April 25, 2011

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Eureka
Europa Exploration

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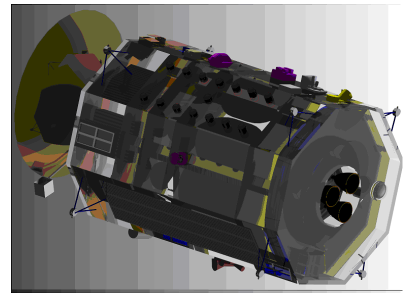

<u>TEAM EUREKA</u>
Europa Exploration
Monday, April 25, 201

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Mission: June 7th, 2010 - November 29th 2029

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

- Certifies that the statements made in this proposal are true and complete to the best of his/her knowledge;
- Agrees to accept the obligations to comply with NASA award terms and conditions if an award is made as a result of this proposal; and
- Confirms compliance with all provisions, rules, and stipulations set forth in the 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 provision of false information in this proposal and/or its supporting documents, or in reports required under an ensuing award, is a criminal offense (U.S. Code, Title 18, Section 1001).

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Proposal Summary Information

"Eureka" is an expression used when a person has discovered something significant or accomplished a great achievement. Europa has an abundant amount of opportunities for new discoveries. There is not much known about this moon of Jupiter. Most of the information about Europa is based upon assumptions. In collaboration with scientists and engineers, Team Eureka is designing a mission that will turn these assumptions into facts. This mission will search for traces of past or present life in a habitable environment; study the structure and composition of the surface, near-surface, and interior; and investigate the geologic activities that encompass Europa, and the processes that drive it. To achieve this, Team Eureka will analyze alternatives for launch vehicle, orbital path, and science objectives based upon the Figures of Merit for each individual part of the mission and decide which alternative is the best fit for the mission.

Proposal Summary Question

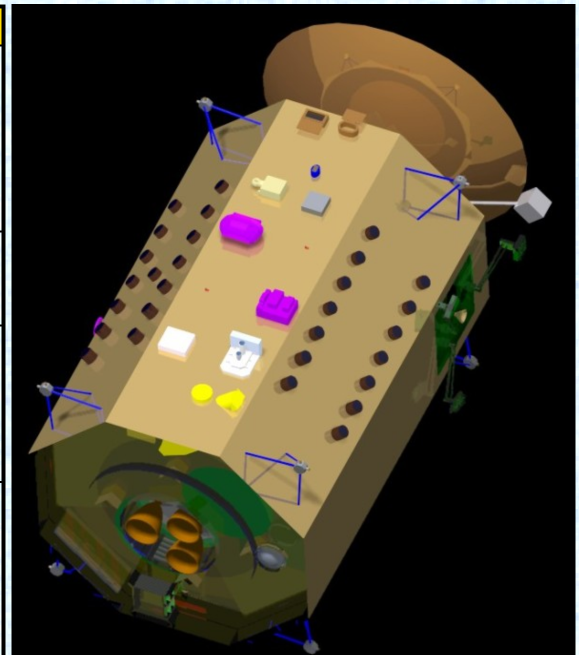
- Is proprietary/privileged information included in this application? –
    - Answer: No;
- Does this project involve activities outside the U.S. or partnership with non-U.S. collaborators?
    - Answer: Yes; ESTCA is located in Paris France
- Are NASA civil servant personnel participating as team members on this project (include funded and unfunded)?
    - Answer: No
- Does this project have an actual or potential impact on the environment?
    - Answer: Yes;
- Has an exemption been authorized on an environmental assessment (EA) or an environmental impact statement (EIS) been performed?
    - Answer: No
- Does this project have the potential to affect historic, archeological, or traditional cultural sites (such as Native American burial or ceremonial grounds) or historic objects (such as an historic aircraft or spacecraft)?
    - Answer: No
- Identify target of investigation
    - Answer: Jupiter Europa Orbiter (JEO)
- Which launch vehicle performance class is proposed?
    - Answer: Standard with 5m fairing
- Is the use of NEXT proposed?
    - Answer: No
- Is the use of AMBR proposed?
    - Answer: Yes
- Is the use of aerocapture proposed?
    - Answer: No
- Is the use of ASRG proposed?
    - Answer: Yes
- Is use of radioisotope heater units, or radioactive material sources for science instruments proposed?
    - Answer: No
- Is a student collaboration (SC) proposed?
    - Answer: Yes
- Is a science enhancement option (SEO) proposed?
    - Answer: Yes
- Total Mission Cost in real year dollars (RY$) and in FY 2010 dollars
    - Answer: 1,169M

- This proposal contains information and/or data that are subject to U.S. export control laws and regulations, including Export Administration Regulations (EAR) and International Traffic in Arms Regulations (ITAR).
    - Answer: No
- The proposer acknowledges that the inclusion of such material in this proposal may complicate the Government's ability to evaluate the proposal.
    - Answer: Yes
- Statement of contributions to development or operations (but not science) by any non-U.S. partner. Identify the non-U.S. partner(s), the non-U.S. funding agency/agencies, and the approximate value of the non-U.S. contributions, if any;

Fact Sheet: The ICESSS Mission

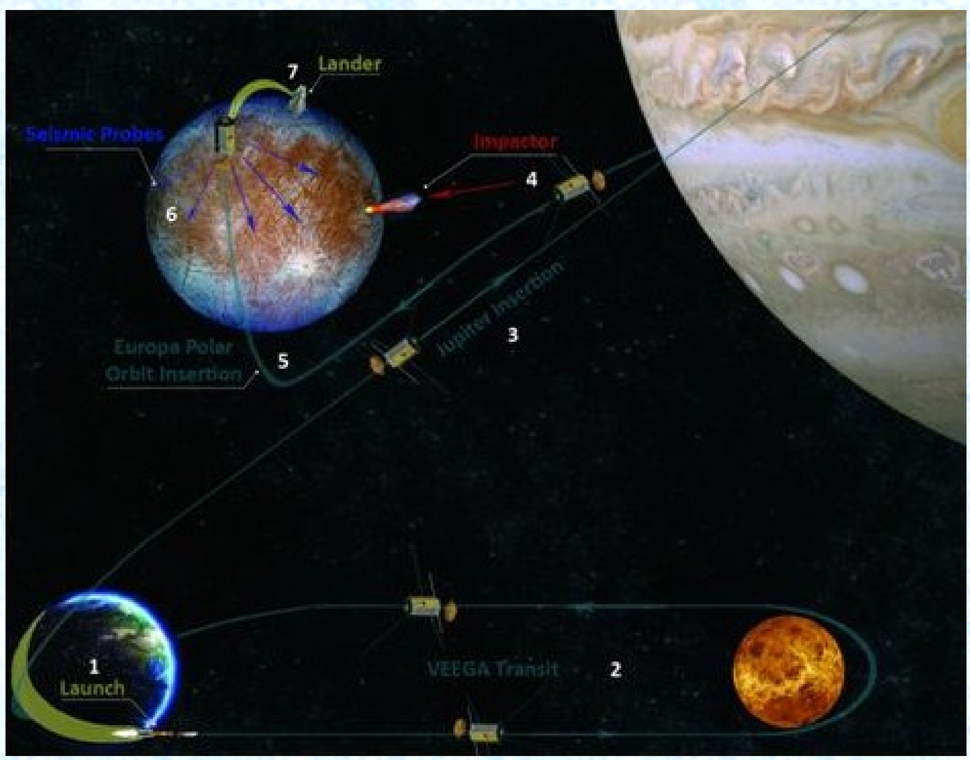
Inner **C**rustal **E**uropa **S**eismic and **S**pectral **S**urveyor

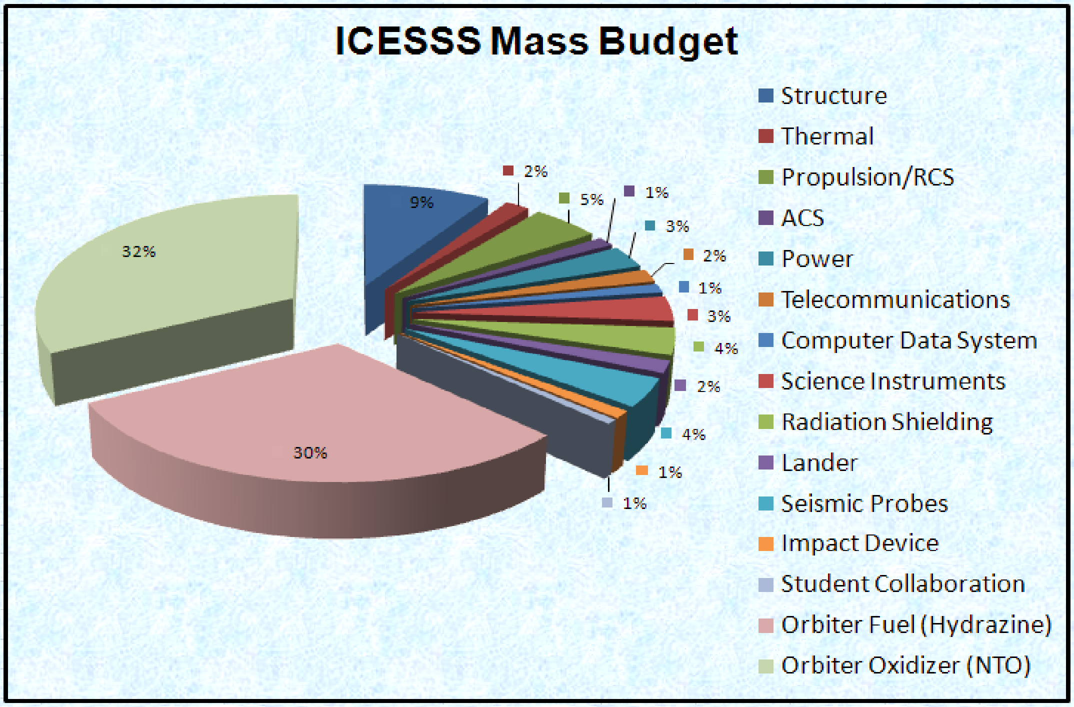
Science Objective	Instruments
Characterize Europa's icy shell and any subsurface water as well as the nature of the surface-ice-ocean exchange Characterize and determine the extent of subsurface oceans and their relations to the deeper interior	Ice Penetrating Radar Seismic Probes
Characterize the deep internal structure, differentiation history and intrinsic magnetic field	Magnetometers Radio Science
Compare the exospheres, plasma environments and magnetospheric interactions	Radio and Plasma Wave Science Instrument Thermal Emission Imaging System Ion and Neutral Mass Spectrometer Magnetospheric Imaging System
Determine global surface compositions and chemistry, especially related to habitability	Near Infrared Mapping Spectrometer UV Spectrometer Seismic Probes Ion Neutral Mass Spectrometer Lander with Gas Chromatograph and Mass Spectrometer
Understand the formation of surface features, including sites of recent or current activity and identify and characterize candidate sites for future *in situ* exploration	Narrow angle camera Wide angle camera Near Infrared Mapping Spectrometer



Launch Vehicle
Atlas V 551 with 4615kg capacity
Dry Mass
2006 kg (including 29% contingency)
Propellant Mass
1168kg of Hydrazine and 1252kg of NTO
Total Wet Mass
4492kg with 123kg of additional margin

Telecommunications
Frequencies
Ka-band and X-band
Antennas
3 meter parabolic high gain antenna
1 medium gain and 2 low gain antennae
Power Requirement
85 W

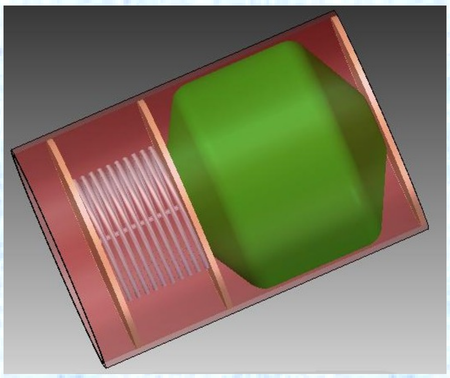




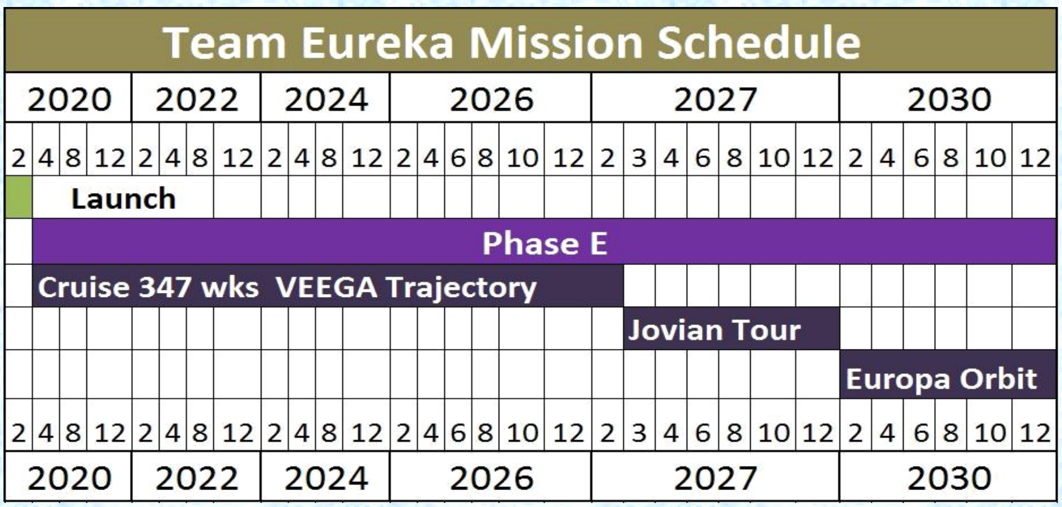
Propulsion System
ΔV Requirement
2260m/s
System
Hydrazine and Nitrogen Tetroxide
Dual mode system utilizing AMBR Engine
335s Isp for primary maneuvers
210s Isp for attitude control maneuvers

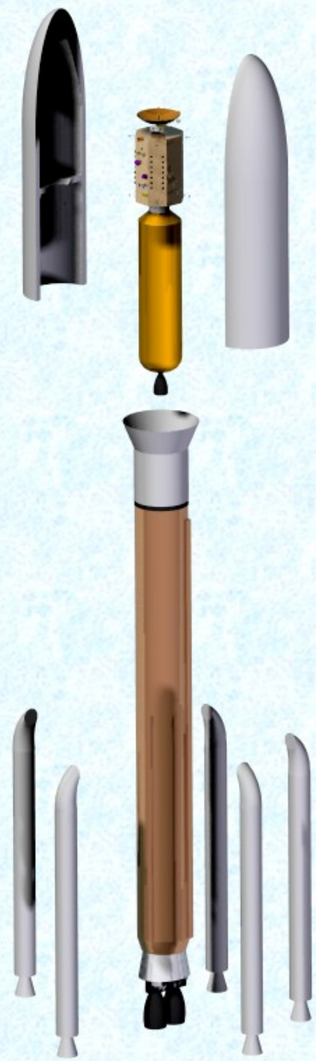
Power Budget (in Watts)

System	Standard Operations	Telecom Operations	Safe Mode
Scientific Payload	125	90	0
Thermal	0	0	0
ACS	49	49	49
CDS	51	51	51
Communications	50	85	85
Propulsion	5	5	5
Total Power (W)	280	280	190



Seismic Probes
Scientific Studies
Quantify tidal flexing
Determine fundamental surface composition
Hydrazine and Nitrogen Tetroxide
Ejection System
29 separate ejection canisters
Spring ejects probe with 50lbs of force
Release actuated by thermal knife





Cost Allocation – Baseline Mission			
Fixed Cost Element	**Cost (FY2010$)**	**Variable Cost Element**	**Cost (FY2010$)**
Launch Vehicle	$68M	Orbiter	$773M
Additional ASRG	$27M	Lander	$134M
NEPA compliance	$20M		
Total	$115M	Total	$907M
Total Mission Cost: $1022M			

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D. Science Investigation

D.1 Scientific Background, Goals, Objectives

The primary objectives of this mission involve studying the geology and composition of Europa. The mission is designed to directly sample the surface of Europa and determine its chemical composition and density. The geological structure and history of Europa, especially in regard to the tidal flexing of the moon caused by gravitational interactions with Jupiter and other Galilean moons, is studied. Additionally, the exosphere, plasma environment, and magnetospheric interactions of the moon is analyzed.

D.2 Baseline

A baseline Inner Crustal Europa Seismic and Spectral Surveyor (ICESSS) mission will conduct a polar orbit around Europa. Before inserting into Europan orbit, an impact device is launched from the orbiter in order to generate surface renewal and a plume of material. The melted ice and dust plume is analyzed by the spectroscopy instruments onboard the orbiter. The data these instruments yield will enable us to determine subsurface chemistries, as well as potentially yielding the composition of the subsurface ocean, and will enable us to determine the habitability of Europa. Additionally, a near infrared mapping spectrometer, micro-imager, and ultraviolet spectrometer onboard the orbiter will provide surface composition, including the composition of potential organic and non-ice non-organic materials.

During the course of ICESSS's orbit, 29 impact probes is deployed to the surface of the planet. The probes will each house a seismometer and thermal instrument, and upon impact, the seismometer will obtain data on the possible seismic activity of Europa's surface, and map its location. This data is used to characterize the tidal deformation Europa experiences. The Europa Orbiter Radar Sounder onboard the orbiter will constantly be used during the first week of ICESSS's polar orbit around Europa. The seismic probes and radar sounder instrument is used to characterize the icy shell and any subsurface water as well as the nature of the surface-ice-ocean exchange and to characterize and determine the extent of subsurface oceans and their relations to the deeper interior. Additionally, the Thermal Emission Imaging System (THEMIS) is used to characterize sub-surface and surface thermal patterns to study internal processes and surface-ocean interactions.

Once the impact probes are no longer transmitting data due to battery termination, a lander, designated R^2D^2 is dropped to Europa's surface. The lander will house a microscopic imager to take high resolution pictures of the surface in order to determine surface characteristics and composition. A gas chromatograph/mass spectrometer onboard the lander will analyze the composition of the atmosphere of Europa, determine noble gas abundance, isotopic ratios, and analyze organic material if present with ice samples prepared by the ice abrasion tool. Two fluxgate magnetometers onboard the lander will take magnetic field measurements in order to study the overall configuration and dynamics of the magnetosphere at the surface level.

To understand the formation of surface features, including sites of recent or current activity and identify and characterize candidate sites for future *in situ* exploration, the Narrow Angle Camera/Wide Angle Camera and Light Detection and Ranging (LIDAR) is utilized. The NAC/WAC is used to attain high resolution images of surface features to yield insight into their formation, as well as potential landing sites. LIDAR is used to map the topography of Europa's surface to yield potential landing sites.

To compare the exospheres, plasma environments and magnetospheric interactions of Europa, a radio and plasma wave science instrument, thermal emission spectrometer, radio science instrument, ion and neutral mass spectrometer, and a magnetospheric imaging system onboard the orbiter is used throughout the polar orbit of Europa. Additionally, two fluxgate magnetometers onboard the lander will generate magnetic field measurements at the surface in order to characterize the magnetic field at the surface and subsurface ocean characteristics.

To characterize the deep internal structure, the differentiation history and intrinsic magnetic field, the Radio Science Subsystem, Radio and Plasma Wave Science System, Magnetospheric Imaging System, and the Europa Radar Sounder onboard the orbiter is used throughout the polar orbit of Europa.

The data obtained by the ICESSS mission will provide us with a greater understanding of the characteristics of Europa's surface and subsurface ocean composition and their interactions with each other, the exospheres and plasma environments, the current seismic activity of Europa's surface and renewal, the processes that drive the observed geologic features, future landing sites, and the prospect of life below the icy surface.

D.3 Threshold

A threshold ICESSS mission will conduct a polar orbit around Europa. Prior to inserting into Europan orbit, an impact device is launched from the orbiter in order to generate surface renewal and a plume of material. The melted ice and dust plume is analyzed by the spectroscopy instruments onboard the orbiter. The data these instruments yield will enable us to determine subsurface chemistries including the composition of potential organic and non-ice non-organic materials, as well as potentially yielding the composition of the subsurface ocean, and will enable us to determine the habitability of Europa.
During the course of ICESSS's orbit, 29 impact probes is deployed to the surface of the planet. The probes will each house a seismometer and thermal instrument, and upon impact, the seismometer will obtain data on the possible seismic activity of Europa's surface, and map its location. This data is used to characterize the tidal deformation Europa experiences. The Europa Orbiter Radar Sounder onboard the orbiter is used during the first week of ICESSS's polar orbit around Europa. These two instruments is used to characterize the icy shell and any subsurface water as well as the nature of the surface-ice-ocean exchange and to characterize and determine the extent of subsurface oceans and their relations to the deeper interior. Additionally, THEMIS is used to characterize sub-surface and surface thermal patterns to study internal processes and surface-ocean interactions.

To understand the formation of surface features, including sites of recent or current activity and identify and characterize candidate sites for future *in situ* exploration, the Narrow Angle Camera/Wide Angle Camera and LIDAR is utilized. The NAC/WAC is used to obtain high resolution images of surface features to yield insight into their formation, as well as potential landing sites. LIDAR is used to map the topography of Europa's surface to yield potential landing sites.
To compare the exospheres, plasma environments and magnetospheric interactions of Europa, a radio and plasma wave science instrument, thermal emission spectrometer, radio science instrument, Ion and neutral mass spectrometer, and a magnetospheric imaging system onboard the orbiter is used throughout the polar orbit of Europa.

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The data obtained by the ICESSS mission will provide us with a greater understanding of the characteristics of Europa's surface and subsurface ocean composition and their interactions with each other, the exospheres and plasma environments, the current seismic activity of Europa's surface and renewal, the processes that drive the observed geologic features, future landing sites, and the prospect of life below the icy surface.

D.2 Science Requirements

The science requirements traceability matrix, shown in Table 1, shows the logical decomposition required to derive a list of scientific instruments and requirements from the desired scientific investigations described in section D.1 above. Table 2 illustrates the scientific investigations each scientific instrument enables. As shown, many of the scientific instruments serve to gather data useful for multiple scientific investigations.

As illustrated by the science requirements traceability matrix, the mission's first goal is to characterize Europa's ocean. The first objective under this goal is to determine definitively if an ocean exists, and characterize its physical parameters. The scientific requirements for this objective are as defined as follows: the observable parameter is thermal profiles; the physical parameters are global surface measurements; and instrumentation is the thermal imager. The mission functional requirements are consistent pointing accuracy of areas to allow for global surface mapping and spectral imaging.

The second objective is to characterize the surface-ocean interaction. Scientific measurement requirements are as follows: the observable parameters are thermal profiles and topographical changes; the physical parameters are global surface measurement; and the instrumentation includes a thermal imager and a high resolution camera. The mission functional requirement is consistent pointing accuracy of areas to allow for global surface mapping and spectral imaging.

The third objective is to determine how the ocean interacts with the silicate interior. The science measurement requirements are as follows: the observable parameters are topographical changes and heat flow patterns, the physical parameters are global surface measurement; and the instrumentation is the thermal imager and high resolution camera. The mission functional requirement is consistent pointing accuracy of areas to allow for global surface mapping and spectral imaging.

The forth objective is to determine the composition of the ocean and of the icy surface. The scientific measurement requirements are as follows: the observable parameters are spectral emissions and absorption lines; the physical parameters are surface reflectance spectral measurements; and the instrumentation is spectroscopy suite IR - EUV. The mission functional requirements are substantial irradiance of the surface to allow for adequate light collection from the sun.

