Mission Concept and Design for the Orbiting Aerosol Observatory

Frank Rutherford, Erica Jenson, Kieran Wilson
University of Florida
Gainesville, FL
frutherford2@ufl.edu

Faculty Advisor: Riccardo Bevilacqua
University of Florida

ABSTRACT

Preliminary design and analysis has been performed for a 6U CubeSat to carry a miniaturized aerosol polarimetry sensor in order to examine the effect of atmospheric aerosols on the Earth's climate This hypothetical satellite, the Orbiting Aerosol Observatory, is intended to perform partial mission objectives of the Glory spacecraft, which experienced a launch failure in 2011. The Orbiting Aerosol Observatory will collect data on the types and concentrations of aerosols in the atmosphere by observing incident sunlight reflected from the oceans from a position within the Afternoon Constellation. Subsystem requirements and component selection will be discussed. The Orbiting Aerosol Observatory consists primarily of off-the-shelf components with prior flight heritage to minimize cost, accelerate development, and maximize reliability. A series of simulations were created in MATLAB, Simulink, and Systems Tool Kit to model the satellite's operation in orbit and ensure propulsive, attitude determination and control, power, communications, and thermal systems could perform to the system requirements. Development, build, and test plans were created, and a budget was developed to project costs throughout the mission life cycle.

OVERVIEW

This paper outlines the design and analysis work conducted for a 6U CubeSat, the Orbiting Aerosol Observatory (OAO). This mission concept was proposed by Kennedy Space Center to perform partial mission objectives of the Glory spacecraft, which experienced a launch failure in 2011 [1]. Mission requirements and budgetary constraints were provided by Kennedy Space Center [1].

Mission

The mission is planned to operate on orbit for two years as part of NASA's earth observing Afternoon Constellation (A-Train). Orbiting in a sun synchronous trajectory, the OAO is intended to monitor the types and concentrations of aerosols around the world, with the goal of providing data to understand how aerosols influence climate systems [1]. This earth science observation mission fits within NASA's Technology Area (TA) 8.1, for remote sensing systems [2].

The sole payload on the satellite bus is the hypothetical miniaturized Aerosol Polarimetry Sensor (mAPS), a miniature version of the sensor designed to fly on the Glory mission. The sensor is designed to look at scattered sunlight reflecting off the surface of the ocean (sun glint) and use a polarimeter to extract spectral information that can be used to infer aerosol composition and concentration [1] [3].

System requirements

The mission requirements specified that the bus be a 6U CubeSat, comply with all specifications and requirements governing NASA missions as well as CubeSat standards and the requirements drafted to apply to the original APS [1] [3].

Concept of Operations

The OAO is scheduled to be deployed into an initial orbit 160° out of phase with the target position in the A-Train on January 1 2020 at 12:00 GMT [1]. Upon orbital insertion, the OAO will first de-tumble, neutralizing any initial rotational rates while the stowed solar panels begin generating power; this initialization phase is scheduled to last approximately 72 hours. Solar panel deployment will follow the de-tumble, and after the batteries are fully charged the electrospray thruster will be used to perform the required phasing maneuver.

Seven days after insertion, the OAO will perform the maneuvers required to calibrate the mAPS, after which the science mode can begin. Science mode, which will span the remainder of the two-year mission, consists of tracking the sunlight reflecting off the ocean's surface (the sun glint vector) with the mAPS whenever it is available and uploading data during each orbit as the OAO passes over the Thule Tracking Station in Greenland. Thruster firings will periodically be used to maintain position within the A-Train, and, at the end of the mission, the thruster will be used to place the OAO into a lower orbit to ensure rapid reentry.

ORBITAL MECHANICS AND PROPULSION

The OAO will be inserted into a nearly circular sun synchronous orbit with a semi major axis of 7071 km and an inclination of 98°, completing an orbit around the earth every 98.6 minutes and passing over the equator at the same local time every orbit (11 minutes behind the Aqua spacecraft) [1]. To determine attitude information for the spacecraft as it tracked the sun glint and performed the phasing maneuver, the orbit was modeled in AGI STK 11 and quaternions for periods of interest were exported to Simulink.

Three types of maneuvers will require propulsion during the mission: the initial phasing maneuver, station keeping maneuvers to compensate for drag and other perturbations, and the deorbit burn. The thruster selected is the Busek BET-1 electrospray thruster, with a 200 mL propellant reservoir. This thruster flew on the LISA mission, and uses an electrical field to accelerate an ionic liquid as a propellant, generating 0.7 mN of thrust while consuming 15 watts. Despite its low thrust, the total impulse is 2420 N-s (a total velocity change of 239 m/s), which provides large margin on expected requirements over the operational life of the OAO. The I_{sp} is 800 s, making it far more efficient than conventional cold gas or chemical thrusters, while generating larger thrusts for less power than ion engines [4]. Additionally, the 1U size and 1.15 kg mass fit well within the 6U frame.

