<u>Preliminary Design Review</u>

Akash Systems Team 61116

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Table 1: List of Acronyms

ADCNS	Attitude Determination, Control, and Navigation System
AOS	Acquisition of Signal
BPSK	Binary Phase Shift Keying
C/N	Carrier-to-Noise Ratio
CDH	Command and Data Handling
CDR	Critical Design Review
COTS	Commercial Off the Shelf Component
CSLI	CubeSat Launch Initiative
DC	Direct Current
DSN	Deep Space Network
EDU	Engineering Development Unit
EIRP	Equivalent Isotopically Radiated Power
EOL	End-of-Life
EPS	Electrical Power System
FSPL	Free Space Path Loss
GMSK	Gaussian Minimum Shift Keying
GPIO	General Purpose Input Output
GPS	Global Positioning System
ISS	International Space Station
I2C	Inter-Integrated Circuit
IP	Internet Protocol
JPL	Jet Propulsion Laboratory
LEO	Low Earth Orbit
LV	Launch Vehicle
NASA	National Aeronautics and Space Administration
OBC	Onboard Computer
PDR	Preliminary Design Review
P-POD	Poly-Picosatellite Orbital Deplorer
RF	Radio Frequency
RWA	Reaction Wheels
SLV	Separation from LV
SPI	Serial Peripheral Interface
T&C	Telemetry and Command
TBD	To Be Determined
TRL	Technology Readiness Level
UART	Universal Asynchronous Receiver/Transmitter
VCRM	Verifications Cross Reference Matrix

I. Storyboard

The life and mission of the spacecraft can be divided into three main sequences: The Start-Up Sequence, Payload Operations, and End-of-Life. In addition, the video of the storyboard that was presented in the PDR presentation can be viewed here: https://tinyurl.com/Akash-storyboard

a. Start-Up Sequence

1) Separation from LV *(Mission Elapsed Time: Separation from LV)*
The spacecraft is ejected from the launch vehicle into LEO. As it leaves the 6U Dispenser (P-POD), the deployment switches are unsuppressed which will turn on only the EPS and start the subsystem's clock. All other subsystems will remain powered off. The launch mechanism of the P-POD will exert a torque on the spacecraft and provide it with a slight tumble.

Figure 1: Akash-Sat is separated from the launch vehicle into LEO and has a slight tumble (green arrows).

2) Turn-On Subsystems *(Mission Elapsed Time: SLV + 30 minutes)*
The flight computer, ADCNS and payload are turned on 30 minutes after separation from the LV. The spacecraft will perform a pre-determined checkout of all its hardware to ensure proper functionality of all subsystems. There will be no radio frequency transmissions during this phase.

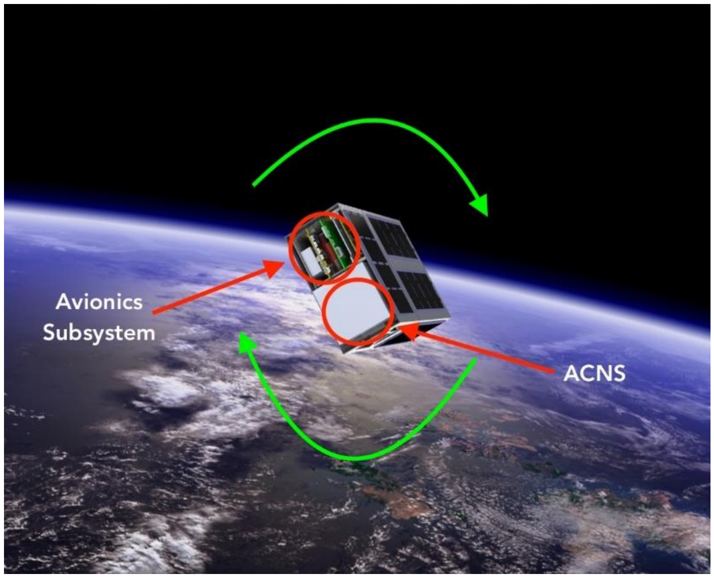


Figure 2: All subsystems except for communications are turned on. The spacecraft is tumbling (green arrows).

<u>3) De-Tumble</u> *(Mission Elapsed Time: SLV + 35 minutes)*
After a successful checkout of the ADCNS, the spacecraft will de-tumble and orient itself into a thermal and power safe attitude. To de-tumble, the spacecraft will use its magnetic torquers to dump the angular momentum provided to the spacecraft by the P-POD's launch mechanism. After its de-tumble maneuver, the spacecraft will orient itself so that it's +Y face is directed toward the sun, regardless of whether the sun is visible at that instant in the orbit.

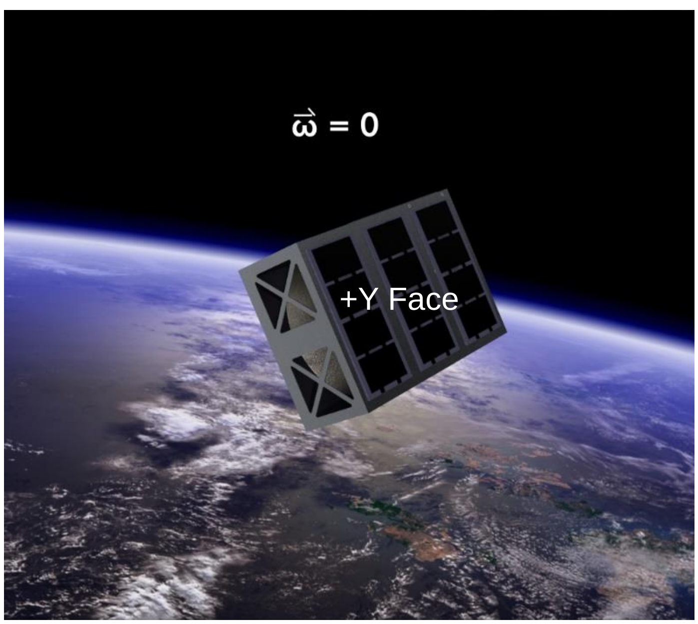


Figure 3: The spacecraft has successfully de-tumbled and is oriented in a thermal and power safe attitude. The +Y face is oriented towards the Sun.

4) Deployment of Payload Door *(Mission Elapsed Time: SLV + 40 minutes)*

The payload door on the –Y face of the spacecraft will deploy. The door will swing 180 degrees about its hinge on the edge that intersects the –Y and –Z faces. This will allow the payload's parabolic antenna to operate and orient the solar cells of the door in the +Y direction.

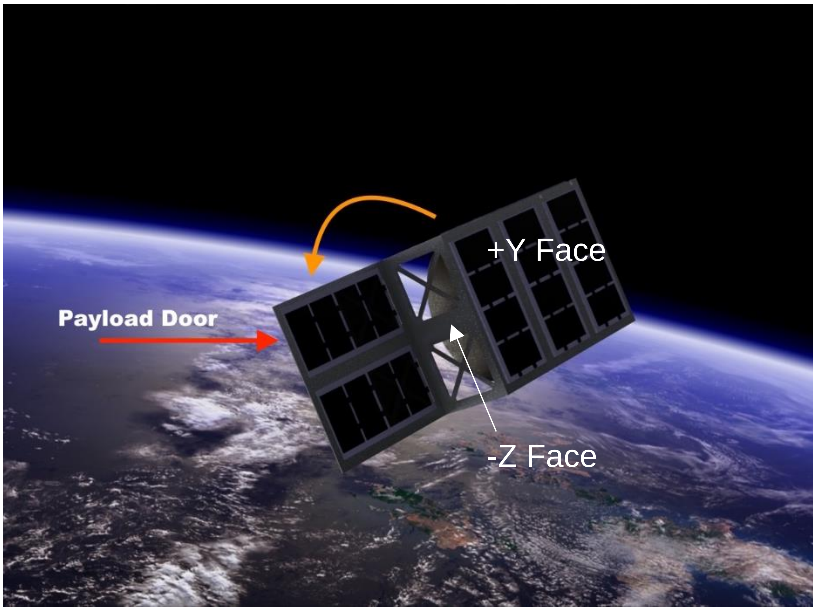


Figure 4: The payload door has been deployed (orange arrow) and now has its solar cells pointing in the spacecraft +Y direction, which is currently towards the sun.

5) Turn-On Communications *(Mission Elapsed Time: SLV + 45 minutes)*

The spacecraft's RockBlock radio will turn on 45 minutes after separation from the LV. The radio will communicate with the ground station by transmitting to the Iridium satellite constellation. By communicating with the Iridium constellation and utilizing Iridium's communication infrastructure, the mission operators will be able to communicate with the spacecraft through the internet. The initial communication between the spacecraft and mission operators will be a series of pings.

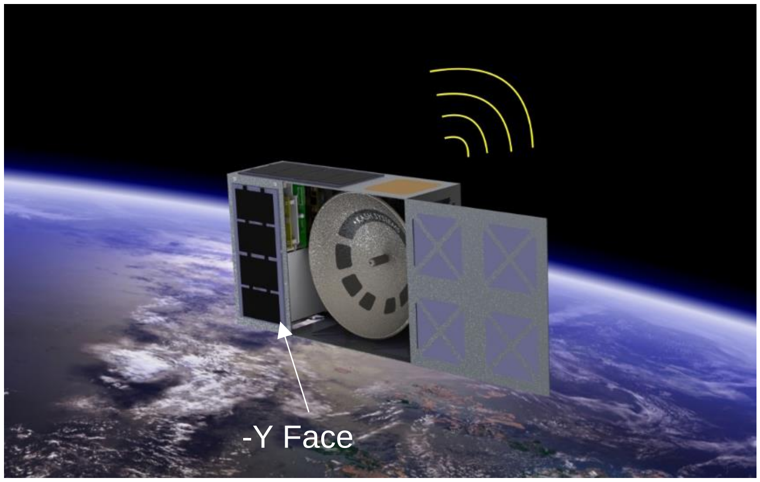


Figure 5: The spacecraft's radio is turned on and has begun transmitting to the Iridium constellation to communicate with mission operators on the ground.

6) Health Check *(Mission Elapsed Time: SLV + 50 minutes)*
After successful communication is established between the spacecraft and mission control, the spacecraft will transmit data on its health and status. The spacecraft will confirm the success of the checkout of its subsystems, de-tumbling, re-orientation into a thermal and power safe attitude, and deployment of the payload door. If the success of one or more of these events is not confirmed, the mission operators will send the appropriate manual commands to the spacecraft to complete them.

b. Payload Operations

7) Initial Payload Demonstration *(Mission Elapsed Time: SLV + 1 day)*
The spacecraft will coast in its orbit as it approaches the section of its orbit for initial payload demonstration. Seconds before reaching this section, the spacecraft will use its reaction wheels to provide it with a constant angular speed that will allow the payload's antenna boresight to remain pointed at the Ground Segment during the transmission section. Once the spacecraft reaches this section with the appropriate constant angular speed, it will downlink a known packet to the Ground Segment at any data rate. **At this point, the minimum mission success criteria will be met.** After passing this section of the orbit, the spacecraft will dump all its angular momentum and then re-orient itself into its thermal and power safe orientation.

8) Partial Payload Demonstration *(Mission Elapsed Time: SLV + 4 days)*
The spacecraft will coast in its orbit as it approaches the section of its orbit for partial payload demonstration. Seconds before reaching this section, the spacecraft will use its reaction wheels to provide it with a constant angular speed that will allow the payload's antenna boresight to remain pointed at the Ground Segment during the transmission period. Once the spacecraft reaches this transmission section with the appropriate constant angular speed, it will downlink a known packet to the Ground Segment at a 100 Gbit/s data rate for

the entire duration of the section. **At this point, the partial mission success criteria will be met.** After passing this section of the orbit, the spacecraft will dump all its angular momentum and then re-orient itself into its thermal and power safe orientation.

9) Full Payload Demonstration *(Mission Elapsed Time: SLV + 7 days)*
The full payload demonstration will consist of repeated transmissions at a 100Gbit/s data rate from the payload to the Ground Segment over a period of three months. The process for transmissions is identical to that of the partial payload demonstration phase, the key difference being that instead of one transmission at a 100Gbit/s data rate there will be multiple transmissions at a 100Gbit/s data rate over a three-month period. **At the end of three months of repeated transmissions at a 100Gbit/s data rate, the full mission success criteria will be met.**

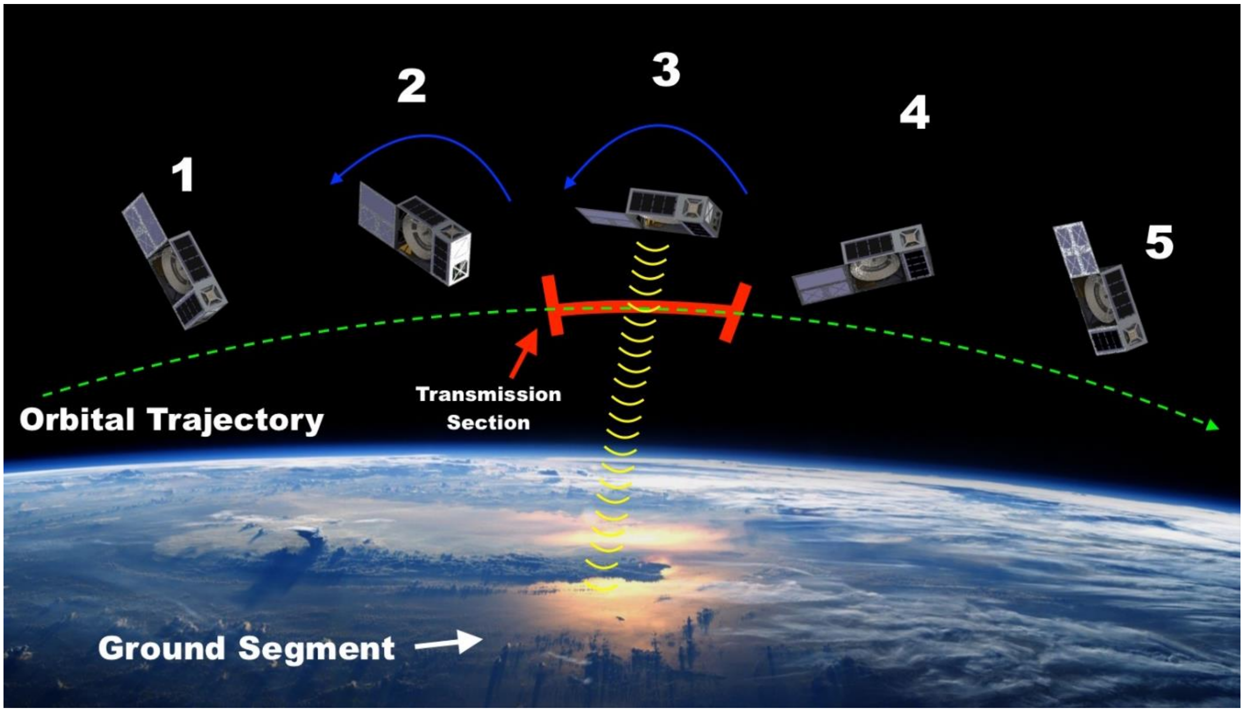


Figure 6: This is a depiction of one cycle of the payload demonstration. The spacecraft is (1) initially coasting with a fixed attitude, (2) then begins to rotate as it approaches the five-second orbit section, (3) transmits for five seconds, (4) de-tumbles and (5) reorients to a fixed attitude for coasting.

c. End-of-Life

10) End-Of-Life *(Mission Elapsed Time: SLV + 100 days)*
The final sequence in the life of the spacecraft will be activated by a command sent from the mission operators. After the three-month period of the full payload demonstration sequence and after the ‘kill’ command is received, the spacecraft will shut down all its subsystems except for the EPS. The spacecraft will re-enter and burn up in the atmosphere approximately three to six months after EOL.

II. Mission Success Criteria

MSC 1 Minimal mission success requires that the Akash Systems payload successfully downlinks a known packet to the Ground Segment.

> Minimal mission success verifies the ability of the Akash Systems payload to communicate with the Ground Segment at any data rate. This verifies that the payload radio can modulate a signal and that the payload power amplifier functions. Minimal mission success also verifies the ability of the Ground Segment to receive data from the on-orbit payload.

MSC 2 Partial mission success requires that the Akash Systems payload successfully demonstrates a 100 Gbit/s symbol downlink rate to the Ground Segment.

> Partial mission success verifies the ability of the Akash Systems payload to transmit a continuous stream of packets at the symbol downlink rate specified by the customer. Partial mission success verifies the successful interfacing of the payload and all peripheral subsystems during full functionality, and most crucially verifies that the spacecraft can provide the payload with sufficient power for full functionality.

MSC 3 Full mission success requires that the Akash Systems payload successfully maintains the ability to demonstrate a 100 Gbit/s symbol downlink rate to the Ground Segment over a 3-month mission duration.

> Full mission success verifies the ability of the Akash Systems payload to remain operational over an extended period of time in a space environment. Full mission success requires that the payload demonstrates functionality after 3-months. The concept of operations may include additional demonstrations of functionality in the time interval between the verification of partial mission success and the verification of full mission success.

III. Requirements Definition

a. System-Level Requirements

SYS 1 The spacecraft shall be a CubeSat [Customer Request].

SYS 1.1 The spacecraft shall interface with the 6U Dispenser.

SYS 1.2 The maximum mass of the spacecraft shall be 12.00 kg. [CP-CDS 3.2.9]

SYS 2 The spacecraft shall accommodate the Akash Systems payload during full functionality.

SYS 3 The spacecraft shall survive the launch environment.

SYS 4 The spacecraft shall be operational in any LEO for at least 3 months.

SYS 4.1 The spacecraft shall be operational in any LEO radiation environment for the duration of the mission. [SYS 4]

SYS 5 All spacecraft components shall remain within their survival temperature ranges at all times.

SYS 6 All spacecraft components shall remain within their operating temperature ranges during operation.

SYS 7 The spacecraft shall have a safe configuration for indefinite survival in any LEO.

SYS 7.1 The spacecraft shall have an indefinite thermal-safe state without ground intervention.

SYS 7.2 The spacecraft shall have an indefinite power-safe state without ground intervention.

SYS 8 Thermal sensors shall monitor the temperature of all spacecraft electronics.

SYS 9 The wiring harness shall be shielded from electromagnetic radiation.

SYS 10 The spacecraft shall verify the health of all subsystems.

SYS 11 The spacecraft shall communicate with the Ground Segment.

SYS 12 The spacecraft shall accept commands from the Ground Segment.

SYS 13 The spacecraft shall be single fault tolerant.

