The Design of a Low Altitude Lunar Surface Mapping Satellite

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The Faculty of the Department of Aerospace Engineering
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Master of Science in Aerospace Engineering

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ABSTRACT

A satellite is designed to be placed in a low polar orbit of the Moon to photograph the surface in preparation for a series of manned operations near the lunar poles. Previous lunar missions are analyzed, and their specifications are functionalized to create predictive models for the required specifications for the planned satellite. Further estimates are made using mission requirements based on optical and orbital specifications derived from mission requirements. Specifications for other subsystems of the satellite are given and used to create budgets for various requirements to be referenced later in the design process. Further improvements and complications are described for the future of the project.

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Acknowledgements

Table of Contents

Table of Figures

List of Symbols

ρ – Angular Radius of the Scanning Swath

R_M – Radius of the Moon

H – Altitude of Final Orbit

SR – Spatial Resolution

P_T – Power Required

FOV – Field of View

M – Mass

V_g – Ground speed

ΔL – Node Shift

P – Orbital Period

ε – Minimum Elevation Angle

η – Nadir Angle

λ – Moon Central Angle

D – Distance to the Horizon

S – Longitudinal Degree Spacing

θ_r – Angular Resolution

X – Linear Spatial Resolution

Zc – Cross Track Pixels

Za – Swaths Recorded per Second

Z – Number of Pixels in Square

T_i – Period Required to Integrate Pixels

Q – Image Quality Factor

f – Focal Length

Δv - Change in Velocity

M_{dry} – Dry Mass of Satellite

M_P – Propellant Mass of Satellite

1. Introduction

1.1 Motivation

NASA is currently pursuing the Artemis mission, a plan to return humans to the moon this decade. The main focus of this mission is a permanent base on the surface of the moon, which will be preceded by manned landings on the lunar surface as soon as 2025 [1]. These landings and future operations could be aided by further understanding of the conditions of the lunar surface. The most recent project to map the surface of the moon was the Lunar Reconnaissance Orbiter, which photographed the lunar surface from a polar orbit, creating a map of the surface with 100-meter resolution [2]. This mission was launched in 2009, however, and the scope of our lunar missions has changed. NASA estimates that there is enough fuel in the spacecraft to last another 7 years. This is not long enough to cover the duration of the Artemis mission, meaning that there will be no way to analyze the surface of the moon in regions that Artemis may need to expand to in greater detail than we already have. Technology has improved and we may be interested in a more detailed analysis of a smaller section of the moon. This could be an opportunity to deploy a new surface mapping satellite to a similar orbit as the LRO that would allow it to use new camera technology to observe more specific regions of the moon useful for Artemis in greater detail and in more ways than the LRO.

1.2 Literature Review

The Lunar Reconnaissance Orbiter is the most similar mission performed by NASA to map the surface of the moon. It was initiated in 2004 and was launched in 2009. Its scientific mission was successful, and it was able to create a three-dimensional map of the moon using its onboard cameras while flying in a polar orbit over the course of a year. This orbit was highly elliptical to make the orbit easier to maintain and so that the spacecraft would pass over the entire lunar surface as the moon rotated beneath it [1]. Images taken by the LRO have helped to find landing locations for the Artemis mission, which will begin landing on the moon in 2025. This mission will continue for years afterward and will change in scope and target based on the findings of the first years of the mission. Because of this, Artemis will need to be flexible, as developments beyond base camp might be predicated on what the local topography and resource makeup of the lunar surface are [3]. Since the project will be using a similar mission architecture to the LRO, analyzing its mission plan in detail will be useful.

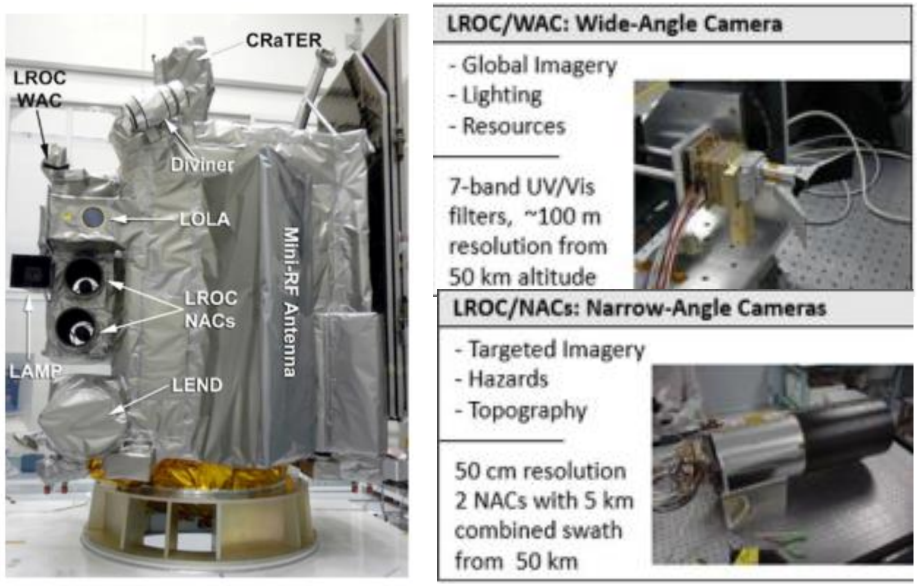


Figure 1.1 The LRO and its cameras during assembly. [1]

The LRO was designed for a one-year reconnaissance mission in a low polar orbit. It was launched to the moon using a simple direct transfer (with one correction burn during transit) and was then brought into its initial low polar orbit using a series of insertion burns. After these burns it entered its commissioning phase for 60 days, then was transferred to its final mission orbit as described previously. It has changed its orbit since then and remains operational and maneuvers to this day. In the plans for the LRO, the spacecraft would be put into the lunar holding orbit using three orbital maneuvers, allowing for multiple contingency options to ensure that the spacecraft will be captured into a proper lunar orbit. The orbit was tracked by an S-band tracking station. ¼ of its orbit would be tracked to predict its flight path and compare that to the mission plan. Several factors were also considered when tracking the prediction of the orbit, including gravity perturbations created by the shape and makeup of the moon. The factors were tracked at the Flight Dynamics Facility (FDF) at Goddard several times during the mission to predict precise future conditions for the mission [4].

The camera aboard this vessel will be of significant importance to how the mission will be designed and what limiting factors will be encountered during systems engineering. To get an impression of what these conditions will be, NASA should be studied for how it had to design their own LRO and more modern designs with components that may be able to overcome these limitations. One of the main mission objectives of the LRO was to “Acquire multi-temporal synoptic 100 m/pixel imaging of the poles during every orbit to unambiguously identify regions of permanent shadow and permanent or near-permanent illumination.” [5] This was so that NASA could reliably determine what areas of the Moon near the poles would be of interest to a future lunar mission, among other things. This is one of the factors that the Artemis mission had to consider, meaning that a more thorough analysis from a mission like the one being proposed could be useful [3]. The cameras used to achieve this mission objective were two narrow angle cameras (NACs) which would scan the lunar surface with a telescope that could cover 5 km of ground when in a 50 km orbit of the moon with a resolution of 0.5 m per pixel.

The second set of cameras were designed to produce a lower resolution image (100m per pixel) over a wider swath of land (up to 105 km). The camera was equipped with the ability to report a multispectral dataset, which would give scientists access to information about the mineral makeup of the lunar crust, another element of interest to the Artemis mission since rare materials like hydrogen and helium could be used to produce water and fuel for the mission. The WAC also provides scientists with information about the craters of the moon at a high incidence angle, allowing them to understand more about the craters' composition. This is important in scientific research on the age of the moon and other lunar characteristics related to crater composition. [5]

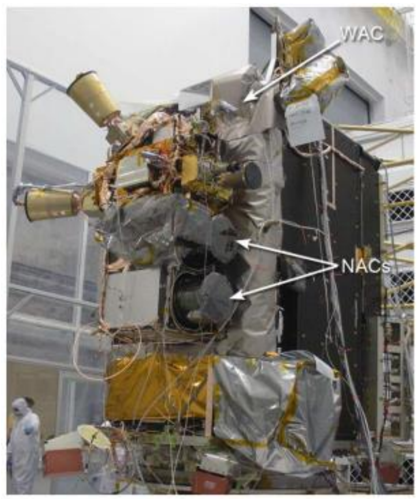


Figure 1.2 The constructed LRO with arrows showing where the NAC and WAC are located. [5]

The NACs are mounted to be perpendicular to the spacecraft's x-axis. Both are aimed and aligned so that their images slightly overlap, however they are oriented in a way that requires one of their images to be reoriented for them to match. This process is confirmed by the overlapping images. The WAC is oriented in a different direction so that it can take oblique images of the craters of the moon while the NAC takes more detailed photos of the lunar surface beneath the spacecraft.

The long-term condition of the LRO's orbit is important for our understanding of the spacecraft that will be designed. Since our mission will be interested in assisting the Artemis mission, it will be necessary to assume its mission will be longer than the one-year mission of the LRO, it will be useful to analyze the orbit of the original LRO mission to see what unexpected elements have shaped its orbit. The measurement of the orbit is maintained using the same techniques as before, combining an altimeter system with an S-Band array to observe its position relative to the moon with less precision [1]. After seven years, an analysis of the orbit was done to understand its properties as they have changed over time. The study found that the orbit has been stable due to its eccentric polar nature. The data found during this survey will be used to estimate the orbital characteristics of our own mission using data found in the Planetary Geodynamics Archive, which the scientists involved used to catalog their findings [6].

To model our own orbital characteristics, it will be useful to employ software. The most useful tool in this process will be the General Mission Analysis Tool (GMAT). This tool was developed by NASA between 2001 and 2013, when it was approved for public use. It has been used extensively since and includes a wide range of features, many of which will be useful for our analysis. These include tracking of important information, including launch window analysis, shadow predictions, and end of life modelling [7]. Information is processed as a combination of scripts and resources which are taken as input. These are turned into a navigable output. From this output the characteristics can be determined based on how they have been defined within the program to be tracked. The output is not flexible, however, and new executions of the software must be done to change any of our inputs after computation has already been done [8].