Phasing Maneuver

The initial phasing maneuver is designed to place the OAO into its operational position in the A-Train, and must be completed before the calibration maneuver, which in turn must be completed within 7 days of deployment, as per mission requirements [1]. The phasing maneuver will consist of a 13,233 second burn of the thruster in the positive velocity direction, followed by a 180° rotation about the nadir, a three-day coast period, and an identical burn in the opposite direction to reestablish the initial orbital velocity. This continuous thrust phasing maneuver minimizes the time required to achieve operational position. Simulation and calculation of this burn were performed by modeling the orbit to account for the J2 perturbation in MATLAB and assuming a constant acceleration from the thruster. The effect of solar and lunar gravity as well as atmospheric effects were assumed to be negligible for the relatively short duration of the maneuver.

Station keeping

Over the two-year mission life, it is anticipated that atmospheric drag will result in a velocity decrease of 0.35 m/s, which, although small relative to the orbital velocity of the OAO, is sufficient to have the OAO lag its desired orbital position in the A-Train by over a full orbital period. This model was based on the MSISE-90 atmospheric profile, assuming mean solar activity, the projected frontal area of the satellite, and the drag coefficient of a cube. Based on these calculations, periodic thruster burns will be required to ensure that the OAO remains within 15 seconds of its desired position; such burns, based on the MSISE-90 atmospheric model and assuming mean solar activity, would need to occur every 74 days and require a thruster burn of 604 seconds to maintain the desired orbital position [5]. This could be done during the dark phase of the orbit to minimize disruption to the science mission. The precise frequency of burns can be determined on-orbit based on the actual rate of orbital decay, measured by the GPS.

Deorbiting

Atmospheric drag is insufficient to guarantee deorbit within 25 years as required by NASA [1] [6]. Therefore, after the completion of the primary science objectives, the remaining propellant on the spacecraft will be used to lower the orbit of the OAO so that it decays more rapidly. Based on the anticipated propellant consumption for phasing and station keeping, the OAO should have enough reserve to decrease the velocity by 140 m/s, which is sufficient to lower the orbit to a near-circular orbit less than 490 km in altitude. Below 490 km, the OAO is expected to re-enter within 24 years per STK simulations based on the projected area of the satellite and a drag coefficient of 2; any additional propellant available will further lower the orbit and significantly accelerate reentry.

ATTITUDE DETERMINATION AND CONTROL SYSTEM

The attitude determination and control system (ADACS) is responsible for de-tumbling after orbital injection and precisely orienting the satellite for the mAPS lunar calibration maneuver, tracking the sun-glint vector, and any other maneuver to orient the OAO for solar power generation, heat balancing, and communications with the ground station.

Attitude Determination and Control System Selection

There are several commercially available ADACS. A decision matrix was constructed comparing the MAI-400 from Maryland Aerospace Inc. and XACT from Blue Canyon Technologies. Both were comparable in the fields of cost, weight, and magnitude of magnetic dipole moment; however, the maximum torque and pointing accuracy of the XACT system better satisfy the OAO mission requirements. During the lunar calibration maneuver, the OAO must slew to a fraction of a degree to either side of the moon. The MAI-400 system is accurate to 0.1° along its axes, while the XACT system is almost three orders of magnitude more accurate (0.003° to 0.007° depending on the axes) [7] [8]. Moreover, the XACT has an angular momentum storage of 0.0154 mNms when fitted with Blue Canyon Technologies MicroWheels [8]. By contrast, the reaction wheels of the MAI-400 store only 0.00935 mNms [7]. Ultimately any advantages the MAI-400 system had over the XACT were outweighed by the significant discrepancy in angular momentum storage and pointing accuracy.

De-tumbling

A de-tumbling simulation was performed to verify that the magnetorquers on the XACT system are capable of de-tumbling the satellite in the 72-hour time window. Simulink code developed by Sanny Omar served as foundation for modeling the system dynamics of the spacecraft [9]. Assuming a worst case ejection angular velocity of 5 revolutions per minute in each axis, if stabilization was left solely to the torques provided by the reaction wheels, the wheels would need to spin beyond their allowable angular velocities [10]. Thus, a magnetorquer incorporated in the ADACS was chosen to help de-tumble the OAO. The de-tumbling simulation employed the following equations:

$$\dot{\vec{\omega}} = \boldsymbol{I}^{-1}(\vec{\tau} - \vec{\omega}\times\boldsymbol{I}\vec{\omega}) \tag{1}$$

$$\vec{\tau} = \vec{m}\times\vec{b} \tag{2}$$

$$\vec{m} = \frac{-k}{\|\vec{b}\|^2}\left(\dot{\vec{b}}\times\vec{\omega}\right) \tag{3}$$

Equation (1) represents the dynamics of the spacecraft de-tumbling and how the angular acceleration ($\dot{\vec{\omega}}$) is affected by its inertia matrix (***I*),** angular velocity ($\vec{\omega}$), and applied torque from the mangetorquer ($\vec{\tau}$).

