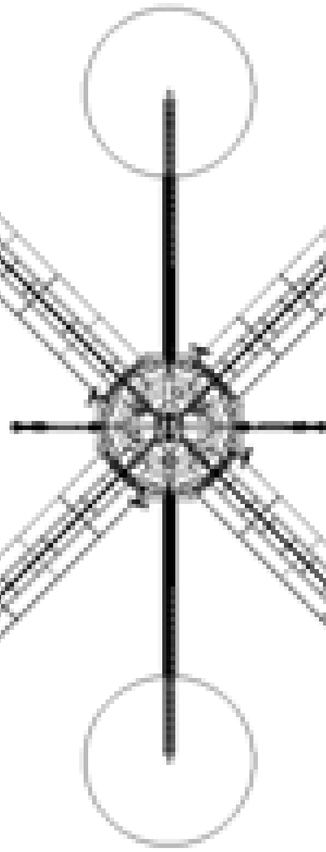
The University of Alabama in Huntsville | Integrated Product Team | MAE 491/492

Mission Concept Review

Radio Astronomy on the Moon

LIBRA



UAH
The University of Alabama in Huntsville

Thomas Bender
Daniel Morrow
Andres Buse
Tyler Earley
David Perrin
Dustin Drake
Sharon Deerman
Jonathon Griffis
Robert Smith
Susan Gardner
Alexandra Aruwajoye

COLLEGE of CHARLESTON

Alex Greene
Samantha Geltz
Jonathon Hegler

Matthew W. Turner, Ph.D
MAE 491 & MAE 492 Instructor
Mechanical and Aerospace Engineering Department
The University of Alabama in Huntsville

Team LIBRA

RAM

Radio Astronomy on the Moon

April 29, 2011

Alex Greene
Principle Investigator
66 George Street
Charleston, SC 29424
greeneax@gmail.com, 513-309-2866

Thomas Bender
Project Manager
685 Providence Main St. NW
Huntsville, AL 35806
tb7792003@yahoo.com
256-557-0228

Matthew Turner, PH.D.
Integrated Product Team Manager
The University of Alabama in Huntsville
301 Sparkman Drive
Huntsville, AL 35899
turnerm@uah.edu, 256-509-6118

Daniel Morrow
Chief Engineer
dcm0004@uah.edu
205-527-8868

Philip Farrington, PH.D.
Engineer Program Director
The University of Alabama in Huntsville
301 Sparkman Drive
Huntsville, AL 35899
paf@eng.uah.edu, 256-824-6568

Andres Buse
Lead Systems Engineer
agb0003@uah.edu
256-990-8897

Mission: December 14, 2017 – December 14, 2022

Certification of Compliance with Applicable Executive Orders and U.S. Code by submitting the proposal identified in the Graphic Cover sheet and/or the Proposal summary Information in response to this Announcement of Opportunity, the Authorizing Official of the proposing organization (or individual proposer if there is no proposing organization) as identified on the cover of this proposal:

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- 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 stipulation set forth in the three Certifications contained in this AO (namely, (i) the Assurance of Compliance with the NASA Regulations Pursuant to Nondiscrimination in Federally Assisted Programs, (ii) the Certification Regarding Debarment, Suspension, and Other Responsibility Matters Primary Covered Transactions, and (iii) Certification Regarding Lobbying).

Willful provisions of false information in this proposal and/or its supporting documents, or in reports required under an ensuing award, are a criminal offense (U.S. Code, Title 18, Section 1001).

Thomas Bender
Project Manager, UAHuntsville
tb7792003@yahoo.com, 256-557-0228

Daniel Morrow
Chief Engineer, UAHuntsville
dcm0004@uah.edu, 205-527-8868

Alex Greene
Principle Investigator, College of Charleston
greeneax@gmail.com, 513-309-2866

Samantha Geltz
Co-Investigator, College of Charleston
sngeltz@edisto.cofc.edu, 843-513-2240

Jonathon Hegler
Co-Investigator, College of Charleston
jhhegler@gmail.com, 803-960-7346

Andres Buse
Lead Systems Engineer, UAHuntsville
agb003@uah.edu, 256-990-8897

Tyler Earley
Mass Properties and ALHAT, UAHuntsville
twe0001@uah.edu, 405-831-2294

David Perrin
Communications, UAHuntsville
dmp0002@uah.edu, 256-655-0175

Robert Smith
Power and Thermal, UAHuntsville
smithr3@uah.edu, 205-706-1388

Jonathan Griffis
Structures, UAHuntsville
jdg0004@uah.edu, 205-305-3115

Sharon Deerman
Cost and CAD, UAHuntsville
sad0002@uah.edu, 256-497-2979

Dustin Drake
Propulsion, UAHuntsville
dwd0001@uah.edu, 256-652-9865

Alexandra Aruwajoye
Technical Editor, UAHuntsville
ana0005@uah.edu, 256-520-5394

Susann Gardner
Technical Editor, UAHuntsville
seg0005@uah.edu, 256-489-0125

Summary

In accordance with the Discovery Amendment of Opportunity, LIBRA designed a mission to implement radio astronomy technology on the far side of the moon. This will be accomplished by two government supplied Atlas V 551 launch vehicles that will utilize the help of two solid rocket motors and an orbiter to place a lander on the far side of the moon. The mission will start from the launch of the two launch vehicles from Cape Canaveral Air Force Station on November 4, 2017. Once the payload separates from the launch vehicle, the orbiter will correct any trajectory misalignment. The solid rocket motor will then insert the lander and orbiter into lunar orbit. The solid rocket motor will then be jettisoned and the orbiter will proceed to correct any trajectory error. Once this occurs the lander and orbiter will orbit for at least one orbit, where after, the lander will start its descent to the lunar surface. During its descent, another solid rocket

motor will be utilized to slow the orbiter down enough so a controlled approach can be implemented. After the jettison of the second solid rocket motor, the lander will utilize its main thrusters and attitude control system to divert from any landing spots not deemed safe by the autonomous landing hazard avoidance technology (ALHAT) system on board. Once the lander has safely touched down, four coilable booms with kapton radio antenna sheets will be deployed horizontally. A central coilable boom will be deployed vertically with cables attached to the end of the four horizontal coilable booms to add stability to the system. Once everything is deployed, radio astronomy data collection will commence. The data will be either be relayed to the orbiter by antennas or will be stored on hard drives when orbiter communication cannot be established. The data will then be relayed, using deep space network (DSN), to earth where it will be analyzed by scientists at the College of Charleston.

Required Proposal Summary Information

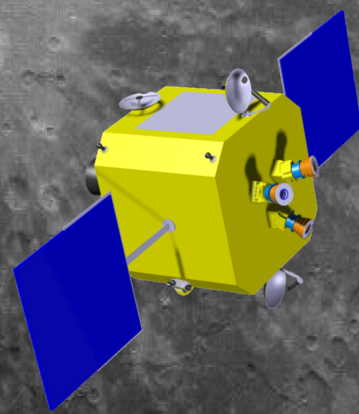
- Proprietary/privileged information is NOT included in this application.
- This project DOES involve activities outside the U.S. or partnership with non-U.S. collaborators.
- NASA civil servant personnel are NOT participating as team members on this project which also includes funding.
- This project DOES NOT have an actual or potential impact on the environment.
- An exemption has NOT been authorized on an environmental assessment (EA) and an environmental impact statement (EIS) has NOT been performed.
- This project DOES NOT have the potential to affect historic, archaeological, traditional cultural sites, and historic objects.
- This proposal DOES NOT contain information or data that are subject to U.S. export control laws and regulation, including Export Administration Regulations (EAR) and International Traffic in Arms Regulations (ITAR).
- The use of radioactive materials is NOT proposed.
- Student Collaboration IS included with the proposed mission.
- NO Science Enhancement Options (SEOs) are proposed.
- There were NO contributions to development or operations from non-U.S. partners.
- The use of NEXT, AMBR, ASRG, and Aerocapture are NOT proposed.
- The proposing institution IS a University.
- This proposal is in response to the Radio Astronomy on the Moon (RAM) concept from the Discovery AO.
- Two Atlas V 551 launch vehicles are proposed and is the highest performance launch vehicle in its class.
- The total Mission Cost is $800M in FY 2010 dollars. (See H. Cost and Cost Estimating Methodology)

LIBRA'S RADIO ASTRONOMY MOON MISSION

A FUTURE MISSION CONCEPT

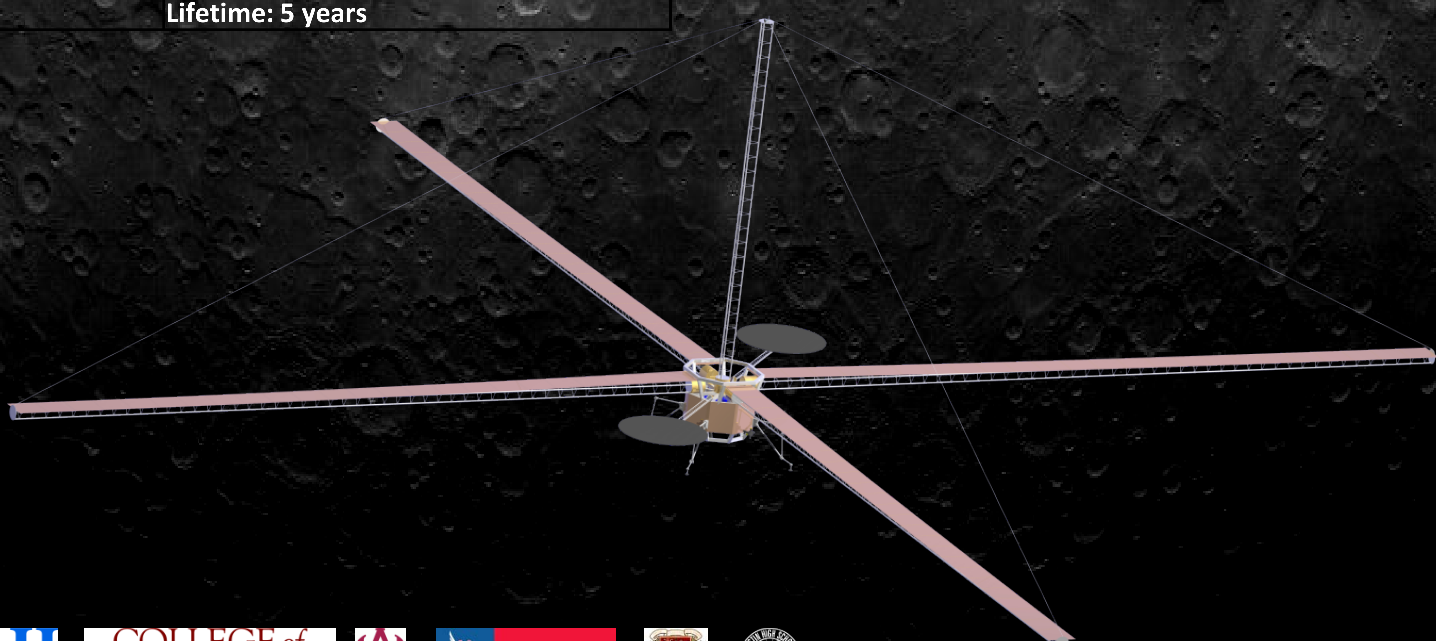
Science Goals	Science Objectives
Understand and probe the structure and evolution of the early universe	Determine the structure of neutral hydrogen. Determine when the first stars formed. Determine the physics of the epoch of reionization. Determine when the global transition between a neutral and ionized universe happen. Create a topographic map of the epoch of reionization. Detect and study early galaxy evolution. Explore the power spectrum of the 21-cm transitions.
Understand the sun and its effects of the solar system	Trace coronal mass ejections as they propagate towards earth. Improve space weather predictions
Observe interaction between the lunar regolith and high energy particles	Understand the origin and nature of ultra high energy cosmic rays. Detect ultra high energy cosmic rays.

Science Payload	
Lander	Lifetime: 5 years
	DALI: frequency range (40-150MHz), temperature sensitivity (10mK)



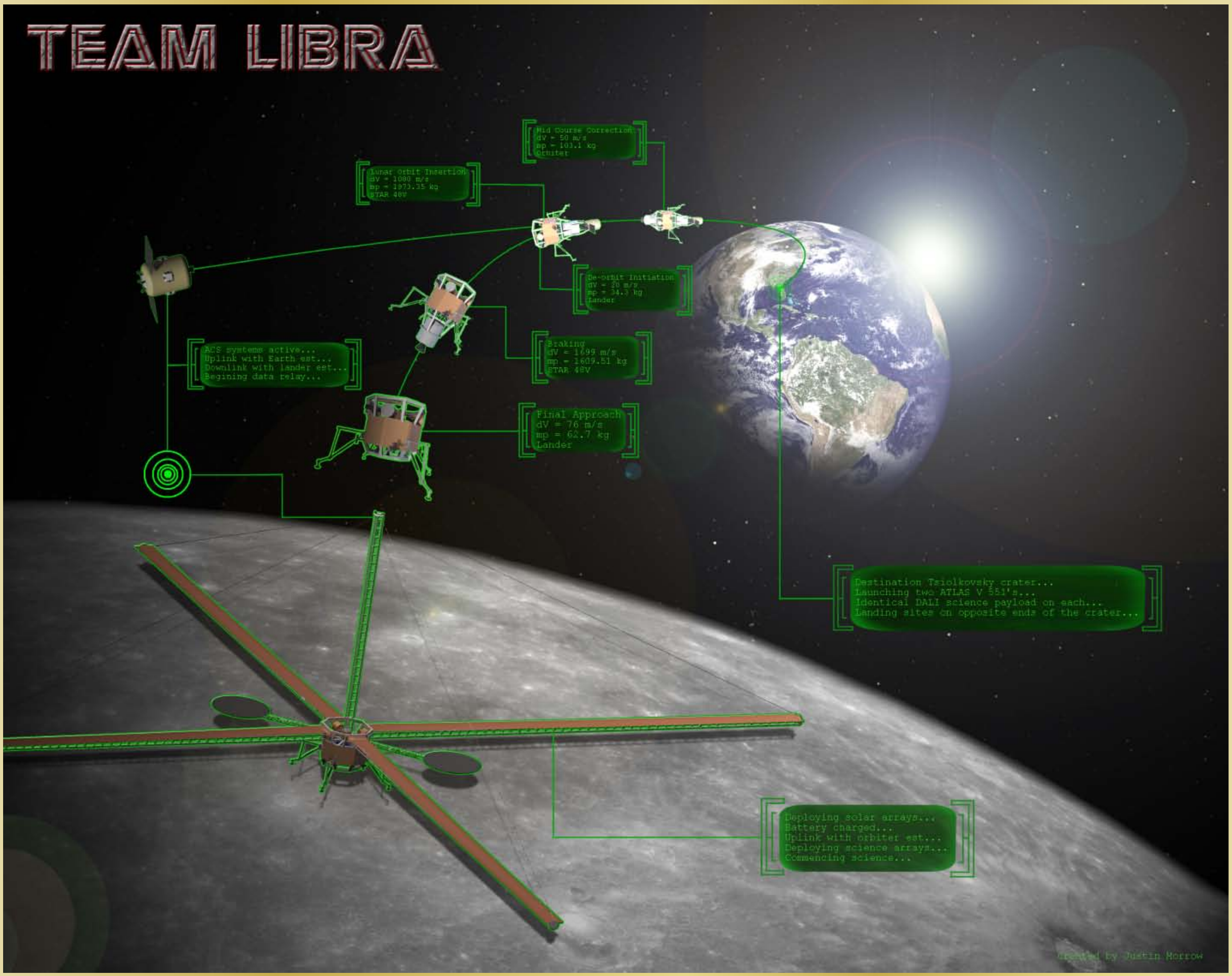
Lander	
Mass	790.9 kg (w/o propellant)
Power	15 m^2 GaAs triple junction solar cells
	250 kg of batteries at 220W*hr/kg
Communication	Ka-Band
Propulsion	3 Aerojet MR-80B Main Thrusters
	12 MR-106L and 4 MR-120 ACS Thrusters
Functions	Collect data from LRA and relay data to orbiter.
	Three burn maneuvers
	Lifetime: 5 years

Orbiter	
Mass	127 kg (w/o propellant)
Power	1.28 m^2 QIOPTIQ Solar Relectors
	20 VES-180 Batteries
Communication	Ka-Band
Functions	Relay data from lander on lunar surface to earth.
	Two burn maneuvers
	Lifetime: 5 years



LIBRA'S RADIO ASTRONOMY MOON MISSION

A FUTURE MISSION CONCEPT



Maneuver	Purpose	Performed By	ΔV (m/s)	Isp (s)	m_p (kg)
1	Correct trajectory after centaur jettison	Orbiter	50	312	105.7
2	Slow payload to allow entry into lunar orbit	Solid Rocket Motor	1080	294.2	1973.35
3	Correct any solid rocket motor thrust vector misaligment	Orbiter	30	312	39.9
4	Push lander out of lunar orbit	Lander	20	231-200	34.3
5	Slow lander to allow for reasonable approach speed	Solid Rocket Motor	1699	294.2	1609.51
6	Slow lander enough for minimum 9g landing	Lander	76	231-200	62.8
7	Divert to a suitable landing spot determined by ALHAT	Lander	19	231-200	15.1

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D. Science Investigation

D.1 Scientific Background, Goals, and Objectives

The primary scientific drivers for LIBRA are the science goals, each of which represents significant advances in the scientific community that can be achieved with LIBRA. In order to understand and probe the structure and evolution of the early universe, LIBRA will focus on several key scientific objectives using the DALI concept for dipole antennas. DALI outlines a concept for an array of dipole antennas with a frequency range of ~40-150 MHz and a temperature sensitivity of ~10 Mk. They are arranged on polymer sheets of up to 1000 antennas on a polymer sheet ~100m x 1m x 20 microns. The first objective LIBRA has is to determine the structure of neutral hydrogen at high redshift ($6 < z < 30$) through observation of the 21-cm line of neutral hydrogen in emission and absorption. The expected frequency range for this signal is 60-150 MHz requiring integration times of 2 hours to 20 days and stable antenna conditions. The far-side lunar surface provides an excellent surface for this radio array because the far side shields these sensitive dipoles from terrestrial inference, solar radio bursts, and terrestrial radio burst all of which dominate this ultra low frequency spectrum. Secondly, LIBRA will attempt to determine the distribution of dark matter throughout the early universe through an examination of the emission and absorption of the 21-cm line of neutral hydrogen and redshifts of ($10 < z < 100$). The expected frequency range is 30-45 MHz with an integration time of .3 to 30 years requiring stable antenna conditions and very precise band-pass calibration for foreground subtraction. The third objective is to determine the physics of the epoch of re-ionization, which will largely be coupled to the observation of the structure of neutral hydrogen. LIBRA will observe the 21-cm line of hydrogen in absorption over time as the temperature of the gas heats up to the temperature of the cosmic microwave background.

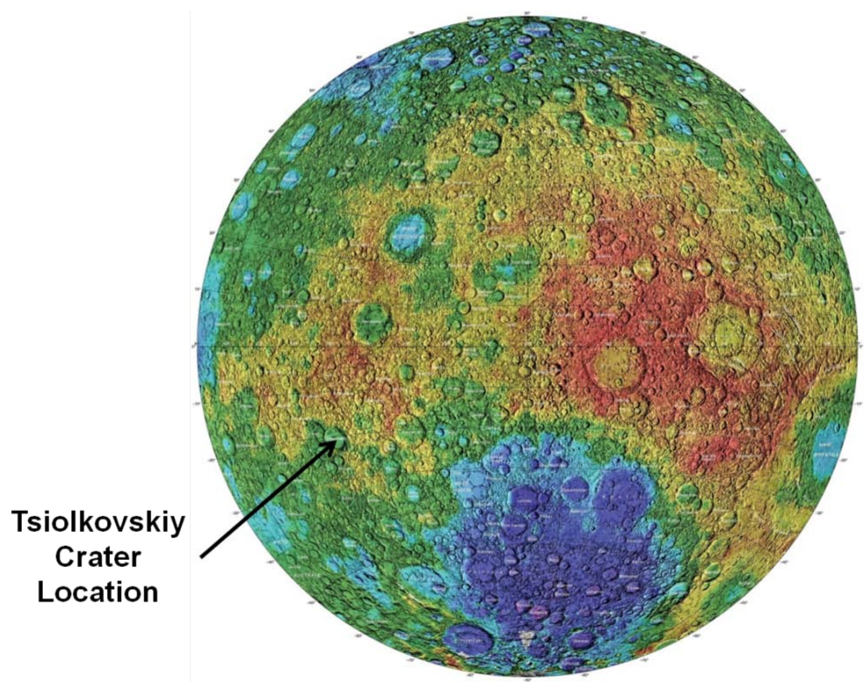


Figure 1. Tsiolkovskiy Crater Location

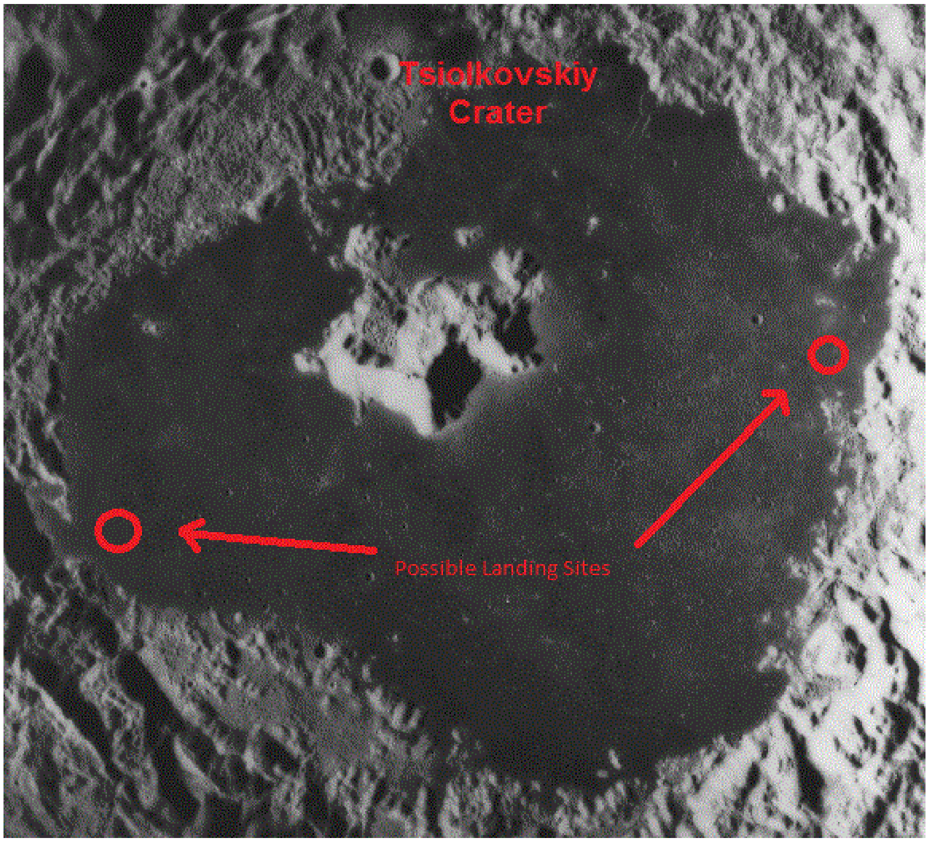


Figure 2. Landing Sites

With the expected frequency being 50-150MHz and an integration time of 2hours – 20 days, this should reveal detailed physics of re-ionization. The fourth objective is to determine when the global transition between a neutral and ionized universe happened. Through observations of the emission and absorption of the 21-cm of hydrogen and measuring the brightness temperature change at frequencies of 50-150 MHz, precise measurements should be able to constrain this transition to a few million years. The fifth objective is to create a tomographic map of the Epoch of Re-ionization by mapping the emission of the 21-cm line of hydrogen at frequencies of 50-150 MHz over a period of time on the order of one half to 5 years depending on the filling factor of the arrays of LIBRA. The detailed history of re-ionization can be traced by observing the two-dimensional structure of neutral and re-ionized gas around luminous objects. The sixth objective is to detect and study early galaxy evolution. This will be done through the study of synchrotron emission from the radio lobes in a frequency range of 50-150 MHz and an integration time of 2 hours to 20 days. Due to the need for small angular resolution interferometer is needed to study the galaxy in detail. The eighth objective is to observe the 21cm power spectrum, which would yield much more information about the density fluctuations in the early universe than direct observations of the CMB power spectrum. This makes the 21-cm line a powerful tool to constrain all model parameters necessary to describe the Universe.

The second goal is to understand the sun and its effects on the solar system through the study of type II and III Coronal Mass Ejections (herein referred to CME's). LIBRA's objectives are to trace CME's as they propagate towards earth, which will be done by observing the synchrotron emission from propagating electron streams from type III bursts and coronal shock waves from type II bursts. By tracing these bursts we will have a better understanding the origin and nature of these bursts in the hopes to improve space weather predictions for our robotic and human explorers.

The third goal is to understand the origin and nature of ultra-high energy cosmic rays. Ultra-energy particles can be detected through their intense particle cascades, which they initiate when encountering a target. Due to the finite length of the cascade of a few meters, the emission must be coherent at long radio wavelengths. The cascade will then show up as coherent ultra-short radio bursts of frequencies 1 – 100 MHz. Additionally, by observing the synchrotron emissivity towards different HII regions a 3d map of the electron density distribution can be built. Since the frequency of synchrotron emission scales with the particle energy, low-frequency observations can be used to trace the energy distribution of the lowest energy particles. This could help constrain the “injection problem” of supernova remnants where thermal particles are not accelerated by the Fermi mechanism to very high and can be studied by LIBRA. The lunar regolith provides a unique environment for the study of cosmic rays due to its radio quiet environment. Interaction between the lunar regolith and cosmic rays can be studied by a radio array as Ultra-high energy particles are deflected and interact with the array. These interactions can be measured and if the interaction is seen by 3 or more antennas the location and origin can be established.

D.2 Science Requirements

Hydrogen is the dominant component of the IGM, and neutral hydrogen (H I) displays a hyperfine spin-flip transition at a frequency of 1420 MHz. The primary scientific drivers for LIBRA are the science goals, which would provide the scientific community with one of the greatest data sets in recent history. Through a detailed mapping of H I line brightness temperature LIBRA hopes to understand and probe the structure and evolution of the early universe. To do so LIBRA will focus on several key scientific objectives using the DALI concept for dipole antennas. DALI outlines a concept for an array of dipole antennas with a frequency range of ~40-150 MHz and a temperature sensitivity of ~10 Mk. Made up of crossed dipole antennas they are arranged on polymer sheets of up to 1000 antennas on a polymer sheet ~100m x 1m x 20 microns. Their low cost and mass make them ideal for a mission to the far side of the Moon. A fundamental question of current cosmological research is the nature of structure formation in the universe and how the observed structures formed from the initial conditions after the big bang. After the Epoch of Recombination the universe became opaque to visible light due to the neutral hydrogen absorbing visible and infrared light and emitting them in random directions. This is now called the called the ‘Cosmic dark ages’ at the redshift range $1000< z <30$. Once early hydrogen and helium had cooled substantially the first stars and galaxies could start to form to emit enough UV and X-ray photons to re-ionize the neutral hydrogen. This time is called the Epoch of Re-ionization. The first objective LIBRA has is to determine the structure of neutral hydrogen at high redshift ($6 < z < 30$) through observation of the 21-cm line of neutral hydrogen in emission and absorption in this Epoch. It is unknown whether this re-ionization happened more or less instantaneously. The precise point where emission transitions to absorption directly reveals when re-ionization occurs and the redshift evolution encodes the detailed physics of re-ionization. The expected frequencies for this signal is 50-150 MHz requiring integration times of 2 hours to 20 days and stable antenna conditions to step through the different redshifts corresponding to this frequency range. The far-side lunar surface provides an excellent surface for this radio array because the far side shields these sensitive dipoles from terrestrial inference, solar radio bursts, and terrestrial radio burst all of which dominate this ultra low frequency spectrum.

Secondly, LIBRA will attempt to determine when the first stars formed in the early universe through an examination of the emission and absorption of the 21-cm line of neutral hydrogen and the spin temperature evolution. Due to the relatively weak signal of the H I 21cm line, the lunar far side provided a pristine environment to study this. When the first stars form, the spectrum will move from absorption to emission and LIBRA should provide a very precise time constraint of this transition. The expected frequency range is 30-60 MHz with an integration time of .up to 1 year requiring stable antenna conditions and very precise band-pass calibration for foreground subtraction.

The third objective is to detect and study early galaxy and black hole evolution. After the first stars formed, the neutral hydrogen signal will have reset back to its relaxation temperature, which as galaxies and black holes form, will begin to be seen in absorption and emission again. This will be done in a frequency range of 60-150 MHz with an integration time of 2 hours to 1 year. Due to the need for high angular resolution, an interferometer is needed to really study the galaxy in detail.

The fourth objective is to observe the 21cm power spectrum, which would yield much more information about the density fluctuations in the early universe than direct observations of the CMB power spectrum. The CMB radiation itself observed today carries information about cosmological parameters mainly at the largest angular scales. However, by contrast, the angular power spectrum in the redshifted 21-cm line carries cosmological information at much smaller angular scales of 1’ or less. In addition, redshifts in the range 30-50 yield independent samples of the cosmological parameters while the CMB suffers from cosmic variance. This makes the 21-cm line a powerful tool to constrain all model parameters and density fluctuations necessary to describe the early Universe.

