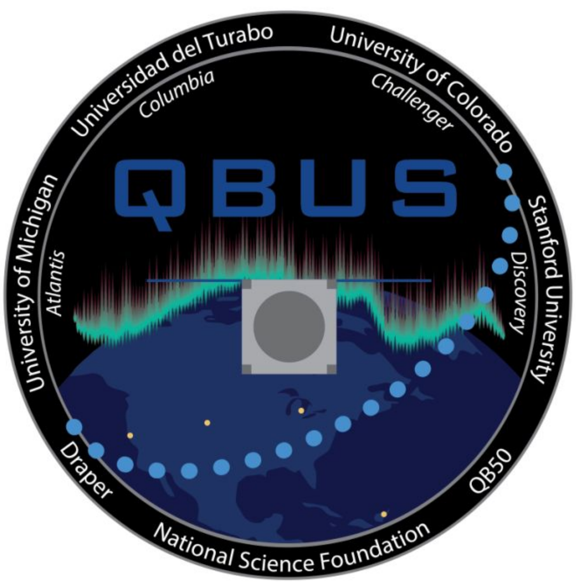
| | | |
|---|---|---|
| | | |
| **CubeSat name / number** | US02 Atlantis and US04 Columbia | |
| **Lead institute** | University of Michigan, Ann Arbor | |
| | | |
| **Author** | | |
| **Checker** | -- | -- |
| **Approver** | Aaron Ridley | ridley@umich.edu |

# QB50

# Preliminary Design Review Data Package




**Satellite Team Change Log**

| Revision | Date | Author | Change Log |
|---|---|---|---|
| 1 | 30 April 2016 | | Initial revision |
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| NR-SRD-029 | NanoRacks CubeSat Deployer (NRCSD) Interface Control Document | 0.36 |
| | QB50 System Requirements and Recommendations | 7 |
| | Cal Poly CubeSat Design Specification | 12 |
| | | |
| | | |

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**Acronyms**

| | |
|---|---|
| 1U, 2U, 3U | One, two, and three-unit CubeSat sizes, respectively |
| ABF | Apply Before Flight |
| ACRR | Adjacent Channel Rejection Ratio |
| ADCS | Attitude Determination and Control Subsystem |
| BOL | Beginning of Life |
| BPB | Battery Protection Board |
| BTJM | Triple-Junction with Monolithic Diode Solar Cells |
| CalPoly | California Polytechnic State University, SLO |
| CDR | Critical Design Review |
| CIC | Solar Cell + Interconnects + Cover glass |
| COTS | Commercial Off The Shelf |
| EOL | End of Life |
| EPS | Electrical Power System |
| FCPU | Flight Computer |
| FIPEX | Flux-Φ-Probe Experiment |
| FPGA | Field-Programmable Gate Array |
| GMSK | Gaussian Minimum Shift Keying |
| GND | Ground |
| GRIFEX | GEO-CAPE ROIC In-Flight Performance Experiment |
| HK | Housekeeping |
| ICD | Interface Control Document |
| IMU | Inertial Measurement Unit |
| ISIS | Innovative Solutions In Space |
| JPL | Jet Propulsion Laboratory |
| M-Cubed | Michigan Multipurpose Minisat |
| MET | Mission Elapsed Time |

| | |
|---|---|
| MoBo | Mother Board |
| MXL | Michigan eXploration Laboratory |
| NASA | National Aeronautics and Space Administration |
| NRL | United States Naval Research Laboratory |
| NSF | National Science Foundation |
| OBC | On-Board Computer |
| OBDH | On-Board Data Handling |
| ORB | Output Regulation Board |
| PCB | Printed Circuit Board |
| PIB | Payload Interface Board |
| PIM | Payload Interface Module |
| RAX | Radio Aurora eXplorer |
| RBF | Remove Before Flight |
| RF | Radio Frequency |
| RTC | Real Time Clock |
| SEU | Single Event Upset |
| SME | Space Mission Engineering: The New SMAD |
| SMM | Solar Maximum Mission |
| SPRL | Space Physics Research Lab |
| SSID | Secondary Station Identifier |
| STK | Systems/Satellite Tool Kit |
| SU | Science Unit |
| SVN | Subversion |
| TBD | To Be Determined |
| TCB | Torquer Control Board |
| TLE | Two-Line Element |

| | |
|---|---|
| TMI2C | Telemetry I2C Line |
| TML | Total Mass Loss |
| TT&C | Telemetry, Tracking and Command |
| TUD | Dresden University of Technology |
| UHF | Ultra High Frequency |
| UTJ | Ultra-Triple-Junction |
| VHF | Very High Frequency |
| VKI | von Karman Institute for Fluid Dynamics |
| WDT | Watchdog Timer |
| WOD | Whole Orbit Data |
| WSTPC | Wilson Student Team Project Center |

# 1 Design Overview

## 1.1 Mission Concept

### 1.1.1 Overview

The thermosphere and ionosphere are unique regions of the atmosphere because they represent boundaries for chemical, neutral, and plasma processes and dynamics. Below the thermosphere, the atmosphere is dominated by neutral dynamics, which are often slow and dominated by forces such as gradients in pressure and the Coriolis Effect. For example, the largest disturbances in the troposphere, tropical cyclones, take days to develop, many more to propagate, and a few days to decay. In the ionosphere, the time-scales for physical processes are extremely fast. For example, the high-latitude electric field can take only 10-30 minutes to change after an interplanetary magnetic field change [Ridley et al., 1997, 1998, Ruohoniemi and Greenwald, 1998, Lu et al., 2002], meaning that the global ion dynamics can change within 30 minutes. In the thermosphere, the neutral dynamics are often coupled with the ion dynamics - forces such as ion drag drive winds [Killeen and Roble, 1984, Deng et al., 1993, Deng and Ridley, 2006]. With ever changing ion drifts [Matsuo et al., 2002, Shepherd et al., 2003, Foster et al., 2004], the ion drag can change quite suddenly, which often does not allow the winds to ramp up to the ion speeds, causing a frictional heating that occurs between the ions and neutrals [Codrescu et al., 1995, Deng and Ridley, 2007]. In the auroral zone, the frictional heating can turn on rapidly [e.g., Yiğit et al., 2012] and can cause traveling atmospheric disturbances that propagate from the high latitudes to the equatorial region [Sutton et al., 2005].

This statistically driven, simplified vision of how the atmosphere works has been built up over many NASA missions and ground-based campaigns. The problem with this large scale, slowly varying vision of the gross dynamics of the system is that the atmosphere responds much faster to changes in the aurora and the corresponding ion drifts [Conde and Smith, 1998b,a, Innis and Conde, 2001, Innis and Conde, 2002]. There is a fundamental lack of knowledge about the actual dynamics of how the atmosphere responds to forcing and feedbacks on short temporal and small spatial scales. With a typical orbit of 90 minutes, the thermosphere can change dramatically due to auroral forcing - so much so that it is sometimes impossible to determine what actually occurred between passes.

This Preliminary Design Review describes two satellites within the QB50 constellation mission. This constellation of satellites will measure how the thermospheric and ionospheric density changes in the auroral zone across multiple spatial and temporal scales in the hope of determining the dynamics and propagation of high latitude effects to the equatorial zone. This mission is a unique opportunity to measure the dynamics of the lower thermosphere and ionosphere with a variety of sensors spread throughout an orbital plane.

This mission is a unique opportunity in many regards. The QB50 mission, if successful, will provide upper atmospheric data continuously over an entire orbit. The community has never had such an opportunity before, and, most likely, will not for many years to come. The CubeSats that are described within this document integrate thoroughly tested commercial off the shelf (COTS) components into satellites that have been designed for missions with lower costs and shorter lifetimes than traditional NASA-funded satellites. The majority of these components have flown on at least one mission and many of them will fly on at least two other missions before the two satellites described here are delivered for flight.

### 1.1.2 Space Physics Research Laboratory

The Space Physics Research Laboratory (SPRL) was established in the early post-war years with the mission of experimental and theoretical studying the natural world with state-of-the-art instruments and models. In the most recent years, SPRL faculty and engineers have built over 30 space instruments, instrumented numerous sounding rockets, balloons and aircraft, and developed ground-based instruments. SPRL has a strong reputation for designing, constructing, testing, operating, and analyzing data from space flight instruments. Most recently, SPRL has been developing the Cyclone Global Navigation Satellite System (CYGNSS) constellation of eight small satellites with a payload using GPS bi-static scatterometry to measure ocean surface wind speed.

## 1.2 Systems Overview

### 1.2.1 Requirements

*Table 1: General Requirements*

| ID | Requirement | Section Addressed | Notes |
|---|---|---|---|
| QB50-SYS-1.7.1 | The CubeSat shall be designed to have an in-orbit lifetime of at least 6 months. | | |
| QB50-SYS-1.7.2 | The CubeSat shall not use any material that has the potential to degrade in an ambient environment during storage after assembly, which could be as long as approximately 2 years. | 1.4.3 | |
| QB50-SYS-1.7.3 | The CubeSat shall withstand a total contamination of 3.1 mg/m2 at all phases of the launch vehicle ground operation and in flight. | TBD | Ignore |
| QB50-SYS-1.7.4 | The CubeSat shall withstand a maximum pressure drop rate of 3.92 kPa/sec. | TBD | Ignore |
| QB50-SYS-1.7.5 | If a CubeSat has any special requirement in terms of cleanliness, handling, storage or shipment, these shall be communicated to the QuadPack integrator (ISIS) and also be approved by ISIS, 12 months before delivery of the CubeSat and also highlighted in the User Manual. | N/A | Ignore |
| QB50-SYS-1.7.6 | Apply Before Flight (ABF) items, including tags and/or labels, shall not protrude past | N/A | Possible change |

| | | | |
|---|---|---|---|
| | the dimensional limits of the CubeSat extended volumes (as defined in Figure 13) when fully inserted. | | |
| QB50-SYS-1.7.7 | All Remove Before Flight (RBF) items shall be identified by a bright red label of at least four square centimeters in area containing the words "REMOVE BEFORE FLIGHT" or "REMOVE BEFORE LAUNCH" and the name of the satellite (CubeSat QB50 ID) printed in large white capital letters. | 3.4.4 | Possible change |
| QB50-SYS-1.7.8 | The CubeSat QB50 ID (e.g. BE05) shall be printed, engraved or otherwise marked on the CubeSat and visible through the access hatch in the door of the QuadPack. | 1.4.3 | Ignore |
| QB50-SYS-1.7.9 | The CubeSat provider shall transfer the whole orbit data and science data to the QB50 central server within 24 hours following reception on the ground. | 1.3.6 | |
| QB50-SYS-1.7.10 | All of the whole orbit data and science data downlinked to the ground shall be stored in the individual CubeSat server up to 6 months after the completion of the mission. | 1.3.6 | |
| QB50-SYS-1.7.11 | CubeSats carrying the standard atmospheric sensors shall be able to commence the science payload operations within one week after deployment in orbit. | 1.2.6 | |
| QB50-SYS-1.7.12 | CubeSats carrying the standard atmospheric sensors shall operate it for a period of at least 2 months. | 1.2.7 | |
| Nanoracks-1.1 | CubeSats shall be passive and self-contained from the time they are loaded into the NRCSD for transport to the ISS and until after deployment from | | |

| | | | |
|---|---|---|---|
| | the NRCSD. No charging of batteries, support services, and or support from ISS crew is provided after final integration. | | |
| Nanoracks-1.2 | CubeSats shall not contain pyrotechnics unless the design approach is pre-approved by Nanoracks. Electrically operated melt-wire systems for deployables that are necessary controls for hazard potentials are permitted. | | |
| Nanoracks-1.3 | CubeSats must have a timer (set to a minimum of 30 minutes) before satellite operation or deployment of appendages. If deploy switches should be released causing the timer to run, the timer must automatically re-set whenever the Remove Before Flight (RBF) feature is replaced and/or the deploy switches are returned to the open state. | | |
| Nanoracks-1.4 | CubeSats should not have detachable parts or create any space debris during launch or normal mission operations. | | |
| Nanoracks-1.5 | CubeSats shall use a secondary locking feature for fasteners external to the CubeSat chassis. An acceptable secondary locking compound is LocTite. Contact NanoRacks for the proper locking compound application procedure. Other secondary locking methods must be approved by NanoRacks. | | |
| Nanoracks-1.6 | A description of frangible materials (e.g. solar cells) must be provided to NanoRacks for approval. | | |

### 1.2.2 Overview

As part of the QB50 Mission, Atlantis and Columbia are two 2U CubeSats carrying the FIPEX payload. The attitude determination and control block includes magnetorquers, magnetometers, coarse sun sensors, and an inertial measurement unit. The magnetorquers are made in house and the CubeSat will consist of three iron core magnetorquers. The electrical power system will consist of solar cells, a voltage regulation system, a simple set-point tracker and lithium batteries. The solar cells will be in strings of five and placed on three body panels. The on-board data handling system will employ a computer processor and a watchdog timer. This will allow the telemetry, tracking, and control system to employ a monopole antenna, which has been flight proven, along with heritage radio systems, in order to relay FIPEX data back to our ground stations at the University of Michigan, or any of the other ground stations in the United States QB50 network. Structurally, the CubeSats will utilize a backplane configuration; therefore, subsystem boards and side panels will be plugged into a backplane PCB. Finally, in order to help ensure that all systems function nominally during flight, a passive thermal control system will be implemented. A 1U example of these heritage components is shown in Figure 1. The full analysis and description of each of these subsystems, along with their associated requirements, can be found later in the document.

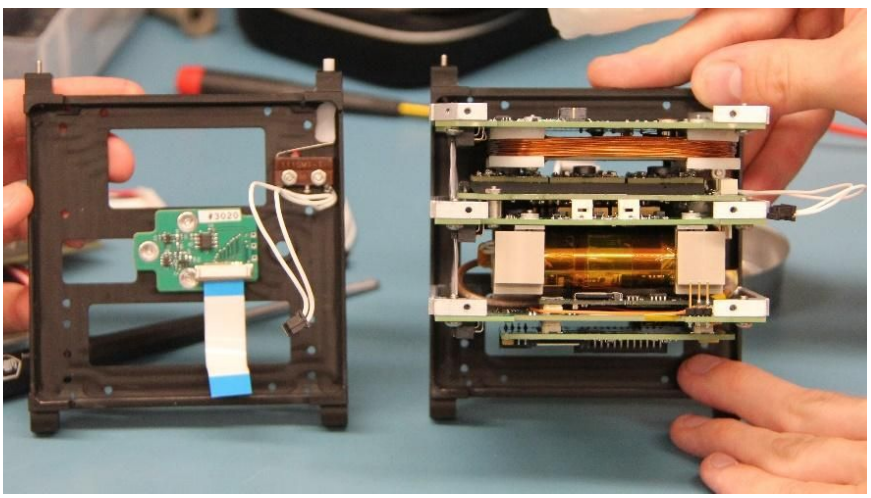


*Figure 1: Integration of M-Cubed-2, a 1U CubeSat built at The University of Michigan*

### 1.2.3 Modes of Operation

Atlantis and Columbia will have four main modes of operation. Safe Mode, FIPEX OFF, FIPEX ON, and UHF Downlink. These modes will be cycled through during an initial set of post launch phases and may continue to be used for the duration of its lifetime.