Equation (2) defines $\vec{\tau}$ as the cross product between the magnetorquer's magnetic dipole moment ($\vec{m}$) and the earth's magnetic field vector ($\vec{b}$). The $\vec{b}$ vector changes at every position in the earth's atmosphere across all three dimensions, and was generated in STK.

Equation 3 defines the magnetic dipole moment $\vec{m}$, where k is constant gain. This is the root of the B-Dot Law. The $\dot{\vec{b}}$ vector is orthogonal to $\vec{b}$ and the negative gain, -k, ensures the torque applied is opposite of the spacecraft's tumble. [9].

The gain was tuned to match the performance capabilities of the XACT system. Assuming a worst-case slewing scenario of five revolutions per minute after orbital injection, the de-tumble can be completed in approximately 26 hours as seen in Figure 1.

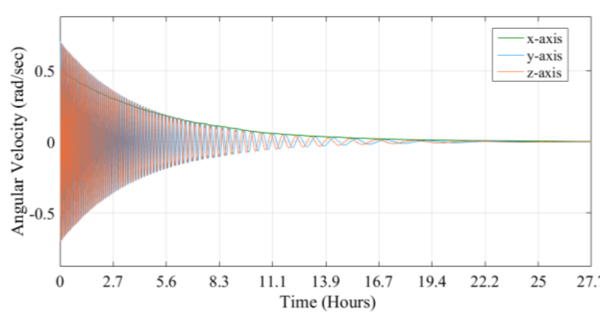

Figure 1: Angular Velocities During De-tumble

Proportional-Derivative Attitude Control Algorithm

Reaction wheel sizing, maximum angular velocity, and torque requirements were verified by modeling the attitude control system with a proportional-derivative (PD) control algorithm in Simulink that was modified from the simulation in Ref. [9]. Attitude quaternions for sun glint tracking describing the orientation of the satellite body frame relative to the Earth-centered inertial frame were generated in STK. The PD controller computed the desired angular acceleration of the system as follows:

$$\begin{bmatrix}\alpha_x\\ \alpha_y\\ \alpha_z\end{bmatrix} = \begin{bmatrix}k_{p_1}q_{e_1}\\ k_{p_2}q_{e_2}\\ k_{p_3}q_{e_3}\end{bmatrix} - k_d\begin{bmatrix}\omega_x\\ \omega_y\\ \omega_z\end{bmatrix}. \tag{4}$$

where $[\alpha]$ is the angular acceleration, q_{e_i} is the i-th element of the error quaternion, k_{p_i} is the proportional gain assigned to q_{e_i}, k_d is the derivative gain, and $[\omega]$ is the angular acceleration of the spacecraft with respect to the inertial frame. The angular momentum of the spacecraft was calculated as follows:

$$[H] = [I]_{SC} \begin{bmatrix} \omega_x \\ \omega_y \\ \omega_z \end{bmatrix} + I_w \begin{bmatrix} \omega_{wx} \\ \omega_{wy} \\ \omega_{wz} \end{bmatrix}. \quad (5)$$

where $[I]_{SC}$ denotes the moment of inertia matrix of the spacecraft. The moment of inertia and angular velocity of the reaction wheels are denoted I_w and $[\omega_w]$, respectively. The desired torque was calculated as

$$[T] = [I]_{SC} \begin{bmatrix} \alpha_x \\ \alpha_y \\ \alpha_z \end{bmatrix} - \begin{bmatrix} H_x \\ H_y \\ H_z \end{bmatrix} \times \begin{bmatrix} \omega_x \\ \omega_y \\ \omega_z \end{bmatrix}. \quad (6)$$

The simulation then calculated the angular acceleration caused by the commanded torque to complete the feedback control loop.

Gain Scheduling

Gain scheduling was implemented to create a proportional gain that varied linearly with error, such that the system did not command torques beyond its operating capabilities when high errors were present. The proportional gain, k_{p_i}, can vary between values of k_o and $k_o(1 - k_g)$. Both k_o and k_g were tuned simultaneously. The proportional gains were calculated as follows:

$$k_{p_i} = k_o - q_{e_i} k_o k_g. \quad (7)$$

Reaction Wheel Performance

The principal coordinate axes of the OAO, with respect to the mAPS line of sight, is depicted in Figure 2.