The second goal is to understand the sun and its effects on the solar system through the study of type II and III Coronal Mass Ejections (herein referred to CME’s). LIBRA’s objectives are to trace CME’s as they propagate towards earth, which will be done by observing the synchrotron emission from propagating electron streams from type III bursts and coronal shock waves from type II bursts. By tracing these bursts we will have a better understanding the origin and nature of these bursts in the hopes to improve space weather predictions for our robotic and human explorers.

The third goal is to understand the origin and nature of ultra-high energy cosmic rays. Ultra-energy particles can be detected through their intense particle cascades, which they initiate when encountering a target. Due to the finite length of the cascade of a few meters, the emission must be coherent at long radio wavelengths. The cascade will then show up as coherent ultra-short radio bursts of frequencies 1 – 100 MHz. Additionally By observing the synchrotron emissivity towards different HII regions a 3d map of the electron density distribution can be built. Since the frequency of synchrotron emission scales with the particle energy, low-frequency observations can be used to trace the energy distribution of the lowest energy particles. This could help constrain the "injection problem" of supernova remnants where thermal particles are not accelerated by the Fermi mechanism to very high and can be studied by LIBRA. The lunar regolith provides a unique environment for the study of cosmic rays due to its radio quiet environment. Interaction between the lunar regolith and cosmic rays can be studied by a radio array as Ultra-high energy particles are deflected and interact with the array. These interactions can be measured and if the interaction is seen by 3 or more antennas the location and origin can be established.

Table 1. Science Traceability Matrix

Scientific Goals	Scientific Objectives	Scientific Measurement Requirements		Instruments	Instrument Functional Requirements	Mission Functional Requirements
		Observables	Physical Parameters			
Goal 1: Understand and probe the structure and evolution of the early universe	Determine the structure of neutral hydrogen	Emission of the 21-cm line neutral hydrogen	Differences in the background emission and absorption temperatures	DALI frequency range: ~40-150 MHz Temperature Sensitivity: ~10mK	Frequency: 50-150 MHz Integration time: 2h-20d	Stable temperature conditions, Dedicated signal processing chain
		Absorption of the 21-cm line neutral hydrogen				
	Determine the when the first stars and galaxies formed	Emission of the 21-cm line neutral hydrogen	Differences i the Background emission and absorption temperature compared to CMB		Frequency: 40-55 MHz Integration time: Up to 1yr	Stable antenna conditions, Band-pass calibration of the antenna for foreground subtraction
		Absorption of the 21-cm line neutral hydrogen				
		Spin temperature evolution				Stable antenna conditions
	Determine the physics of the epoch of Reionization	The redshift evolution of the 21-cm brightness temperature	Change of the 21-cm brightness temperature as a function of redshift		Frequency: 50-150 MHz Integration time: 2h-20d	
	Detect and study early galaxy evolution	Emission of synchrotron radiation	Fossil radio galaxy lobes		Frequency: 50-150 MHz Integration time: 5h-1yr	Long baseline, radio quiet environment, long integration times, accurate subtraction of spectral and noise
	Explore the power spectrum of the 21cm-	Changes in the slope of the spectrum	Rate at which the composition of the universe			

	Explore the power spectrum of the 21cm-transitions	Changes in the slope of the spectrum	Rate at which the composition of the universe changes			
Goal 2: Understand the sun and its effects of the solar system	Trace coronal mass ejections as they propagate towards earth	The propagation of electron streams (Type III bursts)	Coronal Mass Ejections		Frequency: 30-50 MHz Integration time: 1min-110hr	Day side observing, and interferometer for angular discrimination between different burst sources
		Propagation of Coronal Shock waves (Type II)				
	Improve space weather predictions	Low-frequency radio emission from the sun				
Goal 3: Observe interaction between the lunar regolith and High energy particles	Understand the origin and nature of Ultra-high energy cosmic rays	Synchrotron emissivity towards different HII regions	Radio Pulses originating below the detector		Frequency: 1-100 MHz Integration time: N/A (bursts)	Radio quiet environment
	Detect Ultra-high energy cosmic rays					

D.3 Threshold Science Mission

D.3.1 Baseline

LIBRA's baseline mission would in have two landing sites (one at each pole) resulting in an interferomic baseline diameter of 2-3 Kilometers yielding in a higher angular resolution on this mission then ever achieved at this frequency. Integration times would range anywhere from one second to the entire lifetime of the mission. Under these conditions our mission would yield definitive science in several key areas of astrophysical and heliocentric physics. LIBRA’s primary objective would delve into the early universe by analyzing the 21-cm hydrogen line spectrum as it evolves through redshifts. As this is one of the last completely unexplored frontiers LIBRA stands to yield one of the richest cosmological data sets in history. Due to the moons day and night cycles, 14 days of daylight present a unique opportunity to analyze type II and III solar coronal mass ejections and provided man and robotic explorers in space with an early warning system for this type of space weather.

D.3.2 Threshold

LIBRA's threshold mission would have one landing site to detect signal from the epoch of re-ionization, dark ages, ultra high-energy particles and solar bursts resulting in longer integration times and less angular resolution. Still using the DALI concept for the array our dipole will operate at its frequency range of ~40-150 MHz. As only one site is being considered for the threshold mission all sky coverage is not possible however a plethora of science is still accessible. From one site decreased baselines mean longer integration times but as the mission's expected lifetime is on the order of years this is still within the acceptable requirements of the AO and still provide a preponderance of new and accessible data to warrant the launch of the mission.

E. Science Implementation

E.1 Instrumentation

In order to understand and probe the structure and evolution of the early universe LIBRA will focus on several key scientific objectives using the DALI concept for dipole antennas. DALI outlines a concept for an array of dipole antennas with a frequency range of ~40-150 MHz and a temperature sensitivity of ~10 Mk. They are arranged on polymer sheets of up to 1000 antennas on a polymer sheet ~100m x 1m x 20 microns. The Array will employ "multi-beaming" to acquire a sufficient field of view and two identical arrays kilometers apart will ensure redundancy and high angular resolution. Star trackers and triangulation with the orbiters will allow exact distances between sites to be determined upon landing. The minimum specified distance between the sites is one kilometer.

E.2 Data Sufficiency

The instrumentation delivers spectra in the range 40 MHz to 150 MHz and will be the final science data once various foreground contributions have been removed. Data validation and calibration occur once the array has been fully deployed and determined to be operational. Several basis on which we will decide whether the data is acceptable or not are: (1) Whether observed power levels (i.e., brightness temperature *TB*) are consistent with those determined from the known sky temperature distribution, the known power pattern of the Array. (2) Whether the spectra obtained are consistent with expected performance based on the array status. During Instrument calibration, spectra must be combined to verify that the root mean square (RMS) noise levels in the combined spectra decrease as expected with integration time. The data acquired during this phase are not of sufficient quantity to detect any turning points, but an RMS noise level decreasing as t-1/2 is a requirement for science analysis. Other tests focus on looking for variations in spectra, as a function of time or frequency, at a level exceeding that expected from statistical variations. During science operations, spectra not meeting the validation criteria are discarded, a standard procedure in ground-based radio astronomy. Because the signal-to-noise ratio increases with time as t1/2, discarding occasional spectra does not impact mission lifetime. When the data is determined to be acceptable it is then processed for foreground removal and archived.

E.3 Science Mission Profile

After deployment and calibration the array will start recording spectra from the early universe. Integrations times will vary according to the intended target outlines in the science goals and objectives. In order to examine the early universe, the sun, and Ultra high energy particles the integration times will vary between seconds and 1 year according to background calibration and intended target. The array is intended to integrate for the entire lunar night while it recharges its batteries and relays information during the lunar day. Each objectives outlined in the science goals and objectives requires the "radio quiet" environment of the lunar far side as the earth and galactic foreground can saturate the detector signal.

E.4 Data Plan

E.5 Science Team

James A. Greene Principle Investigator

- Leadership of the science team
- Primary link between science and engineering teams
- Generator of all cool and note worthy ideas of any kind of interest
  - All science ideas
  - Having two landing sites to increase angular resolution/sky coverage
  - Using the DALI concept
- Writing the entire proposal
- Attach resume

Samantha Geltz Co-Investigator

- General disruption
- Creation of the poster
- Picking the landing sites
- Sitting still and doing what she is told
- Flirting with the review panel so we win the competition (sex appeal?)
- Attach resume

Johnathan Hunter Hegler Co-Investigator

- Making witty comments
- Creating the presentation
- General amusement
- Attach Resume

E.6 Plan for Science Enhancement Options

As all available weight is to be used on the array for maximum coverage the only science enhancement options are based on longer integration times. Over the course of the mission lifetime a signal from the "Dark Ages" could be detected. The minimal lifetime is expected to be 5 years and it could take anywhere from 5 to 20 years for these low gain dipoles to detect a signal. Additionally given enough time the array could form a tomographic survey of the Epoch of Re-ionization however that would require 20 years or more. This mission does also provide the chance for additions. Arrays could easily be integrated into the system to provide a larger surface array of detection, which would reduce integration times.

F. Mission Implementation

F.1 General Requirements and Traceability

The proposed mission design in this report is based upon science objectives put forth by the Principal Investigator (PI). The science objectives are described in Sections D and E of this proposal. The intent of the mission is to position and implement interferomic radio arrays on the far side of the moon. The arrays will observe the existence of dark matter and coronal mass ejections in outer space that will assist the PI in determining the origins of the universe. The arrays will collect scientific data and transmit it to Earth via satellite. The proposed mission has been designed to implement and fulfill the science objectives. The mission architecture consists of 2 lunar landers and 2 lunar orbiters. The 2 landers will land on the lunar surface and implement interferomic radio arrays for scientific data collection. The 2 orbiters will be inserted into 100 km altitude circular selenocentric orbit and will be responsible for scientific data transmission to Earth via the Deep Space Network. The mission elements have a threshold time window of 5 years of operation.

The Mission Traceability Matrix conveys the synthesis of the mission design, spacecraft, ground systems, and operations requirements from the mission functional requirements and science requirements. The Mission Traceability Matrix is illustrated in Table 2. The overall Concept of Operations (CONOPS) is illustrated in Figure 8.

NASA has also prescribed requirements in the Discovery Announcement of Opportunity (AO) to shape the mission design. These requirements were imposed to ensure that the proposed mission complies with the NASA framework of operations. The incorporation of the AO requirements into the proposed mission is further elaborated on in Sections F.6, G, and H of this proposal.

Table 2. Mission Traceability Matrix

Mission Functional Requirements	Mission Design Requirements	Spacecraft Requirements	Ground System Requirements	Operations Requirements
Stable temperature conditions, dedicated signal processing chain. Stable antenna condistionas, band-pass calibration of the antenna for foreground subtraction. Stable antenna conditions. Stable antenna conditions, band-pass calibration of the antenna for foreground subtraction. Long baseline, radio quiet environment, long integration times. Day side observing and interferometer for angular discrimination between different burst sources. Radio quiet environment.	Launch Vehicle: Atlas V 551 Launch date: November 4, 2017 Mission length: Minimum 5 years Orbit altitude requirement and rationale: 100km Equitorial landing site so orbiter is on a equitorial orbit Type of orbit: Lunar orbit	Spinning at 6 rpm Mass: 6524 kg based on a C3 of -1.8 Power: 82W (Total after margin) Volume: approximately 129 m^3 Data Rate: 1 MB/s data in, 0.01311 MB/s data out Temperature Range for spacecraft systems: approximately -130°C to 130°C Pointing Control: ACS system	Passes per day and duration: 12passes/orbiter for 750s Assumed antenna size: < 8oz. Data volume per day: 2034 Mb (per site/per day) Real time data transmission requirements Transmit frequency: Will vary between sites. Power available for comm (Watts): peak 20W (per site) Downlink data rate: 1kB/s (per antenna) Number of data dumps per day: 24 per day Spacecraft data destination: DSN Science data destination	General spacecraft maneuver requirements and frequency: MCC, LOI, Correction, DOI, Braking, Approach, Divert Rationale for maneuvers Ephemeris requirements Changes in viewing modes and directions per orbit, per day or over longer time periods. Rationale for these changes Other

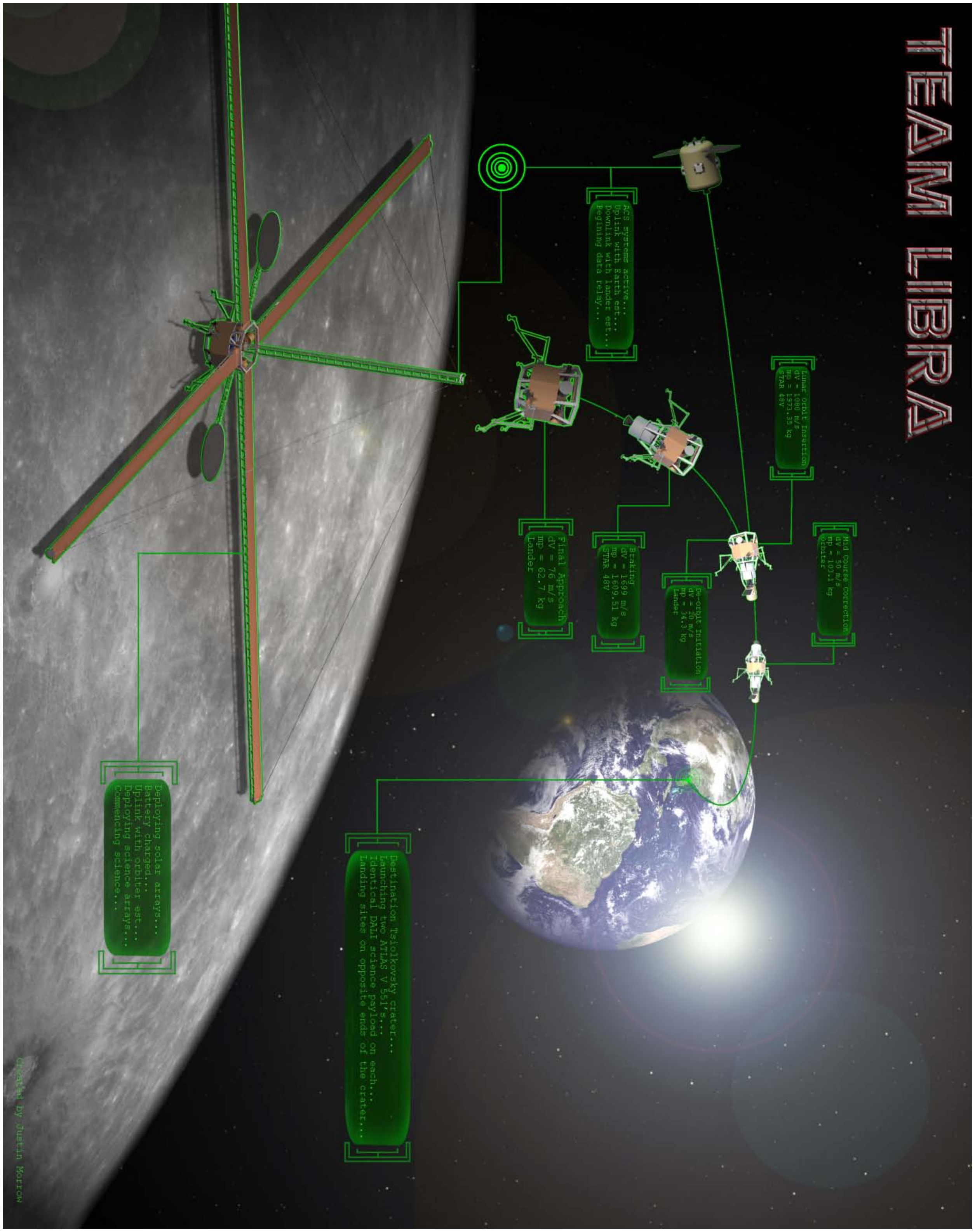


Figure 3. CONOPS

F.2 Mission Concept Descriptions

The mission will begin with the launch of two Atlas V 551 launch vehicles, shown in Figure 1, with a 5 meter short shroud. The Atlas V 551 has a main central Pratt & Whitney/NPO Energomash RD-180 liquid booster engine with five solid rocket boosters [herein referred to as SRB]. Once the SRB's and RD-180 have broken away, a common centaur stage with a Pratt & Whitney RL 10A engine will propel the payload on a trajectory to the moon.

Figure 4. Atlas Launch

The payload is identical for both launch vehicles. It is comprised of an orbiter, two STAR 48B-short solid rocket motors [herein referred to as SRM], and a lander which is shown in Figure 2 below.

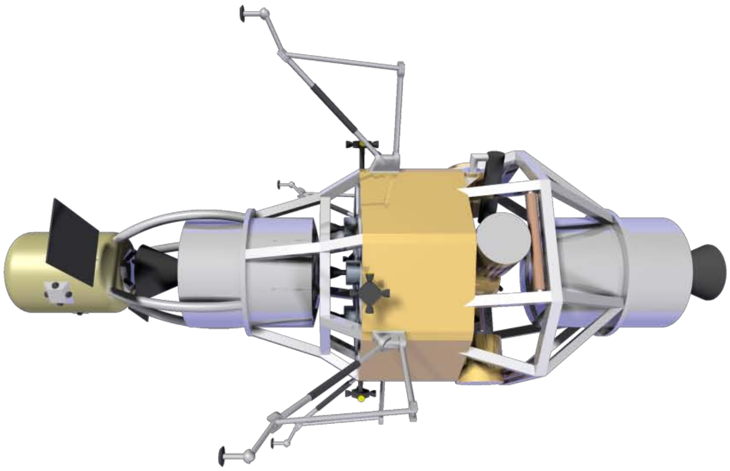


Figure 5. Payload Diagram

The orbiter will be the utilized as the relay point of data between the lander on the lunar surface and earth. It uses a liquid bi-propellant propulsion system to help propel the payload into lunar orbit where it will be jettisoned from the lander. It has an attitude control system [herein referred as ACS] to the orbiter remain in orbit for the life of the mission. It will have the ability to receive and transmit data from the lunar surface to the earth, respectively.

The lander was designed to be the main central hub for the radio arrays and the communication to the orbiter from the lunar surface. It will utilize a liquid mono-propellant pressure fed propulsion system for landing. The radio arrays and solar panels will be deployed from the lander horizontally by coilable booms, developed by ATK. The radio array booms will be arranged in a "+" pattern. There will be a central vertical boom, which has cables connected to the ends of the horizontal booms to help with the stability from the weight of the radio arrays. The science will be conducted from the kapton radio array sheets mentioned above in section E.1. The data will be stored in hard drives aboard the lander until communication can be established with the orbiters. There will be two orbiters that will be placed in opposite sides of orbit so that there is no duplicated data uploaded to both orbiters. This allows us to use the full 750s window of each orbital pass. The data will be relayed from the lander to the orbiter. The orbiter will have the ability to store the data until communication can be reached with earth. Once communication with earth has been reached, the data will be relayed there from the orbiter and can then be compiled and analyzed by the scientists. This process will be repeated constantly until the life of the mission comes to an end.

F.2.1 Mission Element Description

F.2.1.1 Trajectory

For this mission there will be seven different burns that will be utilized to put the orbiter in lunar orbit and the lander on the lunar surface. The calculations have only been done from the point where the payload is jettisoned from the payload shroud after the common centaur stage has been utilized. The initial payload mass that is able to be put into space for the selected launch vehicle was calculated using the C3 in Table 3 below. The breakdown of these burns applies to both launch vehicles because they are identical.

Table 3. Trajectory Information

Trajectory Information	
Launch Vehicle	Atlas V 551
Payload Mass	6524 kg
C3	-1.8 km^2/s^2

Once the payload has separated from the common centaur stage, there will be a mid-course correction [herein referred to as MCC] burn. This burn, done by the orbiter, will allow for any correction in trajectory. As the payload gets closer to the moon, it will need to be inserted into a lunar orbit. The lunar orbit insertion [herein referred to as LOI] burn will be accomplished using one of the SRM. Once this SRM has been used it will be jettisoned. The exhaust from a SRM does not necessarily come out perfectly straight which could set the payload up to a degree off trajectory. For this, a correction burn, done by the orbiter, will be implemented to ensure that the payload will be in a 100km orbit above the lunar surface. Once in orbit, the orbiter will separate from the lander. The lander will utilize a de-orbit initiation [herein referred to as DOI] to put it out of orbit and begin its descent to the lunar surface. At this point the lander is still traveling extremely fast and needs to be slowed down. To slow it down, another SRM, identical to the one that did the LOI burn, will be used for this braking burn. Just like the LOI burn, once the SRM has been used it will be jettisoned and the lander will begin its approach to land on the lunar surface. During this approach phase is when ALHAT will be used. ALHAT will scan the lunar surface as the lander is descending, to find a suitable landing spot. Once one has been found, the lander will rotate vertical at 30 meters above the surface and utilize a divert burn to move the lander to the landing spot determined by ALHAT. The burns that have been mentioned are further illustrated in Table 4 below.

Table 4. Burn Breakdown

Burn Breakdown				
Maneuver	Purpose	Performed By	ΔV (m/s)	Propellant Mass (kg)
1	MCC	Orbiter	50	103.1
2	LOI	STAR 48V	1080	1973.35
3	Correction	Orbiter	30	38.9
4	DOI	Lander	20	34.3
5	Braking	STAR 48V	1699	1609.51
6	Approach	Lander	76	62.7
7	Divert	Lander	19	15.1

F.2.1.2 Concept of Operations

F.2.2 Launch vehicle compatibility

The selected launch vehicles for this mission chosen from a trade study are two Atlas V 551. This launch vehicle is a product of ULA whom provide customers with an Atlas User's Guide. This User's Guide was used for information on the launch vehicle adapters for the different payload fairings. According to this User's Guide, all 5 meter payload fairings come standard with a C22 Payload Adapter shown in Figure 3. The C22 gives a bolt hole pattern to interface the payload with, which is necessary to keep the payload stationary during the launch of the launch vehicle.

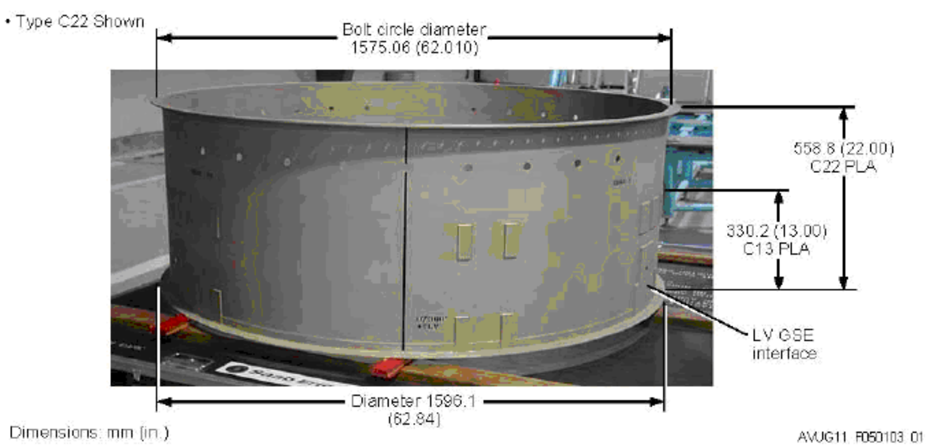


Figure 6. C22 Payload Adapter

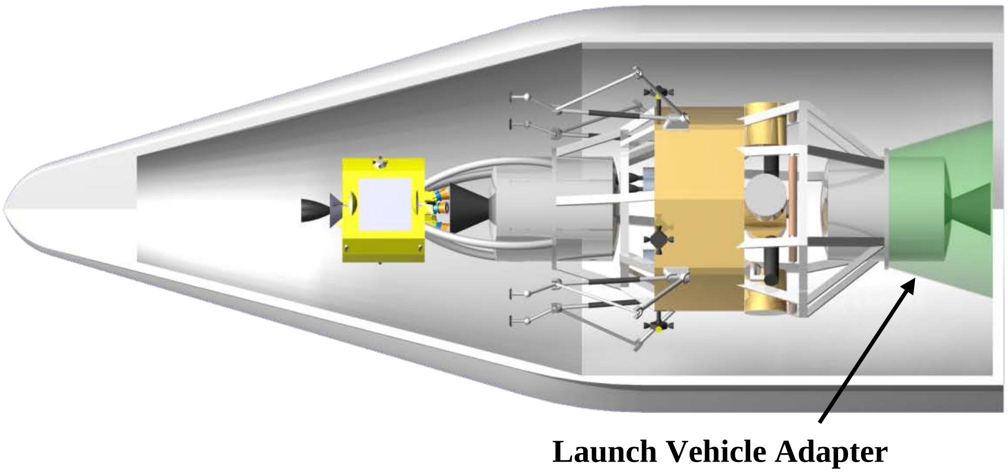


Figure 7. Payload in 5-m Short Shroud

F.2.3 Flight System Capabilities

F.2.3.1 Lander

The lander will be built around the ”Scout ETL” frame. It is a modular frame that is easily applied to other missions. The frame will house all of the subsystems. It will be built to withstand 9g’s. Lander capabilities will be discussed in depth in subsection F.2.3.5.

F.2.3.2 Payload

The payload is the science array. It is 4 1m x 25m sheets of kapton with the array collectors embedded within. They will be deployed using the coilable boom technology. When deployed, the panels will form an “X” shape as required by the science.

F.2.3.3 Mass Breakdown

Before any mass breakdown could be done it was necessary to figure out how much useful mass could be put on the lunar surface. To accomplish this, Tsiolkovsky’s rocket equation

, was used to see how much propellant would be burned during the separate stages mention in section F.2.1.1. For this equation m_p is the mass of the propellant required for the burn, m_o is the mass before the burn takes place, g_o is the gravity constant of earth, ΔV is the change in velocity required for the burn, and I_{sp} is the specific impulse of the element doing the burn. This was done for the stages shown in table 4 above until a final useful mass of 1437.2kg was obtained which is the maximum amount of useful mass that can be put on the lunar surface. From here, LIBRA decided that the mass allotted for the science equipment would be 43% of that final useful mass. From here, it was necessary to divide up the remaining 57% of the mass. The weight of the lander structure was already taken out of the final useful mass, so the structures had mass already allotted for it. The way LIBRA decided to divide up the remaining 57% of the useful mass was to go from one subsystem to another until all the subsystems had been designed. Using this method the propulsion system was designed first.

After the propulsion system was sized for the lander, it was then decided to work on power. Power was sized to figure out how much power consumption there was and then from that power consumption, the number of batteries and solar panels was designed for our lander to run for the lifetime of the mission. Next thermal was looked at and then structures. Structure's was the last subsystem to be fully designed because it was decided to build the structure around all the other components.

F.2.3.4 Power

Power is a critical component of the LIBRA mission. The mission to collect data on the lunar surface involves a massive expenditure of time, money, and knowledge. Team LIBRA's science objective involves using a power intensive radio array to collect scientific data to determine universal origins. This mission will be performed during night operations, and will draw off of an allocated battery system. Secondary Science will be performed by the radio array during lunar day to analyze electromagnetic solar emissions. This will draw primarily from the solar panel collection system. Tertiary science will also be performed by a system designed in the InSPIRESS design completion, and will operate day and night to collect data. Power profile for a current technology readiness level is proposed in Table 4.

Power for the mission will be generated by a system of batteries and solar panels. For night-time operation, Lithium-Ion batteries in conjunction with a light heater system and insulation will provide ~ 27.5 kW*hr of overall use. During daylight hours, LIBRA will be powered by a 15 square meter solar array. The Solar panels chosen from this mission are from Spectre Laboratories. These panels were chosen for their extremely high levels of efficiency and durability. Spectre has several versions of their UTJ cells already in orbit, deeming LIBRA's UTJ solar panels as TRL 9.

The Dark Ages Lunar Interferometer (DALI) concept envisions a series of small radio arrays clustered around a central processing hub that sends information to an orbiter. Each of these arrays will require a preamplifier to send data back to the storage hub. Each pre-amplifier is expected to have a draw of .1W, but current technology is still approximately 1W of draw. For a real world benchmark, LIBRA found a Japanese mission launched in 1997. The "Highly Advanced Laboratory for Communications and Astronomy" (HALCA) mission used a radio telescope placed in an elliptical orbit around Earth for a combination of Earth and Space based interferometer. Using HALCA as a worst case modern example of maximum power draw, calculations show that LIBRA can expect a constant radio array draw of 13.9 W/M^2. This number is much higher than the Lunar Radio Array (LRA) papers would indicate, but there is no concrete counter to this assumption. All of LIBRA's analysis is based on worst case [current technology] calculations. Team LIBRA's assessment of power calculations are the current basis for power consumption, and represents technology that flew in 1997.

Table 5. Current State of the Art Power Profile

	POWER IN OPERATING MODE (W)					
SYSTEM	Cruise	Brake	Landing	Deployment	Day Operation	Night Operation
GN&C	1	19	19	0	0	0
Avionics	9	9	26	10	10	10
Power	1	1	1	150	0	0
RF/Comm	20	20	20	20	15	5
Thermal	30	30	20	20	20	20
Battery Recharge	0	0	0	0	45	0
INSPIRES Miss.	0	0	0	0	10	10
SCIENCE	0	0	0	0	1116.8	50
Harness Loss	1.83	2.37	2.58	6	36.504	2.85
Total Load	62.83	81.37	88.58	206	1253.304	97.85
TL + 30% Margin	81.679	105.781	115.154	267.8	1629.2952	127.205

Power consumption during nighttime hours will be split into 4 main points of consumption. The critical nighttime system is the heater system. As shown in table 5, this system will always draw power until the end of the mission. The second main power draw will be the Ka-Band transmitters. A pair of Octane Wireless Ka-12 non-directional transmitters will consume an average of 5 W over the course of the night mission. This average value represents a total draw of 20W that will be utilized during all satellite overpasses. The functionality of this system is second only to thermal protection of the batteries, given that unfiltered data rates into the LRA are a fraction of a percent of available Ka uplink bandwidth. Priority 3 is the DALI array, which will consume a calculated average of 50W over the period of lunar night. Full power at the worst case of 14 W/m^2 would drain the batteries to 50% in approximately 12 hours. This limited time of data collection is sub-optimal, but the power availability is a similar limitation to the data collection and compression. The last power allocation is for the InSPIRESS science mission. The winning design is allocated an average draw of 10W through the lunar night for a secondary science mission.

