Pole-Sitter Based Space Domain Awareness for Cislunar Regions

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ABSTRACT

Modern space missions are increasingly venturing across cislunar space, requiring expansion of space awareness functions. Legacy Space Domain Awareness (SDA) systems were not originally built to detect and track cislunar objects, and this could require acquisition of new sensor systems. There are numerous parameters, including sensing type, altitude, and number of platforms that could be varied for each system. One key advantage to any "pole-sitter" is that it has a position well outside the ecliptic plane and offers a unique, in some cases orthogonal viewing geometry that here to fore have not been developed for operational deployment. In this paper, the physics of the pole-sitter trajectory, the trade of fuel against altitude and updates in technology, which all point towards the feasibility of demonstrating in the near term a pole-sitter SDA capability, are discussed. In addition, this paper devises a proposed prototype using small spacecraft working in conjunction with ground-based sensors with descriptions of current technology ready for deployment.

1. INTRODUCTION

US Space Force (USSF) needs to continuously monitor Earth orbits as well as cislunar space (the vast region of space above geostationary orbit (GEO) and extending beyond the orbit of the moon) to assure freedom of action in space [1]. The current Space Domain Awareness (SDA) architecture consists of ground and space assets, composed of optical and radar assets that provide various levels of capability to detect and track resident space objects (RSO) in low Earth orbit (LEO), medium Earth orbit (MEO), and GEO. A review of existing cislunar SDA architecture can be found in [2]. Ground-based assets can make intermittent observations of cislunar RSO - but are fundamentally limited by atmospheric effects, weather, day-night cycles. To provide low latency, continuous monitoring of cislunar objects requires a space-based observing network. Fig. 1 displays numerous different orbits and their possible missions and describes the importance of the cislunar region.

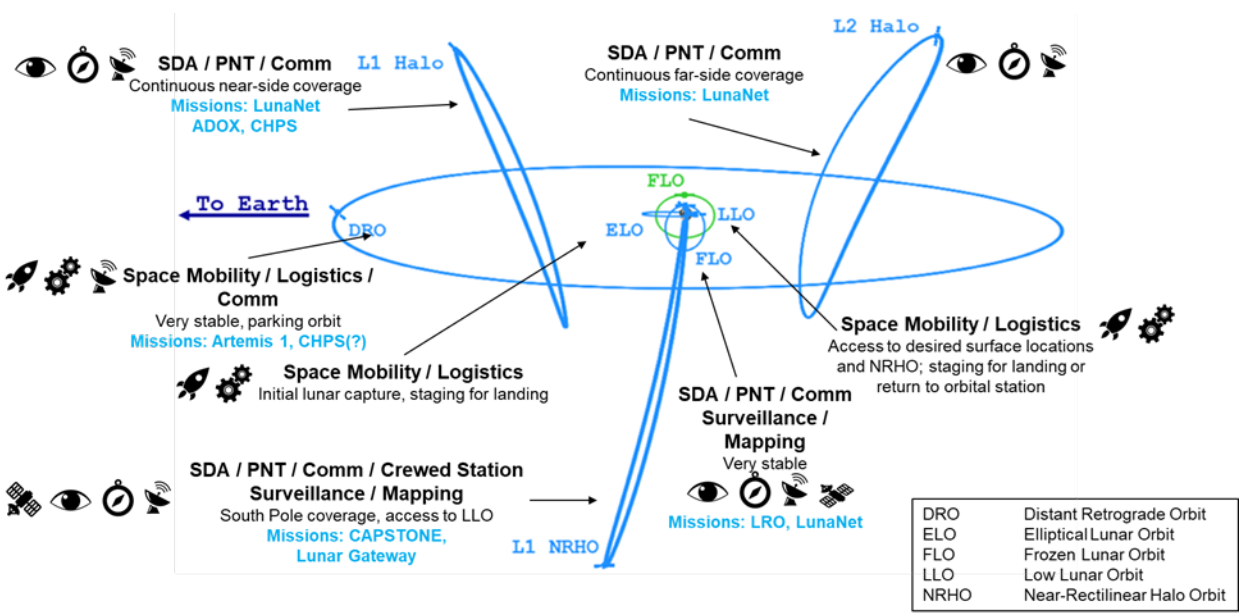

Fig. 1. Typical Orbits of Cislunar Missions

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The orbits cited in Fig. 1 describe the wide range of missions that may be flown by friends and adversaries in cislunar space. These orbits will encompass the typical targets that would be observed by the cislunar SDA infrastructure (a combination of ground and space-based instruments). One of the overarching requirements for observing these targets is to maintain sufficient custody of the trajectories so that the combined observers compensate for the black-out periods of each other – providing quasi-continuous tracking. The near-rectilinear halo orbit (NRHO), commonly used by NASA (e.g., [3]), is of particular interest as it will be the orbit flown by the NASA Gateway for the Artemis mission back to the moon and is presently being explored by the CAPSTONE mission, a CubeSat pathfinder to explore trajectory management to and at the NRHO. Once the Artemis mission ramps up, heavy traffic is expected in the NRHO.

The USSF is presently conducting Project Rocket [1], an extensive measurement campaign, to determine the capabilities of ground-based telescopes to acquire, track and characterize cislunar objects. The capabilities of present and future ground-based stations to initially acquire cislunar objects for queuing of orbital sensors have been recently studied [1]. Preliminary results confirm that the ground arrays have significant data collection features but are limited by timely instrument availability, day-night cycle, weather, atmospheric absorption, and turbulence, etc. The next phase of Project Rocket will enhance the measurement sets, and deposit them into the Universal Data Library (UDL) for examination by the interested community. Given that ground-based sensors are insufficient to always cover the entire cislunar region, data being collected by ground-based sensors as part of Project Rocket will establish the needed gap-filling requirements for a future space-based observer.

The exclusive use of ground-based sensors has drawbacks such as numerous gaps in coverage for both the optical, passive RF, and radar sensors. Additional complications are large exclusion areas near the moon, that are further worsened due to Earth's atmospheric effects – a space-based observer would have clearer viewing of areas much closer to the moon, an important part of the cislunar volume. Cislunar RSOs are illuminated primarily by the Sun, but also by moonlight and Earth light that may come from different directions. One of the difficulties of observing such RSOs is that unlike typical GEO spacecraft that have large solar arrays, these are typically "boxes" with only a few square meters of illuminated surface. The challenge is clear: closing the coverage gaps and finding objects that have different characteristics than spacecraft traditionally tracked.

