AN OVERVIEW OF THE ANDE RISK REDUCTION FLIGHT

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ABSTRACT

The Atmospheric Neutral Density Experiment (ANDE) Risk Reduction flight is a mission proposed by the Naval Research Laboratory to monitor the thermospheric neutral density at an altitude of 400km. The primary mission objective is to test the deployment mechanism from the Shuttle for the ANDE flight in late 2005. Scientific objectives of the ANDE risk reduction flight include; monitor total neutral density along the orbit for improved orbit determination of resident space objects, monitor the spin rate and orientation of the spacecraft, provide a test object for polarimetry studies using the HI-CLASS system.

The mission consists of two spherical spacecraft fitted with retro-reflectors for satellite laser ranging (SLR). Each spacecraft contains a small light-weight payload designed to determine the spin rate and orientation of the spacecraft from on-orbit measurements and from ground based observations. A unique design requirement of one satellite is to telemeter the data to the ground without external protrusions from the spherical spacecraft (i.e. an antenna). This satellite will carry a communications system developed by the USNA that uses two aluminum hemispheres as the ends of a dipole antenna, at amateur radio frequencies.. This system will act as a backup communications platform in the full ANDE mission, planned for 2005. The techniques for determining spin rate and orientation of the satellite include: an orthogonal array of photovoltaic cells, an orthogonal array of laser diodes (to be observed by the HI-CLASS sensors at the Air Force Maui Space Surveillance Site), and variations in the light reflected by the ANDE sphere, which will have a specific pattern of different surface finishes.

This paper presents a mission overview and emphasis will be placed on the design, optical layout, performance, ground station, and science capabilities of the mission.

1. INTRODUCTION

The Atmospheric Neutral Density Experiment Risk Reduction (ANDERR) flight is a mission in development at the Naval Research Laboratory. The ANDERR mission consists of two spherical spacecraft with retro-reflectors for satellite laser ranging (SLR). The major goals of this mission are to test the deployment system intended for the Atmospheric Neutral Density Experiment (ANDE) / Modulating Retro-reflector Array in Space (MODRAS) combined mission [1,2], to determine the spin rate and orientation of the spacecraft, and to provide calibration objects for both atmospheric drag research and the Naval Networks and Space Operations Command radar system. The MODRAS experiment has a tight operational temperature range. Thermal balancing will be achieved via rotation of the spacecraft in the ANDE mission, therefore determining the initial spin rate and orientation of the spacecraft (and the evolution of these parameters) has direct relevance to the ANDE/MODRAS mission. The launches and orbital insertion for both the ANDE and ANDERR missions are provided by the Department of Defense (DoD) Space Test Program (STP).

2. ANDERR SPACECRAFT

The two spherical spacecraft for the ANDERR mission, shown in Figure 1, are designated the Mock ANDE Active (MAA) and the Fence Calibration (FCal). Each spacecraft is fitted with a set of thirty 12.7 mm diameter optical retro reflectors for SLR located along latitudinal bands at ±90º, ±52.5º and ±15º with one, six and eight retros per band respectively. Figure 2 presents the number of retro reflectors illuminated at an altitude of 400 km as a function of location on the sphere (in longitude and latitude) to the observer.

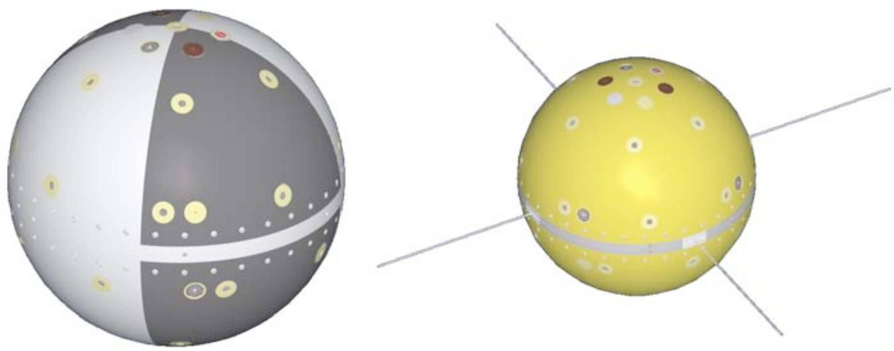

Fig. 1. Models of the ANDERR MAA (left) and FCal (right) spacecraft (note: not to scale).