Table 6. Low Power Operational Contingencies

Lander Power Priorities		
	System	Draw [avg]
Priority 1)	Thermal	20 W
Priority 2)	Communications	5 W
Priority 3)	Radio Array	50 W
Priority 4)	INSPIRESS	10 W

Daytime power is sourced from a 15m^2 array of GaAs triple junction solar cells. These cells are provided by Spectre Laboratories. Solar Power calculations are based on current production models, and solar perpendicularity produces an EOL 25.3% efficiency. The daytime mission has calculated that the power required to run the arrays should be less than 1500 W. This means that the all science may operate continuously, at maximum power, for approximately 250 hours of lunar day. This 250 hour period was calculated do compensate for the lack of solar tracking. Calculations assume no power generation under 30 degrees of solar horizon. Requirements are modeled after Spectre UTJ cells on a Germanium substrate. These cells are currently in product and in use, and offer an extremely potent mass density, coupled with yearly power production degradation of approximately 1%.

There are many trade-offs associated with a battery system and the LRA concept. It is given with the high power utilization rate of the HALCA system, that no batteries can run the full panel system at full power for the duration of lunar night. With the current iteration of LIBRA design, running full power for the entire night cycle would require 3500 kg in battery mass per site. This battery mass is considerably beyond available mission architecture. With this limitation in mind, LIBRA's mission is planned for a worst case scenario. Power goals are set with modern technology. From there, further incremental advances in technology serve to increase mission ability beyond threshold. With current tech in mind, LIBRA has planned to run partial sampling at night. LRA power and data usage are designed to be scalable, and running in low power/data collection is a perfect way to trade threshold for mass requirements.

Radioisotope generators were not considered for this mission. RTGs were declared unusable as a mission constraint, but their feasibility must be recognized as an alternative. The Stirling engines that NASA is developing show real promise, but may not be capable of managing the high power draw the LIBRA array would need to draw. If RA requirements drop below 5W/m^2, a feasibility study on the mass savings of a RTG would be reconsidered.

With the LIBRA mission launch window 6 years in the future; there is room for technological development beyond the current state of the art. The LRA documentation calls for power draw of 1W from a signal amplifier for each square meter receptor. This number's current feasibility has been asserted, but remains unverified. When this number were achievable and coupled with expected battery power density gains of 50% beyond current state of the art availability, LIBRA will be able to operate fully through the lunar nights. A power density of 350W*hr/kg is a legitimate goal within the mission development cycle. Non-Rechargeable batteries from Saft using a Lithium-Thionyl Chloride core currently are capable of 400 Watt-hours per kilogram. A breakdown of idealized achievable future power consumption is shown in Table 6. Table 4 and Table 6 represent the differences between current and ideal technology levels. These tables display the average draw of the LRA, but leave out one crucial detail, the time of exposure for power draw. Current state of the art technology allows us to meet threshold, but with only 12 hours of night-time data collection. Increasing battery power density and reducing power throughput to match non-rechargeable state of the art will allow for 14 days of continuous collection. A breakdown of idealized achievable future power consumption is shown in Figure 6b. If this power density is also achievable in the near term with Li-Ion rechargeables, the science goals of LIBRA will be more readily achievable beyond current mission planning. Current state of the art allows for data collection beyond threshold, but battery technology development allows a considerable improvement in data collection and quality.

Table 7. Idealized Future Power Profile

	High Power, low Draw: POWER IN OPERATING MODE (W)					
SYSTEM	Cruise	Brake	Landing	Deployment	Day Operation	Night Operation
GN&C	1	19	19	0	0	0
Avionics	9	9	26	10	0	0
Power	1	1	1	150	20	20
RF/Comm	20	20	20	20	15	5
Thermal	30	30	20	20	20	20
Battery Recharge	0	0	0	0	45	0
INSPIRES Miss.	0	0	0	0	10	10
SCIENCE	0	0	0	0	500	150
Harness Loss	1.83	2.37	2.58	6	18.3	6.15
Total Load	62.83	81.37	88.58	206	628.3	211.15
TL + 30% Margin	81.679	105.781	115.154	267.8	816.79	274.495

F.2.3.5 Structures

Currently, the lander is based on a former design known as SCOUT-ETL. However, the lander will be of octagonal shape using the leg design of SCOUT-ETL. The lander will contain the booms for the antenna, solar panels, and radio array, respectively, mounted to the top of the lander in the stowed position. Inside the lander chassis, the propulsion systems, thermal systems, and power systems will occupy the majority of the space. The exact dimensions and layout are yet to be determined but will be finalized (for our purposes) within the next week. Once this is completed, CAD (computer aided drawings) will be rendered.

F.2.3.6 Thermal

Thermal requirements for a lunar mission are of a brutal nature. Given the lack of a protective atmosphere, and long day/night cycles, the heat cycles on the lunar surface are extensive and extreme. Lunar missions near the equator will see temperature variations from 125 K to 400 K. The antenna and station electronics must be able to operate during the lunar night and survive the lunar day. With these considerations in mind, LIBRA has calculated for mass and power allocations of a thermal system.

The space borne thermal system for LIBRA has been modeled after the JPL study on a Lunar Polar Volatiles Explorer (LPV). The LPV mission is of a similar design and battery mass to the resultant carrier proposed within this document. Using JPL's model as a rough estimate allows a baseline for thermal systems that adequately meet the LRA thermal threshold requirements. The mass of the system has been allocated as 25 kg. This mass is sized from the LPV battery mission, which has a similar battery mass and an ASRG lite to provide heat. With this baseline intact, the mission has the electrical and thermal requirements to meet and move beyond science threshold.

For nighttime operations, a Warm Electronics Box (WEB) will house the batteries and electronic controllers for the mission. This system will be well insulated, and contain resistive heaters spaced carefully within the WEB to maintain nighttime temperatures around 288K. This WEB and insulation will require an average power draw of 20 Watts for both State of the Art and Future design power modes. The temperature window required for the mission is based current technology limitations. The system will be maintained between the range of 270 K and 310 K. Future research should be applied to expand the electronic operations window. An increase in temperature operability will reduce power availability in the primary power loop.

Insulation and heater requirements are subject to change with mission profile alterations. From a power perspective, thermal control is the priority and will receive power while all other systems are idled during any battery depletion beyond 55%. As mission development and power profiles are altered, thermal requirements will be adjusted.

F.2.3.7 Propulsion and Attitude Control Systems

The overall propulsion system for this mission includes both solid and liquid propulsion engines. Table 1 below shows the overall propulsion sequence for the mission along with ΔV, Isp, and propellant masses for each maneuver. The first maneuver after launch will be a mid course correction (MCC) burn that will be provided by the orbiter once the spacecraft separates from the launch vehicle. From there the spacecraft will coast to the moon until the first STAR48V solid motor is fired to begin lunar orbit insertion. Any corrections needed during and after this burn will be provided by the lander’s propulsion system. After LOI, de-orbit initiation will take place where the orbiter is separated from the spacecraft and continues to operate in lunar orbit while the spacecraft will begin its decent to the lunar surface. The braking burn will be initiated shortly after DOI and is provided by a second STAR48V. The approach and divert burns are provided by the mono-propellant propulsion system which ultimately lead to a soft landing on the moon.

Table 8. Propulsion Firing Sequence

Maneuver	Purpose	Performed by	ΔV [m/s]	Isp [s]	Propellant mass used [kg]
1	Mid course correction	Orbiter	50	320	103.1
2	Lunar orbit insertion	STAR48V	1080	290	1973.35
3	Correction burn	ACS / Lander	30	220	38.9
4	De-orbit initiation	ACS / Lander	20	220	34.3
5	Braking	STAR48V	1699	290	1609.51
6	Approach	ACS / Lander	76	220	62.7
7	Divert	ACS / Lander	19	220	15.1

F.2.3.8 Solid rocket propulsion

Conservative ΔV budgets were calculated for the lunar orbit insertion and braking burns on this mission. The LOI burn requires a ΔV of 1080 m/s and the braking burn requires 1699 m/s. The LOI and braking burns are relatively large, one time burns for the mission so the best option was to use solid rocket motors for these maneuvers due to weight and simplicity. After evaluating several options for the required performance, a STAR 48V with a 2% offload and another STAR48V with a 20% offload were chosen as the solid rocket motors for the LOI and braking burns, respectively.

F.2.3.9 Liquid Rocket Propulsion

The final approach and landing as well as attitude control are handled by the liquid propulsion system. A pressure fed mono-propellant system utilizing hydrazine as the propellant was designed to accommodate this phase of the mission. A mono-prop system was chosen over a bi-prop for simplicity but more importantly due to the pulsing requirements of landing. The system consists of three Aerojet MR-80B main engines as well as the sixteen additional rocket engines for attitude control which are covered in more detail later. The system can generate enough thrust to operate with two MR-80Bs, but three are utilized to ensure stability of the spacecraft. A regulated pressure vessel with helium pressurant is used to ensure stable thrust levels for the spacecraft during landing. The pressure vessel is monolithic titanium constructed in lieu of a composite overwrapped design due to the small size of the tank. Two monolithic titanium propellant tanks were also custom designed per the system specifications. Two diaphragm propellant tanks were designed according to the propellant mass of the system and are also monolithic titanium. There could be potential weight savings with a composite overwrap design, but due to the relatively small size of the tanks it would require a custom design and an off the shelf tank could not be found for reference. Two tanks are utilized to optimize the spacecraft layout and weight distribution. See Figure 3 for a diagram of the entire liquid propulsion system. Additional information and calculations detailing the liquid propulsion system design can be found in appendix J.16.

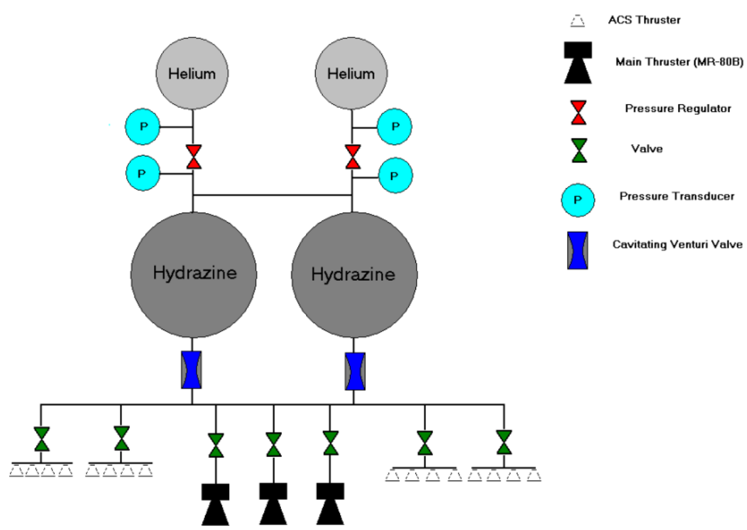


Figure 8. Liquid Propulsion System Diagram

F.2.3.10 Attitude Control System

The attitude control system consists of four sets of four engines to maintain control of the spacecraft. Each set of thrusters include one twenty pound Aerojet MR-120 thruster and three five pound MR-106L thrusters. The twenty pound thrusters will be pointing in the aft direction to allow for some thrust vector control during the firing of the solid rocket motors.

The ACS system provides 3-axis control of the spacecraft to ensure a soft landing and provide the necessary maneuverability to ensure the spacecraft is orientated properly upon landing. Orientation upon landing is extremely important due to the large diameter of the spacecraft after deployment of the science arrays. There must not be any interference between the science arrays and the terrain. ALHAT will allow for appropriate tolerances in the surrounding environment for protection of the lander and science equipment.

F.2.3.11 Orbiter

The orbiters will provide the communications between the landers and the earth. They will be parked in a 100km orbit and will be in range of the landers once every hour for 750s. Each orbiter will be equipped with two antennae so that it can communicate to both stations simultaneously.

F.2.4 Additional Mission Elements

F.2.4.1 Throttling Cavitating Venturi Valve

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.

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 Aerojet MR-80B:

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.

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.

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.

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 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.

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

Table 9. Pressure

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

Table 10. Materials

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 equations shown in the appendix.

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

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.

F.2.4.2 CoilABLE Booms

For the mission project Radio Astronomy on the Moon team Libra is using a lander system with CoilABLE booms to extend and support radio-telescope arrays consisting of kapton material imbedded with radio telescope nodes after lander touchdown. With this system team Libra hopes to provide a wide area of which to scan and observe the reaches of space from the far side of the Moon with the radio-telescope equipment away from the great deal of radio interference present on Earth hindering data collection.

The type of boom being used will be an ATK Canister deployed CoilABLE boom. This type of boom was chosen due to the method of its deployment. The lanyard method is required to rotate during deployment. This rotation would prove problematic when trying to unravel a radio-telescope sheet that needs to be mostly flat to operate properly. The canister deployed method does not rotate as it is deployed making it a more viable method to extend the radio-telescope sheets without damaging them.

The radio telescope sheets will be about twenty-five meters long by one meter wide by approximately 0.025 millimeters thick. Four CoilABLE booms will be used to deploy these sheets from each side of the lander. The booms will provide a small amount of support along the twenty-five meter length. A fifth twelve and a half meter boom will provide vertical support to the structure by extending vertically upward with cables attached to the ends of the four horizontally positioned booms on which the radio-telescope sheets are mounted. Each sheet will also have a rod perpendicular to the booms at regular intervals to provide further support to the radio-telescope sheet and keep the sheets flat. This method should provide enough support to the structure to withstand the Moon's gravity.

CoilABLE booms, however, were not designed for terrestrial missions therefore it is questionable how well exactly this system will work. Calculations indicate that the structure will withstand lunar gravity but CoilABLE booms are designed for zero-gravity situations in space and not under constant strain of gravitational forces. Unforeseen problems may occur due to the use of the booms on a terrestrial body in the presence of a gravitational force.

This non-mobile design was chosen over a rover due to the extreme sensitivity of the radio-telescope sheets and the greater ease of deployment. The radio-telescope sheets must be laid out flat in order to work properly. Debris and uneven land on the Moon's surface could cause the sheets to not function properly and give misleading data, because of this the rover design was abandoned in favor of the lander that would not have to place the radio sheets on the ground. It would consume too much energy and time to precisely locate a suitable position for the array and then clear potential debris from the deployment location. The lander can be set up and deployed much easier with the CoilABLE booms that would allow the arrays to almost completely ignore any issues with the terrain. This design will also consume less energy than a large rover clearing debris.

Using CoilABLE boom technology with the use of radio-telescope sheets is an effective and relatively easy way for scientists to study the reaches of space with radio telescopes away from the interference of radio waves on Earth. Due to the versatility of a lander it could be deployable at nearly any location on the far side of the moon giving scientists many possibilities for research.

F.2.4.3 Orbiter

The power network is based on the power conditioning and distribution unit provided by Thale Alenia Space and EADS. All power is received from this unit.

The solar panels provide onboard electrical energy during the solar exposure periods. This energy is saved in the batteries for the non-exposure phase and gives back when the light goes out. The power unit ensures the management of the power. The power unit provides electrical energy to the transmitter that sends data from the orbiter to the earth. The power unit feeds the receptors that communicate with the moon. The power unit also supplies energy to the general data processing unit, which manages all the onboard systems. The Solid State Data contains all the mission data and the transmitted data finds its energy in the power unit. The navigation control system contains an inertial measurement unit, which locates the orbiter in space. The altimeter also provides an altitude reference to the inertial central unit.

It was necessary to size the batteries that would supply the orbiter with energy. Several ranges of rechargeable batteries were compared in order to optimize the weight of onboard batteries. According to the specifications, the orbiter must be powered during the eclipses. For safety reasons, an accumulator was added in order to be sure the required voltage was achieved. It was necessary to compute the number of chains of accumulators in parallel in order to get the intensity and the power in the connections. According to the calculations, the total useful power is P = 194.855 W. The batteries have to supply all of the power during the eclipse. The aim is to size the battery for the most critical, or longest, eclipse.

The intensity of battery discharge and the amount of battery discharged during eclipse was calculated. All batteries have a Depth of Discharge (DOD) given in percentage. In geosynchronous orbit, this DOD is usually equal to 80% during around 15 years of cycles. It is assumed that the DOD of the batteries will remain at 80% as the orbiter rotates around the Moon. With this information, a minimum battery capacity was calculated.

With the factory accumulator capacity provided, the number of accumulator chains was calculated. The necessary battery mass and volume was then determined.

A VES 180 Lithium ion battery was chosen based on trade studies conducted with Nickel-Cadmium batteries and Nickel-Hydrogen batteries. The VES 180 battery was chosen based on weight, filled volume, and power characteristics.

Solar panels were chosen to help power the orbiter because it is an easy and simple method of acquiring power in space. Since the mission is around the Moon and in feasible proximity with the Sun, the use of solar cells is appropriate for the mission. In order to size the solar panels, an inventory was taken of electrical components in the orbiter. The sum of the power consumptions of the components gives the necessary power required from the solar arrays.

Some of the orbiter elements, like the receiver and the transmitter, are doubled in quantity. This redundancy was implemented in the aim of increasing the reliability of the system. It was not possible to double every component because of the orbiter dry mass limit. The total consumption of the orbiter is approximately 295W.

Mass and surface calculations were carried out with panel power/mass and power/surface ratios of 70 W/kg and 230 W/m^2, respectively. These ratios were chosen in order to optimize the size and the weight of the power system. Multiplied with the electric consumption, a mass of 4.21 kg and a surface of 1.28 m2 was calculated for the arrays. The mass of electrical wires was estimated to be 15% of the full power system mass, which is comprised by the battery, array, and wires. A wiring mass of 5.44 kg was calculated for the system.

A bipropellant system was necessary for the orbiter propulsion system. The MMH (Monomethylhyrazine)/N2O4 (Dinitrogen tetroxide) combination was chosen for the orbiter. The advantage of these propellants is that they are hypergolic, meaning the two chemicals ignite upon contact without a separate ignition source.

There are three main operations that the orbiter must perform with its engines:

- The first boost is the mid-course correction (MCC) to change the trajectory of the spacecraft .
- The second boost is for the lunar orbit insertion (LOI) to put the entire component in the circular selenocentric orbit.
- The last phase is to put the satellite in a spin (rotating around the axis of symmetry) to cancel the effects of transverse thrust created by the failure axisymmetric of nozzle.
- To finish the balance, propellant is needed in order to counter the drift of the orbiter during its life.

Initial calculations were made with an engine Isp of 320s. However, the best statistic found was an engine with an Isp = 312s. For our main engine, a R-4D Marquardt engine was chosen to accomplish the MCC burn. For the MCC burn, the mass of the structure is 6524 kg. The mass of propellant for the burn was calculated to be approximately 105.7 kg. After the MCC, a solid rocket motor is utilized to insert the orbiter into lunar orbit. After this solid rocket motor is jettisoned, a correction burn ($\Delta V = 30$ m/s) will be utilized to correct any trajectory error due to the solid rocket motors thrust vector misalignment. At the end of this burn, the mass is 4088.4 kg. For the LOI burn, with a $\Delta V = 30$ m/s, 39.9 kg of propellant is necessary.

For the control of attitude or correction of orbit operations during the life of the orbiter, twelve small engines will be used. These engines are TIROC bipropellant engines from the Kayser Company. The TIROC engines utilize the MMH/N2O4 propellant combination.

During the mission lifecycle, the orbiter will experience inclination disturbance from the gravitational pull of the Sun, Earth, and Moon. Due to this disturbance, correction burns must be implemented to maintain circular selenocentric orbit. According to calculations, the amount of propellant required for orbit correction burns throughout the mission lifecycle is approximately 9.75 kg. An approximate total propellant mass of 158.81 kg will be required for the entire mission lifecycle.

In the propellant calculations, a five percent margin was implemented to account for residual propellant. According to calculations, an approximate total of 98.88 kg of MMH and 59.93 kg of N2O4 will be needed to complete the mission. Tank sizing was looked at and according to the calculations, the necessary volume for the MMH and N2O4 tank are 0.11 m^3 and 0.04 m^3, respectively.

The absence of pumps in the propulsion system necessitates the use of gaseous Helium (GHe) to pressurize the system. A restrictive chamber pressure of 10 bars was used in sizing the Helium tank. A regulator will be used to correctly distribute the GHe during the mission. An array of fluid control equipment (valves, filters, injectors, etc.) will be used in the hydraulic system. Heritage use of this equipment has provided statistics of pressure loss and efficiency. According to calculations for loss and safety, the Helium tank must be pressurized to 27 bars.

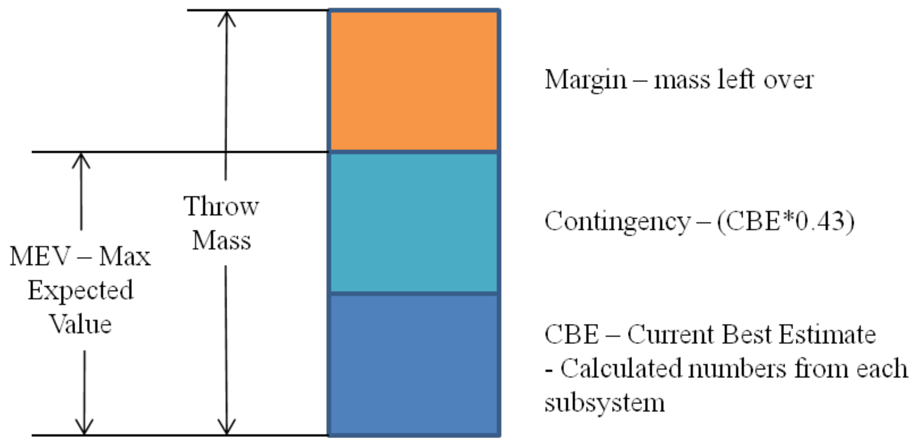
The orbiter will experience varying temperatures throughout the mission lifecycle. The most restrictive temperature requirement on the orbiter is that of the hydrazine propellant, which must be maintained between 7°C and 35°C. The orbiter must remain in this temperature range for proper functioning. An Optical Solar Reflector will be used to protect the orbiter. The surface mirrors on the reflector will reflect the solar flux and cool the orbiter. The orbiter will also be covered with Kapton and Mylar to protect all the surface and instruments. The electronic systems on board the orbiter create a dissipated flux of 100 W.

The orbiter is sized to support the launch. The propelled flight will be the most strenuous period for the orbiter. The spacecraft will be subjected to a wide range of dynamic excitation and vibration during launch. All the frequencies are defined by the following categories: low frequency, mid-frequency and high frequency. The low frequency vibration is the design driver for the orbiter structure. The high frequency vibration the spacecraft will experience is primarily due to the acoustic field noise, with a very small portion being mechanically transmitted through the spacecraft interface. The random vibration environment is the design driver for lightweight components and small structural supports. The noise can reach 130 dB. The highest acoustic level occurs for approximately 10 seconds during liftoff. This will be when the acoustic energy of the engine exhaust is being reflected by the launch pad. The other significant level occurs for approximately 20 seconds during the transonic portion of flight. This is due to aerodynamic shock waves and a high boundary level. Acoustic levels inside the payload fairing (PLF) are spatially averaged. These levels vary with different spacecraft due to acoustic absorption that varies with spacecraft size, shape, and surface material properties.

The orbiter structure is designed to withstand a maximum of 4.6 g acceleration due to the motor thrust. The orbiter structure is designed to withstand different stresses and thermal conditions, but it is also designed to be light to cut down on mass. For this reason, the orbiter has been designed with composite materials. The structure of the orbiter load structure and panels will consist of a honeycomb carbon core and carbon face-sheets. The orbiter structure has been designed to limit the deformation due to the temperature differential between the orbiter and outer space. The orbiter has also been designed with several sensors to navigate correctly.

F.2.5 Flight System Contingencies and Margins

Team LIBRA will abide by the NASA Jet Propulsion Laboratory standard for systems safety rules to provide the margin and contingency. These rules state a contingency of 30% used all across the mission. The following table graphically describes the relationships between margin, contingency and current best estimate and how to calculate them.



MEV = (CBE)*(1.43)
- Cost for Orbiter and Lander

Figure 9. Margin and Contingency Standard

F.2.6 Mission Operations

Once the Lander touches down on the surface of the moon, it will wait 24 hours for the lunar dust to settle. Then the booms deploy from their canisters by the use of frangible bolts. The first set of booms to deploy is the solar arrays. They deploy from opposing sides of the Lander simultaneously and parallel to the lunar surface so they can begin to gather energy immediately. Then the central boom and antenna deploy straight up off the top of the Lander. This also deploys the data array support cables attached to the central boom and the end of the data array booms. Next, all four data array booms deploy parallel to the lunar surface pulling the data arrays off the rolls positioned above the booms. The arrays rest on top of the flat surface of the deployed boom. The deployed booms are supported on their ends by the cable attached to the central boom and the end of each data array boom. With all seven booms deployed, the Lander begins taking data as required.

F.2.6.1 Ground Systems and Facilities

The Deep Space Network (DSN) will be utilized to communicate with Earth. All launch and ground operations will follow standard center procedures.

F.2.6.2 Telecommunications, Tracking, and Navigation

The telecommunications equipment for the mission will be subjected to the environmental conditions of outer space and the minute lunar atmosphere. The acceptable operations temperature range of the telecommunications equipment for the orbiters and the landers is between -130°C and 130°C. This window of acceptable operations temperatures is based on heritage values from past lunar missions. Each orbiter shall carry 4 wideband Ka-Band antennas. This number of antennas was chosen for mission redundancy and also for simultaneous communication between each orbiter and both landing sites. Each lander shall carry 2 wideband Ka-Band antennas. This number of antennas was chosen for mission redundancy and simultaneous communication with each orbiter. The particular antennas being utilized have not been flown in a spacecraft mission before.

The use of Ka-Band is a requirement necessitated by the AO. Ka-Band is defined as the radio frequency range of 26.5 GHz – 40 GHz. It is the intention of NASA to transition all deep-space missions after 2016 for the use of science data return. In efforts to comply with this transition, Ka-Band is being utilized for all forms of communication throughout the mission life cycle. In the proposed mission, the use of Ka-Band will entail transmission of mission commands and engineering data from Earth to the lunar surface and also the transmission of scientific and engineering data from the lunar surface to Earth.

Data transmissions from Earth to the lunar surface include mission commands and engineering data necessary to the completion of the mission. Data transmissions from the lunar surface to Earth include scientific and engineering data necessary to completion of the mission. The scientific data includes the detection of dark matter and coronal mass ejections from disturbances sensed in electron fields in the minute lunar atmosphere.

The scientific data gathered on the lunar surface requires no data encoding before transmission to the Earth over the Deep Space Network (DSN). The scientific data will require 10x compression prior to transmission to Earth. The data compression will provide for more efficient rate of data transmission over the DSN. Threshold science objectives can be achieved without data compression, but data compression allows a maximization of scientific data transmission with minimal resources.

The mission commands and engineering data from the Earth requires no data encoding before transmission to the lunar surface over the DSN. The mission commands and engineering data can be assumed to undergo a certain level of data compression prior to transmission to the lunar surface. The data compression will provide for more efficient rate of data transmission over the DSN.

Each lander on the lunar surface will transmit data continuously everyday throughout mission duration. Each lander will see both orbiters once per hour, for every hour, of each day. The landers will have 750 seconds of data transmission for each orbiter, per orbiter pass. Assuming nominal data collection and no losses, the total data transmitted per lander per day is approximately 407 MB. From this figure, an annual data transmission total of 297,110 MB can be nominally assumed.

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F.2.6.3 Description of Approach for Acquiring and Returning Critical Event Data

Increased telemetry will not be required on either of the orbiters during any part of the mission due to the antennas being omni-directional and the orbiters' close proximity to the Earth.

F.3 Development Approach

F.3.1 System Engineering Approach

A top-down systems engineering approach was used to develop the mission by analyzing the requirements from both the integrated product team class and the science objectives. The requirements in the class come from UAHuntsville, parent organization of team LIBRA and director of the class, who imposed drafts and briefings, the use of Discovery AO as a guide, deadlines and documentation. The science objectives on the other hand imposed requirements such as science instrumentation, mission duration and related mission requirements.

F.3.1.1 Decision Making Progress

The decision making process of the mission design was carried out based first and foremost on the science objectives. Using a system engineering approach, team LIBRA was aware that the science was the final product therefore placing its requirements as first priority. Given this, the science traceability matrix was used to help design our mission traceability matrix, which in case would stem down requirements to each subsystem, giving a starting point for research to contemplate different options for our decisions.

Once the team had an idea of the possible options for a certain decision, they were analyzed for feasibility given the constraints from class. These requirements included mostly the Discovery AO, cost and launch vehicle requirements.

