Selenological Survey Using Satellite-Based Ground Penetrating Radars

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Executive Summary

The performance of a hypothetical space-based ground penetrating radar in lunar orbit is explored. The system is modeled after an actual space-based radar mission. Five uncertainties are identified and compared with a sensitivity analysis which reveal that the radar target reflectivity and scientific discount rate are the most impactful uncertainties on the mission's value. A series of Monte Carlo simulations reveal the baseline mission has a 29% chance of failing to yield scientific value due to altitude constraints. Real technical options are explored in order to reduce this risk; this risk can be reduced by adding an additional transmitter operating in a different power and/or wavelength. Real options are also explored for managing the mission's lifetime: a satellite with a five-year design life and the option to deploy a second satellite both out-performs and potentially costs less than a single satellite with a ten-year design life. This analysis highlights several lessons learned regarding the design of real options, to include the use of minimum and maximum functions as well as the role of learning.

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Introduction

The purpose of this analysis is to demonstrate how real options can be used to improve the design of a lunar remote sensing mission under uncertain conditions.

Motivation

As the Earth's closest natural celestial body, the Moon has been subject to scientific study for millennia; however, significant fundamental questions remain regarding the Moon to include its origin and composition. A better understanding of the Moon's physical features—referred to as Selenography from the Greek word Σελήνη (Selene, meaning Moon)—can support a number of applications to include lunar habitation, a lunar economy, and human exploration beyond Earth's orbit. Although the surface of the Moon has been surveyed by a number of different missions, very little is known about what lies beneath the Moon's surface in the lunar regolith. Several missions to the lunar surface have provided some knowledge of the regolith but these missions have only touched a very small fraction of the lunar surface.

Ground penetrating radar is an emerging technology that is currently used to understand what lies beneath the Earth's surface. Although ground penetrating radars have not been deployed in space to date, a study in 2005 outlined the technical feasibility for an Earth-orbiting ground penetrating radar (Marco D'Errico, 2005). Similar technology deployed in orbit around the Moon can provide scientists with a better understanding of the Moon's regolith. This paper explores a potential implementation of a space-based ground penetrating radar orbiting the Moon.

Scenario

The analysis presented here will suppose that the US National Aeronautics and Space Administration (NASA) and the Indian Space Research Organisation (ISRO) have decided to copy the upcoming NASA-ISRO Synthetic Aperture Radar (NISAR) mission and deploy the copy in lunar orbit as a ground penetrating radar. "The NASA-ISRO Synthetic Aperture Radar (SAR), or NISAR mission, is a multidisciplinary radar mission to make integrated measurements to understand the causes and consequences of land surface changes" by measuring changes in topography (National Aeronautics and Space Administration, 2019). Assuming the success of the NISAR mission on Earth, this analysis supposes NASA and ISRO will capitalize on the technology by deploying a similar satellite into lunar orbit as a ground-penetrating radar.

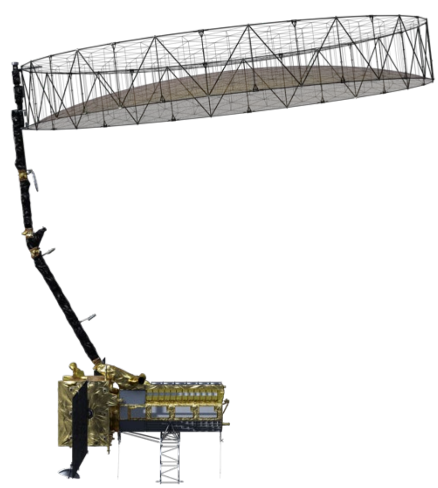

Figure 1: NISAR Artistic Rendering

System Design

The system in question is a constellation of one or more satellites in lunar orbit with the objective of searching for and discovering selenological areas of interest. Each satellite is equipped with a radar payload designed to penetrate the lunar regolith from orbit. The performance of the radar payload, measured by the signal-to-noise ratio S/N, is governed by the monostatic radar equation

$$S/N = \frac{P_t G^2 \lambda^2 \sigma}{(4\pi)^3 R^4 k T_s B_n L}$$

Equation 1: Radar Signal-To-Noise Ratio

where P_t is the power of the transmitter, G is the gain of the radar's antenna, λ is the wavelength, σ is the radar cross section, R is the range from the transmitter to the target, k is Boltzman's constant, T_s is the system noise temperature, B_n is the noise bandwidth of the receiver, and L are the losses of the radio frequency components (Massachusetts Institute of Technology Lincoln Laboratory). The antenna gain is modeled with the equation

$$G = \frac{4\pi\eta A}{\lambda^2}$$

Equation 2: Gain Formula

where η is the antenna efficiency and A is the area of the reflector (Salam, 2014). The following values are used for this analysis:

- 5 kilowatts for the transmit power P_t. This is an assumed improvement over the actual NISAR's S-Band power of 3.96 kilowatts (Misra, Bhan, Putrevu, Mehrotra, & Chakrabarty, 2016)
- 8 dB of losses L in the radar. This is an assumed improvement over the 15 dB of assumed losses with NISAR's S-Band system (Misra, Bhan, Putrevu, Mehrotra, & Chakrabarty, 2016)
- C-Band transmitter (6 GHz) instead of S-Band and L-Band, which defines λ. C-Band was chosen in order to increase signal to noise ratio and improve resolution at the expense of the field of view.
- The bandwidth B_n of the receiver is assumed to be 10% of the center frequency of the radar. This is larger NISAR's bandwidth of 10-75 MHz to account for uncertainty in radar returns (Misra, Bhan, Putrevu, Mehrotra, & Chakrabarty, 2016).
- The reflector's diameter d is 15 meters, which defines $A = \pi\left(\frac{1}{2}d\right)^2$ in Equation 2. This is an assumed improvement over the NISAR's 12 meter reflector (Misra, Bhan, Putrevu, Mehrotra, & Chakrabarty, 2016).
- The aperture's efficiency η is 50%. No data for NISAR's aperture efficiency could be found so a standard figure for antennas is used.
- The duty cycle of the radar τ is assumed to be 20%. This is assessed as reasonable given the uncertainties in the thermal environment surrounding the Moon.
- The required signal to noise ratio for radar returns is 10 dB. This is higher than the NISAR mission because of the ground penetration aspect covered in this analysis. 10 dB

was chosen based on previous research of space-based ground penetrating radars (Marco D'Errico, 2005).

Given that the radar payload is a synthetic aperture radar, the radar must be pointed to the side of the direction of travel to generate the motion required for the synthetic aperture.