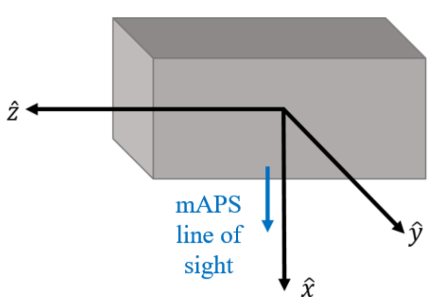

Figure 2: Coordinate System with Respect to mAPS Line of Sight

Simulations indicated that the torques required for continuous sun glint tracking were on the order of magnitude of 1×10^{-5} Nm, well under the maximum allowable torque. Therefore, it can be assumed that the XACT can perform sun glint tracking at its specified pointing accuracy of 0.003°. Moreover, the simulation indicated that large errors can be corrected in the range of 100-150 seconds without exceeding the maximum torque or angular velocity of the reaction wheels. Relative quaternion magnitude, reaction wheel angular velocities, and reaction wheel torques from one such simulation are presented in Figure 3, Figure 4, and Figure 5, respectively. This simulation is representative of the satellite transitioning from mAPS lunar calibration to sun glint tracking. Note that the error is corrected in roughly 100 seconds. The angular velocities and torques remain under the maximum values of 628 rad/s and 4 mNm, respectively [10].

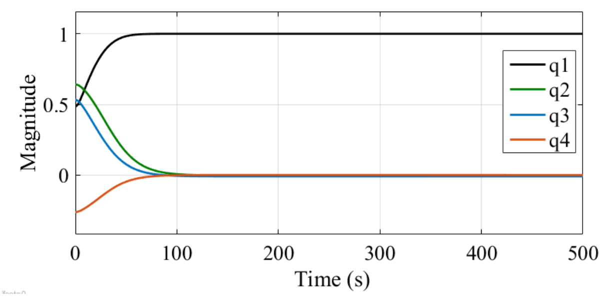

Figure 3: Relative Quaternion Magnitude While Reducing Initial Error

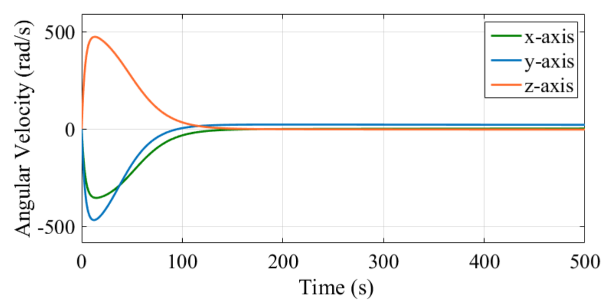

Figure 4: Reaction Wheel Angular Velocities

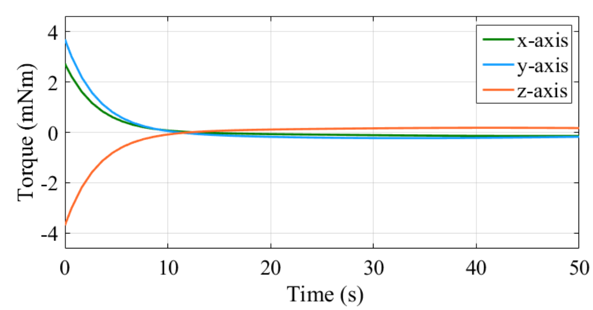

Figure 5: Reaction Wheel Torques

mAPS Lunar Calibration Maneuver

The mAPS utilizes the brightness of the moon to perform an on-orbit calibration maneuver. The lunar calibration maneuver will occur on January 7, 2020, roughly three days before the full Moon, and will require the ADACS to command the following maneuvers [1] [11]:

1. At a lunar phase angle of 24$\pm$3°, point the mAPS to a target 1° from the center of the Moon (duration 250 seconds).
2. Slew 2° to the other side of the Moon at a rate of 0.0133°/s (duration 150 seconds).
3. Slew 2° to the other side of the Moon, returning to the original target 1° from the center of the Moon, at a rate of 0.0133°/s (duration 150 seconds).
4. Slew 1.5° to the other side of the Moon at a rate of 0.0133°/s (duration 131.25 seconds).
5. Slew to 1.5° to the other side of the Moon at a rate of 0.133°/s (duration 112.5 seconds).
6. Return to nadir pointing to commence ground station tracking (duration 250 seconds).