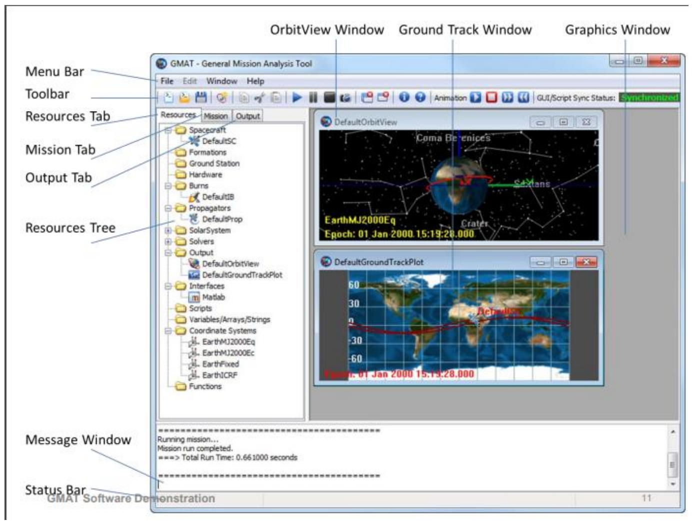


Figure 1.3 The GMAT software GUI with labels for different components. [8]

GMAT was thoroughly tested and refined before it was released, and those refinements have continued since its initial debut. The requirements for the software are incredibly strict, and the older features were removed as time went by as they did not meet the requirements of the agency. A study was done on the accuracy of GMAT, attempting to find any weaknesses in the software that could lead to any inaccuracies. These inaccuracies are of interest to our project since it will be necessary to be sure that the orbital characteristics of a highly precise and multiyear mission are well understood so that specific segments of the moon that are mapped are being categorized correctly. The study found that the results of the rigorous testing before the release of the 2013 version of the software were in excellent agreement with industry standards in areas such as modelling and numerical integration [9]. This makes GMAT the most viable candidate for the flight path analysis and planning phase of this project.

Other components of the system need to be understood in detail for our planning phase to be accurate and meet the mission requirements. One of the goals of the project is to create an outline for the layout of the BUS of the spacecraft and the connections between the components of the space craft with each other, and their connections to the ground. The LRO is the most relevant example available, but other missions have performed similar functions and their own BUS layouts and components that may be of interest to this project as well. The LRO will also be discussed for reference.

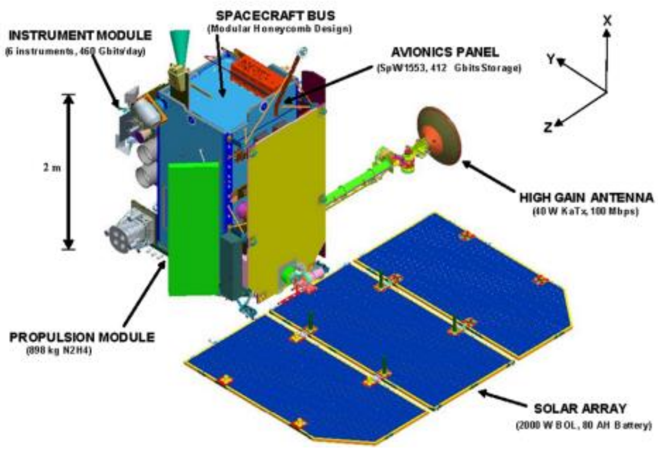


Figure 1.4 The LRO aligns different components along the axis of its flight, allowing the cameras to observe simultaneously while not obstructing the high gain antennae or the solar array.

The layout of the bus for the LRO indicates that the components are oriented to fit the mission objectives of the system. These priorities are to identify potential landing sites, resources and observe the lunar space radiation environment [10]. Since the project will take on some of these objectives as well, it will be useful to adopt the component configuration principles found in Figure 4.

The Cassini mission was intended to last many years. Our mission intends to do the same to facilitate the Artemis program. A study into the lifetime of spacecraft in regard to their design showed that increasing the lifespan of a spacecraft from 3 to 15 years could incur a cost in weight of up to 30-40% [11]. It is important therefore to analyze the techniques used by long term missions like Cassini to understand how it was able to achieve these goals. A basic analysis of Cassini shows that the bus was housed in a protective shell called the upper equipment module (UEM) This along with different support structures would protect fragile electronic components in the bus from environmental hazards, decreasing the need for specific protection of components due to the structure of the shell also serving as the anchor points for Cassini's communication system. The bus is also protected by a conductive cap that produces a Faraday cage [12].

A similar mission to our proposed project was flown by the Chinese in 2007. This Chang'E-1. This mission used a similar series of cameras as carried in the LRO. These were used to create a three-dimensional model of the moon's surface, however, a feature that the LRO mission could not achieve since its two cameras did not overlap enough to create stereo images. The altitudes of these images were measured by a laser altimeter in both, however. A microwave radiometer was also included to measure the thickness of the lunar soil beneath the craft [13].

This could be useful information for the Artemis mission as well and may be something to consider adding to the project to create a greater support system for the Artemis mission.

Finally, the communication system for the spacecraft will be an important component in both the planning of the spacecraft design and also in the structure of data transfer for the mission as a whole. Information will have to be relayed through a series of satellites and ground stations, each of which will have to interface with each other and with data from the spacecraft, meaning each connection becomes an integration problem that must be tracked [14].

To fully understand the structure of the system, it will be important to implement a basic form of model-based system engineering. Viewing the system as a model and keeping information organized along those principles rather than separately organized as unique and distinct documentation will be important for keeping track of the requirements of each of the component parts of this project, since design cost and flexibility are low for a two-semester project. The empirical benefits of this structural understanding have not been documented; however, the structure fits the needs of our project, most notably in the improvement of reliability and availability of information for the project. While the larger evidence of MBSE's uses are expectation, the organizational structure of the method is the most reasonable way to connect the different levels and sectors of development for the project [15].

1.3 Project Proposal

The goal of this project is to create a plan for placing a satellite capable of photographing the surface of the moon in high detail into a low altitude eccentric polar orbit around the moon to create a composite map in preparation for future crewed missions to the moon. The project will be composed of systems engineering documentation that addresses multiple levels of design. Plans will be made for the mission plan and the properties required for the satellite will be estimated using multiple methods. The spacecraft will leverage previously used components and mission plans to minimize design and production costs.

1.4 Methodology

The design of the mission will focus on three different levels of design. Each level of analysis will use a combination of diagrams and software to describe the goals, requirements, and restrictions of their mission element.

The first level of design is the mission plan. Research will be done on similar missions and will be used to design a high-level systems engineering plan for the mission, including components such as the flight path of the mission (including required burns), defined phases of the mission, and design limitations for these phases (indirect communications, solar interference, etc.). The models for the orbits will be designed in GMAT and physical characteristics observed through this modeling will be documented. The planned flight path including the final orbit will be documented.

The second level of design will be leveraging similar system designs from the past to find components that can be used for the mission and designing a configuration of these components for the bus of the spacecraft along with a reasonable launch system for the spacecraft. Relevant requirements from the first level of design will be considered for the requirements for each component (how much power will be available to the system, satellite mass, etc.). Components will be selected based on their similarity to previous estimations and how well they can achieve the mission requirements. To find information about the expected values of these components and the overall mission without analyzing each individually, prior missions will be studied for relevant system requirements. These systems will be parameterized and the information from each put into scatter plots of data for each possible comparison between the various studied parameters. Then the scatterplots will be functionalized and component values with high correlation will be considered during the design process using their produced functions and regressive analysis of the functions and the data.

The third level of design will be the connections between these components. This will include how the payload affects the power requirements, weight, and propulsion systems of the spacecraft. Future considerations for the specific planning and implementation of satellite subsystems will be considered along with possible improvements to the estimated mission requirements.

2. Historical Data

2.1 Historical Camera Data

To understand the limitations of our mission, it is important to build a model of the relationships between our parameters based on historical data from similar missions. To do this, the values of these parameters will be compared to each other and functionalized. To use relevant data for these missions, similar missions from multiple countries were used to create relationships concerning the camera system used to map the surface of the moon, while more readily available data from NASA about the performance of American unmanned missions to the moon were used to determine more general performance measures for the spacecraft.

First, missions from multiple countries to map the surface of the moon were chosen and the specifications of their camera equipment were recorded. The relevant missions selected were the Lunar Reconnaissance Orbiter, the Chang'e-1, the Korean Pathfinder Lunar Orbiter, Selenological and Engineering Explorer, and the Chandrayaan-1. Each of these are missions to create composite images of the lunar surface for future missions by their respective agencies, meaning the camera specifications for these spacecrafts are important for understanding our own mission to do so. The parameters collected for each of these missions were the spatial resolution, field of view, required power, and mass of the cameras used to scan the lunar surface.

Table 2.1 Each of the parameters for the cameras studied.

	Name	Spatial Resolution (m)	FOV (deg)	Power (W)	Mass (kg)
LRO	LROC	0.5	2.86	10	5.4
CE1	CCD Imager	120	40	25	15.7
KPLO	LUTI	2.5	5.72	12	15
SELENE	Terrain Camera	10	22.4	16	14
CHYN1	TMC	5	11.42	13	6

As stated previously, the Lunar Reconnaissance Orbiter had sets of two cameras for each of the functions of the orbiter. A more detailed map image was required to produce the map of shadows near the lunar poles. This is where the value of 0.5-meter spatial resolution was taken from. This is the more relevant data point since the cameras used for this mission objective created the lunar surface map that will be most closely related to this project. The high level of detail relative to other projects hoping to achieve similar goals was of interest to this project since budgeting mission constraints will be an important part of analyzing the mission using these functionalizations. One element to note for this mission is that the cameras are smaller than the other ones in the table both in weight and in power demand. It will be important to take into consideration the effect mission design will have on the resolution of the cameras used. The LRO can take such high-resolution images of the ground due to its proximity to the ground during the low point of its eccentric polar orbit. Its low FOV helps with this process, narrowing the range of ground covered in its swath as it passes over the poles.

The Chan'e-1 was part of a larger Chinese program to perform unmanned landings on the surface of the moon. Its goal during these flights was to provide the program with a map of the shape of the surface of the moon rather than the detailed map NASA was seeking from the LRO [17]. The mission was at a higher altitude due to this and would take pictures of the general lunar features rather than detailed composites like the shadows being observed by the LRO. This meant that its FOV would need to be higher, and the altitude of the mission would need to be higher as well [18]. The mission was able to provide information for landings for the later parts of the Chang'e program, a goal that this project hopes to facilitate with the NASA Artemis missions. Incorporating the Chang'e 1 data may help understand the requirements for a less detailed scan and how that affects both the mission design and spacecraft design.