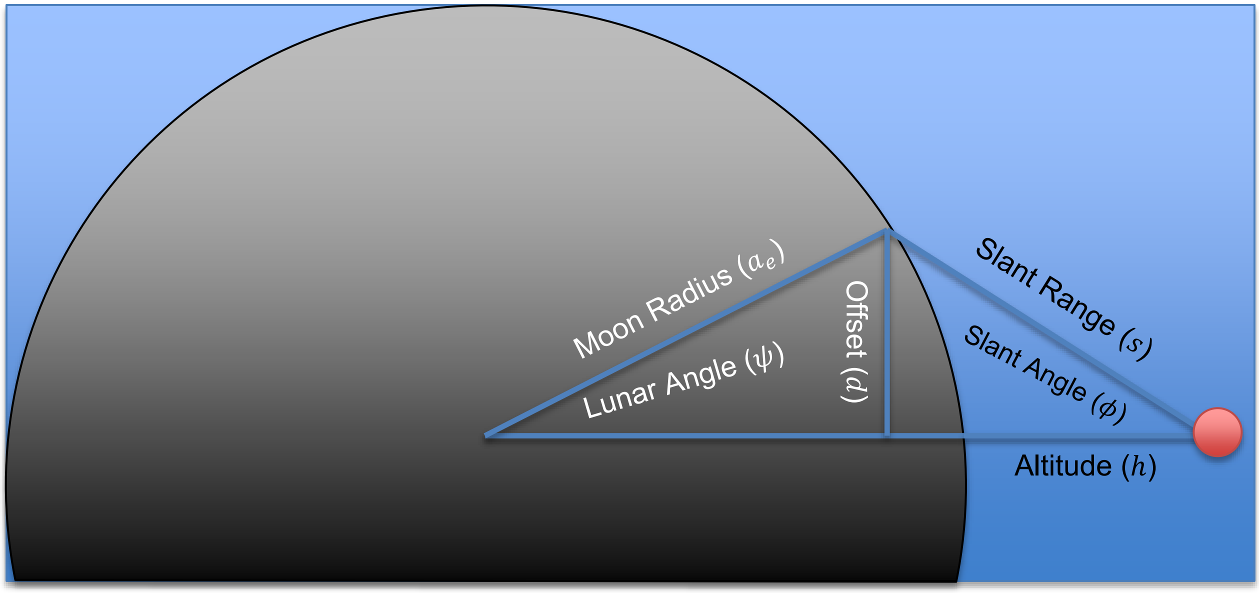

Figure 2: Diagram of SAR Geometry

The required offset p can be calculated through trigonometry using the slant range s and the slant angle ϕ:

$$p = s * \sin(\phi)$$

The lunar angle ψ is then derived from the offset using the is the Moon's volumetric mean radius (1737.4 kilometers as reported by NASA), written as a_e:

$$\psi = \arcsin\left(\frac{p}{a_e}\right)$$

The altitude h of the spacecraft can then be derived from the lunar angle:

$$h = (s * \cos(\phi)) - (a_e * (1 - \cos(\psi))$$

The slant angle for this analysis is 30 degrees, 7 degrees shallower than NISAR's 37-degree slant angle (Misra, Bhan, Putrevu, Mehrotra, & Chakrabarty, 2016).

System Utility

Given that the mission is scientific in nature, it does not generate value in the form of financial gains; instead, the value of the mission manifests in scientific observations. The scientific value of the mission will depend on how much of the Moon is able to be surveyed: the more area surveyed, the more value the mission has. To determine the amount of area surveyed in a specific time period, this analysis will assume the radar is scanning the field of view. Under this assumption, the scan rate is a function of the beamwidth θ of the radar and the orbital velocity of the spacecraft:

$$\frac{dA_n}{dt} \approx 2s\tan(\theta) * v$$
Equation 3: Scan Rate

where A_n is the area surveyed during period n and v is the velocity of the spacecraft parallel to the surface. The equation is an approximation because it assumes the lunar surface in view of the sensor is flat and does not account for the curvature of the Moon. Given the relatively small beam width of a radar sensor and the relatively large radius of the Moon, this assumption is assessed as reasonable. The beamwidth of the radar is governed by the wavelength and antenna diameter:

$$\theta = \frac{\lambda}{d}$$
Equation 4: Beamwidth

For the purposes of this analysis, the spacecraft is assumed to follow a circular Keplerian orbit around the Moon such that the velocity of the spacecraft is

$$v = \sqrt{\frac{\mu}{a_e + h}}$$
Equation 5: Orbital Velocity

where μ is the standard gravitational parameter of the moon (4904.8695 million cubic meters over square seconds).

Given that Equation 5 does not vary with time, Equation 3 can be integrated over time to calculate the area surveyed within a given time t and sensor duty cycle:

$$A_n \approx 2s\tan(\theta) * v * t * \tau$$
Equation 6: Survey Area

Although the value the mission is proportional to the amount of area surveyed, not all surveys are equally valuable: surveys that occur earlier in the mission are more valuable than surveys that occur later because early results can be exploited sooner. To model the time-dependent value of survey area, a discount rate will be used. The discount formula used will be

$$V_n = A_n(1-r)^n$$
Equation 7: Discounted Scientific Value

where V_n is the value of the area survived during period n, and r is the discount rate.

System Constraints

The key system constraint in this model is the orbital altitude. Altitude affects both the signal to noise ration and the radar's field of view: a decrease in altitude will increase the signal to noise ratio but decrease the radar's footprint on the lunar surface. The optimal altitude is the highest altitude that provides the required signal to noise ratio. This analysis will assume the spacecraft is inserted in to the optimal altitude. Given the relatively small range of possible altitudes, maneuvering to the required altitude will consume an inconsequential amount of fuel as compared to the station keeping fuel requirements.

Since the moon does not have a “sea” to reference for altitude (i.e. sea level), altitude is measured from the moon’s mean radius. Some of the moon’s features rise over ten kilometers from the mean radius, as depicted below (NASA/Goddard Space Flight Center/DLR/ASU, 2024).