#### Safe Mode

Safe mode utilizes only the systems that are most essential to spacecraft health. It draws minimum power, and is the default operating mode. This mode is designed to remain power positive. This is the primary mode used while de-tumbling, and will also be used as a base for checking out subsystem hardware in the checkout phase.

**FIPEX OFF**
This mode operates all subsystems with exception of the science payload FIPEX and UHF downlink. Normal beaconing will occur.

**FIPEX ON**
This mode operates all subsystems including FIPEX, but excluding UHF downlink. Normal beaconing will occur.

**UHF Downlink**
This mode operates all subsystems, excluding FIPEX. It also encompasses data transmission to a ground station, with more information than what is sent with simple beaconing.

### 1.2.4 Operational Phases

There are five main phases which the spacecraft will execute after launch. These phases will shift the spacecraft through its four modes of operation in a way that allows system components to be tested while minimizing risk to other subsystems.

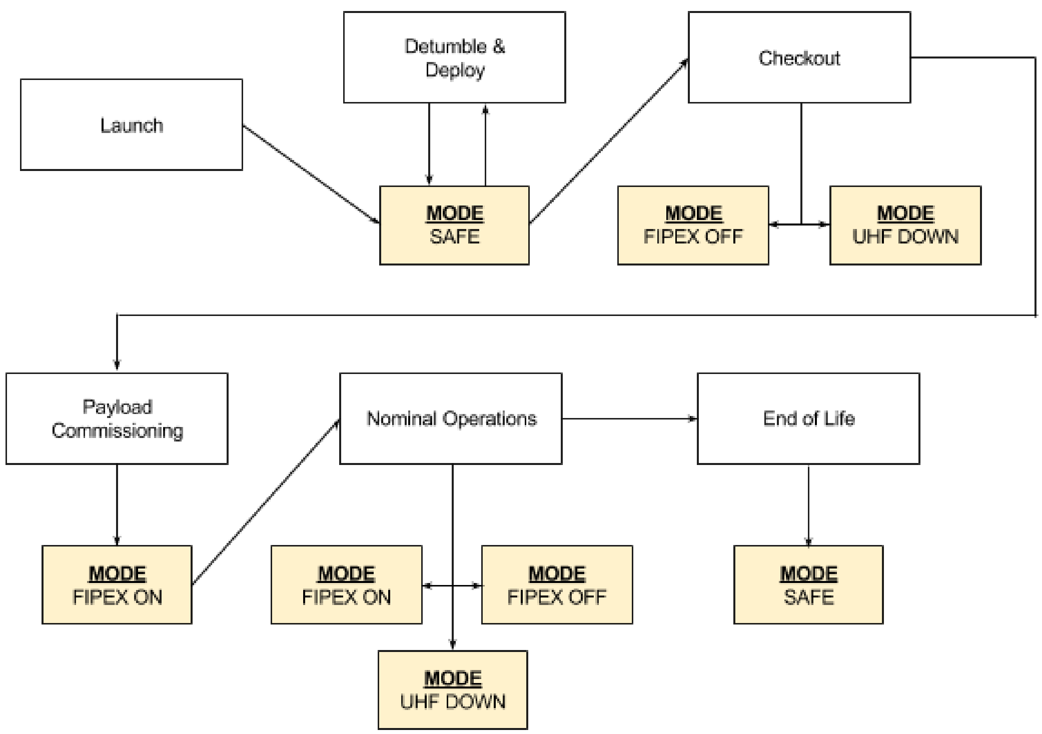



*Figure 4: Operational phases and use of four modes*

**Launch Phase**
The launch phase of the mission begins at launch and continues until Atlantis and Columbia are ejected from the NanoRacks CubeSat Deployment system onboard the ISS. During ascent, the CubeSats will be powered off neither recording or transmitting telemetry. The batteries will be optimally charged however, so as to power on once the separation switches signal ejection. Two plunger switches are positioned

diagonally in the structural feet of the –Z face with a single roller/slide switch embedded in the side of the CubeSat rail.

## Deployment and De-tumble Phase

Once the NRCSD door is opened and the CubeSats are ejected, the separation switches are flipped and the spacecraft bus is powered on. The mission clock begins at first power on and will not be reset for the duration of the mission. By default, Atlantis and Columbia will always boot into Safe Mode. At this point, both the flight computer (FCPU) and watch-dog timer (WDT) are powered on. The WDT monitors a heartbeat pulse from the FCPU, and will power cycle the spacecraft if the signal is not received. The key events and times during the deployment phase are below (Table 10). Time is listed in approximate Mission Elapsed Time (MET) which is cumulative, and does not stop. Since the elapsed time to receive the first health and status beacon will depend on the orbital parameters and time of launch, these times are shown as approximate times based on revisit times typical of a polar orbiting LEO spacecraft.

*Table 10: Deployment Timeline*

| **Deploy + time (minutes)** | **Event** |
|---|---|
| 00:00 | CubeSat deployed from the NRCSD and CubeSat switches indicate successful deployment. |
| 00:00 | CubeSat powers on |
| 30:00 | UHF antenna deploys |
| 35:00 | Begin RF Beaconing |
| After ground contact | Subsystem checkouts/ commanding tests |
| Automatic | ADCS sensor calibration and detumble |
| +7 days | Payload commissioning |

### Antenna Deployment

The UHF antenna is first to deploy. It is folded over the top cover (+Y face) and is secured with a dedicated monofilament deployment circuit. The antenna will be deployed autonomously by the FCPU at 30 minutes MET. UHF health and status beacons also begin transmission at this time. Photodiodes shaded by the stowed antenna will confirm deployment in sunlight. The FCPU autonomously determines whether or not the antenna has successfully deployed and if it appears that it has not, it will attempt to deploy the antenna five more times until it exits into a ninety (90) minute hold to allow the spacecraft to charge before attempting again. Once the FPCU determines successful deployment of the antenna, Atlantis and Columbia enter a low power state. When the health and status beacon is received at the Ann Arbor ground station, the ground station will verify antenna deployment. This command may or may not be sent during the same pass that the deployment confirmation beacon is received.

**De-Tumble**

After the antenna deployment, the FCPU will power on the magnetorquer control board to provide magnetic damping of the spacecraft in order to stabilize its tumbling spin rate. This also serves as the checkout for the magnetorquer control board (TCB). This will remain on during subsystem checkout until the ADCS subsystem is fully operational and begins active attitude control.

### Subsystem Checkout Phase

While the spacecraft are de-tumbling, each subsystem proceeds through its checkout procedures in the order given below (Table 11). Between each checkout procedure, Atlantis and Columbia enter a low power state. Following successful checkout of each subsystem, the spacecraft enter FIPEX OFF mode. FIPEX requires additional checkout procedures outlined in the Payload Commissioning Phase.

*Table 11: Checkout Phase Timeline*

| Event | MET |
|---|---|
| ADCS Core Checkout | ≈ 2 days |
| ADCS Calibration | ≈ 3 days |
| Standby Hold | ≈ 3 days |
| Payload Commissioning (separate phase) | ≈ 3 days |
| Nominal Operations (separate phase) | ≈ 5 days |

**ADCS Core Checkout**

The ADCS core checkout verifies that the ADCS subsystem is operating correctly. It steps through communications between the Atlantis and Columbia FCPUs and the coarse attitude sensors (photodiodes and magnetometers) before proceeding to checkout of fine attitude sensors (gyro). Once sensor data quality is confirmed by the ground, the ground station commands the FCPU to test control of the Telemetry I2C line (TM I2C). Once the ground station has determined that the ADCS is functioning normally, the ground station commands the spacecraft to enter its low power state.

**ADCS Calibration**

The ADCS calibration phase follows the ADCS subsystem checkout and is used to fine-tune the magnetometers, gyros, sun sensor, and magnetorquers as a system. First a full set of sensor outputs is taken and relayed to ground over the UHF beacon. After the ground determines any calibration parameters needed, those parameters are uploaded and a full set of sensor outputs is taken again. This is repeated once per orbit until sensors are calibrated satisfactorily. Over a minimum of one orbit, the spacecraft attitude is logged and evaluated by the ground to confirm that Atlantis and Columbia are meeting their pointing requirements. Finally, once successful pointing is confirmed, the ground commands each spacecraft to enter its low power state.

**UHF Radio Checkout**

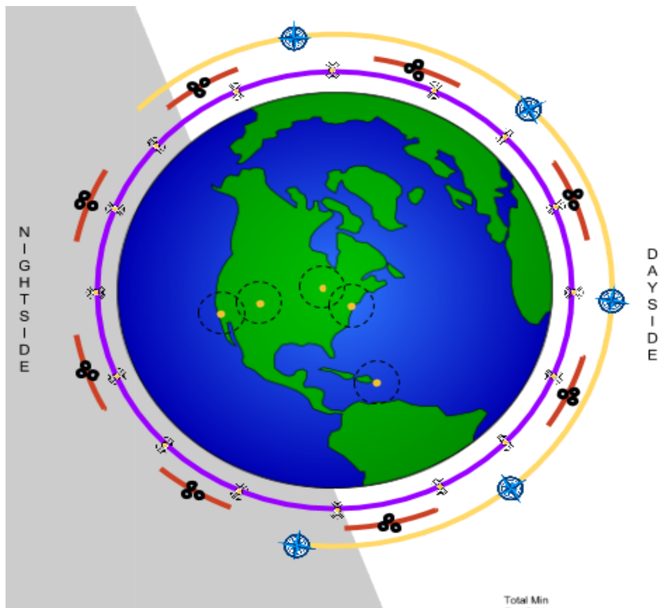
Following ADCS calibration, Atlantis and Columbia will confirm radio and antenna performance by downloading pre-stored packets of information that are not usually transmitted during normal beaconing. Once transmitting is complete the spacecraft will return to FIPEX OFF mode. Each spacecraft will beacon every 30 seconds. Each beacon will contain bytes of information regarding spacecraft health and telemetry, as identified with the unique satellite ID.

### 1.2.5 Payload Commissioning Phase

FIPEX commissioning will begin no later than five days after launch. (see above at 7+ days)

### 1.2.6 Nominal Operations

Atlantis's nominal operations will consist of toggling between three modes; FIPEX ON, FIPEX OFF, and UHF Downlink, while streaming health and status beacons over UHF. Beacons will be received by the ground stations in 30 second intervals.




| | | Total Min Per Orbit |
|---|---|---|
| RF Beacon | Every 10 Seconds for Entire Orbit | 9.5 |
| FIPEX | On 20 minutes per orbit, only in day side | 20 |
| Sun Sensor | Only on while sun is in view | 63 |
| Ground Stations | Downlink 5-20 min per orbit | ~ 16 |

*Figure 6: Depiction of duty cycles during nominal operations*

### 1.2.7 End of Life Phase

At a certain point Atlantis and Columbia will begin to fail due to battery storage, solar cell efficiency, memory failure due to radiation, thermal damage, or an assortment of other reasons. When it begins to fail, the satellites will likely enter safe mode in order to preserve power and transmit telemetry and health beacons for as long as possible.

## 1.3 Payload Design

### 1.3.1 Requirements

*Table 12: FIPEX Requirements*

| ID | Requirement | Section Addressed |
|---|---|---|
| FPX-RQ-1 | The CubeSats carrying the FIPEX SU shall have an attitude control with pointing accuracy of ±20 and pointing knowledge of ±2 from its initial launch altitude down to at least 200 km. | |
| FPX-RQ-2 | The CubeSat carrying the FIPEX SU shall determine its position to within 10 km accuracy. | |
| FPX-RQ-3 | On receiving a science or telemetry packet from FIPEX, the OBC shall attach the following information within 500 ms after receiving the first Byte to it: current real time, current spacecraft attitude and position. The full packet shall be stored in the OBC Mass Memory for later request of transmission to ground. The format of this packet is defined in section 3.4 and section 5 of the FIPEX ICD. | |
| FPX-RQ-4 | It shall be possible to request and transmit science and telemetry packets separately. | |
| FPX-RQ-5 | The implementation of error handling of the mass memory storage should assure that the latest data can be transferred to ground | |
| FPX-RQ-6 | Each CubeSat carrying a FIPEX SU shall communicate a volume of at least 0.3 Megabits of SU data per day to the ground station that is operated by the university providing the CubeSat. | |
| FPX-RQ-7 | The CubeSat shall have two independent memory storage units, where SU data can be store. At least 10 MB should be reserved for the science unit. | |
| FPX-RQ-8 | The OBC shall have the ability to run a command script of the format specified in section 3.2 at the specified STARTTIME. The OBC shall execute each command consecutively. | |
| FPX-RQ-9 | The OBC shall wait for the delay condition DELAYx on each command, before executing the next command. During waiting for the next command execution the OBC must be able to receive data from the SU. A command can contain an instruction for the OBC only, or an instruction which shall be send to the SU unit | |
| FPX-RQ-10 | In case of an error during execution of the command script the OBC shall ABORT the current script, request a housekeeping packet and TURN OFF the SU. | |
| FPX-RQ-11 | The CubeSat shall provide a method to inform the SU operator upon script completion or abort and in case of an error whether or whether not a housekeeping package was acquired. | |