ICESSS's second goal is to characterize Europa's ice shell and surface features. The first objective under this goal is to determine thickness of the ice shell, the uniformity of that thickness, and the distribution of liquid water in the shell. The scientific measurement requirements are as follows: the observable parameters are radar profiles/gravity profiles/density profiles; the physical parameters are dopler shifts in rebounding of signals; and the instrumentation is the ice penetrating radar, gravitometer, seismic probes. The mission functional requirements are global viewing to allow for measuring of entire surface.

The second objective is to determine how surface features form, and if there has been a change in formation mechanism over time. The scientific measurement requirements are as follows: the observable parameters are changes in morphological features and heat flow/thermal profiles; the physical parameters are observing the temporal evolution of the surface features in tidal cycle; and the instrumentation is thermal imagers. The mission functional requirements are viewing over an extended time span.

The third objective is to determine the surface age of Europa and the ages of its various surface features and regions. The scientific measurement requirements are as follows: the observable parameters are collected samples and high resolution images; the physical parameters are surface sample and geologic features such as ridges; and the instrumentation is the lander instruments and imaging suite. The mission functional requirements are landing on the surface and transmitting data back to the obiter.

The forth objective is to characterize the global stratigraphy of Europa. The scientific measurement requirements are as follows: the observable parameters are topographical images and density profiles; the physical parameters are mountains, ridges, trenches, and other features on the surface; and the instrumentation is the imaging suite and geological analysis from the lander. The mission functional requirements are high pointing accuracy to allow for non-blurred images.

The third goal of the ICESSS mission is to investigate Europa's geologic activity. The first objective under this goal is to determine Europa's activity and where that activity is expressed. The scientific measurement requirements are as follows: the observable parameters are heat flow patterns; the physical parameters are hot spots on surface and morphological features; and the instrumentation is the thermal imager. The mission functional requirements are viewing over and extended time span and global viewing to allow for analysis of the entire surface.

The second objective is to identify the location of tidal dissipation eg. (ice shell vs. rocky mantle). The scientific measurement requirements are as follows: the observable parameters are heat flow patterns; the physical parameters are hot spots on surface and morphological features; and the instrumentation is the thermal imager. The mission functional requirements are viewing over an extended time span and global viewing to allow for analysis of the entire surface.

The third objective is to examine the heat flow through out Europa's interior and surface. The scientific measurement requirements are as follows: the observable parameters are thermal profiles, the physical parameters are global surface, and the instrumentation is the thermal imager. The mission functional requirements are viewing over an extended time span and global viewing to allow for analysis of the entire surface.

The forth objective is to determine the seismic activity in the shell. The scientific measurement requirements are as follows: the observable parameters are seismic data and visual changes in topography; the physical parameters are vibrations of the surface; and the instrumentation is the impact device and seismic probes with accelerometers. The mission functional requirements are communications with the seismic probes specifically distributed across the surface.

The fifth science objective is to determine to resurfacing rate and style of Europa and its uniformity or episodicity. The scientific measurement requirements are as follows: the observable parameters are changes in morphological and surface features; the physical parameters are areas of potential resurfacing eg. mountainous areas; and the instrumentation is the imaging suite. The mission functional requirements are viewing of an extended time span and global viewing to allow for analysis of the entire surface.

The sixth objective is to determine if there is any active cryvolcanism on Europa. The scientific measurement requirements are as follows: the observable parameters are high resolution images; the physical parameters are weak spots in the surface such as cracks or trenches, and the instrumentation is the imaging suite. The mission functional requirements are viewing over an extended time span and global viewing to allow for analysis of the entire surface.

The seventh objective is to compare the exospheres, plasma environments and magnetospheric interactions. The scientific measurement requirements are as follows: the observable parameters are composition of exosphere ion ratios, and magnetic field analysis; the physical parameters are different chemical species comprising each and magnetic field measurement fluctuations; and the instrumentation is the plasma analysis system, the neutral ion spectrometer, and the magnetometer. The mission functional requirements are irradiance of exosphere by sun where the exosphere is being illuminated by the sun and also direct passes through Jupiter's magnetosphere are required.

Table 1 Science Requirements Traceability Matrix

Scientific Goals	Scientific Objectives	Scientific Measurement Requirements			Mission Functional Requirements
		Observable	Physical Parameters	Instrumentation	
Characterize Europa's Ocean	Determine if there is definitively an ocean, and characterize its physical parameters	Thermal profiles	Global surface measurement	Thermal imager	Consistent pointing accuracy of areas to allow for global surface mapping and spectral imaging
	Characterize the surface-ocean interaction.	Thermal profiles/ topographical changes		Thermal imager	
	Determine how the ocean interacts with the silicate interior	Topographical changes / heat flow patterns		High Resolution camera	
	Determine the composition of the ocean and of the icy surface	Spectral emission and absorption lines	Surface reflectance spectral measurements	Spectroscopy suite IR-EUV	Substantial irradiance of surface to allow for adequate light collection from sun
Characterize Europa's Ice Shell and Surface Features	Determine thickness of the ice shell, uniformity of that thickness, and distribution of liquid water in the shell	Radar profiles/	Doppler shifts in rebounding of signals	Ice penetrating radar/ gravitometer/ nano-bots and their accelerometers	Global viewing to allow for measuring of entire surface
		Gravity profiles/density profiles			
	Determine how surface features form, and if there has been a change in formation mechanism over time	Changes in morphological features/ heat flow and thermal profiles	Looking at temporal evolution of surface features in tidal cycle	Thermal imager	Viewing over an extended time span
				Imaging suite	
	Determine the surface age of Europa and the ages of its various surface features and regions	Collected samples,	Surface sample	Lander instruments	Landing on surface and transmitting data back to orbiter
		high resolution images	Geologic features e.g. ridges	Imaging suite	
	Characterize the global stratigraphy of Europa	Topographical images and density profiles	Mountains, ridges, trenches and other features on surface	Imaging suite and geological analysis from lander	High Pointing accuracy to allow for non blurred images

Investigate Europa's Geologic Activity	Determine Europa's activity and where that activity is expressed.	Heat flow patterns	Hot spots on surface and morphological		Viewing over an extended time span Global viewing to allow for analysis of entire surface
	Identify the location of tidal dissipation eg.(ice shell vs. rocky mantle).	Heat flow patterns			
	Examine the heat flow throughout Europa's interior and surface.	Thermal profiles	Global surface	Thermal imager	
	Determine if there is seismic activity in the ice shell	Seismic data, visual changes in topography	Vibrations of surface	Penetrators and nano-bots accelerometers	Communications with nano-bots and penetrators specifically distributed across the surface
	Determine the resurfacing rate and style of Europa and its uniformity or episodicity	Changes in morphological and surface features	Areas of potential resurfacing e.g. mountainous areas	Imaging suite	Viewing over an extended time span Global viewing to allow for analysis of entire surface
	Determine if there is any active cryovolcanism on Europa	High resolution images	Weak spots in surface {cracks ,trenches}		
	Compare the exospheres, plasma environments and magnetospheric interactions	Composition of exosphere, ion ratios, and magnetic field analysis	Different chemical species comprising each Magnetic field measurement fluctuations	Plasma analysis system Neutral-Ion spectrometer Magnetometer	Irradiance of exosphere by sun where exosphere is being illuminated by sun and also direct passes through Jupiter's magnetosphere is required

Table 2 Functions of Instrumentation

Science Objectives	EORS	MIMI	Radio Science	RPWS	THEMIS	INMS	NIMS	UV Spec	NAC/WAC	LIDAR	Micro imager	Athena Microscopic Imager	GCMS	Fluxgate Magnetometer	Impact Device	Seismic Probe
Characterize and determine the extent of subsurface oceans and their relations to the deeper interior	X	X								X						
Characterize the icy shell and any subsurface water as well as the nature of the surface-ice-ocean exchange	X				X				X	X	X				X	X
Characterize the deep internal structure, differentiation history and intrinsic magnetic field	X	X	X	X										X		
Compare the exospheres, plasma environments and magnetospheric interactions		X	X	X		X							X	X		
Determine global surface compositions and chemistry, especially related to habitability						X	X	X	X		X	X	X		X	X
Understand the formation of surface features, including sites of recent or current geologic activity and identify and characterize candidate sites for future in situ exploration	X								X	X						X
Map temperature anomalies and thermal inertia of surface materials on Europa					X											
Map composition of non-ice components on Europa, including potential organic materials						X	X	X	X		X	X				

E. Science Implementation

E.1 Instrumentation

E.1.1 Athena Microscopic Imager:

The Athena Microscopic Imager (MI) shown in Figure 1 is placed on the lander to Europa. The Athena MI was selected to fulfill two of our science objectives. One, to map the composition of non-ice components on Europa, including potential organic materials; two, to determine global surface compositions and chemistry, especially related to habitability. It will return data in the form of optical light images. The Athena MI is also ideal for this mission because of its low weight and power requirements. The heritage for this instrument is the Mars Exploration Rovers (MER) and has proven to be an effective and useable instrument. The images from the MER's shown in Figure 2 are an example of this success. Environmental effects such as radiation is minimized due to the polar orbit of out flight around Europa. The specifications for the Athena MI are shown in Table 3.

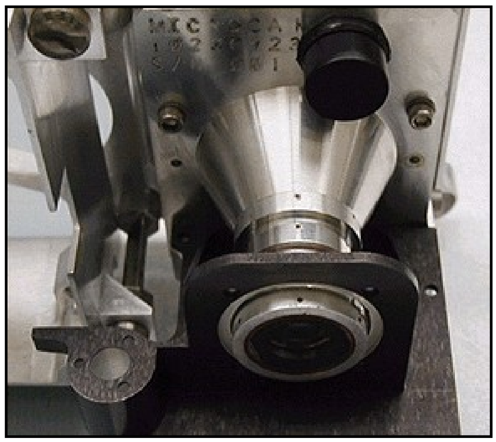


Figure 1 Athena Microscopic Imager

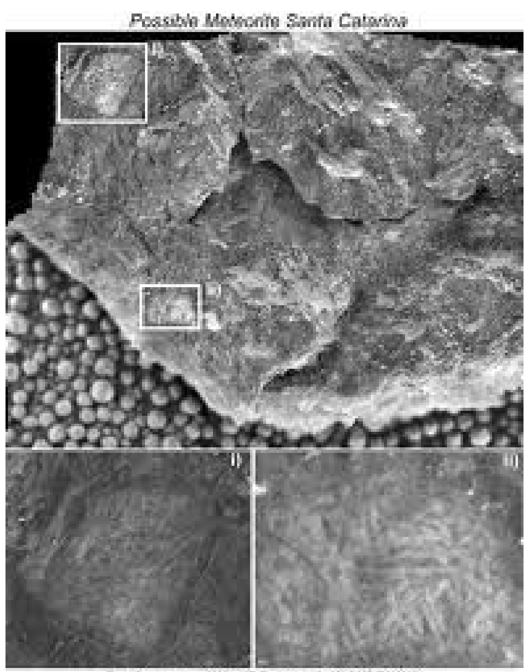


Figure 2 Sample Images from Athena MI

Table 3 Athena Microscopic Imager Specifications

Mass	Power	Resolution	Field of View	Temperature	Signal to Noise Ratio
.075kg	.1 watts	1024x1024 pixels	31x31mm	218.15-278.15K	At least 100 for exposures of 20%
Focal Length	*Working Distance*	*Object to Image Distance*	*Spectral Bandpass of the Optical System*	*Modulation Transfer Function (at best focus)*	*Radiometric Calibration Performance Accuracy*
20mm	63 mm from the front of the lens barrel to the object plane	100mm	400-680 nm	At least .35 at 30 lp/mm	Relative (pixel to pixel) accuracy is ≤ 5% Absolute accuracy is ≤ 20%
Depth-of-View					
Fixed focus design at f/15 that provides ±3 mm depth-of-field at 30µm/pixel sampling					

E.1.2 Gas Chromatograph and Mass Spectrometer

The Gas Chromatograph Mass Spectrometer (GCMS) shown in Figure 3 is placed on the lander to Europa. It was chosen to accomplish the following Science Objectives: To compare the exospheres, plasma environments and magnetospheric interactions, and to determine global surface compositions and chemistry, especially related to habitability. It measures the isotropic make-up of the major constituents such as water, carbon dioxide, carbon monoxide, methane, and ammonia. The gas chromatograph takes samples of gases and purifies them. They are then analyzed for their isotropic properties in the mass spectrometer. The mass and power fit into the payload for the spacecraft. The heritage for the GCMS is the Rosetta Mission; specifically the Rosetta MODULUS. Environment effects such as radiation is minimized due to the polar orbit of out flight around Europa.   The specifications for the GCMS are shown in Table 4.

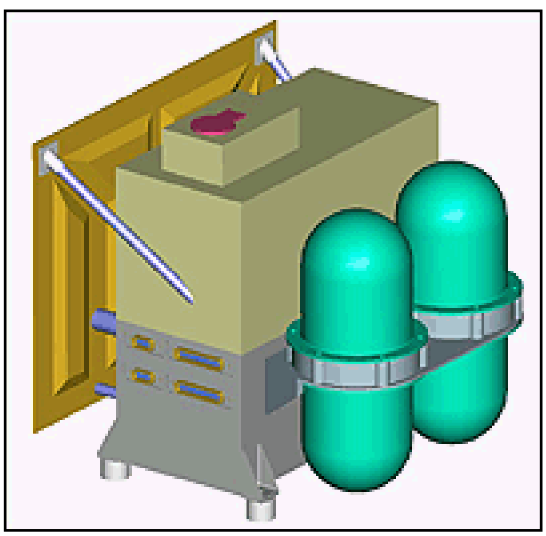


Figure 3 Gas Chromatograph Mass Spectrometer

Table 4 Gas Chromatograph Mass Spectrometer Specifications

Mass	*Power*	*Resolution*	*Principle Isotropic Ratios*
3kg	5 watts	mass resolution m/delta-m=100	Oxygen 18/16 and 17/16, carbon 13/12, nitrogen 15/14, and D/H

E.1.3 Fluxgate Magnetometer

The Fluxgate magnetometers are the primary method of determining if Europa has its own magnetic field. R^2D^2 uses two magnetometers that is placed in two different locations on the lander. This gives data that can potentially distinguish a separate magnetic field if one exists. The lander can accomplish this by comparing the two datasets from each orbit and try to normalize the field intensities by subtracting out the intense field that Jupiter produces. The two separate positions give us two vector components of the magnetic field since the vector fields are not the same at every position in the orbit. The lander gathers data that systematically isolates potential Europan magnetic field data and then allows the scientists to describe Europa's magnetic field in terms of its magnitude and vector field lines. Figure 4 shows are examples of a fluxgate magnetometer and the orientation of the rings that would comprise the magnetometer in order to distinguish miniscule magnitude magnetic fields.

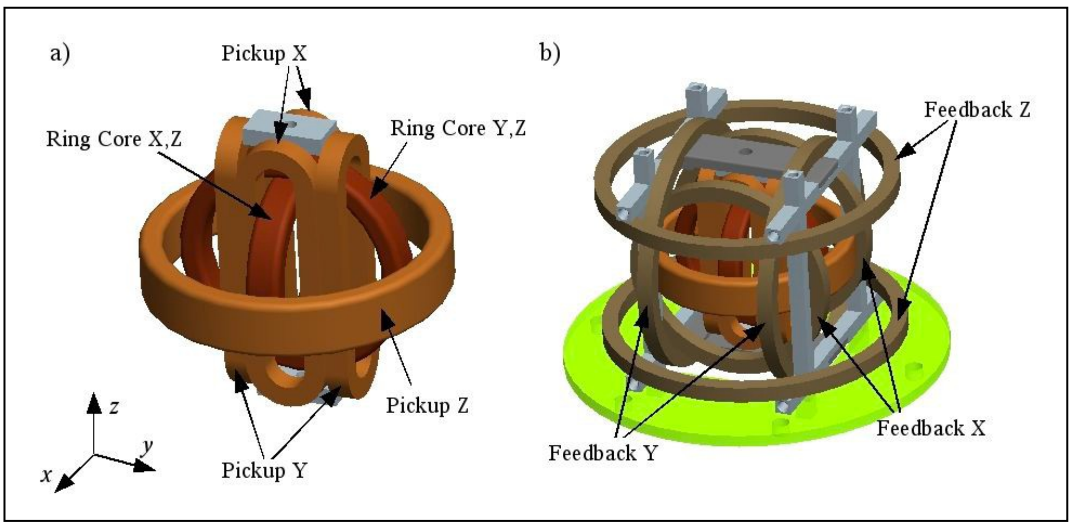


Figure 4 Fluxgate Magnetometer

E.1.4 Ion and Neutral Mass Spectrometer

The Ion and Neutral Mass Spectrometer or INMS, shown in Figure 5, is used to compare the exospheres, plasma environments and magnetospheric interactions, to determine global surface compositions and chemistry, especially related to habitability, and to map the composition of non-ice components on Europa, including potential organic materials. The INMS, operates in three different modes to collect information about the number density and composition of neutral species and low energy ions in its field of view. A "closed source neutral" mode is used for analyzing non-reactive, neutral chemicals and compounds. An "open source neutral" mode is used for analysis of neutral, reactive chemicals and compounds. An "open source ion" mode is used to examine positive ion. The instrument is useful for examining magnetospheric interactions and plasma environments. The heritage of this instrument is the Cassini Mission. Environmental effects such as radiation is minimized due to the polar orbit of out flight around Europa. INMS was tested at Goddard Space Flight Center in a high vacuum station with thermal neutral and ion sources to characterize instrument performance. The specifications for the instrument is shown in Table 5.

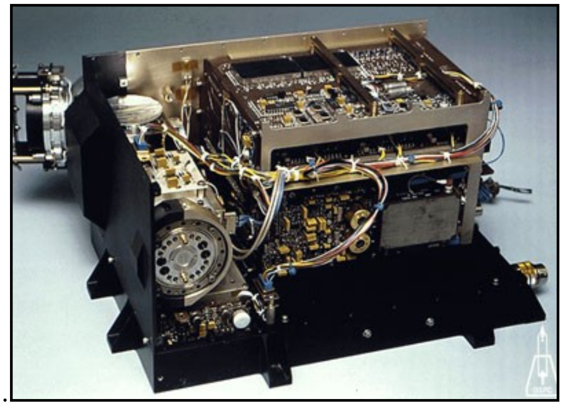


Figure 5 INMS

Table 5 Specifications for INMS

Mass	Power	Resolution	F o V	Date Rate	Bandpass Filter
9.25 kg	27.7 watts	mass range of 1 to 99 Daltons and a mass resolution of M/(deltaM) of 100 at 10% of the mass peak height	8.6 degrees	1.5 kilobits/s (average data rate)	Solid-state switched band pass filter (preforms frequency selection)

E.1.5 Radio and Plasma Wave Science

Radio and Plasma Wave Science Instrument (RPWS) consists of three electric field sensors, three search coil magnetometers and a Langmuir probe along with some receivers that cover the frequency range of 1Hz to 16MHz. It characterizes the deep internal structure, differentiation history and intrinsic magnetic field of Europa and compares the exospheres, plasma environments, and magnetospheric interactions of Europa. The heritage for this instrument is the Cassini-Huygens Mission. Environmental effects such as radiation are minimized due to the polar orbit of out flight around Europa. The specifications for the instrument is shown in Table 6.

Figure 6 RPWS

Table 6 Specifications for RPWS

Mass	Power	Data Rate	Frequency	Physical Orientation of the Electric Monopole Antennas
6.8 kg	7 watts	0.9 kilobit s/s	1Hz to 16MHz	Eu = 107.5(theta), 24.8 (phi) Ev = 107.5 (theta), 155.2 (phi) Ew = 37.0 (theta) 90.0 (phi)

E.1.6 THEMIS

This instrument has a thermal infrared spectrometer and a high-resolution camera. It will satisfy the following science objectives: To characterize the icy shell and any subsurface water as well as the nature of the surface-ice-ocean exchange and to map temperature anomalies and thermal inertia on the surface materials on the surface of Europa. The heritage for this instrument is the 2001 Mars Odyssey Mission. Environmental effects such as radiation is minimized due to the polar orbit of out flight around Europa. The specifications for the instrument is shown in Table 7.

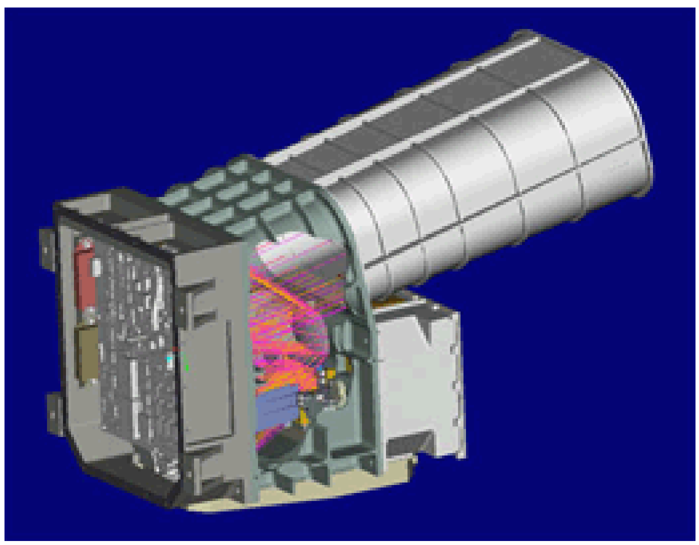


Figure 7 THEMIS Instrument

Table 7 Specifications for THEMIS

Mass	Power	Resolution	F o V	Temp.	Data Rate	Size	*Effective Aperture*	*Effective Focal Length*	*Infrared Imager Detector*	*Field of VIew*	*Filter Bands (micrometers)*
11.2kg	14 watts	Infrared = 100m/pixel	Infrared: 4.6 x 3.5 degrees	245 - 270 K	0.6 megabits/s	54.5 x 37.0 x 28.6 cm	12 cm	20cm	320 x 240 micro-bolometer array	4.6 degrees crosstrack and 3.5 degrees downtrack.	(9 bands) = 6.62 (1.01), 7.88 (1.09), 8.56 (1.18), 9.30 (1.18), 10.11 (1.10), 11.03 (1.19), 11.78 (1.07), 12.58 (0.81), 14.96 (0.86)
		Visual = 18m/pixel	Visual: 2.9 x 2.9 degrees								

E.1.7 Seismic Probes

The seismic probes seen in Figure 8 are similar to the probes used on NASA's Deep Space 2 mission. The probes is modified by removing the aeroshell, removing the atmospheric accelerometer, adding more batteries to increase battery life, and adding a seismic accelerometer to the probes. Additionally, the probe is monolithic, compared with the Deep Space 2 mission wherein the probes split apart into two sections on impact. As with the Deep Space 2 mission, the probe is able to measure deceleration on impact and determine the density of the Europan surface. The probes will take samples of Europa using the drill and spectrometer system included on the probes for the Deep Space 2 mission. These probes will have a battery life between 4 and 5 Earth days, allowing them to collect data regarding tidal flexing over the course of an entire orbit of Jupiter. The seismic probes will be evenly distributed over the Europan surface. All of the probes are not able to be active at the same time; however, by compiling the data from all of the probes, including the times at which the measurements occurred, a complete picture of the tidal flexing of the entire surface of Europa over the course of one complete orbit of Jupiter can be obtained. The probes impact the Europan surface at an angle of 20 degrees from the horizontal, skipping along the surface until their horizontal velocity is decreased enough to allow them to embed in the surface of Europa.