SYS 13.1 The spacecraft shall survive any single fault for a minimum of 24 hours without operator intervention. [Command Media]

SYS 14 Flight operations shall require no time-critical operator involvement for nominal operations. [Command Media]

SYS 15 All parts shall remain attached to the spacecraft during launch, ejection, and operations. [CP-CDS 3.1.1]

SYS 16 The spacecraft shall not exceed 100 Watt-hours of total stored chemical energy. [CP-CDS 3.1.5]

SYS 17 The spacecraft mission and design hardware shall be in accordance with NPR 8715.6 to limit orbital debris. [CP-CDS 3.4.3]

SYS 18 All spacecraft components shall re-enter with energy less than 15 Joules. [CP-CDS 3.4.3.1]

SYS 19 The spacecraft shall not utilize pyrotechnics. [CP-CDS 3.1.2]

SYS 20 All deployables shall wait to deploy a minimum of 30 minutes after the spacecraft deployment switches are activated during ejection. [CP-CDS 3.4.4]

SYS 21 The spacecraft shall not generate or transmit any RF signal for a minimum of 45 minutes after on-orbit deployment from the 6U Dispenser. [CP-CDS 3.4.5]

SYS 22 The spacecraft shall comply with all United States radio license agreements and restrictions. [CP-CDS 3.4.2]

SYS 23 All spacecraft hazardous materials shall conform to AFSPCMAN 91-710, Volume 3. [CP-CDS 3.1.6]

SYS 24 All spacecraft materials shall have a Total Mass Loss (TML) of 1.0% or less. [CP-CDS 3.1.7.1]

SYS 25 All spacecraft materials shall have a Collected Volatile Condensable Material (CVCM) of 0.1% or less. [CP-CDS 3.1.7.2]

SYS 26 The CubeSat shall be designed to accommodate ascent venting per ventable volume/area of less than 2000 inches. [CP-CDS 3.1.9]

SYS 27 The spacecraft shall be maintained in a class 100,000 level facility at all times.

b. Subsystem-Level Requirements

i. Power Requirements

PWR 1 PWR shall provide all subsystems with the necessary electrical power for the lifetime of the mission. [SYS 2, SYS 4]

PWR 1.1 PWR shall provide a peak power output of 325W. [PWR 1]

PWR 2 PWR shall control the powered state of all other subsystems. [SYS 2, SYS 10]

PWR 2.1 Power distribution shall be centralized. [Command Media]

PWR 2.2 Power regulation shall be centralized. [Command Media]

PWR 3 PWR shall monitor all distribution lines voltage. [SYS 10]

PWR 4 PWR shall monitor all distribution line currents. [SYS 10]

PWR 5 PWR shall directly interface with T&C. [SYS 10, SYS 11]

PWR 6 All solar panels shall have flyback protection diodes. [SYS 13]

PWR 7 PWR shall not exceed 100 Watt-hours of stored chemical energy. [SYS 16]

PWR 8 PWR shall be in a powered off state while integrated in 6U Dispenser. [CP-CDS 3.3.1, SYS 1.1]

PWR 9 PWR shall have at least one deployment switch. [CP-CDS 3.3.2, SYS 1.1]

PWR 9.1 Deployment switch shall electrically disconnect power system from powered functions. [CP-CDS 3.3.2.1, SYS 1.1]

PWR 9.2 Deployment switch shall be in actuated state at all times while integrated in 6U Dispenser. [CP-CDS 3.3.2.2, SYS 1.1]

PWR 9.3 The spacecraft shall reset to the pre-launch state if a deployment switch toggles from the actuated state. [CP-CDS 3.3.2.3]

PWR 10 PWR shall have a RBF pin. [CP-CDS 3.3.3]

PWR 10.1 RBF pin shall protrude no more than 10 mm from rails when fully inserted into satellite. [CP-CDS 3.3.3.1]

PWR 10.2 RBF pin shall be removed before integration into 6U Dispenser if the 6U Dispenser does not have access ports. [CP-CDS 3.3.3.2, SYS 1.1]

PWR 11 PWR shall incorporate battery circuit protection. [CP-CDS 3.3.4]

PWR 12 PWR shall have at least three independent RF inhibits. [CP-CDS 3.3.5]

ii. Telemetry and Command Requirements

T&C 1 T&C shall establish communication link with the Ground Segment. [SYS 11]

T&C 1.1 T&C shall transmit mission critical data. [T&C 1]

T&C 1.2 T&C shall have sufficient margin in its link budget for stable operation. [T&C 1]

T&C 2 T&C shall receive commands from the Ground Segment. [SYS 12]

T&C 3 T&C shall comply with FCC Part 97. [FCC Code of Federal Regulations Title 47,SYS 22]

T&C 4 T&C shall not generate or transmit any RF signal from the time of integration into the 6U dispenser through 45 minutes after on-orbit deployment from the dispenser. [SYS 21]

iii. Command and Data Handling Requirements

CDH 1 CDH shall execute commands from the Ground Segment. [SYS 12]

CDH 1.1 CDH shall reject all invalid commands. [CDH 1]

CDH 2 CDH shall provide the necessary computing power for all flight operations. [SYS 4]

CDH 3 CDH shall allow for communications between all spacecraft components. [SYS 2]

CDH 4 CDH shall monitor the health and status of all spacecraft subsystems. [SYS 10]

CDH 5 CDH shall recover from data communication single fault upsets. [SYS 13]

CDH 5.1 CDH shall survive any single fault without losing the ability to accept ground commands. [CDH 5]

CDH 6 CDH shall put the spacecraft in an indefinite safe-mode when off-nominal conditions are detected. [SYS 7]

CDH 7 CDH shall employ a watchdog timer. [SYS 13]

iv. Structure Requirements

STRUCT 1 STRUCT shall mechanically interface with all other subsystems. [SYS 1, SYS 2]

STRUCT 2 The spacecraft shall use the coordinate system as defined in Appendix A. [CP-CDS 3.2.1]

STRUCT 2.1 The origin of the spacecraft coordinate system shall be located at the geometric center of the spacecraft. [STRUCT 2]

STRUCT 3 The spacecraft configuration and physical dimensions shall be per the design specifications in CP-CDS 3.2.1.1. [CP-CDS 3.2.1.1, SYS 1]

STRUCT 4 The –Z face of the spacecraft shall be inserted first into the 6U Dispenser. [CP-CDS 3.2.2, SYS 1.1]

STRUCT 5 Components shall not exceed 10 mm normal to the outer surface of the spacecraft. [CP-CDS 3.2.2, SYS 1.1]

STRUCT 6 Deployables shall be constrained by the spacecraft, not the 6U Dispenser. [CP-CDS 3.2.4, SYS 1.1]

STRUCT 7 Rails shall have a minimum width of 8.5 mm. [CP-CDS 3.2.5, SYS 1.1]

STRUCT 8 Rails shall have a surface roughness less than 1.6 µm. [CP-CDS 3.2.6, SYS 1.1]

STRUCT 9 The edges of the rails shall be rounded to a radius of at least 1 mm. [CP-CDS 3.2.7, SYS 1.1]

STRUCT 10 At least 75% of the rail shall be in contact with the 6U Dispenser rails. [CP-CDS 3.2.8, SYS 1.1]

STRUCT 11 The spacecraft center of gravity shall be located within 4.5 cm from its geometric center in the X direction, within 2 cm from its geometric center in the Y direction, and within 7 cm from its geometric center in the Z direction. [CP-CDS 3.2.10]

STRUCT 12 Aluminum 7075, 6061, 6082, 5005, and/or 5052 shall be used for both the main spacecraft structure and the rails. [CP-CDS 3.2.11]

STRUCT 13 The spacecraft rails and standoff shall be hard anodized aluminum. [CP-CDS 3.2.12, SYS 1.1]

STRUCT 14 Redundant fasteners shall be used such that failure of a fastener does not cause a hazardous situation. [SYS 13]

STRUCT 15 The spacecraft structure shall be designed to withstand the forces and vibrations during launch. [SYS 3]

STRUCT 16 Epoxies, adhesives, or tape shall not be used to join structural components. [SYS 3]

v. Attitude Determination and Control Requirements

ADCS 1 ADCS shall control the orientation of the spacecraft in 3 DOF. [SYS 2]

ADCS 2 ADCS shall stabilize the spacecraft in three axes. [SYS 2]

ADCS 2.1 ADCS shall achieve pointing accuracy sufficient for full mission success. [ADCS 2]

ADCS 3 ADCS shall have positon and attitude knowledge in 6 DOF. [SYS 2]

ADCS 3.1 ADCS shall determine relative attitude sufficient for full mission success. [ADCS 3]

ADCS 3.2 ADCS shall determine absolute attitude for full mission success. [ADCS 3]

ADCS 4 ADCS shall provide accurate sensor measurements in the space environment. [SYS 4]

ADCS 5 ADCS shall report sensor measurements outside of the nominal range. [SYS 13]

ADCS 6 ADCS shall orient the spacecraft in a thermal and power safe orientation during safe mode. [SYS 7]

vi. Payload Requirements

PLD 1 PLD shall satisfy full mission success criteria. [SYS 2]

PLD 2 PLD shall interface with an external antenna. [SYS 2]

PLD 3 PLD shall occupy no more than 4U of volume. [SYS 1, SYS 2]

PLD 4 PLD shall not exceed a peak power draw of 300W. [PWR 1]

PLD 5 PLD shall mechanically fasten to the spacecraft structure. [SYS 2]

PLD 6 PLD shall survive the space environment for at least 3-months. [SYS 4]

PLD 7 PLD shall comply with FCC Part 97. [SYS 22]

PLD 8 PLD shall not affect the rest of the spacecraft in the event of a single fault. [SYS 13]

PLD 9 PLD shall not generate or transmit any RF signal for a minimum of 45 minutes after on-orbit deployment from the 6U Dispenser. [SYS 21]

V. Subsystem Analysis

a. Payload

i. Orbit & Lifetime

The spacecraft will be deployed in a LEO orbit with an altitude of 400km and inclination of about 52°. This is similar to the ISS orbit and is a common orbit for CubeSats. This orbit was selected because it is a common orbit meaning that the spacecraft is more likely to receive a launch from CSLI as launch providers are frequently delivering spacecraft to this orbit. In addition, this orbit provides frequent coverage of the ground segment and allows for re-entry in 160 days, mitigating the risk of space debris.

Full mission success requires that the payload operates for three months; accordingly, the spacecraft orbit must have a lifetime of at least three months before deorbiting occurs. Preliminary analysis utilizing tables suggested that Akash-Sat1 would remain in orbit for approximately five to six months. This initial estimate was confirmed and refined by the AGI STK model of Akash-Sat1.

The Lifetime tool was utilized to determine that the lifetime of the Akash-Sat1 spacecraft is 160 days. This verifies that the spacecraft can remain in orbit and operate for its full three month mission duration.

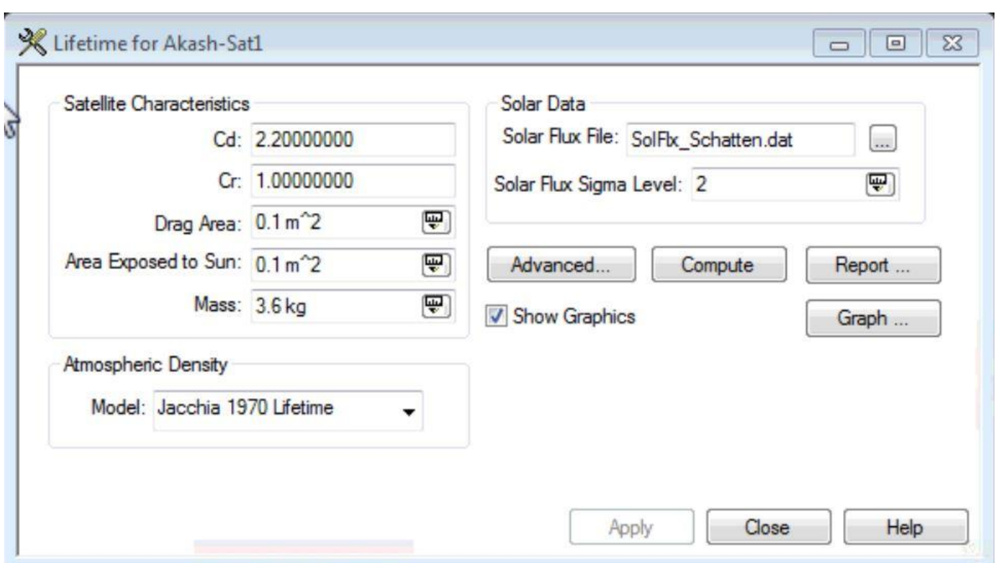


Figure 7: Screen capture of the inputs to AGI STK Lifetime tool for Akash-Sat1.

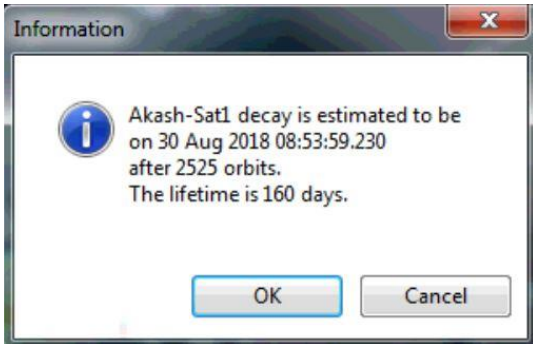


Figure 8: Output of the AGI STK Lifetime tool model of Akash-Sat1. The start date of the simulation was March 23, 2018.

ii. Payload Downlink Budget

Table 2: Payload Downlink Budget. Yellow indicates inputs. Blue indicates outputs. Green indicates margin. A Eb/N0 greater than 3dB verifies that the link budget closes.

	40GHz System MIN	40GHz System MAX	90GHz system
R (Data rate)	100,000,000,000	100,000,000,000	100,000,000,000
Bandwidth(MHz)	5,000	5,000	10,000
Modulation Scheme	QPSK	QPSK	QPSK
Pwr (Tx Power in W)	75.00	75.00	75.00
Pwr (Tx Power in dBm)	48.75061263	48.75061263	48.75061263
Ll (Line Loss in dB)	1	1	1
Antenna variation factor (in dB)	0	0	0
Gt (Tx Total Antenna Gain in dB)(60% eff.	36.30088715	36.30088715	43.34
EIRP	84.05149978	84.05149978	91.09061263
Raos (Dist to SAT @ AOS in km)	400	2292.771772	400
f (Desired Tx Frequency in GHz)	40	40	90
FSPL (Space Loss @ AOS in dB)	176.63	191.80	183.68
ISAB (Ionospheric Absorption Loss in dB)	1	1	1
Latmo (H20 & O2 Atmo-/Ionospheric)	1	1	15
Gr (Rx Antenna Gain in dB)	45.9	45.9	45.9
Power Received dBm	-48.68	-63.85	-62.69
Eb/No	15.60667162	0.440654926	1.602134113
Figure of Merit(G/T)(dB/K)	21.58636236	21.58636236	21.58636236
Eb/No Required (1e-05 BER)	10.00	10.00	10
Eb/No margin	5.61	-9.56	-8.40

iii. Space Segment

The payload consumes 300W of DC power for 5 seconds every 500 seconds at the minimum duty. See the power system sizing for more details on DC power consumption and supply. We were given that the spacecraft is to demonstrate up to 100Gbit/s data-rate, at a carrier frequency of 40GHz to 90GHz. The power amplifier was also given to be 25% efficient, providing an RF output power of 75W, or 48.75dBm. We were also given a bandwidth of 5GHz. Given that the payload will be supplied from the customer, an appropriate antenna system was posited. Using the standard circular parabolic antenna equation, a typical aperture efficiency of 60%, and a diameter of 0.195 meters (to fit within the 2Ux2Ux1U volume), a significant gain of 36.3dB is yielded. Such a high gain is due to the wavelength of the carrier being incredibly small relative to the size of the parabola. The aperture efficiency of high frequency systems could be lower than our assumed value.

See the link budget above for the payload system, for which all the gains and attenuations within the path of propagation have been described. With a typical minimized coaxial line loss of 1dB, the estimated isotropic radiative power is 84.05 dBm is found at a distance of 400km. The first two columns of the budget will show the maximum and minimum values of the system operating at 40GHs. The third column shows the least amount of FSPL for a 90GHz system. The first column shows a nominal orbital zenith distance of 400km, while the second shows the distance from a ground target 5 degrees off the horizon. Using the G/T and gain of the Deep Space Network's ground systems, the overall Eb/N0 margin is approximately 5.61 and -9.56 dB for the best and worst cases of a 40Ghz system respectively. When the margin is above 3dB, then the signal is at least twice the power of the noise floor given by the G/T figure of merit. This is known as the minimum detectable signal. We see that at 5 degrees of horizon, the 40GHz system does not make link. We also see that at the minimum FSPL for a 90GHz system at 400km, the link is not made. This calculation include the gain boost on the antenna from higher frequencies. Note that the atmospheric loss increases drastically.

iv. Ground Segment

Akash Systems has specified that they shall be using the DSN as their ground segment. They have also requested an operational duty cycle of at least 1%, representing a downlink period of 5 seconds every 500 seconds; however, the capability to meet this requirement and the maximum operational duty cycle will be limited by the DSN coverage of Akash-Sat1.

To determine the coverage, we modeled Akash-Sat1 in AGI STK and specified the ground stations as the 3 DSN stations located in the Mojave Desert of California (Goldstone) Madrid (Robledo), Tidbinbilla (Canberra). Assuming the orbit described above and utilizing the "Coverage" tool of STK over a 3-month duration, we found that Akash-Sat1 is under DSN coverage for an average of 11% of a given orbit. The average period of continuous coverage was further determined to be approximately 450 seconds. Accordingly, these parameters shall be used in all following analysis.

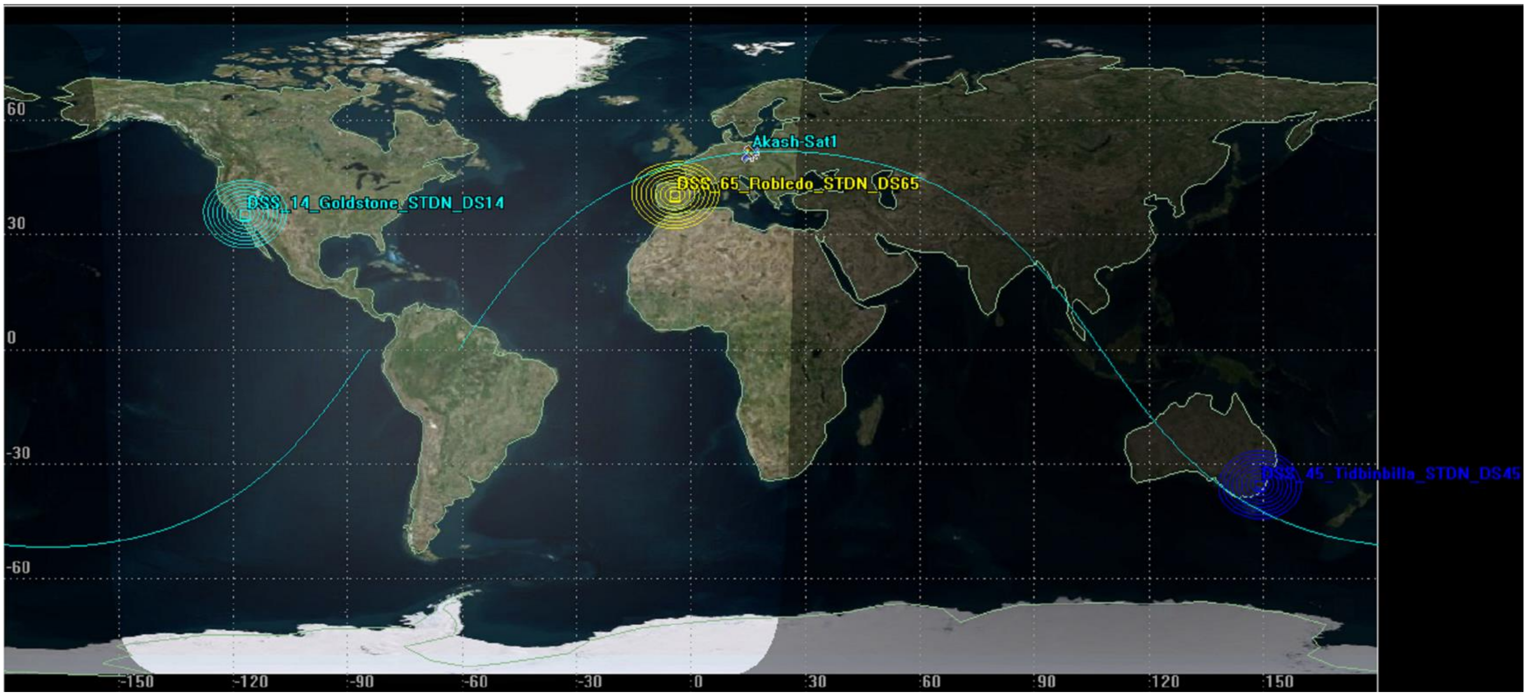


Figure 9: Screenshot of AGI STK Model of Akash-Sat1 and DSN. Please see the associated animation here: https://tinyurl.com/mjfactk

Table 3: Results of coverage analysis using data from AGI STK Coverage Tool

Average Downlink Per Orbit [s]	633.79071
Average Coverage Per Orbit	11.40%
Average Continuous Downlink Time [s]	458.63144

The plot below shows the Eb/N0 margin that the downlink has with the DSN as it passes over. If we want the Eb/N0 margin to always be 3dB above floor, we will be able to communicate from +45 degrees above the horizon. However, if the atmospheric attenuation fluctuates more than 4dB above the listed losses, then communication cannot be established.

