SSC15-P-40

Design and Testing of a CubeSat-Sized Retroreflector Payload

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ABSTRACT

In a previous paper, the feasibility and trade space of a CubeSat with modulated retroreflector optical communications was discussed. In this paper, the design and testing of a surrogate retroreflector payload to assess key components of a modulated retroreflector payload is presented. In addition to passive retroreflectors, the surrogate payload has an optical beacon to aid in acquisition and tracking as well as an optical detector. The payload design is comprised of six subsystems: mechanical structure, optical electrical circuits, flight computer, retroreflectors, optical beacon, and optical detector. The mechanical structure has been tailored to accommodate the retroreflectors, while preserving roughly a 1.5U size. The circuits are designed to perform electrical power conversion, drive a high current optical source, and drive an optical detector. The flight computer will be comprised of a commercially available FPGA and/or a microcontroller on a custom circuit board to interface with the optical beacon and optical detector as well as collect telemetry. Previously developed models are used to design custom retroreflectors to provide a specific returned optical intensity pattern. The optical source and optical detector are commercially available and designed according to link analysis and electrical power restrictions. Individual components will be benchmarked, environmentally tested, and reassessed for performance prior to integration into the mechanical structure.

INTRODUCTION

A CubeSatellite offers a versatile, responsive, and relatively inexpensive solution for space applications. While the reduced size, weight, and power (SWaP) enables these characteristic, it is ultimately the limiting factor in payload performance. Previously, we discussed the challenges associated with optical communications on a CubeSatellite platform [1]. Particularly, the modulated retroreflector (MRR) was evaluated as an optical communication solution for a CubeSat, both for ground-to-space downlinks and interspace crosslink communications.

In this paper, the design and engineering is presented for a CubeSat-sized retroreflector payload. The payload is intended to reduce many of the risks identified in our previous paper for a 3U free-flyer CubeSat. These risks include retroreflector optical design and survivability, optical channel characterization, spacecraft pointing requirements, link budget validation, and optical ground station performance. The payload is being built as a hosted payload.

The payload is designed to act as a surrogate for an MRR optical communication payload. It contains all of the required components as the communication payload except the MRRs. The MRRs are under development and will be presented in a later paper. The reported payload was designed to mimic the envisioned MRR payload, and accordingly the mechanical structure has approximately 1.5U of internal payload space. Within the reported payload are 4 passive hollow retroreflectors, an 808nm laser beacon used for acquisition by the ground station, a Silicon detector and transimpedance amplifier used to record the interrogating laser beam, a power distribution board, and a flight computer.

The payload will be designed to perform several on-orbit experiments to validate payload performance models. The retroreflectors are custom designed and the reflected laser power received by the ground station will be used to validate optical system design and link budget models for the next design increment. The laser beacon will have selectable modes to operate in either a continuous wave or modulated output; this data will be input to ground station tracking and receiver sensitivity models. The detector onboard the payload will log the raw illumination laser data to be used in atmospheric transmission and turbulence modeling. The flight computer/processor board will log telemetry as well as set the operational modes of the laser beacon and detector. The telemetry from the electronics will be used to verify system operation as well as thermal modeling.

PAYLOAD MECHANICAL STRUCTURE

The payload mechanical structure is comprised of three parts: the main structure, zenith cover, and nadir cover. The main structure is machined from a single block of aluminum and serves as the strength member and housing for the optical and electronic components. The nadir cover has cutouts to accommodate the optical field of view of the retroreflectors, laser beacon optics, and detector optics. The zenith cover is machined for the single electrical connector. Both covers are removable for installation and access of components during assembly. These three parts are indicated in Figure 1.

The physical dimensions of the mechanical structure are representative of a 1.5U CubeSat form factor (10 cm x 10 cm x 15 cm). Aluminum 6061-T6 was selected as the structure material in order to meet the high vacuum and extreme temperature survivability requirements of a space environment. In addition, Aluminum 6061-T6 exhibits low outgassing, high thermal conductivity, and has a high strength to weight ratio. The thermal properties aid in thermal management of the laser, as well as add strength to better survive launch vibrations. An Iridite chromate conversion coating was chosen for the mechanical structure to prevent corrosion in space while maintaining electrical and thermal conductivity; the coating conforms to MIL-DTL-5541.