Finally as fewer options were available, decision analysis, heuristics and engineering judgment were used to decide among them. It is important to note that communication parts a key role when making these decisions because all decisions entail very many factors and having many people with different ideas and experiences helps to make better decisions. It is important to note that some tools were used to aid in decision making, such as trade studies and communication tools, as well as outside help from expert advice from Dr. Matt Turner and Dr PJ Benfield.

F.3.1.2 Tools

Team LIBRA utilized a number of tools to make decisions as a team. As for any systems engineering approach, team LIBRA recognized that a key concept to make decisions was communications.

The tools used for communications within the internal subsystems of the team were Drop box, email and telephone. In the case of Drop box, this online software was utilized to share files with each individual team member and partners online and keep them updated. This worked really well because most people had access to the internet making communication simple. As a side benefit, Drop box works as a backup for the files, which makes the project more secure. Even in the case when files are deleted, those can be recovered from the deleted files area in the Drop box website. Email was another efficient tool for communication when decisions had to be made. The project manager would communicate assignments, deadlines and decisions to each individual in the group in order to accomplish them faster. Finally the telephone (which includes texting) was used as a form of communication that influenced decision making in the group because it enabled verbal communication and discussion between subsystems and leaders of the team.

Team LIBRA also used tools to communicate with the partners. The software Skype was used to have verbal communication with both ESTACA and The College of Charleston representatives in order to keep them informed of any changes in the design. This would help them make decisions that relate better with the most current design and iterate the current requirements and configurations of the project.

In addition, the tool used for communication with the high school InSPIRESS level 1 teams was the electronic mail. Team LIBRA managed requirements and deadlines along with any feedback or change in the design that led to configuration changes. This communication was vital for these teams since team LIBRA gave them the criteria for winning the competition, therefore basically giving them parameters from which to gauge their own decisions.

One important tool utilized for the decision making in the mass subsystem was a spreadsheet designed by one of the team members, Tyler Early. This spreadsheet had details of the mass breakdown using the rocket equation and going through the iterations of the trajectory elements. This way it had the ability to change the mass breakdown upon changes in requirements which was widely used for configurations management. This tool was shared in the previously discussed Drop box tool.

Moreover team LIBRA utilized one technical tool for the choice of engines, a spreadsheet supplied by Dr. Turner and Dr. Benfield that allowed team LIBRA to size the different solid rocket motors. This spreadsheet took in consideration mass calculations delta V's to recommend the most adequate engine choice. Given the amount of information this spreadsheet considered and the lack of real world experience, team LIBRA was inclined to considerate the engine choice in this fashion.

F.3.1.3 Interfaces

All elements considered in team LIBRA and its partners interface with most other elements in some sort of fashion. This is very important to note since a change in configuration of one element will change many others. Keeping this in mind, an N2 diagram refers to Appendix J for N2 diagram) was created to show the interfaces of all the elements of the mission.

Moreover, even though one element affects many, two behaviors were found. First, in many cases there were elements that were affected most from particular changes. In those cases the different partners would already know to interface more often with that particular element than any other. As particular examples, the propulsion element was highly affected by the mass numbers, and the high school InSPIRESS level 1 teams were highly affected by the criteria imposed to them by the lead systems engineer.

The second behavior found was that, in general terms, the chief engineer and project manager served as the most influential interfaces among all. In particular the chief engineer would interface with every subsystem to deal with technical developments while the project manager would usually deal with overarching roles, deadlines and responsibilities with every subsystem.

F.3.1.4 Configurations Management

Configuration management deals with changing requirements and configurations within the mission design to fulfill overall objectives and to overcome constraints. In the case of team LIBRA, these changes happened plenty and often. The mission design and configuration mainly saw changes in mass, power, lander configurations, science deployment, area for data collection, launch vehicle and engine choice which were dealt with different tools. Both the mass and power subsystems, whose values changes numerically, were placed in Drop box where they were updated regularly and where everyone was able to see them. The configurations of the structure of the lander and all that entailed to the science deployment and area of data collection was kept track on the whiteboards in the room as well as an agenda kept by the chief engineer to discuss during meetings. Finally the launch vehicle and engine choice was closely kept under observation by the chief engineer since those decisions most affected other subsystems and therefore the most urgent.

One particular area of interest for configurations management that team LIBRA paid plenty of attention to was the science instrumentation configuration because it was really important not to jeopardize the science objectives and because everyday communications with the principal investigator were not guaranteed. As mentioned before, a track of any changes of this subject was kept under close observation on the whiteboards of the team room.

F.3.2 Mission Assurance Approach

Team LIBRA ascribes to the NASA Jet Propulsion Laboratory standard for systems safety rules for mission assurance in order to assure that the design requirements are met throughout all the phases of the project.

F.3.2.1 Product Assurance

The product of this mission is the data collection. In the case of team LIBRA's project, the instrument that will collect the data is called the Dark Ages Lunar Interferometer holds a TRL two therefore it doesn't assure the data collection but it must be used because it is the mission enabling technology. Team LIBRA will therefore use redundancy in the system to assure the collection of data the best possible.

There are a few things that team LIBRA has done to provide redundancy. First of all the threshold mission requires one landing site while the baseline has two from two different launch vehicles, so if one fails you can still meet threshold. Moreover, each launch vehicle has an orbiter so one of them can fail and you still have the other one. In addition, we have good amount of memory available and plenty of solar panels where if a few batteries died we could still work during daylight. Finally each landing site has 4 panels to have a total of eight. This is great because one can fail as long as there is more than 1 non-parallel on the same site we can perform science.

F.3.2.2 Reliability

Team LIBRA understands the importance of reliability and even though redundancy greatly improves the overall safety of the system, reliability is still an issue, mostly where the TRL's aren't high. In the case of team LIBRA's project there are two technologies that are important in the project.

The mission enabling technology called DALI only has a TRL level two therefore the future testing on this technology will be key in future assessments of reliability. Even though reliability of this technology is not proven, team LIBRA believes that it will confirm to fulfill the expectations in order to be a part of a reliable overall system.

Moreover team LIBRA will utilize ALHAT, with a current TRL four. This technology will assist in landing by scanning the terrain and assessing the best place to land. In this case ALHAT's purpose is technology demonstration and in reality the lander could land without ALHAT and still meet the mission requirements. This fact therefore takes ALHAT out of consideration and will not be considered further as a reliability problem.

Finally the only other element that can play a role in reliability will be the booms that will support the DALI and solar arrays for the landers. Even though these booms have been used before, team LIBRA will add a hinge and have them rotate in order to be parallel to the lunar surface. This new aspect might require further testing as it should be considered as a reliability risk.

More generally team LIBRA believes that analysis and tests of the overall system should be performed in the future stages of the mission but that only the few elements described above should play an important role given that all other technology (such as engines and antennae) have been used in space before.

F.3.3 Instrument to Spacecraft Interfaces

Our main science instrument is the DALI technology which will be utilized in the form of thin rolls of kapton to be deployed by the lander. This instrument will only start its interface once landed and after the allotted time for regolith decay has passed.

In physical terms, there will be four rolls of kapton of size 25 x 1 m deployed using vertical booms. Each "panel" as team LIBRA calls them, will be deployed through its own boom and a main supporting beam will be deployed right in the middle of them. These panels will then fold towards the ground through a hinge at each of the panels origin, all supported by the top of the supporting beam. The weight of each panel should balance the forces to produce a balanced lander as a final product.

In terms of data interface, each of the four rolls of kapton will collect data and send it to our antenna located also at the top of the supporting beam to be later sent to Earth via the orbiter. Specifically these panels will produce continuous voltage that will be interfaced with the orbiter to produce the Ka-band and communicate with the orbiter.

F.3.4 Design Maturity and Heritage

The mission elements that do not have high maturity and heritage are the science instrumentation and the landing assistance hardware and software. The first one called the DALI (for Dark Ages Lunar Interferometer) has been tested on Earth and is being used in this mission to receive the data for the science objectives. This technology has a technology readiness level (TRL) of two, since there is proof-of-concept but hasn't been tested in the lab environment. Team LIBRA fully understands the significance of this level DALI is the mission enabling technology therefore it must be utilized in order to perform any type of science on the moon.

On the other hand, ALHAT, the Autonomous Landing Hazard Avoidance Tester has a TLR of four but it is carried in this mission for technology demonstration purposes. In other words, team LIBRA will test this technology in the mission but it is actually not needed to meet the mission requirements therefore it will not affect the outcome of the mission. If by any chance this technology doesn't work team LIBRA will simply land using conventional procedures and the mission shouldn't be at risk.

F.3.5Essential Trade Studies

In this project team LIBRA used decision analysis during pre-phase A to decide between engines and burns to be used. Later on during phase A we found that these would change as part of our configurations management process but the initial decision analysis helped plenty in guiding towards the best alternative solutions.

In the future team LIBRA believes that the testing and development of the booms in this project will change the requirements and shall require decision analysis due to the lack of testing with hinges on booms and the overall structure of the lander.

Furthermore the mission design has stability in terms of data and power but team LIBRA foresees the need for more decision analysis in that area once the design matures and requirements and configurations keep changing due to the constant changes and constraints of the project.

F.3.6Management Approach

Team LIBRA made decisions based on the area of expertise of each individual member along with the support of the chief engineer and the expert advice of the course instructors. Team LIBRA's division of work was very detailed into subsystems where every subsystem would report to the chief engineer about technical issues. On tough decisions both would discuss the problem, come up with possible solutions and make a decision based on several resources. Later those decisions were communicated to ensure that the configurations of the project would all fall into place, most predominantly the mass and power requirements.

Moreover team LIBRA discussed discrepancies or disagreements through open discussions. These type of situations occurred when the decision affected many elements of the mission. In these cases all the alternatives were discussed as a group but ultimately the chief engineer would be responsible for the decision.

Finally, in the future stages of this mission, test anomalies should be addressed very carefully. In the case of the mission enabling technology, DALI, if the tests prove that it doesn't collect the data needed to meet the science requirements, mission termination should be considered an option; by taking in consideration the maturity of the project, alternative science instrumentation and other factors. In the case that test anomalies are established at a non-critical location, configurations management along with decision analysis should be performed to come up with the best solution.

F.4 New Technologies/Advanced Developments

For this mission, team LIBRA will utilize the mission enabling technology called the Dark Ages Lunar Interferometer (DALI) which holds a TRL of three. This technology is the mission enabling technology of the mission therefore it must be utilized in order to perform the science. In order to assure that the technology performs under requirements further testing and analysis will be performed to try to mature the TRL to six. In case this science instrument does not mature, team LIBRA will add and combine other higher level subsystems later in the mission design to increase the TRL for the system.

On the other hand this mission will utilize ALHAT, with a current TRL four. This technology will assist in landing by scanning the terrain and assessing the best place to land. In this case ALHAT's purpose is technology demonstration and in reality the lander could land without ALHAT and still meet the mission requirements. This fact therefore takes ALHAT out of consideration for these purposes.

F.5 Assembly, Integration, Test, and Verification

F.5.1 Integration and Test Plan illustration, discussion, and time-phased flow

Spacecraft integration and testing was contracted to Boeing. This decision was based upon the outstanding heritage work in the spacecraft industry provided by this company. Spacecraft integration and testing will take place at Marshall Space Flight Center in Huntsville, Alabama. It is recommended that trade studies be conducted for later design phases to ensure that testing procedures will qualify the spacecraft for the proposed mission.

F.5.2 Verification approach

The verification approach for later phases will come from the NASA System Safety Handbook. The verification approach will be performed to ensure all requirements from previous design phases have been satisfied before progressing to further design phases. Verification will occur during reviews between design phases.

F.6 Schedule

Following team LIBRA completed a schedule to lay out the system engineering phases of this project. First, team LIBRA considered pre-phase A and phase A as both semesters for the IPT class where phase A finishes in May of 2011. Secondly phase B was allotted plenty of time since costs will increase as the design matures, therefore the more time this design is considered during the early phases, the better. In any case, phase B should be allowed plenty of time to work on the decision analysis to be made on booms and power constraints.

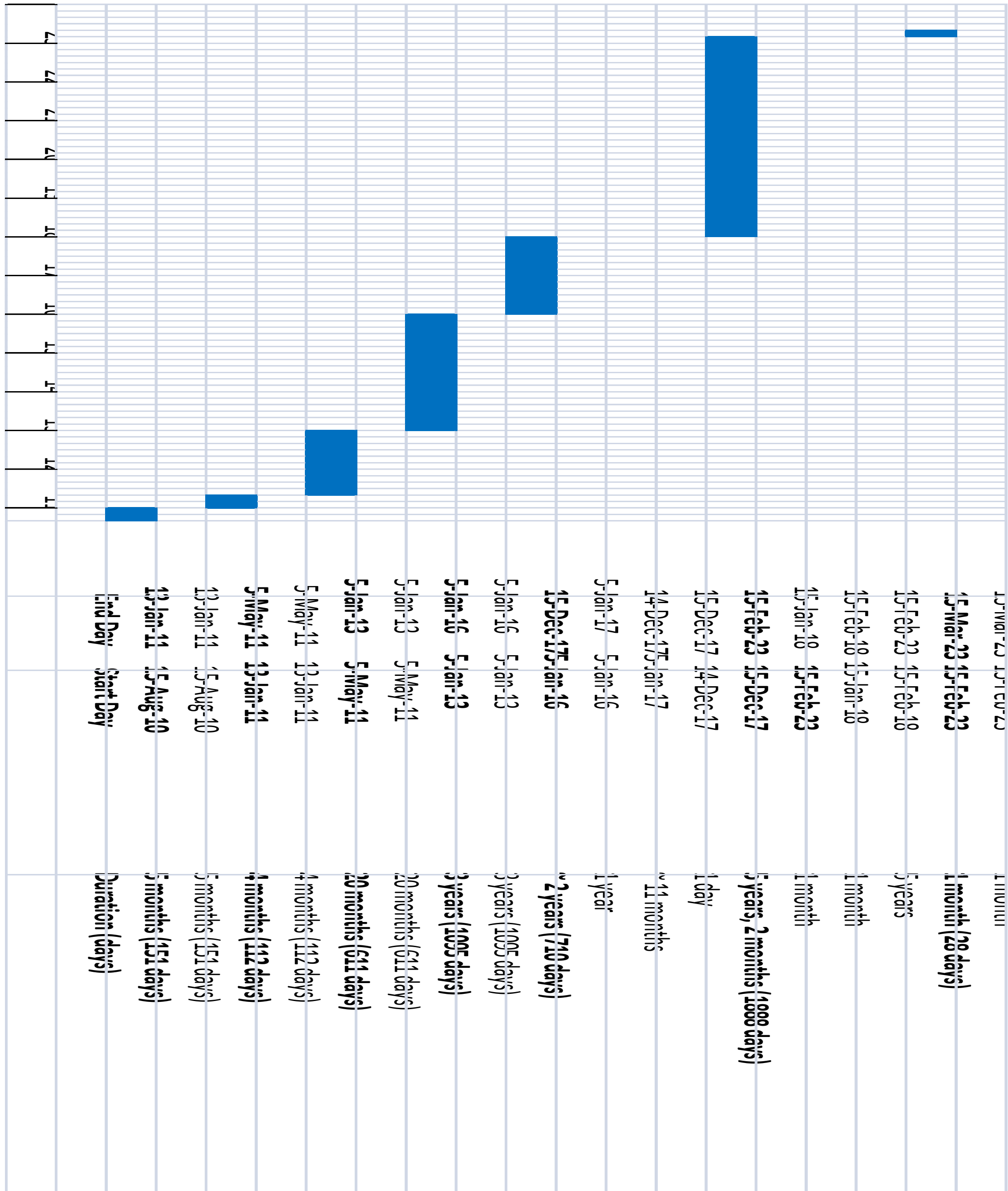
Phase C and D duration were calculated based on engineering judgment. Phase C is considered as final design and fabrication and team LIBRA doesn't have any accurate data on how long it would take to fabricate the DALI technology. Moreover, the lander is custom ordered to the mission therefore it will also take time to be finished. In any case there are a few things which construction time data is more at hand such as time to build the solid rocket motors and the engines, three years being estimated for their construction. In addition team LIBRA assigned two years for the system assembly and integration from engineering judgment by analyzing similar mission and a day to launch both vehicles.

Once launch is taken place team LIBRA knows that the time of transit will be three days due to the trajectory of the mission.

What's more, after transit team LIBRA assigned one month’s time to allow regolith and other debirs to settle on lunar surface to avoid having issues with the solar panels and science instruments. After that the science observation given by the mission threshold was allotted a time frame of five years which is the established duration of the mission.

Finally team LIBRA allocated one month for the closeout by comparison with other missions.

Following is a table of times along with a Gantt chart.


End Day Start Day Duration (days)
13-Jan-11 15-Aug-10 5 months (151 days)
13-Jan-11 15-Aug-10 5 months (151 days)
5-May-11 13-Jan-11 4 months (112 days)
5-May-11 13-Jan-11 4 months (112 days)
5-Jan-13 5-May-11 20 months (611 days)
5-Jan-13 5-May-11 20 months (611 days)
5-Jan-16 5-Jan-13 3 years (1095 days)
5-Jan-16 5-Jan-13 3 years (1095 days)
15-Dec-175-Jan-16 ~2 years (710 days)
5-Jan-17 5-Jan-16 1 year
14-Dec-175-Jan-17 ~11 months
15-Dec-17 14-Dec-17 1 day
15-Feb-23 15-Dec-17 5 years, 2 months (1888 days)
15-Jan-18 15-Feb-23 1 month
15-Feb-18 15-Jan-18 1 month
15-Feb-23 15-Feb-18 5 years
15-Mar-23 15-Feb-23 1 month (28 days)
15-Mar-23 15-Feb-23 1 month

G. Management

G.1 Management Organization

Team LIBRA made decisions for this project mostly through order of authority. The main concept is that each subsystem and element of the project (which includes all partners) had specific requirements based on overall mission requirements and therefore the issues would arrive mostly from the subsystems and taken upwards through management to make decisions. These decision making situations would often involve a change in some configurations therefore the changes needed to be kept on record depending on the type of change, being on Drop box or the whiteboards. The decisions would be made mostly depending on expertise and engineering judgment where the chief engineer, project manager, systems engineer and the principal investigator (for the science requirements) would be mostly responsible for those decisions. An organization chart follows (Figure 7) to describe this management approach.

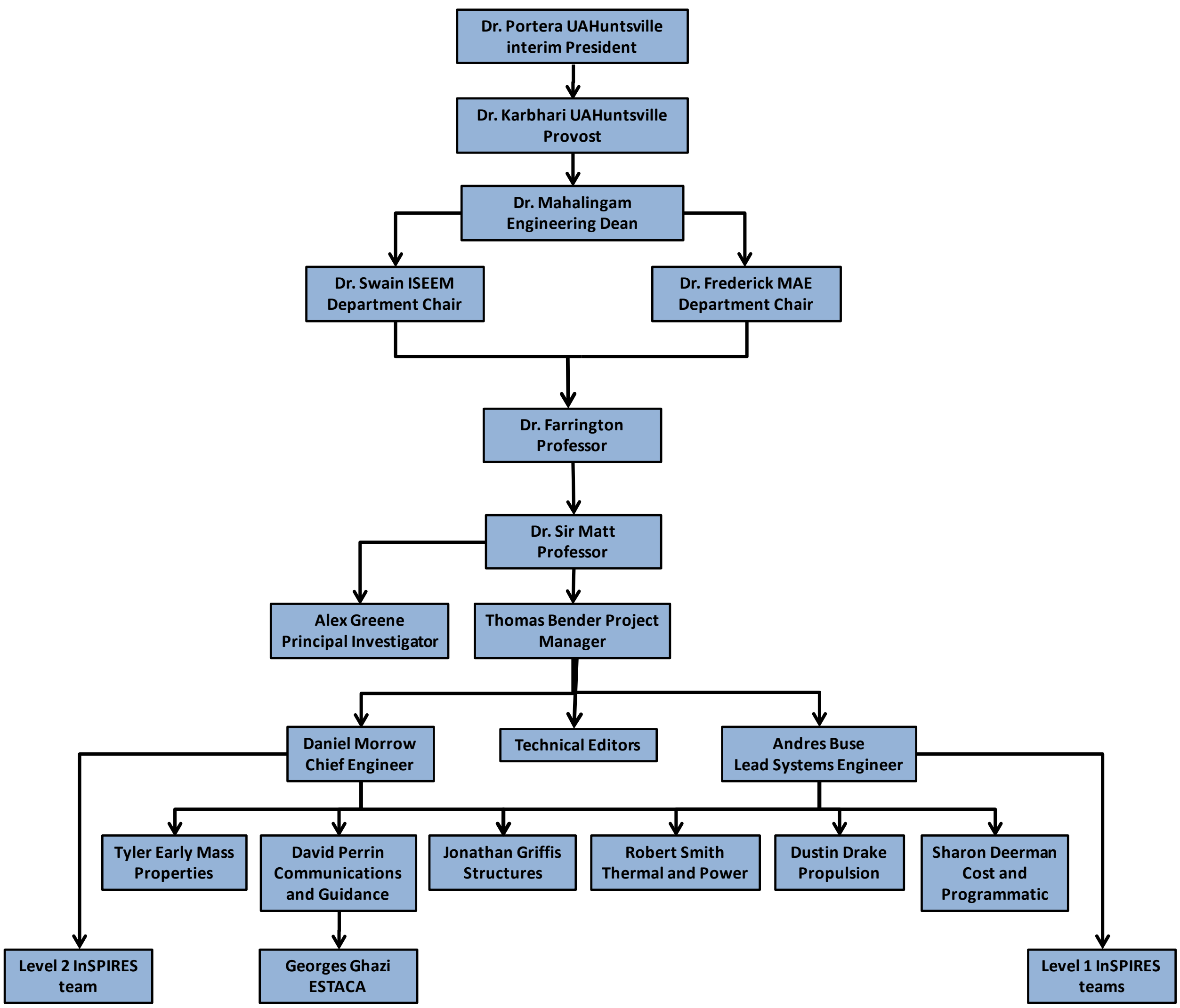


Figure 10. Organization Chart

Moreover team LIBRA had some teaming arrangements with its partners. These were carried out through the compliance of a memorandum of understanding between team LIBRA and its partners mainly to establish the commonality of deigning a mission given the science requirements. Also a team charter document was prepared to ensure that all participants of team LIBRA were in accordance with the same objective by having clear and defined information on each person in the group.

G.2 Roles and Responsibilities

G.2.1 Project Manager – Thomas Bender

The project manager will be responsible for the overall performance of the project. He will be in charge of meeting team deliverables and deadlines, making ultimate managerial decisions, improving team integration and increasing the efficiency of the team's production as a whole. In the case of Thomas Bender, following are the qualifications and experience he brings to team LIBRA.

- Led a team of student engineers to design an airplane in a Design/Build/Fly (DBF) competition held by the American Institute for Aeronautics and Astronautics (AIAA)
- Vice-president of the American Institute for Aeronautics and Astronautics student chapter at the University of Alabama in Huntsville. As part of his duties he wrote a research paper related to combustion instabilities and presented its results at a AAIA student conference
- Led a small group to design, build and program a Lego mind storm as part of a class project where small robots raced each other in an obstacle course

G.2.2 Chief Engineer – Daniel Morrow

The chief engineer will be responsible for the overall technical requirements of the mission. He will be in charge of the ConOps and mission design and will overview all the subsystems and try to help them in making important decisions. He will also be in charge (along with LSE) of keeping track of technical configurations management as it is a vital part of the technical development of the project. In the case of Daniel Morrow, following are the qualifications and experience he brings to team LIBRA.

- Currently a mechanical design engineer for Rotorcraft Systems Engineering and Simulations Center
- Built and flown a personal aircraft as a personal project
- Led a group of student engineers to design a portable wind tunnel for pre-engineering classes in high school
- Active remote control aircraft and model rocket enthusiast
- Lead with leadership since early on as a boy scout
- Trained in structural analysis and machining

G.2.3 Lead Systems Engineer – Andres Buse

The lead systems engineer will be in charge of interface and configurations management as well as aiding all subsystem using the systems approach to problem solving. The lead systems engineer also shall be in charge of the system engineering requirements in the deliverables. In the case of Andres Buse, following are the qualification and experience he brings to team LIBRA.

- Vice-president of the Institute of Industrial Engineers student chapter at the University of Alabama in Huntsville. As part of his duties he helped organize meetings and events for the student in the ISE department such as plant tours, talks and social meetings.
- Led a group of student engineers to write a paper and present a briefing about the situation of NASA after the elimination of the Constellation program and the path that NASA should take thereon forwards.
- His natural qualifications as a logic systems approach thinker are proof of his success as a student in technical classes as an engineer.
- He was co-captain of the UAH tennis team his senior season where he learned on the burden and responsibility of leadership.

G.2.4 Principal Investigator – Alex Greene

The principal investigator will be in charge of developing the science design and requirements for the mission. He will be in charge of the science instrumentation, deliverables and developmental concept of the scientific research. In the case of Alex Green, following are his qualification and experience.

- He was the Co-Investigator for the NIRO mission (came in second during the IPT competition in 2010) which gave him experience working with others and interfacing with engineers.

- He has successfully gathered knowledge during his four years in college.

- He experienced a private tour of the Very Large Array and gained a basic understanding of how radio telescopes operate.

- He has been working as an undergraduate research assistant for the past two years, giving him the experience with professional level deliverables.

- He has led numerous physics and astronomy group projects during his college years which gave him the experience necessary to be an effective leader.

G.2.5 Primary Institutions

Even though there are a several institutions as partners in this project, there are two main institutions.

The University of Alabama in Huntsville is the overarching institution in charge of the project. This institution has a highly respected and ABET accredited engineering program which greatly qualifies them for the technical requirements of the project. Moreover, the university has plenty of experience working as the overarching institution of project proposals, dating back a few decades. In summary, under the leadership of the well experienced and qualified instructors Dr. Matt Turner and Dr. P.J. Benfield the institution is more than capable of sponsoring this proposal.

In addition, the College of Charleston is the other main institution in charge of the design of the science requirements and implementation of the mission. The College of Charleston is well qualified as it has a highly respected college of science and has plenty of experience as it has been working with the University of Alabama in Huntsville in developing mission designs for years. The College of Charleston has the qualifications and experience to sponsor the science portion of this mission.

G.3 Risk Management

While there are many risks for the mission at hand, a full risk analysis is outside the scope of this proposal.

G.3.1 Mission Risk Analysis

Team LIBRA performed a risk analysis based on the likelihood and impact of any risk. Both of those variables were given a numerical value given from the risk assessment table provided below. Soon after, Team LIBRA utilized a risk matrix taken from the NASA Systems Engineering Handbook (how should I reference this?) and analyzed the level of importance of each of them. Once the level of importance was obtained, the moderate and high risks were evaluated to assess the possibility for mitigation strategies.

Table 11. Risk Assessment Table

	Likelihood	**Impact**
1	Near certain to occur	Catastrophic
2	Highly likely to occur	Critical
3	Likely to occur	Moderate
4	Not likely to occur, improbable	Marginal
5	Impossible to occur	Negligible

G.3.2 Risk Matrix

The risk matrix used to classify the importance level of a risk is below. Given the value for likelihood and impact, each risk is assign a specific risk level where the green color means low, yellow moderate and red high risk level.

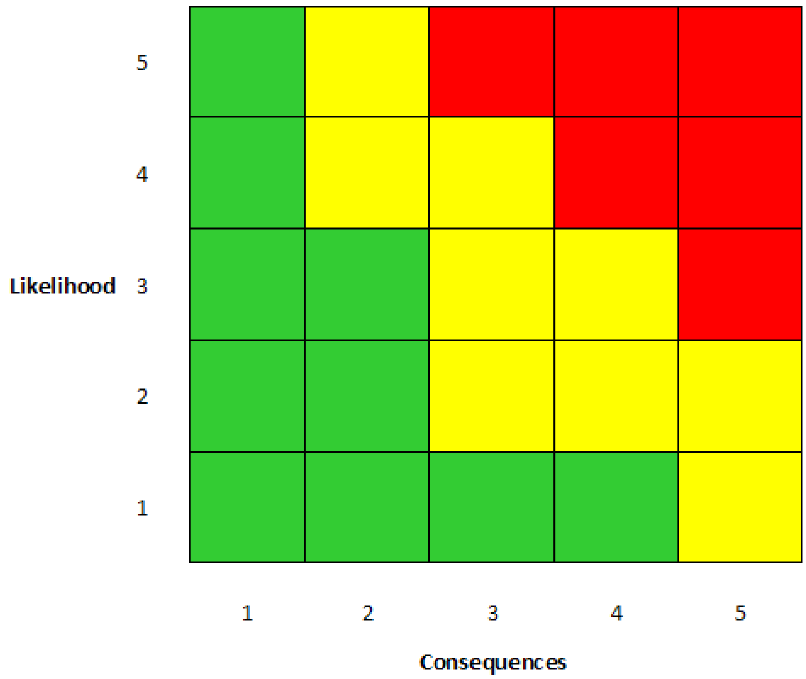


Figure 11. Risk Matrix

G.3.3 Primary Risks

The primary risks evaluated in the project are summarized in the following table and later explained in detail.