The LUTI system aboard the Korean Pathfinder Lunar Orbiter was designed after the LRO and Chang'e-1 so its cameras have been designed with those systems in mind. Due to the weight and altitude requirements a camera was selected that would give a resolution of 2.5 meters for an operating altitude of 100 km. This was done to accommodate the mission objectives with the requirements for operation. Mission designers considered the properties of the moon and restricted the operating altitude to 100 km [19]. This decision limited the camera operations of the high-resolution scans of the lunar surface. Considering the similarities between the objectives of the KPLO and this project, it will be important to find reasonable tolerance for our spacecraft in the altitude of the orbit and how eccentric this orbit is, since this data shows that it will affect the potential for surface scans and define the resolution more than the specific hardware of the mission.

The SELENE mission was able to create a uniquely detailed topographic map of the surface of the moon. These data readings were taken 100 kilometers over the surface of the moon. The detail provided can be explained by the smaller scope of the mission and therefore the smaller array of scientific instruments. Using a laser altimeter along with cameras, the topographic map was completed in greater detail than other missions to map the topography of the moon up until that point. This was done to facilitate future lunar exploration by the Japanese space program. The orbiter also deployed separate smaller satellites meant to relay the data taken by the orbiter, among other functions [20]. Since this project is focused on topographical information about the lunar surface, this information may be of interest to the Artemis program and a similar method for obtaining this topographic detail for NASA might be used that could reveal greater detail than the method used by the Chang'e-1. Further research into the array of instruments for this successful observation could give a better understanding of what would be required to replicate these results for a similar mission.

Chandrayaan-1 was operated for multiple purposes, one of which was the development of deep space capability of the Indian space program. Like many of the other missions studied for this project, the operational altitude of the satellite was 100 km. The satellite overheated and was brought into a higher orbit to protect itself from the heat it was experiencing. It was not able to overcome this problem and failed after its power supply overheated in 2009. This shows the limitations of missions to study the lunar surface and the issues that could be caused by selecting our altitude during the mission planning phase of the project. The satellite was not able to overcome its overheating issues at an altitude of 200 km, higher than was initially planned for the mission [21]. This shows that while altitude will be an important factor when designing the mission objectives of the project, the importance of onboard requirements will be of high priority as well. Other studied missions indicate that high performance can be extracted at the operational altitude of 100 km without losing instrument capacity. The resilience of onboard instruments will be a critical component of the mission especially when mission objectives call for spatial resolution and therefore FOV that can only be achieved at lower altitudes.

2.2 Historical Spacecraft Requirements

Since most of the lunar mapping missions used to compile camera data were launched by foreign space agencies and many are still ongoing, specific information about the parameters of the spacecraft is hard to find as readily as the camera specifications. Information about the cameras is usually referred to in press kits more frequently than the specifics of the satellite design, and ongoing missions tend to maintain ambiguity about the spacecraft specifications. NASA, on the other hand, has launched many missions to create orbiting satellites around the moon. NASA is also obligated to release more information about each mission to the public, meaning the data for each mission can be compared more reliably than those of the various space programs studied before. Four unmanned lunar missions launched by NASA were studied. These were Lunar Reconnaissance Orbiter, the Lunar Orbiter Program 5, Clementine, and the Lunar Atmosphere and Dust Environment Explorer.

Table 2.2 Each of the parameters for the spacecraft studied.

	Thrust (N)	Isp (s)	Total Power (W)	Mass (kg)	Total dv (m/s)
LRO	350	214.5	824	1846	1258
LOP	444.822	312	375	385.6	1400
CLEMENTINE	489	309	360	424	1900
LADEE	455	323	295	383	1374

Since this section focuses on the expected requirements for transferring the payload into orbit around the moon, a broader base of information could be leveraged. Instrument data was kept within the last 30 years to account for the rapid pace of technological developments in instrumentation. This section, however, is based on the larger history of NASA lunar exploration. Relationships between thrust and efficiency for missions requiring a certain change in velocity are similar over the history of lunar exploration and the tradeoffs associated with this are still being made in recent history. Our mission will use a conventional liquid propellant rocket to park our spacecraft in its operational orbit, so the detailed documentation of NASA missions is invaluable for this analysis.

The Lunar Reconnaissance Orbiter was placed in an eccentric orbit to specifically study the poles of the moon for future missions that would be conducted there. Small adjustments to its orbit were also made, a common characteristic of each of these missions. Its orbit was changed to be closer to the lunar surface later, bringing its orbit into a circular orbit meant to study the surface of the moon uniformly [4]. The LRO is the closest analogy to this project meaning that this approach of gradually descending the orbit will be an important component to study for this project and the spacecraft requirements will likely be similar to the LRO’s. This is interesting because the LRO is the heaviest satellite in the group and has the lowest total thrust of its thrusters combined. The extended gradual burns of the mission may have to be compared to other NASA missions which did not segment their orbital maneuvers this way.

The Lunar Orbiter Program 5 was an early NASA mission to take photographs of the potential landing site for the Apollo mission. This mission was selected because it is an early attempt to perform similar scans to this mission for a similar mission. It is notable that the performance of an older system like this is within the range of modern missions with the same goal [22]. This shows that the principles of traditional chemical rockets have not changed substantially since this mission which was launched during the early days of space exploration. Knowing the parameters of several missions over a long time will therefore be able to predict the efficiency of this project more accurately than fields that are subject to more change such as the instrumentation of a satellite. The latter develops very quickly as computing and miniaturization technology improves.

The Clementine mission is notable for its more complex flight path. The mission had many objectives and was inserted into lunar orbit and afterwards set to intercept an asteroid. This complexity gives a reason for the increase in necessary Δv for the mission [24]. This gives more range in the data set for missions to the moon allowing our estimation to be more accurate at different required Δv's. If the project changes in scope, these estimations could be useful for determining what changes would need to be made to other systems to accommodate this change.

The Lunar Dust Environment Explorer was a mission with different objectives than the other lunar orbiters studied in this report. Until now, lunar orbiters that have been discussed were designed to take pictures of the lunar surface from similar altitudes. LADEE orbited the moon as low as 75 km, lower than the previous missions [25]. For more detailed images, this project may need to lower its orbital altitude to increase the spatial resolution of a given FOV. The information from the LADEE spacecraft can be useful in this approximation process, despite LADEE not having similar onboard components as this mission.

3 Functionalization and Analysis

3.1 Camera Data Functionalization

Using this data, each of the parameters of the cameras are compared to each other on a series of scatter plots. These scatter plots are then used to create a function that describes the relationship between the parameters. This acts as a model to estimate the requirements for the mission if a realistic correlation is found between the data points. Figure 5 shows the six graphs produced by this process and their corresponding trendlines. Three of these show no reasonable correlation and have R^2 outside the acceptable range and can therefore be disregarded. These three are related to the mass of the camera, which interestingly also does not relate to the energy requirement of the camera, a conclusion that seemed counterintuitive. The three that showed correlation were the relationships between field of view, spatial resolution, and required power.

More parameters were recorded for this section due to the broader subject being studied. The total thrust and specific impulse of the propulsion system, the total power required by the satellite's power system, the mass, and the total delta-v available to the spacecraft were all recorded and plotted against each other as before. The relationships between each and the functionalization of each scatter plot are shown in Figure 7.

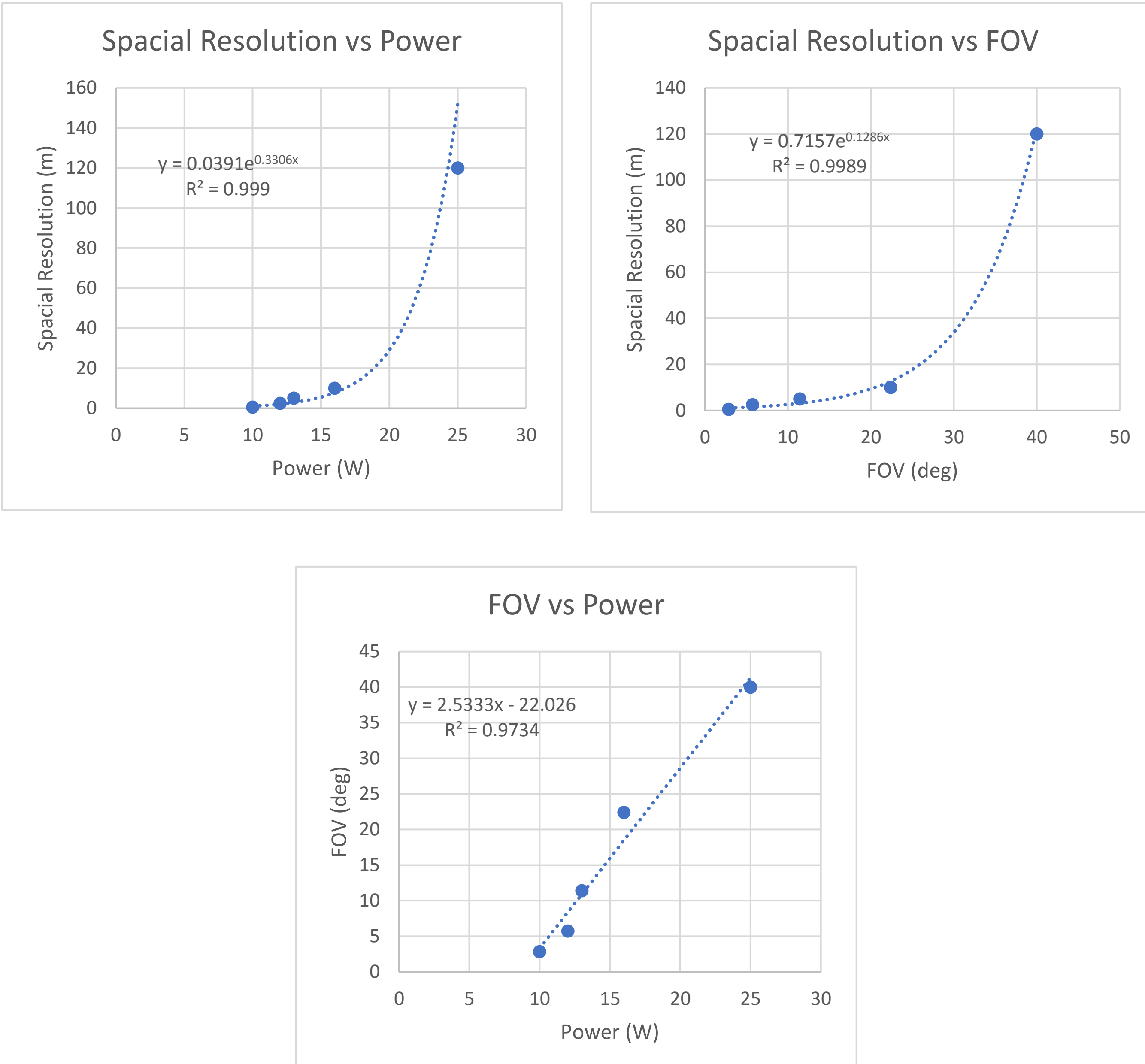


Figure 3.1 The camera parameters that showed correlation.