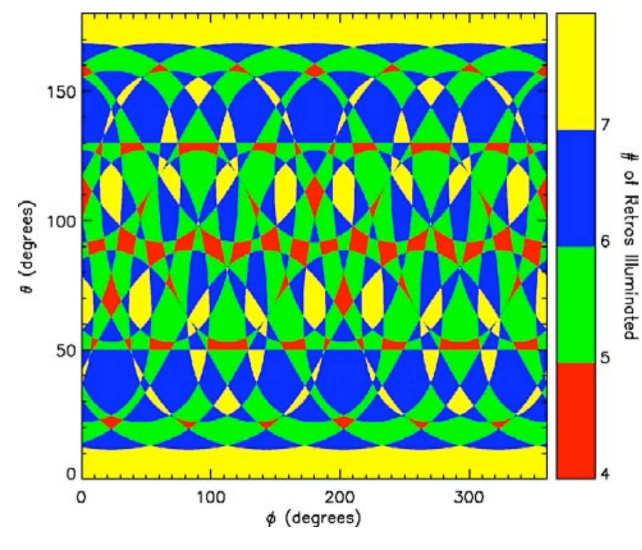

Fig. 2. The number of retros illuminated as a function of longitude and latitude on the sphere.

2.1 ANDERR MAA

The MAA spherical 50 kg spacecraft is 19.0 inches in diameter and constructed from two anodized aluminum (6061-T6) hemispheres. The hemispheres are formed by spin-casting aluminum. They are then rough machined, heat treated, finish machined, and finally anodized. The equator of this sphere is constructed from Delrin which provides a non-conductive separator between the hemispheres allowing the spacecraft shell to perform as a dipole antenna for the communications system developed at the United Sates Naval Academy. The payload mounting decks for each hemisphere are fabricated from anodized aluminum (6061-T6). The communications system operates at 145.8 MHz with a power of 2 Watts. The system is powered by a total of 112 battery cells (Tadiran TL-5930) in four battery boxes, each wired as 7 strings of four cells per string for a 12 V system bus voltage. The capacity of this system is 7450 WHrs, an estimated lifetime of about 1.5 years. One disadvantage of the large parallel structure of the batteries is that they would all approach the end of their life at the same time and with the very flat discharge curve for these primary cells, there would be no way to judge the remaining life of the system. To overcome this problem and provide a means to evaluate the condition of the battery system on orbit, the 4 battery banks will be depleted one at a time to drive the laser system. Thus, an assessment of the life of the complete power system may be performed by monitoring the status of each battery bank in the telemetry.

The payload is split into the two hemispheres, with two battery banks, a communications box and the laser driver box stacked in a vertical configuration. An exploded view of the MAA spacecraft is presented as Figure 3. The MAA sphere is painted with a pattern of four 90º longitudinal segments, alternating bare black anodized aluminum and Aeroglaze A276 gloss white paint. The purpose of this paint scheme is two fold. It provides an easy visual pattern to observe the initial spin rate and orientation and it also provides means to determine spin rate and orientation from the polarization return, discussed in section 4.

Onboard instrumentation for the MAA spacecraft consists of a set of six Copper Indium Gallium Diselenide (CIGS) photovoltaic cells that are mounted flush with the surface of the sphere. These light sensors are located at the endpoints of three nearly orthogonal axes and are used for attitude and spin rate determination; they do not recharge the power system. Thermistors are placed at several places within the spacecraft to monitor the temperature of the various components of the spacecraft. The temperature and photovoltaic voltage values telemeter to the ground via a “heartbeat” communications system that activates for 2 seconds out of every 20 seconds. If a ground station is detected the data is transmitted, if not, the system returns to a sleep cycle for another 20 seconds. A set of six laser diodes, also located at the endpoints of three nearly orthogonal axes, are commanded “on” during passes over Maui. These diodes emit light at 810 nm which is observed from the Air Force Maui Optical and Supercomputing facility.

Fig. 3. Depicts an exploded view of the ANDERR MAA spacecraft. The payload stack in each hemisphere consists of two battery banks, a communications system and a laser diode driver.

2.2 ANDERR FCal

The FCal spacecraft is a 17.5 inch sphere with a mass of 75 kg. The size requirement of FCal was driven by the resonant frequency of the Navy radar fence. The two hemispheres are fabricated by spin-casting brass, followed by finish machining. The exterior of each hemisphere is nickel coated for durability in the harsh space environment. The equator consists of an anodized aluminum (6061-T6) deck which incorporates an antennae deployment system, and mounting locations for the FCal payload. An exploded view of the FCal spacecraft is presented in Figure 4; note a cubesat ballast mass is included for center of gravity symmetry.