| | | |
|---|---|---|
| FPX-RQ-12 | The next start time for the script shall be STARTTIME + REPEATTIME. The script shall be repeated even though it may have failed the last execution time. This does not apply in case the satellite has an over current protection of the SU power lines, which was triggered during last operation of the script. | |
| FPX-RQ-13 | The OBC shall implement the specified commanding interface of the SU as defined in section 3.3 with the process charts Figure 3-2 and Figure 3-3 of the FIPEX ICD. Timeout for a response of the SU to a command shall be 100 ms. | |
| FPX-RQ-14 | The Science Unit Thermistor (SU_TH_GO, SU_TH_RET) can be used monitor and log the SU temperature but should not be used to decide on board about SU command script execution. | |
| FPX-RQ-15 | The SU electrical connector shall provide the signals as shown in Table 2-1 of the FIPEX ICD. Signals are still TBC. | |
| FPX-RQ-16 | The electrical interface shall be switchable by the CubeSat. +5 V, 3V3, TX, RX and its screens shall be switched ON/OFF together | |
| FPX-RQ-17 | The supplied voltage (3V3 and 5V) shall have a tolerance of ±1,0%. | |
| FPX-RQ-18 | The power lines shall tolerate a switch on peak current of 50% above the specified maximum current for 50ms. | |
| FPX-RQ-19 | Protection circuits shall allow a maximum current of 10% above the maximal current for each power line listed in Table 2-2 of the FIPEX ICD. | |
| FPX-RQ-20 | The Science Unit electronics shall be electrically grounded to the CubeSat GND via the SU Connector GND. The Science Unit Chassis shall be electrically connected to the CubeSat structure at its points of attachment. The electrical resistance shall be <1. In case of a non-metallic structure the SU Chassis shall be grounded using the contact points specified in the Interface Control Drawing | |
| FPX-RQ-21 | The serial interface shall have the following settings: BAUD Rate: 115200, Data Width: 8-bit, Parity bit: No Parity, Start bit: ONE, Stop bit: ONE | |
| FPX-RQ-22 | The CubeSat shall provide the required accommodation according to the Interface Control Drawing. | |
| FPX-RQ-23 | The electrical interface connector shall be attached via cable to take manufacturing uncertainties into account. | |
| FPX-RQ-24 | The following thermal environment shall be maintained by the CubeSat side for the FIPEX SU: Operational temperature range:-20 deg to+40 deg. | |

| | | |
|---|---|---|
| FPX-RQ-25 | The following thermal environment shall be maintained by the CubeSat side for the FIPEX SU: Non-operational temperature range: -30 deg to +65 deg. | |
| FPX-RQ-26 | The following thermal environment shall be maintained by the CubeSat side for the FIPEX SU: Switch on temperature -10 deg. | |
| FPX-RQ-27 | A minimum of one fit check shall be conducted, in which SU shall be integrated into the CubeSat. | |
| FPX-RQ-28 | A minimum of one interface test shall be conducted, in which the OBC operates the SU according to predefined test command files | |
| FPX-RQ-29 | The FIPEX SU shall be stored in a dry environment (<20% humidity), at room temperature (19 – 21 deg). | |
| FPX-RQ-30 | The FIPEX SU storage time should not exceed 2 years. | |
| FPX-RQ-31 | The Science Unit shall be handled in a cleanroom environment (class 100.000, ISO 8). ESD protection protocols shall be followed. | |
| FPX-RQ-32 | Handling of the Science Unit Flight models must be documented in the life book of the SU. The maximum number of integrations/ deintegration is 10 (mechanical and electrical). Connector Saver can be used. | |
| FPX-RQ-33 | Sensor Units with Flight sensors shall only be operated under low pressure conditions (max. allowed pressure $10^{-4}$ bar). Flight sensors must not be touched at all. | |

### 1.3.2 Overview

Flux-φ-Probe Experiment (FIPEX) is an oxygen flux probe developed by the Technical University Dresden, Germany (Figure 7). FIPEX is able to distinguish and measure the time resolved behavior of atomic and molecular oxygen as a key parameter of the lower thermosphere. Atomic oxygen is the dominant species in these regions and therefore its measurement is crucial in the correlation and validation of atmosphere models. FIPEX in the QB50 mission is an extension of FIPEX on ISS – an experiment launched on the STS-122 (1E) Shuttle flight on 07 February 2008, which deployed on the COLUMBUS External Payload Facility on the platform EuTEF (European Technology Exposure Facility). It provided the first measurements of the time resolved behavior of atomic oxygen and oxygen molecules. The next natural step would be a time and spatial resolved measurement, which is possible with QB50.

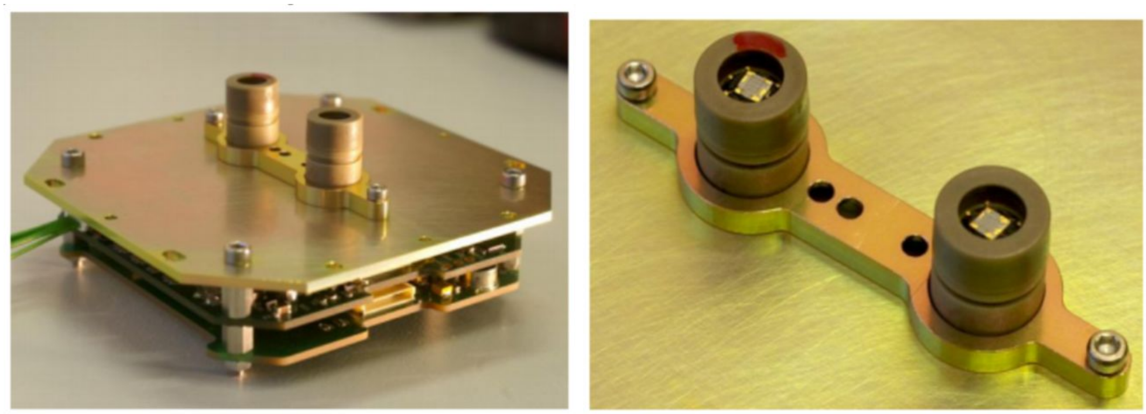


*Figure 4: FIPEX Science Unit*

### 1.3.3 Design Drivers

The FIPEX instrument was developed by TUD as a science payload for the QB50 missions, and dictates the above set of requirements onto the various subsystems. The major challenges with the FIPEX instrument will be maintaining attitude control at low altitudes, providing sufficient power to minimize duty cycling, and interfacing the instrument with standard US bus designs. However, The University of Michigan QB50 Team is confident that the chosen design successfully deals with all of these criteria.

### 1.3.4 Plans for Documentation

### 1.3.5 Determination

Requirement FPX-2 states that the CubeSat carrying FIPEX will need to determine its position to within a 10 km accuracy. In order to fulfill this requirement, two-line element (TLE) sets will be used to determine the spacecraft's position and velocity along with coarse sun sensors in the ADCS. The TLE's will provide several different orbital elements in order to calculate the CubeSats' position and velocity; the position calculation using this method has an accuracy on the order of 1 km.

### 1.3.6 Downlink

The CubeSat will downlink data to a network of five ground stations. The University of Michigan ground station, located in Ann Arbor, MI, will be the only station in the network to uplink commands to the spacecraft. Data received by each ground station will be forwarded to the Ann Arbor facility before being stored in the VKI repository. This data will be stored at the University of Michigan for at least 6 months after mission success. Figure 8 shows a diagram of the communication network.

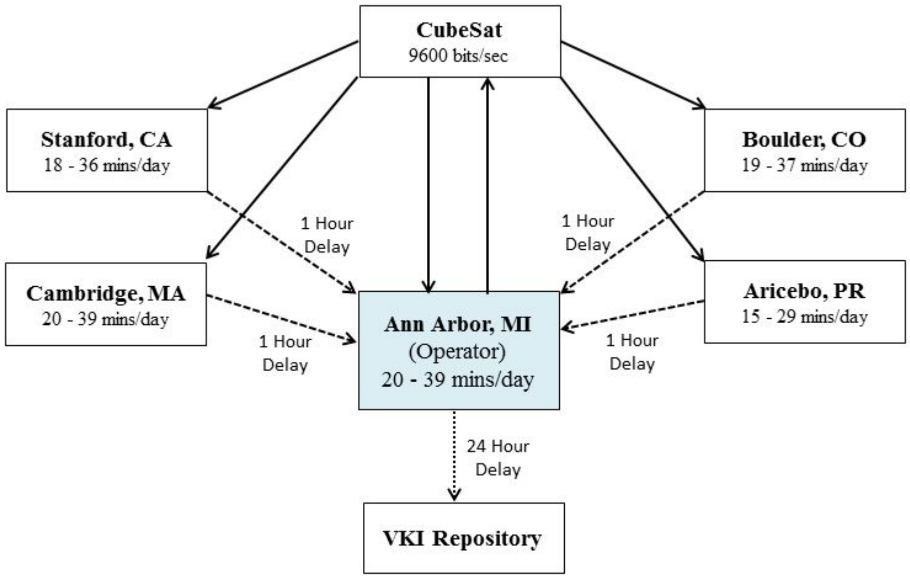



*Figure 8: Diagram of Data Flow, Access Times, and Delays in the Ground Station Network*

### 1.3.7 Connections and Interfaces

#### Electrical Interface

The EPS will utilize a backplane configuration that provides an electrical infrastructure for the CubeSat as each subsystem will be designated a slot and access to a regulated power bus. The subsystems will have power regulated using buck regulators (+8.4 unregulated & +5, +3.3, +1.5 V regulated).

#### Serial Interface

The payload interface module (PIM) will connect to FIPEX via a standard 25 pin MDM (female)connector, with the connections specified in the FIPEX ICD, and below in Table 15. There will also be UART available on the PIM at 115.2 kbaud, with an 8-bit data width, no parity, and start and stop bits of one, in accordance with default serial settings. In addition, the PIM will allow the electrical interface to the SU to be switchable by the CubeSat. When switching occurs, all lines shall be switched on or off together.

*Table 15: FIPEX Connector Pinout*

| Pin | Signal Name | Comment |
|---|---|---|
| 1 | +5 | SWITCHED Power for +5V |
| 2 | +3V3 | SWITCHED Power for +3V3 (logic power supply) |
| 3 | STM_TH_GO_0 | Surface Thermal Monitor -signal for CH0, positive lead on AD590 |
| 4 | STM_TH_GO_1 | Surface Thermal Monitor -signal for CH1, positive lead on AD590 |
| 5 | STM_TH_GO_2 | Surface Thermal Monitor -signal for CH2, positive lead on AD590 |
| 6 | STM_TH_GO_3 | Surface Thermal Monitor -signal for CH3, positive lead on AD590 |

| | | |
|---|---|---|
| 7 | STM_TH_GO_4 | Surface Thermal Monitor -signal for CH4, positive lead on AD590 |
| 8 | FIPEX_TH_GO_SU | temperature sensor - signal GO, positive lead on AD590 |
| 9 | GND | System GROUND |
| 10 | TX | SWITCHED Serial line to SEND data from SU to CubeSat |
| 11 | RX | SWITCHED Serial line to RECEIVE data from CubeSat to SU |
| 12 | GND | System GROUND |
| 13 | GND | System GROUND |
| 14 | +5 | SWITCHED Power for +5V |
| 15 | +3V3 | SWITCHED Power for +3V3 logic power supply |
| 16 | STM_TH_RET_0 | Surface Thermal Monitor - RETURN for CH0, negative lead on |
| 17 | STM_TH_RET_1 | Surface Thermal Monitor - RETURN for CH1, negative lead on |
| 18 | STM_TH_ RET_2 | Surface Thermal Monitor - RETURN for CH2, negative lead on |
| 19 | STM_TH_ RET_3 | Surface Thermal Monitor - RETURN for CH3, negative lead on |
| 20 | STM_TH_ RET_4 | Surface Thermal Monitor - RETURN for CH4, negative lead on |
| 21 | FIPEX_TH_RET_SU | temperature sensor - signal RETURN, negative lead on AD590 |
| 22 | GND | System GROUND |
| 23 | RX | Screen for TX |
| 24 | TX | Screen for RX |
| 25 | GND | System GROUND |

## 1.4 Structure

### 1.4.1 Requirements

*Table 16: Structural Requirements*

| ID | Requirement | Section Addressed |
|---|---|---|
| QB50-SYS-1.1.1 | CubeSats dimensions shall be as shown in Table 17 | |
| QB50-SYS-1.1.2 | The CubeSats shall use the reference frame as shown in Figure 16 such that it will be in line with the reference frame of the deployment system. | |
| QB50-SYS-1.1.3 | In launch configuration the CubeSat shall fit entirely within the extended volume dimensions shown in Figure 13 for a 2U CubeSat including any protrusions | |
| QB50-SYS-1.1.4 | After integration into the QuadPack, the CubeSat shall only require access, for any purpose, through the access hatches in the door of the QuadPack. The position and dimensions of these hatches are shown in Figure 14. | |
| QB50-SYS-1.1.5 | The CubeSat mass shall be no greater than 2.0 kg. | |
| QB50-SYS-1.1.6 | The CubeSat center of gravity shall be located within a sphere of 20 mm diameter, centered on the CubeSat geometric center. | |
| QB50-SYS-1.1.7 | Deployment switches shall be non-latching (electrically or mechanically). | |
| QB50-SYS-1.1.8 | The CubeSat rails and standoffs, which contact the QuadPack rails, pusher plate, door, and/or adjacent CubeSat standoffs, shall be constructed of a material that cannot cold-weld to any adjacent materials. | |

| ID | Requirement | Section Addressed | Notes |
|---|---|---|---|
| NanoRacks | 2U CubeSats shall have a maximum mass of 5.657 kg | | |
| Nanoracks | CubeSats nominal envelope maximum dimensions are shown in Figure 5. No external components other than CubeSat rails or rail roller/slider switch, if used, shall contact the NRCSD interior. Additional envelope provided by a cylindrical recess within the NRCSD pusher plate is available subject to approval. | | |
| NanoRacks | A CubeSat shall have four (4) rails, one per corner, along the Z axis | | |
| Nanoracks | Each rail shall have a minimum width of 6mm +0.1mm/ -0.0mm tolerance. | | |
| NanoRacks | The edges of the rails shall be rounded to a radius of at least 0.5mm +/-.1mm | | |

| | | | |
|---|---|---|---|
| Nanoracks | Each rail end face shall have a minimum surface area of 4mm x 4mm for contact with the adjacent CubeSats. | | |
| NanoRacks | The minimum extension of the CubeSat rail standoffs beyond the CubeSat +/-Z face shall be 6.5mm (see Figure 7) | | |
| Nanoracks | Rail length variance in the Z axis between rails shall not exceed ± 0.1 mm | | |
| Nanoracks | CubeSat rail surfaces that contact the NRCSD guide rails shall have a hardness equal to or greater than hard anodized aluminum (Rockwell C 65-70). | | |
| Nanoracks | CubeSat developers can verify mechanical compatibility by a fit check with a gauge built to the requirements in Figure 8 | | |

The NRCSD interior envelope is shown in Figure 5.