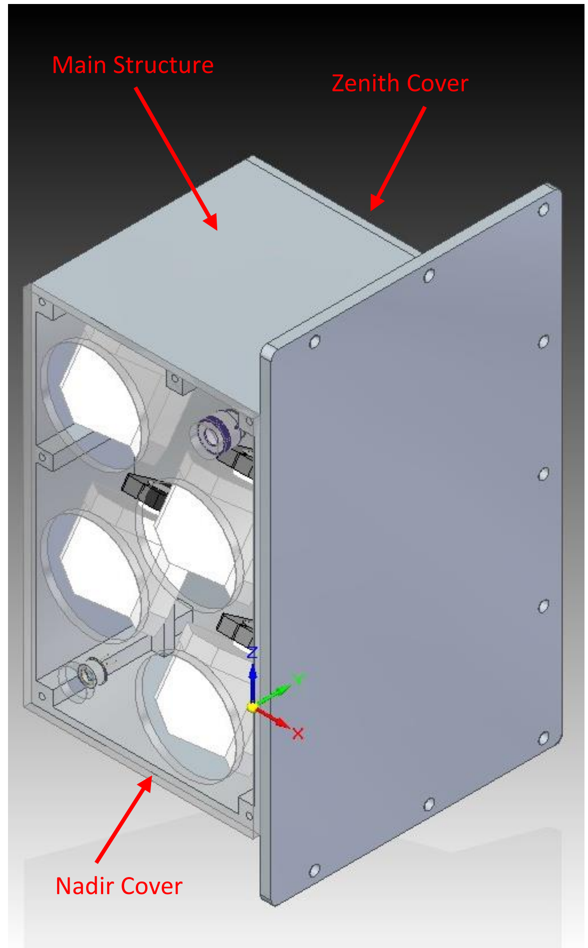

Figure 1: Isometric view of payload showing retroreflectors behind Nadir cover.

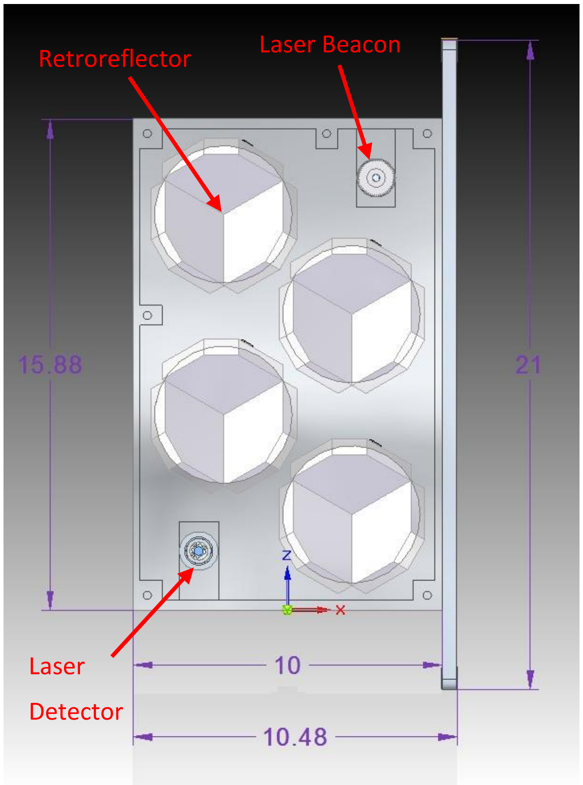

Figure 2: Front view of payload showing arrangement of components; dimensions are in centimeters.

Figure 2 shows the arrangement of the optical components as well as some critical dimensions of the mechanical structure. In order to physically bolt to the spacecraft payload deck, an adapter plate flange was designed for the payload. The adapter plate is machined as part of the payload, not attached with fasteners. Figure 2 and Figure 3 show the approximate 1.5U form factor of the payload. The payload is slightly longer than 15cm in order to allow for ample retroreflector spacing during the high stress launch conditions. The retroreflectors are mounted in a staggered position to optimally fill the payload and allow for movement during vibration to prevent the retroreflectors from contacting each other and the payload walls. The laser beacon is mounted directly to the flange side of the main structure as a means for direct conductive cooling to the host spacecraft payload deck. Remaining space is allotted to the laser detector which is positioned as shown.