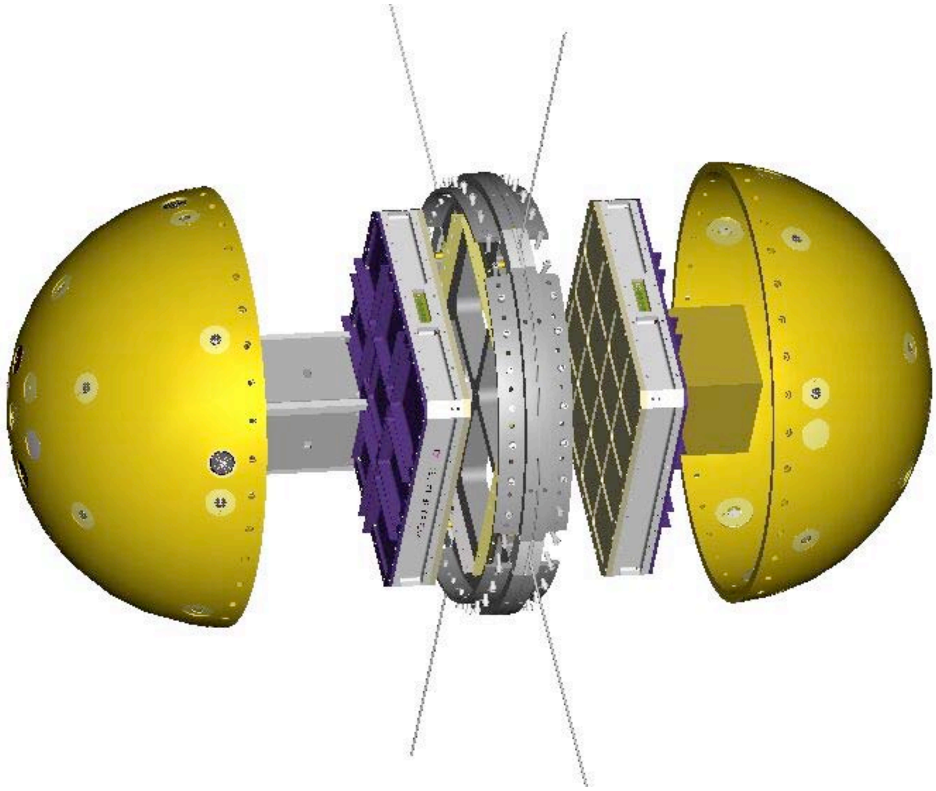

Fig. 4. Depicts an exploded view of the ANDERR FCal spacecraft. The payload stack in each hemisphere consists of a battery bank and a cubesat (or mass cubesat model).

The FCal payload is a Stensat Group LLC cubesat. The cubesat is a four-inch cube pico-satellite originally designed as an educational platform. It is a fully functional satellite with solar power (not used in FCal), battery power, a computer, and a communications subsystem. It can accommodate small payloads in the internally available space. The goal of the structural design is to minimize the mass of the cubesat while providing a robust structure that can survive launch stresses. The structure is highly integrated with the electronics; the electronic circuit boards are part of the structure. The assembly is fabricated from 6061-T3 aluminum. The cubesat core consists of a stack of eight circuit boards on 11 mm centers. Board-to-board connections are by means of surface mount high density connectors on the top and bottom of each board. Since the only inter-board communication is via these connectors, otherwise empty slots must be populated with a blank feed-through board, as seen in Figure 5.

For the FCal, the cubesat structure is used with minimal modifications. The bottom corner blocks are replaced with solid blocks that have a 10-32 threaded hole to secure the cubesat structure to the FCal internal structure. Normally, the cubesat has solar panels on all faces. Each face includes a temperature sensor and a light sensor. The sensor functions are needed on the FCal without the solar cells. The same sensor interface circuit is used for the sensor modules on FCal. A board is designed to mount on the ANDE universal mounting system. The sensors detect light levels for determining the orientation of the satellite and temperature.

The processor board houses a Microchip PIC16C77 processor. It is an 8-bit single chip computer with built-in program memory, data memory, and various interfaces. The processor board performs the communications modulation with an AFSK encoder using the MX614 IC. The processor board generates the AX.25 protocol telemetry and routes the bit stream to the MX614 for the modulation. The modulated signal is routed to the communications board. The CM8870 decodes DTMF tones and generates a 4-bit pattern, which is interpreted by the processor. The processor board controls the operations of the satellite. It controls the transmitter power. The processor board stimulates the watchdog timer located on the power board.

Fig. 5. A picture of the cubesat with the cover plate removed payload is shown.
Note that the four antennae are visible.

The communications board contains a transmitter and receiver. The transmitter operates at 436.81 MHz. The transmitter is an FM transmitter consisting of an exciter and power amplifier. The exciter is crystal controlled and has a modulation input port for an audio signal or AFSK signal. The output level of the exciter is 10 mW. The exciter output is connected to the amplifier circuit. The amplifier is a single IC device capable of a maximum of 800 milliwatts output. The output level is adjustable and has been set to 250 milliwatts. The output of the amplifier is single ended with a low pass filter. It is routed to splitters with a 180 degree phase shift between the two outputs. The outputs are routed via coaxial cable to two antenna boards on opposite sides of the FCAL equator.

The power board was redesigned to support the FCAL battery configuration, which uses two battery banks identical to the MAA battery banks. The cubesat power board was designed to support voltages from 3 volts to 5 volts. The FCAL battery configuration provides 9 volts to 15 volts. The power board contains high efficiency power converters to reduce the voltage to 3.2 volts with an efficiency of about 85%. The power board contains the satellite watchdog circuit, which will cycle power to the whole satellite if the processor suffers a single event upset. Onboard diagnostics capabilities have been included which do not require operation of the processor board. The processor board includes an eight channel analog-to-digital converter which allows the monitoring of the power converters, the batteries, and current draw of the satellite.