One co-author previously proposed three space-based SDA options for monitoring Earth orbiting targets including 1) two satellites at a 12-day highly elliptical orbit (HEO), 2) one satellite at L1 Lagrange point, and 3) a pole-sitter satellite 2.5 MKm above either the North or South Pole [4, 5, 6]. Benefits of such orbits offering continuous tracking of candidate RSOs are twofold, efficiency, and characterization. The efficiency comes in the act of not having to frequently reacquire the RSO when the custody chain is broken due to Earth eclipse. Near continuous tracking of all space objects significantly enhances SDA capabilities and increases warning times for potential attacks in orbit. The second benefit is that once detected and continuously tracked, any behavior of the RSO begins to indicate "its pattern of life" and this leads to a better understanding of the intent of the motion or action of the RSO. Among the three originally studied, pole-sitter based SDA is considered most prominent with the advantages over the other two options in terms of 1) continuous coverage, 2) the effort an adversary would need to expend to physically deny the gathering of the RSO data, requiring 400 times of combined energy and time expenditure to reach 2.5 MKm altitude than to arrive at GEO [4, 5, 6].

Using pole-sitter as shown in Fig. 2 for persistent overhead observations of near-Earth orbital objects first appeared in the literature over a decade ago and its conceptual mission design was proposed [7, 8, 9]. The orbital dynamic model utilizing a circular restricted three-body problem (CR3BP) defined rotating frame and mission scenarios were analyzed [7], while trajectory designs to reach pole-sitter location and orbit maintenance trades on solar electric propulsion (SEP) vs. solar sail were performed [8, 9]. Note that the Earth orbit is slightly elliptical with a perihelion in the winter and aphelion in the summer but the small differences to a circular orbit are negligible in the analysis.

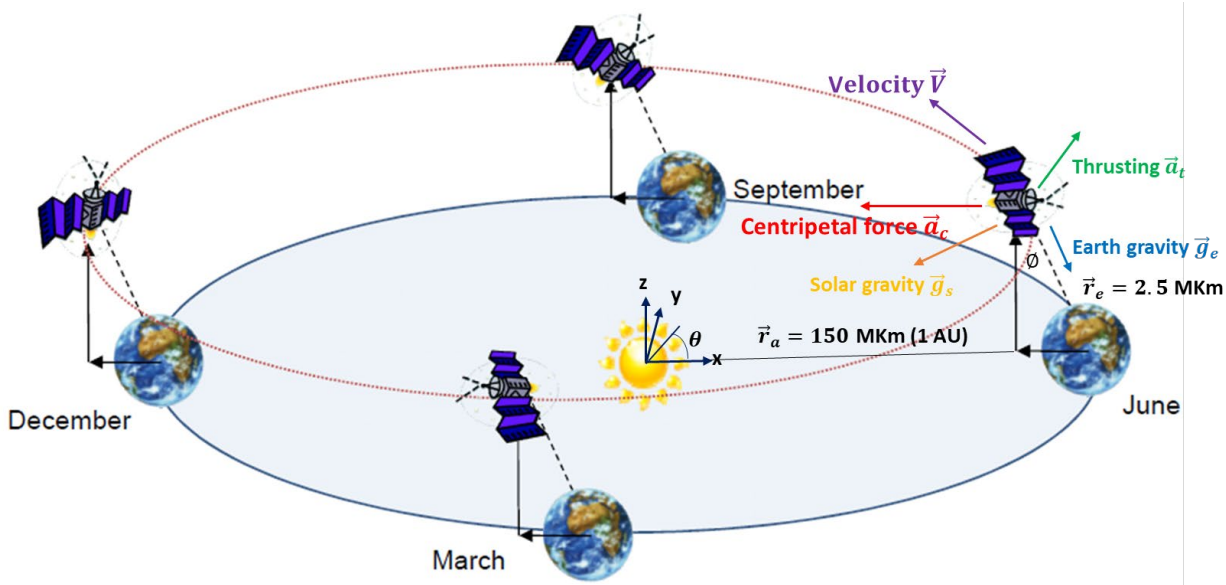

Fig. 2. Pole-Sitter Concept in Sun-Centered Inertial Frame (revised from [8, 9] not to scale)

Due to the high altitude of a pole-sitter spacecraft at between 1.0 and 2.5 MKm, it could monitor cislunar space. This paper proposes deployment of a pole-sitter sensor system in accordance with the co-author's prior publications as part of such a space-based and ground-based SDA observing and tracking capability. This SDA system would detect and track RSOs in the cislunar space and provide state vector and intensity data to ground for further characterization. Many cislunar RSOs can be occasionally observed by ground-based instruments but continuous custody will be needed which reinforces the need for space-based observing to provide quasi-continuous monitoring.

2. POLE-SITTER MISSION DESIGN

The 2.5 MKm altitude used in the co-author's prior publications is far enough from Earth to allow spacecraft to "hover" over the North or South pole and conduct mission operations. Due to the booming popularity in low-cost small satellites (SmallSat) and the associated technological advancements, it would be of interest to develop a pole-sitter design based on smaller vehicle as shown in Fig. 3, which could be demonstrated much sooner than a large full-scale space vehicle. Such SmallSat would follow the trend of high-performance deep space CubeSats (e.g., [3]) and with a replenishment strategy that would allow periodic technology upgrades.

