MicroNimbus: A CubeSat Mission for Millimeter-Wave Atmospheric Temperature Profiling

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MicroNimbus: A CubeSat Mission for Millimeter-Wave Atmospheric Temperature Profiling

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MicroNimbus is a small satellite mission being developed by the Georgia Institute of Technology and Georgia Tech Research Institute that will utilize a frequency-agile mm-wave radiometer to measure and update the temperature profile of the atmosphere from a 3U CubeSat platform. The on-board radiometer instrument will provide atmospheric temperature profile data at an altitude resolution of 10 km, a geographic resolution of 0.5°, and a temperature resolution of 2K RMS. The mission strongly aligns with the goals set forth in NASA's Science Plan and will generate data valuable to researchers in the fields of weather forecasting, LIDAR, and laser communications. MicroNimbus has passed its Preliminary Design Review (PDR) phase and is moving towards the Critical Design Review (CDR) for the mission. If successful, MicroNimbus will serve as a first step towards the creation of a constellation of satellites designed to perform near real-time temperature profiling of the atmosphere.

Nomenclature

ADC	=	Attitude Determination and Control System
CDH	=	Command & Data Handling System
COM	=	Communications System
$COTS$	=	Commercial Off-The-Shelf
EPS	=	Electrical Power System
GPS	=	Global Positioning System
HDR	=	High Data Rate
IMU	=	Inertial Measurement Unit
ISS	=	International Space Station
$LIDAR$	=	Light Detection and Ranging
LEO	=	Low Earth Orbit
LDR	=	Low Data Rate
$MMIC$	=	Monolithic Microwave Integrated Circuit
NAV	=	Navigation System
PCB	=	Printed Circuit Board
$SCAMS$	=	Scanning Microwave Spectrometer
STR	=	Structural and Mechanical System
UHF	=	Ultra High Frequency

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I. Introduction

Atmospheric sounding spacecraft have played a critical role in NASA's and NOAA's ability to monitor, analyze, and predict weather over both short and long time scales. One example of this is the NASA Nimbus 6 satellite − part of the Nimbus series of missions (to which MicroNimbus' name pays homage). One of the many instruments carried by Nimbus 6 was the Scanning Microwave Spectrometer (SCAMS) − a passive microwave radiometer in the ∼60 GHz regime that was used to retrieve atmospheric temperature profiles. Another similar, and currently active, mission by NASA is the Aqua (EOS PM-1) satellite. Specifically, the Advanced Microwave Sounding Unit (AMSU) instrument onboard Aqua is capable of scanning between 10 channels in the ∼60 GHz range where O_2 absorption lines occur [1]. The most recent iteration of this type of radiometer is the Advanced Technology Microwave Sounder (ATMS) flown on the NASA Joint Polar Systems Satellite (JPSS) which can be seen in Figure 1. Note that the dimensions on ATMS show that it is much bigger than most typical small satellites (3U or 6U).

Flagships weather satellites such as these provide high radiometric resolution over areas of interest such as tropical storms, heatwaves, etc. However, because few such spacecraft are launched, the data is often of poor temporal resolution with high revisit times.

Thus, frequent repeat measurements of extreme weather events in the microwave regime are not possible with the current suite of NASA and NOAA satellites. These rapid measurements would reveal a significant amount of new information on the formation and evolution of extreme weather phenomenon such as heat waves, cold waves, and tropical cyclones. The capability to obtain this data will allow for the development of higher fidelity weather forecasting models [2]. One method to achieve measurements at higher temporal resolution is to deploy a constellation of satellites in Low Earth Orbit (LEO). Due to the economic infeasibility of launching a constellation of full scale NASA Earth Observing Satellites (EOS), small satellites present the only known feasible option for implementing such a constellation at a reasonable total mission cost. MicroNimbus serves as a pathfinder demonstration mission for this type of mission architecture; with the goal of performing scientific experiments similar to those carried out by Nimbus 6 and Aqua at a fraction of the cost and development time. This will enable weather forecasters to obtain temperature sounding data over specific regions of interest which are revisited on the order of minutes rather than hours or days.

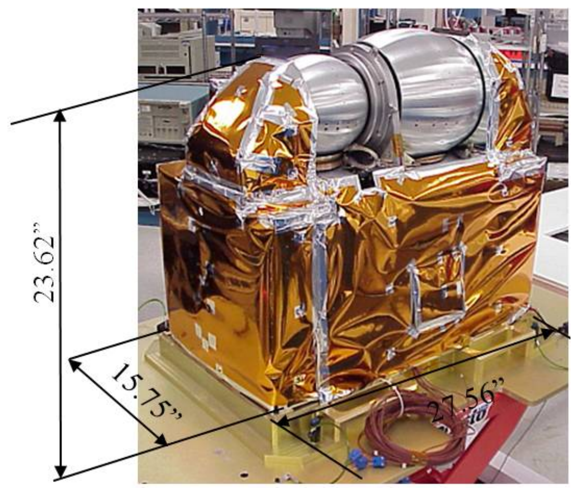

Figure 1: Advanced Microwave Sounding Unit (with Dimensions)

This shift towards smaller spacecraft, however, is not limited to passive microwave radiometry. In recent years, CubeSats have been utilized by the scientific community in a number of areas of remote sensing due to their low cost, fast development schedules, and unique mission architecture configurations (formations, constellations, etc.) [3]. However, in general, this small satellite architecture applied to areas of microwave remote sensing is rapidly expanding. A few missions to note are MicroMAS [4] and MiRaTA [5], both from the Massachusetts Institute of Technology (MIT). The first mission was dedicated to performing radiometric measurements in the ∼118 GHz regime for oxygen absorption lines. The latter is dedicated to measuring weather temperature (52-58 GHz), water vapor (175-191 GHz), and cloud ice (207 GHz). These mission show that there is scientific interest in miniaturizing the satellites that obtain microwave radiometric data of the atmosphere.

Specifically, the MicroNimbus mission miniaturizes the design of a microwave radiometer, such as the Nimbus 6 SCAMS[a] and the AQUA's AMSU instrument, through the use of a silicon-germanium (SiGe) integrated receiver front end and a corrugated horn antenna design. While SCAMS and AMSU focused on scanning three and ten different O_2 absorption bands respectively, MicroNimbus will scan through seven. Furthermore, MicroNimbus will make use of the CubeSat platform in order to reduce the cost and development times required to obtain this type of atmospheric sounding data.

[a]http://nssdc.gsfc.nasa.gov/nmc/experimentDisplay.do?id=1975-052A-10, Accessed 19 April 2018

II. Background

A. Passive Microwave Radiometry

Microwave radiometers are remote sensing instruments that measure the passively emitted electromagnetic radiation by a medium of interest in specific frequency bands in the microwave regime (∼3 to 300 GHz[6]) of the electromagnetic spectrum. Satellite based radiometers have been used for decades by NASA to collect global-scale observations of the Earth which are used by the Earth science community to improve global climate models. These measurements are vital to our understanding of the planet as a system both spatially and temporally and must be collected from satellites orbiting the Earth. Microwave radiometric observations can yield useful information on media ranging from solids (vegetation, ice sheets, snow), liquids (oceans, lakes), and even components of the Earth's atmosphere (water vapor, ozone, oxygen).

Many of these media absorb radiation in a specific frequency regime that can be detected by radiometers sensitive to those frequencies. Specifically, for this mission, radiometric measurements of Earth's atmosphere will be considered as the focal point of the research. In the ∼60 GHz frequency range, multiple frequency bands exist in which atmospheric oxygen (O_2) absorbs the radiation emitted by the Earth's surface and produces absorption lines that a microwave radiometer senses. Some of these frequency bands correspond to different optical depths in the atmosphere, as seen in Figure 2. Measuring these frequency bands can be used to derive temperature profiles of the atmosphere, an example of which can be seen in Figure 3. If enough of these vertical temperature profiles are observed, a global temperature map for specific pressure altitudes can be developed, as seen in Figure 4.

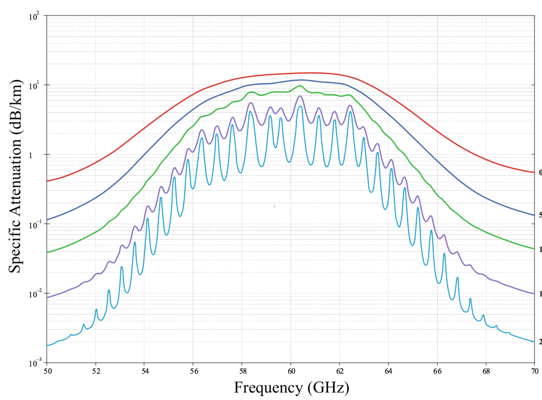

Figure 2: Specific Attenuation in the Range 50-70 GHz [7]

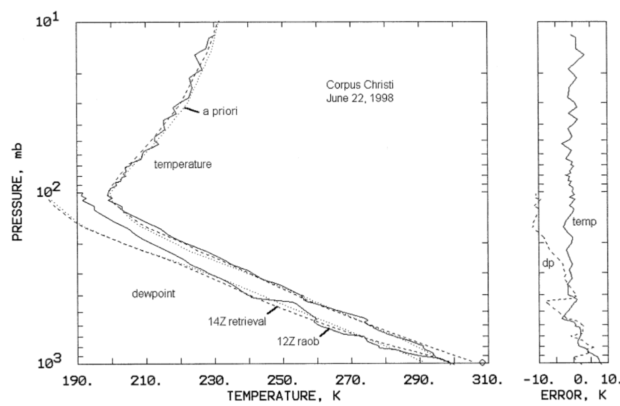

Figure 3: Atmospheric Temperature Retrieval at Corpus Christi, TX [8]

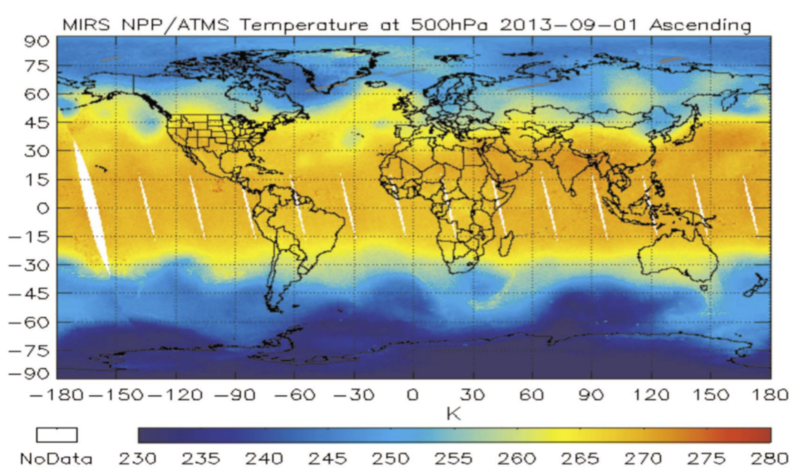

Figure 4: ATMS Global Temperature at 500 hPa on September 1, 2013 [9]

Microwave measurements have certain advantages over smaller wavelength sensors (i.e. optical) in that their measurements are not significantly affected by cloud cover. This property has led to the widespread use of microwave sensors as instruments for weather and climate modeling on both short and long timescales respectively. In the short-term, these radiometers can provide observations of extreme weather events such as hurricanes by penetrating cloud cover and determining key thermodynamic profiles of the storm system such as temperature, water vapor, cloud and precipitation [2]. In the long-term, these measurements remain metrically consistent as local cloud and weather patterns have little effect on the data obtained.

B. Traceability to NASA's Objectives

One of the high level goals presented in the NASA 2014 Strategic Plan is to "Advance knowledge of the Earth as a system to meet the challenges of environmental change, and to improve life on our planet"[10]. This high level goal flows down into two sub-goals; scientific understanding of the climate system and technology development of Earth based remote sensing instruments. More specifically, the NASA 2014 Science Plan calls for researchers to "improve the ability to predict climate changes by better understanding the roles and interactions of the ocean, atmosphere, land, and ice in the climate system"[11] and the NASA 2015 Technology Roadmap calls for researchers to "improve remote sensing capabilities and performance" through "investments in microwave, millimeter-... receiver component technology include low-noise receivers...and field demonstration of active and passive instruments from mm to sub-mm wavelengths" [12]. The MicroNimbus mission directly addresses both of these sub-goals by creating a single integrated mm-wave radiometer front end and using this device in space to provide near real-time atmospheric temperature profile data to researchers for verification and improvement of current weather models over daily and monthly time scales. A NASA objectives traceability summary is shown in Figure 5.

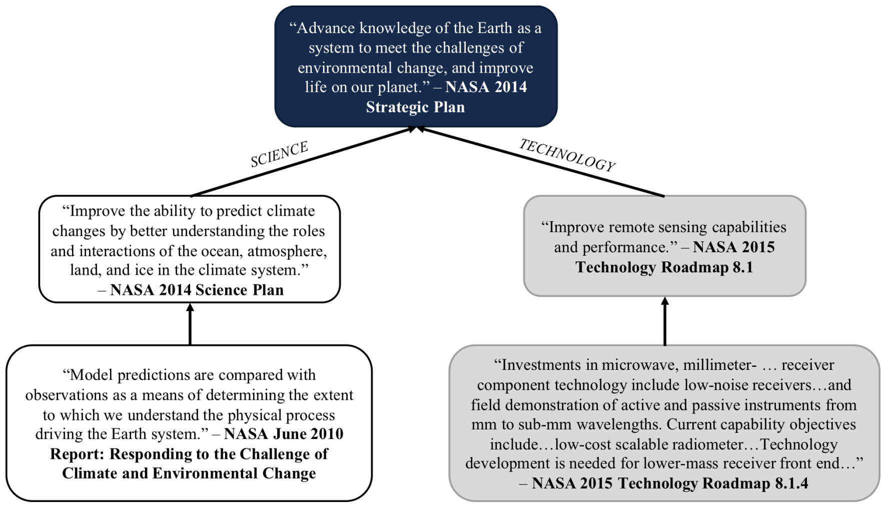

Figure 5: Traceability to NASA's Objectives

C. Applications

Researchers can use the data generated by MicroNimbus to understand a variety of climate and weather related phenomena. For example, the data generated by the AMSU instrument has been used to observe tropical storms because temperature measurements are not significantly affected by cloud cover that typically resides over these storm systems. Thus, these passive measurements can penetrate through the layers of cloud cover, allowing researchers to determine thermodynamic properties of the storm as a function of its altitude. This type of vertical temperature data has been used by researchers to determine the relationships between temperature anomalies and surface wind speeds and central pressure of a tropical storm system[13]. An example of this temperature altitude profile is shown in Figure 6.