The FCal operation is fairly straight forward. When the FCal is deployed, the processor boots and initializes all the satellite components. It then waits one minute and deploys the antennas one at a time. There is a one second delay between the deployment of each antenna. Once deployed, the satellite starts monitoring all the sensors and internal parameters and transmits the telemetry once every two minutes. The satellite can accept commands during this period. The telemetry transmission interval can be set and the receiver audio can be routed to the transmitter to turn the satellite into a transponder. Telemetry data includes voltage and temperature from each of the six phototransistors, battery and power data, and internal temperature data. The purpose of the telemetry is to determine if the FCAL is spinning and at what rate and orientation.

3. DEPLOYMENT

The orbit requirements for the ANDE and ANDERR missions are to deploy a set of two spherical spacecraft into a circular orbit at an altitude of 400 km and an inclination of 51.6º with an initial in-track separation of less than _ orbit between the two spacecraft. Mission unique design requirements of the deployment system are: ANDERR spacecraft masses of 50 kg and 75 kg (25 kg and 50 kg for ANDE), no permanent mechanical interface between the system and the spacecraft that changes the spherical envelope of the spacecraft, an initial spin rate of 1 to 10 rpm (one spacecraft only), and the payload exterior must be protected. No existing Space Shuttle ejection systems could meet these requirements, therefore the DoD STP requested that a study be commissioned to determine the necessary hardware. The Canister for All Payload Ejections (CAPE) under development by Muniz Engineering, Inc. in conjunction with the DoD STP, is in response to this request to the needs of the research community for a single ejection platform capable of ejecting payloads with requirements that are not compatible with current NASA developed ejection systems.

The decision to develop the Canister for All Payload Ejections (CAPE) was made after an extensive review of all existing payload ejection systems was performed. The review examined the capabilities of the Shuttle Small Payloads Program Office (SSPPO) systems at Goddard Space Flight Center (GSFC). These systems were unable to meet the requirements for the ANDE/ANDERR payloads without imposing design or operational restrictions on the payloads. Also, the existing ejection systems would have required similar containment support structures for the spherical spacecraft. An additional manifest burden for using the GSFC SSPPO would be the need for an ejection system for each ANDE/ANDERR satellite. Also examined were un-conventional methods of using the crew to “jettison” the spacecraft from either ISS or the Space Shuttle. This was discounted due to the need for accurate velocity and direction data for the re-contact analysis, and because of projected EVA airlock and timeline constraints.

The CAPE is a system that is designed to transfer safety and integration requirements from the payloads to the CAPE. To meet the requirements for the near simultaneous ejection of the two spheres, the CAPE system uses an Internal Cargo Unit (ICU). The Internal Cargo Unit meets the requirement for no traditional mechanical contact between the satellites and the ejection system by utilizing Viton O-Rings for the interface plane. This interface is pliable and prevents marring of the surface from metal to metal contacts. The ICU also allows for insertion of both satellites into the same orbit, eliminating the possibility of a failure in a second ejection mechanism. CAPE is designed to mount to a Get Away Special (GAS) Beam on the orbiter sidewall, the weight and center of gravity of the overall CAPE/ICU/ANDE system govern the installation location of the system in the Space Shuttle. For the overall system weight of the ANDE RR flight, only one payload bay location is within the weight and CG limitations on the GAS Beam, whereas the overall system weight of the ANDE Payload, five payload bay locations are within the limits.

The CAPE is a cylindrical tube that has a payload volume of 21 inches in diameter and 53 inches long. The diameter is driven by the need for an ICU that can hold two 19 inch diameter spheres. The length is driven by the need to support the two 19 inch diameter spheres and an avionics box that controls the separation of the ICU from the two satellites, and yet stay within the payload envelope of the shuttle cargo bay. For EMI/RF transmit power level reduction, the CAPE is designed to have two 90 degree corners in each joint. Since electromagnetic waves tend to attenuate drastically at hard corners, the corners designed in should provide between 5 and 20 decibels reduction in any payload or ICU generated EMI/RF.

The CAPE, shown in Figure 6, is basically a mechanical and electrical interface to the Orbiter. The mechanical interface to the Orbiter GAS beam is provided by the GAS Beam mounting plate. The electrical interface, called the Cape Inhibit Box, is for controlling and providing power to the deployment mechanism that initiates the Primary Separation System and subsequent ICU ejection. The ejection is initiated by the astronauts via the Orbiter Standard Switch Panel.