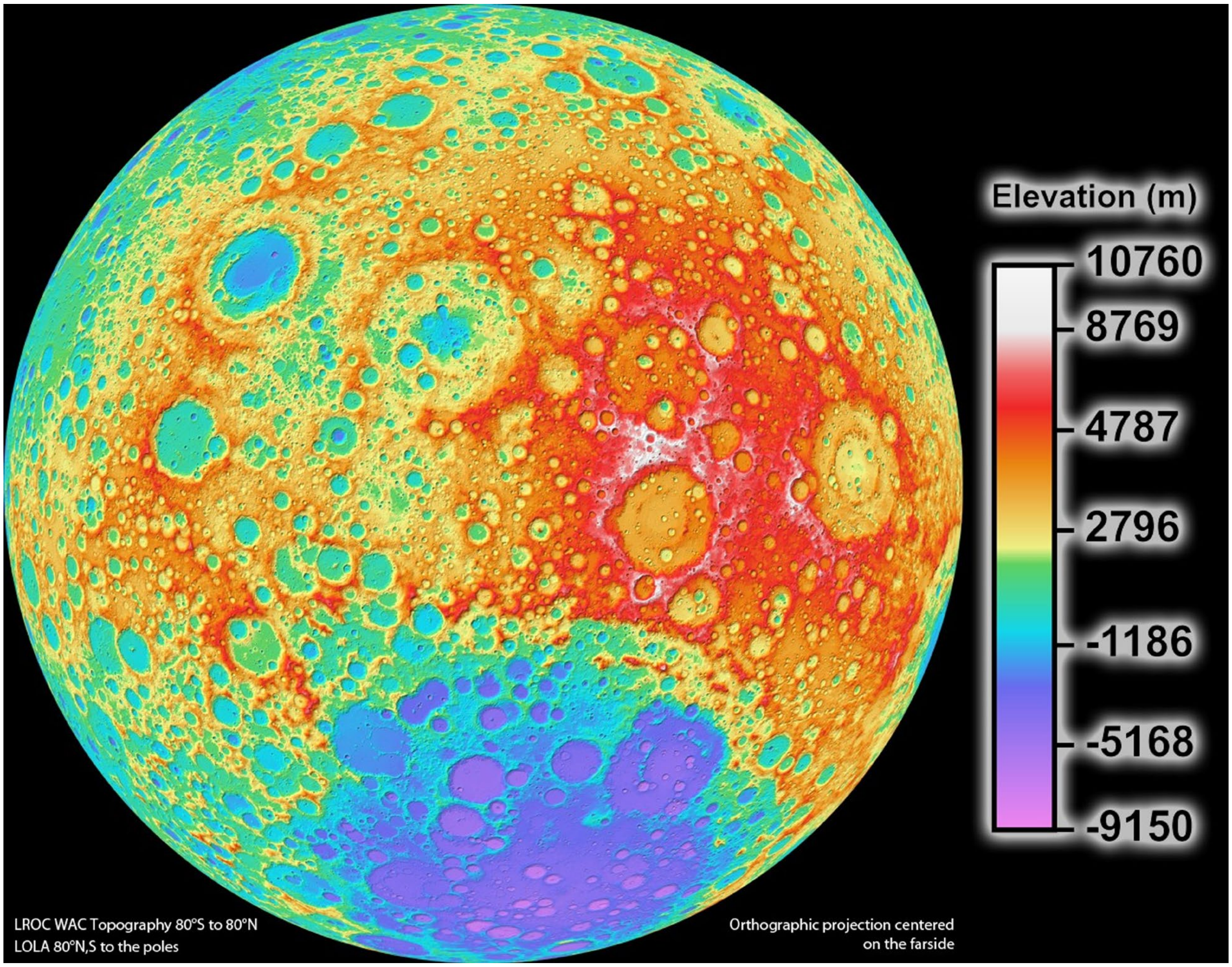

Figure 3: Lunar Elevation Map

Since the purpose of this mission is to look under the surface of the moon, the model of the satellite’s radar assumes the radar is targeting the moon’s mean radius. However, the spacecraft should not operate below the moon’s highest peaks for flight safety reasons. Therefore, 15 kilometers will be the “hard deck” which the spacecraft cannot operate safely under. **For the purposes of this analysis, if the spacecraft has insufficient signal to noise ratio at 15 kilometers then the mission is considered a failure and produces no scientific value.**

Uncertainties

A significant number of uncertainties exist in the scenario. This analysis identifies five uncertainties and examines their effect on the outcome of the supposed mission.

Radar Performance

The terms of Equation 1 can be grouped into four categories:

- Terms that are mathematical constants: 4π, k
- Terms that can be calculated: R
- Terms that can be measured: P_t, G, λ, B_n, L
- Terms that must be assumed in this context: σ, T_s

Although the five terms that can be measured (P_t, G, λ, B_n, and L) will be uncertain until the hardware is built, they are assumed to be "within specification" and will not be treated as random variables for the purposes of this analysis. Radar cross section and system noise temperature are assumed to be time-invariant random variables for the purposes of this analysis.

Radar Reflectivity

In traditional radar applications, the radar cross section is a measurement of the energy that is reflected from the target directly back to the radar; any returns that bounce off of another surface are considered multi-path noise and are discarded. For ground penetrating radars, most of the radar energy is expected to be reflected by the surface and only a subset of energy will actually penetrate the surface. The actual reflectivity of lunar subsurface targets is not known.

σ will be treated as a random variable uniformly distributed between -10dB and -30dB. *Note that this is a uniform distribution on a logarithmic scale*.

System Noise Temperature

The thermal environment in Low Earth Orbit (LEO) is well-understood given the number of missions flown in LEO for decades. The thermal environment of a low lunar orbit is not as well understood, which will affect the radar's system temperature.

T_s will be treated as a time-invariant random variable normally distributed around 950°K with a standard deviation of 100°K.

Scientific Value Discount Rate

As discussed in the System Utility section, a discount rate is used to account for the fact that surveys conducted earlier in the mission provide more value than surveys conducted later in the mission. A small discount rate implies that future surveys of the moon will still have significant value whereas a large discount rate implies that future surveys will be of little additional value. The discount rate is not known a-priori and will therefore be treated as a random variable.

> The scientific discount rate will be a time-invariant random variable uniformly distributed between 0.05 and 0.25.

Risk of Spacecraft Loss

Space is a harsh environment and lunar orbit is no exception. With no magnetosphere to protect the spacecraft, the spacecraft is subject to electromagnetic storms and solar flares which can permanently damage spacecraft components and result in the loss of mission capability. The exact risk is difficult to quantify.

Let P_n^a be the probability that the spacecraft is alive after n periods. P_n^a can be calculated using the equation

$$P_n^a = (1 - l)^n$$

where l is the probability of the spacecraft failing in any given period. If the probability of the spacecraft being alive after n periods is known, the risk of failure in any given period can be written as:

$$l = 1 - e^{\frac{\ln P_n^a}{n}}$$

According to a 2019 report by The Aerospace Corporation, "~87% of U.S. military and civil satellites and ~75% of commercial satellites met or exceeded their design life" (Ferrone).

> The probability that a spacecraft is meeting mission requirements after n periods will be modeled as a binomial distribution. The probability of loss in a given period will be derived from the design life assuming there is an 80% chance of the spacecraft meeting design life d:
>
> $$l = 1 - e^{\frac{\ln P_n^a}{n}}$$
>
> The choice of 80% for P_d^a is a round arbitrary average of the U.S. military, civil, and commercial satellites derived from The Aerospace Corporation report (Ferrone, 2019).

