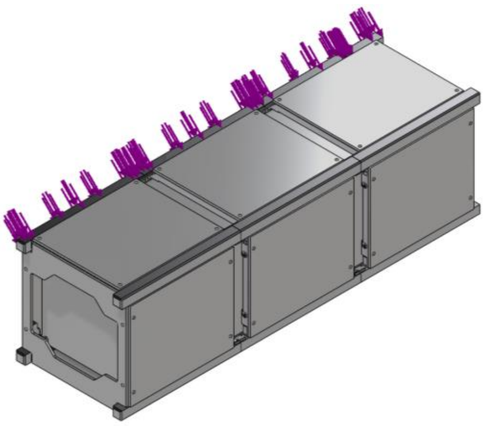
Red arrow depicts the direction of 15 times gravity acceleration



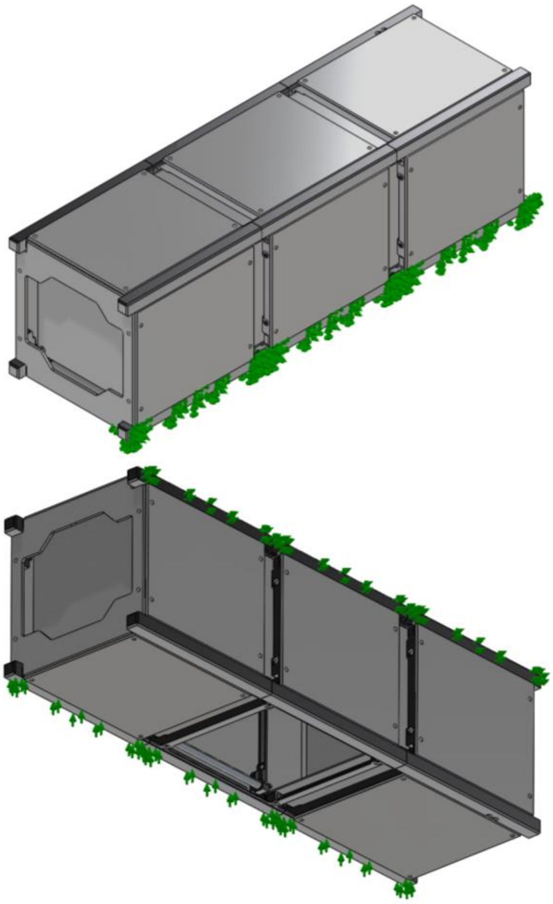
Arrows pointing normal into the face of the structure represent a total force of 12 N



Arrows pointing normal into the face of the structure represent a total force of 45 N



Distributed force of 10 N at a 45 degree angle passing through the center of the Ex-Alta 1 structure



The outside surfaces of the rail are fixed to stabilize the structure, this is the natural point of contact the that above force of 10 N would cause

Both faces highlighted are fixed in the normal direction and free in all other degrees of freedom

Figure 3-6: Boundary conditions and loading of the quasi-static model

3.3.5 Boom

Figure 3-7: Boom Model

A custom model was created for the boom assembly matching the profile given for the boom assembly. The new model uses a "surface" to create the geometry from the shell mesh, this is necessary because imported models are meshed as solid bodies. Two masses are added in the model shown on the right of **Error! Reference source not found.**. The C.O.G (center of gravity) for the magnetometer is placed at the far end of the boom, where as the satallite structure C.O.G is placed on the central axis of the boom and 40 mm from the end; This 40 mm will center the boom in the satallite structure when it is in its folded state. **Error! Reference source not found.** shows the surface the remote masses used in the simulation will be connected to the boom.

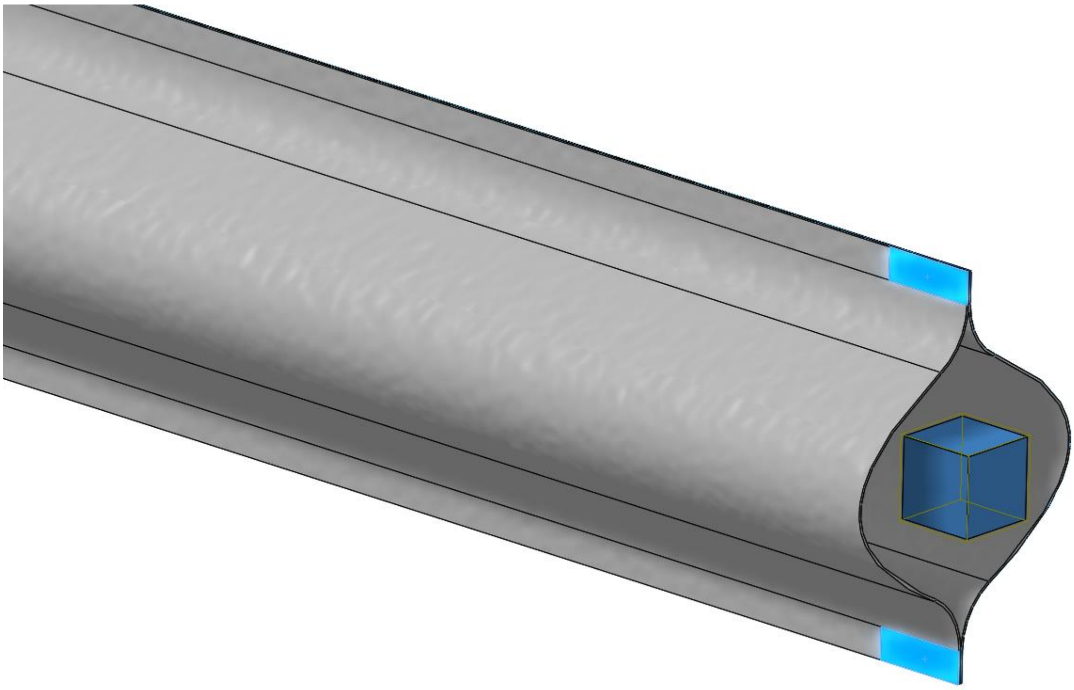


Figure 3-8: Magnetometer remote mass connection geometry, dark blue mass (magnetometer) has a rigid connection to the light blue highlighted areas on the boom. The mirrored surface of the boom is included in this connection.

The CFRP (carbon fiber reinforeced plastic) properties were not given and can vary significantly on a variaty of factors so a substite material estimate was chosen. The material properties were taken from the Advanced Composites Group and classified as a "high preformance areospace" composite with 45 degree construction. This particular material was chosen because its properties were readily available and it has ideal flexability for the application. CFRP is not an isotropic material but Solidworks can only create orthotropic materials for constant radius or flat materials, therfore the material must be treated as isotropic due to its variable radius. Only the first resoanance mode is of interest for this study of a slender, long beam. Stress from this mode will be caused by bending moments perpendicular to its longitudal axis, therefore the over estimation of the modulus in shear will not be critical for this simulation.

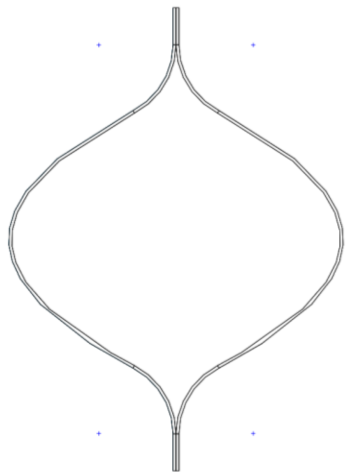


Figure 3-9: Cross section showing variable radius curvature

Properties	Direction	Value	Averaged Properties
Tensile Strength [Mpa]	WARP	540.0	530
	WEFT	520.0	
	in-plane	95.0	
Compression Strength [Mpa]	WARP	490.0	477.5
	WEFT	465.0	
	in-plane	95.0	
Tensile Modulus [Gpa]	WARP	54.5	53.75
	WEFT	53.0	
	in-plane	3.8	
Compression Modulus [Gpa]	WARP	49.0	48
	WEFT	47.0	
	in-plane	3.8	
Poisson's Ratio	-	0.05	0.05

Table 3-2: CFRP VTM264 199 GSM properties and averages for simulation

3.4. Results

3.4.1 Structure- Resonance

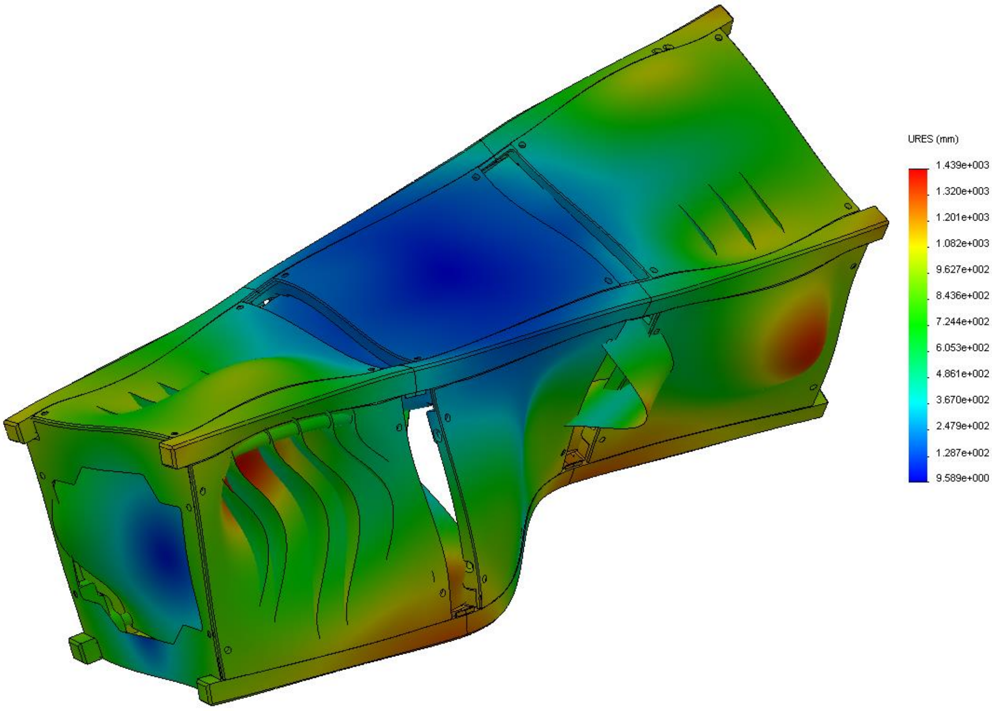


Figure 3-10: First mode shape of the Ex-Alta 1 at 314 Hz. Note the displacement scale does not show true values.

The resonance solution given by SolidWorks has the mode shape above, which occurs at 314 Hz. The displacement scale on the figure gives displacement values relative to a point in free space and not relative to any point on the model. The shape shows “clipping” surfaces moving through each other due to scaled displacement to show the mode shape. For clarification, the model is twisting around the axis of the longest dimension.

3.4.2 Structure- Quasi-Static

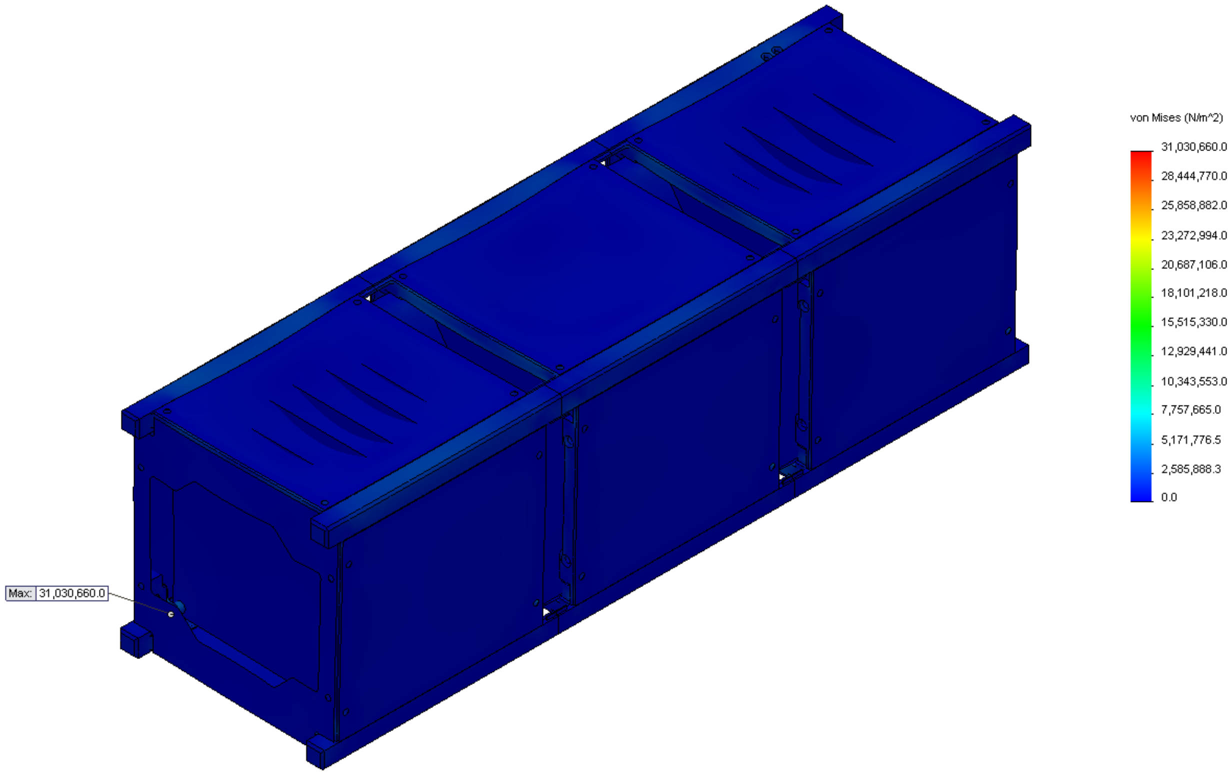


Figure 3-11: Stress plot of Ex-Alta 1 showing the max stress of 30 MPa where the PCB support connects to the structure. Deflection shown on a scale of 240:1.

Quasi-static loading of 15 G produced a maximum stress of 30 MPa and deflection of 0.2 mm. A higher deflection of 0.2371 was found for the interstage boards but since the material was assumed to be isentropic and not the focus of this study, these results are omitted.