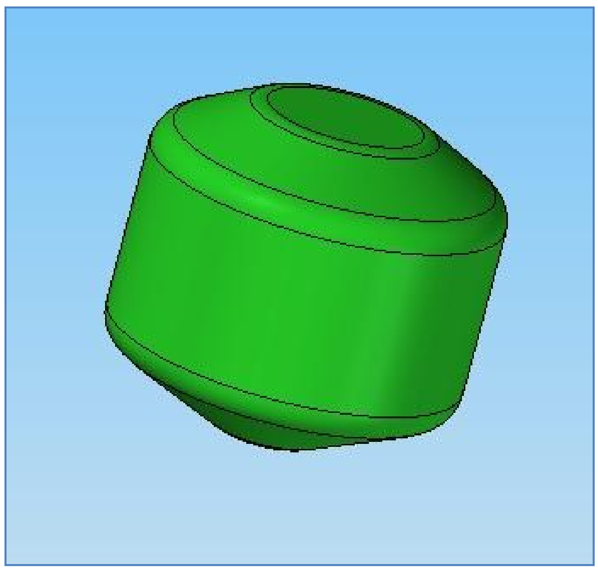


Figure 8 Seismic Probe

E.1.8 Europa Orbiter Radar Sounder

The Europa Orbiter Radar Sounder (EORS) was selected in order to map Europa's ice layer up to a depth minimum of 20 km. The device has a beam width of 22 degrees and a center frequency of 50MHz. The instrument can map the three dimensional structure of Europa with a resolution of 100m. The Radar Sounder offers excellent resolution with low weight and power requirements, and has been engineered to withstand high-radiation environments. The Europa Radar Sounder is currently in development for the Nasa Europa Orbiter. The Radar Sounder will provide data that can be used to create cross-sectional diagrams of the ice layer, leading to observations of a subsurface ocean, water pockets, and cross-section views of ongoing geologic processes. The spacecraft adequately accommodates the physical and power constraints needed by the Radar Sounder with a clear field of view. The Europa Radar Sounder is a TRL 3-5, however is up to 7 by the time of the mission. Radar sounders such as MARSIS have been proven effective in other missions, and the Europa Orbiter Radar Sounder builds off current proven technology. Figure 9 shows an example of the data return form EORS. The specifications for EORS are listed in Table 8.

Table 8 Specifications for EORS

Instrument	Mass	Power	Resolution	F o V	Temperature	Data Rate	Pointing Accuracy	Pointing Precision
Europa Orbiter Radar Sounder	10kg	100 watts	66 m(r)by 13 km(d) at 100 km	22 degrees	273-323 K	2 megabits per second	0.04 degrees	1 degree

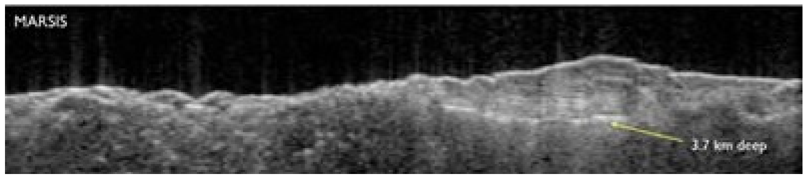


Figure 9 MARSIS example of the science return of EORS

E 1.9 Radio Science

The Radio Science Subsystem (RSS) was selected in order to map Europa's atmospheric circulation, ionospheric structure, and internal structure. The Radio Science Subsystem offers low weight requirements, with a high power requirement preventing the Radio Science Subsystem from operating constantly during orbit. However, the power requirements are adequate, allowing the Radio Science Subsystem to provide data that contribute to the mapping of the internal and ionospheric structure, as well as mapping any potential atmospheric circulation that Europa may experience. The Radio Science Subsystem transmits radio waves which are read by RSS sensing devices on Earth at the Deep Space Stations in California, Spain and Australia. The radio signals are transmitted in S-,X-, and Ka-band with the primary down link being X-band. The spacecraft radio equipment receives waves from earth with the High Gain Antenna, a processing unit transforms the signal to a predetermined downlink frequency, amplifies the signal, and transmits the signal back to Earth. The returning signal is detected, amplified, and converted to useful data. The data is used to determine the composition of materials which the radio waves pass through on the journey between the orbiter and Earth.

The spacecraft adequately accommodates the physical and power constraints needed by the Radio Science Subsystem with a clear field of view. The Radio Science System is a TRL 8, having been proven extremely effective in the Cassini mission, working as planned during the entire scope of the mission. The specifications for the instrument are shown in Table 9.

Table 9 RSS Specifications

Instrument	Mass	Power	Resolution	F o V	Temperature	Data Rate	Pointing Accuracy	Pointing Precision
Radio Science	14.38kg	80.70 watts	265-375 MHz, centered at 320 MHz	N/A	273-323 K	N/A, unmodulated carrier is transmitted. RSS sensing devices are on earth.	0.04 degrees	0.04 degrees

E.1.10 Rock Abrasion Tool

The Rock Abrasion Tool as selected in order to abrade select areas of Europa's surface for further analysis by Gas Chromatograph and Mass Spectrometer. The RAT offers low weight and power requirements, and is able to drill through rock at an adequate rate, allowing all areas of desired study to be ground for analysis until the lander is no longer operational. The RAT exposes an area nearly 5 cm (2 inches) in diameter, and grinds down to a depth of about 5 mm (0.2 inches). The RAT is designed to preserve petrologic textures of the prepared rock surfaces, allowing useful data to be recovered from the samples. The lander adequately accommodates the physical and power constraints needed by the Rock Abrasion Tool with a clear field of operation. The RAT is a TRL 8 after being proven successful by both Mars Rovers Spirit and Opportunity. The RAT completed its objectives with no mechanical failures. The RAT is shown in Figure 10. The specifications for the instrument are shown in Table 10.

Table 10 Rock Abrasion Tool Specifications

Instrument	Mass	Power	Resolution	F o V	Temperature	Data Rate	Pointing Accuracy	Pointing Precision
Drill	.687kg	30 watts	N/A	N/A	233-313 K	N/A	N/A	N/A

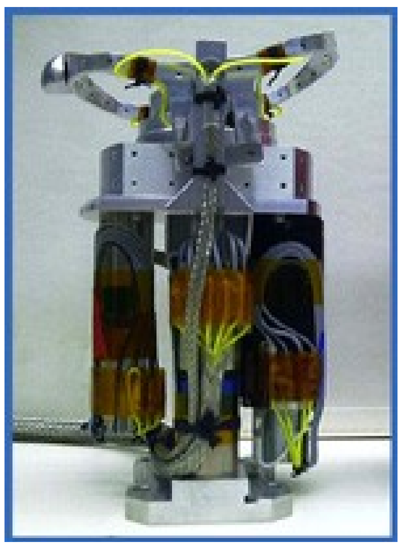


Figure 10 Rock Abrasion Tool

E.1.11 QE65000 UV Spectrometer AKA ALICE

The QE65000 UV Spectrometer shown in Figure 11 was selected because the ice minerals expected to be found on Europa have a wavelength signature lying in the UV range. This spectrometer offered excellent performance combined with low weight and power requirements. Flight heritage of this spectrometer is the Rosetta mission. The spectrometer takes scans in the UV range of a wide area of Europa's surface, providing data in the form of spectra that is matched with known mineral spectra to provide data in the form of mineral compositions of the selected areas. The spacecraft adequately accommodates the physical and power constraints needed by the spectrometer with a clear field of view. The UVS is a TRL 9, having proven itself in space in the Rosetta mission, the New Horizons mission to Pluto/Charon and the Kuiper Belt and the LCROSS mission. A mounting and slewing device must be developed for the QE65000. Reengineering is also available and previously successful by Aurora Design & Technology to withstand the extreme temperature, radiation, shock, and vibration of space. The New Horizons ALICE UV spectrometer was successfully launched on 19 January 2006 and is operating normally in space. All in flight performance tests to date have shown performance within specification; the pointing and AGC sensitivity tests completed in September 2006 are in analysis, and the initial results of these tests indicate nominal performance with no degradation yet observed. The specifications for the instrument are shown in Table 11.

Table 11 ALICE UV Spectrometer Specifications

Instrument	Mass	Power	Resolution	F o V	Temperature	Data Rate	Pointing Accuracy	Pointing Precision
Ultraviolet Spectrometer	1.18 kg	3.5 A	0.0015-0.033 microns	1 x 0.1 degree 0.4 x 0.1 degree	273-323 K	3559956 bits per second	0.0225 degrees	0.1 x 0.1 degrees

Figure 11 ALICE UV Spectrometer

E.1.12 LIDAR

A Light Detection and Ranging instrument (LIDAR), shown in Figure 12, was selected because it provides a reliable shape model of Europa's surface including topographic information. LIDAR offers reliability and high performance at adequate physical and power levels. Flight heritage of this instrument is the Hayabusa mission. The LIDAR provides data in the form Europa's surface profile, topographic information, and range from spacecraft that is corrected for orbit and pointing errors. The spacecraft adequately accommodates the physical and power constraints needed by the spectrometer with a clear field of view. The UVS is a TRL 7, having proven itself in space in the Hayabusa mission. The current radial inaccuracies of the LIDAR data do not permit the generation of global grids of the surface of Itokawa. Additional refinements are still required. LIDAR functioned without flaws for the entire 3 month period of the encounter with Itokawa with no observed degradation. The specifications for the instrument are shown in Table 12. Figure 13 shows a sample of the data returned by LIDAR.

Table 12 LIDAR Specifications

Instrument	Mass	Power	Resolution	F o V	Temperature	Data Rate	Pointing Accuracy	Pointing Precision
LIDAR	3.56kg	22 watts (heated, 17 watts w/no heater)	300 m by 133 m at 100 km	1 mrad	283-333 K	3.008 kilobits per second	50 m at 100 km	+/- 2m at 100 km

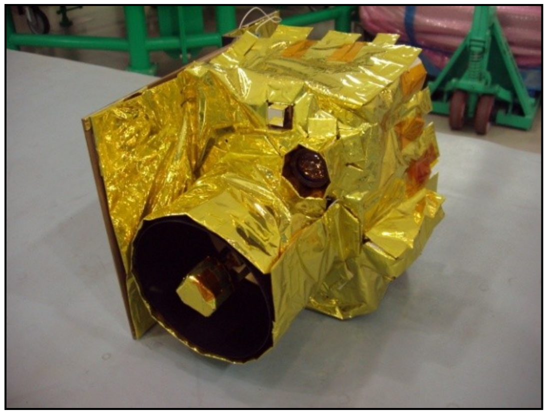


Figure 12 LIDAR

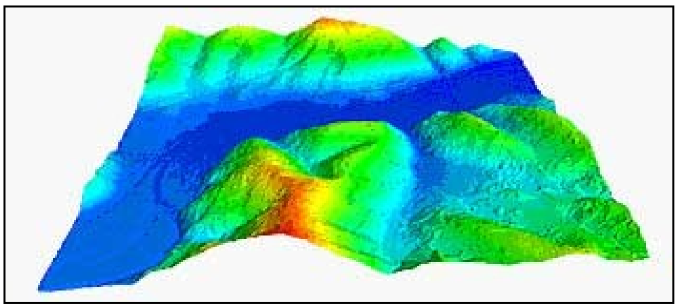


Figure 13 LIDAR Data Sample

E.1.13 OSIRIS NAC/WAC

The OSIRIS NAC/WAC was selected because it consists of two independent camera systems sharing common electronics. This reduces the weight and size of the camera while providing the performance of both an NAC and WAC. This spectrometer offered excellent performance combined with low weight and power requirements. Flight heritage of this spectrometer is the Rosetta mission. The NAC and WAC each capture pictures in the visible spectrum of Europa, from an extremely small to a wide area, providing data in the form of visible light pictures that is analyzed for each asteroid's surface geology. The spacecraft adequately accommodates the physical and power constraints needed by the OSIRIS with a clear field of view. The OSIRIS is a TRL 9, having proven itself in space in the Rosetta mission. Since March 2005 instrument health has been monitored in a checkout every 6 months. The instrument proved to be in good health in the checkouts performed so far. Operations have included mechanism tests, instrument calibration, alignment between the boresights of the different remote sensing instruments on Rosetta, and interference check between OSIRIS and other instruments.

E.1.13.1 System Characteristics

The OSIRIS cameras are unobstructed mirror systems, equipped with two filter wheels containing 8 position each, and with backside illuminated CCD detectors comprising 2048 x 2048 pixels with a pixel size of 13.5 µm. Both cameras use identical image acquisition systems, consisting of the Focal Plane Assembly and the CCD Readout Box. OSIRIS comprises two cameras NAC (Narrow Angle Camera) and WAC (Wide Angle Camera). The NAC is designed to obtain high-resolution images of a body at distances from more than 500,000 km down to 1 km. The camera also should be able to detect small ejected particles (brightness ratio = 1/1000). The NAC is equipped with 12 filters to characterize the reflectivity spectrum of the nucleus surface over a wide spectral range from 250 to 1000 nm. The NAC has a square field of view (FOV) of width 2.2 degrees, has an instantaneous field of view (IFOV) of 18.6 µ rad (3.8 arcsec) per pixel, and is a moderately fast system (f/8). The system has a 717 mm focal length. A flat-field, three anastigmatic mirror systems is adopted. It has a mass of 13.2 kg. WAC the principal objective of this camera is to study the surface characteristics of Europa, as well as the ejecta generated by the Europa Impactor. The WAC is accomplished by 14 filters from 240 to 720 nm. Seven of the narrow band filters isolate and gas emissions from double ridges; the others filters measure the dust continuum at wavelengths close to that of the gas emissions. The WAC has a FOV of 12x12 degrees, has an angular resolution of 101 μ rad (20.5 arcsec) per pixel, and is a system with a fast focal ratio of f/5.6. The system has a 140 (sag)/131 (tan) mm focal length. A two aspherical mirror system is adopted. It weighs 9.5 kg. The specifications for the OSIRIS camera system are shown in Table 13. Images of OSIRIS NAC and WAC are shown in Figure 14 and Figure 15, respectively. Figure 16 shows an example of the expected images OSIRIS will return.

Table 13 OSIRIS NAC/WAC Specifications

Instrument	*Mass*	*Power*	*Resolution*	*F o V*	*Temperature*	*Data Rate*	*Pointing Accuracy*	*Pointing Precision*
NAC/WAC	35kg	20 watts	1.66 km x 1.66 km (WAC) and 0.562 m x 0.562 m (NAC) at 100km	12x12 degrees (WAC) 2.2x 2.2 degrees (NAC)	285 operational; 233-323 K non-operational	19173.96 kilobits per second (NAC)19173.96 kilobits per second (WAC)	0.022 degrees	0.1 millidegree

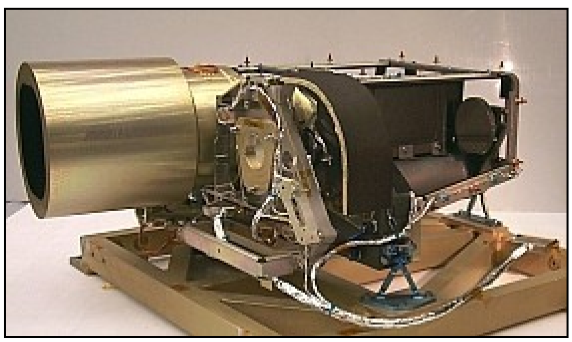


Figure 14 OSIRIS NAC

Figure 15 OSIRIS WAC

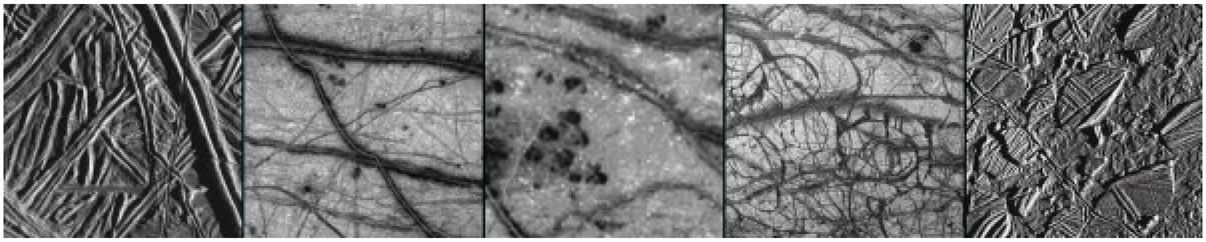


Figure 16 Example of Images OSIRIS is Expected to Return

E.1.14 Magnetospheric Imaging System

The MIMI (Magnetospheric Imaging System) is composed of three detectors which are found in Figure 17, Figure 18, and Figure 19. The Ion and Neural Camera (INCA), Charge Energy Mass Spectrometer (CHEMS), and Low Energy Magnetospheric Measurement System (LEMMS) perform a variety of measurements permitting the characterization of internal structure and intrinsic magnetic field. The units are connected via coax cable to a central Main Electrical Unit (MEU), Figure 20. Table 14shows the specifications for the MIMI system.

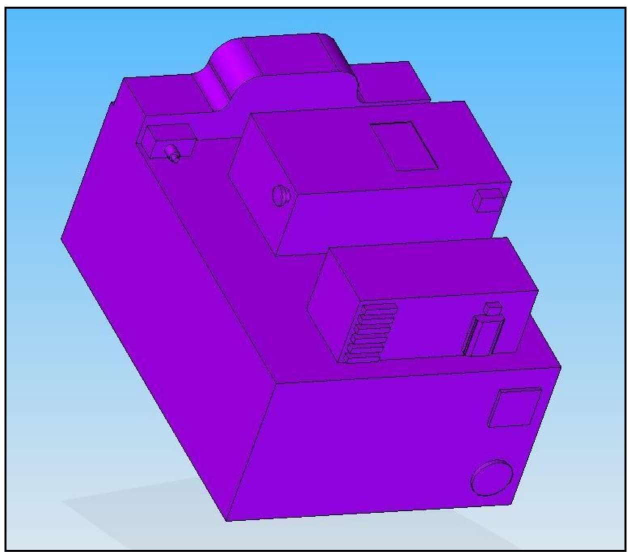


Figure 17 Charge Energy Mass Spectrometer

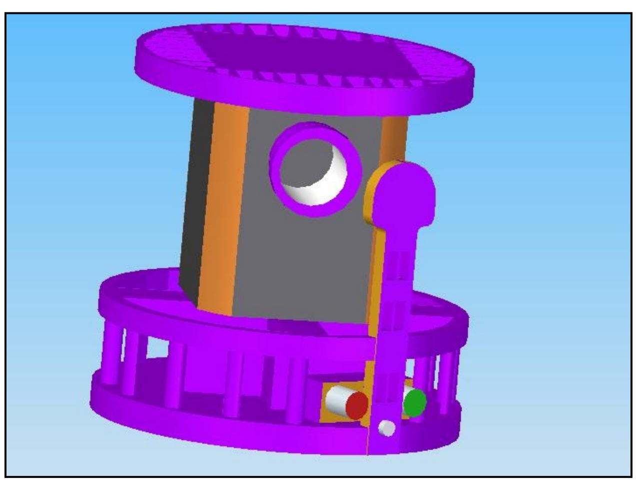


Figure 18 Low Energy Magnetospheric Measurement System (LEMMS)

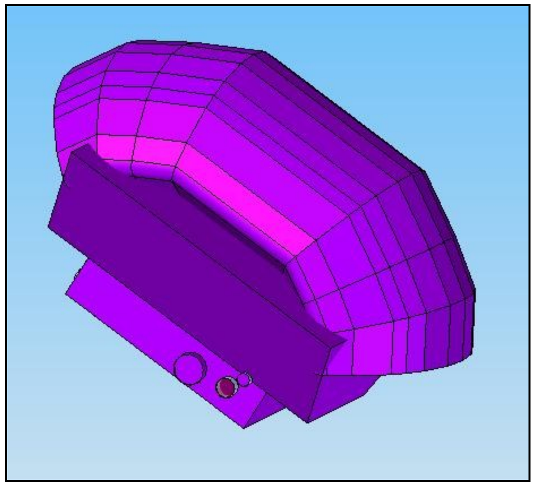


Figure 19 Ion and Neural Camera (INCA)

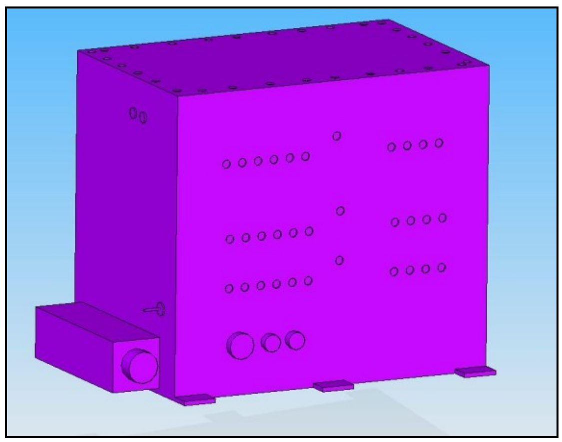


Figure 20 Main Electrical Unit (MEU)

Table 14 MIMI Specifications

Instrument	Mass	Power	Resolution	F o V	Temp	Data Rate	Pointing Accuracy	Pointing Precision
Magnetosphereic Imaging System*	16kg	14 watts	+/-0.04 n	CHEMS: 160 INCA: 120 LEMMS: 15/30	273-323 K	7 kilobits/s	0.04 degrees*	1 degree*

E.1.15 Near Infrared Mapping Spectrometer (NIMS)

The NIMS instrument shown in Figure 21 consists of seventeen detectors allowing multiple near-simultaneous measurements spaced evenly across the wavelength region. The unit has a dispersion element which is dual-blaze grating and allows seventeen additional wavelength sets with small offsets to be obtained. The instrument acquires spatial information by utilizing the motions of the spacecraft scan platform and motions of a secondary mirror. The primary objectives of the instrument are measurement of the composition if present of the Europan atmosphere and the composition of the surface. Table 15 shows the specifications for the NIMS system.

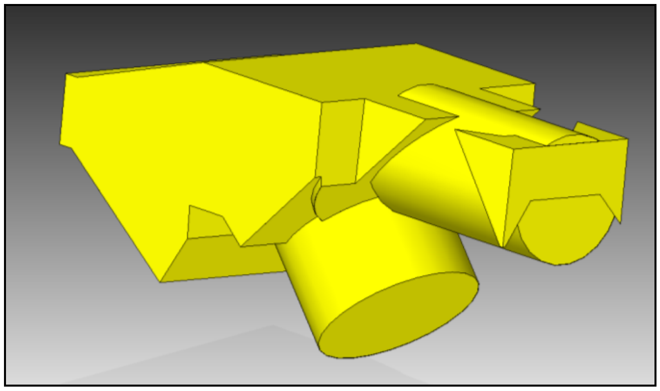


Figure 21 Near Infrared Mapping Spectrometer

Table 15 NIMS Specifications

Near Infrared Mapping Spectrometer	18kg	12 watts	.0125 @ wavelengths less then 1micron .0250 @ wavelengths greater than 1 micron	10 mrad x .5 mrad	150 K	11.52 kilobits/s	0.6 degrees*	0.12 - 0.27 degrees*

E.1.16 Micro-Imager

The Asteroid-Moon Micro-Imager Experiment (AMIE) shown in Figure 22 is designed to take multi-band images of the Moon. It is a 5.3 degree field-of-view silicon CCD camera which will provide a 1024 x 1024 pixel image with an average resolution of 80 m/pixel.

