Microsatellites—A Light in the Darkness

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ABSTRACT

Digital Fusion Solutions, Inc. is currently developing a microsatellite payload for the US Army Space and Missile Defense Command that will serve as an exo-atmospheric infrared calibration source for a variety of ground and airborne infrared sensors. In addition to the primary mission, it will also serve to demonstrate a low-cost satellite development approach. This mission is well-suited to microsatellites because it requires low average power, limited pointing control, and modest pointing knowledge. Challenges for this active infrared payload include achieving the required radiometric accuracy, power handling (high peak power at low duty cycle), and ensuring coverage of airborne sensors. Payload design, performance testing, and initial environmental test plans will be discussed.

INTRODUCTION

Small and microsatellites have enjoyed increasing attention in the last few years. Interest in responsive space missions has grown as spacecraft component technology has reaped the benefits of miniaturization and mission demands have put more pressure on shorter acquisition timelines. Digital Fusion Solutions, Inc. is currently developing a microsatellite payload in support of the US Army Space and Missile Defense Command The goal is to develop a microsatellite (SMDC). payload to meet SMDC requirements to support lowcost, on-demand access to space. The focus is on reducing the high cost of satellites through employing a modular design, using commercial parts, and applying Pareto's 80/20 Rule (in this context, achieving the first 80% of the capability requires 20% of the resources). A survey of potential payloads was conducted, considering mass and power limitations of a 50 kg microsatellite, simplicity needed for the first payload, and value to the end customer. Although the focus of this task is on providing R&D for the space environment, much effort has been undertaken to also provide operationally useful capability. Results of the initial satellite design efforts have been documented previously,¹ and the purpose of this paper is to present the results of the detailed payload design.

Based on feedback from SMDC, the payload selected was an exo-atmospheric IR calibration unit to assist in calibrating Missile Defense Agency (MDA) sensors, such as the High Altitude Observatory (HALO) airborne testbeds. The HALO testbeds are Gulfstream IIB aircraft operated for MDA by Tulsa, OK based L-3/Aeromet.² The aircraft are used to support tests of the Ballistic Missile Defense System using a variety of electro-optic and infrared (IR) sensors. The sensors provide high-accuracy acquisition, pointing, and tracking to collect single and multi-band IR to support mid-course and terminal missile defense tests.³ The microsatellite payload requirements were focused on simplifying the calibration procedure currently used by the HALO II crew, while providing the same level of precision provided by the current techniques.

SATELLITE OVERVIEW

One area of emphasis for this program is developing a low-cost microsatellite design to support Army science and technology testbed needs, and the concept design has been published in Ref. 1. For simplicity, the satellite has been envisioned using a gravity gradient boom for stabilization. Once a satellite has been slowed to a minimum rotational speed following deployment from the launch vehicle, a gravity boom provides a simple, effective, low-power means to maintain system stability for a nadir pointing spacecraft. However, the gravity boom typically only provides pointing accuracy to within about 5° because of the variation in atmospheric drag and coupling between the non-uniformities in the gravitational field and the pitch of the satellite. A model of the satellite concept is shown in Figure 1.



Figure 1. Early concept design for payload attached to microsatellite bus; launch configuration (left) and on-orbit configuration (right).

In order to avoid real-time commanding of the satellite and three axis control to point the IR source toward the sensor, the concept of operations calls for the payload to slowly rotate about the vertical axis. The footprint of the IR emission then sweeps out a "donut" shape on the ground. As the satellite orbits, repeated sweeps will bring the center of the pattern across the sensor at increasing irradiance levels.

Two methods have been considered for spinning the payload: rotating the entire satellite or rotating just the payload. The long gravity boom raises a challenge for rotating the satellite. Objects freely rotating about the axis with the lowest moment of inertia will eventually become unstable due to the inevitable conversion of rotational kinetic energy to vibrational energy, reducing the angular speed during this process. The axis of rotation will slowly change to one with a higher moment of inertia so that the angular momentum remains constant. Furthermore, the angular momentum vector of the satellite tends to remain pointed in a constant direction, while the gravity boom tends to rotate as the satellite orbits, so that the boom remains pointed toward zenith. These effects can result in instability of the satellite. Finally, the reaction wheels required to spin the satellite are reasonably expensive components (typically about \$100k each for the size needed here). On the other hand, rotating the payload significantly increases the complexity of the payload