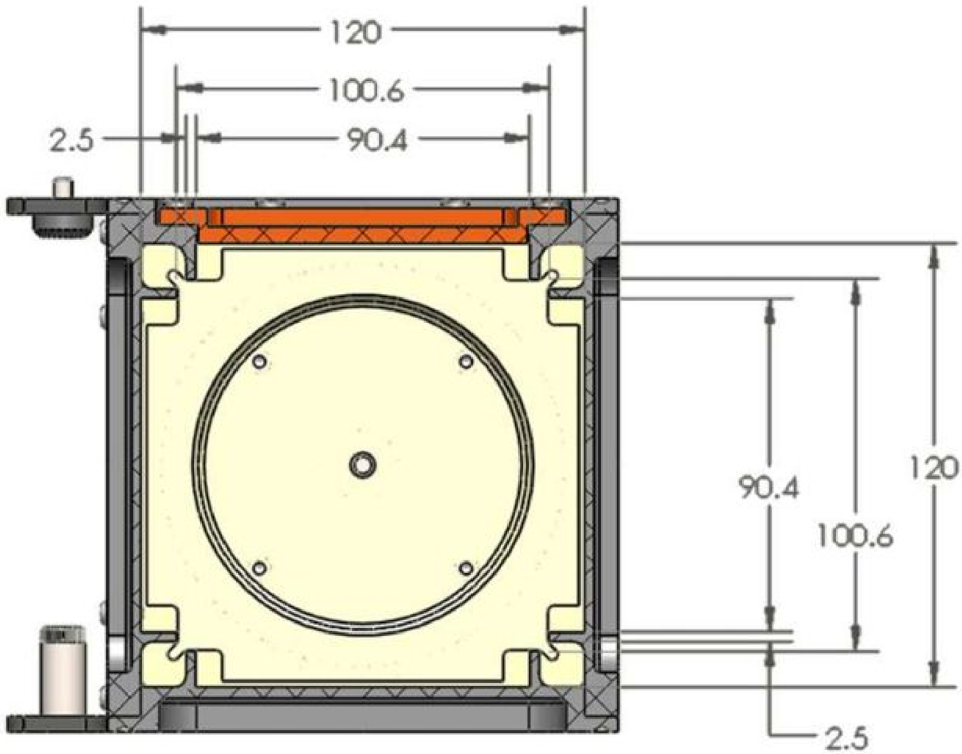



**Figure 5 NRCSD Axial Cross-Section (+Z view).**

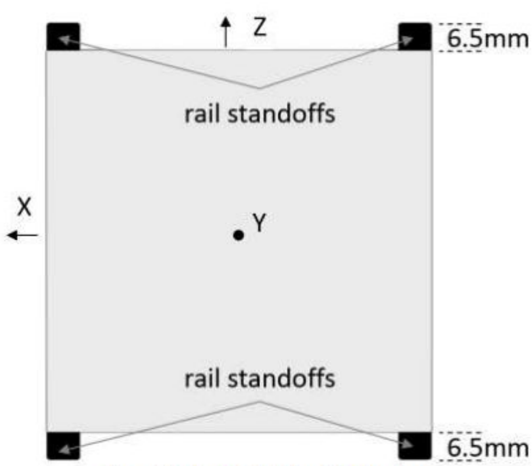



**Figure 7 CubeSat Rail Standoff Clearance**

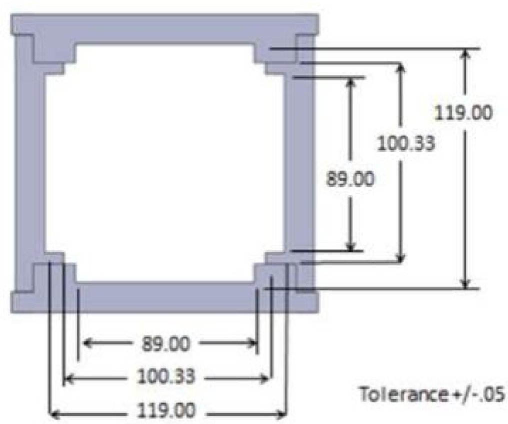



**Figure 8 Dimensional fit check gauge cross section view**

| ID | Requirement | Section Addressed | Notes |
|---|---|---|---|
| NanoRacks | CubeSats shall have a minimum of three (3) mechanical deployment switches corresponding to inhibits in the main electrical system (see section on electrical interfaces). | | |
| Nanoracks | Deployment switches can be of the pusher variety, located on the –Z face on one or more of the rail end faces, or roller/lever switches embedded in a CubeSat rail and riding along the NRCSD guide rail. | | Two pusher and one roller |
| NanoRacks | A roller or slider shall be centered on the deployer guide rail, allowing for placement accuracy, the roller or slider shall maintain a minimum of 75% (ratio of roller/slider width-to-guide rail width) contact along the entire Z-axis (see Figure 9) | | |
| Nanoracks | Deployment switches force exerted shall not exceed 3N. | | |
| NanoRacks | CubeSats, except 6U, shall have separation springs. Separation springs shall be located at the –Z end face of a diagonal pair of CubeSat rails as shown in Figure 10. | | |
| Nanoracks | Each spring shall be captive. When compressed the spring shall be contained within the maximum rail length. Separation spring and the rail end face alignment are shown in Figure 11. | | |
| Nanoracks | Individual separation spring force shall not exceed 3.34 N (0.75 lbs) with the total force for both springs not to exceed 6.67 N (1.5 lbs). | | |
| Nanoracks | During deployment, the CubeSats must be compatible with deployment velocities between 0.5 m/s to 1.5 m/s and accelerations no greater than 2g's in the +Z direction. | | |

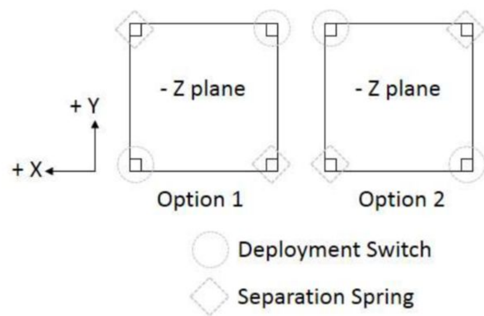



Figure 10 Separation spring placement options

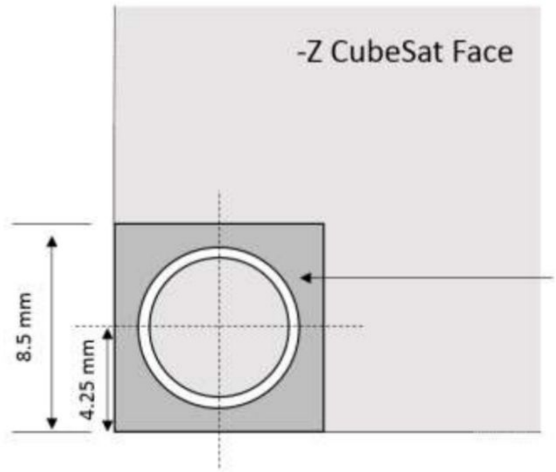



Figure 11 Separation spring rail standoff geometry

### 1.4.2 Design Drivers

The primary system drivers for our structural design involve simplicity, power, mission lifetime, mass, and spacecraft stability.

### 1.4.3 Specifications

The QB50 CubeSat will fit to the 2U specifications outlined by the NanoRacks CubeSat Deployer (NRCSD) configuration. This also contains the Interface Control Drawings that provide specific dimensional constraints for both 2U CubeSats. Our QB50 CubeSats will fit into the NRCSD, utilizing the cylindrical recess in the pusher plate to accommodate for the FIPEX instrument. The material used to fabricate the CubeSat structure will be Aluminum 6061-T6, which is an accepted material used in past missions at the University of Michigan. The anodized aluminum resists corrosion and functions as a shield against solar radiation making it ideally suited for the extent of the mission as well as a range of prior storage conditions. Furthermore, this anodized material will minimize the spacecraft's direct contact with the deployment rails preventing cold welding. In order to allow the structure to be used for electrical grounding, a section of the anodized surface inside of one of the satellite walls must be masked or removed. For proper identification of our CubeSats, their names and Universities will be etched on the +Z Block where it will be visible when the NCRSD latch is opened (Figure 15). Further coordinate definition can be seen in Figure 16.

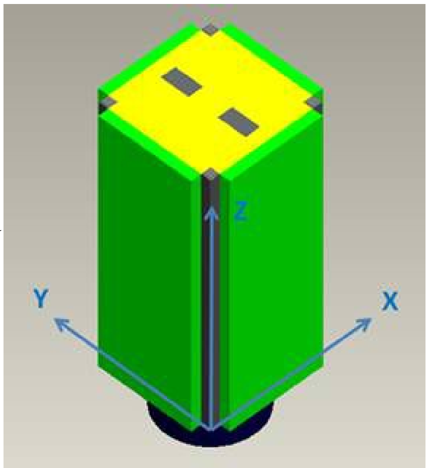
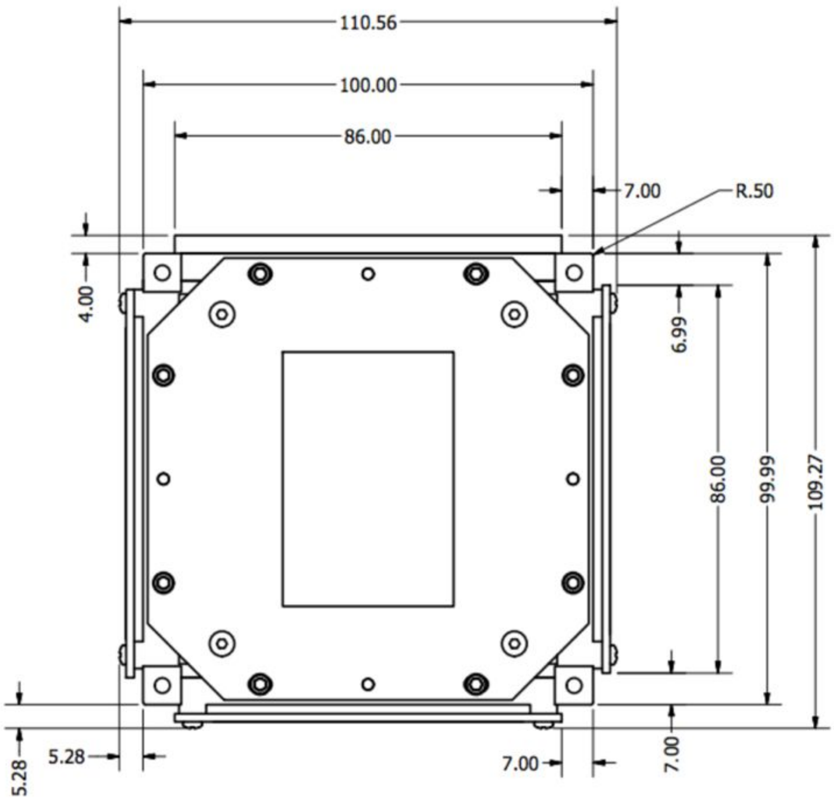
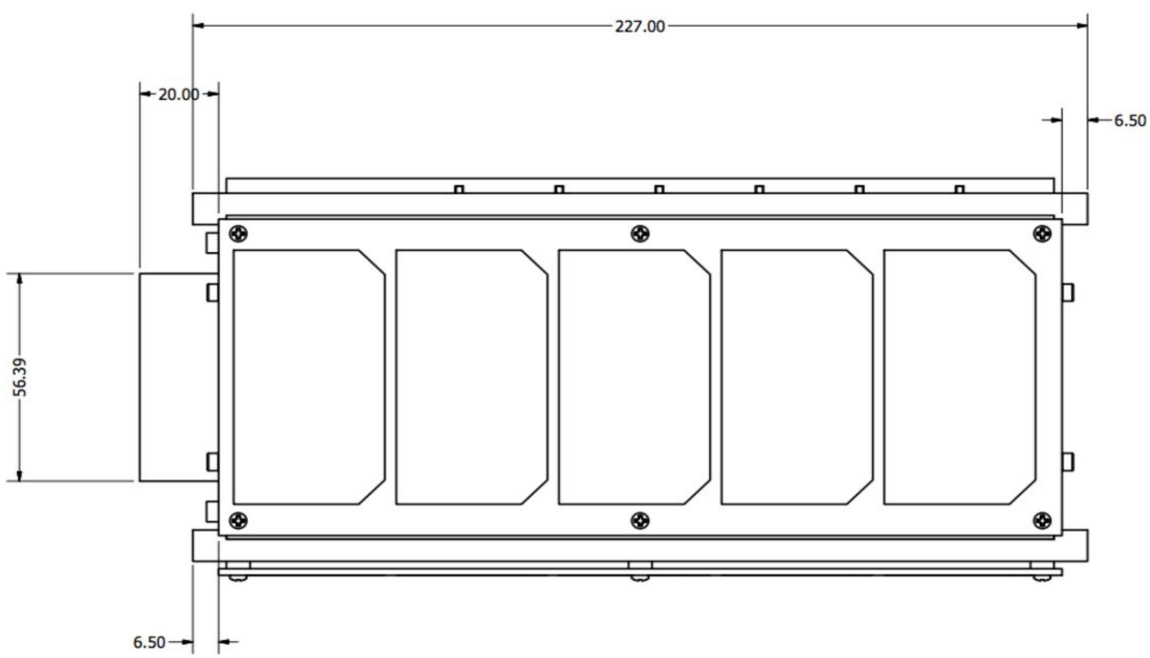



*Figure 16: CubeSat Coordinates*

*Table 17: CubeSat Dimensions*

| **Property** | **2U** |
|---|---|
| Footprint | 100 × 100 ± 0.1 mm |
| Height | 227 ± 0.1 mm |
| Feet | 8.5 × 8.5 ± 0.1 mm |
| Rails | External edges shall be rounded 0.5mm +/-.1mm |


110.56
100.00
86.00
7.00
R.50
4.00
6.99
86.00
99.99
109.27
5.28
5.28
7.00
7.00


227.00
20.00
6.50
56.39
6.50

### 1.4.4 Deployment

Two plunger switches are positioned diagonally in the structural feet of the –Z face with a single roller/slide switch embedded in the side of the CubeSat rail. Once the NRCSD door is opened and the CubeSats are ejected, the separation switches are flipped and the spacecraft bus is powered on.
The UHF antenna is first to deploy. It is folded over the top cover (+Y face) and is secured with a dedicated monofilament deployment circuit. The antenna will be deployed autonomously by the FCPU at 30 minutes MET. UHF health and status beacons also begin transmission at this time. Photodiodes shaded by the stowed antenna will confirm deployment in sunlight. The FCPU autonomously determines whether or not the antenna has successfully deployed and if it appears that it has not, it will attempt to deploy the antenna five more times until it exits into a ninety (90) minute hold to allow the spacecraft to charge before attempting again. Once the FPCU determines successful deployment of the antenna, Atlantis and Columbia enter a low power state. When the health and status beacon is received at the Ann Arbor ground station, the ground station will verify antenna deployment. This command may or may not be sent during the same pass that the deployment confirmation beacon is received.