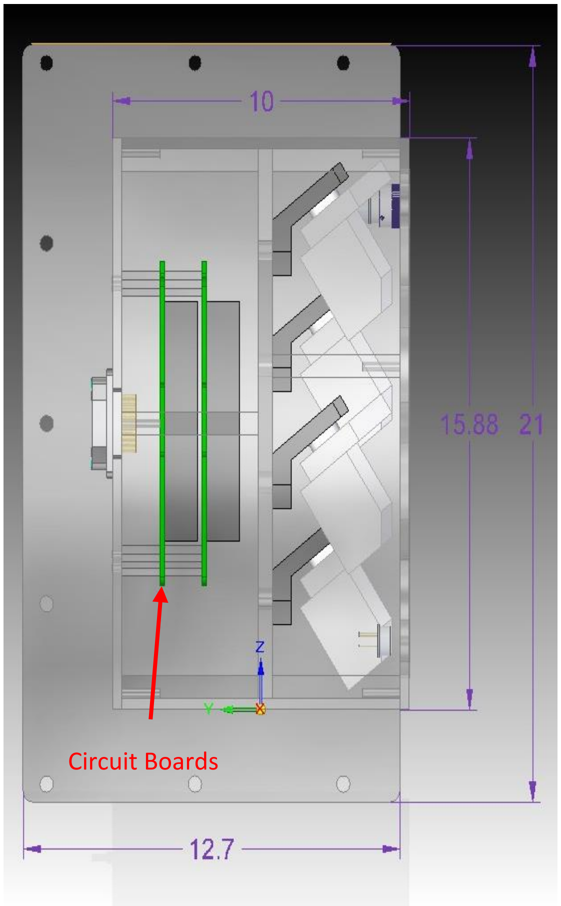

Figure 3: Top-down transparent view showing divider partition and mounting of retroreflectors and circuit boards; dimensions are in centimeters.

Figure 3 shows a transparent view of the payload structure, a divider plate separates the optical components from the electrical components which include the flight computer, power distribution board, and 15 pin D-sub connector. The divider plate serves as a mounting surface for the retroreflectors as well as reinforcement for the strength of the payload. Modal analysis of the model shows that the natural frequencies of the structure are at the far end of the vibration power spectral density (PSD) profile; approximately 2kHz. The PSD profile declines at higher frequencies near the far end, indicating that the mechanical structure will experience minimal fatigue stress. To ensure the optical and electrical devices don't vibrate loose, self-locking fasteners will be used.

An initial estimate of the overall weight is 3.5 lbs, which is well under the imposed weight limitation of a CubeSat this size. Accordingly, the payload volume measures 21 cm x 12.7 cm x 10.5 cm which is also well within the volume limitation.

POWER CONVERSION CIRCUIT BOARD

The primary role of the power conversion and analog processing board is to provide the necessary voltage levels to operate the various electronic elements of the payload. Additionally, this board acts as an interface between the analog sensing and communication systems, and the digital processing which occurs on the flight computer. A block diagram of the various power conversion circuits is shown in Figure 4. From a power conversion point of view, the primary input for the power conversion board is the DC voltage from the host spacecraft. This system was designed for a DC input voltage of 28V (±6V) capable of sourcing 2A of current.

The power conversion and analog interface board is partitioned into several functional blocks including power conversion, power and temperature monitoring, and an analog/digital interface. Additionally, there is a connector to the flight computer which provides power and communications between the two sub-systems.

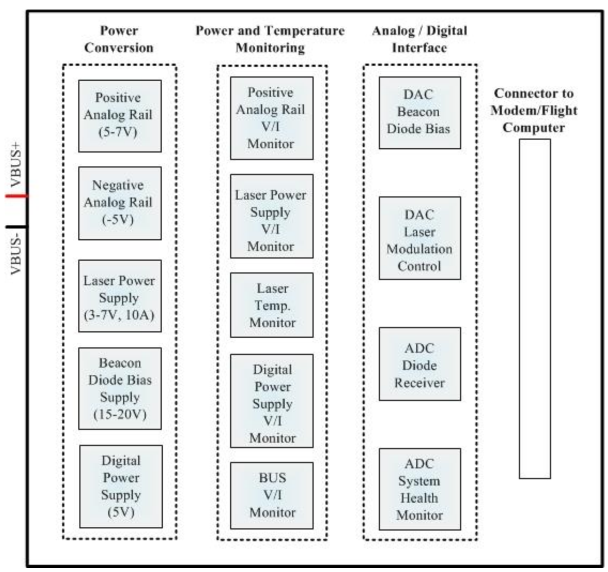

Figure 4: Block diagram of power conversion circuit.