and the satellite. Instead of a rigid interface, a slip ring would have to be used to pass power and data to the payload. However, in order to spin up the payload in a sufficiently short period of time, the satellite will need actuators to keep it from rotating in the opposite direction. Cold gas thrusters have a very limited supply of propellant and magnetic torque rods do not supply enough torque. Therefore, the apparent solution is again reaction wheels. If reaction wheels are to be used anyway, spinning the satellite appears to be the better approach. More detailed analysis will be done with the bus designers to reach a final solution.

The launch vehicle anticipated for this program was the Super-Strypi, which is being developed by Sandia National Laboratories.⁴ The SMDC payload will share the initial launch with a University of Hawaii payload being developed under their LEONIDAS program. Thus, although the payload capacity of the Super Strypi is expected to exceed 200 kg, the mass allocation for the SMDC microsatellite is only 50 kg. Of that 50 kg, 15 kg has been allocated to the payload and 35 allocated to the bus. However, both the payload and the bus have been designed to accommodate the widest possible set of launch vehicles, including the SpaceX Falcon 1 and secondary payload adapters for the Evolved Expendable Launch Vehicles (ESPA) and the Ariane 5 (ASAP). This was done to enable maximum flexibility in launch vehicle selection for future applications of the bus and payload.

PAYLOAD OVERVIEW

Currently, the calibration function assigned to the microsatellite is accomplished using stars. Stars are often used in space environments for calibration purposes. Stars behave very much like blackbody sources and their emission spectra may be approximated as a blackbody. Ground-based and airborne platforms collect information on well-known stars to provide absolute calibration information.⁵

Stellar calibration has performed adequately to date, however there are certain aspects of using stars for calibration that create difficulties for telescope and sensor platforms. Calibration generally requires the use of several intensities to obtain a correction curve. A star emits only one particular radiant intensity value; additional values must come from other stars. Obtaining irradiance data from multiple stars can become logistically challenging to mission planners and operators. The time of day and the season must be considered in determining which stars are appropriate for calibration. Furthermore, the platform must often be repositioned in order to place a particular star into the system's field of regard. The ability to obtain all system calibration data at one location would be beneficial. A satellite platform with a controllable infrared source would decrease the logistical challenges associated with mission planning and save valuable time during calibration. For airborne platform systems, this can translate to less money spent on flight time or more time available for data collection.

Key characteristics for the calibration source should include low cost, technical suitability, repeatability, and availability. Although not driven by cost alone (or even primarily), the cost of the calibration source should be offset by savings in operations cost (at several thousand dollars per hour) to the maximum extent possible. Technical suitability is based on the range of radiance values provided by the source and the accuracy required, both of which will be sensor specific. Repeatability is closely tied to accuracy, but not Ideally, the source should provide svnonvmous. identical values of irradiance with each use; however, excursions from the average irradiance are acceptable so long as the excursions are well characterized and still provide values in the needed range.

Several different types of IR sources were considered in this research for testing purposes. These included thermal emitters, quantum cascade lasers, quantum dot lasers, optical parametric oscillators, and a hydrazinebased hot plate. Quantum cascade lasers do not emit in a broadband blackbody spectrum so these devices would not generally be a viable option, as the sensors generally need to be calibrated in multiple IR bands. The quantum dot laser and the optical parametric oscillator also did not meet requirements for a broadband emission spectrum. In general, solid state electro-optic sources were eliminated due to the difficulty of obtaining broadband IR radiation from these devices. The hydrazine-based hot plate was eliminated due to cost resulting from added complexity and safety precautions required for incorporating hydrazine into the payload. In addition, use of the expendable propellant would place a strict limit on mission life.

The conclusion of the trade study effort was that Kanthal®-based wire-wound thermal emitters were the current best choice. These infrared emitters generate a blackbody spectrum by heating up a filament with the application of electrical current. A number of commercially available Kanthal® emitters were reviewed, and a commercial emitter from Hawkeye Technologies, LLC was chosen for its power handling capacity, mass, and construction (see Figure 2).⁶



Figure 2. Photo and drawing of Hawkeye IR emitter.

