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Forgoing Time and State – The Challenge for CubeSats on Artemis-1

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Abstract

While launch opportunities for small satellites increase with each passing year, there remain gaps in planning for the functional needs of the satellites. For example, small satellites on the Artemis-1 launch will be released from the unpowered Space Launch System (SLS) rocket with no a priori knowledge of their time, position, or velocity (state), and must be able to communicate back with Earth. To accommodate this, there are a variety of different decision pathways that can be taken to ensure that communications can be established with the satellite. One option is to constantly be transmitting time and state from a ground station to the satellite upon deployment, in hopes that the Earth and the satellite will be in a geometry that allows reception of the transmissions. Another option is to coordinate a transmission schedule from that satellite that will have the satellite transmit, wait a period of time to receive any ground communications, then move forward in a specified search pattern until total sky coverage has been obtained. The two final methods are both hardware-based, and include the use of a star tracker located on the CubeSat to find and locate the Earth-Moon system and transmit/receive in that direction, or using a telecommunication system design that provides full-sky coverage. These approaches are important for small satellites, as Deep Space Network assets cannot support the expected growth of deep space CubeSats, thus requiring different approaches. The application and downsides to each of these approaches, along with technical details, will be discussed in this paper, as well as potential downsides with each option.

1. Introduction

The opportunities for small spacecraft launches with the Artemis-1 mission are the first to allow spacecraft not coupled to a main mission, allowing deep space flight opportunities with deployment opportunities along the path of flight of the Space

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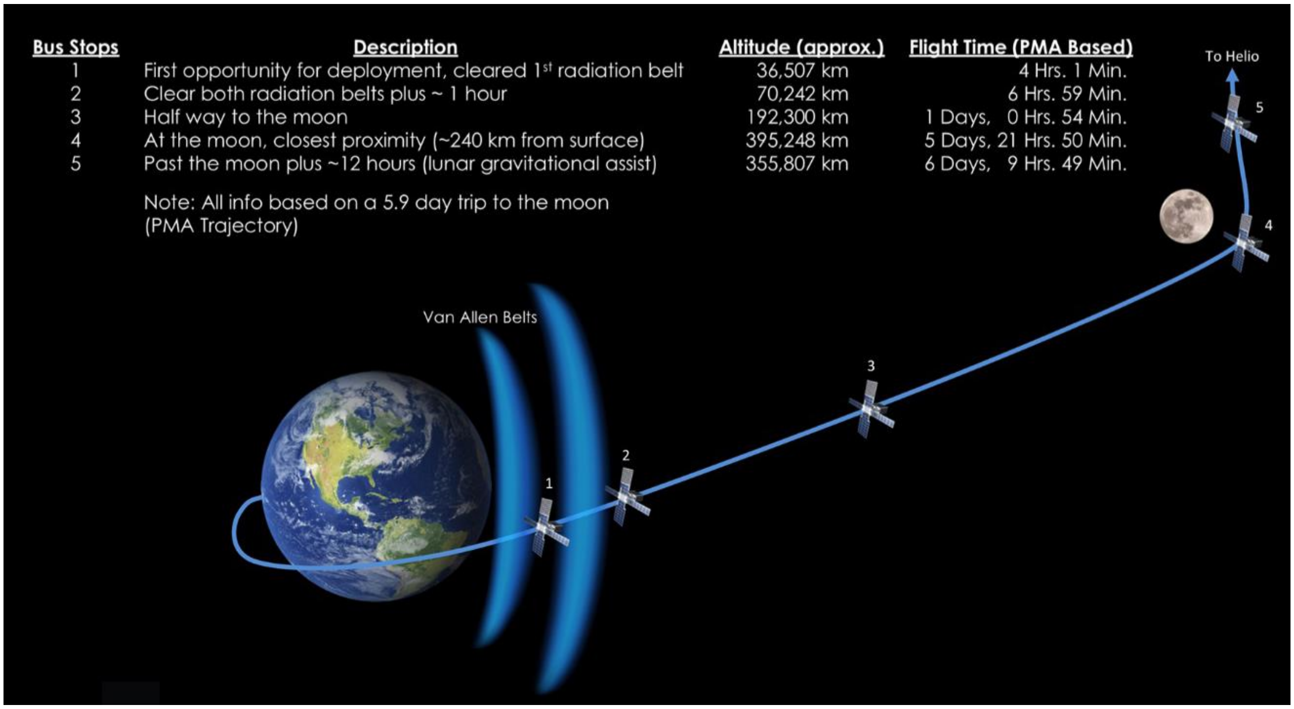
Launch System (SLS) rocket. The Colorado Earth Escape Explorer (CU-E^3) will be one of these small spacecraft that will be deployed from the SLS. The CU-E^3 CubeSat is a student-designed and built CubeSat, and is competing in the Deep Space Derby portion of the Cube Quest Challenge, which is a communications-based competition (Hyde, 2017).