The relationships between these values are less surprising than the uncorrelated examples. It is logical that the field of view and spatial resolution of the cameras are almost perfectly related. Increasing the field of view of the camera means that the camera will be observing a wider swathe of lunar surface, decreasing the detail of the images produced. This decrease in detail means that each pixel of the image represents more of the lunar surface. What is notable is that increasing spatial resolution and field of view increase the power requirements for the camera to function. This will be important to note when considering the scope of our camera system with regard to the weight of the spacecraft devoted to meeting electrical needs.

The exponential relationship given by the Spatial Resolution functions presents an interesting design question. Lowering the standards for terrain resolution will result in exponential savings, while the diminishing returns of increasing this resolution will need to be considered when budgeting elements such as power consumption for the mission. Mission objectives will need to be considered carefully due to this relationship, as reducing the standards for resolution in the images produced could significantly improve the budget of the mission, especially if other mission objectives would need to be achieved as well.

3.2 Historical Spacecraft Functionalization

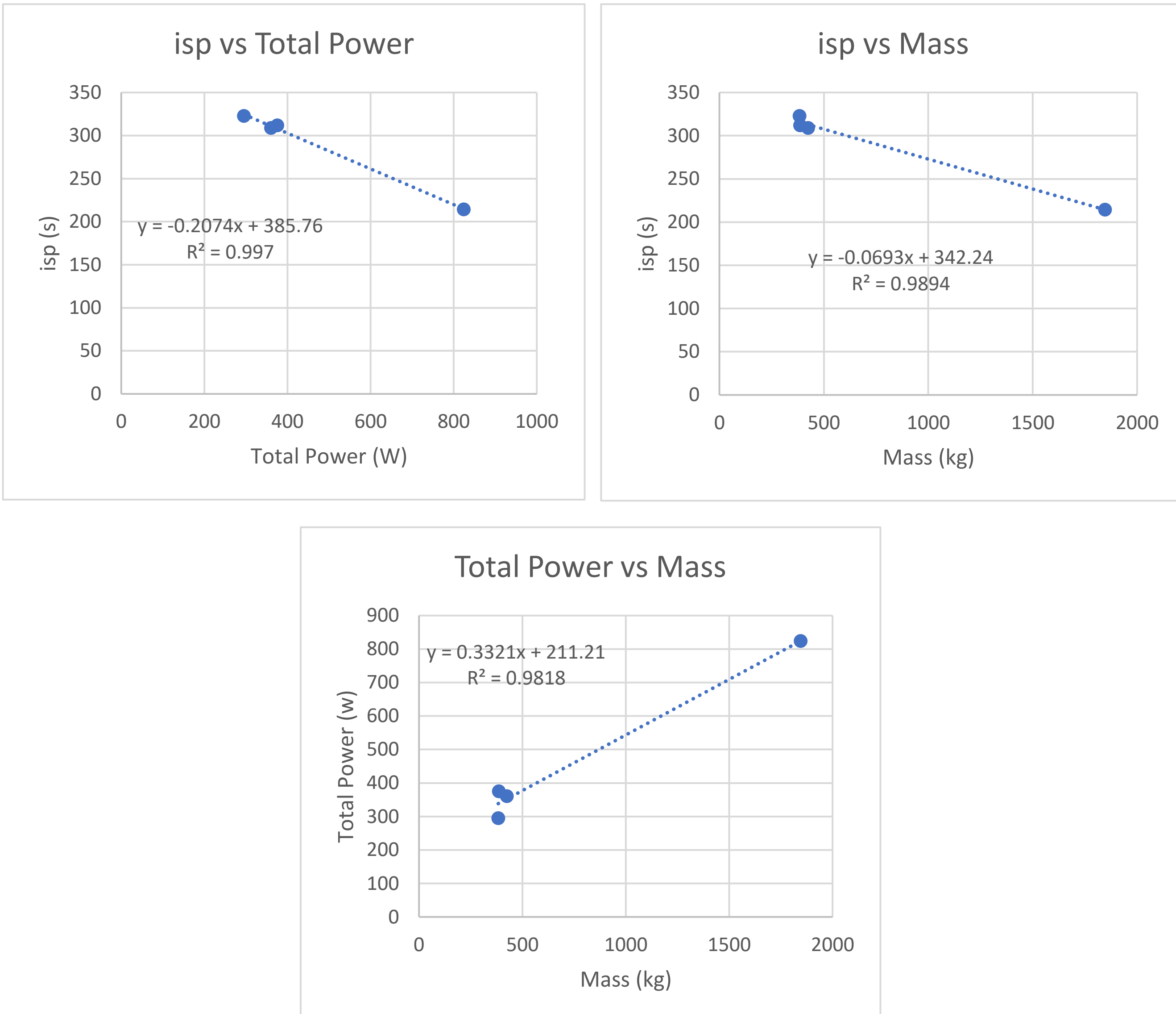


Figure 3.2 The relationships between total power, mass, and specific impulse of the spacecraft.

In this set of data, only the total amount of delta-v available to the spacecraft showed no correlation to the rest of the data points. This is surprising, but the remaining six correlated trend lines tell an interesting story considering this lack of correlation. Most notably, the specific impulse of the engine is one of the most direct correlations when compared to the total power and mass of the spacecraft. As the mass of the spacecraft increases or the spacecraft requires more electricity to function, the specific impulse of the propulsion system decreases. This also ties into the relationship between the mass of the spacecraft and its electric requirements, where the next strongest correlation shows that as the power required by the spacecraft grows, the mass increases linearly. This could be a design constraint for our spacecraft, since the size of the electrical system, and therefore the mass of the spacecraft, limits the efficiency of our propulsion system.

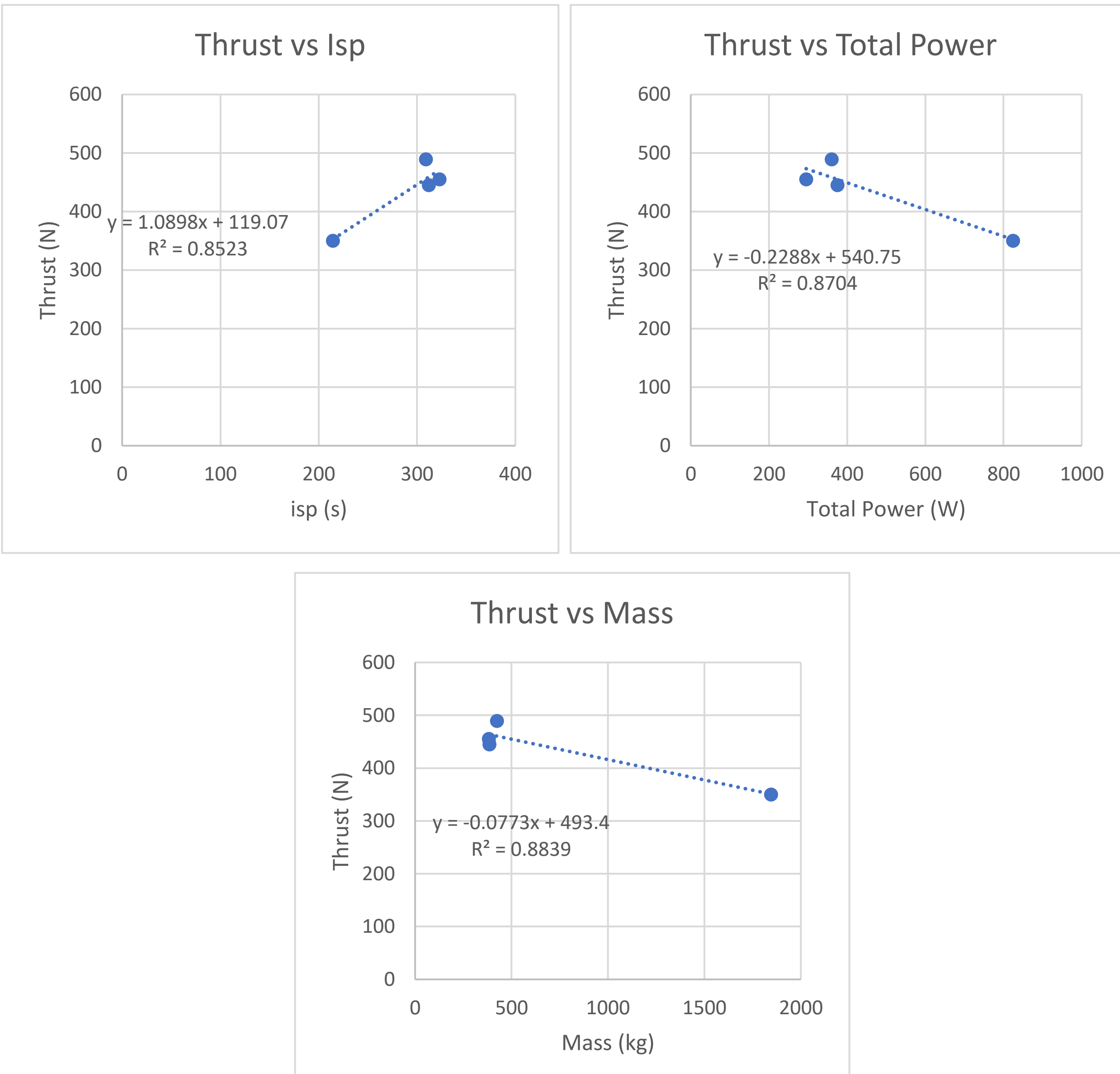


Figure 3.3The correlative relationships between thrust and other spacecraft parameters.

Total thrust also showed a looser correlation to the performance of our spacecraft. Counterintuitively, as the spacecraft's mass increased, thrust tended to decrease. In line with the observation from before, the increase in mass also lead to a decrease in specific impulse. This showed that when thrust increased, specific impulse increased with it. Since this was a function with a lower R^2 value, the large mass of the LRO may have influenced this trend and should be considered when using the functions in the model to determine the link between thrust and weight.

