Preliminary CubeSat Design for Laser Remote Maneuver of Space Debris at the Space Environment Research Centre

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ABSTRACT

The Space Environment Research Centre (SERC) endeavors to demonstrate the ability to maneuver high area to mass ratio objects using ground based lasers. Lockheed Martin has been leading system performance modeling for this project that includes high power laser propagation through the atmosphere, targeted interactions, and subsequent orbital maneuver of the object. This paper describes a CubeSat that could be used as a potential target to demonstrate the maneuver system. The model assumptions and performance estimates for an on-orbit laser maneuver demonstration are discussed.

1. INTRODUCTION

There are an estimated 29,000 pieces of large debris (>10 cm) in orbit, 670,000 pieces from 1 to 10 cm, and over 170 million pieces smaller than 1 cm. These debris can travel at speeds up to 17,500 mph and, even at the size of a paint fleck, can inflict critical damage to a satellite or spacecraft.[1] While the current state of debris in Low Earth Orbit (LEO) is navigable to some extent, the Kessler syndrome describes a scenario in which the amount of debris is sufficiently large that the debris from previous collisions exponentially results in more collisions, leaving large bands of debris where satellites cannot exist.[2] To avoid this critical point, the Federal Communications Commission and NASA enforce requirements for spacecraft to prove their ability to deorbit within a specified time frame. Additionally, the Department of Defense Space Surveillance Network maintains a highly accurate catalog of objects in orbit that are larger than a softball in size. Because 90% of the space object catalog is composed of debris objects with no active maneuvering capability, there is no current solution to avoid the majority of collision scenarios.

One potential mechanism for collision avoidance is the use of a ground-based laser that utilizes photon momentum transfer (*i.e.* radiation pressure) to perturb debris into neighboring orbits, avoiding collision. In this paper, we outline the design for a CubeSat that can be used to calibrate a ground based laser for this purpose. Principal mission objectives for such a satellite are: 1) quantify the photon momentum transfer capabilities and the ability to track objects from a ground station, 2) measure the actual radiated flux achieved in-orbit, including temporal variations induced by a ground based laser system designed to exert enough photon pressure to sufficiently alter the orbit of space debris, and 3) to serve as an active, cooperative target with a sufficiently high area to mass ratio and known optical properties to support an empirically verifiable demonstration of the feasibility of an active collision avoidance system.

The proposed design solution incorporates a deployable reflective sail to convert photon momentum into a sufficiently perturbing force applied near the center of mass of the satellite. The satellite uses a photodiode to simultaneously characterize the laser, and an advanced 2U attitude control system maintains required attitude and rejects laser-induced perturbations. Attitude determination and payload operations are downlinked using an S-band patch antenna. The satellite is divided into seven subsystems: payload, attitude determination and control, electrical power, structure, command and data handling, communications, and thermal.[5] Herein, we discuss the payload, attitude determination and control, and the electrical power subsystems because they are the most unique to this mission.

2. MISSION AND SYSTEM OVERVIEW

The primary objective of this mission is to provide calibration data to the Mount Stromlo Observatory to determine the feasibility laser-induced orbit perturbation. The specific technical requirements of the satellite payload are to reflect a ground-based laser (~1 kW/m2 at 1064 nm) and measure momentum transfer, measure the flux of the laser as perceived by the satellite, track precise attitude and location information, and to relay all data to a ground station for further analysis. If successful, ground models will be calibrated using the information from this mission, and photon momentum transfer may be implemented in the future to predictably mitigate space debris collisions.

In designing this satellite, multiple subsystems are defined to address these challenges: attitude determination and control, command and data handling, telecommunications, electrical power, thermal control, structures, and the payload itself. A 3D model of the CubeSat is shown in Fig. 1. The payload components have been selected as an InGaAs photodiode with a radiation-hardened photoconductive circuitry, a deployable Al-Mylar sail developed by Clyde Space, and a dome of 3-4 corner-cube fused silica retroreflectors. Attitude determination (0.1° accuracy) and control (1° accuracy) can be achieved using a combination of reaction wheels and magnetorquers in conjunction with sun sensors, magnetometers, Global Navigation Satellite System (GNSS), and potentially star trackers. Command and data handling requirements are determined based on the various communication protocols utilized by subsystem components, as well as the required rate of payload data processing (1 KHz). Telecommunication frequencies have been predetermined by the SERC to be in the S-Band (downlink) and UHF (uplink) frequencies, requiring a patch and a dipole antenna, respectively. The electrical power system includes solar power generation, storage, and distribution, and is highly dependent on the orbit in which the satellite travels and the surface area available for solar panels. Similarly, the thermal subsystem is heavily reliant upon orbit, being responsible for ensuring heat is maintained and dissipated effectively for safe operation of all components, especially sensitive payload objects such as the photon flux detector. It is likely that a minimal thermal infrastructure is needed, however this will be further clarified depending on the orbit. The structural components must be designed such that the satellite can withstand takeoff and changes in orbital trajectories, but it must also implement the deployable sail necessary for this mission. A 2U cube satellite is preferred, however a 3U satellite is also explored due to mass and power constraints.