The lunar calibration maneuver was also simulated with the PD control algorithm. STK was used to point the satellite body x-axis toward the center of the moon and export a set of Moon tracking quaternions. The body *z*-axis was constrained to the Earth inertial *z*-axis to reduce *x*-axis rotation. The Moon tracking quaternions were converted to direction cosine matrices and multiplied by a rotation matrix about the satellite body *z*-axis that specified the angular rotation of the satellite relative to the nominal moon-pointing attitude. The degree of rotation, θ, was varied linearly based on the calibration requirements. The calibration DCM was calculated as

$$C = \begin{bmatrix} \cos\theta & -\sin\theta & 0 \\ \sin\theta & \cos\theta & 0 \\ 0 & 0 & 1 \end{bmatrix} [M] \tag{5}$$

where *M* is the Moon tracking DCM at each time step. Calibration matrices were converted to quaternions for use in the PD control simulation. Results of the simulation indicated that the calibration maneuver can be accomplished with torques on the order of magnitude of $1\text{x}10^{-5}$ Nm. Therefore, it can be assumed that the XACT will be capable of completing the calibration maneuver with its specified 0.003° pointing accuracy.

COMMUNICATIONS

The Thule Tracking station located in Greenland will serve as the sole ground station for communication with the OAO. The high latitude of the Thule tracking station permits access in every orbit and allows for the mAPS and the transceiver to operate at separate times, which is ideal for power system and data handling factors. The OAO will spend approximately 5 minutes in communication range over the ground station each orbit. The mAPS will be collecting 364 MB of data each day, as derived from the mAPS data collection rate of 139 kbps and average sun glint access of 25 minutes per orbit. If access can be achieved each orbit, the OAO is only required to transfer data at a rate of 640 kbps; however, system access may be inhibited due to weather or ground conditions. As a safety precaution, the transmitter must be capable of transmitting data from a full day in one pass. This transfer will require a data rate of 8.96 Mbps, which is too high for most commercially available S-band transceivers.

A Syrlinks EWC 27 HDR-TM X-band transceiver can provide ample bandwidth to satisfy the design constraints and was chosen for use on the OAO. The EWC 27 is designed specifically for CubeSats and fits easily in 1U, while offering a data transfer rate of up to 100 Mbps [12]. At the maximum data rate, approximately 10 days of data can be transmitted in a single pass.

An 11 dB gain X-band patch antenna supplied by the Antenna Development Corporation was chosen for its simplicity, ease of mounting, and low volume and mass [13]. There is no need for deployment of a patch antenna, reducing system complexity and possible failure modes.

A SkyFox Labs PocketQube pqNAC-L1/FM GPS Receiver and SkyFox Labs piPATCH-L1 Active GPS Patch Antenna module were selected to provide global positioning data required for mission operations.

COMMAND AND DATA HANDLING

The on-board computer system manages data and commands for the entire satellite. The system chosen for use in this satellite is manufactured by GomSpace, the NanoMind Z7000 Field Programmable Gate Array (FPGA) with a Nano Dock SDR motherboard [14] [15]. The flight software will be developed in coordination with the supplier, GomSpace, using their preexisting framework to satisfy mission requirements.

POWER SYSTEM

To accurately predict the power requirements for the satellite, the power consumption and the length of operation for each component must be considered. The nominal power consumption of each component was given by the vendor. Table 1: Power Consumption [Watts]Table 1 displays each of the major components' power consumption for the different operational modes within the mission. The "Thrust" category represents the phasing maneuver, station keeping, and deorbit. A typical orbit during the science portion of the mission only includes the collect data, transmitting, and umbra modes.

Table 1: Power Consumption [Watts]

	Thrust	Calibrate mAPS	Collect Data	Trans-mitting	Umbra
mAPS	0.55	0.55	3.36	0.55	0.55
OBC	2.3	2.3	2.3	2.3	2.3
Comms.	1	1	1	7	1
GPS	0.2	0.2	0.2	0.2	0.2
ADACS	1	5.5	1	1	1
Thruster	15	0	0	0	0
EPS	1.25	1.25	1.25	1.25	1.25
Total	21.3	10.8	9.11	12.3	6.3

The amount of operation time in each mode was obtained from STK access windows. During the phasing maneuver the 15 W thruster will have to operate constantly for a period of 2.2 hours. During this time, the satellite will require an estimated total of 33 Wh. Over this same span of time, the solar panels will produce a predicted output of 68.2 Wh. The surplus power generated will be used to keep the battery charged and in turn power the thruster on the dark side of the orbit. Once the phasing maneuver is complete, the satellite will only require 12.0 Wh per orbit since some components, including the mAPS, transceiver, and attitude control, only run for part of each orbit.

Solar Power Analysis

An STK simulation was run to estimate the power generation by the solar panels during sun glint tracking. In this simulation, a 1U CubeSat model was used to determine incident sunlight on all six faces. STK provided the estimated power generated by all six sides of the cube; this data was normalized so that the maximum power generated had a value of one. Figure 6 shows the normalized data for all six sides of the satellite throughout one orbit, with the coordinate system illustrated in Figure 2. Using this data, the power output of several different solar array configurations can be extrapolated by multiplying the normalized value by the specified solar panel output and the length of time in the sun.