## 1.5 Attitude Determination and Control System

The Attitude Determination and Control System design philosophy attempts to satisfy the requirements while adhering to the following fundamental Design Principles:

1) Minimize footprint: low mass, low volume, low power, low processing, low cost
2) Achieve robustness with simplicity
3) Minimize risk through heritage

### 1.5.1 Requirements

*Table 18: Attitude Determination and Control Requirements*

| ID | Requirement | Section Addressed |
|---|---|---|
| QB50-SYS-1.2.1 | The CubeSat shall be able to recover from tip-off rates of up to 10°/sec within 2 days. | 1.5.3 |
| QB50-SYS-1.2.2 | The Science Unit will be accommodated at one end of the CubeSat, on a 10 mm×10 mm face—the -Z face using the CubeSat reference frame as shown in Figure 16. The vector normal to this face shall be in the spacecraft ram velocity direction. The face shall not be available for solar cells, or for any other subsystem and nothing must forward this face. | 1.5.3 |

### 1.5.2 Velocity Vector Tracking Considerations

In order to satisfy the FIPEX 10° absolute error velocity vector tracking requirement, simultaneously providing for the satisfaction of requirement 1.2.2 above, the spacecraft will need to a) counteract disturbance torques and b) rotate in accordance with the periodic rotation due to orbital motion. The most significant disturbance torques are expanded on below.

### 1.5.3 Attitude Determination and Control Methodology

In order to counteract the disturbance torques and match the required rotation due to orbital motion such that the attitude requirements are satisfied, the spacecraft will employ a suite of sensors and actuators. On the attitude knowledge side, the spacecraft will use coarse sun sensors located on all of the faces (excluding the FIPEX SU face) and gyroscope readings from an inertial measurement unit (IMU). For actuation, the spacecraft will use a set of magnetorquers that will interact with the local magnetic field using proven control algorithms supplied with field component determination via magnetometers.
The x, y and z-axes will be controlled with iron core magnetorquers. Figure 18 presents a diagrammatic summary of the major ADCS components and their relationships.
The sun sensors will not always be able to aid in attitude determination due to field of view and eclipse duration constraints. For this reason, they are coupled with an IMU.

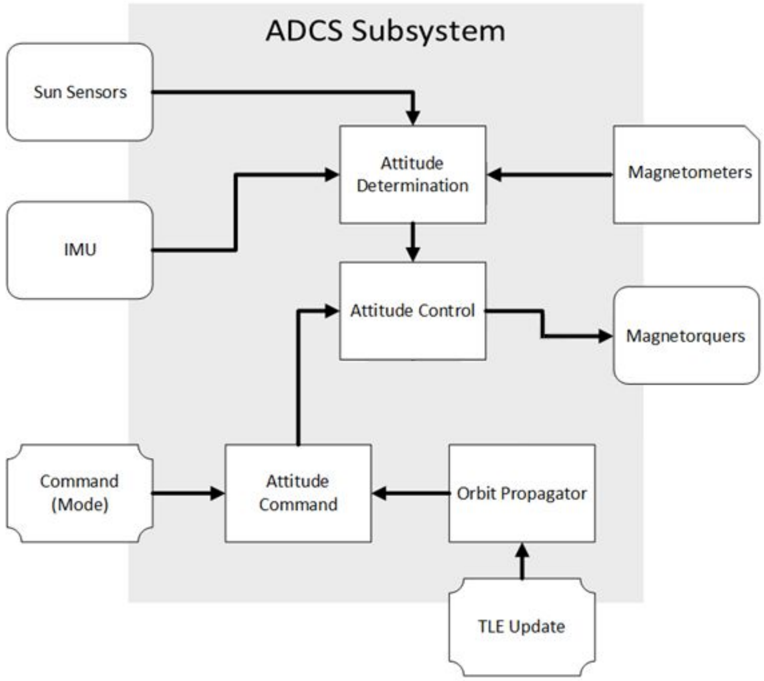



*Figure 18: Major ADCS subsystem components and relationships*

## 1.6 Electrical Power

### 1.6.1 Requirements

*Table 19: Electrical Power Requirements*

| ID | Requirement | Section Addressed |
|---|---|---|
| QB50-SYS-1.3.1 | The CubeSat shall provide sufficient power at the appropriate voltage, either by solar array generation or battery, to meet the power requirements of all satellite subsystems in all modes of operation. | |
| QB50-SYS-1.3.2 | The CubeSat shall be able to be commissioned in orbit following the last powered-down state without battery charging, inspection or functional testing for a period of up to 8 months. | |
| QB50-SYS-1.3.3 | The CubeSat shall be powered OFF during the entire launch and until it is deployed from the deployment system. | |
| NanoRacks | The CubeSat operations shall not begin until a minimum of 30 minutes after deployment from the ISS. Only an onboard timer system may be operable during this 30-minute post deploy time frame. | |
| NanoRacks | The CubeSat electrical system design shall incorporate a minimum of three (3) inhibit switches actuated by physical deployment switches (see Deployment Switches section 4.7) as shown in Figure 12. | |
| NanoRacks | The CubeSat electrical system design shall not permit the ground charge circuit to energize the satellite systems (load), including flight computer (see Figure 12). This restriction applies to all charging methods. | |
| NanoRacks | RBF pins are required. Arming switch or captive jumpers may be an acceptable alternative; contact NanoRacks. | |
| NanoRacks | The RBF pin shall preclude any power from any source operating any satellite functions with the exception of pre-integration battery charging. | |
| NanoRacks | RBF pins must be capable of remaining in place during integration with NRCSD. It shall not be necessary to remove the RBF to facilitate loading into the NRCSD. | |
| NanoRacks | All RBF pins, switches, or jumpers utilized as primary electrical system and RBF inhibits must be accessible from the access panels (see Figure 1) for removal at the completion of loading into the NRCSD. | |
| NanoRacks | CubeSats that utilize on-board batteries shall comply with NASA requirements for battery safety. This | |

| | | |
|---|---|---|
| | requirement applies for main power batteries and for batteries associated with real-time clocks or watch-dog circuits i.e. "coin cell" batteries. Contact NanoRacks for the appropriate battery test procedure. Batteries should maintain charge for a minimum of 6 months from time of integration into the NRCSD by NanoRacks. | |

### 1.6.2 Design Drivers

In order to determine which candidate designs are feasible in terms of power generation, the estimates of spacecraft power consumption were performed, and are listed in the following table.

*Table 20: Spacecraft Power Requirements*

| CubeSat Component/Subsystem | Power Consumption (W) |
|---|---|
| Peripheral Sensors | 0.038 |
| TCB/Torque Coil | 0.327 |
| 440MHz System Power | 0.136 |
| 440MHz 1W Transmit | 0.264 |
| Flight Computer | 0.402 |
| FF | 0.020 |
| EPS Battery Board | 0.399 |
| FIPEX | 0.607 |
| Total | 2.193 |

### 1.6.3 Design Overview

The Electrical Power System (EPS) system consists of solar cells, a solar panel interconnect board, voltage regulation system and lithium batteries. The solar cells are mounted on three body panels to minimize complexity and are in strings of five. These solar cells are Emcore BTJM CIC'ed cells. Each solar panel will also have three photodiodes attached to it as sun sensors. These three photodiodes are each connected to an amp circuit and connected to a 4 channel analog to digital converter. The additional channel will be filled with a temperature sensor. The analog to digital converter, the solar cells, and a magnetometer are connected into the solar panel interconnect board, which is then connected to the backplane. Two Panasonic lithium-ion batteries are used in parallel to provide power.
*Each solar panel contains 5 cells, 3 photodiodes and a temp sensor. The diodes and sensor connect to a 4 channel analog to digital converter. The converter, 5 solar cells and the magnetometer are connected to the solar panel interconnect board.*

### 1.6.4 Solar Array Power Output

Modeling parameters consistent across all three designs are shown in Table 21 below. Solar cell area and beginning of life (BOL) efficiency values were obtained from Emcore Triple-Junction with Monolithic Diode (BTJM) solar cell data. End of life (EOL) efficiency values were calculated based on data sheets and data from NASA's Solar Maximum Mission (SMM). Spectrolab data indicates an EOL 24.3% solar cell efficiency at a 1 MeV electron fluence of 1E15 e/cm2.

*Table 21: Power Analysis Parameters*

| **Parameter** | **Value** |
|---|---|
| Solar Cell Area | 26.26 cm2 |
| # Solar Cells per 2U face | 5 |
| BOL Solar Cell Efficiency | 28.3% |
| EOL Solar Cell Efficiency | 27.5% |
| Simulation Start Time | 1 Jan 2016 17:00:00 |
| Simulation End Time | 1 Feb 2016 17:00:00 |
| Simulation Timestep | 60 sec |
| Orbit Altitude | 330 - 410 km |
| Orbit Inclination | 51.6 deg |

### 1.6.5 Battery Selection

The CubeSat will be power net negative for approximately 53 minutes of every orbit, and will require appropriately sized batteries in order to function. Batteries were sized using a 20% depth of discharge and 2.9W orbit average, and therefore must be capable of supplying 46200 Joules of energy per orbit. The selected design will use two Panasonic 18650A Li-ion batteries in parallel. Each battery cell possesses a minimum 3.2 Amp-hour capacity and operate at a nominal voltage of 3.6 Volts, providing approximately 41500 Joules of energy storage each. The total nominal energy capacity of the batteries is 83000 Joules, which satisfies the requirement with an 80% margin. Battery performance numbers are based off of heritage components built at The University of Michigan, such as the battery board for a 1U CubeSat shown in Figure 23.

Panasonic 18650 batteries have spaceflight heritage, and are expected to last over 5000 cycles at a 30% depth of discharge. Storage characteristics for 18650 battery cells reveal a discharge of approximately 15% for a fully charged cell over the period of a year.

Prior to deployment, the batteries will be isolated from all other components with spring-loaded foot switches held in an electrically open position. After deployment, the switches are released and spring back to an electrically closed position.

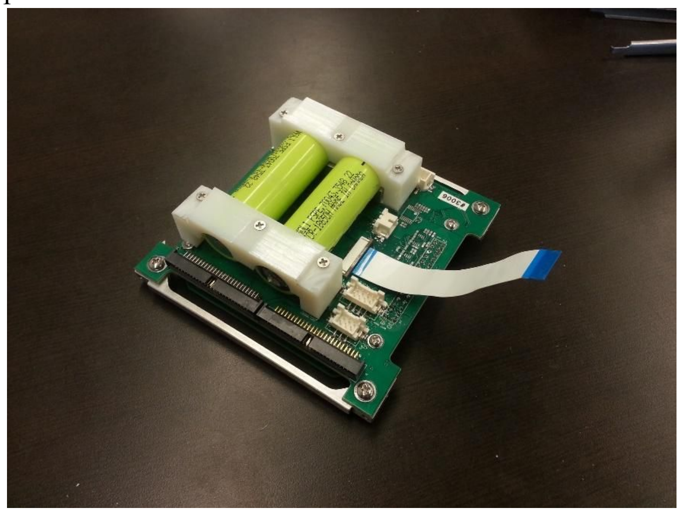


*Figure 23: Two cell battery board from previous CubeSat mission*

### 1.6.6 Solar Cells

After the initial power analyses, changes were made and the solar cells were switched from Spectrolab UTJ cells to Emcore BTJM cells. UnCICed BTJM cells are available to populate the EM and backup FM, and CICed BTJM cells will be purchased for use on the FMs. The solar panels will be fabricated with five cells on each panel. The current bus architecture can support up to ten lines, as it has been designed for a double-sided shuttlecock configuration, even though this design currently uses only three lines (Figure 24).

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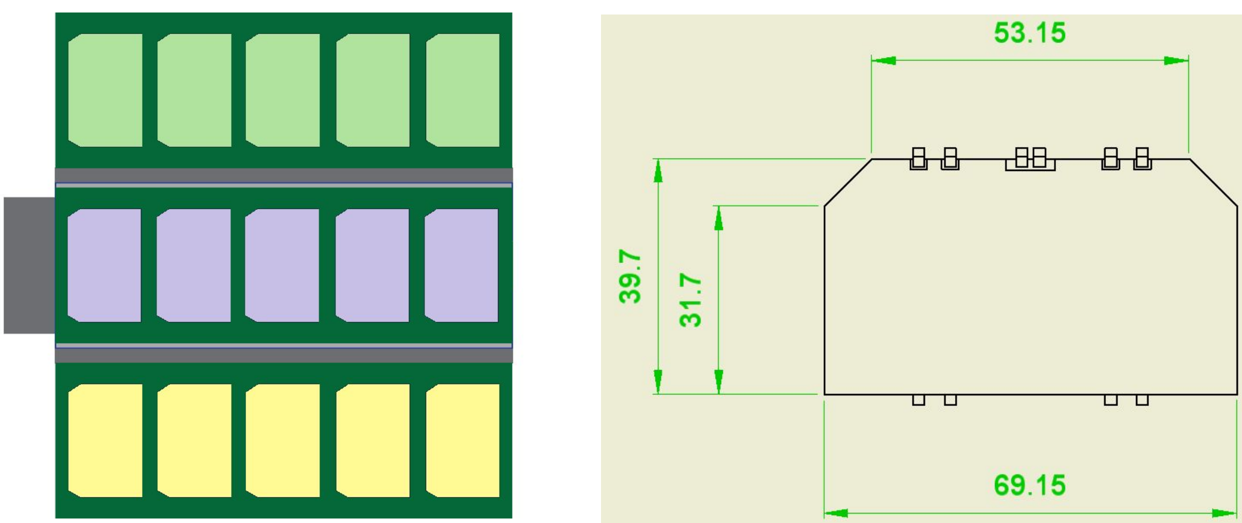


*Figure 24: Solar Panel Lines and Dimensions*

## 1.7 On-Board Data Handling

### 1.7.1 Requirements

*Table 26: Data Handling Requirements*

| ID | Requirement | Section Addressed |
|---|---|---|
| QB50-SYS-1.4.1 | The CubeSat shall collect whole orbit data and log telemetry every minute for the entire duration of the mission, where whole orbit data is defined as the following set of parameters: time, spacecraft mode, battery bus voltage, battery bus current, current on regulated bus 3.3V, current on regulated bus 5.0V, communication subsystem temperature, EPS temperature and battery temperature. The WOD packet format is provided in the reference document [R05]. | |
| QB50-SYS-1.4.2 | The whole orbit data shall be stored in the OBC until they are successfully downlinked. | |
| 1QB50-SYS-1.4.3 | Any computer clock used on the CubeSat and on the ground segment shall exclusively use Coordinated Universal Time (UTC) as time reference. | |
| QB50-SYS-1.4.4 | The OBC shall have a real time clock information with an accuracy of 500ms during science operation. Relative times should be counted or stored according to the epoch 01.01.2000 00:00:00 UTC. | |
| QB50-SYS-1.4.5 | The onboard software (OBSW) shall not be allowed to override hardware inhibits such as the deployment switch. (This is not applicable during check-out via umbilical cord). | |
| QB50-SYS-1.4.6 | The OBSW shall protect itself against unintentional infinite loops, computational errors and possible lock ups. | |
| QB50-SYS-1.4.7 | The check of incoming commands, data and messages, consistency checks and rejection of illegal input shall be implemented for the OBSW. | |
| QB50-SYS-1.4.8 | The OBSW programmed and developed by the CubeSat teams shall only contain code that is intended for use on that CubeSat on ground and in orbit. | |

### 1.7.2 Architecture

The on-board data handling (ODH) subsystem will include the ProASIC A3P1000 FPGA processor, a 12-bit, 8 channel analog-to-digital converter, and a watchdog timer, in order to counter single event upsets. Further work needs to be done in defining the interfaces between CDH and the other subsystems and in defining the structure of the flight code.