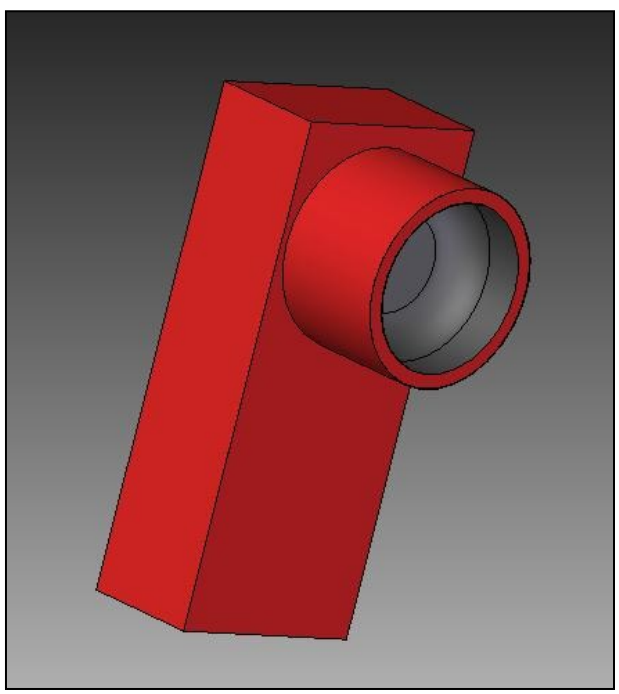


Figure 22 Micro-Imager

E.2 Data Sufficiency

Each instrument collecting data is capable of collecting a sufficient amount of information from measurements to answer the related questions and fulfill objectives. Each instrument provides a quality and quantity sufficient enough for analysis, as stated in the instrumentation Section E (i.e. accuracy, resolution, sensitivity, data rates).

E.3 Science Mission Profile

The proposed mission design and operations plan given in later sections are directly impacted by proposed investigation objectives, selected instruments, and measurement requirements The science observing profile involves the parameters covered by each instrument discussed in the science traceability objectives. The navigation accuracy, operational time lines, observing periods, data transmission periods, techniques, and time-critical events are as follows. The navigation accuracy is challenged by the high resolution of the instrumentation. The data storage is fully capable of satisfying all needs while the communication system seems to be the real limiting factor to the computer data system. Operational time line is best summarized and critical events are established in the concept of operations and detailed in the baseline earlier in this report. The observing periods are dependant primarily on the field of view for each device. The data transition periods for the lander are estimated to be about 15 minutes every 2 hours. The primary communications with the lander implement S band. All telecommunications strategies are influenced heavily by critical event such as flybys, science enhancement options, and celestial body studies.

E.4 Data Plan

A schedule based data management plan is put into place to account for data retrieval, validation, preliminary analysis, and archiving of all data collected during the mission. The data is transmitted from the spacecraft by data downlinks. A PI is responsible for analysis of the data required to complete the science objectives of the mission. A PI is also responsible for the publication of all newfound results of the mission to relevant scientific journals. There is a short period of exclusive access to the data found by the mission in order to calibrate and validate the data. This period does not exceed six months and is followed by the presentation of the data to the public, as is the policy of NASA. The mission data is made presented to the public by means of the Planetary Data System. Raw data, or Level 0 data, is analyzed by the PI before delivery to the archive. All data submitted to the archive is processed data, or Level 1. All Level 2 or higher data products are evaluated according to NASA data archive standards of format.

E.5 Science Team

E.5.1 Principal Investigator

The primary responsibility for implementing and executing selected investigations rests with the PI, who must have significant latitude to accomplish the proposed objectives within committed schedule and financial constraints. This responsibility, however, is exercised with essential NASA oversight to ensure that the implementation is responsive to the requirements and constraints of the Discovery Program. The Mission PI is accountable to NASA for the success of the investigation, with full responsibility for its scientific integrity and for its execution within committed cost and schedule. The Mission PI must be prepared to recommend project termination when, in her/his judgment, the minimum subset of science objectives identified in the proposal as the Threshold Science Mission is not likely to be achieved within the committed cost and schedule.

E.5.2 Co-Investigator

The mission co-investigator shall function as the principal investigators main constituent. He will help in the realization of the mission and ensure that the scientific portion of the mission is being carried out according to the plans of the principal investigator. All other scientists that are to be working on the project shall be designated a role and title by either the principal investigator or the co-investigator during the actual mission execution.

E.6 Plan for Science Enhancement Option (SEO)

Science Enhancement Options are vast for ICESSS. The ICESSS orbiter has the instrumentation necessary to conduct high resolution visual and thermal imaging on Jupiter as well as its other major satellites Io, Ganymede and Callisto. Due to the polar orbit that the ICESSS orbiter is following as it tours the Jovian system, it will encounter all of the satellites numerous times during its proposed mission life. During these orbits when the ICESSS orbiter is not focusing on Europa it could collect valuable information on the magnetic fields of the other satellites and Jupiter using the magnetometers already on board. It also could gather UV, and IR spectroscopy, micro-images, high resolution visible light images, thermal evolution, radio science, and magnetospheric data on the exterior satellites. The high-resolution imaging would provide the means to analyze the temporal and spatial changes due to the tidal friction on the exterior satellites in addition to Europa's. These enhancements significantly amplify the potential scientific information that would be gained by realizing ICESSS.

F. Mission Implementation

The Science Investigation and Science Implementation sections of this proposal describe the scientific investigation to be performed at Europa and the instruments selected to accomplish the study. This section describes, in detail, how the scientific instruments are accommodated onboard a spacecraft which takes them to their destination and allows them to perform their appointed tasks.

F.1 General Requirements and Mission Traceability

The scientific objectives described in this report can best be accomplished utilizing an orbiter which travels to Europa and deploys seismic probes and a lander to the Europan surface. In order to assure mission success, the Inner Crustal Europa Seismic and Spectral Surveyor (ICESSS) is able to perform a large amount of the science investigation without relying on the lander or seismic probes, in case any deployed devices fail. The lander's primary objective is to obtain a direct sample of the surface and analyze the composition and crystal structure of the surface of Europa. The primary objective of the seismic probes is to obtain data on the tidal flexing of Europa over the course of Europa's orbit around Jupiter. Each of these devices has additional objectives which are described in more detail in Section D and E.

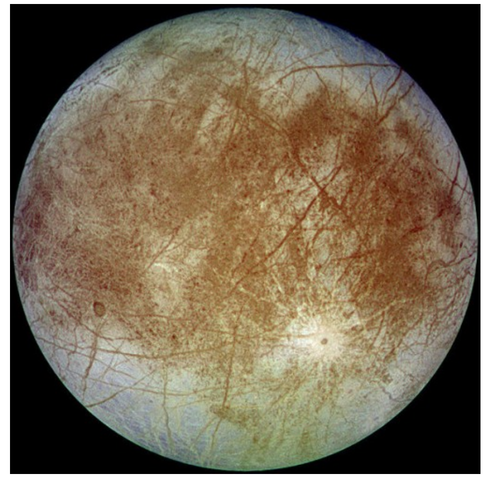


Figure 23 Europa

The general requirements for this mission are as follows:

- Launch from Kennedy Space Flight Center
- Travel to Europa
- Obtain scientific data regarding the composition and structure of Europa, magnetic and plasma environments of Europa, tidal flexing, and geological history of Europa
- Protect scientific instrumentation and recorded scientific data from the radiation and thermal environments encountered in for the entirety of the mission
- Transmit recorded scientific and housekeeping data to the Deep Space Network on earth for analysis

More specific requirements for the mission stem from the requirements and sensitivities of the scientific instrumentation described in Table 1 The mission requirements are described in Table 16. The basic thought process is illustrated from left to right for both Matrices. The Mission Traceability Matrix shows the traceability from mission functional requirements to mission design, orbiter, Lander, ground system and operations. Each column is an established list of design features that directly affects the next column. Rows are not utilized because one feature in the left column may affect multiple features in the next column.

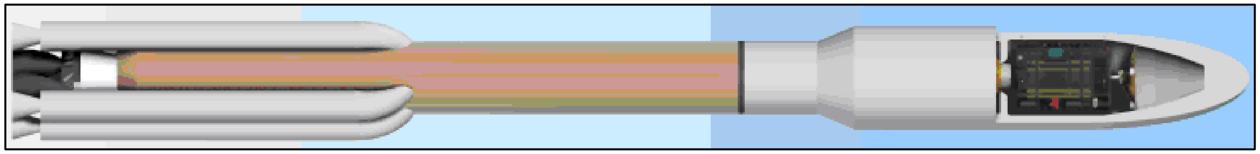
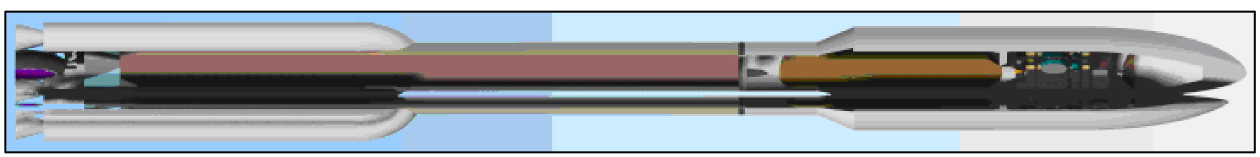


Table 16 Mission Traceability Matrix

Mission Traceability Matrix					
Mission Functional	**Mission Design**	**Orbiter**	**Lander**	**Ground Systems**	**Operations**
• Launch from Kennedy Space Flight Center and Travel to Europa • Obtain scientific data of composition and structure, magnetic and plasma environments, tidal flexing, and geological history • Protect instrumentation and recorded data from radiation and all thermal environments encountered • Transmit all data to earth for analysis • Consistent pointing accuracy to allow for global surface mapping and spectral imaging. • Irradiance of surface to allow adequate sun-light collection; and of exosphere by sun and passes in Jupiter's Magnetosphere • Global viewing to allow for measuring • Viewing over an extended time span • Landing on surface and transmitting data back to orbiter • High Pointing accuracy to allow for non blurred images • Communications with penetrators specifically distributed across the surface	Atlas V 551 C22 adapter with B1194 PSR Launch 2/1/2020 VEEGA Trajectory Mission 139 Months Orbit altitude 50 to 100 km Complete Geographic coverage Julian Time Polar orbit RPS through ASRGs	Mass 4517 kg Total Power 280 W Data Rate 21 Mb/s Temperature Range 290 to 350 K with sheilding Battery for peak power modes Data storage 671.6 Gbit Ka band Data destination: mission ops Power for comm 85 W	Mass 130 kg Total Power 1140 W*hr Data Rate .88 Mb/s Temperature: 70K Data storage 16 Mbit Number of data dumps 6 S band Data destination: Orbiter Power for comm 30 W	Real time data transmission Ka band Downlink data rate:150kbit/s Number of data dumps 12 per day 1hr per pass duration Science data destination: Deep Space Network	Number of Orbits per day: 12 Alignment to release impactor, penetrators, and lander Pointing Control: Nadir Rationale for maneuvers Europa Initial Orbit Inclination: 95 deg Continuous self diagnostics



F.2 Mission Concept Descriptions

F.2.1 Mission Design

The ICESSS mission launches from Kennedy Space Center on board an Atlas V 551 and travel to the Jovian System using a Venus Earth Earth gravity assist (VEEGA). Arriving at the Jovian system, the spacecraft named performs a propellant burn to insert into a Jovian orbit which tours the Jovian system for 2 years, slowing down over time. During this time, ICESSS utilizes onboard equipment to study various bodies in the Jovian system. After 2 years, the spacecraft is slow enough to transition into a polar Europan orbit without requiring a prohibitively large propellant burn. At this time, ICESSS begins a final orbit of Jupiter, on a path taking it to Europa. An impactor device is be deployed from the spacecraft as it nears Europa. This device deploys while the spacecraft is traveling at approximately 2.18 km/s.

The device continues to Europa and impacts the surface, producing a plume of ejecta. A propellant burn injects the spacecraft into a polar Europan orbit. This orbit has a period of approximately 2 hours. The spacecraft orbits the moon, collecting data from its onboard instrumentation. The initial focus of the orbiter is analyzing the ejecta produced by the impactor and collecting data about the three dimensional structure of Europa using the Europa Orbiter Radar Sounder. These priorities exist due finite timeframe during which the ejecta exists at a high enough density to be analyzed properly and due to the 30 day life expectancy of the Europa Orbiter Radar Sounder once it enters the Europan orbit.

After these studies have been completed, the orbiter begins a one week period of magnetospheric studies, initiated by a reaction control system burn to initiate a slow spinning of the spacecraft. This spin allows the instruments composing the magnetospheric imaging system onboard ICESSS to obtain 3-dimensional data and visualizations of Europa's induced magnetic field. A de-spin maneuver follows completion of this 1 week phase of the ICESSS mission, and results in a spacecraft orientation which points all of the scientific imaging devices toward the Europan surface for the rest of the ICESSS mission. As the spacecraft orbits Europa, it begins to deploy twenty-nine seismic probes to the surface of Europa. The spacecraft ejects these probes at separate times, such that they are spread across the surface in order to form a network encompassing the entirety of the moon. The deployments take place over a period of seven days, as new longitudes become visible with the rotation of Europa on its axis. Europa completes one full rotation about its axis in 3.55 days. Each probe is able to determine a local surface density and composition for Europa's icy shell. Seismic accelerometers on each probe are able to measure tidal flexing of the moon over the course of its orbit around Jupiter. The end result from all of these probes is a picture of the composition and density of Europa's ice shell, including its variations with regard to longitude and latitude, and an understanding of the tidal flexing Europa undergoes, again including variations with regard to longitude and latitude.

After all of the seismic probes have been deployed and deactivated upon exceeding their battery life of approximately 5 days, a lander is deployed to the Europan surface. The lander has two fluxgate magnetometers onboard capable of determining the magnetospheric conditions of Europa. This information leads to an understanding of Europa's magnetic field and its interactions with Jupiter's magnetic field and radiation. The lander also utilizes a rock abrasion tool, a Bernice gas chromatograph and mass spectrometer, and a microscopic imager to obtain stereoscopic images and composition data for the Europan surface material.

The overall mission design is accomplished with the spacecraft shown in Figure 24, while Figure 27 illustrates the basic Concept of Operations for the ICESSS mission to Europa, with a more in detailed mission breakdown in Table 17. As shown, after launch the spacecraft transitions from that seen in Figure 28 to that displayed in Figure 24. After about seven years in a VEEGA trajectory, the spacecraft tours the Jovian system before orbiting the moon and beginning its primary mission. Figure 36 illustrates the suggested baseline schedule presented in Table 17. The specific dates, while flexible, provide for a more accurate goal and prevent possible confusion.

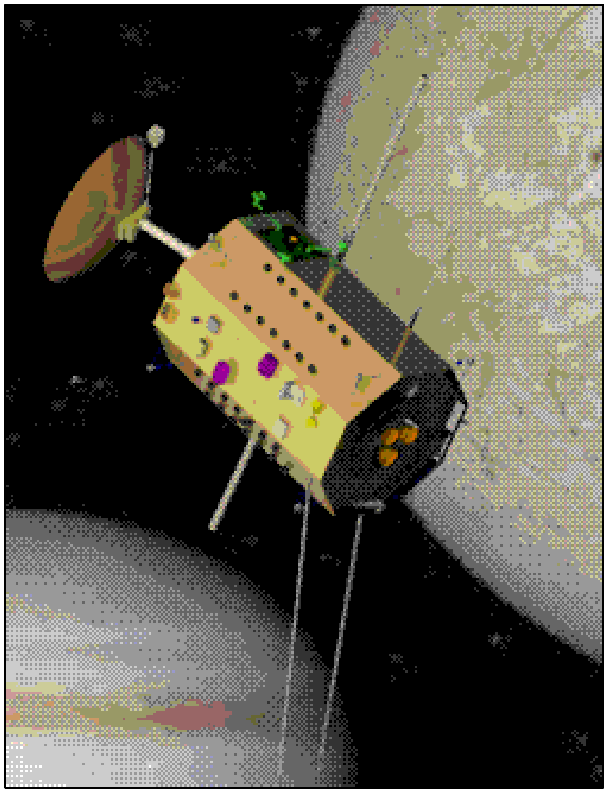


Figure 24 ICESSS Deployed Spacecraft

Table 17 Mission Duration Breakdown(JEO 2009)

Mission Duration Breakdown					
Phase	**Title**	**End of Phase Review**	**Duration (Months)**	**Start**	**Finish**
Pre-Phase A	Proposal KDP-A	Mission Concept Review (MCR)	11	6/7/2010	4/28/2011
Phase A	Concept Study Reports	Concept Study KDP-B	9	4/29/2011	2/1/2012
Phase B	Preliminary Design	Preliminary Design Review (PDR)	20	2/1/2012	9/23/2013
	Downselection of investigation		4	2/1/2012	6/1/2012
	The independent cost estimate	KDP-C	20	2/1/2012	9/23/2013
Phase C	Final Design and Fabrication	Critical Design Review (CDR)	30	9/24/2013	3/12/2016
	NEPA Notice of Intent	KDP-D	21	9/24/2013	6/21/2014
	NEPA Databook		16	9/24/2013	11/18/2014
	Final NEPA Document		1	9/24/2013	2/11/2016
	SAR* Launch Vehicle Databook		3	9/24/2013	12/13/2015
	OSTP Request Launch Approval		6	9/24/2013	9/14/2015
	Risk communication Plan		21	9/24/2013	6/21/2014
Phase D	Integration KDP-E	Systems Integration Review (SIR)	28	3/12/2016	6/30/2018
	Assembly		14	3/12/2016	5/6/2017
	Test		14	5/6/2017	6/30/2018
	Launch			2/1/2020	2/1/2020
Phase E	Operation and Sustainment		133	2/1/2020	1/4/2031
	Cruise 347 weeks		87	2/1/2020	3/26/2027
	Jovian Tour 139 weeks		35	3/26/2027	2/8/2030
	Europa Orbit 41 weeks		11	2/8/2030	1/4/2031
Phase F	Closeout	Decommissioning Review	6	1/4/2031	7/3/2031
Total	21	8	490		

F.2.1.1 Trajectory

A general trajectory is shown in Figure 25 from the JEO report with in depth orbital mechanics information shown in Table 18 Orbital Mechanics and Figure 26. As shown in the figures, the mission utilizes a VEEGA trajectory to decrease the required excess velocity which must be imparted to the orbiter by the launch vehicle. This maximizes the mass available for the orbiter. After completing the VEEGA, the orbiter follows the prescribed path to Jupiter, passes by Io in order to reduce its velocity, and performs a Jupiter Orbit Insertion (JOI) burn. The JOI takes place when ICESSS is at a distance of 5.2 Jovian radii from Jupiter. After inserting into Jovian orbit, the spacecraft performs a two year tour of the Jovian system, passing by the Galilean moons and providing multiple opportunities for scientific studies. After completing the tour of the Jovian system, ICESSS begins a final approach to Europa. The spacecraft inserts into Europan orbit and begins the scientific study of the moon. Table 18 provides the critical dates for delivery of the spacecraft to Europa. These values for velocity are expected at the corresponding dates and should help indicate accuracy to the red line in Figure 25. Refer to Section 3.3.2.2 of the Jupiter Europa Orbiter Final Report, for a more detailed explanation of the mission's flight path.

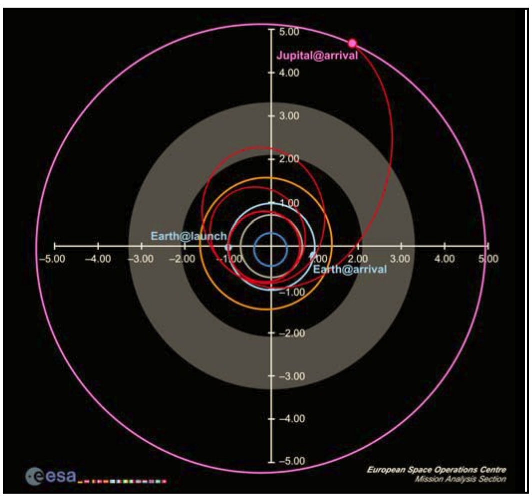


Figure 25 Trajectory (JEO 2009)

Table 18 Orbital Mechanics (JEO 2009)

Orbital Mechanics		
Depart Earth	6/30/2020	
Cruise Duration	2610	Days
Jovian Tour Duration	1050	Days
Jovian Arrival Date	8/22/2025	
European Arrival Date	7/7/2028	
Launch Vehicle	Atlas V 551	
Mass (Total at launch)	4615	kg
Depart C3	14.7	km2/s2
Jovian Arrival V	5.25	km/s
European Arrival V	2.81	km/s
ΔV	2360	km/s

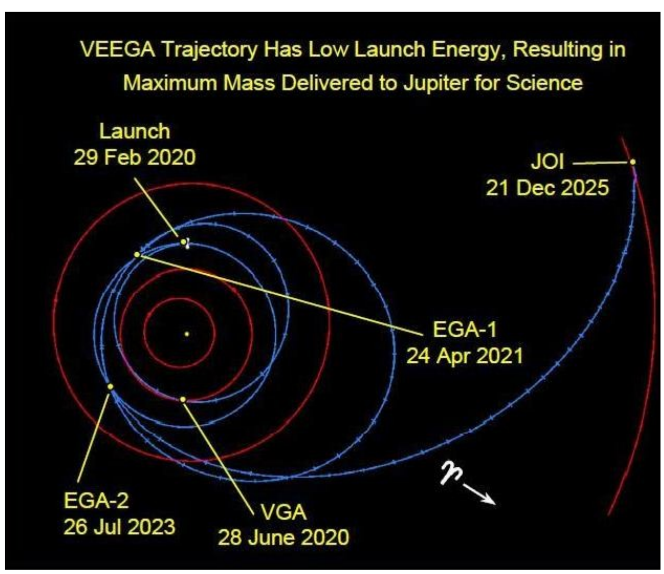


Figure 26 Orbital Information (JEO 2009)