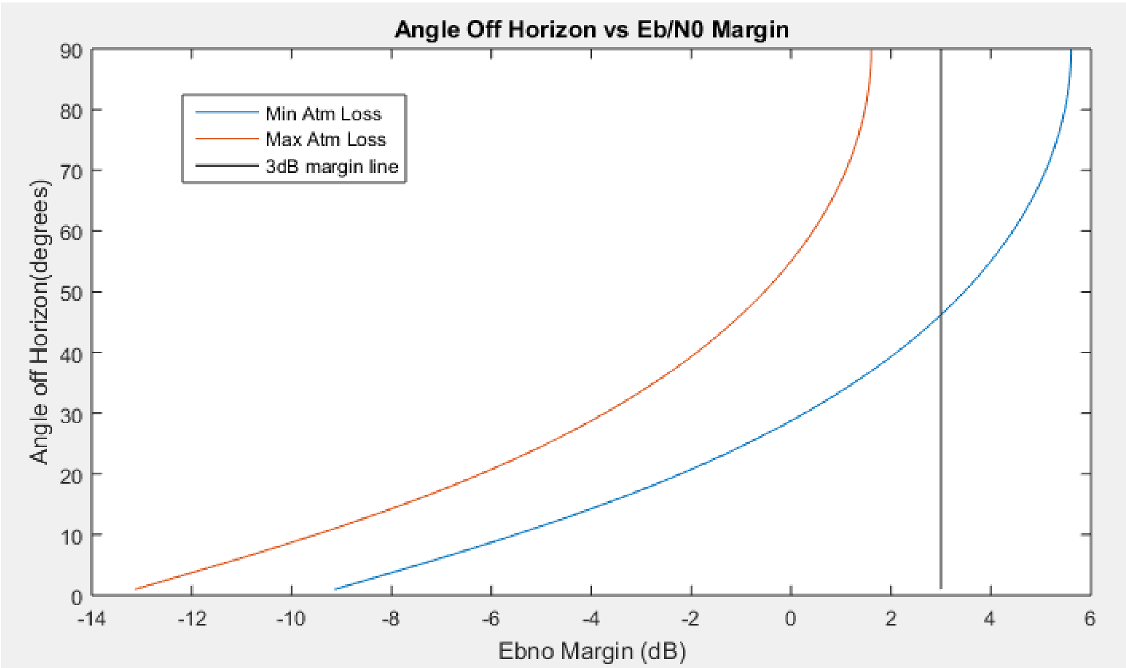


Figure 10: Plot of the Eb/N0 margin as the spacecraft appears over the horizon to the ground segment.

b. Power

i. Average Electrical Power Requirement

On average, the spacecraft's power consumption will be about 6.59W. This value was calculated based on our power budget average power consumption. Further details are shown in the power budget.

ii. Peak Electrical Power Requirement

The payload shall operate with a power of 300W for 5 seconds. This will occur about every 500 seconds, representing a 1% orbital duty cycle. However, the payload could operate anywhere from 1% to 11% duty cycle and still meet all requirements as described above. In addition to the payload, standard avionics will consume about 3.59W. See Appendix A and the battery trade studies for the calculations used to justify this power requirement.

iii. Spacecraft Configuration

The solar cells will be located on all sides of the CubeSat and on the outside face of the deployable door. In the best-case scenario, there will be 25 solar cells facing the sun. For the solar cells, triple junction solar cells with an efficiency of 30% were chosen.

iv. Trade Studies

Batteries: Since the payload will draw a large amount of power it is important to conduct a trade study on what type of batteries will be necessary to meet the power requirements. Based on calculations, the LG Chem 18650 MJ1 was chosen, as it can support up to a 10A continuous discharge. The battery has a nominal capacity of 3500mAh. The mathematical reasoning for this decision is shown in Appendix A. The GomSpace BPX Battery Pack will be used to house the LG Chem 18650 MJ1 batteries. The BPX Battery Pack does come with its own batteries, but those will be removed and replaced with the LG Chem batteries because those have a higher continuous discharge.

Deployable Solar Panels: Power calculations indicate that deployable solar panels will not be necessary to satisfy the power requirements. The current design features deployable solar panels in the form of a door for redundancy and increased capability.

Capacitors: Full power of the system is approximately 303.59W. For an assumed nominal voltage bus of 29.6V (total range is 25.6V to 33.6V) and payload transmission time of 5 seconds, the peak current draw for the system will be about 10.37A. Using the capacitor charging current equation, we can solve for the value of the capacitor necessary to prevent brown-out of the system

$$V_{lowest\ DC/DC} = V_{capacitor} e^{\frac{-t}{RC}}$$

If we assume the system is partitioned into the avionics and payload, where the payload is operated by the battery and the avionics by the capacitor, then the resistance of the avionics is given by

$$R_{partitioned} = \frac{28\ V}{0.25\ A} = 112\ \Omega$$

Where the current draw of the avionics stack is approximately 0.25A. Using this to solve for the capacitor value, assuming $V_{lowest\ DC/DC}$ to be about 7V, we can see that the capacitor would have to be about 0.0322F. Not only is it difficult to find capacitors rated up to 33.6V and have a capacitance of 0.03F, capacitors of this rating are very large (about the size of a soda can). Additionally, given that the capacitor would likely be electrolytic or electric double layer capacitors (EDLCs), it would need to be placed in its own box in space due to outgassing and thermal constraints. Given these demanding requirements, we will move forward with the solution of partitioning the power supply into two parts. We will explain this new architecture solution in the Power Architecture section.

v. Power Architecture

The block diagram below depicts the general architecture of the power system. To maximize efficiency of the system we must make sure to minimize power conversion as much as possible. This is especially important when switching loads as high as 300W. Given that Akash Systems specified that the payload would operate at standard aircraft supply of 28V (abiding by MIL-STD-1275D), we can readily operate the system directly off an 8S lithium-ion system. When the payload is operating, the avionics stack must still be operating. When the battery system is under extreme load, also possibly sinusoidal, it is best to isolate the avionics stack and have it operate on a different supply. Issues like browning out the main DC/DC converters and current robbing become significant and could lead to failure of the system. Therefore, the power system must be partitioned into two parts, one that will operate the payload, and one that will operate the avionics stack. It was shown in the capacitor section that implementing a capacitor bank system to run the main avionics stack would be bulky, require environmental control, and would significantly limit operations at larger duty cycles (greater than the minimum 1%). Consequently, the power system uses a battery-based design. To reduce modifications, component quantity, and cost required of the power system, two things must be ensured: keep the batteries at the same architecture and keep the two power supplies on the same charging rail. If both batteries are an 8S architecture then the maximum power-point tracking systems do not have to be different, and the power boards do not have to be modified. This is the same reason for keeping both batteries on the same rail.

We chose GomSpace BPX battery banks with all cells in series. As mentioned before, we will be replacing the standard GomSpace 18650 cells with LG MJ1 18650s. This is purely for increased current support for the payload, given that 10A will need to be supplied. In the future, we should consider other implementations of this 8S system, as the BPX supplies are nearly $9000 each. A standard CubeSat waiver will also need to be filled out due to exceeding the limit of 100Whr of stored chemical energy.

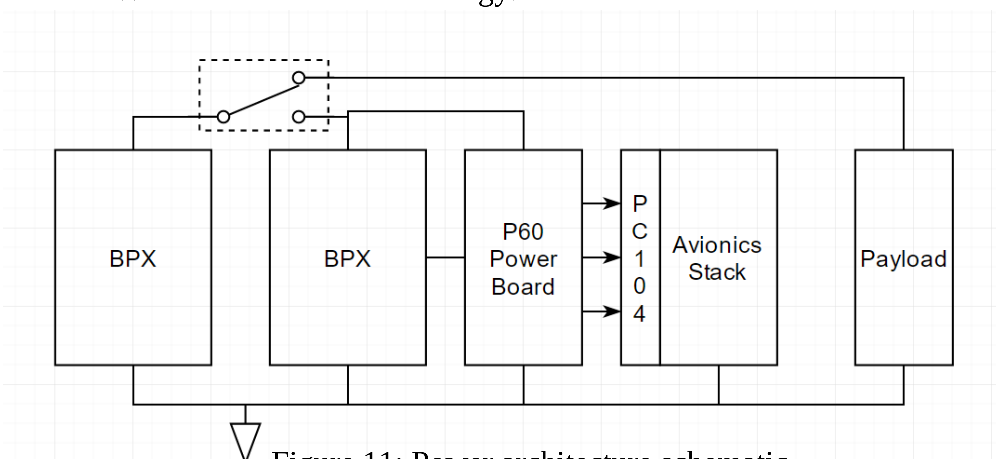


Figure 11: Power architecture schematic

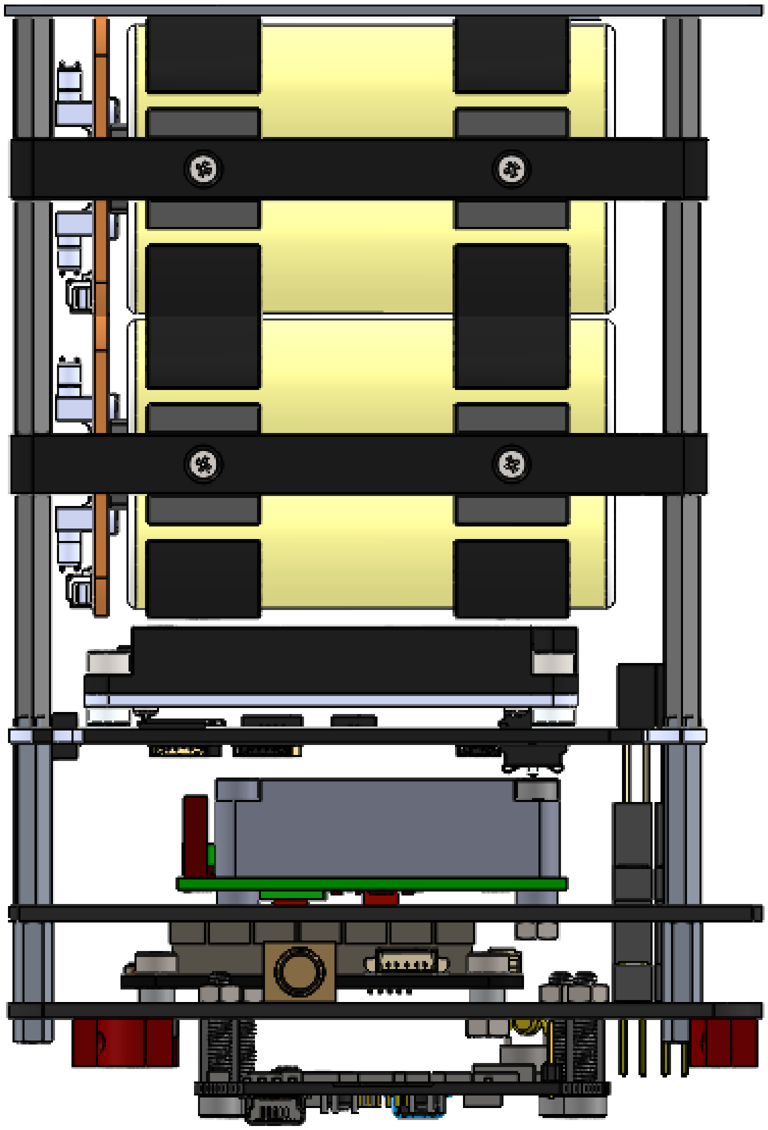


Figure 12: CAD Model of Avionics Stack. From top to bottom: BPX 1, BPX 2, P60 Power Board, Remainder of Avionics Stack (RockBlock, Piksi, BeagleBone Green FC)

vi. Power Budget

Table 4: Power Budget. All values are in units of Watts. See Appendix A for assumptions.

Power Budget						Day Mode		Eclipse Mode		Charge Mode		
Subsystems		Min. Power (W)	Max. Power (W)	Avg. Power (W)	Number Active	Avg. Duty Cycle	Watts	Avg. Duty Cycle	Watts	Avg. Duty Cycle	Watts	Notes
Power												
	Power Board	0.05	1	0.26	1	1	0.26	1	0.26	1	0.26	NanoPower P60 (custom rail)
T&C												
	Radio	0.5	2.25	0.5	1	1	0.5	1	0.5	1	0.5	RockBlock
CDH												
	Flight Computer	1.05	2.05	1.3	1	1	1.3	1	1.3	1	1.3	Beaglebone Green Wireless
ADCNS												
	Reaction Wheels	0.312	2.03	0.34636	3	1	1.0391	1	1.039	0	0	Maxon 32 EC Flat Motor
	Magnetic Torquers			0.25	3	0.2	0.15	0.2	0.15	0.1	0.075	Custom
	Sun Sensor	0.005	0.01	0.0075	5	1	0.0375	1	0.038	1	0.0375	Nano-SSOC-A60 Analog Sun Sensor
	IMU	0.00002	0.002628	0.01325	1	1	0.0133	1	0.013	1	0.0133	Invensense IMU
	GPS Receiver			2.9	1	0.1	0.29	0.1	0.29	0	0	Swift Piksi
Payload												
	Payload	0	300	300	1	0.01	3	0	0	0	0	
Other												
	Solenoid	0	12	0	1	0.009	0	0	0	0	0	
					TOTAL Watts Used		**6.5898**		**3.59**		**2.1858**	
						Eclipse Fraction	0.5					
				Note: Charge mode is entered only when batteries need to be charged								
				Orbit Avg. Consumption		5.08983						
				Orbit Avg. Power In *		21.40176						
				Contigency Factor		1						
				Power Budget Margin		15.31193						
		*This value was calculated assuming no sun pointing, however if we did have sun pointing this value would be 31.449.										

All satellite modes (launch, checkouts, payload demonstration, etc.) are taken into account by the day and eclipse mode and component power consumption is considered by the average duty cycle in day and eclipse mode.

vii. Solar Cells

To get more accurate power generation numbers we calculated the power generated based on the incidence angle to the sun. We used the equation

$$Power = Power\ Generated\ by\ Number\ of\ Solar\ Cells * \cos(angle\ to\ sun\ in\ radians)$$

where the power generated by each solar cell is 1.18675 W. Below are the calculations we used for the best and worst case scenario.

Table 5: Best Case Solar

Best Case Scenario: Front Facing Earth, Back Facing Sun	
Solar Panel	Number of Cells
-Y	25
+Y	5
where the power per cell is 1.18675 W	

Table 6: Best Case Solar Panel Power Generation

Solar Panel Power Generation		
Solar Panel	Efficiency	Power (W)
-Y, sun facing	30%	29.669
+Y, albedo	30%	1.780

Table 7: Best Case Power Yield

Power Yield with Incidence Angle to Sun (W)									
Angle	5	10	20	30	40	50	60	70	85
-Y	29.55585	29.21802	27.87951	25.69389	22.72758	19.0707	14.83438	10.14731	2.585802
+Y	1.773351	1.753081	1.67277	1.541633	1.363655	1.144242	0.890063	0.608839	0.155148
Total Power (W)	**31.3292**	**30.9711**	**29.55228**	**27.23552**	**24.09124**	**20.21495**	**15.72444**	**10.75615**	**2.74095**

Table 8: Worst Case Solar

Worst Case Scenario: Front Facing Sun, Back Facing Earth	
Solar Panel	Number of Cells
-Z	5
+Z	5
where the power per cell is 1.18675 W	

Table 9: Worst Csse Solar Panel Power Generation

Solar Panel Power Generation		
Solar Panel	Efficiency	Power (W)
-Z, sun facing	30%	5.934
+Z, albedo	30%	1.780

Table 10: Worst Case Power Yield

Power Yield with Incidence Angle to Sun (W)									
Angle	5	10	20	30	40	50	60	70	85
-Z	5.91117	5.843603	5.575901	5.138778	4.545516	3.814141	2.966875	2.029462	0.51716
+Z	1.773351	1.753081	1.67277	1.541633	1.363655	1.144242	0.890063	0.608839	0.155148
Total Power (W)	**7.684521**	**7.596684**	**7.248671**	**6.680412**	**5.909171**	**4.958383**	**3.856938**	**2.638301**	**0.672309**

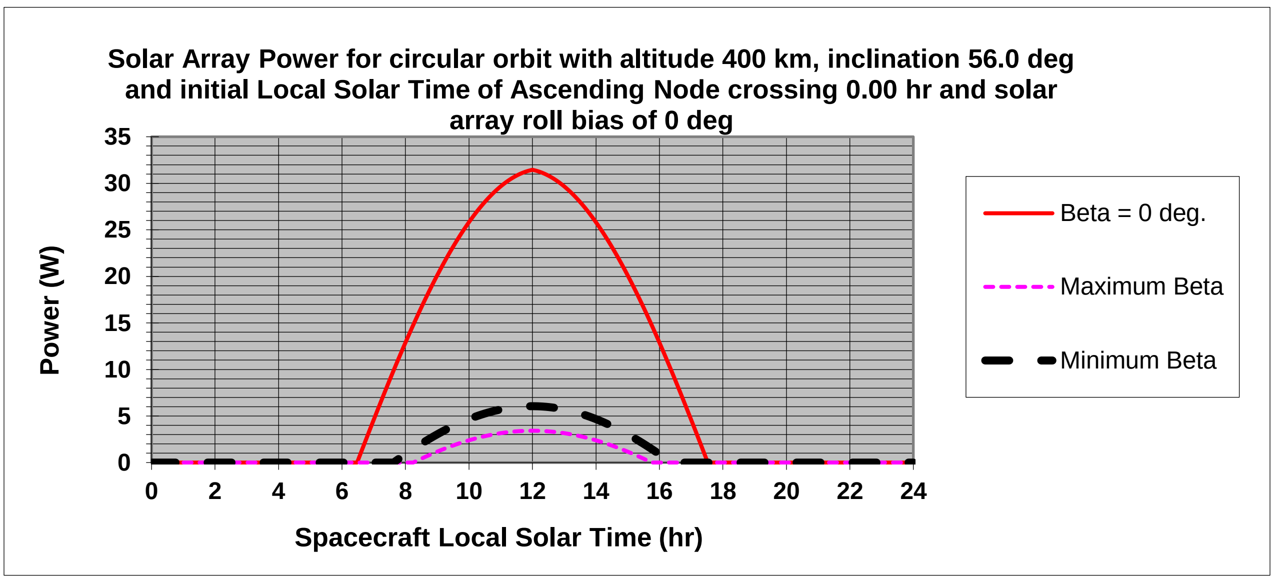


Figure 13: Solar Array Power

viii. Battery State of Charge

The plots below depict the evolution of the state of charge of the batteries throughout operation of the spacecraft. The first plot shoes the orbital duty cycle of 1%, or 5 seconds of operation every 500 seconds. The first plot assumes the spacecraft charges whilst nadir pointing. This means that the space-craft does not continuously point during the mission. This is an accurate calculation because the spacecraft will spend most of its time nadir pointing for the operation of its payload. The zero-degree beta angle will therefore be the best-case scenario for power generation. Below we can see that, with a 1% operational duty cycle, the spacecraft will most likely not need to enter a charging low-power regime for tens of orbits.

We then wanted to see the extent of payload operation that our spacecraft could withstand. Although our batteries are rated for down to 80% depth of discharge, we did not want to operate below 50%. Therefore, we calculated that an 11% operational orbital duty cycle brings the spacecraft to about this level after two orbits. Also, note that this 11% value is important because it is the average line-of-sight per orbit that the spacecraft has with the DSN. To charge the payload battery back up to the 7Ahr capacity, with assumed sun-pointing, it will take approximately 3.6 orbits.

The two plots below the state of charge plots show our power generation at different beta angles and the duration of eclipses respectively. Other beta angles will have longer durations in sunlight, but will not necessarily have the largest solar panel pointed to the sun.

