Radio Astronomy on the Moon

Mission Proposal By:

Aerospace Engineering Technologies Heading Extrasolar Research

AETHER

Project Manager
Joel Grissom

Mission Manager
Dr. Matthew Turner

See Appendix J.2

Principal Investigator
Heather Meyer

Department Chair of Physics and Astronomy- College of Charleston
Dr. Jon Hakkila

See Appendix J.2

A. Proposal Summary Information

A.1 Section I

A.1.1 The Principal Investigator

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A.1.1 Proposal's title

Radio Astronomy on the Moon
Proposed start date: Aug 15, 2017
Proposed end date: Sept 15, 2022

A.2 Section II

A.2.1 Date Submitted

April 25th, 2011

A.3 Section III

A.3.1 Proposing Organization

The University of Alabama in Huntsville (UAHuntsville)

A.1.1 Division/Department

Aerospace Engineering Technologies Heading Extrasolar Research (AETHER)
Mechanical and Aerospace Engineering Department

A.3.2 Mailing Address

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A.4 Section IV

A.4.1 Proposal Point of Contact

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A.5 Section V

A.5.1 Certification of Compliance with Applicable Executive Orders and U.S. Code

By submitting the proposal identified in the Cover Sheet/Proposal Summary Information in response to this Research Announcement, the Authorizing Official of the proposing organization (or the individual proposer if there is no proposing organization) as identified below:

- certifies that the statements made in this proposal are true and complete to the best of his/her knowledge;
- agrees to accept the obligations to comply with NASA award terms and conditions if an award is made as a result of this proposal; and
- confirms compliance with all provisions, rules, and stipulations set forth in the two Certifications and one Assurance contained in this NRA (namely, (i) the Assurance of B-4 Compliance with the NASA Regulations Pursuant to Nondiscrimination in Federally Assisted Programs, and (ii) Certifications, Disclosures, and Assurances Regarding Lobbying and Debarment and Suspension). Willful provision of false information in this proposal and/or its supporting documents, or in reports required under an ensuing award, is a criminal offense (U.S. Code, Title 18, Section 1001).

A.5.2 Authorizing Officials

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A.6 Section VI

A.6.1 List of Team Members

Team Member	Member Role
The University of Alabama in Huntsville	
Joel Grissom	Project Manager
Garrett Gammon	Chief Engineer
Sam Bennett	Lead Systems Engineer
Megan Beattie	Supporting Engineer
Jamison McAllister	Supporting Engineer
David Moore	Supporting Engineer
Clayton Pannell	Supporting Engineer
James Pearson	Supporting Engineer
Matthew Wright	Supporting Engineer
Brittany Gibbs	Technical Editor
Maria Munn	Technical Editor
College of Charleston	
Heather Meyer	Principal Investigator
Ryan Wilkie	Co-Investigator
Jesica Trucks	Co-Investigator
Austin High School	
Lindsay Boonarkat	Industry Partner
Trevor Montgomery	Industry Partner
Phillip Betts	Industry Partner
Sean Goodman	Industry Partner
ESTACA	
Lucas Schoukroun	International Partner
Antoine Bercault	International Partner
David Langlois	International Partner

A.7 Section VII

A.7.1 Proposal Summary

In response to the NASA Discovery Announcement of Opportunity (AO) regarding Radio Astronomy on the Moon, Aerospace Engineering Technologies Heading Extrasolar Research (AETHER) will utilize two Atlas V 551 launch vehicles to establish an array of radio astronomy instruments on the far side of the Moon. AETHER consists of engineers at The University of Alabama in Huntsville; engineers at ESTACA University in Paris, France; scientists at The College of Charleston; and Level 2 Innovative Student Project for the Increased Recruitment of Engineering and Science Students (InSPIRESS) high school students. The Atlas V 551 launch vehicles will contain one orbiter, one lander, and one solid rocket motor of identical design. The landers will have different science payloads. One lander will contain four dipoles arranged in a precise array. The other lander will have four radio telescopes in the same configuration. Information gathered from the experiment will result in a better understanding of the Universe around us.

A.8 Section VIII

A.8.1

There is no proprietary or privileged information included in this application.

A.8.2

This project does involve activities outside the United Sates. The AETHER team includes ESTACA University, who is designing the two orbiters. These orbiters will be identical to one another and each orbiter will be utilized in each launch vehicles. The orbiters exist to capture data received by the radio arrays and transmit that data to Earth.

A.8.3

NASA civil servant personnel will participate as team members for the project.

A.8.4

The proposed project has no an actual or potential impact on the environment.

A.8.5

The proposed project does not have the potential to affect historic, archeological, or traditional cultural sites (such as Native American burial or ceremonial grounds) or historic objects (such as an historic aircraft or spacecraft).

A.9 Section IX

A.9.1

Radio Astronomy on the Moon (RAM)

A.9.2

The University of Alabama in Huntsville

A.9.3

Target of investigation: Astronomical phenomena

A.9.4

AETHER proposes utilizing two Atlas V 551 launch vehicles in the projected mission. These two Atlas V 551s will be identical payloads.

A.9.5

The use of NEXT is not proposed.

A.9.6

The use of AMBR is not proposed.

A.9.7
The use of aerocapture is not proposed.

A.9.8
The use of ASRG is not proposed.

A.9.9
There is no use of radioisotope heater units, or radioactive material sources for science instruments proposed.

A.9.10
Student collaboration (SC) is proposed.

A.9.11
Science enhancement option (SEO) is proposed.

A.9.12
ESTACA: $438.16 million

A.9.13
PI-Managed Mission Cost in FY 2010 dollars is $1.40 billion.

A.9.14
Total Mission Cost in FY 2010 dollars is $1.40 billion.

A.9.15
The NASA Personnel Full Time Equivalent (FTE) information for FY10 is 104 people. These people should be utilized throughout the entire mission life cycle.

A.9.16

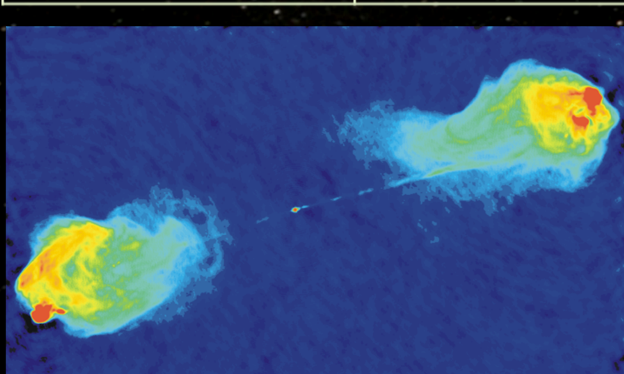
A.1.1.1
The proposal does not contain any information and/or data that are subject to U.S. export control laws and regulations, including Export Administration Regulations (EAR) and International Traffic in Arms Regulations (ITAR).

A.9.16.1
AETHER acknowledges that the inclusion of such material in this proposal may complicate the Government's ability to evaluate the proposal.

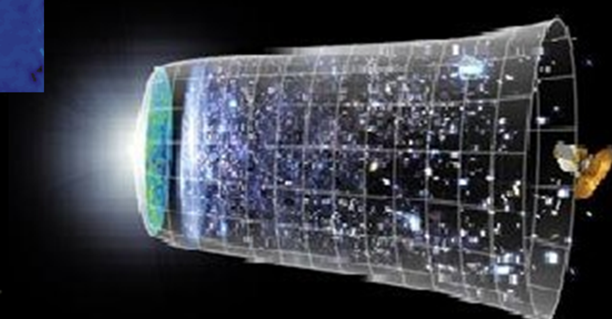
B. Fact Sheet
On the following two pages.

Radio Astronomy on the Moon

Science Goals	Science Objectives
To observe sources of radio emission that will advance scientific knowledge of the origins and evolution of the universe.	To collect data from observing pre-main sequence stars and protostars, the red-shifted neutral hydrogen line, the interstellar medium, quasars, and radio galaxies.
To observe solar phenomena and the terrestrial magnetosphere in order to further our understanding of the space environment, including potential hazards to humans.	To collect data from solar observations.



Above: Far Side of the Moon
Left: Radio galaxy Cygnus A
Below: Timeline of Universe



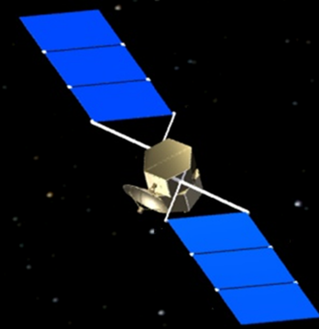
Instrumentation:
Array of Radio Telescopes
Array of Radio Dipoles
Lunar Ejecta and Meteorites experiment

UAHuntsville
THE UNIVERSITY OF ALABAMA IN HUNTSVILLE
COLLEGE of CHARLESTON
AUSTIN HIGH SCHOOL DECATUR 1962 ALABAMA
ESTACA
ÉCOLE D'INGÉNIEURS

Mission Management:
Project Manager- Joel Grissom
Principal Investigator- Heather Meyer
Chief Engineer- Garrett Gammon
Lead Systems Engineer- Sam Bennett

Total Mission Cost: $1.40 Billion
PI-Managed Mission Cost: $1.40 Billion

Radio Astronomy on the Moon



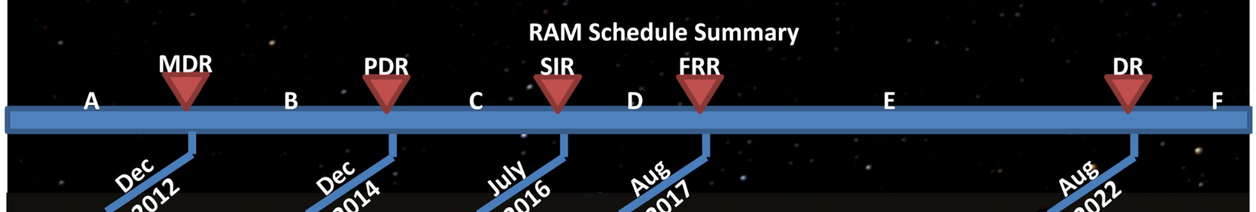
Elements per Launch Vehicle:
1 Data Relay Orbiter (DRO)
1 Radio Astronomy Instrument Lunar Lander (RAILL)
1 STAR 48V Solid Rocket Motor
3 ATK CoilABLE Booms

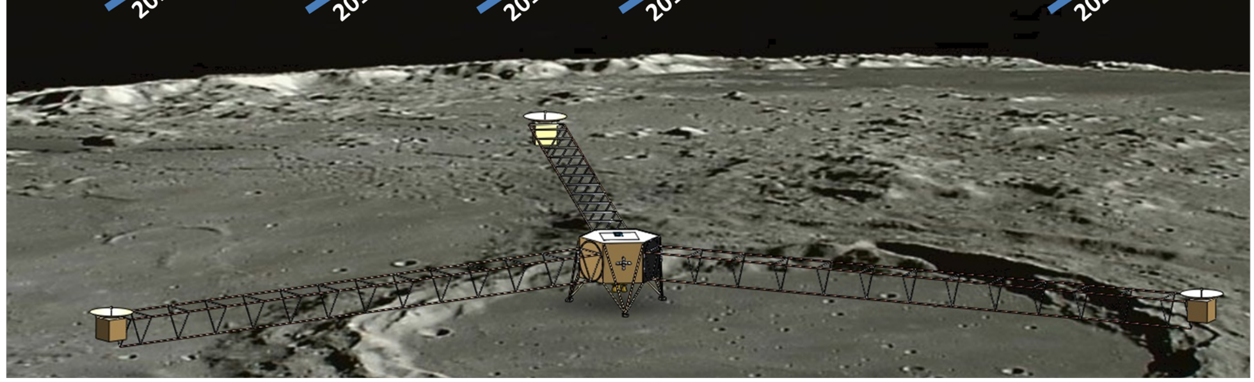
Overview :
Two Atlas V 551 Launch Vehicles will place an array of radio astronomy instruments at the Kholscutter and Aitken craters on the far side of the Moon.

Key Events per Launch Vehicle:
1. Launch from Cape Canaveral
2. Upper stage of Atlas V 551 performs TLI burn
3. DRO performs MCC and LOI burn
4. DRO separates from RAILL and STAR 48V
5. RAILL performs DOI,
6. Star 48V performs a Braking burn and separates
7. RAILL lands on lunar surface
8. ATK Booms deploy to create a radio array
9. Scientific data is relayed through DRO to Earth

Autonomous Precision Landing: RAILL will utilize Autonomous Landing and Hazard Avoidance Technology (ALHAT). This will be a technical demonstration that could prove planetary landings can be made safe and precisely.

Above: Model of DRO
Bottom: RAILL with deployed booms on the lunar surface





C. Table of Contents

D. Science Investigation

D.1 Scientific Background, Goals, and Objectives

D.1.1 The Moon

Before the dawn of the space age, very little was known about the Moon. What was known was limited to what could be seen from the surface of the Earth. Figure 1 shows the near side of the Moon, which is the same side that faces the Earth at all times. Using the naked eye, anyone can see that the Moon's surface is dotted with dark and light spots, called maria, or "seas," because it was thought that the dark areas were water. With the use of telescopes, pre-modern astronomers were able to identify major impact features and bright, hilly highland features. Though these were major discoveries for their time, the evolution of space technology has revealed more about the Moon than can have been imagined previously.

Figure 1 The Near Side of the Moon

From data collected by the Russian Luna missions and NASA's Apollo missions, the truth about the lunar maria and the highlands came to light. Maria are actually ancient basaltic lava flows. Their dark color comes from high concentrations of iron. Maria make up around 16% of the lunar surface and are highly concentrated on the near side. Maria are very rare on the far side of the Moon. The maria provide an excellent landing platform because they are generally smooth and solid. On the other hand, the lunar highlands are coated in meters of light-colored lunar regolith, and as such, they constitute extremely rugged terrain. The highlands are the most ancient areas of the lunar surface. These heavily bombarded and churned up surfaces are composed of anorthosite, a calcium-rich rock. The far side of the Moon is dominated by highland regions.

The nature of the lunar rotation provides us with a natural haven for observations, particularly at wavelengths that are too weak to be properly studied on Earth. The Moon's rotation is approximately equal to its revolution around the Earth. Therefore, the Earth always faces the same side of the Moon. Because of this rare phenomenon, the far side of the Moon escapes the noise and interference of man-made signals, providing a fairly stable, radio-quiet area for observation. Observing from the far side of the Moon has many advantages over Earth-based telescopes. First, the Moon has little or no magnetic field, and therefore cannot trap solar particles. So, in theory the Moon would provide shielding to instruments, unlike space-based telescopes. Second, there is no atmosphere to weather and destroy the telescopes themselves. No atmosphere also means no atmospheric extinction, which is something that astronomers on Earth deal with constantly. Also, at orbital velocity, some residual O_2 impacting structures can excite faint emissions that can contaminate sensitive observations. To avoid this, orbiting telescopes must be pointed downwind. Third, the Moon provides a stable thermal environment, allowing the telescopes more time to thermally equilibrate and preventing tracking errors. Next, the Moon's gravity, at one-sixth that of Earth, makes it easier to deploy and operate a large telescope. Its gravity is just enough to assist with vibration dampening. In addition to its gravity, the slow sidereal rate of the Moon provides astronomers with approximately thirty times more time to observe an object crosses the celestial sphere, making tracking requirements minimal. This extremely long observation time coupled with a 14-day long night period will enable scientists very high integration times to gather faint radiation and to make deep field extra-galactic observations. Perhaps the most important factor to mention is that of radio quiet. All other radio telescopes are subject to man-made interference, whereas the far side of the Moon is free of it. Finally, a lunar array on the far side would be both long-lived and accessible to humans if updates were needed. The ability to study radio waves on the far side of the Moon would certainly provide astronomers with an enormous amount of unique data for interpretation.

D.1.1 Radio Astronomy

The Dark Ages denote a period of time between two major epochs in the universe's timeline: recombination and reionization. Recombination is thought to have occurred around 400,000 years after the big bang. Reionization is thought to have started a few billion years after recombination. Figure 2 shows a blank space near the beginning, between the microwave background radiation and the epoch of reionization. This blank space is the Dark Ages.

To understand why the Dark Ages are important, we must start by explaining what was happening in the universe before this time.

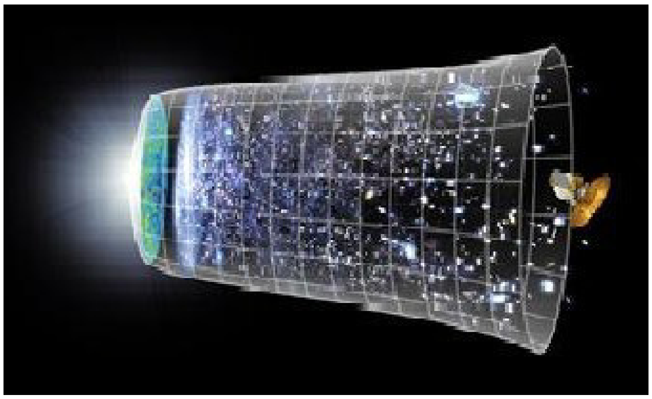


Figure 2 This pictorial representation of the timeline of the universe was created by the NASA/WMAP Science Team.[1]

After most of the matter formed in the universe, the universe was extremely hot, causing the protons and electrons to be too energetic to bond. Because of this "primordial soup" of plasma, the photons' movement was hindered. Photons wouldn't be able to travel very far before colliding with electrons (the Thomson effect). It is because of this that the early universe would have seemed opaque. After the universe relatively cooled to about 3,000K from expansion, the electrons got caught in orbits around the protons, forming the first atoms, neutral hydrogen (See Figure 3). This is the start of recombination.

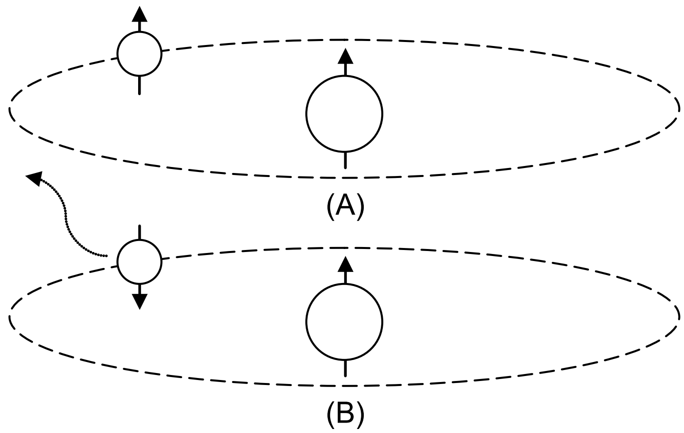


Figure 3 (A) The aligned spin of neutral hydrogen. (B) The antialigned spin of neutral hydrogen, letting of radiation in the transition

The two lowest energy states of neutral hydrogen are (with the electron orbiting on the lowest level) the electron having a spin up or a spin down attribute. The lower of the two energy states occurs when the electron's spin is antialigned with the proton's spin. These neutral hydrogen atoms would get hit by photons and transition from the antialigned state to the aligned state. When the neutral hydrogen transitioned back to the antialigned state, they gave off radiation at a wavelength of 21cm (See Figure 3). Because the energy given off was so low, this era has been aptly named the Dark Ages. The probability that the hydrogen transitions from the aligned state to the antialigned state is calculated to be extremely rare, but this is not a hindrance, for there is such a great amount of hydrogen making up the universe that the transition happens frequently overall.

As the universe continually expanded and cooled, the neutral hydrogen began to coalesce. This is the start of reionization. It is at this time in which enough matter had clumped together due to gravity to start forming larger objects, like stars, galaxies, and black holes. The increased radiation from gravitationally bound atoms was enough to increase surrounding atoms to higher energy states, usually ionizing the atoms. "Bubbles" of this matter formed here and there, but spread quicker over time due to the radiated energy traveling through the universe and interacting with matter in other places. The Dark Ages ended when reionization began.

There are many sources of radio emissions that are also of interest to scientists. Radio galaxies and their close companions, radio-loud quasars, are active galaxies that are very bright at radio wavelengths. These radio emissions generate from synchrotron radiation, fast moving electrons moving along magnetic field lines. These electrons travel along a curved path at relativistic speeds, they experience acceleration as long as their motion is perpendicular to the field, and therefore emit electromagnetic radiation (Ginzburg et al 1965). The radio emissions reveal a structure to the system that includes jets and lobes of irradiated gas (See Figure 4). These regions of ionized gas can be up to distances as far as megaparsecs away. The jets are made up of particles moving at relativistic speeds. These high speeds cause a beaming effect. Radio galaxies are very bright and can be found at very high redshifts (z). These galaxies are generally large elliptical galaxies. They can be useful in observational cosmology, used to see the early evolution of the universe.

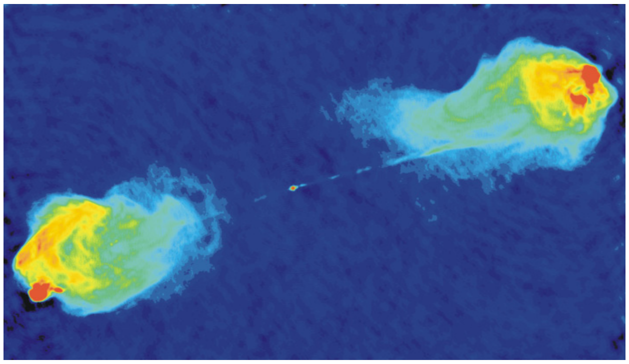


Figure 4 The hyperluminous radio galaxy Cygnus A; this image shows in false color the radio jet and lobes of the galaxy[2]

Protostars are also observable in radio. A protostar is a large mass that forms out a giant molecular cloud. The molecular cloud starts out at virial equilibrium; the gravitational energy and thermal energy are at equilibrium. In order for the cloud to collapse into a star the gravitational potential energy has to be twice the thermal energy. A protostellar cloud will radiate light as it gravitationally collapses. During this period there would be weak activity in the radio. Looking for these compact radio emissions from protostars we would be able to accurately locate these emerging star systems. We can also create false color images from the data, like Figure 5 which is a radio image of a dense molecular cloud in the process of star formation.

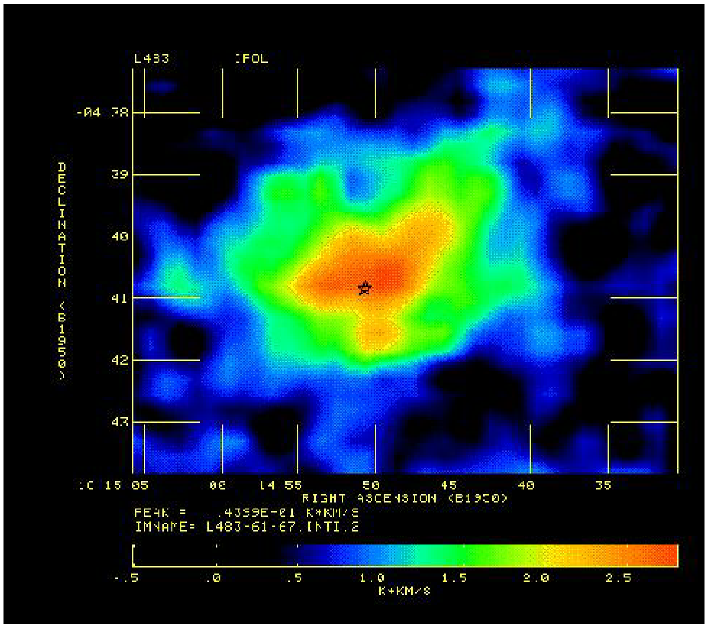


Figure 5 An image taken with the VLA of a dense dark molecular cloud (L483 Molecular Cloud) that is forming a massive protostar.[2]

By accurately locating these objects, they would then be able to be studied in the infrared using adaptive optics, which requires a high precision accuracy due to the small field of view. By observing these types of objects, we get a chance to understand more about stellar and planetary formation.

This mission will provide valuable information that is otherwise not easily collected. Observing the 10 to 15 meter wavelength range will be easier on the far side of the moon, because man-made radio interference will not be a problem there. Also, the moon has no ionosphere to filter or block completely (like on earth) the radio wavelengths we would like to study. The sensitivity is highly important, for the radio waves to be studied are of weak emission. This mission, if done, will be able to provide information never before seen about our early universe. Once the mission has played out, we will have knowledge about this subject in more detail than before, and, if a future of even more valuable information looks promising, the observation sites can be built upon. The following table outlines the overarching Science Goals and Objectives for this mission.

Table 1 Science Goals and Objectives

Science Goals	Science Objectives
To observe sources of radio emission that will advance scientific knowledge of the origins and evolution of the universe.	To collect data from observing pre-main sequence stars and protostars, the red-shifted neutral hydrogen line, the interstellar medium, quasars, and radio galaxies.
To observe solar phenomena and the terrestrial magnetosphere in order to further our understanding of the space environment, including potential hazards to humans.	To collect data from solar observations.

D.2 Science Requirements

Getting the mass power spectrum will allow us to see a three-dimensional view of the universe. This will give us a more detailed view than Wilkinson Microwave Anisotropy Probe (WMAP). We will also be able tell where the first stars and galaxies started forming during reionization. This will allow us to better understand their formation and life cycle. Although we will observe the 21cm hydrogen spin flip emission, we need to consider the cosmological redshift. The redshift

$$z = \frac{v_o}{v_e} - 1$$

for the Dark Ages will be1100 ≥ z ≥ 6, where ν_o is the observed wavelength and is the emitted wavelength. Table 1 shows the comparison of recombination and reionization.

Table 2

	Recombination	Reionization
z	1100	666
Time Ago (yrs)	~13.7 billion	~12.8 billion
ATB (yrs)	372000	951 million
Comoving Distance (Mpc)	14000	8450
Fraction of Observable Radius	.98	.59
Scale Factor(current value)	9.8×10^{-4}	0.1430
Radiation Temperature (K)	3000	19.1
Expansion Rate (km/s/Mpc)	1.56×10^{6}	686
Comoving Volume Within Redshift (Mpc^3)	1.16×10^{13}	1.52×10^{12}
Angular Diameter Distance (Mpc)	12.7	1210
Luminosity Distance (Mpc)	1.55×10^{7}	59100
Observed Wavelength (m)	231.2	1.47
Frequency (MHz)	1.297	203.9

To get a three-dimensional image, we will observe a small range of this redshift – in this case, 70.43 ≥ z ≥ 46.62. This range is compared in Table 2.

Table 3

	15 meter wavelength	10 meter wavelength
z	70.43	46.62
Time Ago (yrs)	~13.7 billion	~13.7 billion
ATB (yrs)	28.5 million	52.8 million
Comoving Distance (Mpc)	12700	12200
Fraction of Observable Radius	.884	.854
Scale Factor(current value)	0.0140	0.021
Radiation Temperature (K)	195	130
Expansion Rate (km/s/Mpc)	22500	12200
Comoving Volume Within Redshift (Mpc^3)	8.49×10^{12}	7.65×10^{12}
Angular Diameter Distance (Mpc)	177	257
Luminosity Distance (Mpc)	904000	582000
Frequency (MHz)	20	30

These wavelengths are chosen because of their closeness in time to reionization, helping us to see how it started. They are also chosen because they are easier to sense than the longer wavelengths which appear from farther back in time and have a higher redshift.

There are several sources of radio emission in addition to neutral hydrogen that are poorly understood and would benefit from an array of radio telescopes on the far side of the Moon where there is no interference from man-made signals. Quasars, pulsars, protostars, pre-main sequence stars, and some solar phenomena will be observed at wavelengths extending from 1mm to 10 m. The addition of these wavelengths requires an array of 3 total parabolic radio telescopes, with dishes of wire mesh, at one site and an array of 4 radio dipoles at another site no closer to the first than 100 km for observation. Each array must form an equilateral triangle with each radio telescope 30m from the other two. Beyond 30 m, the positioning of the telescopes is not of the utmost importance as long as their precise location is known. The wire mesh of the parabolic dish can have gaps no larger than 1/10th of 1 mm. A pointing accuracy of 1 arcsecond or better is required to make these observations.

The movement of regolith across the surface of the far side has never been studied. To do so, a sensor with at least 3 directions of detection and a means of auto leveling is needed to observe the movement of lunar regolith and dust particles over time. The shift from night to day is of particular interest due to the interaction of the charged surface with the Earth's magnetosphere.

The success of these observations depends on their placement. As previously mentioned, observation of the Dark Age neutral hydrogen line requires a stable environment free of manmade interference. The far side of the Moon, being permanently shielded from Earth's transmissions, provides a radio quiet location for observation. The lunar maria provide a perfect landing site because they are solid, smooth, and are not entirely covered with regolith. Though maria are rare on the far side of the Moon, there are a few large craters where these basaltic flows may be found. In addition to a stable, shielded platform, proximity to a location containing regolith is needed in order to study the interaction of that charged regolith with the Earth's magnetosphere. Two sites are suggested for landing on the far side of the Moon: Aitken and Kohlschütter craters (Figure 6). These two sites were chosen based on their large diameter, flat bases with small amounts of regolith nearby for study.

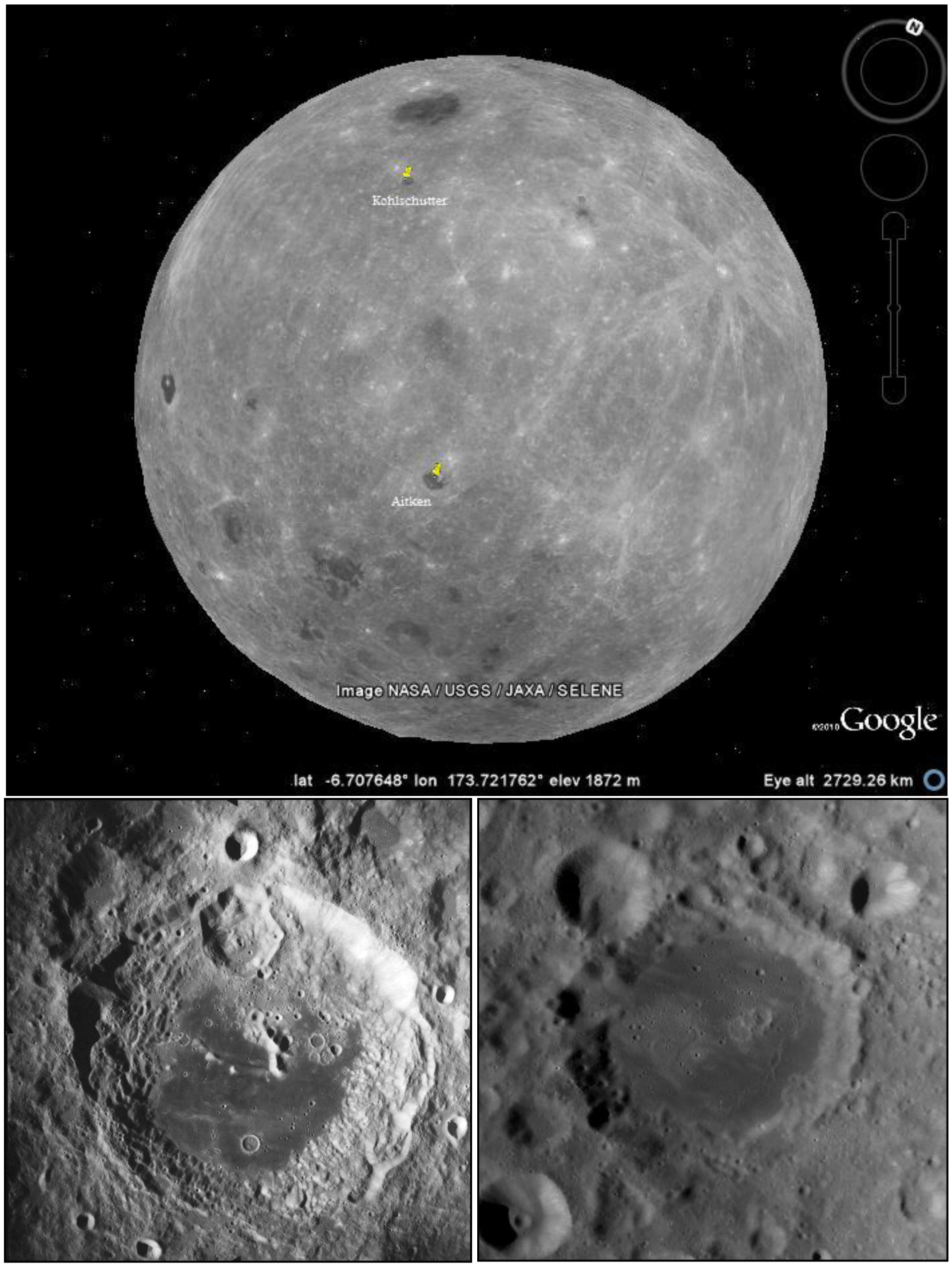


Figure 6 The lunar far side, with close-ups of Aitken crater (left), Kohlschütter crater (right) [3]

Table 4 Science Traceability Matrix

Science Goals	Science Objectives	Scientific Measurement Requirements: Observables	Scientific Measurement Requirements: Physical Parameters	Instrument Performance Requirements		Projected Instrument Performance	Mission Requirements
To observe sources of radio emission that will advance scientific knowledge of the origins and evolution of the universe.	To collect data from observing sources of radio emission.	Emission Line: Redshifted Neutral Hydrogen	Density and Temperature of Emitter	Operating Wavelength Range	λ=10m	λ=1mm-10m	Observing Strategies: Requires two sites with at least three telescopes per site and tracking maneuvers
				Gain/Resolution	TBD	TBD	
				Range of Motion	160˚	160˚	
		Emission Line: Protostars		Operating Wavelength Range	λ=1mm-1cm	λ=1mm-30m	Observing smaller wavelengths requires gaps in the dish material no larger than 1/10th of the wavelength.
		Emission Line: Quasars		Operating Wavelength Range	λ=>1mm	λ=1mm-30m	
		Emission Line: Pulsars		Operating Wavelength Range	λ=3.5cm-7m	λ=1mm-30m	
To observe solar phenomena and the terrestrial magnetosphere in order to further our understanding of the space environment, including potential hazards to humans.	To collect data from observing solar phenomena.	Emission Line: The Sun		Operating Wavelength Range	λ=12-21m	λ=1mm-30m	
	To collect data from observations of the interactions between solar particles, the lunar surface, and the terrestrial magnetosphere.	Energetic Particle Spectrum	Type of Particle	Energetic Ions	5-270 MeV/nucleon	5-270 MeV/nucleon	Observing Strategies: Requires Observation While Facing the Sun
				Protons, Helium	4.3-100 MeV/n	4.3-100 MeV/n	
				Neutrons	2-100 MeV	2-100 MeV	
				Electrons	150 keV-15 MeV	150 keV-15 MeV	
				X-Rays, Gamma Rays	<1.5 MeV	<1.5 MeV	

			Size of Viewing Area	Field of View	36.7 degrees	36.7 degrees	
			Collection Time	Time Resolution	1 hour [1 min]	1 hour [1 min]	
			Amount of Recorded Events	Events Rate	5000 events/s	5000 events/s	
To study the lunar far side regolith to better understand lunar surface processes and the interaction of the space environment with the lunar surface.	To observe the movement of lunar regolith across the far side.	Particle speed, direction, momentum, and kinetic energy	Size of Particle	Sensitivity	>2 microns	>2 microns	
			Location of event	Directivity	3 directions/ 3 sensors	3 directions/3 sensors	
			Amount of Recorded Events	Events Rate	4 per month	4 per month	Need two years of observation to observe variability of the phenomenon.
			Relative orientation of event to observer	Autolevel	Full 360˚	Full 360˚	

D.3 Threshold Science Mission

The main goal for this mission is to observe radio emissions from the time of the Dark Ages in the early universe from the far side of the Moon. Observing emission from this early point in time will help us build a picture of the matter power spectrum of the Dark Ages and help us build a picture of how the universe started to reionize. In order to accomplish the science objectives, an array of radio dipoles must be placed on the far side of the moon in Aitken crater. Observation must take place on the far side of the Moon because it is shielded from man-made radio transmissions, and it provides a stable, semi-permanent platform for observation and for future additions, if deemed necessary. Aitken crater (Figure 6) was chosen because it contains a large, nearly smooth bottom measuring over 100 km.