Table 12. Primary Risks

Primary Risk	Description	Likelihood	Impact	Importance
Attitude Control System Fail	The ACS's control the landing	1	5	Medium
Solid Rocket Motors Fail	The SRM's are the main engines	1	5	Medium
Coiled Boom Failure	The Coiled Booms deploy the solar and kapton panels	3	4	Medium
Kapton Tearing	The kapton panels are the DALI technology that captures the science	2	3	Medium
Solar Panel Fail	Solar panels are our main energy draw	2	5	Medium
Major Electronic Fail	All systems work electronically	1	5	Medium
Antenna Damage	Delivers the product to the orbiter	1	5	Medium
Orbiter Fail	Delivers the product to Earth	1	4	Medium
Hard drive Fail	Stores the data before sending	2	4	Medium

As it is seen on the table, none of the primary risks evaluated by team LIBRA are high, which was great news when the risks are evaluated. On the other hand these risks were not taken lightly and an appropriate assessment was necessary.

First of all it is important to note that there are five risks seen in this system with likelihood of one and a high impact. Those elements are the attitude control system, the solid rocket motors, the electronic system, the antenna and the orbiter; all which have extremely high maturity because have been used and tested for other missions before. Because of this and even though the level of impact is five, team LIBRA will not have a mitigation strategy or fall-back plan since these risks are natural for this type of mission.

Moreover one of the most important risks is the coiled boom failure. Team LIBRA is utilizing booms on the lunar surface, a new concept because they have been used for other mission only in open space. Also, most of the booms will be deployed sideways using support cables coming from a vertical center beam, a new concept to the booms. Finally, each landing site will have six total booms, four for the DALI technology and two for the solar panels. Once deployed the balance of the lander depends on the equal balance of these booms therefore a failure of one boom might affect that balance and steer lander to fall on its side. What is more, even though all these things can happen, the impact of any of these effects will not be as great mostly because data can be extracted using less than four panels per landing site, and only one landing site is needed.

Team LIBRA considered all these factors thoroughly. To start, the lander will have spikes to secure it on the ground and bring balance. What's more, these arrays are very hard to deploy. There really isn’t any other less risky way of deploying these, may it be a rover or some kind of inflation device. Because of this and the array being a requirement, this risk comes naturally to the mission and we have done the best we can to provide redundancy anyway.
The next risk to keep track of is the possibility of the kapton panels tearing when they are being deployed. The process of deployment will be the following. Once the solar panels are deployed, the central vertical beam will be deployed with taut cables connected to each of the four science booms. Then the four booms are deployed simultaneously having the cables as a supporting system as they extend out. There is a risk of tearing the kapton fabric while deploying them therefore two things will be used as countermeasure. First, team LIBRA will provide a support structure and an anti-torque motor for each panel to prevent it from falling and second, the speed at which each panel is rolled out will be calculated to match exactly the speed of the deployment to prevent tears.

The following risk is the solar panel fail and since the risk for the booms has already been addressed, the only risk here is something that obstructs the solar panels. To prevent that from happening team LIBRA will include a support structure on the solar panels and also will not deploy them until the regolith has settled to give them with a more favorable environment to work with.

Finally there is the risk of hard drive fail. In the past hard drives have failed due to weather conditions, especially at cold temperatures. On the other hand hard drives have also proven to stand these cold temperatures in some other missions, therefore the likelihood will have a level two. The impact is critical because if team LIBRA can’t store all the data then most of it will get lost. In order to countermeasure this team LIBRA will increase redundancy by doubling the number of hard drives in each lander.

G.3.4 Allocation of Resources

The control, allocation and release of resources will be handled closely by the PM and will be mostly arranged as the design matures and resources are needed to adjust based on requirement changes. Decisions then will be made as a group utilizing decision analysis tools where the PM will make the final decision in case the entire group can't find agreement.

Currently team LIBRA uses a contingency and margin of 30% and 2% across the board for mass, power and cost. These values are not specified for each subsystem but rather kept as a general extra requirement since changes and allocations of resources in the future will most likely not be spread out evenly through the elements of the system.

Moreover this approach is closely related to the areas that team LIBRA believes will be decision points down the road. Those areas will be the ability of the booms to function properly given the new set of conditions team LIBRA is requiring them to have, as well as the development of the DALI technology and any further unforeseen resource that will be needed in the future. In addition to that, mass numbers should always be kept under observation since those are most likely to change if a major change happens in the requirements that call the need for the allocation of resources.

G.3.5 Descoping

Team LIBRA will use a schedule approach to analyze the status of the mission and the need to descope it. There will be milestone meetings held at the end of every phase, emphasizing mostly on the end of phase A and B since design changes might be too costly if they are made later on.

The strategy of those meetings will be to assess id the mission is still able to meet all the mission and science requirements given the present resources. In the case a need for descoping exists, team LIBRA will use decision analysis to analyze the future options mostly based on the science requirements in an attempt to meet the most of the science threshold mission as possible. These decisions will be made as a group where the PM and PI will make the decision if the group doesn't find an agreement.

More specifically team LIBRA already has ways to descope the mission. The following ideas will be proposed at those meetings at the following order. First, the use of two launch vehicles should be reduced to only one vehicle because team LIBRA has chosen to make them only unequal in location and transit time while everything else is equal. Also, science threshold can be met with only one landing site therefore erasing one launch vehicle is a possibility. Second, the amount of panels could be reduced to three, reducing the chances of failure and still being able to meet threshold. Finally team LIBRA could reduce the size of the panels even more in a final attempt to descope the mission.

G.4 Cooperative Arrangements

G.5 Management and Scheduling Plans

Team LIBRA wasn't able to provide the appropriate details in management and schedule due to several factors, mostly lack of information and experience.

In terms of schedule team LIBRA was unable to provide details because of several reasons. First, no one on team LIBRA has participated in a full project and seen the development of all the phases and therefore phases B through E were very unfamiliar. Team LIBRA basically attempted at the schedule based on comparison for those phases along with engineering judgment. Moreover, some elements of the mission have never been produced or developed therefore there were no references for them. Finally the mission schedule included phases pre-A and A spent during the class while the future phases spent as a continuing project. The speed and development of that following environment was unknown to team LIBRA.

In terms of management team LIBRA was unable to provide with more details mainly in the risk section. Due to the TRL of the science instrument and the performance of the integration of all the subsystems, team LIBRA was unable to obtain more detail than the provided in the risk analysis. This risk management also affected the lack of detail in the schedule because not much detail is known about how a risk could affect the schedule.

H. Cost and Cost Estimating Methodology

H.1 Cost Model

Team Libra's cost analysis for the RAM (Radio Astronomy Mission) was calculated using the Hamaker Cost Model (Hamaker, 2006), designed by Joseph W. Hamaker who was a senior cost analyst for SAIC (Science Applications International Corporation). SAIC solves mission-critical problems with innovative applications of expertise and technology. The project model was created in Excel with functions setup in many of the cells which evaluated input by the user. Heuristics, experience-based techniques, were used within the model in lieu of exhaustive searches for historical data including manpower requirements and labor rates.

Originally this semester, the plan for cost analysis was to use NAFCOM (NASA/Air Force and Cost Model) which is an automated parametric cost-estimation tool that uses historical space data to predict the cost of a new space program. NAFCOM has a data-base type structure where by the user selects certain elements similar to elements of a previous mission and builds a mission via a template. Due to the time constraints to learn this impressive tool, the plan was changed from using NAFCOM to using the Hamaker Cost Model.

H.2 Model Inputs and Outputs

A cost model was created for each individual element (an orbiter and a lander) and these models were summed to get a total cost for the mission. Estimated mass and power consumption values were the major inputs for the cost models. The Threshold for this mission was a single rocket containing an orbiter and a lander with science on board.

On this mission redundancy was an option chosen by the team. The Baseline for this mission was two identical rockets intended to perform exactly the same mission, but with different landing sites. This was decided as a backup plan should a single rocket fail, the possibility of a second rocket would survive the mission. Rather than doubling the cost of a single mission to generate cost for the duplicate mission, the power and mass were doubled within a single cost model to calculate for duplicate elements. Models were summed to get a total cost for two duplicate missions. Individual costs for propellant and DSN (Deep Space Network) were not identified, as these parameters were factored into the cost model via the functions within the model.

The inputs for the orbiter were projected to be: mass was approximately 56.76 kilograms (132 kilograms x 0.43) and power was approximately 110 Watts (for low earth orbit equivalent). Orbiter mass included the orbiter, four antennas, hard drives, and a propulsion system. The inputs for the lander were projected to be: mass of 692.51 kilograms (1610.48 kilograms x 0.43) and used 110 Watts of power. The value of 0.43 is a factor for generating a value form the total cost minus the margin. Reference section F.2.5 for contingencies and margin explanations. The Lander mass included 3 MR-80B engines, two antennas, hard drives, and science arrays (included are solar panels, batteries, kapton sheets embedded with copper, and booms). Table 13 illustrates the projected cost of the mission for a single rocket, orbiter and lander. The total 2006 dollar value was multiplied by a factor of 1.15435 to get the 2010 dollar value for the mission. Figure 14 and 15 are examples of the Hamaker Cost Model used to estimate costs for a single orbiter.

Table 13. Cost of Mission for Threshold (single elements)

Orbiter (1 quantity)	$230,740,000
Lander (1 quantity)	$1,020,560,000
Total Cost of Mission	$1,251,300,000

Table 14. Example of Single Orbiter Costs Using Hamaker Cost Model

Orbiter Projected Cost			
Variables	**Inputs**	**Units/Descriptions**	**Comments**
Spacecraft Bus + Instruments Total Dry Mass	56.76	KG	Dry mass of proposed system
Spacecraft Total Power Generation Capacity	110	W LEO equivalent flux	Power consumption of proposed system
Design Life in Months	60	Months	Estimated life of mission
Number of Science Organizations	1	Count (Enter zero for projects with no science or science organizations involvement)	Science organization exists
Apogee Class	4	LEO=1, HEO/GEO=2, beyond GEO=3, Planetary=4	Mission is in Planetary category
Maximum Data 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 max.	Based on data transfer estimates
Test Requirements Class	2	Less than average testing=1, Average=2, More than average=3, Extensive=4, Very extensive=5	Average testing selected
Requirements Stability Class	3	Very low volatility=1, Low=2, Average=3, High=4, Very high volatility=5	Average stability selected
Funding Stability Class	2	Stable funding=1, Some instability=2, Significant instability=3	Stability dependent on Mission approval
Team Experience Class [Derived from Price Model; used with permission from Price Systems LLP]	4	Extensive experience=1, Better than average=2, Averaged (mixed experience)=3, Unfamiliar=4 [Ref: Price Model]	Unfamiliar with Price Model
Formulation Study Class	2	Formulation Study (1=Major, 2=Nominal, 3=Minor)	Nominal selected
New Design Percent	70%	Simple mod=30%, Extensive mod=70% (average), New=100%	The science arrays of this mission are not historical

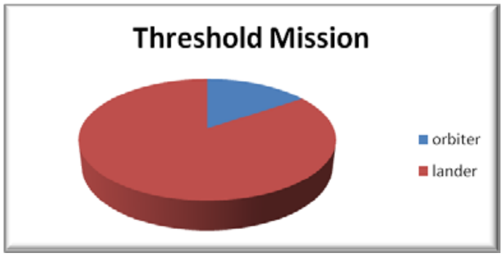
Table 15 Example of Single Lander Costs Using Hamaker Cost Model

Orbiter Projected Cost (Continued)			
Variables	**inputs**	**units/descriptions**	**comments**
ATP Date Expressed as Years Since 1960	51	Years elapsed since 1960	Proposed launch date 2017
Technology Readiness Level (TRL) Penalty Factor	7	Refer to NASA TRL scale (TRL 6 is nominal)	All elements are historical
Platform Factor [Derived from Price Model; used with permission from Price Systems LLP]	2.2	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 Mission
Calculated Size of the Government Project Office (Project Office Only--Excludes Government Functional Line/Laboratory Labor)	47.6	Civil Service annual Full Time Equivalents (FTE's)	Man hours to monitor mission
Final Estimate of the Size of the Government Project Office and other Oversight (excludes government non-oversight labor which is included in subtotal above)	47.6	Civil Service Full Time Equivalents (FTE's)	Man hours to monitor mission
Total (2004$)	**$199.9M**		
Total (2010$)	**$230.74M**		

As stated previously, the parameters for the duplicate mission were input as twice the mass and twice the power in a single cost model. Table 16 illustrates the cost of the mission for the Baseline mission (duplicate rockets, orbiters and landers). See appendices for additional examples using the Hamaker Cost Model. Why not simply multiply the single cost from Table 8, the Threshold cost, to calculate the cost for the Baseline mission? Each input in the cost model generates costs for the mission. For instance when a portion of the mission has been tested and has passed the qualifications, there would be no need to run additional tests for the second mission that is intended to perform the same tasks. Similarly, the amount of personnel to run one mission, could possibly be relied upon to run two simultaneous missions, rather than depending on a completely different group of full time equivalents. The lander is the major consumption of the budget compared to the orbiter as shown in Figure 12.

Table 16. Cost of Mission for Baseline (duplicate elements)

Orbiter (2 quantity)	$305,080,000
Lander (2 quantity)	$1,366,140,000
Total Cost of Mission	$1,671,220,000



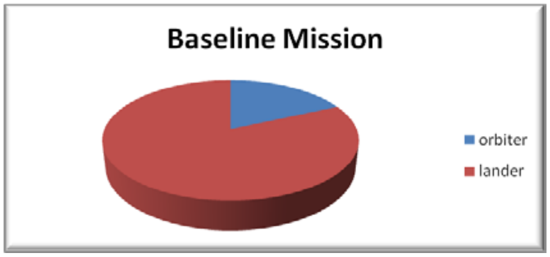


Figure 12. Lander Costs Consumes Budget Compared to the Orbiter Costs

The RAM Mission had a budget of $800 Million with the Atlas V551 rocket incurring no cost to the budget. Currently Team Libra is over budget for the RAM mission. As shown in Table 8, the estimated costs for the Threshold Mission are over budget by just over $400 million. Much of the design was calculated, yet assumed……mass, power, propulsion requirements, and size constraints. Other aspects of the model were estimated: time frame (months) for design life; percentage of new design; manpower required to manage the mission; etc. The science for this mission has yet to be designed and tested, so this part of the mission was not based on historical data. Expectations are high for the science mission, utilizing the DALI (Dark Ages Lunar Interferometer), to be a success.

Additional inputs of the cost model were taken conservatively. The "database median" values were used when conceivable to do so. The mission was considered an unmanned planetary mission. The maximum data rate requirements relative to state-of-the-art for the APT date, was projected to be 50% (as shown in Table 14). The various class requirements listed in the cost model were deemed as average values.

One aspect of the cost model increasing the budget was the TRL (Technology Readiness Level) Penalty Factor. Because the science of this mission was not yet been designed nor tested, the cost estimated was higher than a known, proven element. For the lander the TRL value was set as a "5" because the science is a new portion of the lander element. The cost increased because there would be greater risk to use an untested specimen in the mission. If one aspect of the mission has a low TRL value, the whole mission is said to have the value. The orbiter TRL was set at "9" because of historical testing has been proven for that element.

H.3 Cost Resource Allocation

Cost is a major component that will either make or break a project. In the consumer world, the product must sell and be profitable in order for the project to be a success. In similar terms for space investigation, the amount of data collected should be of significant importance for an expensive space mission to be worth the effort. Experiments performed on this mission shall be cost effective and operate for at least sixty months.

Cost can be reduced by material selection (if a possibility), by duration of project construction (cost of man hours to build) and manufacturing location. Best cost guesses are given by utilizing historical data for comparison. For a US spacecraft, it may not be practical to have inexpensive overseas labor costs for component assembly. It would be practical to manufacture/purchase static components from friendly countries while keeping cost lower than local manufacturing costs. Shipping costs can be astronomical depending on size and weight of large bulky components. The manufacturing of components for this mission will be best administered in close proximity to the launch position if possible.

The resources will be allocated in an effort to extract the most science possible by giving priority to data transfer, which would give more time and mass per landing site. Due to the nature of the cost model, the actual cost values for data transfer are unknown. The reserves from this mission shall be used to correct any anomalies that may occur as the mission progresses.

I. Education and Public Outreach (EPO) and Student Collaboration

I.1 Education Public Outreach

Competition Status

Team LIBRA 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. Tam LIBRA will submit an E/PO plan with my Concept Study Report if the proposal is selected.

I.2 Student Collaboration

Team LIBRA, in accordance with the student collaboration program, worked with two InSPIRES level one high school teams as a competition to include the best proposal into the mission.

The first team is formed with students from Sparkman high school located in Madison, Alabama. As part of an engineering initiative, Sparkman and UAH partnered through an engineering class as part of the high school curriculum where students form groups and develop their engineering skills in projects working along UAH students in the IPT class. Specifically, one of those teams is called team DARKCIDE, led by their project manager Alex Wakefield and did a proposal to be a part of team LIBRA's mission as secondary science. Their project is trying to find out more about the heat flow and insulating properties of the regolith in the far side of the moon. They propose to do that by using a mole previously engineered by DRL to drill the surface and take thermal data at different depths.

The second team is formed with students from Bob Jones high school also located in Madison, Alabama. UAH and Bob Jones also partnered to foster engineering interest in high school students but created an extracurricular activity were students would also form groups and work with UAH students in the same fashion as Sparkman teams. In this case, the team working along team LIBRA and proposing to be part of team LIBRA's mission as secondary science is called SOLAR and is led by their project manager Colin Burleson. This team mission is interested in learning about sun spots, sun flares and coronal mass ejections (CME) in hopes of learning useful information about Earth's closest star. They propose to do this by capturing images of the sun and analyzing its visible and infrared spectra.

Both of these teams, as mentioned before, competed for the opportunity to perform secondary science in team LIBRA's mission and both of their proposals were submitted Friday April 22nd. Team Libra chose DARKCIDE as the winning team of the competition and head LIBRA's secondary science mission based on scientific recollection, mass, cost, power draw, and structural interface with team Libra's mission.

This decision was based on the different factors of the mission. First of all, the scientific investigations have different objectives, SOLAR's objective is to watch the sun while DARKCIDE's was to analyze regolith's insulating properties. Team LIBRA believes that the advantage gained from watching the sun from the moon compared to just watching from Earth is not as great as regolith properties because this information could be very useful in near future missions to the moon.

Moreover, all mission requirements in terms of mass, data and power favor DARKCIDE. In the case of mass, SOLAR asks for a total 11.836 kg while DARKCIE only 6.6 kg. For power SOLAR will require between 8.5 and 18.5 W while DARKCIDE will only require around 11W. Finally the data constraints for the SOLAR mission were not available but the DARKCIDE mission will only require about 0.1 Mb for the entire mission because this mission's data is thermal readings as compared to images, which are usually much smaller.

In addition, the structural interface of both teams was appealing to team LIBRA's structure. SOLAR required to be on the side of the lander upon landing and would stay immobile throughout the entire mission. Team DARKCIDE will use a Nanokhod, an independent system part of their payload, to transport their moles 20-50 meters away from the lander.

Finally, an extra factor considered by LIBRA was redundancy. Team DARKCIDE chose to include two moles per landing site, each independent from each other. Team LIBRA deemed this as very important in terms of product assurance for the secondary mission.

Appendices

I.3 Tables of Proposed Participants

Table 17. Proposal Participants Roles and Budgets

Organization Name	Organization Role	Organization Budget
The University of Alabama in Huntsville	Design Leadership	
College of Charleston	Science Leadership	
Sparkman High School	Secondary Science Leadership	

I.4 Letters of Commitment

The following page contains the Letter of Commitment.

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 James Alex Greene 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

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

UAHuntsville
THE UNIVERSITY OF ALABAMA IN HUNTSVILLE

April 21, 2011

Thomas Bender
Project Manager
Team Libra, IPT Team B
The University of Alabama in Huntsville
Mechanical and Aerospace Engineering Dept.
N274 Technology Hall
Huntsville, AL 35899

Dear Mr. Bender,

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

I.5 Resumes

The following pages contain the resumes of each of Team LIBRA's members.

Alexandra Aruwajoye

(256)-851-2488; (256)-520-5394
ana0005@uah.edu; aaruwajoye@yahoo.com

11012 Rockcliff Drive
Huntsville, Alabama, 35810

CITIZENSHIP

U.S.

TECHNICAL SKILLS

Operating Systems: Windows XP and Windows Vista

Applications: Microsoft Word, PowerPoint, Excel, and AutoCAD

EDUCATION

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

Degree: Bachelor of Arts in English

Minor: Technical Writing, GPA: 2.9/4.0 in major, May 2013

Multi-Disciplinary Integrated Project Team — **Huntsville, AL**

Position: Technical Editor- Edit Documents

The University of Alabama — **Tuscaloosa, AL**
Degree: Bachelor of Arts in English
Minor: Creative Writing, GPA: 3.2/4.0 in Major, May 2010
Degree: Bachelor of Metallurgical and Materials Engineering, 3.1/4.0 in major, May 2009

WORK EXPERIENCE

Jun 2005 – Present **Affordable Janitorial Supply** — **Huntsville, AL**

Secretary

Typed, sorted, and distributed invoices.

Delivered packages.

Sorted incoming and outgoing mail.

HONORS AND AWARDS

The University Of Alabama-Dean's List (Aug 2009- May 2010)

-Academic Engineering Achievement Award

Thomas P. Bender

(256)-557-0228; cell
tb7792003@yahoo.com

685 Providence Main St. NW
Huntsville, AL 35806

3409 Russell St.
Cedar Bluff, AL 35959

CITIZENSHIP U.S.

TECHNICAL SKILLS General Fluid System Simulation Program (GFSSP), MATLAB, Mathcad, C++ coding, Solid Edge modeling, Machining

EDUCATION

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

Bachelor of Science in Engineering with a concentration in Aerospace

GPA: 2.95/4.00 (3.01/4.00 in major), Expected graduation Summer 2011

Senior engineering project

Aug 2010 – Present

Project Manager of Integrated Product Team

- Lead a team of future engineers to design a spacecraft mission to perform radio astronomy on the moon.
- Communicate with partners to accomplish mission goals.

Jacksonville State University **Jacksonville, AL**

GPA: 3.2/4.0

Gadsden State Community College **Gadsden, AL**

GPA: 3.8/4.0

Coosa Christian High School **Gadsden, AL**

GPA: 3.83/4.00

WORK EXPERIENCE

Apr 2009 – Present Propulsion Research Center **Huntsville, AL**

Student Specialist IV

- Assist graduate students in attaining research project goals.
- Designed a thermocouple rake for rocket engine combustor temperature measurements.
- Supported high pressure spray facility assembly for injector characterization testing.
- GFSSP Water Hammer Simulation.
- PRC facility security upgrade.

PUBLICATIONS AIAA Student Paper “Combustion Instability Analysis using a Linear Thermocouple Rake”

AFFILIATIONS AIAA (Vice President of UAH Student Chapter)

Andres Buse

(256)-990-8897
andres_buse@yahoo.com

704 Apt #L
John Wright Dr.
Huntsville, AL, 35805

Av. Del Sur 279 dep. #401
Chacarilla del Estanque, Santiago de Surco
Lima 33, Lima, Peru

CITIZENSHIP Peru

TECHNICAL SKILLS Arena, Solid Edge V20; Microsoft Office; Windows Vista; C+

EDUCATION

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

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

GPA: 3.839/4.0 (3.94/4.0 in major), Expected graduation Dec 2011

Senior engineering project

Aug 2010 – Present

Systems Engineer of Integrated Product Team

- Part of a team of future engineers to design spacecraft mission architecture to perform radio astronomy on the moon.
- Integrate mission elements that include all possible factors and stakeholders being affected by the mission.

WORK EXPERIENCE

Jan 2010 – Present **Student Success Center** **Huntsville, AL**

Position: Tutor

- Tutor calculus, statistics, physics, chemistry, intro to C++, engineering economy and operations research.
- Helping with strategies and problem solving tools.

Jun 2010 – Aug 2010 **Camp Winaukee** **Moultonboro, NH**

Position: Camp Counselor

- Bunk counselor for boys age 15 at a 7-week overnight, all-sports camp in NH.
- Taught tennis skills and strategy to boys ages 13-15.

HONORS AND AWARDS

Engineering's Dean List 2009, 2010

Men's Tennis East Division GSC All-Academic team 2008, 2009, 2010

All GSC Men's Tennis Team 2009

Academic Excellence Award 2009, 2010

AFFILIATIONS

Phi Kappa Phi (2010), Tau Beta Pi (2011), Alpha Pi Mu (2010)

VP of the UAH chapter of the Institution of Industrial Engineers 2010

UAH tennis player Jan 2007 - May 2010

Sharon Deerman

(256)-430-0275; cell: (256)-497-2979
sad0002@uah.edu; sdeerman@knology.net

225 Rosecliff Drive
Harvest, AL 35749

CITIZENSHIP

U.S.

TECHNICAL SKILLS

Operating System: Windows XP

CAD Software: Solid Edge; AutoCad; MicroStation; Adobe Illustrator; Excel; Outlook; Powerpoint; Word; cfDesign; Patran/Nastran; Mathcad; MATLAB; Agile; SAP;

EDUCATION

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

Bachelor of Science in Engineering with a discipline in Mechanical

GPA: 3.5/4.0 in major, Dec 2011

Calhoun Community College — **Decatur, AL**

Degree: Associates of Science in Mathematics; GPA: 3.85/4.0 in major, May 2006

Wallace State Community College — **Hanceville, AL**

Degree: Drafting Technology Diploma; GPA: 3.7/4.0 in major, Jun 1981

WORK EXPERIENCE

Jan 1999 – Present **Emerson Network Power/Avocent** **Huntsville, AL**

Mechanical Engineer

- Produce 3D design models and detailed documentation for sheet metal enclosures, plastic faceplates and related accessories; interface with fabrication shops and track prototype deliverables.
- Utilize design practices of Design for Manufacturability (DFM).
- Analyze product design using fluid flow/thermal analysis software (cfDesign).

Jan 1986 – Nov 1998 **Intergraph** **Huntsville, AL**

Development Engineer

- Produced detailed component/assembly drawings and related documentation.
- Designed protective packaging for branded hardware and software products; analyzed existing packaging solutions for cost reduction.

Jun 1981 – Jan 1986 **Assoc. Steel Detailers, Inc.** **Homewood, AL**

Drafter

- Produced miscellaneous steel and ornamental handrail drawings for fabrication.

HONORS AND AWARDS

Calhoun Community College – graduated Magna Cum Laude

Dustin W. Drake

(256)-652-9865
dwd0001@uah.edu

10011 Conrad Drive
Huntsville, AL 35803

CITIZENSHIP U.S.

TECHNICAL SKILLS Microsoft Office, Mathcad, MATLAB, Solid Edge, MSC Nastran and Patran

EDUCATION

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

Bachelor of Science in Engineering

GPA: 3.0/4.0, Expected graduation May 2011

- Concentration in Mechanical Engineering

Senior engineering project
Jan 2011 - Present
Propulsion Lead

- Performed engine selection, propellant selection and general calculations and design for the overall propulsion system.

WORK EXPERIENCE

Jun 2002 – Jan 2007 **Lee Builders, Inc.** **Huntsville, AL**

Project Engineer

- General contractor in the construction industry with focus on open bid commercial work.
- Assisted project managers on scheduling and project materials deliveries.
- Provided subcontractors and suppliers with necessary informational packages to submit price quotes on competitive bid work.
- Subcontractor / supplier database maintenance.

Tyler Earley

(405)-831-2294
twe0001@uah.edu

Current Address
1211 Grandeview Blvd, Apt# 2524
Huntsville, AL 35824

Permanent Address
3705 Burlington Dr.
Norman, OK 73072

CITIZENSHIP U.S.

TECHNICAL SKILLS Solid Edge, NX, Nastran/Pastran, MATLAB, MathCAD

EDUCATION

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

Bachelor of Science in Engineering

GPA: 3.20/4.0 (3.52/4.0 in major), May 2011

WORK EXPERIENCE

Aug 2010 – Present Integrated Product Team **Huntsville, AL**

- Part of a team of future engineers to design a spacecraft mission to perform radio astronomy on the moon.

PROFILE

Team captain for the UAH Men's Soccer team which has helped me acquire skills in team work, communication and leadership. It has also taught me to be a hard worker when completing any task I set out to accomplish.

HONORS AND AWARDS

Academic Excellence Award (4.0 Night)

Bronze Scholar Award

Susann E. Gardner

(256)-348-8528
seg0005@uah.edu

125 Jefferson Patton St.
Harvest, AL 35749

CITIZENSHIP U.S.

TECHNICAL SKILLS

Operating System: Windows XP, Windows 7, Mac OS X

Microsoft: Office, Publisher, Outlook, PowerPoint, Word

EDUCATION

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

Bachelor of Arts, Psychology

Minor: Technical Writing, Web Cognate, Computer Languages and Systems

GPA: 3.4/4.0 (3.0/4.0 in major), Expected graduation 5/2014

Jan 2011-- Present

Technical Editor, NASA Integrated Product Team Program

- Worked on an editing team preparing proposal, editing résumés and other documents for the Integrated Product Team.

WORK EXPERIENCE

Apr 2001 – Mar 2004 **Exotic Harvest Nursery** **Harvest, AL**

Greenhouse Manager

Developed employee manual.

Set up intranet.

Implemented bar code system.

May 1984 – May 1987 **Ecology Cooperatives** **Philadelphia, PA**

Produce Manager

Developed management team.

Assistant accountant.