The ICU, shown in Figure 7, is a container that is attached to the CAPE via a Planetary Systems Corp (PSC) Lightband ™ separation system and totally contains both spacecraft. The ICU allows both spacecraft to be ejected from the Orbiter at the same time while having no traditional mechanical interface to the spheres. The ICU is a three segment device consisting of bottom, middle and top sections. The sections are held together by two secondary separation systems, also Lightband™ separation devices. The spherical spacecraft are contained inside

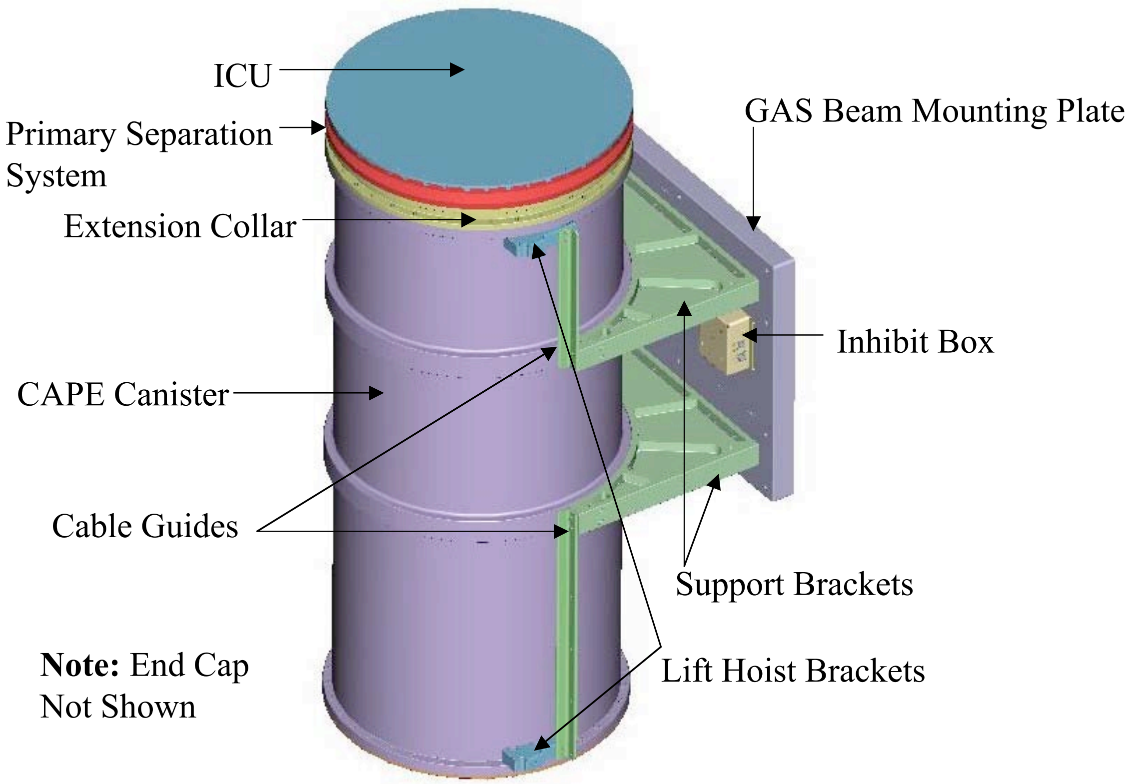

Fig. 6. Canister for All Payload Ejections

the ICU as shown by Figure 8, with the ICU removed for clarity. The spacecraft are held in place by Viton O-rings that are compressed during the ICU buildup process. The o-rings are held in place by pedestals mounted to the ICU structure. Rotation of the spheres is prevented by a cup and cone interface, with the cup and cone not normally in contact with each other. The cone is located on the ICU pedestal along with separation springs to ensure separation of the sphere from the Viton o-ring and pedestal. The separation springs engage the spheres with a spherical Delrin head.

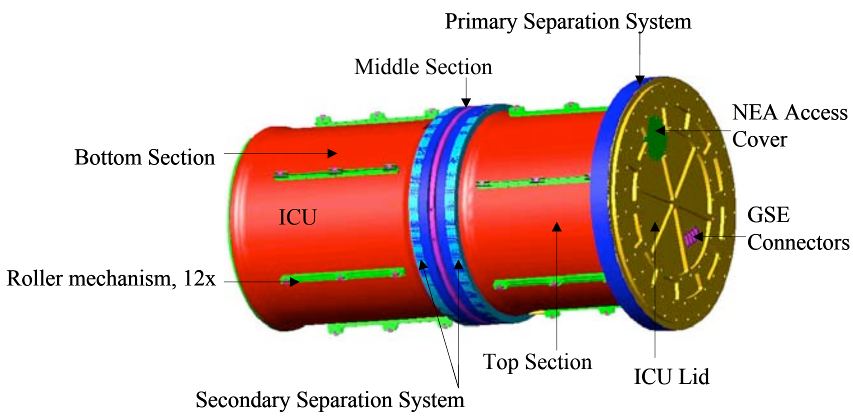

Fig. 7. The Internal Cargo Unit (ICU)

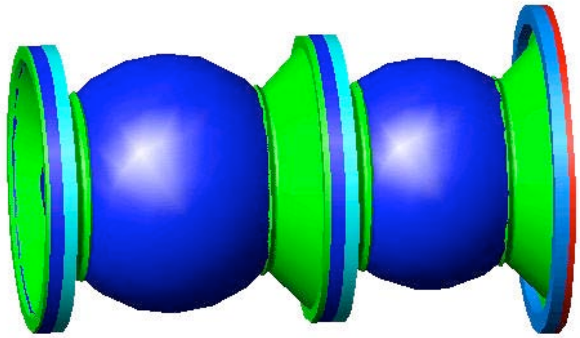

Fig. 8. ICU internal configuration.