### 1.7.3 Flight Computer

**Processor**

The ProASIC A3P1000 FPGA will act as the C&DH processor. It will have a Coldfire IP Core processor.

**Single Event Upsets**

A Single Event Upset (SEU) occurs when an energetic particle causes a flip in the state of memory cells. The most common causes are from alpha particle radiation and atmospheric neutrons. A bit flip may corrupt data but it is cleared within the next write up to memory location, and can easily be fixed by the watchdog timer triggering a power cycling of the spacecraft.

**Watchdog Timer**

The on-board data handling system is monitored by a watchdog timer (WDT) daughter card. The WDT will reset the entire spacecraft if it detects one of three conditions.

**(1) Missing Heartbeat**

The missing heartbeat error condition is meant to protect against errors with the flight computer. The WDT will reset the spacecraft if the flight computer does not toggle the heartbeat pin of the timer within a configurable interval.

**(2) No Communications**

The WDT resets itself internally on an hourly basis. Every time the reset occurs it increments a counter. Once the count reaches a specified number, usually 48, it resets the spacecraft and begins counting again. The WDT monitors the receive line of the UHF radio, and a certain string of bytes will reset the WDT's count, preventing a reset. This condition is designed to protect against communication failures.

**(3) Commanded Reset**

In some cases, the satellite operation team will want to reset the spacecraft. The WDT is always monitoring the receive line of the UHF radio, and will reset the spacecraft if it receives a certain string of bytes. This condition exists simply to allow operators to attempt to fix errors that don't bring down the flight computer or UHF radio.

**Real Time Clocks**

| | |
|---|---|
| QB50-SYS-1.4.3 | Any computer clock used on the CubeSat and on the ground segment shall exclusively use Coordinated Universal Time (UTC) as time reference. |
| QB50-SYS-1.4.4 | The OBC shall have a real time clock information with an accuracy of 500ms during science operation. Relative times should be counted or stored according to the epoch 01.01.2000 00:00:00 UTC. |

To be handled by the ProASIC A3P1000 FPGA.

### 1.7.4 Backup Memory

Everspin magnetic RAM
As fast as static RAM but non-volitile with infinite

### 1.7.5 Telemetry

An analog to digital converter is used for telemetry. It calls two(**three?)** temperature sensors (comm system, EPS and battery), and monitors three currents(regulated bus 3.3V & 5.0V and battery bus), and three voltages(regulated bus 3.3V & 5.0V and battery bus). This is added to the time, spacecraft mode, and other subsystem temperatures, as a collection of whole orbit data (WOD). At a minimum, WOD is recorded every minute, and stored in on-board memory until the spacecraft is commanded for downlink. The same process of time stamping and storing data until downlink is commanded, also applies to the science data produced by FIPEX. However, these are distinct packets, with separate downlink commands.

### 1.7.6 Flight Code

Preliminary flow charts were made for each flight operation to diagram how the flight code will be written. Only the command definitions necessary for orbital operations shall be uploaded to the spacecraft. In order to prevent inconsistency and invalid commanding, valid command definitions are synchronized across both the spacecraft and the ground stations support equipment. An authentication code system is used to verify origin (QB50-SYS-1.4.7). This is in addition to the checksum that is appended to each packet and verified to detect bit errors.
Figure 25 is the flow chart for detumbling mode. EPS will be turned on first, followed by CDH and ADC. ADC will initiate the detumbling procedure, which will be run in a loop until a certain field alignment is achieved. When achieved, the magnetometer with feedback control will be activated, and the final alignment will be returned to CDH.

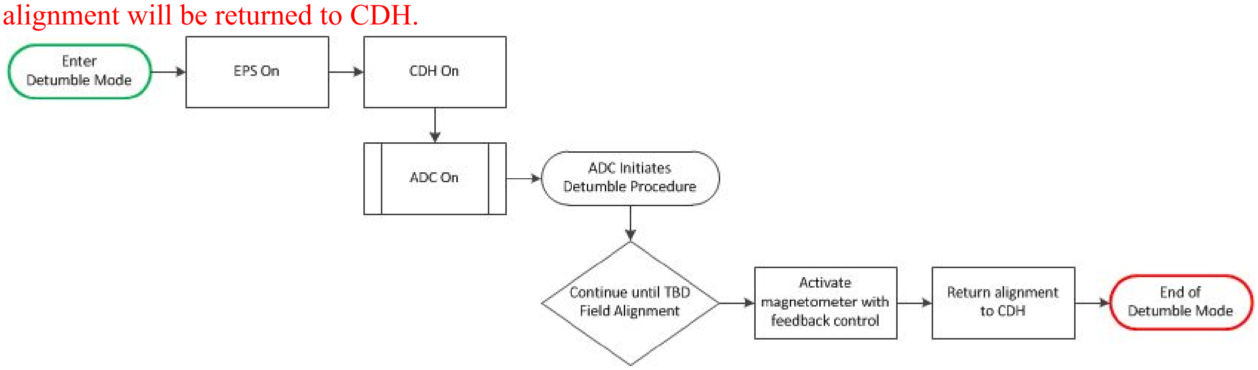


*Figure 25: Detumble Mode*

Figure 26 is the flow chart for the deployment mode. A signal is sent to the thermal knife for a set period of time. A signal confirming deployment will be returned to CDH and ADC is put on standby.

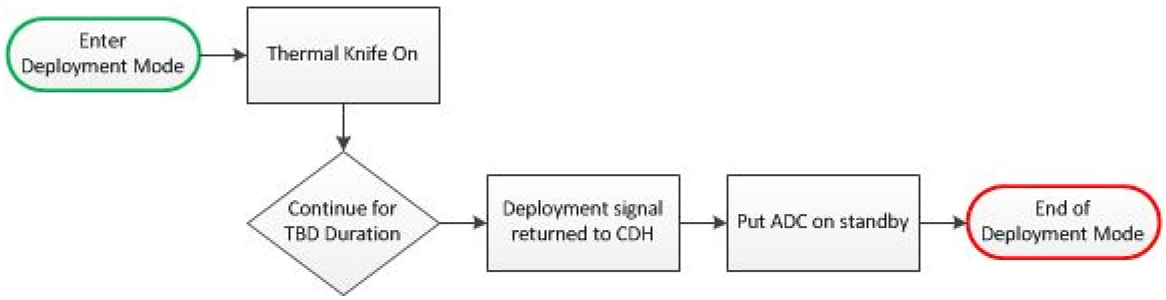


*Figure 26: Deployment Mode*

Figure 27 is the flow chart for the commissioning mode. The beacon is activated and health checks are done for the entire system, the ADCS, and FIPEX. Further research and study will be needed on what values are considered an acceptable state. The system health check includes functionality; accuracy; battery performance, temperature, and voltage; radio response; and error checking. The ADC health check consists of the FPGA checking each component for a response; these responses are then sent to the processor. The FIPEX test check determines if basic functionality is met. If any component fails the health check, they will be reset. Else, normal mode will commence.

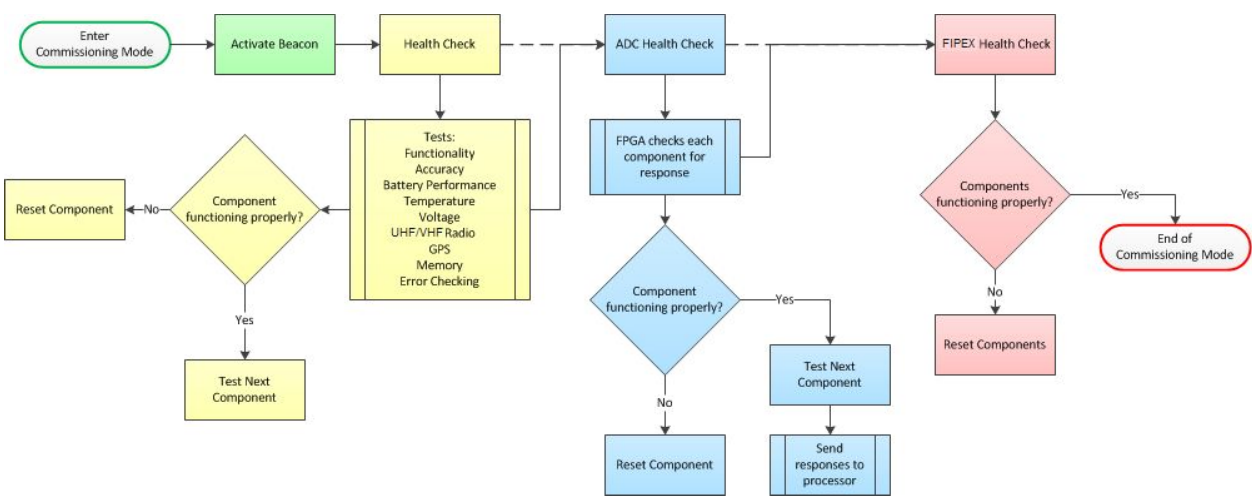


*Figure 27: Commissioning Mode*

## Science Unit Flight Code

In accordance with the FIPEX ICD, science unit commanding by the flight computer will occur such that only one command is executed at a time, with delays between commands. This commanding will be executed according to the format specified in Table 27. When communicating with the ground stations, separate commands will be used for executing operations on the OBC and SU.
In the case that an error has occurred in FIPEX, the procedure outlined in Figure 28 will be followed, in order to ensure a proper retrieval of housekeeping data prior to notifying the operator of the issue.

*Table 27: FIPEX byte encoding*

| **Byte** | **Field** | **Description** |
|---|---|---|
| 0 | LEN | one BYTE value one BYTE value representing the number of 8-bit BYTES to follow the LEN parameter. Maximum value is 254 |

| | | |
|---|---|---|
| 1..4 | STARTTIME | four BYTE value (LS BYTE first) according to chapter 5. This specifies the first execution time |
| 5..6 | REPEATTIM E | two BYTE value (LS BYTE first) in seconds, defining the script execution interval |
| 7 | CMD_CNT Z | one BYTE value containing the number n of commands to follow |
| 8 + x*5 | CMDx | Commands to execute. Details are dened in the FIPEX ICD. x counts from 0 |
| 9 + x*5.. 10 + x*5 | DELAYx | two BYTE Delay in seconds (LS BYTE first) until next command is executed. The DELAY field (d) is ONLY read by the OBC, but NOT sent to the SU as part of the command |
| LEN-2.. LEN | SCRIPT END | Three Byte command OBC SU END signaling the end of the script: 0xFF 0x01 0xFE |

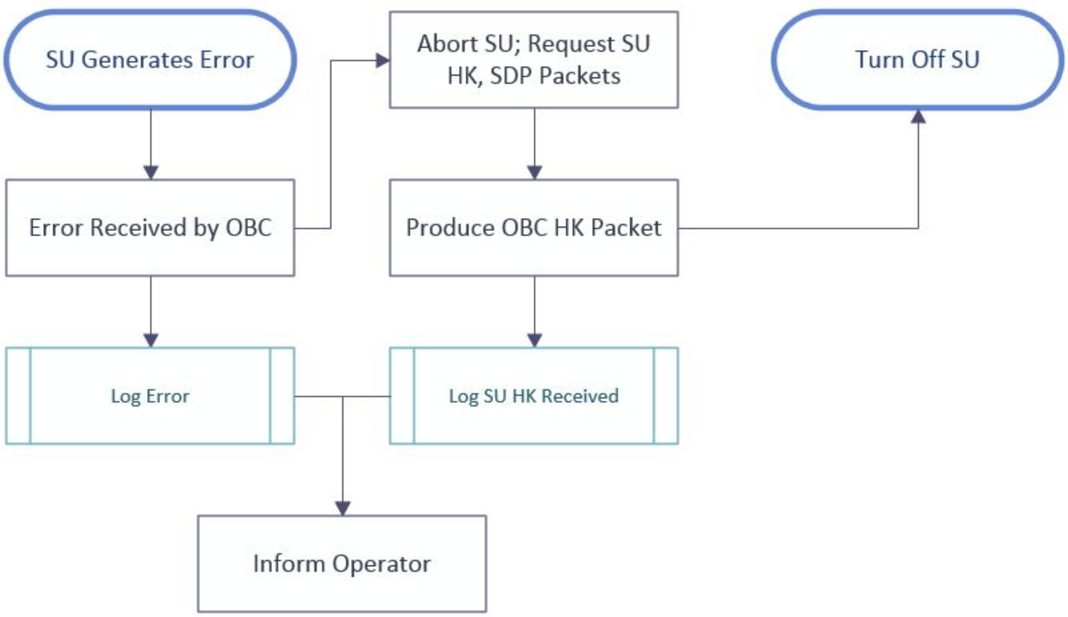


*Figure 28: FIPEX Error Handling*

## 1.8 Telemetry, Tracking and Command

### 1.8.1 Requirements

*Table 28: Telemetry, Tracking and Command Requirements*

| ID | Requirement | Section Addressed |
|---|---|---|
| QB50-SYS-1.5.1 | VHF shall not be used for downlink. | N/A |
| QB50-SYS-1.5.2 | If UHF is used for downlink, the CubeSat shall use a downlink data rate of at least 9.6 kbps. | 1.8.3 |
| QB50-SYS-1.5.3 | If UHF is used for downlink, the transmission shall fit in 20 kHz at -30 dBc, measured without Doppler, but over the entire operating temperature range | 1.8.3 |
| QB50-SYS-1.5.4 | All CubeSats shall have and make use of its unique satellite ID in the telemetry downstream. | 1.2.5.3 |
| QB50-SYS-1.5.5 | If VHF is used for uplink, it shall have a data rate no greater than 1.2 kbps. | N/A |
| QB50-SYS-1.5.6 | If UHF is used for uplink, it shall have a data rate no greater than 9.6 kbps. | 1.8.3 |
| QB50-SYS-1.5.7 | All CubeSats shall have the capability to receive a transmitter shutdown command at all times after the CubeSat's deployment switches have been activated from QuadPack ejection. | 1.8.4 |
| QB50-SYS-1.5.8 | Once a transmitter shutdown command is received and executed by the CubeSat, a positive command from the ground shall be required to re-enable the transmitter. Power reset (e.g. following eclipse) should not re-enable the transmitter. | 1.8.4 |
| QB50-SYS-1.5.9 | The CubeSat provider shall have access to a ground station which has the capability and permission to send telecommands through an uplink to control its satellite. | 1.8.4 |
| QB50-SYS-1.5.10 | *Deleted.* | N/A |
| QB50-SYS-1.5.11 | The CubeSat shall transmit the current values of the WOD parameters and its unique satellite ID through a beacon at least once every 30 seconds or more often if the power budget permits. | 1.2.5.3 |
| QB50-SYS-1.5.12 | If UHF is used for uplink, the radio receiver shall have an Adjacent Channel Rejection Ratio (ACRR) of at least 100 dB. | TBD |
| QB50-SYS-1.5.13 | The CubeSat shall use the AX.25 Protocol (UI Frames). The data type during downlink shall be specified in the Secondary Station Identifier (SSID) in the destination address field of the AX.25 frame. Science data shall be indicated using 0b1111 and Whole Orbit Data with 0b1110. | 1.8.3 |
| QB50-SYS-1.7.14 | User-friendly and documented software consisting of a) CubeSat data Frames Decoder b) CubeSat data Packet Decoder and c) CubeSat data Viewer that | 1.2.5.3 |

| | | |
|---|---|---|
| | complies with radio amateur regulations shall be made available to VKI 6 months before the nominal launch date. This documented software will be made available to the public following the AMSAT regulations. | |