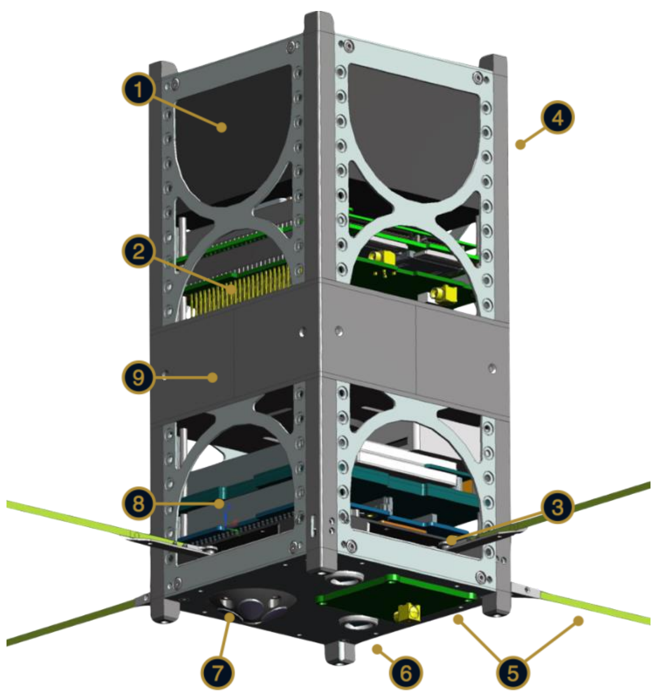

Fig. 1. CAD model of volumetric study using 2U structure. The components are as follows: 1. ADCS 2. CDH 3. TCS 4. STR 5. COMM 6. PAY: Photodiode 7. PAY: Retroreflector 8. EPS.

3. PAYLOAD

The payload subsystem will pursue the primary mission of the satellite along with two other secondary objectives. The primary mission is to transfer momentum from a high-energy ground-based laser to the satellite, referred to as Laser Momentum Transfer (LMT), to validate the space debris removal method. The satellite will also include a Photon Flux Detector (PFD) to quantify the quality of received radiation from the high-energy laser and Laser Retroreflectors (LRR) for ranging from a separate ground-based laser.

3.1 LASER MOMENTUM TRANSFER (LMT)

Electromagnetic radiation (*i.e.* light) possesses an energy and momentum proportional to the frequency of radiation. This momentum may be transferred elastically or inelastically over a known area, resulting in a pressure. Thus, a high-energy laser is able to impart a notable acceleration on a low mass object, especially in a space environment where drag is minimized. The force due to radiation momentum transfer is given by the following equation,

$$F = \frac{C_r}{c} \int I(x, y)\, dA \tag{1}$$

where c is the speed of light, I is the spatial intensity distribution of the laser, and C_r is a radiation pressure coefficient characteristic of the material's reflectivity. The radiation pressure coefficient ranges from 0 to 2, where in perfect reflection (*i.e.* elastic collision) Cr = 2, perfect absorption (*i.e.* inelastic collision) Cr = 1, and complete transmission Cr = 0. Therefore, a perfectly reflective material with Cr→2 will result in the largest momentum transfer, which will facilitate the observation of LMT. In addition, the irradiated area should be maximized to further increase momentum transfer.

Current physics models suggest an irradiated area of >1 m^2, which would require the deployment of a nadir-pointing or target-tracking "sail" to act as the reflective surface for the high-energy laser. This area is derived from calculations which assume that a 10X ratio of laser force to drag force is necessary to achieve a detectable maneuver; however, a "detectable maneuver" must be more strictly defined by SERC to qualify this requirement. This laser-to-drag ratio requirement has driven much of the design of the LMT system, including the requirement of nadir-pointing. As shown in Fig. 2, a track-pointing scenario would not result in sufficiently resolvable maneuvers due to prohibitively large drag forces. When sizing the sail, it is important to recognize that we are really considering the area-to-mass ratio, commonly written as A/M, thus the decision regarding the final mass of the satellite (largely driven by the decision of 2U versus 3U) will determine sail size. The irradiance of the laser is assumed to follow a Gaussian