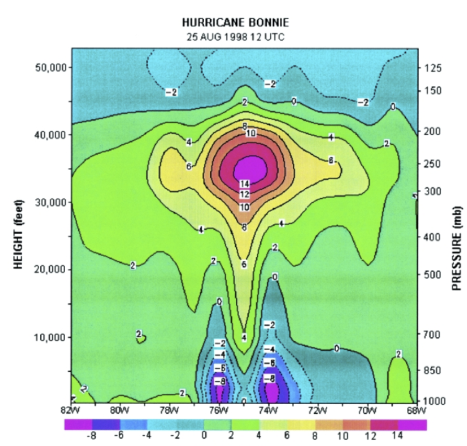

Figure 6: Cross Section of Temperature Anomalies Through Hurricane Bonnie at 1200 UTC, 25 August 1998. Retrieved from AMSU [13]

III. Mission Design

A. Orbit Description

Currently, the mission baseline design is that the satellite will be deployed from the ISS and will perform nominal operations until orbital decay due to atmospheric drag causes the satellite to be destroyed in the atmosphere. Other orbit options have been considered and will be addressed should the launch opportunity provide it. The most scientifically useful of these options is a polar orbit at higher altitudes than the ISS. Polar orbits would be ideal for a remote sensing mission such as this due to the fact that global coverage can be achieved. Furthermore, power requirements would be relaxed if the orbit is constrained to be sun-synchronous. However, higher altitude orbits would cause a lower spatial resolution for the instrument. Because MicroNimbus has no on-board propulsion system for station-keeping, it is predicted to remain in orbit for approximately 6 to 9 months with an ISS deployment. Even at this orbit, nominal operations should yield instrument data for at least one season, demonstrating the utility of the measurements. The classic orbit elements used for the mission analysis are shown in Table 1.

Table 1: Classical Orbital Elements for MicroNimbus

$\boldsymbol{a}$ (km)	$\boldsymbol{e}$	$\boldsymbol{i}$ (deg)	$\boldsymbol{\Omega}$ (deg)	$\boldsymbol{w}$ (deg)	$\boldsymbol{f}$ (deg)
404.5	0.0001935	51.6	100.69	226.1407	-

B. Operational Views

The Concept of Operations (ConOps) visually describes the nominal mission in Figure 7. In general, the spacecraft will deploy from the launch vehicle, perform a de-tumble maneuver, deploy the solar panels, perform initial checkouts, and then begin nominal operations. Nominal operations switch between science data collection, data transmission, and power generation (as detailed in later sections) until the End-Of-Life procedure is initiated − just before the spacecraft begins de-orbiting rapidly due to drag. MicroNimbus must maintain nadir pointing, or as close to it as achievable, during science mode − which occurs only in the daylight portion of the orbit (shown in white). During eclipse (shown in red), no science operations are performed due to attitude determination constraints (as described in Section IV). The spacecraft has two main communications systems, a UHF LDR system which performs both uplink and downlink, and an S-band HDR system which only performs downlink. The UHF is primarily used for command uplink and health/telemetry downlink. The S-band is only used for science data downlink. The exact method to transition between different modes of the mission will be detailed in Section V.

The Operational View 2 (OV2) diagram, in Figure 8, shows the flow of information to and from the physical phenomenon observed through a series of "need lines". In one direction, the Earth's atmosphere emits radiation in the 60 GHz regime and is captured by the radiometer on-board MicroNimbus. MicroNimbus transmits both the raw measurements and its states (position, velocity, and attitude) to the S-band and UHF ground stations respectively. The ground stations send the raw data to the GT mission operations center which then forwards the raw data to the GTRI data servers. From here the data is calibrated and processed, and then sent to atmospheric data models. These models then become publicly accessible through a secure web interface.

In the opposite direction, the GT mission operations center prepares a command script for the spacecraft and forwards it to the UHF ground station. The UHF ground station sends these commands to the spacecraft during the next available pass. Note the colors for UHF and S-band transmission in Figure 8 correspond to those in Figure 7.

C. Scientific Data Collection Requirements

Two types of radiometric sounding data must be obtained for this mission in order to satisfy science requirements, a short-term (day to day) variation and a long-term (weekly to monthly) variation. Because of constraints on attitude knowledge in the eclipse phase of the orbit, the payload will only collect data during the sunlight side of the orbit.

The Level 1 System Requirements call for the mission to be operational for up to 6 months to obtain seasonal variations in atmospheric sounding data. However, due to the power limitations of the mission,

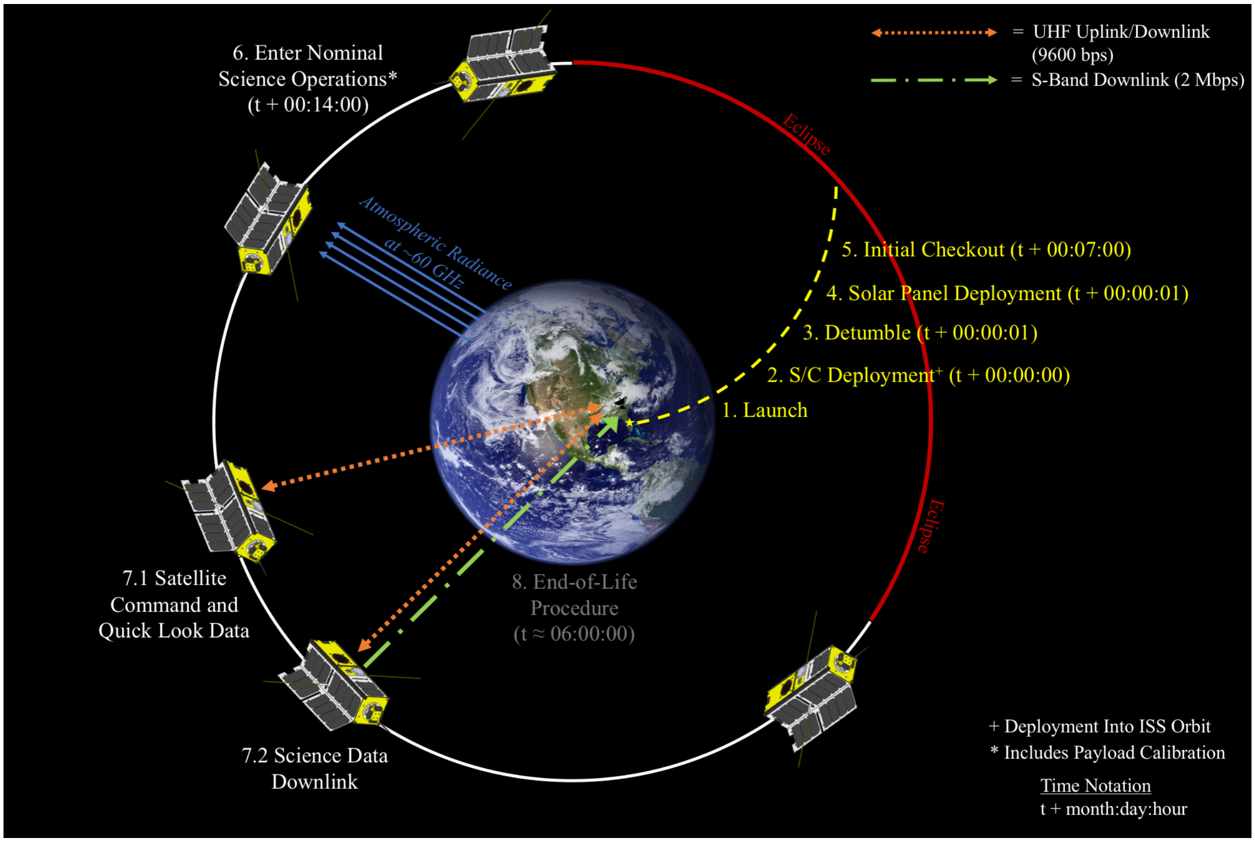

Figure 7: MicroNimbus Concept of Operations

this data will not be continuous. Nominally, the mission is designed to perform three consecutive orbits worth of science data collection followed by one orbit for recharging the on-board batteries. This includes time dedicated for scientific and spacecraft health data downlink, regardless of being in a science orbit or a charging orbit, as seen in Figure 9. Although this will create gaps in the data obtained, for the purposes of observing changes on weekly or monthly time scales, the overall trends in the measurements will still achieve the scientific objectives.

D. TECHBus

This mission will be the first to make use of The Evolved Common Hardware Bus (TECHBus), a generic 3U or 6U CubeSat bus under development at Georgia Tech. Arising from the need of having a satellite bus that can be used to fly a variety of scientific payloads without having to be redesigned for each mission, the TECHBus is an in-house, versatile, reusable, and reliable approach to solving this problem[14]. MicroNimbus will be the first mission to make use of this bus design, serving as a demonstration mission for future Georgia Tech Earth orbiting 3U CubeSats. Because almost every mission requires some level of customization, the payload section of the TECHBus can be customized to incorporate the payload and any required sensors or actuators that are not already present on the standard bus.

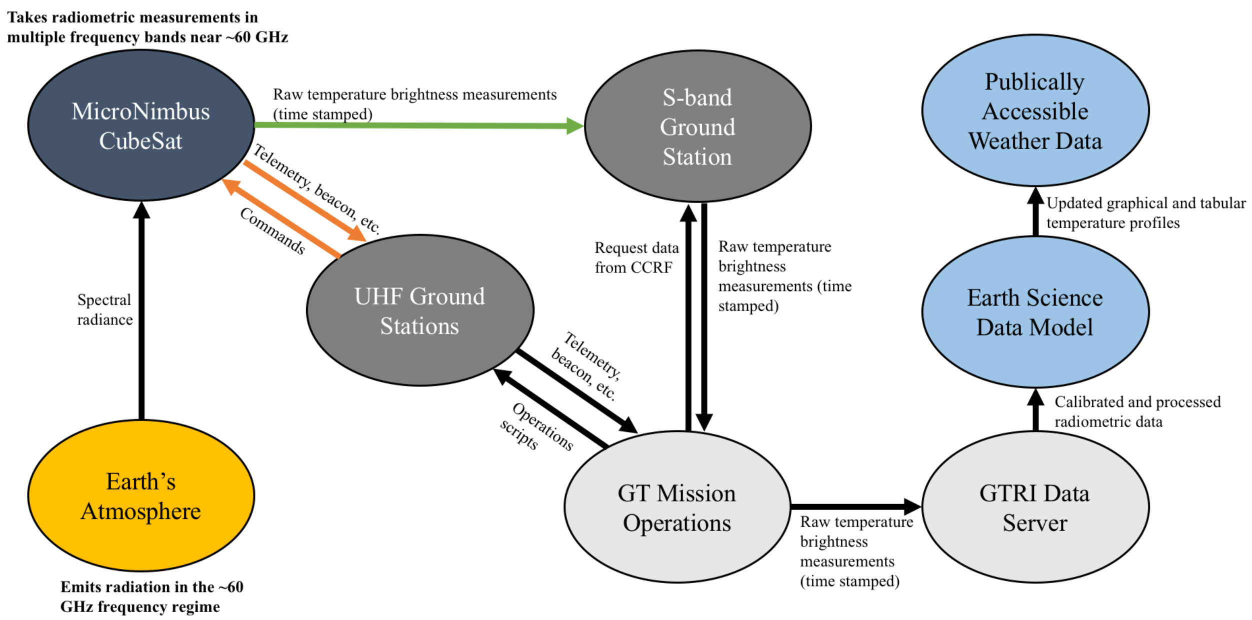

Figure 8: MicroNimbus Operational View 2 (OV2)

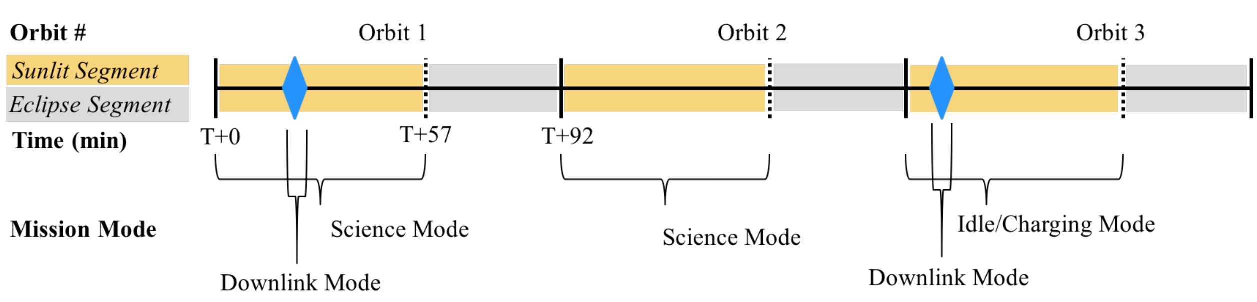

Figure 9: MicroNimbus Example Operations Schedule

IV. Radiometer Payload

To develop a radiometer capable of measuring the physical parameters necessary to accurately derive temperature profiles of the atmosphere, a Science Traceability Matrix (STM) is established. The STM can be seen in Figure 10.

Science Goal	Science Objectives	Measurement Requirements		Instrument Functional Requirements	Mission Top Level Requirements
		Observables	Physical Parameters		
	Primary Science Objective				
Utilize a frequency-agile radiometer to measure and profile the temperature of sections of Earth's atmosphere as a function of altitude	Atmospheric oxygen absorption for seven frequencies and signal bandwidths near 60 GHz will be inverted to yield altitude weighting functions of the atmosphere's temperature	Extremely High Frequency (EHF) radio waves (~60 GHz) emission lines of atmospheric oxygen	Intensity of thermal energy radiated by the Earth's atmosphere at varying altitudes.	Must be capable of measuring seven frequencies (in GHz): 64.47, 60.82, 60.5685, 60.4409, 60.4365, 60.4348, and 58.388	Radiometer must be calibrated at a frequency of 10 Hz while taking measurements
			Absorption and optical depth of atmospheric oxygen near 60 GHz consisting of multiple lines with differing absorption levels	Frequencies measured must have a bandwidth of 200, 200, 1.5, 2.5, 1.0, 1.5, and 30 respectively	The noise level of the spacecraft bus with all components active shall be less than X dBm at ~60 Ghz
				Radiometer output must be sent to a square law detector capable of outputting at a sample rate of ~10 Hz (10 bits/sample)	

Figure 10: MicroNimbus Science Traceability Matrix - STM

The microwave radiometer onboard MicroNimbus will be capable of scanning through seven frequencies, allowing the instrument to sound the atmosphere from an altitude range of 10 - 80 km[15], as summarized in Table 2. Since CubeSats are inherently volume, mass, and power constrained, these limitations also translate into constraints on the radiometer itself. For MicroNimbus, the radiometer payload (including all additional required sensors, boards, etc.) is designed to occupy no more than 1.5U of volume with a mass limit of 0.5 kg and power consumption of less than 1 W. To meet these objectives, the 60 GHz receiver front-end is integrated on a single SiGe integrated circuit, a level of integration previously not achieved at 60 GHz [16].