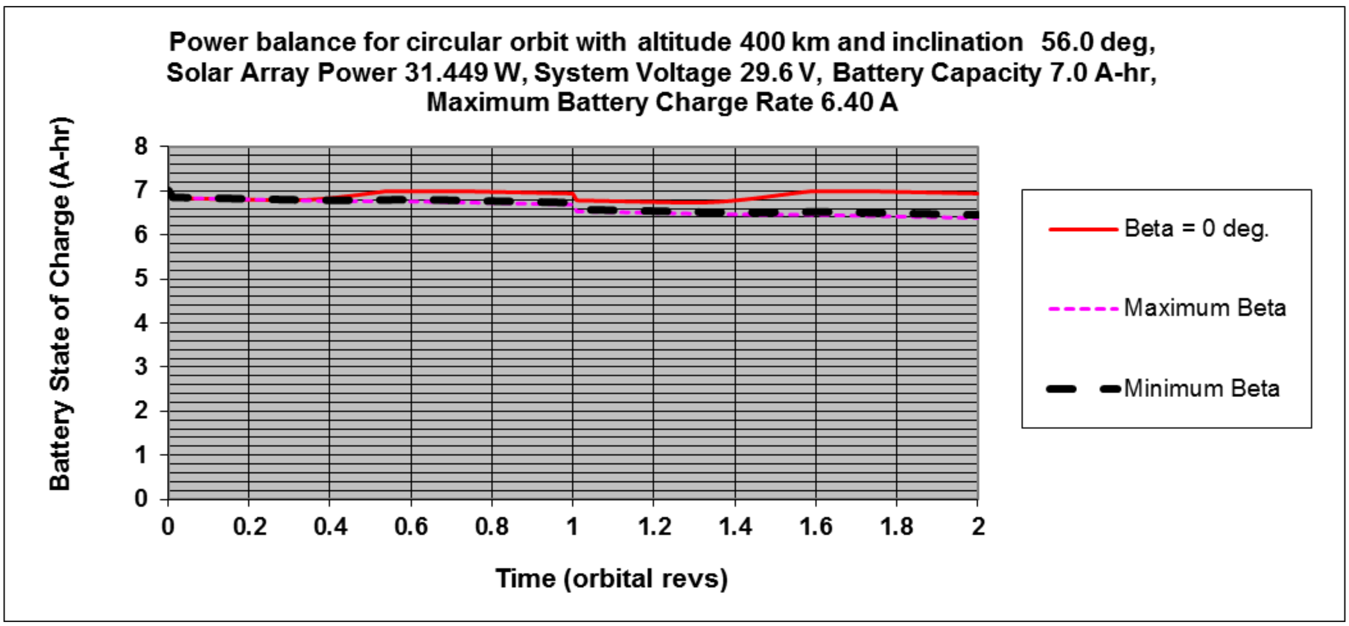


Figure 14: 1% Payload Duty Cycle

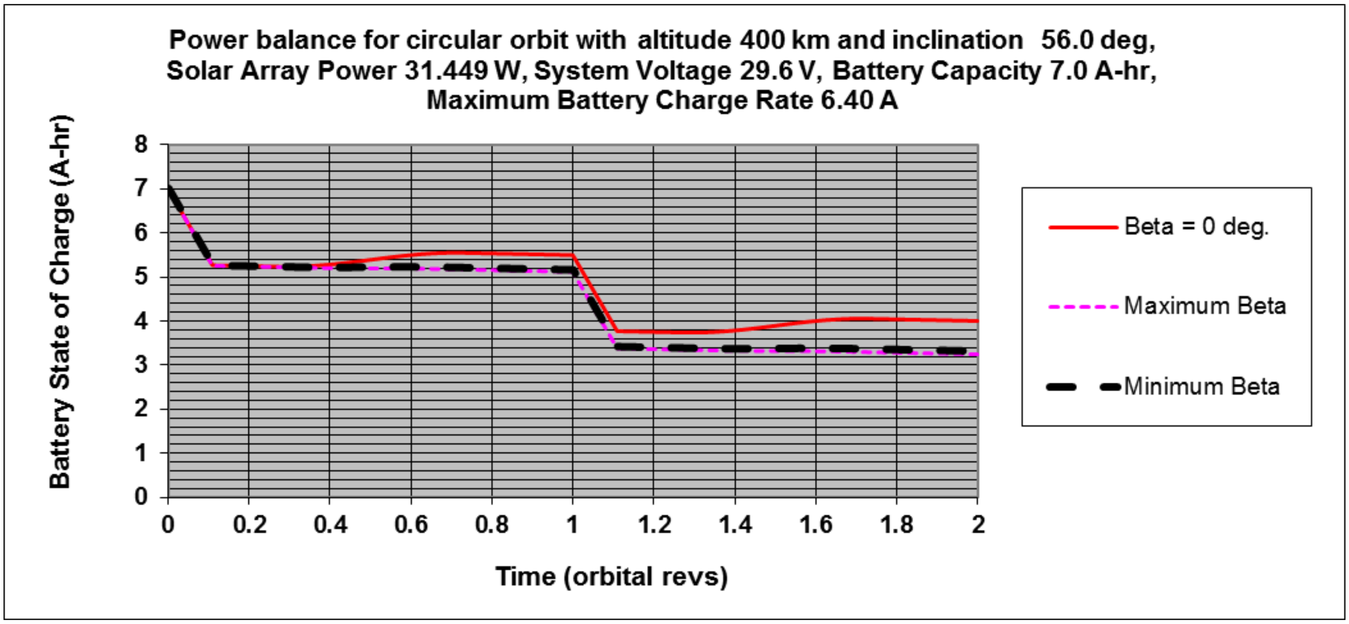


Figure 15: 11% Payload Duty Cycle

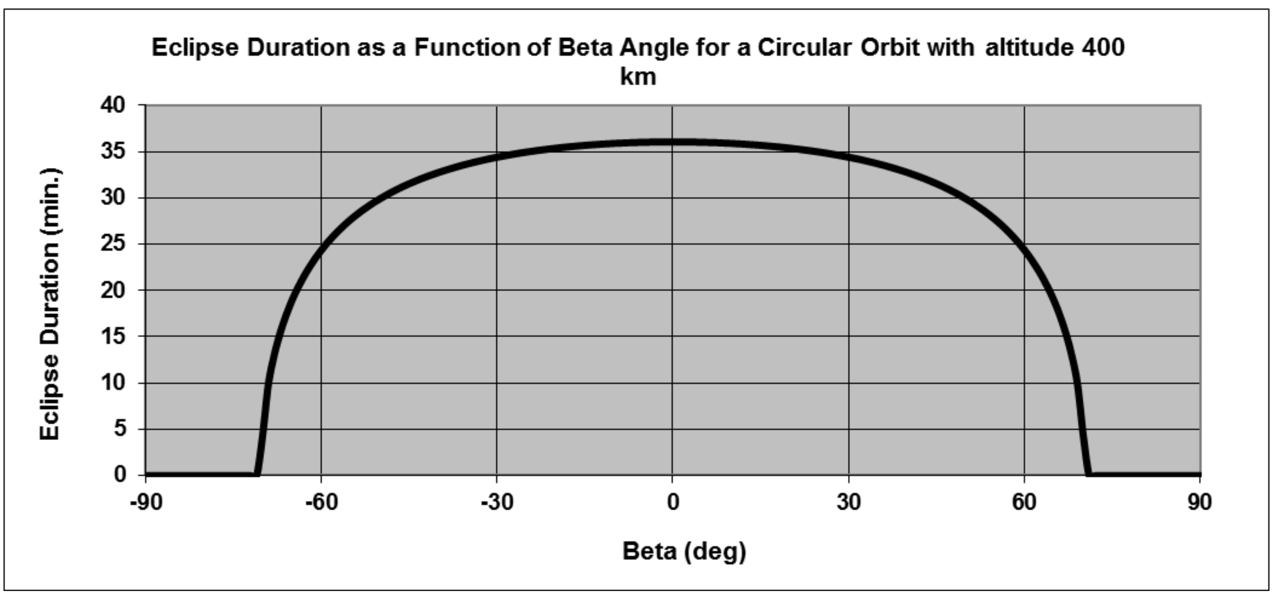


Figure 16: Eclipse Duration

ix. Payload Orbit Duty Cycle

As you can see from the first state of charge plot, the 1% duty cycle case is trivial and can operation continuously for tens of orbits without needing to enter a charging regime. This means that there is significant room to expand our payload operations. Given that the ground coverage tool in AGI STK, using DSN and our specific orbital parameters, gave an average of 11% coverage, and that two orbit operations with this duty cycle brings us down to 57%

depth-of-discharge, we chose to use this as our upper limit. Therefore, everything in between is within our operational limits. Theoretically we could operate up to 22% duty cycle for a single orbit, but given that there exists no ground support infrastructure to receive this data, it does not make sense to do so.

c. Telemetry and Command

i. Return Budget to RockBlock

Table 11: RockBlock Return Budget. Yellow indicates inputs. Blue indicates outputs. Green indicates margin. A Eb/N0 greater than 3dB indicates that the link budget closes at maximum and minimum.

	System MIN	System MAX
R (Data rate)	343	343
Bandwidth(MHz)	100	100
Modulation Scheme	GMSK	GMSK
Pwr (Tx Power in W)	1.60	1.60
Pwr (Tx Power in dBm)	32.04119983	32.04119983
Ll (Line Loss in dB)	0.5	0.5
Antenna variation factor (in dB)	0	0
Gt (Tx Total Antenna Gain in dB)(60% ef	-3	-3
EIRP	28.54119983	28.54119983
Raos (Dist to SAT @ AOS in km)	381	2236.037854
f (Desired Tx Frequency in GHz)	1.63	1.63
FSPL (Space Loss @ AOS in dB)	148.41	163.78
ISAB (Ionospheric Absorption Loss)	0	3
Latmo (H20 & O2 Atmo-/Ionospheric)	0	3
Gr (Rx Antenna Gain in dB)	45.9	45.9
Power Received dBm	-73.97	-95.34
Eb/No	74.96357852	53.592495
Figure of Merit(G/T)(dB/K)	21.58636236	21.58636236
Eb/No Required (1e-05 BER)	12.00	12.00
Eb/No margin	62.96	41.59

ii. Space Segment

Power consumption, frequency delegation, data-rate, and ground system architecture were contributing factors to the preliminary T&C trade study. The two radios that provide the most operational leeway are the RockBlock and LinkStar radios. Both communicate with larger satellite constellations. These constellations, namely Iridium and GlobeStar respectively, communicate with their own web of ground station networks, sending data back via network call. The RockBlock device transmits and receives on L-band. This radio is also very affordable, below $200. These features result in it leading the preliminary trade study.

When spacecraft transmit to each other in an inter-satellite link, reception on the spacecraft in question is considered the *forward* path and transmission is considered the *return* path. Given that the forward path is covered by the network and should be invariant in our situation, only the return link budget is shown above. The line loss is small, around 0.5dB, and the antenna gain will be isotropically -3dB. Free space path loss is calculated from a minimum distance between the constellation and the spacecraft orbit (400km), and a maximum distance of twice the orbital distance to the constellation. Small atmospheric and ionospheric attenuation effects are added. Both worst case and best case margins are at least 3dB. This means they are 3dB above the noise floor, and larger than the minimum detectable signal with a receiver of this type of figure of merit. This means that from nominal operation, through highly non-nominal operation, the link will be secure.

iii. Ground Segment

The ground network side of telemetry and command for this radio simply requires a computer. A command is sent over IP to an Iridium ground station, which then transmits the packet to the constellation. The spacecraft will then receive the packet and respond accordingly. This means that the ground segment architecture has been completely abstracted away, thus significantly reducing operational costs and risks.

d. Command and Data Handling

i. CDH System Overview

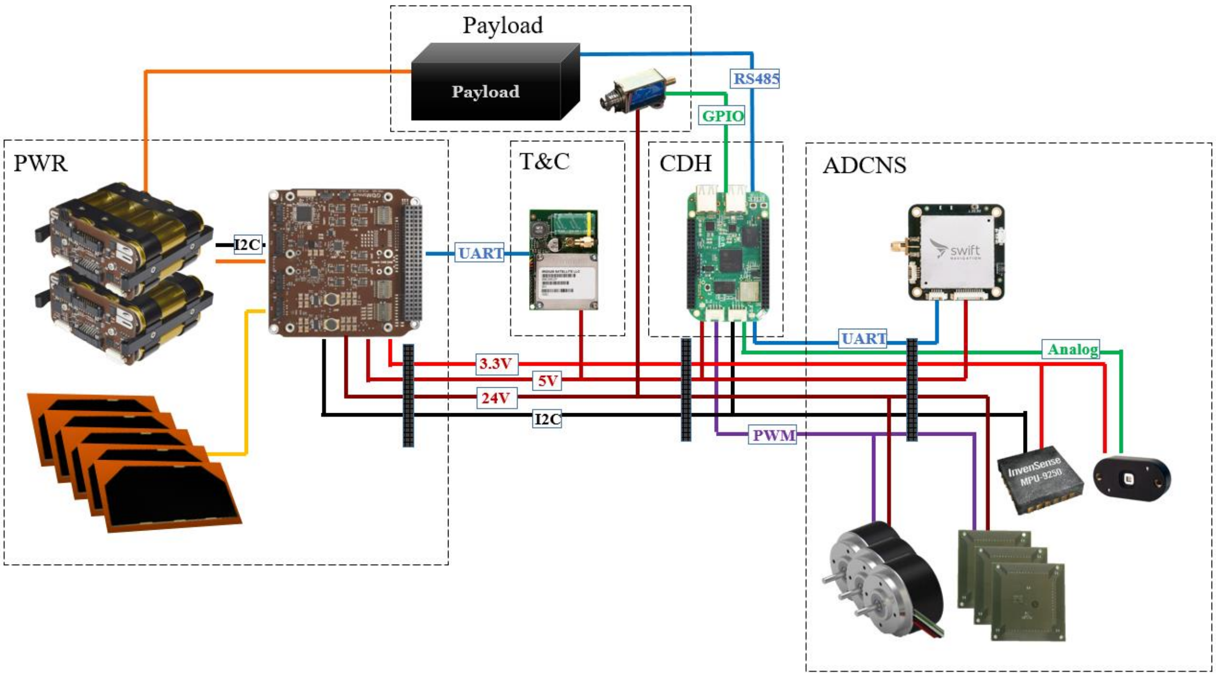


Figure 17: Schematic of data interfaces between spacecraft components.

The CDH system is responsible for processing commands and telemetry, storing data, executing flight software, and monitoring the status and health of the spacecraft. CDH functions are distributed between the power board and flight computer. Commands sent via T&C are first received on the power board so that instructions critical to the powered state of the spacecraft can be executed, while other commands are relayed to the flight computer. CDH reads attitude information from ADCNS sensors, performs computation for controls on the flight computer, and sends instructions to ADCNS actuators.

CDH interfaces with the T&C subsystem via a UART serial port connected from the power board. Any commands destined to flight computer are relayed by the power board to an I2C bus connected to the flight computer. Key interfaces with ADCNS include analog inputs from the sun sensor, digital PWM outputs to ADCNS actuators, an I2C connection to the spacecraft IMU, and a serial connection to the GPS. A separate serial connection is also required to communicate with the payload. The CDH system is tasked with enabling transmission from the payload radio and collecting status of health information, while the payload is responsible for supplying its own data stream to downlink.

A BeagleBone Green Wireless (BBGW) was selected as the flight computer. This board provides the necessary peripherals which allow it to interface with all spacecraft subsystems, while also providing generous amounts of memory and sufficient speed to process commands. The wireless variant of the BeagleBone Green was selected for greater flexibility during development and testing.

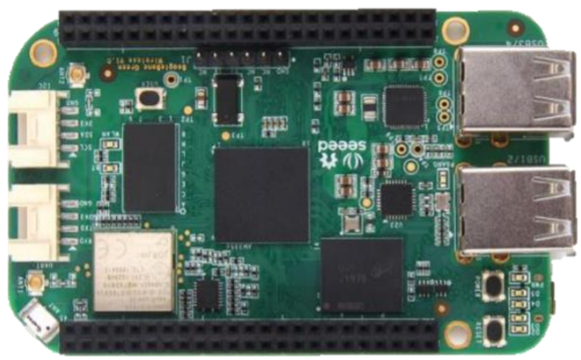


Figure 18: Top view of the BeagleBone Green Wireless board.

BeagleBone Green Wireless Seeed Studio	
Processor	AM3358 1GHz ARM® Cortex-A8
RAM	512 MB
Flash	4 GB
Input Voltage	5 V
Power Draw	1.2 W (typ.), 2.3 W (max)
Weight	40 g
Dimensions	87 x 127 x 54 mm
Temperature Range	0 to 75 C
Peripherals	UART (4), SPI (2), I2C (2), PWM (8), GPIO (65)

Table 12: BeagleBone Green Wireless specifications

ii. Processor

The BBGW employs a Sitara AM3358 processor. The processor relies on a 32-bit ARMv8 computer architecture, and is capable of speeds of up to 1 GHz for a maximum throughput of 2000 MIPS. External sensors and actuators can be accessed via analog inputs, I2C, and UART serial ports. The processor's analog inputs use 12-bit ADCs which can support a data rate of up to 200 kHz, while data is transferred over I2C at a standard rate of 100 kHz.

The BBGW boots from an embedded 4G flash storage device, where both computer and application software is stored. The AM3358 incorporates single-fault detection for data stored on the flash device via error correcting code. Single bit error correction is achieved using Hamming Code, and BCH code is used for up to 8-bit error correction. Single bit error detection is also performed on the processor's own on-chip memory via parity. Finally, a watchdog timer is employed on the processor to reset the BBGW if flight software becomes unresponsive for an extended period.

Component	Quantity	Model	Interface	Data
Power Board	1	NanoPower P60	I2C, UART	Out: Voltages, Currents, Temperatures
Battery Pack	1	GomSpace BPX	I2C	Out: Voltage, Current, Temperature
Reaction Wheels	3	Custom	PWM, GPIO	In: Motor Speed, Motor Direction Out: Speed Monitor
Magnetic Torquers	3	Custom	PWM	None
IMU	1	InvenSense MPU-9250	I2C	Out: Gyro Rate (x3), Accelerations (x3), Magnetic Field Strength (x3), Temperature
Sun Sensor	1	Custom	Analog	Out: Sun intensity (voltage)
GPS	1	Swift Piksi	UART	Out: Position and Settings (byte string)
Radio	1	RockBLOCK	UART	In: Commands (byte string) Out: Telemetry (byte string)
Payload	1	n/a	RS-485	In: Enable/Disable Out: Health and Status
Solenoid	1	Adafruit 412	GPIO	None

Table 13: Component output and data types

iii. Software

The flight software has been separated into modes to command the spacecraft at different phases of the mission, including initializations, normal operations, and demonstrating the payload. The start-up sequence was designed to comply with CubeSat requirements for launch vehicle separation with regards to powered systems. After deployment, the first mode of operation performs initial de-tumble procedures and checkouts to ensure the satellite survived launch. Mode transitions are controlled by CDH, and occur when certain criteria are met such as when low power is detected by the power board or when checkouts are complete.

Initial checkouts include verification from ADCNS sensors that de-tumbling has been successful, minimally actuating ADCNS actuators to confirm the spacecraft's attitude can be controlled, and that a status byte retrieved from the payload. Additionally, nominal power requirements on battery voltage and line currents must be met before the state of the spacecraft can be changed.

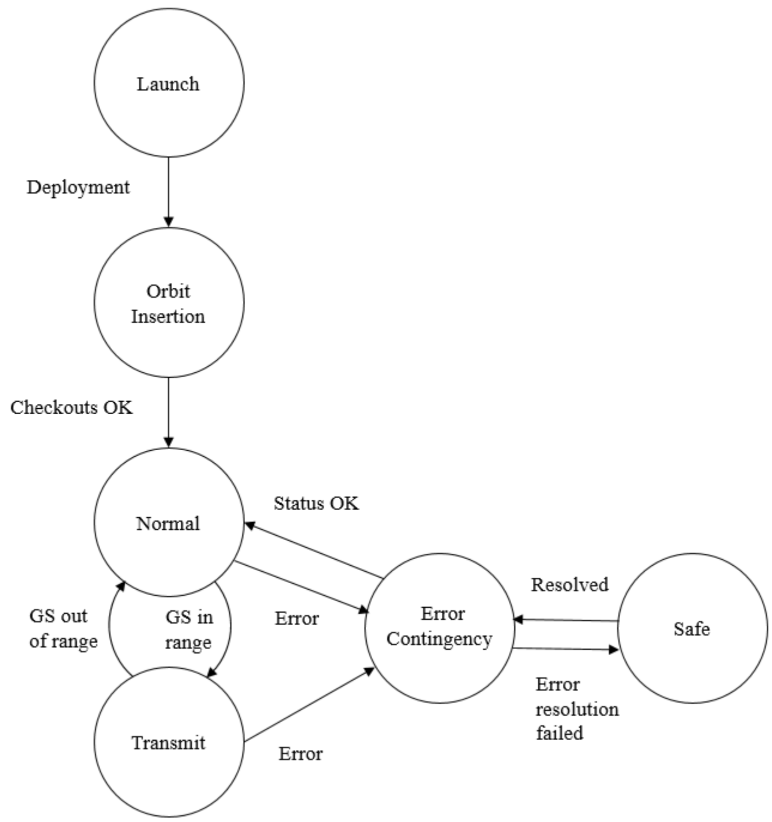


Figure 19: Software Modes

Mode	Description	Subsystems
Orbit Insertion	After deployment, T&C is turned on 45 min. after deployment. De-tumble procedures are performed along with and a status check of all systems.	T&C: On ADCNS: On (De-tumble) PWR: On FC: On PLD: Standby
Normal	The spacecraft operates in normal mode while in orbit. The batteries are set to charge and the status of health from all subsystems is collected periodically.	T&C: On ADCNS: On PWR: On FC: On PLD: Standby

Transmit	This mode is specifically partitioned to demonstrate the payload. The spacecraft is pointed at a Ground Station and transmission is enabled on the payload provided that there is sufficient power.	T&C: On ADCNS: On PWR: On FC: On PLD: On
Error Contingency	The spacecraft enters this mode if off-nominal conditions are detected. The payload is turned off if on, and contingency procedures are performed such as power cycling individual components.	T&C: On ADCNS: On (if possible) PWR: On FC: On PLD: Off
Safe	If standard contingency procedures fail, the spacecraft is placed in a thermally safe orientation and power to non-critical functions are turned off. The spacecraft then waits for updates from the ground.	T&C: On ADCNS: Off PWR: Low-power mode FC: Off PLD: Off

Table 14: Description of software modes

e. Structures

i. Mass Budget

The current best estimate for the mass of the spacecraft was calculated, as seen in the preliminary mass budget below. The mass allocation threshold was set at 98% of the absolute mass allocation, as recommended by the “Design, Verification/Validation and Operations Principles for Flight Systems, Rev. 1” from JPL. The current best estimate of the spacecraft mass is 3.63kg and the margin is 69.1%, which is significantly above the 10% pre-CDR mass margin percentage requirement recommended by JPL. Thus, no significant recovery actions are currently required.

The mass budget was derived from the CAD model of the Akash-Systems spacecraft, which included all components for each subsystem, all specific required fasteners and brackets, the entire structure, all actuators, and the payload. The CAD assembly currently contains over 1000 parts, all of which have been assigned specific materials and associated densities; notably, the payload was modeled as an aluminum parabolic antenna with features for mounting to the bus. The mass budget below avoids unnecessary resolution and rather only displays the masses of subsystems or significant components as determined by the CAD model, as well as estimates the mass of chemicals and harnessing. Moving past PDR, we do not expect the mass budget to significantly change and to remain within 10% of its current figure, leaving it well within the spacecraft mass requirement.

Table 15: Current Best Estimate Mass Budget. All numbers are in kg unless otherwise noted.

Total Mass Allocation	12
98% Mass Allocation	11.76
Avionics Stack	1.44
Solar Panel Assemblies	0.472
ADCNS	0.549
Structure	0.466
Payload	0.451
Fasteners and Brackets	0.131
Harness	0.1
Conformal Coating and Chemicals	0.025
Current Best Estimate	**3.634**
Margin	**69.10%**

ii. Door Deployment

A door has been incorporated so that a greater surface area covered with solar panels can be pointed toward the sun. The door will actuate after deployment once 60 minutes have passed, as per CubeSat specifications, and remain open for the duration of the mission.