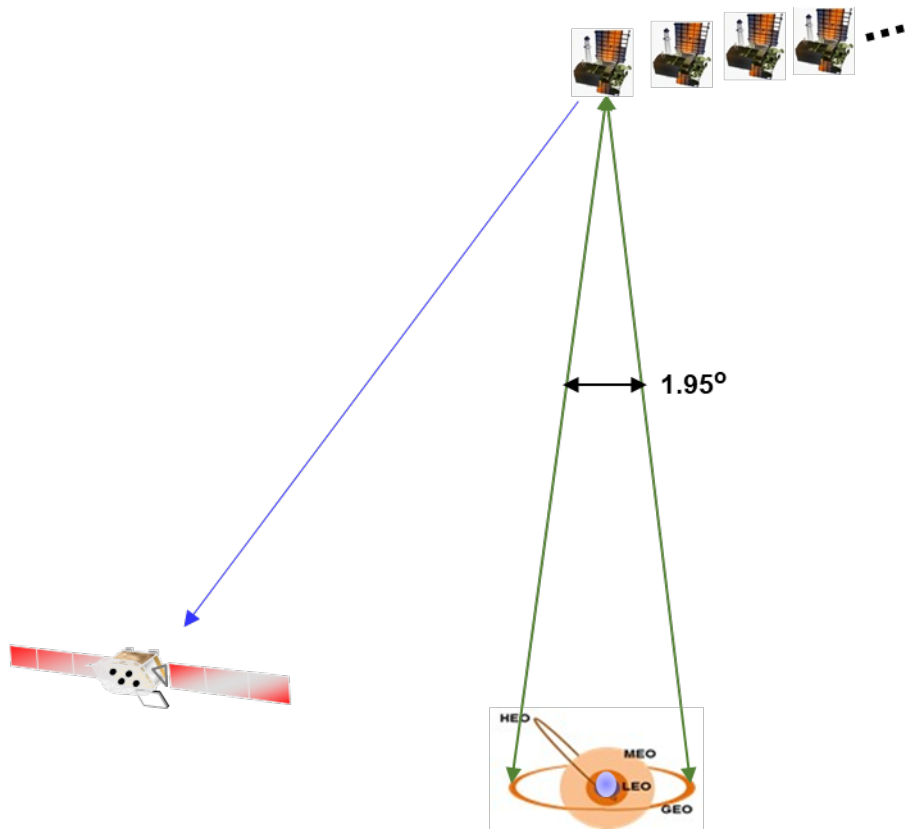

Fig. 3. Pole-Sitter Surveillance Concept (not to scale)

To assess the mission feasibility, a pole-sitter sensor system hovering ~2.5 MKm over the North Pole was analyzed. This vantage point has virtually unobstructed view of cislunar space. The orbital stability of the Pole-Sitting system depends upon a balance of the contending forces. The dynamic model has been formulated in a CR3BP based rotating frame [7]. However, for conceptual design and trades on required orbital maintenance thrust and propellant, a kinematic formulation in a Sun-Centered Inertial Frame as defined in Fig. 2 would be more straightforward and easily understandable. To maintain the circular non-Keplerian orbit (red dash line in Fig. 2) in synchronization with Earth rotating around Sun, the acceleration required is based on a spacecraft hovering at a constant relative position to Earth, the Earth gravity on the spacecraft is also constant and pointing toward North Pole in this case. As shown in Fig. 2, the pole-sitter's non-Keplerian orbit rotational axis is off-centered to Earth's rotation axis centered at the origin of the Sun-Centered Inertial Frame. Therefore, the Solar gravity effect on the pole-sitter varies over the course of the year sinusoidally depending on the distance to the Sun. For example, when the pole-sitter dwells over the North Pole, the Solar gravity effect is stronger in June when the Sun is closer, and weaker in December when the Sun is further away. With the above descriptions, the required orbital maintenance thrust can then be solved, and annual propellant usage estimated. Intuitively, this continuous thrusting needs to both resist the pull-down gravity from Earth and Sun towards the ecliptic and provide corrections to offset variations in the centripetal force from the gravitational effects of the Sun and the Earth.

Back-of-the-envelope worst case annual ΔV, 1st–year fuel usage, and lifetime estimation for a proposed pole-sitter spacecraft flight demonstration are found in Table 1, considering the kinematic model shown in Fig. 2. These calculations assume a spacecraft hovering over the North Pole at 2.5 MKm, a spacecraft wet mass of ~200 Kg consisting of ~100 Kg dry mass plus ~100 Kg of propellent, and the use of SEP thruster with I_{SP} of 3,200 seconds as used in [7, 8, 9]. It appears that the spring/fall equinox represents the worst case amongst the 4 possible seasonal worst cases in terms of the required thrust and resulting fuel usage. The orbital maintenance fuel required for the first year is approximately 30 Kg and decreasing thereafter over the remaining lifetime due to constantly consumed

propellant mass per the rocket equation in [10]. In that case, the total lifetime is about 4.2 years for the proposed demonstration.

Table 1. Back-of-the-Envelope Worst Case Annual ΔV, 1st–Year Fuel Usage, and Lifetime Estimation

Variable	Summer Solstice	Fall Equinox	Winter Solstice	Spring Equinox	Units
Centripetal acceleration, X-axis	-0.005930	0	0.005930	0	m/sec^2
Earth gravity, X-axis	2.530E-05	2.530E-05	2.530E-05	2.530E-05	m/sec^2
Solar gravity, X-axis	-0.006008	3.960E-05	0.005850	3.960E-05	m/sec^2
Thrusting acceleration, X-axis	5.258E-05	-6.490E-05	5.519E-05	-6.490E-05	m/sec^2
Centripetal acceleration, Y-axis	0	-0.005930	0	0.005930	m/sec^2
Solar gravity, Y-axis	0	-0.005928	0	0.005928	m/sec^2
Thrusting acceleration, Y-axis	0	-2.496E-06	0	2.496E-06	m/sec^2
Earth gravity, Z-axis	-5.819E-05	-5.819E-05	-5.819E-05	-5.819E-05	m/sec^2
Solar gravity, Z-axis	-9.293E-05	-9.108E-05	-8.928E-05	-9.108E-05	m/sec^2
Thrusting acceleration, Z-axis	0.0001511	0.0001493	0.0001475	0.0001493	m/sec^2
Total required thrusting acceleration	0.0001600	0.0001628	0.0001575	0.0001628	m/sec^2
Annual ΔV	5,050	5,137	4,969	5,137	m/sec
1st–year fuel usage	29.75	30.22	29.31	30.23	Kg
Lifetime estimation	4.30	4.23	4.37	4.23	years

In addition to the calculations for the ~2.5 MKm case, there is a trade space to consider for altitude versus fuel expenditure. Fuel usage is a function of altitude; as altitude goes up there is a reduction in the amount of fuel required, reduction in space vehicle wet mass, and increase in system lifetime. Considering the kinematic model shown in Fig. 2, Fig. 4 shows the required thrusting acceleration necessary to maintain a pole-sitter in its location above the pole throughout a one year period . Note that 0 deg in the X-Y plane is summer solstice, while 90 deg is fall equinox, and so forth. Multiple simulation runs with varying altitudes were performed to predict the fuel usage which is strictly related to the magnitude of the required orbit maintenance thrusting acceleration. Since most of the orbital maintenance thrust is used to resist the pull-down force from both Earth and Sun gravities, the Z-axis component in the Sun-Centered Inertial Frame is most dominant as expected. While Earth gravity is constant due to its fixed relative position to pole-sitter, solar gravity is sinusoidal depending on season as reflected in all three axes (Z-axis component looks constant due to scaling). Also, when altitude is low, Earth gravity dominates. However, solar gravity dominates when altitude is high.