The grouping of data points indicates that satellites used for lunar scientific analysis tend to follow similar mass and thrust characteristics under normal conditions. The LRO is an anomaly among these data points due to its large size. Since this project resembles the mission objectives of the LRO, special care should be taken in analyzing what design choices led to the increased mass of the LRO during the design stage of the mission. These special characteristics could indicate some necessary changes to the approach of this project which might require further analysis of missions under the same conditions as the LRO to find how best to budget larger payloads to accommodate specific mission objectives.

3.3 Functionaliztion Regression Analysis

To better understand the more important models developed in this chapter, the relationship between the datapoints and the relationships they have to their corresponding functionalization were analyzed. Two methods were used for Regression Analysis: Sample Standard Deviation and Relative Standard Error. Previously the R^2 value was given for the functionalization of data points. Finding the Relative Standard Error for these functions will give expected spread from this function. Together these values will give the expected deviation of both the functionalized and real data for each of the comparisons of legacy systems studied for this project.

Figure 3.4 The calculated SSD and RSE of the functionalizations.

SSD	Isp	Total Power	Mass	Total dV
Thrust	0.111692	0.490813926	0.543246	0.9610863
Isp		0.936529745	1.080728	1.5181569
Total Power			0.897633	1.380316
Mass				1.2535648

RSE	Isp	Total Power	Mass	Total dV
Thrust	0.061212	0.056941608	0.053841	0.1157541
Isp		0.01099525	0.02015	0.1988474
Total Power			0.118443	0.510338
Mass				0.790351

Sample Standard Deviation is a method to determine how close elements of a data set are to their mean. It is used to indicate whether new elements in the data set will stray far from the mean of the data, which shows the precision of the estimates given by looking at the data points to understand the expected range of future values. When estimating the values of future data points using functionalization, it will be useful to know how precise those values will likely be and how much they will vary. The method for determining this value is given by the equation:

$$SSD = \sqrt{\frac{1}{n-1}\sum(x_i - \bar{x})^2} \qquad (3.1)$$

Where x_i is the specific value being summed, $\bar{x}$ is the mean of all the data points being studied, and n is the number of observed values. Calculations were done using Excel and are compiled in the first table of Figure 3.4.

The method for determining the value of Relative Standard Error is given using the equation:

$$RSE = \sqrt{\frac{1}{n-m}\sum(\frac{y_i}{f(x_i)} - 1)^2} \qquad (3.2)$$

Where n is the number of observed values, m is the number of parameters estimated, y_i is the value of the real data for the data point being summed, and $f(x_i)$ is the expected value of that data point based on the functionalization of the data.

Sample Standard Deviation returned somewhat predictable results. For functions with low R^2, SSD was usually close to the average value of the ratio between independent and dependent variables. This was not always true, and some data points were separated by larger amounts than may be considered reasonable for this analysis. When using these corresponding models, extra care will be taken to ensure that the values of outliers do not influence the functionalization of the historical data. This is most notable in the field of view of the Chang'e-1 and the mass of the LRO. Both will be scrutinized more when functionalization is used to predict the limitations of spacecraft design available.

Regression Analysis was more reliable, however. Most of the data points for high R^2 value functions fell within 50% of the estimated value. The lowest percentages were given by linear functions, while higher percentages were given by exponential functions. The linear functions produced RSE as low as 2% while exponential functions created RSE that hovered around 50-60%. Graphs that did not correlate predictably tended to produce values that exceeded 100% RSE, showing how far data points tend to stray from the trend line when R^2values dip below 0.5. This supports the accuracy of these high R^2 value functions in determining expected data. They will therefore be used to estimate budgets for the project and analyze the relationships between subsystems.

4 Mission Requirements

4.1 Mission Necessities

To use the information collected so far, the specific goals and requirements to attain those goals must be clarified. This will give the specific information needed to estimate the larger project's necessary combined specifications. First, goals of our mission in relation to the Artemis program will be defined, then these goals will be used to derive the requirements for their respective components. Finally, the specific requirements derived from those goals in our previously derived models functionalizing the relationships between the different components on the spacecraft as well as the cost of the mission will be used.

First, what services our lunar orbiter will provide to the Artemis mission must be understood. Its primary goal will be to provide imaging for the specific area around the landing site of the lunar base; its secondary goal will be to analyze the contents of the lunar surface in the area around the landing site as well as any region that could be utilized later. Both operations will require different sensors that will need to operate at different altitudes.

The optical parameters for the cameras used to map the lunar surface will be based on the required resolution of the images as well as the intended field of view of the cameras. Composite images can be created using different sized focal lengths, so the specific requirements for the cameras are not necessarily directly related to one or the other specifications for the camera. However, a set amount of detail will be revealed using any method, with the tradeoff being the amount of surface area that can be mapped relative to the number of orbital passes and field of view of the camera. The higher the field of view of a camera is, the more resolution it must have to map a similar amount of detail. Using a lower field of view for the camera will also require that the satellite will need to orbit closer to the moon, creating new challenges for the mission along with the requirement for the satellite to pass over the same region multiple times to fully map it.

A second set of cameras could also be used to watch the horizon as the spacecraft orbits the lunar surface. These cameras can use spectral imaging to analyze the contents of the lunar surface. This could be useful around the base site of the mission and other potential future areas of interest across the lunar surface. Understanding the contents of the lunar soil is an important part of the exploration being done by Artemis. The possibility of their use will be determined in future iterations based on the requirements of onboard subsystems.

Since a moon-based orbit will be used, different altitudes and FOV values that will be able to cover the surface of the moon will need to be understood. Using equations for the Earth and adjusting them to analyze the ground track of satellites orbiting the Moon, the amount of coverage the satellite will be able to provide to the Artemis mission can be found.

Since boulders on the surface of the moon can be as small as 0.13 m, the spatial resolution required to analyze the smallest boulders in the landing area should be estimated to be 0.1 m. Using the relationship described in our estimation model it will require a FOV of about 1.1 degrees. If we assume that we will be scanning a swathe of the moon with an angular radius of ρ, we can use the equation:

$$\sin(\rho) = R_M/(R_M + H) \qquad (4.1)$$

Where R_M is the radius of the moon and H is the altitude of the spacecraft. If we assume that the altitude of the orbiter will be similar to the altitude of the LRO, we can estimate that the value of H will be 50 km. Using this information, the value of ρ of 76.4 degrees can be determined. This will be important when we need to understand what parts of the landing site will be photographed and the angle of the second set of cameras that will be observing the horizon.

Using this information, the FOV of the camera will be 1.1 degrees and the altitude the satellite will best orbit is 50 km. This is an important set of requirements since we need to understand how the camera will integrate into the rest of the spacecraft and will also determine the size and shape of the orbit when we begin budgeting for the deltaV of the mission. This will be important when determining the propulsion system for the mission.

4.2 Subsequent Required Specs

Specifications for the satellite will be affected by the values we determined in the last section. Since we were unable to find correlations for other values, we can determine the values for spatial resolution and power requirements. This allows us to find the spatial resolution required for the FOV using our estimation models and also allows us to use these values to determine an estimate for the amount of power required to run the spacecraft overall.

The estimated mass of the spacecraft will be determined by using the power requirement. This value also correlates with other estimation models. We can use these to get a general understanding of what we need to configure other parts of the craft to accommodate. This will be iterated on when a specific orbital transfer plan is established and the deltaV requirements are no longer an estimation. The values that will need to be re-estimated will be the thrust and the specific impulse of the engine.

The equations used to estimate the values of power and spatial resolution are:

$$SR = 0.0391e^{0.3306P} \quad (4.2)$$

$$SR = 0.7157e^{0.1286FOV} \quad (4.3)$$

$$FOV = 2.5333P - 22.026 \quad (4.4)$$

The equations used to estimate the isp and thrust of the engine will be based on the power estimation as follows.

$$P_T = 0.3321m + 211.21 \quad (4.5)$$

$$isp = -0.2074P_T + 385.76 \quad (4.6)$$

$$isp = -0.0693m + 342.24 \quad (4.7)$$

4.3 Estimated Values

Using the techniques from the previous section we can estimate that the value of the spatial resolution is 0.1 m and the power requirement will be 9.13W. This section is less likely to be iterated on since it is directly related to the mission requirements defined earlier. What can be changed is the altitude of the orbit since we have not finalized the transfer path.

These values are important to understand the later requirements for the sizing and control systems. We will discuss the implications of these values in the next chapter. We will need to iterate on these as the design changes and the mass of the spacecraft necessarily changes with it. For now, we can use the equations from before to estimate the value of the mass.

This value, based on the power requirement derived from the necessary specifications for the camera, will be used to estimate the specifications for the engine. For now, we can assume power consumption will be similar to previous missions using cameras that drew around 9.13W. Similar missions used around 800 W in total. Using the second set of equations we can determine that the engine can be estimated to have a specific impulse of at least 219.84 seconds and the craft will have a total mass of 1772.9 kg. These are the most likely values to be iterated on multiple times throughout the design process as both the mass of the spacecraft and the requirements for the orbit are solidified rather than approximated. Using this information in the next chapter we will determine the specific requirements for the onboard systems of the spacecraft.

If needed the models will be used again as specific values are narrowed down and their corresponding values need to be estimated again. Until those specific values are narrowed down, we will need to approximate them using historical data as a placeholder.

4.4 Orbital Requirements

To evaluate the necessary component requirements derived from different orbital designs and compare different configurations, multiple orbits were evaluated. The various coverage maps and other characteristics related to the final orbit were also studied in GMAT to determine the relationship between vehicle efficiency and orbital efficiency. The results support previous mission descriptions and an orbit similar to the orbit of the LRO was determined to have the best combination of payload requirements and orbital stability.

The final orbit was determined to be most efficient as a circular orbit 50 km above the surface and with an inclination of 86 degrees. This configuration provided ample coverage of the potential base sites for the Artemis mission and reduces maintenance required for the orbit by leaving the satellite in what is known as a frozen orbit. This is important because at low altitudes the gravitational field of the Moon disturbs the orbit of spacecraft over long periods of time. Frozen orbits exist in a small range of inclinations and allow the spacecraft to travel in a consistent orbit without needing extra design accommodations for the maneuvers required to maintain the intended final orbit [4]. This is important for this mission since the spacecraft will already be high mass, and any savings in the budget for propellant weight could mean more freedom in areas where requirements would otherwise increase the scope of the mission [26].