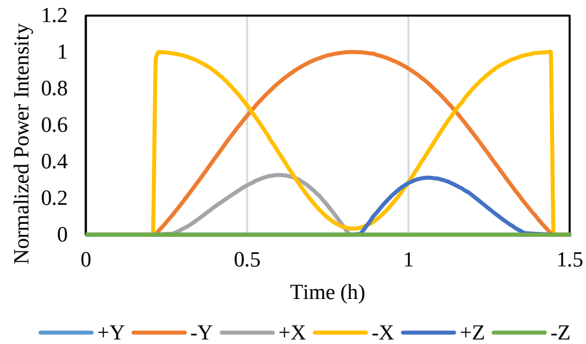

Figure 6: Normalized Power Generation for all Sides of a 1U Cubesat Over an Orbit

Power generation, storage, and management is accomplished through three components: the battery, the solar panels, and the electrical power system. Two deployable 3U solar arrays and two deployable 2U arrays will be used on the satellite. Together the arrays are rated to produce 40.4 W in maximum sunlight; however, due to inefficiencies and inconsistent sunlight, the estimated theoretical output is 30.6 Wh per single orbit [16]. This value was obtained by calculating the average power production over one orbital period. The panel arrays will be placed on the negative *y*-face of the satellite since it receives the most direct sunlight for the longest amount of time. In a standard orbit, the satellite will spend 1.08 hours with the solar panels in the sun. A power generation curve for one orbit is shown in Figure 7.

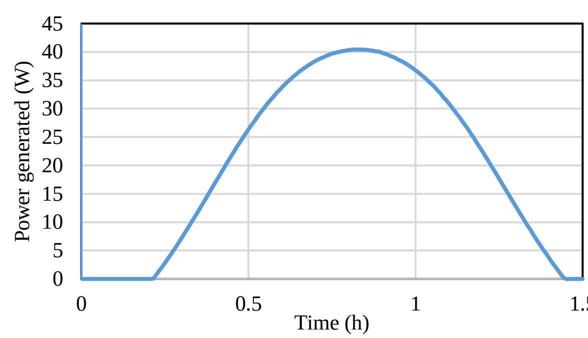

Figure 7: Solar Power Generation of the OAO Over One Orbit

Batteries

The batteries that were chosen were the GOMspace BPX, capable of storing 87.4 Wh of power, a capacity sufficient to power the OAO even during the peak power demand of the phasing maneuver [17]. The batteries also meet the FAA requirement of less than 100 Wh stored on the CubeSat [6]. The battery pack comes with a built-in heater which will keep it from losing its effective storage during its mission [17].

Electrical Power System

The electrical power system (EPS) will manage the power distributed throughout the system. The NanoPower P60 was chosen for its compatibility with the BPX batteries and the ability to manage over 30 W of power, as required during the phasing maneuver [18].

STRUCTURES

Mission requirements provided by the customer specified that the OAO must adhere to the 6U CubeSat form factor [1]. The Innovative Solutions in Space (ISIS) 6-Unit Frame was chosen as the satellite structure for its flight heritage and adherence to the size, material, and mass requirements [1]. The frame also includes adjustable shelving units for component mounting and center of mass adjustment. A CAD model was constructed in Solidworks using supplier-provided solid models. The model was utilized to estimate the overall mass, center of mass, and moment of inertia matrix of the OAO for use in orbital mechanics and attitude control simulations.

Mass Budget

The estimated mass budget is presented in Table 2. Note that the total mass is under the maximum allowable 12 kg that was allocated in [1].

Table 2: Estimated Mass Budget

Component	Mass (kg)
mAPS	5.000
ADACS	0.910
Structure	1.100
Solar Panels	0.770
Battery	0.500
EPS	0.064
X-band Transceiver	0.225
X-band Antenna	0.050
GPS Receiver	0.012
GPS Antenna	0.050
Thruster	1.150
Shelves	0.500
On-board computer	0.283
Hardware	0.500
Total Mass	**11.114**

Center of Mass

The deviation in center of mass relative to the geometric center can be seen in Table 3. The 6U CubeSat design specification document provides the allowable deviation in the center of mass from the geometric center [6]. In addition to meeting 6U CubeSat requirements, the center of mass must also be closely aligned with the thrust vector to prevent moments from being created by thruster firings. Misalignments between the thruster and the center of mass will require use of the ADACS and additional power consumption.