Table 2: Radiometer Frequencies and Corresponding Altitude Windows

Frequency (GHz)	Bandwidth (MHz)	Altitude Window (km)	Window Width (km)
64.47	200	12	11
60.82	200	18	7
58.388	30	27	9
60.4409	2.5	40	12
60.4365	1	50	20
60.5685	1.5	60 (equator), 54 (pole)	21 (equator), 26 (pole)
60.4348	1.5	73 (equator), 66 (pole)	20 (equator), 26 (pole)

A. SiGe Integrated Receiver Front End

The main reason why an integrated receiver front end was selected for the payload was due to the large reduction in size, weight, and power consumption that comes as a result of packaging the instrument into one integrated circuit. While typical radiometers make use of multiple chips from multiple material technologies, the approach for this design was to use a single semiconductor material for the entire device − an approach that is enabled by the use of SiGe technologies. Although this approach does have some drawbacks in achievable sensitivity, it does however offer major advantages in the areas of manufacturing, radiation tolerance, and thermal management[17].

The first of these advantages is manufacturing; SiGe technologies have better manufacturing tolerances than other semiconductor materials (GaAs, InP, etc.) allowing for low chip-to-chip and circuit-to-circuit variation. Another advantage is that SiGe heterojunction bipolar transistors (HBTs) have the best low-frequency noise characteristics of all high-frequency semiconductor technologies − a critical consideration for minimizing the out of band sensitivity in radiometers. Additionally, they are far more resistant to degradation due to total dose radiation, suffering almost no performance change up to multi-Mrad doses[17]. Because CubeSats are weight and volume constrained, there is often little radiation shielding available and thus radiation tolerant components are of high value for these types of missions. Finally, silicon integrated circuits have high thermal conductivity, leading to more stable thermal properties on orbit and requiring less active thermal management − once again reducing overall size, weight, and power consumption.

The SiGe integrated receiver comprises most of the major components of the radiometer[16] as seen in Figure 11. Note that all components which are integrated onto a single SiGe chip are bounded by a box in Figure 11.

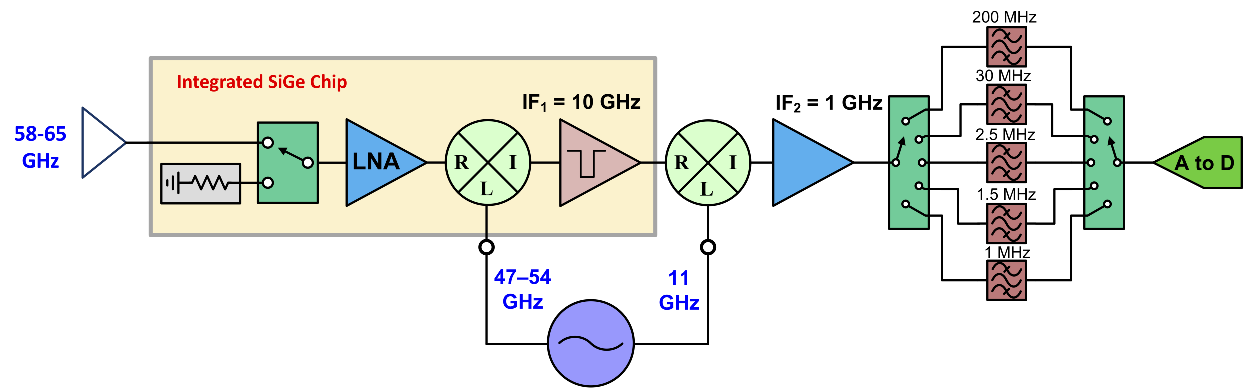

Figure 11: 60 GHz Profilometer Radiometer - Block Diagram

B. Radiometer Horn and Structure

The volume constraint of 1.5U was the driving factor in the selection and design of the radiometer horn antenna design. For this mission, a corrugated horn was selected because it has similar efficiencies (low loss and good match) to that of a larger antenna but fits within a smaller form factor, thus allowing the antenna to fit inside the payload volume[18]. Although this antenna is more difficult to manufacture than traditional horn antennas, the performance combined with compactness of the design outweighs the additional schedule and cost increases required for manufacturing. The final integrated payload module can be seen in Figure 12.

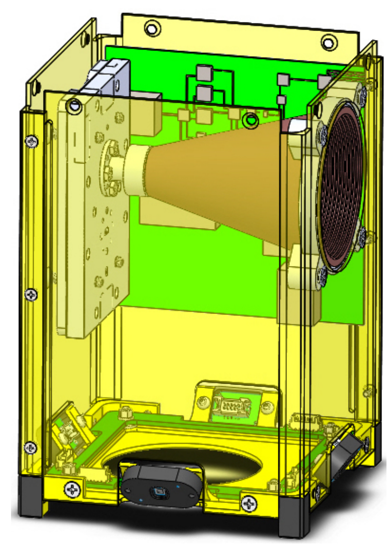

Figure 12: MicroNimbus Payload Module - Integrated

V. Requirements

Requirements for the MicroNimbus mission are differentiated into tiers, from Level 0 to Level 3. Level 0 requirements start with the mission statement and from there, the mission objectives flow down. The mission objectives are then used to create the mission requirements and mission success criteria. Next, system level requirements (Level 1) are generated which satisfy the mission requirements. The system level requirements are divided into two categories; flight system and mission operations. The flight system deals with any requirements related to the spacecraft and its respective subsystems (ADC, EPS, PAY, etc.). The mission operations system deals with the two ground stations, GSE, and various test facilities used to support the mission. Level 3 requirements stem from Level 2 requirements and deal with component level requirements. For example, the Level 3 requirements for the ADC subsystem will include requirements on the reaction wheels, torque rods, sun sensors, etc. A summary of the requirements flow down can be seen in Figure 13.

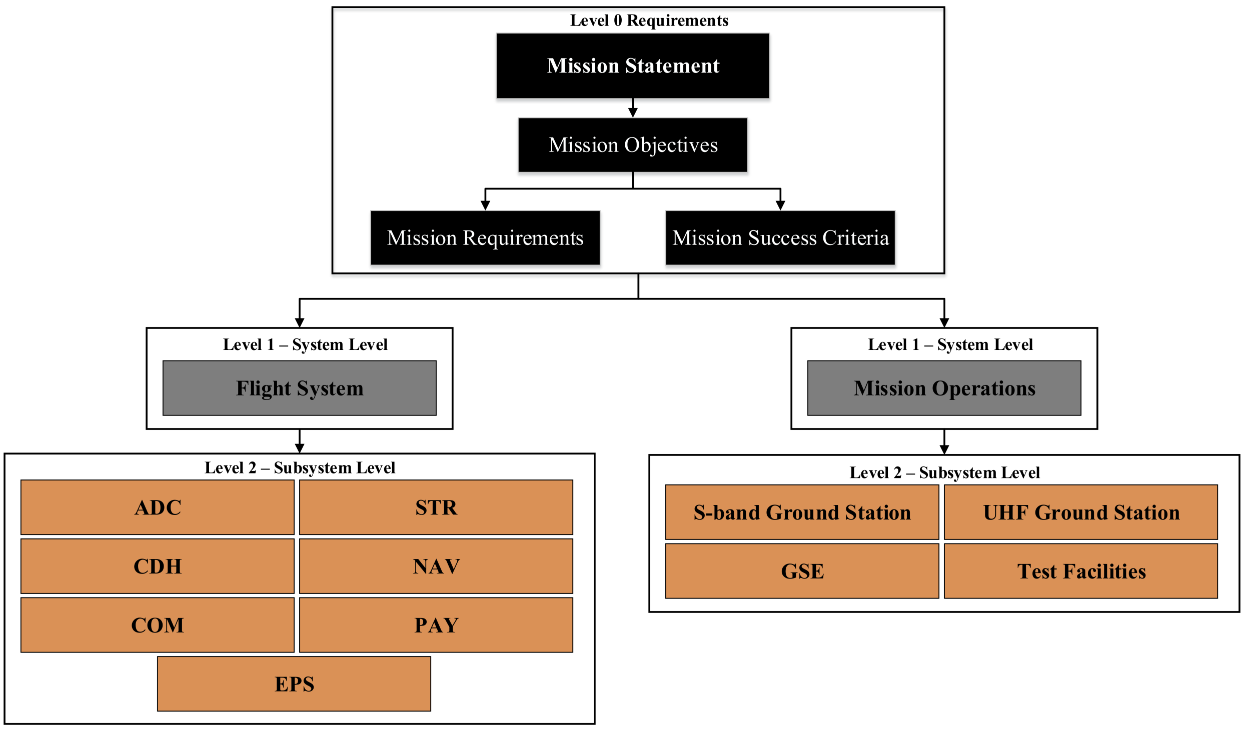

Figure 13: MicroNimbus – Requirements Flow Down

A. Level 0

The mission statement describes the entire mission and, from it, all other requirements are derived. The mission statement for MicroNimbus is:

MicroNimbus will utilize a frequency-agile mm-wave radiometer to measure and update the temperature profile of the atmosphere from a CubeSat platform.

From this statement two mission objectives are derived, one for the science/payload (MO-1) and one for the spacecraft bus design (MO-2). These two mission objectives are shown in Table 3.

MO-1 places two major constraints on the instrument and the type of data collected. The first constraint is that the instrument must be capable of measuring multiple frequency bands of the atmosphere within the ∼60 GHz band. The second constraint is that the instrument must fit within a 1.5U volume available for the payload (typical for 3U CubeSats). MO-2 places constraints on the design of the overall spacecraft. Specifically, that the mission will use the Georgia Tech in-house CubeSat bus design known as TECHBus.

The mission objectives next help to define both the mission requirements and the mission success criteria – each of which serve a specific purpose. The mission requirements are directly answerable to the the Level 1

Table 3: MicroNimbus – Level 0 Requirements – Mission Objectives

ID	Mission Objectives: "The objective of MicroNimbus is to..."	Source
MO-1	Enable future real-time weather monitoring by demonstrating a frequency-agile radiometer capable of measuring radiant energy in multiple frequency bands within the 60 GHz range within a 1.5U payload volume.	MS
MO-2	Enable quicker development schedules for future Georgia Tech in-house small satellite missions by demonstrating the use of a CubeSat platform known as TECHBus.	MS

requirements. For example, a Level 1 requirement should only trace back to the Level 0 mission requirements (not the Level 0 mission objectives for instance). The Level 0 mission requirements can be seen in Table 4.

Table 4: MicroNimbus – Level 0 Requirements – Mission Requirements

ID	Mission Requirements: "The MicroNimbus mission shall..."	Source
MR-1	Gather information on atmospheric temperature as a function of altitude and orbit states by measuring brightness temperatures of the atmosphere in multiple frequency bands.	MO-1
MR-2	Validate the TECHBus platform as a reuseable CubeSat design for future science and technology demonstration missions.	MO-2
MR-3	Pass any scientific data gathered on-orbit to scientists on the ground for analysis.	MO-1, MO-2
MR-4	Shall adhere to small satellite programmatic guidelines and standards set forth by NASA, NOAA, and the FCC wherever possible.	MO-1, MO-2

The mission success criteria (both minimum and full) are a set of "check lists" that mission operators use to assess the progress of the mission. While the mission success criteria are answerable to the mission objectives, no lower level requirements are answerable to the mission success criteria. The minimum and full mission success criteria can be seen in Table 5 and 6.

Table 5: MicroNimbus – Level 0 Requirements – Minimum Mission Success

ID	Minimum Mission Success: "The MicroNimbus mission shall..."	Source
MMS-1	Achieve successful launch vehicle separation and detumbling.	MO-2
MMS-2	Establish communication link and perform health checks for all subsystems.	MO-2
MMS-3	Successfully calibrate the science payload on orbit.	MO-1

Table 6: MicroNimbus – Level 0 Requirements – Full Mission Success

ID	Full Mission Success: "The MicroNimbus mission shall..."	Source
FMS-1	Downlink 30 days (with at least 30 daily minutes) of science payload data to the ground station.	MO-1
FMS-2	Downlink 30 minutes worth of science payload data to the ground station.	MO-1
FMS-3	Transmit a periodic beacon on amateur radio bands for public outreach	MO-1
FMS-4	Operate on-orbit for a minimum of 4 months.	MO-2
FMS-5	All science data downlinked is registered on ground to 2 K precision.	MO-2

B. Level 1

Level 1 requirements address the flight system and mission operational requirements. The flight system requirements are written to put programmatic and high level technical constraints on the spacecraft. Similarly, the mission operations requirements are written to put both programmatic and high level technical constraints on all necessary ground support equipment (facilities, ground station, etc.). Level 1 requirements for the flight system are shown in Table 7.

Table 7: MicroNimbus – Level 1 Requirements – Flight System

ID	Requirement: "The MicroNimbus satellite shall..."	Source	Verification
SR-1	Be operational for 4 months in order satisfy full mission success criteria.	MR-1	Analysis
SR-2	De-orbit within 25 years after the start of the mission.	MR-4	Analysis
SR-3	Provide a space of 1.25U in the payload module for the radiometer.	MR-1, MR-2	Inspection
SR-4	Operate continuously throughout all phases of mission (sunlit, eclipse, etc.) without going into low power mode	MR-1, MR-2	Analysis, DITL Test
SR-5	Store and downlink both payload and subsystem data	MR-2	DITL Test
SR-6	Communicate command and telemetry data regardless of attitude control ability	MR-2	Analysis
SR-7	Store and execute commands sent by Georgia Tech ground stations	MR-2	Analysis
SR-8	Have the ability to point and maintain pointing within $\pm 1°$ from nadir	FMS-4	Analysis
SR-9	Operate using frequency bands and modulation schemes which are compatible with Georgia Tech ground stations	MR-2	Analysis
SR-10	Meet all structural, electrical, and testing requirements set forth by Cal Poly and NanoRacks	MR-2, MR-4	Analysis
SR-11	Downlink payload data to Georgia Tech ground stations	MR-3	Analysis
SR-12	Provide a stable 10 MHz and 100 kHz GPS-disciplined oscillator to the science payload.	MR-1	Analysis
SR-13	Know the spacecraft position within 1 km RMS	MR-1	Analysis

C. Level 2

Level 2 requirements are placed on each of the individual subsystems based on the Level 1 requirements. For example, for the flight system, Level 2 requirements are placed on subsystems such as the ADC, EPS, and PAY. For the mission operations system, the Level 2 requirements are placed on subsystems such as ground stations, GSE, and test facilities. One example of Level 2 requirements (for the EPS) can be seen in Table 8. The full set of Level 2 requirements for each subsystem can be seen in `\16043048-MicroNimbus\Documents\SYS\top\sys-top-Requirements.xlsx`