### 1.8.2 Link Design

The Communication and Ground system for the QB50 mission is aimed at downlinking FIPEX data to the ground. The design goal of the communication system is to analyze and optimize the RF chain from selected hardware to maximize data throughput. The ratio of received energy per bit to noise density ($\frac{E_b}{N_0}$) is a good indicator of overall link performance. This ratio is calculated using several known parameters and constants.

$$\frac{E_b}{N_0} = \frac{P L_l G_t L_P G_r}{k_b T_s R}$$

Where P=output power, $L_l$ = line Loss, Gt=antenna gain and $L_P$ = path loss, $k_b$ = Boltzman Constant (1.38E-23 J/K), $T_s$ = system noise temperature and R = data rate. Because most values in the RF chain are discussed in decibels, the received energy per bit to noise ratio can be simplified to:

$$\frac{E_b}{N_0} = (EIRP) - L_P + G_r - 10log(k_b T_s R)$$

Where EIRP is the Equivalent Isotropic Radiated Power and is equal to $P*L_l*G_t$. When this ratio is compared to a desired ratio, it is called a Link Margin which is frequently used to characterize a communications system's performance. It is acceptable to maintain a link margin from 5-10, with 3 being the very minimum. Certain parameters were estimated and ratios were calculated for both spacecraft and ground station.

*Table 29: Link Budget*

| Parameter | Uplink | Downlink | Units |
|---|---|---|---|
| Output Power (P) | 20 | 3 | dBWatts |
| Line Loss ($L_l$) | 4 | 4 | dB |
| Trans. Antenna Gain ($G_t$) | 18.9 | 0 | dBiC |
| Path Loss ($L_p$) | 155.5 | 155.5 | dB |
| Receiver Ant. Gain ($G_r$) | -4 | 13.4 | dB |
| System Noise Temperature ($T_s$) | 11000 | 372 | K |
| Data Rate (R) | 9600 | 9600 | b/s |
| Energy to Noise Ratio ( $E_b/N_o$) | 19.7 | 19.9 | - |
| Desired ( $E_b/N_o$) | 14 | 14 | - |
| Link Margin | 5.8 | 5.9 | - |

With a desired noise ratio of 14 this link budget projects a link margin of 5.8 for uplink and 5.9 for downlink; both acceptable margins as they are well over 3. 20 dBWatts of output power (P), was estimated based on previous CubeSats designed and owned by the MXL. Values for line loss ($L_l$), transmitter antenna gain ($G_t$), receiver antenna gain ($G_r$), path loss ($L_p$), and system noise temperature ($T_s$)

were estimated using the characteristics of our current ground station used for communication with CubeSats. The data rate (R) of 9600 bps is the specification listed for the Astrodev's Li-1 transceiver.

### 1.8.3 Communications System

The communication system will be designed around Astrodev's Lithium Li-1 transceiver mounted on a communications board. The system will communicate in half-duplex mode using Gaussian Minimum Shift Keying (GMSK) modulation at 9600bps and a frequency of 437.35 MHz, within a bandwidth of 20kHz, for both uplink and downlink. AX 25 packet protocol will be implemented, according to the requirements. Serial communication with the ODH subsystem will be performed at 115.2kbps. Power will be supplied to the transceiver by the 5V bus from the EPS system. The antenna system will consist of a deployable quarter wavelength tape measure monopole connected to an RF switch from Astrodev that will be controlled by the ODH unit. This configuration has heritage at The University of Michigan, as well as on many other CubeSat missions. Figure 29 shows a tape measure antenna as part of a previous 1U fit check.

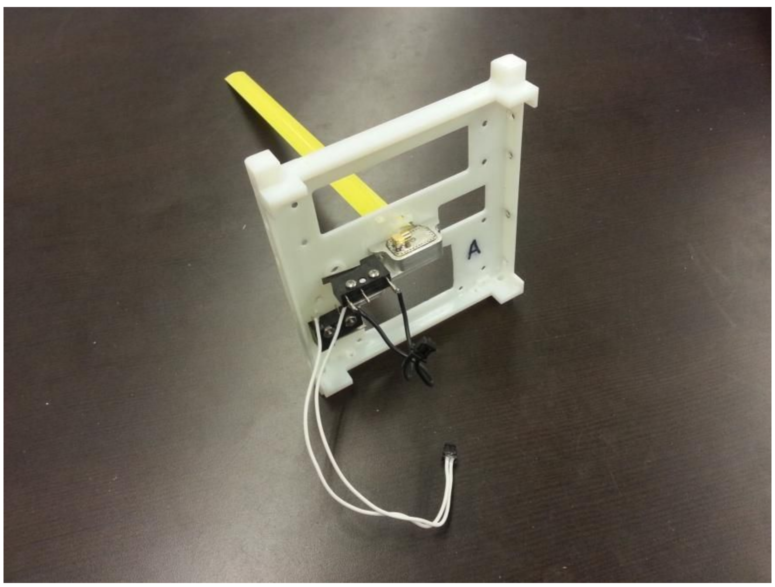

*Figure 29: Previous 1U fit check at The University of Michigan*

### 1.8.4 Ground Station

The main ground station for The University of Michigan CubeSat program is located at 42.294412, -83.711391, with the antenna on top of the Space Research Building.

**Commanding the Spacecraft**

A repository of pre-designed commands will be tested on a replica engineering model of Atlantis. These are listed in Table 30 below.

*Table 30: List of Commands*

| Command | Response |
|---|---|
| Deploy Part 1 | First portion of deployment phase |
| Deploy Part 2 | Second Part |

| | |
|---|---|
| Deploy Part 3 | Third Part |
| Reset WDT | Resets internal WDT. If clock is not reset after 48 hours S/C will reboot |
| Test | Atlantis will beacon immediately |
| Check Disk Usage | Atlantis will send back percentage of disk space being utilized |
| Safe Mode | Atlantis will enter safe mode |

In addition to these preset commands numerous other commands will be generated to accomplish a variety of tasks with the majority crafted to operate FIPEX, downlink data or delete data. These commands will also be tested on the aforementioned engineering model.

## 1.9 Thermal Control

### 1.9.1 Requirements

*Table 31: Thermal Requirements*

| ID | Requirement | Section Addressed |
|---|---|---|
| QB50-SYS-1.6.1 | The CubeSat shall maintain all its electronic components within its operating temperature range while in operation and within survival temperature range at all other times after deployment. | 1.9.3 |
| QB50-SYS-1.6.2 | The CubeSat shall survive within the temperature range of −20°C to +50°C from the time of launch until its end of life. | Table 32 |

### 1.9.2 Design Drivers

The primary temperature constraints on the thermal control system are the operating temperature range of the batteries (5 to 35°C) and the solar cells (-30 to 55°C). The batteries have the most limited thermal range of all internal components, in addition to being located near multiple heat sources, including the UHF radio. In contrast, the solar cells maintain standard operations within a much larger range, but due to the nature of their design they are highly absorptive and emit very little heat in return. This creates the potential for overheating, especially on areas of the body corresponding to areas of high internal heat generation and on the deployable panels. If this occurs, the power generating abilities of the solar cells will be increasingly limited as temperature increases until they no longer function. Beyond the temperature requirements, the secondary constraint on the system is the maintenance of a passive system for any necessary controls, as adding a fan or heater would add mass to an already highly constrained budget.

### 1.9.3 Overview

The current architecture of the thermal controls system is passive implying that the system only uses component placement, coatings, and thermally conductive pathways in order to regulate spacecraft temperature. For Atlantis and Columbia, this control system includes radiative area on the external faces of the body and deployable panels facing the Earth, as they do not receive direct sunlight, and are therefore able to emit heat more readily. In addition, the exterior of the spacecraft may be coated with a relativity low absorptivity (a=0.05) and high emissivity (e=0.9) paint such that it will not absorb excessive amounts of sunlight while still being able to dissipate the internal heat of the spacecraft. This ability to emit heat is especially important on the deployable panels, where the high absorptivity of the solar cells has the potential to retain excessive amounts of heat, and diminish the total energy intake. However, there is some radiative area on the backs of the panels, which may need to be coated with a very low absorptivity and high emissivity coating in order to ensure that the heat can be radiated into space efficiently. The specific panel coating, if any, will be determined after further analysis. The interior of the spacecraft will have very few components that will require thermally control. Due to the sun synchronous orbit expected for the QB50 Mission, we are primarily concerned with the hot case, in which components will no longer function as required to maintain satellite functionality. In order to prevent overheating of temperature critical components, thermal pathways may be required, pending a more formal analysis of internal conditions. The table below shows a list of the main components, both internal and external, and the operational temperature range for each.

*Table 32: Component Operating Temperature*

| Component | Operational Temperature Range |
|---|---|
| FIPEX | -20°C to 40°C |
| Sun Sensors | -25°C to 50°C |
| Magnetorquers | -40°C to 70°C |
| Solar Cells | -30°C to 55°C |
| Lithium-Ion Batteries | 5°C to 35°C |
| ProASIC A3P1000 FPGA | |

As mentioned above, the batteries are the limiting factor of the thermal system. However, the passive thermal system that will be set in place should be sufficient to accommodate the temperature range. Currently, MCubed-2 is in orbit and experiencing temperatures as low as 0°C and as high as 40°C. Given this information, we are confident that both Atlantis and Columbia will survive the temperature extremes in the space environment.

## 2 Risk Analysis and Mitigation Plan

### 2.1 Summary of Risks

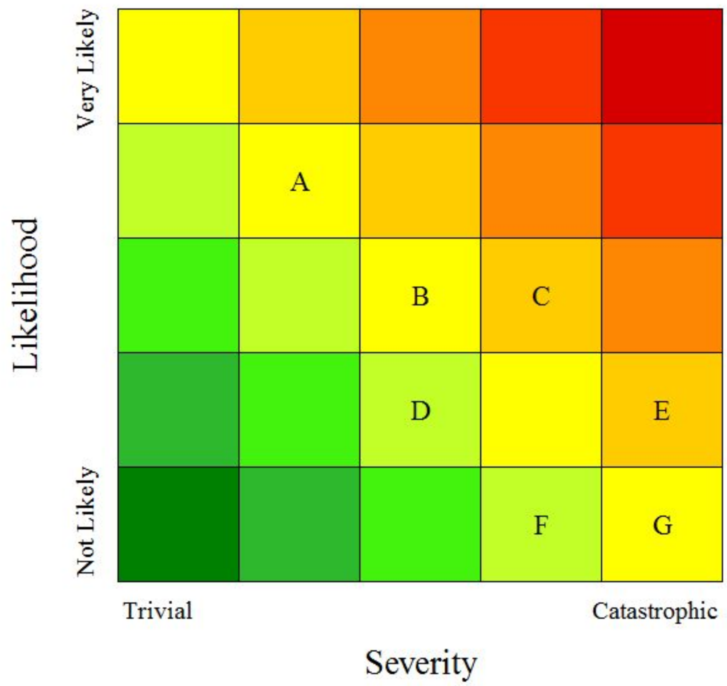


*Figure 31: Risk Matrix for US02 and US04*

*Table 33: Listing of Risks Shown in Matrix*

| Designato r | Risk |
|---|---|
| A | SD Card Failure |
| B | Gyro Failure |
| C | Pointing Failure |
| | |
| E | Solar Cell Failure |
| C | Delay from Component Lead Times |
| G | FIPEX Integration Failure |

### 2.2 Estimating Risk

Seven major risks have been identified – the majority of them being technical in nature, meaning they present some degree of risk to the final operation of the satellite. The remaining risks are programmatic, meaning they present risk to the engineering schedule. These risks were quantified with the use of 5 x 5 risk matrix - a method that separates each risk into two main factors: severity (Table 34) and likelihood (Table 35).

*Table 34: Degree of Potential Severity*

| Severity Level | Technical Degradation | Schedule / Cost Degradation |
|---|---|---|
| Trivial | Little or no impact | Little or no impact |
| Minor | Minor reduction in technical performance or supportability; usually tolerated with little or no impact on program | Able to meet key dates. Slip < 1 week |
| Moderate | Moderate reduction in technical performance or supportability with limited impact on program objectives. | Minor schedule slip. Able to meet key milestones with no schedule float. Slip < 2 weeks |
| Major | Significant degradation in technical performance or major shortfall in supportability; may jeopardize program success. | Program critical path affected. Slip < 4 weeks |
| Catastrophic | Sever degradation in technical performance. Cannot meet key mission requirements; will jeopardize program success. | Cannot meet key program milestones. Slip > 4 weeks |

*Table 35: Degree of Potential Likelihood*

| Likelihood | Probability of Occurrence |
|---|---|
| Not Likely | ≤ 10% |
| Possible | 11 – 25 % |
| Very Possible | 26 – 50 % |
| Likely | 51 – 70 % |
| Very Likely | ≥ 71 % |

## 2.3 Technical Risks & Mitigation

**Pointing Failure**

FIPEX requirements dictate the CubeSat must maintain a ram pointing error of less than 10 degrees. To achieve this, the baseline design aims to use three magnetorquers for active control, aided by passive aerodynamic stability. The lack of darts in the chosen configuration introduces heavy reliance upon magnetorquer actuation in all three rotational axes.