The door is designed to be mounted to the -Z face of the spacecraft, opposed to the +X face. The spacecraft must perform a slew when the payload is transmitting, and placing the door in this orientation creates a small moment of inertia about the spacecraft's axis of the rotation for the slew relative to the other axes. Additionally, this orientation places the magnetometer farthest away from other spacecraft avionics, thereby reducing sensor noise. However, attaching the door to the +X face of the spacecraft would result in a more symmetric inertia matrix and improve overall agility. Further analysis must be done before CDR to determine if the benefit in increased agility about the X axis exceeds the benefit of high agility about the Z axis. The placement of the door does not otherwise affect power generation or the thermal characteristics of the spacecraft.

Redundant rigid hinges with torsion springs were selected to attach the door to the spacecraft bus. This method of actuation was selected because of its flight heritage and simplicity. A latch mechanism locks into place once the door is deployed to prevent it from imparting dynamics on the spacecraft during the mission.

During launch, the door will be held closed by a linear solenoid to counteract the torque from the loaded torsion springs in the hinges. The maximum pull force of the solenoid was sized to overcome friction between the solenoid pin and the door.

$$F = \frac{\mu K \theta}{L}$$

Where μ is the coefficient of friction between steel and aluminum, K is the torque constant of the torsion spring, θ is the pre-loaded angular displacement of the torsion spring, and L is the distance between the hinges and the solenoid.

Since the time duration necessary to deploy the door is not mission critical, a torsion spring with a low spring coefficient was selected to minimize any disturbances to the system during deployment. For a torsion spring with a spring constant of 0.03 N.m/rad and a stowed angular displacement of 180 degrees, the minimum force required for the solenoid to release the door is 0.61 N. This calculation was used to select a candidate solenoid, incorporating a force margin of 8.2.

Figure 20: Solenoid Latch

Table 16: Solenoid Specifications

Vendor	**Adafruit Industries LLC**
Voltage	24 V
Maximum Pull Force	5 N
Stroke Length	5.5 mm
Weight	39 g
Current	240 mA

iii. Preliminary Trade Studies

Bus Material: For the structure subsystem, a trade study on the material of the bus frame was conducted, as shown in the table below. The alternatives were limited to Aluminum 7075, 6061, 6082, 5005, and 5052. The ultimate strength, tensile yield strength, fatigue strength, density, and manufacturability were maximized while cost/mass was minimized. Appropriate weighting factors were assigned to each attribute. Aluminum 7075 scored the highest under this trade study; therefore, Aluminum 7075 will be used for the primary structure of the spacecraft. This material is used for the spacecraft structure in the CAD model.

Table 17: Primary Structure material trade study.

Structural Material (maximizing)		**Alternatives**				
Weight	Attribute	Al 7075	Al 6061	Al 6082	Al 5005	Al 5052
0.01	Ultimate Strength [MPa]	507	280.5	310	180	268.5
0.02	Tensile Yield Strength [MPa]	444.5	241.5	265	165	210.5
0.01	Fatigue Strength @ 10^7 cycles [MPa]	160	95	95	71.3	130
10	1/(Cost/kg)	0.211193242	0.416666667	0.420168067	0.422832981	0.417536534
0.003	Density [kg/m^3]	2800	2700	2700	2690	2685
0.1	Manufacturability (1 = least manufacturable)	1	2	2	3	4
	Total Score	26.17	21.05	21.85	18.41	20.83

iv. CAD Model

As discussed above, Akash-Sat1 was modeled in Solidworks. All subsystems and components were individually modeled and assigned material properties. Figures of the Akash-Sat1 CAD model are shown below.

Figure 21: Akash-Sat1 Rear View 1

Figure 22: Akash-Sat1 Rear View 2

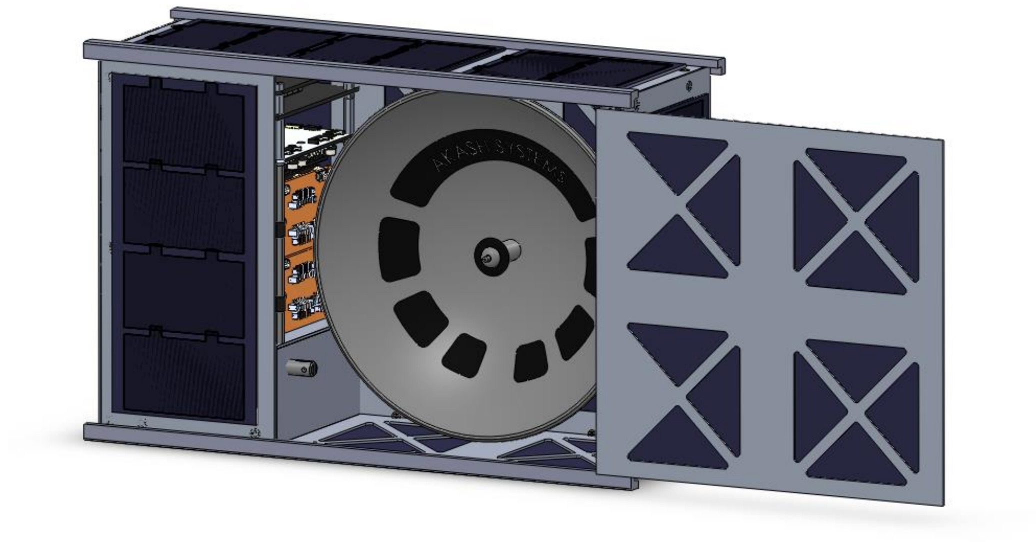


Figure 23: Akash-Sat1 Front View

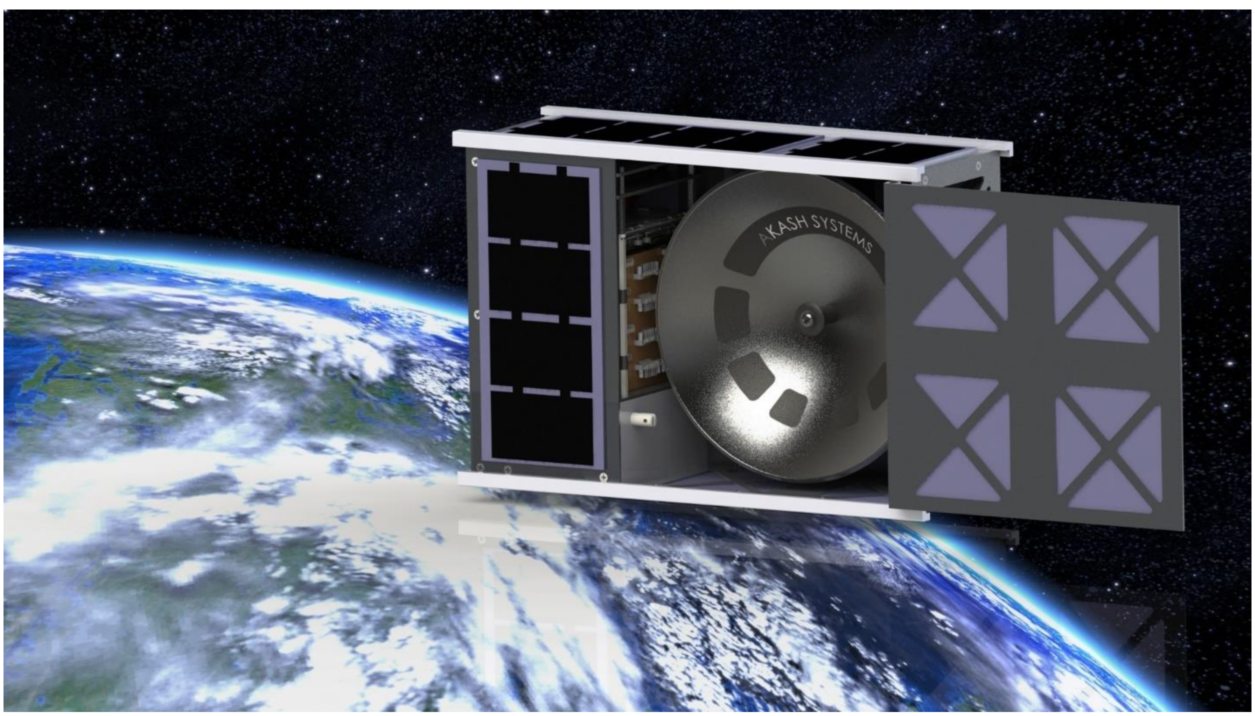


Figure 24: Akash-Sat1 Rendered CAD Model. Marketing Image.

f. Thermal

Additional information and analysis on the thermal subsystem can be found in the Appendix.

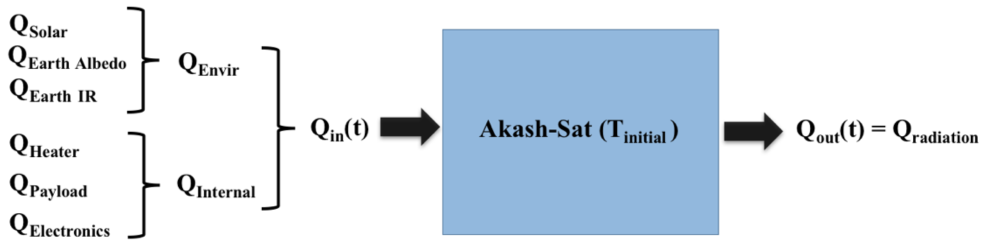
i. Flight Allowable Temperature Range

The thermal subsystem is responsible for determining the flight allowable temperatures for the spacecraft and controlling the actual mission temperatures to within a safe margin away from the flight allowable temperature range. Preliminary calculations were conducted in MATLAB to determine the expected heat flux to the spacecraft assuming ISS orbit. As shown in the table below, the current flight allowable temperature range for the spacecraft is 0 to 45 Celsius. While the flight computer is setting the lower bound of the flight allowable temperature change rather than the batteries, it is a matter of choice. The benefits of BeagleBone Green were traded against its risks and it was deemed to be viable. As recommended by the ""Goddard Space Flight Center Rules for the Design, Development, Verification, and Operation of Flight Systems", the thermal design margins shall be 5 degrees Celsius at the PDR level. Therefore, the thermal subsystem shall aim for a temperature range of 5 degrees Celsius to 40 degrees Celsius.

Table 18: Operational temperature ranges for preliminary spacecraft components

	Operational Temperature Range [◦C]	
Components	**Low**	**High**
Solar Arrays	-150	110
NanoPower BPX Board	-40	85
NanoPower BPX Batteries (charge)	-5	**45**
NanoPower BPX Batteries (discharge)	-20	60
BeagleBone Green Wireless	**0**	75
ADCNS	-30	85
Parabolic Antenna	-100	100
Payload	-40	120
RockBLOCK	-30	60
Piksi GPS Receiver	-20	63

ii. Thermal Model



$$(Q_{in}(t) - Q_{out}(t))dt = mc_p(T_{final} - T_{initial})$$

Figure 25: Preliminary Thermal Model for Akash-Sat1

The general approach of the thermal model is to determine the worst case (hot and cold) transient environmental and internal heating and conduct thermal balance on Akash-Sat1 to calculate the temperature evolution of the spacecraft. The purpose of this model is to facilitate the initial selection and trade studies of thermal coatings and/or active thermal control elements such as a heater. 3 modes: Payload transmission, tumble, and safe modes will be considered as part of the analyses.

Table 19: Modes of Operation

Mode	**Normal**		**Tumbling**		**Safe Mode**	
Cases	**Hot**	**Cold**	**Hot**	**Cold**	**Hot**	**Cold**
Solar Constant (S) [W/m^2]	1419	1317	1419	1317	1419	1317
Beta Angle [degrees]	75	0	75	0	75	0
Earth Albedo (a)	0.45	0.22	0.45	0.22	0.45	0.22
Earth IR (Φ) [W/m^2]	261	217	261	217	261	217
Electronics Heat [W]	5.2	5.2	5.2	5.2	5.2	5.2
Heater [W]	0	6	0	6	0	0
Payload Heat Dissipation [W]	225 @ 11%	225 @ 1%	0	0	0	0
Thermal Model	6-sided box with door	6-sided box with door	sphere	sphere	6-sided box with door	6-sided box with door
Initial Temperature [Celsius]	23.8	23.8	23.8	23.8	23.8	23.8
Attitude Control?	Yes	Yes	No	No	No	No

The three modes of operation, normal, tumble, and safe, and their respective thermal characteristics are listed in Table 19. Hot and cold cases were considered for each mode of operation. Firstly, transient environmental heating was determined using the equations listed below:

$$Q_{solar} = SA_p\alpha_{avg}$$

$$Q_{albedo} = SaAF\alpha_{avg}$$

$$Q_{IR} = \Phi AF\epsilon_{avg}$$

where A is the area of the plate in question, A_p is the projected area towards the Sun, α_{avg} is the average surface absorptance, ϵ_{avg} is the average surface emissivity, and F is the view factor from the plate to the Earth. It is also important to note that the albedo factor will decrease is a sinusoidal manner from the subsolar point to the terminator, where the Earth albedo is near 0. Next, internal heat generation was determined and shown in Table 11 based on baseline power usage, presence of heater, and payload power, efficiency, and duty cycle. Since the payload operates at 300W at 25% efficiency, 225W will be heat dissipation at the specified duty cycle.

Heat dissipation from the spacecraft is only in terms of radiation, which is determined by:

$$Q_{out} = A\epsilon_{avg}\sigma T^4$$

where T is the temperature of the surface in Kelvin. By conducting thermal balance and incorporating the influence of the mass and specific heat capacity of Akash-Sat1, the temperature evolution of Akash-Sat1 can be calculated after many orbits.

iii. Thermal Model Results

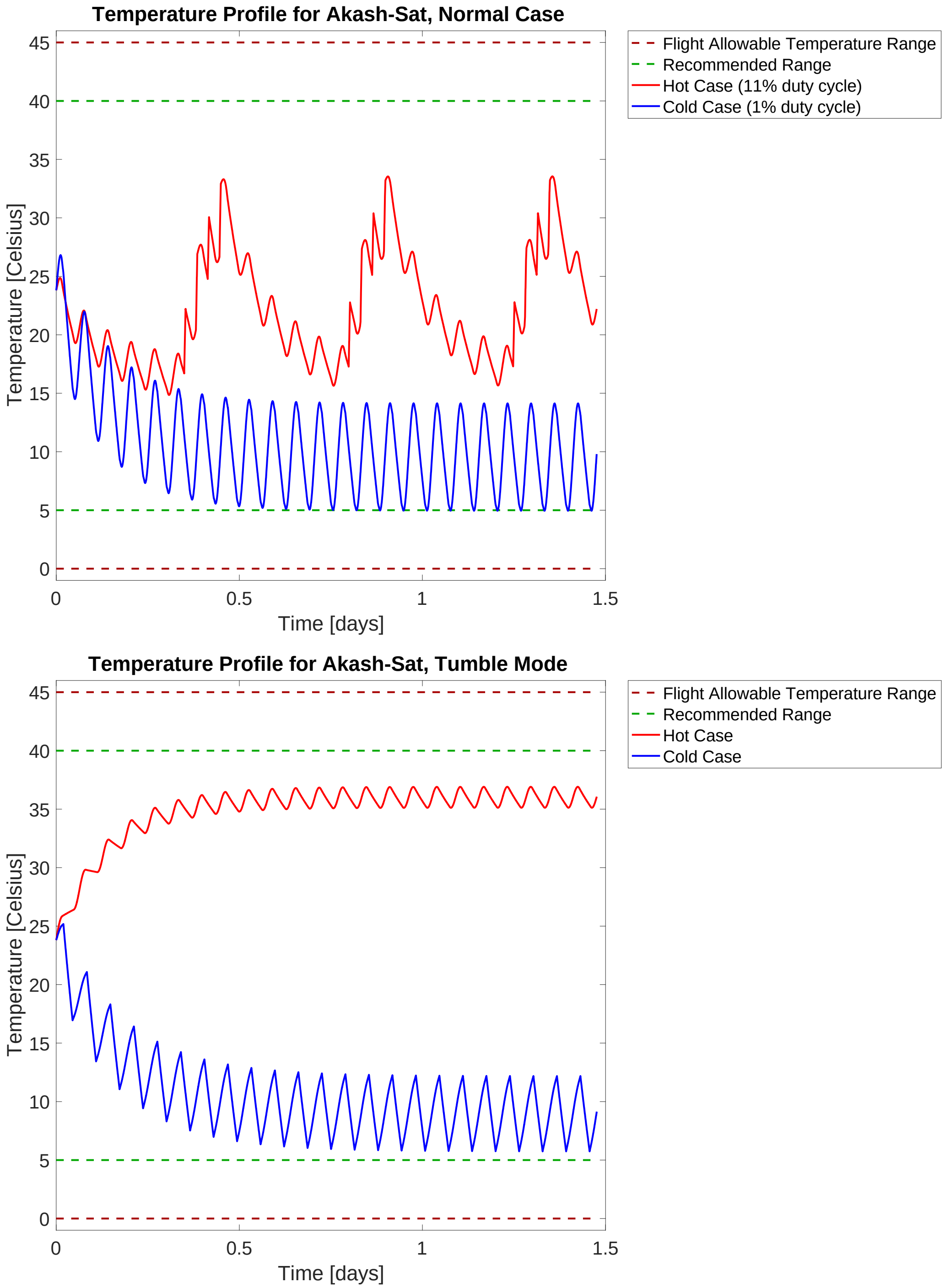
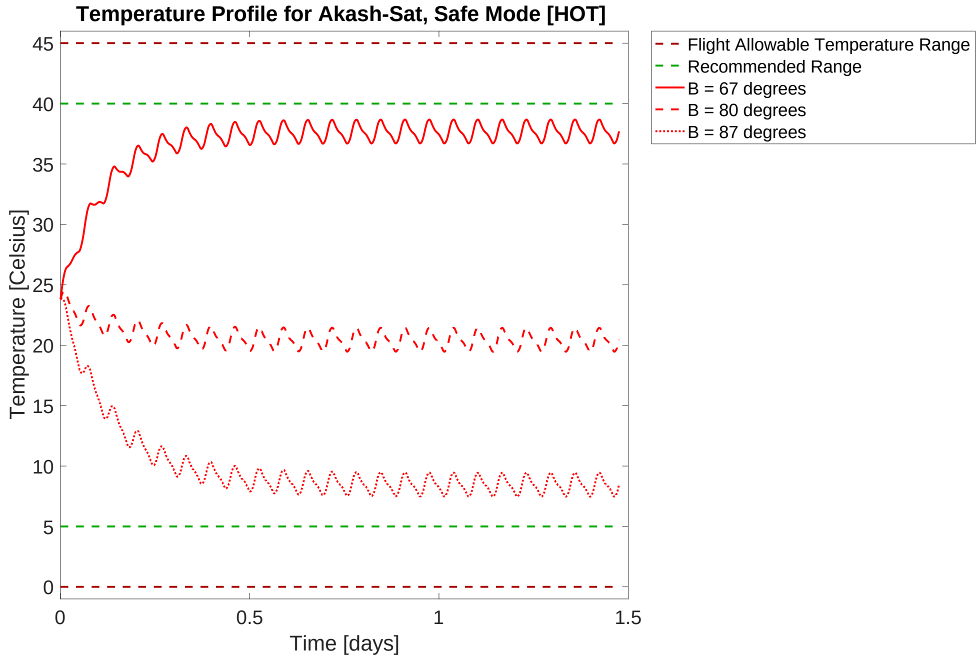


Figure 26: Normal and Tumble Mode Temperature Profile

As shown in Figure 26, both hot and cold cases for normal and tumble modes satisfy the recommended flight allowable thermal margin.



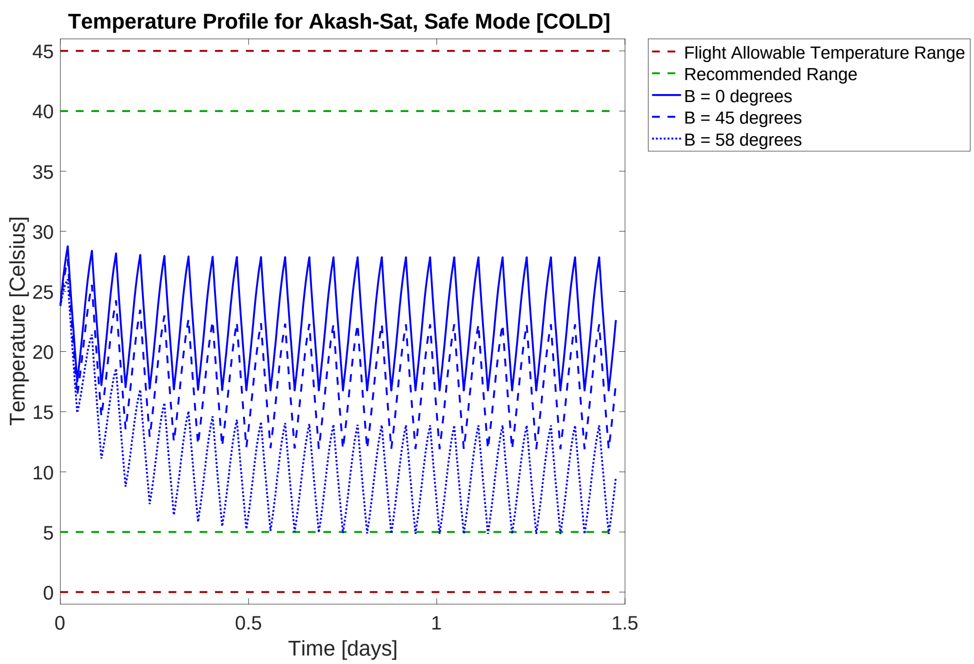


Figure 27: Safe Mode Temperature Profile

The purpose of the safe mode is to orient the spacecraft in a thermal and power safe attitude   without the need of ground intervention. This mode is characterized by no heater input and attitude control availability.