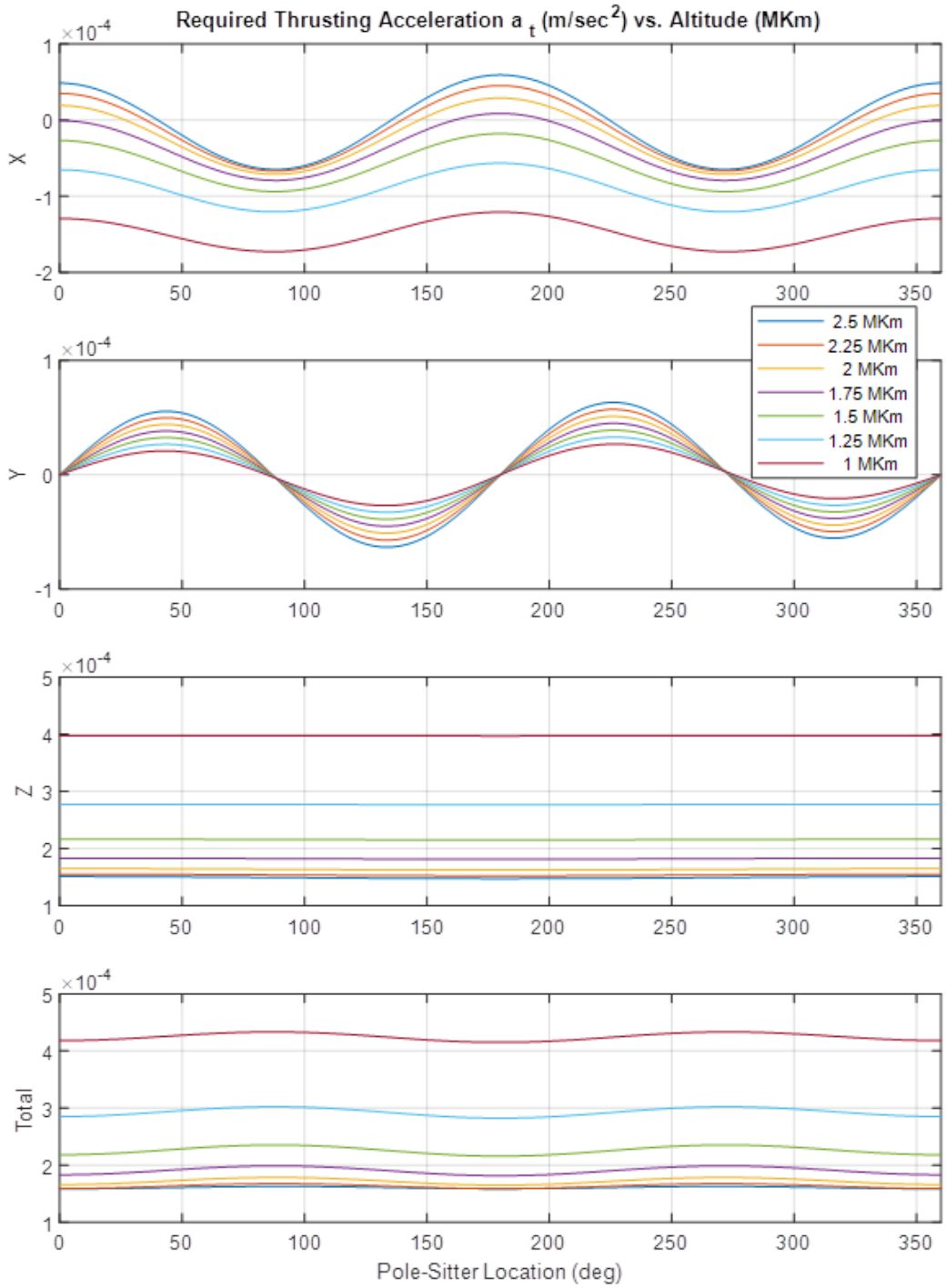

Fig. 4. Required Thrusting Acceleration to Maintain the Pole-Sitting Orbit at Different Altitudes

The annual ΔV required for orbital maintenance is calculated by integrating the thrusting acceleration in Fig. 4 over time. The expected lifetime is then calculated by the rocket equation in [10]. The results for different altitudes are shown in Fig. 5.

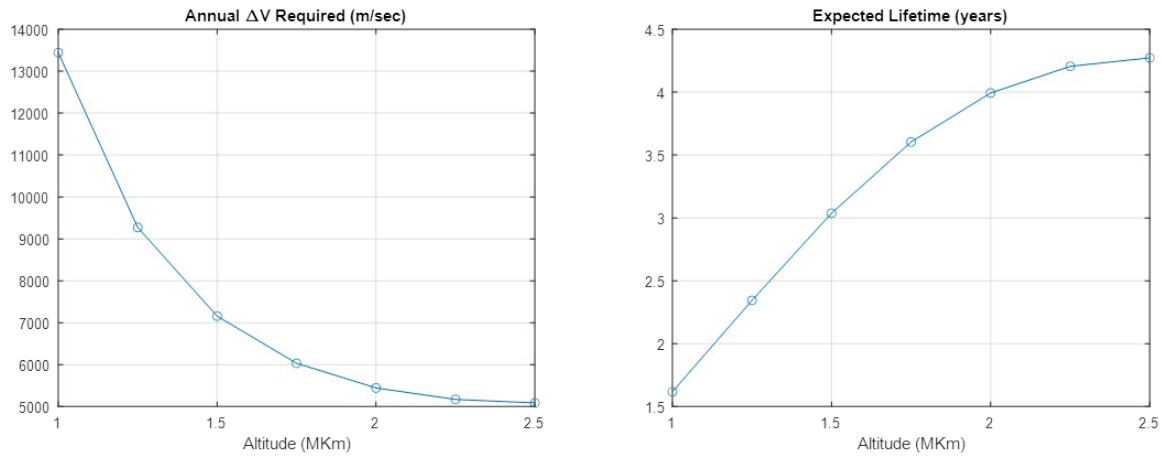

Fig. 5. Annual ΔV Required for Orbit Maintenance and Expected Lifetime

3. PROPULSION CONCEPT

Propulsion is a challenge for a pole-sitter satellite. The first requirement will be the orbit transfer out of the Sun-Earth orbital plane, that will most likely be accomplished by the third stage of the booster to minimize fuel needed on the spacecraft. However, to maintain a non-Keplerian orbit shown in Fig. 2 requires continuous station-keeping, which becomes the most critical requirement for pole-sitter satellite design as no other disposable replacement is available. Therefore, both solar sail and solar electric propulsion (SEP) are proposed in [7, 8, 9].