The power conversion block provides regulated power for all of the analog and digital systems, and is comprised of 5 separate power converters. Positive and negative power rails, nominally set at ±5V, provide power for the majority of the analog components. The negative voltage is included to allow for amplification of small signals close to ground. Power for the laser beacon is provided on a dedicated converter, due primarily to the large current draw and potential for noise as the laser is modulated. The diode receiver requires a higher voltage level than the other components in order to be properly biased. This is realized with a separate converter. Finally, a 5V digital power rail for the flight computer is provided. Special care was taken to ensure that the potentially noisy digital ground was isolated from the analog ground.

The power and temperature monitoring block allows the flight computer to monitor critical system voltages, currents, and temperatures. Both voltage and current are monitored for the analog rail, digital rail, laser power supply, and the input bus voltage. The temperature of the laser is also monitored as this is directly related to the modeled laser performance. Thermal management of the laser is a concern, and monitoring the laser temperature allows it to be shut down if necessary

The analog/digital block provides the interface to the digital processing of the flight computer. Two digital to analog converters (DACs) are included on this board. One is used to drive the laser beacon intensity and the other controls the bias voltage on the diode receiver. Two analog to digital converters (ADCs) are also present. One ADC is used to digitize the optical signal received by the photodiode for processing by the flight computer. The other ADC is multi-channel, and provides the flight computer with all of the telemetry data from the power and temperature monitoring block.

FLIGHT COMPUTER

The payload has a single processer board to carry out all computing functions. The processor board serves as the flight computer to interface with the host spacecraft, a data acquisition module to record the telemetry and optical detector data, and a modem to modulate the laser beacon. The interface between the host spacecraft and the payload is RS-422 serial. Over the RS-422 communication interface, commands are issued to the payload by the spacecraft and payload data are sent to the spacecraft computer. Two hardware implementations are being considered: an FPGA-based solution and a microcontroller solution based on the Beagle Bone Black [2].

The processor board performs three functions, data acquisition, command and data handling, and an optical communications modem. The primary purpose of the processor board is to record raw data sampled from the optical sensor and transfer it to the host spacecraft. Second, the processor board operates as a payload command and data handling (C&DH) flight computer. As a flight computer the processor board monitors I/O, services system interrupts, interprets commands issued by host spacecraft, and handles telemetry. Third, as a modem, the processor board demodulates data received from the optical receiver and modulates the optical beacon. For the envisioned MRR payload, the processor board will function as the modem to modulate the MRR with data to be downlinked.

The data collected during an experiment is stored in nonvolatile memory on the processor board as well as sent out to the spacecraft in real time. The data is transferred to the spacecraft during the experiment and for a predetermined time afterward to ensure a complete data transfer. The data recorded and stored on the payload will be maintained until completely transferred to the host spacecraft. Upon request, all or some of the experimental data can be retransferred to the host spacecraft. Payload data remains on the spacecraft until it is able to be downloaded via the RF channel. Figure 5 shows the data flow between ground and payload.

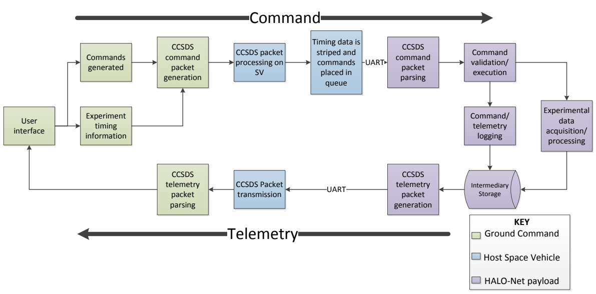

Figure 5: Command and Telemetry Concept of Operation

As a hosted payload, real time control and communication with the payload will be unavailable. Accordingly, the experiment must be planned ahead of time and all commands must be pre-planned in a time-tagged schedule. The schedule will be uploaded to the host spacecraft at least 24 hours ahead of engagement with the optical ground station. The command schedule will be aggregated onboard the host spacecraft and commands will then be issued to the payload according to the time tags for real time execution. Figure 6 illustrates the communication chain with relevant interfaces.