Upon activation the ICU is deployed from the cargo bay with an ejection velocity of at least 1.6 ft/sec. Timers within the ICU are initiated as the ICU separates from the CAPE. These timers count up to a pre-determined separation time at which point the ICU will deploy the ANDE spacecraft. This time is determined by the minimum safe distance for a deployment of five free objects near the Space Shuttle. Currently NASA Safety is reviewing the deployment sequence to determine this minimum safe distance. A secondary separation system will deploy the top and bottom ICU cylindrical sections in equal and opposite directions at less than 0.5 ft/sec, and similarly the ANDE or ANDERR spacecraft will deploy via springs in equal and opposite directions at less than 0.2 ft/sec. The mass, area and coefficient of drag properties of the three ICU pieces will cause them to separate from the ANDE/ANDERR spacecraft due to differential drag forces. Figure 9 depicts the system shortly after deployment. The system employs a pivot point to induce a spin of 1 to 10 rpm on one of the spacecraft. Several agencies within the DoD are interested in these types of on-orbit separation events.

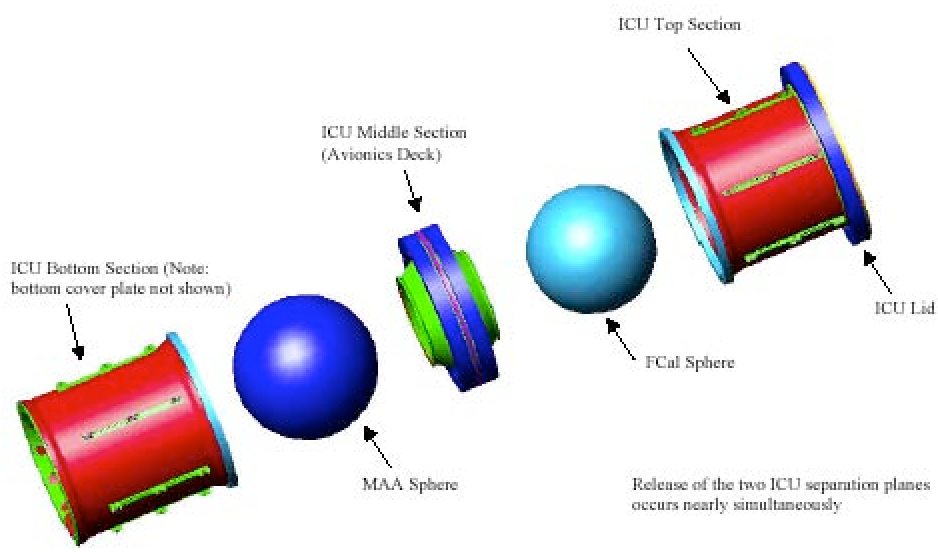

Fig. 9. Separation of the ICU.

4. DETERMINING ANDE MAA ROTATION RATES AND ORIENTATIONS

The rotation rates of the ANDE MAA spacecraft must be known with some confidence in order to model the operation environment for the MODRAS experiment. Since it is not known what exact rotation rate will be induced by the deployment from the CAPE, an important goal of the ANDERR flight is to determine the rotation rate.

Several methods will be used to determine the rotation rate and orientation of the ANDE MAA sphere as a function of time. An astronaut will acquire video and still images of the deployment through a window on the Space Shuttle, providing a visual record of the initial rotation and orientation. Six photovoltaic cells, at opposite ends of three nearly orthogonal axes, will provide different voltages based on their orientation relative to exposure to solar illumination, Earthshine, and deep space. Thus, the temporal variation in the solar cell outputs will provide a time history of the ANDE MAA spacecraft orientation relative to the sun. In addition, the deployment has been requested to be done within sight (in terminator) of the AF Maui Optical and Supercomputing facility (AMOS). On successive passes over Maui, signals from the six onboard laser diode beacons (810 nm) will be received at the AMOS facility and used to deduce the sphere spin rate and orientation. Ground based observations of glints from sunlight reflecting off of the front surface of the retro reflectors may also be used to determine the spin rate of the spacecraft. Finally, the MAA sphere will have a pattern of several longitudinal segments, with alternating surface materials. Thus, when the sphere is illuminated by a laser beam from AMOS, variations in the light returned to detectors at AMOS will be used to determine the sphere's rotational motion.