Fuel Requirements

Most spacecraft require thrusters for a number of purposes to include navigating gravitational fields, adjusting orbits perturbed by external forces, and desaturating angular momentum. Refueling a lunar survey mission is not considered in this analysis; therefore, fuel can become a life-limiting factor of the mission. The actual fuel requirement will depend on a number of factors, to include the actual insertion orbit, the perturbations in the orbit, and thruster efficiency.

For this analysis, the mission will launch with a certain maneuverability capability measured in meters per second.

The spacecraft's fuel requirements, measured by the required change in velocity per period will be a random variable normally distributed around 100.0 meters per second with a standard deviation of 20.0 meters per second.

Baseline Analysis

In the baseline case, the satellite operates an altitude of 21.4 kilometers above the mean equatorial radius of the moon which provides the required signal to noise ratio of 10 dB. In the baseline case, all uncertain values were set to their expected value with the exception of the risk of spacecraft loss: since the risk of spacecraft loss models discrete events and the baseline case does not, the baseline cased uses an operational availability figure which is derived from the cumulative distribution function of a binomial distribution. The period's scientific value is then scaled by the operational availability within the period.

Using this approach, the operational availability of the spacecraft decreases linearly to 80% over five years and then drops to 0% after year six when the spacecraft's fuel reserves are depleted. Under these conditions, the spacecraft surveys 6.0 million square kilometers (approximately 16% of the Moon's 37.9 million square kilometers of surface area). **The discounted value of the mission is 4.2 million square kilometers of value.**

Sensitivity

In order to understand the implications of uncertainty in the analysis, a tornado diagram was derived from the following ranges:

Table 1: Sensitivity Analysis

Parameter	Expected	High	NPV at High	Low	NPV at Low
Reflectivity	-20 dB	-10 dB	7.5M km^2	-30 dB	0.0M km^2
System Noise Temperature	950°K	1050°K	4.1M km^2	850°K	4.3M km^2
Scientific Value Discount Rate	15% per year	25% per year	3.4M km^2	5% per year	5.3M km^2
Risk of Loss	20% failure rate at 5 years	30% failure rate at 5 years	3.9M km^2	10% failure rate at 5 years	4.5M km^2
Fuel Requirement	100 m/s per year	120 m/s per year	3.8M km^2	80 m/s per year	4.7M km^2

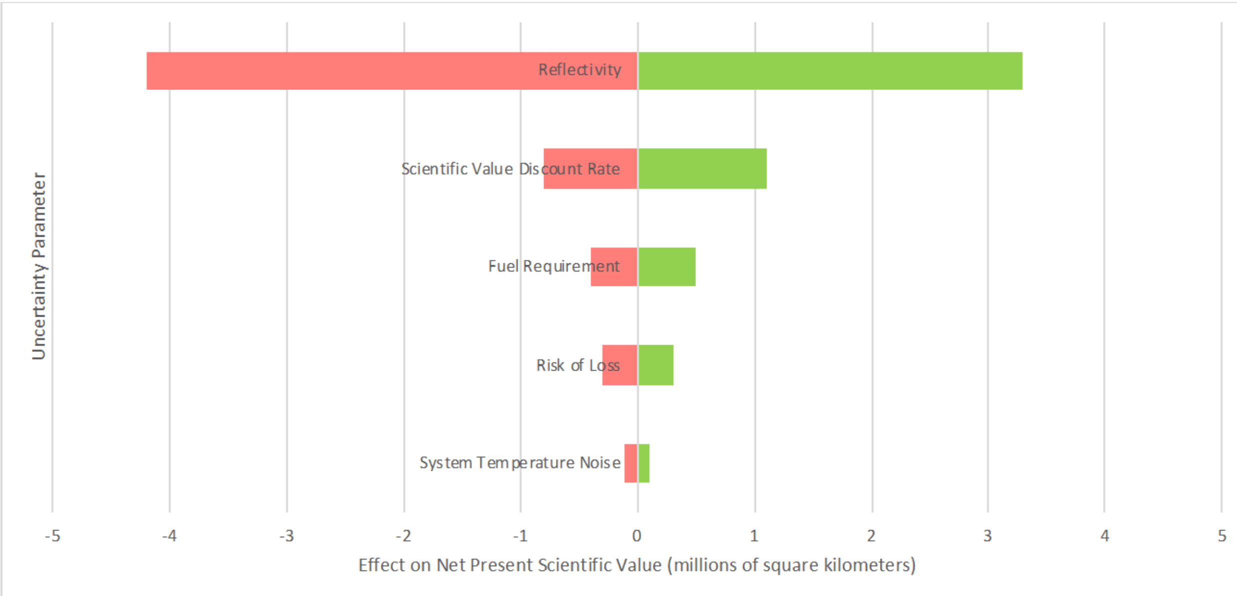

Figure 4: Sensitivity Analysis Results

The sensitivity analysis reals that the radar reflectivity is the most influential uncertainty in system performance, followed by the scientific discount rate, fuel requirements, risk of loss, and system temperature noise, respectively. Reflectivity's large impact comes from its influence on the orbital altitude, as discussed in the System Constraints section: in the best-case scenario, the satellite can operate significantly higher thus producing more value over time due to the larger footprint, whereas the worst-case scenario requires the satellite to operate at an orbital altitude below the minimum altitude allowed and therefore does not produce any value. It is also interesting to note that the radar reflectivity has more potential downside than upside but scientific value has more potential upside than downside.

Performance Under Uncertainty

In order to understand how the uncertainties in combination affect system performance, 2000 simulations of the mission were conducted. The cumulative distribution function, derived from the histogram of the simulations, is shown below.