3.4.3 Boom Modes

The boom has a best-fit resonance curve as a function of boom length, as shown by the formula below. The best-fit function for resonance frequency (f_n) as a function of boom length x is described as:

$$f_n = 5.78 * 10^7 x^{-1.903}$$

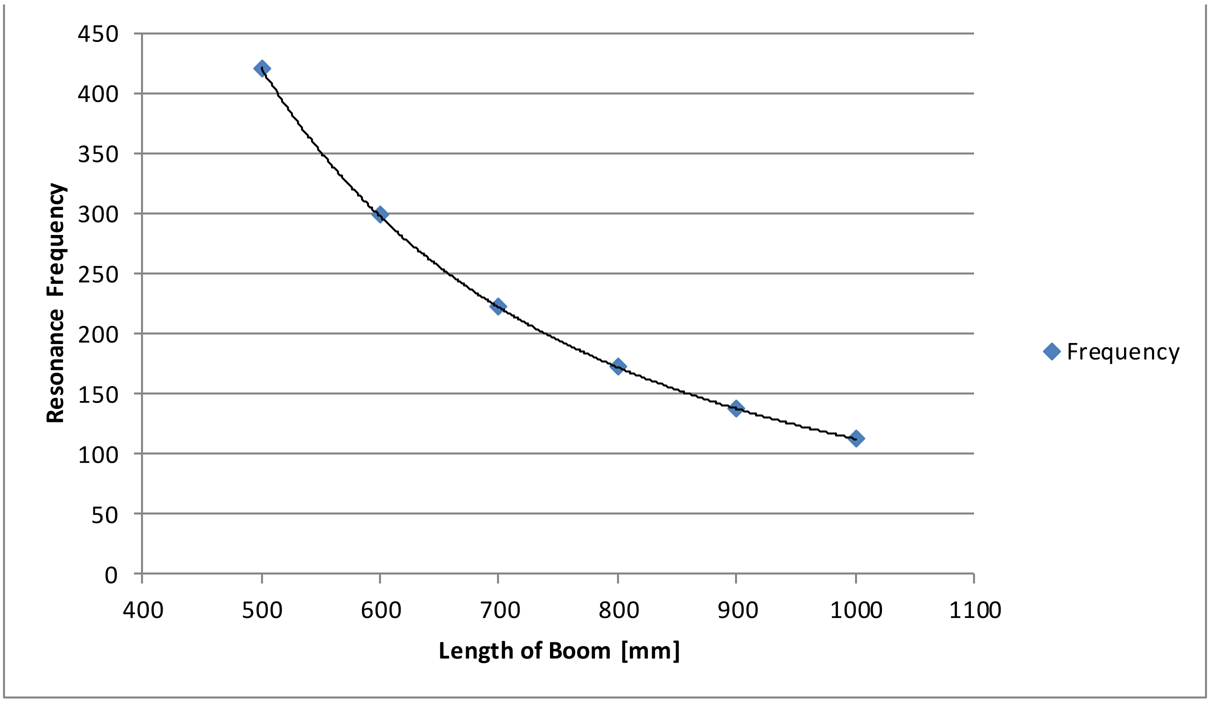


Figure 3-12: Boom resonance frequency as a function of length

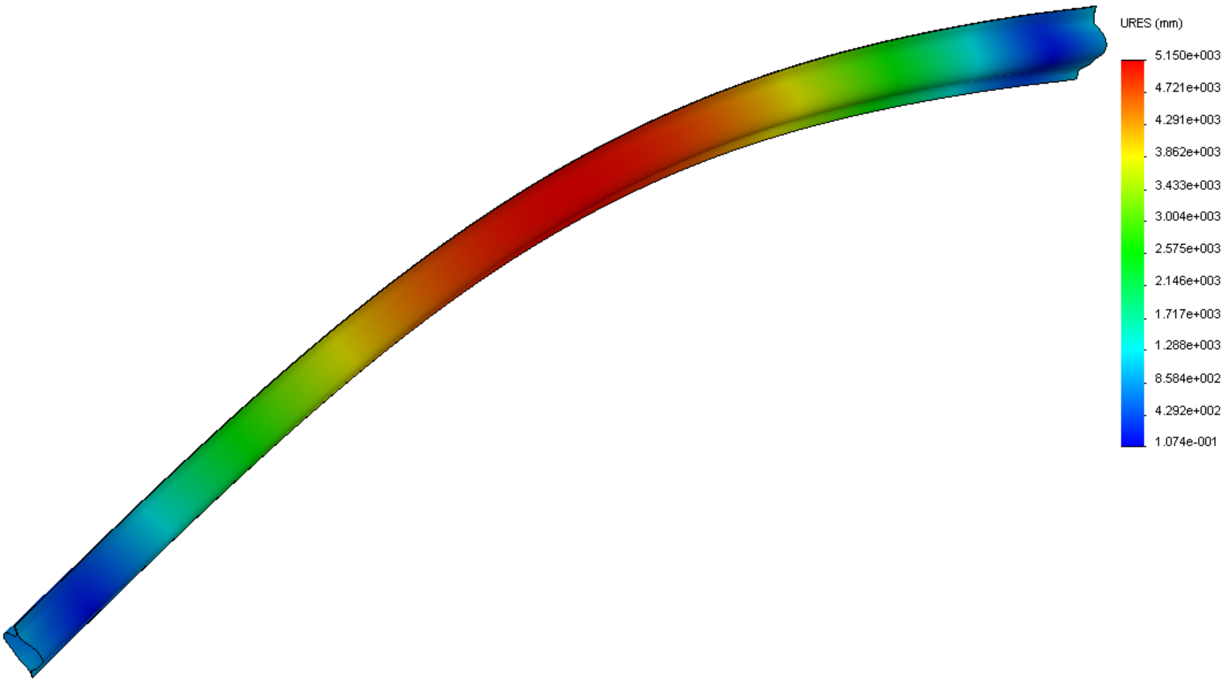


Figure 3-13: First mode shape of the boom assembly. Note displacement scale does not show true values.

3.5. Boom Design

Ex-Alta 1 will be equipped with a lightweight composite boom from DLR in Germany. The boom is made of a shape-memory smart material, whose permanent shape is a rigid cylinder-like structure. It can be rolled up like a tape measure and stowed safely during launch. To deploy, the memory material of the boom will roll out and reform the rigid cylinder-like shape that will provide the necessary support for the fluxgate magnetometer to perform stable measurements.

DLR has successfully performed low-gravity testing of the boom design on parabolic flights, and a customized mounting solution is under investigation by the U of A and DLR. During the preliminary design stage of the satellite, the DLR boom, the ATK coilable boom, the Surrey Space Weitzmann Boom and an in house design were traded off to come to the final decision of using the DLR boom. The more traditional designs employ trussed structures and spring deployment.

3.5.1 Trade-Off Analysis with More Traditional Designs

The magnetometer boom's main function is to put a finite distance between the spacecraft and the magnetometer payload as possible. By "as possible" it is meant that the boom shall maximize length while minimizing effects on the spacecraft that would otherwise compromise any system requirements. The criteria outlined for the Ex-Alta 1 boom is as follows:

1. The boom shall be light weight.
2. The boom shall be easy to deploy.
3. The boom shall be rigid.
4. The boom shall have a low cost.
5. The boom shall maximize length while minimizing the effects of aerodynamic drag.
6. The boom shall conform to extended volume requirements.
7. The boom shall be magnetically clean.
8. The boom shall be capable of passive deployment.

	DLR Solar Sail Boom	ATK Coilable Boom	SSTL – Weitzmann Boom	In House Boom (Estimated)
Mass	18 g/m	35 g/m	3.2 – 15.2 kg	500 g
Rigidity	Good	Good	Good	Acceptable
Cost	$10,000.00 In Kind Donation	$750,000.00	TBC	$0-$5000
Risk	Low	Very Low	Very Low	Critcally High
Stowed Volume	50mm x (length dependent)	160 mm x (length dependent)	102 x 115 x 264 mm	Custom
TLR	7	9	9	0
Total Capable	40 m	100m	6 m	1 m

Length				
Deployment Method	Active or Passive (Length Dependent)	Passive	Passive	Passive
Magnetic Signature	None	Can be made magnetically clean	None	Low
Aerodynamic Drag	Medium	Low	Medium	Medium

Table 3-3: Boom Trade Study Comparison Matrix

Four booms were compared during this trade study. The DLR solar sail boom, the ATK coilable boom, the SSTL Weitzman boom and an estimation of a custom made satellite boom. The lowest mass is the DLR boom. This is because of the lack of components and the use of a carbon fiber mesh to create a memory material. The DLR boom is rolled up flat when stowed and springs into its deployed profile during deployment. The ATK coilable boom has an acceptable mass, the custom made boom is at the highest end of the acceptable range and the Surrey Space boom exceeds the required mass of the satellite. In this aspect the DLR boom wins.

The rigidity of all booms is acceptable; this is unsurprising as the nature of a boom is to extend an object a finite distance away without compromising rigidity. The cost of the booms covers a wide range. The DLR boom is around $10,000.00 (prototypes and flight models) however it is given “in-kind” which means that no fund raising needs to be done. The ATK boom is entirely out of the cost range (costing many times the base cost of the whole satellite) and the Surrey Space boom has a cost that has yet to be determined. The cost of the custom made boom was estimated to be approximately $5000.00 assuming that the design work was done by volunteers and the boom was built in the Mec E machine shop using reduced rates.

The risk of all four booms varies based on TRL. The DLR boom has a TRL of 7 (as it has been tested with a parabolic flight) while the ATK and Surrey Space booms are a TRL of 9 due to the extensive flight experience for each. The custom boom currently has a TRL of 0 as no design work (aside from this conjecture) has been completed. The stowage volume for the boom on the satellite cannot exceed 1U. As a result, the only booms that pass this requirement are the DLR and custom boom. The total capable length of each boom exceeds or meets the expected length (1 m) and thus there are no concerns. Each of the considered booms is capable of passive deployment. Each of the booms (with the exception of the custom boom) have no magnetic signature.

From the basic analysis performed the best candidate for the boom is the one from DLR. The boom from DLR is basically free, has the lowest mass, is magnetically clean, has a simple deployment mechanism which utilizes a minimum number of parts, has plenty of length, has an acceptable aerodynamic drag and is capable of deploying passively. The custom made boom is far too risky to design. The ATK boom is comparable to the DLR boom in capability but is incredibly expensive and would need to be modified in order to fit in the satellite (adding to the expense). The Surrey Space boom exceeds, alone, the required mass of the satellite. Thus the DLR boom shall be used for this design.

3.5.2 Material & Physical Properties

The boom is made up of a carbon fiber weave which allows for a great deal of strength for a low amount of mass (18 g/m). The boom is a memory material, flattening out when it is rolled up and assuming its rigid profile during deployment. The DLR boom was originally designed for use with solar sails and has been tested in a micro gravity environment at a length of 40 m. The DLR boom is capable of both passive and active deployment. The capability for active deployment has been tested using an electric motor and by inflating the boom with air, however it is not expected that an active deployment will be necessary for the Ex-Alta 1 satellite. Material & physical properties of the DLR boom are listed below:

Matrix	Cyanate ester resin LTM123 (Cytec)
Fibers	Torayca T700
Type	Plain weave fabric 94g/m²
Laminate	1 ply ±45deg (2 plies for interfaces)
$E_{11} = E_{22}$	70GPa
G_{12}	4.5GPa
ν_{12}	0.1
Effective laminate thickness	0.12mm
Mass	18 g/m
Property	Bending Stiffness [Nm²]
EI_Y	8.0
EI_Z	13.9

Table 3-4: DLR Magnetometer Deployment Boom Mechanical Properties

3.5.3 Dimensions

The boom length is currently undergoing refinement (as a function of aerodynamic effects and orbital lifetime) and the expected length shall be 1 meter at a maximum. The boom is capable of flattening for stowage due to its memory nature, however the deployed profile is displayed below.

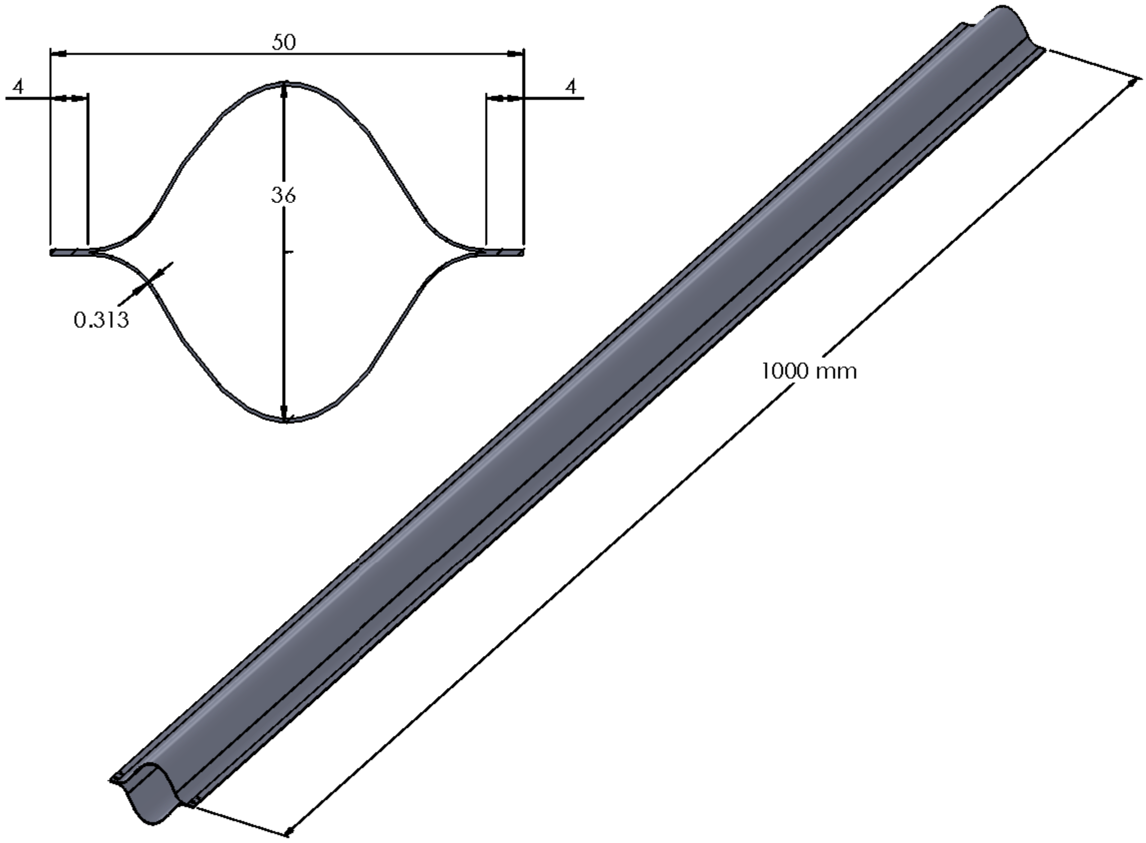


Figure 3-14: DLR Magnetometer Boom Dimensions

3.5.4 Deployment Mechanism

The magnetometer boom shall deploy in the nadir direction through an opening in the middle U of the nadir face. The boom is capable of passive deployment through the use of its internal spring energy which can be initially harnessed by rolling the boom into a coil while fixing one end to the interior of the satellite opposite the opening in the nadir face. The boom is secured to the satellite by burn wire which is burned off after the detumbling phase.

3.6. Mass budget

The following table summarizes the mass budget of Ex-Alta 1.