In order to obtain the irradiance values needed, the IR radiation from the emitters had to be concentrated using parabolic reflectors. The reflectors available from Hawkeye were unsuitable for a number of reasons, and electroformed reflectors were chosen instead. Based on initial analytical modeling and ray tracing results, a specific commercial reflector was acquired for characterization studies.

EMITTER CHARACTERIZATION

Surprisingly, detailed characteristics of the Hawkeve emitter were scarce, so a significant effort was placed into characterizing the emitter. The purpose of this section is to give an overview of the characteristics; details are available elsewhere.⁷ The Hawkeye emitter has been in use for over twenty years, and consists of a coiled Kanthal® filament that is wound around a grooved cylindrical alumina substrate. This grooved substrate provides electrical isolation between the windings and produces a more uniform radiating source.⁸ Important aspects of the emitter that were studied included the thermal characteristics, stability, dependence of radiance on the voltage, and thermal power lost to modes besides radiation. Tests were first conducted in the lab and in a vacuum chamber using the 1" reflector available from Hawkeye, then in the lab using the 4" electroformed reflector.

Emitter Thermal Modeling

To better understand the distribution of energy, a rudimentary model was developed to calculate the energy going into each of the three possible modes: radiation, convection, and conduction. The emitter was treated as a cylinder with conduction occurring in one dimension along the axis of the cylinder and radiation and convection occurring normal to the surface of the cylinder using a finite difference algorithm. Conduction through the posts was calculated as a point sink. The energy absorbed by the sink was equal to the energy conducted down the posts, calculated analytically assuming one dimensional thermal conduction with one end of the post at the temperature of the emitter and one end at a fixed temperature. Radiation was modeled by treating the emitter as a cylinder, using the emissivity for Kanthal (0.7) and alumina (0.2) in the Stefan-Boltzmann law. The model accounted for the energy input to the emitter in the form of Joule heating $(P=I^2R)$ only (ignoring energy gains from the environment). The resistivity of Kanthal has a weak temperature dependence, which could result in non-uniform heating. However, the model assumed the input power was uniformly distributed within the Kanthal, and the final temperature profile showed the assumption was accurate. The most challenging aspect was modeling the convection, where an empirical model for natural convection from a horizontal was used.⁹ The cylinder was divided into 20 longitudinal elements, and the temporal step size was adjusted to ensure the finite element solution remained stable. At the anticipated power levels (6-8 W), less than half of the input power was radiated. About 40% of the wasted power went into convection and 60% into conduction through the support posts. In vacuum, no power was lost to convention, but the model predicted about half of the input power was still lost to conduction. Therefore, after consultation with the manufacturer, a new design was developed using posts from a different product. The modified design used longer, thinner posts, which substantially reduced the conductive loss. With the new posts, only about 5% of the input power was conducted down the posts, reducing both the power requirements and thermal management issues. Having established the emitter design, the emitters were then characterized first in the lab and then in a vacuum chamber.

Initial Laboratory (Atmospheric) Testing

All radiometric measurements were accomplished using a pair of IR imaging radiometers (essentially IR cameras) that were calibrated by imaging standard blackbodies at known temperatures. For all measurements, an independent two-point nonuniformity correction was applied to every pixel in the image. This resulted in an image based on quantitative radiance values, and an example can be seen in Figure 3. The cameras were identical, using 256x256 HgCdTe focal planes, cooled to 77 K with liquid N2. However, one set of optics was designed for use in the long-wave IR (LWIR) and one set for the mid-wave IR (MWIR). For all tests, data was collected at 30 Hz for 1 second in order to reduce the effects of shot noise from the focal plane. The radiance image (in W/cm²-sr) was then integrated to obtain the radiant intensity (in W/sr).

Although it was known the 1" reflectors were not suitable for the final payload, they were used for emitter characterization testing. This choice was driven by the size of IR window available for vacuum chamber testing. The clear aperture of the ZnSe window was only 1.9". Given the minimum focusing distance on the camera and size of the vacuum chamber, the entire 4" reflector could not be imaged through the window. In order to avoid vignetting, the smaller reflector was used.



Figure 3. Raw MWIR image and same image corrected for non-unifomity.