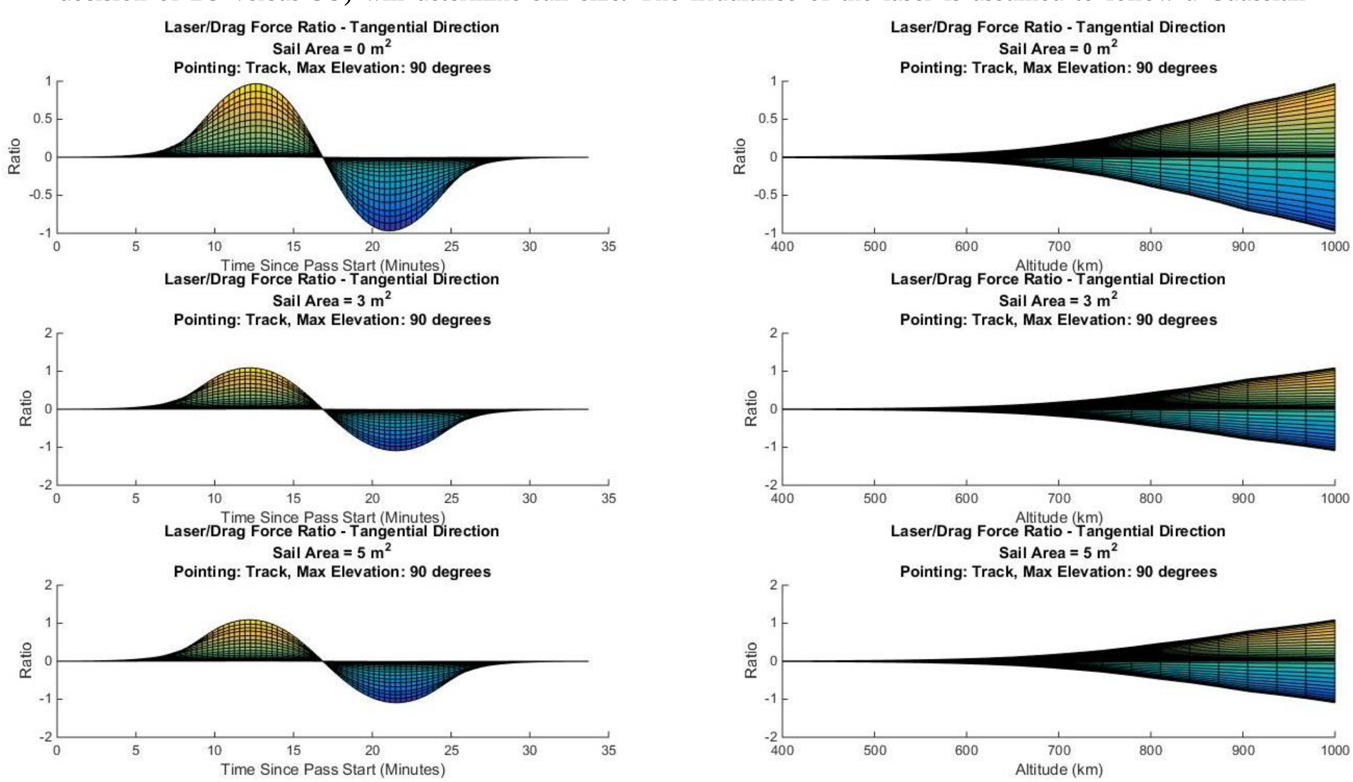

Fig. 2. Laser-to-drag ratio calculations for nadir-pointing scenario.

distribution that is attenuated nonlinearly through the atmosphere, and its final irradiance as a function of altitude is predicted using Lockheed Martin models. This irradiance information is also used for work on the PFD.

ClydeSpace has developed a deployment mechanism for a 3 m^2 sail which is commercially available. The system employs a tape spring deployment that has been performed reliably on-ground.[6] Current conversations with ClydeSpace indicate that the device is highly customizable for our purposes because the system is not readily available. In fact, ClydeSpace would develop this product over a 6-8 month period in tandem with our project, such that cable pass-throughs, sail design, and structure integration would be highly flexible. The system, referred to as AEOLDOS, possesses several unique capabilities that have been addressed by a Request for Information: mid-stack configuration, re-foldable sail cartridges for testing, no power consumption, and cable pass-throughs. However, this system has no flight heritage and is not readily available. Concerns have been expressed regarding the pass-through size and deployment trigger, neither of which have been finalized. Knowing these issues, we continue to pursue to AEOLDOS system for several reasons: there are no other commercial options, ClydeSpace is flexible in overall price to get flight heritage on this system, and the benefit of custom-developing this product with ClydeSpace for our mission. In addition, the 3 m^2 sail size has been deemed sufficiently large for this mission after calculating the minimum altitude necessary to achieve a 10X laser-to-drag force ratio at this sail size.

Regarding the sail material, a membrane should be developed such that it is reflective specifically in the range on the irradiating laser (~1000-1100 nm). Reflection, absorption, and transmission outside of this wavelength does not affect LMT from the laser, although other sources of irradiance exist. Aluminized Mylar (Al-Mylar) has been tentatively selected as the sail material. The aluminum exhibits favorable reflectivity characteristics in the near-IR, and the Mylar exhibits favorable UV stability. This material has been developed extensively for solar sailing applications, and it is already the default material provided for the AEOLDOS deployment system. Further characterization using FTIR would provide higher fidelity information at the near-IR and enable modeling of thermal performance of the sail.

3.2 PHOTON FLUX DETECTOR (PFD)

The high-energy laser will necessarily be affected by the atmosphere as it traverses through space, therefore characterizing the radiation incident on the satellite will provide insight into radiation pressure analysis. Computational models are already in place to provide such analyses, and the PFD will provide validation and calibration to these models. The PFD will measure the flux of impinging radiation from the high-energy laser as a function of time, although it will not need to discern a spectral flux from that radiation.