F.2.1.2 ConOps

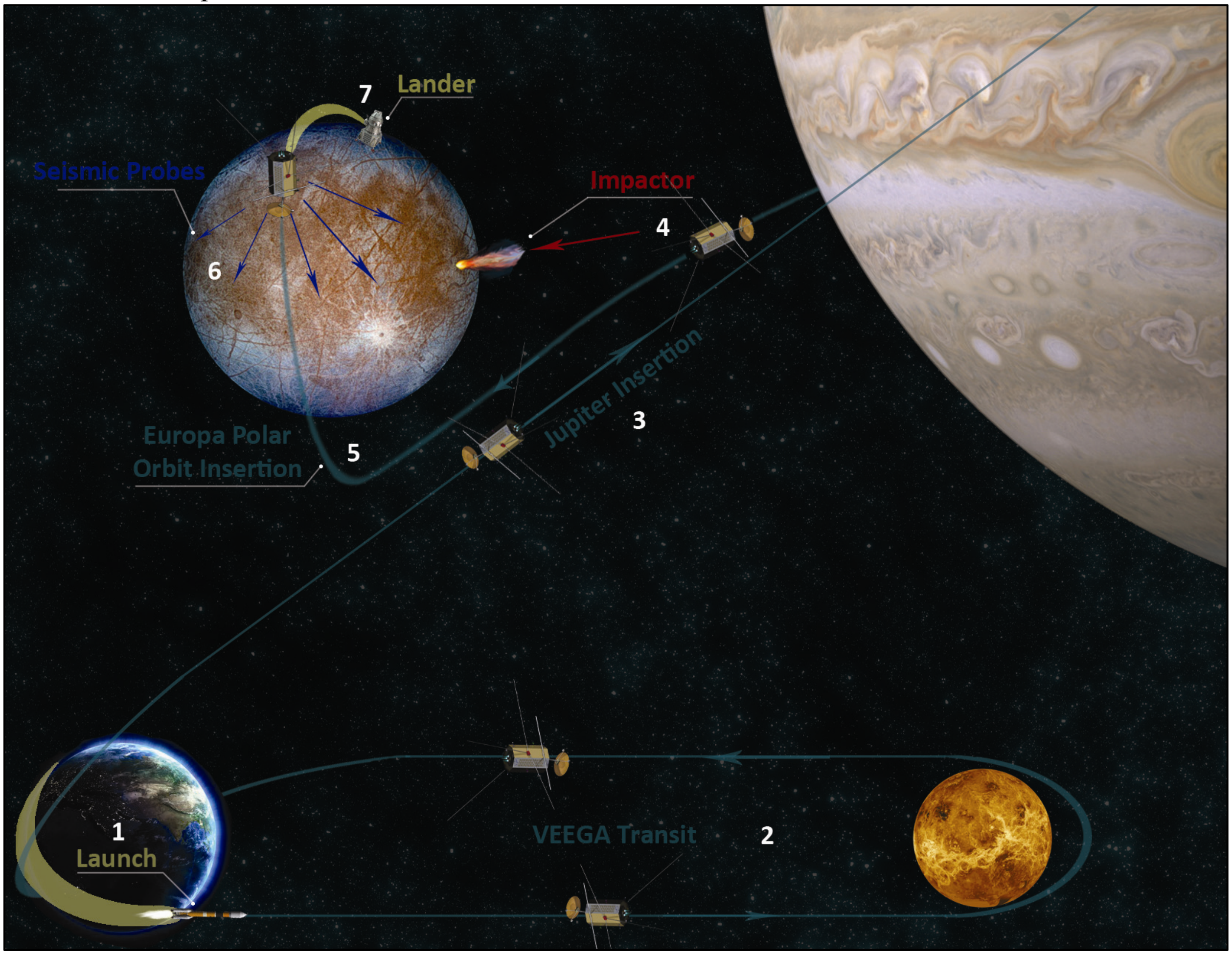


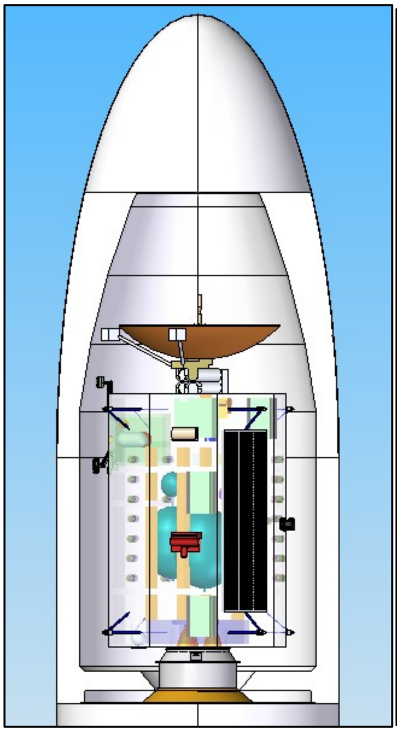
Figure 27 Concept of Operations

F.2.2 Launch Vehicle Compatibility

The Atlas V 551 launch vehicle, operated by the United Launch Alliance (ULA), is utilized to facilitate the spacecraft reaching orbit and traveling to Europa. Atlas Launch System Mission Planner's Guide is provided by NASA and ULA to aid the design teams in preparing to use the Atlas V. The launch site is Space Launch Complex 41 via Cape Canaveral Air Force Station (CCAFS) in Florida. The spacecraft is designed to fit easily within the 5 meter shroud of the launch vehicle, with the footprint of the spacecraft fitting within a 4.35m circle. The payload capability of the Atlas V 551 is 4615kg, and the total wet mass of the spacecraft is 4517kg, including an average of 29% contingency. For additional launch flexibility, the threshold mission can still be met without the utilization of the lander onboard the spacecraft. The mass of the spacecraft without the wet mass of the lander and without the necessary fuel to carry the extra mass of the lander is 4116kg including contingency. The payload capability of the Atlas V 541 is 4205kg.

F.2.2.1 Vehicle Shroud

The launch vehicle decided upon for this mission is the Atlas V 551. This is a high class, five meter fairing vehicle. Given a C3 of 14.7 km^2/sec^2 his launch vehicle provides a total payload mass of 4615 kg, which is used for the scientific mission. The spacecraft designed for this mission must be able to fit within the shroud provided for the Atlas V 551 and must be able to be attached to the adapter provided. Figure 28 ATLAS V Faring shows a CAD rendering of the shroud, including the ICESSS spacecraft containing the lander. The spacecraft must be attached to the adapter via the separation ring and must fit within the space provided by the shroud of the vehicle. The CAD model to the left demonstrates attachment to the vehicle while the CAD model to the right demonstrates the spacecraft fits within the shroud for the Atlas V 551.



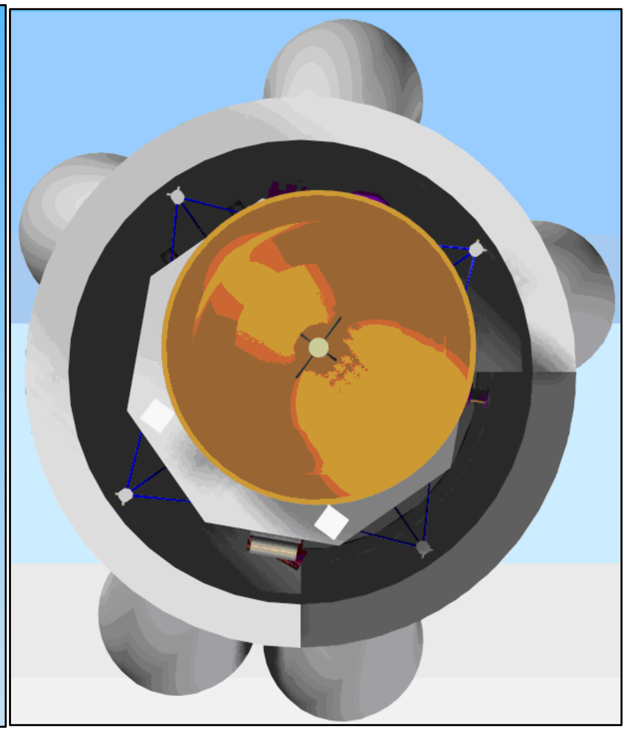


Figure 28 ATLAS V Faring

F.2.3 Flight System Capabilities

F.2.3.1 Orbiter

The Inner Crustal Europa Seismic and Spectral Surveyor (ICESSS), designed by Team Eureka at University of Alabama in Huntsville (UAH) has a total dry mass of 2010kg, including contingency. This value includes the dry mass of several deployed systems, including an impact device, 29 seismic probes, and a lander. The lander is covered in more depth in F.2.3.2 Lander. The mass budget outlined in Table 19 shows the mass of each subsystem for the orbiter. The science instrumentation onboard ICESSS is described in Section E.1. All other subsystems are described in detail in subsections of this section. Figure 24 Figure 24 ICESSS Deployed Spacecraft displays ICESSS in its fully deployed state. As shown, there are several antennae and booms deployed from the orbiter. One of these is a 10 meter boom which holds two magnetometers for investigation the induced magnetic field of Europa. Other extended devices include a 10m dipole antenna which is used by the Europa Orbiter Radar Sounder, several antennae deployed for the Plasma Wave and Radio Science instrument, and the telecommunications antenna. Figure 29 shows how the various subsystems of the orbiter interact with one another.

Table 19 Mass Breakdown

Dry Mass Budget (in kg)			
System	Current Estimate	Maximum Expected Value	Contingency
Structure	345	493	30%
Thermal	67	96	30%
Propulsion/RCS	187	260	28%
ACS	48	69	30%
Power	115	164	30%
Telecommunications	67	96	30%
Computer Data System	47	63	25%
Science Instruments	130	173	25%
Radiation Shielding	140	200	30%
Lander	130	186	30%
Seismic Probes	149	213	30%
Impact Device	40	50	20%
Student Collaboration	25	36	30%
Total Dry Mass	1489.6	2097	29.0%
Propellant Mass Budget			
Orbiter Fuel (Hydrazine)	1168.3	Orbiter Oxidizer (NTO)	1252.1
Total Propellant Mass			2420.4
Total Wet Mass (including contingency)	4517.0	Launch Vehicle Capability	4615
Additional Margin on Launch Vehicle:			98

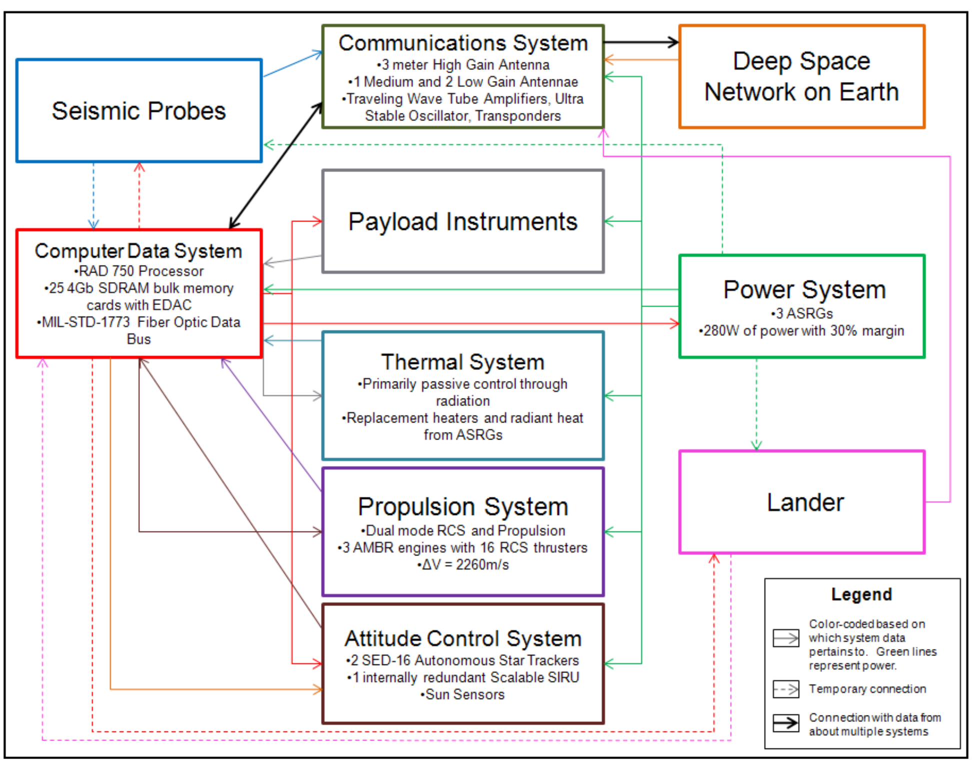


Figure 29 ICESSS Block Diagram

F.2.3.1.1 Structures

The structural design of the orbiter must be able to accommodate the internal and external components of the orbiter. The structural subsystem includes the internal and external subsystems layout, mechanisms, and deployments. 2014-T6 Aluminum is used for the primary framework of the spacecraft main bus in an octagonal layout. This layout can optimize the amount of surfaces for scientific instrumentation, thermal regulation, communication, propulsion, and component configuration.

The internal structure/layout of the spacecraft bus must be arranged in such a way as to shield components from radiation, regulate temperatures, limit cable mass, and minimize surface area. This can be accomplished by compartmentalizing and layering subsystems and components, and optimizing the layout of the subsystems in such a way as to limit any excess volume. The geometric center of gravity must be taken into account when designing the internal structure.

The octagonal spacecraft bus must accommodate the 3 AMBR engines, propulsion tanks, and support structure for the engines. This can be accomplished by having a hollow center in the spacecraft bus. This allows for mounting of the engines and all of the components in line with the center of gravity of the orbiter. The truss support structure for the AMBR motor mount is utilized in order to propel the spacecraft via thrust transfer to the spacecraft.

The aluminum framework for which the spacecraft bus is built around is the most critical part of the structural subsystem. At launch the orbital spacecraft is subjected to a vertical acceleration of 5.5 g's (Guide 2010). This is the maximum amount of force the spacecraft is subjected to therefore the structural framework must be designed to accommodate such a load to a certain degree of safety greater than two. The total launch mass of 4590 kg is taken into account to meet the design requirements. Multiplying the launch mass by the vertical acceleration of 5.5 g's yields a maximum force of 247,653.5 N. Using this force and the yield strength of 2014-T6 Aluminum the minimum area of a support column can be found. The resulting area is a cross section of 6 cm^2. Using a factor of safety of 3 gives each supporting column a cross sectional area of 18 cm^2. Figure 30 of the designed orbiter in proves satisfaction of these design parameters and exaggerates the deflection during the applied loads to indicate locations of greatest impact.

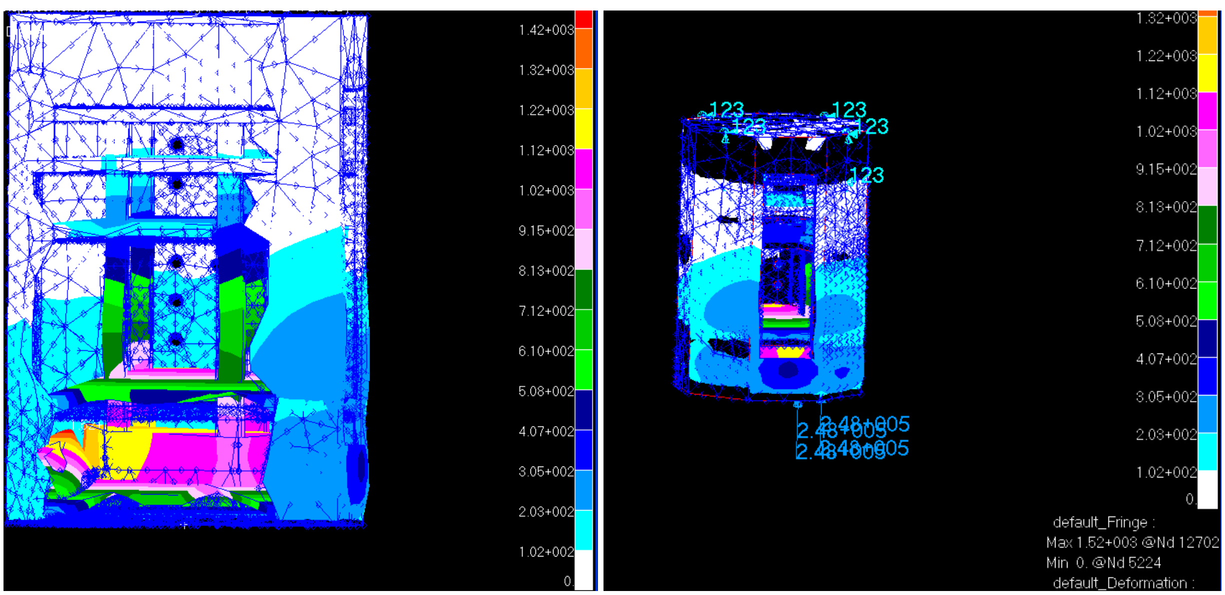


Figure 30 Finite Element Analysis

F.2.3.1.2 Power System

The power source for the orbiter consists of three Advanced Stirling Radioisotope Generators (ASRGs). The 3 ASRGs comprising the power system are capable of providing 401W at end of life. ICESSS is predicted to operate at or below 280W at all times. This allows for slightly over a 30% power margin for the orbiter. The power profile of the orbiter includes 125W of power allocated to the scientific payload on average, and 50W allocated to communications on average. When the telecommunications system is in heavy use and is downlinking its scientific data to Earth, it is projected to draw 85W of power, and the scientific payload is limited to 90W of power use. The power allotments of other subsystems are not dependent on the power use of the telecommunication system, and remain unchanged when the orbiter is sending scientific data to Earth. In safe mode, which is activated when a problem with one of the orbiter's systems is detected, scientific data collection stops, communications are utilized to transmit engineering data to Earth and receive commands from Earth.

Table 20 Power Budget shows the estimated power budget for the spacecraft during normal operations.

Table 20 Power Budget

System	Standard Operations	Telecom Operations	Safe Mode
Scientific Payload	125	90	0
Thermal	26	26	26
ACS	49	49	49
CDS	25	25	25
Communications	50	85	85
Propulsion	5	5	5
Total Power (W)	280	280	190

F.2.3.1.2.1 ASRG

The ASRGs use Pu-238 to produce electrical power. For the purposed Europan mission, three ASRG are used for power generation. As specified in the Discovery Announcement of Opportunity NNH10ZDA007O, two ASRGs are provided. The third ASRG poses an additional cost of 27 million dollars. Table 21 displays ASRG performance parameters obtained from "Space Radioisotope Power Systems" (Nuclear 1-2). Table 21 is the CAD drawing of the ASRG. The ASRG dimensions are 50cm x 50cm x 80mm. (Nuclear 2008)

Table 21 ASRG Performance Parameters

Power at launch	Power Degredation	Mass	System Efficiency	Lifetime	Operating Temperature
145-155W	0.8%/yr	32kg	27%	14yrs	90-850C

Each ASRG is projected to produce 145-155 Watts of power at launch using less than 1 kg of Pu-238 fuel. The power degradation of the ASRG is approximately 0.8 percent loss per year with a lifetime of 3 years storage plus 14 years operation. ASRGs have a system efficiency of greater than 31 percent. The mass of each ASRG is approximately 32 kg. Including 23 kg of cabling, the power system mass allotted for this mission is 115kg leaving 23 kg in excess for cabling and mounting fixtures. The operating temperature of the power system is 90C to 850C. Since the internal temperature of the space craft had a minimum temperature of 204C and a maximum temperature of 519C, the ASRG temperature is within the operating range. (Nuclear 2008). The temperature range expected for the ASRGs ranges from 300K to 400K. The temperature of the ASRGs is controlled by louvers. The louvers are placed on the wall near each ASRG, and control how much heat each ASRG radiates outward to space. This directly affects the temperature of the ASRG itself and the average internal temperature of the spacecraft. The louvers are designed to maintain a cabin temperature near 290K, ensuring that the hydrazine propellant does not freeze. The ASRGs are placed nearer to the hydrazine tank than the NTO tank, allowing the ASRGs to radiate heat to the hydrazine tank, ensuring that the hydrazine propellant does not freeze, while not increasing the temperature of the NTO significantly. The ASRGs are designed to be decommissioned without violating planetary protection.

F.2.3.1.3 Propulsion System

This mission utilizes the Advanced Material Bipropellant Rocket (AMBR) engine in a dual mode propellant system which handles primary propulsive maneuvers as well as reaction control maneuvers. This implementation utilizes hydrazine and nitrogen tetroxide (NTO) for the fuel and oxidizer respectively. The system is pressure fed with helium as the pressurant. The primary propulsion system utilizes three 200 pound AMBR main thrusters, to allow for a thrust to weight ratio of approximately 0.5 during the Jupiter orbit insertion burn. This high thrust to weight ratio effectively minimizes gravity losses during propellant burns.

Accounting for the reduced mass after the Jupiter Orbit Insertion burn, the thrust to weight ratio for the Europa orbit insertion is 0.7. This is a high thrust to weight ratio; however, there are benefits to allowing a0.5 thrust to weight ratio for the Jupiter orbit insertion burn. If one of the AMBR engines fails, the specific impulse of the system remain constant, with only the thrust and thrust vector being affected. This was a secondary consideration for utilizing three engines. In the event that one engine fails, thrust from the two remaining engines remains high enough to ensure low gravity losses during propellant burns. If this happens, the reaction control thrusters serve to counteract the change in the thrust vector caused by the loss of the engine. This increases the redundancy of the propulsion system.

The primary propulsion system has a specific impulse of 335 seconds and a delta-V requirement of 2260 m/s. The reaction control thrusters have a specific impulse of 210 seconds and utilize hydrazine fuel for a monopropellant system delivering 5 pounds of thrust. Sixteen reaction control thrusters are being used in order to retain 3-axis control in the event of failure of one of the thrusters. The delta-V budget for the reaction control system is 56m/s. This value is 2.5% of the delta-V requirement of the primary propulsion system. The mass of hydrazine for the propellant system is 1168 kg. This includes the fuel for both the primary thrusters and the reaction control thrusters. The mass of NTO for the propulsion system is 1252kg.

F.2.3.1.4 Attitude Control System

The attitude control system for the spacecraft is responsible for obtaining information regarding orientation and trajectory for the spacecraft throughout its mission. For this mission, a 7.1kg internally redundant Scalable Space Inertial Reference Unit (SSIRU) and two 2.9kg autonomous star trackers perform the functions for the attitude control system. The SSIRU has 4 hemispherical resonating gyros, insuring availability of three-axis knowledge in the event that one of the gyros fails. The SSIRU also utilizes two power supplies and two sets of sensor processors with input/output electronics, to ensure that loss of power to one of the inputs does not end the functionality for the SSIRU. The SSIRU requires 38W of power (Grumman 2011). Additionally, 4 0.3kg sun sensors are onboard the spacecraft to assist with orientation determination. The star trackers are not aligned with one another, and are not located near one another on the spacecraft. This allows redundancy in the event of failure of one of the star trackers, and ensures that a single event impact with a micrometeoroid does not disable both star trackers. Each star tracker uses 10.7W of power on average (SODERN 2006).

F.2.3.1.5 Telecommunication System

The telecommunications system of the orbiter is comprised of a 3 meter diameter high gain antenna which is mounted on a boom and articulated in order to maximize the ability of the orbiter to communicate with the Deep Space Network on Earth. The high gain antenna is capable of transmitting on both X-band and Ka-band frequencies. Additionally, one medium gain antenna and two low gain antennae, all operating on X-band frequencies, is onboard the orbiter. Redundant cross-strapped X/Ka-band Small Deep Space Transponders (SDSTs) will allow for radio frequency transmission and reception, as well as radio metric functions. Redundant cross strapped Traveling Wave-Tube Amplifiers (TWTAs) is used to increase signal strength for both Ka and X-band frequencies, to provide higher data transmission capabilities while remaining within power constraints.

An Ultra Stable Oscillator (USO) and a Ka-band transponder is included in the telecommunication system for use in radio science investigations. When sending scientific data back to Earth, the telecommunications system requires 80W of power. When the system is not communicating with Earth, it utilizes 30W in order to remain capable of receiving uplink commands. In order to ensure that the mass memory capacity of the spacecraft is never exceeded, the telecommunications system must be capable of transmitting all of the data collected each orbit to Earth within the one hour communications window available each orbit. The scientific downlink rate of the communications system is approximately 150kbps.

As shown in Table 22 below, the maximum combined data rate for the orbiter, if all of the scientific instrumentations are operating at once, is approximately 41Mbps. Based on the 150kbps science downlink, assuming a compressing ratio of 75% for images and 90% for test data, for every second of data collection, the telecommunications system requires 65.3 seconds to transit all of the data back to Earth. For this reason, many of the scientific instruments are not collecting data continuously. For example, the wide angle camera of the OSIRIS camera package only collects data when over regions of Europa which it does not already have images for. In particular, as the camera crosses the poles of the planet, a spherical cap with a base diameter of 208km exists at each pole for which the OSIRIS WAC has complete data for after one half of a eurosol, or 1.75 days. Additional areas for which the camera already has coverage over the lower latitudes of the moon is encountered with increasing frequency as ICESSS spends more time imaging the Europa surface. The telecommunications system for the orbiter is modeled after the telecommunications system designed for the Jupiter Europa Orbiter Mission Study. This system estimates an average scientific data collection rate of less than 25Mbps (JEO 2009). For ICESSS, this translates into a design constraint requiring the scientific data collection rate to be approximately 21Mbps on average. This ensures that the communication system is able to return all of its data to Earth and operate with no net accumulation of stored data over time. This design constraint imposes a restriction on the amount of scientific instrumentation which can be operating at any one time. The OSIRIS NAC and WAC are the primary contributors to the scientific data collection rate, with each collecting approximately 19Mbps. The limitation on scientific data collection results in a general inability to operate the OSIRIS NAC and WAC simultaneously. This is not an unbendable rule, as ICESSS could maintain an average scientific data rate of 21Mbps by operating both cameras for a small amount of time followed by an equal amount of time during which neither camera is operated. Based on the limit on average scientific data rate, approximately 50% of the scientific equipment can be operating at any one time.