HONORS AND AWARDS

Community Service Award Harvest Meadows Community

The University Of Alabama in Huntsville-Dean's List (Jan 2010- Dec 2010)

Samantha Geltz

1665 Mulberry Street Apt D
Charleston, SC 29485
(843)-513-2240

EDUCATION

Completing three undergraduate degrees at the College of Charleston

- Astrophysics
- Classics
- History

WORK EXPERIENCE

2009 - Present **College of Charleston** **Charleston, SC**
Research Assistant, Physics Department

- Researched the winds and accretion belt of a black hole with Dr. Chartas
- Analyzed data using computers

2007-2010 **College of Charleston** **Charleston, SC**
Resident Assistant, College Lodge Residence Hall

- Individually managed 44 people, as a team 200+ people
- First line of security

2008-2009 **College of Charleston** **Charleston, SC**
Campus Activities Board, Campus

- Fun Fridays director
- Campus wide event planning for 10,000+ students

GHAZI Georges
Nationnalité : Française
Adresse : 7 Avenue Eugénie,
Batiment B1 - 92210
Saint-Cloud
@ : ghazi.georges@gmail.com
Tel : +33 (0)6 44 17 39 85

Elève ingénieur à l'ESTACA
Objectif : Ingénieur Recherche & Développement

FORMATION

Depuis 2008 : 2^e,3^e,4^e année à **l'Ecole Supérieur des Techniques Aéronautiques et en Construction Automobile - (ESTACA) - Dominante ESPACE.**

Janvier 2010 : Semestre à l'étranger dans le cadre d'un programme d'échange avec l'**Ecole Polytechnique de Montréal en Génie Aérospatial.**

2006 - 2008 : 1^{ere},2^e année en **Classe Préparatoire aux Grandes Ecoles** (PCSI-PSI) au **Lycée Technologique de Raspail**, Paris.

Depuis 2006 : Obtention du Baccalauréat Scientifique, spécialité Physique-Chimie, mention assez bien, au **lycée Jean Mermoz**, Dakar (Sénégal).

PROJETS

- Projet IPT : Conception d'un orbiteur pour une mission sur la Lune, en collaboration avec une université Américaine et la NASA.
- Conception et modélisation d'un étage de lanceur avec une équipe du CNES.
- Cours d'intiation à Matlab/Simulink donné aux 2e années de l'ESTACA dans le cadre d'un monitorat.
- Réalisation d'un Arbre de défaillance et étude AMDEC sur la perte de poussée du moteur Vulcain 2.
- Création d'un modèle d'asservissement pour un moteur à courant continu (correcteur, observateur et régulateur).
- Analyse en méthodes numériques des paramètres de vol d'une fusée sous Matlab.
- CFAO/IAO : Modélisation, conception, analyse et fabrication d'un VTT sous catia.

EXPERIENCES PROFESSIONNELLES

Eté 2010 : **Sénégal Handling Services (S.H.S) :** Assistant technicien dans l'assistance au sol des compagnies aériennes (Sénégal) - Gestion des plans de vol et supervision des vols.

Mai 2010 : **Participation à la Bromoscopie du Concorde** avec l'association Olympus 593.

Eté 2009 : **Agence Nationale de l'Aviation Civile du Sénégal (A.N.A.C.S)** Assistant inspecteur de la Navigabilité - Inspections d'aéronefs. Etudes de dossiers d'immatriculation et renouvellement d'un certificat de navigabilité

COMPETENCES ET CONNAISSANCES LINGUISTIQUES

Langues :

Anglais : Bon niveau (Séjours au Canada et Etats-Unis).
Espagnol : Niveau BAC.
Wolof (Dialect Sénégalais) : Bon niveau.
Libanais : Notion.

Notion de pilotage : 1h de pilotage sur Pipper.

Logiciels :

Solidworks · Catia · Matlab · Office 2007 · Patran
Inkscape · Blender

Langages Informatiques : C, Matlab et Maple.
Outils Informatiques : Simulink, Simscape et Simmechanics.

CENTRES D'INTERETS

Association : Membre de l'association Estaca Space Odyssey (E.S.O) : conception de fusées expérimentales.
- Trésorier en 2011, Vice-Secrétaire en 2010 et Webmaster depuis 2008.
- Chef de projet sur les fusées Marianne et Dragoon.
- Responsable mécanique sur la fusée Bi.
- Connaissances en usinage, réalisation de tubes en carbone et fibre de verre.

Loisirs : Pratique de la guitare, conception de site web, photographie et sport.

James Alexander Greene

Telephone: (513)-309-2866
Email: jagreene@edisto.cofc.edu

EDUCATION

College of Charleston — **Charleston. SC**

B.S. Astrophysics and B.S. Physics

GPA: 3.4

TECHNICAL SKILLS

Proficient in Mathmatica, IDL, Microsoft Office, and some HTML

WORK EXPERIENCE

2010 – Present **Undergraduate Research Assistant**
GRB Pulse Fitting
Dr. Jon Hakkila Department of Physics and Astronomy Chair

Used IDL to fit GRB pulses using statistical models, generated and interpreted correlations adding to existing catalogue.

The Predictability of Pulse Evolution Models: Towards Explaining Complex GRB Properties
Dr. Jon Hakkila Department of Physics and Astronomy Chair

Applied Monte Carlo analysis to several BATSE GRBS to explore whether GRB's exhibit simple, correlated observable characteristics indicative of hard-to-soft evolution.

2010 – Present **Department of Physics and Astronomy Teaching Assistant**

Chris True

Helped professors run lab, answered questions on material discussed, and prepared/broke down materials for lab.

PUBLICATIONS

The Predictability of Pulse Evolution Models: Towards Explaining Complex GRB Properties, James A. Greene and Jon Hakkila. 2011

The Progenitor-Independent Nature of Gamma-Ray Burst Pulses, Jon Hakkila and Robert D. Preece, 2011 (Contributed)

Jonathan Griffis

161 Clover Ridge Drive
Madison, AL 35758
(256)-864-9532; (205)-305-3115
Jdg0004@uah.edu

CITIZENSHIP U.S.

TECHNICAL SKILLS MathCAD, MATLAB, Microsoft Office Suite, C++, ProE, Solid Edge v.20, NX3.0

EDUCATION

The University of Alabama in Huntsville — **Huntsville, AL**
Bachelor of Science in Engineering with a concentration in Aerospace Engineering
GPA: 3.13/4.0 (Cumulative), Aug 2011

The University of Alabama in Birmingham — **Birmingham, AL**
Bachelor of Science in Criminal Justice
GPA: 3.19 (Cumulative), Graduated Dec 2003

Bevill State Community College in Sumiton — **Sumiton, AL**
Honors Certification in EMT-Basic
GPA: 3.88 (Cumulative), Certified Jun 2003

WORK EXPERIENCE

Aug 2010 – Present — **Integrated Product Team** — **Huntsville, AL**

- Part of a team of future engineers to design a spacecraft mission to perform radio astronomy on the moon.

Jun 2010 – Aug 2010 — **Northrup Grumman** — **Huntsville, AL**
Senior Intern
Worked on the Enhanced Command and Control, Battlefield Management, and Communications (EC2BMC) program

Jul 2008 – Dec 2008 — **Aerospace Testing Alliance** — **Tullahoma, TN**
Coop 1 and 2
Participated in a support role for ATA Engineers and Clients
Performed the role of Engineering Technician 1

Nov 2004 – Jun 2006 — **Birmingham Police Department** — **Birmingham, AL**
Sworn Police Officer

- Completed 22 weeks of Police Academy Training, Certified Draeger operator, Certified Radar Operator
- Certified Field Sobriety, American Red Cross First Aid, Adult CPR/AED Operator

CLEARANCE SECRET, Northrup Grumman Information Systems (most recent)

JOHNATHAN HUNTER HEGLER
46b State St.
Charleston, SC 29401
(803)-960-7346 ● Jhhegler@gmail.com

EDUCATION

College of Charleston — **Charleston, SC**
Bachelor of Science in Computer Science,
May 2012

RELEVANT COURSES COMPLETED

Computer Programming I (Dr. Isaac Green), Game Programming (Dr. Bill Manaris), Computer Programming II (Walter M. Pharr, Jr.), Discrete Structures I (Dr. Dinesh G. Servate), Introduction to Computer Organization and Assembly Language (Dr. Isaac Green) Data Structures and Algorithms (Walter M. Pharr, Jr.)

CURRENTLY ENROLLED

Software Architecture and Design (Dr. James F. Bowring), Operating Systems (Dr. Bill Manaris),

TECHNICAL SKILLS

Operating Systems: UNIX, MAC, Windows (XP/ Vista/ 7), Linux
Programming Languages: Java , Python , C , MIPS Assembly
Other Software: Adobe Illustrator, Adobe Photoshop, Final Cut Pro, Microsoft Office (Word, Excel, PowerPoint, etc.)

EXPERIENCE

2010 – Present — **College of Charleston** — **Charleston, SC**
Undergraduate Research Assistant, (CIRDLES)
Cyber Infrastructure Research and Development for the earth sciences
Pursued Innovation and understanding in unit testing, and testing in general, for object oriented Programming languages, mostly focusing on Java.

2008 – 2010 — **College of Charleston** — **Charleston, SC**
Lab Attendant
Center for the Documentary
Attendant to Post Production Lab**,** Photo Editing (Adobe Photoshop)**,** Project Editing and Archiving**,** Media Transfer and Duplication**,** Photography and Videography, Assist Students with their Homework and Projects

2006 – 2008 — **BI-LO** — **Chapin, SC**
Bookkeeper
BI-LO
Supervised Cashiers and Baggers, Attended the Customer Service Desk, Collection and counting of cash, checks, as well as debit and credit receipts

INTERESTS

Computer Assisted Illustration (Adobe Illustrator), Photo Editing (Adobe Photoshop), Photography, Guitar

Daniel Craig Morrow

(205)-527-8868
dcm0004@uah.edu

1500 Sparkman Dr NW
APT 43D
Huntsville, Al 35186

1245 Sullivan Rd
Sumiton, AL 35148

CITIZENSHIP U.S.

TECHNICAL SKILLS Solid Edge, Pro/Engineer Wildfire, Nastran, Patran, MATLAB, Mathcad, Geomagic, Faro Arm, C Programming, Composite & Metal Fabrication, IPT

EDUCATION

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

- Bachelor of Science in Engineering with in Aerospace
- GPA: 3.025/4.0 (3.13/4.0 in major)
- Expected graduation: Jul 2011

WORK EXPERIENCE

Jul 2010 – Present **RSESC** **Huntsville, Al**

(Rotorcraft Systems Engineering and Simulation Center)

Student Intern

Government level IPT experience

Composite/metal fabrication

Computer Aided Design

Mechanical design consultation

Aug 2008 – Nov 2009 **The University of Alabama in Huntsville** **Huntsville, AL**

Student Intern Level II

Solid Edge instructor

Student work grader

HONORS AND AWARDS Academic Excellence Scholarship

AFFILIATIONS American Institute for Aeronautics and Astronautics

Pierre PASCAL

Appartement 234 – Les Estudines Victor-Hugo
171, boulevard Victor Hugo
92110 Clichy
FRANCE
Phone : +33 (0)6 48 36 45 02
E-mail : pierre.pascal@estaca.eu

EDUCATION

- *Now* : Student in 4th year at **ESTACA** engineering school (Ecole Supérieure des Techniques Aéronautiques et de Construction Automobile) in **Levallois-Perret (FRANCE)**. Major in Aeronautics engineering.

  Programme includes: aerodynamics, aircraft architecture, aircraft structure, structure dynamics, aircraft performance, flight mechanics, aircraft propulsion, helicopter, aircraft regulation and certification, materials study, statistics, electronic, ...
- *From September 2007 to June 2008 :* Preparatory courses for engineering schools in PCSI section in **Vauvenargues High School** in **Aix en Provence.**
- *In July 2007 :* Baccalauréat diploma (equivalent to High School Diploma) in sciences in **Saint-Tropez High School**. Graduated with honours.

INTERNSHIP EXPERIENCE

June 2009 to August 2009

Snecma – Safran group - Gennevilliers
Operator internship
- Operated in a production sector
- Measured and coated fan blades of the CFM56 and M53 engines

Mars 2003

DCN – Saint-Tropez
Observation internship
Assisted several engineers at their work

WORK EXPERIENCE

July 2007

Caisse d'Epargne – Sainte-Maxime
Summer job
Managed several bank accounts

EDUCATIONAL PROJECT EXPERIENCE

- Executed a safety mechanism study (FMECA and fault tree) of the event « loss of the Vulcain 2 engine thrust »
- Architecture and performance of a supersonic liner Concorde type. Designed the aircraft 3D modelisation
- Sized the wing structure of a Gulfstream G450
- Conducted an IPT projet on a Moon mission with the University of Alabama in Huntsville (UAH)

COMPUTER EXPERIENCE

Software: *Microsoft Office, Solidworks, CATIA V5,* Matlab/Simulink, *NASTRAN/PATRAN, Relex, Maple, Blender*
Hardware: *Windows and Macintosh*
Languages: *C language*

LANGUAGE

French (Native speaker), English (Fluent, TOEIC 915 in May 2010), Spanish (Basic level)

ASSOCIATIVE WORK

President of the association ESO (ESTACA Space Odyssey) : conception, manufacture and launch of experimental rockets in partnership with the CNES (French Space Agency)

Project Styx : Built a mini rocket launch from a plane as part of the PERSEUS project (Scientific and Universitary European Space Research Student Project) initiated by the CNES

HOBBIES

Electric guitar, drums, electronical music, sports (Basket-Ball, skiing, jogging, ...), aeronautics readings, informatics, preparation of the Private Pilot License PPL(A).

Driver's license

François Peraud
1, rue Jules Verne
92300 Levallois-Perret
francois.peraud@estaca.eu
06 32 25 15 70

Elève Ingénieur en 4ème année à l'ESTACA
Ecole Supérieure des Techniques Aéronautiques et des Constructions Automobiles (Levallois-Perret)

FORMATIONS ET DIPLÔMES

Septembre 2010 : Admission en 4e année dominante « Espace » à l'ESTACA.

Juin 2010 : obtention du TOEIC.

Septembre 2009 : Admission en 3 e année à l'ESTACA.

Septembre 2007- Mai 2008 : cycle préparatoire intégré à L'ESTACA.

Juin 2007 : Obtention du baccalauréat S, mention Assez Bien.

EXPERIENCE PROFESSIONNELLE

Juillet 2009 : Emploi saisonnier d'un mois à l'OCP de Saint-Ouen (Seine-Saint-Denis 93) chauffeur-livreur auprès de pharmacies parisiennes.

Juillet 2008 : Stage ouvrier d'un mois à AIRBUS TOULOUSE au département Aircraft Performance (Organisation et classification des dossiers de certification).

COMPETENCES

- **Anglais** (niveau intermédiaire, stage linguistique d'un mois à Londres en **Juillet 2010**).
- **Allemand** (niveau moyen)
- **Informatique** : Solidworks, Catia. Programmation en langage C et Matlab.

Obtention du Permis B en avril 2007.

LOISIRS

Modélisme, surf, astronomie. Responsable Sponsor du BDE 2010-2011 de l'ESTACA.

David M. Perrin

3031 Flint Mill Run
Owens Cross Roads, AL 35763
Cell: (256)-655-0175
dmp0002@uah.edu

CITIZENSHIP U.S.

TECHNICAL SKILLS Microsoft Office applications including Microsoft Visio, MATLAB, MathCAD, Solid Edge, Pro/ENGINEER Wildfire 3.0.

EDUCATION

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

Bachelor of Science in Engineering with a concentration in Aerospace Engineering

GPA: 3.60/4.00; Expected graduation Aug 2011

WORK EXPERIENCE

Aug 2010 – Present **Integrated Product Team** **Huntsville, AL**

- Part of a team of future engineers to design a spacecraft mission to perform radio astronomy on the moon.

Jan 2010 – May 2010 **Jacobs ESTS Group** **Huntsville, AL**

Co-op Student, MSFC Structural and Mechanical Design Branch

Completed NASA Spacecraft Launch & Transportation System course.
Trained using Pro/ENGINEER Wildfire 3.0; completed CAD drawings for Ares I vehicle.

May 2008 – Dec 2009 **Jacobs ESTS Group** **Huntsville, AL**

Co-op Student, MSFC Spacecraft and Auxiliary Propulsion Systems Branch
Developed concept design for Ground Support Equipment Pneumatic Test Panel for Roll Control System (RoCS) acceptance testing for Ares I first stage.
Developed assembly drawings, associated parts lists, and specifications.
Prepared Test Operations Requests (TOR) for over 200 developmental tests for the RoCS System Development Test Article (SDTA).

HONORS AND AWARDS

Sigma Gamma Tau, National Honor Society of Aerospace Engineering (2009 – Present)

The National Society of Leadership and Success (2009 – Present)

Alpha Lambda Delta National Academic Honor Society (2007 – Present)

UAH Foundation Presidential Scholarship Recipient, 2006

AFFILIATIONS American Institute of Aeronautics and Astronautics (AIAA)

Robert Smith

14207 Pulaski Pike
Ardmore, Al 35739
(205)-706-1388
smithr3@uah.edu

CITIZENSHIP U.S.

TECHNICAL SKILLS Solid Edge, MathCAD, IDEAS 10, Nastran / Patran. Experience with FAC, GLG, IST, Appendix J. Experience in handling and tracking NCM.

EDUCATION

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

Bachelor of Science in Engineering with a concentration in Mechanical Engineering,

GPA: 2.97/4.0, Expected graduation May 2011

WORK EXPERIENCE

Jul 2010 – Current **Tennessee Valley Authority** **Decatur, AL**

Engineering Programs Intern

Assisted in maintaining of FAC program for two outages. Field and Office experience.

Assisted in NRC documentation for the IST & Appendix J Programs

Sep 2006 – Dec 2007 **International Diesel of Alabama** **Huntsville, AL**

Machining Intern

Maintained NCM database

Conducted testing for manufacture defects on the Production Troubleshooting Team

Functioned as an engineering assistant

Jan 2005 – Apr 2010 **PMC, Inc** **Huntsville, AL**

City Manager 2008-2010

Hired/fired ~ 30 employees

Managed Customer Relations in North Alabama

Managed local finances and operations

CLEARANCE Nuclear Security (Power Operations)

Jean-Baptiste VINCENT
92, rue Louise Aglaé Crétté.
94400 Vitry sur seine FRANCE
Phone : +33 6.24.66.45.62
Email : jean-baptiste.vincent@estaca.eu

Availability: from may 2011 to august 2012

Aeronautics engineering student
Seeking a full year industrial experience in Aircraft industry.

Education

2007 - 2011	**E.S.T.A.C.A**: a 5 years program in Automotive, Aerospace & Railway Engineering currently in the 4th year of the Aeronautics major http://www.estaca.fr/en/home.html **Program includes:** Aircraft structure, Aircraft architecture and performances, Engine integration, Helicopter. **Projects:** Architecture and performances study of a Supersonic liner, Safety mechanisms study on the event "Loss of the Vulcain 2 engine thrust", Structure study of wing surface gulfstream G450, Engine integration in a stealth aircraft, International space project proposed by NASA.
June 2007	**Baccalaureate** in science specialty physics and chemistry with honours (equivalent to A levels) at Saint Jeanne Elisabeth grammar school Paris 7ième.

Work Experience

2010 - 2011	**Private tuition** in Mathematics, Physics and Chemistry to gramma school students **Skills acquired**: how to explain things in different ways and improved oral communication skills.
2009 to 2011	**CERCLE AERONAUTIQUE DE L'ESTACA**, a student association, ESTACA – Paris – *Communications Manager* • Introduce students in aeronautics world with industrial site visits. • Organization of a symposium on the future of supersonic with French Aeronautics Industry, student in Aerospace Engineering School and a political representative.
July 2008	**Air France Industrie** – Roissy en France – Department of general options. • Update of the organization maps of the offices. • Update of the offices acces database. **Skills acquired**: strictness, respect of strict time delays and rigor.
August 2006 to 2010	**Ville de Paris** – Activity leader in a day care centre at the city of Paris for childs of 6 to 11 years old. **Skills acquired**: patience and tolerance.

Skills & Extra-Professional Activities

Language	**French**: Mother tongue / **English**: TOEIC 765. / **Spanish**: Intermediate Level.
IT	Proficient on pack Office SolidWorks, Matlab, CATIA V5, Nastran & Patran, MS office and Knowledge of programming in C Language.
Aviation	Diploma BIA (Aeronautics Initiation License).
Sports	All terrain bike with friends and scuba diving
Travel	USA – China – Thailand – Poland – Austria – Czech Republic

I.6 Summary of Proposed Program Cooperative Contributions
I.7 Draft International Participation Plan
I.8 Planetary Protection Plan
I.9 Discussion of End-of-Mission Spacecraft Disposal Requirements
I.10 Compliance with Procurement Regulations by NASA PI Proposals
I.11 Master Equipment List (MEL)

Table 18. Master Equipment List

Master Equipment List (MEL)				
Subsystem	**Equipment**	**Mass [kg]**	**Quantity**	**Total Mass [kg]**
Science Instruments	Coilable Boom	11.15	7	78.1
	Support Structure for RA	35.2	1	35.2
	Kapton Panels (RA)	35.5	4	142.0
Power (Orbiter)	Li-Ion Rechargeable Batteries	0.81	24	19.4
	Spectre UTJ Solar Panels	2.1	2	4.2
Power (Lander)	Spectre UTJ Solar Cells	0.002688	4668	12.5
	Li-Ion Rechargeable Batteries	250	1	250.0
Propulsion (Orbiter)	Inertial Wheel	4.2	1	4.2
	Interstage for orbiter and SRM	20	3	60.0
	R4-4D Marquart	3.63	1	3.6
	Helium Valve	0.075	2	0.2
	Pressure Sensor	0.06	7	0.4
	Misc Valving	0.3	4	1.2
	Filter	0.15	4	0.6
	Piping	1.5	1	1.5
	Hydrazine	0.15	1	0.2
	N2O4 Valve	0.15	1	0.2
	Helium Tank	1.256	2	2.5
	Hydrazine Tank	5.64	1	5.6
	N2O4 Tank	2.1	1	2.1
	Nitrogen TetraOxide (N2O4)	59.5	1	59.5
	Hydrazine	98	1	98.0
Propulsion (Lander)	MR-80B	7.94	3	23.8
	Pressurant Tank	1.078	2	2.2
	Piping/Valves/Hardware	36.2	1	36.2
	Hydrazine	137.04	1	137.0
	Propellant Tank	4.798	2	9.6

ACS (Orbiter)	TIROC	0.08	12	1.0
SRM Braking	Propellant	1609.51	1	1609.5
	Inert Mass	124.35	1	124.4
SRM LOI	Propellant	1973.35	1	1973.4
	Inert Mass	124.35	1	124.4
ACS (Lander)	MR-120	0.41	4	1.6
	MR-106-L	0.59	12	7.1
Thermal (Orbiter)	Heat Shielding	0.5	1	0.5
Thermal (Lander)	MLI Insulation	15	1	15.0
	Resistive Heaters	10	1	10.0
Structures (Orbiter)	Misc Bolting/Connections	3.26	1	3.3
	Primary Load	4.03	1	4.0
	Solar Panel Structure	7.55	1	7.6
	Honeycomb Mass	18.92	1	18.9
Structures (Lander)				0.0
	Structural Mass	144.19	1	144.2
				0.0
Communications (Orbiter)	KA Receptor	2	0.5	1.0
	KA-12 Transmitter	1.75	4	7.0
Communications (Lander)	RA SSD for Data Storage	0.45	2	0.9
	KA-12 Transmitter	1.75	2	3.5
Command and Data Handling (Orbiter)	RADAR Altimeter	1.4	1	1.4
	ALHAT	25	1	25.0
	Solid State Drive	0.475	1	0.5
	Star Tracker	1.8	3	5.4
	CPU/Processor	3.2	1	3.2
	Orbiter Power Unit	4.5	1	4.5
Command and Data Handling (Lander)	Star Tracker	1.8	1	1.8
	ALHAT	25	1	25.0
Total				5113.9

I.12 Heritage

Table 19. Heritage

Heritage			
Subsystem	**Element**	**Heritage Level**	**Heritage Examples**
Science Instruments	Coilable Boom	Medium	Shuttle Missions
	Kapton Panels (RA)	Medium	HALCA, Land Based Ras
Power	Li-Ion Batteries	High	Venus Express
	UTJ Solar Panels	High	Venus Express, Spirit of Opportunity
Propulsion	Atlas V551	High	New Horizons
	R4-4D Marquart	High	Apollo Service, Lunar Module
	Inertial Wheel	High	
	MR-80B	High	Dervived from Viking TDE
ACS	TIROC	High	Classified DoD
	MR-106-L	High	NEAR, Genesis, Mercury Messenger
	MR-120	High	Small ICBM
	A-STR Star Tracker	High	Messenger
	MIMU	High	MRO, LRO, DMSP
Thermal	Heat Shielding	High	Mercury, Apollo
	Resistive Heaters	High	Apollo, Mercury
	MLI Insulation	High	Venus Express
Structures	Modular Structure	Low	SCOUT-ETL (Europa Terrestrial Orbiter)
Communications	Ka Receptor	High	
	Ka-12 Transmitter	High	
Command and Data Handling	SSD (HD)	High	EO-1, Landsat-VI7, Hubble
	ALHAT	Low	Theorhetical/Design State
	RADAR Altimeter	High	

I.13 List of Abbreviations and Acronyms

Table 20. List of Abbreviations and Acronyms

Acronym	Phrase
ACS	Altitude Control System
ALHAT	Autonomous Landing and Hazard Avoidance Technology
DALI	Dark Ages Lunar Interferometer
DOI	De-Orbit Initiation
DSN	Deep Space Network
HALCA	Highly Advanced Laboratory for Communications and Astronomy
LOI	Lunar Orbit Insertion
LRA	Lunar Radio Array
MCC	Mid Course Correction
SRM	Solid Rocket Motor
TRL	Technology Readiness Level

ULA	United Launch Alliance
WEB	Warm Electronics Box

I.14 List of References

Hirabayashi, H., et al. "The VLBI Space Observatory Programe and the Radio-Astronomical Satellite HALCA. *Astronomical Society of Japan*. Vol. 52, No. 6, 2000, pp. 955-965

Lazio, J., et al. "*Technology Development for The Lunar Radio Array*."

NASA, "Systems Engineering Handbook," NASA/SP-2007-6105 Rev1, 2007.

Stephan, R., "Overview of the Altair Lunar Lander Thermal Control System Design and the Impacts of Global Access," NASA Johnson Space Center, Houston, TX.

NASA, "Announcement of Opportunity Discovery 2010," NNH10ZDA007O, 2010.

Shearer, C., and Tahu, G., "Lunar Polar Volatiles Explorer (LPVE) Mission Concept Study."

Dawson, M., et al., "Joint Propulsion Conference & Exhibit," Monopropellant Hydrazine 700 lbf Throttling Terminal Descent Engine for Mars Science Laboratory, 2007.

EADS : European Aeronautic Defence and Space Compagny : ASTRIUM website : http://www.astrium.eads.net/en/equipment/

THALES ALENIA SPACE website : http://www.thalesgroup.com/aerospace/

"Element of spacecraft design" by Charles Brown

Parabolic antenna courses : http://f5zv.pagesperso-orange.fr/RADIO/RM/RM09/RM09i03D.html

Earth view : http://www.fourmilab.ch/cgi-bin/Earth/action?opt=-m

The control for a satellite : http://www.je-comprends-enfin.fr/index.php?/Technologies-des-satellites/maintien-a-poste-dun-satellite/id-menu-53.html

Thermal control for a satellite : http://www.techno-science.net/?onglet=glossaire&definition=10190

Altas V user guide

Courses from ESTACA School: "Mécanique spatial", "Structure", Satellite", "Propulsion liquid", „Architecture lanceur".

Lester, D., "*Astronomy From The Moon: Science Up On The Rocks?,* " Department of Astronomy, University of Texas, Austin, Texas.

Burns, J., et al., "The LUNAR Consortium," *Science From the Moon: The NASA/NLSI Lunar Univeristy Network for Astrophysics Research (LUNAR)*.

Lester, D., "*Does the Lunar Surface Still Offer Value As a Site for Astronomical Observatories?,"2004.*

Abbot, J., Pixton, S., and Roberts, C., *"Lunar Interferometric Radio Array L.I.R.A.,"* Embry-Riddle Aeronautical University, USA.

Burns, J., "Lunar University Node for Astrophysics Research (LUNAR): Exploring the Cosmos From the Moon," Center for Astrophysics and Space Astonomy, University of Colorado at Boulder, Boulder, CO.

I.15 NASA-Developed Technology Infusion Plan

I.16 Description or Enabling Nature of ASRG

Team LIBRA shall not utilize ASRG for this spacecraft mission.