The PFD has been selected as an InGaAs *p-i-n* photodiode, which absorbs light in the 800-1600 nm range. A bare InGaAs photodiode can withstand irradiance flux values up to 3 kW/m^2, whereas the laser being used for this project is expected to illuminate the satellite at irradiance flux values around 1 kW/m^2, thus well within the operating range of the photodiode. In addition, InGaAs is a III-V semiconductor, which has a history of radiation tolerance in space. InGaAs possesses a particularly linear response within 1000-1100 nm, making this ideal for an Nd:YAG laser at a wavelength of 1064 nm for testing purposes.

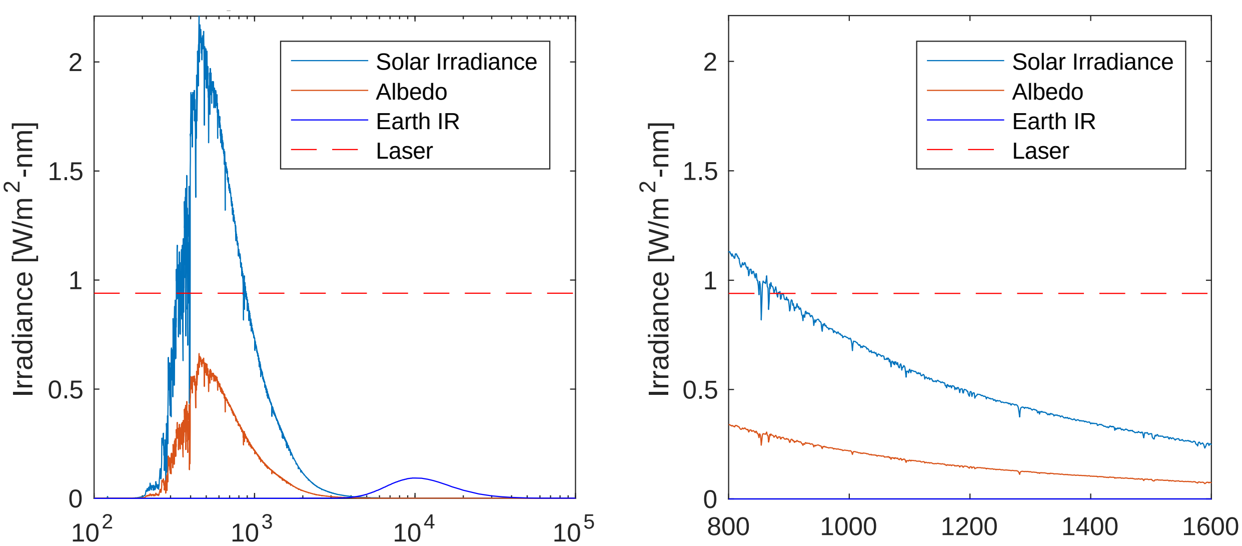

Fig. 3. Spectral irradiance in LEO from the sun, albedo, earth-IR and the Nd:YAG laser.

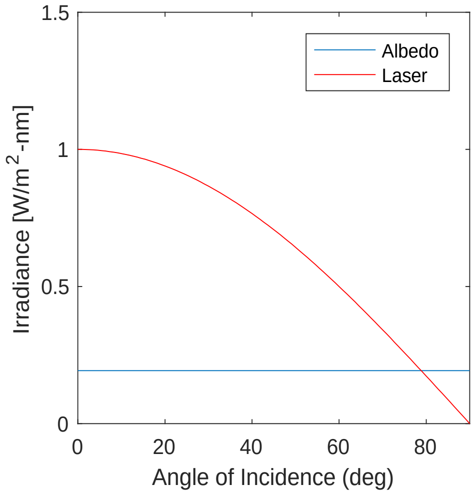

Fig. 4. Angular dependence of irradiation from Nd:YAG laser.

The PFD will be expected to operate in two modes: sun-pointing and laser-pointing. The sun-pointing mode is for a rough calibration of the PFD with the sun. Fig. 3 displays all significant sources of irradiance in LEO, include solar irradiance, Albedo from earth, earth-emitted IR, and compared with the laser flux. In the sun-pointing mode, the PFD will see the solar irradiance in the 800-1600 nm range, which integrates to be 462 W/m^2, which is well above the detectable level of the PFD. Once laser-pointing, the PFD will be confronted with noise from the earth's albedo and emitted-IR, however both prove to be reasonably low at 132 W/m^2 combined. It is notable that this will be received as significant noise in the signal, although the laser should be discernable at >800 W/m^2, which is more than double in magnitude than that of the noise. Baffling and optical bandpass filters are being discussed to minimize noise, although the current consensus is that they are unnecessary and my further induce uncertainty in the signal response. An alternative laser configuration has been proposed in which several lasers (including the 1064 nm Nd:YAG laser) are spectrally combined to increase overall power. This proposal has not yet been proved feasible, but it is being explored by SERC.