The simulations above were conducted assuming a solar cell coverage of 75%, a radiator on the underside of the deployed door with $\alpha = 0.1$ and $\epsilon = 0.9$, and the rest of the area with $\alpha = 0.9$ and $\epsilon = 0.1$. Anodized aluminum can be used for both instances. The cost is accounted for in the cost analysis. In addition, the battery heater that comes with the NanoBPX battery board with a power output of 6W will be used only for the cold cases.

Table 20: Thermal Margin

	Low [Celsius]	High [Celsius]	
Flight Allowable	0	45	
Normal Mode	8.1	34.9	
Margin	+8.1	+10.1	
Tumbling Mode	5.4	34.6	
Margin	+5.4	+10.4	

The margin for safe mode is expressed via the range of rotation in the X-axis allowed. The variable B from Figure 27 is the rotation of the spacecraft about its X-axis in degrees. As shown in Figure 27, Akash-Sat1 will be thermally safe when rotated 67 ~ 87 degrees about the X-axis for the hot case and 0 ~ 58 degrees for the cold case.

iv. Future Work

Future work includes the formulation of a transient ANSYS thermal model using the environmental heating values calculated via MATLAB to validate the choices for surface finishes and battery heater determined from the preliminary model. Additionally, thermal cycling tests in a thermal vacuum chamber will need to be conducted on the Akash-Sat1 engineering development unit.

g. Attitude Determination, Control and Navigation

i. Environmental Disturbance Torques

In LEO, the spacecraft will experience multiple disturbance torques from environmental forces such as solar radiation pressure, atmospheric drag, earth's magnetic field and earth's gravity. The change in angular momentum per orbit and instantaneous torque in one axis from these environmental forces were calculated from a MATLAB model and are listed below in Table 21. The ADCNS must have the angular momentum storage and torque capability to overcome these disturbances. However, as it will be shown in the next subsection, these angular momentum and torque values are orders of magnitude lower than the angular momentum and torque requirements for payload operations so they are not considered in the design and sizing of ADCNS hardware.

Table 21: Disturbance Torques due to environmental factors

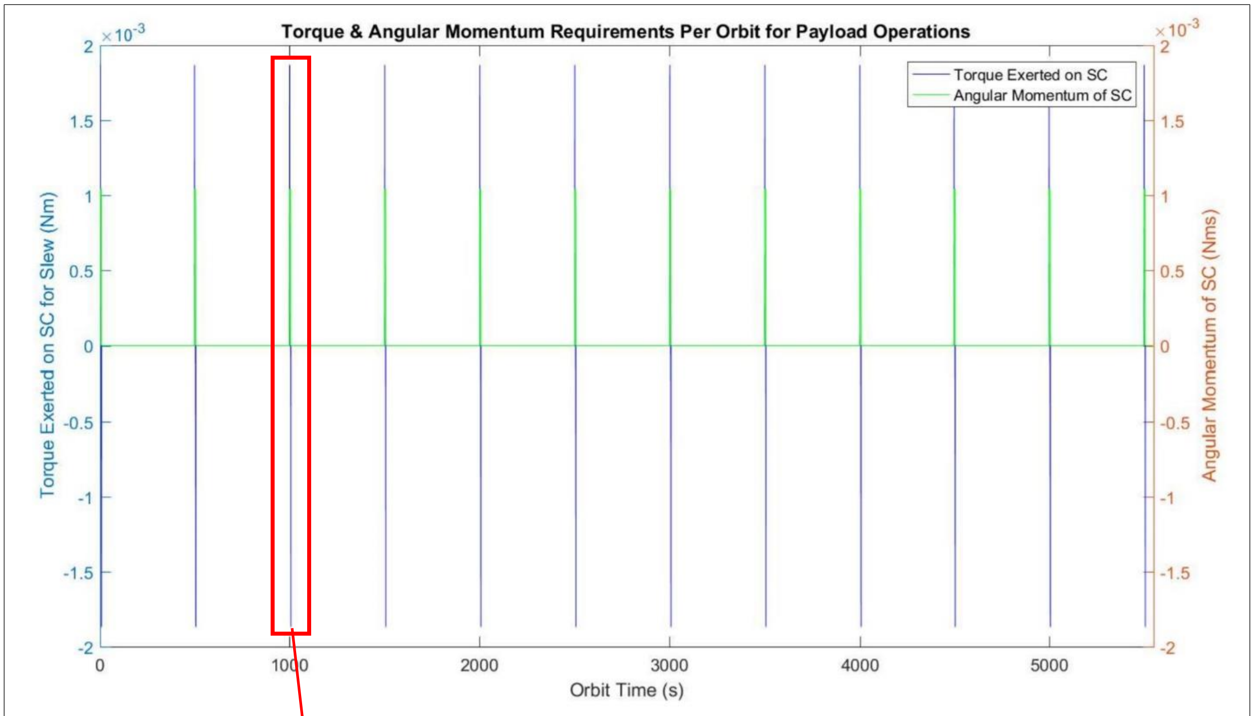
Environmental Force	Change in Angular Momentum per Orbit (Nms)	Torque on Spacecraft (Nm)
Solar Radiation Pressure	5.67×10^{-5}	1.64×10^{-5}
Atmospheric Drag	7.13×10^{-5}	1.28×10^{-5}
Earth's Magnetic Field	2.66×10^{-5}	4.76×10^{-5}
Earth's Gravity Gradient	6.46×10^{-5}	1.14×10^{-5}

ii. Payload Operations Requirements

To perform its payload operations, the spacecraft must perform a slew (as described in the 'Payload Operations' sequence of the Storyboard) to maintain a downlink connection with the ground segment. Using a MATLAB model that models the spacecraft's velocity, attitude, altitude and transmission time, it was determined the spacecraft must perform a 5.5° slew in 5 seconds to demonstrate the payload. The ADCNS requirements from this slew are shown in Table 22. Included in this table is the accuracy requirement as set by the payload's antenna ½ power beam width and provided by the Payload Link budget.

Table 22: Payload Operational Requirements

	Needed for Operation	ADCNS Requirement, 200% of Payload Requirement
Average Angular Speed of Spacecraft during payload op. (°/s)	1.1	2.0
Angular Acceleration of Spacecraft ($°/s^2$, assuming 1 s)	1.1	2.0
Angular Momentum Storage Capability of RWA (Nms)	1.042×10^{-3}	2.084×10^{-3}
Torque Capability of RWA (Nm)	1.868×10^{-3}	3.736×10^{-3}
Accuracy (°)	2.0	1.0



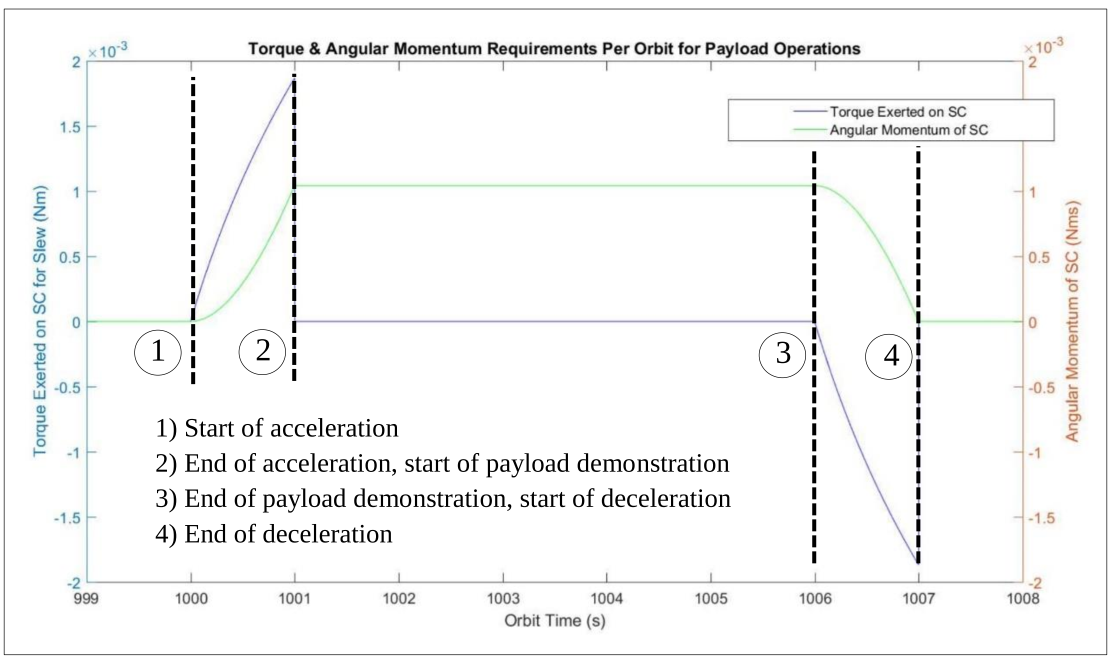


Figure 28: Torque and Angular Momentum

iii. Spacecraft Configuration

The ADCNS will take up 0.3U (10cm x 6cm x 5cm) of space, weigh less than 1.0 kg, and nominally use 2.15W of power per orbit. The subsystem will include attitude control actuators to rotate the spacecraft, angular momentum control devices to dump the spacecraft's angular momentum, attitude determination sensors to determine the spacecraft's attitude, and a GPS receiver to determine position and orbital parameters.

iv. Hardware Sizing & Trade Studies

All trade studies for ADCNS are provided in the appendix. In this section, the results of the trade studies are presented and explained.

Attitude Control Actuators: Reaction Wheel Actuators will be used to rotate the spacecraft. In selecting the RWAs, certain strict requirements had to be met such as having an angular momentum storage capability of 2.084×10^{-3} Nms, torque capability of 3.736×10^{-3} Nm and a maximum diameter of 50 mm. Other factors that were optimized for in the trade study were cost, weight, lead time, reliability, control type, motor type and power. There are several COTS options that were considered for the RWAs and the result of the trade study was the Maxon EC 32 Flat Motor IE (Figure 29 and Table 23).

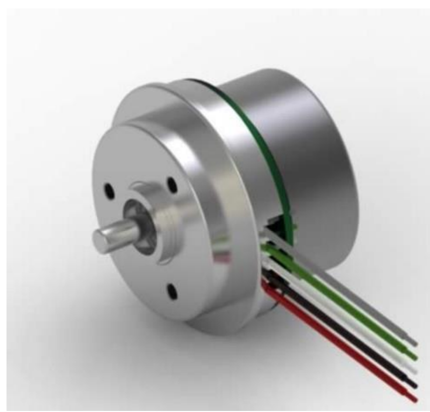


Figure 29: Maxon EC 32 Flat Motor IE

Table 23: Specifications of Maxon EC 32 Flat Motor IE

Angular Momentum Storage Capability (Nms)	2.084×10^{-3}
Torque Capability (Nm)	35.0×10^{-3}
Diameter (mm)	40

This motor meets all the highest priority requirements of the trade study while being the cheapest option. This is because the Maxon EC 32 Flat Motor IE is not a traditional RWA but a brushless motor that has a relatively large rotor inertia, allowing its own internal rotor to serve as the 'flywheel' of a traditional RWA. In addition, being a brushless motor with integrated electronics gives the efficiency benefit of a brushless motor (as compared to a brushed motor) but with the simplicity of operating as a DC motor.

Angular Momentum Control: Magnetic torquers will be used to dump angular momentum in the spacecraft and RWAs. The torquers must be able to dump the momentum in the wheels very rapidly so the strict requirement that must be met is a magnetic dipole of at least 0.2 Am^2. This value is a standard for CubeSats and proves acceptable in the MATLAB model to dump angular momentum rapidly. Other factors that were considered and optimized in the trade study were power, size, cost, lead time, residual moment, and precision. There are several COTS options available that were considered but the overwhelming cost of COTS magnetic torquers and the fact that magnetic torquers are easy to produce resulted in the magnetic torque being custom made in-house.

There are two custom magnetic torquer options presented in the trade study: magnetic torque rods and the magnetic torque boards. To calculate magnetic dipole $\vec{M}$ for an electromagnet, the following equation must be considered:

$$\vec{M} = Kni\vec{A}$$

where K is the relative permeability of the electromagnet's core, n is number of loops, I is current in amps, and $\vec{A}$ is area enclosed by the loop in m^2. Magnetic torque rods contain a coil of copper wire around a ferrite iron or mu-metal core that could have a gain factor of 10-1000 or 1000-100000, respectively, making them very power efficient. However, metal cores tend to have a residual moment and tend to have a latency in their activation due to the core becoming magnetized.

A magnetic torque board is a PCB board with embedded torque coils as copper traces within the PCB's layers as shown in Figure 30. Although the enclosed area of the coils on these boards are larger than the magnetic torque rods, they do not contain a metal core so their gain factor is 1 and therefore require more power than the magnetic torque rods to achieve the same magnetic dipole. However, the lack of a metal core is a clear benefit because the magnetic torque boards will have no residual moment and upon activation they will achieve their intended magnetic dipole almost instantaneously. In addition, the orbital duty cycle of the magnetic torquers is 2% so power consumption is weighed much less heavily in the trade study. Most importantly, the addition of magnetic torque boards requires no extra space in the spacecraft because the coiled traces can be embedded in the spacecraft's solar panels. As a result, the magnetic torque boards are the optimal option for momentum control and dumping.

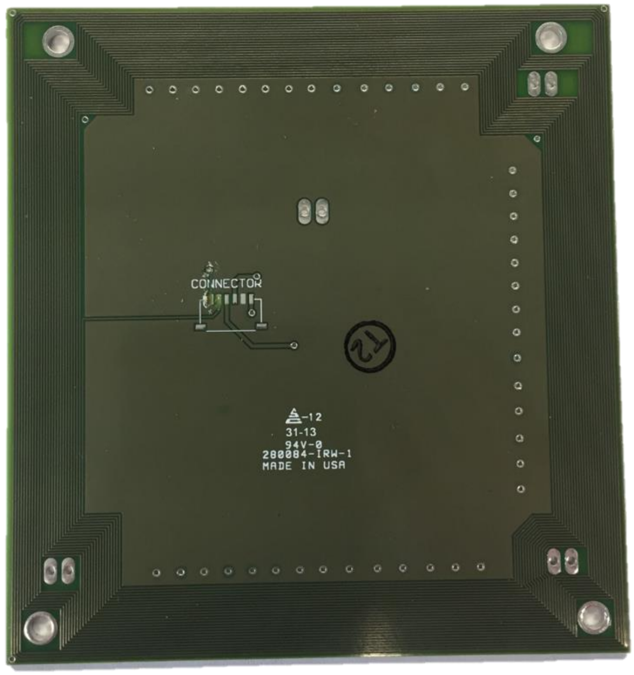


Figure 30: PCB Printed Magnetic Torque Board from KickSat-1

The specifications of the magnetic torque boards are provided in table 24. Although the magnetic torque boards on the different faces have a different dipole moment, they all surpass the magnetic dipole requirement of 0.2 Am^2 and the power input into each board can be easily adjusted to produce identical dipole moments.

Table 24: Magnetic Dipole of Magnetic Torque Coils Embedded in Solar Panels

Solar Panel Face(s)	**# of Solar Panels**	**Size (m)**	**Magnetic Dipole Total (Am^2)**
+Y	1	0.3 x 0.2	0.518
+- X	2	0.3 x 0.1	0.588
+- Z	2	0.2 x 0.1	0.522

Attitude Determination Sensors: In LEO for small satellite missions, the most common attitude determination sensors are sun sensors, magnetometers and gyroscopes. The strict requirement was having an accuracy of 1° while factors that were considered in the trade studies and optimized were size, power consumption, cost and lead time. Magnetometers and gyroscopes are available in single units that are cheap, small, effective and have no lead time so there were many COTS options to pursue. For the sun sensor, one with a high accuracy is needed so making a custom sun sensor would not be feasible and a COTS option must be considered. The trade studies are provided in Appendix A and the results are presented in Table 25.

Table 25: Attitude Determination Sensors

Component	**Hardware**
Fine Sun Sensor	Nano-SSOC-A60 Analog Sun Sensor
Gyroscope	9-DOF Razor IMU SEN-14001
Magnetometer	9-DOF Razor IMU SEN-14001

According to SMAD, a combination of a fine sun sensor and an accurate magnetometer (assuming position is known) is able to achieve a 1° accuracy. Both these sensors achieve the near maximum accuracy achievable for that type of sensor so the accuracy requirement is assumed to be met with the current design.

Five sun sensors will be used so that one can be placed on each face (excluding the -Y face that contains the parabolic antenna) to give the spacecraft a large field of view. This is because the spacecraft is frequently slewing based on the location of the ground segments and independent of the position of the sun so the spacecraft's orientation relative to the sun is constantly changing. In addition, the spacecraft is frequently entering and exiting eclipse so having multiple sun sensors will allow the spacecraft to differentiate between light from the sun and albedo from the earth to accurately determine attitude.

The magnetometer readings are susceptible to noise if the magnetometer is placed in the ADCNS box because of the presence of magnetic fields from the brushless motors and other electronics. Therefore, the magnetometer will be placed on the outside edge of the payload door to minimize noise from electronics in the spacecraft's bus. This is a technique used on large spacecraft as magnetometers are placed on very long boons. Based off hand calculations on the strength of magnetic field and the magnetic fields present in the spacecraft bus, placing the magnetometer on the end of the payload door is estimated to reduce noise in the magnetometer readings by 1,000% - 10,000%. In addition, magnetometer readings will only be taken when the magnetic torque boards are powered off to prevent inaccurate measurements being taken.

GPS Receiver: A GPS receiver is needed as the magnetometer and sun sensor utilize position and orbital parameter data to determine attitude. Factors to consider are cost, lead time, size, power consumption, modularity, accuracy and space compatibility. However, the Swift Piksi GPS receiver is a standout contender as there are few options that perform as well.

Microprocessor: There will be no trade study on the microprocessor because there is no component in the ADCNS that requires extreme processing power (such as a star tracker). Therefore, the ADCNS will connect to the spacecraft's BeagleBone Green as that component has sufficient processing power and available pins.

VII. Cost Assessment

a. Akash-Sat1 Cost Assessment

Cost assessment for the Akash-Sat1 program is significantly simplified by the fact that it will be designed and integrated by students at Cornell University, who are unpaid for their work. Accordingly, time and labor as a function of development duration are not factored into the cost assessment. This does not include summer labor, for which $20,000 is allotted for each summer during the program ($40,000 total). Furthermore, the following cost assessment is solely for the development of Akash-Sat1 and does not include overhead taken by Cornell University, which will factor into the final requested payment from Akash Systems.

As all software will be written by unpaid students, software developments costs are also not factored into the cost assessment. In addition, it is assumed that many tools and lab space are already available in the Cornell University Space Systems Design Studio. However, it is assumed that engineering software programs, lab computers and clean room equipment will be purchased for the program.

Still, the majority of the cost is determined by component costs. Because the components for Akash-Sat1 have been already selected at this stage, top-down assessments utilizing cost-estimating relationships such as subsystem weight and measures of complexity are not required and would represent an inaccurate assessment of cost.

A significantly higher accuracy cost assessment can be obtained by simply summing component quantities and costs. First, the unit quantity per spacecraft for each hardware component is determined. Then, depending on the component, a flight set, a spare set, and an EDU set is defined. To obtain the final cost, shipping and environmental testing are estimated and a management reserve is added. The final projected cost of the program is $285,363. The cost assessment breakdown can be seen in the table below.