Solar sail does not require propellant and nowadays is mature and capable of up to several AU per year [11]. Solar sails under development by NASA programs will provide both TRL maturity and expected performance versus weight. Examples of the state of the art of potential solar sails are shown in Fig. 6. Note that solar sail propulsion suffers the pitfall of controllability [3] and is only effective when thrust direction is outward from Sun. As such, solar sail propulsion alone, with the orbital geometry shown in Fig. 2, will not work in all seasons. Solar sail propulsion is useful only when gravity forces are larger than the required centripetal force. Therefore, either SEP or a more conventional propulsion or a hybrid system with solar sail is needed. In essence, the SEP thrusters provide the upward component of force to overcome both Sun's and Earth's gravities, while the sails provide the lateral force needed to overcome the excessive centripetal force from solar gravity in the summer due to shorter distance to Sun than that between Earth and Sun. On the other hand, solar sails would not work in the winter, while solar gravity is insufficient to support the required centripetal force for the circular orbit.

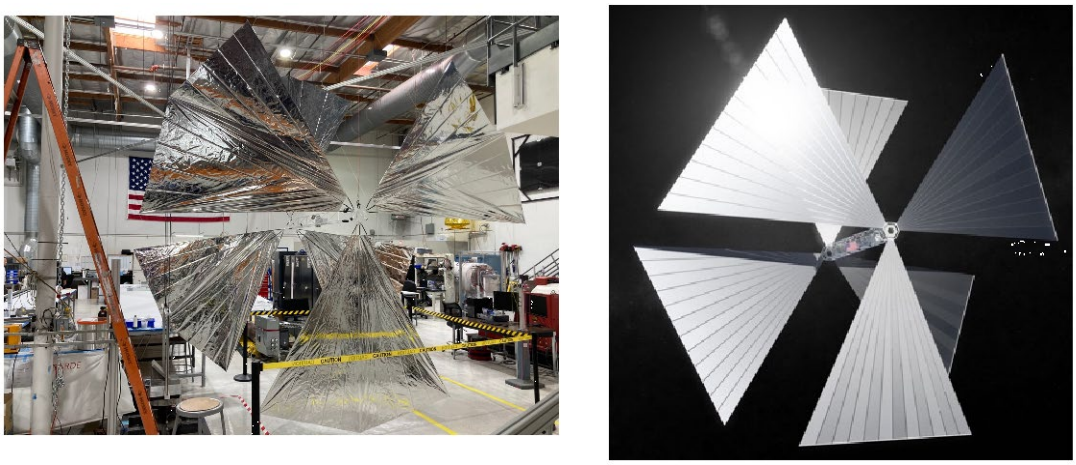

Fig. 6. Prototype Solar Sails (credit and permission: Nxtrac)

SEP on the other hand, due to its low propellant weight required, is also gaining popularity in recent years, and becoming mature for deep space applications. For example, the spacecraft to be launched soon for exploring the Psyche asteroid, exceeding 3 AU from Earth where solar power is limited, employs commercial-off-the-shelf (COTS) Hall-effect thrusters [12]. Nowadays mature Hall-effect thruster technology is typically capable of 1500-2500 sec I_{SP} and ion thruster technology can extend to 3000-4000 sec I_{SP}. The newest liquid metal-based field-emission electric propulsion (FEEP) [13] advanced SEP system is even pushing the boundary beyond 6000-7000 sec I_{SP}. Therefore, SEP propulsion would be a natural baseline for this pole-sitter spacecraft.

To allow for solar sail in all seasons for maximum fuel saving, References [7, 8, 9] proposed an optimization of varying altitude between 2 MKm and 2.8 MKm. This altitude variation range results in minimum SDA performance change. In the summer season higher altitude is allowed due to stronger solar gravity (shorter distance to Sun) resulting combined Earth and solar gravities over the required centripetal force, while in the winter the altitude is lowered to increase the combined Earth and solar gravities beyond the required centripetal force. Fig. 7 shows the comparison of pole-sitter trajectory and altitude using SEP with and without solar sail, where βo is the ratio of the maximum solar radiation pressure acceleration at 1 AU (sail characteristic acceleration) to the gravitational acceleration. A 15% reduction in terms of maximum thrust by using the solar sail together with SEP is reported.

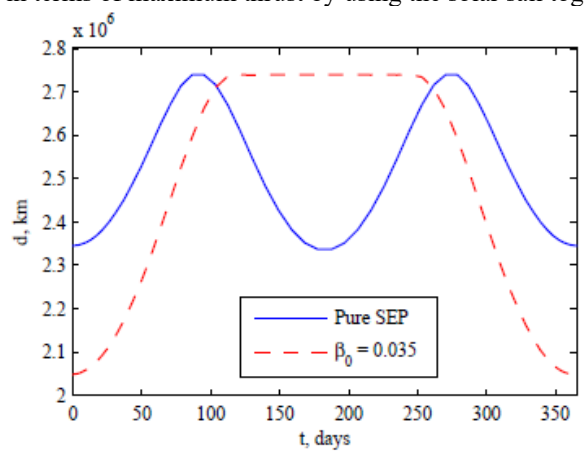

Fig. 7. Optimization of Trajectories and Altitude Using SEP with and without Solar Sail [9]

4. DEVISING A SENSOR SYSTEM FOR A POLESITTER DEMONSTRATION

Due to the extreme distance between the pole-sitter location and the RSOs of interest, only passive detection methods are practical. Moreover, because of the geometries involved, even target reflectance due to sunlight illumination will not always be present. Therefore, to guarantee the ability to detect and track objects, the imager needs to be sufficiently sensitive to RSO natural black-body emission spectra. Assuming an average temperature of $T \approx 300°$ K, RSO emission spectra should exhibit broad peaks spanning the long wave infrared (LWIR) range of $\lambda =$ 8-12 μm. Note that the definition in this paper of LWIR, may not be the same as the definition in other papers. In addition, the MWIR and LWIR definitions can also change in various communities.

RSO also present emissive signatures in the MWIR band, but at a lower intensity than LWIR. MWIR could provide better resolution than LWIR, but the driver on aperture size for the pole-sitter mission is sensitivity rather than resolution because its mission is to detect RSO and not produce an image. The LWIR band requires a smaller aperture than MWIR would because the signal is stronger in LWIR, so a LWIR focal plane array (FPA) imager is the best candidate. In previous studies a larger spacecraft with one aperture was designed. For this smaller demonstration cluster pole-sitter, the design would be to compose a larger aperture from the distributed smaller apertures. In addition, there would be trades as to how the field of view (FOV) of each of the cluster spacecraft would be combined to obtain the systems sensor performance. These additional trades for sensor design will need to trade pixel sizes and detectivity. The original large pole-sitter design was intended to allocate one pixel per IR target with the FOV equating to a GEO graveyard to GEO graveyard distance. For the smaller spacecraft demonstration,

especially with a smaller focal plane, the one pixel per target could be verified, as well as looking at multiple pixel or pixel crossing tests and especially expanding beyond the GEO to cislunar region.