This site will require a series of at least four stacked dipoles in an area of at least $1257m^2$. The dipoles will be set out by booms and unfolded. They will be about one half-wavelength, or 5 meters, in size after deployment. With these dipoles, the redshifted 21cm neutral hydrogen spin flip emission line will be observed with a resolution of 1″ or better, with about the same precision. No tracking is necessary, for the dipoles view the whole sky. Two orbiters are necessary for efficient communication between the landing site and Earth.

The data collected will be sent to Earth to be analyzed and put together in an orderly fashion. Out of the collected data, the mass power spectrum of the Dark Ages can be put together and the data will be analyzed to show where reionization first started to occur.

D.4 Science Traceability

The following science traceability profile corresponds to Table 3: Science Traceability Matrix.

The first science goal: to observe sources of radio emission that will advance scientific knowledge of the origins and evolution of the universe. The main objective in this goal is to collect data from observing sources of radio emission. The observables in this category will be the emission lines of the following: redshifted neutral hydrogen, protostars and the sun. The instrument performance requirements are wavelengths of 1mm up to 21m, with a collecting area of $1257m^2$, with 160° range for motion. Our projected instrument performance is wavelength of 1mm to 30m with a collecting area of $1257m^2$ and range of motion of 160°.

The second science goal is to observe solar phenomena and the terrestrial magnetosphere in order to further our understanding of the space environment, including potential hazards to humans. The first objective is to collect data from observing solar phenomena by observing the energetic particle spectrum from wavelengths of 12-21m, with our projected instrument performance being able to observe wavelength from 1mm-30m. The second objective is to collect data from observation of the interactions between solar particles, the lunar surface, and the terrestrial magnetosphere. The observable in this objective would be the particle speed, direction, momentum, and kinetic energy. The types of particles observed being energetic ions, protons, helium, neutrons, electrons, x-rays, and gamma rays operating on particle energies from

150keV-270MeV per nucleon. Operating over the projected instrument performance of a field of view of 36.7°, time resolution of 1 hour [1 min], event rates of 5000 events/s.

The final science goal of to study the lunar far side regolith to better understand the lunar surface processes and the interaction of the space environment with the lunar surface by observing the movement of the lunar regolith across the far side. This would be done with a sensitivity of particles greater than 2 microns, directivity of 3 directrions/3 sensors, event rate of 4 per month, and a relative orientation to the observer an autolevel that would cover 360°. We would need 2 years of to observe the variability of this phenomenon.

E. Science Implementation

E.1 Instrumentation

E.1.1 Radio Telescope Array

There is no model for a radio telescope array for operation in space. All previous extraterrestrial radio observations have been made using space-based telescopes. That being said, all of the parameters for these arrays come from model telescopes operating on Earth and telescopes operating in space. Some adjustments have been made to compensate for the change in environment. Each 3m telescope will be 100kg including the receiver and mount. The dish will be composed of a metal mesh with gaps no larger than $1/10^{th}$ of a millimeter. The dipoles will be much smaller at one half wavelength long and 20kg each including the dipole itself and the bar mount. The viewing direction will change from objective to objective for the 3m telescopes, but it is expected that they will have access to the entire sky at some point. Tracking ability is required. The dipoles will have full-sky observing capabilities, requiring no tracking. The 3 meter telescopes will have a pointing accuracy of 0.01 arcseconds. Both the dipoles and the telescopes must be able to withstand temperatures ranging from +265F to -170F. The electronics corresponding to the lander and the telescopes must be protected against radiation levels of 10 mSv per year. The risk of contamination is minimal.

E.1.2 LEAM

The Lunar Ejecta and Meteorites experiment, or LEAM, was selected for this mission because it has been used in the past to study dust and impact particles on the Moon. We know that there are major shifts in the distribution of the regolith thanks to LEAM's previous work, and to the astronaut's that sketched the strange phenomena they saw while orbiting the Moon (Figure 5). LEAM was first used on the Apollo 17 lunar module to measure the frequency with which the lunar surface was impacted by cosmic dust particles and the effect of lunar ejecta emanating from sites of meteorite impacts.[4] LEAM had six primary science objectives for the Apollo 17 mission which could greatly benefit from new information: (1) To determine the background and long-term variations of cosmic dust influx rates in cislunar space (2) To determine the extent and nature of lunar ejecta produced by meteorite impacts on the lunar surface (3) To determine the relative contribution of comets and asteroids to the earth's meteoroid ensemble (4) To study possible correlations between the associated ejecta events and the times of the earth's crossing of cometary orbital planes and meteor streams (5) To determine the extent of the contribution of interstellar particles toward the maintenance of the zodiacal cloud as the solar system passes through galactic space (6) To investigate the existence of an effect called 'earth focusing of dust particles.'

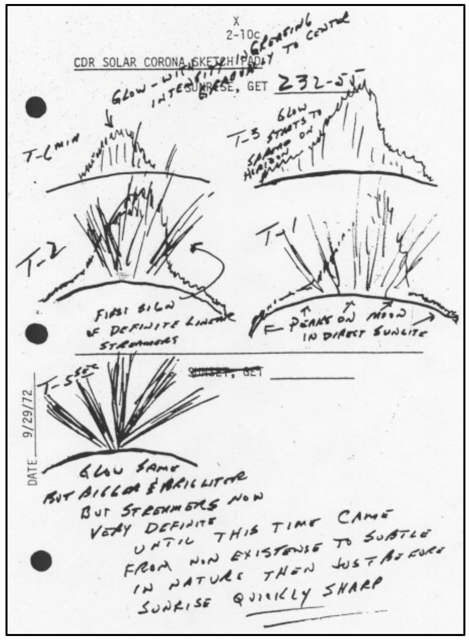


Figure 7 Sketches by Apollo 17 astronauts of immense clouds of dust at sunrise.

LEAM has three sensors which are well suited to our proposed investigation of migrating, charged regolith: the finest grains of which correspond in size to particles that LEAM was originally designed to measure. The sensors have a field of view of plus or minus 60 deg. They measure particle impacts in an energy range of 1 to 1000 ergs with a primary frequency of measurement of 1.E-4 impacts/sq m/s. The unit should be aligned to plus or minus 5 deg of the scan-shadow line and leveled to plus or minus 5 deg. Table 5 provides the estimated performance characteristics.

The equipment for this experiment, which was part of the Apollo 17 ALSEP, includes one deployable unit with detector plates, ALSEP central station electronics, and the cable and astromate connector for mating the external unit with the central station. The external unit components or sensors consist of suppressor and collector plates, impact plates, film frames, and microphones. Figure 6 is a basic schematic of the instrument. Originally, the external unit was erected and deployed on the lunar surface about 8 m south of the ALSEP central station; however, this instrument can be mounted to the lander if necessary as long as the detectors are externally located within view of the lunar surface.

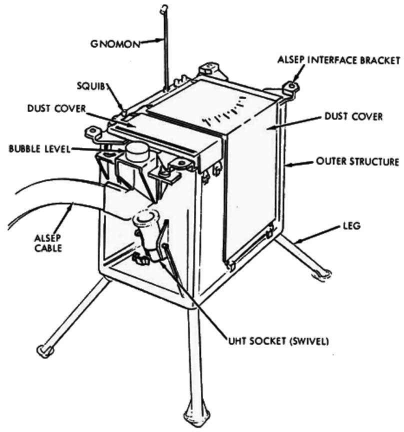


Figure 8 LEAM instrument sketch

LEAM requires minimal adjustments to make it a viable instrument for this mission's SEO. It must be thermally insulated to protect the instrument from temperature fluctuations on the far side ranging from +265F to -170F. The angular resolution, originally plus or minus 26 degrees, must be increased to plus or minus 10 degrees by updating the detectors.
No development plan is needed because LEAM is already TRL 6. Radiation is a concern for this instrument, but will not have a significant effect on its operation over the course of this mission. No onboard data processing is necessary.

Table 5: Estimated performance characteristics:

Mass	7.4 kg	**Angular Resolution**	10 degrees
Power	6.6 W	**Vertical Alignment**	+/- 5 degrees
Volume	0.01948 cubic meters, stowed	**Horizontal Alignment**	+/- 5 degrees
Data Rate	33.1 bits/sec	**Field of View**	+/- 60 degrees
Thermal	-170F to +265F	**Standby Power**	3 W

E.2 Data Sufficiency

In order to achieve the science objectives and goals, observations must be collected over a period of three years. The length of this mission provides for long exposure times for collecting radio waves. This data will be stored on the lander until such time as one of the two orbiters passes overhead. Expected real time data transmission: 900 Kbit/sec, Data volume per day: 1 Gigabit. Though no data processing will take place on the lander or on the orbiters, the integration of this data will require a substantial effort on the part of the Mission Specialist.

E.3 Science Mission Profile

Since the lunar radio telescope is to be placed on the far side of the moon, there is no direct radio contact with Earth possible. Hence, we need one or multiple additional satellites to be able to communicate with the earth. Due to the specific landing sites necessary for the success of this mission, two orbiters will be employed. Observations will take place for a period of 24 hours at a time, in order to provide sufficient exposure time. Observing the entire sky with the dipole array, or even a 2 arcsecond slice of it with both arrays, produces massive amounts of data; therefore, data must be dumped 14 times per day.

E.4 Data Plan

Within six months of the first observations, the raw data that was be relayed to Earth will be integrated, validated, and preliminary analysis will begin. The retrieved raw data will be archived beginning no later than six months after the first data is retrieved. All data, once preliminary analysis is complete, will be archived and accessible to the public no later than one year after the start of data collection. The Co-Investigator and Mission Specialist shall assume all responsibility for the software/hardware technologies required to work with the observed spectra. Out of the collected data, the mass power spectrum of the Dark Ages can be composed and the data will be analyzed to show where reionization first started to occur. The Principle Investigator shall assume all responsibility for the software required to work with the observed particle energies and locations. All final data will be archived in NASA's Planetary Database for public use.

E.5 Science Team

Heather Meyer is the Principle Investigator for this mission and is responsible for the integration of scientific experiments corresponding to the aforementioned Science Goals & Objectives. The PI is also responsible for the selection of landing sites and the SEO project. Ryan Wilkie, Co-Investigator, is responsible for determining the physical parameters of the dipoles and telescopes necessary to achieve the goals and objectives. Jesica Trucks, the Mission Specialist, is responsible for the integration and interpretation of all raw data to be obtained by the dipole-telescope array and relayed back to Earth. The science team will be funded through NASA.

E.6 Plan for Science Enhancement Options

The movement of regolith across the surface of the far side has never been studied, but it easily could be with a small addition to the proposed lander. To do so, a sensor with at least 3 directions of detection and a means of auto leveling is needed to observe the movement of lunar regolith and dust particles over time. The shift from night to day is of particular interest due to the interaction of the charged surface with the Earth's magnetosphere. As you can see in Figure 7, the Moon passes through the Earth's magnetotail every month for about 6 days. During this time, it is suspected that there is much migration of regolith due to the interaction between regolith that was positively charged by solar particles during the day and the Earth's magnetotail. An updated form of the Apollo Lunar Ejecta and Meteorite experiment outlined in Section E.1 is suggested for this project. This experiment would provide unique data in addition to enhancing previously collected data. Table 4 outlines the additional objectives that could be accomplished with the addition of this instrument to this mission.

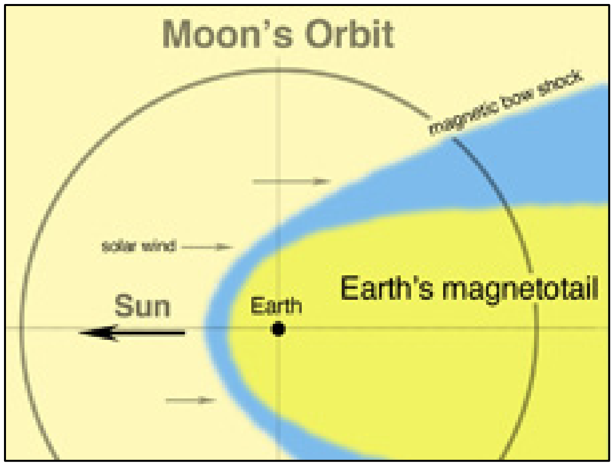


Figure 9 The Moon's orbit relative to Earth's magnetic field

Table 4

Secondary SEO Objectives
To study the lunar regolith of the far side to better understand lunar surface processes and the interaction of the space environment with the lunar surface.
To collect data from observations of the terrestrial magnetosphere's interactions with solar particles.
To observe the movement of lunar regolith across the far side.
To determine the background and long-term variations of cosmic dust influx rates in cislunar space.
To study possible correlations between the associated ejecta events and the times of the earth's crossing of cometary orbital planes and meteor streams.
To determine the extent of the contribution of interstellar particles toward the maintenance of the zodiacal cloud as the solar system passes through galactic space.
To investigate the existence of an effect called 'earth focusing of dust particles.'
To determine the extent and nature of lunar ejecta produced by meteorite impacts on the lunar surface
To determine the relative contribution of comets and asteroids to the earth's meteoroid ensemble.

F. Mission Implementation

F.1 General Requirements and Mission Traceability

The Mission Traceability Matrix in Table 5 shows information from the most current iteration of the mission architecture. The mission requirements column is a list of criteria that would make the science portion of the mission possible. The mission design column is an overall list of criteria that will be required to achieve the mission requirements. The following three requirement columns represent the spacecraft while en route to the moon, the lander requirements, and Earth based requirements.

Most of the information not included in the current iteration of the mission traceability matrix is being verified and will be received from other sources. ESTACA is designing the orbiter and the College of Charleston is designing the science portion. The DRO requirements will be assessed and calculated by ESTACA and the College of Charleston will do the same for the science information. At this point, no substantial information is known about the required earth based systems.

The first launch date is the 15th of August, 2017. The launch date for the second is the 15th of September, 2017. The mission is considered a success if it performs for three years but the components may have longer life cycles. The orbits of the communication satellites will provide information transfer to and from landers in the Aitken and Kholscutter craters. The data rate to the satellites is assumed to be 100Mbps from the landers, but data transmission from the Kholscutter crater to either of the satellites is only possible around 11% of the month. The average daily transmission from the Kholscutter crater is approximately one gigabit per day. Data transmission will occur about 14 times per day for around 11 minutes per pass. The available time for data transmission for the Aitken crater is significantly higher.

Table 5 Mission Traceability Matrix

<table>
<tr><th>Mission Requirements</th><th>Mission Design Requirements</th><th>Spacecraft Requirements</th><th>Lunar System Requirements</th><th>Other Requirements</th></tr>
<tr><td rowspan="2">Observing Strategies: Requires Tracking Maneuvers</td><td>Rocket Type: 2 Atlas V 551's C3 -1.85</td><td>Mass: 6105 Kilograms</td><td>Passes per day and duration: 14 daily 11 minute windows</td><td rowspan="16">TBD by further trade studies</td></tr>
<tr><td rowspan="2">Launch Date: 15 Aug 2017- 15 Sep 2017</td><td rowspan="2">Power: 5 kW</td><td rowspan="2">Antenna size: 3 meter</td></tr>
<tr><td rowspan="4">Observing smaller wavelengths requires gaps in the dish material no larger than 1/10th of the wavelength.</td></tr>
<tr><td rowspan="2">Mission Length: 3 years</td><td>Volume: TBD</td><td>Data Volume Per Day: 107 Gigabit</td></tr>
<tr><td rowspan="3">Data Rate: 100Mbit/sec</td><td rowspan="3">Real time data transmission: 100 Mbit/sec</td></tr>
<tr><td rowspan="2">Lunar Orbit Altitude: 100 Km</td></tr>
<tr><td rowspan="5">Observing Strategies: Requires Observation While Facing the Sun</td></tr>
<tr><td rowspan="4">Geographic Coverage: The orbit must cover the Kholscutter and Aitken Crater</td><td rowspan="2">Temperature Range: -200°C to 200°C</td><td rowspan="2">Transmit frequency: KA Band</td></tr>
<tr></tr>
<tr><td>Pointing Control: Small Thrusters</td><td>Power available for comm: 105 watts</td></tr>
<tr><td rowspan="6">Detector Radiation Shielding Requirements: Mylar foil sheeting</td><td rowspan="2">Downlink data rate: TBD</td></tr>
<tr><td rowspan="5">Need two years of observation to observe variability of the phenomenon.</td><td rowspan="3">Orbit Local Time: GMT</td></tr>
<tr><td>Dumps per day:14</td></tr>
<tr><td rowspan="2">Spacecraft data destination</td></tr>
<tr><td rowspan="2">Type of Orbit: 2 Inclined Circular Lunar Orbits</td></tr>
<tr><td>Science data Destination</td></tr>
</table>

F.2 Mission Concept Descriptions

This mission requires dipoles and radio telescopes to be placed on the far side of the Moon in the Aitken and Kholscutter craters. This environment will allow for interference free reception of radio waves due to a lack of atmosphere on the Moon and shielding from human-made transmissions. For further explanation on location refer to section D.1.1.

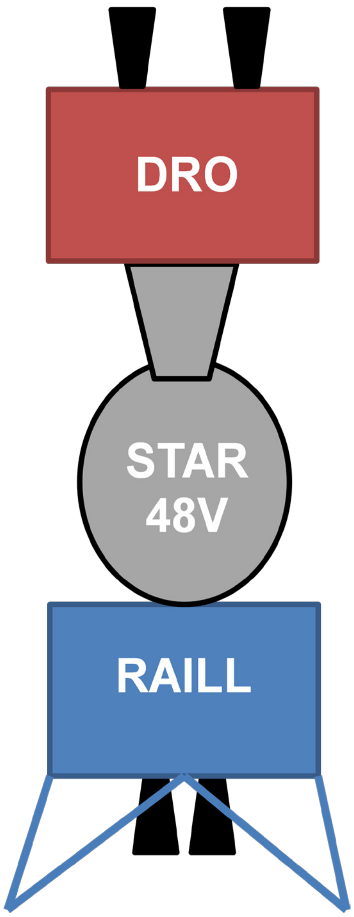


Figure 10 Configuration of Key elements

In order to meet the threshold science mission, one Atlas V 551 (AV551) launch vehicle (LV) will be used to land at Kholscutter. The baseline science mission will require a second AV551 LV to land at Aitken. The payload for both launch vehicles will consist of three key elements: an orbiter, an autonomous lander, and a solid rocket motor. The term "system" will be used when referring to all three of these key elements in the proposal and is depicted in Figure 10. The orbiter will be referred to as Data Relay Orbiter (DRO). The lander will be referred to as Radio Astronomy Instrument Lunar Lander (RAILL). The science package on RAILL will be the only difference between the two LV payloads. The first LV will contain yagi-una dipoles and the second LV will contain radio telescopes. The mobility system on RAILL, detailed in F.2.4.2, was developed by the InSPIRESS Level 2 team, consisting of the high school students listed in A.6.1. The solid rocket motor will be referred to as STAR 48-V. STAR 48-V is an off the shelf solid rocket motor from ATK.

This mission provided two options for launch vehicles, one Delta-IV H or two AV551. AETHER conducted a quantitative decision analysis, found in J.11, and concluded that two AV551 launch vehicles would best serve the science and instrumentation requirements.

F.2.1 Mission Design

The mission will place two arrays of radio astronomy instruments on the far side of the Moon. A pictorial summary of events is located below in Figure 11.

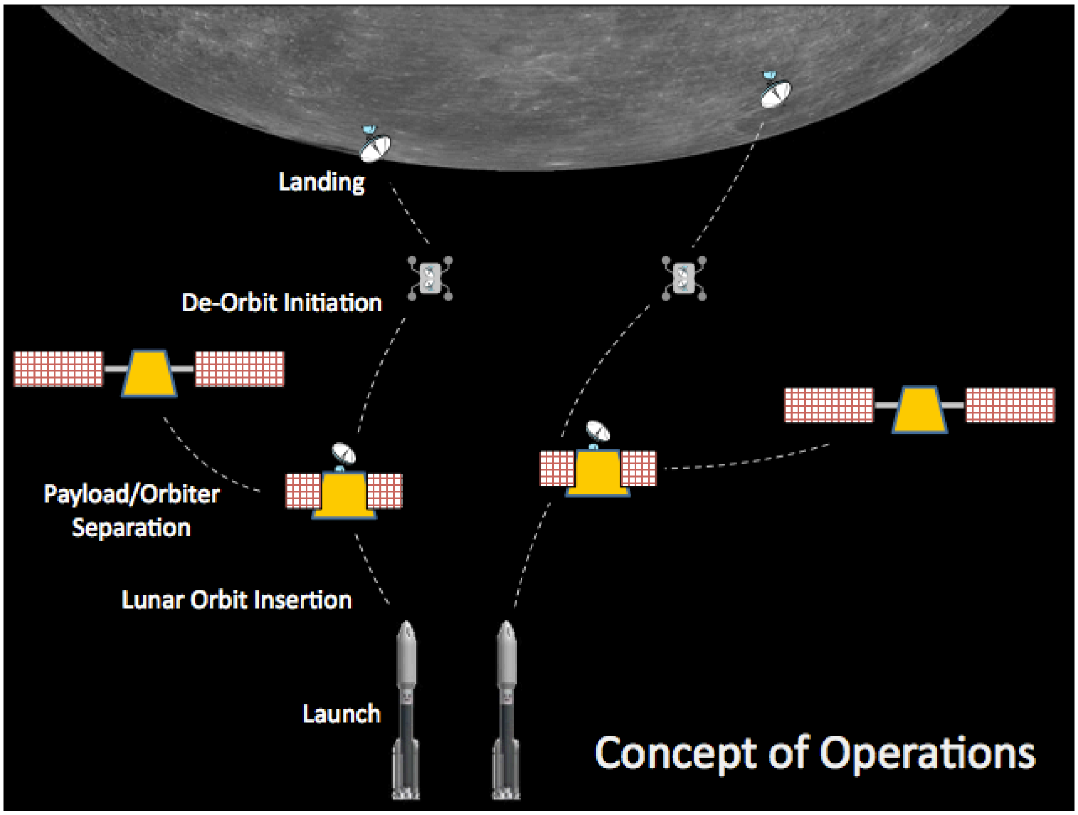


Figure 11: Concept of Operations

The first AV551 LV will launch August 15, 2017. The launch will have a c3 of -1.85. After the systems leaves the atmosphere the shroud will separate and the LV upper stage will perform trans-lunar injection. At this point DRO will take control of the trajectory. DRO will perform mid-course correction and lunar orbit insertion. The system will arrive in lunar orbit on August 18, 2017. This trajectory can be seen in Figures 12 and 13. The second AV551 LV will undergo the same events starting September 15, 2017 and arriving in lunar orbit on September 18, 2017. The second launch is dependent on the lunar cycle as well as the cycle of the precession of the first DRO. In order to maximize communication the first DRO must be halfway through a precession cycle when the second DRO reaches orbit. More explanation on this can be found in section F.2.6.

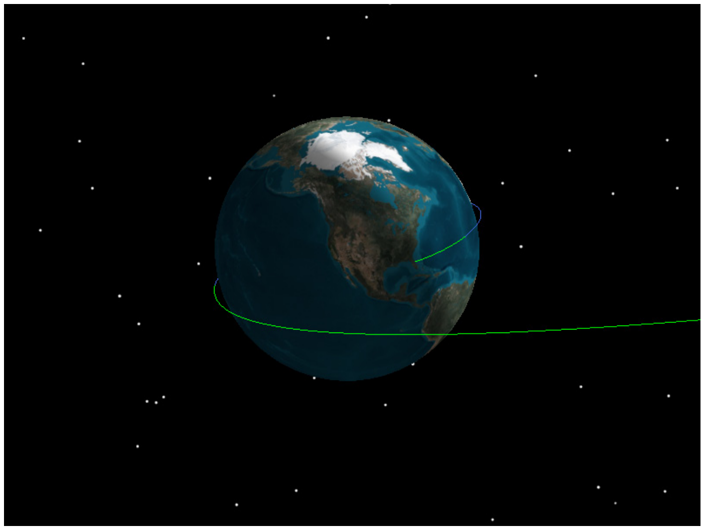


Figure 12 Launch Trajectory

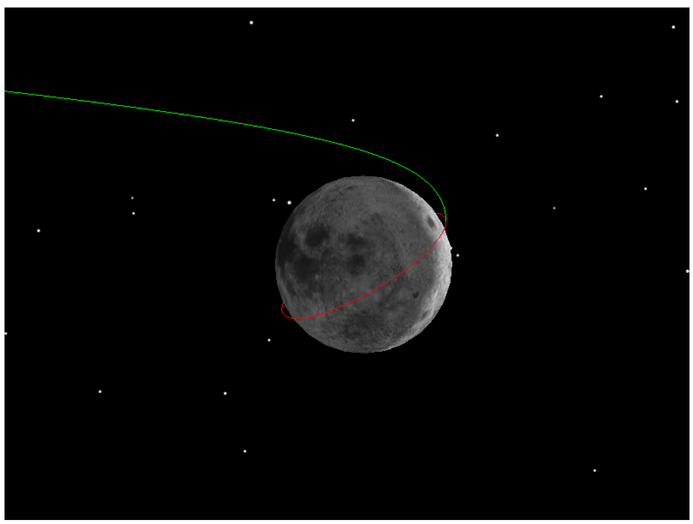


Figure 13 Lunar Orbit Insertion Trajectory

In order to facilitate communication with both Kholscutter and Aitken craters the system will have a 15 degree inclined orbit at 100km. The altitude is required by Autonomous Landing and Hazard Avoidance Technology (ALHAT). ALHAT will require several orbits to gather the necessary data to accurately land RAILL. At this point RAILL will perform the de-orbit initiation. STAR 48-B will perform the braking maneuver to handle the majority of the delta-v. Once STAR 48-B has finished braking, it will separate and RAILL will perform the final approach and landing burn. The delta-v's can be found in Table 6. These events, delta-v's and trajectories are the same for both launch vehicles.

Table 6 Delta V Requirements

Maneuver Number	Maneuver	Delta V (m/s)	Element Responsible
1	Mid Course Correction	51	DRO
2	Lunar Orbit Insertion	800	DRO
3	De-Orbit Initiation	20	RAILL
4	Braking Maneuver	1665	STAR 48-B
5	Final Approach & Landing	161	RAILL

RAILL will perform systems checks and wait for instructions from operations control on Earth. RAILL will then deploy the radio astronomy instruments. In order to facilitate the array placement requirements specified in section D.2 three ATK CoilABLE booms will be used to place the instruments. This is illustrated in Figure 14 and 15.

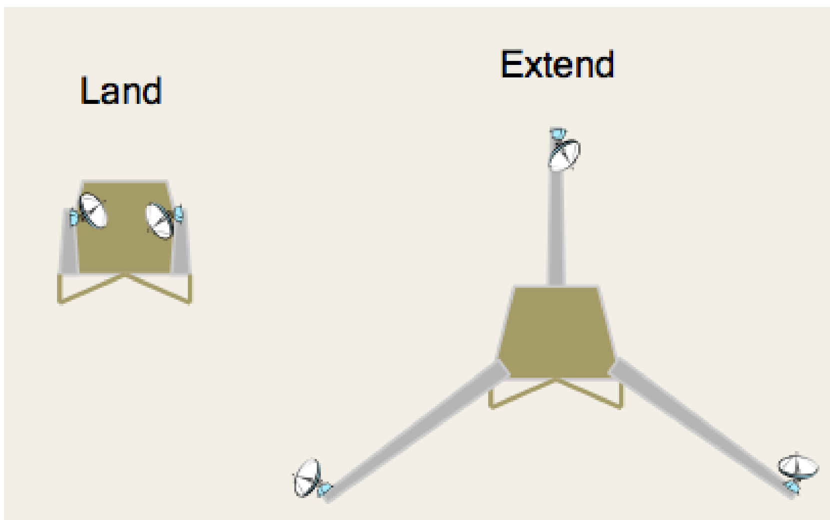


Figure 14: Deployment of Science Packages using the Mobility System

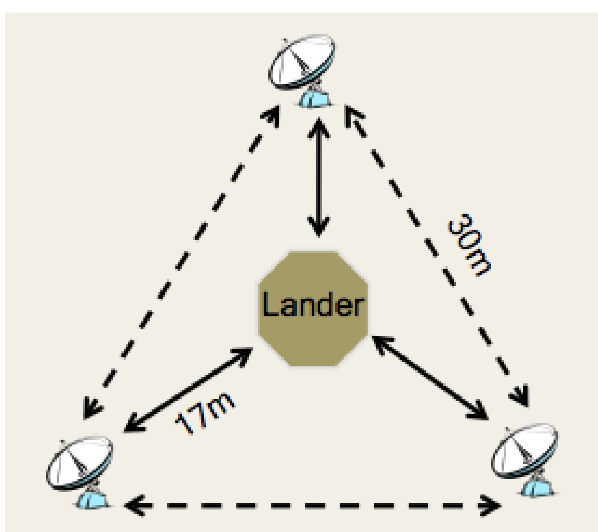


Figure 15: Arrangement of Science Packages

The RAILL's will send data through the DRO's to the Deep Space Network (DSN). The DSN is an international network of antennas that supports interplanetary spacecraft missions and radio and astronomy observations for the exploration of the solar system and the universe. Using DSN will allow facilitate communication with both the Kholscutter and Aitken landing sites.

The mission has a planned life cycle of five years. The threshold baseline mission will be met with a life cycle of three years. At the end of the mission life the DRO's will perform a 20 m/s de-orbit initiation burn for a controlled collision in the lunar highlands.

F.2.2 Launch Vehicle Compatibility

For this mission, two AV551 LV's will launch from Cape Canaveral to transport the critical mission systems to the Moon. The pair of departures will take place no later than 15 August 2017 and 15 September 2017, respectively. These launch vehicles are equipped with a medium length fairing, a C15 Launch Vehicle Adapter (LVA) and a B1194 Payload Spacer Ring (PSR). The LVA and PSR were chosen based on the highest allowable payload center of gravity among the available LVA and PSR. Each of these launch vehicles has a rated throw mass of 6105 kilograms. This accounts for the mass of DRO, RAILL, STAR 48-B, LVA, PSR, extension to the medium fairing, required spacecraft hardware (acoustic panels, thermal shield, environmental verification package, and centaur systems package). These elements can be found seen in Figure 16. The nearly identical launch vehicle payloads yield a high degree of redundancy to maximize the total data collected and minimize design costs.

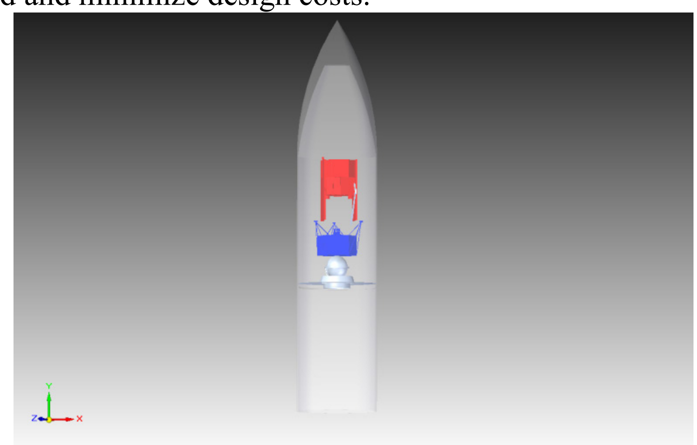


Figure 16 Spacecraft Stack in Shroud

F.2.3 Flight System Capabilities

F.2.3.1 Radio Astronomy Instrument Lunar Lander

RAILL will transport the science payload from lunar orbit to the lunar surface. During the transit the STAR 48V will remove the majority of the descent velocity allowing the lander to be more mass efficient. RAILL will utilize a Hydrazine monopropellant propulsion system to perform attitude control and main propulsion. Once RAILL has safely landed it will deploy the science payload using extending booms. After the deploying the science packages, RAILL will power the science equipment and transmit the science data to the orbiter to be sent back to Earth. A block diagram for RAILL is displayed in Figure 17 and the mass budget is in Table 7.