Table 8: MicroNimbus – Level 2 Requirements – EPS

ID	Requirement: "The EPS subsystem shall..."	Source
EPS-1	Include an RBF inhibit that turns off all power supply rails when inserted	SR-10
EPS-2	Include a latching kill switch inhibit that turns off all supply rails when active	SR-10
EPS-3	Reset all RF transmission and deployable actuation timers if the deployment switch(es) are reengaged	SR-10
EPS-4	Incorporate battery circuit protection for charging/discharging	SR-10
EPS-5	Charge the battery through the GSE harness when one or more inhibits are active	SR-10
EPS-6	Provide battery voltage/current, all supply rail voltage/current, solar panel voltage/current, and battery temperature to the CDH upon request	SR-5
EPS-7	Provide a minimum of 3A on a regulated 3.3V $\pm$ 5% supply	SR-1, SR-4
EPS-8	Provide a minimum of 3A on a regulated 5V $\pm$ 5% supply	SR-1, SR-4
EPS-9	Provide unregulated battery voltage to the payload	SR-4
EPS-10	Discharge at least 6 W-h for ecplise operation	SR-4
EPS-11	Have sufficient solar panels to charge the batteries at least 6 W-h during the daylight phase of the orbit	SR-4
EPS-12	Process three independent solar panel strings, each string rated for 18V and 1A input	SR-4
EPS-13	Be compatible with the PC-104 Cubesat Standard	SR-10
EPS-14	Not be active from integration into the deployer to separation from the deployer	SR-10

D. Level 3

Level 3 requirements are requirements placed on each individual component of a specific Level 2 subsystem. For example, The EPS subsystem consists of multiple components such as the batteries, power distribution board, solar panels, and wire harnessing. Each of these components has a set of requirements on it that trace back and satisfy Level 2 EPS subsystem requirements. For brevity, no Level 3 requirements are shown in this paper. The full set of Level 3 requirements can be seen in The full set of Level 2 requirements for each subsystem can be seen in `\16043048-MicroNimbus\Documents\SYS\top\sys-top-Requirements.xlsx`

VI. Spacecraft Design

A. Attitude Determination and Control - ADC

1. Components

The ADC subsystem consists of a variety of sensors, actuators, and interface boards. The attitude is determined primarily through the use of eight sun sensors, which are photodiode arrays that provide a sun vector to the spacecraft, and an on-board magnetometer. The attitude is mainly actuated with reaction wheels, which are desaturated using magnetic torque rods. The eight sun sensors present on the spacecraft allow for full-sky coverage to be achieved in daylight and the reaction wheels allow for enough fine pointing to meet both the payload and COM subsystem requirements. The main requirements enforced onto the ADC subsystem stem from the payload and the COM subsystem. The radiometer payload requires nadir attitude pointing to within 1° while the COM subsystem requires a maximum slew rate of up to 1.5° per second for downlinking to the ground station.

Due to the need for low cost, availability, and flight heritage, most of the sensors and actuators are COTS components except for the magnetic torque rods and certain printed circuit boards. The magnetic torque rods are manufactured in-house using a stainless steel core with magnet wire wrappings and are designed to provide enough magnetic moment to desaturate the reaction wheels and de-tumble the spacecraft. Due to the variety of communication protocols used by the ADC components, a customized ADC interface board has been made in order to connect all sensors and actuators to the flight computer. A summary of all components present in the ADC subsystem is shown in Table 9.

Table 9: ADC Subsystem Components

Type	Component	Quantity	Manufacturer
Sensor	Sun Sensor	8	SolarMEMS
Sensor	IMU	1	Epson
Sensor	Magnetometer	2	Honeywell
Actuator	Reaction Wheel	3	Sinclair Interplanetary
Actuator	Magnetorquer	3	In-house
PCB	ADC Interface Board	1	In-house
PCB	Sun Sensor Interface	2	In-house

2. Modeling & Simulation

A full ADC simulation is developed within NASA Goddard Space Flight Center's 42 dynamic simulation software. This simulation imports spacecraft parameters and performs high-fidelity orbit and attitude propagation with the controller in the loop. 42 allows users to import CAD models of the spacecraft with variable mass moments of inertia. Next, sensor and actuator models are implemented within the spacecraft. For example, the torque rods for the spacecraft are custom made so the exact magnetic moment generated by the custom rods was imported into the model. Furthermore, the field of view of each sun sensor was modeled obtain attitude estimates when the sun was in the field of view of the sensors.

Finally, numerous controllers were placed in the loop and tested to determine if they performed according to requirements. For brevity, full details of the controller design and implementation [19] will not be explained. However, a brief explanation of the types of controllers available on the spacecraft along with each respective use is shown in Table 10.

Table 10: ADC Controllers

Controller	Purpose
De-tumble	To negate spacecraft rotation rate with respect to the Earth's magnetic field. Used anytime the spacecraft boots/re-boots.
Nadir Pointing	Points the radiometer nadir to collect measurements. Requires highest accuracy sensor measurements and actuation.
Slew	Points the S-band patch antenna towards the ground station continuously during a ground station pass. Can be used to achieve higher downlink times.
Sun-Pointing	Points the zenith face of the spacecraft parallel to the sun vector and maintains this orientation. Used for maximally charging the spacecraft batteries.
User-Defined Pointing	Points the spacecraft nadir axis at an arbitrary direction with respect to the LVLH frame. Used for potential calibration or sun avoidance maneuvers.

B. Command & Data Handling - CDH

1. *Components*

Because the bus is designed to be robust and have redundant components, one of the biggest challenges in designing the CDH subsystem was finding a flight computer with the required peripherals to be able to interface with all components. The peripherals needed on the flight computer are summarized in Table 11.

Table 11: Flight Computer Peripheral Summary

Type	Quantity	Use
Ethernet	2	Payload, GSE
I^2C	3	(Current Sensor, EPS, S-band Com), Magnetometer, Reaction Wheels
SPI	4	Sun Sensor A2D Converter, S-band Data
UART	2	AX100, IMU, GPS, Console

Additionally, the ability to recover from single event upsets (SEU) through the use of error correcting code (ECC) was also desired. After extensive searching, no single computer embedded processor was found that had the required peripherals, ECC, and fit within the volume allocated within the CubeSat. Thus, other options were explored that could meet these requirements. One potential option was to use a heterogeneous processing architecture that combines a single embedded processor (the ARM processor for example) along with a field programmable gate array (FPGA). This allows communication protocols which don't exist on the processor side to be "created" within the FPGA fabric and mapped to a series of generic input pins. It was also discovered that most heterogeneous systems that include an FPGA come with ECC memory as standard (typically a specialty feature on single embedded processors). Furthermore, FPGA's are, in general, more tolerant to radiation compared to standard embedded processors − a desirable feature for satellite missions. Based on these desired characteristics, the flight computer selected is the NOVSOM-CV. This embedded heterogeneous module is based on the Cyclone V SOC − a combination of an FPGA and ARM processor. The specifications for the NOVSOM-CV can be seen in Table 12.

Table 12: NOVSOM-CV Specifications [20]

Characteristic	Specification
Clock	Up to 925 MHz
Operating Temperature	$-40°$C to $+85°$
Dimensions	2.83" X 2.33"
RAM	2 GB DDR3 with ECC
Processor	Altera Cyclone V

2. Software

Software development pre-CDR has occured on a combination of a BeagleBone Black and a NOVSOM-CVLite (which contains the same chip as the NOVSOM-CV). Because the NOVSOM-CV has a lead time of 14 weeks for the version required for flight (with high temperature tolerance and ECC memory), pure software development (including all drivers and applications) was tested and built using the BeagleBone while all FPGA specific peripheral generation occurred in the CVLite. Since all three computers run a linux operating system and an ARM computer architecture, the code will be portable to either the CV or CVLite from the BeagleBone. Also, the CVLite has the same chip and FPGA fabric as the CV, allowing any FPGA specific work to be transferable to the CV upon delivery.

The MicroNimbus flight software utilizes NASA's Core Flight Software (cFS)[21], as it provides a baseline framework for developing flight software for use on small satellite missions. The functionality provided by cFS includes software bus messaging, event management, time services, and file management systems. Each one of these systems in cFS is an application or "app." The software bus message system allows for the transfer of commands to an application, or telemetry data between applications, through a data subscription architecture. The event managing application is a system that allows an application to track that a predetermined requirement has been met in the software (an "event" has occurred), as well as notify any applications expecting a particular event occurrence. The time services application is a scheduling system that is set to send messages to applications at specified periodic intervals (e.g., commanding a system to send telemetry data at a pre-determined rate). The file management system manages the onboard file storage, creating and deleting files. Users of cFS create applications and utilize the aforementioned cFS services. The cFS package also provides a way for users to create and manage threads without worrying about operating system (OS) specific syntax and usage. Currently the software dependency of MicroNimbus as used in cFS is summarized using in Figure 14.

C. Communications - COM

1. Components

The COM subsystem is designed to satisfy two major driving requirements. The first, and most important requirement, is to be able to communicate command and telemetry data regardless of attitude control ability. This drove the selection of a LDR communication system that uses UHF (430 to 440 MHz) with an omni-directional dipole antenna. The second driving requirement is to be able to communicate the large amount of scientific data to existing Georgia Tech ground stations. This requirement drove the selection of a HDR communication system that uses S-band frequencies (2200 to 2250 MHz) with a patch antenna ($\pm 60°$ beamwidth).

The LDR radio chosen is the GOMSpace AX100. This radio is a full-duplex radio in the UHF band and is paired with the ISIS UHF deployable dipole antenna. The combination of these two components will provide MicroNimbus with an omni-directional link to the ground with a data downlink capability of 9.6 kbps. The HDR radio chosen is CPUT's STXC. This radio operates in the S-band and is paired with a Haigh Farr patch antenna with a 60° half-cone angle. The patch antenna will be placed on the nadir face of the spacecraft so that downlink can be easily transitioned into from science mode. The combination of these two components will provide a 2 Mbps downlink capability for raw payload data (with 1 Mbps expected for science data throughput).

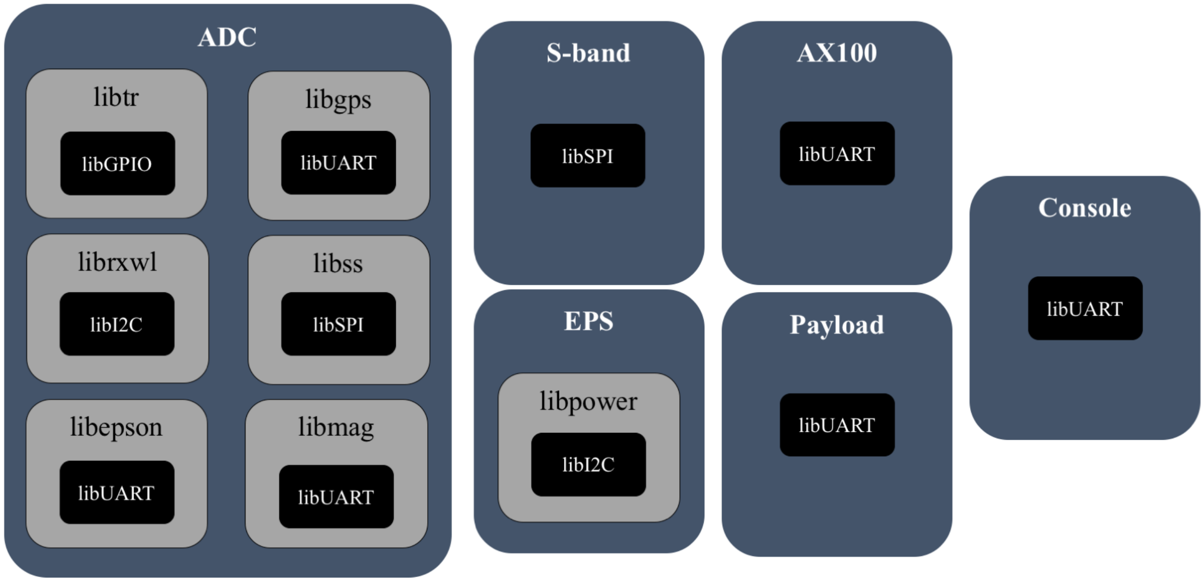

Figure 14: Software Dependency Diagram

2. Simulation

By placing the UHF dipole antenna at the center of the spacecraft, there was concern that the omni-directional gain pattern of the antenna may be non-uniformly attenuated. To determine if this is the case, antenna simulations were performed [22] using the ANSYS High Frequency Electromagnetic Field Simulation (HFSS) software tool. First a low fidelity outer structure model of the MicroNimbus spacecraft (with wings deployed to 90°) with the UHF dipole antenna was imported into HFSS and gain pattern simulations were run. The simulation results for this design can be seen in Figure 15.

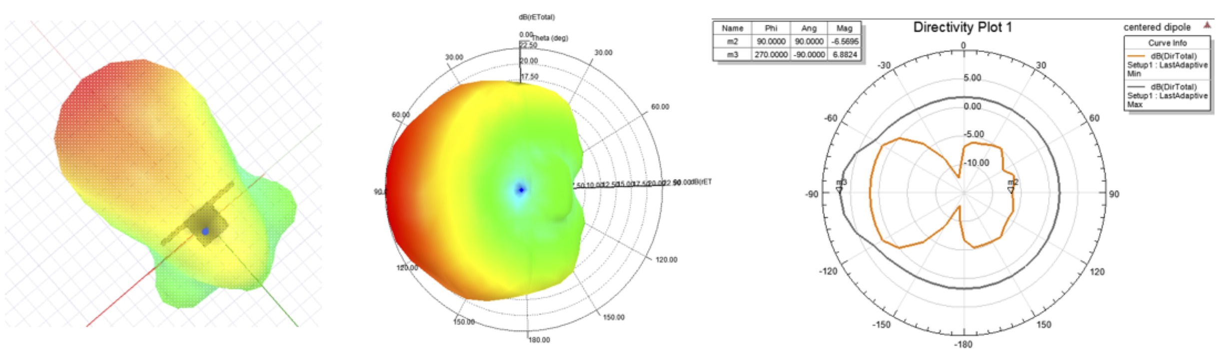

Figure 15: HFSS Simulation – 90° UHF Antenna Deployment

Figure 15 shows that the gain pattern for the 90° wing deployment is highly non-uniform. The top down view (left) shows that the gain is attenuated on the nadir facing side of the spacecraft and is increased on the side facing zenith. Furthermore, looking at the directivity plot (right), it shows that the gain is attenuated significantly in nearly all regions around the antenna such that the link margin can not be maintained in most regions. This simulation shows that a 90° deployment angle does not meet the link margin requirement regardless of orientation.

Thus, a series of design changes were considered and simulated to determine how to mitigate this issue. The changes considered include; re-orient the antenna dipole axis to face nadir, use a dual dipole or quadrupole, change deployment angle of the solar panels, and move the UHF antenna to the bottom of the spacecraft. Eventually, the design change which required the least overall spacecraft re-design while still meeting the link margin was to change the deployment angle of the spacecraft solar panels to 120°. Figure 16 shows the simulation results when the panels are deployed to 120°.