The stability test was conducted to determine the consistency of the radiated power. Given the environmental changes that cannot be completely controlled and the presence of convection in air, it was recognized this result would be a worst case bound for the realistic conditions in a vacuum chamber. However, it provided the opportunity to test the procedures before attempting them in the vacuum chamber. The test setup was designed to replicate the procedures to be used in the vacuum chamber, including the use of a 1.9" ZnSe viewport. The measurements showed that the radiance of the emitter varied by about 0.7% or less over a period of 1.5 hours.

Another test was performed to determine the functional dependence of the radiant intensity upon the voltage applied to the emitter. This dependence would form the basis of the trade study to determine the optimum operating voltage. The power radiated from the emitter is a complex function of the IR radiation, conduction through the posts holding the emitter, and convection into the surrounding air. The results are shown below in Figure 4 for both the laboratory measurements and the vacuum chamber measurements (discussed below). Key points extracted from this test are that (1) the MWIR radiation rapidly exceeds the LWIR, making the LWIR requirements the most stressing for this approach and (2) increasing the voltage beyond about 4-5 V is not an effective means for increasing the LWIR output. At this point, the increased electrical power is radiated in the MWIR band and little increase goes into the LWIR band.



Figure 4. Graph showing how radiant intensity depends on operating voltage

Vacuum Chamber Testing

A vacuum chamber at the National Space Science and Technology Center in Huntsville, AL was used to characterize an emitter in conditions representative of those in orbit. The emitter performed largely as expected based on the lab testing and modeling. The emitter was hotter in the vacuum, although not quite as much as predicted by the simple model discussed above. This is most likely due to a discrepancy in the thermal convection model due to the parameters falling at the limits of the empirical model. In vacuum, the emitter required substantially longer to reach thermal equilibrium, about 2.5 min, compared to < 1 min in air. It was also more stable, varying <0.3% over the 1.5 hour test.

Reflector characterization

Having characterized the performance of the emitters, the next phase was characterizing the performance of the larger 4" reflector in combination with the emitters. In order to achieve the needed radiometric accuracy, including impact of the accuracy of the pointing knowledge of the spacecraft, the profile of the radiant intensity needed to be about 23° full width at half maximum. The profile of a single reflector with the emitter at the focus was much narrower, so two options were investigated. The first approach was to combine the reflectors with slight angular offsets, resulting in a broader total beam. However, to achieve sufficient beam width, the offsets resulted in a very irregular pattern. The second option was to shift the emitter from its location at the focus of the reflector, resulting in a broader beam. While this did produce significant broadening, it also produced a very irregular profile near the center of the beam, as shown in Figure 5. The final design merged both concepts, by tilting each reflector by 2° in both azimuth and elevation relative to

the adjacent reflectors. This approach resulted in a smooth, approximately Gaussian beam profile.



Figure 5. Radiant intensity (W/sr) for an emitter at the focus of a reflector (color) and shifted 5 mm along the axis (gray) as a function of orientation angle (degrees).

PAYLOAD DESIGN CHALLENGES

As with any spacecraft design, a number of challenges had to be overcome. Of these, the most important were size, weight, and power (SWaP); optimizing performance based on a point design; and ensuring the resulting design could be space-qualified. All of these challenges have been overcome, some to a greater extent than others.

Size, Weight, and Power

Because the payload is being designed in advance of the bus, a number of final design requirements remain to be determined. Therefore, the ultimate SWaP limitations have been stated in terms of goals rather than firm requirements. Payload size was the least restricting, based on early designs of the Super-Strypi fairing. The payload diameter is required to be less than 43", allowing 87" for the height of the spacecraft, the LEONIDAS spacecraft, and launch vehicle adapter.¹⁰ Alternative launch vehicles, such as the Falcon I, have also been considered. The Falcon I payload volume is larger in both height and diameter than the Super-Strypi.¹¹ The Pegasus, however, is slightly smaller and may require modification to fit, depending on the size of the size of the other payload elements.¹² Fitting the payload in the envelope for ESPA proved to be the most difficult,¹³ and support for using ESPA for this program was eventually dropped.