The laser diode beacon system is being designed and built by the AF Research Laboratory Semiconductor Laser Applications Group (known as the ScorpWorks) at Kirtland AFB, NM. Outputs from six pulsed laser diodes (810 nm, 500 mW each) will be transmitted via fiber optics to the ends of three nearly orthogonal axes on the sphere. The transmit apertures will be formed simply by the cleaved ends of the 50 micron core of the fibers, which will form beams with a full-angle divergence of ~ 10°. The ends of the fibers will be inserted through holes in the sphere. The fibers will be secured with standard fiber couplers. The diodes will be pulsed at six unique rates (0.5 Hz, 0.75 Hz, 1.25 Hz, 1.75 Hz, 2.75 Hz, and 3.25 Hz) so that at the expected spin rate, of 1-10 rpm, it will be possible to identify which beacon is oriented directly toward the observer.

The observations of the laser diode beacons and the laser illumination will be done using the High Performance CO_2 Ladar Surveillance Sensor (HI-CLASS) system, which is integrated with the Advanced Electro-Optical System (AEOS) 3.67 m telescope at AMOS [3]. HI-CLASS uses a 12 Joule per pulse, Carbon-13 based CO_2 laser, at a wavelength of 11.15 microns, to perform heterodyne detection. Coherent detection is used, as opposed to direct detection, to mitigate the effects of detector and thermal background noise that would otherwise overwhelm return signals from targets in space. The divergence of the 11.15 micron laser beam transmitted by the AEOS telescope will produce a beam spot about 3 m in diameter at 400 km, the initial insertion altitude of the ANDERR spacecraft, large enough to flood-illuminate the spherical spacecraft. The broadband, all-reflective optical system makes the HI CLASS system very convenient for testing passive and active sensors at other wavelengths. Thus, HI-CLASS will be able to simultaneously observe the laser beacon signals (at 810 nm) while transmitting and receiving signals at 11.15 microns.

The HI-CLASS laser will transmit a linearly-polarized beam. The heterodyne receiver will have two orthogonal, linearly-polarized channels, one aligned with the polarization of the transmitter and one orthogonal to it. As a result, in addition to detecting variations in the intensity of the return signals, variations in the polarization of the return signal can be detected.

Simulations of the returns from the active illumination of the MAA sphere [4] were carried out using the Time-Domain Analysis Simulation for Advanced Tracking (TASAT) software toolkit [5]. The TASAT simulations take into account tracking radiometrics, viewing aspect, glint effects, polarization effects, and illumination profile effects, as well as surface optical properties. The surface optical properties are based on a Maxwell-Beard Bi-directional Reflection Distribution Function (BRDF) model. The simulations also include first-order effects due to atmospheric turbulence. Transmission and path radiance effects on both the uplink and downlink beam propagation through the atmosphere are also taken into consideration. The Maxwell-Beard BRDF model provided overall reflectivities of the two surface finishes chosen, as well as estimates of their polarization characteristics. Some experimental BRDF measurements were made of black anodized 6061-T6 aluminum samples, as well as of white Aeroglaze A276 paint. However, the data is limited at 11.15 microns and only selected angles of incidence (30° and 60°) were used. Thus, additional experimental characterizations of the materials are needed to be confident in the TASAT simulation results.

For the TASAT simulations, a number of assumptions about the ANDE orbit were made. The spacecraft was given an orbit that passed within sight over the AMOS site. The altitude of the simulated orbit was 400 km and the orbital inclination was 51º. A typical culmination angle of 10º was chosen to assure a real world scenario, typical of many satellite-pass geometries. The duration of the simulated pass was set at 200 seconds. A sampling rate of 5 Hz was used for the HI-CLASS detection system.

Three MAA surface patterns were analyzed with TASAT. The first pattern consisted of two hemispheres, one coated with white Aeroglaze paint, the other simply anodized aluminum. The second pattern consisted of four 90º longitudinal segments, of alternating white Aeroglaze and anodized aluminum. The third pattern used eight 45º longitudinal segments. These three potential ANDE paint patterns are shown in Figure 10.

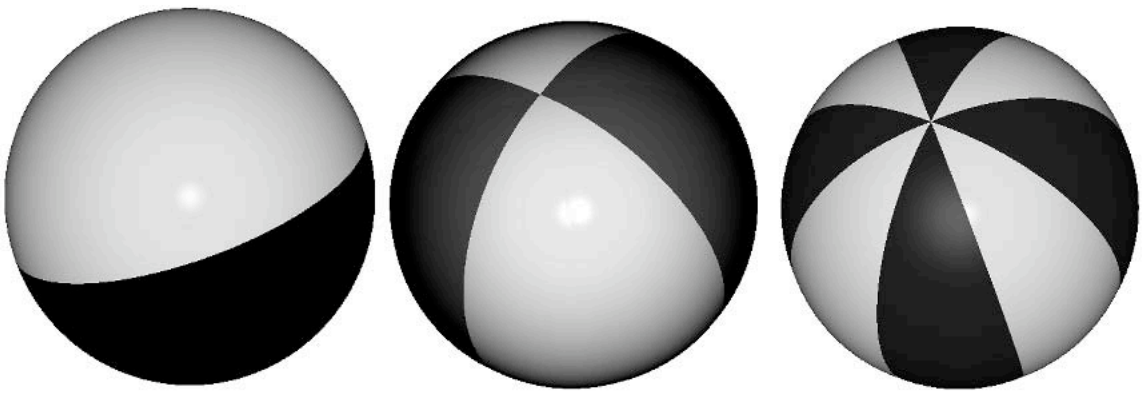

Fig. 10. ANDERR MAA paint pattern options.