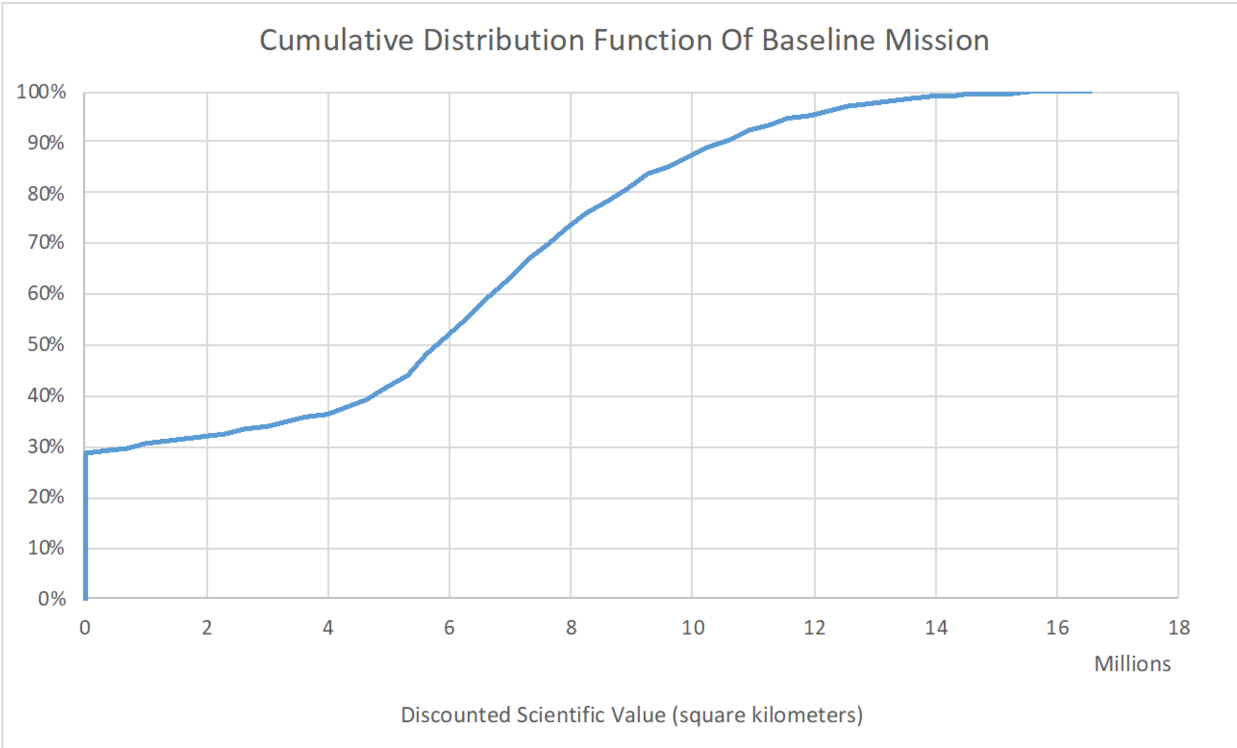

Figure 5: Approximate Cumulative Distribution Function of Scientific Value

In the Monte Carlo simulation, the average scientific value was 4.6 million square kilometers, 0.4 million square kilometers more than the baseline case. The difference can be explained by the asymmetry of the scientific value uncertainty, which has more potential upside than downside, as well as the discontinuity involved with the altitude constraint.

The cumulative distribution function of the discounted scientific value has a conspicuously sharp increase at the beginning of the target curve; this steep increase is associated with the altitude constraint. In the simulations, approximately 29% of the simulated missions failed to derive scientific value because the signal to noise ratio was still too low even at minimum altitude: stated another way, **the signal is too weak to produce scientific value in 29% of simulations** in the baseline case. This can be seen in the cumulative distribution function of the maximum operating altitude from the Monte Carlo simulation, shown below.

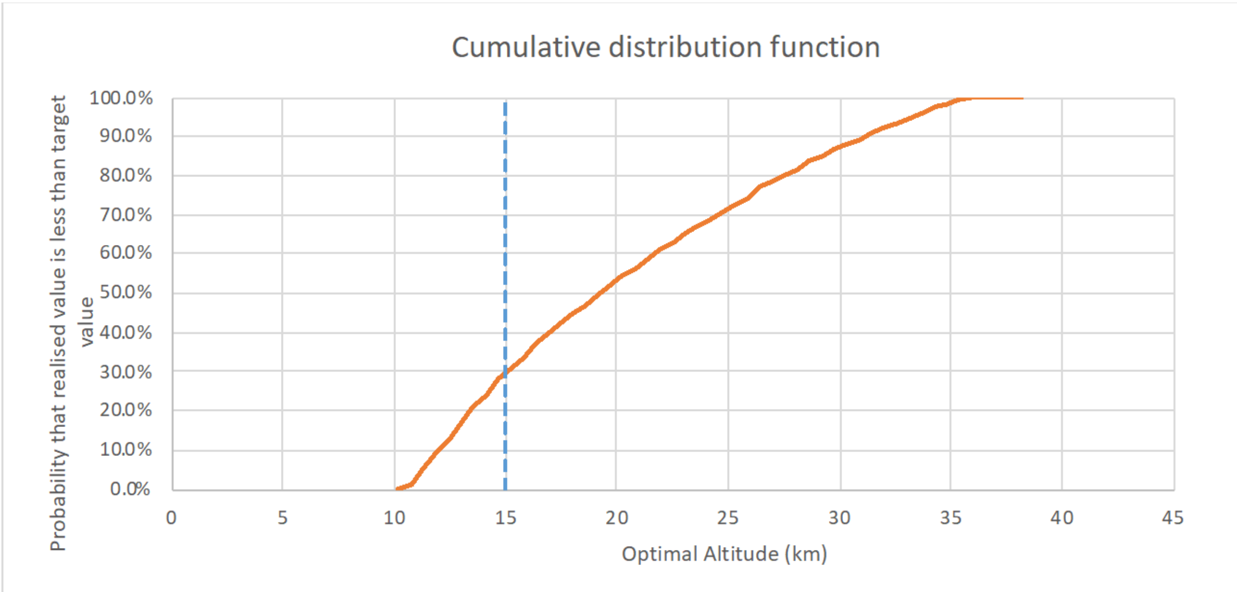

Figure 6: Maximum Operating Altitude Required By Radar

Based on these results, the three criteria used to evaluate different mission configurations will be the risk of mission failure (which will be the percentage of missions that yield no scientific value), the expected discounted scientific value, and the discounted scientific value 95th percentile P_{95}, which is a measure of maximum upside. The results of the baseline analysis are summarized in the following table.

Table 2: Baseline Analysis Results

Metric	Value
Average Value	4.6M km^2
Risk of Mission Failure	29%
P_{95}	10.7M km^2

Real Options

This analysis will now consider four real options to limit downside and increase upside:

- Operating in X-Band instead of or alongside C-Band
- Operating at a higher power level
- Extending the design life of the system
- Planning for a potential follow-on mission

The former two real options address technical uncertainties with the radar target reflectivity and system noise temperature whereas the latter two options address uncertainties in the scientific discount rate, fuel requirements, and risk of failure.

Real Options for Technical Uncertainties

As shown in the sensitivity analysis, radar performance can have the largest impact on the scientific value of the mission to the point of mission failure if the radar target reflectivity is too low. The ability to change the altitude of the spacecraft to respond to uncertainty is an inherent real option but it cannot solely prevent scenarios where the mission fails (i.e. no scientific value is produced) because of the altitude constraint. Additional real options can limit potential downside in the event altitude reductions are not sufficient for generating the required signal to noise ratio.