There are five deployment locations, or Bus Stops, from which a CubeSat can be dispensed during the Artemis-1 mission. CU-E^3 will be exiting the SLS rocket at Bus Stop 5, close to 350,000 km from Earth, as it is the most convenient and economical stop to fulfill the CubeQuest competition requirements. Figure 1 shows the current locations of these bus stops based on the most recent trajectory available for the Artemis-1 launch.

When CU-E^3, and all other CubeSats, are deployed from the SLS upper stage, they will not have any *a priori* knowledge of their time, position, or velocity (state) from the deployer on the rocket body. This is a new challenge that the deep space CubeSats on the Artemis-1 mission will face, as the only other deep space CubeSats, of the MarCO mission, were provided status information upon deployment (Klesh, 2018). In most deployment scenarios, if a CubeSat should need time, position, and velocity, a GPS receiver can be added to the mission to obtain an initial state report; otherwise, ground information can be sent to the CubeSat based on TLE information. GPS is not available for deep space, and communicating with the CubeSat upon deployment poses a challenge, which will be discussed in section 3. The CubeSats also must be powered off, following delivery and during the launch and cruise phase of the SLS rocket. These power off requirements come directly from the Spacecraft Payload Integration & Evolution (SPIE) Office’s Secondary Payload Interface Definition and Requirements Document (SPIE 2018). This poses a significant challenge, as CU-E^3 must be able to communicate back with Earth as quickly as possible in order to work through commissioning, before the CubeQuest competition starts at four million kilometers, ~60 days post launch. The farther CU-E^3 is from the Earth, the slower the communication rate, therefore completing commissioning before four million kilometers is important, so that CU-E^3 can begin the competition phase exactly at four million km.



Bus Stops	Description	Altitude (approx.)	Flight Time (PMA Based)
1	First opportunity for deployment, cleared 1st radiation belt	36,507 km	4 Hrs. 1 Min.
2	Clear both radiation belts plus ~ 1 hour	70,242 km	6 Hrs. 59 Min.
3	Half way to the moon	192,300 km	1 Days, 0 Hrs. 54 Min.
4	At the moon, closest proximity (~240 km from surface)	395,248 km	5 Days, 21 Hrs. 50 Min.
5	Past the moon plus ~12 hours (lunar gravitational assist)	355,807 km	6 Days, 9 Hrs. 49 Min.

Note: All info based on a 5.9 day trip to the moon
(PMA Trajectory)

Figure 1: Bus stop location for Artimes-1 launch (Subject to change based on launch changes) (Robinson et al., 2018).

Appendix 1 shows a full map of the Artemis-1 mission with CubeSat deployment locations marked.

1.1. CU-E^3 Overview

The CU-E^3 satellite, diagrammed in Figure 2, is a 6U CubeSat consisting of four main subsystems: Structures & Mechanisms; Command and Data Handling (CDH)/Flight Software (FSW); Electronic Power System (EPS); and Communications. CU-E^3 will not be flying an on-board propulsion system, and will be using solar radiation pressure to manage the accumulation of system momentum. The CU-E^3 structure was designed to meet the standards of the Planetary Systems Corporation Canisterized Satellite Dispenser (CSD), which will eject CU-E^3 from the SLS with a small delta-V of ~1.5 m/s. This ejection will be at a 34 degree angle off of the along-track direction of the rocket body. Within the CubeSat structure, a Blue Canyon Technologies XB-1 avionics module contains the flight computer as well as the reaction wheels, sun sensors, and star tracker necessary for high fidelity attitude determination and control. The XB-1 flight software will be pre-loaded prior to CU-E^3 integration into the CSD and consists of two main portions of functionality: general station-keeping and communication with the commercial ground station network. A primary focus of the flight software is ensuring the safety of the CubeSat prior to initial uplink, at which point a consistent uplink and downlink schedule will be established. This will be accomplished through on-board monitoring of the CU-E^3 electronics status as well as autonomous transitions between power generation and reaction wheel desaturation attitudes. The XB-1 and CU-E^3 communications systems are powered by a 21,000 mAh battery pack that will be charged on orbit by two deployable 3U solar panels and a 6U solar panel.

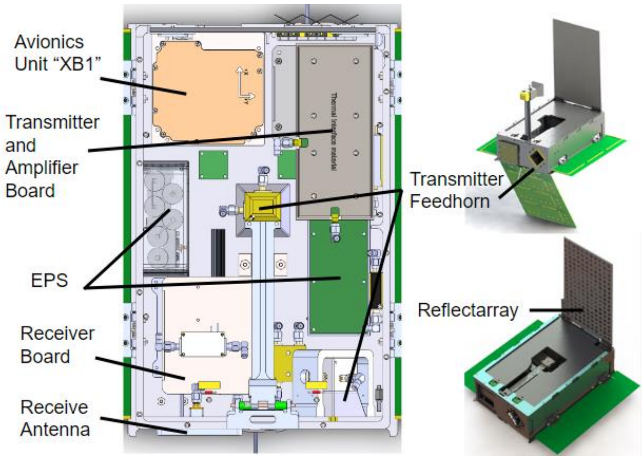


Figure 2: Overview of the CU-E3 CubeSat.