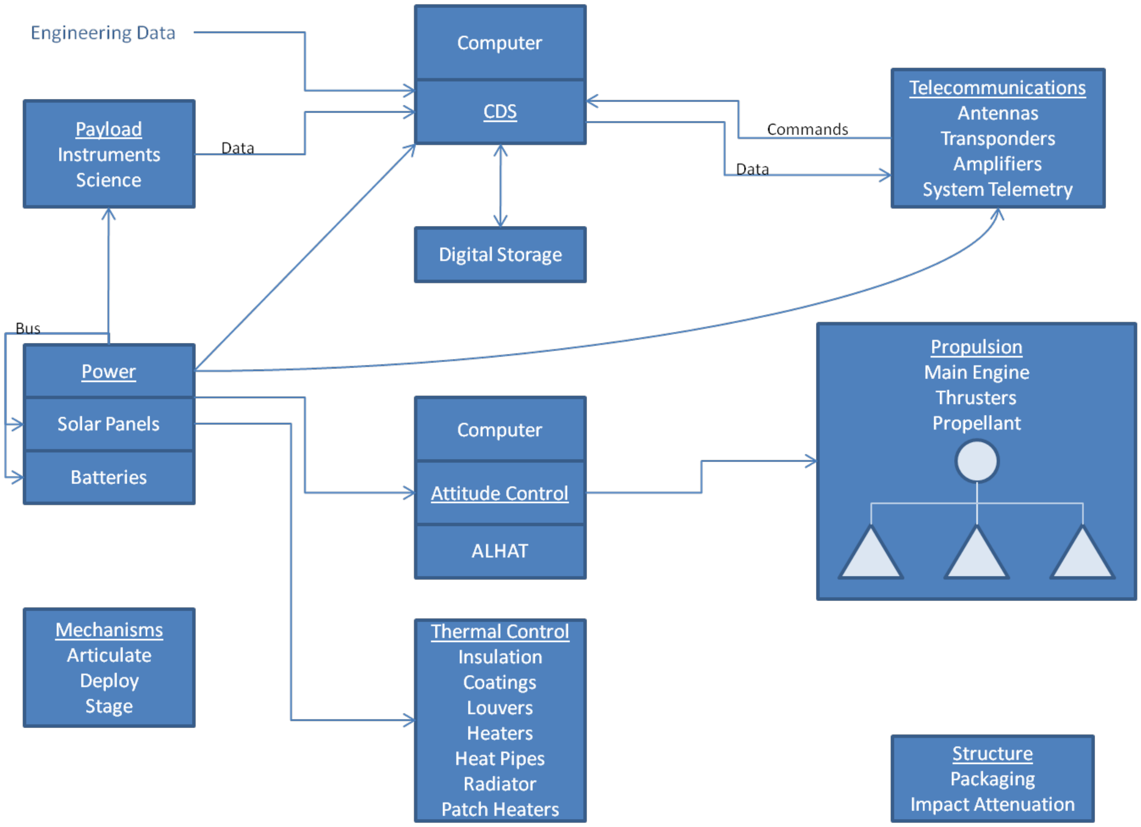


Figure 17 RAILL Block Diagram

Table 7 Mass Budget

Subsystem	Current Best Estimate	Contingency (%)	Contingency (kg)	Max Expected (kg)
Propulsion/ACS	67	30	28.7	95.7
Thermal	39.1	30	16.8	55.9
Power	126.2	30	54.1	180.3
Structures	423.7	30	127.1	550.8
Communications Command & Data Handling	21.8	30	9.4	31.2
Science Payload	256.2	30	109.9	366.1
Mobility System	151.8	30	65.1	216.9
Lander (Dry Mass)	1085.8	30	411.1	1496.9
Propellant				386
Lander + Propellant	1471.8	21.8	411.1	1882.9
Margin		5	99.1	1982
Reserve		5	104.6	2086.6
ATK Star 48V				1907.5
Dry Mass	154.1			154.1
Propellant (13% offload)	1753.4			1753.4

F.2.3.1.1 Propulsion

The total Delta-V for the lander's descent will be 1.962km/s, including allotment for attitude control. The maneuvers for the descent include descent initiation, braking burn, ALHAT maneuvers, and final approach and landing. The delta-V and propellant requirement for each maneuver is tabulated in Table 8. The braking burn will require the largest delta-V (1.665km/s) and will be performed by an ATK STAR 48V vector controlled solid rocket motor. The thrust vector capability will reduce the load on the attitude control system during the braking burn. This solid rocket motor has a total mass of 1908kg and a propellant load of 1753kg, which is a 13% offload. For the remaining maneuvers, and attitude control, the lander will utilize a Hydrazine monopropellant engine system with a specific impulse of 230s for steady burns and 208s for attitude control pulsing. The lander will use three MR-80B (3100N) thrusters for main propulsion and 12 MR-120 (90N) thrusters for attitude control. The MR-80B thruster was chosen because of its maximum thrust (3184N) and its wide throttling range (3184-31N).

Table 8 Delta-V Budget

Burn	Delta-V	Engine	Propellant Mass (kg)
De-Orbit	20	MR-80B	35
Braking	1665	Star 48B	1780
Final Approach and Landing	161	MR-80B	140
ACS	116	MR-120	197

The monopropellant system has a dry mass of 61kg and a wet mass of 420kg, which includes 360kg of Hydrazine and 2.2kg of Helium as a pressurant. The total required propellant mass is 360 kg but an additional 6.5% propellant mass will be loaded onto the lander for loading and expulsion uncertainties. The additional 6.5% of propellant accounts for 3% residual propellant and 0.5% loading uncertainty and 3% reserve to allow for unexpected maneuvers and additional attitude control burns.

In order to have adequate throttling control during landing, Alabama A&M designed a Throttling Cavitating Venturi Valve (TCaV) seen in Figure 18. The TCaV will allow for the necessary hydrazine flow rate of 4.2 kg/s at a pressure of 2MPa. The full report on the TCaV is appendixed.

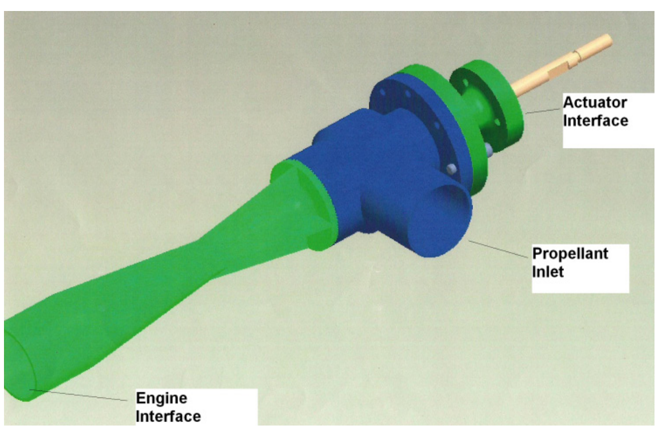


Figure 18 TCaV Assembly

F.2.3.1.2 Command and Data Handling

AETHER will be using a RAD750 radiation hardened computer in each lander for command, control, and data handling. The RAD750 has a processor speed of 132MHz and 128MB SDRAM. It can withstand temperatures between -55°C and 125°C. In addition to the computer there will be a 2 Gigabit solid state recorder to store the science data during the time between orbiter passes.

F.2.3.1.3 Communications

For communications to the orbiter, the lander will use a 0.7m parabolic antenna and Travelling Wave Tube Amplifier (TWTA) similar to what is used on the Lunar Reconnaissance Orbiter (LRO).[5] This provides a high data rate of 100Mbps which will significantly increase the transfer of science data to the orbiter. For communication from the orbiter to earth, AETHER will utilize the deep space network at Ka-band frequencies.

F.2.3.1.4 Power

One of the largest obstacles to overcome in this mission is powering the science package during the (14-Earth day) night cycle on the moon. This mission will use a large array of batteries to power the science during the night and recharge the batteries using solar arrays during lunar day-cycle. The solar arrays will deploy from the lander into a fixed position.

During the lander's descent it will require about 57W-hr of power. The onboard computer, heaters, and ACS instrumentation will require 52W-hr while the propulsion will require 5W-hr for heating and operating the valves.

The maximum power required is during the orbiter flyover, when the science instruments will be running and the lander will be transmitting data to the orbiter. The majority of the power draw comes from the TWTA, which has a high power draw in order to achieve the high data rate. The power requirements can be seen in the power profile (Figure 19) and the power summary (Table 9).

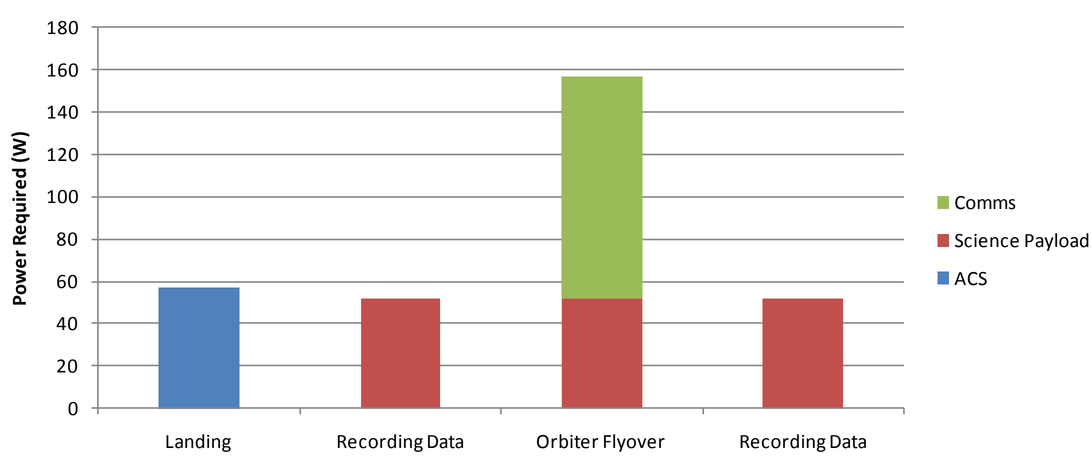


Figure 19 Power Profile

The maximum power draw for RAILL is 152W for each lander. This requires 126kg of ABSL designed battery pack containing SONY 18650NL Lithium Ion batteries. In order to charge the batteries and provide power to the lander, RAILL will be equipped with 21 SpectroLab XTJ solar cells. These cells have an efficiency of 29.5%, a mass of 10g and an area 60cm^2 each. The solar cells will be mounted together in groups of 7 between the booms for the science telescopes. The cells will be deployed into a fixed position from the side of the lander.

Table 9 Power Summary

Subsystem	Power Required(W)			
	Cruise	Landing	Comm.	Recording Data
Propulsion	0	5	0	0
ACS	0	27	0	0
CD&H/Comm.	0	13	106	10
Thermal	16	12	0	12
Instruments	0	0	0	42
Total Power Loads (CBE)	16	57	106	64
Total Power Loads (CBE plus contingency)	22.9	81.5	151.5	91.5

F.2.3.1.5 *Attitude Determination and Control*

RAILL will utilize the lander's processor, Star Tracker, gyroscope, velocimeter, Terrain-Relative Navigation position sensor, and altimeter to control the descent of the lander.

Comtech AeroAstro's Miniature Star Tracker was chosen because of its low power consumption (<2W) and mass (375g). It has a 24°x30° field of view, a maximum pitch/yaw rate of 10°/second, and accuracy better than ±70 arc seconds about yaw and pitch axes and ±120 arc seconds about the roll axis. Honeywell's Miniature Inertial Measurement Unit (MIMU) was chosen to serve as the gyroscope due to its small mass (4.5kg) and its available velocity channel so that it could serve as the velocimeter.

The attitude control system will be comprised of 12 AEROJET MR-120 (90N/20-lbf) hydrazine thrusters connected to the fuel system of the main propulsion system. These thrusters will provide attitude control during the solid motor burn, while the three main engines (MR-80B) will be throttled to provide the major attitude control during the remainder of the landing.

F.2.3.1.6 *Thermal Control*

The landers will require a combination of heaters and louvered radiators to maintain operation temperatures during the long day/night cycle of the moon. The temperature on the lunar surface will vary from -233°C to 123°C while the batteries and electronics of the science payload and lander require a temperature range of -20° to 45°C. To maintain the operational temperature range a Warm Electronics Box (WEB) will house all the electronics and maintain a temperature range of -10° to 35°C. The WEB will contain patch heaters controlled by Tayco solid-state controllers and heat pipes to maintain the required temperature range. The battery pack and computer will mount to cold plates inside the WEB which will be connected by a heat pipe to a louvered radiator to dissipate excess heat.

Passive thermal control is obtained using 12-layer multi-layer insulation (MLI) which reflects sunlight to shade the spacecraft against overheating, and retains internal spacecraft heat to prevent too much cooling. Gold is a very efficient infrared reflector and the MLI's gold color results from a reflective silvery aluminum coating behind sheets of amber colored Kapton material.

MLI will surround the telescope equipment in order to maintain the standard operating temperature. To keep the solid rocket motor within a safe operating temperature between -1°C and 37°C, the motor will be wrapped in MLI and the spacecraft will roll during the lunar transit to evenly heat the solid motor. The hydrazine will be maintained at a safe temperature (17-50°C) by wrapping the tank and lines with MLI blankets and monitoring the temperatures with heaters and thermostats to prevent over cooling. Radiation shields will be used to block the radiation coming from the thruster nozzles.

F.2.3.2 Data Relay Orbiter

The DRO is a contribution designed by the partnering team from ESTACA.

From launch to separation, DRO provides the major functions for RAILL. During lunar transit, the solar panels will deploy to provide power to the orbiter and lander. The solar panels will be retracted during mid-course correction (MCC) to avoid breaking them with the acceleration of the first Delta-V. The MCC is done in the direction of the advancement. After have been performed, the solar panels are re-deployed. In preparation for the lunar orbit insertion (LOI), the solar panels will be retracted again and the spacecraft rotated around because the second thrust is used to slow down the spacecraft. After the boosts have been performed, the lander is ejected and the solar panels are re-deployed. Then the orbiter begins its primary role in orbit sending and receiving data. The mass budget for DRO can be found in Table 10.

Table 10 DRO Mass Budget

Subsystem	Current Best Estimate	Contingency (%)	Contingency (kg)	Max Expected (kg)
Propulsion/ACS	157	30	67.4	224.4
Thermal	50.8	30	21.8	72.6
Power	60.0	30	125.7	85.7
Command & Data Handling	28.8	30	12.4	41.2
Communications	10.2	30	4.4	14.6
Structures	84.6	30	36.3	120.9
Orbiter (Dry Mass)				**559.4**
Propellant	1408			
Lander+Propellant				1967.4
Margin		5	103.6	**2071**

F.2.3.2.1 Propulsion

The propulsion part of the lunar orbiter will be in charge of two maneuvers, the MCC ($51ms^{-1}$) and the LOI ($800ms^{-1}$). The value of this Delta-V is 51m.s-1. The second one is called Lunar Orbit Insertion (now referred as LOI). In addition to these primary burns, the orbiter will perform station keeping burns to maintain a stable orbit.

To meet these requirements, AETHER decided to use bi-propellant engines with dinitrogen tetroxide (NTO, N_2O_4) as an oxidizer and Hydrazine (N_2H_4) as a fuel. Pure hydrazine was chosen over other hydrazine compounds so the attitude control thrusters and main engine could use the same propellant tank. These propellants have a temperature range of 263.85 to 373.5K for NTO and 215.2 to 518.5K for Hydrazine at a tank pressure of 400psi. Helium will be used to pressurize the tank and will be filled at a pressure of 4500psi.

For the main engine, 3 were considered: the TR-308 from Northrop Grumman, the HiPAT from Aerojet and the very promising AMBR (Advanced Material Bi-propellant Rocket) developed by NASA. Each of these has a thrust in the desirable range and an Isp higher than 320s. Using the AMBR provides a reduction of 42kg in propellant. Also, the AMBR engines have twice the thrust of the TR-308 and HiPAT, cutting the thruster mass in half.

The propulsion system will have two tanks for each of the propellants (NTO & Hydrazine) and two tanks for the pressurant (Helium). The propellant tanks are made of titanium and the pressurant tank is made of Composite Overwrapped Pressure Vessels pressurized at 310 bars (4500psi). All tanks will be made with a factor of safety of 1.5.

All the sub-systems here are of technology readiness 9, that mean they have been used in space on many successful missions. It also means that it will cost less since we will only need to recalibrate them and make them suited for a fly to the moon. Only the AMBR engine has not proven flight capability but the gain in terms of mass makes it worth the risk.

F.2.3.2.2 Command and Data Handling

The computer will monitor the status of the DRO and RAILL while they are together to detect, diagnose, and activate bypass solutions for any system failures. The computer will be a RAD750 equipped with an inertial measurement unit. The computer will manage the movements of the solar panels, the charging of the batteries and antenna during flight. The same computer will also be used for avionics control. Along with the computer, DRO will have 8 Gigabits of solid-state data storage to store the science information while the orbiter is on the far side of the moon.

F.2.3.2.3 Communications

The telecommand and telemetry system handles communication to the ground. The telecommand functions (ground ⇒ satellite) receive and decode the instructions or data sent by the control center and carry out the task of distributing them to other subsystems. The telemetry functions (satellite ⇒ ground) gather the data relevant to the satellite's performance and the science data, and compress the data to transmit to Earth.

DRO will be equipped with one parabolic dish to use for transmitting and receiving data. It can move in both axes to maintain sight of the earth or moon. Like RAILL, DRO will be equipped with a TWTA similar to the one used on LRO for its high data rate.

F.2.3.2.4 Power

The power subsystem has been sized to supply 1kW, and solar panels will be used to provide the energy the satellite needs. The solar panels have an end of life power output of 240W/m^2 and efficiency of 20%. In order to meet the power needs 4.2m^2 of solar panels will be required which gives a total mass of 55.4kg. The panels will be divided into 4 identical panels of 1.05 m^2 each so they can be folded up to fit within the payload shroud.

During eclipses the Solar panels will not generate power so batteries become the primary power source. DRO will have Lithium ion rechargeable batteries to provide this power. It will require 4.4kg of Sony's

During each orbit, the batteries complete charge-discharge cycles. To last through the whole life-span of a satellite at such a high rate (more than 50000 cycles), laws of control were established after testing. These laws are described by the CNES (French government space agency) as follows:

- Control of the discharged electricity amount in proportion to the battery's nominal capacity: the discharged quantity depth shall not exceed 25 %.
- Control of the charged and discharged electricity amounts: their ratio, called 'charge ratio', must be close to 1. Its value depends on temperature. The on-board computer supervises and controls the charge state of the batteries.
- Control of the charge voltage of each battery relating to a certain threshold (36.5 Volts), also depending on temperature. The charge current must be limited to a maximum of 12A. Upholding this condition is an electrical component, the shunt junction regulator (RSJ). The RSJ regulates the charge voltage and current while also ensuring the satellite is correctly supplied.

F.2.3.2.5 Attitude Determination and Control

The flight control system maintains the satellite's trajectory and orientation. This task is achieved by software that utilizes data supplied by star trackers and gyroscopes. It calculates the deviations and corrects them by means of actuators and engines. The attitude control system will be made up of 12 Aerojet MR-106E thrusters, which have a thrust of22N and Isp of 229s. These use hydrazine from the main propulsion tank.
Before performing the MCC, the attitude control system will orientate the spacecraft in order to perform the first acceleration. To perform the second maneuver (LOI), the attitude control system will have the spacecraft to make a flip in order for the second boost to make the orbiter decelerate.

F.2.3.2.6 Thermal Control

The primary thermal control method for DRO will be passive, utilizing 10-layer MLI, louvered radiators, and multiple patch heaters for redundancy. The MLI will cover the entire outer surface of the orbiter except for the radiators..

Once every six months the orbiter will pass through an eclipse in which the temperatures will drop significantly and will require active thermal control. In order to compensate for these extreme temperatures DRO will be equipped with heaters and thermal switches. The heaters will be placed in the spacecraft near the propellant tanks and electronics, while the thermal switches will be placed between the patch heaters and radiator to prevent heat loss.

F.2.3.2.7 Structures

The satellite main design was inspired from pre-existing communication satellites and orbiters. The main structure of the satellite is hexagonal. For a given cylindrical fairing it offers more room than a cubic structure. It allows the big spherical fuel tanks to fit without much waste of space. The satellite is composed with two identical hexagonal sections set on top of each other, each 1m tall having a diameter (corner to corner) of 2m. The top section contains the hardware (electronics, memory devices, batteries, etc.), and the bottom section contains the tanks for the propulsion system and thrusters. The engines are diametrically opposed to allow the thrust resultant to be aligned with the center of gravity.

F.2.4 Additional Mission Elements

There are two additional mission elements for this mission: the ATK Boom deployment system, and the STAR 48V solid rocket motor. For the descent from lunar orbit to the lunar surface, the STAR 48V will reduce descent velocity, allowing for a more mass efficient lander. After RAILL has arrived at the lunar surface, the booms will retract, for the placement of the radio telescopes and yagi-uda dipoles.

F.2.4.1 ATK CoilABLE Booms

CoilABLE ATK Booms will be used for the deployment of the telescopes and dipoles. A diagram showing how the booms fit in to the entire deployment system is shown below. These booms use strain-energy for positive force deployment, meaning there are no moving mechanical parts. This is a commonly used method of deployment for this type of boom. These booms were chosen because they use very little space (stowed length is 2% of deployed length), can be very strong, can perform the required extension, and are cost efficient. These booms have also been used on many other missions, such as Galileo, Cassini, and Lunar Prospector. When stowed, the structure is compressed and twisted; stowed inside a canister style lanyard. Upon deployment, the strain-energy from being stowed will retract the structure; dampened by a power operated crank (winch motor).[6] The interfaces of this deployment system are modeled in Figure 20.

The chosen booms will retract 17.5 meters, and will support a 72kg load (telescope mass, including mounting hardware; this is more than the dipoles mass). This gives a required bending strength of 2016 N*m. Due to this, the booms will be of 2.27m diameter. This results in a boom mass of 151.8 kg, which is equal to a weight of 1488N on Earth, and 242.8N on the Moon.

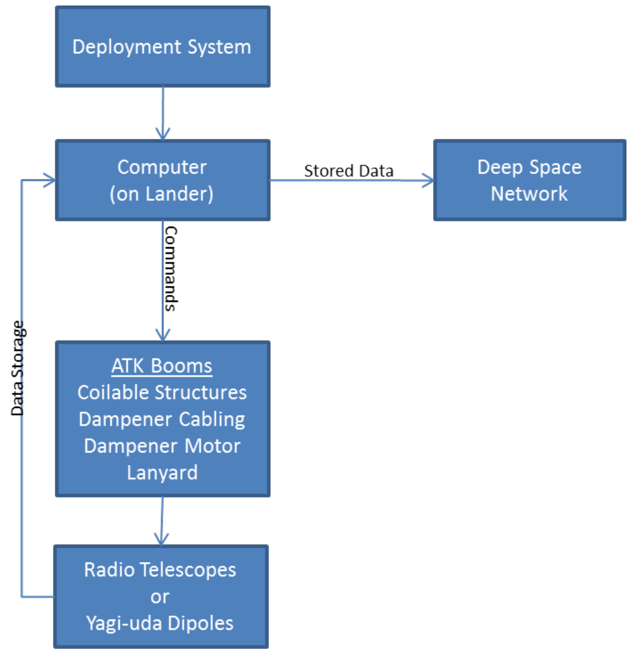


Figure 20 Deployment System Block Diagram

The longerons are made of s-gloss epoxy, and have cross sectional area of 1.5% of the boom diameter (0.034m). The battens – the buckled compression members that are compressed to preload the structure for positive strain-energy – are also made of s-gloss epoxy, with a diameter smaller than the longerons. The battens will be assembled into a triangular frame. The diagonals will be made of stainless steel cable, or s-gloss epoxy; these provide torsional and shear stiffness, countering the batten preload. The junctions between the longerons, battens, and diagonals will use a corner fitting for attachment. There is a transition zone, which is a section of the boom in mid-deployment, for the transition from stowed to deployed arrangements. The distance between the batten frames is 58% of the boom diameter (1.314m).[6]

F.2.4.2 STAR 48

From the mission architecture, the Star 48V (vectorable) solid rocket motor, Figure 21, was chosen, because its characteristics meet the impulse requirements, and because of its vectorable capability. Solid rocket motors have an unstable velocity vector, and the vectorable Star 48V corrects for that within 4˚. The Star 48V is essentially a modified Star 48B solid rocket motor. The Star 48V has a total impulse of 5799 kN*sec, propellant specific impulse of 294.2 sec, and an effective specific impulse of 294.1 sec. The Star 48V has a percent offload of 11%, gives the required impulse, and is able to perform the correct Delta-V; all in the smallest possible package. The 48B was first used on the Space Shuttle in June 1985, so there are some benefits to its flight-proven production status. The case features mounting flanges, and tabs for external hardware, that can be relocated or modified without the need for requalification (as is also true for the Star 48V). This gives an almost infinite variety of possible mounting configurations. The Star 48V received its qualification as the upper stage for EER System's Conestoga Vehicle in 1993 (Star 48V production status: qualified).[7]

The motor is enclosed in a high-strength titanium case, for the use of high-energy TP-H-3340 propellant. There is a carbon-phenolic exit cone, and 3-D carbon-carbon throat on the submerged short nozzle. Below, an image of the Star 48V is shown. The motor has dimensions of 49 inches in diameter, with an 81.7 inch length. The Delta-V that this motor will perform is 1665 m/s.[7]

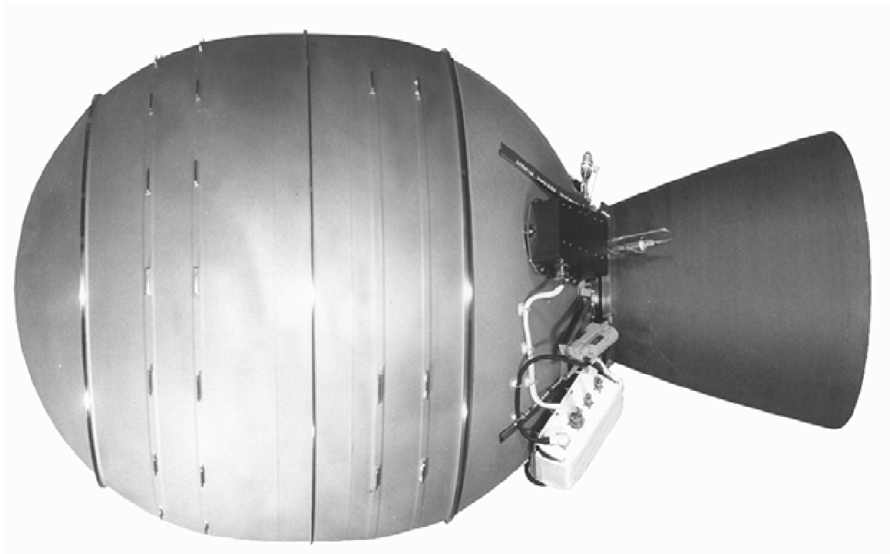


Figure 21 Image of STAR 48V Solid Rocket Motor[7]

F.2.5 Flight System Contingencies and Margins

The margins and contingencies can be seen in Table 11. The spacecraft, lander and orbiter have a 30% contingency on the mass to allow for typical growth during the design process. There is also an additional 5% margin on the major system elements. There is a 3% reserve on the propellant mass to allow for any unexpected maneuvers or additional attitude control maneuvers. The power requirements of the spacecraft have a 30% contingency and the data storage has a 400% contingency based on the JPL recommendation for the proposal stage.

Table 11 Contingency and Margin

	Contingency	Margin
Subsystem Mass	30%	0
Element Mass	0	5%
Power	30%	0

F.2.6 Mission Operations

F.2.6.1 Ground Systems and Facilities

For this mission, NASA's Deep Space Network (DSN) is used. The DSN consists of an international network of communications centers located in three countries. Each base has a variety of antennae that are capable of supporting all data for this mission. These bases are located so that a constant observation of spacecraft is possible. These communication facilities communicate directly with the Deep Space Operations Center (DSOC) located at the Jet Propulsion Laboratory (JPL) facilities. The antennas at all three DSN Complexes communicate directly with the Deep Space Operations Center (DSOC) located at the JPL facilities in Pasadena, California. DSOC officials handle all operations and data handling from spacecraft guidance and navigation.[8]

The facility used for development of the RAILL's is NASA's Goddard Space Flight Center (GSFC), which is located in Greenbelt, Maryland. GSFC is a major U.S. laboratory for developing and operating unmanned scientific spacecraft. GSFC provides facilities for the construction and development of spacecrafts, as well as the spacecraft software and scientific instruments that will be utilized.[9]

The facility used for testing is the Marshall Space Flight Center (MSFC), which is located on the Redstone Arsenal in Huntsville, AL. MSFC offers many facilities that are capable of testing all subsystems and elements. MSFC contains the Huntsville Operations Support Center (HOSC). HOSC monitors missions from Cape Canaveral, which is the location of the departure of the two Atlas V 551 launch vehicles.[10]

F.2.6.2 Telecommunications, Tracking, and Navigation

Two DRO are used to relay data from the two RAILL at two different landing sites. Both DRO's have the same inclination of 15 degrees. However, the Moon precesses underneath the orbit. So the peak and valleys of the sine wave move with each orbit. Because of this, an orbiter will directly pass over the northern site for a few passes and then it will no longer be in contact with the site. The orbiter will then come in contact with the southern site. Because of the precession, the orbiter switches between which site it is in contact with. To maintain constant communication, AETHER is using two DRO's. Both DRO's are in the same inclination of 15 degrees, but the first orbiter is allowed to get halfway through the precession cycle before the second orbiter gets put in orbit. Therefore, when one orbiter is in contact with the northern site, the other orbiter is in contact with the southern site. The crater walls can interfere with the communication time between the landers and orbiters. However, the crater walls are small compared to the radius of the craters; therefore, the interference is expected to be negligible.

A Traveling Wave Tube Amplifier is used, which gives a date rate of 100 Mbps.[11] The RAILL in the Aitken crater is capable of transmitting a total volume of uplink data of about 108 gigabits per day, or 39,271 gigabits per year. The duration of the data uplink from the Aitken site is approximately 37 days of the year, or about 10% of the year. The RAILL in the Kohlscutter crater is capable of transmitting a total volume of uplink data of about 115 gigabits per day, or 42,103 gigabits per year, while the duration of the data uplink from the Kohlscutter site is approximately 40 days of the year, or 11% of the year. The total amount of uplink data capability is expected to be 223 gigabits per day or 81,374 gigabits per year. The Earth access time is approximately 327 days of the year, or 89%. The uplink data volume is limited by the communication access between the orbiters and landing sites rather than between the orbiters and Earth.

All uplink information was obtained using STK. STK models the flight of both orbiters and shows duration of communication time between both orbiters and the landers for a year. 90 percent of the uplink volume is science data, while 10 percent is overhead. The uplink data capability for both landing sites is summarized in Table 12 shown below.

Table 12 Uplink Capacity

Landing Site	Data Rate	Uplink Volume						Duration	
		Science Data		Overhead		Total			
		Gb/ day	Gb/ year	Gb/ day	Gb/ year	Gb/ day	Gb/ year	days/ year	% of year
Aitken	100 Mbps	96.8	35344	10.8	3927	107.6	39271	37	10%
Kohl-scutter	100 Mbps	103.8	37892	11.5	4210	115.3	42103	40	11%
	Total	200.6	73236	22.3	8137	222.9	81373		

All data transmission is in the Ka band frequency between 25.5 and 25.8 GHz. The landing sites have 14 passes per day from the orbiters, most lasting about 12 minutes, but some lasting as little as 3 minutes. The time between data transmission periods is approximately 1 hour and 46 minutes.

F.2.6.3 Acquiring Critical Event Data

Data is taken during critical events to monitor all equipment. The critical events include launch, in-orbit maneuvers, mid-course corrections, lunar orbit, landing and science package deployment. The mission has communication redundancy for science data as there will be two landing sites and two orbiters. All scientific data is stored on the RAD750 computer and is sent back to Earth's ground systems for further analysis.

F.2.6.4 Discussion of Operations Plan

The mission operates from the Jet Propulsion Laboratory (JPL). JPL conducts astronomy missions and is also responsible for operating NASA's Deep Space Network.[8] Experts are needed for critical events such as launch, maneuvers, and landing. A crew of scientists is trained to analyze all scientific data and to monitor the landers and equipment for the duration of the mission. The required size of the crew is yet to be determined by College of Charleston.

F.3 Development Approach

F.3.1 Systems Engineering Approach

F.3.1.1 Requirements management

AETHER utilizes requirements through consideration of a top down philosophy. That is, AETHER observes the over-arching philosophies and guidelines that are required to address this proposal first. After the top priorities are observed and considered, AETHER addresses next tier requirements. The Announcement of Opportunity would be considered the high level requirements so AETHER has taken it into account from the start of requirements development. The Principal Investigator objectives provide the next tier in requirements, so they are considered next priority for the mission design. Finally, the engineering capabilities defined the final tier AETHER utilized for mission requirements.

F.3.1.2 Decision Making

In AETHER, the Lead Systems Engineer (LSE), Sam Bennett, is responsible for conducting a quantitative analysis in which each team member can provide input on criteria, weightings, and ratings. The criterion lists subject matter important to the specific decision that is being addressed. Weights are allocated to each criterion and assigned a quantitative value to justify mission significance and magnitude. Quantitative ratings are assigned to each option being observed in the decision analysis to allocate proper fulfillment to each criteria. This process was as impartial as possible in order to make the best engineering choice for critical aspects of the mission architecture without bias.