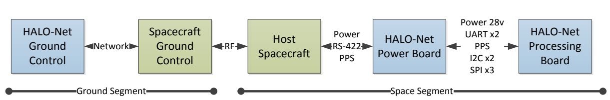

Figure 6: Communication Path Interface

All communications between the host spacecraft and the payload will occur over RS-422 UART using a CCSDS recommended standard protocol [3]. Figure 7 illustrates the command packet structure. All command data transmitted from ground control to the host spacecraft for the payload must fit within the 223 Byte Packet Data section. The Telemetry packet structure is similar but has a larger data payload section of 1110 Bytes.

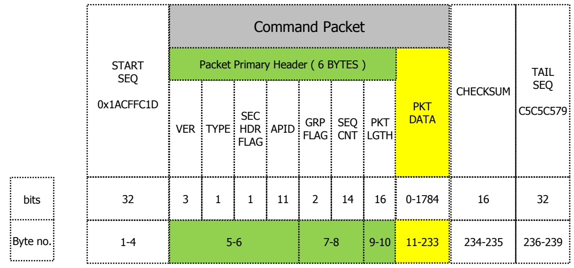

Figure 7: Payload CCSDS Command Packet

The processor board collects experimental digital data over serial peripheral interface (SPI) from the ADC on the payload power conversion board. The data is then down sampled and processed for storage. CCSDS Telemetry packet generation begins when 1024 Bytes of data have been collected. The packets are numbered and placed in a queue to be transferred to the host SV over RS-422. A copy of the packets is retained on board in nonvolatile memory in case of error during transmission and can be requested by the ground control station to be retransmitted. The packets are erased on a first in first out basis to free up memory.

C&DH is part of the supervisor module; it starts tasks, monitors I/O, maintains a command parser and telemetry packet generator, collects health and status data, and "pets" the watchdog timer. Health and status telemetry is collected periodically based on the operational state of the C&DH module. Voltage, current, and temperatures are collected from every voltage domain on the power conversion board as well as temperatures of processor board memory and the CPU. The system health and status block is added to every telemetry packet that is transmitted to the ground. If there was no experimental data collected, only health and status information is stacked into a telemetry packet.

A time reference is required between the host spacecraft and the payload to time stamp experimental data. The time reference used by the payload is kept by the host spacecraft and is made available over RS-422 on boot and upon request. This timing counter is synchronized once per second by a differential pulse per second (PPS) from the host spacecraft. The time is generated from GPS and has an accuracy of +/- 0.5 milliseconds.

Two flight computer solutions are being pursued: a low power microcontroller solution and a more capable FPGA solution. The processor board based off the Beagle Bone Black is a relatively un-modified commercial off the shelf processor that resides on a PC/104 form factor motherboard. The processor board has a 1GHz ARM processor, 512 MB industrial RAM, and 32 GB of onboard flash memory in addition to a removable SD card. The processor board based off the FPGA is a custom board with a microcontroller to act as the flight computer and an FPGA to offload intensive processing. The FPGA would be desired in the envisioned MRR payload to carry out the modem functionality.

RETROREFLECTORS

The heart of this experimental payload is in the retroreflectors. The primary risk in implementing an MRR communication payload is the unknown performance and survivability of the retroreflectors in a space environment. Although the reported payload is a surrogate, it was designed to be as close as possible to the envisioned MRR payload. Therefore, the physical payload size will limit the clear aperture area available for the retroreflectors, optical detector, and optical beacon, i.e. the diameter of the retroreflector and optics. Furthermore, the retroreflector will be restricted by the MEMS mirrors to be used on the final modulated version. These factors must be taken into consideration when designing the optical systems for the highest optical returns and overcoming velocity aberration.