1.2. CU-E^3 Communication System

The CU-E^3 communications subsystem, and particularly the transmit chain, is the primary focus of the mission, as the Deep Space Derby prizes are associated with communication system downlink performance from distances beyond four million kilometers, including highest data rate, furthest communication distance, and CubeSat survivability (longest period between initial and final communication). The CU-E^3 receive chain consists of a C-band antenna operating at a frequency of 5182 MHz that is mixed down to UHF and received by an AstroDev Lithium-2 receiver. The CU-E^3 transmit chain contains an X-band transmitter initially developed at the University of Colorado Boulder, a high-power amplifier, and two transmitting antennas: a fixed X-band feedhorn and a high gain deployable X-band reflectarray and feedhorn system. Both uplink and downlink communications use binary phase shift keying (BPSK) modulation, and the downlink uses a half-rate convolutional coding scheme. CU-E^3 has the receive and transmit apertures on the same face and pointed in the same direction, enabling the system to transmit and receive simultaneously (Withee, 2019). A link budget covering the downlink at 8447.6 MHz is shown in Table 1, along with the receive chain for the distances CU-E^3 will see in the competition.

2. Scenarios

With the delays in the SLS rocket production, and subsequent delay in the Artemis-1 launch for CubeSats, the trajectory scenarios for deployment of the CubeSats are continually changing. An overview of the current trajectory and previous trajectory are outlined below with the impact that it has on the ability

Table 1. CU-E3 Link Budget

Slant Range	500,000 km	4,000,000 km	10,000,000 km
UPLINK 5182 MHz			
Tx EIRP @ Ground Station	111 dBm		
Rx Data Rate	218,000 bps	3,400 bps	540 bps
Symbol Rate	436,000 sps	6,800 sps	1,080 sps
Narrowest Rx Filter BW	523.2 kHz	8.16 kHz	1.30 kHz
Uplink Rx SNR	15.7 dB	15.7 dB	15.7 dB
Uplink Eb/No Margin	6.00 dB	6.03 dB	6.04 dB
DOWNLINK 8447.6 MHz			
Tx EIRP @ CU-E^3	56 dBm *(using Reflectarray antenna)*		
Tx Data Rate	28,000 bps	440 bps	70 bps
Symbol Rate	56,000 sps	880 sps	140 sps
Narrowest Rx Filter BW	67.2 kHz	1.06 kHz	0.17 kHz
Downlink Rx SNR	10 dB	10.0 dB	10.0 dB
Downlink Eb/No Margin	6.02 dB	6.00 dB	6.02 dB

of the CubeSat to recover time and state information, as well as considerations for future trajectories.

2.3. Current EM-1 Scenario

The current trajectory uses data that was released from United Launch Alliance (ULA) (ULA PMA, private communication, Oct. 2017) and represents the path of the Space Launch System (SLS) rocket second stage, called the Interim Cryogenic Propulsion Stage (ICPS). This data includes the Mean Equatorial J2000 Cartesian position and velocity of the ICPS with respect to Earth. Previous trajectory data, which included 3,000 CubeSat dispersion trajectories in addition to the nominal ICPS path, also included attitude data, which was not part of the file for this new launch data. This most recent file was released as a special one-time run for the secondary payloads to use to plan their missions. The attitude of the ICPS cannot be guaranteed, as the avionics unit on the ICPS is shut down before the CubeSats start deploying. This new trajectory run starts at bus Stop 1, the first opportunity for secondary payloads to deploy from the ICPS, and ends at Bus Stop 5, defined as 12 hours of flight past perilune, where CubeSats have the final option of deploying, and where CU-E^3 will deploy.

CU-E^3 will be coasting throughout its trajectory, because it does not carry propulsion. This means the CubeSat will be following the path of the ICPS from Bus Stop 5 onward, plus a small deployment delta-V of ~1.5 m/s. The trajectory simulation for CU-E^3 was performed using GMAT (General Mission Analysis Tool) (Hughes, 2016). To simulate the trajectory, the analysis takes the final state vector from the provided ICPS data, with a delta-V of ~1.5 m/s added, and assumes this is the state vector of CU-E^3. The trajectory is then integrated over the one-year length of the Cube Quest Challenge competition. Figure 3 illustrates the distance from Earth over the one-year competition phase that CU-E^3 will be placed in after the lunar flyby.