Since the original pole-sitter papers [4, 5, 6] were written there have been numerous improvements and lessons learned for 12 µm LWIR technologies. The original paper described multiple technologies including HgCdTe [14] and those approaches are all still valid candidates for a demonstration sensor spacecraft tradeoff. The addition to that sensor tradeoff family would be the technology flying in the James Webb Space Telescope (JWST). The JWST has launched to L2 recently and its "LWIR" system, Mid Infrared Instrument (MIRI), is operating at sensitivities beyond what pole-sitter would need at 12 µm. The MIRI is 1K × 1K SiAs Impurity Band Conduction (IBC) sensor array developed by Raytheon Vision Systems [15]. This sensor has fewer pixels which imply either a stitching into a larger format or an analysis that separate sensors would be assigned separate regions of space to view. The original intent of the pole-sitter was to be a synoptic sensing system with constant custody and no hand-offs amongst sensors so the approach of stitching together physically separated sensors flying on separate spacecraft would require a concept of operations tradeoff.

The MIRI operates as a cooled device to obtain its exquisite performance and has Responsive Quantum Efficiency (RQE) > 70% at 12 µm. The sensor also has dark current versus temperature performance over a wide bias range with one data point being < 0.1 e-/sec @ 7.1° K. The original pole-sitter design specified a dark current (≤ 25 e−/s/pixel) at 77° K for comparison. This noise to temperature tradeoff will be explored more fully, but it appears some type of cooling is necessary for the 12 µm sensors. The JWST MIRI sensor has an advanced cooling system that does not use any coolant and its operating life is only limited by wear to its moving parts. The system is intended to operate for 20 years or more. This cooling system would likely have to be downsized or compacted to fit into a small spacecraft format.

While the JWST is pushing the state of the art for LWIR technology in space, terrestrial based LWIR sensors are being developed for space applications that are possible candidates for deployment on smaller spacecraft. One very exciting area is the growth of arsenic doped silicon (SiAs) detectors for the 5-28 µm range in a very small format. An example of this new format sensor is the Teledyne BOSON 640 IR camera family which has been successfully flown on a CubeSat mission and is slated to fly on another effort between The Aerospace Corporation and NASA JPL currently called CASTOR. This is a very small and uncooled family of devices that demonstrates a low-cost approach to LWIR sensor development. The BOSON comes with multiple optics options including a 640 x 512 12 µm pitch option and multiple FOV optic choices. There will be additional trade off studies of the various sensor technologies needed, from the MIRI implementation down to the CASTOR implementation and against multiple thermal backgrounds. CASTOR will operate in a higher thermal noise background in LEO and is proposing a 3D formation flying testbed using LWIR sensors. CASTOR could be a great risk reduction effort for the cluster pole-sitter mission by testing how three small satellites with LWIR could perform together.

5. DEEP SPACE NAVIGATION AND COMMUNICATION

Long distance to the ground station imposes another major challenge for pole-sitter spacecraft in two folds: navigation and pointing control without GPS as well as limited bandwidth for communication with ground.

5.1 Deep Space Navigation

While the majority of USSF's high value assets (HVA) are located from LEO to GEO with GPS signals either directly available (main lobe) or manageable (side lobe), NASA has been in the forefront of exploring interplanetary missions for decades and developed mature deep space navigation technologies without GPS. NASA's Deep Space Network (DSN) managed by its Jet Propulsion Laboratory (JPL) at Goldstone in California, Madrid in Spain, and Canberra in Australia equipped antenna up to 70 meter in diameter is capable of collecting two-way Doppler, two-way ranging, and Delta Differenced One-way Ranging (DDOR) measurements for correcting the spacecraft position predicted by its sophisticated orbit determination model as well as performing pointing control from such a long distance, once spacecraft location is known [16]. NASA's deep-space spacecraft typically equips a high gain antenna (HGA) for regular communication at high throughput rate along with a low gain antenna (LGA) with wide FOV for safe-mode communication at low data rate for essential information. Lately, DSN is further enhanced with

the state-of-the-art optical communication terminals. This further pushed the lunar orbit ephemeris accuracy down to center-meter level as proven by the Lunar Laser Communication Demonstration (LLCD), the world's first laser communication between Moon and Earth, on the Lunar Atmosphere and Dust Environment Explorer (LADEE) spacecraft [17].

However, DSN has been oversubscribed among numerous simultaneous NASA missions. The time-share allocated for USSF's use would likely be limited. Therefore, USSF needs to either build its own full-scale deep space navigation infrastructure in the long run or find a viable short-term low-cost alternative.

A near-term proposed solution may be available for a pole-sitter demonstration is the newly developed inverted Positioning, Navigation, and Timing (iPNT) Concept for cislunar [18]. iPNT uses a network of Earth-based beacons to provide one-way navigation signals to range the pole-sitter satellite then correct the orbit determination model via a Kalman filtering scheme. But the proposed use of 2.5-meter C-band ground antenna would not provide the same level of ephemeris accuracy as DSN, whose antenna is up to 70 meter in comparison. Additionally, the non-Keplerian orbit model does not yet exist in the iPNT and is subject to further development. The commonly used two-line element (TLE) method for Keplerian orbit is not applicable anymore. Instead, JPL uses unique Inertial Vector Propagation (IVP) or Thrust Vector Propagation (TVP) techniques for modelling ephemeris of interplanetary mission [16].

Another low-cost alternative utilizes pole-sitter's SDA optical sensor to triangulate lines of sight to known GEO RSOs and stars rather than ground beacons as in the iPNT method. Similar to line-of-sight calibration commonly using star measurements for wide FOV sensor, e.g., Space Based Infrared System (SBIRS) [19] and its next generation design architectures [20], the pole-sitter sensor can be used as a classical sextant to determine the position of the spacecraft. This merely requires the measurement of the angle between selected GEO RSOs and stars. By making these measurements as often as needed, the ephemeris of the pole-sitter can be determined by ground analysis to < 1 Km (3σ), as extrapolated from past SDA orbit determination data. This is like the sextant process used in the Apollo missions and has no impact on spacecraft resources. This method should achieve similar accuracy as the iPNT method. However, ephemeris accuracy would improve with greater number of RSOs at GEO as well as stars being used for the orbit determination. In particular, the ranging accuracy is directly related to the distances among RSOs used for triangulation.