In order to meet the design goals of the low-cost bus concept, the payload mass needs to be approximately 15 kg. Again, this is not a firm requirement, but is the value we have targeted in the design. Because of the active nature of the payload, power was a very stressing requirement. The initial bus concept allocated 2.7 W daily average power to the payload. The initial system performance goal was two calibration operations of 10 minutes per day. A total of 20 minutes of operation per day allowed the goal payload power to be set at 200 W. The threshold performance was one operation per day. Reducing the daily operating time to 10 minutes boosted the threshold power to 400 W. Additional analysis has relaxed the duration of the operation to 8 minutes, allowing for additional power margin if required.

Optimization trades

The point design from which trade studies commenced is shown in Figure 1. The trade study was based on maximizing the overall utility of the system, based on the following assumptions:

- Orbital dynamics was approximated as a two-body interaction with a circular orbit and geoidal Earth.
- A contact was defined based on the irradiance at the sensor exceeding the minimum value (threshold or goal) for a specified duration.
- The utility of the contact was evaluated using the following weights: Target areas outside the continental US, 1.0; targets areas inside the continental US, 0.8; aircraft maintenance site, 0.2.
- The total utility over an extended period was calculated by summing the utility of each contact.

The trade study was conducted by approximating the emitter performance with a polynomial that determined radiant intensity as a function of applied voltage. The irradiance at the sensor was calculated based on the slant range between the target and the satellite, with approximate effects of the atmosphere based on the aircraft altitude and calibration source elevation angle included. Atmospheric effects were modeled using 40 kft altitude at 30° latitude during summer with 5-23 km visibility). The design of the payload was varied by adjusting the angle between the normal to the array and the nadir direction (henceforth referred to as the array normal). Finally, the orbital altitude and inclination were varied. The resulting four-dimensional trade provided the following conclusions. As expected, the optimal inclination was near the 30° latitude of the northern target sites. Likewise, the optimum value of array normal was almost independent of the orbital inclination, as shown in Figure 6. The trades between the altitude and array normal proved to be very interesting. As the altitude decreased, the array could be pointed more horizontally while maintaining a constant slant range. This trade provided a more favorable geometry for targets further away, increasing

the number of possible contacts (and hence utility). However, at lower altitudes, the region accessible to the satellite decreases. Because of the high altitude of the sensors, atmospheric losses are minimal. The decrease in optimum array normal with increasing altitude is demonstrated in Figure 7. With the inclination fixed at 30° , this figure shows the optimum value for array normal decreased with increasing altitude. Furthermore, the lower the altitude, the higher the net utility.



Figure 6. Net utility score for 300 km altitude.



Figure 7. Trade between array normal and altitude.

Eventually, the decrease in the region accessible to the satellite would dominate and the utility would decrease with decreasing altitude. However, the optimum altitude appeared to be limited by atmospheric drag on the spacecraft. In order to maintain a 1-year mission life, the lower altitude limit was set at 300 km, which was the optimum value found in our study.

A separate trade study was performed to determine the optimum voltage and number of emitters to obtain the For this application, the LWIR required irradiance. band is the limiting case. The peak of Planck's blackbody radiation curve occurs in the LWIR band at about 90 °C. To get maximum electrical power efficiency, the emitters should operate near this peak. However, the amount of energy at this temperature is so small that the size of the payload would be prohibitively large. Using the relationship developed during the characterization tests for the intensity of a single emitter as a function of applied voltage, a relationship for intensity as a function of electrical power and number of emitters was derived. This relationship, illustrated in Figure 8 for a fixed power of 400 W, demonstrates that a larger number of emitters provide greater intensity for a given rate of electrical power consumption. Therefore, the design was based on the largest number of emitters that could be achieved within the 15 kg total mass goal.





The final payload trade was for the electrical design. To provide electrical power, the 28 V unregulated bus voltage would have to be converted to regulated voltage by one or more DC-to-DC power supplies. A number of commercial power supplies with space heritage were available, none of which could power the entire array Therefore, a design was developed which alone. combined a string of emitters in series, with multiple strings arranged in parallel. While the number of emitters was fixed, the arrangement in series and parallel was subject to trade. A larger number of strings provided for graceful degradation in the event of a failure, but required additional materials and labor for wiring. Within a family of power supplies, there was little variation in output power, efficiency, or mass.