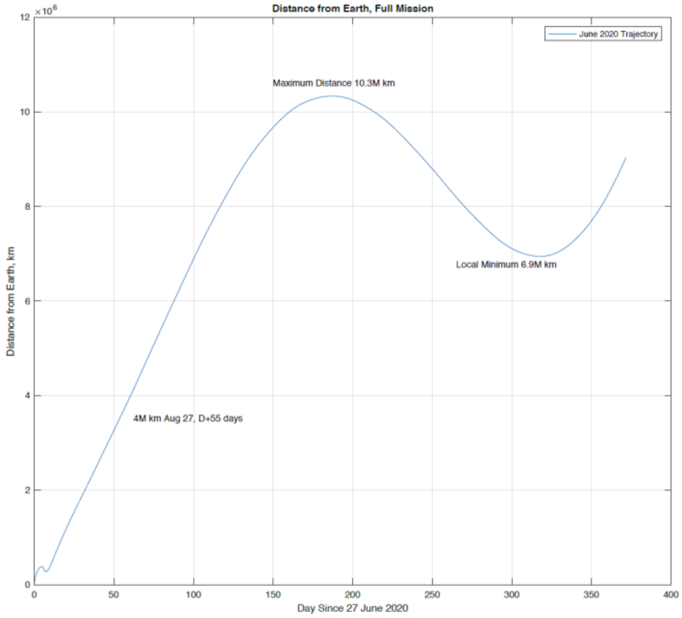


Figure 3: Distance of CU-E^3 from Earth over the one-year competition timeline. Deployment occurs at six days.

The trajectory analysis shows that CU-E^3 will obtain a maximum distance 10.3 million km over the course of the year-long competition phase. This distance will be used to analyze the star tracker option for establishing communication. Fortunately for the competition, even though the distance to Earth decreases from roughly 200 days from launch (early February 2021) until about the 300-day mark, the CubeSat will stay farther than four million km throughout the remainder of the mission. The orbit into which CU-E^3 will be inserted is a heliocentric orbit just ahead of the Earth with an aphelion that is greater than 1 AU ($1.51*10^8$ km) and a perihelion that is less than 1 AU ($1.46*10^8$ million km). Figure 4 shows how the spacecraft distance from the Sun will change over the course of the year.

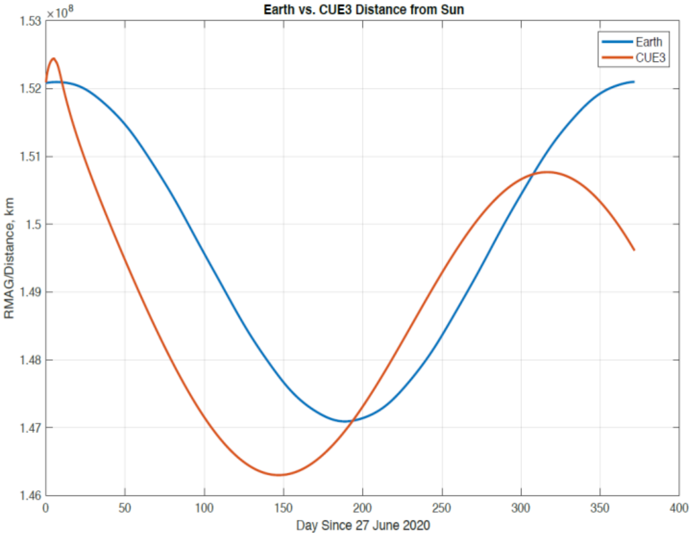


Figure 4: CU-E^3 distance from the Sun compared to the Earth

2.4. Previous Scenarios

In October 2017, NASA delivered to its secondary payloads Preliminary Mission Analysis (PMA) (ULA PMA) data, which represents a simulation of the ICPS. The data consisted of timestamps about a minute apart, starting at BS1 and stopping at BS5, with a provided state vector at each time stamp. The state vectors consisted of Cartesian J2000 position, velocity, Julian date, body-centered frame rotation rates, and a constant to convert from the ICPS frame to the J2000 frame. Payloads were instructed to perform their own dispersion analysis and return that to NASA based on the 3,001 trajectories that were sent, one nominal trajectory, and 3,000 dispersed trajectories. Since CU-E^3 is a deep space mission, BS5 is the choice of deployment location, and thus used for the simulations. Since the payload is entering a heliocentric orbit, the risk of recontact with NASA's Orion capsule was calculated to be less than one in a million, as was the risk of lunar impact, ICPS recontact, or lunar-bound secondary payload recontact. The PMA analysis will therefore simulate a deployment from the ICPS at BS5 in each of the 3,000 dispersion trajectories and propagate the resulting trajectory for 28 days per NASA recontact requirements. The nominal case, called Case 0, exhibits the nominal flight conditions of the ICPS (e.g., a 6.5-day trip to the Moon). This nominal case is used for trajectory analysis and design decision for CU-E^3, while the remaining dispersions are used to perform the PMA analysis and provided to NASA. Figure 5 illustrates the final heliocentric orbit that the lunar flyby places CU-E^3 in. The orbit is an Earth-trailing orbit with a semi-major axis greater than 1 AU.