**Solar Cell Failure**

If one or more solar cells are damaged during launch, the ability of the CubeSats to remain power positive is severely limited. In order to aid in the prevention of this damage, the FMs will use CICed Emcore BTJM solar cells. The addition of cover glass will help prevent damage to the delicate layers of the cells. Second, the EM will use unCICed Emcore cells, which will test their ability to withstand the most extreme rocket environments.

**SD Card Failure**

SD card failure on CubeSats is a common occurrence. The use of multiple SD cards or a flash chip will decrease chances of data storage failure due to radiation.

**Gyro Failure**

The ADCS system will rely heavily on a gyro sensor to establish an inertial frame of reference. Fortunately, there is also significant flight heritage with gyro units onboard CubeSats constructed at the University of Michigan. Other methods of determination, such as coarse sun sensors, will also be useful in determining attitude.

## 2.4 Programmatic Risks & Mitigation

**Component Lead Times**

Components used for space systems often have large lead times. We will mitigate this logistical risk through the design and planning process. By finalizing our architecture early and maintaining a schedule we can limit some of this risk. Additionally, some components are already on hand from prior CubeSat projects.

**FIPEX Integration**

This science payload has never been integrated into a CubeSat designed at the University of Michigan. The integration process spans over multiple subsystems and carries a high level of complexity. To successfully accomplish this, extensive and early testing will be completed. Additionally, reliable communication with the designers at the University of Dresden, through the VKI, will be established and upheld.

# 3 Assembly, Integration and Test Plan

## 3.1 Overview

The following is the anticipated assembly plan for Atlantis and Columbia, based on the University of Michigan's heritage CubeSat designs. This assembly is for one CubeSat.

## 3.2 Tools

Screw drivers and wrenches for 4-40, 2-56 and M3 screws and nuts
Personal protective equipment
- Safety Glasses
- Nitrile Gloves
- ESD Protection

## 3.3 Satellite Layout

### 3.3.1 Hardware

*Structure*
- 2 Walls
- +Z plate
- 3 Bus PCBs, no cells
- 1Bus PCB, fully integrated
- 4 Custom Hinges
- 2 Deployable Panels, fully integrated
- 12 Beam Board Mounts
- 2 Battery Board Mounts
- 4-40, 2-56, and M3 mounting screws and nuts

*Attitude Determination and Control System*
- 2 X/Y axis iron core magnetorquers
- 1 Z axis air core magnetorquer

*Boards/Electronics*
- ADCS Board (Mother Board, MoBo)
- Gyro Board (Daughter Board)
- Torquer Control Board (TCB)
- Battery Pack, fully integrated
- Output Regulation Board (ORB)
- Solar Board (SB)
- FCPU Board (UHF Radio)
- SD Mezzanine
- PIM
- +Z Interface Board
- Mounted antenna

*Solar Panels*
- 15 Solar Cells, integrated into PCB panels

*Payload*
- FIPEX

### 3.3.2 Pre-Integration Steps

## 3.4 Satellite Assembly

### 3.4.1 Interior Assembly

### 3.4.2 Exterior Assembly

### 3.4.3 Before Flight

During integration, any items to be removed before flight will be labeled with a large, bright red "Remove Before Flight" tag stating the name of the spacecraft. Prior to flight, any remove before flight tags will be addressed. After that point, the team does not anticipate needing further access to the satellites within the QuadPack.

# 4 Management Plan

## 4.1 Scope

This document provides information on the management of the QB50 CubeSat development at the Space Physics Research Laboratory (SPRL) at The University of Michigan, Ann Arbor. It helps The University Team and the Von Karmin Institute (VKI) to structure the work to be carried out and to identify potential management-related problems. It thus helps to increase the chance of the QB50 mission success. The content of this document is held confidential between SPRL and VKI.

## 4.2 Work Break down

## 4.3 Information on Funding and Funding Source

## 4.4 Team Organization

## 4.5 Short Curriculum Vitae

**Principle Investigator, Aaron Ridley**
*Current Position*
Professor, 2013 - present
Associate Professor, 2006 - 2013
Associate Research Professor, 2005 - 2008
*Previous Experience*
Constellation Scientist, NASA's CYGNSS mission, 2012-present
Principle Investigator, CubeSat CADRE, 2011-present
*Education*
BS in Physics, Eastern Michigan University, 1992
MS in Atmospheric and Space Sciences, University of Michigan, 1995
PhD in Atmospheric and Space Sciences, University of Michigan, 1997

**Student Team Lead, Kathryn Luczek**
*Current Position*
Student Project Manager/Chief Engineer, 2014 - present
*Previous Experience*
Structures Team Member, CubeSats M-Cubed 2 and CADRE, 2011-2014
*Education*
BSE in Aerospace Engineering, University of Michigan, 2014
MEng in Space Systems Engineering, University of Michigan, 2015

## 4.6 Facilities

The following facilities may be used in the development and testing of the QB50 CubeSats at The University of Michigan:

**Internal Facility, Student Space Systems Fabrication Laboratory**
Within the Atmospheric, Oceanic, and Space Sciences Department at The University of Michigan, S3FL is a student managed laboratory working on space projects. This laboratory includes a TVac chamber which will be used to bake out the flight model satellites.

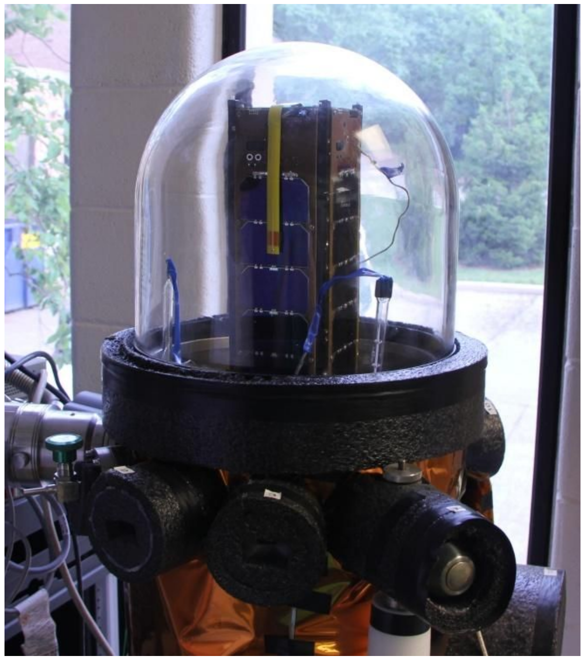

*Figure 35: S3FL TVac Chamber testing the 3U CubeSat GRIFEX*

**Internal Facility, Space Physics Research Lab**

Within the Atmospheric, Oceanic, and Space Sciences Department at The University of Michigan, the Space Physics Research Laboratory is constantly developing novel space science research projects. With the assistance of their professional engineers, students can use the vibe table for conducting structural vibration testing, in addition to consulting with engineers on designs, documentation, and other aspects of the mission. TVac Cycling will also occur in SPRL, using the chamber originally designed for the CYGNSS mission.

**Internal Facility, Wilson Student Team Project Center**

Within the College of Engineering at The University of Michigan, the Wilson Student Team Project Center (WSTPC) serves as a machine shop and work space for undergraduate engineering student teams. The Wilson Center provides training on mills, lathes, and other machines used in the production of student projects, including any structural prototypes, and the engineering design units for our CubeSats.

**Internal Facility, Ground Stations**

The main ground station is located in the Space Research Building.

**Annex 01**

**CubeSat Mass Budget**

| Component | Development Status | Per Unit Mass (grams) | | | Mass (grams) | |
|---|---|---|---|---|---|---|
| | | Estimated Mass | Contingency (%) | Total | Quantity | Mass |
| **Structural Subsystem** | | | | | | |
| *Side Wall* | *Heritage Estimate* | *100* | *5* | *105.00* | *2* | *210.00* |
| *Z Plate* | *Heritage Estimate* | *36.8* | *5* | *38.58* | *1* | *38.58* |
| *Beam Board Mounts* | *Heritage Estimate* | *5.6* | *5* | *5.83* | *6* | *34.98* |
| *Battery Mount* | *Heritage Estimate* | *10.3* | *5* | *10.83* | *2* | *21.66* |
| *Battery Mount Beams* | *Heritage Estimate* | *7.4* | *5* | *7.78* | *2* | *15.56* |
| *+Wing Deployable* | *Heritage Estimate* | *37.0* | *5* | *38.85* | *1* | *38.85* |
| *-Wing Deployable* | *Heritage Estimate* | *37.0* | *5* | *38.85* | *1* | *38.85* |
| *Hinges* | *Heritage Estimate* | *5.0* | *5* | *5.25* | *4* | *21.00* |
| *Side PCB +X* | *Heritage Estimate* | *37.0* | *5* | *38.85* | *1* | *38.85* |
| *Side PCB +Y* | *Heritage Estimate* | *37.0* | *5* | *38.85* | *1* | *38.85* |
| *Side PCB -X* | *Heritage Estimate* | *37.0* | *5* | *38.85* | *1* | *38.85* |
| *Side PCB -Y* | *Heritage Estimate* | *37.0* | *5* | *38.85* | *1* | *38.85* |
| *Avg #2-56 Screw* | *Heritage Estimate* | *0.17* | *5* | *0.18* | *74* | *13.32* |
| *Avg #4-40 Screw* | *Heritage Estimate* | *0.42* | *5* | *0.44* | *20* | *8.80* |
| *Avg M3 screw* | *Heritage Estimate* | *0.17* | *5* | *0.18* | *16* | *2.88* |
| **Attitude Determination and Control Subsystem** | | | | | | |
| *MoBo* | *Heritage Estimate* | *60.0* | *10* | *66.00* | *1* | *66.00* |
| *TCB* | *Heritage Estimate* | *120.0* | *10* | *132.00* | *1* | *132.00* |
| *Magnetometer* | *Heritage Estimate* | *0.2* | *10* | *0.22* | *13* | *2.86* |
| *Gyro board* | *Heritage Estimate* | *12* | *10* | *13.20* | *1* | *13.20* |
| **Electrical Power Subsystem** | | | | | | |
| *Battery Board* | *Heritage Estimate* | *6.0* | *5* | *6.30* | *1* | *6.30* |
| *Batteries* | *Heritage Estimate* | *45.6* | *5* | *47.90* | *2* | *95.80* |
| *Solar Cells* | *Hardware* | *2.3* | *5* | *2.36* | *15* | *35.44* |
| *Cover Glass* | *Hardware* | *1.0* | *5* | *1.05* | *15* | *15.75* |
| *Interconnects* | *Heritage Estimate* | *1.0* | *5* | *1.05* | *19* | *19.95* |
| *Battery Charging Connector* | *Heritage Estimate* | *1.9* | *5* | *2.00* | *1* | *2.00* |
| *Output Regulation Board* | *Heritage Estimate* | *68.0* | *5* | *71.40* | *1* | *71.40* |
| *Switches (RBF, SEP)* | *Heritage Estimate* | *8.0* | *5* | *8.40* | *3* | *25.20* |
| *EPS Connector* | *Heritage Estimate* | *4.3* | *5* | *4.53* | *1* | *4.53* |
| **On-board Computer and On-board Data Handling Subsystem** | | | | | | |
| *FCPU Board* | *Hardware* | *53.0* | *5* | *55.65* | *1* | *55.65* |
| *STAMP9G20* | *Hardware* | *8.0* | *5* | *8.40* | *1* | *8.40* |
| *Payload Interface Module* | *Hardware* | *15.0* | *5* | *15.75* | *1* | *15.75* |
| *Micro SD Card* | *Hardware* | *5.0* | *5* | *5.25* | *8* | *42.00* |
| *AstroDev Lithium Radio* | *Hardware* | *35.0* | *5* | *36.75* | *1* | *36.75* |
| *Bus Ribbon Cable* | *Heritage Estimate* | *12.0* | *5* | *12.60* | *1* | *12.60* |
| *SD Mezzanine Board* | *Heritage Estimate* | *5.0* | *5* | *5.25* | *1* | *5.25* |
| **Communication Subsystem** | | | | | | |
| *PA Board* | *Heritage Estimate* | *15.0* | *10* | *16.50* | *1* | *16.50* |
| *UHF Antenna* | *Heritage Estimate* | *10.0* | *10* | *11.00* | *1* | *11.00* |
| *UHF Antenna Mount* | *Heritage Estimate* | *10.0* | *10* | *11.00* | *1* | *11.00* |

| | | | | | | |
|---|---|---|---|---|---|---|
| **Interface Board** | | | | | | |
| *Z interface board* | *Heritage Estimate* | *18.0* | *5* | *18.90* | *1* | *18.90* |
| **Thermal Subsystem** | | | | | | |
| *Passive* | *Heritage Estimate* | *0* | *0* | *0.00* | *0* | *0.00* |
| **Payload** | | | | | | |
| *FIPEX* | *Hardware* | *160* | *25* | *200.00* | *1* | *200.00* |
| **Subtotal** | | | | | | 1524.10 |
| **Integration** | | | | | | 21.6 |
| **Total** | | | | | | 1545.70 |
| **Target** | | | | | | 2000 |
| **Margin** | | | | | | 454.30 |

## CubeSat Power Budget

| **Loads** | **Power Consumption (W)** | **Overall** |
|---|---|---|
| Antenna Release | 7.551 | 7.551 |
| Peripheral Sensors | 0.038 | 0.038 |
| TCB/Torque Coil | 0.327 | 0.327 |
| 440MHz System Power | 0.136 | 0.136 |
| 440MHz 1W Transmit | 0.264 | 0.264 |
| Flight Computer | 0.402 | 0.402 |
| EPS Solar Board | 0.020 | 0.020 |
| EPS Battery Board | 0.399 | 0.399 |
| FIPEX | 0.607 | 0.607 |
| Sum loads (W) | | 2.193 |
| Efficiency | | - |
| Power Consumed (W) | | 2.193 |
| Power Generated (W) | | 2.62 |
| Power Margin (%) | | 16.32 |