5.2 Deep Space Communication

As previously mentioned, NASA's DSN has a long heritage of deep space communication using radio frequency (RF) with recent addition of optical communication capability. Optical communication performance is supposed to be 100 times over RF except that it is sensitive to degradation due to terrestrial weather conditions. The latest Deep Space Optical Communication (DSOC) flight demonstration scheduled to launch soon on Psyche spacecraft will explore laser communication over 2 AU [21]. DSOC boasts its expected deep space capability of 531Mbps from 40 MKm [22, 23] by using a 22 cm aperture terminal, a high efficiency photon-counting detector, and operational power up to 100 W. Such new technology will be certainly available for pole-sitter's optical communication.

Using this advanced DSOC design as a baseline, back-of-the-envelope calculations were made to predict data rate versus altitude for a pole-sitter application as shown in Fig. 8. The assumptions include center frequency 300 THz, near-infrared wavelength 1μm, transmit output power of 5 W, detector efficiency of at least 0.19, minimum pointing error loss of 0.57, and 3-dB design margin. This design results in a minimum of 10 Gbps data rate at an altitude of 2.5 MKm. Optical communication data rates can be improved with precise terminal pointing, higher detector efficiency. Weight savings can be achieved by reducing the telescope aperture. More work is recommended to optimize optical communication size, weight, power, and cost (SWaP-C) versus data rate requirement for the pole-sitter application.

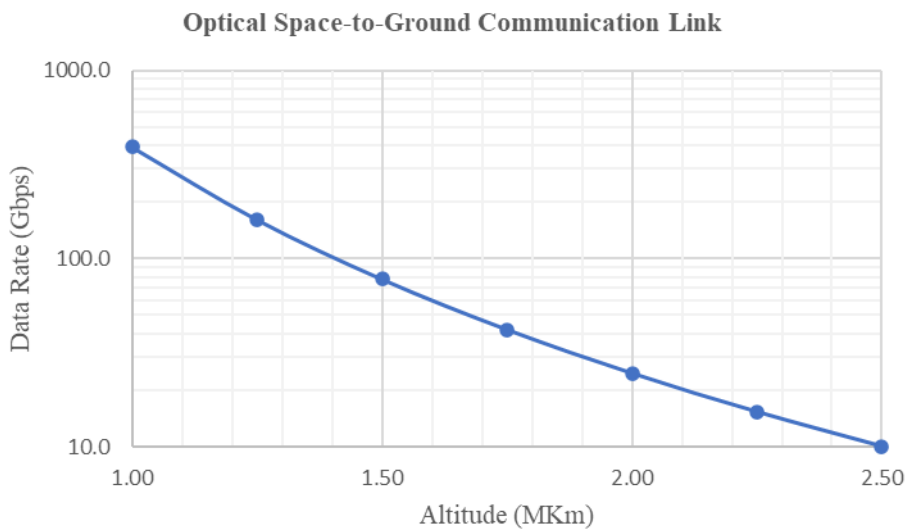

Fig. 8. Optical Space-to-Ground Communication Data Rate as a Function of Altitude

It is likely that pole-sitter RF communications will be necessary for tracking, telemetry, and commanding (TT&C) with presumed low data rate. RF technologies are mature and will be a straightforward design activity to optimize SWaP-C versus data rate requirement. Therefore, it is reasonable to assume that pole-sitter spacecraft may use dual RF and optical communications as shown in Fig. 9.

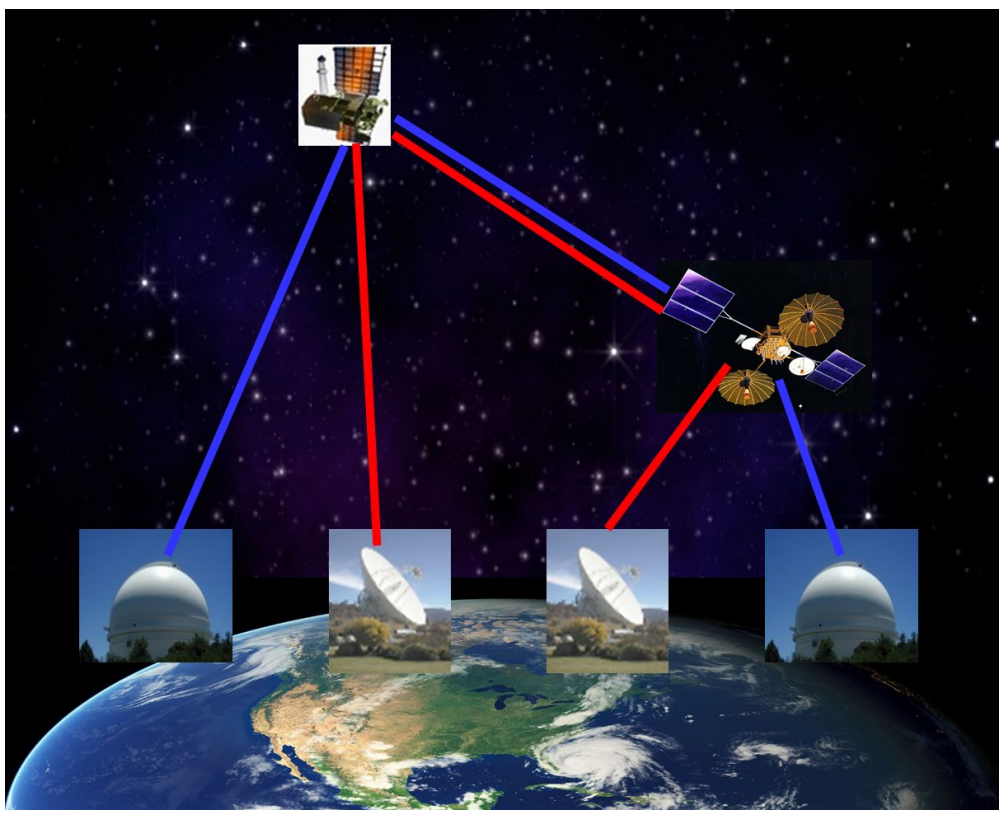

Fig. 9. Conceptual Communication Link

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A minimum data rate required for downlink is further studied. Assume the pole-sitter SDA sensor is required to track 5-10 objects simultaneously. The telescope aims its FOV onto the expected RSOs for observation. The on-board processing removes all background stars that are more or less than 0.5 visible magnitude than the target. Assuming an integration time of 1 minute the information sent down to the ground is the target intensity and position plus a few reference stars. This process continues once per minute going from target to target in a 10-minute cycle. Intensity changes will indicate changes in target configuration or orientation – motion changes will indicate changes in its trajectory due to maneuvers. Each such one-minute snapshot will be of the order of magnitude of 1 Kb – so the total bit rate of 10 bps will be more than enough. Adding some housekeeping data, a 100-bps downlink would be plenty. This allows a small RF or optical communication to do the job.