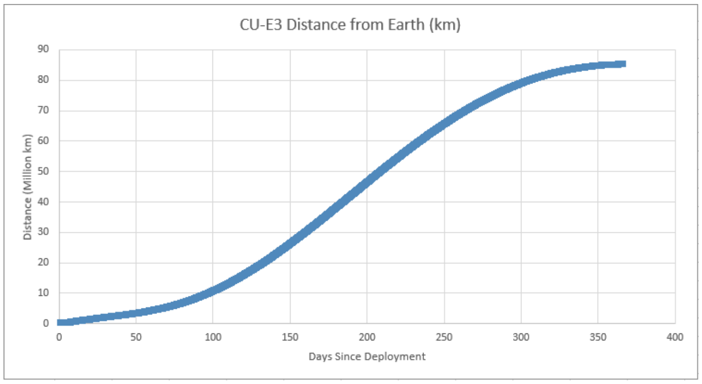


Figure 5: First PMA trajectory analysis showing CU-E^3 trajectory distance from the Earth.

The trajectory analysis showed that CU-E^3 will obtain a maximum distance of ~87 million km in the course of the year-long competition phase. The start of this trajectory is also after the CubeSat has deployed at Bus Stop 5. This much larger distance was due to a completely different launch timeframe and lunar flyby that was planned before NASA settled on the current mission trajectory, described in section 2.1, for the Artemis-1 mission. This further distance was a worse scenario for the CU-E^3 CubeSat, since it is a pure technology demo for communication, and a larger distance would reduce the ability to communicate with Earth at rates capable of winning the competition.

2.1. Future Scenarios

Should NASA release the Final Mission Analysis (FMA) trajectories, the CU-E^3 team will run that analysis to obtain updated launch trajectories. Both trajectories that have been analyzed show vastly different scenarios with different Sun-Earth probe angles and distance from Earth. This has helped shape the design of CU-E^3 to be adaptable to drastically different scenarios including solar panels that can be adjusted before launch, based on the Sun-Earth angle to a potentially different launch and deployment trajectory of the Artemis-1 mission.

3. Solution Scenarios to Obtain Time and State

Four different techniques have been examined to find the best solution to this time and state problem for CubeSats launching on Artemis-1. The transmission and reception apertures are pointed in the same direction for CU-E^3, as shown in Figure 2. Once a solution is formulated to establish communication, the correct time, position, and velocity could be transmitted to the satellites, this state would need to take into account any time delays between transmission, and make sure to transmit a state that accounts for this time delay, thus providing the CU-E^3 with the correct time, position, and velocity.

3.1. Ground Station Transmission

The first method of trying to establish communication and transfer time and state to the CubeSat is to continuously transmit from the ground and hope the spacecraft geometry will be receptive to the transmission from the ground station. The main difficulties with this method are the following:

- Limited ground station resources due to cost and scheduling: CU-E^3 is a student project solely funded from ground tournament winnings and donations. This puts a limit on the amount of resources that can be allocated to the ground station. The Deep Space Network is a costly avenue for continuous communication time for commissioning of this magnitude, and even ground station networks such as ATLAS ground stations are out of reach from the CU-E^3 budget for such long periods of time.

- No tracking data available: As stated by NASA in the Secondary Payload Interface Definition & Requirements Document, the only tracking data that will be done on the CU-E^3 CubeSats is verification that the CubeSat is past four million km, and verification of the distance from Earth at the end of the competition. This will make it difficult to accurately track CU-E^3 which will make it harder to utilize transmitting from ground stations to maximum effectiveness. The ground station will be broadcasting in the blind to the best-known location of the CU-E^3 CubeSat and will be limited by the beamwidth of the ground station.

- Unfavorable geometry of the spacecraft: It is entirely possible that CU-E^3 may end up in a tumble or attitude that is unfavorable to communicating with the Earth initially, while it becomes three-axis-stabilized. With no knowledge of time, even when CU-E^3 obtains an attitude estimation via a star tracker, it will not have any knowledge of the location of Earth after deployment and will be unable to point in the right direction immediately. The antenna configuration increases the likelihood of being in an unfavorable geometry to either transmit or receive towards Earth.

For CU-E^3, the ground station transmission method is unfortunately out of reach, due to cost restrictions. Should unlimited access to DSN or any ground station be available to a CubeSat, this method would require the least amount of design changes and development to any mission as the DSN or ground station is already a completely setup resource.