Table 3-5: Mass Budget

Ex-Alta 1 Satellite Mass Budget

Description	Total Weight W/out Contingency (g)	Contingency (%)	Contingency (g)	Total Weight (g)
Mechanical Subsystem	587.00	0.16	96.20	683.20
Power Subsystem	678.00	0.10	67.80	745.80
Communications Subsystem	105.00	0.10	10.50	115.50
Command and Data Handling Subsystem	100.00	0.10	10.00	110.00
Payload	490.00	0.20	98.00	588.00
Attitude and Control Subsystem	383.00	0.10	38.30	421.30
Integration	250.00	0.10	25.00	275.00
Satellite Total	2593.00	0.13	345.80	2938.80

Contingency was calculated as 10% for COTS hardware, and 20% for hardware in development. This covers uncertainties and mass for harnessing. Additionally, any extra weight from integration is estimated.

4. Attitude Determination and Control Subsystem

4.1. ADCS Overview

The ADCS serves to control the orientation and provide position knowledge to Ex-Alta 1. The Surrey Space ADCS system will be employed, provided by QB50. Using a combination of magnetorquers and a momentum wheel to adjust the spacecraft's orientation, the system will counteract the various torques and forces exerted on the body over the course of its mission lifetime.

4.2. Sensors and Actuators

The following are sensors that each play a role in determining the attitude of the satellite during flight.

4.2.1 Magnetometer

The Surrey ADCS system comes equipped with a single magnetometer, which will have an accuracy of at least 50nT, at a 1 Hz refresh rate. Used in conjunction with the pointing knowledge from the other sensors, the magnetometer can provide the satellite's position with respect to the Earth's dipole field. These values may be used as an auxiliary to the GPS coordinates in order to verify the satellites position and orientation.

4.2.2 MEMS Rate Sensor (Gyroscope)

A Micro Electromechanical System (MEMS) Rate sensor is also included in the ADCS system. This sensor provides measurements of the satellite body Y-axis rotation with respect to its orbital frame of motion with an accuracy of 15 millidegrees per second.

4.2.3 Coarse Sun sensors

Five photodiode sun sensors are to be attached to the satellites structure. These sensors will provide an approximation of the sun vectors to 10 degrees accuracy, allowing a first order approximation of the pointing of the spacecraft. These sensors will be sampled at a rate of 1 Hz. As the MNLP unit will occupy the ram face of the satellite, it will not be possible to mount a sensor on this face, however, the ADCS system will still be able to estimate the sun vector from only five faces.

4.2.4 CubeSense Sun and Nadir Sensors

A pair of 640 x 480 pixel 8bpp grayscale CMOS cameras are used for more refined measurement of both the sun vector and the nadir vector. These cameras will be mounted on the interstage panels and will face out the zenith and nadir faces. The CubeSense processing board interfaces to the two CMOS cameras. Each camera produces sun or nadir vector measurements accurate to at least 2 degrees. The cameras have a 180 degree field of view, however they provide even greater accuracy (half a degree), while the target remains in the center 40 degrees of the field of view. From the sun and nadir vectors the pointing of the satellite may be determined, and deviations from the predicted heading may be corrected for.

4.2.5 GPS

A Novatel OEM615 GPS and receiver module with H-code firmware will be used in conjunction with the Surrey ADCS Module. The GPS will be used for determination of the satellite's location, as well as confirmation of its ephemeris. The GPS is set to turn on for 2 minutes each orbit in order to conserve power. Location information will be propagated from this point each orbit. This module will provide velocity, position, and time measurements to the ADCS.

The GPS antenna consists of a 10x10 mm ceramic patch. The antenna is mounted to the zenith face of the satellite, on an interstage panel, in order to provide optimal access to the GPS satellites while the satellite is in nominal attitude.

4.2.6 Magnetorquers

The Surrey Space ADCS includes three magnetic actuators, two torque rods and a single air core torquer. Each of these actuators has a nominal moment of 0.2 Am^2 per actuator. In addition, the torque rods have been designed for a minimal residual magnetic moment (>1 mA m^2). The two torque rods are mounted to PC104 boards, aligned to the Y and Z axis respectively, the air core torquer is on its own board aligned to the X axis. These actuators are rated for control of cubesats composed of up to 24 unit cubes. The magnetic actuators will be used in conjunction with the momentum wheel to ensure the attitude of the spacecraft remains constant.

4.2.7 Momentum Wheel

The system will feature a reaction wheel, acting as a momentum wheel. The wheel will spin about the Y axis (in the case of Ex-Alta 1's frame of reference it is the X-axis), which will provide stabilization in all axes. The wheel has a maximum momentum storage of 1.7 mN m s and can provide a maximum torque of 15 mNm. The wheel's maximum speed is 8000 rpm.

4.3. ADCS Software

The ADCS software will be provided by Surrey Space. Minimal modifications (if any) will be made to the generic software (in fact, changes are restricted by Surrey Space).

4.3.1 Modes of Operation

The ADCS modes of operation are controlled via the hierarchy outlined in the concept of operations earlier in the document. Three modes of control are available, consisting of an idle state where no action is performed, but pointing knowledge is maintained, a detumbling state where the satellite seeks to reach a mode of spin only around the X-axis (as defined in the satellite reference frame above) and finally a 3-axis X-momentum stabilized state where the satellite will use as many attitude sensors as possible to maintain a stable and ram-facing platform. Additionally, separate modes of maintaining pointing knowledge are available, but these are left to the ADCS system to control based on the mode of control. These consist of an idle state, a MEMS sensor X-rotation filter, a magnetometer rate filter, a full-state EKF filter, and a triad algorithm utilizing the CMOS cameras and magnetometer. Please note that the reference axes included in the Surrey ADCS ICD document are not the same as the satellite reference frame as defined above – the Y-axis defined in the Surrey document maps to the X-axis defined in this document.

4.4. Detumbling

4.4.1 Detumbling Procedure

The Satellite will detumble in two phases. In the first phase, the satellite will halt its rate of rotation with respect to its nominal orbital reference. This phase will be conducted through the use of the magnetorquers, in conjunction with the sun and rate sensors on the Surrey ADCS. In the second phase the satellite will saturate its momentum wheel, entering the ADCS "Y momentum mode" (Defined in the Surrey Space ADCS interface control document [ICD]).

4.4.2 Detumbling Simulation

To simulate the process of phase 1 of detumbling a simulation of the torque provided by the magnetorquers was constructed. By using the magnetic field values from IGRF11 the B field on the satellite body was calculated. Using the maximum dipole moment of the torquers (0.2 Am^2 in each axis), the effective torque that could apply was calculated. Summing this torque over the course of a single day allowed for the calculation of the maximum change in angular momentum that the torquers can provide. Given the moments of inertia of the satellite, in one day at 100% duty cycle, the magnetorquers could recover the satellite from a maximum of a 278 degree per second spin (about the satellite yaw axis).

Given a 10% duty cycle over the course of one day, activating the torquers only when they would provide the largest torque, the satellite can recover from a 54 degree per second spin. This value is much larger than the required 10 degree per second detumble over the course of two days.

Likewise, the saturation of the momentum wheel in phase 2 of detumbling can be simulated by calculating the maximum amount of angular momentum that can be produced by the torquers (used to offset the angular moment generated in the opposite direction by the momentum wheel). At a 10% duty cycle over the course of one day, the torquers can supply a change in angular momentum of 57 mNms, substantially larger than the 1.7 mNms of total momentum storage in the wheel.

4.5. Boom Effects and Stability

The deployable boom shall have a significant effect on the spacecraft as a result of aerodynamic torque, aerodynamic drag, increased inertia, and orbital resonance. These may affect the spacecraft ADCS and orbital decay significantly.

4.5.1 Aerodynamic Drag

Aerodynamic drag for a 0.3m boom and a satellite without a boom was simulated using Cowell's method. Using the 380km altitude and 98 degree inclination from the expected launch parameters, the 0.3m boom shortens the expected flight duration from ~390 days down to ~195 days.

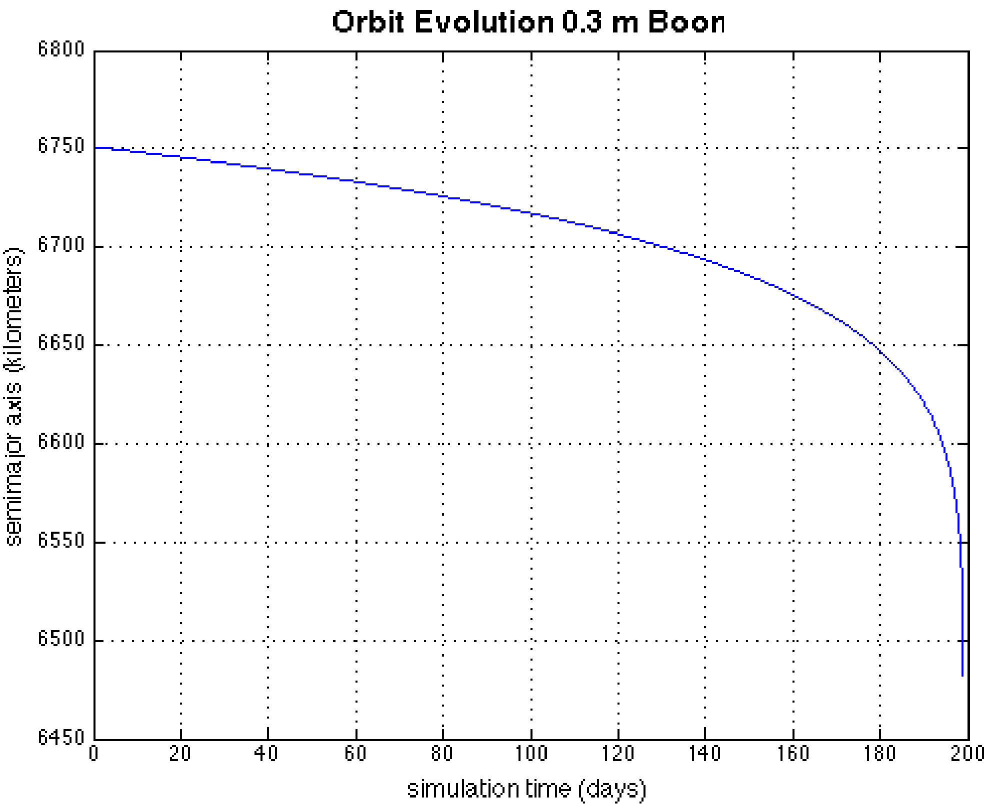


Figure 4-1: Analysis of orbital lifetime using drag on a 0.3 m boom.

4.5.2 Aerodynamic Torque

The deployable boom exerts a torque about the center of mass on the spacecraft as a result of aerodynamic drag on the extended surface presented by the boom. To meet the requirement of maintaining science unit operation down to 200 km altitude, the angular momentum change imparted by the boom torque must not exceed the total available angular momentum change provided by the torque coils. Using a simple analysis summing the torque on a satellite body with an unchanging attitude, and comparing that to the maximum angular momentum dissipation provided by the torque coils in the detumbling simulation, it is found that at an altitude of 200 km a ~0.2 m boom can still be pointed. Keeping in mind that the atmospheric density is dynamic and that the model is an overestimate, it is a reasonable conclusion that the satellite will maintain pointing down to 200 km, but not likely at or past 200 km.

4.5.3 Effect on Inertia

With the boom deployed an increased moment of inertia will help maintain the required pointing of the satellite. The main effect of the boom is to increase the moment of inertia on 2 axes. Once the boom has been deployed, the increase in moment of inertia in these 2 axes will provide a minimal degree of passive stabilization to the satellite. Overall, the boom will have a parallel effect to that of the torque wheel, which employs the same technique of increasing the inertia about two axes in order to add enhanced stability.

4.5.4 Orbital Resonance

In a 380 km orbit, a minimum length of 0.5 m is required to generate any parametric resonance with the orbital period. A boom of 0.3 m length should not significantly affect the attitude of the satellite in this context.

4.6. ADCS Suitability

The ADCS system provided by Surrey Space is being tailored for the QB50 mission. An in-depth pointing budget is currently under construction to verify that a 3U cubesat with boom will still meet the pointing requirements. The ICD for the ADCS system is referenced in [R-05].

5. Electrical Power Subsystem

The Electrical Power Subsystem is described below, including its components and capabilities. Every face of the satellite except for the boom opening and MNLP payload face will have solar panels.

5.1. NanoPower P31uS EPS board

5.1.1 General Description

Figure 5-1: Gomspace P31uS EPS

The NanoPower P-series power supplies are designed for small, low-cost satellites with power demands from 1-30W. The NanoPower interfaces to solar panels and uses highly efficient converters to condition the output power in order to charge lithium-ion batteries and provide power to the subsystems of the satellite. The NanoPower also provides all of the housekeeping for the satellite with an onboard microcontroller, through I2C, and can reset the satellite in the event of hardware or software malfunction in the OBC or elsewhere.

5.1.2 Main Features:

- Photovoltaic power conversion up to 30W
- 3 input channels with independent power-point setting
- Input converter efficiency: 93% average
- Optimized for panels with up to 7 solar cells in a string
- Three different photovoltaic power point options
- Battery under-voltage and over-voltage protection
- Two regulated power buses 5V and 3.3V
- 6 user-controlled output switches with latch-up protection
- Discrete control of output switches
- Onboard housekeeping measurements
- Separation-switch interface with latching mechanism
- Remove-Before-Flight-pin interface
- Onboard microcontroller with I2C interface
- CSP network protocol for seamless integration with other Gomspace products

5.1.3 Schematic View

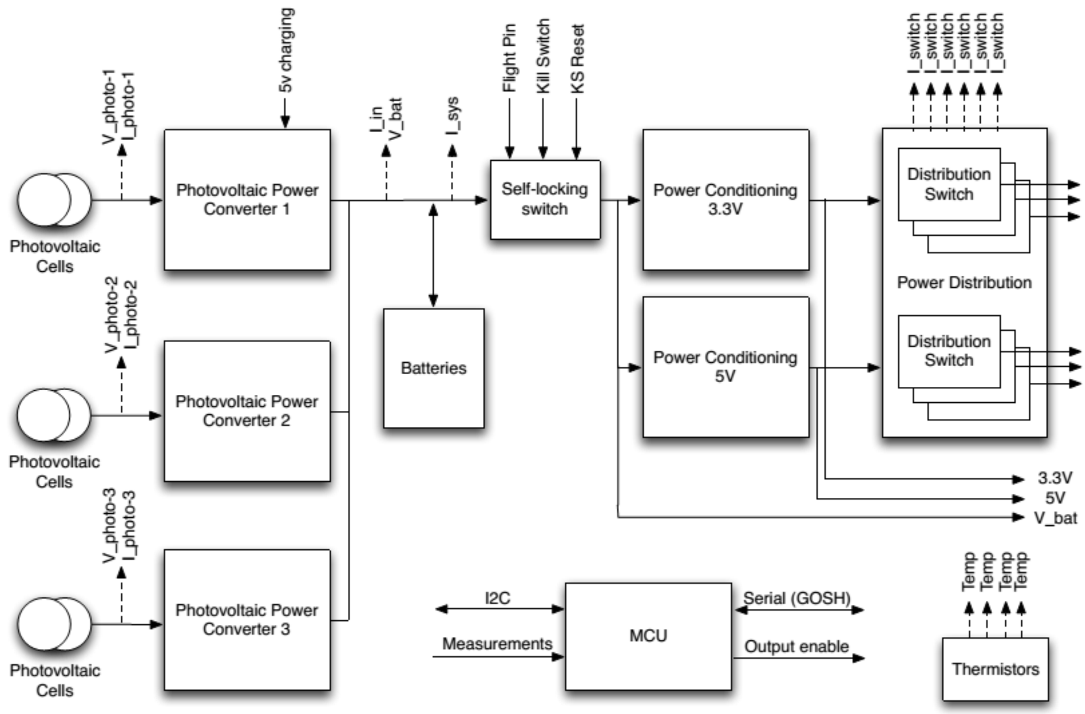


Figure 5-2: P31uS Block Diagram

5.1.4 Power Safeguards

The EPS implements three levels of safeguard for the satellite:

1. If a latch-up on any system power bus occurs, including the NanoMind computer power, the EPS will cycle the power bus and perform a reset of the particular subsystem. The threshold for over-current latch-up is customizable with a default of 2.0A as defined in the GomSpace ordering options datasheet for the NanoPower system.