Two technical configurations considered for limiting the downside of technical uncertainties are increasing the radar transmitter's power and operating at a higher frequency.

- Increasing the radar transmitters power results in a linear increase in the signal to noise ratio; however, the duty cycle of the radar must be reduced in order to prevent overheating.
- Increasing the radar's frequency increases the radar antenna's gain based on a lower wavelength for Equation 2 but it also decreases the radar's beamwidth, which in turn reduces the ground footprint based on Equation 3.

The configurations are summarized in the following table:

Table 3: Design Permutations

Configuration	**Transmitter Configuration**	**Power Configuration**
Baseline	C-Band	5 kW, 20% duty cycle
High Power	C-Band	**10** kW, **10**% duty cycle
X-Band	**X-Band**	5 kW, 20% cycle
High Power X-Band	**X-Band**	**10** kW, **10**% duty cycle

The results of a *rigid* design are shown below: in the rigid design, the parameters cannot be changed on orbit.

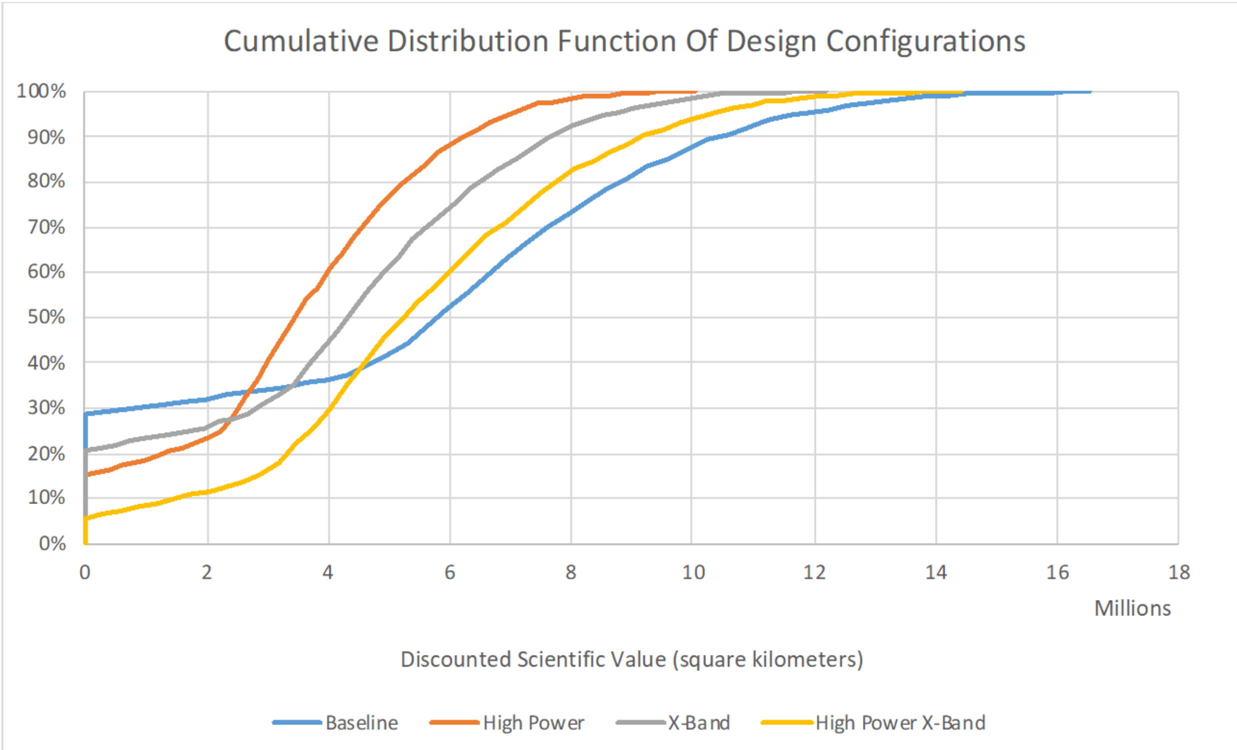

Figure 7: Performance of Rigid Technical Designs

The results captured in Figure 7 show that operating in C-Band at a higher power and a corresponding lower duty cycle reduces the risk of mission failure to approximately 15% but also reduces potential upside by 60% as compared to the baseline configuration. Operating in X-Band at the baseline power level reduces the risk of mission failure to 20% but reduces the potential upside by 25%. Operating in X-Band at a higher power and a corresponding lower duty cycle further reduces the risk of mission failure to 5% but reduces the potential upside by 15%. The high-power X-band approach dominates the high-power C-Band and the baseline-power X-band approaches but none of the three different configuration dominates the baseline configuration: the baseline configuration has the highest risk of mission failure but will yield better results than any of the other configurations approximately 65% of the time. These results are summarized in the following table.

Table 4: Rigid Technical Configuration Results

Configuration	**Risk of Mission Failure**	**Probability of Out-Performing Baseline**	**Maximum Upside**
Baseline	29% (4th)	-	~16M km^2 (1st)
High Power	15% (2nd)	32% (3rd)	~10M km^2 (4th)
X-Band	20% (3rd)	35% (2nd)	~12M km^2 (3rd)
High-Power X-Band	5% (1st)	38% (1st)	~14M km^2 (2nd)

The analysis discussed above shows the trade space of a rigid design—the design cannot be changed once deployed. The advantage of real options is best demonstrated when those options can respond to conditions as those conditions reveal themselves. Consider the following technical baseline:

Table 5: Technical Real Options

Configuration	Transmitter Configuration	Power Configuration
Baseline	C-Band	5 kW, 20% duty cycle
Selectable Power	C-Band	5 kW, 20% duty cycle **or** 10 kW, 10% duty cycle
Dual-Band	C-Band **or** X-Band	5 kW, 20% cycle
Selectable Power + Dual-Band	C-Band **or** X-Band	5 kW, 20% duty cycle **or** 10 kW, 10% duty cycle

In the following analysis, the system chooses the baseline configuration or the option configuration based on which will produce the most scientific value. In most cases, the baseline configuration is preferred but when the baseline configuration has an insufficient signal to noise ratio because of system constraints, the optional configuration is used.