The photodiode is sensitive for both angle of incidence and temperature. Regarding the angle of incidence of the impinging radiation, Fig. 4 shows a simple model of this dependence strictly if the laser angle of incidence were varied. Depending on the orbit, more sophisticated models show that solar irradiance will exceed the irradiance of the laser at certain angles; a baffle should be designed based on this parameter. However, the viewing angle due to the baffle should be discussed again after the orbit is specified. Regarding temperature dependence, it is impractical to maintain <2 K tolerance in temperature for the photodiode using Peltier cooling and resistive heating. It is suggested that diode response is characterized as a function of temperature to a resolution of <1 K and use an RTD monitor the temperature. In this way, the response due to temperature may be backed out if the temperature is known at all times.

The PFD system possesses its own processing unit designed to handle payload data. Data is to be collected at 1 kHz, with an unspecified resolution that we assume to be 1000 points. The circuit for the PFD runs in the photovoltaic mode. The op-amp in this case can be designed to operate at <5 V, and the capacitor "caps" need not be more than 10 nF if deemed necessary at all. This circuit design assumes low frequency operation at 1 kHz, which may be changed to a photoconductive operation if operation frequency significant increases. This noise-mitigated op-amp scenario may be repeated iteratively to reduce noise and maintain gain, although noise from operation is expected to be minimal at 1000 points for 1 kHz. After the photovoltaic circuit, the signal will be processed by an ADC chip operating on a 10-bit platform, the bit rate for which may be easily increased if higher resolution is desired. Finally, the signal is processed by a parallel-to-serial chip operating on the I2C platform to maintain a unify operating language with the other components. This entire circuit is expected to occupy less than half of a 1U PCB board.

3.3 LASER RETROREFLECTORS (LRR)

Precise ranging of this satellite is necessary in order to ensure that the high-energy laser is directed directly at the reflective surface. Laser ranging provides high precision (<3 cm) without the need for additional active subsystems on the satellite, making it ideal for CubeSats. In such a system, a ground-based laser emits light in the direction of the satellite, and retroreflectors reflect that light back to a detector on the ground with a time delay. The satellite must incorporate retroreflectors with adequate reflection and viewing angles to pursue this tracking scheme.

The wavelength of the tracking laser is 532 nm, which requires that the retroreflector be effective in the range of visible light. The half cone angle will be at least 5 degrees, and several retroreflectors will be positioned such that the effective field of view will be nearly 2π sr as to minimize the effect of attitude on tracking capabilities. The International Laser Ranging Service (ILRS) hosts a number of missions that incorporate laser tracking, which will serve as models for the LRR component of this satellite. The retroreflectors can be purchased commercially.

Current modeling suggests that a half-sphere “dome” with 3-4 fused silica corner-cube retroreflectors will be sufficient for our mission. This is based off the requirements of Lightsail-B, which is a 3U CubeSat with similar ranging requirements.

4. ATTITUDE DETERMINATION AND CONTROL

The Attitude and Determination Control System (ADCS) will provide all required functionalities to determine and control the attitude of the satellite and to determine the orbital state vector of the satellite. The ADCS is required to provide 3-axis 0.1° pointing knowledge accuracy, 1° pointing control accuracy, a slew rate of 1°/second, and an orbital state vector determination accuracy of 10 meters. The required slew rate is a result of the cross-tracking requirement during payload operations, which is required to ensure sufficient sail exposure to the laser when not flying directly above the laser. The subsystem components can be categorized into three different functionalities: attitude sensors, actuators, and orbital state sensors. The components associated with each category are detailed below.

4.1 ATTITUDE DETERMINATION COMPONENTS

Various sensors used in conjunction with a Multiplicative Extended Kalman Filter determine an attitude estimate of the satellite throughout orbit. The MEKF processes each sensor measurement along with the associated uncertainty to output a real-time estimate of the attitude. The MEKF also ingests system dynamics model and measurements from rate sensors to propagate the state estimate between updates. The attitude sensors required on the satellite are a magnetometer, a fine sun sensor, and an inertial measurement unit, such as rate gyros. The magnetometer and sun sensor measure the magnetic field vector and sun vector, respectively. These measurements are used along with the known position and alignment of the sensors on the satellite and the current orbital state vector of the satellite to determine attitude estimates based on magnetic field and sun ephemerides. In sunlit orbit, these components provide pointing knowledge estimates of 0.1° according to Clyde Space and approximately 0.3° according to CubeSpace. The rate gyro provides 3-axis angular acceleration measurements.

The need for a star tracker for additional attitude determination accuracy is also being studied. The star tracker would provide attitude determination accuracy on the order of arcseconds, resulting in an order of magnitude of accuracy increase during eclipse orbit as well as improvement in the sunlit orbit. However, this component adds at least 0.2U of volume to the system as well as at least 0.3 W of power draw, which may not close under the current volume and power budget. Lockheed Martin and SERC are currently determining the effect of reduced pointing knowledge accuracy on the ability to correlate orbit perturbation measurements to momentum transfer from the laser rather than disturbance torques from sources such as solar radiation pressure and drag. Based on this analysis, a final decision can be made on the need for a star tracker.