Table 3: Ideal Center of Mass Location

Distance from Ideal Center of Mass		
Direction	Distance to Geometric Center (cm)	Max Allowable Deviation (cm) [6]
x	-0.36	±4.5
y	0.34	±2.0
z	-1.37	±7.0

Component Layout

The CAD model is depicted in Figure 8, Figure 9, and Figure 10. The thruster is positioned such that the thrust vector acts through the center of mass. The mAPS and X-band patch antenna are located on the positive *x* face for Earth-pointing, and the GPS antenna is located on the negative *x* face for communication with GPS satellites. Other components were positioned to adjust the center of mass such that it was better aligned with the thrust vector.

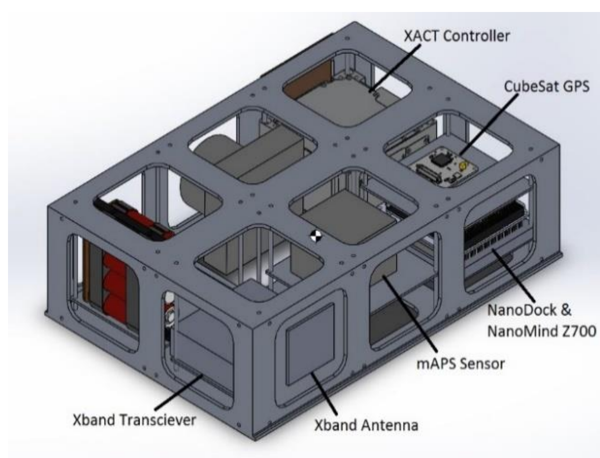

Figure 8: Annotated Final Assembly, Isometric View 1

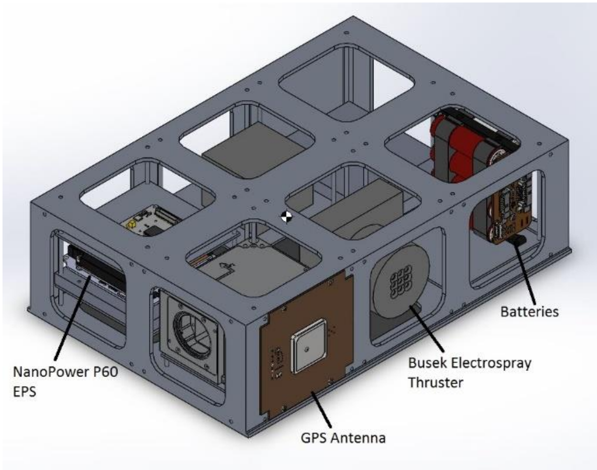

Figure 9: Annotated Final Assembly, Isometric View 2

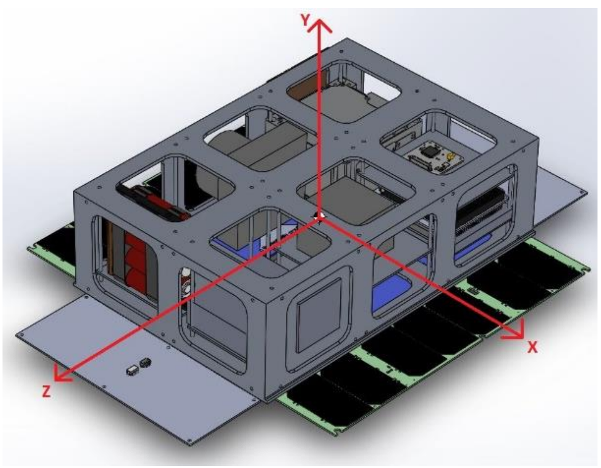

Figure 10: Final Assembly with Solar Panels and Coordinate System

Launch Vehicle Integration

Integration of the OAO CAD model with the 6U Canisterized Satellite Dispenser from Planetary Systems Corporation was verified [19]. The OAO structure fits within the dispenser and has rails that contact the dispenser to guarantee a smooth deployment.

Thermal Radiation

A preliminary thermal analysis indicated that the internal temperature of the satellite would exceed the maximum and minimum operating temperatures of some components. Operating temperature ranges of all satellite components are given in Table 4Table 4: Component Temperature Ranges, excluding the mAPS. The mAPS has integrated temperature sensors, survival heaters, and a passive radiative cooler and provides its own thermal management [3].