Table 26: Akash-Sat1 Cost Assessment

		Units Per Satellite	Flight	Spare	EDU	Total Units	Unit Cost	Price
Power	Solar Cells	52	1	0	0.5	78	$ 220.00	$ 17,160.00
	Electronic Boards	1	1	1	1	3	$ 1,000.00	$ 3,000.00
	External Battery Packet	2	1	0.5	1	5	$ 8,250.00	$ 41,250.00
	EPS	1	1	1	1	3	$ 19,000.00	$ 57,000.00
	Li-On Batteries	16	1	1	1	48	$ 5.25	$ 252.00
	Harness	1	1	1	1	3	$ 1,000.00	$ 3,000.00
	Subtotal							**$ 121,662.00**
ADCNS	RWAs	3	1	1	1	9	$ 125.00	$ 1,125.00
	Sun Sensor	5	1	0.5	1	12	$ 2,398.00	$ 28,776.00
	Magnetic Torquers	1	1	1	1	3	$ 500.00	$ 1,500.00
	9 DOF IMU	1	1	1	1	3	$ 50.00	$ 150.00
	GPS Receiver	1	1	1	1	3	$ 500.00	$ 1,500.00
	Subtotal							**$ 33,051.00**
Structure	Walls	1	1	1	1	3	$ 1,200.00	$ 3,600.00
	Coating	1	1	1	1	3	$ 500.00	$ 1,500.00
	Fasteners	1	1	1	1	3	$ 2,000.00	$ 6,000.00
	Subtotal							**$ 11,100.00**
CDH	Flight Computer	1	1	1	1	3	$ 50.00	$ 150.00
	Subtotal							**$ 150.00**
T&C	Radio	1	1	1	1	3	$ 200.00	$ 600.00
	Data	1	1	0.5	0.5	2	$ 1,400.00	$ 2,800.00
	Subtotal							**$ 3,400.00**
Other	Conformal Coating	1	1	1	1	3	$ 1,000.00	$ 3,000.00
	Clean Room/I&T Materials	1	1	0	1	2	$ 3,000.00	$ 6,000.00
	Engineering Software	1	1	0	0	1	$ 3,000.00	$ 3,000.00
	Summer Labor	1	1	0	0	2	$ 20,000.00	$ 40,000.00
	Subtotal							**$ 52,000.00**
						Components Subtotal		**$ 221,363.00**
						Shipping		$ 2,000.00
						TVC Testing		$ 2,000.00
						Vibration Testing		$ 10,000.00
						Management Reserve		$ 50,000.00
						TOTAL		**$ 285,363.00**

a. Future Akash-Sat Program Costs

Akash Systems has indicated that they may request as many as three spacecraft from Cornell University. These spacecraft will be similar to each other and will utilize similar architecture. If the customer requirements for all three spacecraft are known before the procurement of hardware occurs, less spares and EDU units will be needed to be purchased per spacecraft, as they can be shared among the spacecraft as appropriate. Therefore, the average cost of each spacecraft is decreased. Purchasing more units of a component can earn the benefit of quantity discounts, which typically range between 2% and 5% for hardware. This represents a further reduction of the average cost per spacecraft; however, to maintain a sufficient margin due to uncertainty these quantity discounts are not included in the projected three spacecraft budget.

Using the quantity multiplication factors in the table below to determine the program cost for three spacecraft, it was determined this cost would be $727,084. Included in this cost is a steep increase of Management Reserve from $50,000 to $250,000 because of the strong possibility of adding new subsystems (ex. Propulsion), redesign of current subsystems and purchasing future launches. The launch for Akash-Sat1 will be obtained through CSLI and be free, however, it is assumed it will be difficult to receive two further free launches for the same program and the possibility of purchasing launches must be considered. The program cost for three spacecraft is preliminary and will be reassessed in the future.

Table 27: Akash-Sat Program Prelimary Projected Cost Assessment

Quantity Spacecraft	**3**
Flight Quantity Factor	**1**
Spare Quantity Factor	**0.67**
EDU Quantity Factor	**0.33**

Components Subtotal	**$435,084**
Shipping	**$6,000**
TVC Testing	**$6,000**
Vibration Testing	**$30,000**
Management Reserve	**$250,000**
TOTAL	**$727,084**

VIII. Risk Assessment

a. Fault Tree Analysis

Fault tree analysis was conducted for each primary subsystem of the spacecraft bus. This represents a top-down deductive failure analysis in which Boolean logic is used to combine low-level single events in order to analyze the undesired state of the system. At the system level, this state is simply the failure of the Akash-Sat1 bus, whereas for each subsystem it is the failure of the subsystem.

After the development of the fault trees, each event was assigned an appropriate probability and Boolean logic was used to determine the probability of failure P_F of the subsystem and the system, which is shown at the top of each tree. The probability of success P_S is found as: $P_S = 1 - P_F$.

For the Akash-Sat bus, the Probability of Success was determined to be 92.9%, which represents a confidence level of 1.8σ. Although major spacecraft aim for a confidence level of 3σ, a level of 1.8σ is acceptable at a PDR level and especially for a CubeSat. As further analysis is conducted and risks are retired, the confidence level is expected to rise to at least a 2σ level, representing a 95% probability of success.

i. Systems Level

$$P_S = 1 - 0.07097 = 0.92901$$
$$\boldsymbol{P_S = 92.9\% \rightarrow 1.8\sigma}$$

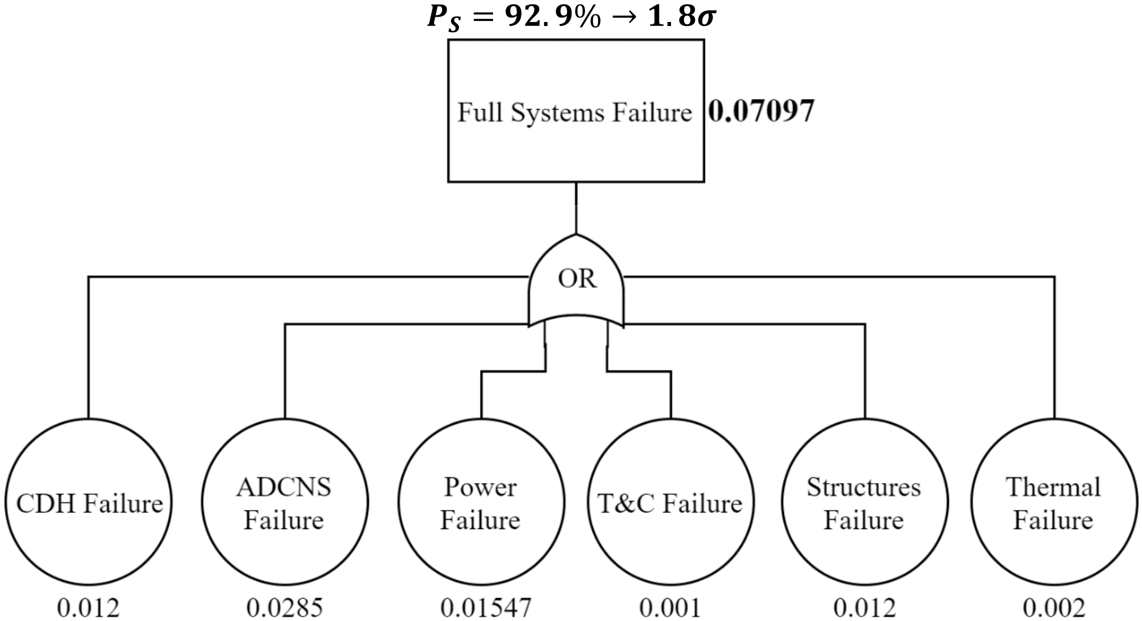


Figure 31: Condensed Systems Level Fault Tree

ii. ADCNS

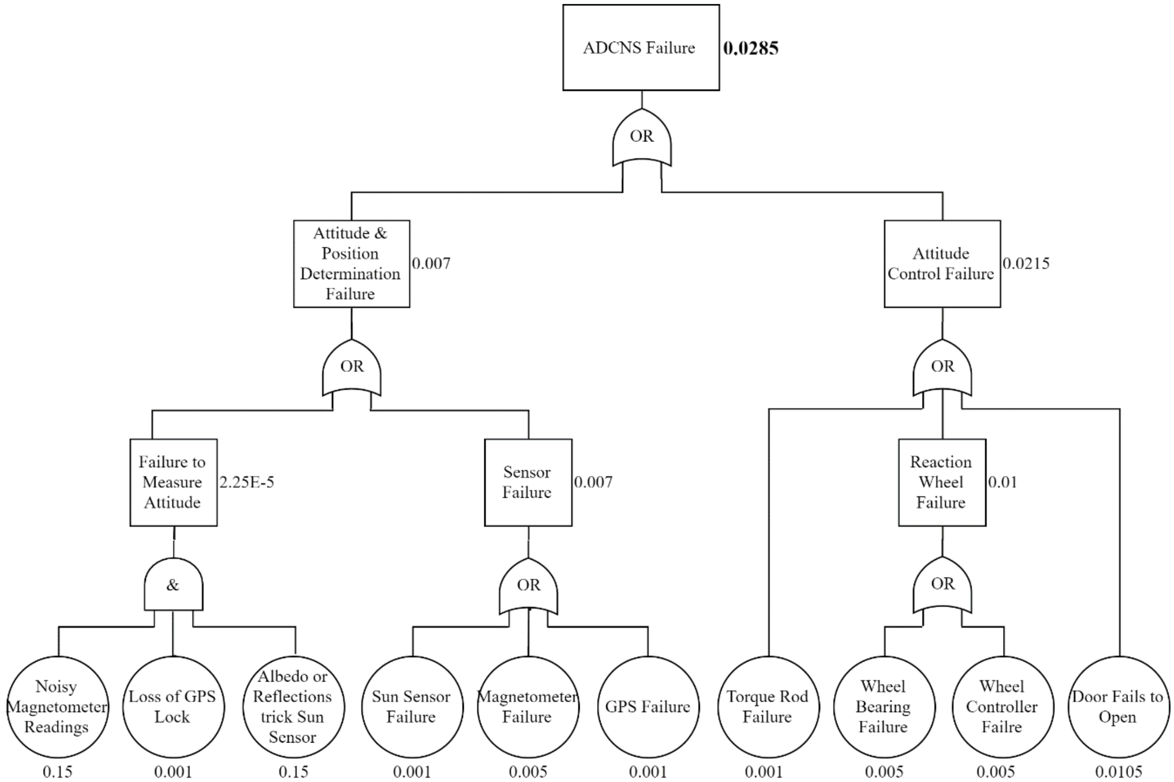


Figure 32: ADCNS Subsystem Fault Tree

iii. CDH

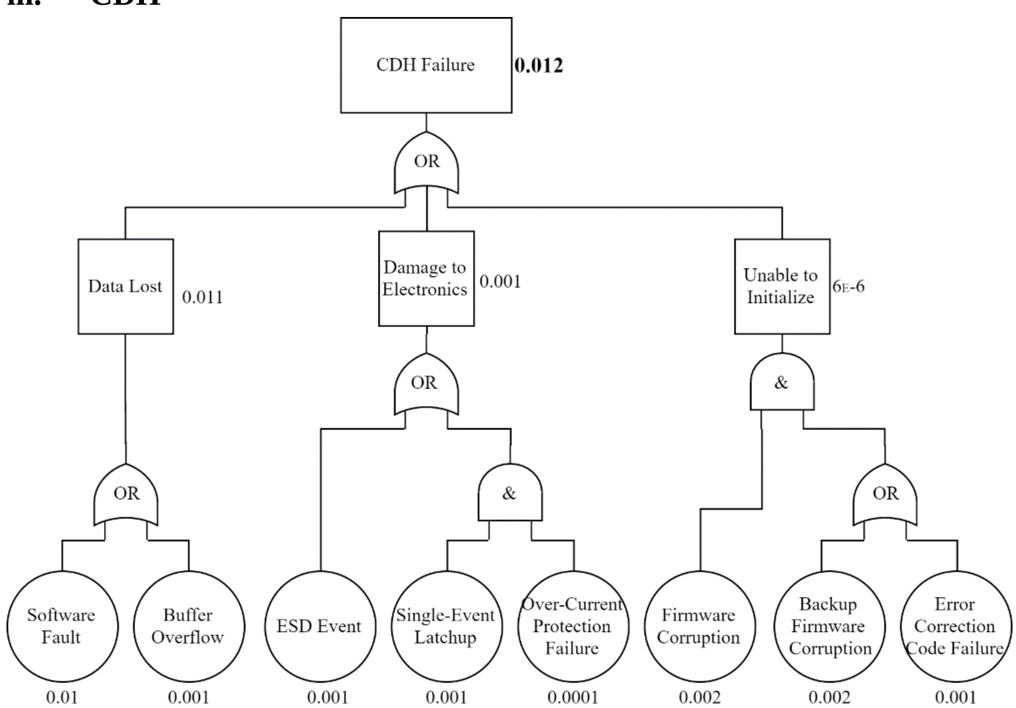


Figure 33: CDH Subsystem Fault Tree

iv. Power

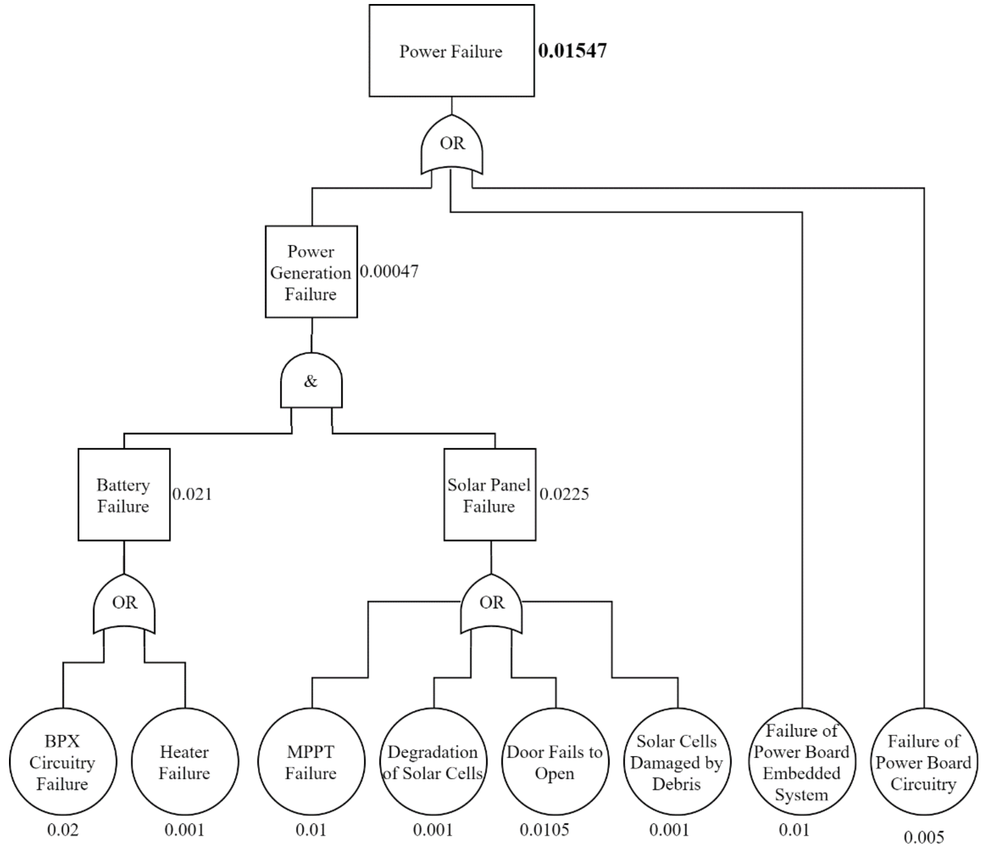


Figure 34: Power Subsystem Fault Tree

v. Structures

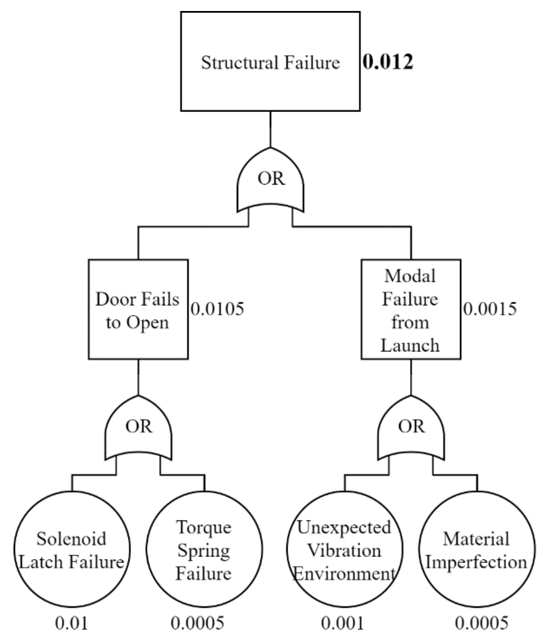


Figure 35: Structures Subsystem Fault Tree

vi. Thermal

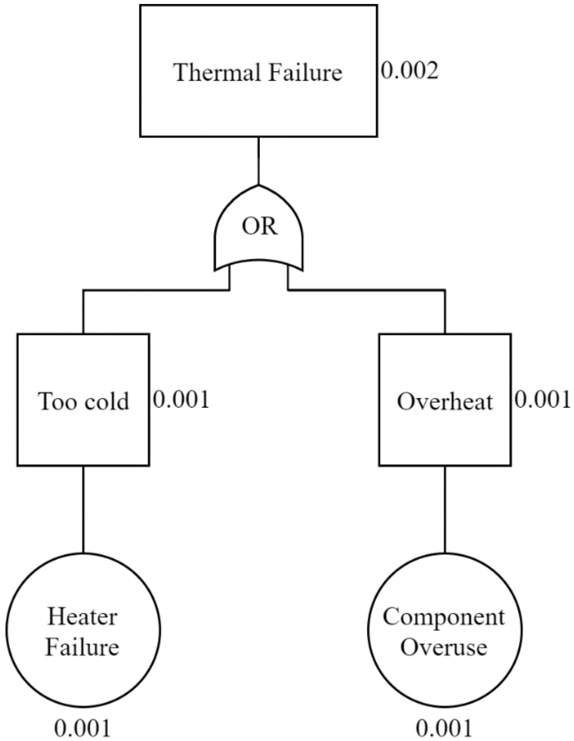


Figure 36: Thermal Subsystem Fault Tree

vii. T&C

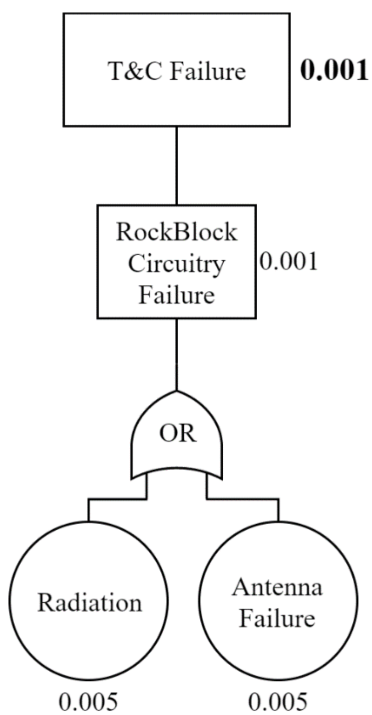


Figure 37: T&C Subsystem Fault Tree

b. Heat Map Analysis

Table 28: Risk Analysis Table for Heat Map

Code	Risk	Description	Mitigation	Impact	Likelihood	Risk Level
1	Structure Failure from Launch	Launch vibrations can cause failure in the primary structure of the Akash-Sat	Mitigation includes conducting thorough modal analysis as well as vibration tests on Akash-Sat1 before launch.	Catastrophic	Low-Medium	**Medium**
2	Payload Door Deployment Failure	It is possible for the payload door to jam, preventing the parabolic antenna from linking to ground. This can occur due to a solenoid latch failure or torque coil spring failure.	Conduct extensive testing and analysis before launch. Make the payload door single-fault tolerant.	Catastrophic	Low-Medium	**Medium**
3	Satellite Temperature Falls Below Minimum Allowable Temperature	The satellite falling below the minimum temperature would lead to subsystem failures and it would likely result from the failure of heaters.	Utilize multiple heaters per component and go into a thermally safe attitude when this minimum is detected.	Significant	Low	**Low**
4	Satellite Temperature Exceeds Maximum Operating Temperature	The satellite exceeding the maximum temperature would lead to subsystem failures and it would likely result from overuse of components.	Active thermal control and decreased power consumption during hot cases.	Significant	Low	**Low**
5	RockBlack Circuitry Failure	The failure of the RockBlock would result from wither radiation or antenna failure.	Incasing the RockBlock in radiation shielding can mitigate this failure. Including a second backup radio for redundancy is optional, but represents an extreme solution.	Catastrophic	Low	**Low**
6	Battery Current Limit	The LG Chem 18650 MJ1 is rated for up to 10A, which is what the spacecraft will operate at. This may be problematic as 10A is the current limit of the battery.	The power architecture could utilize both BPX battery packs in parallel to divide the load in high current cases.	Significant	Medium	**Medium**
7	Battery Flight Qualification	Qualifying the LG Chem 18650 MJ1 for spaceflight and ISS flight may be problematic and could represent significant programmatic and schedule risks.	ISS Qualification testing can be outsourced to firms such as GomSpace. Another COTS battery that has already been flight tested can also be alternately used.	Significant	Medium-High	**High**
8	Solar Panel Failure	The failure of the solar panel will result in the inability to generate power. It can result from degradation, MPPTG failure, debris damage, and a door failure.	Alternate modes of operation in which power generation is limited can be. This may include limiting payload operations.	Moderate	Low-Medium	**Low**