4.2 ATTITUDE CONTROL COMPONENTS

The actuators, or attitude control components, that will be used in the satellite are reaction wheels and magnetorquers. There will be three reaction wheels with orthogonal rotation axes to provide three-axis control. Magnetorquers are required on the satellite to desaturate the reaction wheels.

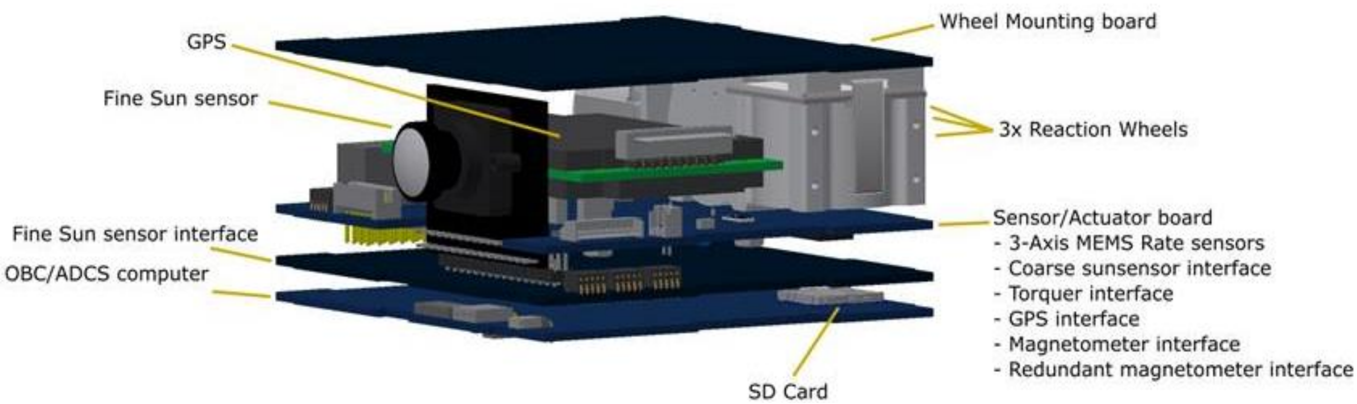

Fig. 5. ADCS configuration from CubeSpace. Figure reproduced from CubeSpace.

A trade study is currently being conducted to determine magnetorquer placement. There are two options: integrate the magnetorquers into the PCB of the solar panels, or have two torque rods mounted orthogonally inside an inductive coil that acts as the third orthogonal magnetorquer. The option of integrating them into the solar panels would save mass and volume but would not be able to provide as large of a moment as the torque rods. The required moment provided by the magnetorquers will determine which configuration to move forward with. This moment depends on the rate at which the reaction wheels will become saturated, which is a function of the disturbance torques on the satellite. These disturbance torques can be mainly attributed to the effects of solar radiation pressure, laser pressure, and drag on the 3 m^2 sail that is offset a maximum distance of 4.5 cm from the center of gravity of the satellite. Once these numbers have been quantified, a final decision will be made as to what configuration will be required for the magnetorquers as well as the required size of the reaction wheels.

4.3 ORBITAL-STATE DETERMINATION COMPONENTS

A GPS receiver will be used to determine the orbital state vector of the satellite throughout orbit with approximately 10 m accuracy. The GPS will require a receiver and antenna to detect and process the signal and determine the inertial state of the satellite. CubeSpace recommends the piNAV-L1 GPS receiver from SkyFox Labs which can provide an accuracy of 10 meters with a 2-sigma confidence level and an update frequency of 1 Hz. The receiver also provides velocity determination accuracy of 10 cm/s with a 2-sigma confidence level. The power draw is also only 0.12 W which is low in comparison with other COTS GPS receivers. The Clyde Space GPS receiver provides similar performance so either supplier should suffice regarding this component.

Having an accurate GPS receiver is crucial to the mission success so that the time required to locate the satellite in the approximately six minutes of useable laser engagement time per day is minimized. The idea of accurately propagating the orbital positon based on the last known position and downlinked attitude data from the prior orbit was studied; however, the results showed that the uncertainty after just one orbit grew to hundreds of meters. This uncertainty cannot be handled under the established mission requirements so the satellite must have an accurate GPS receiver on board.

5. ELECTRICAL POWER

The electrical power subsystem is responsible for generating, conditioning, regulating, and storing power for peak demand operations and distributing it throughout the spacecraft. The major components of the EPS will include: solar arrays, batteries and a power management distribution board.

5.1 POWER GENERATION

Solar arrays are used to generate power for the satellite. The preferred vendor for the solar arrays is ClydeSpace as they have experience with solar panel customization and are the vendor that will be used for the sail and structure. In preliminary talks with ClydeSpace, they are confident that they will be able to customize the panels to maximize the available space on the structure while also working around the area required by the sail deployment mechanism. This will be necessary since the sail requires 0.4U of space, therefore standard 1U solar panels will not be able to be used. The design will require custom sized panels to go above and below the sail on the long face of the satellite if using a 2U or 3U structure. ClydeSpace panels utilize Spectrolab UTJ solar cells manufacturing using PCB substrate with space-grade Kaptop coverlay. They also include built-in sun detectors, temperature sensors, and magnetorquers. The other major vendor that was considered was GomSpace, but for compatibility with other components and given the ability to customize around the sail, ClydeSpace was selected as the preferred vendor.