F.3.1.3 Interfaces

Within AETHER, UAHuntsville interfaces with Level 1 Innovative Student Project for the Increased Recruitment of Engineering and Science Students (InSPIRESS) from Grissom and Sparkman High Schools, as well as Alabama A&M. The Level 1 InSPIRESS Grissom High School team provides an experiment that will be attached the orbiters to measure ground effects on the Moon. The Level 1 InSPIRESS Sparkman High School team provides an experiment measuring the solar effects on electronics through the use of CCD's and measuring solar winds through the use of a Solar Sail. Alabama A&M designed a Cavitating Ventury Value to be used for propulsion. To manage all interfaces appropriately, AETHER distributed the workload for each interface. AETHER's Program Manager is the main Point of Contact (POC) with ESTACA. An Interface Control Document was created to manage the requirements between UAHuntsville and ESTACA. The Chief Engineer is the POC for the College of Charleston. A Memorandum of Understanding was instituted to govern the relationship between UAHuntsville and College of Charleston with regard to requirements and deliverable dates. The Lead Systems Engineer is the POC for the Level 1 and Level 2 InSPIRESS. AETHER maintained communication with the Level 1 and Level 2 InSPIRESS to convey requirements.

The chief engineer (Garrett Gammon) is responsible for communicating technical requirements to subsystem engineers on the lander design team. While these interfaces are more internal, AETHER knows that a subsystem team lead would be necessary for success. To communicate with all interfaces, AETHER utilizes Skype, email, phone calls, and face to face meetings frequently to make sure all interfaces work effectively and efficiently.

F.3.1.4 Configuration Management

Configuration management is performed on a mission level by the UAHuntsville team members. AETHER's Chief Engineer (Garrett Gammon) maintains the estimated mass allocations for elements and subsystems throughout the changes in mission architecture. The expected masses and ConOps figures are updated on whiteboards in the team meeting room, as well as submitted electronically for team members to access anytime. To allow for consistency, the most up-to-date mission architecture is provided on a regular basis in team meetings and updated electronically.

F.3.2 Mission Assurance Approach

F.3.2.1 Fault Tolerance and Management

As this is a Class B mission, few faults that are allowed for this mission and have to be mitigated. Redundancies are in place for most aspects of the mission. The major redundancy that would help mitigate risk is how the mission design is structured. AETHER plans to use two semi-identical payloads for the two Atlas V551s. Each Lander will contain radio astronomy instruments extended by three ATK booms. The threshold mission allows for one Lander to be successful. If the first lander does not successfully the dipole antennas, the second lander will have radio telescopes that work with the solitaire dipole on the first lander. Should the first lander fail all together, the second can be modified to contain the dipole instruments instead of telescopes. This way if only one lander is successful, then the threshold will be satisfied.

To address systems' faults, AETHER will use a Fault Tree Diagram (FTD) to analyze statistical probabilities to find possible faults in the system. The FTD will identify possible hazards affecting the systems and subsystems. Once faults are identified, they can be controlled to decrease the probability of fault occurrence.

To address specific problems that might arise in the mission testing, AETHER will use a cause-and-effect diagram to search for the root of the problem. This cause-and-effect diagram (also referred to as a fish bone diagram) will analyze multiple efficiency areas such as people, materials, management, equipment, measurements, and environment.

F.3.2.2 Product Assurance

To ensure the mission collects the correct science data and science data comes back to earth, AETHER implemented several essentials to the mission design. The necessary science can be achieved if the first lander is successful in deployment or both landers land without deployment. The two DRO's in orbit around the moon will provide the ability to transmit the science data to the earth. Also, having two orbiters provides redundancy for the transmission of scientific data.

F.3.2.3 Reliability

To ensure reliability, redundancy is in place for all critical mission elements. The threshold mission allows for one lander to be successful. If only one lander is successful, then the threshold will be satisfied. For further details see section F.3.2.1.

F.3.3 Identification of Instrument to Spacecraft Interfaces

This missions design has several science instruments that will be implemented. The radio astronomy instrument is of primary concern. The radio astronomy instrument will be mounted at the end of deployment booms. Cabling will run through the boom to power and communicate with the telescopes. Another science instrument is Phase Detectors that will be mounted inside the lander to compile the data from the telescopes to be sent to the orbiter. The last science instrument is Digital Video Camera that will be mounted with a view of the lunar surface. These Cameras (one on each lander) has a connection to the CD&H module to transmit recordings to the orbiter.

F.3.4 Design Maturity and Heritage of Mission Elements

The design maturity for all aspects of the mission is described in detail in section F.4. The heritage of each element is described in full in J.6. AETHER strives for design maturity.

F.3.5 Essential Trade Studies to be Conducted

AETHER has conducted multiple mission critical trade studies to date found in J.11.5. These are the decision to use one Delta IV or two Atlas V 551s, one or two orbiters, orbiting Earth-Moon L2 or lunar orbit, and rovers or booms to place science packages. A description of the quantitative process that was implemented for the trade studies can be found in section F.3.1.2. This quantitative process helped AETHER decided the best unbiased option for each trade study. The conclusion of which brought AETHER to choose to implement two AV551, two orbiters orbiting around the Moon, and the science packaged placed by booms on the landers. AETHER understands that further trade studies will need to be completed. Further trade studies to come would be how to increase the quality and reliability of the mission on a component level. This can by analyzing options on how to raise TRL levels where applicable. Another trade study could be how to improve the quality of science while maintaining capabilities within budget. This could be done by minimizing moving parts through appropriate options.

F.3.6 Approach to Management

AETHER undergoes a thorough decision making process. All potential concepts are critically reviewed so that no back tracking will be required. In the Pre-Phase A to Phase B parts of the mission design, a consistent review of decisions is required and the mission concept may change if necessary. Once the project is beyond Phase B decisions will be set after a critical review. Test anomalies will be reviewed in a case by case situation in order to determine whether the fault lies with the test or in the element tested. Further testing will be conducted to reduce or eliminate anomalies.

F.3.7 Approach for Handling Special Processes

There are no special processes used in this mission.

F.4 New Technologies/Advanced Developments

This mission will require the development of a few technologies. First and foremost are the radio telescopes. A light weight deployable radio telescope does not yet exist for planetary ground based uses. The current concept for the design of the radio telescopes is made of a flexible mesh and will deploy like an umbrella. The second necessary component of the mission that requires further development is the booms. ATK's CoilABLE Booms have never been used in an environment with a significant gravity. The booms will have to be tested and modified to function in lunar environment. The third mission component that will have to be developed is ALHAT. RAILL will have the capacity to land without ALHAT, but ALHAT will need to be ready for the mission for its first technical demonstration. The Technology Readiness Levels for each subsystem can be found in Table 13. STAR 48-V is not included in the table because it is a fully developed off the shelf product from ATK.

Table 13 Technology Readiness Levels

Orbiter		
Technology/subsystems	**TRL**	**Justification**
Thermal	7	Similar systems used on Lunar Orbiter & LRO
Propulsion/ACS	6	Exists not used
Comm/CD&H	7	Similar systems used on Lunar Orbiter & LRO
Power	7	Similar systems used on Lunar Orbiter & LRO
Structure	7	Similar systems used on Lunar Orbiter & LRO
Science	N/A	none

Lander		
Technology/subsystems	**TRL**	**Justification**
Thermal	7	Similar systems used on LRO
Propulsion/ACS	6	Thrusters developed
ALHAT	5	Some ground testing done (GENIE)
Comm/CD&H	6	Technology is used on LRO
Power	7	Standard solar arrays and batteries
Battery	7	
Structure	6	JPL Polygon spacecraft design
Telescope Placement	5	ATK boom used in spacecraft not on landers
Launching/Landing Attenuation	7	Same materials used in other spacecraft
Science	5	Unbuilt
Radio Telescope Array	5	Requires further development

F.5 Assembly, Integration, Test and Verification

Once the spacecraft is built at Lockheed Martin (LM) in Boulder, Colorado, all testing will be concluded at Marshall Space Flight Center (MSFC) in Huntsville, Alabama. At the current time, the complete list of testing is not finished but will be completed once a trade study is complete. Certain tests are a given for spaceflight, though. Thermal-vacuum testing will be done at MSFC in the environmental testing facility to ensure that the thermal components will accurately heat and cool as needed in the vacuum of space. Testing done on all mechanical objects will be done to verify that the design will work. Vibrational testing will also be done for all three axes to verify that nothing in the payload package will break loose during and delta v maneuvers. Once the trade study is complete, a more finalized list of the testing done on the payload will be made to ensure its ability to be safely launched and utilized for the mission.

Using NASA's defined safety requirements, the group-defined requirements based on the official ones will be checked and verified to ensure that all previously defined requirements have been met. This will be done periodically during the process of defining and planning the mission.

F.6 Schedule

AETHER has completed a living scheduling agenda adhering to major deliverables, launch vehicle integration, Instrument development and major review dates, critical path identification, and launch readiness.

F.6.1 Gantt Chart

Adhering to the multiple items due, AETHER developed a Gantt Chart to show items to be worked on with respect to time. Also included are the roles and responsibilities of who is champion of specific items. The Gantt Chart is Figure 22 provided on the next page. Phase A is to be completed April 28, 2011. After which, Phase B will begin to examine concept and technology deployment. This phase is projected be completed by December 30, 2012. Phase C will follow to analyze the final design and fabrication. Phase D will begin after Phase C (July 30th, 2016). This phase will include the assembly of the mission system and testing upon that system. Phase E (beginning August 15th, 2017) will be the start of the implantation of the meeting. The first launch vehicle will lift off August 15th, 2017 and the second launch vehicle is scheduled to lift off September 15th, 2017. Each launch vehicle will travel to the Moon, land on the Moon, place the radio astronomy, and collect necessary science information to be sent to Earth. This whole process phase will end August 15th, 2022 (five years from the lift off date). The mission closeout will be August 16th, 2022.

Activity	Start Date	Duration (days)	End Date	16-Aug-10	28-Apr-11	30-Dec-12	30-Dec-14	30-Jul-16	15-Aug-17	16-Aug-22
Pre-Phase A: Preliminary analysis	16-Aug-10	255	28-Apr-11							
Concept Studies	16-Aug-10	255	28-Apr-11							
Phase A: Preliminary analysis	28-Apr-11	612	30-Dec-12							
Concept and Technology Development	28-Apr-11	612	30-Dec-12							
Phase B: Concept Definition & Refinement	30-Dec-12	730	30-Dec-14							
Preliminary Design and Fabrication	30-Dec-12	730	30-Dec-14							
Phase C: Full-scale development	30-Dec-14	578	30-Jul-16							
Final Design and Fabrication	30-Dec-14	578	30-Jul-16							
Phase D: Assembly, Integration & Test	30-Jul-16	381	15-Aug-17							
System Assembly	30-Jul-16	153	30-Dec-16							
Test	30-Dec-16	228	15-Aug-17							
Phase E: Mission operations	15-Aug-17	1826	15-Aug-22							
1st Launch:	15-Aug-17	1	16-Aug-17							
Travel to Moon	15-Aug-17	3	18-Aug-17							
Lander to Moon	18-Aug-17	1	19-Aug-17							
Place Arrays	19-Aug-17	1	20-Aug-17							
Scientific Information to Earth	20-Aug-17	1821	15-Aug-22							
2nd Launch:	15-Sep-17	1	16-Sep-17							
Travel to Moon	15-Sep-17	3	18-Sep-17							
Lander to Moon	18-Sep-17	1	19-Sep-17							
Place Arrays	19-Sep-17	1	20-Sep-17							
Scientific Information to Earth	20-Sep-17	1790	15-Aug-22							
Phase F: Disposal	15-Aug-22	1	16-Aug-22							
Mission Closeout	15-Aug-22	1	16-Aug-22							
						Key:	Length of Entire Phase			
							Length of Sub-System Phase			

Figure 22 Mission Schedule

F.6.2 Sub-System Roles and Responsibilities

Table 14 below tells who on AETHER's UAHuntsville's engineering team is responsible for each aspect of the engineering design. This is provided to ensure every individual knows specific systems they will be assigned to. This will help breakdown work specifically, evenly, efficiently throughout the team.

Table 14 Lander Subsystem Roles

Category	Person Assigned to Category
Thermal	David Moore and Jamison McAllister
Propulsion	Garrett Gammon
CD&H	Garrett Gammon, Megan Beattie, and James Pearson
Power	James Pearson
Structures	Matthew Wright and Clayton Pannell
Systems Eng	Sam Bennett
Management	Joel Grissom

G. Management

G.1 Management Approach

The Project Manager (PM) for Radio Astronomy on the Moon is Joel Grissom. The PM is affiliated with The University of Alabama in Huntsville, UAH, which acts as the governing organization with the university president at the head. The PM and Principal Investigator (PI), Heather Meyer, will have direct oversight of the mission team. The PM will be responsible for reporting mission progress to the Mission Manager within the organization of UAH.

The PM and PI will communicate on a management level. The Chief Engineer (CE), Garrett Gammon, and the Lead Systems Engineer (LSE), Sam Bennett, will directly report to the PM. The Co-Investigators (Co-I), Ryan Wilkie and Jesica Trucks, will report directly to the PI. The CE and PI will communicate on technical aspects of the mission. Other organizations will report to either the PM, CE, or LSE as seen in Figure 23.

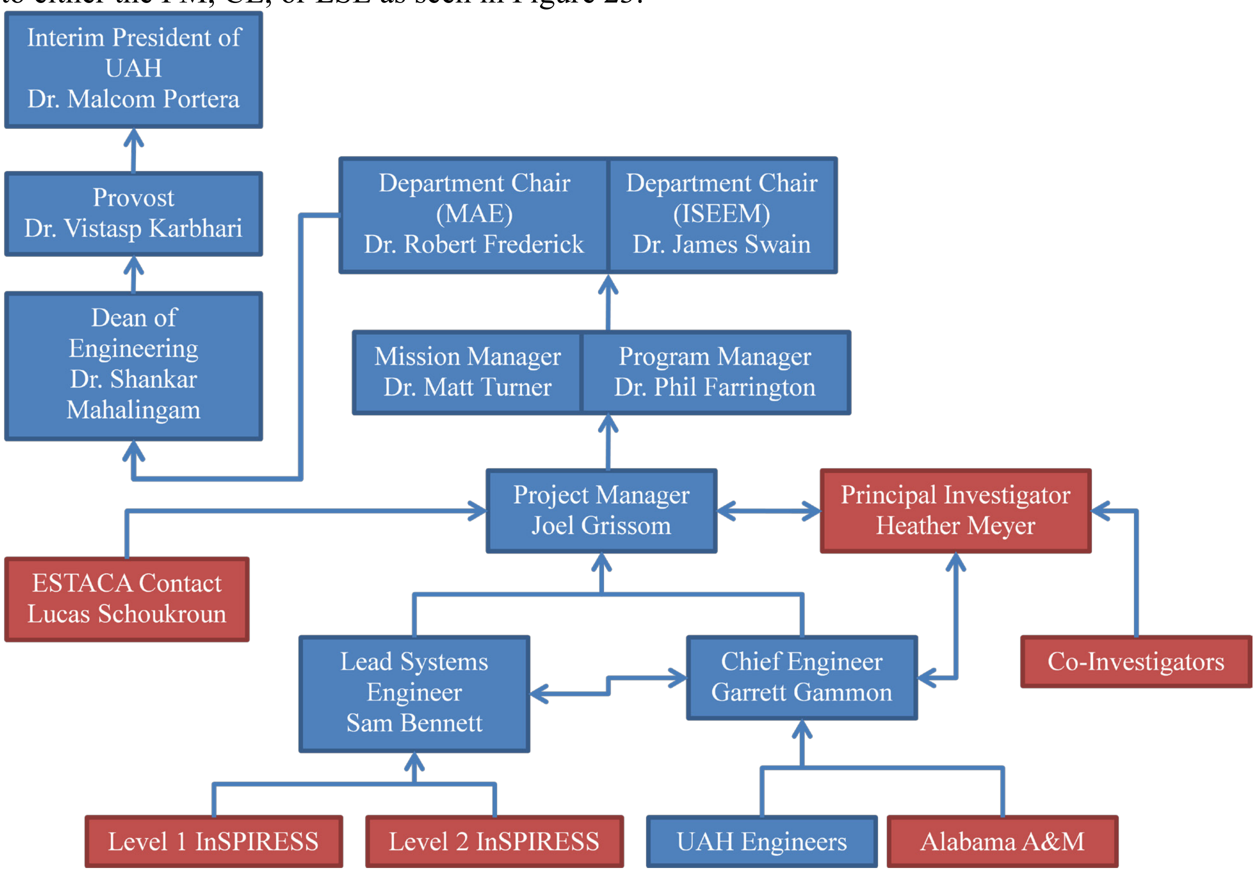


Figure 23 Organizational Structure

The UAH team makes all mission critical decisions given science requirements from the principal investigator. The Chief Engineer has control of decision making for technical aspects of the mission and reports these decisions to the PM. The Project Manager has control of decision making pertaining to scheduling, costs, and risk for the mission. Any conflicts between decisions made by the CE and PM are settled by the PM. Each partnering organization is given leeway for their decision making techniques so long as they meet the requirements given to them in their respective Interface Control Documents (ICD). However all decision making process and conclusions must be reported and validated by the PM before implementation.

G.2 Roles & Responsibilities

G.2.1 Governing Institution- The University of Alabama in Huntsville

The University of Alabama in Huntsville has been developing an Integrated Product Team (IPT) program for over a decade. IPT is in place to put students from various universities together in order to have the capacity to develop mission proposals. UAH also has a history of relationship between professors and NASA. Dr. Phillip Farrington, Dr. Matt Turner, and Dr. P.J. Benfield are providing guidance and oversight for this mission.

G.2.2 Principal Investigator- Heather Meyer

As an astronomy minor, Heather is aware of the importance of this mission, and she has a great wealth of knowledge regarding the implications of a radio telescope array outside of Earth's interference. As a geologist and geomorphologist, she has studied the Moon for three years with her mentor, Dr. Cassandra Runyon, who is currently planning a sample return mission to the Moon. As a former Co-Investigator for a sample return mission to Mars, she has the advantage of understanding what constitutes a well-organized and practical mission. As a graduate of the LeaderShape Institute, she also have a lot of experience with teamwork and leadership. All of this makes Heather uniquely suited to the responsibilities of the P-I, including implementing and executing the selected investigations, managing a team of scientists, maintaining open lines of communication with the engineering team, and communicating our proposed science in a clear and concise manner.

G.2.3 Project Manager- Joel Grissom

Joel Grissom is a senior in Mechanical and Aerospace Engineering at UAH. As such he has finished all but a few courses required for a degree. As Project Manager his primary responsibility is oversight of the technical and programmatic implementation of the project within the allotted resources.

G.2.4 Lead Systems Engineer- Sam Bennett

The LSE is responsible for interface management with internal and external partners. The LSE is also responsible for mission cost analysis, decision analysis, requirements management, etc. Samuel also helped with overall mission design, management approach, logical decomposition, and product integration. Samuel currently works as an Army contractor through the Systems Management and Production Center (SMAP) where he incorporates upper management initiatives into business processes to ensure traceability and organizational success. Samuel frequently uses Systems Engineering tools and techniques in daily work processes to serve in a support role to multiple strategic planning and implementation projects.

G.2.5 Chief Engineer- Garrett Gammon

As Chief Engineer, Garrett Gammon oversees the implementation of the project elements and the technical design of the lander. He is a senior in Mechanical and Aerospace Engineering at the University of Alabama in Huntsville and is able to utilize the knowledge and experience from previous classes in performing his duties as Chief Engineer.

G.3 Rick Management

The PI and PM have compiled a list of significant foreseeable risks involved with RAM. These risks can be found in Table 16. This table, adapted from the IPT Venus in Situ Explorer proposal, lists the risk, root cause, mitigation process, consequences rating, and likelihood rating. The two ratings are from 1 to 5 and applied to the risk matrix found in NASA/SP-2007-6105. The risk matrix is Figure 24. The definitions for a rating of 1 through 5 are taken from the IPT Venus in Situ Explorer proposal and found in Table 15.[12,13]

Table 15 Risk Assessment Scores

	Likelihood	Consequences
5	Near Certain to Occur	Catastrophic
4	Highly likely to occur	Critical
3	Likely to Occur	Moderate
2	Not Likely: Improbable	Marginal
1	Impossible	Negligible

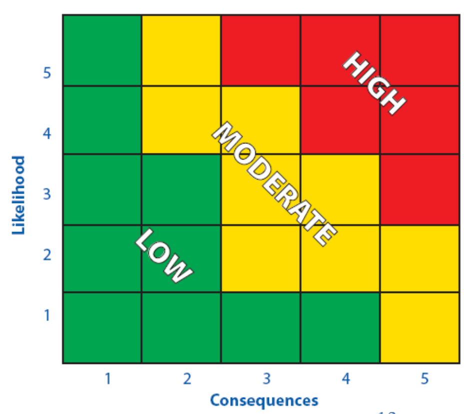


Figure 24 Risk Matrix[13]

Table 16 Significant Risk Mitigation

Risk	Cause	Mitigation	Consequences	Likelihood
Only one successful launch	Limited budget or key system failure	Threshold mission to only require one launch vehicle	Original 5 Mitigated 2	Original 3 Mitigated 2
One dish deployment per landing site minimum not reached	Deployment mechanism damaged during landing or deployment	Threshold mission can still be attained. Redundancy on number of telescopes required per landing site.	Original 3 Mitigated 2	Original 4 Mitigated 2
Severed power connection to instruments	Wiring not deployed along boom.	Redundancy on number of telescopes required per landing site.	Original 3 Mitigated 2	Original 4 Mitigated 3
Data/Comm Failure	Antenna damage or CPU failure	Two sites to record data Redundant CPU's	Original 4 Mitigated 4	Original 3 Mitigated 1
ALHAT Failure	Software malfunction or integration failure	Significant room for landing error due to large craters. All standard landing components in place.	Original 3 Mitigated 2	Original 3 Mitigated 3
DRO contribution delaying mission	ESTACA funding issues	Frequent communication on scheduling. Contribution cut-off date for UAH control of DRO. Cost Reserve.	Original 4 Mitigated 4	Original 2 Mitigated 1
Inaccurate TRL on certain systems or subsystems causing mission to go over budget	Lack of component definition and research	Cost Reserve	Original 2 Mitigated 2	Original 4 Mitigated 2

Scheduling and cost risks will be mitigated by margins and reserves. The PM will be responsible for determining the appropriate release of all margins and reserves.

G.4 Cooperative Arrangements

ESTACA will provide a contribution in form of the twin Data Relay Orbiters. This relationship is and will continue to be governed by an Interface Control Document. ESTACA is an engineering college with many ties to the professional world. ESTACA provides degrees in both space engineering and aerospace engineering. Protoflight testing of the orbiters will be done by ESTACA to ensure mission worthiness. A cost reserve is factored in to the cost analysis of the mission should ESTACA not be able to fulfill its role.

Alabama A&M will provide the cavitating venturi valve as part of TALL's propulsion system. A&M's engineering program has close ties with NASA and has the full capacity to manufacture components. The valve will be provided by A&M for testing with the entire propulsion subsystem. This component factors in with RAILL for cost.

H. Cost and Cost Estimating Methodology

H.1 Cost Model

AETHER used a provided spreadsheet (Hamaker)[14] to analyze the cost estimate for the entire mission. At one time used by NASA for cost estimating of previous missions, this spreadsheet allowed AETHER to input relevant information that characterized the proposed mission. While the cost estimator was beneficial, the estimator made available was very basic and extrapolated data that was unknown. AETHER compensated for this by allowing room for error and by not over-riding any pertinent inputs.

AETHER estimates both landers to cost $954.46 million and both orbiters to cost $438.16 million. Every instrument that is used for the proposed mission is referenced within these cost estimates and is implemented into the total cost. This gave the total cost estimate for the proposed mission as $1.40 billion (FY10 dollars). The original spreadsheet contained a 25% cost reserve, which was changed to 30%. The total calculations include cost reserve.

As both launch vehicles were given to AETHER, those costs were not incorporated (although always considered for any other factor). The Deep Space Network is accounted for the in the cost estimation analysis. The solids (STAR 48V) are bought commercially off the shelf and are incorporated into the cost estimation analysis. The first STAR 48V costs $3.5 million and the second STAR 48V costs $3.25 million (information provided by Dr. Turner). In the case of the baseline of two launch vehicles, the total STAR 48V cost is $6.75 million.

This estimate is over the $800 million budget because costs could not be cut any further without adding risk or complexity to the proposed mission. Section H.2 and H.3 explains further why the estimate is over $800 million. To keep responsible cost levels, AETHER allocated proper TRL levels, masses for all elements, and power for all elements.

H.2 Model Inputs and Outputs

The full description of the inputs and outputs of the cost model is provided in J.11.1.

The Table 17 briefly explains the cost comparison from one launch vehicle (threshold) and two launch vehicles (baseline).

Table 17 Total Cost Comparison

	One Atlas V 551	Two Atlas V 551s	
Orbiter(s) Cost	359.856	438.155	(in Millions of Dollars)
Lander(s) Cost	887.774	954.456	(in Millions of Dollars)
Solid Rocket Motor(s) (STAR 48V)	3.5	6.75	(in Millions of Dollars)
Total Cost (Including cost reserve)	1.251	1.399	(in Billions of Dollars)

When considering mass allocation for one launch vehicle, the total mass for one lander is 1809 kilograms and 720 kilograms for one orbiter. This number was doubled in the cost analysis for two launch vehicles; therefore, 3618 kilograms for two landers and 1440 kilograms for two orbiters. The total power generated was 127 watts for the landers and 50 watts for the orbiters. The design life is for the mission is 5 years or 60 months.

When justifying TRL level in cost for the landers, AETHER took the average of all TRL levels explained in section F.4. The average came to be 6.375, which was rounded conservatively to 6. When justifying the TRL level in cost for the orbiters, AETHER chose 6 as it was the lowest subsystem level analyzed in section F.4.

H.3 Cost Risks

AETHER compensates for risk by allocating conservative measures into the cost analysis. This is completed by not cutting spending dramatically just for the sake of cutting costs. Costs must only be cut appropriately without adding unneeded risk. The estimated cost can be lowered further, but at additional risk. For example, the calculated size of the Government Project Office that is needed for the missions comes to a total of 104 people (69 for the landers and 35 for the orbiters). If AETHER cut the calculated number of Government people to 60 for the landers and 30 for the orbiters, the total would be $895.45 million for the landers and $417.45 million for the orbiters. This total would come to be $1.32 billion. Personnel can be decreased to cut costs; however, AETHER did not change this number as the team feels at least 104 people will be necessary to support this mission throughout its life cycle. Another way to lower costs is by decreasing scheduled duration for Project Phases C and D. In reference to the Gantt chart, AETHER understands that lowering schedule duration further would create unneeded strain and risk into the proposed mission.

I. Acknowledgement of Student Collaboration

I.1 Education and Public Outreach (EPO)

The AETHER Principal Investigator, Heather Meyer, understands the NASA SMD requirements for E/PO and is committed to carrying out a core E/PO program that meets the goals described in the *Explanatory Guide to the NASA Science Mission Directorate Educational and Public Outreach Evaluation Factors* document. AETHER will submit an E/PO plan with the Concept Study Report if this proposal is selected.

I.2 Student Collaboration

AETHER interfaced with Innovative Student Project for the Increased Recruitment of Engineering and Science Students (InSPIRESS) Level 1 teams based from Grissom High School and Sparkman High School. AETHER chose these teams so they may become interested in UAHuntsville's Engineering program. AETHER is very excited to work with both teams from InSPIRESS Level 1 and shall assist them in their scientific efforts. The Level 1 InSPIRESS Grissom High School team provides an experiment that will be attached to the orbiters to measure ground effects on the Moon. The Level 1 InSPIRESS Sparkman High School team provides an experiment measuring the solar effects on electronics through the use of CCD's and measuring solar winds through the use of a Solar Sail.

Both Level 1 InSPIRESS teams have developed excellent science experiments. The experiments proved to be ingenious and well developed. Ideally AETHER would include both experiments in the mission. However, only one experiment can be chosen. AETHER has chosen to include Sparkman High School's experiment in Radio Astronomy on the Moon.

Both experiments would provide valuable information about the Moon so this decision was primarily due to risk and reliability. AETHER decided that Sparkman High School's experiment has a smaller chance for failure. Placing the experiment on the lander will provide it with assured power and communication for its duration.

I.3 Grissom High School Report

INSPIRESS Report Final Draft

The INSPIRESS team from Virgil I. Grissom High school is named Team Dark Side of Virgil I. Grissom High School, and the name of the experiment is G.R.I.S.S.O.M., or Great Research in Space Seismometers On the Moon.

With the Moon at such a close proximity to Earth, one would think that we would know almost everything about the Moon's composition. However, little is known about the interior of the Moon and its potential for resources and scientific research. The purpose of the experiment is to map the inside of the moon by observing Moonquakes, lunar seismic activity, by sending a number of test cartridges to the moon to observe and record these Moonquakes.

These cartridges (Figure 1) are spherical and are each made of an aluminum alloy. They contain a Colibrys SF1500S accelerometer, a data storage & transmitter device (CPU), an antenna, and a space-rated 6V battery as a power source, as seen in figure 2. In order to protect this equipment from the deceleration of the impact, the spheres are surrounded by a metal honeycomb lattice (figure 1), which compresses to gradually absorb the impact as the cartridges roll.

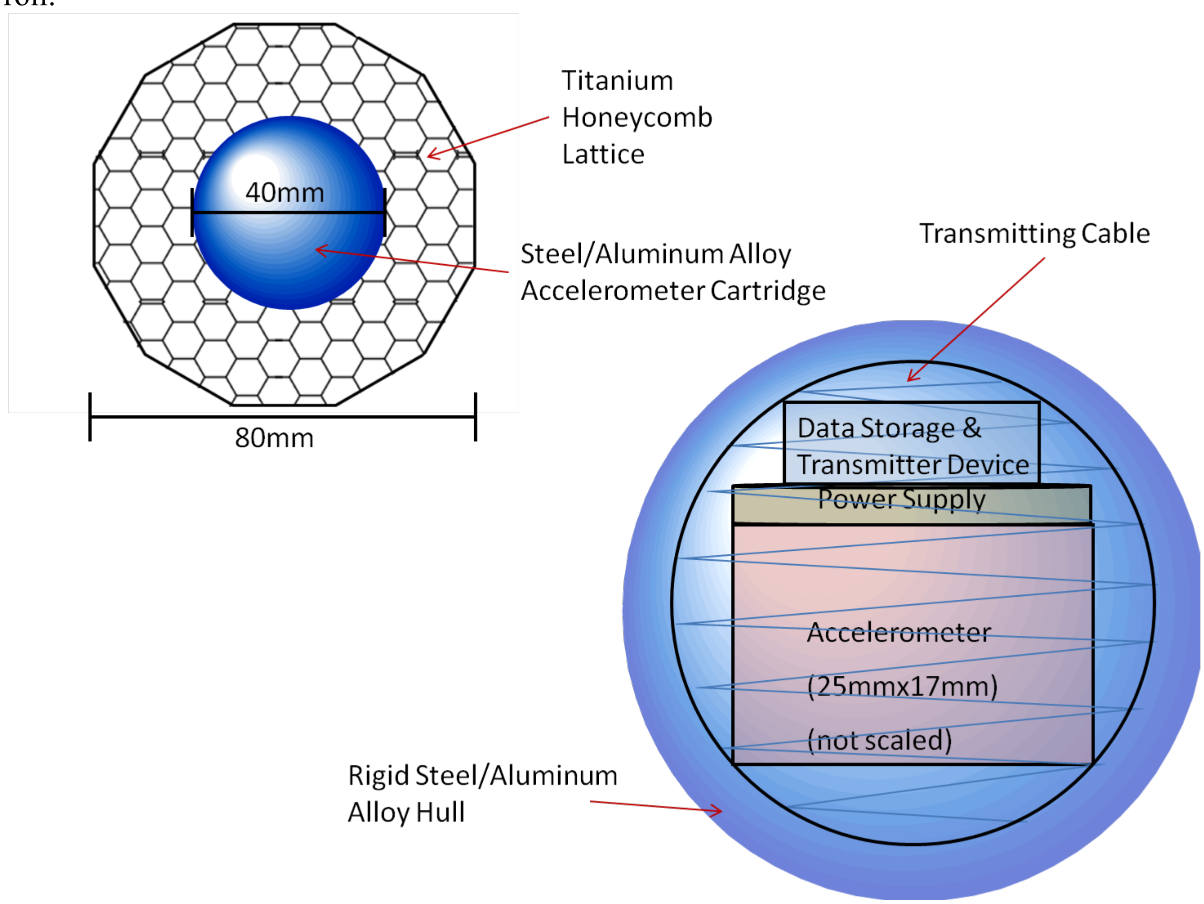


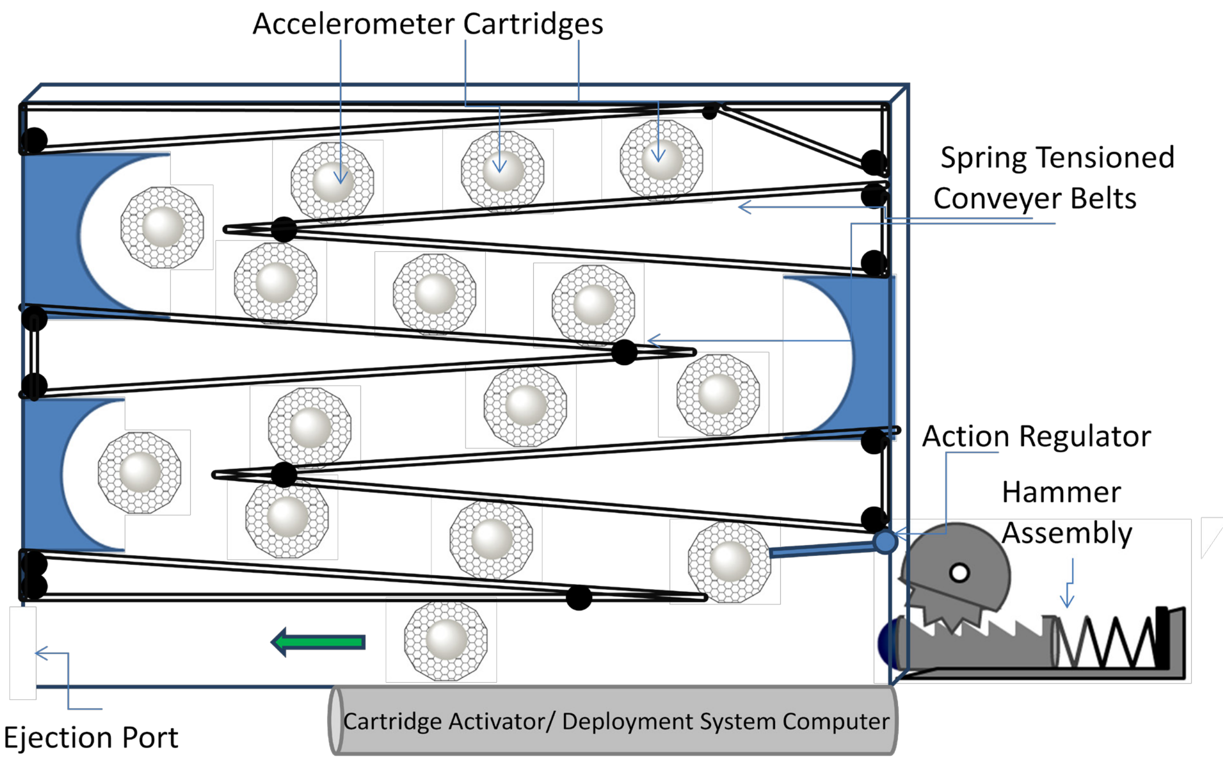
Figure 1. – Accelerometer Cartridge Layout

Figure 2. – Cartridge Equipment Table

Instrument	Purpose Relative to Scientific Question
Accelerometer	Measures the *g* force of Moonquakes as they occur
CPU	Stores and relays available data to orbiter
Antenna	Used in conjunction with the transmitter
Space Batteries(non rechargeable)	Provide power for Accelerometer and CPU; periodically shuts off to save power; supplies power when pinged by deployment device. The batteries are non-rechargeable, and the mission will end when the batteries can no longer supply power.