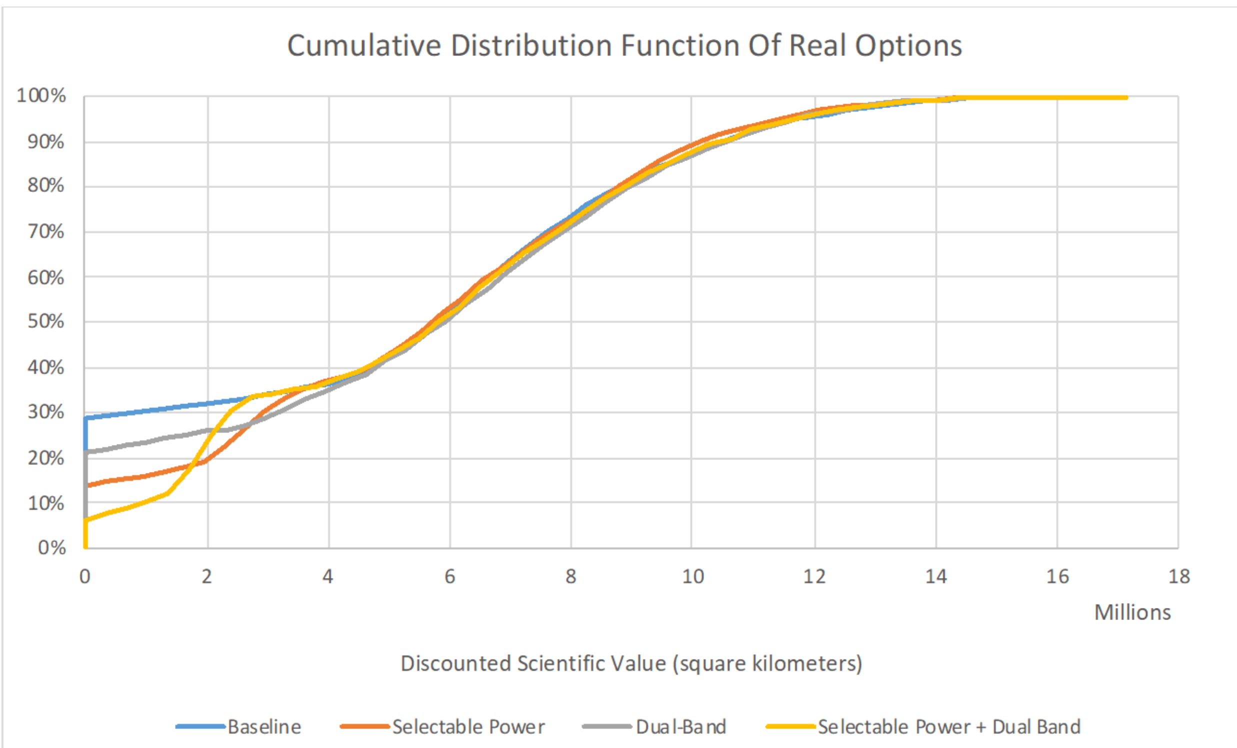

Figure 8: Performance of Real Technical Options

Figure 8 shows that adding real technical options does not change the expected scientific value (P_{50}) nor does it change the potential upside but they do limit downside to the same risks of mission failure documented in Table 4. All target generated using real options dominate the baseline target curve; however, no one target curve generated using real options dominates the others.

Based on these results and in the absence of cost and schedule data, the primary recommendation would be to install a high-power X-band transmitter alongside the baseline-power C-band transmitter; doing so will reduce the risk of mission failure to approximately 5% from 30% without affecting the expected performance or potential upside. However, if cost and schedule data for implementing the different options was available, it would be trivial to generate a cost-benefit analysis. Regardless of the cost and schedule data, real options in the form of increased power and/or higher transmit frequencies provide the flexibility needed to increase the chances of mission success without affecting the expected performance or potential upside.

It is important to note that these technical options are not far-fetched: the actual NISAR mission already uses two transmitters with different frequencies (L-Band and S-Band). The decision to fly two transmitters on the NISAR mission was driven by differing resolution requirements but nevertheless shows that the real options for the ground penetrating radar proposed here need not be unrealistic.

Real Options for Time-Based Parameters

Three key elements of the system model change as a function of time:

- The amount the scanned area is discounted
- The cumulative probability the spacecraft has had a catastrophic failure
- The fuel reserves

The previous section on Real Options for Technical Uncertainties showed that real options can limit downside but do not change the expected value or upside of the system. Although launching multiple satellites simultaneously could increase the mission's value it may not be the most cost-effective, especially in cases where the scientific discount rate is high. Instead, allowing for a longer mission duration can increase expected and upside value while providing cost-saving measures. There are two generic approaches for extending the mission:

- Increase the design life of the satellite
- Launch a second satellite at the end of the first satellite's design life

The former real option provides mission operators the opportunity to end operations before the end of the design life if the marginal benefit of the system drops below a certain threshold. The latter real option allows production costs to be deferred and potentially avoided if an extended mission is not warranted. This analysis will explore whether one real option for extending the mission is preferred over the other.

The following four scenarios are explored:

- Scenario 1: 5-year design life with the option for a follow-on mission but the option is not executed and the mission shuts down after year 5
- Scenario 2: 10-year design life and the mission is operated for 10 years
- Scenario 3: 5-year design life and the option for a follow-on mission is executed
- Scenario 4: 10-year design life but the mission shuts down after year 5

The scenarios are summarized in the following table:

Table 6: Longevity Scenarios

	Mission ends after year 5	**Mission ends after year 10**
5-year design life with option for follow-on satellite starting in year 6	Scenario 1	Scenario 2
Single satellite with a 10-year design life	Scenario 3	Scenario 4

The target curves for all four scenarios are shown in the following figure:

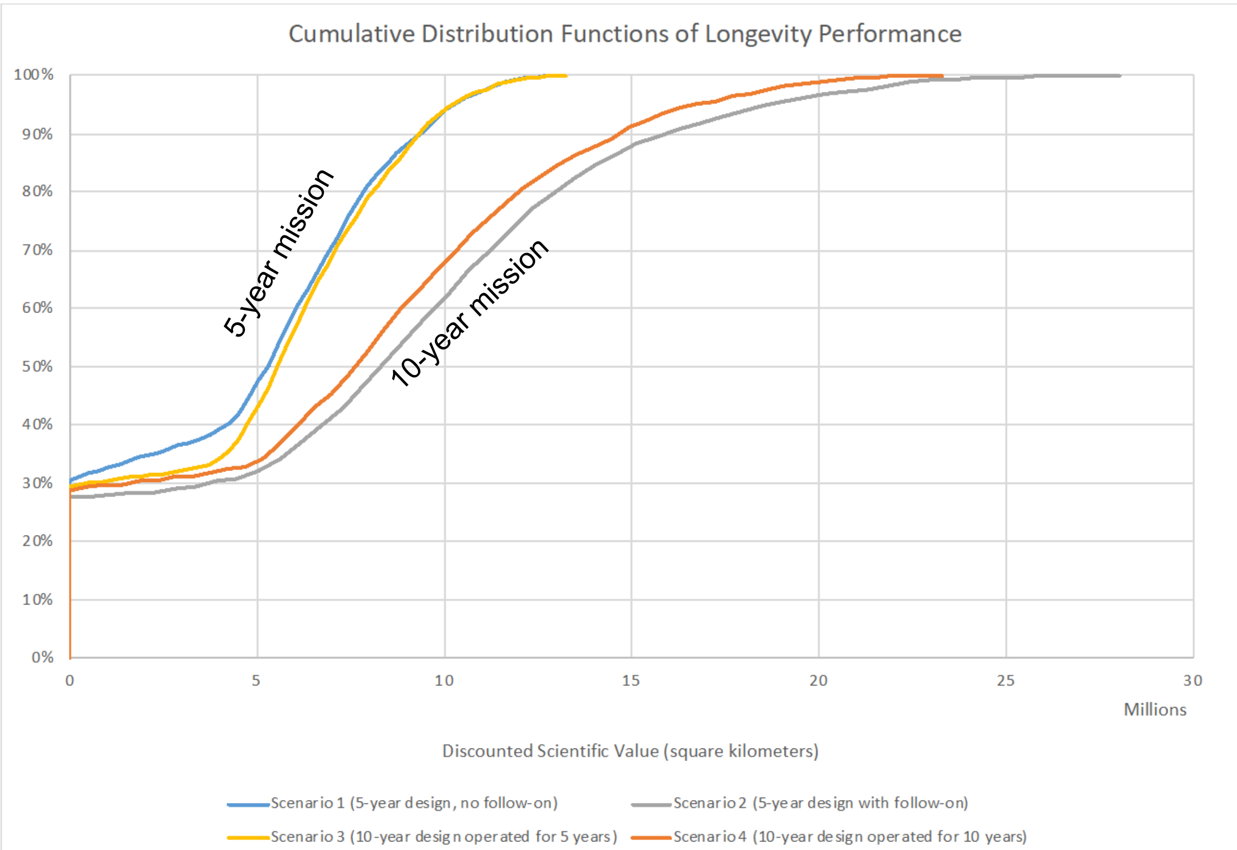

Figure 9: Longevity Scenario Results

Comparing the scenarios, a single satellite with a 10-year design life (Scenario 3) out-performs a single satellite with a 5-year design life (Scenario 1), as expected. Most importantly though, two satellites with a five-year design life launched five years apart (Scenario 2) out-perform a single satellite with a 10-year design life operated for 10 years (Scenario 4). The results are summarized in the following table:

Table 7: Longevity Results

Scenario	**Rank**	**Expected (Mean) Performance**	**Maximum Upside (P_{95})**	**Initial Cost**	**Deferred Cost**

1 *(5-year design, no follow-on)*	4th	5.2M km^2	~10.3M km^2	Lower (~)	None (+)
2 *(5-year design with follow-on)*	1st	8.4M km^2	~18.5M km^2	Lower (~)	Lower (~)
3 *(10-year design operated for 5 years)*	3rd	5.6M km^2	~10.6M km^2	Higher (-)	None (+)
4 *(10-year design operated for 10 years)*	2nd	7.5M km^2	~16.8M km^2	Higher (-)	None (+)

Grouping the results by the approach:

Table 8: Longevity Approach Results

Approach	5-year Performance	10-year Performance	Cost
5-year design life with option for follow-on	Mean of 5.2M km^2 P_{95} of ~10.3M km^2	Mean 8.4M km^2 P_{95} of ~18.5M km^2	Lower initial cost, lower or no deferred cost
10-year design life	Mean of 5.6M km^2 P_{95} of ~10.6M km^2	Mean of 7.5M km^2 P_{95} of ~16.8M km^2	Higher initial cost, no deferred cost

It would be trivial to generate a cost-benefit analysis using Figure 9 and Table 7 to inform an approach. In the absence of cost and schedule data, **opting for the five-year design life with the option for a follow is the preferred solution**. If the mission ends at the five-year mark, it was the lowest-cost approach and performs nearly as well as satellite with a 10-year design life during the first five years. If the mission is extended for the full 10 years, the approach successfully deferred the high cost five years, unlike the 10-year design life approach.

Conclusion

This analysis provides a framework for designing a space-based selenological radar system in lunar orbit accounting for uncertain conditions. Given real cost data, a cost-benefit analysis can be constructed trivially given the results. Even without cost data, the benefit of real options is evident through the change in expected scientific value, mission risk, and potential upside.

- Adding real options for transmitter frequency and/or power can reduce the risk of mission failure without affecting the expected mission performance or the potential mission upside
- Deploying a satellite with a five-year design life and the options for a follow-on mission is likely preferred to deploying a single satellite with a 10-year design life. The two-satellite approach reduces the impact of a catastrophic satellite event by providing a backup and also allows costs to be deferred or avoided as compared a more robust single satellite approach.

This analysis examined some decision rules for real options:

- Earlier iterations of the analysis examined whether it was worthwhile to expend fuel to move from the altitude identified in the Baseline Analysis section to the optimal altitude shown in Figure 6. The analysis revealed that approximately one week's worth of fuel needed to be used but doing so would result in drastic improvements in scientific value.

This results in a trivial real option and decision rule: expend some fuel (and potentially shorten the life of the spacecraft by approximately one week) in order to drastically improve scientific value.

- The decision rule for using a different transmitter frequency and/or higher power is simply a test on whether the different configuration will result in improved radar performance.
- There is a difference between design decisions and real options, as shown in the Real Options for Technical Uncertainties section. Design decisions are the trade space during system design that cannot be changed later, real options are components of the system that can respond to uncertainty as uncertainties reveal themselves.
- This analysis references a decision regarding the mission life but does not propose a decision rule. The decision rule for executing a follow-on mission will depend on a number of factors that are outside the scope of this analysis; however, this analysis shows that a follow-on mission does have the capability of increasing mission value more than a single satellite with an extended design life.

There are several takeaways from this analysis:

- Real options need not only manifest as “if” statements, they can also manifest in other ways including always selecting the minimum or maximum value to improve performance, as evidenced by the selection of the best operating frequency in the Real Options for Technical Uncertainties section.
- Real options are not limited to technically-challenging solutions. This analysis proposes using two transmitters as a real option; this real option is already implemented by the actual NISAR mission.
- Learning can be a way to capture “unintended options”. As operators become increasingly proficient in the system, they can discover new ways to operate the system that generates additional value. Since these are “known unknowns”, they can be modeled via learning. Draft versions of this analysis included a learning factor that was considered uncertain. The factor was ultimately removed because of complexity on how to model the increased value; however, learning can be considered a force counter-acting the scientific discount rate.

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