The most critical parameter in the retroreflector design is the diameter, as this is proportional to the amount of light returned. The two most reasonable options were one large 3-inch diameter or four 1.5-inch diameter retroreflectors, which have equivalent surface areas. Utilizing models developed and presented in our previous paper, these options would have slightly different return patterns [1]. The larger retroreflector would have a narrower, more intense beam, while the smaller retroreflectors would have a less intense, more spread out beam.

Both options would require custom retroreflector engineering in order to overcome velocity aberration (illustrated by the black ring in Figure 8). Alternatively, a very weak lens could be placed in front of the retroreflector to diverge the output beam thus creating a wider beam at the ground station. This would account for velocity aberration, but such a weak lens is difficult to manufacture and physical mounting would be challenging for a hollow retroreflector [4]. As mentioned in our previous paper, a hollow retroreflector can be spoiled by intentionally canting the mirrors from the orthogonal configuration. If all three mirror faces are equally canted, then the illumination on the ground will be approximately a ring of six diffraction-limited beams. The size of the ring will be determined by the amount of canting for the three mirrors. [5] Additionally, the retroreflectors could be clocked, or rotated about their optical axis, to smooth out the difference between the peaks and the valleys of the return illumination pattern. Figure 8 shows the returned two-dimensional spatial intensity pattern at the ground station for an array of four 1.5-inch retroreflectors with no spoiling, uniform spoiling at 13.5-microradians, and uniform spoiling of 13.5-microradians with each retroreflector clocked 30 degrees.

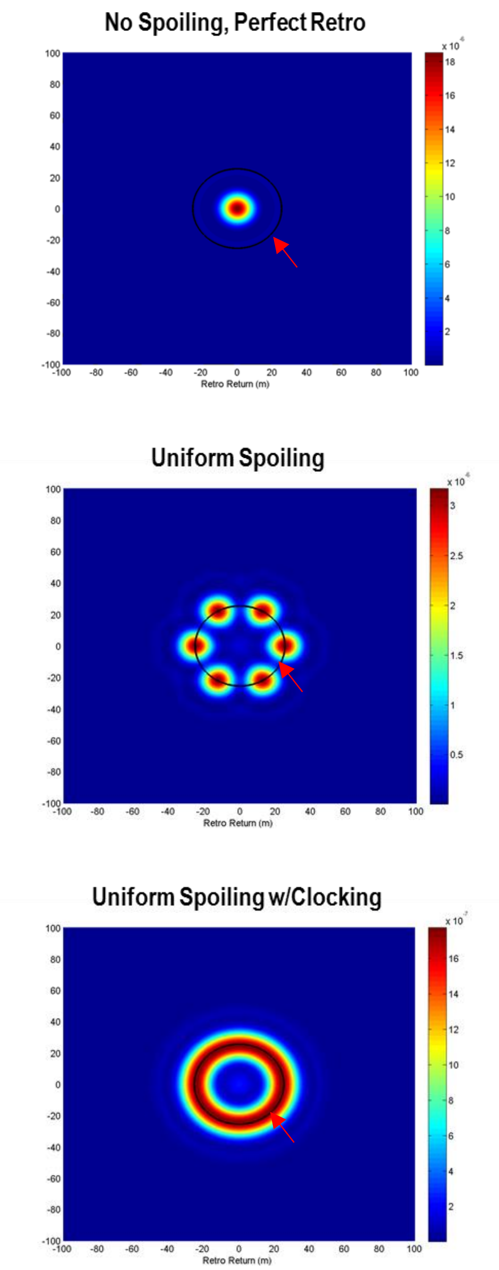

Figure 8: Optical return patterns from an array of retroreflectors: no spoiling, equal spoiling, clocked; the red arrow points to the retroreflected return due to velocity aberration.

After exploring the system tradeoffs, the decision was made to use four 1.5-inch retroreflectors with custom spoiling angles. The cost of engineering more retroreflectors was offset by the mitigation of several risks. First, having four retroreflectors provides redundancy. If any part of one retroreflector fails (the mounting, adhesives, or modulators in the MRR payload), there are three others that will be functional. Second, spoiling and clocking the retroreflectors will create a more uniform and more intense return illumination at the ground station, which makes it easier to close our link budget. Based on our previous models, spoiling the retroreflectors at a 13.5-microradian dihedral angle will yield the highest average return intensities over the anticipated engagement angles for a satellite in low earth orbit. Clocking the retroreflectors will create a more even output and make detection of the return beam highly tolerant to rotation of the spacecraft about the optical axis. Finally, the engineering challenges will be greatly reduced for the future integration of the MEMS modulating mirrors to enable the MRR. Currently, smaller sized MEMS modulators are easier to manufacture and more robust.