The team plans to achieve its goal by sending approximately 10 to 15 cartridges of 0.75 kg each to the Moon to observe and record Moonquakes. The team wishes to place these vehicles on a deployment system on the Moon-side of the orbiter. Deployment Device 2 is a box containing a series of spring tensioned conveyer belts with the cartridges lined up against each other. On command, the action regulator stops and releases the cartridges, which are being forced toward the action by the conveyer belts. The hammer is cranked back by the gear system and then shot forward, propelling the cartridges toward the surface of the Moon. The action once again propels the cartridges out of the ejection port by means of a hammer and gear system. These systems provided the most amount of volume for the cartridges with the least amount of electrically powered parts, thus making the systems easier to make and operate. Once the orbiter is in orbit, the cartridges will be deployed at regular intervals (depending on the number of cartridges) equally around the equator of the Moon. By releasing the vehicles at regular intervals instead of all at once, the cartridges cover larger area and are able to collect a broader spectrum of data. Once on the Moon, the vehicles will send data recorded during the Moonquakes back to the orbiter.

Deployment Device 2



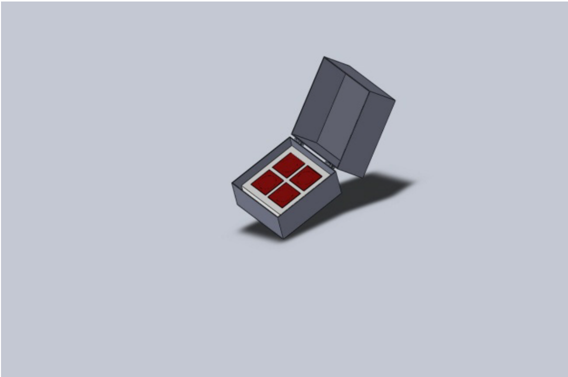
Once launched from the orbiter, the cartridges will either bury themselves beneath the Moon's surface, or skip and roll to a stop on the Moon's surface. As previously described the honeycomb lattice will compress on impact, absorbing the initial impact. The equipment in the cartridges will lay dormant in the Moon's regolith until a Moonquake occurs. The vibrations of the Moonquakes will cause the cartridges to activate, and the time and magnitude of each Moonquake will be recorded. The information will then be transmitted to the orbiter. After the orbiter sends the information back to Earth, the data in coordination with location of the equipment (as determined by the orbiter pinging the cartridges with ultra high frequency) will be used to determine the speed of the vibrations as well as the various distortions caused by traveling through the Moon's core.

G.R.I.S.S.O.M. extensively enhances the science of the mission. The volume of each cartridge allows for 10 to 15 to fit inside the deployment system. The more cartridges deployed for testing, the high the chances that a usable number will survive. The payload requires approximately 0.16 m^3 and approximately 14.00 kg in return for superior scientific knowledge gained by the experiment. With the data gathered, knowledge of the interior of the moon will greatly increase.

I.4 Sparkman High School Report

Measuring Intense Solar Lunar Effects' (MISLE) mission will provide NASA with relevant data in the areas of protective shielding and the potential use of solar winds. The two scientific questions that this project hopes to answer are: Is there a better protective shielding that can be used to protect sensitive electronics from solar radiation? Could a Solar Sail be implemented to help sustain life on the far side of the moon by capturing the natural energy found in solar winds? MISLE looks to answer these questions through two separate experiments. The first being a pad of four charged couple devices (CCD), each with a different protective covering, that will measure the amount of radiation blocked from the individual device by each of the protective coverings. The second experiment is measuring the force placed on a Solar Sail on the far side of the Moon.

Figure 26: Magnetometer Box



The whole assembly is relatively meager having a mass around 3-5 kilograms. Both experiments are contained inside of a 15 inches x15 inches x 13.5 inches box (See Figure 1). Upon arriving on the far side of the Moon the Magnetometer Box will open up using a small Direct Current (DC) brushed 19 millimeter motor drawing 1.5 watts during operation. The Magnetometer box will then open. The Solar Radiation Shielding Pad (SRSP) is affixed to the upper half for maximum exposure. Then, in the lower half, there will be the magnetometer with the plasma monitor mounted on a telescoping boom. In full deployment with both experiments in full operation the dimensions of the magnetometer box will be 31 inches x 15 inches x 48 inches. During start up the estimated power consumption will be 7 watts, after start up the usage will drop to 5.5 watts. The data will be transmitted through a tether between the MISLE box and the main Lander.

Figure 25: Plasma Monitor

Solar Sail Force Testing (SSFT)

Purpose and Function

The purpose of using a magnetometer with plasma monitor (see Figure 2) is to measure the solar wind and evaluate the strength to see if a Solar Sail could be used in future missions to provide a source of energy. An example being a windmill that generates energy off of Earth wind the Solar Sail could be designed to provide energy on the Moon. As the wind turns the mill, energy is stored. Solar Sails work the same way. As the particles hit (or even move) the sail energy is collected and stored. The reason that this device was chosen over a Solar Sail was to insure that there would be less complexity and the experiment would have less failure points. The plasma monitor will not have to unfurl to begin measurement and will not get torn.

Design

The testing of the force of the ions that would hit a Solar Sail will be read by a magnetometer with plasma monitor**.** The magnetometer will be mounted on top of a telescoping boom that will be placed in the other half of the box. The boom will project to a length around 27 inches to properly test the ions. The mass is .7 kilograms and requires from .7 watts to 1.5 watts depending on when in use.

How It Works

Upon the opening of the containment box, the boom will begin to deploy to its full height. At the full height or extension, the magnetometer with plasma display will be activated and begin measuring the ions accordingly. The magnetometer with plasma monitor works by measuring the ions that hit the device itself. This allows a data point that can be used to measure the intensity of the solar wind.

Figures of Merit

Table 1: Figures of Merit (FOMs)

Categories	Plasma Reader	Feather Sail
Mass(9)	9(81)	5 (45)
Volume(3)	3 (9)	2 (6)
Complexity(1)	5 (5)	3 (3)
Power(3)	4 (12)	5 (15)
Size(9)	5 (45)	4 (36)
Durability(3)	5 (15)	1 (3)
Totals	167	108

Solar Radiation Shielding Pad (SRSP)

Purpose

The purpose utilizing CCDs on the far side of the moon is to collect radiation data. This data can display how radiation affects technology on the far side of the Moon. If the need arose to colonize the Moon, this would be valuable information because it would enable the creation of technology that is less venerable to the radiation. With new technology, new material can be adapted to be best suited for the amount of radiation that the far side sees every year. The CCDs are ideal for measuring this radiation because they are small and efficient.

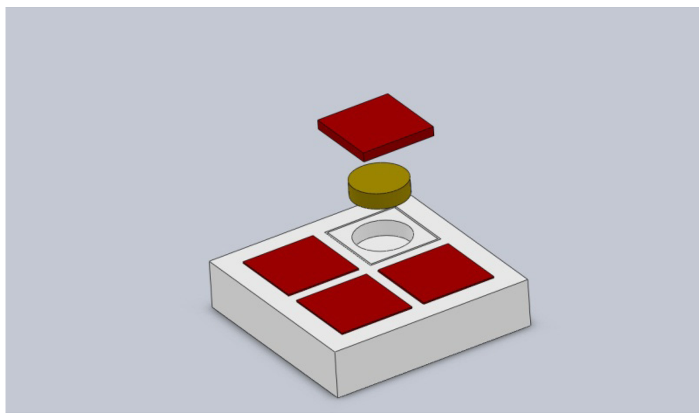
Design

The design of the CCD is relatively small with dimension being at 3.5 inches in diameter. There will be four of the round CCD's placed in a 13 inches by 13 inches box equally spaced apart. The CCD's will be covered with three experimental coverings. The final CCD will be covered with the aeroshell covering that is used in the Mars' rovers in today's rovers.

How it Works

The Charged Coupled Device (CCD) is an instrument that takes a picture of the area around it. This picture is not a traditional picture like one on a digital camera. Instead, it takes a picture of the radiation in the area and shows how well each of the protective coatings blocks radiation.

Figure 27: Solar Radiation Shielding Pad (SRPS)



J. Appendices

J.1 Table of Proposal Participants

Team Member	Email Address	Phone Number	Member Role
The University of Alabama in Huntsville			
Joel Grissom	fjg0001@uah.edu	(615) 848-8033	Project Manager
Garrett Gammon	gjg0002@uah.edu	(205) 913-6182	Chief Engineer
Sam Bennett	kp0288@gmail.com	(256) 757-5142	Lead Systems Engineer
Megan Beattie	beattim@uah.edu	(651) 387-7819	Supporting Engineer
Jamison McAllister	mcallij@uah.edu	(256) 599-8699	Supporting Engineer
David Moore	moorewd@uah.edu	(256) 318-2601	Supporting Engineer
Clayton Pannell	wcp0001@uah.edu	(256) 690-3219	Supporting Engineer
James Pearson	jfp0003@uah.edu	(205) 790-4337	Supporting Engineer
Matthew Wright	mrw0002@uah.edu	(816) 520-7844	Supporting Engineer
Brittany Gibbs	brg0001@uah.edu		Technical Writer
Maria Munn	mam0010@uah.edu		Technical Writer
College of Charleston			
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Ryan Wilkie	rdwilkie@edisto.cofc.edu		Co-Investigator
Jesica Trucks	jmactrucks@gmail.com		Co-Investigator
Austin High School			
Lindsay Boonarkat	whoselineisit04@aol.com	(256) 227-4249	Industry Partner
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Lucas Schoukroun	lucas.schoukroun@estaca.eu		International Partner
Antoine Bercault	antoine.bercault@estaca.eu		International Partner
David Langlois	david.langlois@estaca.eu		International Partner

J.2 Letters of Commitment and Support

Date: April 15, 2011

To: Heather Meyer College of Charleston
Principle Investigator

From: AETHER-UAHuntsville

Re: Letter of commitment

We acknowledge that we have been identified by name as the design team for the proposed project entitled "Radio Astronomy on the Moon", that you are submitting in response to the Announcement of Opportunity, Discover 2010, NNH10ZDA007O, and that we intend to carry out all responsibilities identified for us in this proposal. We understand that the extent and justification of our participation as stated in this proposal will be considered during peer review in determining in part the merits of this proposal. If you have any questions, please feel free to contact us at any time.

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Garrett Gammon
Chief Engineer and Communications, UAHuntsville
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Sam Bennett
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James Pearson
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jfp0003@uah.edu, 205-790-4337

Meagan Beattie
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David Moore
Thermal, UAHuntsville
moorewd@uah.edu, 256-318-2601

Jamison McAllister
Thermal Engineer, UAHuntsville
mcallij@uah.edu, 256-599-8699

UAHuntsville
THE UNIVERSITY OF ALABAMA IN HUNTSVILLE

April 21, 2011

Forrest Joel Grissom
Project Manager
Aerospace Engineering Technologies Heading Extrasolar Research (AETHER), Team A
The University of Alabama in Huntsville
Mechanical and Aerospace Engineering Dept.
N274 Technology Hall
Huntsville, AL 35899

Dear Mr. Grissom,

The University of Alabama in Huntsville is pleased to formally acknowledge your team's design for a Radio Astronomy on the Moon (RAM) mission as part of NASA's Discovery Announcement of Opportunity Program. We believe, should your design be selected, the science gained from this mission will not only provide a greater understanding of our solar system, but will help to distinguish our institution as a premier center for engineering education, research, and technological development. With this said, The University of Alabama in Huntsville is fully committed to support your team in its current and future endeavors. Best wishes on being selected!

Sincerely,

Matthew W. Turner, Ph.D.
Integrated Product Team Mission Manager
The University of Alabama in Huntsville

INTEGRATED PRODUCT TEAM PROJECT OFFICE »
Shelby Center 157 | 301 Sparkman Drive | Huntsville, AL 35899
T 256.824.2976 | F 256.824.4322 | http://ipt.uah.edu

To Whom It May Concern:

"I acknowledge that I have been identified for institutional support of the proposed project entitled “Radio Astronomy on the Moon” on behalf of the College of Charleston, that Heather Meyer is submitting in response to the Announcement of Opportunity, #NNH10ZDA007O. I understand that the extent and justification of institutional support as stated in this proposal will be considered during peer review in determining in part the merits of this proposal. If you have any questions, please feel free to contact me at any time."

Signature: ______________________________

Jon Hakkila, Chair and Professor
Department of Physics and Astronomy
College of Charleston

J.3 Resumes

The following pages contain resumes for each member of AETHER.

Megan Beattie

(651)-387-7819
beattim@uah.edu

Current Address
201 Water Hill Road F3
Madison, Alabama 35758

Permanent Address
1160 Cushing Circle #120
Saint Paul, Minnesota 55018

CITIZENSHIP

U.S.

TECHNICAL SKILLS

MATLAB, Mathcad, Linux, Solid Edge, ProE, technological networking

EDUCATION

The University of Alabama in Huntsville — **Huntsville, Alabama**

Bachelor of Science in Engineering with a concentration in Mechanical Aerospace

Expected to graduate August 2011

WORK EXPERIENCE

April 2010 – Present — **Propulsion Research Center** — **Huntsville, Alabama**

Undergraduate Research Assistant

- Assist with research in high energy density physics

Jan 2010 – Present — **Jet Propulsion Laboratory** — **Pasadena, California**

Solar System Ambassador

- Familiarize the public with NASA missions, specifically interplanetary missions

Oct 2006 – Present — **Von Braun Astronomical Society** — **Huntsville, Alabama**

VBAS Intern, Planetarium Presenter

- Present specific space, science, and astronomy topics for educational shows

Aug 2007 – July 2010 — **Teledyne Brown Engineering** — **Huntsville, Alabama**

Engineering Co-op

- Assisted in Manufactured Products under NASA's Cargo Mission Contract (CMC) for Shuttle and other specific NASA, DoE, and DoD contracts

RESEARCH

NASA Microgravity University – *Performance of Heat Pipes as a Function of G-Level*
Experiment performed on NASA's C-9 "Weightless Wonder" Spring 2007

HONORS AND AWARDS

NASA Group Achievement Award for Ares IX (2010), ASME Service Recognition Award (2009), VBAS Student Leadership Award (2008, 2009, 2010)

Samuel Scott Bennett

(256)-797-5142
112 Joe Phillips Road
Madison, AL 35758
kp0288@gmail.com

CITIZENSHIP U.S.

TECHNICAL SKILLS Proficient in Microsoft Excel, PowerPoint, Word, and Minitab

EDUCATION

The University of Alabama in Huntsville **Huntsville, AL**

Bachelor of Science in Engineering with a concentration in Industrial and Systems Engineering

GPA: 3.667/4.0 Expected graduation: August 2011

Organized and headed a successful 140 hour volunteer project for the Huntsville Botanical Gardens (initially planned to be over 195 hours)

Calhoun Community College

Pre Bachelor of Science in Engineering
GPA: 3.8/4.0

WORK EXPERIENCE

Dec 2009 - Present **20-20 Leadership** **Huntsville, AL**

- Army Contractor internship through the UAHuntsville SMAP Center
- Implemented strategic planning ideals to incorporate upper management initiatives into business processes to ensure organizational success
- Implemented System Engineering tools and techniques in organizational processes

Mar 2008 – Mar 2010 **Applebee's** **Madison, AL**

- Server

CLEARANCE Secret Clearance provided through The University of Alabama Huntsville security office in February 2010.

HONORS AND AWARDS Earned top BSA rank of Eagle Scout with bronze, silver, and gold palms, Super Transfer Scholarship Award, Government Scholarship Award, Summer Scholarship Award

AFFILIATIONS Member of Tau Beta Pi Engineering Honor Society, Treasurer of The University of Alabama Huntsville's Institute for Industrial Engineers, Member of The National Society of Leadership and Success

Garrett Gammon

(205)-913-6182
gjg0002@uah.edu

Current Address	Permanent Address
1000 Airport Rd Sw	915 County Hwy 11
Huntsville, AL 35802	Hayden, AL 35079

CITIZENSHIP U.S.

TECHNICAL SKILLS Solid Edge, MATLAB, Mathcad, Microsoft Office

EDUCATION

The University of Alabama in Huntsville **Huntsville, AL**

Bachelor of Science in Engineering

GPA: 3.89/4.0, Expected graduation August 2011

WORK EXPERIENCE

May 2008 – Present **GATR Technologies** **Huntsville, AL**

Mechanical Engineer Coop

- Test and calibrate outgoing antenna systems to meet FCC regulations
- Pre-deployment and packaging of antenna systems
- Test and evaluate new product additions and modifications

May 2006 – May 2008 **B&R Systems** **Tarrant, AL**

Printer Technician

- Printer and copier repair and delivery

CLEARANCE Type of Secret Clearance GATR Technologies Nov 2009

HONORS AND AWARDS UAHuntsville College of Engineering Dean's List

AFFILIATIONS AIAA student section, ASME, Tau Beta Pi, Reformed University Fellowship

Forrest Joel Grissom

(615)-848-8033
fjg0001@uah.edu

Current Address	Permanent Address
704-H John Wright Dr.	3535 Long Shadow Ct.
Huntsville, AL 35805	Murfreesboro, TN 37129

CITIZENSHIP U.S.

TECHNICAL SKILLS Software Used: Solid Edge, NX, MATLAB, Simulink, Mathcad, MicrosoftOffice

EDUCATION

The University of Alabama in Huntsville **Huntsville, AL**

Bachelor of Science in Engineering with a concentration in Aerospace Engineering

GPA: 3.34/4.0 in major, Expected graduation May 2011

- Projects: Radio Astronomy on the Moon (ongoing), Collaborative Design (ongoing), Marshmallow Gun

WORK EXPERIENCE

May 2010-Aug 2010 **UAHuntsville** **Huntsville, AL**

Undergraduate Research Assistant

- Assist in development of robotics program at UAHuntsville
- Control the motion of various robots using MATLAB, Simulink, and XPCtarget

July 2007-Present **Two Men and a Truck** **Huntsville, AL**

Driver

- Load and unload goods.
- Interface with customer and meet requirements
- Perform cost analysis

HONORS AND AWARDS

National Society of Leadership and Success, Sigma Alpha Pi UAHuntsville chapter

Two Men and a Truck Safe Driver Award-2008

AFFILIATIONS

Member of the National Aerospace Engineering Honor Society, Sigma Gamma Tau UAHuntsville Chapter

Jamison McAllister

(256) 599-8699
mcallij@uah.edu
612 College Rd.
Fyffe, AL 35971

CITIZENSHIP U.S.

TECHNICAL SKILLS

MATLAB, MathCAD, Solid Edge, NX

Scale Gemini Capsule, Radio Astronomy on the Moon

EDUCATION

The University of Alabama in Huntsville
Huntsville, AL Bachelor of Science in Engineering with a concentration in Mechanical Engineering

GPA: 2.7/4.0 (3.3/4.0 in major), Expected graduation date: December 2011

WORK EXPERIENCE

August 2004 – Present Manley Landscaping Owens Crossroads, AL

Landscaper

- Grade Work, Sod, Shrubs, etc.
- Retaining Walls
- Stone Work
- Pavers

April 2009 – Present B&B Poultry Powell, AL

Equipment Maintenance

- Servicing and Maintaining Equipment
- General Shop Labor

William David Moore

(256) 318-2601
moorewd@uah.edu

415 Jack Coleman Dr
Huntsville, AL 35805

CITIZENSHIP

United States of the America

TECHNICAL SKILLS

Adept in the following Software/Applications:

- Solid Edge, C++, MATLAB, MathCad, Patran
- Microsoft Office (Word, Excel, PowerPoint)

EDUCATION

The University of Alabama in Huntsville — **Huntsville, AL**

Bachelor of Science in Engineering with concentration in Mechanical Engineering

GPA: 2.122/4.0, Expected graduation Summer 2011

WORK EXPERIENCE

August 2007 – Present — **Morris, Conchin, King, and Hodge** — **Huntsville, AL**

Runner

- File documents with the court, serve documents on opposing counsel, pick up documents from the court, pick up documents from wherever I am sent to get them from, and run errands for lawyers and secretaries of the firm

January 2006 – August 2007 — **University Center of UAH** — **Huntsville, AL**

Information Desk Assistant

- Answered questions concerning academics and activities for the UAH campus to refer individuals to various departments and/or offices for specific information
- As a Game Room Attendant: Provided change, cleaned the area, maintained the equipment and acted as a resource for Game Room activities
- As a Setup Crew Member: Maintained the overall cleanliness of the building, setup for meeting rooms, and other various duties assigned

AFFILIATIONS

Brother of the Sigma Nu Fraternity, Inc. since January 2006

Heather Meyer

(843) 810-4630
Meyer.HMM@gmail.com

Current Address	Permanent Address
4785 Arco Lane	433 Carriage Lane
N. Charleston, SC, 29418	Charleston, SC 29464

CITIZENSHIP

U.S

TECHNICAL SKILLS

Writing grant proposals, Technical writing, Making formal presentations, Laboratory skills (i.e. ICP-MS, MARS5, SEM, Ro-Tap, LPA), Windows, Microsoft Word, Excel, and Access, Personnel and project management, Interpretation of aerial images and maps, Interpretation of remote sensing data, Rock and mineral identification

EDUCATION

College of Charleston — **Charleston, SC**

Bachelor of Science in Geology

Minor: Astronomy, GPA: 3.431/4.0 (3.358/4.0 in major), Expected graduation May 2012

College of Charleston — **Charleston, SC**

Bachelor of Arts in Religious Studies

Minor: European Studies, GPA: 3.431/4.0 (3.456/4.0 in major), Expected graduation May 2012

WORK EXPERIENCE

Jan 2011 – Present — **Conference Assistant & Presenter** — **Charleston, SC**

College of Charleston

- Plan and conduct educational conferences for the South Carolina Space Grant

Jan 2010 – Present — **Teaching Assistant** — **Charleston, SC**

College of Charleston Department of Geology & Environmental Geosciences

- Assist students in lab work, setup lab, grade assignments, maintain and update grade book

May 2010 – Aug 2010 — **Summer Conference Hall Director** — **Charleston, SC**

College of Charleston Department of Residence Life & Housing

- Manage staff, oversee operations in five on-campus residences

Aug 2007 – May 2010 — **Resident Assistant** — **Charleston, SC**

College of Charleston Department of Residence Life & Housing

- Designed and conducted various programs to improve residents' experience

AFFILIATIONS

Planetary Society Member, Association of Environmental & Engineering Geologists (AEG) Member, National Society of Collegiate Scholars, Member of the Higdon Student Leadership Center, Graduate of the LeaderShape Institute

James Pearson

205-790-4337
Jfp0003@uah.edu

1500 Sparkman Drive Apt 44G
Huntsville, AL, 35816

CITIZENSHIP United States Citizen

TECHNICAL SKILLS MathCAD, MatLAB, Solid Edge

EDUCATION **The University of Alabama in Huntsville**
Huntsville, AL

Bachelor of Science in Engineering

GPA: 3.180/4.0 Expected graduation December 2011

PROFILE

- Expressing ideas
- Facilitating group discussion
- Identifying problems
- Imagining alternatives
- Gathering information
- Solving problems
- Cooperating

Jesica L. Trucks

(803) 347-6411
jltrucks@g.cofc.edu; jmactrucks@gmail.com

Mailing Address
P.O. Box 62532
North Charleston, SC 29419

Permanent Address
2998 Buckfield Dr.
North Charleston, SC 29406

CITIZENSHIP United States

TECHNICAL SKILLS LaTeX, Inkscape, IDL, Mircosoft Office

EDUCATION

College of Charleston **Charleston, SC**

Bachelor of Science in Physics

GPA: 2.570/4.0 (2.526/4.0 in major), Expected graduation May 2012

Bachelor of Science in Astrophysics

GPA: 2.570/4.0(2.404/4.0 in major), Expected graduation May 2012

WORK EXPERIENCE

Jan 2010 - Present **College of Charleston** **Charleston, SC**

Teaching Assistant

- Gather and set up equipment for intro level astronomy classes, take down and put away equipment after the lab ends.
- Help professor answer student questions during the course of the lab.

Sep 2010 – Present **Patriot's Point** **Mount Pleasant, SC**

Astronomy Merit Badge Teacher

- Teach 2 astronomy merit badge classes to boy scouts staying on the USS Yorktown
- During the course of class teach the required material for a merit badge in a classroom setting
- Also an observing portion take the scouts on the flight deck and teach them about the constellations and how to use them for direction.

AFFILIATIONS Society of Physics Students

Ryan Wilkie

(803) 260-7919
ryandwilkie@gmail.com

Current Address	Permanent Address
92 Wentworth St.	123 Carlsbad Ct.
Charleston, SC, 29424	West Columbia, SC 29170

CITIZENSHIP United States

TECHNICAL SKILLS Proficient with Wolfram Mathematica®

EDUCATION **College of Charleston** **Charleston, SC**
Bachelor of Science in Astrophysics
Expected graduation May 2013

PROFILE

- Works well in a team
- Able to take the lead
- Gives best effort, even if uninterested in a task
- Takes responsibilities seriously
- Gives assistance to others when needed
- Enthusiastic about astronomy and physics

HONORS AND AWARDS Eagle Scout

AFFILIATIONS Society of Physics Student at College of Charleston, Boy Scouts of America

Matt Wright

(816) 520-7844
mrw0002@uah.edu

118 Preswick Place NW
Huntsville, AL 35806

CITIZENSHIP

U.S.

TECHNICAL SKILLS

Software: MS Office, Nastran, Patran, MATLAB, Mathcad, NX, Solid Edge, Solid Works

Specialized Equipment: Lathes, load cells (and other materials testing apparatuses)

EDUCATION

The University of Alabama in Huntsville **Huntsville, AL**

Bachelor of Science in Engineering with a concentration in Mechanical Engineering

GPA: 2.9/4.0, Expected graduation: August 2011

PROFILE

- Projects: Design Build Fly, Radio Astronomy Mission
- Chief Engineer on Design Build Fly team, showing leadership skills.
- Involved in martial arts since childhood, building honor and integrity.
- Experience in materials testing environment.
- Some experience in machining and composites.
- Disciplined, quick learner with excellent problem solving capabilities.

HONORS AND AWARDS

Winner, January 2008, PLM Software Student Design Contest (monthly award)

Winner, 2008, PLM Software Student Design Contest (yearly award)

Feature, March 2009, Siemens PLM Software international calendar (only student chosen)

First in Class, July 2009, Temple Classic Push-Pull (powerlifting competition)

AFFILIATIONS

Secretary, Formula SAE, The University of Alabama in Huntsville

Member, Southern Powerlifting Federation

J.4 Planetary Protection Plan

Planetary Protection Plans are implemented for missions to other solar system bodies to ensure there is no biological or non-biological contamination transmitted to and from other solar system bodies. Missions are classified for planetary protection categories based on the mission types, target bodies, and science goals. This mission falls under Category I according to NASA Directive NPR 8020.7G, *Biological Contamination Control for Outbound and Inbound Planetary Spacecraft.* Lunar missions are not considered at risk for contamination due to the extreme environmental conditions. More than 2,000 lunar sample returns have been brought back with no evidence of past or present biological activity found. Missions classified as Category I are not required to have a formal planetary protection plan according to the aforementioned NASA Directive.

J.5 Discussion of End of Mission Spacecraft Disposal Requirements

Though NASA missions are notorious for functioning far beyond their intended capabilities, there is a plan that corresponds with NASA's End of Mission (EOM) spacecraft requirements. This plan includes firing what propellant we have leftover on the DRO's at a delta V of 20 m/s to impact the moon and end the life of the orbiters. This will be done away from the telescope array sites since a future orbiter could possibly use the arrays to conduct further research. The land-based objects will remain where they are, nonfunctioning.

J.6 Master Equipment List

R.A.I.L.L. (Radio Astronomy Instrument Lunar Lander)					
Subsystem/Component	**Unit Mass**	**Quantity**	**Total Mass**	**Total Power**	**Heritage/ Additional Info**
Propulsion			67		
MR-80B	7.94	3	23.8	45	Viking, MSL
MR-50S	0.68	12	8.2	47	Viking, Voyager, 680 have flown
Helium Tank	8.8	2	17.6	0	
Propellant Tank	2.3	2	4.6	0	
Required hardware and tubing	6		6	0	
Propellant			384		Hydrazine
Pressurant			2.3		Helium
Attitude Determination and Control			8.9		
Star Tracker	0.4	1	0.4	2	
MIMU	4.5	1	4.5	22	40 units launched successfully
TRN	4	1	4	10	
Thermal Control			39.1		
12-Layer MLI			2		For entire spacecraft
Heater	2.5	4	10	28	
Louvered Radiator	24	1	24		
Heat Pipes	.34	9	3.1		
Structures					
Command & Data Handling			11.6		
RAD750	0.6	1	0.6	10	
Data Storage	3	1	3	3	
Cabling			8		
Power			126.2		
Solar Cells	.01	21	.210	1388	
Batteries	126	1	126	126	
Communications			10.2		
TWTA	3.0	1	3.0	104	LRO
0.7m Parabolic Dish	3.1	1	3.1	9	
RF Components	4		4	5	
Science Payload			256.2		
Radio Telescope	72	3	216	1.5	
Dipole Antenna	11	3	33		
LEAM	7.2	1	7.2	6.6	Apollo
Mobility System					

Booms	151.8		151.8		
STAR 48V			**1907.5**		
Propellant			1753.4		13% offload
Inert			154.1		

D.R.O. (Data Relay Orbiter)					
Propulsion			**152**		
Pressurant Tank (COPV)	13.2	2	26.4		Calculated Hardware
Propellant Tanks, Fuel (w/ PMD)	22.7	2	45.4		Calculated Hardware
Propellant Tanks, Oxidizer (w/ PMD)	15.7	2	31.4		Calculated Hardware
Required hardware and tubing			30.4		
RCS Thruster (22 N, 5 lbf thrust)	0.6	12	7.2		Aerojet MR-106E 22N
AMBR Thruster (890 N, 200 lbf thrust)	5.5	2	11		To be flight ready in 2014
Propellant			**1405.9**		NTO & MMH
Pressurant			**2.1**		Helium
Attitude Determination and Control			**5.3**		
Star Tracker	0.4	2	0.8	4	
MIMU	4.5	1	4.5	22	40 units launched successfully
Thermal Control			**50.8**		
12-Layer MLI			.2		
Louvered Radiator	17.3	2	34.6		
Heater	4	4	16	110	
Structures					
Command and Data Handling			**28.8**		
RAD750	0.6	1	0.6	10	
Data Storage	8.2	1	8.2	12	
Cabling			20		
Power			**60.0**		
Solar Panels	13.9	4	55.6	1000	
Batteries			4.4	836	
Communications			**10.2**		
TWTA	3.0	1	3.0	104	LRO
0.7m Parabolic Dish	3.2	1	3.2	9	

RF Components	4		4	5	

J.7 Heritage

Combined with MEL in J.6.

J.8 List of Abbreviations and Acronyms

AETHER	Aerospace Engineering Technologies Heading Extrasolar Research
ALHAT	Autonomous Landing Hazard Avoidance Technology
AV551	Atlas V 551
CE	Chief Engineer
Co-I	Co-Investigator
DRO	Data Relay Orbiter
DSN	Deep Space Network
EOM	End of Mission
ESTACA	Ecole Supérieure des Techniques Aéronautiques et de Construction Automobile
InSPIRESS	Innovative Student Project for the Increased Recruitment of Engineering and Science Students
JPL	Jet Propulsion Lab
LEAM	Lunar Ejecta and Meteorites
LM	Lockheed Martin
LV	Launch Vehicle
LVA	Launch Vehicle Adapter
LOI	Lunar Orbit Insertion
LSE	Lead Systems Engineer
MSFC	Marshall Space Flight Center
MLI	Multi-layer insulation
NASA	National Aeronautics and Space Administration
PI	Principal Investigator
PM	Project Manager
POC	Point of Contact
PSR	Payload Spacer Ring
RAILL	Radio Astronomy Instrument Lunar Lander
SEO	Science Enhancement Option
TCaV	Throttling Cavitating Venturi Valve
TRL	Technology Readiness Level
UAH	The University of Alabama in Huntsville
WEB	Warm Electronics Box
WMAP	Wilkinson Microwave Anisotropy Probe

J.9 List of References

1. http://map.gsfc.nasa.gov/media/060915/060915_CMB_Timeline75nt.jpg
2. VLA nrau.edu
3. Google Earth
4. NASA
5. Approach Phase ΔV Considerations for Lunar Landing, Cohanim, Fill, Paschall, Major, Brady; Charles Stark Draper Laboratory: IEEEAC paper #1195, Version 3, Updated Dec 19, 2008
6. http://www.atk.com/capabilities_multiple/documents/Coilable_Booms.pdf
7. http://www.atk.com/capabilities_space/documents/ATK_Catalog_May_2008.pdf
8. deepspace.jpl.nasa.gov. 2010. About the Deep Space Network. 11 November 2010 <http://deepspace.jpl.nasa.gov/dsn/>
9. nasa.gov. 2010. About the Goddard Space Flight Center. 30 March 2010. <http://www.nasa.gov/centers/goddard/about/index.html>
10. nasa.gov. 2011. Marshall Space Flight Center: About Marshall. 29 Match 2011. <http://www.nasa.gov/centers/marshall/about/index.html>
11. Simons, Rainee, and Dale A. Force. High-Efficiency K-Band Space Traveling-Wave Tube Amplifier for Near-Earth High Data Rate Communications. Cleveland, Ohio: Glenn Research Center, 2010
12. IPT Venus in Situ Explorer proposal
13. NASA/SP-2007-6105
14. Hamaker, Joseph W. 2006. *Improving the Predictive Capability of Spacecraft Cost Models by Considering Non-Technical Variables: a Dissertation.* Thesis Ph.D. The University of Alabama in Huntsville.