6. PROPOSED IN-SPACE DEVELOPMENTAL DEMONSTRATION

A flight demonstration utilizing a 200 Kg grade SmallSat to be launched to a pole-sitter altitude of 2.5 MKm is proposed to show its feasibility and gain experience in such orbit. The deployment at 2.5 MKm would provide for a few years of orbital experience to collect object and background data of a prototype sensor, quantify the flight dynamics (orbit determination and control), and provide a baseline for comparison with other high altitude deployment options. For cost and schedule savings, this pole-sitter satellite is proposed to be jointly developed with the authors' another mini-Advanced Space-Based Testbed (mini-XST) technology demonstration [24] at USSF to share as many components and technologies as possible.

In this flight demonstration, the in-space test of the focal plane array technology follows the author's original strategy for developing the pole-sitter's three major technical risk areas shown in Fig. 10 [5]. It will help address two of the three interim risk reduction items. The currently proposed space demonstration will cover the "FPA" risk and the "SEP" risk in the original plan in [5]. This demonstration will fly at a pole-sitter altitude instead of LEO in the original plan to test prototype FPA and SEP with a smaller telescope viewing through SEP plume. The second test points would be taken for disturbance, thermal performance of sensors, relative position station keeping and allocation of FOV or overlay of FOV sensor data at a pole-sitter orbit.

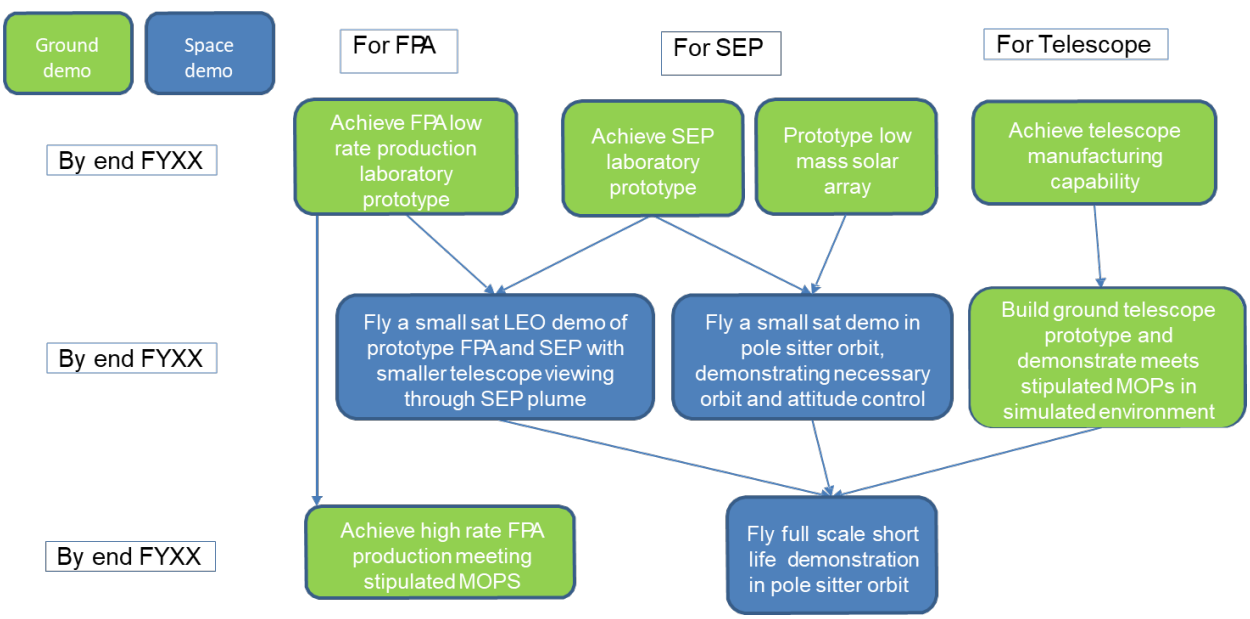

Fig. 10. Demonstration Strategy for Polesitter Risk Reduction [5]

The tentative schedule for this proposed flight demonstration is as in Table 2 in terms of Authority to Proceed (ATP).

Table 2 Tentative Schedule

Event	Milestone Year
Milestone A (MSA) – System Assessment & Spacecraft Development	ATP + 3 yrs
Initial Operational Capability (IOC) – Launch & Transfer Orbit	ATP + 3.5 yrs
Full Operational Capability (FOC) – In-Space Test	ATP + 4 yrs
End-of-Life (EOL)	ATP + 7 yrs ATP + 10 yrs if refueled

7. CONCLUSION

This paper provides a chain of thought and underlying data to illustrate that there is a key effort - persistent SDA for cislunar region - the nation can use to deter adversaries through information dominance. Industry indicates they can produce a pole-sitter system at an affordable price and within the current planning horizon [6]. To expedite the realization of this pole-sitter SDA, a flight demonstration using SmallSat is proposed and essential trades to support the mission were performed.

Additional work will include continuing the sensor design trade space optimization including visible versus IR wavelength technologies. The sensor design will need additional trades with the optical system design, including how the diameter of the primary optic affects resolution and sensitivity. Additional trades studies include sensor frame rates versus sensor/detector integration times, the trades for FPA size, stitching of arrays into larger format sensors, and optimizing sensitivity versus resolution will be required. Additionally, using SEP or other propulsion options may incur additional design considerations such as jitter to the pointing control and contamination to the optical sensor. The near-term plan also includes conceptual spacecraft and payload designs.

While the proposed flight demonstration would be on a 200 Kg or smaller spacecraft, the goal of future work is extended to full scale spacecraft over 1000 Kg with more sophisticated optical sensors and long-life propulsion system. Another alternative to prolong the investment on the persistent SDA satellite for say 7 years or longer, the refueling capability [25] developed under the In-Space Servicing, Assembly, and Manufacturing (ISAM) National Strategy [26] has to be taken into account.

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