Table 22 Mission Data Rates

Instrument	Data Rate (bps)	Instrument	Data Rate (bps)
Europa Orbiter Radar Sounder	2000000	LIDAR	3008
Magnetosphereic Imaging System	7000	Micro Imagers on Orbiter (2)	18000
Radio Science	900	Athena Microscopic Imager	15000
Radio and Plasma Wave Science	900	Gas Chromatograph and Mass Spectrometer	4000
THEMIS	600000	LOLA	10000
Ion and Neutral Mass Spectrometer	1500	Fluxgate Magnetometer (2) on Orbiter	4000
Near Infrared Mapping Spectrometer	11520	Fluxgate Magnetometer (2) on Lander	110
Ultraviolet Spectrometer	144	Engineering data	2000
OSIRIS NAC/WAC	38348000	Uplinkcommands	500
		Totals (Orbiter)	41026582

F.2.3.1.6 Computer Data System

The computer data system for this spacecraft utilizes a radiation hardened single board computer (SBC). The computer selected for this mission is the SCS750A produced by Maxwell Technologies. This computer has a space-qualified performance of 1800 million instructions per second (MIPS). The device has 256 megabytes of volatile Synchronous Dynamic Random Access Memory (SDRAM). This memory is Reed-Solomon protected and has error correction. Additionally, each SCS750A has 8 Megabytes of non-volatile Electrically Erasable Programmable Read-Only Memory (EEPROM). For bulk memory storage, radiation hardened SDRAM is utilized. Based on the analysis discussed in F.2.3.1.5 Telecommunication System, the spacecraft is collecting and storing approximately 21 megabits of information per second, uncompressed. With a worst case scenario of 4.5 hours of time with no telecommunication ability occurring as Europa passes behind Jupiter and does not have a line of sight to Earth, the computer data storage must have 43 4-Gbit SDRAM memory units in order to store all of the collected data, if the data is collected uncompressed.

Due to the high bulk memory capacity required for scientific data to be stored uncompressed, it is necessary to compress the scientific data as it is stored. This is different from the more common method of compressing scientific data as it is being read out of memory to be sent to Earth by the telecommunications system. Assuming a compression of 75% for images and 90% for other data, the amount of 4-Gbit SDRAM memory units is decreased to 11. In order to compress the data for storage, additional RAM must exist for the raw scientific data to be temporarily written to so that it can then be compressed and stored. For this reason, an additional 5 4-Gbit SDRAM memory units are included in the ICESSS computer data system. This is far more temporary storage capacity than required to accommodate the raw scientific data before it is compressed; however, the excess memory serves as contingency in the total amount of data the spacecraft can store, making data loss less likely in the event that telecommunications are ever eclipsed for more than the 4.5 hours projected to be the worst case scenario for the system. The end result is that utilizing a CDS which compressed the scientific data as it stores the data allows for a system with less than half as many SDRAM memory units, and more contingency in the amount of total storable scientific data.

The computer data system utilizes the small explorer data system (SEDS) MIL-STD-1773 fiber optic data bus for data transfer from between instruments. This implementation was chosen for its light weight design. The mass of the Rad750 SBC is 1.5 kg, and it requires between 7W and 25W of power during operation, with an average of 10W (Technologies 2010). The SDRAM memory units requires a maximum of 2.16W each while being read from and written to and have a mass of 0.022kg each (3D-Plus 2008). 3 power converter units, each with a mass of 15gm are included in the computer data system, as well as a science data processor and an engineering data processor. The science data processor has a mass of 15kg and requires 16W of power. The engineering data processor has a mass of 10kg and requires 5W of power. 30kg of mass is allotted for miscellaneous cabling and interfacing components. The total mass of the computer data system is 46.8kg and the power requirement for the system is 51W.

F.2.3.1.7 Thermal System

The thermal system for this mission is designed based on a preliminary design process for thermal subsystems. The temperature limits of the payload usually dominate the design of the subsystem. For example, batteries usually have narrow temperature limits. By assuming an isothermal and spherical spacecraft, first-order estimates of the spacecraft thermal performance can be made. This is done by determining the diameter of a sphere whose surface area is equal to the total outer surface area of the actual spacecraft. (Brown 2002) Polished aluminum (6061-T4) was chosen as the skin material for the spacecraft due to the relatively light weight and relatively low absorptivity/emissivity ratio. It's also used for the majority of the structure.

The spacecraft worst-case hot temperature is calculated from spacecraft orbit around Venus in direct sunlight during the VEEGA transfer. The value of Tmax is 519K which is well above the non-operating temperatures of the payload. The average temperature that the payload needs to be at during the mission is 270K. In an effort to reduce the heat received at Venus. A heat shield is used to block the sun's light while in VEEGA transit. The maximum mass of the heat shield is 60kg and can reduce the Tmax to a minimum of 258K assuming all sunlight is blocked. The shield is composed of MLI with solar reflective paint and covers the side of the spacecraft which is facing the sun. The shield is not released after passing by Venus, as implementing a system to jettison the shield add mass and complexity to the spacecraft. The spacecraft worst-case cold temperature is calculated from orbit around Europa, while in Europa's shadow of Jupiter. This eliminates the thermal radiation emitted from Jupiter. The value of Tmin is 205K. By using multilayer insulation, the power lost at the cold scenario is reduced to 21W. The multilayer insulation consists of 24 layers with separators. The mass of this thermal blanket is .25kg. To counter the heat loss, 2 heat pipes are utilized to transfer heat from the ASRGs to the scientific instrumentation. These pipes are 4m long and have a mass of 1.32kg each. The masses of the heat pipes and thermal blanket are calculated based on equations from Charles D. Brown's <u>Elements of Spacecraft Design.</u> By using thermostats, the temperature of the spacecraft can be monitored. The ASRGs produce most of the heat inside the spacecraft. The ASRGs are positioned to allow them to radiate some of their heat to the propellant tanks in order to ensure that the propellants remain above their melting point. Louvers are placed adjacent to the ASRGs to remove excess heat from the spacecraft when needed. The mass of the louvers is 3.81kg. The total mass of the thermal subsystem is approximately 67kg, which is 11kg under budget for our spacecraft. (Brown 2002)

F.2.3.1.8 Seismic Probe Deployment Devices

As shown in Table 19, there is a seismic probe system onboard ICESSS. The mass for this system includes the mass of the seismic probes and their deployment devices. Each of the 29 seismic probes has a mass of 3.57kg and is mounted onto the orbiter within a deployment canister which has been designed for this mission. The walls of the canisters are 3.175mm (1/8") thick, and each canister has a mass of 1.17kg. Each canister utilizes a compression spring and thermal knife in order to deploy its seismic probe to the Europan surface. The spring in each canister is sized to provide approximately 53 pounds of force to eject the seismic probe from the canister and away from ICESSS with a velocity of

2.5m/s. Figure 31 depicts the deployment canister system. The deployment canisters are positioned to ensure that the seismic probe deploys from the orbiter without impacting any deployed booms.

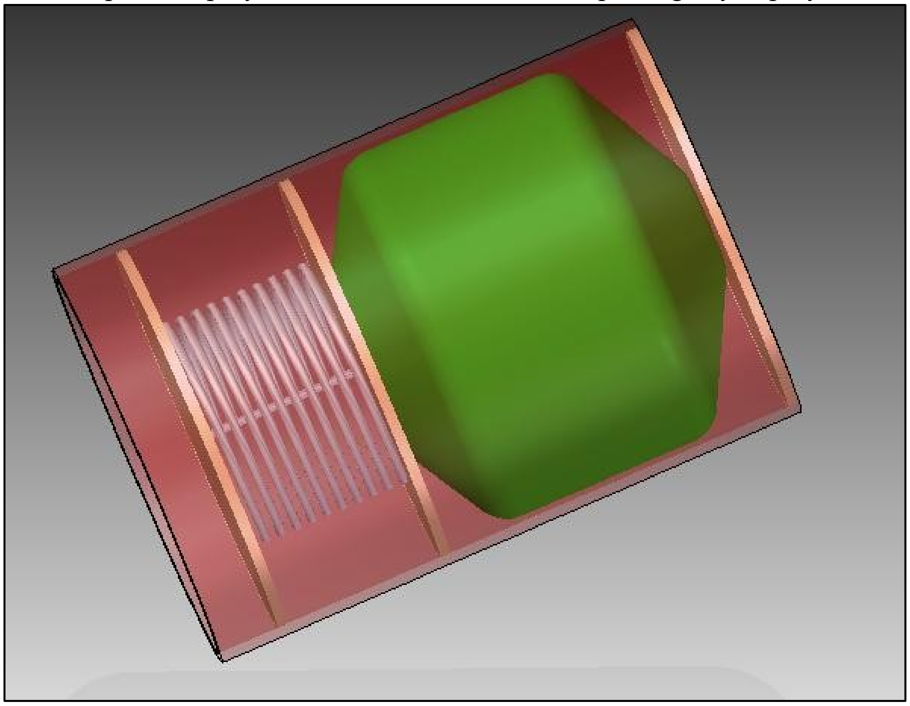


Figure 31 Seismic Probe Deployment Canister

The thermal knife in each canister is designed to counteract the force of the spring throughout the mission until it is time for the seismic probe to be deployed. When the time arrives, electrical power is given to the thermal knife, allowing it to melt the cord whose tension counteracts the compressive force of the spring. Once the cord breaks, the spring is released and forces the seismic probe through the thin covering at the top of the canister. The scientific instrumentation within each probe and the scientific investigation enabled by the probes are discussed in Sections E and D, respectively.

F.2.3.2 Lander

The observations made by previous missions have shown that the surface of Europa was mostly a sea of ice. Indeed, Europa has the smoothest surface of the solar system: the terrestrial spectral observations reveal that its surface is composed for the most part of water ice; also note the cracks and scratches, with relatively few craters. In response, the stability of the Lander during the mission and the landing has become an issue. ESTACA consequently researched for a solution to maintain the Lander and secure it in place to prevent any risk of slipping or tipping over on landing during the collection and also for the transfer of information to the orbiter. Figure 32 presents a possible configuration solution.

The external environment is subjected to an intense magnetic field and at very low temperatures. Radiation is 540 rem per day or 104 times higher that the acceptable level. The Lander remains 12 hours on Europa, and then undergoes a dose of 270 rem. The Structure presented in Figure 32 is expected to be able to hold up to such conditions on Europa.

The lander, called Robotic Reconnaissance Deployed Device (R^2D^2) designed by Ecole Supérieure des Techniques Aéronautiques et de Construction Automobile (ESTACA), must be designed to withstand the extreme conditions on Europa. It must be able to land gently, stabilize quickly, with stand radiation and cold, and to take samples. When the masses of the fuel and instruments were established, the specifications require a mass of 67.9 kg for the structure of R^2D^2. The mass is an important criterion in selecting and sizing materials. All masses is calculated based in a terrestrial environment.

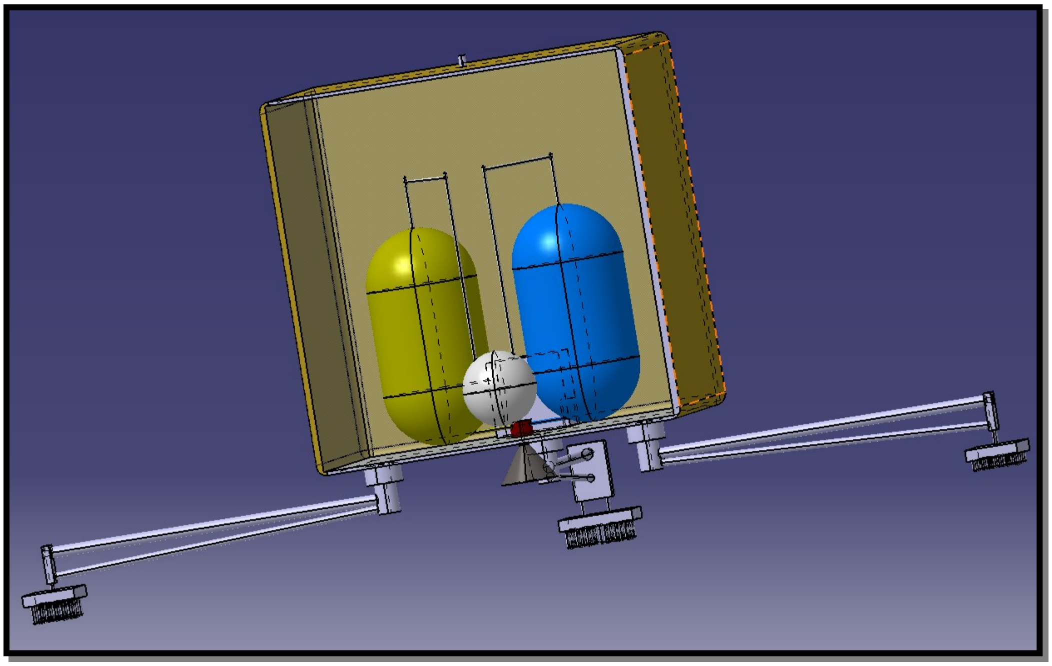


Figure 32 Lander deployment configuration

F.2.3.2.1Previous model

ESTACA first conducted a preliminary study with the main purpose of resistance to extreme conditions of Europa. The instruments are the most sensitive, so a capsule made of 5 mm in mu-metal, which is an alloy of nickel and iron chose to protect. The choice of this alloy was made because it has a very high magnetic permeability, which allows it to attract the magnetic field lines. It is therefore an excellent material for deflecting magnetic fields. The main structure was a cube of 2 millimeters thick composite made of titanium. The choice of this material is explained later.

In the end, the materials used are very dense, especially the mu-metal with 8813.7 kg / m^3. Even if the size of the R²D² is reduced significantly, it does not satisfy the specifications which impose a total mass of 130 kg. Therefore the team had to change materials by taking the risk that materials used have less performance.

F.2.3.2.2 Current model

However the current model remains very close to the former model. The team decided to keep a protective cap for instruments and a cubic structure except that now these two parts are aluminum. The rest of the model does not change. Inside the cube are the following: the capsule, tanks and batteries. R²D² has three feet consisting of shock absorber evenly distributed from the center of gravity. At the end of each leg, there is a device sheet with cleats order to set R²D² in ice and prevent slipping.

F.2.3.2.3 Propulsion system

One of the main specifications of the lander is to lay R²D² on the surface of the moon of Jupiter with zero speed (0g). This specification led to some choices regarding the mode of propulsion of our R²D² and the weight that this system represents. In fact, the weight restriction imposed by the specification has driven many of the technological choices. Liquid propulsion has emerged due to the benefit that shows the multiple ignition of the engine. The propulsion system consists of two cylindrical fuel tanks, a spherical tank of helium and a nozzle. Each of these parts is made of titanium composite. The most common is the alloy Ti-6Al-V4. A mixture of titanium, aluminum and vanadium. It has an excellent mechanical strength, its strength to weight ratio is very high. It can also withstand temperatures up to 900 C °, hence its use for the nozzle. Each fuel tank is 30 cm in diameter and 60 cm in length. With a thickness of 2 mm a mass of 12.3644 kg was obtained for the two reservoirs.

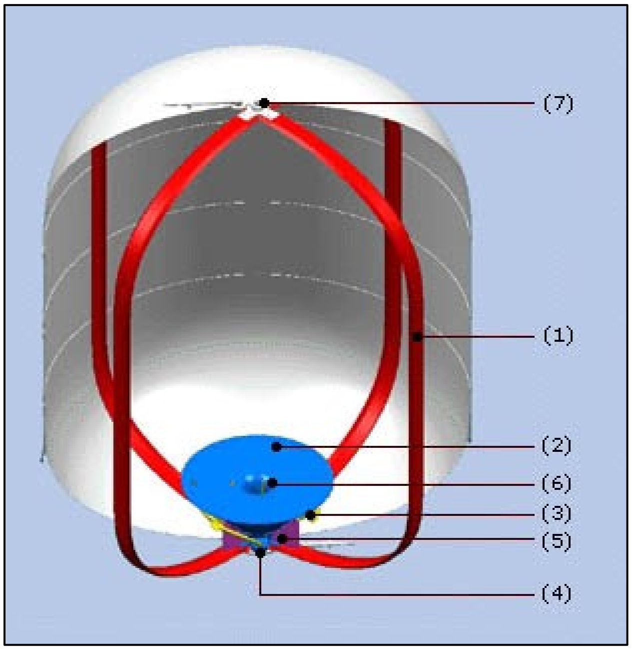


Figure 33Propellant Tank

Propellant Tank
1. Propellant Acquisition Vanes
2. Propellant refillable reservoir
3. Upper screen
4. Lower screen
5. Propellant port
6. Venting tube
7. Gas port

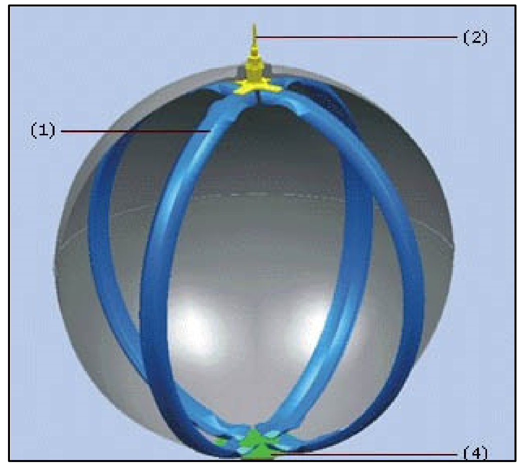


Figure 34 Helium Tank

The helium tank is 18.42 cm in diameter. With the same thickness, we get an empty weight of about 922 g and 1.2687 kg with helium.

F.2.3.2.4 Structure

The main structure is shaped like a cube supported by three feet. They are each independent and equally distributed. They must be able to withstand the impact with the ground, even if it occurs at a speed of 0 m/s. The feet allow a soft landing and stabilize the module. Titanium is a material that is too expensive and dense; therefore the entire structure cannot be designed with this material. Another choice is to use aluminum. It provides a lightweight structure and protection against heat radiation. To have a high rigidity for an extremely low weight, aluminum honeycomb is sandwiched between two sheets, also in aluminum. The cube is 80 cm long and wide, with 70 cm in height and 3 mm thick, assume that each part of the volume is aluminum. Do not take into account the hollow honeycombs. The mass of the empty cube is therefore 28.2775 kg.

F.2.3.2.5 Instrument Protection

Instruments are the most vulnerable to radiation, they must be protected. For that we achieve a kind of "safe" in which they are placed. This capsule serves as protection against magnetic radiation but can also serve as thermal protection. This shield is also made of aluminum. It is considered that a thickness of 7 mm is sufficient to protect the scientific equipment. In addition, the Lander capsule is 3 mm thick, so there are a total thickness of aluminum of 10 mm between the instruments and the outside. We have a shield thick enough to limit the radiation dose. Instruments must take samples of the surface of Europa. To achieve this sampling, was performed a sliding opening on the side to release a telescopic arm. To contain all the instruments, the capsule made 20 cm square. We have a volume of 4.2242 kg.

F.2.3.2.6 Thermal Protection

Due to the very low temperature in Europe, we need to protect the Lander and its equipment. The requirement is that it must operate at temperatures as low as 70 K.°. A passive solution is to coat the walls of the structure by insulating gold paint. To ensure additional protection we can put a layer of Aerogel insulation inside the walls. In addition, batteries and the onboard computer is placed inside the capsule containing the instruments, so they produce heat which will also contribute to limit the temperature.

F.2.3.2.7 Conclusion

R²D² able to make the mass budget and weighs less than 118 kg, 12 kg of it remains available that can be used for a robotic arm and structural elements that we have not taken into account.

MASS BUDGET	
Name	**Mass (kg)**
Instruments	5.602
Fuel	57.1
Protective capsule	4.2242
Helium tank	1.2687
Propellant tanks	12.3644
Lander capsule	28.2775
Battery	5.2
Landing gears	6
Nozzle	1.64
Total	**121.68**

If there are more question on the Lander or details required. They can be found with answers in Section J.10.2 ESTACA's Telecommunication's Calculations and J.10.3 ESTACA's Trajectory and Propulsion Calculations.

F.2.4 Additional Mission Elements

The InSPIRESS Level I and Level II teams develops two additional mission elements, which are explained in detail in I.2 Student Collaboration and F.2.6 Impact Device respectively.

F.2.5 Budget Information

The mass contingencies and margins for the spacecraft systems are shown in Table 19 in F.2.3.1 Orbiter. As discussed in Section F.2.3.1.2 Power System 30% margin is included in the power budget. The propellant required for the mission by the primary propulsion system is 2295kg. This mass of propellant is capable of transporting 4615kg, the maximum mass which the launch vehicle can accommodate, to the destination. A 5% ullage volume and 1% residual propellant are included in this propellant mass. A propellant management device is included in the propulsion system to reduce residual propellant. This device adds an extra 10% of the mass of each tank to the mass of the propulsion system.

F.2.6 Impact Device

The Sparkman High School Level II team was charged with the responsibility of creating an impact device that would penetrate the theorized icy surface of Europa in an effort to study its composition. The main goal was to determine whether or not the Galilean moon is capable of sustaining life—the proposed ejected plume following the impact would aid a team of scientist to determine the composition of the icy shell. By characterizing the shell, scientist can determine the properties of the subsurface ocean and generate a formal conclusion that would state the possibilities of Europa holding life. The high school team—aided by UAH—was to be responsible for the entirety of the impactor design and its mathematical components. In order to determine how the impactor was to be designed, the high school team was required to conduct an extensive amount of research involving materials, shapes for the impactor, and the mathematical concepts that govern the behavior of the plume.

For this reason, the high school team was divided into 2 sub-teams: The Design and Equations teams. The design team was to determine—after reviewing mathematical data proposed by the equations team—the desired dimensions and material for the impactor design. Using the data from the equations team, the design team was to begin a CAD drawing of the impactor and the mount/release mechanism for it. The equations team was to derive a set of equations in order to determine the behavior of the plume. Although this was the main objective, the team was required to analyze and calculate the depth of the penetration and the shape of the resulting crater in order to gain a more accurate understanding of the plume itself. For this reason, the process was divided into three stages: The calculation for penetration depth in relation to four chosen materials, the calculation of the plume rise and density (merely a ratio proposed of two conic functions that highlighted the area where mass was ejected in relation to the orbital height), and the design of the impactor itself.

To calculate for depth, the equations team used the C.W Young/Sandia penetration equation ice and frozen soil at a velocity exceeding 60 m/s. The equation analyzed a given cross-sectional area in conjunction with elemental values such as impact velocities and gravitational accelerations to calculate penetration depth. This equation presented a challenge since the impactor needed to be proportional to the mass and the proposed material's density. The students determined each variable value adjacent to each material's properties and effectively formulated a design that would yield an acceptable result in depth—since depth of penetration would relate to the plume behavior. These values are recorded below in relation to the four best-performing materials. For the penetration equation use Table 23 and Table 24.