The solar arrays can either be body-mounted or deployable. The deployable option also offers customization in the axis of deployment and number of panels deployed. The body-mounted option theoretically provides less power generation due to its limited surface area of exposure. Since the sail will deploy from the center of the satellite, it is likely that it will cause a shadow on the solar panels. An analysis of this was performed and found that between 0-7°, there is still some exposure to all the panels, but once the satellite is orientated at an angle greater than 7°, there is no longer any exposure to the area of the satellite under the sail. Based on this analysis, it can be assumed that if the satellite is oriented off-axis from the sun vector by greater than 7°, then the sail will cast a shadow on half of the usable solar panel area and therefore only half of the 2U/3U face will generate power.

If deployable solar panels are used then it will be important that the panels are deployed in an orientation such that they are always facing the sun. This will be dependent on the type of orbit and the right ascension of the ascending node (RAAN). It is likely that the panels will need to be deployed from one of the long faces of the satellite, as opposed to the short face.

There is an open trade study comparing the use of body-mounted versus deployable panels, which will be discussed further in the Power Budget section of this paper. Although deployable panels generally have greater power output since they can cover a larger surface area, they present a high failure risk regarding their deployment. Additionally, based on the current state of the design, there is a tradeoff between power and mass, as it must be noted that the deployable system would add significant mass additions to the system. Further mass analysis can be seen in the Structures section of this paper.

Another trade study being performed regarding the solar panels is whether to build them in-house or purchase them from a commercial vendor. Georgia Tech has experience building body-mounted solar panels for cubesatellites but they do not include built-in components such as magnetorquers and sun sensors, which commercial panels generally come standard with. They also do not have experience building deployable panels, which may be necessary for the design. There is a cost savings and greater ability for customization with building in-house panels but this adds complexity to the design and fabrication as well as it decreases the confidence of deployment success. The conclusion of this trade will likely depend on direction from Lockheed Martin and SERC as well as an ADCS study on whether the standard magnetorquers and sun sensors that come with ClydeSpace or GomSpace solar panels meet the design specifications.

5.2 POWER MANAGEMENT DISTRIBUTION BOARD (PMAD)

The purpose of the power management and distribution board is to charge the batteries and provide power to the spacecraft. During sunlight operations, the PMAD will regulate the solar power for use by the various subsystems and will shunt excess power generated by the panels for battery charging. During eclipse, the PMAD will regulate the supply of power to the subsystems. GomSpace and ClydeSpace were considered for sourcing the PMAD, but it was concluded that the preferred vendor will be ClydeSpace since we have chosen to source the solar panels and other EPS components from them. It is important to use the same vendor for the EPS components to ensure compatibility and ease of design integration. The chosen system is the ClydeSpace Power Bundle: EPS + 10 Watt-hour (Whr) Battery.

5.3 POWER STORAGE

The purpose of the on-board battery is to store generated power for regulation and distribution by the power management board. The battery will likely consist of individual lithium ion cells connected in series with a built-in automatic heating system to keep it within operation temperatures at all times. Based on the power and mass budget, a 10 Whr battery should be acceptable for use since the maximum power draw per orbit is less than 6 Whr. Moving to a larger battery would add extra weight and does not result in any added benefits since the power require will never exceed the bounds of a 10 Whr battery. The vendor for the batteries must be the same as the vendor for the PMAD, thus ClydeSpace has been chosen as the preferred vendor for the batteries.

5.4 POWER BUDGET

An initial power budget has been created in order to determine the required battery capacity and whether the satellite will have enough power to perform the mission. The power budget is split into three modes where each component is set to on or off depending on what capabilities are required during that operational mode. Based on a 90 min orbit, a typical orbit profile has been decided as 75 min of nadir pointing, 12 min of downlinking only, and 3 min of payload operations. Nadir mode is the primary operational mode of the satellite and involves all components vital to the general health of the satellite to be kept on while minimizing unnecessary power draw by turning off non-critical components. The goal of this mode is to maintain the orbit and health of the satellite while reducing power consumption, therefore it is used for all times during orbit when the satellite is not performing payload operations. Payload operations mode involves all components to be turned on and interacting with the ground station. This mode will only occur for 3 minutes per orbit, and constitutes the time spent when the "science" mission is being performed and the satellite is interacting with the laser and ground station. Downlink only mode occurs when all components are turned on, except for those which are solely necessary for payload operations, and the S-Band transmitter is downlinking information to the ground.

Using these profiles, the power draw during each of the three modes was multiplied by the time in the mode to obtain the energy draw in Whr. That energy draw was subtracted by the initial battery capacity until the end of the orbit where the solar power generation value was added in. To estimate the amount of power generation per orbit,

Table 2. Results of the 2U body-mounted analysis.