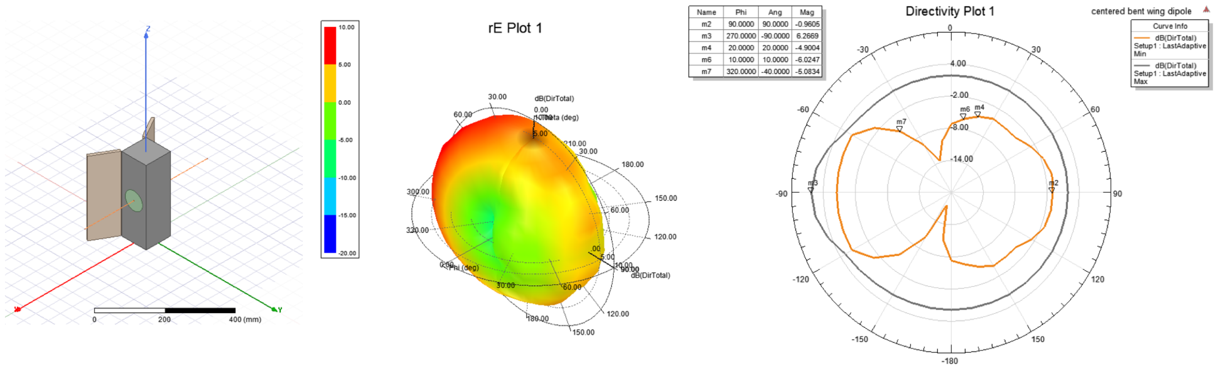

Figure 16: HFSS Simulation – 120° UHF Antenna Deployment

Figure 16 shows that the gain pattern (middle) for the new wing deployment angle of 120° is more uniform than 90°. The directivity plot (right) shows that the link margin required is maintained in almost all directions other than an angular region of ∼35° around the dipole axis. This is deemed to be acceptable to satisfy the omni-directional link requirement as even a perfect dipole will have these "null regions".

Deploying the solar panels further does cause a change in the power budget of the spacecraft. At the 90° deployment, when the spacecraft is in sun-charging mode, the maximum sun angle is achieved for all three panels (sun vector is normal the face of the panels). With the 120° configuration, sun-charging mode will take longer but this will be offset due to high power generation in other operating modes. For example, if the sun-vector is facing directly towards a 3U face, part of the deployable solar panel will be illuminated in the 120° case (whereas no part of the deployable would be illuminated in the 90° case).

D. Electrical Power System - EPS

The EPS subsystem is designed to be able to operate for at least the four month mission lifetime and be capable of operating continuously throughout all phases of orbit (sunlight, eclipse, etc.). These two requirements led to the selection of the GOMSpace BP4 battery pack and the GOMSpace P31us power distribution board. Figure 18 shows the power generation, eclipse times, and power bounds for a 1 year simulation of the spacecraft on orbit. For the worst case power draw and generation, the spacecraft requires two orbits of science followed by one orbit of power generation. However, depending on the orbit geometry and time of year, MicroNimbus can remain power positive throughout all portions of its orbit..

It was determined that the 3U structure with a total of 49 solar cells (requiring two single deployed solar panels off the 3U face of the CubeSat), would be able to sufficiently close the power budget. Furthermore, the battery depth-of-discharge (DoD) was set at 30% so that enough battery cycles can be achieved for the predicted 4 months of nominal operations. Based on this, operational modes for the mission were developed and can be seen in Figure 17.

Mode Definition	MEV Power Draw (W)	Definition						
		Attitude	CDH/EPS	ADC Actuation	ADC Sensors	UHF (RX & TX)	S-Band (TX)	Payload
Safe	2.46	Unknown Attitude	ON	OFF	OFF	Low Rate (Beacon)	OFF	OFF
Eclipse	3.46	~Nadir Pointing	ON	Reaction Wheels, TR	IMU	Nominal Rate (Beacon)	OFF	OFF
Charging	3.55	Max Sun Generation Attitude	ON	Reaction Wheels, TR	IMU, Sun Sensors	Nominal Rate (Beacon)	OFF	OFF
Downlink	12.91	Ground Station Pointing	ON	Reaction Wheels, TR	IMU, Sun Sensors, GPS	Max Rate	ON	OFF
Science	6.50	Nadir Pointing	ON	Reaction Wheels, TR	IMU, Sun Sensors, GPS	Nominal Rate (Beacon)	OFF	ON

Figure 17: MicroNimbus Operational Modes (Red, Yellow, and Green Correspond to Low, Medium, and High Duty Cycles)

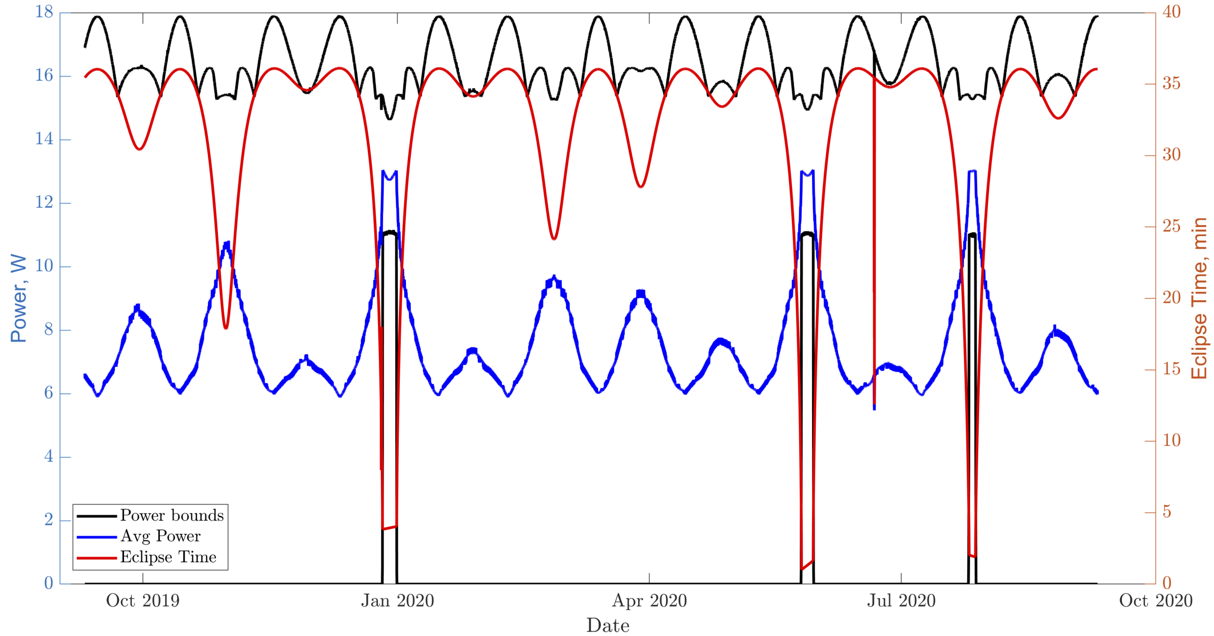

Figure 18: Average Power Generation, Eclipse Times, and Power Bounds For 1 Year

E. Structures - STR

1. Design

The overall structure of the spacecraft is designed specifically such that it not only meets the structural requirements set forth by most major CubeSat specifications (Cal Poly, NanoRacks, etc.), but also that it is easy to machine and integrate. For example, each module (ADC, service, payload) of the CubeSat is comprised of L-shells, each of which is created from individual blocks of aluminum. Then each module is joined together through the use of section connectors. This allows for schedule margin within the machining process to be smaller and poses less risk of large schedule delays for overall delivery of the structure[14]. Furthermore, because the satellite is inherently modular (seen in Figure 19) each module can be tested, during integration, and integrated individually without interfering with other modules. This once again helps alleviate schedule risk during integration. Fully integrated views of the spacecraft (including solar panels) can be seen in Figure 20, 21, 22, 23. Note that Figure 20 shows the body axis of the spacecraft.

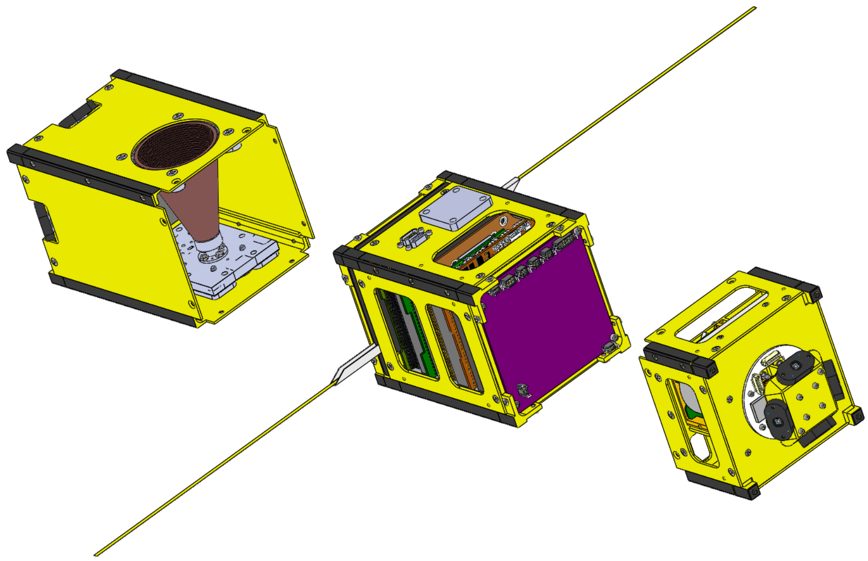

Figure 19: MicroNimbus Modules - Payload (Left), Service (Middle), and ADC (Right)

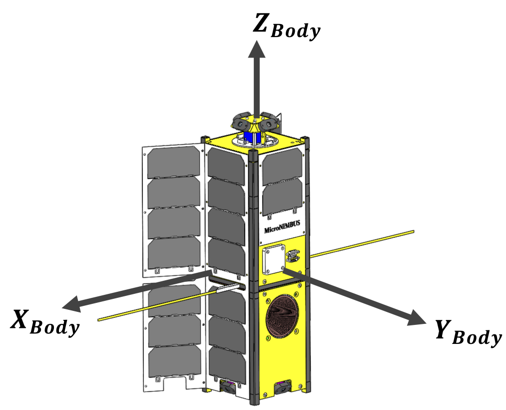

Figure 20: MicroNimbus - Integrated View 1

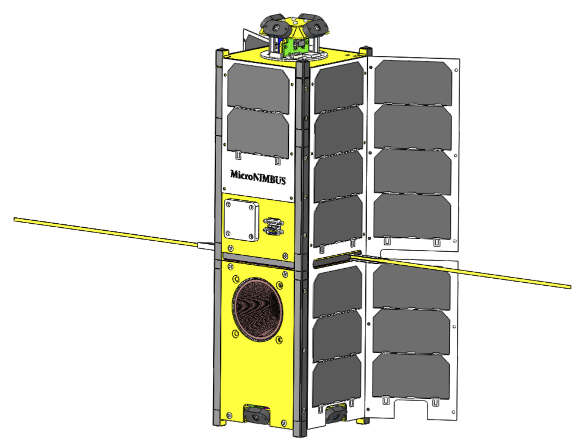

Figure 21: MicroNimbus - Integrated View 2

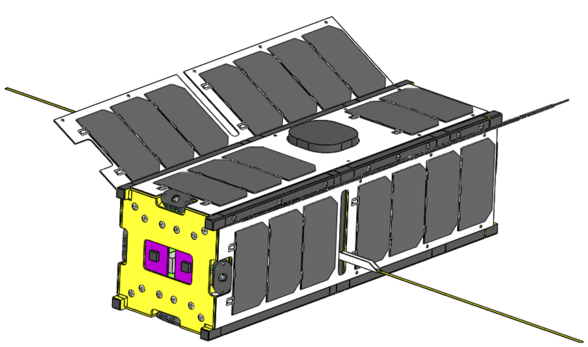

Figure 22: MicroNimbus - Integrated View 3

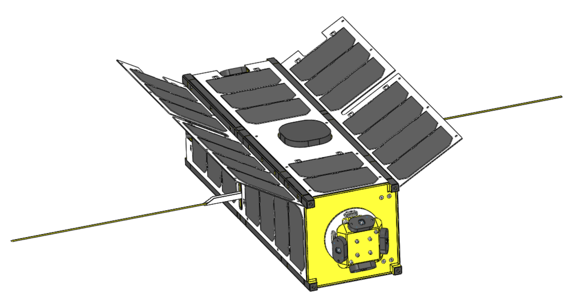

Figure 23: MicroNimbus - Integrated View 4

2. *Solar Panel Hinges*

Custom hinges have been developed for the deployment of the solar panels. The hinges deploy to 120°(rather than 90°) so that the UHF antenna gain pattern is minimally affected. These hinges use a burn-wire to keep the panels folded in. Once commanded, the spacecraft sends current through a resistor, which melts the burn-wire, and a torsion spring forces open the panels. In order to reduce risk, two redundant resistors are used per panel to melt the wire. This design has been tested in both air and vacuum and has been shown to successfully operate [23]. An image of the hinge test prototype can be seen in Figure 24 and a graphic representation of the burn wire deployment mechanism and tension system can be seen in Figure 25 and 26 respectively.

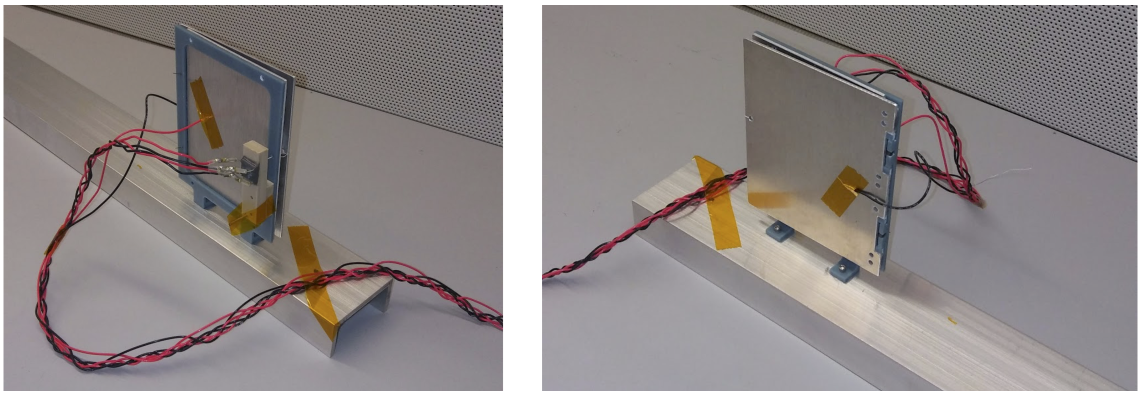

Figure 24: Integrated Test Panel Front (Left) and Back (Right)

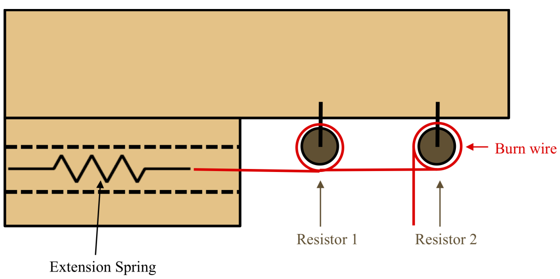

Figure 25: Solar Panel Release Mechanism

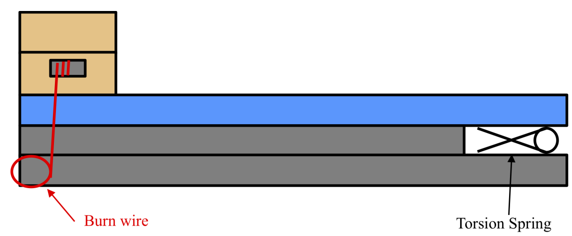

Figure 26: Solar Panel Tension System Diagram

To determine the time between sending a command to direct current through the resistor and solar panel deployment, a contact spring was used. The results from the first test campaign can be seen in Table 13. Note that R_1 and R_2 are each of the two redundant resistors and that deployment tests were conducted in air and vacuum independently.