LASER BEACON

In order to facilitate precise pointing, acquisition, and tracking of a satellite, it is advantageous to have an optical beacon onboard the satellite. The optical beacon is commonly an LED or a laser, but could be any light source so long as it has a finite wavelength that the ground station can use to discriminate it from glints. For this payload we chose a laser over an LED in order to achieve narrower beam divergence and higher spectral purity. A laser also provides comparable electrical –to-optical power efficiency, high optical output powers, and efficient fiber optic coupling. The laser selected is a fiber coupled diode laser, operating at 808nm, capable of 15W optical output. 808nm is a very common laser diode wavelength and therefore offers a wide range of vendor options. Additionally, this wavelength is very close to the peak responsivity of a Silicon camera, which will be used for beacon tracking with the ground station. A MODTRAN analysis was performed to estimate the atmospheric transmission around 808nm. On average, the analysis shows the atmospheric transmission within a 10nm bandwidth centered at 808nm to be approximately 60%. The bandwidth around 808nm was relatively smooth, unlike the transmission of alternative common high power laser diodes around 975nm.

A fiber coupled laser is preferred as it adds flexibility in the placement of the laser, reducing mounting complexity. The laser diode is the primary power consumer for the payload and consequently the primary source of heat. As mentioned in the mechanical section, it is desired to conductively mount the laser diode to the flange of the payload to allow for optimal thermal management. A fiber optic output allows the laser diode to be mounted anywhere in the payload and the fiber to be routed to the optical system. The optical system for the laser beacon was designed to output a 2-degree full-angle beam. As mentioned in our previous paper, we expect the CubeSat platform to be able to actively point at the ground station to better than 1-degree during an engagement. By restricting the satellite platform motion, a 2-degree divergence adds a safety margin for any unexpected platform jitter.

OPTICAL RECEIVER

The detector serves as an optical uplink channel to the satellite. In the MRR payload, it would enable low data rate commanding of the payload via an optical channel. For this surrogate payload it will serve a scientific purpose to record the raw illuminated signal to provide insight into the scintillation effects of the illumination laser. The illumination laser wavelength will be approximately 1060nm, which allows the detector to be made of several materials. Two common detectors exist for this wavelength: Silicon (Si) which is sensitive to the visible and a small portion of near infrared spectrum and Indium Gallium Arsenide (InGaAs) which is sensitive to only the near infrared spectrum. Although a Silicon detector is less efficient at 1060nm, it was chosen to allow a relatively large detector diameter of 2.5mm.

An optomechanical system was designed to mount the detector, a lens, and a filter in a single assembly. The optical design consists of a single UV fused silica lens yielding a five degree full angle field of view for the detector. A long-pass color glass filter is used in front of the detector to help block visible light. Also, the large detector ensures the entire beam is captured if it is broken up, smeared, or wandering due to the turbulence along the uplink path.

FUTURE WORK

Plans for next year include integration and testing of the subsystems discussed in the paper, environmental testing of the payload, and additional focus on the ground station. As the subsystems finish individual benchmarking, they will be integrated into the mechanical structure. Individual components as well as the integrated system must pass thermal cycling in vacuum operation, outgassing, shock, and vibration testing prior to launch. The laser to interrogate the payload has been purchased and will be setup and tested in the lab. An optical system will be designed and implemented to expand the beam to a workable diameter. The laser system will also be integrated into the tracking system and telescope.

CONCLUSION

In this paper, we summarized current progress towards an MRR optical communication payload for a CubeSat. The reported payload will act as a surrogate to our envisioned MRR communications payload. The experimental data collected with this payload will provide valuable data to mature and validate spacecraft and optical communication link models.

ACKNOWLEDGMENTS

This work was supported and funded by the SPAWAR Systems Center Pacific Naval Innovative Science and Engineering (NISE) Program.

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