Furthermore, the power supplies represent a very small portion of the overall payload mass (<5%). Therefore, the design was chosen based on simplicity, cost, and part availability.

Space environment qualification

Environmental testing for space qualification was planned to certify the design for launch, predominately static and vibration load environments; thermal and vacuum environments: and atomic oxvgen environment. In order to maximize flexibility pending final fabrication of the Super-Strypi and the associated uncertainties, the test envelope included a variety of systems, including the Super-Strypi, Falcon I, Pegasus, and secondary payload adapters for the Delta IV and Ariane 5. Each vehicle represents a unique challenge in particular regimes. For example, the Super-Strypi, being spin-stabilized, experiences relatively low lateral g-forces along the axis, but 16 g of centripetal acceleration near the circumference of the payload fairing. Conversely, the Pegasus has maximum axial load of up to 13 g, while most vehicles are below 7 g. A comparison of the lateral and longitudinal loads is shown in Figure 9 for the primary vehicles of interest. Another example serves to demonstrate the variety between launch vehicles. The Ariane 5 had the highest requirement for acoustical loads (at most frequencies) and random vibration loads, despite its very modest axial load. This variety is shown below in Figure 10 and Figure 11. For each type of test, the test envelope was developed which included the worst case of all the potential vehicles, to ensure the payload could be launched on any vehicle and to address any potential changes to the Super Strypi because the payload design is occurring early in the booster design phase.

RESULTING DESIGN AND TEST PLANS

Mechanical

The mechanical design for the payload is shown in Figure 12. The bi-pod structure has proven to be robust, but required substantial interaction with machinists to develop a design that was also affordable to produce. At the time of this writing, the combined loads analysis is still underway and there does not appear to be any serious obstacles because of the conservative design approach employed to date. For example, in Figure 13 the maximum stress is 41 MPa, below the yield strength of general purpose 6061 Al of 55 MPa and well below the ultimate strength of 128 MPa. Furthermore, the peak stress occurs in an isolated region, and additional design work is expected to further alleviate the peak stress.



Figure 9. Comparison of quasi-static acceleration loads



Figure 10. Random vibration environment for selected launch vehicles



Figure 11. Acoustic load levels for selected launch vehicles





Figure 12. Current mechanical design of emitter array



Figure 13. Static stress for the payload structure under maximum longitudinal acceleration of the Super Strypi (12g)

The following table provides a summary of the payload design elements.

Element	Material	Mass (kg)
Reflectors (52)	Rh coated Ni	4.8
Emitters	Kanthal wound on alumina core	0.1
Structure	6061 Al	12.7
Adhesive to bond reflectors to structure	High temperature vacuum epoxy	0.6
Power supplies (4)	Various	0.2
Wiring (30 m)	20 AWG Tefzel coated Cu	0.2
	Total mass	18.6

Table 1. Mass of various elements of payload

Additional mass will need to be removed, and will come from two areas. First, the payload adapter ring is currently included in the payload mass. However, once the bus design has matured, this ring will be replaced with a much lower mass adapter, saving approximately 2 kg. The structure mass will also be reduced by exploring other tempers of 6061 rather than general purpose 6061. With these changes, the final design will be very close to or below the goal mass of 15 kg.

Plans for space qualification testing are underway. An engineering development model (EDM) will be fabricated based on the completed design. The EDM will be a full scale article, but will not be fully populated with reflectors and emitters. In order to reduce cost, the EDM will have one quadrant of the array fully populated and powered by a flight-qualified power supply. The remainder of the components will be replaced with mass mockups. The testing will be completed at a commercial test facility in the local area, using the environments discussed in the previous section. The mechanical structure will undergo static load testing via centrifuge. Shaker tables will be used to perform dynamic testing, including sine sweep and random vibration.

Electrical

The electrical design initially used four 28-V power supplies to energize 56 emitters, one power supply for each quadrant of the array. However, this design was complicated by the need to drop from 56 emitters to 52 in order to save mass. With 56 emitters, each 28-V power supply could each power 14 emitters in 2 parallel strings of 7 emitters. However, with only 52 emitters, the strings were no longer evenly balanced. Therefore, each power supply would still service one quadrant, but with one string containing seven emitters and one string containing six emitters. The precise pattern of which emitters are connected to the string of six and which are connected to the string of seven remains to be determined. The choice will be based on the smoothness of the resulting beam profile. As mentioned previously, the calibration source needs to provide a range of intensities. However, given the limited pointing knowledge provided by the bus (<1°), the profile must vary smoothly on the scale of that uncertainty.