Table 13: Measured Deployment Times For Solar Panel Burn Wire

	$\mathbf{R_1}$ – Air (s)	$\mathbf{R_2}$ – Air (s)	$\mathbf{R_1}$ – Vacuum (s)	$\mathbf{R_2}$ – Vacuum (s)
Test 1	3.80	3.87	3.61	3.45
Test 2	3.76	3.67	3.54	3.52
Test 3	3.62	3.67	3.43	3.61
Average	3.73	3.74	3.53	3.53

F. Navigation - NAV

1. *Hardware*

The navigation subsystem of the spacecraft consists of a GPS receiver and antenna which allows the spacecraft to precisely determine its inertial position and velocity states. The GPS receiver used is the NovAtel OEM615 − paired with an AntCom patch antenna. The antenna is placed on the zenith side of the spacecraft, allowing for greatest visibility for GPS satellites during science operations. Additionally, the NAV subsystem plays a critical role for both the spacecraft and payload. First, for the overall science mission, the position of the spacecraft relative to the Earth (in addition to the attitude) is critical in determining which part of the atmosphere the instrument is sounding. Secondly, the GPS receiver plays a critical role for the radiometer payload itself. The radiometer requires a well-disciplined 10 MHz signal for both the SiGe integrated receiver front end and the down-mixer. However, since the GPS receiver used on the satellite (NovAtel OEM615) only provides a 1 Pulse-Per-Second (PPS) disciplined signal, the payload interface board is used to convert this signal into the required 10 MHz signal. The payload interface board sits in-between the service module and the payload module, and is used to place any communication interfaces between the payload and the flight computer.

2. *De-Orbit Analysis*

In order to comply with NASA standards on orbital debris, an Orbital Debris Assessment Report (ODAR) is required. At the current stage of the mission (with no current launch manifest), a formal ODAR report is not required. However, a preliminary orbital debris assessment was created [24] using the required NASA DAS [25] software package. In its nominal configuration, the maximum expected lifetime of the mission before rapid de-orbit due to drag is between 1.8 and 3.6 years depending on the average orientation of the spacecraft throughout the operational lifetime. These values fall well below the NASA requirement of 25 years and thus are deemed acceptable. A plot of the nominal MicroNimbus orbit history using DAS' most conservative lifetime estimation method can be seen in Figure 27.

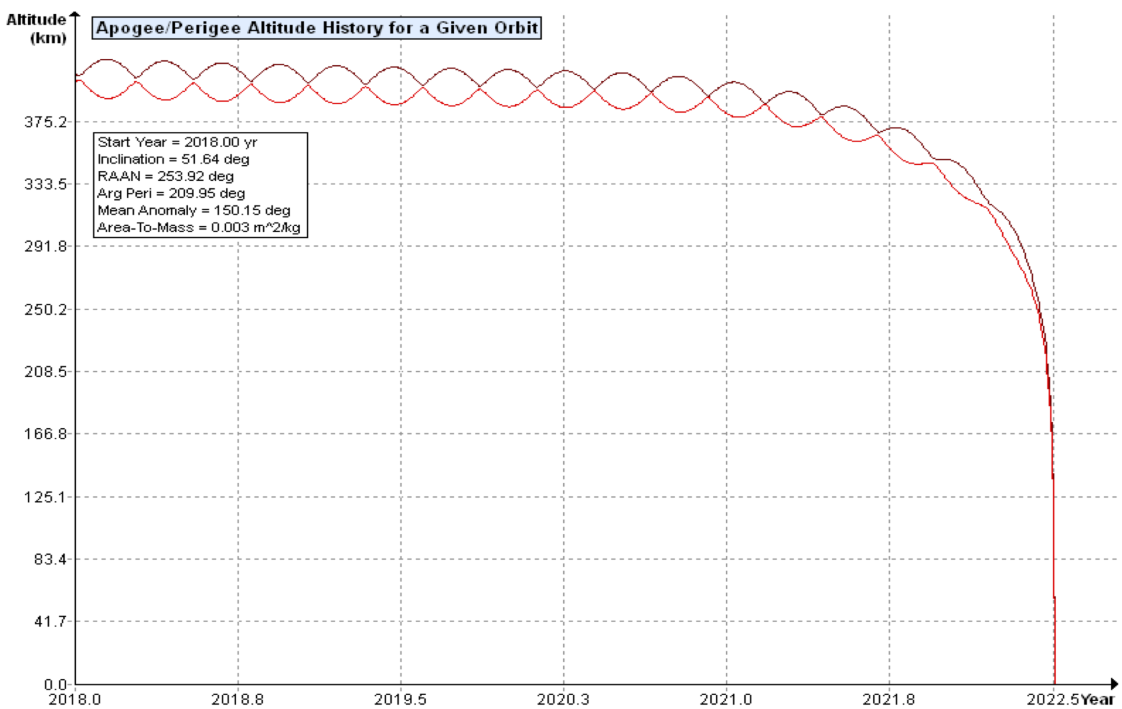

Figure 27: Nominal MicroNimbus Orbit History − Calculated Using DAS A/M

VII. Systems Engineering

A. System Budgets

1. Data Budget

This budget is obtained by first determining the size of the following packets; health, ADC, ADC-mission, and science. The health packets are comprised of telemetry (temperature, voltages, currents, etc.) from each of the spacecraft subsystems. The standard ADC packets are both raw and processed sensor values from each of the attitude determination sensors. The ADC-mission packets consists of only science critical attitude pointing information needed as reference for the science data. Finally, the science data packets are raw measurement and calibration values from the radiometer. Although the HDR system is nominally used only for science and ADC-mission data downlink, the data budget shown includes the standard ADC data and the health packets in the event of a temporary failure in the LDR system.

The data budget for the HDR radio can be seen in Figure 28. This budget shows that over the course of an entire day, the total amount of data generated from all packets is roughly 28 MB. The HDR radio selected for this mission has the capability to downlink at a data rate of 3.4 Mbps. However, this does not translate completely to data throughput. The data throughput is approximated to be 50% of the full data rate – this conservative estimate accounts for the bit overhead during transmission (start/stop bits, encoding, encryption, etc.) This translates to a true downlink rate of 0.25 MB/s. With a conservative estimate of two ground station passes a day totaling five minutes each, the amount of data which can be downlinked per day is 150 MB – far greater than the required 28 MB.

MicroNimbus Data Budget				
	Health	**ADC**	**ADC-Mission**	**Science**
Bytes per sample	294	129	17	4
Samples per hour	360	360	3,600	360,000
Mission duty cycle	100%	100%	75%	75%
Est. Compression	0.3	0.6	0.6	1.0
Bytes/Day	762,048	668,736	660,960	25,920,000
			Total Gen. Data (Bytes per day)	**28,011,744**
			Downlink rate (Bps)	250,000
			Seconds per pass	300
			Bytes per pass	75,000,000
			Passes per day	2
			Bytes per day	150,000,000

Figure 28: MicroNimbus Data Budget – HDR

2. Power Budget

The power budget can be seen in Figure 29. This budget details the net power use of the spacecraft during each of the various operational modes as listed in Section II. The budget is built up by tallying the maximum expected value (MEV) power draw of each component in the spacecraft, along with component and system level margin. The component level margins are determined based on a confidence assessment of the potential of the component to increase its power draw during final assembly and testing. The system margin is a flat 10% to provide additional margin. Based on the MEV power consumption and the orbit average power (OAP) power production, final margin percentages are generated for each of the modes. Safe, charging, and eclipse mode are power positive and downlink and science mode are power negative. The amount of "negative margin" that downlink and science mode have dictates how much science the mission can perform before a charging mode must be initiated to keep the batteries above the desired depth of discharge.

			Safe Mode		Charging Mode		Downlink Mode		Science Mode		Eclipse Mode		Detumble	
Component	**QTY**	**Cont.**	**Duty Cycle**	**CBE Power Draw (W)**	**Duty Cycle**	**CBE Power Draw (W)**	**Duty Cycle**	**CBE Power Draw (W)**	**Duty Cycle**	**CBE Power Draw (W)**	**Duty Cycle**	**CBE Power Draw (W)**	**Duty Cycle**	**CBE Power Draw (W)**
1.0 CDH														
1.1 Flight Computer (ARM+FPGA)	1	30%	1	2.00	1	2.10	1	2.10	1	2.10	1	2.10	1	2.10
1.2 CDH Carrier Board	1	30%	1	0.42	1	0.42	1	0.42	1	0.42	1	0.42	1	0.42
1.3 Payload Interface Board	1	100%	0	0.00	0	0.17	1	0.17	1	0.17	0	0.17	0	0.17
2.0 ADC														
2.1 Actuators														
2.1.1 10 mNm-s Reaction Wheels	3	10%	0	0.00	1	0.11	1	0.11	1	0.11	1	0.11	0	0.11
2.1.2 X & Y Torque Rods	2	5%	0	0.00	0.05	0.47	0	0.47	0.05	0.47	0.05	0.47	1	0.47
2.1.3 Z-axis Torque Rod	1	5%	0	0.00	0.05	0.47	0	0.47	0.05	0.47	0.05	0.47	1	0.47
2.1.4 Torque Rod PCB	1	0%	0	0.00	0.05	0.00	0	0.00	0.05	0.00	0.05	0.00	1	0.00
2.2 Sensors														
2.2.1 Epson G364PDCA IMU	1	5%	0	0.00	1	0.05	1	0.06	1	0.06	1	0.05	1	0.05
2.2.2 SolarMEMS Sun Sensors	8	10%	0	0.00	1	0.01	1	0.01	1	0.01	0	0.01	0	0.01
2.2.3 NovAtel OEM615 GPS	1	5%	0	0.13	0	1.23	1	1.23	1	1.23	0	1.23	0	1.23
2.2.4 Antcom GPS Antenna	1	30%	0	0.00	0	0.05	1	0.05	1	0.05	0	0.05	0	0.05
2.2.5 ADC Sun Sensor PCB	1	30%	0	0.00	1	0.01	1	0.01	1	0.01	0	0.01	0	0.01
2.2.6 Payload Sun Sensor PCB	1	30%	0	0.00	1	0.01	2	0.01	2	0.01	1	0.01	0	0.01
2.2.7 Honeywell HMC 1053 Magnetome	1	5%	0	0.01	1	0.01	1	0.01	1	0.01	0	0.01	1	0.01
2.3 ADC Interface Board	1	100%	0	0.03	1	0.03	1	0.07	1	0.07	1	0.03	1	0.03
2.4 ADC Sliver Board	1	0%	0	0.00	1	0.00	1	0.00	1	0.00	1	0.00	1	0.00
3.0 LDR Communications		0%												
3.1 GomSpace AX100 - RX	1	10%	1	0.15	1	0.26	1	0.40	1	0.26	1	0.26	1	0.26
3.2 GomSpace AX100 - TX	1	10%	0.005	2.48	0.01	2.64	1	2.81	0.01	2.64	0.01	2.64	0.01	2.64
3.3 ISIS Deployable Antenna	1	10%	1	0.04	1	0.04	1	0.04	1	0.04	1	0.04	1	0.04
4.0 HDR Communications		0%												
4.1 ClydeSpace/CPUT STX-C	1	5%	0	0.11	0	0.60	1	3.76	0	3.76	0	0.60	0	0.60
5.0 EPS		0%												
5.1 GomSpace P31us	1	10%	1	0.26	1	0.26	1	0.26	1	0.26	1	0.26	1	0.26
5.2 BP4 Battery Pack - Heater	1	10%	0	0.00	0	3.50	0	7.00	0	7.00	0	7.00	0	7.00
5.3 BP4 Battery Pack - Power Switch	1	0%	0	0.00	0	0.00	0	0.00	0	0.00	0	0.00	0	0.00
6.0 Payload	1	50%	0	0.00	0	0.61	0	0.61	1	0.61	0	0.61	0	0.61
MEV Power Consumption (W)			**3.6**		**4.6**		**13.4**		**7.3**		**4.5**		**5.5**	
System Level Contingency			**5%**		**5%**		**5%**		**5%**		**5%**		**5%**	
Orbit Average Power Production (W)			**6.0**		**6.0**		**6.0**		**6.0**		**6.0**		**6.0**	
Total MEV Power Consumption			**3.82**		**4.79**		**14.10**		**7.64**		**4.71**		**5.79**	
Margin			**57%**		**25%**		**-57%**		**-21%**		**28%**		**4%**	

Figure 29: MicroNimbus Power Mode Budget

3. Link Budget

Link budgets are used to verify that the spacecraft to ground and ground to spacecraft links can be established and maintained during a ground station pass. Since MicroNimbus has two modes of communications, a LDR and HDR system, two separate link budgets are created. The link budgets take into account aspects of both ground and space communications parameters such as spacecraft antenna gain, free space path loss, and ground station antenna gain. Both traditional methods of computing gain margin for satellite downlink are implemented as seen in `\16043048-MicroNimbus\Documents\SYS\bdg`; Eb/No and Spacecraft Alternative Signal Analysis Method (SNR). These budgets can be seen in Tables 14 and 15 for LDR and Tables 16 and 17 for HDR respectively. The link budgets show that both the LDR and HDR systems meet the 3 dB link margin requirement.