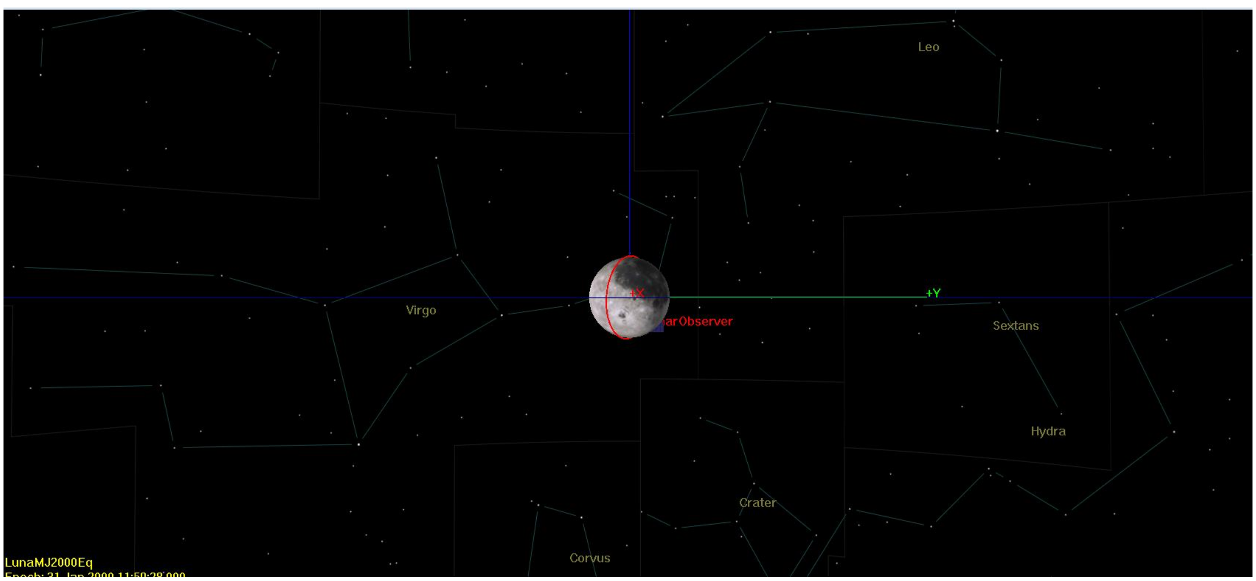
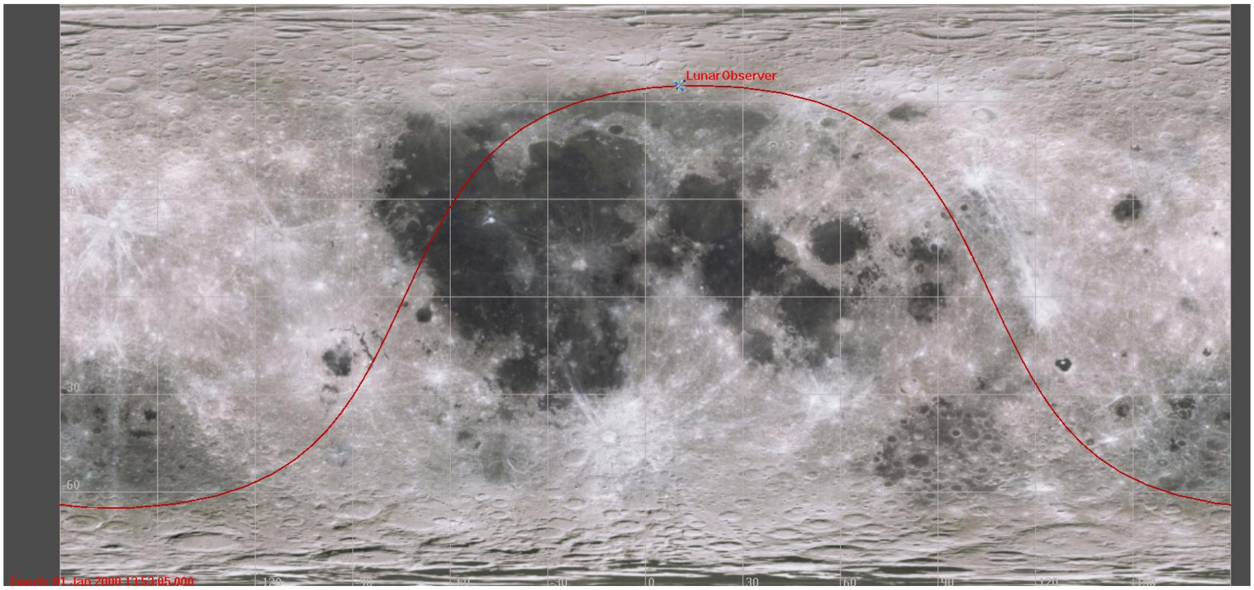


Figure 4.1 The chosen orbit as rendered in GMAT.

Based on this simulation, a ground track plot for the spacecraft can be produced. The most important part of this track is where the coverage can reach and the spacing between the ground tracks for each consecutive orbit. This determines the efficiency of the scanning process since closer spaced bands allow the image to be processed multiple times over an area or the field of view of the camera can be narrowed to improve the size and power usage to capture a similar resolution image.



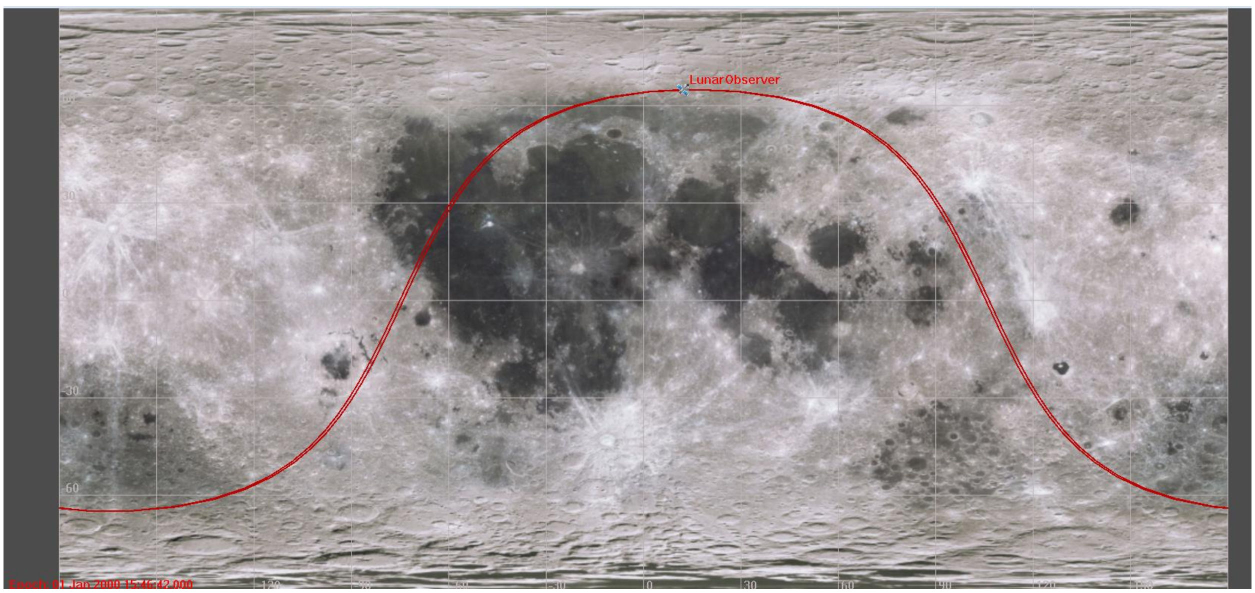


Figure 4.2 The ground track of a single orbit (Top) and the ground track over two orbits (Bottom).

A MATLAB script was used to do the hand calculation for the various possible combinations of ground track of the various inclinations and altitudes that were inspected. The specific calculations for the track of the orbit that was picked will be expanded on in the next chapter where the values derived from those calculations will be used to design the payload for the mission. The ground swath of the chosen orbit is very thinly spaced. This is largely due to the slow rotation of the moon relative to the Earth. The ground track bunches up as the spacecraft crosses the poles; however, this is due to the way the map is projected which stretches the space between longitudinal lines to keep them parallel which makes latitude differences different than they appear at the equator. The hand calculations in the next chapter address this in the calculation of the spacing between ground track lines.

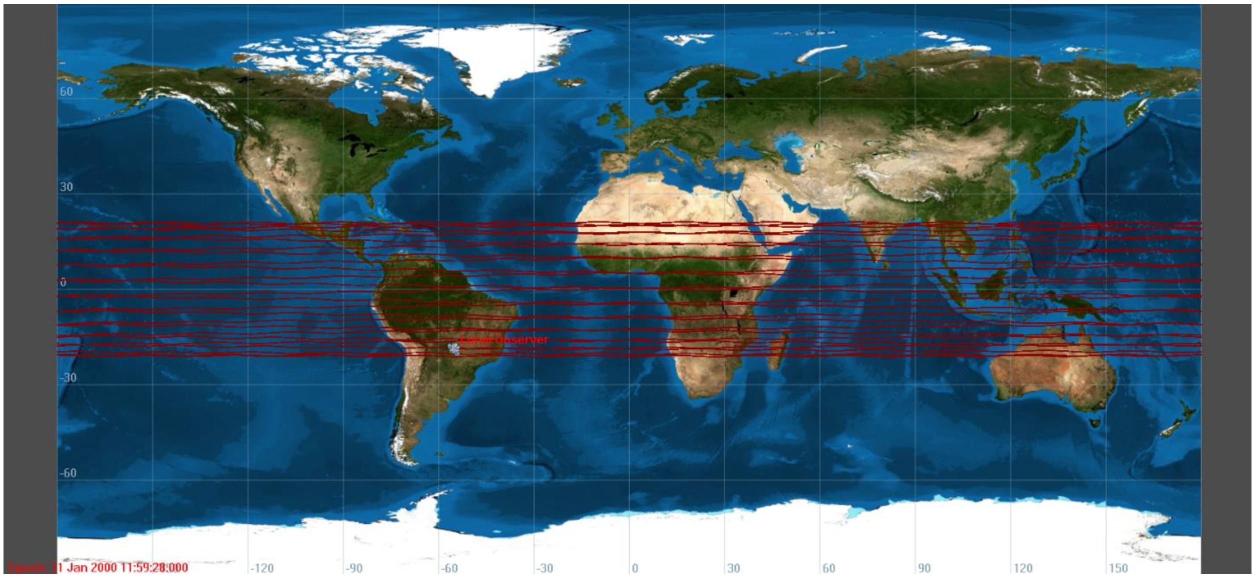


Figure 4.3 The ground track of the satellite over the Earth while orbiting over the Moon over a month.