3.2. Patterned Transmission from Satellite

A more complex method of recovering communication with the CubeSat is to setup a transmission sequence from the spacecraft before it is delivered and subsequently deployed from the SLS. The premise behind this would be to take the beamwidth of the transmission aperture of the CubeSat and devise a pattern that will cover the entire sky in a minimal amount of time, to let the ground station know that the spacecraft is alive and active. A minimal amount of time is desired as it will allow commissioning to be completed more quickly and ensure CU-E^3 is prepared for the competition phase. After an entire sky transmission sweep, the spacecraft would perform a reception sweep to allow the ground station to transmit time and state to the CubeSat. This transmission phase from the ground station will be based on the estimate of the trajectory of CU-E^3 from the last known state of the ICPS, which will be made available to the CubeSats on the rocket. The transmission beamwidth of the CU-E^3 CubeSat is either 10 or 40 degrees half-power beamwidth depending on which aperture is used (the reflectarray or feedhorn). Calculations will be done for both for completeness, but the feedhorn with the 40-degree beamwidth will be utilized for this patterned transmission option, up to 2.5 million kilometers from Earth.

A step further of this method would be to make use of the FMA trajectories, once released by NASA, to create a bounding location for CU-E^3 to ensure a sweep that would find Earth. This bounding sweep would be able to minimize the amount of sky coverage that CU-E^3 would need to cover to find Earth by using the attitude solution obtained from the star tracker and the trajectory that CU-E^3 should be on based on the FMA. Because the CubeSats must be delivered months in advance of the SLS launch, a major downside to this additional step is that if there were any delay in the SLS launch, this analysis would no longer be viable and would potentially cause an issue with location of the Earth in the prearranged sweep. Thus, this method was not used in the analysis.

To calculate the optimal number of transmissions required to do a full sky sweep, the surface area of a sphere can be compared to the surface area of the transmission pattern on that sphere which would be a spherical cap. The following equations are derived to perform this calculation, using Figure 6 as a reference, where:

$$r = radius\ of\ the\ sphere,$$

$$h = height\ of\ the\ spherical\ cap,$$

$$a = radius\ of\ the\ spherical\ cap,$$

$$\theta = half\ the\ beamwidth\ of\ transmission\ apurature,$$

$$Surface\ area\ of\ the\ sphere\ \left(SA_{sphere}\right) = 4\pi r^2,$$

$$Surface\ area\ of\ the\ spherical\ cap\ (SA_{cap}) = 2\pi rh,$$

$$h = r(1 - cos\ cos\ (\theta)\ ).$$

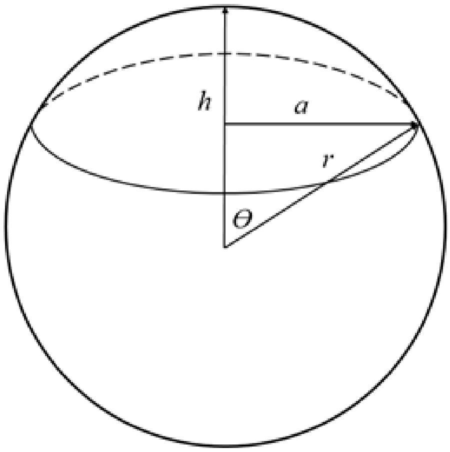


Figure 6: Depiction of a spherical cap.

Since the number of spherical caps that can fit in the surface area of the sphere is needed, the equations can be rearranged as follows:

$$\frac{SA_{sphere}}{SA_{cap}}, \tag{1}$$

$$\frac{4\pi r^2}{2\pi r * r(1 - coscos\ (\theta)\ )}, \tag{2}$$

yielding:

$$\frac{2}{1 - coscos\ (\theta)}\ . \tag{3}$$

Using this equation and the different aperture beamwidths of CU-E^3, the number of transmissions to complete a full sky sweep with no overlap can be obtained, but an overlap percentage must be added to account for the circular nature of the aperture beamwidths. The optimal overlap of identical circles occurs when six circles surround one circle and intersect at identical distances as shown in Figure 7. (Kamer, 2013)

With identical circles overlapping, the coverage of one circle becomes the same as a regular hexagon with the same radius. This can be expressed as:

$$A_{circle} = \pi r^2,$$

$$A_{hexigon} = 3r^2\ sin\ sin\ \left(\frac{\pi}{3}\right), \tag{4}$$

$$\frac{A_{hexigon}}{A_{circle}} = \frac{3r^2\ sin\ sin\ \left(\frac{\pi}{3}\right)}{\pi r^2} = \frac{3\ sin\ sin\ \left(\frac{\pi}{3}\right)}{\pi}$$

$$= .8270.$$

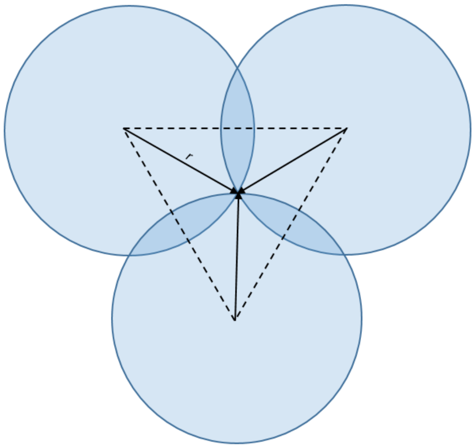


Figure 7: Identical circles of equal radius overlapping.