I.17 Images and Tables

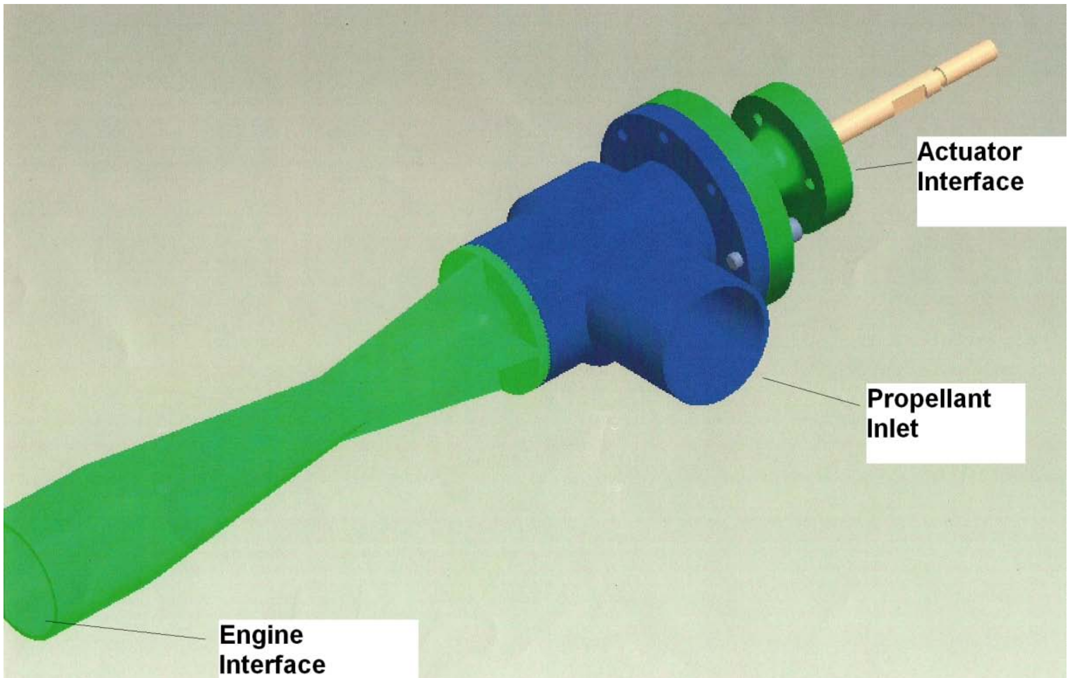


Figure 13. TCaV Assembly

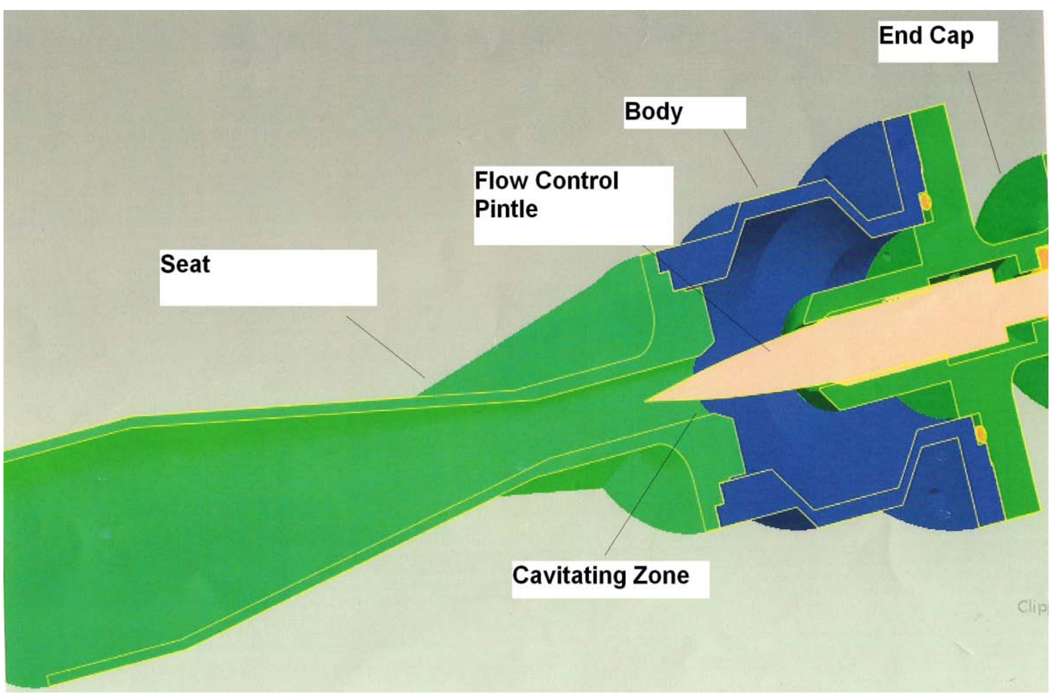


Figure 14. TCaV Cross-section

QuickCost Model
A Project Cost Model
Joe Hamaker, Cost
Version:May 9 2005

The regression equation is:
LnCostUN = -0.169 + 0.344*LnMass + 0.090*LnPower + 0.0807*LnLife
+ 0.0951*LnSciOrg + 0.0865*Apogee + 0.240*Data + 0.115*Test
+ 0.0815*Reqmts + 0.0980*Funding + 0.239*Team + 0.0918*Study
+ 1.46*New - 0.00946*ATP

Or rewritten in non-log terms as power equation:
Cost = 0.845 * Mass^0.344 * Power^0.090 * Life^0.0807 * SciOrgs^0.0951
* e^(0.0865*Apogee) * e^(0.240*Data) * e^(0.115*Test)
*e^(0.0815*Reqmts) * e^(0.0980*Funding) *e^(0.239*Team)
*e^(0.0918*Study) * e^(1.46*New) * 1 / [e^(0.00946*ATP)]

Project Name: Template With Representative Inputs

RAM-B-Dual Landers

Basis of CER	Variable Description	Make Inputs In Blue Cells	Conversion	Variable Units	Range In Database	Database Average	Database Median
Regression	Enter Spacecraft Bus + Instruments Total Dry Mass	680.18		KG	76 to 14,475 KG	1263	664
	Enter Spacecraft Total Power Generation Capacity (LEO Equivalent)	220	220	W LEO equivalent flux	90 to 7500 W	1142	700
	Enter Design Life in Months	60.0		Months	12 to 180 months	48	36
	Enter Number of Science Organizations	1.0	1	Count (Enter zero for projects with no science or science organization involvement)	0 to 36	9.21	6
	Enter Apogee Class	4.0		LEO=1, HEO/GEO=2, beyond GEO=3, Planetary=4	1 to 4	2.32	2
	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	0% to 100%	0.5	50%
	Enter Test Requirements Class	2.0		Less than average testing=1, Average=2, More than average=3, Extensive=4, Very extensive=5	1 to 3	2.16	2
	Enter Requirements Stability Class	3.0		Very low volatility=1, Low=2, Average=3, High=4, Very high volatility=5	1 to 5	3.01	3
	Enter Funding Stability Class	2.0		Stable funding=1, Some instability=2, Significant instability=3	1 to 3	2.14	2
	Enter Team Experience Class [Derived from Price Model; used with permission from Price Systems LLP]	4.0		Extensive experience=1, Better than average=2, Average (mixed experience)=3, Unfamiliar=4 [Ref: Price Model]	1 to 4	2.93	4
	Enter Formulation Study Class	2.0		Formulation study (1=Major, 2=Nominal, 3=Minor)	1 to 3	2.17	2.0
	Enter New Design Percent	70%		Simple mod=30%, Extensive mod=70% (average), New=100%	38% to 130%	71%	69%
	Enter ATP Date Expressed as Years Since 1960	51		Years elapsed since 1960	1 to 45	25.42	28.0
	Regression Model Result	$240.7		DDT&E + TFU (Phases C/D/E) in Millions of 2004 Dollars including fee, excluding full cost			
			Factor				
Heuristics	Enter Technology Readiness Level (TRL) Penalty Factor	5.0	1.30	Refer to NASA TRL scale (TRL 6 is nominal)			
Based on Price Model	Enter Platform Factor [Derived from Price Model; used with permission from Price Systems LLP)	2.20	1.27	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 Mdoel]	2.0 to 2.7	2	2.0
	Enter Functional Complexity Factor	To Be Added Later	1.00	To be added later			
	Subtotal (Non Full Cost Subtotal)	$396.9	-----	Subtotal (Millions of 2004 Dollars including fee)			
Heuristics	Calculated Size of the Government Project Office (Project Office Only--Excludes Government Functional Line/Laboratory Labor)	47.6	-----	Civil service annual full time equivalents (FTE's)			

QuickCost Model
A Project Cost Model
Joe Hamaker, Cost
Version:May 9 2005

The regression equation is:
LnCostUN = -0.169 + 0.344*LnMass + 0.090*LnPower + 0.0807*LnLife
+ 0.0951*LnSciOrg + 0.0865*Apogee + 0.240*Data + 0.115*Test
+ 0.0815*Reqmts + 0.0980*Funding + 0.239*Team + 0.0918*Study
+ 1.46*New - 0.00946*ATP

Or rewritten in non-log terms as power equation:
Cost = 0.845 * Mass^0.344 * Power^0.090 * Life^0.0807 * SciOrgs^0.0951
* e^(0.0865*Apogee) * e^(0.240*Data) * e^(0.115*Test)
*e^(0.0815*Reqmts) * e^(0.0980*Funding) *e^(0.239*Team)
*e^(0.0918*Study) * e^(1.46*New) * 1 / [e^(0.00946*ATP)]

Project Name: Template With Representative Inputs

Basis of CER	Variable Description	Make Inputs In Blue Cells	Conversion	Variable Units	Range In Database	Database Average	Database Median
Regression	Enter Spacecraft Bus + Instruments Total Dry Mass	162.46		KG	76 to 14,475 KG	1263	664
	Enter Spacecraft Total Power Generation Capacity (LEO Equivalent)	220	220	W LEO equivalent flux	90 to 7500 W	1142	700
	Enter Design Life in Months	60.0		Months	12 to 180 months	48	36
	Enter Number of Science Organizations	1.0	1	Count (Enter zero for projects with no science or science organization involvement)	0 to 36	9.21	6
	Enter Apogee Class	4.0		LEO=1, HEO/GEO=2, beyond GEO=3, Planetary=4	1 to 4	2.32	2
	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	0% to 100%	0.5	50%
	Enter Test Requirements Class	2.0		Less than average testing=1, Average=2, More than average=3, Extensive=4, Very extensive=5	1 to 3	2.16	2
	Enter Requirements Stability Class	3.0		Very low volatility=1, Low=2, Average=3, High=4, Very high volatility=5	1 to 5	3.01	3
	Enter Funding Stability Class	2.0		Stable funding=1, Some instability=2, Significant instability=3	1 to 3	2.14	2
	Enter Team Experience Class [Derived from Price Model; used with permission from Price Systems LLP]	4.0		Extensive experience=1, Better than average=2, Average (mixed experience)=3, Unfamiliar=4 [Ref: Price Model]	1 to 4	2.93	4
	Enter Formulation Study Class	2.0		Formulation study (1=Major, 2=Nominal, 3=Minor)	1 to 3	2.17	2.0
	Enter New Design Percent	70%		Simple mod=30%, Extensive mod=70% (average), New=100%	38% to 130%	71%	69%
	Enter ATP Date Expressed as Years Since 1960	51		Years elapsed since 1960	1 to 45	25.42	28.0
	Regression Model Result	$147.1		DDT&E + TFU (Phases C/D/E) in Millions of 2004 Dollars including fee, excluding full cost			
			Factor				
Heuristics	Enter Technology Readiness Level (TRL) Penalty Factor	7.0	0.90	Refer to NASA TRL scale (TRL 6 is nominal)			
Based on Price Model	Enter Platform Factor [Derived from Price Model; used with permission from Price Systems LLP]	2.20	1.27	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 Mdoel]	2.0 to 2.7	2	2.0
	Enter Functional Complexity Factor	To Be Added Later	1.00	To be added later			
	Subtotal (Non Full Cost Subtotal)	$167.9	-----	Subtotal (Millions of 2004 Dollars including fee)			
Heuristics	Calculated Size of the Government Project Office (Project Office Only--Excludes Government Functional Line/Laboratory Labor)	47.6	-----	Civil service annual full time equivalents (FTE's)			

	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)			
	Final Estimate of the Size of the Government Project Office and other Oversight (excludes government non-oversight labor which is included in subtotal above)	47.6	-----	Civil Service Full Time Equivalents (FTE's)			
	Enter Civil Service Loaded Annual Labor Rate Including Center and Corporate G&A	$280,000	-----	Thousands of 2004 Dollars	(Years)		
Regression	Calculated Project Phase C/D Schedule Duration (Excludes O&S Phase E)	32	-----	Months	2.6		
	Enter Override of Calculated Phase C/D Schedule Duration (or leave zero to accept calculated duration)	36	-----	Months			
	Final Estimate of the Project Phase C/D Schedule Duration	36	-----	Months			
	Calculated Cost of the Government Project Office	$35.3	-----	Millions of 2004 Dollars			
Hueristics	Government Service Pool Use Intenstiy Factor	3	0.0900	1=Minimum use of service pools, 2=Less than average, 3=Average, 4=More than average, 5=Significantly more than average			
	Calculated Cost of Government Service Pool Use	$15.1	-----		1 to 1.25	1.19	1.25
	Enter Override of Calculated Cost of Government Service Pool Use (or leave zero to accept calculated service pool cost)	$0.0					
	Final Estimate of the Cost of Government Service Pool Use	$15.1					
	Subtotal (2004$)	$218.3					
	Ground System	$19.6					
	Enter Override of Calculated Ground System Cost	$0.0					
	Final Estimate of the Cost of Ground System	$19.6					
	Subtotal (2004$)	$237.9					
	Enter Launch Services Cost	$0.0					
	Enter Cost Reserves	$59.5					
	Total (2004$)	$297.4					

QuickCost Model
A Project Cost Model
Joe Hamaker, Cost
Version:May 9 2005

The regression equation is:
LnCostUN = -0.169 + 0.344*LnMass + 0.090*LnPower + 0.0807*LnLife
+ 0.0951*LnSciOrg + 0.0865*Apogee + 0.240*Data + 0.115*Test
+ 0.0815*Reqmts + 0.0980*Funding + 0.239*Team + 0.0918*Study
+ 1.46*New - 0.00946*ATP

Or rewritten in non-log terms as power equation:
Cost = 0.845 * Mass^0.344 * Power^0.090 * Life^0.0807 * SciOrgs^0.0951
* e^(0.0865*Apogee) * e^(0.240*Data) * e^(0.115*Test)
*e^(0.0815*Reqmts) * e^(0.0980*Funding) *e^(0.239*Team)
*e^(0.0918*Study) * e^(1.46*New) * 1 / [e^(0.00946*ATP)]

Project Name: Template With Representative Inputs

RAM-B-Single Lander

Basis of CER	Variable Description	Make Inputs In Blue Cells	Conversion	Variable Units	Range In Database	Database Average	Database Median
Regression	Enter Spacecraft Bus + Instruments Total Dry Mass	340.09		KG	76 to 14,475 KG	1263	664
	Enter Spacecraft Total Power Generation Capacity (LEO Equivalent)	110	110	W LEO equivalent flux	90 to 7500 W	1142	700
	Enter Design Life in Months	60.0		Months	12 to 180 months	48	36
	Enter Number of Science Organizations	1.0	1	Count (Enter zero for projects with no science or science organization involvement)	0 to 36	9.21	6
	Enter Apogee Class	4.0		LEO=1, HEO/GEO=2, beyond GEO=3, Planetary=4	1 to 4	2.32	2
	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	0% to 100%	0.5	50%
	Enter Test Requirements Class	2.0		Less than average testing=1, Average=2, More than average=3, Extensive=4, Very extensive=5	1 to 3	2.16	2
	Enter Requirements Stability Class	3.0		Very low volatility=1, Low=2, Average=3, High=4, Very high volatility=5	1 to 5	3.01	3
	Enter Funding Stability Class	2.0		Stable funding=1, Some instability=2, Significant instability=3	1 to 3	2.14	2
	Enter Team Experience Class [Derived from Price Model; used with permission from Price Systems LLP]	4.0		Extensive experience=1, Better than average=2, Average (mixed experience)=3, Unfamiliar=4 [Ref: Price Model]	1 to 4	2.93	4
	Enter Formulation Study Class	2.0		Formulation study (1=Major, 2=Nominal, 3=Minor)	1 to 3	2.17	2.0
	Enter New Design Percent	70%		Simple mod=30%, Extensive mod=70% (average), New=100%	38% to 130%	71%	69%
	Enter ATP Date Expressed as Years Since 1960	51		Years elapsed since 1960	1 to 45	25.42	28.0
	Regression Model Result	$178.2		DDT&E + TFU (Phases C/D/E) in Millions of 2004 Dollars including fee, excluding full cost			
			Factor				
Heuristics	Enter Technology Readiness Level (TRL) Penalty Factor	5.0	1.30	Refer to NASA TRL scale (TRL 6 is nominal)			
Based on Price Model	Enter Platform Factor [Derived from Price Model; used with permission from Price Systems LLP)	2.20	1.27	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 Mdoel]	2.0 to 2.7	2	2.0
	Enter Functional Complexity Factor	To Be Added Later	1.00	To be added later			
	Subtotal (Non Full Cost Subtotal)	$293.8	-----	Subtotal (Millions of 2004 Dollars including fee)			
Heuristics	Calculated Size of the Government Project Office (Project Office Only--Excludes Government Functional Line/Laboratory Labor)	47.6	-----	Civil service annual full time equivalents (FTE's)			

	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)			
	Final Estimate of the Size of the Government Project Office and other Oversight (excludes government non-oversight labor which is included in subtotal above)	47.6	-----	Civil Service Full Time Equivalents (FTE's)			
	Enter Civil Service Loaded Annual Labor Rate Including Center and Corporate G&A	$280,000	-----	Thousands of 2004 Dollars	(Years)		
Regression	Calculated Project Phase C/D Schedule Duration (Excludes O&S Phase E)	38	-----	Months	3.2		
	Enter Override of Calculated Phase C/D Schedule Duration (or leave zero to accept calculated duration)	36	-----	Months			
	Final Estimate of the Project Phase C/D Schedule Duration	36	-----	Months			
	Calculated Cost of the Government Project Office	$42.6	-----	Millions of 2004 Dollars			
Hueristics	Government Service Pool Use Intenstiy Factor	3	0.0900	1=Minimum use of service pools, 2=Less than average, 3=Average, 4=More than average, 5=Significantly more than average			
	Calculated Cost of Government Service Pool Use	$26.4	-----		1 to 1.25	1.19	1.25
	Enter Override of Calculated Cost of Government Service Pool Use (or leave zero to accept calculated service pool cost)	$0.0					
	Final Estimate of the Cost of Government Service Pool Use	$26.4					
	Subtotal (2004$)	$362.9					
	Ground System	$32.7					
	Enter Override of Calculated Ground System Cost	$0.0					
	Final Estimate of the Cost of Ground System	$32.7					
	Subtotal (2004$)	$395.5					
	Enter Launch Services Cost	$0.0					
	Enter Cost Reserves	$98.9					
	Total (2004$)	$494.4					

QuickCost Model
A Project Cost Model
Joe Hamaker, Cost
Version:May 9 2005

The regression equation is:
LnCostUN = -0.169 + 0.344*LnMass + 0.090*LnPower + 0.0807*LnLife
+ 0.0951*LnSciOrg + 0.0865*Apogee + 0.240*Data + 0.115*Test
+ 0.0815*Reqmts + 0.0980*Funding + 0.239*Team + 0.0918*Study
+ 1.46*New - 0.00946*ATP

Or rewritten in non-log terms as power equation:
Cost = 0.845 * Mass^0.344 * Power^0.090 * Life^0.0807 * SciOrgs^0.0951
* e^(0.0865*Apogee) * e^(0.240*Data) * e^(0.115*Test)
*e^(0.0815*Reqmts) * e^(0.0980*Funding) *e^(0.239*Team)
*e^(0.0918*Study) * e^(1.46*New) * 1 / [e^(0.00946*ATP)]

Project Name: Template With Representative Inputs

RAM-B-Single Orbiter

Basis of CER	Variable Description	Make Inputs In Blue Cells	Conversion	Variable Units	Range In Database	Database Average	Database Median
Regression	Enter Spacecraft Bus + Instruments Total Dry Mass	81.23		KG	76 to 14,475 KG	1263	664
	Enter Spacecraft Total Power Generation Capacity (LEO Equivalent)	110	110	W LEO equivalent flux	90 to 7500 W	1142	700
	Enter Design Life in Months	60.0		Months	12 to 180 months	48	36
	Enter Number of Science Organizations	1.0	1	Count (Enter zero for projects with no science or science organization involvement)	0 to 36	9.21	6
	Enter Apogee Class	4.0		LEO=1, HEO/GEO=2, beyond GEO=3, Planetary=4	1 to 4	2.32	2
	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	0% to 100%	0.5	50%
	Enter Test Requirements Class	2.0		Less than average testing=1, Average=2, More than average=3, Extensive=4, Very extensive=5	1 to 3	2.16	2
	Enter Requirements Stability Class	3.0		Very low volatility=1, Low=2, Average=3, High=4, Very high volatility=5	1 to 5	3.01	3
	Enter Funding Stability Class	2.0		Stable funding=1, Some instability=2, Significant instability=3	1 to 3	2.14	2
	Enter Team Experience Class [Derived from Price Model; used with permission from Price Systems LLP]	4.0		Extensive experience=1, Better than average=2, Average (mixed experience)=3, Unfamiliar=4 [Ref: Price Model]	1 to 4	2.93	4
	Enter Formulation Study Class	2.0		Formulation study (1=Major, 2=Nominal, 3=Minor)	1 to 3	2.17	2.0
	Enter New Design Percent	70%		Simple mod=30%, Extensive mod=70% (average), New=100%	38% to 130%	71%	69%
	Enter ATP Date Expressed as Years Since 1960	51		Years elapsed since 1960	1 to 45	25.42	28.0
	Regression Model Result	$108.9		DDT&E + TFU (Phases C/D/E) in Millions of 2004 Dollars including fee, excluding full cost			
			Factor				
Heuristics	Enter Technology Readiness Level (TRL) Penalty Factor	7.0	0.90	Refer to NASA TRL scale (TRL 6 is nominal)			
Based on Price Model	Enter Platform Factor [Derived from Price Model; used with permission from Price Systems LLP)	2.20	1.27	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 Mdoel]	2.0 to 2.7	2	2.0
	Enter Functional Complexity Factor	To Be Added Later	1.00	To be added later			
	Subtotal (Non Full Cost Subtotal)	$124.3	-----	Subtotal (Millions of 2004 Dollars including fee)			
Heuristics	Calculated Size of the Government Project Office (Project Office Only--Excludes Government Functional Line/Laboratory Labor)	47.6	-----	Civil service annual full time equivalents (FTE's)			

	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)			
	Final Estimate of the Size of the Government Project Office and other Oversight (excludes government non-oversight labor which is included in subtotal above)	47.6	-----	Civil Service Full Time Equivalents (FTE's)			
	Enter Civil Service Loaded Annual Labor Rate Including Center and Corporate G&A	$280,000	-----	Thousands of 2004 Dollars			
Regression	Calculated Project Phase C/D Schedule Duration (Excludes O&S Phase E)	27	-----	Months	(Years) 2.2		
	Enter Override of Calculated Phase C/D Schedule Duration (or leave zero to accept calculated duration)	36	-----	Months			
	Final Estimate of the Project Phase C/D Schedule Duration	36	-----	Months			
	Calculated Cost of the Government Project Office	$29.5	-----	Millions of 2004 Dollars			
Hueristics	Government Service Pool Use Intenstiy Factor	3	0.0900	1=Minimum use of service pools, 2=Less than average, 3=Average, 4=More than average, 5=Significantly more than average			
	Calculated Cost of Government Service Pool Use	$11.2	-----		1 to 1.25	1.19	1.25
	Enter Override of Calculated Cost of Government Service Pool Use (or leave zero to accept calculated service pool cost)	$0.0					
	Final Estimate of the Cost of Government Service Pool Use	$11.2					
	Subtotal (2004$)	$165.0					
	Ground System	$14.9					
	Enter Override of Calculated Ground System Cost	$0.0					
	Final Estimate of the Cost of Ground System	$14.9					
	Subtotal (2004$)	$179.9					
	Enter Launch Services Cost	$0.0					
	Enter Cost Reserves	$45.0					
	Total (2004$)	$224.8					

I.18 Calculations

VARIABLES:

$Mass := 7 \cdot 1kg \cdot 1.3 = 9.1\,kg$	Mass Gifted for 13 additional antennae (on orbiter)
$Draw_{RA} := 13.5\frac{W}{m^2}$	Radio Array Power Consumption
$Data_{in} := \frac{1}{s\,(m)^2}$	Data Rate (MB/s)
$Comp := 10$	Compression Ratio
$A_{optimal} := 10m^2$	Solar Panel Area (Max available)

Energy Calculations: State of the Art

Fixed: Data Rate, Batt Density, Batt mass, Solar Panel Data
Variable: Power Consumed

$Mass_{Ant} := 1kg \cdot 1.3$	Antenna Mass (8 oz antenna)
$Mass_{Batt1} := 200kg$	Single Antenna Battery Mass
$Mass_{Batt2} := Mass_{Batt1} - 1.3 \cdot 3 \cdot Mass_{Ant} = 194.93\,kg$	Dual Antenna Battery Mass
$X := 2$	Variable Antenna Number
$Mass_{BattX} := Mass_{Batt1} - (2X - 1) \cdot Mass_{Ant} = 196.1\,kg$	X Antenna Battery Mass
$M_{Ant} := Mass_{Ant} \cdot X = 2.6\,kg$	Antenna Mass Allocation
$Batt_{\rho} := 220W \cdot \frac{hr}{kg}$	Battery Power Density
$Solar_{Raw} := 1300 \frac{W}{m^2}$	Raw Solar Energy
$t_{orb} := 750s$	Transmission Window
$Pass := 1$	Passes per Hour
$P_{ant} := 10W$	Power per Antenna
$P_{thermal} := 1.3 \cdot 20W$	Thermal Systems Power
$Life := 5$	Mission Length: Years
$t_{night} := 354hr \qquad t_{day} := 24 \cdot 14 \cdot hr$	Day/Night
$A_{Panel} := 25m^2 \qquad A_{RA} := A_{Panel} \cdot 4 = 100\,m^2$	RA Area per site

$Data_{out} := \frac{.01311}{s}$ Data Rate (per antenna): MB/s

$NightData_{1Out} := Comp \cdot Data_{out} \cdot t_{orb} \cdot 354$

$NightData_{1Out} = 3.481 \times 10^{4}$ Single Antenna: Nightly Output (MB)

$NightData_{2Out} := 2 \cdot Comp \cdot Data_{out} \cdot t_{orb} \cdot 354 = 6.961 \times 10^{4}$

$\left(NightData_{2Out}\right) = 6.961 \times 10^{4}$ Dual Antenna: Nightly Output (MB)

$NightData_{XOut} := X \cdot Comp \cdot Data_{out} \cdot t_{orb} \cdot 354 = 6.961 \times 10^{4}$ X Antennae: Nightly Output (MB)

$\left(NightData_{XOut}\right) = 6.961 \times 10^{4}$

$HD_{1size} := NightData_{1Out} \cdot 1.3 = 4.525 \times 10^{4}$ 1 Antenna Hard Drive Size (MB)

$HD_{2size} := NightData_{2Out} \cdot 1.3 = 9.05 \times 10^{4}$ 2 Antenna Hard Drive Size (MB)

$HD_{Xsize} := NightData_{XOut} \cdot 1.3 = 9.05 \times 10^{4}$ X Antennae Hard Drive Size (MB)

$\frac{HD_{Xsize}}{1000} = 90.498$ X Antennae Hard Drive Size (GB)

$P_{harness} := .03\left(P_{ant} + P_{thermal}\right)$ Power Loss: Harness & Connections

$\rho_{solar} := 84 \frac{mg}{cm^{2}} = 0.84 \frac{kg}{m^{2}}$ Density of UTJ-X Cells

$\varepsilon_{panels} := .283$ Panel Efficiences

$\varepsilon_{DNI} := .5$

$P_{Harddrive} := 10W$ Power HD

$P_{solar} := Solar_{Raw} \cdot \varepsilon_{panels} \cdot \varepsilon_{DNI} = 183.95 \cdot \frac{W}{m^{2}}$ Solar Power (BOL)

$Lossrate_{Solar} := .99$

$P_{EOL} := .99^{Life} - 1.3\left(1 - .99^{Life}\right) = 0.887$

EOL Power (Margin)

$T_{Solar} := \left(\frac{110}{180}\right) \cdot 14 \cdot 24 \cdot hr = 205.333 \cdot hr$

Available Charging time (30* Solar Angle)

$t_{RAin1} := \frac{HD_{1size}}{Data_{in} \cdot A_{RA}} = 7.542 \cdot min$

GIVEN VARIABLES, TIME TO FILL Hard Drive (SINGLE ANTENNA)

$t_{RAin2} := \frac{HD_{2size}}{Data_{in} \cdot A_{RA}} = 15.083 \cdot min$

GIVEN VARIABLES, TIME TO FILL Hard Drive (DUAL ANTENNA)

$t_{RAinX} := \frac{HD_{Xsize}}{Data_{in} \cdot A_{RA}} = 15.083 \cdot min$

GIVEN VARIABLES, TIME TO FILL Hard Drive (X ANTENNA)

$Pcons_{thermal} := P_{thermal} \cdot t_{night} = 9.204 \cdot kW \cdot hr$

Nighttime Thermal Power Use

$Pcons_{ant1} := P_{ant} \cdot 354 \cdot t_{orb} = 0.738 \cdot kW \cdot hr$

Nighttime Single Ant. Power Use

$Pcons_{ant2} := 2 \cdot P_{ant} \cdot 354 \cdot t_{orb} = 1.475 \cdot kW \cdot hr$

Nighttime Dual Ant. Power Use

$Pcons_{antX} := X \cdot P_{ant} \cdot 354 \cdot t_{orb} = 1.475 \cdot kW \cdot hr$

Nighttime Five Ant. Power Use

$Pcons_{RA1} := A_{RA} \cdot Draw_{RA} \cdot t_{RAin1} = 0.17 \cdot kW \cdot hr$

Nighttime RA Power Use (1 ANTENNA)