2. If the battery pack voltage drops below VCritical (13.2V), all user configurable switches will be powered off until the voltage rises above VSafe (14.4V).

3. The NanoPower has 4 optional and user configured watchdog systems. One watchdog can be enabled to monitor I2C communication on the system bus, and if no activity on the bus is detected in a set period of time, the NanoPower will cycle the entire spacecraft power system. Another watchdog can be enabled to wait for a specific command sent to the NanoPower from the ground station or other hardware and cycle the spacecraft power if time runs out. Lastly, two watchdogs are available to actively ping other CSP-enabled hardware on the satellite, such as the radio or OBC. If the hardware in question does not reply to the ping, the NanoPower can either cycle spacecraft power or cycle the specific switched output powering the hardware in question.

5.1.5 Housekeeping

As outlined in the QB50 requirements annex 1, housekeeping data will be collected as whole orbit data once per minute and transmitted to the ground station when a link is available. From the QB50 document, WOD will contain:

- the CubeSat mode
- raw battery voltage
- battery bus current
- 3V3 bus current
- 5V bus current
- temperature of the COMM system
- temperature of EPS
- temperature of batteries

The CubeSat mode will be calculated by the OBC based on the battery voltage status and the operating mode of the satellite. Since the EPS has 6 configurable power switches controlling the outputs to each subsystem, total 3V3 and 5V0 current will need to be calculated by adding the individual current draw of each 5V0 and 3V3 switched output. The current draw by the battery bus can be calculated by subtracting the total switched current from the total system current out of the battery. Any non-zero result is the current present on the unregulated battery bus since the permanent 5V0 and 3V3 outputs are not connected to any load. EPS temperatures are available from four locations – one on each input boost converter, and one on the EPS circuit board. The board temperature will be used for the WOD value. Two battery temperatures are available as well, and both will be averaged for the temperature WOD value. The same is performed for the two communications board temperature readings.

Beyond the WOD housekeeping, various values are available from each subsystem to get a better sense of their operation, failures, and configuration. A complete list of the housekeeping values is included as reference document [R-06]. A refined concept of operations concerning which housekeeping values will be included after testing has been done on each subsystem to determine the usefulness and required cadence of the values.

5.1.6 Flight Preparation Panel

For details concerning the electrical configurations of the flight pin, kill switch, and kill switch reset, see section 5.1.3 above.

5.2. Battery Dimensioning and BP4 Li-Ion Battery pack

5.2.1 Battery Dimensioning

The following equation was used to determine required battery dimension:

$$N = \frac{P_e T_e}{(DOD) n C_r}$$

N - Number of battery cells required

P_e - Power required during eclipse; we used the raw power consumption of the loads in Science Mode (see Sum Loads (W) in Power Budget Summary Table below).

T_e - Time duration of eclipse; longest 45 min = 0.75 hours

DOD - Depth of Discharge of battery; we aim to maintain this below 10% for optimal battery life cycle.

n - Battery to load transfer efficiency; this value depends on the total load on each of the power buses (Bat, 3.3V, 5V) and the individual transfer efficiency of the buck converters. We took the ratio of the sum of the raw loads on the each converter to the sum of the adjusted loads on each converter. This returned a value for transfer efficiency of 89%.

C_r - Battery capacity in Watt-hours; we are using 2.6 Ah (9.6 Wh) cells from Gomspace

The result of this calculation show that FOUR 2.6 Ah cells (~39 Wh) will be sufficient for the power demands while maintaining a depth of discharge below 10%.

5.2.2 BP4 General Description

Figure 5-3: Gomspace BP4

The BP4 is a set of four Mitsubishi 18650 Li-ion batteries that are connected in a 4S1P configuration. The package is implemented as a daughterboard for the EPS that allows full control of all electrical elements, including two thermally controlled heaters dedicated to assure minimum temperature of -5ºC for the batteries. The batteries in the 4S1P configuration boast 39 Wh of capacity and the onboard heater and temperature sensors ensure maximum battery life and prevent the satellite from getting too cold.

5.3. Solar Panels

5.3.1 General Description

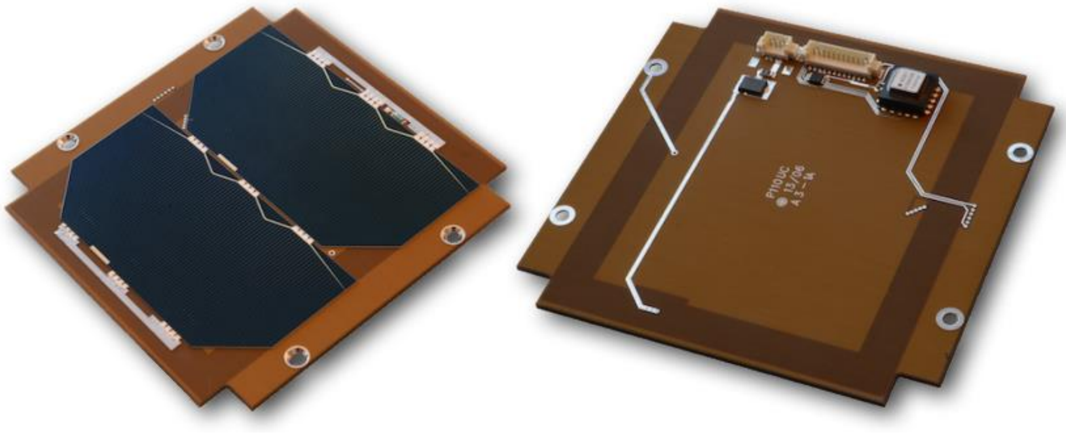


Figure 5-4: Gomspace P110 solar panels Azur 30% TJC

NanoPower P110 Series solar panels are an integrated side panel solution for CubeSat satellites on a single 1.6 mm thick PCB. The photovoltaic string consists of two series-connected AzurSpace 3G30A space qualified triple junction solar cell assemblies with CMX 100 cover glass. The cells are interconnected by three welded, silver plated kovar interconnectors, and the same type of interconnectors are used to connect the anode and cathode to 70um copper tracks on the panel front. The interconnectors are welded to the top of the cells by AzurSpace using a classified process, and the welds are covered by adhesive and cover glass. The rear-side welds are performed by Gomspace using an ultra-sonic heavy-bonder applying 12 welds per interconnector.

5.3.2 Main Features

- 30% efficiency
- Two series-connected AzurSpace 3G30A space qualified triple junction solar cell assemblies with 60.36 cm^2 effective cell area
- CMX 100 cover glass - 100um
- Cell base material: GaInP/GaAs/Ge on Ge substrate
- Panel base material: Space qualified glass/polyimide laminate with 2 internal 70um copper ground planes (10 planes in panels with magnetorquer)

- Cell bonding substrate: 1mil polyimide film with silicone pressure sensitive adhesive
- Plated, countersunk mounting holes with ground connection
- Silver-plated kovar interconnectors - 3 parallel interconnectors per string
- Operational temperature: -40 C to +85 C

5.3.3 Communication and House keeping

Since Ex-Alta 1 will be using the Surrey Space ADCS system attitude determination systems will not be required on the panels. However, the above panels can be equipped with attitude determination electronics that interface seamlessly to the NanoMind via a single connector including power supply and SPI bus for temperature sensor and output from sun sensor. The solar cells use a separate connector to connect to the power supply system on the NanoPower.

5.4. Power Budget Summary

	Assumptions
Altitude (km)	350
Albedo (%)	??
Attitude	Inertial sun stare of the smallest face
Orbit (minutes)	90
Max Harnessable Power per solar cell (W)	1.2
Any other assumptions	30% TJ GaAS cells

Loads	Number of Units ON	Power Consumption (W) Direct from BAT (14.8V)	+5V Regulator	+3V Regulator	Average Duty Cycle by Mode (%) Off mode	Power Safe Mode	Detumbling mode	Active Mission
GPS	1	0.00	0.000	1.000	0%	0%	0%	2%
P31us	1	0.00	0.000	0.125	100%	100%	100%	100%
Nanomind	1	0.00	0.000	0.459	0%	100%	100%	100%
Nanohub	1	0.00	0.000	0.066	0%	0%	0%	100%
UHF Rx	1	0.00	0.000	0.231	0%	100%	100%	100%
UHF Tx	1	0.00	0.000	5.000	0%	2%	2%	5%
Active THCS - Heaters	1	0.00	7.000	0.000	5%	5%	0%	0%
Fluxgate Magnetometer	1	0.00	0.400	0.000	0%	0%	0%	50%
MNLP operational	1	0.00	0.600	0.250	0%	0%	0%	50%
MNLP standby	1	0.00	0.150	0.180	0%	0%	0%	50%
SSADCS Y momentum control (daylight)	1	0.07	0.240	0.337	0%	0%	0%	50%
Sum loads (W)					0.475	1.265	1.567	2.576
Efficiency					89%	89%	89%	89%
Orbit Average Power Consumed (W)					0.536	1.427	1.768	2.906
Orbit Average Power Generated (W)					4.97	4.97	4.97	4.97
Power Margin (%)					89%	71%	64%	42%
Energy Available to Recharge Battery (watt-min.)					399.24	319.05	288.37	185.92
Battery Depth of Discharge After One Eclipse (%)					1.0%	2.8%	3.4%	5.7%
Time Required to Recharge Battery After E.O.E. (min)					3.2	10.7	14.6	37.2

Table 5-1: Ex-Alta 1 Power Budget Summary

Note: The above summary is for a 50% sunlight orbit. Based on the results of the thermal modeling for the batteries (section 8), the heaters will only be necessary in safe mode.

5.5. Power Budget Calculations

The above tables and calculations, and all that now follow, come from the Jan A. King power budget spreadsheet.

5.5.1 Orbit Average Power

Key Values		
Constants		
Solar Constant:	0.1361	w/cm^2
Efficiencies		
Photovoltaic Boost Converter Efficiency:	92%	
+5V Regulator Efficiency:	96%	
+3.3V Regulator Efficiency:	95%	
Battery Charge Efficiency:	85%	
Overall Average Efficiency	89%	
Solar Cell Properties		
Solar Cell Size (Area):	30.02	cm²
Solar Cells Per String Per Spacecraft Side:	6.00	
Parallel Strings Per Side (Facet):	1.00	
Solar Cell Efficiency at Reference Temperature:	29.50%	
Reference Temperature:	80.00	C°
Orbit Properties		
Semimajor Axis:	350	km
Eccentricity:	0.001	
Inclination:	98	deg.
Period:	90	minutes
Sunlit Orbit Fraction:	0.5	

Table 5-2: Power budget parameters

- The solar constant was obtained from http://atmospheres.gsfc.nasa.gov/climate/?section=136

- Regulator efficiencies were obtained from NanoPower P-Series datasheet from Gomspace.

- Solar cell efficiency at 80 degrees C obtained from Azurespace Solar cell datasheet- Sunlit orbit fraction was chosen to be 50% to demonstrate the worst case scenario values.

$$P_{peak\_out} = \text{Sum of subsystem power consumptions on each bus} = 18.13\text{W}$$

$$P_{peak\_in} = \text{Sum of subsystem power consumptions on each bus considering efficiency} = 20.46\text{W}$$

$$\text{Overall Efficiency} = \frac{P_{peak\_out}}{P_{peak\_in}} = 0.89$$

$$A_C = \text{Solar cell area}$$

$$N_C = \text{Number of cells per string per side}$$

$$E_C = \text{Solar cell efficiency}$$

$$C_{Solar} = \text{Solar constant}$$

$$T_{Sunlit} = \text{Sunlit orbit fraction}$$

$$P_{Min;\ 1\ face} = A_C \times N_C \times E_C \times C_{Solar} = 7.23W$$

$$P_{Typical;\ 2\ faces} = P_{Min} \times 2 \times \cos\left(\frac{\pi}{4}\right) = 10.23W$$

$$P_{Max;\ Vertex} = P_{Min} \times \left(2 \times \cos(0.7473) + \frac{\cos(0.7473)}{3}\right) = 12.38W$$

$$P_{Avg.} = 9.94\text{W}$$

$$OAP = P_{Avg.} \times T_{Sunlit} = 4.97W$$

The above approximation using the average of power generated with:

1. Sunlight shining on one single face with three panels (P_{Min}).
2. Equal illumination on two faces ie. 6 panels at 45 degrees ($P_{Typical}$).
3. Illumination of a vertex ie. 7 panels at 43 degrees (P_{Max}).

where both ±Y sides and the +X side are clad with three 1U solar panels, the –X side is clad with two 1U solar panels, and the –Z side is clad with one solar panel (each with two 30% efficiency solar cells) facing the sun was used to obtain Orbit Average Power (OAP).

Key Values		
Total power consumed in Active Mission mode during Eclipse	2.96	W
Time in Eclipse	45.00	min.
Battery Capacity	37.44	Wh

Table 5-3: Power budget key values

Knowing the depth of discharge we can obtain the energy drawn during eclipse in each mode. Dividing this value by the surplus energy generated (in watt min./min.) as calculated using the power margin, we obtain the time required to recharge the battery.

The power budget summary, Table 5-1, above indicates positive power margins, and thereby positive OAP, for all modes, however since the MNLP payload duty cycle has not yet been characterized by the developers, the duty cycle values for both payloads (the DFGM duty cycle will depend partly on remaining power) in Active Mission Mode are approximate.

6. Command and Data Handling Subsystem

The CDH subsystem is made of up two complementary boards – the NanoMind and NanoHub – to provide a full suite of command, data, telemetry, and storage interfaces.