Altitude	Avg Power	Max Power	RAAN @ Max	Min Power	RAAN @ Min
600	3.574	4.242	300	2.728	60
650	3.509	4.297	140	2.538	330
800	3.489	4.439	150	2.519	330

Table 1. Power budget for 2U body-mounted panels above the sail.

	Mode	Time (hr)	Power Draw (W)	Energy Draw (Whr)	Battery Power (Whr)
				Initial SOC	10
Orbit 1	Nadir	1.497	-3.80	-5.69	4.31
	Full Power	5.6E-04	-11.30	-0.01	4.30
	Downlink	2.2E-03	-11.30	-0.03	4.28
	Power Generation			4.40	8.68
Orbit 2	Nadir	1.497	-3.800	-5.69	2.99
	Full Power	5.6E-04	-11.3	-0.01	2.98
	Downlink	2.2E-03	-11.3	-0.03	2.96
	Power Generation			4.4	7.36
Orbit 3	Nadir	1.497	-3.800	-5.69	1.67
	Full Power	5.6E-04	-11.3	-0.01	1.66
	Downlink	2.2E-03	-11.3	-0.03	1.64
	Power Generation			4.4	6.04
Orbit 4	Nadir	1.497	-3.800	-5.69	0.35
	Full Power	5.6E-04	-11.3	-0.01	0.34
	Downlink	2.2E-03	-11.3	-0.03	0.32
	Power Generation			4.4	4.72

Systems Tool Kit (STK) was used to model various scenarios. It was found that 5.7 Whr of power would need to be generated per orbit, at a minimum, in order to remain power positive throughout the duration of the mission.

5.5 2U BODY-MOUNTED SOLAR PANELS ONLY

An analysis was performed to estimate the amount of power that would be generated during a 90-minute orbit with body mounted panels on all faces of the satellite except for the nadir pointing face (where the photon flux detectors and s-band transmitter will be placed). Due to limitations in the STK model and the inability to model the effects of the sail, a few assumptions were made:

1) The sail would shadow all solar panels below it, therefore only the solar panels above the sail would generate power

2) The solar panels above the sail would be 1U in size on each face (with no panel on the nadir pointing face), since the sail is modeled to be placed far enough down the structure that 1U worth of panels can fit above it and 0.6U panels would fit below.

This was performed for three different scenarios: Sun-Synchronous orbit at 600km, 650km, and 800km. For each of these cases, the right ascension of the ascending node was changed from 0-360° to characterize which cases would generate enough power to close the power budget. A summary of the results of this analysis can be seen in Table 1.

It was found through this analysis that there are no scenarios in which a 2U satellite with just body-mounted panels above the sail would close the power budget. The maximum power generated in this configuration would be 4.439 Whr at 800km and 150° RAAN, which would result in a loss of power by the fourth orbit as seen in Table 2.

5.6 3U Body-Mounted Solar Panels

An analysis was performed to estimate the amount of power that would be generated during a 90-minute orbit with body mounted panels on all faces of the satellite except for the nadir pointing face (where the photon flux detectors and s-band transmitter will be placed). Due to limitations in the STK model and the inability to model the effects of the sail, a few assumptions were made:

1) The sail would shadow all solar panels below it, therefore only the solar panels above the sail would generate power

2) The solar panels above the sail would be 2U in size on each face (with no panel on the nadir pointing face) and 1U in size on the small face opposite the nadir face

This was performed for three different scenarios: Sun-Synchronous orbit at 600 km, 650 km, and 800 km. For each of these cases, the right ascension of the ascending node was changed from 0-360° to characterize which cases would generate enough power to close the power budget. It was found through this analysis that there are a handful of cases that would allow the power budget to close, as well as some that would not. If the decision was made to move to a 3U structure, then there would still be constraints on the orbits that could be used.

Based off the analysis performed in STK, it was concluded that a 2U structure with body-mounted panels above the sail would not be able to close the power budget in any scenario. There are scenarios in which a 3U structure with body-mounted panels above the sail would close the budget, but constraints would have to be placed on the specific orbit and RAAN in order to ensure that the satellite would remain power positive throughout the duration of the mission. It was also concluded that more analysis should be performed to estimate the effects of deployable panels.

6. CONCLUSION

It is shown that a 2U CubeSat structure is capable of hosting all the necessary instrumentation to calibrate a ground-based laser for photon momentum transfer. The design includes a predeveloped sail deployment system from ClydeSpace to dramatically increase the area to mass ratio, making the satellite an ideal target for momentum transfer. This system relies on a reliable tape-spring deployment mechanism and is fitted in Al-Mylar for high reflectivity in the laser's spectral range. Additionally, an InGaAs photodetector has been designed that provides information regarding laser irradiance as a function of altitude, attitude, and orbit. This detector is robust against background radiation from the sun and earth and can therefore be used as validation and calibration to ground-based models regarding radiation attenuation. Along with a highly accurate ADCS, these systems constitute a cooperative target to test and refine laser-induced orbit perturbation of space debris, thus providing the first opportunity to mitigate space debris collision from a ground-based system.

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