The ground track can also be shown over the Earth to understand the range of latitudes the satellite will be above as it passes over Earth. The small perturbations are made by the satellite orbiting the Moon. The larger slow changes in latitude are caused by the moon orbiting the earth. These bands are filled in by the Earth's rotation while the moon slowly drifts across the latitude lines over the course of its orbit. This is because the Moon's orbit does not align with the Earth's equator. This information is useful for communications systems as images will need to be transmitted to Earth from the satellite over a long period of time and so the pointing budget of the spacecraft will be affected by the rate at which adjustments in the spacecraft's orientation will need to be made.

5 Payload Design and Sizing

5.1 Orbital Coverage Determination

To understand the requirements for our payload, the largest component will be studied to get an estimate of what requirements will be affected by the objectives of the mission. Numerical analysis of the requirements of the mission will be used to determine information about the camera that we will need for the requirements listed earlier. This will be used to find specific values for the camera and its capabilities that will be needed. This will allow the necessary components of the camera to be compared to a more specific model that has already been used and then scaled to match the specific requirements determined by the numerical analysis.

To understand what will be required of the camera, general information about the satellite's orbit must be determined. These characteristics determine the relationship between the amount of time the spacecraft remains over an area and the coverage of that area. This will determine the characteristic features of the optical system. Based on these features it will then be possible to estimate the requirements for the payload in general. Since the payload is the element of the spacecraft that interacts with the subject, the focus of this analysis will only be the camera system since that is the role it fills in the mission.

Since we know the required payload capability that was defined earlier, along with the altitude the moon will be studied from, this information can be used to derive what specific orbital characteristics can be determined [27]. First, the orbital period is determined using:

$$P = \frac{2\pi h^{3/2}}{\sqrt{GM}} \qquad (5.1)$$

The orbital period is therefore 113.6 minutes, just shy of 2 hours. Since the orbit is very low for a lunar orbit, the period is around as short as it can be which will aid in quickly returning new information as the swaths of the surface are covered quickly and composite images of the surface can be put together in a relatively short time [27]. From here the velocity of the spacecraft over the lunar surface is determined as well using the angular velocity of the craft and the radius of the moon as shown here:

$$V_g = 2\pi \frac{R_M}{P} \qquad (5.2)$$

This gives us a value of 1.6 km/s as the speed the spacecraft passes over the ground. This can be used to determine the rate at which data must be collected when imaging the surface.

From here information about the frequency the spacecraft passes over specific areas can be derived along with information about the area that will be observable from the spacecraft's position over time. The longitudinal spacing is directly related to the period we determined earlier. Using the amount of time in a day and the orbital period, the amount of rotation of the planet can be determined for each time the spacecraft passes the equator [27]. This variable, also referred to as the node shift (ΔL), is given by:

$$\Delta L = \frac{P}{39343} 360° \qquad (5.3)$$

The node shift is therefore only 1.039 degrees every time the spacecraft passes the equator. It is important to note that this is a much lower value than would be expected for spacecraft orbiting Earth since the moon spins much more slowly.

The minimum elevation angle defines how high the spacecraft must be above the horizon of the target area to continue studying that area properly. Similar optical designs in the past have had a minimum elevation angle of 20 degrees. This value will be the base of our estimations for the required camera operations. However, the number can be increased or decreased based on the cost analysis of the mission and the specific requirements of the candidate camera that is being considered. Iteration in this area will be important so the variables associated with this value will need to be studied again when more specific candidates for the payload are selected using information from those candidates [27]. With this assumption in mind, the information needed to numerically analyze the field of view, size, and general effectiveness of the camera can be determined.

The first variable to be determined based on this is the maximum sensor look angle, which is also known as the nadir angle (η). The important value to understand for this angle is its maximum value, since this defines the size of the swathe taken of the lunar surface as the spacecraft passes overhead. This maximum can be given using:

$$\eta = \sin\rho \cos\varepsilon \qquad (5.4)$$

The value for the max nadir angle is therefore 65.97 degrees. This will be important mostly in the context of what ground imaging can be done each time the spacecraft orbits the moon.

The value for the Moon Central Angle is based on the values for both variables using the simple relationship of:

$$\lambda = 90 - \eta - \varepsilon \qquad (5.5)$$

Where λ is the maximum value of the Moon Central Angle, η is the maximum value of the nadir angle, and ε is the minimum value of the elevation angle. The Moon Central Angle is therefore 4.03 degrees.

Using these values, the maximum distance to the horizon of the spacecraft's observation ability can be determined. This will form the basis of the understanding of the swathe taken by the spacecraft when orbiting the moon and therefore how much coverage the camera will be able to perform and what requirements will be needed for the hardware in the camera based on this [27]. The maximum value of D can be determined using the equation:

$$D_{max} = R_M\left(\frac{\sin\lambda}{\sin\eta}\right) \qquad (5.6)$$

The maximum value of D can therefore be calculated as 133.89 km. This is also known as the slant range and will be an important factor when determining the range of the spacecraft's camera.

Using this information, more characteristics of the ground swathe of the satellite can be determined. One of the more important components to understand is the spacing between swaths since it gives more of an understanding of what the overlap will be when pictures are compiled and therefore how much detail can be gained in a set amount of time. The spacing between parallel swaths in the longitudinal direction can be determined using:

$$S = \sin^{-1} \sin \Delta L \sin i \qquad (5.7)$$

Where S is the value for longitudinal degree spacing in degrees. This gives a value of S as 1.036 degrees. Since the swath width is given by a very simple relationship to the Moon Centered Angle (λ) these values can be compared by quickly determining the swath width using λ.

$$swath\ width = 2\lambda \qquad (5.8)$$

Since swath width is 8.06 degrees, it is apparent that there will be large overlap between each swath of images taken of the landing sites that the mission is interested in documenting. This is good, since it allows us to either limit the resolution of the images to reduce the number of pixels that must be saved and transmitted, or it could allow the camera to be operated over a narrower range of terrain and therefore reduce the complexity required for the optical systems. Both options offer opportunities to reduce the necessities for the mission if the budget for subsystems needs to be reduced or the complexity of components could be an issue for reliability [27].

5.2 Optical Sensor Design

The information about the orbital characteristics of the spacecraft can be used to determine the necessary requirements for the camera system used on the spacecraft. First, however, a few qualitative design questions need to be addressed. When designing an imaging system for a passive subject like the moon there are several options for the basic design of the camera. The most notable design decision that will need to be made for the camera used on this satellite will be how it captures the ground as it moves beneath it. There are two similar solutions to the question, the whisk broom scanner, and the push broom sensor. These both operate on the same principle of quickly storing information about the image as pixels using the movement of the spacecraft over the ground to advance the lines of pixels being captured. The whiskbroom method sequentially captures a sequence of vertical pixels (these are oriented in the direction the spacecraft is moving) in a horizontal direction to create the image. The push broom configuration instead captures all the pixels in a horizontal line as the spacecraft moves new horizontal lines into the capture as it moves. The push broom is the more expensive option of the two since it must compute more pixels at a time, however, this allows it to dwell on the pixels for longer. This reduces the computational strain and the bandwidth required if the images are being transmitted back in real time. For now, the tentative design assumes that the whisk broom will be used due to its low cost and complexity and since the images will not have to be taken continuously as the spacecraft orbits the Moon. This allows the spacecraft to process data and transmit it for longer than other optical satellites that are taking continuous scans of their passive subjects. If iterations are required where the push broom method is adopted instead, the related variables for the camera requirements will be updated as well.

Another option is the Step and Stare Imager which actively aims at specific areas of the lunar surface over a period to capture the details of the area over an extended period of time. This creates a much more detailed image with better defined geometry; however, it is much more expensive and complex. It also requires extensive calibration to ensure that the area being observed aligns with what is expected over that period of time. The satellite would only need to use this method of recording data if the other methods proved to be too costly for the satellite in terms of bandwidth, or if the level of detail required by the mission objectives exceeded the reasonable expectations of the previously mentioned methods.

Since the method for data collection has been determined, reasonable estimations of the necessary specifications for the optical system can now be made. Since the required resolution for the spacecraft is 0.5 square meters, the angular resolution can be calculated using:

$$\theta_r = SR/h \qquad (5.9)$$

The value for angular spatial resolution is therefore 0.1 degrees. This is a reasonable value for the camera system planned to be used aboard the spacecraft. From here, based on the linear spatial resolution, the instantaneous field of view (IFOV) can be calculated using the following equation:

$$X = IFOV * h(\frac{\pi}{180°}) \qquad (5.10)$$

From this the instantaneous field of view for the camera can be calculated where X is the linear spatial resolution. IFOV is therefore 5.7296×10^{-4} degrees [27]. This will be important when determining the relationship between pixel resolution and the time it takes to scan the image.

Using this information, the number of pixels observed along a track can be determined. The number of cross track pixels (Zc) can be determined along with the number of swaths recorded along the track per second (Za). These are combined to create a square of pixels that constitute the image. The number of pixels in this square (Z) will be important when understanding the budget for data aboard the spacecraft since this defines the number of pixels that need to be stored every second. These values can be found using:

$$Z_c = \frac{2\eta}{IFOV} \qquad (5.11)$$

$$Z_a = \frac{V_g * 1s}{Y} \qquad (5.12)$$

$$Z = Z_c Z_a \qquad (5.13)$$

The respective values can therefore be found to be: Zc = 2.3028×10^5 pixels, Za = 3200 swaths, and therefore Z = 7.3689×10^8 pixels. This Z value will be used to determine the number of pixels and therefore data needs to be stored per second. The data rate can be found by multiplying the number of pixels stored per second by the number of bits required to store a single pixel. Assuming each pixel requires 8 bits to be stored or transmitted, the data rate required to either store or transit the information is 5,895.14 Mbps. This number is high and iterations on the design process should work to find ways to distribute this data collection over a longer period of time, ideally without using the step and stare method described earlier. The large overlap between different swaths might be useful in reducing the number of pixels required by either reducing the resolution of the camera or reducing the width of each swath [27]. For now, the small area being studied implies that the data processing and transmission can be distributed over the two-hour orbital period to reduce the computational requirements for the spacecraft.

Standard rates for whiskbroom optical sensors tend to capture 256 pixels per instant. Using this information, the period required to integrate the necessary number of pixels each instant (Ti) can be determined using:

$$T_i = \frac{Y}{V_g}\frac{N_m}{Z_c} \qquad (5.14)$$

Ti is therefore 3.47 microseconds. Using this, the frequency of these integrations (Fp) can be determined to be 287.5 kHz. This limits the possible material to build each sensor to a few elements, unfortunately. This seems to be less avoidable than the other possible budget complications since fewer variables are involved that can be easily changed with reallocation of resources.