6.1. On-Board Computer – NanoMind A712D

Figure 6-1: Gomspace NanoMind A712D

6.1.1 Main Features:

- ARM7 processor - 8-40 Mhz
- 2MB RAM, 4MB Code storage, 4MB data flash
- CAN and I2C interfaces
- RTC - real time clock w/backup power keeps time 30-60 minutes without external power
- FreeRTOS OS and driver library included
- CSP network protocol for seamless integration with other Gomspace products

- Integrated Development Environment based on Eclipse
- MicroSD socket for up to 2GB storage using AF2GUDI-OEM SD Card from ATP
- On-board magnetometer
- 3x PWM drivers (Compatible with NanoPower Solar P110U)
- 6x sun-sensor inputs (Compatible with NanoPower Solar P110U)
- 3x Rate-gyro inputs (Compatible with NanoPower Solar P110U)
- Includes flying-lead cables for I/O connectors
- Space Heritage: Since 02/2012 (PWSAT, ArduSat-1, ArduSat-X, GOMX-1, CubeBug-1, Triton-1)

6.1.2 General Overview

The NanoMind OBC is designed as an efficient system for space applications with limited resources. All computing, HK and the communication functions within the satellite are pefomed by the OBC.

6.1.3 Microcontroller

The computer is based on the ARM7TDMI embedded processor. This processor has a high performance 32-bit RISC architecture with a high-density 16-bit instruction set and very low power consumption.

6.1.4 I2C Interface

NanoMind has an I2C bus supporting bidirectional data transfer between masters and slaves, multimaster bus, and arbitration between simultaneously transmitting masters without corruption of serial data on the bus. Serial clock synchronization allows devices with different bit rates to communicate via one serial bus and is used as a handshake mechanism to suspend and resume serial transfer.

The I2C bus provides a high-speed of 400kbit/s, with a transmit hardware buffer of 68 bytes and a receive hardware buffer of 68 bytes.

The multimaster functionality is enabled by use of the CSP to communicate efficiently and easily with other CSP enabled hardware. The traditional master-slave functionality is also usable in concert with the multimaster CSP to communicate with sensors or payloads not using CSP. CSP adds layers 3 and 4 of the OSI framework model to I2C and enables a TCP-IP similar functionality. See addendum document for further details. (GS-CSP-1.1)

6.1.5 Software

Included with the NanoMind are four software libraries offering a complete hardware API and interfacing package. A source copy of FreeRTOS optimized for the NanoMind is included for use as the real time operating system. A driver library containing all of the hardware abstraction necessary for the operation of software on the NanoMind seamlessly integrates with the FreeRTOS system. The third library contains standard C functions from the NewLib C library for use on the NanoMind. Lastly, the CSP networking library and subsystem communication API is included for use with the other Gomspace products on the satellite, and provides all of the interface controls required by the other Gomspace products. An Eclipse IDE is available for the creation and debugging of additional software that will need to be written for the flight.

The flight code for the satellite will be written using the aforementioned software libraries provided by Gomspace. The following list of blocks and subroutines outlines the flight code.

- Satellite Timekeeping
  - Real Time Clock interface
  - Real Time Clock comparison to GPS time
  - Science script scheduling updater
- Science Scripts
  - Script reading
  - Script uploading and execution
  - Script error handling
  - Script scheduler
- Housekeeping
  - Housekeeping logger
  - Housekeeping timestamper
  - Housekeeping calculator and packager
  - Housekeeping beacon handler
  - Housekeeping ground station downlink manager
  - Housekeeping file manager
- File System
  - Read file
  - Write file
  - Delete file based on expired timestamp
  - Delete file based on successful downlink
  - File error and corruption detection manager
  - Science data download manager
- Firmware
  - Firmware upload manager
  - Firmware error checking
  - Firmware bootloader
  - Firmware failsafe handler
- Uplink Commands and Encryption
  - Command origin verifier
  - Command rejection manager
  - Command scheduler
  - Command execution verifier
- SurreySat ADCS Interface

- I2C driver
- Command forwarding manager
- Command reply manager
- Satellite mode selector and controller
- Camera photo manager
- Pointing knowledge and position manager

6.2. On-Board Data Handler – NanoHub

Figure 6-2: Gomspace NanoHub

6.2.1 Main Features:

- Dual electrical knife system with sense feedback
- Two latch-up protected power channel outputs
- Breakout of pins from stack connector
- 3 axis high precision gyros
- USB interface to 4 RS232 ports for easy ground support interfacing
- Flight preparation panel interface

- Digitial IO
- Analog inputs (ADC)
- SPI
- RS232 (TTL)
- I2C interface with CSP protocol
- IPC-A-610 Class 3 assembly

6.2.2 Description

The NanoHub is both an interface and a utility system. It provides a number of necessary interfaces for the seamless integration of a satellite, and integrates ground support interfaces like USB to subsystem serial ports and Flight Preparation Panel connector together with inter subsystem interfaces like digital IO, ADC and SPI and utilities like electrical knives, power switches, and gyros into one compact package.

6.2.3 Electrical Knives

The electrical knives are controllable power switches used to drive the release mechanisms for the booms and antennas. Two switches are available: Knife 0 and Knife 1. Both are powered by the battery voltage from the stack connector (H2-45, H2-46).

6.2.4 Knife Watch Dog Timer

Watch dog timer is included that will trigger knives activation independent on sense input when it times out. It is intended for antenna deployment, where the WDT could be reset from the ground station. If the deployment failed and the sense switch is broken, reporting a successful deployment, no communication is established with ground. The WDT will then timeout and a new deployment attempt will commence.

6.2.5 NanoCom Tx Inhibit

The NanoHub has the option for sending TX inhibit message to NanoCom when the Knife 1 ARM input is pulled low (disarmed). This can be used to ensure that NanoCom does not transmit when inside the launch pod or in the 30 minute safe period after deployment.

6.3. Satellite Telecommands

Commands are transferred across the satellite I2C bus and across the radio link using the same CSP protocol. CSP allows any Gomspace board to be directly interfaced with via the radio link, and any other systems such as the ADCS and payload systems will be interfaced via the NanoMind OBC. References R-01 through R-05 are the relevant datasheets containing the TM/TC commands to date. An additional document will be added when the software is finalized and custom commands are defined.

7. Communication Subsystem

Communication system for Ex-Alta 1 is based on space proven NanoCom U482 transceiver and ANT430 turnstile UHF deployable antennas.

7.1. NanoCom U482

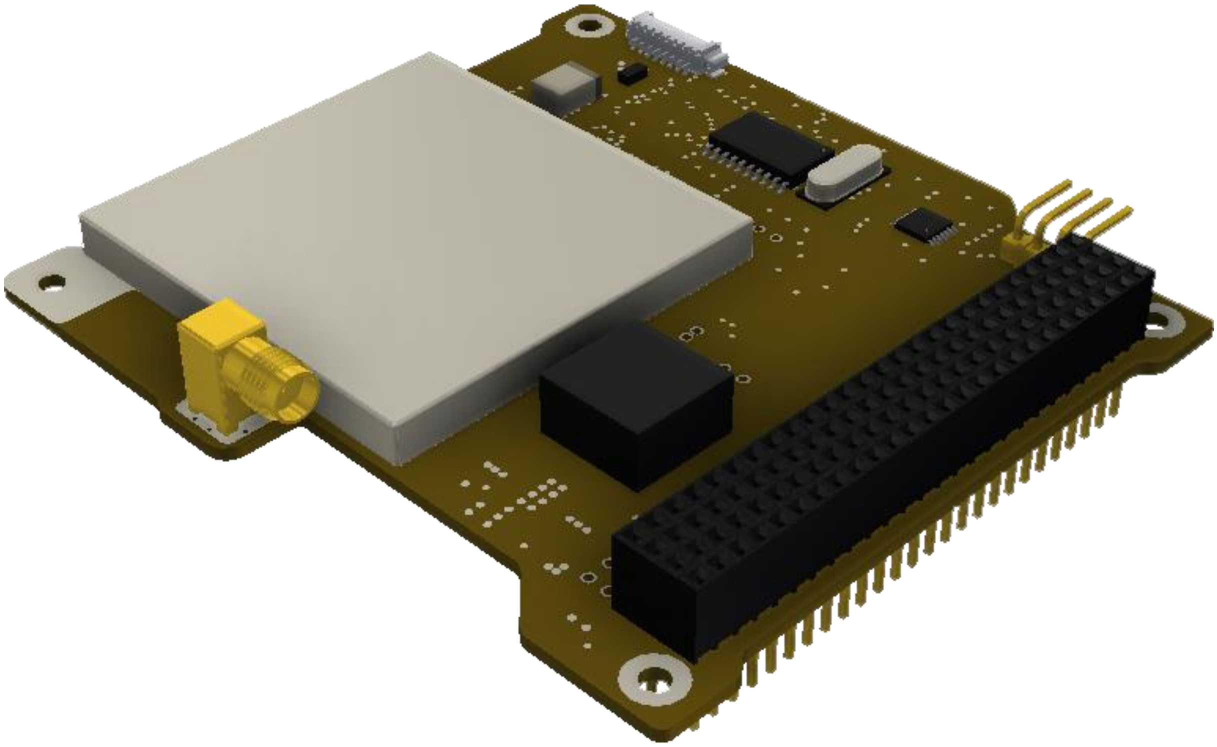


Figure 7-1: Gomspace NanoCom U482

7.1.1 Main Features:

- 435-438 MHz half duplex
- -126 dBm sensitivity
- Up to 34dBm transmitted power (PAE >50%)
- AFSK mod/demod
- 4800 baud uplink
- 9600 baud downlink
- I2C interface
- CCSDS frame support with FEC and Viterbi coding

- CSP router functionality for seamless networking with other Gomspace products
- On-board switched-mode power supply for TX (with current measurement)
- 60 dB ACRR

7.1.2 Description

The NanoCom U482 communication system is designed to provide the space-segment-part of a space-link for nano- and pico-satellites by means of a half-duplex UHF transceiver operating in the amateur radio 70cm band. The internal interface is an I2C bus allowing command and data exchange with the NanoCom board via CSP.

All packet framing plus the optional Viterbi encoding/decoding is performed by the NanoCom, taking CSP packets and transmitting them unmodified to the ground station. Vital housekeeping measurement points are sampled at user-defined intervals and stored for retrieval via the I2C bus or spacelink upon request.

A CW/FM morse beacon is available on the radio, but it will not be used. This CW beacon will be disabled by default. Instead, beaconing will be handled by the OBC sending a packet to be broadcast by the radio containing the required housekeeping values as specified by the QB50 requirements.

7.1.3 Receiver

The receiver is a super heterodyne receiver with two intermediate frequencies (IF) of 21.4 MHz and 455 kHz. The first IF of 21.4 MHz is high enough to give good image frequency suppression by the 437 MHz filter after the low noise amplifier (LNA), while the second IF at 455 kHz ensures good channel separation due to a steep +/-6 kHz (or optionally +/- 17.5 kHz) filter. The frequency modulation (FM) detector is an integrated device, SA606, with high sensitivity which provides the audio output plus a received signal strength indication (RSSI) measurement. An indication of the radio frequency (RF) error on the input is taken from the DC-level of the filtered audio-signal before it's AC-coupling.

7.1.4 Transmitter

The direct-FM transmitter consists of a voltage controlled oscillator (VCO) built around a silicon tuning diode which allows the TX frequency to be adjusted by a few kHz by moving the DC level of the AF input up or down. After the VCO, there is a filter and a buffer leading the low-distortion signal into the power amplifier which outputs 30-34 dBm with an efficiency of around 50% depending on the temperature. The transmitter frequency is temperature-dependent in a range of maximum ±10 kHz.

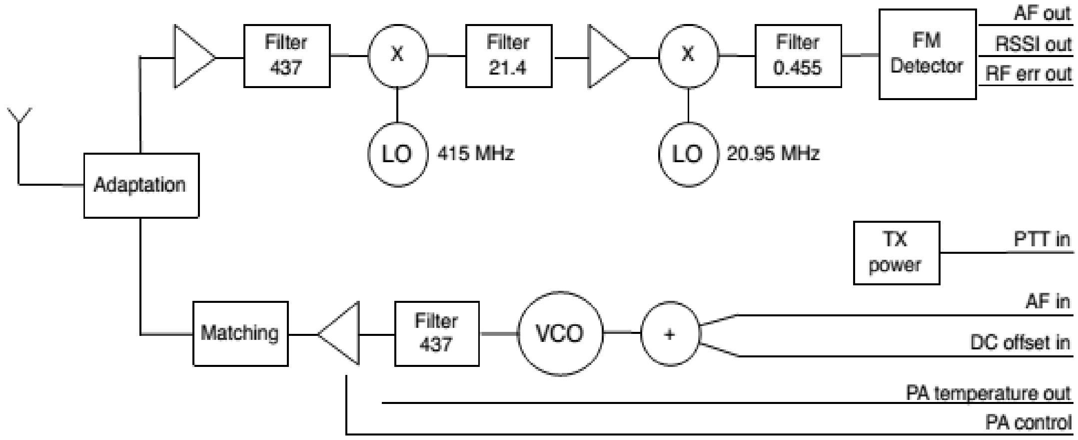


Figure 7-2: NanoCom RF design

7.1.5 Baseband

The baseband processing is done by an AFSK modem allowing 4800 baud half-duplex operation. The transmitter supports 9600 baud, which means baseband bandwidth of less than 20 kHz without distortion.

7.1.6 Package format

The NanoCom does not use AX.25 encoding of data, but implements modern and proven methods from the CCSDS suite of communication protocols for space communication. This approach provides two distinct advantages over AX.25:

- A 32bit sync words is applied over the 8bit in AX.25 providing a much higher success rate for data-link synchronization.
- Data is encoded with forward error correction (FEC) and this means the data-link is much more robust and reliable. Especially this will help optimize operations in an environment with many satellites located close to each other operating in the same frequency band.
- FEC also increases the effective power efficiency of the transmitter, saving power for other mission critical systems without sacrificing link quality.

In addition the NanoCom uses the CubeSat Space Protocol that provides additional benefits over a typical AX.25 UI implementation for telecommand and telemetry:

- Packets are routable to any Gomspace system and commands can be sent directly without involvement from the OBC, allowing for the utmost system control and diagnostic ability.
- Packets do not require alternate formats to be passed through the radio link, saving time and complexity of the system and OBC processes.
- Uplink packets can be encrypted and authorized, allowing safer control of the satellite.
- Firmware updating is available for all boards directly through the radio link.