J.10 Infusion Plan for NASA-Developed Technology

This mission does not propose the use of any NASA-developed technology.

J.11 Calculations

J.11.1 Cost Analysis Tables

The full cost analysis for the landers is as follows:

Cost Item	Input	Units & Explanations	Justification
Enter Spacecraft Bus + Instruments Total Dry Mass	**3618**	KG	Two landers at 1809 KG each
Enter Spacecraft Total Power Generation Capacity (LEO Equivalent)	**127**	W LEO equivalent flux	Total power Generation Capacity
Enter Design Life in Months	**60.0**	Months	5 years life cycle
Enter Number of Science Organizations	**1.0**	Count (Enter zero for projects with no science or science organization involvement)	CoC
Enter Apogee Class	**4.0**	LEO=1, HEO/GEO=2, beyond GEO=3, Planetary=4	Planetary
Enter Maximum Data Rate Requirements Relative to SOTA Expressed as Percentile	**50%**	Kbps requirement relative to the state-of-the-art for the ATP date expressed as a percentile where 0%=very low, 50%=SOTA, 100% is maximum	SOTA
Enter Test Requirements Class	**3.0**	Less than average testing=1, Average=2, More than average=3, Extensive=4, Very extensive=5	

Enter Requirements Stability Class	**2.0**	Very low volatility=1, Low=2, Average=3, High=4, Very high volatility=5	
Enter Funding Stability Class	**2.0**	Stable funding=1, Some instability=2, Significant instability=3	
Enter Team Experience Class [Derived from Price Model; used with permission from Price Systems LLP]	**3.0**	Extensive experience=1, Better than average=2, Average (mixed experience)=3, Unfamiliar=4 [Ref: Price Model]	Mixed knowledge of team members lead by experienced teachers
Enter Formulation Study Class	**2.0**	Formulation study (1=Major, 2=Nominal, 3=Minor)	Normal
Enter New Design Percent	**70%**	Simple mod=30%, Extensive mod=70% (average), New=100%	Only thing new is the telescopes and ALHAT; therefore, used the average
Enter ATP Date Expressed as Years Since 1960	**51**	Years elapsed since 1960	
Regression Model Result	$331.5	DDT&E + TFU (Phases C/D/E) in Millions of 2004 Dollars including fee, excluding full cost	Done by database
Enter Technology Readiness Level (TRL) Penalty Factor	**6.0**	Refer to NASA TRL scale (TRL 6 is nominal)	Everything is off the shelf except for the telescopes

Enter Platform Factor [Derived from Price Model; used with permission from Price Systems LLP)	**2.20**	Platform factor (Airborne Military=1.8, Unmanned Earth Orbital=2.0, Unmanned Planetary=2.2, Manned Earth Orbital=2.5, Manned Planetary=2.7) [Ref: Price Model]	Unmanned Planetary
Enter Functional Complexity Factor	**To Be Added Later**	To be added later	Not in current cost model
Subtotal (Non Full Cost Subtotal)	**$420.4**	Subtotal (Millions of 2004 Dollars including fee)	Done by database
Calculated Size of the Government Project Office (Project Office Only--Excludes Government Functional Line/Laboratory Labor)	**68.6**	Civil service annual full time equivalents (FTE's)	Done by database
Enter Override of Calculated Government FTEs (or leave zero to accept calculated size of project office)	**0.00**	Civil service annual full time equivalents (FTE's)	Did not override
Final Estimate of the Size of the Government Project Office and other Oversight (excludes government non-oversight labor which is included in subtotal above)	**68.6**	Civil Service Full Time Equivalents (FTE's)	Done by database
Enter Civil Service Loaded Annual Labor Rate Including Center and Corporate G&A	**$280,000**	Thousands of 2004 Dollars	Did not override
Calculated Project Phase C/D Schedule Duration (Excludes O&S Phase E)	**70**	Months	
Enter Override of Calculated Phase C/D Schedule Duration (or leave zero to accept calculated duration)	**32**	Months	Congruent with Gantt scheduling
Final Estimate of the Project Phase C/D Schedule Duration	**32**	Months	
Calculated Cost of the Government Project Office	**$112.6**	Millions of 2004 Dollars	Done by database
Government Service Pool Use Intensity Factor	**4**	1=Minimum use of service pools, 2=Less than average, 3=Average, 4=More than average, 5=Significantly more than average	Done by database

Calculated Cost of Government Service Pool Use	**$50.5**		Done by database
Enter Override of Calculated Cost of Government Service Pool Use (or leave zero to accept calculated service pool cost)	**$0.0**		Done by database
Final Estimate of the Cost of Government Service Pool Use	**$50.5**		Done by database
Subtotal (2004$)	**$583.5**		Done by database
Ground System	**$52.5**		Done by database
Enter Override of Calculated Ground System Cost	**$0.0**		Done by database
Final Estimate of the Cost of Ground System	**$52.5**		Done by database
Subtotal (2004$)	**$636.0**		Done by database
Enter Launch Services Cost	**$0.0**		Done by database
Enter Cost Reserves	**$190.81**		30% cost reserve
Total (2004$)	**$826.8**		Done by database
Total (2010$)	**954.45592**		Multiplied by 1.15435 to get 2010 dollars. We were told to do this.

The full cost analysis for the orbiters is as follows:

Cost Item	Input	Units & Explanations	Justification
Enter Spacecraft Bus + Instruments Total Dry Mass	**1440**	KG	Two orbiters at 720 KG each

Enter Spacecraft Total Power Generation Capacity (LEO Equivalent)	**50**	W LEO equivalent flux	2 RAD 750s at 20 watts total, 15 watts for comm, and 15 watts for thermal
Enter Design Life in Months	**60.0**	Months	5 years life cycle
Enter Number of Science Organizations	**1.0**	Count (Enter zero for projects with no science or science organization involvement)	CoC
Enter Apogee Class	**4.0**	LEO=1, HEO/GEO=2, beyond GEO=3, Planetary=4	Planetary
Enter Maximum Data Rate Requirements Relative to SOTA Expressed as Percentile	**50%**	Kbps requirement relative to the state-of-the-art for the ATP date expressed as a percentile where 0%=very low, 50%=SOTA, 100% is maximum	SOTA
Enter Test Requirements Class	**3.0**	Less than average testing=1, Average=2, More than average=3, Extensive=4, Very extensive=5	
Enter Requirements Stability Class	**2.0**	Very low volatility=1, Low=2, Average=3, High=4, Very high volatility=5	
Enter Funding Stability Class	**2.0**	Stable funding=1, Some instability=2, Significant instability=3	
Enter Team Experience Class [Derived from Price Model; used with permission from Price Systems LLP]	**3.0**	Extensive experience=1, Better than average=2, Average (mixed experience)=3, Unfamiliar=4 [Ref: Price Model]	Mixed knowledge of team members lead by experienced teachers

Enter Formulation Study Class	**2.0**	Formulation study (1=Major, 2=Nominal, 3=Minor)	Normal
Enter New Design Percent	**30%**	Simple mod=30%, Extensive mod=70% (average), New=100%	Simple modification because most everything in element has been used before
Enter ATP Date Expressed as Years Since 1960	**51**	Years elapsed since 1960	
Regression Model Result	$123.8	DDT&E + TFU (Phases C/D/E) in Millions of 2004 Dollars including fee, excluding full cost	Done by database
Enter Technology Readiness Level (TRL) Penalty Factor	**6.0**	Refer to NASA TRL scale (TRL 6 is nominal)	Most everything in element has already been used
Enter Platform Factor [Derived from Price Model; used with permission from Price Systems LLP)	**2.20**	Platform factor (Airborne Military=1.8, Unmanned Earth Orbital=2.0, Unmanned Planetary=2.2, Manned Earth Orbital=2.5, Manned Planetary=2.7) [Ref: Price Model]	Unmanned Planetary
Enter Functional Complexity Factor	**To Be Added Later**	To be added later	Not in current cost model
Subtotal (Non Full Cost Subtotal)	**$157.0**	Subtotal (Millions of 2004 Dollars including fee)	Done by database
Calculated Size of the Government Project Office (Project Office Only--Excludes Government Functional Line/Laboratory Labor)	**34.5**	Civil service annual full time equivalents (FTE's)	Done by database

Enter Override of Calculated Government FTEs (or leave zero to accept calculated size of project office)	**0.00**	Civil service annual full time equivalents (FTE's)	Did not override
Final Estimate of the Size of the Government Project Office and other Oversight (excludes government non-oversight labor which is included in subtotal above)	**34.5**	Civil Service Full Time Equivalents (FTE's)	Done by database
Enter Civil Service Loaded Annual Labor Rate Including Center and Corporate G&A	**$280,000**	Thousands of 2004 Dollars	Did not override
Calculated Project Phase C/D Schedule Duration (Excludes O&S Phase E)	**45**	Months	
Enter Override of Calculated Phase C/D Schedule Duration (or leave zero to accept calculated duration)	**32**	Months	Congruent with Gantt scheduling
Final Estimate of the Project Phase C/D Schedule Duration	**32**	Months	
Calculated Cost of the Government Project Office	**$36.2**	Millions of 2004 Dollars	Done by database
Government Service Pool Use Intensity Factor	**4**	1=Minimum use of service pools, 2=Less than average, 3=Average, 4=More than average, 5=Significantly more than average	Done by database
Calculated Cost of Government Service Pool Use	**$18.8**		Done by database
Enter Override of Calculated Cost of Government Service Pool Use (or leave zero to accept calculated service pool cost)	**$0.0**		Done by database
Final Estimate of the Cost of Government Service Pool Use	**$18.8**		Done by database

Subtotal (2004$)	**$212.1**		Done by database
Ground System	**$19.1**		Done by database
Enter Override of Calculated Ground System Cost	**$0.0**		Done by database
Final Estimate of the Cost of Ground System	**$19.1**		Done by database
Subtotal (2004$)	**$231.2**		Done by database
Enter Launch Services Cost	**$79.0**		Done by database
Enter Cost Reserves	**$69.36**		30% cost reserve
Total (2004$)	**$379.6**		Done by database
Total (2010$)	**438.154861**		Multiplied by 1.15435 to get 2010 dollars. We were told to do this.

J.11.2 ATK Booms

ATK COILable Booms Calculations:

$L_e := 17.5m$ Extended Length

$L_s := .02 \cdot L_e = 0.35\,m$ Stowed Length

$m_t := 72kg$ Telescope total mass (mounting hardware included)

$G_m := 1.6\dfrac{m}{s^2}$ Gravity on the Moon

$W_{t.m} := m_t \cdot G_m = 115.2\,N$ Weight of Telescope on Moon

$T_e := W_{t.m} \cdot L_e = 2.016 \times 10^3 \cdot N \cdot m$ $\quad T_e = 1.784 \times 10^4 \cdot lbf \cdot in$ Extended length Torque/Moment

Boom radius of 7" gives bending strength of 1000lbf*in: $Bending_7 := 1000lbf \cdot in$ $\quad Radius_7 := 7in$

Boom radius of 5" gives bending strength of 200lbf*in: $Bending_5 := 200lbf \cdot in$ $\quad Radius_5 := 5in$

Gradient: $$M_s := \frac{Bending_7 - Bending_5}{Radius_7 - Radius_5} = 400 \cdot \frac{lbf \cdot in}{in}$$

Boom diameter (to get bending strength = to extended torque):

$$D := \left(\frac{T_e}{M_s}\right) \cdot 2 = 2.266\,m \qquad D = 89.216 \cdot in \qquad D = 7.435 \cdot ft$$

Boom radius of 7" gives weight of 30lbf $\quad W_7 := 30lbf$

Boom radius of 5" gives weight of 15lbf $\quad W_5 := 15lbf$

Gradient: $$M_w := \frac{W_7 - W_5}{Radius_7 - Radius_5} = 7.5 \cdot \frac{lbf}{in}$$

Weight of boom with said radius, on Earth:

$$W_b := \left(\frac{D}{2}\right) \cdot M_w = 334.558 \cdot lbf \qquad W_b = 1.488 \times 10^3\,N$$

Mass: $$m_{boom} := \frac{W_b}{g} = 151.753\,kg$$

Weight on moon:

$$W_{b.m} := m_{boom} \cdot G_m = 54.585 \cdot lbf \qquad W_{b.m} = 242.805 \cdot N$$

Other information about the booms:

Longerons: Cross section of 1.5% of boom diameter $X_{section.longeron} := 0.015 \cdot D = 0.034\,m$

Battens: "Buckled compression members to preload structure". Typically smaller than longeron frame.

Distance b/w battens: 58% of diameter $D_{bw} := 0.58 \cdot D = 1.314\,m$

J.11.3 Thermal Calculations

Definitions of Terms:
F_{se} - View Factor between Spacecraft and Earth
K_a - Factor for the reflection of collimated solar energy from Earth
G_s - Solar Flux
q_{EIR} - Earth emitted IR
q_{MIR} - Lunar emitted IR
a_{moon} – Lunar Albedo
a_{earth} - Earth Albedo
$Radius_m$ - Radius of moon
H – altitude
F_{sm} - View Factor between Spacecraft and Moon
Q_w - Heat dissipated
σ – Stephan-Boltzmann constant
D – Diameter of sphere with surface area equal to Spacecraft/Lander
α_s - Absorptivity
ε_{IR} – Infrared Emissivity
A_R - Area of Radiator
k_{Al} - Thermal Conductivity of Aluminum
k_{MLI} – Thermal Conductivity of MLI
L_{Al} - Thickness of Aluminum radiator panel
L_{MLI} – Thickness of MLI
L_{AlWEB} - Thickness of Aluminum WEB
U – Overall Heat Transfer Coefficient

Temperature Requirements:

Component	Operating Temp(C)	Survival Temp(C)
Batteries	0 to 45	-10 to 25
Computer	-10 to 60	-20 to 100
Hydrazine Tank/lines	15 to 45	5 to 50
Antennae	-40 to 80	-120 to 120
Solar Panels	-150 to 110	-200 to 200
Instruments	-55 to 125	
Solid Rocket Booster	-1 to 37	

Heat Dissipation From Batteries:

$\text{Maxdraw} := 300\text{W}$ $\qquad$ $\text{maxinefficiency} := 7\%$

$\text{Heatdissipated} := \text{maxinefficiency}\cdot\text{Maxdraw} = 21\cdot\text{W}$

Max Spacecraft Surface Temperature In Flight
Spacecraft recieves solar, earth IR and albedo, Moon IR and albedo

Assumptions:
Spacecraft is modeled as an isothermal sphere
Spacecraft is Rotating
Lunar Orbit Altitude=100km
Direct Solar Input
Heat to be rejected form radiator=200W

Values From Elements of Spacecraft Design:

$F_{se} := .283$ $\qquad$ $K_a := .993$ $\qquad$ $G_s := 1371\dfrac{\text{W}}{\text{m}^2}$ $\qquad$ $q_{EIR} := 237\dfrac{\text{W}}{\text{m}^2}$ **(max value)**

$a_{earth} := .35$ $\qquad$ $q_{MIR} := 430\dfrac{\text{W}}{\text{m}^2}$ **(max value)**

Values from Lunar Sourcebook

$a_{moon} := .07$ $\qquad$ $\text{Radius}_m := 1738\text{km}$

Equation 7.25: Elements of Spacecraft Design Textbook (ESD):

$$F_{sm} := \frac{Radius_m^{\ 2}}{\left(Radius_m + H\right)^2} = 0.804$$

$Q_w := 200W$ $\qquad \sigma := 5.67 \cdot 10^{-8} \frac{W}{m^2 \cdot K^4}$ $\qquad D := 5m$

Properties of White Paint: $\qquad \alpha_s := .20 \qquad \varepsilon_{IR} := .92$

From Equation 7.36: Elements of Spacecraft Design Tesxtbook (ESD)

$$T_{max} := \left(\frac{\frac{G_s \cdot \alpha_s}{4} + q_{EIR} \cdot \varepsilon_{IR} \cdot F_{se} + G_s \cdot a_{earth} \cdot \alpha_s \cdot K_a \cdot F_{se} + q_{MIR} \cdot \varepsilon_{IR} \cdot F_{sm} + G_s \cdot a_{moon} \cdot \alpha_s \cdot F_{sm} + \frac{Q_w}{\pi \cdot D^2}}{\sigma \cdot \varepsilon_{IR}} \right)^{\frac{1}{4}} = 311.853\,K$$

$325.329 - 273.15 = 52.179$ **Celsius**

Minimun Spacecraft Surface Temperature In Flight:
Will occur when spacecraft is not in view of earth or sun and recieves no albedo from moon.

Assumptions:
Minimum Heat Dissipation is 1 Watt

$Q_w := 1 \cdot W$ $\qquad F_{sm} = 0.804$ $\qquad q_{MIR} = 430 \frac{W}{m^2}$

From Equation 7.37 (ESD):

$$T_{min} := \left(\frac{q_{MIR} \cdot \varepsilon_{IR} \cdot F_{sm} + \frac{Q_w}{\pi \cdot D^2}}{\sigma \cdot \varepsilon_{IR}} \right)^{\frac{1}{4}} = 279.461\,K$$

$279.463 - 273.15 = 6.313$ **Celsius**

Radiator Size for Worst Case Hot Scenario in Flight:

Assumptions:
No Environmental Heat Input
Upper Temperature Limit for Batteries = 45C
10 Degree Temperature Margin
Maximum Heat Dissipation

$Q_w := 200W$ $\quad \varepsilon_{rad} := .8$ $\quad T_R := (35 + 273.15)K$

From Equation 7.10 (ESD)

$$A_R := \frac{Q_w}{\left(\sigma \cdot \varepsilon_{rad} \cdot T_R^{\,4}\right)} = 0.489 \cdot m^2$$

Maximum Lander Surface Temperature on Moon (Daytime)
Lander recieves Solar Flux, Moon albedo, and Moon IR

Assumptions:
Direct Solar
Direct Lunar Albedo and IR
Max Heat Dissipation

$G_s := 1371 \frac{W}{m^2}$ $\quad q_{MIR} := 430 \frac{W}{m^2}$ $\quad q_{EIR} := 237 \frac{W}{m^2}$ $\quad a_{moon} := .07$

$Q_w := 200W$ $\quad \sigma := 5.67 \cdot 10^{-8} \frac{W}{m^2 \cdot K^4}$ $\quad \alpha_s := .20$ $\quad \varepsilon_{IR} := .92$ $\quad D := 5.m$

From Equation 7.36 (ESD)

$$T_{max} := \left(\frac{G_s \cdot \alpha_s + q_{MIR} \cdot \varepsilon_{IR} + G_s \cdot a_{moon} \cdot \alpha_s + \frac{Q_w}{\pi \cdot D^2}}{\sigma \cdot \varepsilon_{IR}} \right)^{\frac{1}{4}} = 339.322\,K$$

$339.322 - 273.15 = 66.172$ **Celsius**

Minimum Lander Surface Temperature on Moon (Night Time)
Lander Recieves only Moon IR

Assumptions:
View factor between spacecraft and Lunar surface = 0.2
Minimum Heat Dissipation

$q_{MIR} = 430\frac{1}{m^2}\cdot W$ $\quad F_{sm} := .2$ $\quad Q_w := 1W$ $\quad D = 5m$ $\quad \varepsilon_{IR} = 0.92$

From Equation 7.37 (ESD):

$$T_{min} := \left(\frac{q_{MIR}\cdot\varepsilon_{IR}\cdot F_{sm} + \frac{Q_w}{\pi\cdot D^2}}{\sigma\cdot\varepsilon_{IR}}\right)^{\frac{1}{4}} = 197.354\,K$$

$197.354 - 273.15 = -75.796$ **Celsius**

Radiator Size For Hot Case Scenario on Moon:

Assumptions:
No Environmental Heat Input
Upper Temperature Limit for Batteries = 45C
10 Degree Temperature Margin
Maximum Heat Dissipation

$Q_w := 200W$ $\quad T_R := (35 + 273.15)K$

From Equation 7.10 (ESD)

$$A_R := \frac{Q_w}{\left(\sigma\cdot\varepsilon_{rad}\cdot T_R^{\,4}\right)} = 0.489\cdot m^2$$

Radiator Temp based on worst case cold scenario on moon:

Assumptions:
Minimum Heat Dissipation
No Environmental Heat Inputs

$Q_w := 1 \cdot W$

From Equation 7.10 (ESD)

$$T_R := \left(\frac{\frac{Q_w}{A_R}}{\sigma \cdot \varepsilon_{rad}} \right)^{\frac{1}{4}} = 81.942\,K$$

$145.715 - 273.15 = -127.435$ **Celsius**

WEB Calculations:

WEB Dimensions= .4m x .4m x .4m
The WEB will be made of Aluminum and mounted to the radiator. The WEB will be isulated with 10 Layer MLI. Batteries have the smallest temperature range between 0 and 45 Celsius. A ten degree temperature margin is used to give a range of 10 to 35 Celsius.

Assumptions:
Leaks in WEB are neglected

Values from Space Mission Analysis and Design Textbook (SMAD) $k_{Al} := 185.2 \frac{W}{m \cdot K}$ $k_{MLI} := .0004 \frac{W}{m \cdot K}$

$Area_{rad} := .489m^2$ $L_{Al} := \frac{1}{8} in$ $L_{MLI} := 2.33mm$

$L_{AlWEB} := \frac{1}{8} in$ $Area_{WEB} := .16m^2$ **one side**

Equations From Fundamental of Heat and Mass Transfer Textbook:

$$U = \frac{1}{\sum_{i=1}^{n} \text{Resistances}}$$

$$R_1 := \frac{L_{Al}}{k_{Al} \cdot Area_{WEB}}$$

$$R_2 := \frac{L_{AlWEB}}{k_{Al} \cdot Area_{WEB}} \qquad R_3 := \frac{L_{MLIWEB}}{k_{MLIWEB} \cdot Area_{WEB}}$$

$$U := \frac{1}{R_1 + R_2 + R_3} = 0.027 \frac{1}{K} \cdot W$$

Cold Case

$T_1 := (10 + 273.15)K \qquad T_2 := 81K$ **(From Worst Case Cold Condition)**

$$Q_{WEBcold} := U \cdot (T_1 - T_2) = 5.553\, W$$

Hot Case

$T_1 := (35 + 273.15)K \qquad T_5 := 340K$ **(From Worst Case Hot Condition)**

$$Q_{WEBhot} := U \cdot (T_1 - T_5) = -0.875\, W$$

J.11.4 Mission Architecture

Atlas V-551 Throwmass: $m_{throw} := 6105\text{kg}$

Mass allotted to interstaging: $m_{interstage} := 20\text{kg}$

Engine Specifications:

$Isp_{mono} := 230\text{s}$

$Isp_{biprop} := 320\text{s}$

$Isp_{acs} := 208\text{s}$

Average Orbiter Propellant Mass Fraction:

$PMF_o := 0.7$

Accel. due to gravity: $g := 9.81\dfrac{m}{s^2}$

ΔV Budget:

$\Delta V1 := 51\dfrac{m}{s}$ $\Delta V3 := 20\dfrac{m}{s}$ $\Delta V_{acs} := 90\dfrac{m}{s}$

$\Delta V2 := 800\dfrac{m}{s}$ $\Delta V4 := 1755\dfrac{m}{s}$ $\Delta V6 := 161\dfrac{m}{s}$ $\Delta V_{acs2} := 26\dfrac{m}{s}$

Mid-Course Correction:

$$m_{p1} := m_{throw}\cdot\left(1 - e^{\frac{-\Delta V1}{g\cdot Isp_{biprop}}}\right) = 98.382\,\text{kg}$$

$$m_{1pl} := m_{throw} - m_{p1} = 6.007\times10^{3}\,\text{kg}$$

Lunar Orbit Insertion :

$$m_{p2} := m_{1pl}\cdot\left(1 - e^{\frac{-\Delta V2}{g\cdot Isp_{biprop}}}\right) = 1.351\times10^{3}\,\text{kg}$$

$$m_{p2wet} := \frac{m_{p1} + m_{p2}}{PMF_o} = 2.071\times10^{3}\,\text{kg}$$

$$m_{odry} := m_{p2wet} - \left(m_{p2} + m_{p1}\right) = 621.273\,\text{kg}$$

$$m_{2pl} := m_{throw} - m_{p2wet} - m_{interstage} = 4.014\times10^{3}\,\text{kg}$$

De-Orbit Initiation:

$$m_{p3} := m_{2pl}\cdot\left(1 - e^{\frac{-\Delta V3}{g\cdot Isp_{mono}}}\right) = 35.424\,\text{kg}$$

$$m_{3land} := m_{2pl} - m_{p3} = 3.979\times10^{3}\,\text{kg}$$

Braking Burn:

$m_{p4} := 1753.4kg$ from ATK solid catalog

$m_{solid} := 1907.5kg$

$$m_{p4acs} := m_{3land} \cdot \left(1 - e^{\frac{-\Delta V_{acs}}{g \cdot Isp_{acs}}}\right) = 171.674\,kg$$

$$m_{4desc} := m_{3land} - m_{solid} - m_{p4acs} - m_{interstage} = 1.879 \times 10^{3}\,kg$$

Final Approach and Landing:

$$m_{p5} := m_{4desc} \cdot \left(1 - e^{\frac{-\Delta V6}{g \cdot Isp_{mono}}}\right) = 129.44\,kg$$

$$m_{p5acs} := m_{4desc} \cdot \left(1 - e^{\frac{-\Delta V_{acs2}}{g \cdot Isp_{acs}}}\right) = 23.797\,kg$$

$$m_{landed} := m_{4desc} - m_{p5} - m_{p5acs} = 1.726 \times 10^{3}\,kg$$

Total Wet Mass of Lander:

$$m_{acsprop} := m_{p5acs} + m_{p4acs} = 195.471\,kg$$

$$m_{landerprop} := m_{acsprop} + m_{p5} + m_{p3} = 360.334\,kg$$

Allow for 3%reserve and additional 3.5% for loading uncertainties and residual propellant

$m_{propreserve} := m_{landerprop} \cdot .03$ $m_{propunusable} := 0.035(m_{propreserve} + m_{landerprop})$

$$m_{proploaded} := m_{landerprop} + m_{propreserve} + m_{propunusable} = 384.135\,kg$$

$$m_{wetlander} := m_{landed} + m_{landerprop} = 2.087 \times 10^{3}\,kg$$

$$m_{drylander} := m_{wetlander} - m_{landerprop} = 1.726 \times 10^{3}\,kg$$

Tank Sizing:

Average Hydrazine Density: $\rho_{hydrazine} := 1008 \frac{kg}{m^3}$

$$V_{prop} := \frac{m_{proploaded}}{\rho_{hydrazine}} = 0.381 \cdot m^3$$

$V_{0gdev} := .03 \cdot V_{prop}$ $\qquad$ $V_{ullage} := .03 \cdot V_{prop}$

$V_{proptank} := V_{prop} + V_{0gdev} + V_{ullage} = 0.404 \cdot m^3$

Pressurant Requirement:

$P_{prop} := 300psi$ $\qquad$ $P_{press} := 4500psi$

$R_{helium} := 2077.3 \frac{J}{kg \cdot K}$ $\qquad$ $k_{helium} := 1.67$

$$m_{press} := \frac{P_{prop} \cdot V_{prop}}{R_{helium} \cdot 300K} \cdot \left(\frac{k_{helium}}{1 - \frac{P_{prop}}{P_{press}}} \right) = 2.263\,kg$$

$$V_{presstank} := \frac{m_{press} \cdot R_{helium} \cdot 300K}{P_{press}} = 0.045 \cdot m^3$$

J.11.5 Trade Studies

Orbiter Options					
		Rating		(Weight factor) * (Rating)	
Criteria	Weight factor	Orbit Around Moon	Orbit Earth-Moon L2	Orbit Around Moon	Orbit Earth-Moon L2
Mission Cost	6	6	4	36	24
Communication Period	6	3	5	18	30
Number of Array Sites	9	6	3	54	27
Risk	3	6	6	18	18
Complexity of Placement	4	3	6	12	24
Useful payload on ground	5	9	3	45	15
			Total	183	138

1 Delta IV H or 2 Atlas V 551					
		Rating		(Weight factor) * (Rating)	
Criteria	Weight factor	1 Delta IV Heavy	2 Atlas Vs	1 Delta IV Heavy	2 Atlas Vs
Cost	3	6	6	18	18
Throw Mass	9	6	9	54	81
Throw Volume	6	3	6	18	36
Landing Difficulty	6	3	3	18	18
Launch Risk	9	9	6	81	54
Complexity	6	3	6	18	36
			Total	207	243

Second LV options					
		Rating		(Weight factor) * (Rating)	
Criteria	Weight Factor	Same as first	Larger lander only	Same as first	Larger lander only
Redundancy	6	9	3	54	18
Complexity	3	9	6	27	18
Communication	6	6	3	36	18
Cost	3	6	9	18	27
Ease of Design	3	6	3	18	9
Landed Mass	9	6	9	54	81
			Total	207	171

Satelite Deployment							
		Rating					
Criteria	Weight Factor	A-Rover	B-Rover	A-Boom	B-Boom	A-Gas propelled	B-Gas Propelled
Placement Accuracy	9	6	6	9	9	3	3
Power (Placement)	6	3	3	6	6	9	9
Power (maintain)							
Reliability	9	6	6	6	9	6	6
System Mass	6	3	3	3	6	9	9
System Volume	3	3	3	9	9	9	9

	Weight Factor*Rating					
	A-Rover	B-Rover	A-Boom	B-Boom	A-Gas propelled	B-Gas Propelled
	54	54	81	81	27	27
	18	18	36	36	54	54
	0	0	0	0	0	0
	54	54	54	81	54	54
	18	18	18	36	54	54
	9	9	27	27	27	27
Total	153	153	216	261	216	216

Report From ESTACA

ESTACA
University of Alabama in Huntsville

Radio Astronomy on the Moon

IPT Team A: AETHER (Aerospace Engineering Technologies Heading Extrasolar Research)

Astrid Robert
Antoine Bercault
David Langlois
Lucas Schoukroun

Report From ESTACA

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Introduction

Radio Astronomy on the Moon is a mission that aims for collaboration between two NASA directorates. The first goal falls under the Science Mission Directorate. The goal is to place an array of radio telescopes on the far side of the Moon. This allows for a clear viewing of astronomical objects without interference from man-made transmissions and the Earth's atmosphere. The second goal falls under the Exploration Systems Mission Directorate. The goal is to test the use of precision landing in the form of Autonomous Landing and Hazard Avoidance Technology. Precise autonomous landing would be useful for frequent cargo trips to a planetary body because it would not require a pilot for a reliable landing.

To meet these goals the team of engineers and scientists has undergone research of the moon's surface, the Delta IV Heavy and Atlas V 551 launch vehicles, radio astronomy and precision landing technology. Initially a trade tree was constructed with several conceivable options for mission functions. The team then conducted a quantitative decision analysis for launch vehicle and orbit options.

Aerospace Engineering Technologies Heading Extrasolar Research has determined that two Atlas V 551 launch vehicles with a lunar orbit for communications is the best concept of operations to meet the mission goals. With this option the total payload for both launch vehicles will be 3870 kg. The two launch vehicles will consist of a total of one orbiting communication station and three autonomous landers. Each lander will carry a package of radio telescopes to three different craters on the far side of the Moon. These telescopes will all act together as a large array to view astronomical objects with greater resolution.

I/Orbit calculation and procedures

A/Orbit

The orbiter will turn above the moon at an altitude of 100 km. The period can then be deduced.

Where R_{moon}=1737.4 km (1079.570 miles) is the radius of the moon, h=100km (62.1371 miles) is the altitude of the orbiter, G=6.67428x$10^{-11}m^3kg^{-1}s^{-2}$ the gravitational constant and M_{moon}=7.3477x10^{22}kg is the mass of the moon.

We have a period of 1h57'46''550'''.
Since we have an inclination of 15° from the equatorial plane, we can't make a sun synchronous orbit. We will have to charge batteries when we are seeing the sun and use it when we don't.