$Pcons_{RA2} := A_{RA} \cdot Draw_{RA} \cdot t_{RAin2} = 0.339 \cdot kW \cdot hr$

Nighttime RA Power Use (2 ANTENNA)

$Pcons_{RAX} := A_{RA} \cdot Draw_{RA} \cdot t_{RAinX} = 0.339 \cdot kW \cdot hr$

Nighttime RA Power Use (5 ANTENNA)

$$P_{1night} := Mass_{Batt1} \cdot \frac{Batt_{\rho}}{1.3} = 33.846 \cdot kW \cdot hr$$

Battery Power Avail (Marginal)
1 Antenna

$$P_{2night} := Mass_{Batt2} \cdot \frac{Batt_{\rho}}{1.3} = 32.988 \cdot kW \cdot hr$$

Battery Power Avail (Marginal)
2 Antenna

$$P_{Xnight} := Mass_{BattX} \cdot \frac{Batt_{\rho}}{1.3} = 33.186 \cdot kW \cdot hr$$

Battery Power Avail (Marginal)
X Antenna

$$P_{HS} := 10W$$

$$P_{consHS} := P_{HS} \cdot t_{night}$$

High School Power Use

$$Pcons_{har} := P_{harness} \cdot t_{night} = 0.382 \cdot kW \cdot hr$$

Nighttime harness loss

$$Pcons_{harddrive} := P_{Harddrive} \cdot t_{night} = 1.274 \times 10^{7} J$$

$$P_{1ant} := 1.3\left(Pcons_{thermal} + Pcons_{ant1} + Pcons_{RA1} + Pcons_{har} + P_{consHS} + Pcons_{harddrive}\right)$$

$$P_{1ant} = 22.846 \cdot kW \cdot hr$$

Night Power: 1 Antenna

$$P_{2ant} := Pcons_{thermal} + Pcons_{ant2} + Pcons_{RA2} + Pcons_{har} + P_{consHS} + Pcons_{harddrive}$$

$$P_{2ant} = 18.481 \cdot kW \cdot hr$$

Night Power: 2 Antenna

$$P_{Xant} := Pcons_{thermal} + Pcons_{antX} + Pcons_{RAX} + Pcons_{har} + P_{consHS} + Pcons_{harddrive}$$

$$P_{Xant} = 18.481 \cdot kW \cdot hr$$

Night Power: X Antenna

$$RemainingCharge_1 := \frac{\left(P_{1night} - P_{1ant}\right)}{P_{1night}} \cdot 100 = 32.502$$

% Power Remaining (Single)

$$RemainingCharge_2 := \frac{\left(P_{2night} - P_{2ant}\right)}{P_{2night}} \cdot 100 = 43.978$$

% Power Remaining (Dual)

$$RemainingCharge_X := \frac{\left(P_{Xnight} - P_{Xant}\right)}{P_{Xnight}} \cdot 100 = 44.312$$

% Power Remaining (Five Antennae)

Daylight Operations:

$$P_{Daytime} := 1.3Pcons_{thermal} + 1.3Pcons_{antX} + 1.3Pcons_{RAX} + 1.3Pcons_{har} + 1.3P_{consHS} + 1.3 \cdot P_{Xnight} = 62.565 \cdot kW \cdot hr$$

$$A_{min} := \frac{P_{Daytime}}{T_{Solar} \cdot P_{solar}} = 1.656\,m^2$$

$$ShadeFactor := 2$$

$$A_{Solar} := 1.3ShadeFactor \cdot A_{min} = 4.307\,m^2$$

$$M_{solarmin} := A_{Solar} \cdot \rho_{solar} = 3.618\,kg$$

$$M_{solaroptimal} := A_{optimal} \cdot \rho_{solar} = 8.4\,kg$$

Battery Charge Rate

$$C_r := \frac{P_{2ant}}{t_{day}} = 55.002\,W$$

Energy Calculations, Future Requirements:

Fixed: Data Rate, Batt Density, Batt mass, Solar Panel Data

Variable: Power Consumed

$P_{ConstantDaytime} := C_r + 2 \cdot P_{ant} + P_{thermal} + P_{HS} + Draw_{RA} \cdot A_{RA} = 1.461 \times 10^3 \, W$

VARIABLES:

$A_{Max} := \frac{1.3 P_{ConstantDaytime}}{P_{solar}} = 10.325 \, m^2$

$Draw_{RA} := 1.15 \frac{W}{m^2}$ Radio Array Power Consumption

Full Time Running:

$Data_{in} := \frac{1}{s \, (m)^2}$ Data Rate (MB/s)

$Mass_{Max} := 250 kg$

$Comp := 10$ Compression Ratio

$Batt_\rho := 220 W \cdot \frac{hr}{kg}$

$A_{optimal} := 15 m^2$ Solar Panel Area (Max available)

$P_{max} := Mass_{Max} \cdot Batt_\rho = 55 \cdot kW \cdot hr$

$P_{avail} := \frac{P_{max}}{2} - Pcons_{thermal} - Pcons_{ant2} - P_{consHS} - Pcons_{har} = 12.899 \cdot kW \cdot hr$

$t_{avail} := \frac{P_{avail}}{Draw_{RA} \cdot A_{RA}} = 0.398 \cdot day$

$Mass_{Ant} := 1 kg \cdot 1.3$ Antenna Mass (8 oz antenna)

$Mass_{Batt1} := 200 kg$ Single Antenna Battery Mass

$Mass_{Batt2} := Mass_{Batt1} - 1.3 \cdot 3 \cdot Mass_{Ant} = 194.93 \, kg$ Dual Antenna Battery Mass

$X := 2$ Variable Antenna Number

$Mass_{BattX} := Mass_{Batt1} - (2X - 1)\cdot Mass_{Ant} = 196.1\,kg$ X Antenna Battery Mass

$M_{Ant} := Mass_{Ant}\cdot X = 2.6\,kg$ Antenna Mass Allocation

$Batt_{\rho} := 300W\cdot\frac{hr}{kg}$ Battery Power Density

$Solar_{Raw} := 1300\frac{W}{m^2}$ Raw Solar Energy

$t_{orb} := 750s$ Transmission Window

$Pass := 1$ Passes per Hour

$P_{ant} := 10W$ Power per Antenna

$P_{thermal} := 1.3\cdot 20W$ Thermal Systems Power

$Life := 5$ Mission Length: Years

$t_{night} := 354hr \qquad t_{day} := 24\cdot 14\cdot hr$ Day/Night

$A_{Panel} := 25m^2 \qquad A_{RA} := A_{Panel}\cdot 4 = 100\,m^2$ RA Area per site

$Data_{out} := \frac{.01311}{s}$ Data Rate (per antenna): MB/s

$NightData_{1Out} := Comp\cdot Data_{out}\cdot t_{orb}\cdot 354$

$NightData_{1Out} = 3.481\times 10^4$ Single Antenna: Nightly Output (MB)

$NightData_{2Out} := 2\cdot Comp\cdot Data_{out}\cdot t_{orb}\cdot 354 = 6.961\times 10^4$

$(NightData_{2Out}) = 6.961\times 10^4$ Dual Antenna: Nightly Output (MB)

$NightData_{XOut} := X\cdot Comp\cdot Data_{out}\cdot t_{orb}\cdot 354 = 6.961\times 10^4$

$(NightData_{XOut}) = 6.961\times 10^4$ X Antennae: Nightly Output (MB)

$HD_{1size} := NightData_{1Out}\cdot 1.3 = 4.525\times 10^4$ 1 Antenna Hard Drive Size (MB)

$HD_{2size} := NightData_{2Out} \cdot 1.3 = 9.05 \times 10^4$ 2 Antenna Hard Drive Size (MB)

$HD_{Xsize} := NightData_{XOut} \cdot 1.3 = 9.05 \times 10^4$ X Antennae Hard Drive Size (MB)

$\frac{HD_{Xsize}}{1000} = 90.498$ X Antennae Hard Drive Size (GB)

$P_{harness} := .03(P_{ant} + P_{thermal})$ Power Loss: Harness & Connections

$\rho_{solar} := 84\frac{mg}{cm^2} = 0.84\frac{kg}{m^2}$ Density of UTJ-X Cells

$\varepsilon_{panels} := .283$

$\varepsilon_{DNI} := .5$ Panel Efficiences

$P_{Harddrive} := 10W$ Power HD

$P_{solar} := Solar_{Raw} \cdot \varepsilon_{panels} \cdot \varepsilon_{DNI} = 183.95 \cdot \frac{W}{m^2}$ Solar Power (BOL)

$Lossrate_{Solar} := .99$

$P_{EOL} := .99^{Life} - 1.3(1 - .99^{Life}) = 0.887$ EOL Power (Margin)

$T_{Solar} := \left(\frac{110}{180}\right) \cdot 14 \cdot 24 \cdot hr = 205.333 \cdot hr$ Available Charging time (30* Solar Angle)

$t_{RAin1} := \frac{HD_{1size}}{Data_{in} \cdot A_{RA}} = 7.542 \cdot min$ GIVEN VARIABLES, TIME TO FILL Hard Drive (SINGLE ANTENNA)

$t_{RAin2} := \frac{HD_{2size}}{Data_{in} \cdot A_{RA}} = 15.083 \cdot min$ GIVEN VARIABLES, TIME TO FILL Hard Drive (DUAL ANTENNA)

$t_{RAinX} := \frac{HD_{Xsize}}{Data_{in} \cdot A_{RA}} = 15.083 \cdot min$ GIVEN VARIABLES, TIME TO FILL Hard Drive (X ANTENNA)

$Pcons_{thermal} := P_{thermal} \cdot t_{night} = 9.204 \cdot kW \cdot hr$ Nighttime Thermal Power Use

$Pcons_{ant1} := P_{ant} \cdot 354 \cdot t_{orb} = 0.738 \cdot kW \cdot hr$ Nighttime Single Ant. Power Use

$Pcons_{ant2} := 2 \cdot P_{ant} \cdot 354 \cdot t_{orb} = 1.475 \cdot kW \cdot hr$ Nighttime Dual Ant. Power Use

$Pcons_{antX} := X \cdot P_{ant} \cdot 354 \cdot t_{orb} = 1.475 \cdot kW \cdot hr$ Nighttime Five Ant. Power Use

$Pcons_{RA1} := A_{RA} \cdot Draw_{RA} \cdot t_{RAin1} = 0.014 \cdot kW \cdot hr$ Nighttime RA Power Use (1 ANTENNA)

$Pcons_{RA2} := A_{RA} \cdot Draw_{RA} \cdot t_{RAin2} = 0.029 \cdot kW \cdot hr$ Nighttime RA Power Use (2 ANTENNA)

$Pcons_{RAX} := A_{RA} \cdot Draw_{RA} \cdot t_{RAinX} = 0.029 \cdot kW \cdot hr$ Nighttime RA Power Use (5 ANTENNA)

$P_{1night} := Mass_{Batt1} \cdot \frac{Batt_{\rho}}{1.3} = 46.154 \cdot kW \cdot hr$ Battery Power Avail (Marginal) 1 Antenna

$P_{2night} := Mass_{Batt2} \cdot \frac{Batt_{\rho}}{1.3} = 44.984 \cdot kW \cdot hr$ Battery Power Avail (Marginal) 2 Antenna

$P_{Xnight} := Mass_{BattX} \cdot \frac{Batt_{\rho}}{1.3} = 45.254 \cdot kW \cdot hr$ Battery Power Avail (Marginal) X Antenna

$P_{HS} := 10W$

$P_{consHS} := P_{HS} \cdot t_{night}$

High School Power Use

$Pcons_{har} := P_{harness} \cdot t_{night} = 0.382 \cdot kW \cdot hr$

Nighttime harness loss

$Pcons_{harddrive} := P_{Harddrive} \cdot t_{night} = 1.274 \times 10^{7}\, J$

$P_{1ant} := 1.3\left(Pcons_{thermal} + Pcons_{ant1} + Pcons_{RA1} + Pcons_{har} + P_{consHS} + Pcons_{harddrive}\right)$

$P_{1ant} = 22.644 \cdot kW \cdot hr$

Night Power: 1 Antenna

$P_{2ant} := Pcons_{thermal} + Pcons_{ant2} + Pcons_{RA2} + Pcons_{har} + P_{consHS} + Pcons_{harddrive}$

$P_{2ant} = 18.17 \cdot kW \cdot hr$

Night Power: 2 Antenna

$P_{Xant} := Pcons_{thermal} + Pcons_{antX} + Pcons_{RAX} + Pcons_{har} + P_{consHS} + Pcons_{harddrive}$

$P_{Xant} = 18.17 \cdot kW \cdot hr$

Night Power: X Antenna

$RemainingCharge_1 := \frac{\left(P_{1night} - P_{1ant}\right)}{P_{1night}} \cdot 100 = 50.939$

% Power Remaining (Single)

$RemainingCharge_2 := \frac{\left(P_{2night} - P_{2ant}\right)}{P_{2night}} \cdot 100 = 59.607$

% Power Remaining (Dual)

$RemainingCharge_X := \frac{\left(P_{Xnight} - P_{Xant}\right)}{P_{Xnight}} \cdot 100 = 59.848$

% Power Remaining (Five Antennae)

Battery Charge Rate

$C_r := \frac{P_{2ant}}{t_{day}} = 54.078\, W$

Battery Charge Rate

Approximate Daytime Draw

$$P_{ConstantDaytime} := C_r + 2 \cdot P_{ant} + P_{thermal} + P_{HS} + P_{harness} + Draw_{RA} \cdot A_{RA} = 226.158\,W$$

Daylight Operations:

$$P_{Daytime} := 1.3Pcons_{thermal} + 1.3Pcons_{antX} + 1.3Pcons_{RAX} + C_r \cdot t_{day} + 1.3Pcons_{har} + 1.3P_{consHS} + 1.3 \cdot P_{Xnight} = 96.02 \cdot kW \cdot$$

$$A_{min} := \frac{P_{Daytime}}{T_{Solar} \cdot P_{solar}} = 2.542\,m^2$$

$$ShadeFactor := 2$$

$$A_{Solar} := 1.3ShadeFactor \cdot A_{min} = 6.61\,m^2$$

$$M_{solarmin} := A_{Solar} \cdot \rho_{solar} = 5.552\,kg$$

$$M_{solaroptimal} := A_{optimal} \cdot \rho_{solar} = 12.6\,kg$$

Battery Charge Rate

$$C_r := \frac{P_{2ant}}{t_{day}} = 54.078\,W$$

$$P_{ConstantDaytime} := C_r + 2 \cdot P_{ant} + P_{thermal} + P_{HS} + P_{harness} + Draw_{RA} \cdot A_{RA} = 226.158\,W$$

$Mass_{Max} := 250kg$ Mass Battery Max

$Batt_{\rho} := 350W \cdot \frac{hr}{kg}$ Ideal Battery Power Density

$P_{max} := Mass_{Max} \cdot Batt_{\rho} = 87.5 \cdot kW \cdot hr$ Ideal Battery Power

Power Available for RA: Ideal Situation

$$P_{avail} := .6P_{max} - Pcons_{thermal} - Pcons_{ant2} - P_{consHS} - Pcons_{har} = 37.899 \cdot kW \cdot hr$$

$t_{avail} := \frac{P_{avail}}{Draw_{RA} \cdot A_{RA}} = 13.731 \cdot day$ RA Operational Time: Ideal Situation

$\frac{350}{220} = 1.591$ Ideal % Increase

Monopropellant Propulsion system calculations (regulated pressure fed)

Contents:
Propellant Inventory
Main engines required for landing
Attitude Control System (ACS)
Propellant Tank Volume
Propellant Tank Mass
Pressurant Mass
Pressurant tanks
Total Mono-propellant Sub-System Mass
Notes, comments, problems
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****

Propellant Inventory

Propellant Use

MR80-B
ACS (5%)
Reserves (30%)
Subtotal - Usable
Residuals (3% usable)
Loading Uncertainty (0.5% usable)
Loaded Propellant
Propellant Masses

$mp_{mr80b} := 30kg + 54kg + 13kg = 97\,kg$ Need to reference these values

$mp_{acs} := .05 \cdot mp_{mr80b} = 4.85\,kg$

$reserves := \left(mp_{mr80b} + mp_{acs}\right) \cdot .3 = 30.555\,kg$

$mp_{usable} := mp_{mr80b} + mp_{acs} + reserves = 132.405\,kg$

$residuals := mp_{usable} \cdot .03 = 3.972\,kg$

$uncertainty := mp_{usable} \cdot .005 = 0.662\,kg$

$mp_{loaded} := mp_{usable} + residuals + uncertainty = 137.039\,kg$

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Main Engines required for Landing

$m_{lander} := 1361.543kg$

$g = 9.807 \dfrac{m}{s^2}$ Gravity on Earth

$W_{lander} := m_{lander} \cdot g = 13352.176\,N$

Initial Thrust to weight ratio of .4

$T := W_{lander} \cdot 0.4$

$T = 5340.87\,N$ $T = 1200.675 \cdot lbf$

MR80-B is capable of generating 716 lbf of thrust (maximum). Divide required thrust by 716 to determine the number of engines.

$$Engines := \frac{T}{716lbf} = 1.677$$

This mission will require 2 MR80-B's for landing on the moon, but use 3 for stability.

$m_{engine} := 6.35kg$ Masses found in Aerojet catalog on Angel

$m_{valve} := 1.59kg$

$m_{mr80b} := \left(m_{engine} + m_{valve}\right)$

$m_{mr80b} = 7.94\,kg$ Remember there are 3 engines for total masses calcuations

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ACS (Attitude Control System)

ACS will consists of four sets of four thrusters to perform spacecraft manuevers and control.
System will also perform thrust vectoring during SRM burns.

$m_{mr106l} := .59kg$ 5 lbf thrusts, 3x4.

$m_{mr120} := .41kg$ 20 lbf thrusters, 1x4.

ACS system will require 4 sets of 4 direction rocket engine modules

$m_{acs} := 12 \cdot m_{mr106l} + 4 \cdot m_{mr120} = 8.72\,kg$

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Propellant Tank Volume

Diaphragm Tank

Assume maximum temperature of 30C (tanks will be insulated). Need to find a source for a typical value.

$Temp := 30\,°C$

$$\rho_{hydra} := \left[1025.817 - \frac{0.8742(Temp)}{1\,°C} - \frac{.0005\,(Temp)^2}{(1\,°C)^2}\right] \cdot \frac{kg}{m^3}$$

page 199 'Elements of Spacecraft Design' by Brown.

$$\rho_{hydra} = 1024.85\,\frac{kg}{m^3}$$

$m_p := mp_{loaded}$

$m_p = 137.039\,kg$ Mass of the Hydrazine loaded onto the spacecraft

$V_p := \frac{m_p}{\rho_{hydra}} = 0.134 \cdot m^3$ Hydrazine volume

$mp_{usable} = 132.405\,kg$ Mass of the usable propellant

$V_u := \frac{mp_{usable}}{\rho_{hydra}} = 0.129 \cdot m^3$ Volume of usable propellant

$B := 5$ Blow down ratio (Initial guess) The maximum blowdown ratio is determined by the inlet pressure range the engines can accept.

$V_{gi} := \frac{V_u}{B - 1} = 0.032 \cdot m^3$ Initial Ullage volume (Volume that the pressurant occupies above the propellant)

$$r_b := \left[\frac{.75 \cdot \left(V_p + V_{gi}\right)}{\pi}\right]^{\frac{1}{3}}$$

$r_b = 34.095 \cdot cm$ Internal radius for the diaphram (approximation) pg 194 Brown.

$A_b := 2 \cdot \pi \cdot \left(r_b\right)^2 = 7303.858 \cdot cm^2$ Area of the diaphram

The diaphram thickness can be expected to be about 0.20 cm (pg 194 Brown) therefor

$V_b := 0.2cm \cdot A_b = 0.001 \cdot m^3$

$\boxed{V_{total} := V_p + V_{gi} + V_b = 0.167 \cdot m^3}$ Approximate total volume needed for propellant tanks

We need two tanks... need to find reference to confirm this.

$V_{tanks} := \frac{V_{total}}{2} = 0.084 \cdot m^3$ Internal Volume required for each propellant tank

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Propellant Tank Mass (Assuming Spherical tank design, titanium material)

Process taken from example 4.8 on page 196 of Brown, Elements of Spacecraft Design.

Assumptions:

$\sigma := 690000kPa$ Allowable stress

$P_{max} := 4653.96kPa$ Max working pressure

$\rho_{titanium} := 4429.89 \frac{kg}{m^3}$ Assumed material density for titanium (tank material)

$r_m := \left(\frac{0.75 \cdot V_{tanks}}{\pi}\right)^{\frac{1}{3}} = 0.271 \, m$ Membrane Thickness

$thickness := \frac{P_{max} \cdot r_m}{2 \cdot \sigma} = 0.092 \cdot cm$ Minimum acceptable thickness

$t_{min} := thickness + .002cm = 0.094 \cdot cm$ A .002 cm tolerance is added

$R_o := r_m + t_{min} = 27.234 \cdot cm$ Outside tank radius

$W_{mem} := \frac{4}{3} \cdot \pi \cdot \rho_{titanium} \cdot \left(R_o^{\ 3} - r_m^{\ 3}\right)$

$W_{mem} = 3.848\,kg$ Mass of the membrane

Estimated calculations for the reinforced areas on the tank which include girth weld land, penetration land, and structural attachments.

$W_{girth} := 2 \cdot \pi \cdot R_o \cdot t_{min} \cdot 2 \cdot 5cm \cdot \rho_{titanium}$ Girth weld land mass

$W_{girth} = 0.709\,kg$

Penetration land weight assumes two 15-cm diameter disks

$W_{pen} := 2 \cdot \pi \cdot (7.5cm)^2 \cdot t_{min} \cdot \rho_{titanium} = 0.146\,kg$ Penetration land weight

Structural attachment weight should be roughly 2% of the supported weight

The tank shell weight is a combination of the membrane, girth land, penetrations, and structural attachment weight.

$m_{tank} := 1.02\left(W_{mem} + W_{girth} + W_{pen}\right) = 4.798\,kg$ Mass for 1 tank

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Initial Ullage Pressurant mass

Assume that the system will use Helium

Procedure taken from page 199, Brown.

$V_{gi} = 0.032 \cdot m^3$ Ullage volume, solved in the tank volume section above.

$R_{helium} := 2078.5 \frac{joule}{kg \cdot K}$ Specific gas constant for helium

Assumptions:

$P_{tank} := 3619750\text{Pa}$ tank pressure

$T_{tank} := 290\text{K}$ internal tank temperature

$$m_{pp} := \frac{P_{tank} \cdot V_{gi}}{R_{helium} \cdot T_{tank}} = 0.194\,\text{kg}$$ Mass of pressurant initially in propellant tanks

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Pressurant spheres sizing for regulated system

Followed example 4.10 from page 217 of Brown's Elements of Spacecraft Design

Helium pressurant is used

$P_r := 3551\text{kPa}$ Regulated propellant tank pressure

$V_u = 0.129 \cdot \text{m}^3$ Volume of useable propellant

$P_1 := 33095\text{kPa}$ Initial pressurant sphere pressure

$P_2 := 3965\text{kPa}$ Final pressurant sphere pressure

$$V_s := \frac{P_r \cdot V_u}{P_1 - P_2} = 0.016 \cdot \text{m}^3$$ Volume of pressurant for pressurant sphere $V_s = 961.066 \cdot \text{in}^3$

$$m_{ps} := \frac{P_1 \cdot V_s}{R_{helium} \cdot T_{tank}} = 0.865\,\text{kg}$$ Mass of pressurant in pressurant tanks

Mass of pressurant sphere

Process taken from example 4.8 on page 196 of Brown, Elements of Spacecraft Design.

Assumptions:

$\sigma := 690000\text{kPa}$ Allowable stress

$P_{max} := 4653.96\text{kPa}$ Max working pressure

$\rho_{titanium} := 4429.89 \frac{\text{kg}}{\text{m}^3}$ Assumed material density for titanium (tank material)

$$r_m := \left(\frac{0.75 \cdot V_s}{\pi}\right)^{\frac{1}{3}} = 0.155\,\text{m}$$ Membrane Thickness

$$thickness := \frac{P_{max} \cdot r_m}{2 \cdot \sigma} = 0.052 \cdot \text{cm}$$ Minimum acceptable thickness

$t_{min} := thickness + .002\text{cm} = 0.054 \cdot \text{cm}$ A .002 cm tolerance is added

$R_o := r_m + t_{min} = 15.604 \cdot cm$ Outside tank radius

$W_{mem} := \frac{4}{3} \cdot \pi \cdot \rho_{titanium} \cdot \left(R_o^{\ 3} - r_m^{\ 3}\right)$

$W_{mem} = 0.735\,kg$ Mass of the membrane

Estimated calculations for the reinforced areas on the tank which include girth weld land, penetration land, and structural attachments.

$W_{girth} := 2 \cdot \pi \cdot R_o \cdot t_{min} \cdot 2 \cdot 5cm \cdot \rho_{titanium}$ Girth weld land mass

$W_{girth} = 0.236\,kg$

Penetration land weight assumes two 15-cm diameter disks

$W_{pen} := 2 \cdot \pi \cdot (7.5cm)^2 \cdot t_{min} \cdot \rho_{titanium} = 0.085\,kg$ Penetration land weight

Structural attachment weight should be roughly 2% of the supported weight

The tank shell weight is a combination of the membrane, girth land, penetrations, and structural attachment weight.

$\boxed{m_{presstank} := 1.02\left(W_{mem} + W_{girth} + W_{pen}\right) = 1.078\,kg}$

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Total Mono-Propropellant Sub-system Mass

The total mass for the propulsion sub-system will include the propellant, pressurant, pressurant spheres, main thrusters, ACS, propellant tanks, and associated lines, valves, and hardware needed for installation.

$m_p = 137.039\,kg$ Propellant mass includes residuals, reserve, ect.

$m_{pressurant} := m_{pp} + m_{ps} = 1.059\,kg$ Pressurant is helium

$m_{thrusters} := 3 \cdot m_{mr80b} = 23.82\,kg$ There are 3 main thrusters

$m_{acs} = 8.72\,kg$ Includes the mass of 4x4 modules

$m_{protank} := 2m_{tank} = 9.595\,kg$ Propellant tank mass

$m_{presstank} = 1.078\,kg$ Pressurant tank mass

The mass of the associated lines, valves, and installation hardware is estimated to be 20% of the total subsystem mass. *Need a reference for this assumption*.

$$Mass_{prop\_system} := 1.20\cdot\left(m_p + m_{pressurant} + m_{thrusters} + m_{acs} + m_{protank} + 2m_{presstank}\right)$$

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$$Mass_{prop\_system} = 218.868\,kg$$

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Thrust vector misalignment verification

$Dia := 180in$ — Shroud Diameter

$Radius := \frac{Dia}{2} = 2.286\,m$ — Max radius of spacecraft in shroud

$A\cdot Te\cdot Loff = Tc\cdot Lc$

$A := 1$

$Loff := .1in$

$Te := 16000lbf$

$Lc := 85in$

A = a constant depending on what you want to do (actually "control" or overtake the burn, or just mitigate any thrust vector off-sets)
Te = solid motor thrust level
Loff = the thrust vector offset in the solid motor
Tc = thrust level of the "control" (ACS) thruster
Lc = the moment arm of the control thruster

$$Tc := \frac{A\cdot Te\cdot Loff}{Lc} = 18.824\cdot lbf$$

System requires atleast 19 lbf thrusters to correct for any thrust vector misalignment during SRM burns.

LOI SRM

%2			Offload
		48B (short)	SRM#
kg	2097.7		Total
kg	124.35		Inert
kg	1973.35		Propellant

Braking

%	20		Offload
		48B (short)	SRM#
kg	1733.86		Total
kg	124.35		Inert
kg	1609.51		Propellant

Useful Mass

kg	610.9101	Payload
kg	1433.176	Mass

Divert

kg	279.7089		Lander Wet
kg	15.01342	Mp	
kg	1712.885	Mo	

Approach

kg	62.64596	Mp
kg	1810.487	Mo

Braking

kg	1734.68	Mwet
kg	1610.33	Mp
kg	3602.524	Mo

DOI

kg	34.22418	Mp
kg	3710.269	Mo

Correction

kg	127	Orbiter Dry
kg	284.5	Orbiter Wet
kg	38.90875	Mp
kg	4090.897	Mo

LOI

kg	2097.7	Mwet
kg	1973.35	Mp
kg	6292.493	Mo

MCC

kg	123.0887	Mp
kg	6524	Mo

Gravity

m/s^2	9.81	

Isp

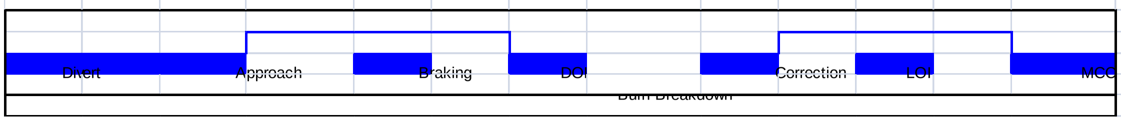
s	220	Lar
s	290	Soli
s	320	Orb

PMF

0.4	Lar
0.7	Orb
0.9	SRM

Delta V

m/s	19	Div
m/s	76	A
m/s	1699	Br
m/s	20	DOI
m/s	30	C
m/s	1080	LOI
m/s	50	MCC



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:

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