This yields a 17.3% coverage factor that needs to be applied to yield the number of transmissions to achieve full sky coverage with overlap, thus leaving no space uncovered. Table 2 shows the number of transmissions required to cover the sky and also the time it would take, assuming a 10 second transmission time and a 20 second slew time for a total of 30 seconds per observation point. Also presented in the table is the 120-degree receiving beamwidth of the patch

Table 2. Transmissions and Receptions Required to Complete Full Sky Coverage

Beamwidth	*Number of transmissions to do an entire sky sweep*	*Time to Complete*
10 deg	132	66 minutes
40 deg	9	4.5 minutes
120 deg	3	6 minutes
10 deg (Overlap)	154	77 minutes
40 deg (Overlap)	10	5 minutes
120 deg (Overlap)	4	8 minutes

antenna. For receiving, the CubeSat will wait for two minutes in each location for a transmission from the ground.

These results show that using the 10-degree beamwidth reflectarray would consume the most time of the two transmission options that the ground station would have to listen for CU-E^3. With the 40-degree beamwidth feedhorn, full sky coverage can be obtained in just a matter of minutes, allowing a quick transition into a reception sweep. The procedure will be processed as follows:

- CU-E^3 initiates transmission sequence with feedhorn. This transmission will be used as a way to notify the ground of the status of CU-E^3 and also inform the ground that reception sweeps are about to be performed.
- Once the transmission sweep is completed, CU-E^3 will go into a reception sweep allowing ample time for the ground to send an updated time and state data.
- CU-E^3 can transmit and receive simultaneously so as soon as the ground hears a transmission it should begin transmitting.

Times can be adjusted to follow power constraints and ground station limitations. This pattern would be repeated on some predefined interval so that the ground station would not need to be set to continuously receive.

3.3. Locating Earth with a Star Tracker

Recovering communication with the Earth could involve utilizing the star tracker located on the CubeSat to find Earth and point the communication array towards the Earth. When the CU-E^3 CubeSat is deployed from the EM-1 rocket body at Bus Stop 5, it will be ~330,000 km from the Earth and ~48,000 km from the Moon; this corresponds to a magnitude of -15.12 and -17.26, respectively. By modifying the star tracker software, it can also be used to identify planets. (Enright et al., 2010). For the specific scenario of the SLS launch, the Earth and Moon will be large objects in the field of view of the star tracker, making them easy to identify.

Using a Sinclair star tracker (Sinclair, 2018) as a baseline, the size of Earth and the Moon on the camera array can be calculated as follows:

$$h_o = \frac{f * H * h_i}{D * h_s}, \qquad (5)$$

where h_o is the object height in the camera, f is the focal length of the camera in km, H is the real height (diameter) of the object in the camera in km, h_i is the image height in pixels, D is the distance to the object in km, and h_s is the sensor height of the camera in km. For the Sinclair star tracker these values become, $f = 25.6\,mm$, $h_i = 2592\,pixels$, $h_s = 7.13\,mm$, and D and H vary based on the selected object. Using a distance of ~330,000 km from the Earth and ~48,000 km from the Moon for D and the diameters of Earth and Moon yields the following Figures 8 and 9 of what each object's apparent size would be in the Sinclair star tracker which has an array of 2592 x 1944 pixels and a 10-degree field of view, half angle. The Sinclair star tracker can also see down to apparent magnitude of 7.5 for objects in the field of view.

These Figures show that at BS5, the Moon will be larger in apparent size on the Sinclair star tracker than the Earth. Both the Earth and Moon will be larger than the apparent size of the Sun, which is shown in Figure 10, although not as bright as the Sun, which has an average apparent magnitude of -26.74 at 1 AU.

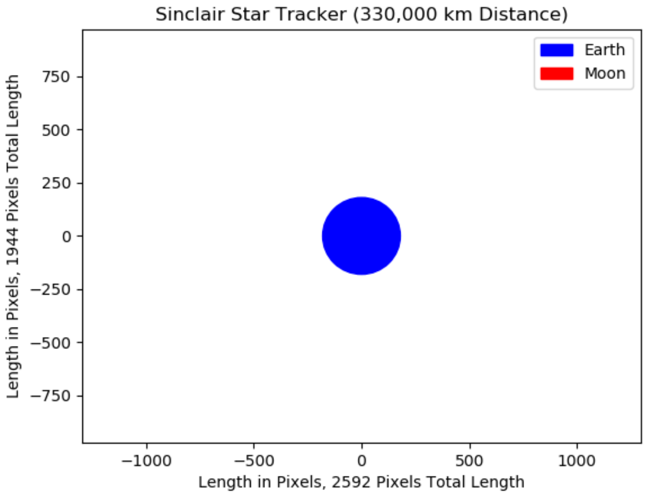


Figure 8: Apparent size of Earth at 330,000 km using Sinclair star tracker.