B/Mission recap

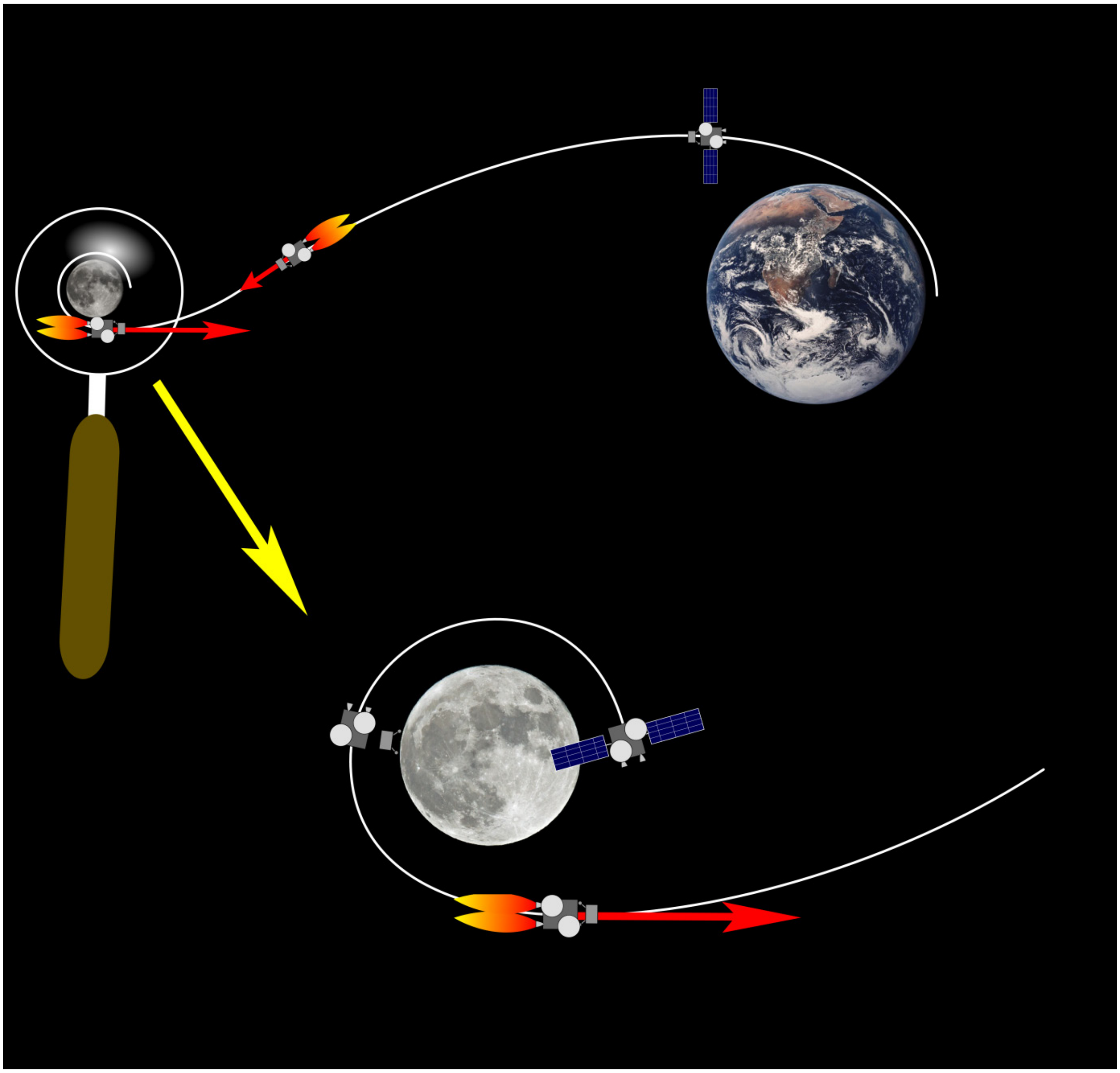

Figure 28: Mission recap

The orbiter and the lander are on a transfer orbit. During all the transit, the orbiter will need a little amount of power. Our computer will also need power, that's why we will be having the solar panel deployed. When reaching mid-course, the orbiter retract is solar panel to avoid breaking them with the acceleration of the first Delta-V. The Mid-Course Correction is a small ΔV ($51 m.s^{-1}$). It is done in the direction of the advancement. After the MCC have been performed, the solar panels are re-deployed. Later, they are retracted as the orbiter performs the second ΔV. The second ΔV is about $800 m.s^{-1}$. The orbiter have to make a flip because the second thrust is used to slow down the orbiter and the lander. After the boosts have been performed, the lander is ejected and the solar panels are re-deployed. Then the orbiter starts is life in orbit receiving and sending data.

C/The computer

We will need a computer to perform all our tasks. The computer will be in charge of performing the thrusts, diagnose permanently the orbiter and the lander while they are together. It will also manage the charging of the battery. It will also store the information while the orbiter is on the far side of the moon. The computer will be equipped with an inertial measurement unit. This inertial measurement unit will be unsettled quickly. That's why it will be frequently re-aligned by measurement made on earth. The computer will be in charge of performing the movements of the solar panels and the antenna. We were unable to determinate neither a precise mass nor a precise envelope for the said computer.

II/Propulsion

The propulsion part of the lunar orbiter will be in charge of two deltaV. The first one called the Mid-Course Correction (now referred as MCC). It is used to slightly change the direction of the orbiter and correct the deviation that occurred during the transfer. The value of this Delta-V is $51 m.s^{-1}$. The second one is called Lunar Orbit Insertion (now referred as LOI). This Delta-V is made to put the orbiter and the other components on a circular orbit at 100km above the surface of the moon. The value of this second Delta-V is $800 m.s^{-1}$.
This is our needs in propulsion. In addition to that, the orbiter should be able to stay on a stable orbit.

A /Chose of the propellant

With those requirements, we decided to use bi-propellant engines. We chose to use Dinitrogen tetroxide (now referred as NTO N_2O_4) as an oxidizer and Hydrazine (N_2H_4) or one of its derivatives like Mono-Methyl Hydrazine (now referred as MMH) or Unsymmetrical Di-Methyl Hydrazine (now referred as UDMH) as a fuel. Those propellants are storable at standard temperature (between a 263.85 °K and 294.3°K for NTO and 215.15°K to 336.2°K for UDMH) at standard pressure but as the fuel tank will be pressurized, the boiling point will be higher (see Temperature and Pressurization of the tank). These kinds of bi-propellant engine are often used as apogee engine for Geosynchronous Earth Orbit satellites. It has been used on several probes too, as the Cassini probe and on several observation spacecraft such as the Chandra X-ray Observatory. Those two propellants have the good taste of being hypergolic, that means that we do not need to ignite them, they do it by themselves when in contact. It makes the engine way simpler and it can be virtually re-ignite an infinite number of times.

B/Sizing the need for thrust

High performance Bi-Liquid engine have a very low thrust. As we aimed for 320 seconds of Isp (the Isp is the indicator of the performance of the engine; the higher it is the more efficient the engine is), we landed on small engine with about 445N (100lbf) of thrust. But to shorten the duration of the second Delta-V, we thought about using two or four of them. Also, having one big engine would have been a mistake since we will be using them only and having a heavy big engine would have been a monkey on our back. Small engine have a debit of aproximatively

0.14kg.s^{-1} to 0.30kg.s^{-1} (respectively 0.308lb.s^{-1} to 0.661 lb.s^{-1}). With only one engine, we had propulsion duration for about 9500 seconds but since those small engine burns for 3000 to 3600 seconds max; we decided to use 4 engines. The engines will be placed all around the launcher interface to provide a centered thrust. None of those engines will be able to move and the orientation of the space train (*i.e.* the orbiter, the lander and the science experiment) will be ensured by another propulsive system (see Orientation and guidance). The duration of the transfer will be function of the thrust of our engines (see Engine Trade-Off).

C/Composition of the engine

All the bi-propellant engines use the same composition. The only things that are changing are the surface ratio, the inlet pressure and the Isp. The inlet pressure is about 27.57 to 17.23 bars (400 Psi to 250 Psi). Both propellant react in the combustion chamber, raising the temperature of the gas. The gas is then accelerated to match mach 1 at the neck then is accelerated and is unwinding in the nozzle. The area ratio is an indicator of the performance of the engine. It represents the ratio between the surfaces of the exit of the nozzle as compared to the surface of the neck of the nozzle. High performance hypergolic engines have an Area ratio of more than 300.

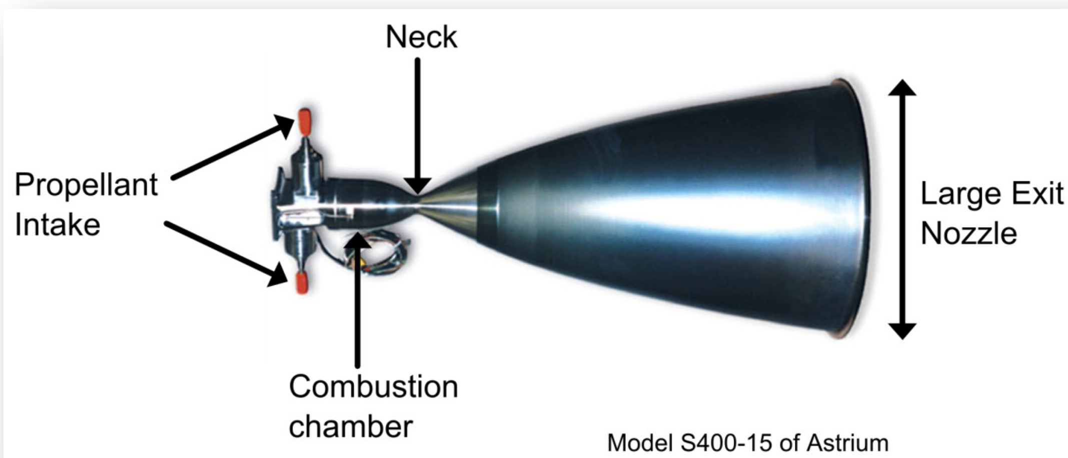

Figure 29: The different parts of en engine

D/Temperature and pressurization

To ensure a high level of Isp, we need to have a high temperature of the combustion gas and a high inlet pressure. We decided to use 400 psi (27.57 bars) tank pressure since many spacecraft systems have been designed to be used at such pression. With those pression, the boiling point of the propellant has increased. Using the Clapeyron's formula, we are able to see the boiling point of the propellant we want to use.

Giving us

Report From ESTACA

For the oxidizer, we can only use Dinitrogen tetroxide.

Temperature at boiling point	21,15	°C
Enthalpy of vaporization	38120	$J.mol^{-1}$
Tank pressure	27,57	bar
Boiling temperature at tank pressure	373,5121962	°K

Chart 1: Temperature at boiling point of NTO

This made us gain about 80°K of boiling point. It made the tank way easier to warm or to cool because the range of temperature between the freezing point and the boiling point is now 110°K instead of 30°k on an unpressurized tank.

	UDMH	**Hydrazine**	**MMH**	
Temperature at boiling point	63,05	113,55	87	°C
Enthalpy of vaporization	35550	41800	40900	$J.mol^{-1}$
Tank pressure	27,57	27,57	27,57	bar
Boiling temperature at tank pressure	454,190135	518,451644	475,0602261	°K
Temperature at boiling point a 27.57 bars	117.990135	131.751644	114.9102261	°C

Chart 2: Temperature at boiling point of Fuel

It is the same for the other propellant; we gain more than a 100°K of boiling point by pressurizing the tank. We will be more strained by the temperature of the oxidizer than by the temperature of the fuel. In order to pressurize the tank, we will be using Helium since it is very light, absolutely neutral (react with nothing since it is a noble gas) and electrically free. It can be stored as a gas at very high pressure on a composite tank at 310 bars (4500 Psi). It will also be used to clean the propellant line while in flight because these propellant could be very corrosive for the materials of the pipe.

E/Tank and hydraulic diagram

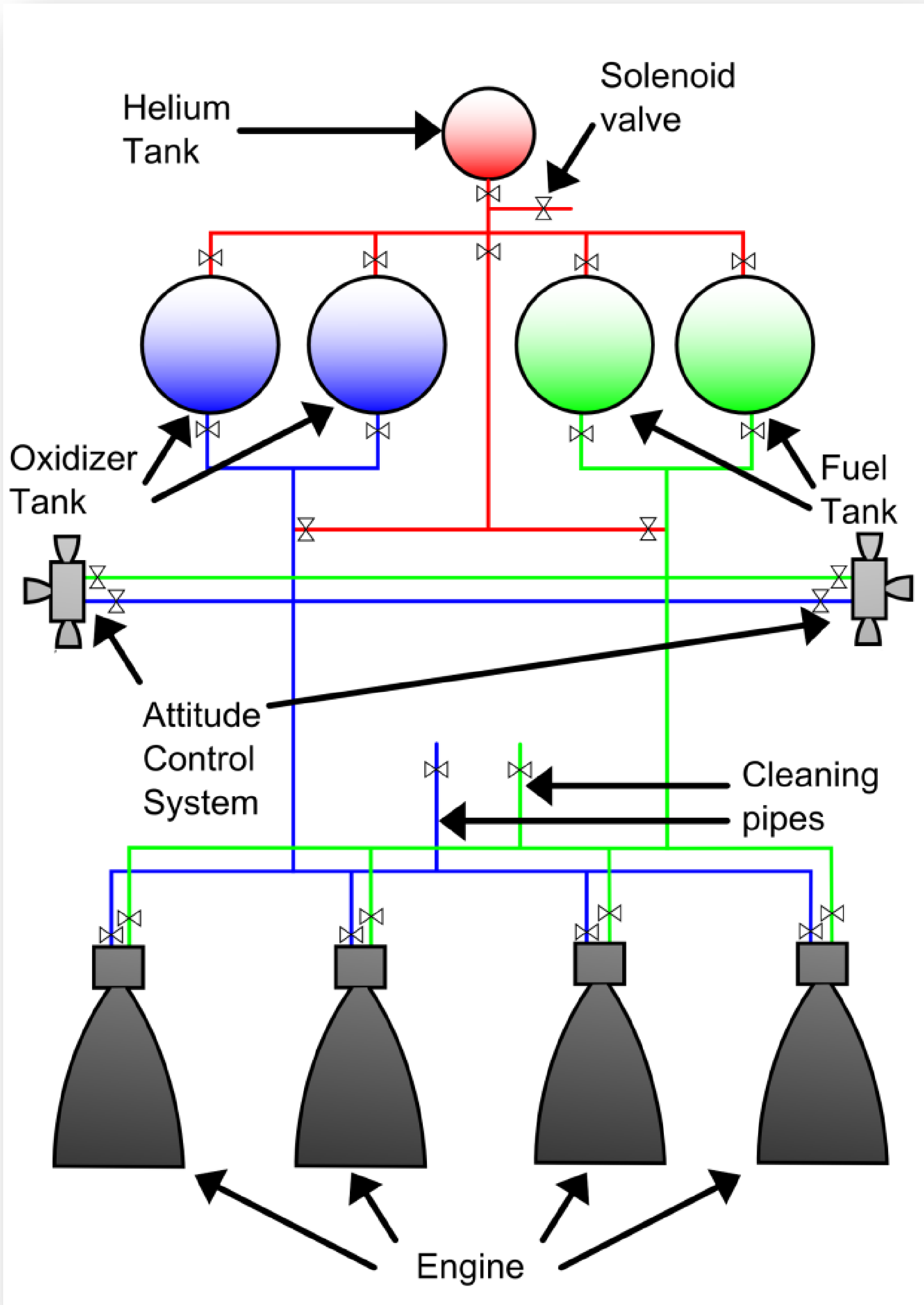

Figure 30: Simplified Hydraulic Diagram

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This is a simplified diagram of the pipes of the hydraulic system. To be more easily balanced, we splited both tank in two. The sizes of the tank depend on the fuel but are about the same size. We could also use a fuel known as Aerozine 50, which is a cocktail of 50% of Hydrazine and 50% of UDMH. If we use Aerozine 50, one fuel tank will be filled with Hydrazine and the other with UDMH. It is not necessary to develop the hydraulic diagram of the attitude control system since it's the same kind of diagram, only with 8 to 12 nozzle. If we use only hydrazine, we do not need an oxidizer on the attitude control system because the hydrazine can react with itself when passing through a catalytic bottom. On high pressure and high debit pipes, the valve are piloted by pressurized Helium but since the pressure is not very high on the pipes and the debit is very low, we will be using solenoid valves to open or close the pipes. All the valves of the engines must be opened at the same time if we want a thrust in the axis of the center of gravity. The Helium will be stored as a gas so we have no problem of propellant settling. For the propellant, we will be using a natural fact in weightlessness that the fluid sticks to the edges of the tank instead of the center. When experiencing weightlessness, the pressurant gas will be stuck in the middle of the tank; we just have to put the end of the helium pipe in the middle of the tank and the intake on the surface of the tank, as shown on the next diagram.

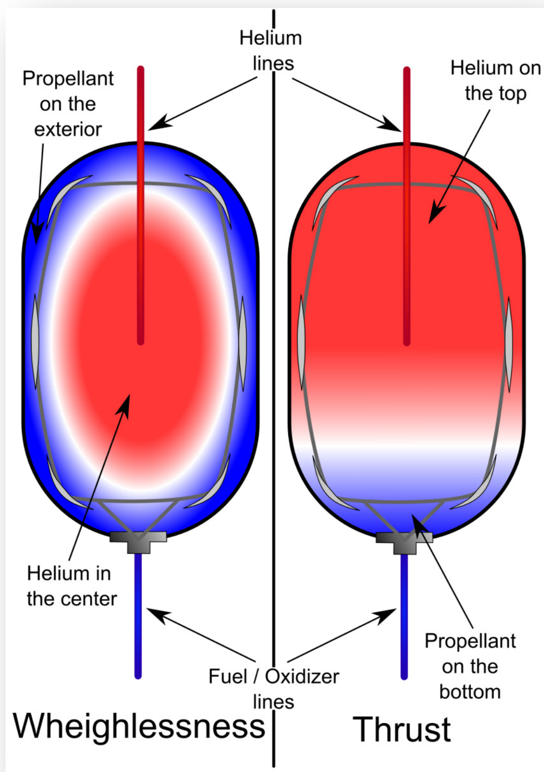


Figure 31: The repartition of helium and propellant

This configuration works very well in weightlessness but when the engines will work, the thrust will have the effect of settle the fuel and the oxidizer on the bottom of the tank. That's why the main intake is on the rear end. The tank will be heated to avoid the propellant to freeze but not too much for them not to boil.

F/Orientation and guidance

The orientation will be performed by two attitude control system. Those systems are always used for controlling the roll, the pitch and the yaw of the spacecraft. It is present on the front of the space shuttle, it could have been found on the Apollo module and it is used on the European Authomatized Transfer Vehicle. It is composed of many small nozzles ejecting combustion gas on various directions to spin the spacecraft. It is used in pair to make a couple on one of the axis of the spacecraft. Then the other nozzle on the opposite side does the same impulse to stop the rotation. As said before Hydrazine could be used as a propellant alone. The Isp of the nozzle is about 200s to 250s because the temperature is not very high. The thrust is about 20N to 10N (4.5 lbf to 2.25lbf). Before making the first Delta-V (MCC), the attitude control system will orientate the spacecraft in order to perform the first acceleration. To perform the second Delta-V (LOI), the attitude control system will have the spacecraft to make a flip in order for the second boost to make the orbiter decelerate.

G/Engine Trade-Off

To choose the engine, we selected the engine within our range of thrust and with an Isp of more than 320s we made a comparison of engines. The entire engine use NTO and Hydrazine/MMH/UDMH. We selected also the engine from an American manufacturer. We landed on 3 engines: the TR-308 from Northrop Grumman, the HiPAT from Aerojet and the very promising AMBR (Advanced Material Bi-propellant Rocket) developed by the NASA.

	TR-308			
Propellant	N_2O_4		N_2H_4	
Thrust	106	lbf	471,51	N
Mixture ratio	1			
Specific Impulse	322	s		
Area Ratio	204			
Inlet pressure	205	psi	14,13	bar
Engine length	27,8	in	70,6	cm
Exit diameter	11,8	in	30	cm
Engine Weight	10,5	lbm	4,76	kg
Qualification life	24190	s		

Maximum Firing	3000	s		
Propellant mass	653,9856676	lbm	1441,791597	kg
Debit	0,067706719	$lbm.s^{-1}$	0,149267765	$kg.s^{-1}$
Propulsion time	9659,095528	s		
Propulsion time with 4 engine	2414,773882	s		

Chart 3:TR-308 datasheet

Those are the data for the Northrop Grumman engine. It have the lowest Isp but it's the oldest and so it is maybe more reliable since there is more data about it.

	HiPAT			
Propellant	N_2O_4		$(H_3C)HN\text{-}NH_2$	
Thrust	100	lbf	445	N
Mixture ratio	1			
Specific Impulse	324	s		
Area Ratio	375			
Chamber pressure	250	psi	21,4	bar
Inlet pressure	137	psi	9,44	bar
Engine length	28,6	in	72,6	cm
Exit diameter	14,25	in	36,3	cm
Engine Weight	11,5	lbm	5,2	kg
Qualification life				
Maximum Firing	3600	s		
Propellant mass	650,4651915	lbm	1434,030276	kg
Debit	0,063505558	$lbm.s^{-1}$	0,140005789	$kg.s^{-1}$
Propulsion time	10242,64987	s		
Propulsion time with 4 engine	2560,662466	s		

Chart 4: HiPAT dataheet

The HiPAT of Aerojet is slightly better because the Isp is shortly ahead. With this engine we gain more than 8kg of propellant. The HiPAT is frequently used on probes or spacecraft so it will still be usable in the next years.

	AMBR			
Propellant	N_2O_4		N_2H_4	
Thrust	200	lbf	890	N
Mixture ratio	1,2			
Specific Impulse	335	s		
Area Ratio	400			
Chamber pressure	400	psi	27,57	bar
Inlet pressure	275	psi	18,95	bar
Engine length	28,6	in	72,6	cm
Exit diameter	14,25	in	36,3	cm
Engine Weight	11,5	lbm	5,2	kg
Maximum Firing	3600	s		
Propellant mass	631,75636	lbm	1392,784363	kg
Debit	0,122840601	$lbm.s^{-1}$	0,270817168	$kg.s^{-1}$
Propulsion time	5142,895382	s		
Propulsion time with 4 engine	1285,723846	s		

Chart 5: AMBR datasheet

Once again, we are able to gain 42kg of propellant by using a motor with a highest Isp. The AMBR engine is a state of the art engine. It is an experimental engine developed by the NASA as an improvement of the HiPAT. Higher Isp is achieved by getting warmest combustion gas. The combustion chamber is in a new metal alloy using Iridium and Rhenium. The injectors have also been re-designed, the pressure in the combustion chamber has been increased and the area ratio had been slightly increased too.

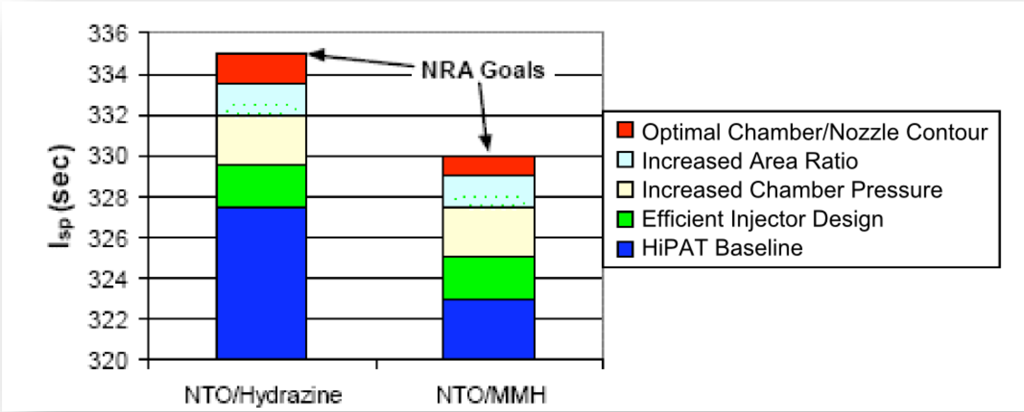


Figure 32: Gain of Isp over the HiPAT

We also see that to achieve high Isp, we have to use Hydrazine. The main reason why we would have liked to use MMH or UDMH is that it is more stable than regular hydrazine. Hydrazine is denser than MMH or UDMH: 1013kg.m^{-3} for Hydrazine (63.24 lb.ft^{-3}) compared to the 880kg.m^{-3} of the MMH and UDMH (54.94 lb.ft^{-3}). This makes the tank smaller and then lighter. As seen before, if we use hydrazine-only fuel we can make the attitude control system use monopropellant engines that make it more reliable (even if it's less effective). The Hydrazine is supposed less stable than UDMH and MMH but as it is commonly used nowadays, we can suppose that we won't have problems during our short mission (compared to GEO satellites that work for 10 to 15 years). The AMBR is by all comparison a far better choice for our engine. It is still not produced but it will be by 2014.

We will finally use 2 AMBR engines since they have two times more thrust than the HiPAT and the TR-308. It will be powered by Dinitrogen Tetroxide (NTO) as an oxidizer and Hydrazine as a fuel.

H/Weight budget for the propulsion system

We were able to make a precise weight budget for the propulsion system by using one found on the internet about the Europa Mission. The system was exactly the same: two AMBR engines, Hydrazine fuelled, pressurant, 12 thrusters for the attitude control system. The only change is that there are two tanks for the pressurant (Helium). The propellant tanks are made of titanium and the pressurant tank is made of Composite Overwrapped Pressure Vessels pressurized at 310 bars (4500psi). Both tanks are made with a factor of safety of 1.5.

	RAM Orbiter	Unit Mass		Total Mass		Comments:
Qty	Propulsion System Components	(kg)	(lbm)	(kg)	(lbm)	
2	Pressurant Tank (COPV)	13,15	29,00	26,31	58,00	Calculated Hardware
3	Fill and Drain Valve, High Press He	0,10	0,22	0,30	0,66	Messenger Hardware
6	Filter, He	0,11	0,24	0,70	1,54	Messenger Hardware
7	Pyro Valve, Pressurant	0,20	0,44	1,40	3,09	Messenger Hardware
2	Pressure Regulator	2,31	5,09	4,60	10,14	STS OMS
1	High Pressure Transducer	0,23	0,51	0,20	0,44	Messenger Hardware
4	Check Valves	1,36	3,00	5,40	11,90	STS OMS
4	Transducer, Low pressure	0,23	0,51	0,90	1,98	Messenger Hardware
0	Burst Disk	0,10	0,22	0,00	0,00	STS OMS
0	Relief Valve	2,31	5,09	0,00	0,00	STS OMS
4	Ground Checkout Hand Valve	0,07	0,15	0,30	0,66	Messenger Hardware
2	Propellant Tanks, Fuel (w/ PMD)	22,68	49,99	45,35	99,99	Calculated Hardware
2	Propellant Tanks, Oxidizer (w/ PMD)	15,66	34,53	31,32	69,05	Calculated Hardware
3	Pyro Valve, Propellant	0,20	0,44	0,60	1,32	Messenger Hardware
2	ISO Valve, Propellant, RCS	0,65	1,43	1,30	2,87	Messenger Hardware

6	Fill and Drain Valve, Propellant	0,15	0,33	0,90	1,98	Messenger Hardware
3	Filter, Propellant	0,29	0,64	0,90	1,98	Messenger Hardware
6	Transducer, Low pressure	0,23	0,51	1,40	3,09	Messenger Hardware
12	RCS Thruster (22 N, 5 lbf thrust)	0,65	1,43	7,80	17,20	Aerojet MR-106E 22N
2	AMBR Thruster (890 N, 200 lbf thrust)	5,50	12,13	10,90	24,03	
	Miscellaneous Hardware	10%		12,00	26,46	
	Design Contingency	10%		13,20	29,10	
	Total Dry Weight			**165,78**	**365,49**	
	Propellant: Usable			1392,00	3068,83	
	Residuals			13,92	30,69	
	Pressurant: Helium			2,13	4,70	
	TOTAL PROPULSION SYSTEM			**1573,83**	**3469,71**	

Chart 6: Propulsion Weight Budget

All the sub-systems here are of technology readiness 9, that mean they have been used in space on many successful missions. It also means that it will cost little money since we will only need to recalibrate them and make them suited for a fly to the moon. Only the AMBR engine has not proven flight capability but as said before, the gain in term of masses make the challenge worth the risk. The components that can be found on the table were for the Messenger probe. Our orbiter does not have the same goal but is similar to the Messenger probe in terms of safety and reliability. Other components come from the space shuttle. Those have a level of reliability even higher.

Report From ESTACA

III/On board energy management

A/On-board energy needs

The on-board management controls the satellite's functioning. It contains the following subsystems:

- Telemetry, telecommand ;
- Satellite surveillance and control ;
- Data processing.

1-Communication

The telecommand and telemetry system handles communication to the ground. The telecommand functions (ground ⇒ satellite) receive and decode the instructions or data sent by the control center and carry out the task of distributing them to other subsystems. The telemetry functions (satellite ⇒ ground) gather the data relevant to the satellite's functioning and the data transmitted by the instruments, and after data compression transmit these to the control center when in sight of the stations.

2-Trajectory control

The flight control system upholds the satellite's trajectory and orientation. This task is achieved by a software that utilizes data supplied by different types of sensors. It calculates the deviations and corrects them by means of actuators (orientation) and (generally chemical) engines (trajectory).

3-Data storage

Data gathered by the instruments is stored in mass memories until it's transfer to the stations during the overflight of a receiver antenna. Internal satellite communication takes place via a bus. The transmitted data flow must be preserved from the charged particles bombarding the satellite.

4-Secondary functions

On-board management also executes the following functions:

- Surveillance of the satellite's functioning, detection of potential failures, diagnosis and activation of bypass solutions ;
- Verification of compliance with the thermal constraints;
- Temporal synchronization between the different subsystems ;

- Triggering of programmed tasks concerning the payload (taking of pictures, …)

Part of these tasks can be carried out either by ground stations or the satellite itself.

5-Total needs

In total, we have seized the energy need for the satellite's functioning to 5kW for its emission.

B/Solar panels

We have opted for solar panels to provide the energy the satellite needs.

1-Principle

Every satellite needs to be autonomous, be it for thermal control or for the transmission of radio waves. To meet this requirement it is necessary to be able to exploit an energy source capable of providing the electricity the various components need. In case of a satellite, the required electricity is procured chiefly by solar panels that transform solar energy into electricity.

Solar energy is the radiating energy produced in the sun as a result of nuclear fusion. It is transmitted through space to Earth and to the satellites as photons. Photons are particles transporting some luminous energy or energy corresponding to other electromagnetic radiation. The energy 'E' of a photon can be formulated as such: $E = h.\nu$ (" h " being Planck's constant and " ν " being the frequency of light).

This solar energy is changed into electricity by photovoltaic cells made of semiconductor material (generally thin layers of P-N junction crystalline silicon). This material directly converts solar radiation into electricity.

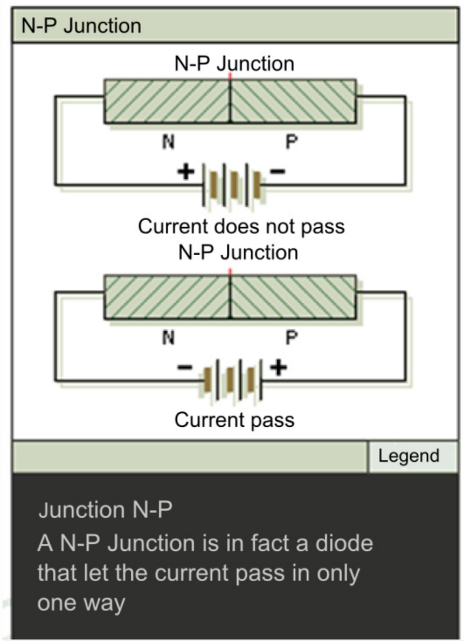


Figure 33: N-P Junction

The amount of current and voltage generated by a solar panel depends on the way the photovoltaic cells are assembled. If they are series-connected, the voltage is high; if they are connected in parallel, then the current is higher. A series of cells is called a string. A series of string is a block.

Every cell is formed off of two layers of silicon. When a light photon reaches the panel, its energy creates a breach between a silicon atom and an electron. This brings about a voltage between the positively charged atoms and the negatively charged electrons. The mechanism of solar panels won't be deepened here, as it isn't the studied subject.

2-Degradation of solar panels

Solar panels wear out over time. A solar panel won't supply as much energy after several years, due to the silicon cells losing efficiency. A 'degradation factor', 'Δ', is used to estimate the wear. The amount of energy 'lost' after 'x' years is calculated in Watts with the formula .

In addition to this chemical degradation, the extensive surface area of the solar panels makes them a frequent target of micrometeorites, thereby reducing the yield. The solar panel's performance is also affected by 'polluting' particles transported by solar winds.

When designing a solar panel, one has to take into account the ageing factor and the decreasing efficiency of said solar panels over the years. The needed size must therefore be chosen to be able to supply the necessary power at the end of the solar panel's life-span.

Taking into account the loss of efficiency, the solar panels are designed to be much greater in size than what would be needed at the beginning of the mission.

3-Performance

i-Delivered power

Solar panels have their own efficiency which depends on the model and the state of wear of the panel. Denoted η, it amounts to

This efficiency is estimated to be around 20%.

The amount of solar energy received by a surface area of 1sqm, at the distance Earth-Sun, that would be exposed at a right angle to solar radiation is denoted ϕ.

If the angle of incidence θ is not 0°, the power input is not ϕ (Watts) but is instead ϕcos θ (Watts).

If we consider the solar panels to always be perpendicular to solar radiation, we have :

Taking into account the ageing of the satellite (life-span of 4 years), the estimation is that the power output will be reduced by 12.5% (with security margin).

The end-of-life power output would be 240 .

This is the value chosen as most dimensioning.

ii-Weight

The solar panel's weight is evaluated to be approximately m , fastenings and framework included.

4-Kinematics an geometry

Solar panels form a mechanical structure that needs to be folded up during launch, when the satellite is still in the rocket. Once the satellite is in orbit however, the structure must be mechanically deployed, and any jamming during this operation can jeopardize the whole mission.

We opted for 2 rectangular panels with simple unfolding on either side of the satellite. Each panel will be an assembly of 3 square panels.

Figure 34: The architecture of our solar panels

5-Recommended solar panels

As previously determined, the satellite needs a power of in order to function properly. All calculations are made using the most dimensioning state: the satellite's end of life. This power will be needed when the satellite will be sending data and charging is batteries.

2 iterations will be used to attain the final result. The first iteration will give a first estimate of a panel's surface and mass, based on the amount of power needed by the satellite. For the second iteration, results will be rounded up to the nearest hundredth and will give a final estimate of the mass and total surface area of the solar panels.

i-Surface area of the solar panels

At the end of the solar panel's life-span, the power output of the panels will be approximately 240 .