7.2. ANT 430

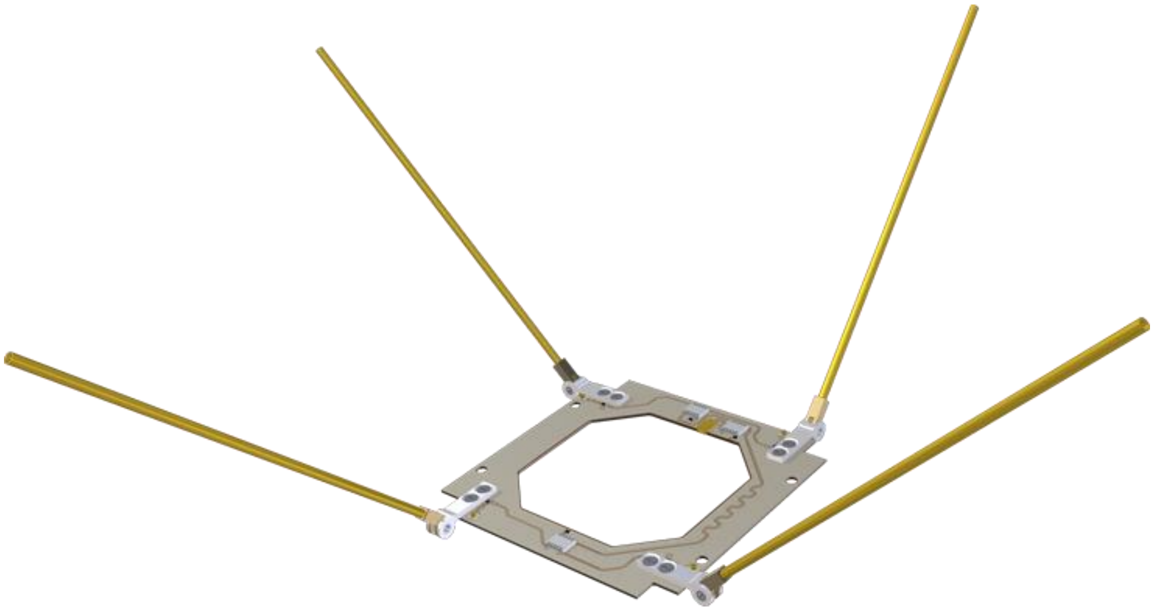


Figure 7-3: Gomspace ANT430 UHF turnstile antenna

7.2.1 Main Features

- Omnidirectional Canted Turnstile CubeSat Antenna
- Gain: 1.5 dBi to -1 dBi
- 400 - 480 MHz
- Rigid antenna tubes (no risk of antenna deformation while stowed)
- Matched to 50 Ω
- IPC-A-610 Class 3 assembly

7.2.2 Description

The turnstile antenna system consists of four monopole antennas combined in a phasing network in order to form a single circular polarized antenna. The antenna radiation pattern is close to omnidirectional and there are no blind spots which can cause fading with tumbling satellites.

The antenna PCB is open in the center to provide access to the QB50-required access ports on the +Z face of the satellite. This is where the FPP will be mounted.

7.2.3 Frequency

The antennas can be tuned to work with any frequency in the range from 400 to 550 MHz, depending on what frequency is allocated by IARU for use with the mission.

7.2.4 Polarization

The antenna is circularly polarized when seen from the top (left hand) and the bottom (right hand).

7.2.5 Connector

The antenna has one SSMCX connector for RF coax attachment to the NanoCom.

7.2.6 Gain

The antenna is designed with the goal of avoiding dead-spots in the radiation pattern making it close to omnidirectional. The actual gain characteristics depend on the shape of the spacecraft and its deployables. A radiation model was constructed using all structural elements, the needles for the mNLP payload, and a conductive boom length (either the material itself, or the ground wire running to the DFGM at the end of the boom) to best approximate the pattern given off by the antenna. The Numerical Electromagnetics Code was used to make the patterns.

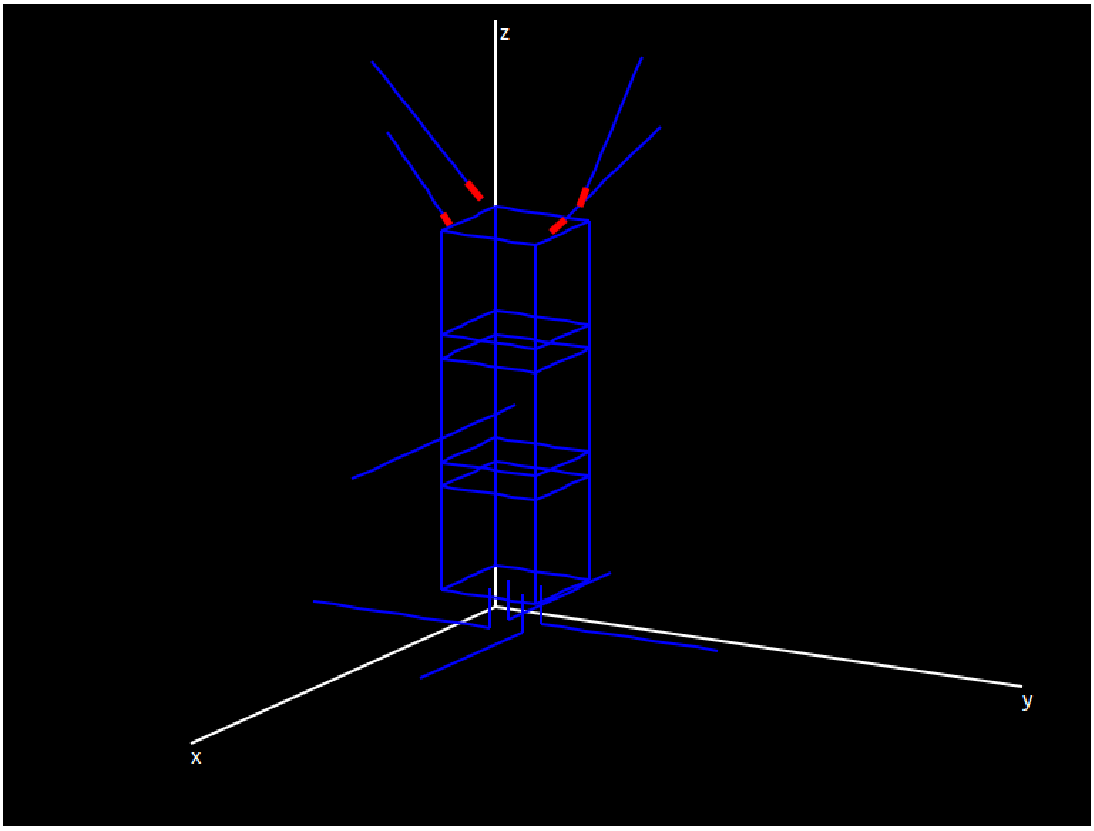


Figure 7-4: Conductive elements of the structure used in the radiation pattern analysis. Note that the reference frame is not equivalent to the previously mentioned frames in this document. The boom is parallel to the x axis, with the body parallel to the z axis.

Figures 7-5 and 7-6 detail the radiation pattern as seen along the boom axis and along the body axis. While the ram face maintains a node of relatively poor gain, the rest of the pattern provides a useable gain. The boom and MNLP will likely affect communications at the very start of a ground station pass, but the link and power budget has adequate tolerances to allow for an increase in radiated power should the radiation pattern be worse off than expected.

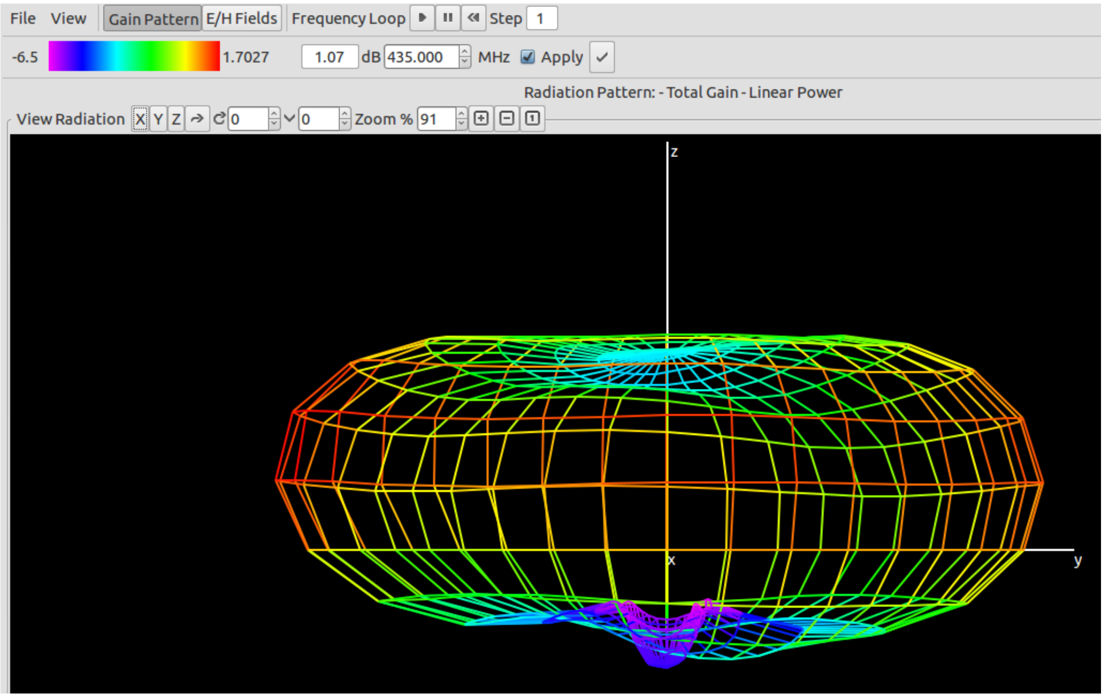


Figure 7-5: Radiation pattern in yz plane.

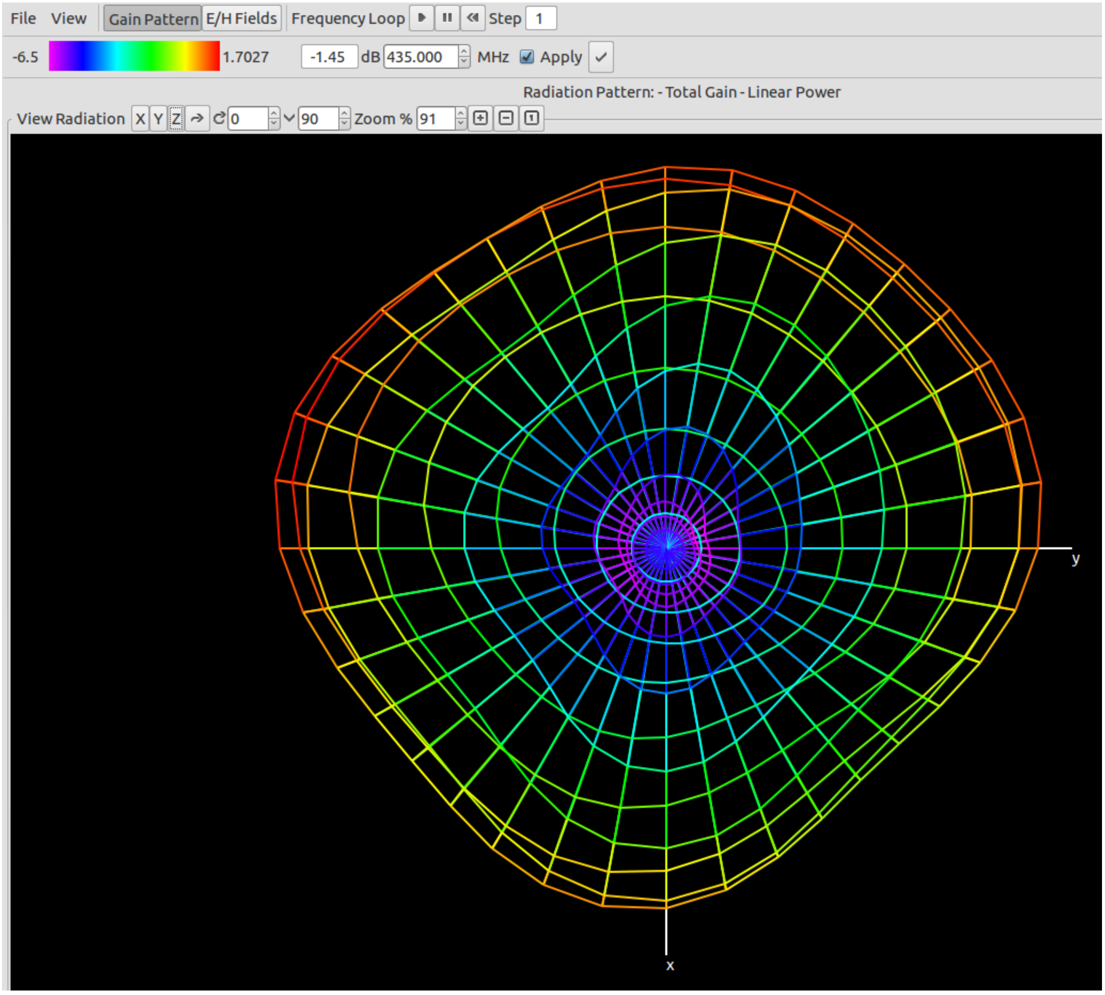


Figure 7-6: Radiation pattern in the xy plane.

7.2.7 Deployment

All four antenna elements are individually mounted on torsion-spring loaded hinges which, when released, rotate the antenna elements to an angle of 45 degrees above the PCB. The spring is only tensioned to appropriate half its safe rating in stowed mode and it is thus safe to keep the antennas stowed indefinitely without effecting the reliable deployment.

7.2.8 Active deployment

The antennas will be deployed using an active burn wire mechanism. Individual release systems are fitted for each antenna element on each face using the interstage panels and a small wire deployment PCB. The antenna element is tied to the deployment mechanism using Dyneema® and is tied over redundant burn resistors and spring loaded to ensure a tight fit, even during vibrations. The small deployment PCB also includes a micro switch to sense deployment with connection to the knife systems on the NanoHub.

7.3. Ground Station

Ex-Alta 1 will be controlled from a ground station set up on the U of A campus. The best location being investigated is on the south campus farms, providing clear horizons and low ground clutter. A simple trailer or container will be used to securely house the radio equipment and provide an elevated mount for the tracking yagi antenna. High-speed internet is provided through the university for remote operation of the station as well as upload and download of data to and from QB50 servers.

7.3.1 Antenna

A circularly polarized yagi-uda antenna will be used, with at least 14 dB of gain and an azimuth-altitude tracking mount. This will provide the necessary half-duplex operation in the UHF band.

7.3.2 Radio

The radio being considered is an ICOM 9100 or Kenwood TS2000, both of which come fully equipped to handle all aspects of satellite command including Doppler shift compensation. The radio will accept data packets using a modem provided by Gomspace.

7.3.3 Software

Ex-Alta 1 will make use of custom ground station software that will interface with a Gomspace CSP client. This will allow for uploading of commands, downloading of housekeeping and science data, and autonomous downlinking using orbit prediction updated with TLE's and GPS data from the satellite. Collaboration of the development is being investigated with the SUSat team from Australia.