Finally, the specific optical properties can be determined. Assuming that industry standards are used (square detector width (d) of 30 μm, Image Quality factor (Q) of 1.0, and an operating wavelength (λ) of 4.2 μm) [27] the focal length can be calculated using:

$$f = \frac{h*d}{X} \qquad (5.15)$$

The focal length is therefore 3 m. This allows the necessary diameter of the aperture to be solved as well. This is important for determining the size of the payload and will be used to scale the satellite to be comparable to a similar payload that has already been used extensively. The diameter of the aperture (D) can be found using:

$$D = \frac{2.44\lambda f Q}{d} \qquad (5.16)$$

The diameter of the aperture is therefore 1.0248 m. This is a large aperture for a satellite, but it is not unheard of for observation missions like this one. From this calculation multiple aspects of the spacecraft can be determined. The scale of the spacecraft relative to a similar spacecraft can be determined. This will allow the other components of the payload to be scaled to the size of the aperture of the camera, which often determines the size power draw and weight for the other components of a camera like this one. It also allows more precise estimates for the field of view and other elements of the optical sensor to be made:

$$F\# = f/D \qquad (5.17)$$

$$FOV = IFOV * N_m \qquad (5.18)$$

The value for the F# of the camera is 2.9274. The FOV is 1.4668 degrees, which is significantly different from earlier estimates.

5.3 Payload Scaling

Finally, an estimate can be made using the properties of the camera of the size and requirements of the payload as a whole. The size of the aperture for the camera is a good indicator for how it will compare to other systems that have been used in past missions. A similar payload is listed in the textbook called the Multi Spectral Mid-IR. The aperture of that camera was 1 m [27]. This value is nearly identical to the camera needed for this satellite. Using the formula:

$$R = D/1m \qquad (5.19)$$

The ratio between the aperture of the required camera and the example camera from the textbook can be found to be 1.0248. This ratio can then be used as the scaling factor for other components of the payload.

The suitable components to be compared are the size, weight, and power consumption of the spacecraft payloads. Each of these scaling issues have their own associated equation. A factor (K) is used in some cases when scaling the components. It is used to diminish the scaling down of the payload if the theoretical aperture is much smaller than the aperture being studied. Since the apertures of the two spacecraft are nearly identical, the value of K in the equations will be set to 1 and therefore ignored.

Starting with the weight of the spacecraft, the formula for this component's scaling is given with the formula:

$$W_i = KR^3W_0 \qquad (5.20)$$

Since the weight of the payload from the textbook is 800 kg, the theoretical optical system can be estimated to weigh 861 kg.

Next the power consumption of the payload needs to be analyzed. The formula for this component is the same as the weight component:

$$P_i = KR^3P_0 \qquad (5.21)$$

Since the textbook example draws 900 W of power, the theoretical optical system can be expected to draw 968 W.

The formula to find the values for the dimensions of the camera are simpler than the ones for the other components. Each of the dimensions are instead multiplied by the ratio determined earlier. These dimensions are taken from the size of the textbook example as well. The formula can be given as:

$$L = RL_0 \qquad (5.22)$$

Applying this to each of the dimensions provided in the example (1.5 m and 1.0 m) the size of the spacecraft payload can be estimated to be 1.5372 m x 1.0248 m.

The comparison of these estimations to the values given earlier from the broader mission analysis will be explored in more detail in the budgeting section. Ultimately, the payload information determined through the sizing of the optical sensor is used as the basis for estimations about the spacecraft as a whole and the subsystems that accommodate the craft in that chapter.

6 Spacecraft Design, Sizing, and Budgets

6.1 Spacecraft Weight

Since the weight of the payload has been estimated, knowledge from prior missions can be gathered to determine some of the subsequent estimates for the spacecraft as a whole and its subsystems. These are only early estimates, which means that further iterations will bring new specific estimates for the values that can only be approximated here.

The general weight of the spacecraft without its propellant can be determined based on the average relative weight of spacecraft and its payload. Of the missions that have been launched in the past 50 years, the total weight of a spacecraft was on average 3.3 times heavier than its payload specifically [28]. Since the estimated mass of the spacecraft's payload is 861 kg, the mass of the spacecraft without propellant can be estimated to be 2,841.3 kg. This kind of estimation is required to be conservative, so a large value is for the dry mass results. This conservative estimate allows the future planning to not exceed expectations and means that budget requirements will most likely be lower than their expected values.

The next subsections will concern the expectations for the subsections based on these estimations and the requirements derived from the previous chapters. The necessary propellant mass will be derived and the preliminary power subsection requirements for the spacecraft will be found.

6.2 Propellant Requirements and Density

The required propellant budget will be based on the amount of propellant needed to perform the maneuvers after the spacecraft has been separated from the launch vehicle. A MATLAB script was used to estimate the values of the Δv required to perform the lunar orbit insertion and the subsequent lowering of the orbit into a polar orbit around the moon. The values of these estimations were combined with the planned mission structure of the LRO, which flew a similar route to a polar lunar orbit. Comparing the estimation to the past mission, an average estimate of around 1400 m/s of Δv will be required to perform the insertion and bring the satellite to its final orbit [4].

Using the rocket equation, the weight of the propellant can be solved algebraically. This allows the final estimated mass of the payload to be estimated. Using this as a base weight, the size of the spacecraft can be estimated, and preliminary design considerations can begin for the launch vehicle among other subsystems. The rocket equation is given as:

$$\Delta v = I_{sp} g_0 \ln \frac{M_{dry} + M_p}{M_{dry}} \qquad (6.1)$$

Which when solved for M_p returns a value of 2115.01 kg. This can be combined with the value of M_{dry} to give the final weight of the satellite: 4956.3 kg.

This is a much larger spacecraft than the LRO, meaning that new consideration needs to be given to the launch vehicle for the satellite [4]. The LRO used an Atlas V rocket to launch into orbit and into its lunar transfer orbit. The mass of the satellite exceeds the weight limit of Atlas V to reach a lunar transfer orbit. The satellite falls within the range for a medium sized lifter, such as the Atlas IV Heavy or Falcon 9 Block 5. Since the Atlas rockets have been retired, this limits the options for the launch vehicle and may restrict the spacecraft to a set of dimensions that may not have otherwise been required due to the limited number of launch vehicles capable of bringing the satellite to its destination.

To understand the sizing requirements for the spacecraft, the average density of other space missions can be used to estimate the volume of the fully constructed satellite. A conservative average of the density of older missions is ρ = 79 kg/m^3 [28]. This means that using the value of total mass obtained earlier, the volume of the spacecraft can be expected to be 62.47 m^3. This is a large payload and will need to be configured based on the necessary subsystems to fit into a limited number of potential fairings, which will need to be considered during any of the iterations of the design that will be considered.

6.3 Power Subsystem

A conservative estimate of the power requirements for a medium or large size satellite assumes that 40% of the power used by the satellite will be used by the payload [28]. Using this information, the estimated value of the total power usage of the spacecraft will be 2420 W.

This amount of power requires the spacecraft to bring in an equivalent amount of power through a solar array and be able to store the required amount of power for when the sun is not in view. This will be dependent on when the polar orbit does not cross between the Moon and the Sun, meaning that the duties being performed while the spacecraft is out of sunlight may vary and will need to be a design consideration for the subsequent iterations that describe the functions and requirements of the subsystems in more detail.

To produce this much power an estimated value of 100 W/m^2 of solar panel can be used to find the area of required solar panels [28]. This means that 24.2 m^2 of solar panels will be needed to produce the required power for the spacecraft. Since the shape of the spacecraft is unknown, it is unlikely that solar panels on the surface of the craft will be able to support the amount of power used by the spacecraft. This means a folding array of solar panels will need to be constructed and deployed. The size and shape of these solar arrays will determine the shape of the satellite in the payload fairing of the launch vehicle. A single square solar array would need to be 5 m x 5m to produce the required electricity. Two arrays, which could be folded toward the spacecraft more simply within the dimensions of a conventional fairing would need to be 2.5 m x 2.5 m each. Ultimately the decision for what shape and size the solar arrays will need to be will depend on the configuration of the spacecraft, the size limitations of the launch vehicle, and the practical tradeoff between how complex the folding mechanisms must be and the space those more complex mechanisms will save in the fairing.

6.4 Considerations for Future Iterations

The previous sections have described the top-down method of organizing and understanding basic estimates for characteristics of the spacecraft. Future work done on the specific subsystems of the satellite will give more specific limitations for the spacecraft's size and configuration. These bottom-up component design requirements will be part of the iterative process and will require the space system as a whole to change in response to new challenges and requirements driven by subsystem specifications [28].

Other decisions will need to be made that will affect the estimates made here. Changes will be made to these specific subsystems and the ones that have not been studied yet. Decisions will need to be made regarding the autonomy of the spacecraft, since it will need to be able to operate on the far side of the Moon for periods of time. The degree to which the spacecraft acts autonomously will be an important decision to make when the command and data handling subsystem's requirements have been determined in more detail. Increasing the autonomy of the satellite will give the spacecraft more ability to operate on the far side of the moon performing operations like processing data. However, increasing this autonomy means that more points of failure exist and less of the spacecraft's functions can be corrected on Earth.

Other subsystems, such as the structural and thermal systems, will be determined when the configuration of the spacecraft has been solidified and more information on the operating environment can be studied. The affect these subsystems will have on the weight, power and other mission specifications for the satellite will be important and will need to be integrated into the iterative process as soon as is possible due to the affect they have on the budgets for these requirements.

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Appendix A – Satellite Weight Budget

Table A.1 Weight and percent of budget of spacecraft components.

Component	Weight (kg)	Percentage
Payload	861	17.30%
Propellant	2115.01	42.67%
Other Dry Mass	1980.3	39.96%
Total	**4956.3**	**100%**

Table A.2 Power and percent of budget of spacecraft components.

Component	Power (W)	Percentage
Payload	968	40%
Propulsion	121	5%
Other Subsystems	1331	55%
Total	**2420**	**100%**

Appendix B – Payload Optical and Data Specifications

Table B.1 Specifications for optical and data properties of the payload.

Property	Value
D_{max}	133.89 km
S	1.036 deg
Swath Width	8.06 deg
θ_r	0.1 deg
IFOV	5.7296×10^{-4} deg
Z	7.3689×10^{8} pixels
Ti	3.47 ms
Fp	287.5 kHz
f	3 m
D	1.0248
F#	2.9274
FOV	1.4668 deg