Table 23 Results from Penetration Equations

Variables	Units- SI	Iron	Lead	Tungsten	Platinum
D (Calculated Depth)	m (meters)	4.15	4.83	5.85	6.20
S (Target Penetrability)	N/A	2.75	2.75	2.75	2.75
N (Nose Performance Coefficient)	N/A	0.65	0.65	0.65	0.65
M (mass)	kg (kilogram)	30	30	30	30
v_f (Impact Velocity)	m/s	2210.27	2210.27	2210.27	2210.27
L_n (length of impactor nose)	m	0.142	0.125	0.1065	0.1015
A (Cross-sectional Area)	m^2	0.063	0.049	0.036	0.032
d (Impactor diameter)	m	0.284	0.250	0.213	0.203
g (gravitational acceleration)	m/s^2	1.314	1.314	1.314	1.314
v_i (release velocity)	m/s	2150	2150	2150	2150

Table 24 Material Given

Material	Mass (kg)	Volume (m^3)	Density (kg/m^3)
Iron	30	.0038	7860
Lead	30	.0026	11350
Tungsten	30	.0016	19250
Platinum	30	.0014	21450

From these calculations, the equations team used two methods to develop a general understanding of the plume. First, they calculated the diameter of the resulting crater by comparing the impactor's diameter and its kinetic energy in joules with the crater's diameter and conversion of that energy. Also, Pi-scaling was used to approximate the values of the crater's diameter. The resulting crater would determine the "volume" of the ejecta. With the aid of the UAH personnel, the high school team determined a rather efficient way of calculating the amount of mass that would be present in the ejecta of mass after penetration: by assuming that the ejecta would form an ideally perfect conical plume, the amount of mass in such a given plume could be measured as the ratio of the volume from the penetration site to the surface and the surface to orbital height. To create a comparison, the team evaluated the volume based on 45° slants and at the actual crater diameter. In essence, by determining the volume of mass ejecta at the crater site, the students calculated the amount of mass that would be ejected upward and therefore, the height of the plume.

Below, Table 25, are recorded values of crater diameters relative to each material and resulting plume behavior.

Table 25 Recorded Values of Crater Diameter and Height

Material	Diameter (measured in meters (m)) of the resulting crater	V_2 (Volume (m^3) of crater at 45° slants)	V_3 (Volume (m^3) of ejecta at 45° slants)	Plume Height at 45° slants (m)	V_{2d} (volume of crater in relation to diameter)	V_{3d} (volume of ejecta in relation to diameter)	Plume Height in relation to diameter (m)
Iron	2.63	74.85	7.48×10^5	85.25	7.51	7.51×10^4	4166.81
Lead	2.7	118	1.18×10^6	99.23	9.22	9.22×10^4	3773.29
Tungsten	2.84	209.65	2.1×10^6	120.18	12.35	1.24×10^5	3446.84
Platinum	2.83	249.58	2.5×10^6	127.4	13.00	1.30×10^5	3229.40

As seen on the table above, the angles created a significant gap in the calculation of the plume height. Also, it was determined that the height of the plume would be proportional to the mass ejected and the density of the object. The deeper the impactor went—and the denser it was—the lesser the plume height; however, the mass ejected would rise significantly.

Based upon the data resulting from the Sparkman team's analysis, UAH determined that the impact device would not satisfy the science requirements describe in Section D. This conclusion was a result of the low height of the plume, which presents a twofold problem. First, the height of the plume affects the ability of the scientific instrumentation to measure the composition of the plume. A plume only 4.2km high is not high enough for the scientific instruments to properly analyze. Additionally, the low height of the plume limits the amount of time the plume remains in the air. Knowing the local gravitational acceleration of Europa, ejecta particles at an altitude of 4.2 km descends back to the moon's surface in under 2 minutes. This is not enough time to allow the orbiter to reach Europa and view the ejecta.

F.3 Development Approach

F.3.1 System Engineering Approach

Team Eureka used a top-down approach to designing its mission. The first step was to determine technical requirement definitions for the overall mission. There are functional requirements, environmental requirements, and AO requirements. Functional requirements outline the basic functions the mission must perform, see Figure 35. Environmental requirements outline the internal and external environments. Internally, the science equipment must be capable of maintaining proper functionality. Externally, the spacecraft will come into contact with atmospheres of Earth, Venus, Jupiter, and Europa. Each environment must be compatible to maintain functionality. The Discovery Announcement of Opportunity outlines specific NASA requirements that must adhere to throughout the mission.

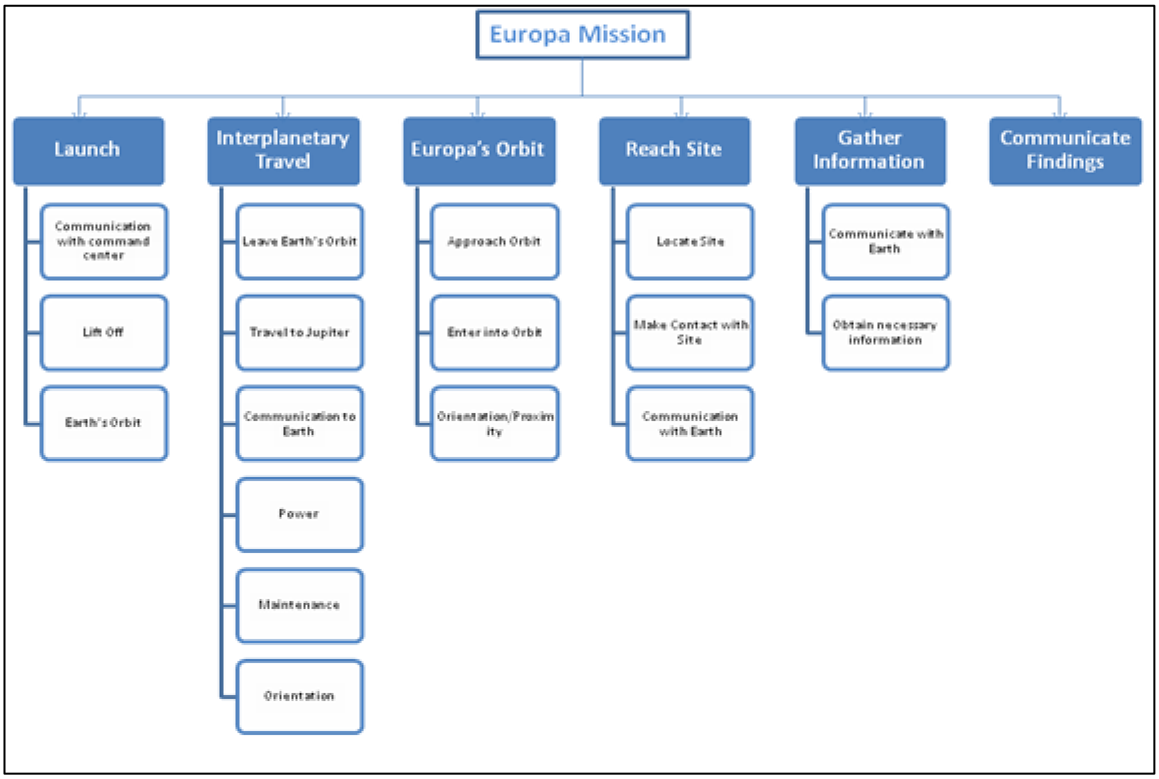


Figure 35 Mission Functional Analysis

The second step was the mission design. The science objectives were established as outlined in SCIENCE TRACEABILITY MATRIX. The equipment needed to achieve the science objectives were chosen. Each function of the mission was assessed to determine where decisions needed to be made. A decision was needed if there were multiple alternatives that could be used to achieve the same goal. The formal decision making procedure is outlined in Section G.1.2 Decision-Making Procedure and Guidelines. Each alternative was reviewed to ensure conformity to the technical requirements.

Once each element and equipment was determined, the cost analysis was performed. The cost analysis determined the total cost for the proposed mission. The total cost of the proposed mission was compared to the allowed budget. If the proposed mission cost more than the allowed budget, elements were analyzed to determine a revised mission to reduce cost.

F.3.2 Mission Assurance Approach

F.3.2.1 Faults

A risk analysis was performed on mission components, and these can be seen in Section G.3 Risk Management. Also, mitigation strategies have been implemented in the Table 28, which reduces the likelihood for catastrophic failure if problems occur.

F.3.2.2 Product Assurance

The mission utilizes high gain antennas to transmit the data. To ensure that the data collected from the science instruments is relayed earth, the mission uses Ka-band radio frequencies for science data returns. Low gain antennas are used to beam back data in incase of a critical event, which includes a failure in the high and medium gain antennas.

F.3.2.3 Reliability

To insure success of the mission many subsystems have levels of redundancy built into the design, which permits failure of various components of the subsystem without compromising the integrity of the mission. Reliability is increased by redundancy that is built into the system components, but redundancy also increases the complexity of the system. Therefore, redundancy increases failure modes.

F.3.3 Instruments to Spacecraft Interfaces

There is a multitude of interfaces which dominate the design process, while every system interfaces with another in some way many systems must interface with multiple other systems. The responsibility of designing the interface lies within both subsystems which interact with one another. This allows for an order of redundancy designed into the system which ensures proper interaction between subsystems should one subsystem fail to address an interface design. Each subsystem engineer reports on any interface which is needed to complete their design. Interfaces is also tracked by the systems engineer to confirm compatibility and assure each subsystem becomes a part of the entire system.

The spacecraft structure is a major and very important interface as every subsystem and component on the mission must attach to the structure at critical interfaces. These interfaces are important as they position the component in the desired location and orientation on the spacecraft via integrated interface points. These interfaces must be capable of withstanding the forces seen at launch and prevent any component from shifting during the mission duration. The interfaces which attach the seismic probes, lander, and impact device must securely hold the devices until the appropriate release location is reached. Once the devices are released the release process must transition smoothly to prevent any undesirable trajectory and/or impact with any spacecraft component. Many interface points are determined by the subsystems which have been previously designed and integrated from prior missions. Other subsystem interfaces with the spacecraft may be designed simultaneously with spacecraft structure. One central hub of the subsystem interfaces is the Computer Data System (CDS). The CDS has unidirectional communication with the following subsystems: Communication, on-board science payload, attitude control, propulsion, thermal, power, and lander and seismic probes via on-board communication. The power subsystem also has a multitude of interfaces due to the power requirements of each individual subsystem. For clarification of subsystem interfaces and how the systems interact with one another refer to Figure 29.

F.3.4 Technical Readiness Levels

All technologies for proposed investigations must be at a established Technology Readiness Level of 6 or higher (AO 2010). For a detailed explanation of each level and the team’s common TRL terminology please refer to the NODIS Library’s NASA Procedural Requirements NPR 7120.8 appendix

J (NASA, NODIS Library's NASA Procedure Requirements NPR 7120.8 Appendix J 2008). Technical Readiness is ensured by using international resources through a NASA and ESA international competition (JEO 2009). The JPL developed Ka band translator just recently achieved level 6 in 2010 which clears it for this mission (JEO 2009). For a system to be considered compliant the overall estimate for a system's Technology Readiness Level is 6 or greater. The derivation for this value is demonstrated in Table 26 (NASA, NASA Systems Engineering Handbook 2007). This values impact on cost is discussed in Section H.3 Cost Model Inputs. The final judgment of unique instrumentation TRL was left to the individual team member researching the instrument. This generally lead to indications of previous successful mission flight such as Cassini or Galileo. TRLs were considered major figures of merit when conducting the trade studies discussed in Section F.3.5 Essential Trade Studies and determining the overall success of the mission. The Technology Readiness Level also was considered when establishing mass margins as seen in the EJSM (EJSM 2010). The Europa Orbiter Radar Sounder and Seismic Probes are discussed in Section F.4 New Technologies/Advanced Developments. In Table 26 is a layout of the levels each piece of Technology on the orbiter has achieved. Notice that the Radar sounder and seismic probes are at a level 3,4,5 which causes the whole orbiter system to be at a level 3,4,5 even though the rest of the elements are level 6 or higher.

Table 26 TRL Assessment Matrix

TRL ASSESSMENT

(Red)	Red = Below TRL 3		
(Yellow)	Yellow = TRL 3,4 & 5		
(Green)	Green = TRL 6 and above		
(White)	White = unknown		
X	Exists		

		Demonstration units					Enviroment				Unit Description				
	Concept	Breadboard	Brassboard	Developmental Model	Prototype	Flight Qualified	Laboratory Enviroment	Relevant Enviroment	Space Enviroment	Space/Launch Operation	Form	Fit	Function	Appropraite Scale	Overall TRL
Orbiter															Yellow
Communications System															Green
Ka Band Translator	X			X											Green
Power System															Green
ASRG					X		X		X	X					Green
Propulsion System															Green
AMBR Engine	X				X	X	X	X							Green
Computer Data System															Green
SCS750ATM SUPER COMPUTER					X	X	X	X							Green
Europa Orbiter Radar Sounder															Yellow
Magnetosphereic Imaging System										X					Green
Radio Science Instrument										X					Green
Radio and Plasma Wave Science															Green
THEMIS	X			X			X	X	X	X					Green
Ion and Neutral Mass Spectromete										X					Green
Near Infrared Mapping Spectrome										X					Green
Ultraviolet Spectrometer										X					Green
NAC/WAC															Green
LIDAR	X						X	X		X					Green
Micro imager(2)															Green
Penetrators	X				X			X							Yellow
Lander															Green
Power System							X	X							Green
Communications System															Green
Propulsion System							X								Green
Computer Data System															Green
Athena Microscopic Imager						X		X							Green
Gas Chromatograph							X								Green
Mass Spectrometer							X	X							Green
Drill						X									Green
Fluxgate Magnetometer (2)							X	X							Green

F.3.5 Essential Trade Studies

Trade studies have been performed on the launch vehicle, trajectory, and power systems. All trade studies were conducted based upon the Figures of Merit (FOM) the team decided were most important. The optimal alternative was chosen to be used in the mission for each system.

Trade studies need to be conducted on the OSIRIS NAC/WAC camera system, seismic probes, thermal materials, and radiation shielding because the team believes more efficient and cost effect materials can be research and compared to the chosen systems. Regarding the OSIRIS camera system, a trade study needs to be performed to determine if the high resolution of the photographs is more desirable than total camera coverage of the Europan surface. It is possible to completely image the entire Europan surface at a lower resolution, which would increase knowledge about the surface of Europa as a whole, at a cost of the knowledge which can be gained from the extra information on structure formation intrinsic to the high resolution photographs taken by the OSIRIS NAC/WAC camera system.

The seismic probes need to be investigated further to ensure that they is able to sustain the impact force imparted by the Europan surface as they impact the moon at an angle of 20 degrees from the horizontal. Based on this angle, the probes are expected to rebound off of the Europan surface rather than immediately penetrating into the moon. The probes continues to bounce along the surface, losing velocity until they have a low enough velocity in the horizontal direction to allow them to penetrate into the Europan soil. The effects of this indirect entry into the Europan surface on the ability of the seismic probes to obtain local surface density measurements need to be analyzed. If this entry method causes the probes to lose the ability to obtain surface density measurements as they penetrate the Europan soil, the result is a mild loss of utility for the probes. The loss is not enough to render the probes ineffective, as their primary function of quantifying the tidal flexing of Europa during the course of its 3.55 day orbit of Jupiter is fulfilled, and the local surface density measurement is only an additional benefit to the use of the probes.

Another trade study needs to be performed regarding the impact device designed by the Sparkman team. The high level analysis performed by the Sparkman team showed that the impact device may not be an viable system; however, after analysis of the equations and assumptions used by the Sparkman team, it was realized that the system should not be dismissed without first performing a more in-depth analysis of the ejecta plume. In particular, the shape of the crater created by the impact device is assumed to remain constant – a cone with a 45degree angle – as the shape of the impactor changes. It stands to reason that a long, thin object creates a cone with a different angle than a spherical object or short cylinder with a relatively large diameter.

In order to properly analyze the height of the ejecta, the shape of the crater should be taken into account, as a narrower cone results in particles being released at a velocity which is more nearly vertical than particles released from a 45 degree cone. Additionally, the amount of mass ejected from the narrower cone would change, affecting the velocity of the ejecta particles, as they are governed by the law of conservation of energy. Specifically, the product of the mass of the impactor and the square of the impactor's velocity directly relates the mass of ejecta particles and the square of the velocity of the ejecta particles. Thus, the mass of the ejecta is inversely proportional to the square of the velocity of the ejecta particles. This results in a desire to design the impactor to reduce the total mass of ejecta and reduce the angle of the cone representing the crater formed by the collision. This increases the height, visibility, and window of opportunity for studying the ejecta cloud.

F.3.6 Management Approach

A task is considered complete when the designed, fabricated with a successful testing evaluation for devices below the TRL of 6, and then assembled on the orbiter or lander respectively.

F.4 New Technologies/Advanced Developments

Modified versions of the penetrator probes that were used on NASA's Deep Space 2 mission is used as seismic probes to support specific science needs during the Europa Exploration Mission. Modifications to the probes include: aeroshell removal, atmospheric accelerometer removal, addition of extra batteries, and the addition of a seismic accelerometer. The aeroshell, which acted as a heat shield to protect the probes during descent through the Martian atmosphere was not needed due to the then Europan atmosphere. The atmospheric accelerometer was removed due to the extremely thin atmosphere of Europa, which renders the accelerometer ineffective. Batteries were added to seismic probes to increase the total life of the system to meet the science objectives of the seismic probe. A seismic accelerometer was added to the seismic probes to collect seismic data from the Europan surface. Additional divergences from the Deep Space 2 probes were made in order to make the system more robust, lowering the likelihood of failure. Each probe is designed as a monolithic device, compared to the Deep Space 2 probes which separated into two parts upon impact with the surface. The probes are designed to be less cylindrical and more spherical, in order to accommodate the indirect impact angle of 20 degrees from horizontal. The ejection mechanism for the probes has been changed to be actuated by a thermal knife, rather than pyrotechnic ejection. This avoids pyro shock and reduces the effect of each ejection on other components of the spacecraft. The design modifications made for the seismic probes and the difference between the operating environments for the seismic probes and the Deep Space 2 probes impacts the element enough to solidify a TRL rating lower than 6 by the NASA NPR 7120.8—Appendix J.

The Europa Orbiter Radar Sounder is also used to support specific scientific needs for the Europa Exploration Mission. A radar sounder is used in NASA's Europa Orbiter Mission. This mission component uses ice penetrating radar technology to obtain detailed information of the geological structure Europa's icy shell and subsurface ocean. The radar has the ability to map Europa's icy surface with a resolution of 100m to a depth of 20km. The mission component operates at a frequency of 50MHz and have a beam width of 22 degrees. The same technology has been used in studies of the Earth's ice sheets. In order for this component to fly it must meet a level 6 rating by the NASA NPR 7120.8—Appendix J.

The seismic probes and Europa Orbiter Radar Sounder has to mature to a TRL 6 before the mission components are used to support the mission. To reach TRL 6, a Europa Orbiter Radar Sounder prototype must be built and tested in a relevant environment. Full-scale prototypes must be built and tested in critical relevant environments without failure. Also, the software used on the two mission components would also need to be tested on full-scale problems as the NASA NPR 7120.8 states. Lastly, empirical data will need to be in compliance with the analytical data for a full TRL rating of 6 to occur.

F.5 Assembly, Integration, Test, and Verification

F.5.1. Assembly, Integration, and Test Plan Illustration

The assembly of the spacecraft is contracted to Lockheed Martin to be built at facilities of their choosing. The spacecraft integration and testing is subcontracted to Lockheed Martin due to their historic expertise in the spacecraft industry. Testing will take place in Huntsville, AL at Marshall Space Flight Center. A trade study is recommended for later design phases to determine the testing schedule necessary to qualify the spacecraft for the proposed mission. Testing can include, but is not limited to: shake and bake, hardware in the loop, and environmental testing. To account for the possibility of a failed test or nonconformity, a margin of error is built into the subsystems in the form of extra mass and power available for use.

F.5.2. Verification approach

The NASA System Engineerig Handbook verification approach is applied in later phases of design to ensure that all requirements developed and agreed upon in earlier phases are met and verified before moving to the next design phase. Verification is provided through reviews to occur between phases. (NASA, NASA Systems Engineering Handbook 2007)

F.6 Schedule

Figure 36 shows the overall mission schedule for Team Eureka. The specific dates are given in the Table 17. These major milestones are governed primarily by the Discovery Announcement of Opportunity (AO 2010). The phases are based off the NASA System Engineering Handbook (NASA, NASA Systems Engineering Handbook 2007) and the JEO (JEO 2009). This schedule is to be utilized merely as an estimated baseline on which to anchor actual execution of the mission. For example it is acknowledged that Phase E may not be initiated for 30 to 90 days after launch. Unforeseen future achievements in better science capabilities or overall efficiency could possibly be implemented due to such an extended mission schedule. On the other hand new developments in issues like planetary protection or instrumentation procurement could post pone progress. Therefore alternate launch dates discussed in the JEO are encouraged should any of these issues arise. This would give a launch window of 2020. While this is the master schedule it could be quite beneficial to establish related schedules for subsystems' development, implementation, and execution.

Each phase contains specific milestones shown under it. All Phases and Milestones possess their own row in the chart. The colored region of the row shows when it is to occur relative to the other milestones. Every milestone is considered critical and is due during the last colored month in its row of the chart. The top and bottom of the chart shows the corresponding month and year for every milestone. These are targeted months. These dates should be seen as a baseline. As the current milestone and phase shortens or extends, the entire chart from that month leftward should also shift.

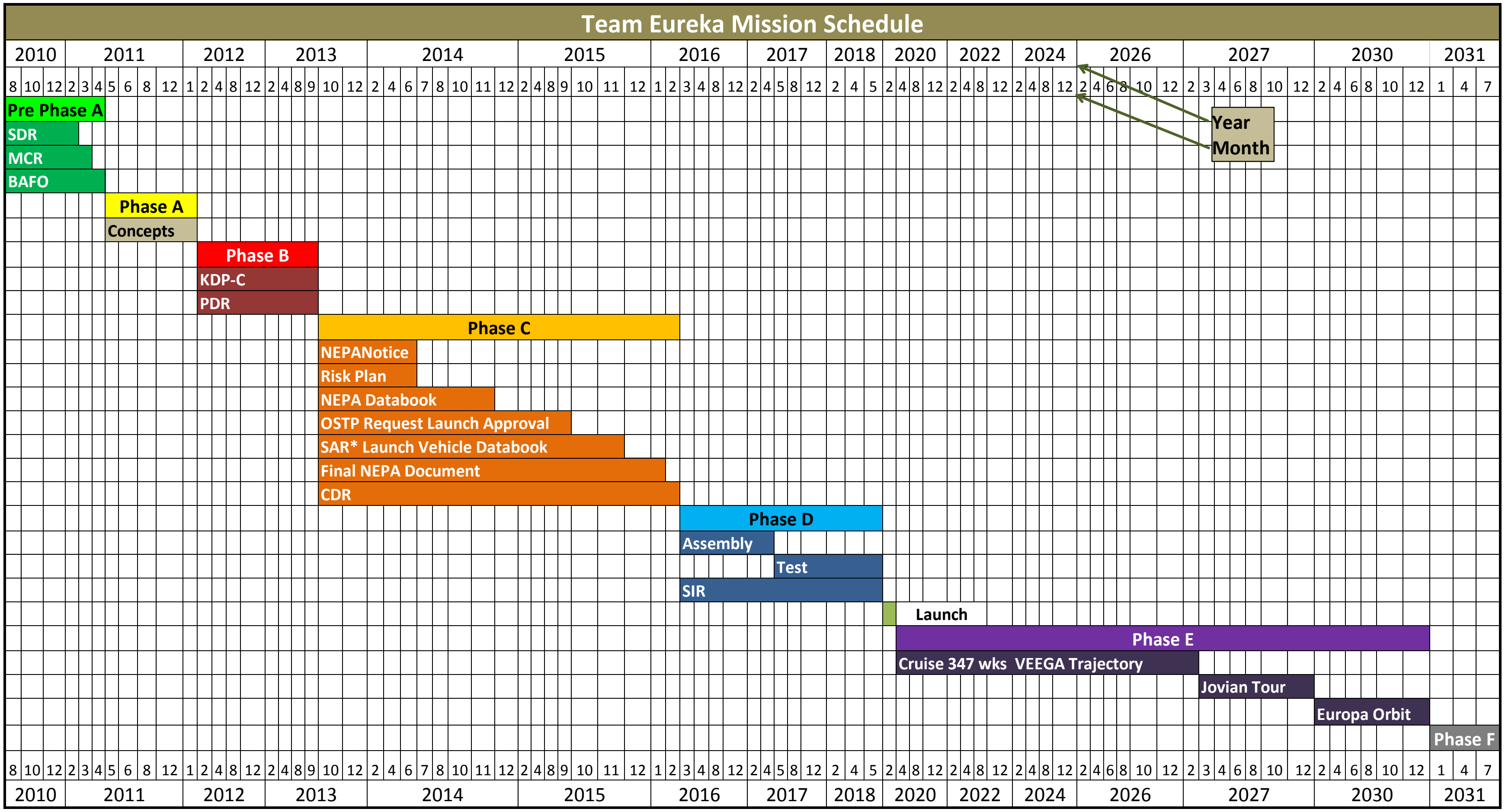


Figure 36 Gantt Chart (Mission) (JEO 2009)

G. Management

G.1 Management Approach

The main functions of the mission are outlined in Figure 35, in Section F.3.1 System Engineering Approach.