Functional testing for the array will use the same procedure that was used for a single reflector. Although the intensity cannot be measured in the vacuum chamber, the scaling relationship between testing in the lab and testing in the chamber has been established. Final characterization tests of the array will be deferred until the unit is on orbit, and will be conducted using ground based sensors with stellar sources as references. The primary concern for the electrical system from the environment is thermal management. Thermal/vacuum chamber testing will be conducted to ensure heat loads are not excessive during operations with unfavorable solar loading. Likewise, deformation to the structure from thermal expansion will be carefully characterized to ensure it does not degrade the accuracy of the pointing knowledge. Modeling to date has not shown any significant problems in these areas and thermal management is not anticipated to present severe challenges.

Two other issues were addressed in the electrical design, electromagnetic interference/electromagnetic compatibility (EMI/EMC) and residual magnetic moment. During launch, when the EMI/EMC issues are most significant, the payload will not be powered and therefore neither susceptible to nor generating EMI. On orbit, the payload will not be operating during communication with the ground station due to power limitations, thus mitigating the other primary EMI risk. Because of the relatively large current (about 14 A) flowing in the various parts of the payload, all connections will be use twisted pair conductors. This approach will ensure the current does not generate a significant magnetic moment, resulting in undesired torques and variation in the rotation rate of the satellite. Finally, the large amount of nickel, almost 5 kg, could result in significant residual magnetic moment. Therefore, each reflector will be de-gaussed before installation in the array.

Atomic Oxygen

Most of the components have either space heritage or are not susceptible to the effects of atomic oxygen. However, no data have been found on the effects of atomic oxygen on the rhodium coating used on the parabolic reflector. Therefore, plans are underway to

test witness samples of rhodium-coated nickel in the Characterization of Combined Orbital Surface Effects (CCOSE) chamber at the Arnold Engineering Development Center.¹⁴ The CCOSE chamber can provide sequential or simultaneous testing of atomic oxygen and ultraviolet, proton, electron radiation. For this program only the atomic oxygen capability is needed. In order to support the one year projected mission life, an atomic oxygen fluence of $7x10^{21}$ atoms/cm² is required. This is the fluence expected at an altitude of 400 km during solar maximum. The test fluence assumes that the rhodium will be continuously exposed to the ram direction, a very conservative assumption. In the past, this would be a very difficult measurement to make because of the low flux available from atomic oxygen sources. However, a new source is currently being installed in CCOSE, which is expected to have a flux as high as 10^{16} atoms/cm²-s.¹⁵ The CCOSE chamber is equipped with an apparatus for conducting in situ measurements of the reflectance. This test will demonstrate if atomic oxygen has a detrimental effect on the rhodium coating. If it does, there are several alternatives. The atomic oxygen fluence can be significantly reduced by pointing the emitter array away from the ram direction during eclipse. Further reductions can be calculated based on the angle between the velocity vector and vector toward sun, since the satellite will be pointing toward the sun most of the time it is not in eclipse. Finally, if these do not reduce the fluence sufficiently, the rhodium coating can be replaced with gold. Although considerably more expensive, gold serves as an even superior reflector in the bands of interest and has been shown to be unaffected by atomic oxygen at fluences of up to 8×10^{21} atoms/cm². ^{16, 17}

CONCLUSION

The results to date have shown that a simple payload consisting of commercial IR emitters and reflectors can be constructed to serve as an exo-atmospheric IR calibration source for terrestrial sensors. The payload was based upon principles of simplicity, risk-tolerance, and capabilities-driven design. At the current time, final design details are being completed in preparations for fabrication of the EDM and subsequent environmental testing. Despite its simplicity, the payload has provided several engineering challenges, but is on-track to continue to support Army space science and technology requirements. SSC07-I-5, 21st AIAA/USU Conference on Small Satellites, 2007.

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