Table 14: UHF Link Budget - Eb/No

Parameter	**Value**	**Units**
G.S. Antenna Pointing Loss	0.6	dB
G.S. Antenna Gain	18.9	dBi
G.S. Transmission Line Loss	1.8	dB
G.S. Effective Noise Temp	1003	K
G.S. Figure of Merit (G/T)	-13.0	dB/K
G.S. SNR Power Density (S/No)	53.4	dBHz
Desired Data Rate	9.6	kbps
Telemetry System Eb/No	13.5	dB
Demod. Implementation Loss	0	dB
Telemetry System Req. Eb/No	9.6	dB
Eb/No Threshold	9.6	dB
System Link Margin	3.9	dB

Table 15: UHF Link Budget - SNR

Parameter	**Value**	**Units**
G.S. Antenna Pointing Loss	0.6	dB
G.S. Antenna Gain	18.9	dBi
G.S. Transmission Line Loss	1.8	dB
G.S. Effective Noise Temp	1003	K
G.S. Figure of Merit (G/T)	-13.0	dB/K
Signal Power at G.S. LNA Input	-145.2	dBW
G.S. Receiver Bandwidth	9.6	kHz
G.S. Receiver Noise Power	-158.8	dBW
SNR Power Ratio at G.S. Reveiver	13.5	dB
Required S/N	9.6	dB
System Link Margin	3.9	dB

Table 16: S-band Link Budget - Eb/No

Parameter	Value	Units
G.S. Antenna Pointing Loss	0.0	dB
G.S. Antenna Gain	37.0	dBi
G.S. Transmission Line Loss	2.5	dB
G.S. Effective Noise Temp	125	K
G.S. Figure of Merit (G/T)	13.5	dB/K
G.S. SNR Power Density (S/No)	78.0	dBHz
Desired Data Rate	2	Mbps
Telemetry System Eb/No	14.9	dB
Demod. Implementation Loss	0	dB
Telemetry System Req. Eb/No	9.6	dB
Eb/No Threshold	9.6	dB
System Link Margin	5.3	dB

Table 17: S-band Link Budget - SNR

Parameter	Value	Units
G.S. Antenna Pointing Loss	0.0	dB
G.S. Antenna Gain	37.0	dBi
G.S. Transmission Line Loss	2.5	dB
G.S. Effective Noise Temp	125	K
G.S. Figure of Merit (G/T)	13.5	dB/K
Signal Power at G.S. LNA Input	-129.7	dBW
G.S. Receiver Bandwidth	2	MHz
G.S. Receiver Noise Power	-144.6	dBW
SNR Power Ratio at G.S. Reveiver	14.9	dB
Required S/N	9.6	dB
System Link Margin	5.3	dB

4. Mass Budget

The mass budget for MicroNimbus is seen in Figure 30. This budget is created from a "bottom up" approach to mass estimation. To obtain this estimate, the mass of each component within a subsystem is tallied up. Each of these components receives a current best estimate (CBE) value in the table along with a contingency added to it. The contingency is chosen based on how well each component's mass is known. For example, if a flight equivalent component is available to weigh, then the lowest contingency value (2%) is assigned to it. For COTS components which have data sheets available, a moderate 5% contingency is given. For custom components that are still being designed or have not been fabricated, a contingency of 10% is given. While this is not as conservative as higher class missions it is reasonable for a CubeSat mission. The mission has experience in ordering and fabricating these custom components (PCBs for example). For fasteners and wiring, the estimate is taken from CAD estimates along with a significant contingency factor (∼50%) to account for additional conformal coating or staking which may be necessary during integration. Overall, the budget estimates that the spacecraft will have a worst case mass of 3.7844 kg, leaving a 5% contingency from the total mass allocation of 4 kg. Note that mass waivers are available for secondary payloads, providing additional margin if necessary.

5. Cost Budget

The Master Equipment List and associated hardware costs of MicroNimbus bus can be seen in Figure 31 below. This budget details the amount of money (in $USD) required to buy all components necessary to assemble the spacecraft in 2018. This budget does not include ancillary costs to the parts such as labor, facilities, overhead, etc. The budget has two main sets of columns − quantity and cost. Each of these columns are broken up into two sections; one for only the flight unit, and the other for all units (EDU, flight, and flight spare). When totaled together, the mission costs are as follows; ∼$170k for the flight unit, and ∼$341k for all units. Note that contingency values (5%, 10%, 15%) are assigned to the cost of each spacecraft component based on confidence levels (quoted, moderate, and low respectively).

MicroNimbus Mass Budget						
		Level 2				Level 1
Component	**CBE per [g]**	**Quantity**	**CBE total [g]**	**Contingency**	**Allocated [g]**	**Total [g]**
1.0 CDH						138.0
1.1 phyCORE Vybrid	25	1	25	5%	26.3	
1.2 CDH Carrier Board	75	1	75	5%	78.8	
1.3 Payload Interface Board	30	1	30	10%	33.0	
2.0 ADC						642.4
2.1 Actuators						
2.1.1 10 mNm-s Reaction Wheels	123	3	369	2%	376.4	
2.1.2 X-, Y-axis Torque Rods	27	2	54	2%	55.1	
2.1.3 Z-axis Torque Rod	76.22	1	76.22	2%	77.7	
2.1.4 Torque Rod PCB	5	3	15	10%	16.5	
2.2 Sensors						
2.2.1 Epson IMU	10.5	1	10.5	5%	11.0	
2.2.2 SolarMEMS Sun Sensors	3.7	8	29.6	2%	30.2	
2.2.5 ADC Sun Sensor PCB	7.16	1	7.16	2%	7.3	
2.2.6 Payload Sun Sensor PCB	15	1	15	10%	16.5	
2.2.7 Magnetometer PCB	5	2	10	10%	11.0	
2.3 ADC Interface Board	27.32	1	27.32	2%	27.9	
2.4 ADC Sliver Board	12.52	1	12.52	2%	12.8	
3.0 NAV						84.9
3.1 NovAtel OEM615 GPS	21.5	1	21.5	2%	21.9	
3.2 Antcom GPS Antenna	60	1	60	5%	63.0	
4.0 LDR Communications						189.0
4.1 GomSpace AX100	24.5	1	24.5	2%	25.0	
4.2 ISIS Deployable Antenna	86.5	1	86.5	10%	95.2	
4.3 GomSpace NanoDock	67.5	1	67.5	2%	68.9	
5.0 HDR Communications						109.3
5.1 ClydeSpace/CPUT STX-C	84.5	1	84.5	2%	86.2	
5.2 Haigh-Farr S-band Antenna	22	1	22	5%	23.1	
6.0 EPS						1195.4
6.1 GomSpace P31us	98.5	1	98.5	2%	100.5	
6.2 BP4 Battery Pack	262	1	262	2%	267.2	
6.3 Solar Panels (Body Mounted)	733.88	1	733.88	10%	807.3	
6.4 Inhibit Switches (4)	5	4	20	2%	20.4	
7.0 Structure						700.4
7.1 ADC Structure	241.5	1	241.5	5%	253.6	
7.2 Service Structure	188	1	188	5%	197.4	
7.3. Payload Structure	237.5	1	237.5	5%	249.4	
8.0 Fastener/Wiring						240.3
9.0 Payload						482.9
9.1 Radiometer	439.00	1	439.00	10%	482.90	
					Total Mass:	**3782.5**
					Mass Limit:	**4000.0**
					Margin:	**5.44%**

Figure 30: MicroNimbus Mass Budget

Subsystem	Component	Cost	Confidence	Contingency	Quantity		Cost ($USD)	
					Flight Unit	All Units	3U	All Units
CDH	Cyclone V5 FPGA	$1,800.00	Quoted	5%	1	4	$1,890.00	$7,560.00
	CDH Carrier Board	$2,000.00	Low	15%	2	4	$4,600.00	$9,200.00
	Payload Interface Board	$500.00	Moderate	10%	2	4	$1,100.00	$2,200.00
ADC	Sinclair 10 mNms Rxn Wheels	$20,000.00	Quoted	5%	3	6	$63,000.00	$126,000.00
	Torque Rod	$250.00	Quoted	5%	3	6	$787.50	$1,575.00
	Torque Rod PCB	$25.00	Quoted	5%	3	6	$78.75	$157.50
	Honeywell Magnetometer	$200.00	Quoted	5%	2	6	$420.00	$1,260.00
	Epson IMU	$1,000.00	Quoted	5%	1	2	$1,050.00	$2,100.00
	Solar MEMS Analog Sun Sensor	$2,100.00	Quoted	5%	8	16	$17,640.00	$35,280.00
	ADC Interface Board	$500.00	Moderate	10%	2	4	$1,100.00	$2,200.00
	ADC Sliver Board	$40.00	Moderate	10%	2	4	$88.00	$176.00
	Sun Sensor Interface Board - ADC	$500.00	Moderate	10%	2	4	$1,100.00	$2,200.00
	Sun Sensor Interface Board - PAY	$500.00	Moderate	10%	2	4	$1,100.00	$2,200.00
STR	3U Structure	$7,500.00	Moderate	10%	1	2	$8,250.00	$16,500.00
EPS	P31us EPS Board	$6,000.00	Quoted	5%	1	2	$6,300.00	$12,600.00
	BP4 Batteries	$2,700.00	Quoted	5%	1	2	$2,835.00	$5,670.00
	Solar Panel PCBs	$400.00	Moderate	10%	4	8	$1,760.00	$3,520.00
	Solar Cells	$275.00	Quoted	5%	50	100	$14,437.50	$28,875.00
	Inhibit Switches	$10.00	Moderate	10%	4	10	$44.00	$110.00
COM	NanoCom AX100 UHF	$5,500.00	Quoted	5%	1	2	$5,775.00	$11,550.00
	ClydeSpace STXC S-band	$8,900.00	Quoted	5%	1	2	$9,345.00	$18,690.00
	ISIS Deployable UHF Antenna	$5,000.00	Quoted	5%	1	2	$5,250.00	$10,500.00
	Haigh-Farr S-band Patch antenna	$850.00	Quoted	5%	1	2	$892.50	$1,785.00
NAV	NovAtel OEM615 Receiver	$9,500.00	Quoted	5%	1	2	$9,975.00	$19,950.00
	Antcom Antenna	$441.00	Quoted	5%	1	3	$463.05	$1,389.15
GSE	GSE box and components	$300.00	Moderate	10%	1	2	$330.00	$660.00
	Computers/Monitors	$2,000.00	Moderate	10%	1	2	$2,200.00	$4,400.00
FS/W	All fasteners	$2,000.00	Moderate	10%	2	3	$4,400.00	$6,600.00
	All wiring	$2,000.00	Moderate	10%	2	3	$4,400.00	$6,600.00

Figure 31: MicroNimbus Cost Budget

B. Risk

1. *Likelihood & Consequence Definition*

The likelihood that an event occurs that creates a consequence to the mission is defined by the likelihood probability scale. This scale can be seen in Table 18. While this scale can be chosen at will, these specific levels of probabilities were deemed appropriate due to the known risks associated with small satellite university missions. For example, student turn over will almost certainly occur, thus the probability space should account for this event occurring (P(event) $\geq$ 90%).

Table 18: Likelihood Probability Definition

Likelihood	Value	Probability
Certain	5	P(event) $\geq$ 90%
Likely	4	50% $<$ P(event) $<$ 90%
Moderate	3	10% $<$ P(event) $<$ 50%
Unlikely	2	3% $<$ P(event) $<$ 10%
Rare	1	P(event) $\leq$ 3%

Next, the a scale establishing the severity of the consequences to the mission is developed and can be seen in Table 19. Each level of severity corresponds to an impact on the mission that is related to its technical, schedule, or cost aspects. For example, a consequence deemed to be of "Level 3" would produce a moderate reduction in technical performance, a schedule slip on the order of 3 months, and a cost increase of no more than $50k.

Table 19: Consequence Definition

Level	Technical	Schedule	Cost
1	Minimal to no consequence to technical performance	Minimal or no impact	Minimal to no impact
2	Minor reduction in technical performance or supportability, can be tolerated with little or no impact to program	Able to meet key dates. Slip $<$ 1 month	Budget increase or unit production cost increases $\leq$ $10k
3	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 $<$ 3 months.	Budget increase or unit production cost increases $\leq$ $50k
4	Significant degradation in technical performance or major shortfall in supportability; may jeopardize program success	Program critical path affected. Slip $<$ 6 months.	Budget increase or unit production cost increases $\leq$ $100k
5	Severe degradation in technical performance; cannot meet key technical/supportability threshoold; will jeopardize program success	Cannot meet key program milestones. Slip $>$ 6 months	Budget increase or unit production cost increases $\geq$ $100k

2. *Selected Risks and Mitigation*

Numerous risks exist which span aspects of the mission such as personnel, schedule, spacecraft, etc. While not all risks can be discussed in this paper due to brevity, some risks are important enough to worth mentioning. The full list of selected risks and mitigation techniques can be found on the server under `\16043048-MicroNimbus\Documents\SYS\top\sys-top-Risk.xlsx`

The first, and most obvious, personnel risk (PE-1) is having insufficient and/or unqualified personnel. Two root causes exist for this risk. The first is that both undergraduate and graduate students have personal priorities for graduation and this can cause high turnover rates. The second is that urgent project deadlines may not give time for new students to receive proper training. The first cause is given first priority with a likelihood of 5 and a consequence of 3. To mitigate this risk, the mitigation technique implemented is to control the risk. Controlling the risk will require training undergraduate and graduate students throughout the project lifecycle and keep extensive documentation. Implementing these mitigation techniques causes the consequence to drop to 2, while the likelihood remains at 5.

One significant schedule risk (SC-1) is a failure to pass critical milestones due to schedule delays. The first root cause for this risk are delays during integration and testing due to COTS components being discontinued or unsupported by the original vendor. The second root cause is additional manufacturing time due to a lack of stringent monitoring of hardware interfaces between the bus and payload. The first cause is given the highest priority and carries a likelihood of 3 and a consequence of 4. Two mitigation techniques are implemented to reduce this risk. The first is to control the risk by building in schedule margin and start embedded systems development early. The second technique is to simply assume the risk and change vendor for acquiring parts. With these mitigation techniques implemented, the likelihood drops to 2, while the consequence remains at 4.

The components of the spacecraft each carry certain risks. Specifically, the solar panel deployment mechanism poses a risk (SP-3) due to two main root causes. The first cause is that the resistor used to heat and melt the burn-wire fails to heat up. The second is that the torsion spring fails to push open the solar panels. The first cause is given priority with a likelihood of 4 and consequence of 5. To mitigate the risk, the first cause will be controlled by using redundant burn wire resistors along with an extensive ground test campaign in a thermal vacuum chamber. Implementing these mitigation techniques changes the likelihood to 3, while consequence remains at a 5. A summary of risks with both unmitigated and mitigated likelihood/consequences can be seen in Figure 32.