9	Failure of Power Board Embedded System	The failure of the Power Board embedded system would result in the inability to control the powered state of all subsystems.	A redundant secondary system can be used, but there are no other mitigation methods.	Catastrophic	Low	**Low**
10	Failure of Power Board Circuitry	The failure of the Power Board circuity would result in a total power board failure and the inability to control the powered state of all subsystems.	A redundant secondary system can be used, but there are no other mitigation methods.	Catastrophic	Low	**Low**
11	Single Event Latch Up	Solar radiation causes a change of state in the computer that creates a short circuit. Latch-up causes high current draw which can damage components and drain power more quickly.	Power board must be able to detect higher than nominal current draw and reset the system.	Mild	High	**Medium**
12	Single Event Upset	Solar radiation causes a bit error. Information, either in the form of a command or measurement, can become corrupted.	Components must be tested to meet acceptable radiation hardness, and periodic checks should be performed to confirm software integrity.	Insignificant	High	**Low**
13	Electrostatic Discharge	Charge built-up on spacecraft due to ionizing radiation.	Electrical components will be grounded to the chassis of the spacecraft, and bonded surface components of the spacecraft should have a resistance of less than 2.5 mOhm.	Moderate	Medium	**Medium**
14	Software Runtime Error	Invalid command sent or received, or another undetected software fault causes the CDH to become unresponsive.	A watchdog timer will be included to reset the CDH if the system is unresponsive.	Mild	Low-Medium	**Low**
15	Failure to Initialize	A firmware corruption in addition with wither a backup firmware corruption or error correction code failure will result in the CDH being unable to initialize and thus activate.	CDH can be thoroughly environmentally tested to mitigate risk of initialization failure. A Secondary firmware backup can also be included or a redundant flight computer architecture can be utilized.	Catastrophic	Low	**Low**
16	RWA Failure	Reaction wheels spin at very high speeds and consistently undergo large torques. Overtime, this can wear the bearings or other mechanical components and break the RWA.	Having a dormant spare RWA or more than three RWAs will mitigate this problem. This way if one is lost than the remaining three (or more) will still have 3 degrees of rotational freedom.	Catastrophic	Medium	**Medium**

17	Magnetic Torquer Failure	Magnetic torquers are used to dump momentum from the wheels. If a torquer fails and the spacecraft has three torquers, the spacecraft will not be able to dump momentum in one rotational direction. This will lead to one or more RWAs saturating.	One way to mitigate this problem is to buy robust, COTS magnetic torquers. They are used widely in industry so there are robust options that would be less susceptible to failure. Another way would be to have a spare torquers whose torque axis points in the same direction of the sum of the torque axes of the other torquerss. Therefore, if one of them fails then the spacecraft will still have 3 degrees of rotational freedom from its torquers.	Catastrophic	Low	**Low**
18	Sensor Failure	For the Akash mission, appropriate attitude sensors include sun sensors, magnetometers and gyroscopes. The sensors work together to provide attitude determination so a failure of one them will make the attitude more susceptible to error.	To mitigate this risk there must be multiple of each sensor. All three are small and cheap so it would be relatively easy to include multiple on the spacecraft. In addition, the solar panels can act as a rough attitude sensor.	Significant	Medium	**Medium**
19	Noise from Torquers Disrupts Other Operations	The torquers produce a magnetic field that interacts with earth's magnetic field to produce a torque on the spacecraft. However, this magnetic field can possibly affect some electronics on board. The magnetometer will be greatly affected and the reaction wheels, likely to be brushless motors, run via internal magnets that could be disrupted by the torquer.	Shielding the wheels with a Faraday Cage will help insulate them from the magnetic field of the magnetic torquers. Having the magnetometer on the extended door of the spacecraft may mitigate this problem, but to avoid it all together, the magnetometer data should only be used when the magnetic torquers are off.	Moderate	Medium	**Medium**
20	RWA Saturation	Angular momentum of the spacecraft exceeds the angular momentum capability of the RWA.	Sizing the RWAs such that there is a margin over the largest predicted angular momentum storage capability.	Moderate	Low	**Low**
21	RWA Jitter	Static or dynamic imbalance in the RWAs can cause vibrations in the wheels that undermine pointing accuracy and degrade the condition of the wheels.	Purchase COTS RWAs as they will be professionally balanced. Making RWAs is difficult, time expensive and can lead to static or dynamic imbalance.	Moderate	Medium	**Medium**

22	ADCNS Development Uncertainty	Due to the ADCNS subsystem being largely custom, it will require significant development and its capabilities may be changed. It also may lengthen the schedule of the spacecraft development period and purchasing an alternate COTS subsystem is expensive.	Dedicate significant work hours to ADCNS development and regularly evaluate the status of the subsystem. Allocate budget margin in the case a COTS ADCNS in required.	Moderate	Medium	**Low**
22	Operation of payload adds noise	The payload induces significant electrical noise on power and signal busses	Shield components and provide RF isolation.	Mild	Medium-High	**Medium**
23	Payload Specifications Significantly Change	As the payload is in development, a change in its operating specifications is possible. This is significant for all subsystems.	Keeping in regular contact with the payload provider and designing with margin for differing specifications.	Significant	High	**High**
24	Design Waiver Rejected	Rejection of the waiver for total chemical energy storage would require an appeal or a significant redesign of the spacecraft power architecture and the mission capabilities.	Overdesign battery and power architecture, or consider an alternate architecture as a contingency.	Significant	Low-Medium	**Medium**
25	Program Cost exceeds budget.	Solar cells, reaction wheels, sensors, and radios can be extremely costly and may cause the program to exceed the allotted budget.	More price-conscious trade studies may be studied and procured hardware should be protected from damage.	Significant	Medium	**Medium**
26	Student Turnover	Students have classes and exams, and they graduate. This represents significant fluctuations in the work schedule and turnover of experience and knowledge.	Proper and significant documentation, along with required and documented hours each week.	Mild	High	**Medium**

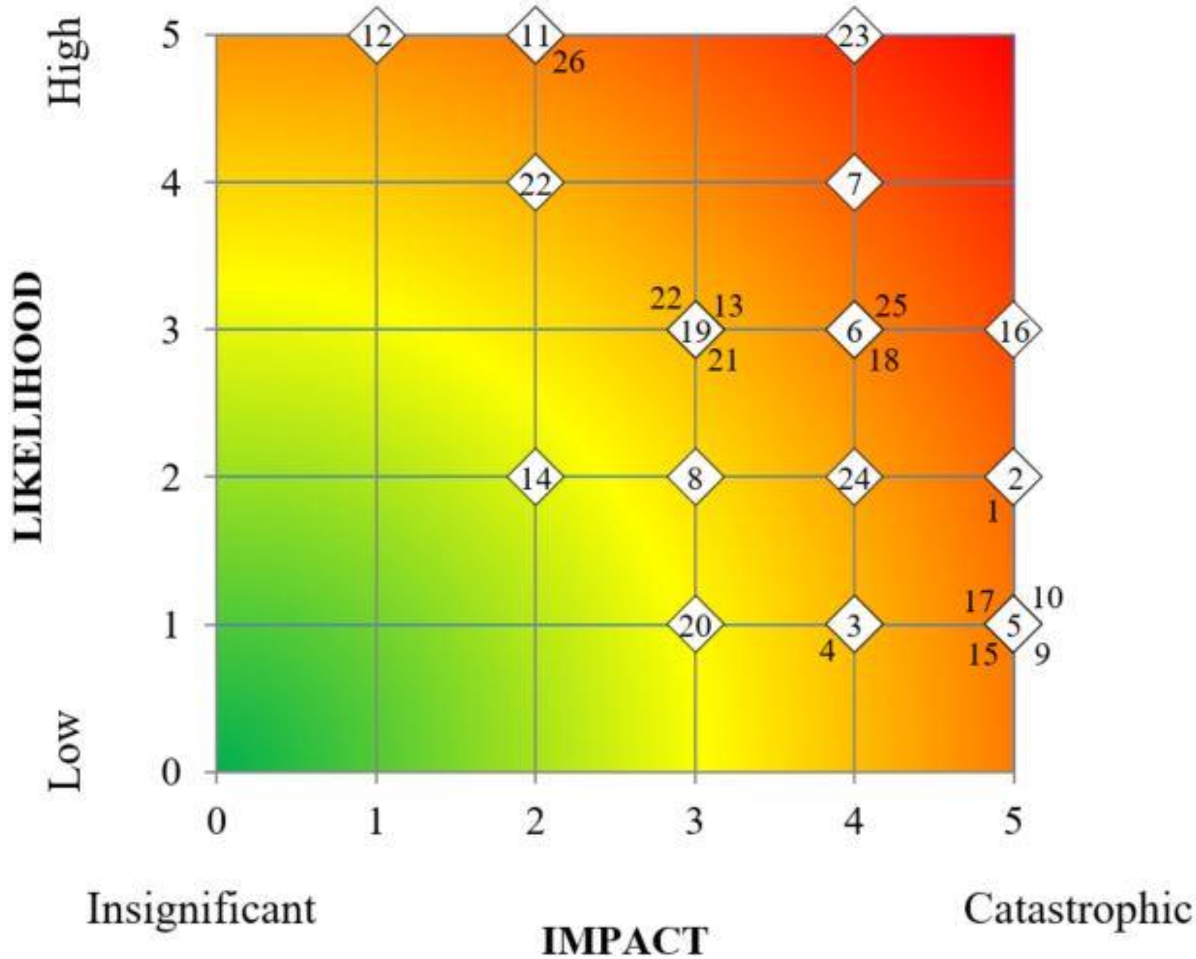


Figure 38: Risk Heat Map. Risks Specified in Table Above

IX. Long Lead Items

Below is a list of items with lead times of 4 weeks or greater. Most of the spacecraft's components are relatively common COTS components that have shorter lead times. The CubeSat industry, especially in the United States, has grown significantly in the past decade. This, along with the exceptional improvement in electronics, allows for many of the spacecraft's components to be COTS with short lead times.

The primary make/buy trade will be on the ADCNS subsystem. As of PDR, the ADCNS is expected to be made primarily using individual COTS components, many of which are not intended for ADCNS purposes. For instance, the RWAs are going to be brushless DC motors which satisfactorily function as RWAs after modification. Additionally, the magnetic torque boards are currently planned to be made, as they are relatively simple systems.

Thus, the ADCNS will be largely custom and developed at Cornell University, representing a significant cost reduction. The primary lead time for ADCNS is therefore the development period, which represents a significant schedule risk; to mitigate this, a make/buy trade will be regularly conducted on the ADCNS subsystem to evaluate if spacecraft-purposed COTS components or even an entire ADCNS subsystem must be purchased to offset any impacts to the program schedule before they occur.

Waivers and licenses are necessarily factored into the lead times for their respective components and subsystems. Specifically, the battery banks will require a chemical energy storage waiver which will require significant time for approval; if this waiver is rejected, then a trade must be conducted on alternate battery systems. FCC licensing will also be required for use of the RockBlock radio and this can take up to a year; it is highly unlikely it is rejected, but with a projected 18-month development time for Akash-Sat1, the licensing process will start immediately.

The structure must be designed and manufactured by a machinist. Although machining at Cornell University will be significantly cheaper, the added precision of a professional machinist is worth the cost, especially when compared to the cost of the rest of the spacecraft. Accordingly, it will take up to 4 weeks to have the structure machined, depending on the machinist used.

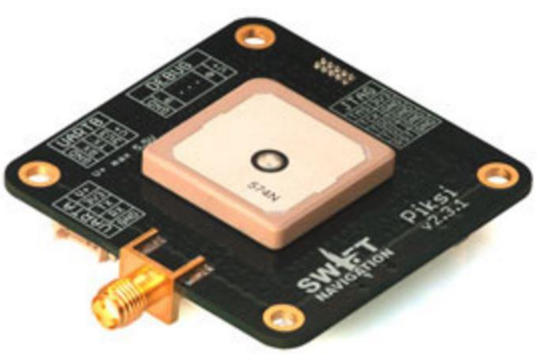


Figure 39: Swift Piksi GPS Receiver

Figure 40: RockBlock Radio

Table 29: Long Lead Items and thier respective procurement processes

Subsystem	Item	Lead Time	Make/Buy/Trade Needed
Power			
	Solar Panel Assembly	3-4 Months	Make
	Power Board	1-2 Months	Buy
	Battery Bank	1-2 Months	Buy
CDH			
	Beagle Bone Green Wireless	1-2 Months	Buy
T&C			
	Radios	1 Month	Buy
	FCC Licensing	1 Year	Buy
ADCNS			Trade Needed
	GPS	1 Month	Buy
	Reaction Wheels	8-12 Weeks	Buy
	Fine Sun Sensor	1 Week	Buy
Structure			
	Structure Bus Frame	4 Weeks	Buy
Payload			
	Payload	N/A	N/A
Launch Vehicle		1-2 Years	Buy

Figure 41: GomSpace NanoPower BPX

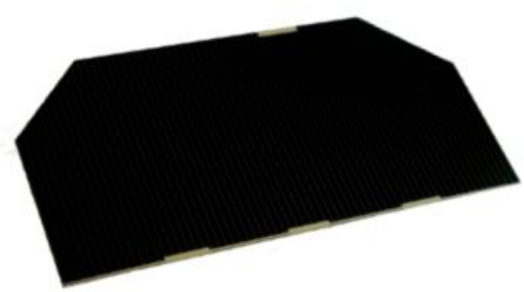


Figure 42: Azur Space Solar Cells

X. References

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XI. Appendix A

a. Power Budget Calculations

With the NanoPower BPX batteries configured to operate at 29.6V at 3500mAh (3.5Ah), if the payload demonstration power requirement is assumed to be 300W, then this will draw 10.13A.

$$\frac{300\ W}{29.6\ V} = 10.13\ A$$

If operation uses 10.13A, then a 3.5Ah battery will last for approximately 0.218 hours or 17 minutes. This is assuming an efficiency of 85%.

$$\frac{3.5\ Ah}{10.13\ A}\eta = 0.294\ hours = 17.621\ minutes$$
$$where\ \eta = 0.85$$

If there exists 60,000 mm (0.06 m^2) and an additional 40,000mm (0.04 m^2) of surface area from the satellite door and all that area is used for solar cells, then the power generation could be shown by multiplying this by the radiance constant, triple junction solar cell efficiency of 30%, surface area, and an assumption of 85% factor for the actual surface area covered. This gives approximately 33.19W of input power.

$$\begin{aligned} Power_{in} &= (Radiance)(30\%)(0.1\ m^2)(\eta_s) \\ &= (1370\ Wm^{-2})(30\%)(0.1\ m^2)(0.85) \\ &= 34.935\ W \end{aligned}$$

However, this is slightly inaccurate, as one should understand the actual geometry of the cells in question. Doing this, we see that this number of cells architecture maximizes power [+X: 6,-X: 6,+Y: 25,-Y: 5,+Z: 5 ,-Z: 5] assuming the use of Azur Space 3G30C cells. Please note that the +Y surface area includes the opened door and cells from the –Y wall. Using the power figures from the data sheet, the total power maximum comes out to be 31.449W. Assuming a zero-degree beta angle with sun-pointing, the spacecraft will be charging via the solar cells for 54.9 minutes or 0.915 hours during an orbit, and continuously be sun pointing then the spacecraft will receive approximately 28.772Wh per orbit.

$$31.449\ W * 0.915\ hours = 28.772\ Wh$$

This obviously changes with the beta angle and orientation, but is a good starting assumption. With the initial assumption of a 5 second transmission every 500 seconds, the full duration of transmit would be 27 seconds of an orbit. At 300W of power consumption would mean the spacecraft expends about 2.25Wh just from the payload. With approximately a 90-minute orbit, a payload orbit duty cycle of 1%, and the assumption that the spacecraft will only transmit during half the orbit, the total transmission time is determined to be 27 seconds.

$$Total\ Transmission\ Time = (90\ \text{minutes}) * (0.01) * (0.5) = 0.45\ minutes = 27\ seconds$$

A total transmission time of 27 seconds at 300W means that a total of 2.25Wh of energy consumption is required by the payload per orbit.

$$(27\ seconds) * (300\ W) = 8100\ WattSeconds = 2.25\ Wh$$

Given that the spacecraft generating 28.7720Wh per orbit as shown above and that the payload consumes 2.25Wh per orbit, 26.522Wh of energy per orbit remains for the rest of the spacecraft.

b. Link Budget Assumptions and Calculations

Table 30: Constants Used in Link Budget Calculations

Constants	Values
Orbit (km)	400
Temperature of Satellite (K)	270
Mean Earth Radius R_e (km)	6371.003
Equatorial Earth Radius R_{ee} (km)	6378.13649
Boltzmann's Constant k (J/K)	1.38E-23
Speed of Light c (m/s)	3.00E+08

i. Gain Calculation

$$G = 10\log_{10}(G_l) = 10\log_{10}(\left(\frac{\pi d}{\lambda}\right)^2\ e_A)$$
$$with\ e_A = 0.6$$

$$G = 36.3\ dB$$

ii. Half-Power Beam Width Calculations

$$\frac{1}{2}\ power\ beamwidth = 60\ \left(\frac{\lambda}{D}\right) = 2.25\ degrees$$

c. ADCNS Trade Studies

Reaction Wheel	BCT RWA	Maxon EC 32	Maxon EC 45	Sinclair RWA		Weight	Norm
Angular Momentum	5	4	5	5		8	0.216216
Torque	4	4	5	4		8	0.216216
Size	3	2	1	3		6	0.162162
Power	5	3	1	5		4	0.108108
Cost	1	5	4	1		8	0.216216
Adaptibility	3	2	2	3		3	0.081081
						37	
Sum	**3.43**	**3.62**	**3.46**	**3.43**			

Magnetic Torquers	NSS MTR	PSS MTR	Electromagnet	Custom Torque Rod	Custom Torque Boards	Weight	Norm
Dipole Moment	5	5	3	5	5	8	0.258065
Power	4	4	4	4	2	3	0.096774
Size	3	3	3	2	5	5	0.16129
Cost	1	1	5	5	3	7	0.225806
Adaptibility	3	3	1	3	5	8	0.258065
						31	
Sum	**3.16**	**3.16**	**3.03**	**3.90**	**4.26**		

9-Axis IMU	Razor IMU	Adafruit Imu	Anaglog Decives IMU			Weight	Norm
Accuracy	5	4	3			8	0.380952
Size	3	3	3			3	0.142857
Power	3	3	3			2	0.095238
Cost	2	4	5			3	0.142857
Adaptibility	4	4	3			5	0.238095
						21	
Sum	**3.86**	**3.76**	**3.29**	**0.00**			

Sun Sensor	NSS FSS	CubeSatShop FSS	Custom CSS- Squar	Custom CSS- Strip		Weight	Norm
Accuracy	5	5	3	3		9	0.3
Size	5	4	2	3		5	0.166667
Power	5	5	2	2		2	0.066667
Cost	1	1	5	5		9	0.3
Adaptibility	4	4	2	3		5	0.166667
						30	
Sum	**3.63**	**3.47**	**3.20**	**3.53**			

GPS Receiver	Piksi					Weight	Norm
Accuracy	5					8	0.296296
Size	4					3	0.111111
Power	2					6	0.222222
Cost	3					5	0.185185
Adaptibility	4					5	0.185185
						27	

ADC Full Subsytem	BCT XACT	Clyde Space ADCS	CubeSpace CubeControl	Custom		Weight	Norm
Angular Momentum	4	4	4	4		8	0.173913
Torque	4	4	4	4		8	0.173913
Size	2	4	4	3		2	0.043478
Power	5	3	3	2		5	0.108696
Cost	0	1	1	5		10	0.217391
Accuracy	5	3	3	2		8	0.173913
Adaptibility	5	3	4	2		5	0.108696
						46	
Sum	**2.57**	**2.43**	**2.54**	**3.04**			

d. Chemical Storage Requirement Waiver

CubeSat Design Specification Waiver Inquiry Request

Created on 15 June 2015

Date: 05/15/2017

Name of request initiator: Jamil Fares

Organization: Cornell University

Contact Information:

email: jef283@cornell.edu
cell: (617) 997-9505

Mission Name: Akash-Sat1

Brief CubeSat Description (size, mission goal, etc.):

GaN on diamond telecommunications technology demonstration, aimed to achieve a symbol downlink rate 0f 100Gbit/s

Which are requirements not met?

3.1.5 Total stored chemical energy will not exceed 100 Watt-Hours.

Describe how the requirement will not be met.

The total stored chemical energy will be 207 Watt-Hours, and will be in the form of Li-ion cells.

Please explain how this deviation from the standard requirements will not present an added risk to the mission. Include pictures as necessary.

This quanitity of chemical storage is required for mission success and payload operations.

All cells shall be qualified for ISS flight via testing per NR-SRD-139 in order to verify functionality and significantly mitigate any risk of failure or uncertainty in their properties. The batteries shall also be controlled and regulated in 8S battery packs and all cells shall be individually monitored and controlled. Software protocols shall be implimented to prevent thermal runaway in the battery packs.

XII. Senior Design Appendices