Table 4: Component Temperature Ranges

Component	Minimum Temperature (ºC)	Maximum Temperature (ºC)
XACT	-35	70
11 dB X-band Antenna [13]	-40	70
EWC27 [12]	-35	45
piPATCH-L1 GPS Antenna [21]	-40	85
PocketQube GPS/Flight Model [22]	-40	85
NanoMind Z7000 [14]	-40	85
Nanopower P60 System [18]	-40	85
NanoPower BPX [17]	-40	85
NanoDock SDR [15]	-40	85
BET-1mN [4]	-40	85

A single aluminized Kapton insulation layer was added to the design. Aluminized Kapton was selected for its low absorptivity and emissivity, to reduce the absorption of solar radiation in the sunlit portion of the orbit and reduce heat loss in Earth's umbra [20]. A sphere-based thermal model was generated in STK using the radiative heat transfer specifications for Aluminized Kapton film [23]. Results of this simulation predicated that satellite temperatures will vary between -20 ºC and 50 ºC during each orbit, which exceeds the maximum operating temperature of the X-band transceiver. The STK SEET Thermal Model incorporates solar radiation, Earth albedo, and internal heat generation of satellite components, but does not consider the thermal inertia of the system. A more thorough simulation of heat exchange within the satellite will be performed to investigate the need for multi-layer insulation or options for additional cooling.

TESTING

An end to end test plan was developed, based on GSFC-STD-7000A and the Falcon 9 Payload User's guide as a baseline for anticipated launch environments [23] [24]. The mAPS is assumed to be a mature technology, and all other components have prior flight heritage; therefore, only systems level testing will be undertaken with the OAO. To qualify the OAO for flight, two complete, identical satellites will be built (though one will use a mass mock in place of the mAPS in an effort to reduce unnecessary cost). The article built with the mass mock will be used to demonstrate structural and functional margin to the acoustic, shock, vibe and thermal environments anticipated on orbit and during launch.

The system will be tested to +6 dB above the shock, vibe and acoustic environments, and then exposed to a series of operational tests while undergoing thermal cycling in a vacuum chamber for a minimum of 350 hours of nominal function [23]. Upon successful completion of the tests and verification of full functionality of all systems and components, the design can be considered qualified. Following qualification, the actual spacecraft will undergo a shortened test campaign of system checks, thermal cycling and structural tests to verify functionality with no additional margin to the shock, vibe, and acoustic requirements [23] [24].

All systems have been chosen to be rad-tolerant; however, a total accumulated dose of over 1000 rad was anticipated over the two year lifetime of the OAO using STK's SEET Radiation Environment. Therefore, potentially sensitive electronic components from the test article, including the on board computer, ADACS, GPS and XBand transcievers, will undergo 1200 rad total accumulated does of ionizing radiation after system qualification. Successful function of those components can be used to demonstrate margin to the anticipated radiation environments for those components deemed most at risk.

Assembly will be conducted in a class 10000 or better cleanroom, and the flight article will be maintained in a clean environment at all times prior to launch, with a continuous gaseous nitrogen purge for the mAPS required from integration with the satellite through launch [1].

BUDGET

The overall budget outline included costs for the construction, launch, and operation of the OAO with an additional portion for development of the mAPS [1]. The cost breakdown is represented in Figure 11.

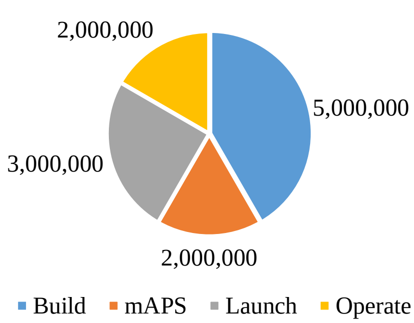

Figure 11: Mission budget in USD

Construction

Each vendor was contacted for an average quote of their product. The total cost for the hardware and software of two identical satellites came to approximately $1.6 million. In addition, two qualified engineers, salaried at $200,000 per year with a 40% overhead to include benefits and compensation, are expected to finalize the design and conduct the build and test of the two satellites. The remaining $2.72 million will be provisionally allocated further into development as testing and unforeseen construction costs become more prevalent.

Operation

The OAO is expected to operate with very little need for human intervention. The primary operation cost will come from data transfer from the Thule Tracking Station and upload to servers to provide access to researchers. An engineer will be contracted for 10 hours a week to review the operation of the spacecraft as well as the data received, for which a budget of $100,000 has been reserved for the two-year operation.

The commercial availability of parts and self-sustainability of the spacecraft significantly bring down the costs of the project, and are projected to lead to large structural margins and significant savings.

CONCLUSION

All major systems for the Orbiting Aerosol Observatory have been selected and, through simulations, were proven to be well suited for the mission requirements. Every phase of the OAO's mission, from deployment into the initial orbit to reentry, was modeled and used to verify the capabilities of the components selected. By focusing on commercial off-the-shelf components throughout, except for the mAPS, a significantly accelerated development time can be achieved while maintaining high reliability and low cost. However, further work remains to be done in integrating all systems with flight software, and ensuring robustness to off-nominal events in orbit. Additionally, thermal analysis should be refined to determine active thermal management needs for nominal operation of all components.

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