The surface area obtained through calculus is:

2nd iteration:

ii-Total take-off mass

At take-off, the panel's framework weights 13.3 kg/m². We can deduce the total mass :

2nd iteration:

iii-Unit panel

The solar panel is assembled from 6 unit panels. Each unit panel will have a surface area of :

2nd iteration:

Mass of one unit panel:

2nd iteration:

Length of a square unit panel's side:

This length will be used as reference length for the 2nd iteration.

C/Battery

The majority of satellites using solar energy as chief source of energy are equipped with storage batteries. These batteries supply the systems with DC-current during eclipses or peaks in consumption. In those cases, the batteries become the principal source of electrical energy. As of today, storage batteries are made of NiH_2.

Report From ESTACA

A battery is a string of photovoltaic cells. It can be rechargeable (secondary battery) or non-rechargeable (primary battery). Primary batteries are charged before launch and can only supply energy for y day, which is why they are primarily used on launchers or during launch, to replace the folded up solar panels. Secondary batteries are also charged on the ground before launch, but can be recharged later on. Thus their weight is greater than that of primary batteries (approx. 45 kg).
The energy is transferred via electrochemical reactions, which is to mean via transfer of electric charges constituted of one or more electrons. These reactions are also Redox reactions.

Like solar panels, the batteries can be connected in series or in parallel.

During each orbit, the batteries complete charge-discharge cycles. To last through the whole life-span of a satellite at such a high rate (more than 50000 cycles!), laws of control were established after testing. These laws are described by the CNES (French government space agency) as follows:

Control of the discharged electricity amount in proportion to the battery's nominal capacity: the discharged quantity depth shall not exceed 25 %.

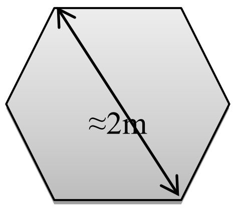
Control of the charged and discharged electricity amounts: their ratio, called 'charge ratio', must be just about 1. It's value depends on temperature. The SPOT 4 on-board computer supervises and controls the charge state of the batteries.

Control of the charge voltage of each battery relating to a certain threshold (36.5 Volts), also depending on temperature. The charge current must be limited to a maximum of 12A. Upholding this condition is an electrical equipment, the shunt junction regulator (RSJ). The RSJ regulates the charge voltage and current while also ensuring the satellite is correctly supplied.

IV/Architecture

A/Structure

For the satellite main design we got inspired from pre-existing communication satellites and orbiter.
The main structure of the satellite is hexagonal; the cross section of the body is a regular hexagon.



For a given cylindrical fairing it offers more room than a cubic one. It allows the big spherical fuel tanks to fit in without any waste of space.

In terms of rigidity it's also a good choice.

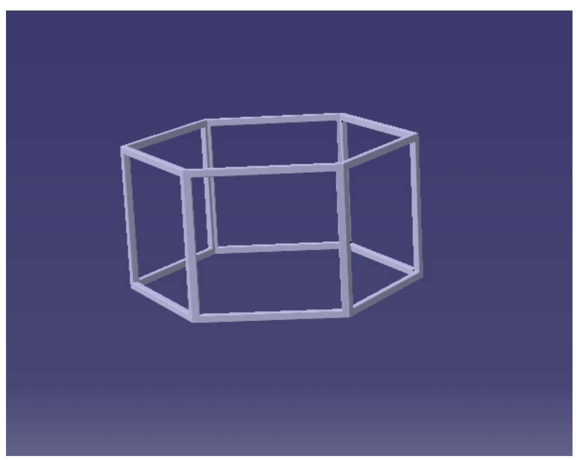


Figure 35: The frame of the orbiter

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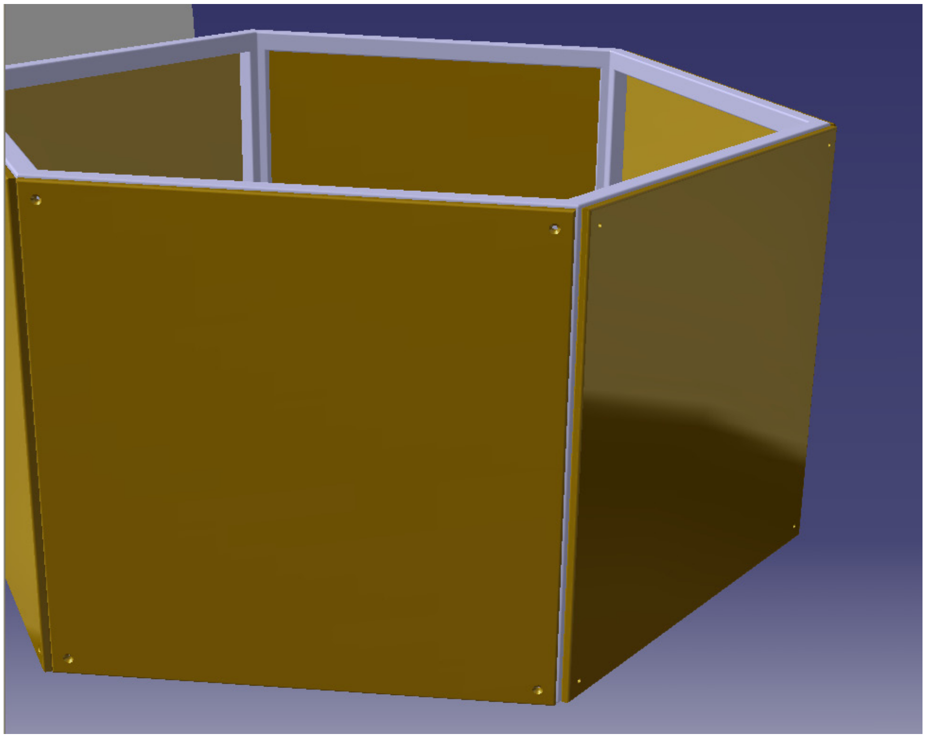


Figure 36: The orbiter, covered by mylar panels

Square panels are attached to each side of the structure. A classic Mylar foil sheeting is added on each panels to protect the satellite from harmful sun rays and provide a good thermal insulation. There is no need for structural armor plate. Those are only needed to prevent satellites for being hurt by other space wreckage but since we will orbit around the moon, there is not so many risks.

The satellite is composed with two identical structures.

The one on the top contains the hardware (memory devices, battery…)

The fuel tanks and the engines are fixed in the second one. Both structures are 1 meter high.

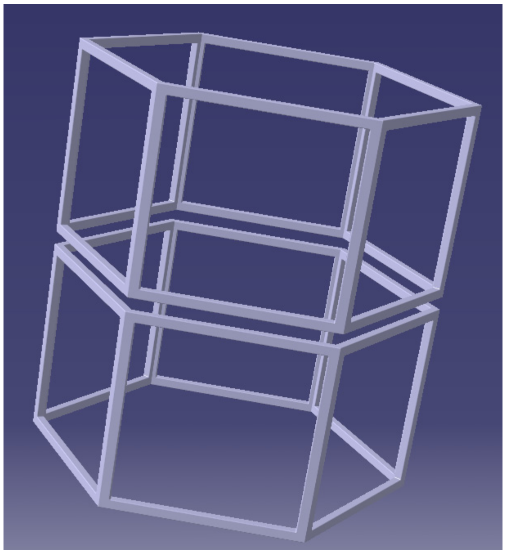


Figure 37: The two frames

B/Engines and tanks

The twin engines are diametrically opposed to allow the thrust resultant to be aligned with the center of gravity.

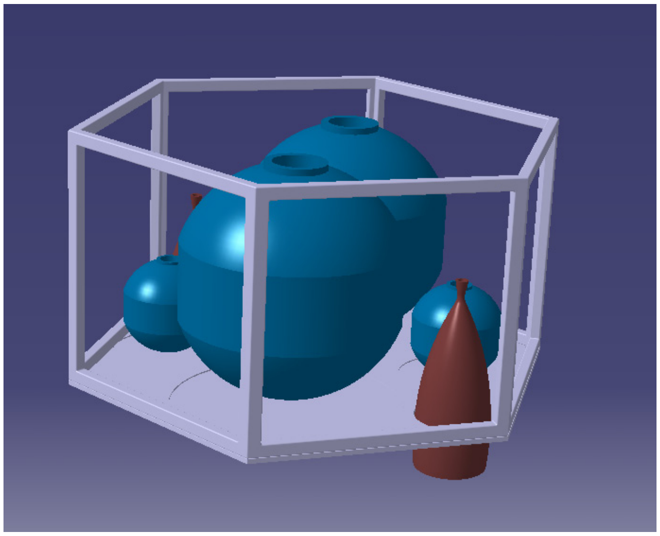


Figure 38: The tank and the engine

C/Attitude control system

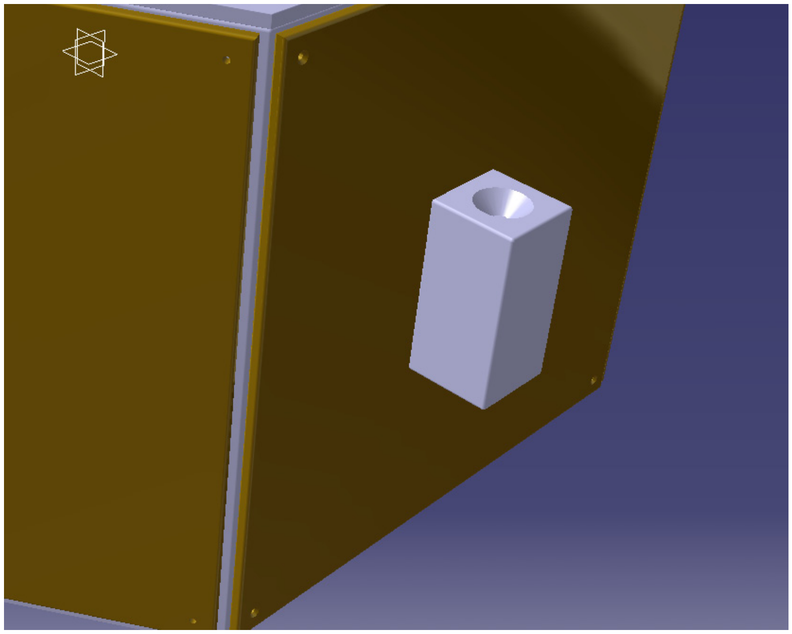


Figure 39: A closer picture of the attitude control system

The attitude control system is disposed on two sides of the orbiter, diametrically opposed.

D/Solar panels

The area of solar panel needed is 42 m². So we design 6 panels of 7 m².
Each panel is 3.5 m long and 2 m large. They are pliable in order to fit in the payload fairing.

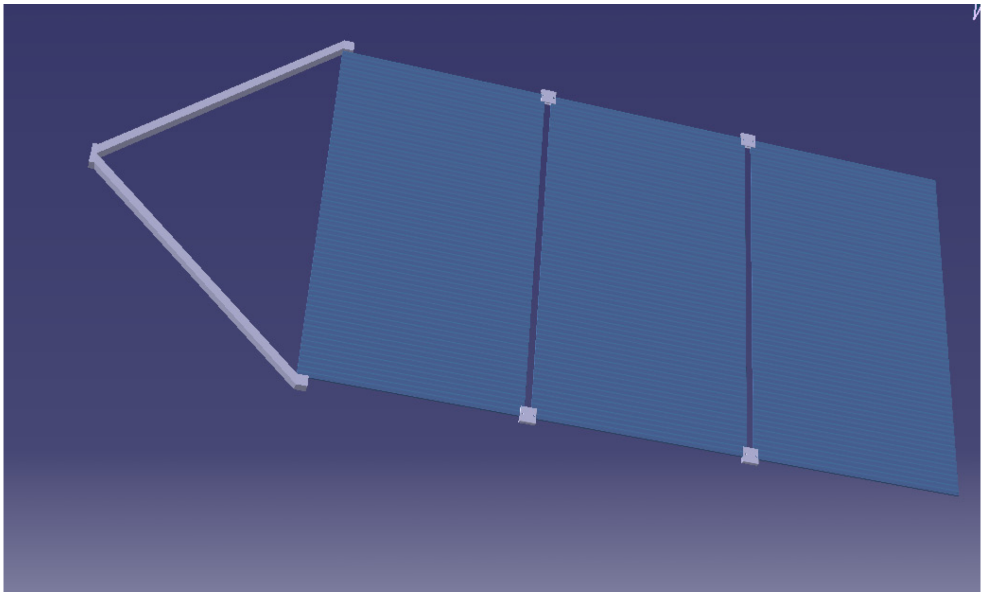


Figure 40: One of the solar panels

We chosed to use the most common way to fold solar panels because this technology have been mastered time ago and it won't cost many money.

E/Satellite dish

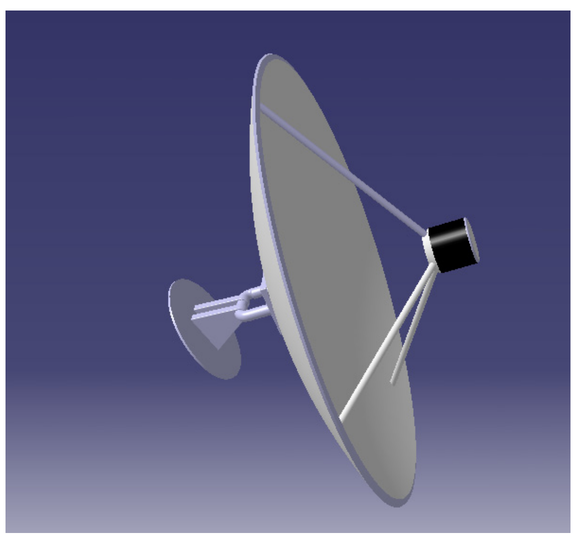


Figure 41: The Satellite Dish

The satellite dish is the same for reception and emission. It can move one both axes to be always on sight of the satellite or the earth, depending if it is actually sending or receiving data.

F/Assemblage

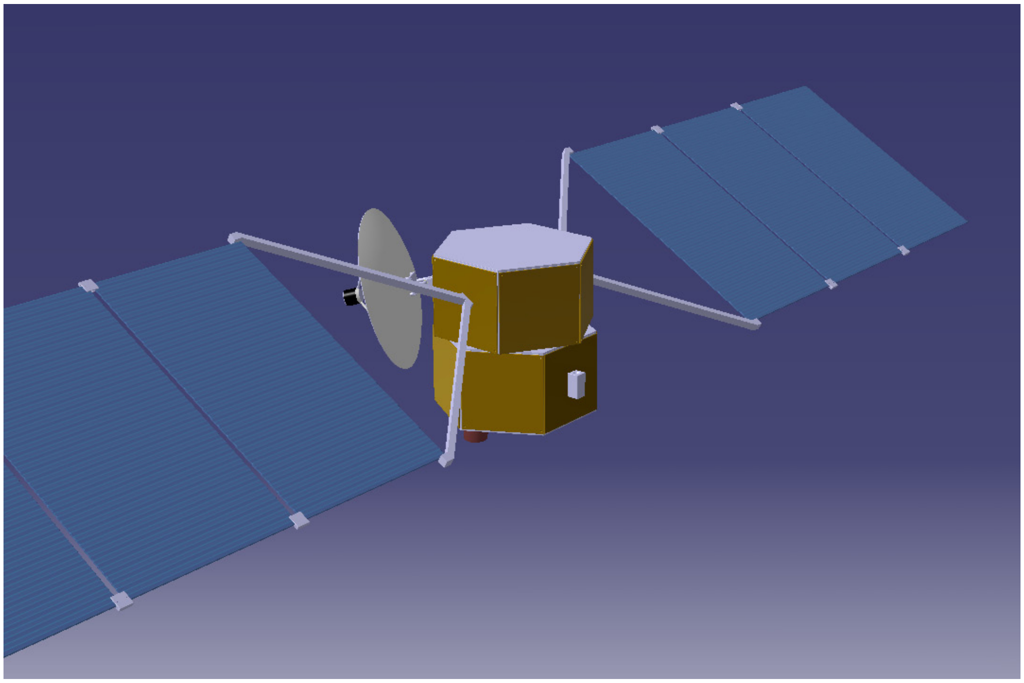


Figure 42: The whole orbiter

V/Tables

Figures

Charts

VI/Credits and sources

Introduction: Introduction from the Mission Definition Review Report

Figure 2: The S400-15 apogee motor of Astrium

<http://cs.astrium.eads.net/sp/spacecraft-propulsion/apogee-motors/index.html>

Figure 5: Gain of Isp: AMBR Engine from NASA In Space Propulsion Technology (ISTP) Program

<http://ntrs.nasa.gov/archive/nasa/casi.ntrs.nasa.gov/20090001339_2008047216.pdf>

Other sources

New Frontiers AO

<http://ntrs.nasa.gov/archive/nasa/casi.ntrs.nasa.gov/20080047426_2008047215.pdf>

ISPT Overview

<http://ntrs.nasa.gov/archive/nasa/casi.ntrs.nasa.gov/20090001339_2008047216.pdf>

TR-308 Dual Mode Liquid Apogee Engine Datasheet

<http://www.as.northropgrumman.com/products/bipropellant_engines/assets/TR-308_DMLAE.pdf>

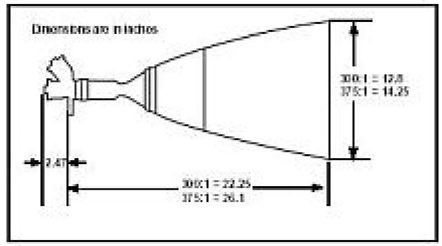
Aerojet's HiPAT Engine Datasheet

<http://www.astronautix.com/engines/hipat.htm>

Report From ESTACA

Appendix

HiPAT™ - 445N (100 lbf) DUAL MODE HIGH PERFORMANCE LIQUID APOGEE THRUSTER



Design Characteristics

- Propellant .... Hydrazine/NTO(MON-3)
- Thrust/Steady State .... 445 N (100 lbf)
- Inlet Pressure Range .... 21.4-15.2 Bar (310-220 psia)
- Chamber Pressure* .... 9.4 Bar (137 psia)
- Expansion Ratio 300:1, 375:1
- Oxidizer/Fuel Ratio 0.85
- Flowrate .... 141 g/sec (0.31 lbm/sec)
- Valve .... Aerojet Solenoid, Dual Coil, Single Seat
- Valve Power .... Various (46 Watts) @ 28 Vdc Typical
- Mass .... 5.2 kg (11.5 lbm)

*at rated thrust

Performance

- Specific Impulse (lbf-sec/lbm) .... 300:1 = 326 ... 375:1 = 329
- Total Impulse .... In Excess of 9.55×10^6 N-sec (2.15×10^6 lbf-sec)
- Total Pulses 672
- Total Thermal Cycles 345
- Minimum Impulse Bit (lbf sec) 8
- Steady State Firing (sec) 1800

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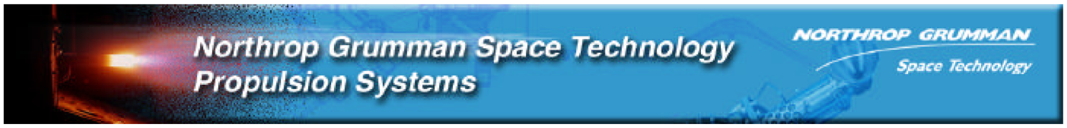
AEROJET

Appendix 1: HiPAT datasheet



TR-308 Dual Mode Liquid Apogee Engine

The TR-308 dual mode liquid apogee engine provides reliable, high-performance capabilities for long-life spacecraft operations. The engine, an improved version of the TR-306 LAE, features Northrop Grumman's high performance pintle injector design and has a specific impulse of 322 seconds. Designed for multiple starts, the engine was qualified for 24,190 seconds and a maximum single firing duration of 3,000 seconds. The engine's integral thrust chamber/nozzle extension (E=204) is manufactured of all-welded R512E silicide-coated C103 columbium. Four TR-308s serve as the on-board propulsion system for NASA's Chandra X-ray Observatory.

Heritage

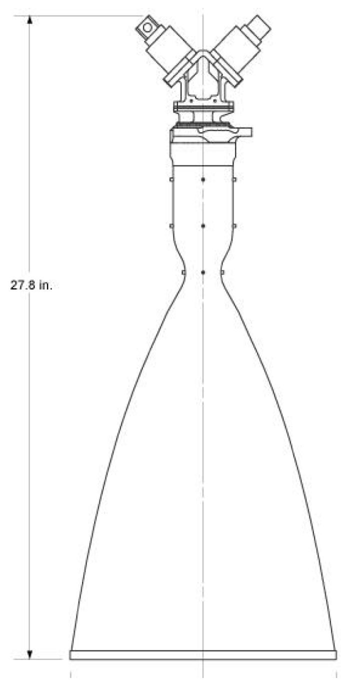
Four engines flown on Chandra X-ray Observatory.

Availability

12 months after receipt of order.

Characteristics

Propellants	N_2O_4 and N_2H_4
Thrust	106 lbf
Mixture Ratio	1.00
Specific Impulse	322 seconds
Nozzle Expansion Area Ratio	204
Inlet Press	205 psia
Engine length	27.8 inches
Nozzle Exit Diameter	11.8 inches
Engine Weight	10.5 lbm
Qualification Life	24,190 seconds
Maximum Firing Duration	3,000 seconds
Throughput	7,983 lbm
Vibration (Qual)	7.36 g-rms



Appendix 2: TR-308 Datasheet

Report From Alabama A&M

1. Introduction and Background

The Throttling Cavitating Venturi Valve (TCaV) is a flow control valve that uses the cavitating effect, which is the formation of vapor bubbles of a flowing liquid in a region where the pressure of the liquid falls below its vapor pressure, to regulate the flow of propellant to the inlet of the engine.. For our project we were given the task of redesigning an existing valve in a collaborative effort between Alabama A&M, UAH and NASA. UAH's task was to design a lunar landing vehicle and the requirements for lunar landing and use the A&M designed TCaV as their main propellant valve. The current design of the valve is bulky, weighs 43 pounds, and is made of Monel k500 and 304L stainless steel materials. The overall of goal of the redesign is to make the valve more flight ready by reducing the weight by at least 40 percent to help reduce the cost.

2. Valve Sizing

In order to meet the engine requirements for fuel delivery, the team needed to assess the flow characteristics of the TCaV to determine if it could deliver the needed amount of propellant (hydrazine) to the engine. To determine the appropriate orifice size for the valve, an Equivalent Sharp Edged Orifice Diameter or ESEOD was calculated. The ESEOD tells us what flow path size internal to the valve is needed in order to flow a fluid of a particular density at a given pressure and flow rate. Applying a valve sizing software by Valcor which uses the following equation, we calculate the ESEOD for a valve that will deliver the required flow rate for the Aeroject MR-80B:

$$q = C_d A\sqrt{(2gh_L)}$$

where,

q = flow rate, in cubic feet per second
C_d=Discharge coefficient (0.93 for TCaV)
A=flow area in square feet
g=gravity
h_L=head loss in feet of water

Based on this calculation, TCaV will provide a flow rate of 9.25 lb/s (4.2 kg/s) of hydrazine with an inlet pressure of 300 psia. This gives a maximum ESEOD of 0.464in. The current configuration of TCaV provides a maximum ESOD with the pintle fully retracted of 0.467in. Therefore, no internal modifications of TCaV would be needed to meet the MR-80B requirements.

3. TCaV Design Concept

Interface Requirements: TCaV will require a 2 inch line size. Welding is the preferred method of fastening as it will allow for a significant reduction in mass at the interfaces.

Materials: TCaV will be made using 304L Stainless Steel and Monel.

Actuator Interface: An Electro-mechanical actuator will be used to drive to TCaV pintle.

The following illustrations constitute the conceptual TCaV proposed for use with the MR-80B engine for this mission. Figure 1 is the assembled valve. Figure 2 is a cross section showing the internal geometry. This concept is not the team's final design but is similar to the design that is being proposed for manufacturing. The stress analysis that follows is based on this concept. However, the structural thicknesses listed in the stress analysis spreadsheet (Appendix A) will reflect the required thicknesses needed for the NASA's flight requirements. The internal geometries are the same and satisfy the needs of the proposed engine configuration.

The End Cap

While some of the material that makes the end cap can be removed, it cannot be reduced too much. The first design idea for the end cap is to weld the end cap to the body. This is oppose to using bolts, which is the current design for the mating of the feature to its body. If welded, this will cut out the need for any screws/bots. Welding also then leaves the possibility that the thickness of the lip of the end cap can be reduced. The second proposed redesign is to minimize the size of the lip directly as well as reduce the number of bolts and/or the size of the bolts being used. Last is the proposed idea to extend the innermost section of the end cap to eliminate the change in diameter between the tip of the end cap and its mated surface with the body. This will allow for the end cap to serve the purpose of housing the pintle and keep the pintle aligned without having unnecessary material.

The final design that was selected was to weld the end cap to the body. Welding of this part will allow for a better seal of the parts together and it's cheaper to manufacture. There was not much that was able to be changed because of the requirements needed for the actuator, and also for an easier manufacturing process. Once the requirements were met then calculations were done to prove that the redesign that was done will actually be capable of being made and capable of being used in an actual flight.

The Body

This feature will interface with both of the other components. Similar to the other components, the strategy is to get rid of as much excess material as possible with as minimal impact to the interfaces as possible. The corners of the body are over designed and as a result, material will be removed. Fluid initially enters the body at the location marked propellant inlet in Figure 1. The reduction in material of the body was taken primarily from the inlet port walls and from replacing the inlet flange with a prepared end for welding to a 2 inch line. The exit connections (at the seat and end cap) of the body have the limiting factor of only being able to reduce as far as the mating areas of the features connecting to them.

The Seat

The strategy for the seat was to optimize mass reduction by segmenting the seat and performing stress analyses on each segment. This was done because the diameter profile of the seat is not constant and therefore the stresses varied from end to end. This allows us to optimize the wall thickness based on the variation in the diameters along the length of the seat. Another mass reduction opportunity was replacing the engine interface flange with a tube stub for welding to the engine inlet. The inner diameters cannot be changed however, because it will change the proper functioning of the valve. The seat walls will be very thin and will have to be reinforced by machining gussets at the wall near the body interface. This will protect against line loads such as torque and bending moments.

Report From Alabama A&M

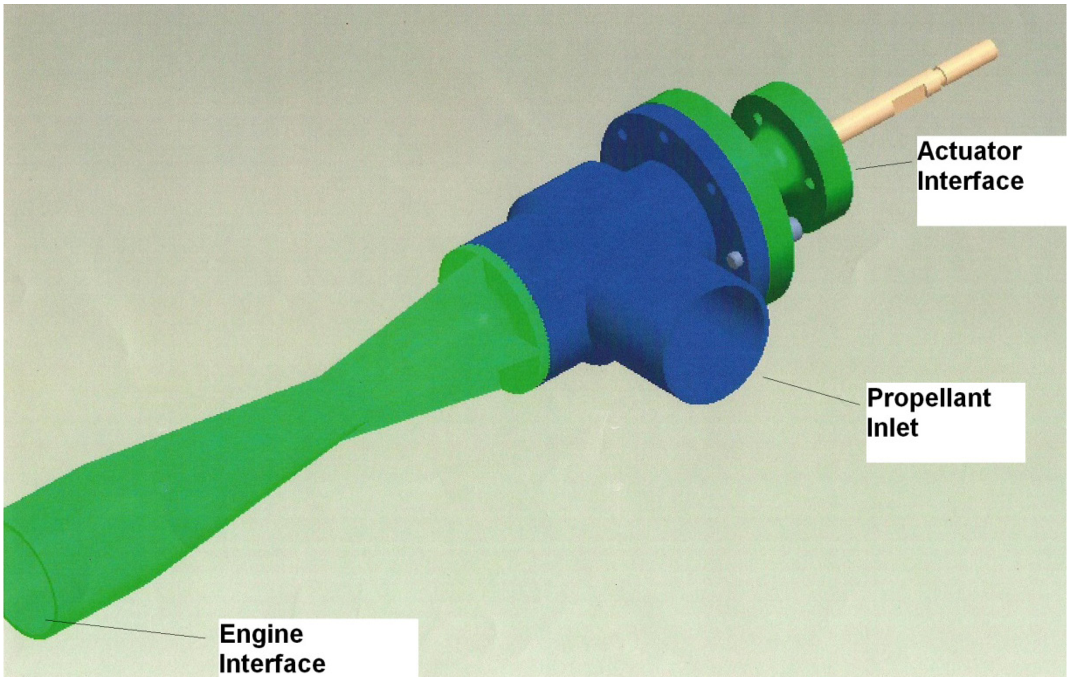


Figure 1. TCaV assembly.

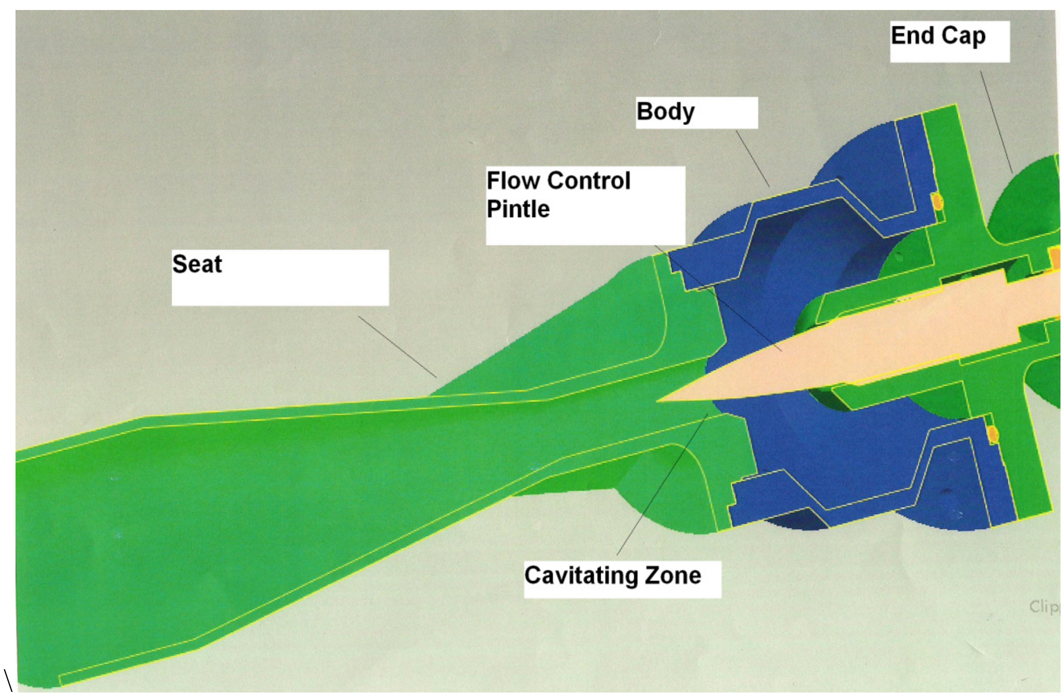


Figure 2. TCaV cross section.

4. Stress Analysis

Structural integrity of TCaV was assessed based on pressures and loads given from NASA's requirements. The following requirements are used for this analysis:

Pressure

Maximum Design Pressure (MDP) will be 2000 psig
Proof Pressure will be 1.5 times MDP = 3000 psig
Burst Pressure will be 2.5 times MDP = 5000 psig

Proof Factor of Safety=1.1
Burst Factor of Safety=1.4

Materials

	Yield (psi)	Ultimate (psi)
304L	25,000	70,000
Monel	55,000	84,000

Stresses created by pressure loads for TCaV were calculated using the following equations:

Longitudinal Stress:

$$\sigma 1_i := \frac{P_i \cdot FS \cdot R_i^{\,2}}{R_o^{\,2} - R_i^{\,2}}$$

Maximum Hoop Stress at inner most point

$$\sigma 2_i := P_i \cdot FS \cdot \left(\frac{R_o^{\,2} + R_i^{\,2}}{R_o^{\,2} - R_i^{\,2}} \right)$$

Radial Stress:

$$\sigma 3_i := -P_i \cdot FS$$

Sheer Stress:

$$\sigma shear_i := P_i \cdot \frac{R_o^{\,2}}{R_o^{\,2} - R_i^{\,2}}$$

Where,

P = inlet pressure

Ro=Outer diameter
Ri=Inner diameter
FS=Factor of safety

Since the combined loads (pressure and line loads) are not yet fully defined, body dimensions in Appendix A only reflect pressure loads.

5. Summary and Future Work.

Based on the engine requirements, the proposed valve configuration will provide a mass flow of hydrazine equal to 9.25lb/s (4.2 kg/s) at 300psia (inlet pressure). The valve flow diameter is approximately 0.464in. A lightweight body has been designed consisting of 304L stainless steel and monel. Pressure loads have been analyzed to ensure structural integrity. Combined loading (line loads + pressure) are still in work but the proposed design includes features that should mitigate any effects of these loads. The gussets located on the valve body are incorporated to prevent failure from torque and bending. Manufacturing and water flow testing are planned to verify flow capabilities.

All AETHER team members have read and signed the following statement: "I acknowledge that I have been identified by name as a team member for the proposed project entitled “Radio Astronomy on the Moon”, which is being submitted in response to the Announcement of Opportunity, Discovery 2010, NNH10ZDA007O, and I intend to carry out all responsibilities identified for me in this proposal. I understand that the extent and justification of my participation as stated in this proposal will be considered during peer review in determining in part the merits of this proposal. If you have any questions, please feel free to contact me at any time."

Team Member	**Signature**
Joel Grissom	[illegible]
Garrett Gammon	Garrett Gammon
Sam Bennett	Samuel Bennett
Megan Beattie	Megan Beattie
Jamison McAllister	[illegible]
David Moore	
Clayton Pannell	
James Pearson	James Pearson
Matthew Wright	Matt Wright
Brittany Gibbs	
Maria Munn	
Heather Meyer	Heather Meyer
Ryan Wilkie	Ryan Wilkie
Jesica Trucks	
Lindsay Boonarkat	
Trevor Montgomery	
Phillip Betts	
Sean Goodman	
Lucas Schoukroun	
Antoine Bercault	
David Langlois	
Astrid Robert	