The simulations used aligned and cross-polarization configurations, in which the illuminator emitted linearly polarized light and the receiver had a polarized filter placed in front of the focal plane, either aligned with the direction of the transmitted beam or perpendicular to it. Simulations were also performed without considering any polarization effects. These simulations were carried out to see if the rotational properties of the test sphere could be deduced from the unpolarized BRDF properties of the materials. The simulations involved the transmission of laser pulses upward to the space target, followed by scattering, reflection, and absorption from target materials. First-order atmospheric effects were then added to the return signal. The return signals were received and stacked to form what is called a “sinogram.” A sinogram is simply the resulting “signature pattern” formed by the stacked returns, over the entire pass. The characteristics of the return signals from the target depend on both the BRDF properties of the materials coating the target’s surface and the polarization properties of the materials. The final detected signal was essentially a time line of returned energy as a function of time.

Visual inspection of the sinograms provided only a rough estimate of the rotational properties of the target. In general, higher spatial frequencies in the sinograms implied fast rotation rates, and complex harmonic patterns implied rotation about multiple axes or some type of tumbling motion. However, to get a quantitative estimate of the rotational characteristics of the sphere, Fourier analysis was used to deduce the rotation rate and possibly the orientation of the rotation axis. The Fast Fourier Transform (FFT) of the polarization contrast of the return signal, summed over all columns (pixels or spatial extent of each profile), was computed. The contrast is defined as:

where “aligned” is the summed returns for the aligned polarization state and “cross” is the summed returns for the cross polarization state. Taking the contrast of the return signals enhanced the ability to deduce the rotational properties of the sphere by taking advantage of the polarization properties of the materials coating the sphere’s surface. The contrast also “normalized” the return signal data, eliminating the effects due to most of the random fluctuations in the total return intensity, which was helpful in carrying out the frequency analysis of the data. The power spectrum of the total return, the sum of aligned and crossed polarizations, was also analyzed. In this case, the spectral component from the rotation rate was “swamped out” by variations in the return level due to pointing jitter, fluctuations in laser output, and changes in the target range. Thus, it would be very difficult to determine the rotation rate using the total return. Laser speckle noise is not reduced by the polarization contrast processing, however, since this noise source is independent for the two orthogonal polarization return states. Thus, contrast processing does not help with mitigating the effects of speckle noise. It may be possible to use a moving-average technique to improve the speckle noise-limited SNR.

The Fourier analysis of the polarization contrast data revealed strong peaks in the power spectra. In addition, several harmonics were also apparent in the power spectra. The strength of these harmonics changed as a function of the orientation of the axis of rotation of the sphere.

Taking the contrast of the aligned and cross polarization states enhanced the ability to determine the rotational rate for the spinning sphere, especially in the case where pulse-to-pulse return variations were significant. Incoherent (no speckle noise) simulations produced a single clean power spectrum spike at the rotational rate of the target. Effects due to speckle-noise, pointing jitter, laser output power fluctuations, and target range variations, which would cause pulse-to-pulse signal variations in the return signal, were also considered. It was found that the polarization contrast measurements enhanced the ability to determine the rotation rates and axis orientations [6] when most of these sources of variations in the return signal were present. Speckle noise would have to be dealt with in another manner. In general, it was found that using polarization contrast data from a coherent heterodyne detection system would enable rotation rate and axis determinations that would not have been possible with direct detection techniques due to detector and background noise.

The rotation rate for the spinning ANDERR MAA spacecraft was easily deduced using the sinogram analysis of the polarization contrast data for both the hemisphere and 4-segment paint patterns. The rotation rate and orientation could not be determined for the 8-segment pattern. This was probably due to the fact that multiple stripes were illuminated at the same time by the laser beam. Thus, either the hemisphere or 4-segment patterns for ANDERR MAA would be acceptable in order to use this technique in deducing rotation rates and axes.

5. SUMMARY

The ANDERR mission is a low cost constellation of micro satellites designed to support improvements in orbit determination and prediction. The mission will provide a proof of concept flight for the CAPE deployment system intended for use in the ANDE/MODRAS combined mission. The arrangement of the MAA and FCal in a lead-trail orbit provides an exceptional set of targets to study C_D modeling and its effect on satellite drag. The MAA satellite contains instrumentation to test the feasibility of an optical communications link from space to ground. This mission will demonstrate the effectiveness of flying low cost calibration targets to support Department of Defense and NASA requirements for precision orbit determination and collision avoidance. In addition, data from this mission will be used to improve the scientific understanding of the interaction between a spacecraft and its environment.

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