7.3.4 Data Budget

An interactive data budget shows the various bandwidth used by each mode of encryption. The budget was calculated using a 10 degree minimum elevation and 60 second minimum duration. STK was used to calculate the estimated ground station visibility duration for one, two and three ground stations. One ground station will be housed at the U of A, and the other will be operated by the SUSat team in Australia. A third possible station was included for a complete analysis based on the yet to be confirmed status of the usage agreement. Frame efficiencies based on error correction and protocol are used to find the total percentage of available visibility used. It can be seen that with full forward error correction the data requirements are met using two ground stations until the orbit decays below 350 km altitude. At 200 km the efficiency of the error correction headers allocates too much data for downlink, but at a lower altitude the two modes of active error correction can be turned off separately as needed to ensure the data requirements are met. At a lower altitude the increased power available at the ground station will help negate the loss of full error correction capabilities.

7.3.5 *Downlink Telemetry Summary Budget*

Ex-Alta 1	NOTE:	
Downlink Telemetry Budget:		
Parameter:	*Value:*	*Units:*
Spacecraft:		
Spacecraft Transmitter Power Output:	1.0	watts
In dBW:	0.0	dBW
In dBm:	30.0	dBm
Spacecraft Total Transmission Line Losses:	1.9	dB
Spacecraft Antenna Gain:	2.0	dBi
Spacecraft EIRP:	0.1	dBW
Downlink Path:		
Spacecraft Antenna Pointing Loss:	6.1	dB
S/C-to-Ground Antenna Polarization Loss:	0.2	dB
Path Loss:	148.4	dB
Atmospheric Loss:	1.1	dB
Ionospheric Loss:	0.4	dB
Rain Loss:	0.0	dB
Isotropic Signal Level at Ground Station:	-156.1	dBW
Ground Station (EbNo Method): ***------- Eb/No Method -------***		
Ground Station Antenna Pointing Loss:	0.2	dB
Ground Station Antenna Gain:	14.1	dBi
Ground Station Total Transmission Line Losses:	1.7	dB
Ground Station Effective Noise Temperature:	370	K
Ground Station Figure of Merit (G/T):	-13.3	dB/K
G.S. Signal-to-Noise Power Density (S/No):	58.9	dBHz
System Desired Data Rate:	9600	bps
In dBHz:	39.8	dBHz
Telemetry System Eb/No for the Downlink:	19.1	dB
Demodulation Method Seleted:	**AFSK**	
Forward Error Correction Coding Used:	**Conv. R=1/2,K=7 & R.S. (255,223)**	
System Allowed or Specified Bit-Error-Rate:	1.0E-05	
Demodulator Implementation Loss:	1	dB
Telemetry System Required Eb/No:	5.5	dB

Eb/No Threshold:	6.5	dB
System Link Margin:	**12.6**	dB
Ground Station Alternative Signal Analysis Method (SNR Computation):		
---------- SNR Method ------------		
Ground Station Antenna Pointing Loss:	0.2	dB
Ground Station Antenna Gain:	14.1	dBi
Ground Station Total Transmission Line Losses:	1.7	dB
Ground Station Effective Noise Temperature:	370	K
Ground Station Figure of Merit (G/T):	-13.3	dB/K
Signal Power at Ground Station LNA Input:	-144.0	dBW
Ground Station Receiver Bandwidth (B):	17500	Hz
G.S. Receiver Noise Power (Pn = kTB)	-160.5	dBW
Signal-to-Noise Power Ratio at G.S. Rcvr:	16.5	dB
Analog or Digital System Required S/N:	10.0	dB
System Link Margin	**6.5**	dB

Table 7-1: Ex-Alta 1 Downlink Budget

Assumptions

1. 400km circular orbit, 98.6 degree inclination
2. Calculated for worst case slant range at 10 degree elevation

7.3.6 Uplink Command Budget Summary

Ex-Alta 1 **Uplink Command Budget:**	NOTE:	
Parameter:	*Value:*	*Units:*
Ground Station:		
Ground Station Transmitter Power Output:	100.0	watts
In dBW:	20.0	dBW
In dBm:	50.0	dBm

Ground Stn. Total Transmission Line Losses:	2.4	dB
Antenna Gain:	16.3	dBi
Ground Station EIRP:	34.0	dBW
Uplink Path:		
Ground Station Antenna Pointing Loss:	0.3	dB
Gnd-to-S/C Antenna Polarization Losses:	0.2	dB
Path Loss:	148.4	dB
Atmospheric Losses:	1.1	dB
Ionospheric Losses:	0.4	dB
Rain Losses:	0.0	dB
Isotropic Signal Level at Spacecraft:	-116.5	dBW
Spacecraft (Eb/No Method): ***------- Eb/No Method ---- ---***		
Spacecraft Antenna Pointing Loss:	6.1	dB
Spacecraft Antenna Gain:	2.0	dBi
Spacecraft Total Transmission Line Losses:	1.6	dB
Spacecraft Effective Noise Temperature:	373	K
Spacecraft Figure of Merit (G/T):	-25.4	dB/K
S/C Signal-to-Noise Power Density (S/No):	80.6	dBHz
System Desired Data Rate:	4800	bps
In dBHz:	36.8	dBHz
Command System Eb/No:	43.8	dB
Demodulation Method Seleted:	**AFSK**	
Forward Error Correction Coding Used:	**Conv. R=1/2,K=7 & R.S. (255,223)**	
System Allowed or Specified Bit-Error-Rate:	1.0E-05	
Demodulator Implementation Loss:	1.0	dB
Telemetry System Required Eb/No:	5.5	dB
Eb/No Threshold:	6.5	dB
System Link Margin:	**37.3**	dB
Spacecraft Alternative Signal Analysis Method (SNR Computation): *---------- SNR Method ------------*		
Spacecraft Antenna Pointing Loss:	6.1	dB

Spacecraft Antenna Gain:	2.0	dBi
Spacecraft Total Transmission Line Losses:	1.6	dB
Spacecraft Effective Noise Temperature:	373	K
Spacecraft Figure of Merit (G/T):	-25.4	dB/K
Signal Power at Spacecraft LNA Input:	-122.2	dBW
Spacecraft Receiver Bandwidth:	17500	Hz
Spacecraft Receiver Noise Power (Pn = kTB)	-160.5	dBW
Signal-to-Noise Power Ratio at G.S. Rcvr:	38.2	dB
Analog or Digital System Required S/N:	10.0	dB
System Link Margin	**28.2**	dB

Table 7-2: Ex-Alta 1 Uplink Budget

Assumptions:

1. 400km circular orbit, 98.6 degree inclination
2. Calculated for worst case slant range at 10 degree elevation

8. Thermal Control Subsystem

8.1. Subsystem Constraints

The following table summarizes the thermal budget of Ex-Alta 1. The simulation method used is described below.

Component	Operational Temperature			
	Design Minimum [°C]	Design Maximum [°C]	Calculated Minimum [°C]	Calculated Maximum [°C]
Mechanical Subsystem				
3U STS	-40	80	TBD	TBD
Interstage Panels	-40	80	TBD	TBD
FPP	-40	80	TBD	TBD
Deployable Boom	TBD	TBD	TBD	TBD
Command and Data Handling Subsystem				
NanoMind A712D	-40	85	TBD	TBD
NanoHub	-40	85	TBD	TBD
Power Subsystem				
NanoPower P31uS	-40	85	TBD	TBD
BP4 Charge	-5	45	15.35	44.45
BP4 Discharge	-20	60	-2.65	25.05
Solar Panels	-40	85	TBD	TBD
Communications Subsystem				
NanoCom U482C	-30	60	TBD	TBD
ANT430	-55	100	TBD	TBD
Attitude Determination and Control Subsystem				
NovAtel OEM615	-40	85	TBD	TBD
Taoglas AP.10F	-40	85	TBD	TBD
Surrey Space ADCS	-10	60	TBD	TBD
Payload Subsystem				
mNLP	-20	60	TBD	TBD
DFGM board	-20	60	TBD	TBD
DFGM	-40	60	TBD	TBD

Yellow	Daylight Bounds
Blue	Eclipse Bounds
Pink	Value is not within ±5 degree C marginal limits

Table 8-1: Thermal budget

8.2. Thermal Analysis Model

A 2-D nodal thermal model was developed to calculate the thermal gradients that will be encountered while in orbit. By employing a 2-D nodal analysis, each surface of the satellite in its respective 10x10x10cm unit forms a node ($T_{s1\text{-}6}$) for which the surface temperature is obtained and the various layers between the outer surface and the stack of boards are converted into resistance values for the thermal circuit.

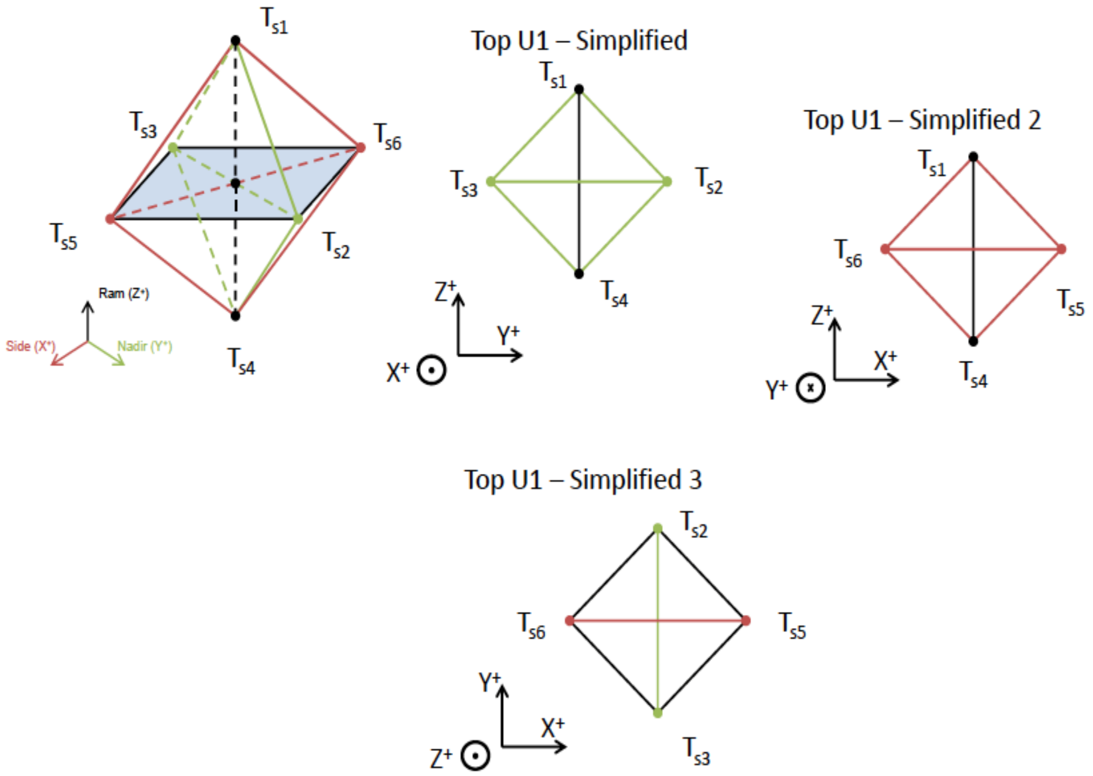


Figure 8-1: Simplified diagram of nodes used in thermal analysis.

Kirchhoff's Law for current was used to determine the equation for heat at each node, using the temperature difference between nodes. The satellite was broken down into its three 10x10x10cm sections to generate the appropriate equivalent thermal resistance circuit diagrams (Figure 8-2) for each 2-D set of coordinates. All diagrams were labeled with respect to satellite position/geometry as can be seen in the diagram.

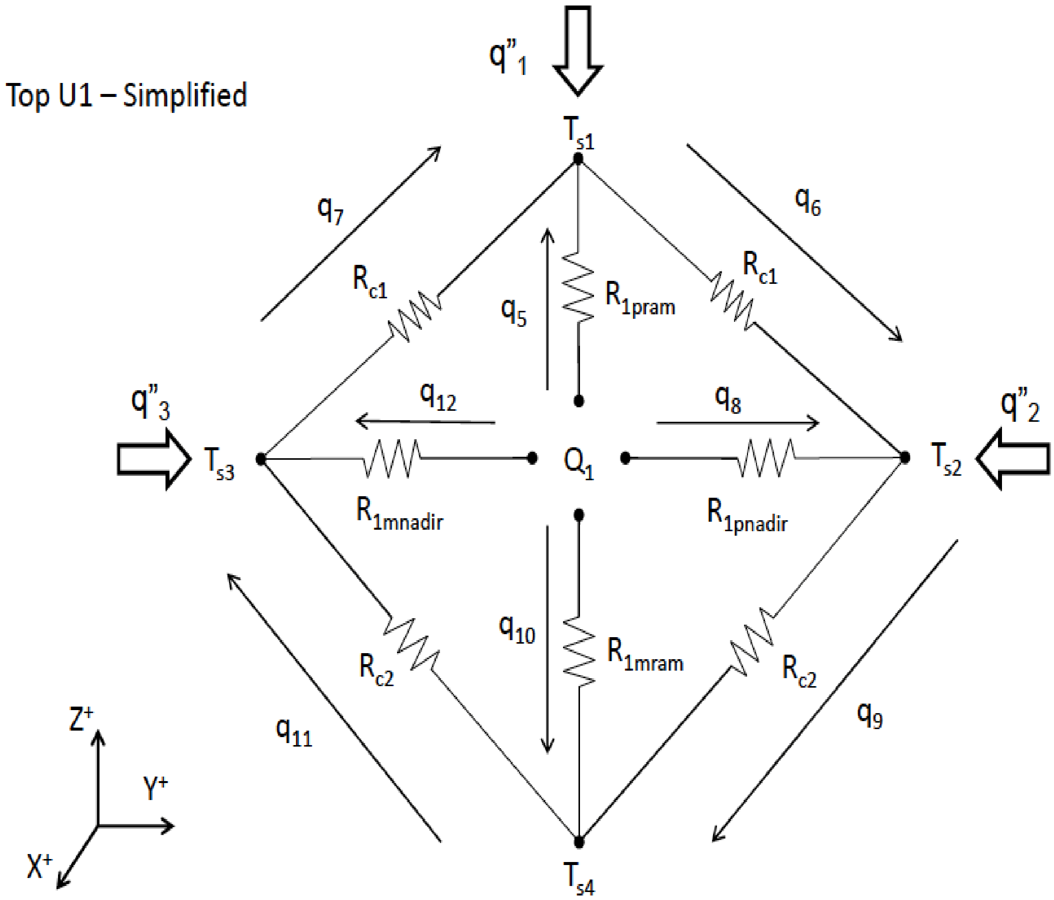


Figure 8-2: Thermal circuitry of unit 1 (this unit contains the batteries). Q1 (center) represents the heat generated by the batteries; $q''_{1\text{-}6}$ are the solar flux values at each face (4-6 are in other perspective); $R_{c1\text{-}2}$ are the resistances of the frame (R_{c3} is in the other perspective); $q_{1\text{-}12}$ are the heat transfers; $R_{1pram\text{-}mrma\ldots}$ are equivalent resistances on specific faces – p = plus, m = minus; $T_{x1\text{-}6}$ are the external spacecraft temperatures (outer top layer of solar panels).