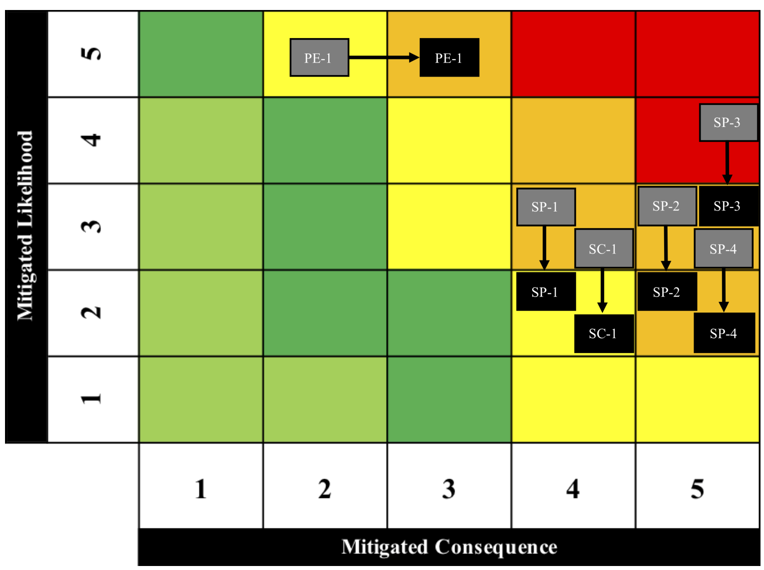

Figure 32: MicroNimbus Risk Matrix

C. Schedule

The overall schedule for the MicroNimbus mission can be seen in Figure 33. This schedule details the major milestones that the mission has gone through and which milestones are upcoming. For example, MicroNimbus passed its Preliminary Design Review (PDR) in April 2017 and is now heading towards the Critical Design Review (CDR). Note that if all major milestones are met for the mission, the mission could launch in the late 2019 to early 2020 timeframe (assuming the mission is manifested), with operations lasting roughly 6 months afterwards.

		Summer 2016	Fall 2016	Spring 2017	Spring 2018	Spring 2019	Fall 2020	Summer 2020	Fall 2020
	Activity*	**JUN**	**SEP**	**APR**	**MAY**	**DEC**	**JAN**	**JUN**	**OCT**
PRE-PHASE	***Conceptual Study***								
	MicroNimbus Documentation Review (*)	***							
PHASE A	System Requirements Review (*)		***						
	System Design Review (if needed)		***						
PHASE B	***Definition***								
	Preliminary Design Review (*)			***					
PHASE C	***EDU Design & Fabrication & Testing***								
	Critical Design Review (*)				***				
PHASE D	Deliver Spacecraft (*)					***	***		
	Flight Readiness Review (*)					***	***		
PHASE E	***Operations***								
	Launch							*	
	Post-Flight Assessment Review (*)								*
	End-of-Mission								*
PHASE F	***Closeout / De-orbit***								
	Data Archive								
	Decomission								
	Publish paper								

Figure 33: MicroNimbus − Schedule

D. Operations

1. *Ground Station Network*

Georgia Tech currently has three ground stations available for full time use to conduct bi-directional satellite communications. Two of the ground stations operate in the UHF (430 – 440 MHz) and VHF (144 – 146 MHz) bands and the third is S-band (2200 – 2310 MHz) [26]. All three stations employ software defined radios, allowing them to support different RF modulation schemes and bandwidths without needing to switch hardware. This approach allows new space missions to be added without disrupting the operations of missions in flight – thus creating a robust ground station network and reducing overall risk for each mission and workload for spacecraft operators.

2. *Operations Plan*

Along with the ground station network, Georgia Tech has built a strong interdisciplinary ground station operations team. Members consist of full-time faculty and research engineers, students from multiple engineering disciplines (aerospace, mechanical, electrical, computer, etc.) and members of the university amateur radio club. Training procedures are implemented by senior members to ensure that new members can successfully conduct operations. These procedures have been vetted through on-orbit spacecraft operations and practice. For example, initial checkout and operations of the LightSail-A spacecraft was completed using the Georgia Tech ground station. The procedure to send commands to spacecraft in orbit goes through a series of verifications to reduce risk and ensure that no fault scenarios are triggered. This process is outlined in Figure 34 (red arrows are satellite commands, purple arrows are rotor commands, and cyan arrows are downlinked data).

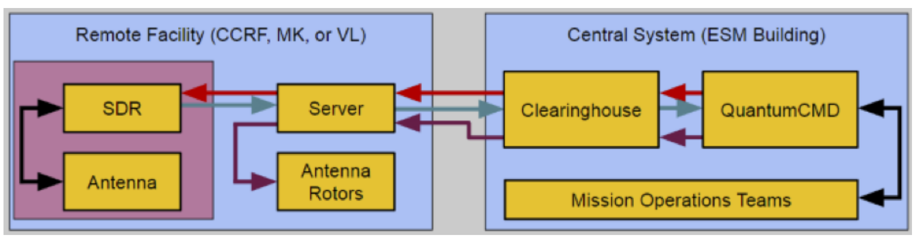

Figure 34: Ground Station – Flow Diagram

Command scripts are generated in advance by a minimum of two trained ground station operators for a given satellite pass. These scripts are processed using a ground station operations software. One option is to use QuantumCMD, a professional satellite mission operations software tool. The commands are tested on an Engineering Development Unit (EDU) version of the spacecraft, an equivalent bench setup that behaves like the on-orbit spacecraft in terms of operational behavior and response. This allows mission operators to determine if the command has the desired effect and if all subsystems are responding as expected. The EDU command execution is verified by both operators and then the command is deemed ready for uplink to the spacecraft. The operations software then forwards the commands to a central routing server known as the Clearinghouse. The Clearinghouse first verifies the identity of the operator and then, when the satellite pass occurs, opens a stream between the operations software and the local ground station server. The stream simultaneously transmits the command packets to the remote facility and the remote facility records all downlinked data along with parameters such as orientation of the ground station antennas during the pass. The process of propagating the satellite trajectories and pointing the ground station during the pass is performed autonomously using software.

3. *State Flow Diagram*

On orbit operations for most spacecraft are, in general, complex. For MicroNimbus specifically, numerous physical and subsystem constraints exist which will dictate the operation state of MicroNimbus. Because of this web of complexity, a state flow diagram was created which determines what state MicroNimbus is in, given any set of conditions. The state flow diagram can be seen in Figure 35 and a legend for the diagram can be seen in Table 20.

Table 20: State Flow Diagram Legend

Type	**Color**	**Definition**
Line	Black (solid)	Nominal Path
Line	Black (dashed)	Component Required for Mode
Line	Red (solid)	Error Path
Box	White	Event Occurance
Box	Dark Gray	Components
Box	Blue	Mode
Box	Cyan	Background Process
Diamond	Black	Logical Operator

While the flow diagram makes it easy to trace through each portion of the mission, some important subtleties should be noted. First, regardless of the method by which nominal operations is entered, the "Error Mask" must be updated. The error mask allows the flight software to recognize and update the error states of all components from a ground station command. For example, if a sun sensor is faulty and no longer transmits its data to the flight computer, the ground station can send a command for the flight software to update its error mask and no longer flag the faulty sun sensor as being in an error state. Second, note that the "Fault Mitigation Tree" event which occurs in the Off-Nominal Operations bounding area is treated as a black box function for the purposes of brevity. In reality, this tree is extensive and could require the ground station to help resolve the fault.

4. Data Downlink and Archival Plan

After the spacecraft data have been received at a specific ground station, multiple fail-safes have been implemented to ensure that satellite telemetry and payload data are secured and archived. First, while the stream between the Clearinghouse and local server is open, the local server stores both the raw data and meta-data (start and end of packets, etc.) received from the spacecraft. The local ground station has no method to process this raw data and it is stored only temporarily as a backup (on the order of days). In parallel, the data incoming to the Clearinghouse is stored on the central secure server in both raw and processed form (decrypted, Doppler corrected, etc.). Level 0 telemetry processing is done through the mission operations software (QuantumCMD) and this data is stored for much longer periods of time ($>$ 1 year) but is still rolled over when storage runs low. Simultaneously, the mission operations software automatically takes the data from the Clearinghouse server and stores it (essentially permanently) on respective servers for the various satellite missions at Georgia Tech. Thus, there are three storage locations (local ground station server, Clearinghouse server, and mission specific server) with increasing periods of storage and post-processing which archive the satellite payload and telemetry data.

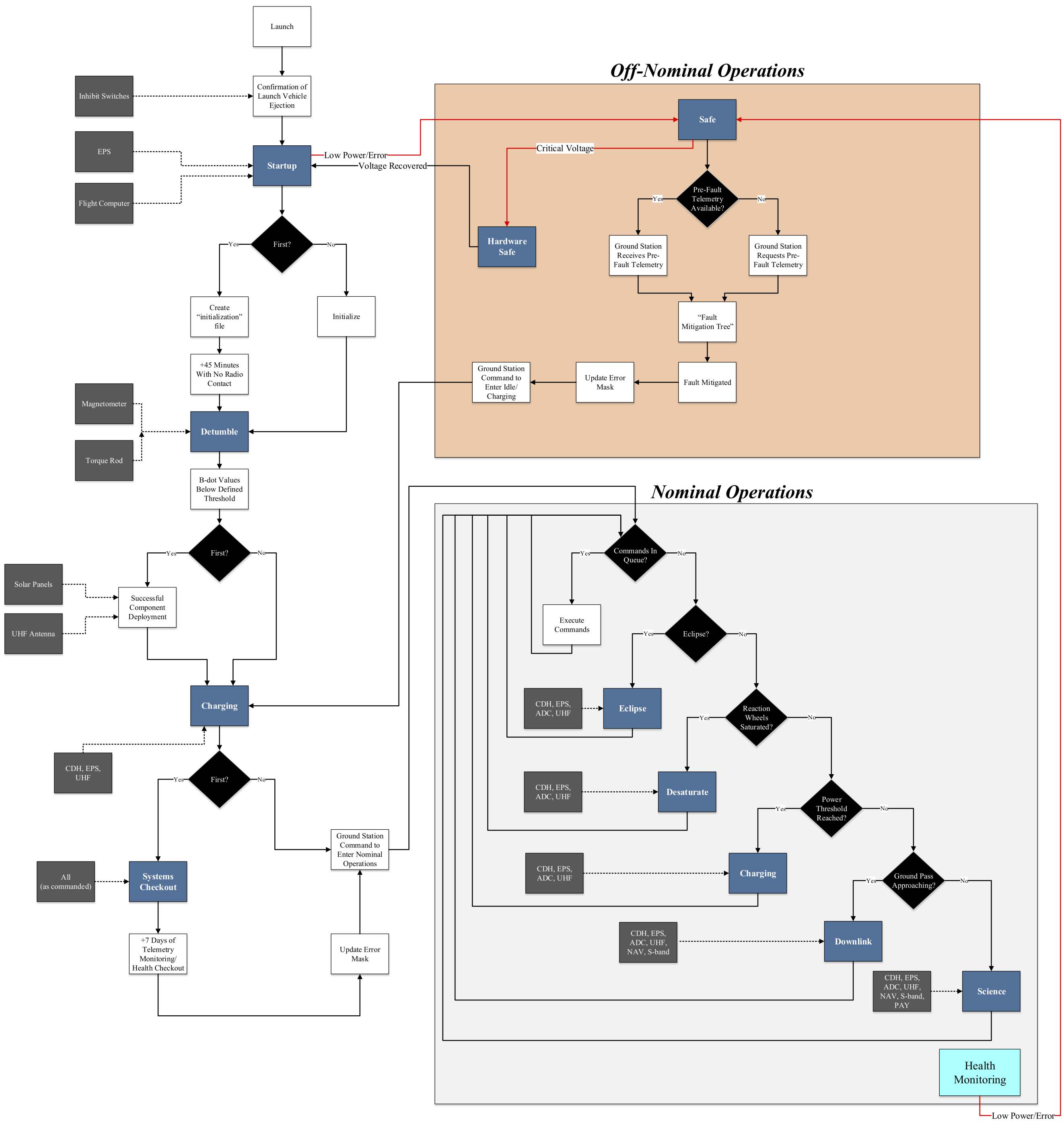

Figure 35: Operations State Flow Diagram

VIII. Conclusion

This paper presents a CDR level design of an Earth atmospheric remote sensing CubeSat mission known as MicroNimbus. The purpose of this mission is to use a frequency tunable mm-wave radiometer in order to measure and update the vertical temperature profile of the atmosphere. The 3U satellite is composed of a standardized bus design known as TECHBus which houses the payload. The radiometer payload itself is composed of an integrated SiGe radiometer receiver and a corrugated horn antenna that allows for the observation of the atmosphere in the ~60 GHz regime where O_2 absorption bands exist. Data gathered from this mission can help with weather forecasting and serves as a step towards the creation of a constellation of remote sensing CubeSats dedicated to near real-time global atmospheric temperature profiling.

IX. Ackowledgements

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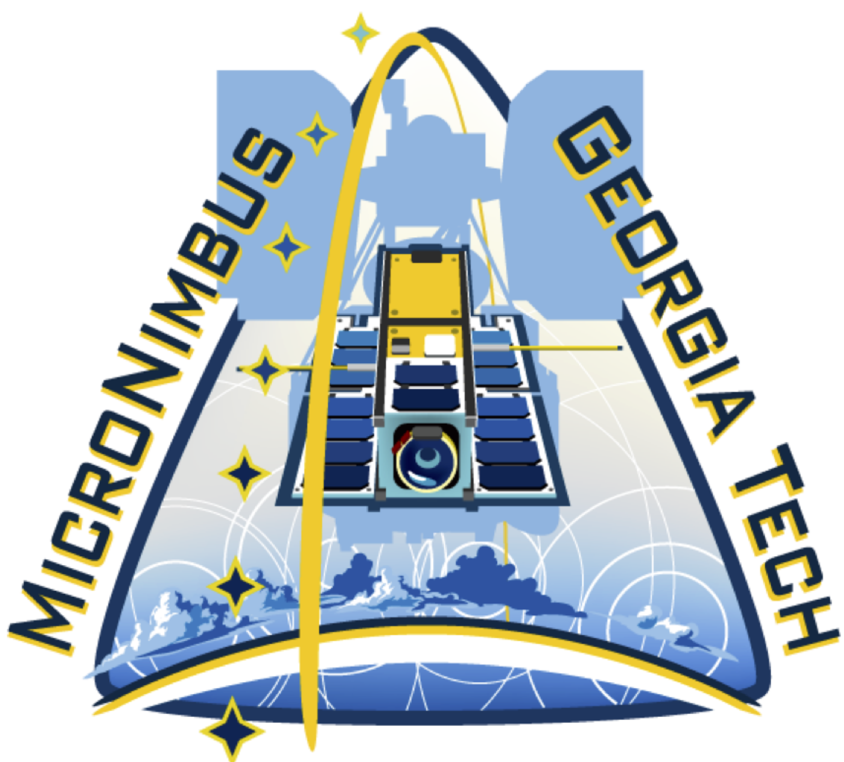

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