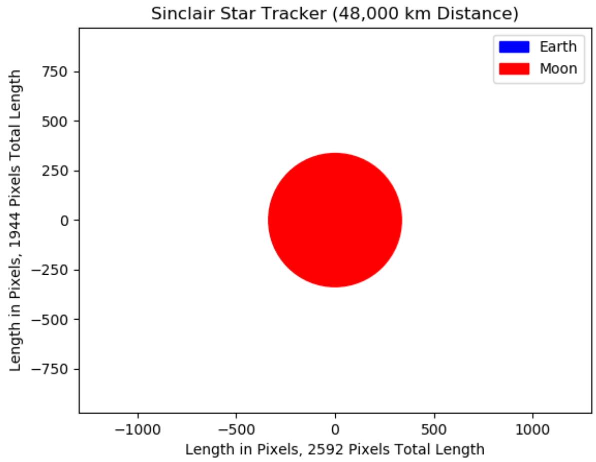


Figure 9: Apparent size of Moon at 48,000 km using Sinclair star tracker.

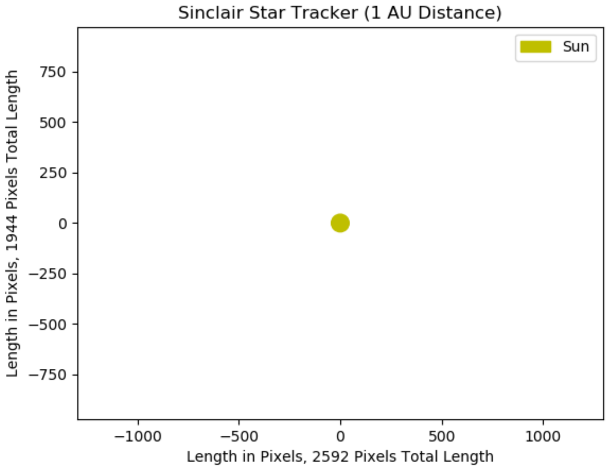


Figure 10: Apparent size of the Sun at 1 AU using Sinclair star tracker.

The star tracker is designed not to look at the Sun via keep-out zones algorithmically ingrained into the software, but can look at the Sun for a short period of time without damage, though if the Sun is in the scene, then the Earth or Moon would not be able to be seen. The size comparison shows that even if the star tracker looks at the Sun, it should not confuse it for the Earth or the Moon.

Using these Figures, a method can be devised with the star tracker to do an entire sky sweep until it finds a bright object that fills a large portion of the aperture. The sun will be too bright and much smaller than the Earth and Moon, and thus can be eliminated as a target. This will allow CU-E^3 to then transmit to either target, the Earth or the Moon, and receive at the same time, to try and obtain time and state from the ground station. Transmission to the Moon is only necessary because positive identification of the Earth versus the Moon may be impossible once CU-E^3 reaches a distance where the objects are equal in size. It is possible that the brightness of the objects could be used, but given the possibility of a change in trajectory, the phases of the Earth and Moon and position of the spacecraft may cause the Earth and Moon to appear at the same magnitude. Once the targets are identified, it is easy to set up a sequence of transmissions to each and planned receiving times to accommodate any ground station requirements.

3.4. Full Sky Receiving Coverage

Full sky reception can be done by having receiving antennas located on each face of the CubeSat, allowing reception to the CubeSat regardless of orientation. This does pose a few constraints on the design of the CubeSat in the way of cost, volume, and logistics. With CU-E^3 being a university-funded and student-built CubeSat, the mission operates on a limited budget, so the cost to have six C-band patch antennas, one on each face of the CubeSat, was prohibitive. There is also a volume and logistics constraint where there was not enough room on one face of CU-E^3 and because there is the desire to use semi-rigid cables for the least loss of gain, one antenna could not be attached during the final integration of the CubeSat. Another way to obtain full sky coverage is to have a ground station powerful enough to allow the CubeSat to use two 1/4-wave monopole antennas on the CubeSat that are orthogonal to each other. Two 1/4-wave monopole antennas are not feasible due to the constraint of funding and availability of the ground station. Should a mission have the available funding, these options would be potentially viable in helping to communicate with the CubeSat.

4. Conclusion

Of the four methods presented, a star tracker used to locate the Earth would require the least amount of additional cost, as CU-E^3 will already have a star tracker located on board. This method and the patterned transmission method would both require some minor software development to make sure CU-E^3 could perform the functions correctly. While making use of multiple receiving antennas and unlimited ground station resources are the simplest in terms of execution, they are the largest costs. For future missions, it would be beneficial for time and state data to be provided to the CubeSats and the CubeSats should be allowed to maintain a small power source, such as a button cell, to retain clock timing from delivery.

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APPENDIX 1: Artemis-1 Mission Timeline (Image Credit: NASA Space Launch System Program)