For simplicity, this first nodal thermal analysis relies on the following assumptions:

Assumptions for 2-D Thermal Nodal Analysis

- Steady state
- Earth's albedo ignored (also to produce the worst possible cold case)
- Complete vacuum – therefore there is no convection and the only modes of heat transfer are radiation and convection
- Antenna mounts are minimal with respect to rest of the ram and antiram faces and can therefore be neglected.
- Rails were assumed to be pure conduction
- Boom (U2) section has no effect, as it mostly empty space when deployed. Thus most heat is lost to vacuum rather than transferred to the other Unit

- Calculated flux does not vary along the specified face of the satellite
- Emissivity of 1
- Calculated values for the heat generated by the battery are constant and known.
- Each stack of boards can be treated as a grey box with internal heat generation
- Heat generation for the first U is assumed to be the battery heat generation (for simplicity)

It is important to note that, following the completion of this model, a more specific simulation will be developed using FEA software to perform a 3-D analysis.

Using these assumptions, along with the equations for each node based on the thermal circuits, a system of equations was developed for each 10x10x10cm unit of the satellite. These equations also took into account the internal heat generated by each grey box as well as the calculated heat flux values from the STK orbital simulation data. This was then converted into a coefficient matrix of equivalent resistances, from which the spacecraft /external surface temperatures or the temperatures of the nodes could be solved. A Matlab script was then developed to solve this matrix and use the new values for the temperature nodes as inputs for the Euler method to converge on a solution for what the internal board stack temperature would stabilize to. Euler's method was chosen for this analysis since creating a Runge-Kutta 4^{th} order scheme would be more computationally expensive and require more coding time for a very small increase in accuracy.

Upon analyzing the thermal design specifications of each board/component (see Thermal budget), it was determined that the battery was the most critical component with the narrowest operational temperature range. The battery is designed to operate at a minimum temperature of – 5 °C and at a maximum temperature of 45 °C as stated by its corresponding data sheet obtained from the manufacturer. Thus, the output from this thermal analysis is the internal temperature range for both hot and cold cases, which will be compared to the design standards for the battery.

Assuming that 15% of the power will be converted into heat (the batteries have a listed charge/discharge efficiency of 85%), each mode has its amount of heat generation calculated from data obtained from the power budget for each part of the orbit (eclipse, recharge, sun). During the eclipse portion, it was assumed that the battery is only discharging whereas during the first portion of the sun side of the orbit, it recharges which produces the heat from recharge in addition to the heat from discharge.

The figures below indicate the heat generated by the batteries in the extreme orbit cases – 50% & 65% sunlight orbit (as specified by QB50) – for the two extreme modes of operation – safe mode and active mission (science) mode.

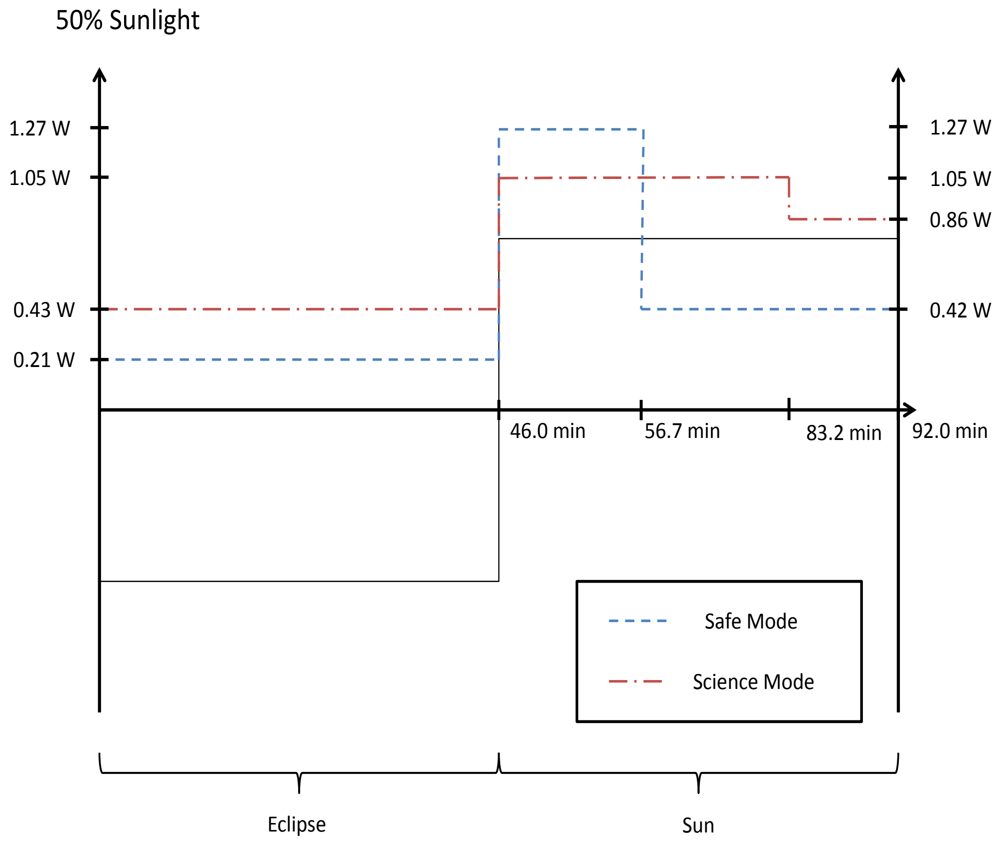


Figure 8-3: Power-time diagram for 50% sunlight (cold case).

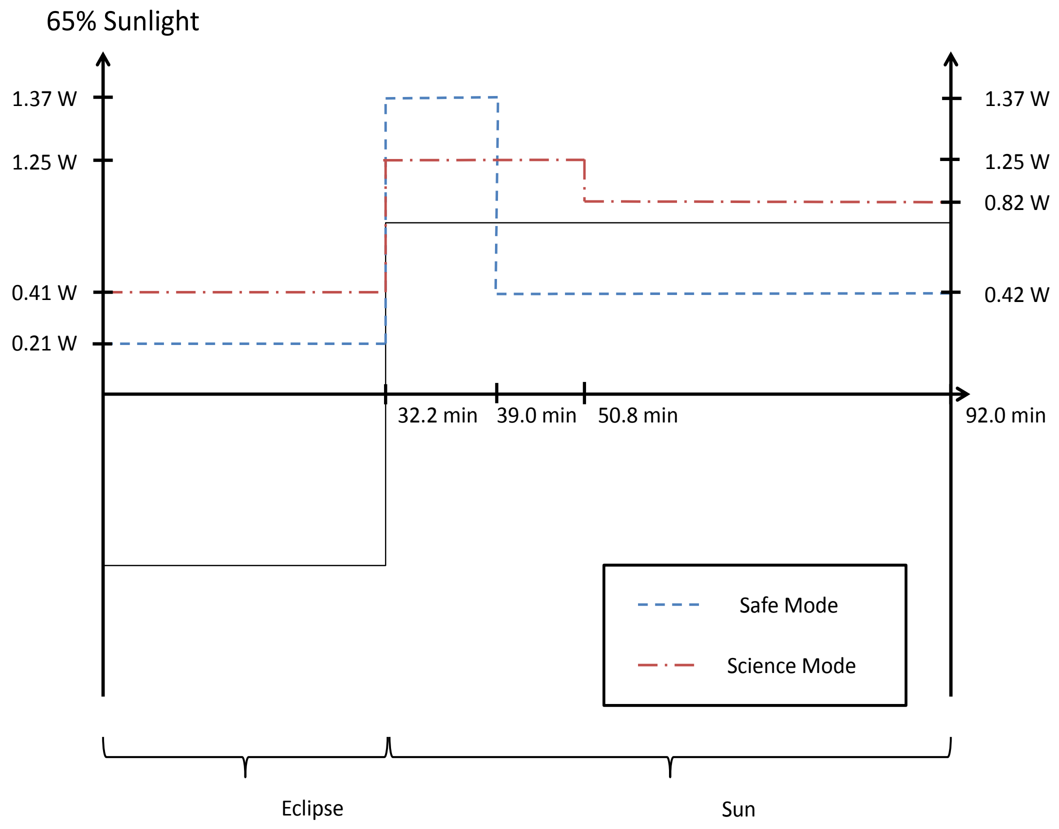


Figure 8-4: Power-time diagram for 65% sunlight (cold case).

The coldest case - shown in both figures above - results from 0.21W of heat produced by the batteries during eclipse in safe mode. The thermal fluctuation of the satellite for the first 12.5 hours of flight in this case is shown in figure 10-5 below:

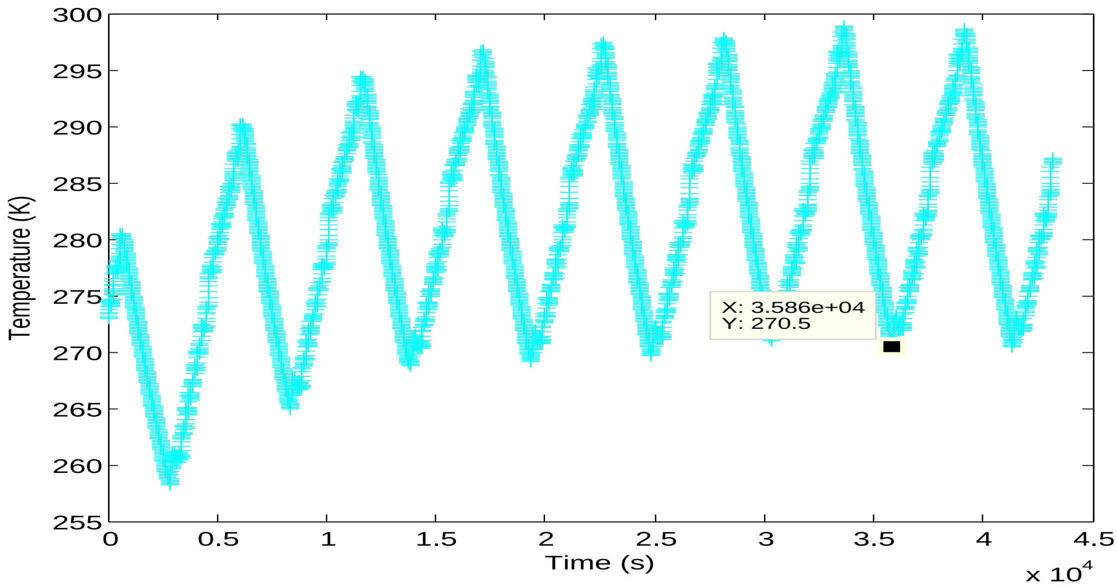


Figure 8-5: Thermal evolution of cold case, 50% sunlight.

Note that the initial temperature of the satellite was set to 273.15K (0 °C) in the simulation and there is a transient state. Steady state of the simulation is achieved at the 6th cycle approximately. T_{min} at steady state is indicated by the data point as 270.5K (-2.65 °C), which is within the marginal allowable temperature of the batteries (see thermal budget). T_{max} at steady state is 298.2kK (25.05 °C), which is also within the marginal allowable temperature of the batteries.

The hot case results from 1.37W of heat produced by the batteries in the 65% sunlight orbit while in safe mode. The thermal fluctuation of the satellite for the first 12.5 hours of flight in this case is shown in figure 10-6 below:

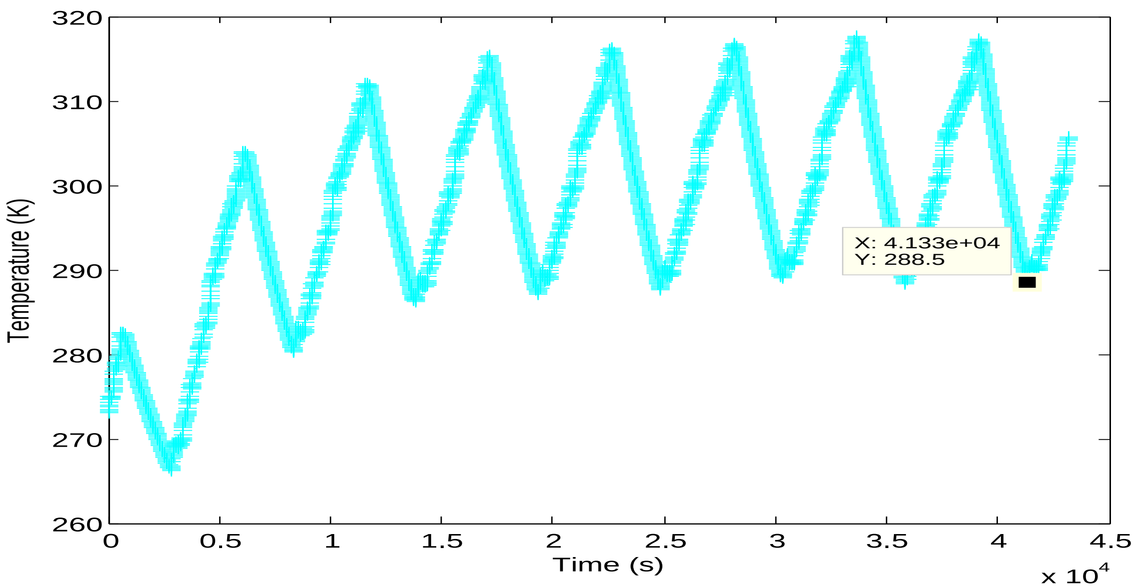


Figure 8-6: Thermal evolution of hot case, 65% sunlight.

Note once again that the steady state of the satellite is not reached until the 6th cycle approximately. T_{min} at steady state is shown by the data point as 288.5K (15.35 °C), which is within the marginal allowable temperature of the batteries. T_{max} at steady state is 317.6K (44.45 °C) which is within the design temperature of the batteries (see thermal budget), but not within ±5 °C margin.

In summary, these results are within the design specifications of the battery, but not within a marginal temperature of ±5 °C, hence some passive thermal control will need to be implemented. We will explore the use of insulation or heat pipe, or thermal coatings for components. Furthermore, the model will be refined to include the transient state and have more accurate assumptions (ie. examining individual boards and other components in the thermal budget), especially since the absolute maximum temperature from these results is rather close to the design maximum of the battery (45 °C).

9. Annex 1: CubeSat Design Overview Spreadsheet

Please see attached file "Annex 1 **Ex-Alta 1** _CDR_Overview_Spreadsheet.xlsx"

10. Annex 2: QB50 System Requirements Compliancy Matrix

Please see attached file "Annex 2 **Ex-Alta 1** _CDR_System_Requirements_Compliancy_Table.xlsx"

11. Annex 3: Science Sensor Requirements Compliancy Matrix

Please see attached file "Annex 3 **Ex-Alta 1** _CDR_MNLP_Compliancy_Table.xlsx"

12. Annex 4: Request for Waiver Form

Please see attached file "Annex 4 **Ex-Alta 1** Request for Waiver Form.docx"

13. Annex 5: QB50 CDR CA03 RID's

Please see attached file "Annex 5 QB50 CDR CA03 RID's.xlsx"