G.1.1 Formal Decision-Making Procedure and Guidelines

Each function shall be evaluated to determine those decisions that require a formal decision-making procedure. A formal decision-making procedure shall be used for those decisions that affect cost. The formal decision-making procedure shall be conducted as follows:

1. Determine alternatives.
2. Determine characteristics to be compared between alternatives.
3. Establish trade tree for each alternative.
4. Determine weights for each characteristic
5. Establish trade study to rank each characteristic for each alternative.
6. Make decision choice based on highest score.

The project manager, or PM, shall assign the trade studies to individuals on the team. Each trade study shall be reviewed with the entire team. The individual conducting the trade study shall express the recommended alternative. The entire team shall discuss the recommended alternative. Any disagreements shall be presented and discussed. If no agreement can be made, the PM shall have the final decision.

G.1.2 Decision-Making Procedure and Guidelines

Those decisions that do not affect cost shall not be required to follow the formal decision-making procedure. Every decision shall be reviewed and discussed with the entire team. There should be some agreement among the team. If no agreement can be made, the PM has the final decision.

G.2 Roles and Responsibilities of Team Members

The main institution that all scientists and engineers report to is The University of Alabama in Huntsville. The lead department for the proposal is the College of Engineering at UAH, led by the Project Manager, Seth Webb. The Principal Investigator, Nathan Towles from The College of Charleston, is in charge of researching and determining appropriate science objectives and equipment for the Europa mission described in Section XX. ESTACA is responsible for designing and creating a 3D CAD rendering of the lander for the mission as shown in Figure XX/Section XX. An organization scheme showing the flow of information and communication line is shown in Figure 37.

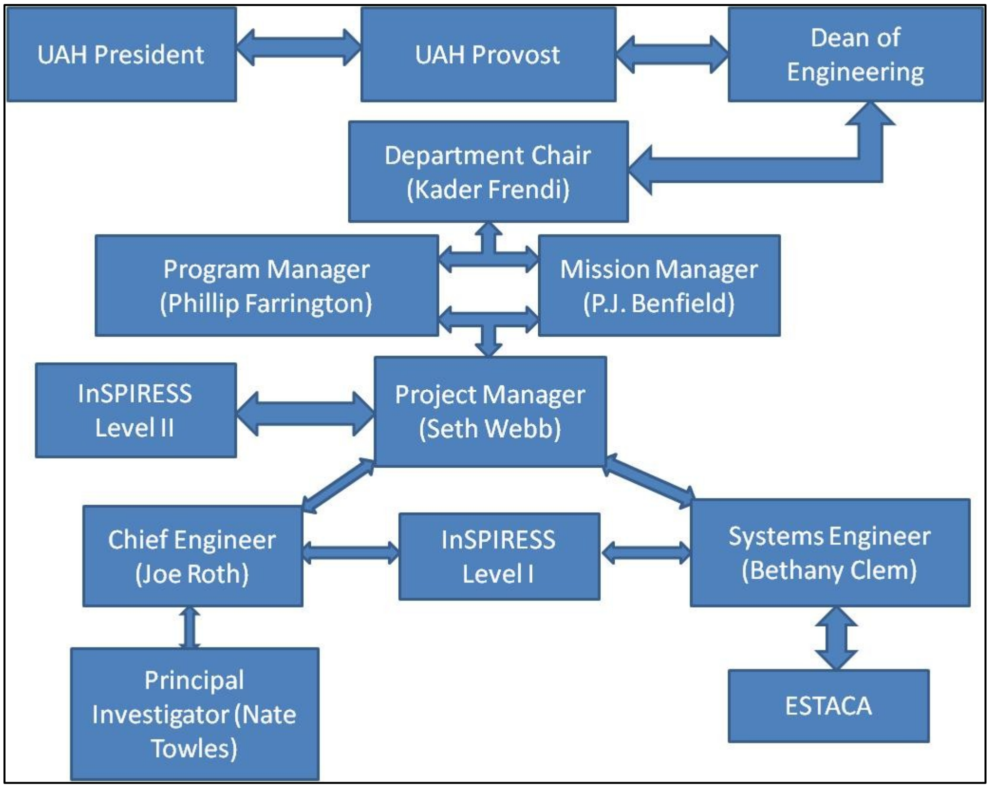


Figure 37 Organization Chart

G.3 Risk Management

Several team defined mission critical risks are defined in Table 28. The table defines the risk, root cause, mitigation, and impact of the risk to the mission. To understand the impact of the risk to the mission, the team developed at risk scoring method in Table 27. To best clarify the risk explained in Table 28, a risk matrix was developed and is shown as Figure 38 where the Table 28 shows the team's thoughts about possible risks and their importance and Figure 37 shows the overall importance of each risk.

Table 27 Risk Matrix Definitions

Scale		Consequence		Likelihood
		Mission Risk	Implementation Risk	
5	Very High	Mission Failure	Overrun budget and contingency, cannot meet launch with current resources	Almost certain
4	High	Significant Reduction in Mission Return	Consume all contingency, budget or schedule.	More likely than not
3	Moderate	Moderate Reduction in Mission Return	Significant reduction in contingency or launch slack	Significant likelihood
2	Low	Small reduction in Mission Return	Small reduction in contingency or launch slack	Unlikely
1	Very Low	Minimal / No Impact on Mission	Minimal reduction in contingency or launch slack	Very unlikely

Table 28 Risk Analysis Cause Mitigation

	Risk	Root Cause	Mitigation	Impact	Likelihood	
					Original	Mitigated
1	Overheating of Orbiter due to Solar Radiation at Venus	The intense solar radiation at Venus causes the spacecraft to become too hot for instrumentation to survive	A heat shield has been designed specifically for the portion of the trajectory which takes the spacecraft near Venus. The shield will reflect the majority of the solar radiation.	5	4	1
2	Loss of Orientation and Location Information from the ACS system	A star tracker malfunctions or experiences a complete failure due to radiation enviroment.	Testing and analysis of the star tracker to determine and reduce possibility of failure. Inclusion of a second, redundant star tracker.	3	3	1
3	Impact Device Overshoots Europa	The deployment of the Impact Device is a risk due to the fact that if the device is ejected on the wrong trajectory there is no attitude correction due to a lack of thrusters on the device.	Extensive testing and analysis of the release mechanism and communication between systems to ensure the device releases at the appropriate time and trajectory.	2	2	1
4	Lander does not transmit all of its collected data to the orbiter before ceasing operations.	The lander does not land near one of Europa's poles, minimizing communication opportunities between the lander and orbiter, decreasing the frequency with which the lander transmits its data.	Extensive testing and analysis of the release mechanism and communication between systems to ensure the device releases at the appropriate time and the field of view of the communication of both lander and orbiter allows sufficient uplink/downlink.	1	3	1
5	ASRG Failure	An ASRG is damaged due to radiation environment or micro-meteoriod impact.	Testing and analysis of the power system to determine and reduce possibility of failure. Three ASRGs are onboard the orbiter, allowing enough power for the threshold mission to still be completed in the event of one ASRG failing.	4	3	1
6	Data corruption in storage	The volatile nature of the environments experienced at Venus and Jupiter could release enough radiation and/or magnetic interferences to cause corruption of data.	Use of radiation shielding. Use of non-volatile SDRAM with Reed Solomon code for Error Detection and Correction.	5	4	1
7	Lander Crash	Failure of one or more thrusters resulting in the lander impacting Europa with unsustainable g-loads	Testing and analysis of the lander's propulsion system. Redundancy in propulsion system.	2	2	1
8	Reaction Control System failure	Due to the harsh environments encountered along with unforeseen problem causes components of the ACS may fail reducing the ability of the spacecraft to alter and/or maintain desired orientation.	Testing and analyis of the orbiter's ACS system. Inclusion of 16 Reaction Control Thrusters with redundant couples in order to maintain 3-axis stability in the event of a thruster failing.	5	3	1
9	Temporary loss of communication with Earth	Jupiter or Europa eclipse the orbiter's line of sight to Earth	Inclusion of enough on-board bulk memory to allow for several hours of data storage without writing over data.	2	5	1

Likelihood		Impact: Very Low	Low	Moderate	High	Very High
	Very High		9			
	High					1 6
	Moderate	4		2	5	8
	Low		3 7			
	Very Low					

Figure 38 5x5 Matrix

G.4 Contributions/Cooperative Agreements

The mission's only contributing partner is the development of the lander from ESTACA. If ESTACA is unable to contribute to the mission, the mission still meets the Mission Threshold. The lander would be the obvious mission element to write off due to the redundancy of the science instrumentation in the orbiter, refer to Section XX for a list of redundancies. Also terminating the lander would allow compatibility with all high class launch vehicles.

H. Cost and Cost Estimating Methodology

H.1 Introduction

For the baseline mission, there are fixed costs and variable costs. The fixed costs include the cost for the launch vehicle, an additional ASRG, and NEPA compliance. A standard launch vehicle is provided at no additional charge (AO). Also, two ASRGs are provided (AO). The Atlas V 551 has been chosen as the launch vehicle for this mission. This mission also requires an additional ASRG. The additional ASRG is a fixed cost. Since ASRGs are to be used, there is a requirement for NEPA compliance. The total fixed cost for the launch vehicle is outlined in Table 29. The variable costs include the cost for the orbiter and lander. A cost model was used to determine the variable costs. The variable costs are also outlined in Table 29.

Table 29 Cost Allocation

Cost Allocation	
Fixed Cost Element	**Variable Cost Element**
Launch Vehicle	Orbiter
Additional ASRG	Lander
NEPA compliance	

H.2 Cost Model

The cost of the orbiter and lander was estimated by using the Spacecraft Cost Model developed by Joe Hamaker (Hamaker). This cost model uses mission defined inputs and variable inputs to determine the cost. Variable inputs are elements which are determined by the user. The mission defined inputs and variable inputs are outlined in Table 30.

Table 30 Cost Inputs

Inputs					
Mission Defined			**Variable Inputs**		
				Orbiter	**Lander**
Number of Science Organizations	1	College of Charleston	**Dry Mass**	2010 kg	65 kg
Apogee Class	4	Planetary	**Power**	32W	3.77 W
Team Experience Class	4	Unfamiliar	**Design Life**	113.4	113.4
ATP Date	51	2010 is 51 years since 1960	**Max Data Rate Requirement**	50%	50%
Platform Factor	2.2	Unmanned Planetary	**Test Requirement Class**	2.5	2.0
			Requirements Stability Class	3.0	3.0
			Funding Stability Class	2.0	2.0
			Formulation Study Class	1.0	1.0
			New Design Percent	50%	50%
			TRL Penalty Factor	5.0	6.0
			Override of FTEs	0	0
			Civil Service Labor Rate (FY2004$)	$280,000	$280,000
			Override of Phase C/D Schedule	58	58

H.3 Cost Model Inputs

The mission defined inputs remains the same regardless of the mission design. The variable inputs changes depending upon the mission design. For the mission defined inputs, several inputs are constant. Total dry mass, total power generation capacity, design life, data rate requirement, and phase C/D schedule have been determined by the designed mission.

The inputs for the test requirement class, requirements stability class, funding stability class, formulation study class, government FTEs, and labor rate are based on the nominal values outlined in the cost model (Hamaker). Due to the team's unfamiliar experience, using the nominal allowed for a more accurate estimate.

The inputs for the new design percent and total readiness level, TRL, required additional analysis to determine a feasible estimate. For the orbiter, there are two pieces of science equipment, penetrator probe and Europa Orbiter Radar Sounder, that has either not been flown or has some modification to a previously flown device. The details of these pieces of equipment are described in Section F.4. Furthermore, using this equipment has an impact on the new design percent as well as the TRL. For the new design percent, the cost model defines simple modifications as 30% and extensive modifications as 70% (Hamaker). Since the modifications are neither simple nor extensive, a 50% new design percent seemed feasible. The overall TRL is determined by the TRL of the subsystems and equipment. Using Figure 38, it is determined that the overall TRL level for the orbiter is coded as yellow, or a TRL range of 3.0 to 5.0. The penetrator probes, 223 kg, and Europa Orbiter Radar Sounder, 10 kg, account for only 11% of the total orbiter, 2010 kg. Since this is a small portion of the orbiter, the low TRLs have little impact on the overall TRL for the orbiter. Therefore, the higher of the range, 5.0, was chosen as the TRL for the orbiter. For the lander, all of the equipment has been flown before. Since it doesn't require extensive modifications, a new design percent of 50% was used. Using Figure 38, it is determined that the overall TRL is coded as green, or a TRL of 6.0 or above. Since 6.0 is nominal, it was used as the overall TRL for the lander.

H.4 Cost Reserve

The TRL and New Design Percent in the cost model has a major impact on the cost of the orbiter. Figure 39 shows how the TRL and New Design Percent affect cost. As TRL decreases and New Design Percent increases, the cost increases. The solid lines represent the different new design percent. The dashed lines represent the various cost reserves. There are several configurations for TRL and New Design Percent as shown by the asterisks on the graph. Any configurations that fall below the cost reserve line is covered by that cost reserve. For example, the asterisks that are below the 25% cost reserve line is covered by the 25% cost reserve. The proposed configuration, noted by the bold dot on the graph, is a TRL of 5.0 and a New Design Percent of 50% as outlined in the previous section. At a 0% cost reserve, the total cost of the orbiter is $618.58M. Section 5.6.3 in the AO requires a minimum cost reserve of 25% (AO). With a 25% cost reserve, the total cost of the orbiter is $773.23M. We do not expect the TRL to increase, or the New Design Percent to decrease. The 25% cost reserve is not be sufficient if the TRL of 5.0 decreases, the New Design Percent increases, or a combination of both. We propose that, while substantially over budget, a 45% cost reserve is more appropriate. This accounts for any deviations from the proposed configuration of TRL and New Design Percent. The total cost for the orbiter with a 45% cost reserve would be $896.941. The 25% cost reserve would only be sufficient if the New Design Percent changes to 60%. The 45% cost reserve would allow for different configurations of TRL and new design percent. This cost reserve would account for any unknowns that may arise.

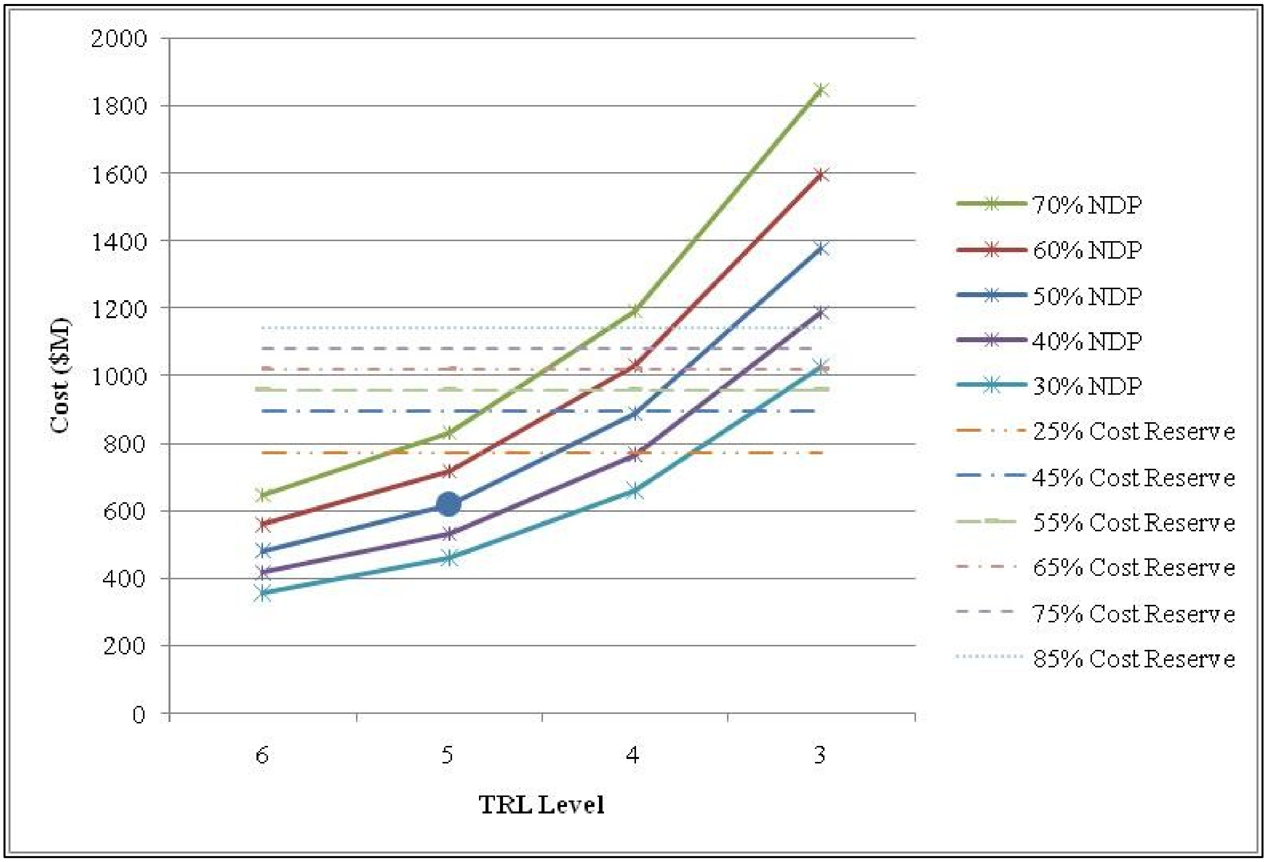


Figure 39 TRL-New Design Percent vs. Cost

H.5 Cost Allocation

The total cost of the baseline mission is $1,169M FY2010$. This total includes a proposed cost reserve of 45% as specified in Section H.3. The total cost has been allocated to the launch vehicle, the orbiter, the lander, the additional ASRG, and NEPA compliance. The cost allocation for each element is outlined in Table 31.

Table 31 Baseline Mission Cost

Cost Allocation – Baseline Mission			
Fixed Cost Element	**Cost (FY2010$)**	**Variable Cost Element**	**Cost (FY2010$)**
Launch Vehicle	$68M	Orbiter	$897M
Additional ASRG	$27M	Lander	$157M
NEPA compliance	$20M		
Total	$115M	Total	$1054M
Total Mission Cost: $1,169M			

I. Acknowledgement of EPO and Student Collaboration

I.1 Education Public Outreach

A plan for a core E/PO program is developed during the Phase A concept study and is included in the Concept Study Report if the proposal is selected. However the quality of the E/PO plans is not a consideration in the selection of the Step 1 proposals for Phase A concept studies. (AO 2010) Therefore, E/PO plans are not needed at this time. These requirements, or lack thereof, are described in Discovery AO, Section 5.5.2 Core E/PO Program

I.2 Student Collaboration

I.2.1 Introduction

In the beginning of the 2010-2011 school year, Lee High School's Superstar's of Modern Engineering began to study the topography and atmosphere of Europa. Much of the information centered on the way the planet flexes and its magnetic field. The research acquired led the team to the final question and mission.

I.2.2 Mission

SOMA's mission is to discover how Jupiter's magnetic field and Europa's tidal flexing affect Europa's magnetic field? To complete this task, the payload must characterize the strength of Jupiter's magnetic field, Europa's induced magnetic field and tidal flexing on Europa. In order to do this, the payload must measure the amount of flexing using a laser altimeter every 15 degrees in orbit. Two magnetometers run continuously. One measures the induced magnetic field of Europa and Jupiter while the other measures the magnetic field around the spacecraft. When the measurements are taken, the team has the ability to look at the measurements and compare the strength of Jupiter's magnetic field and Europa's flexing to see if there are any consistent patterns.

I.2.3 Laser Altimeter

The altimeter measures the distance between our payload and the surface of Europa and would make a high resolution 3D map of the surface. This instrument measures the amount of flexing on Europa.

- The Lunar Orbiter Laser Altimeter (LOLA) has five laser beams and five receiver channels.
- It was previously used in satellite (moon) such as the moon and could be used in the Jupiter's Europa moon.
- Consists of a single stage diode-pumped and a Q-switched Nd: YAG laser at 1064-nm wavelength, 2.7-mJ pulse energy, 6-ns pulse width, 28-Hz pulse rate, and 100 radio beam divergence angles.
- Its diffractive optic element is made of fused silica with an etched-in diffraction pattern that is used to split the single incident laser beam into five slightly off pointed beams.
- The reflected signal is collected by a 14-cm diameter telescope and a 5-optical-fiber array at the focal plane, each one of those sees one of the five laser spots on the lunar surface and delivers the signal to one of the five avalanche photodiode.
- The transmitted laser pulse and the five received laser pulses are time stamped with respect to the spacecraft mission elapsed time using a set of time-to-digital converters at <0.5 ns precision

Table 32 Laser Altimeter

Laser Altimeter					
Laser Oscillators		**Receiver Optics**		**Detectors**	
Number of Laser Oscillators:	2	Number of Receiver Telescopes:	1	Number of Detectors:	5
Wavelength (nm):	1064.3	Objective Lens Size (mm, dia.):	150	Detector Type:	SIAPD
Pulse Energy (nm):	2.7	Collecting Clear Aperture (cm²):	154	Detector Size (mm, dia.):	0.7
Pulse Width (ns, FWHM):	6	Number of Receiver FOV's:	5		
Repetition Rate (Hz):	28	FOV (mrad, FWHM):	400		
Divergence (mrad, $1/e^2$ dia.):	100	FOV spacing (mrad):	500		
Number of Beams/ Lasers:	5	Fiber-Optic Coupled:	Yes		
Beam Spacing (mrad):	500	Bandpass Filter (nm, FWHM):	0.7		
MASS (kg):	9.7	POWER (W):	15	VOLUME (cm):	35 x 45 x 31
THERMAL (F):	-3/10	DATA RATE (Hz Pulse):	26		

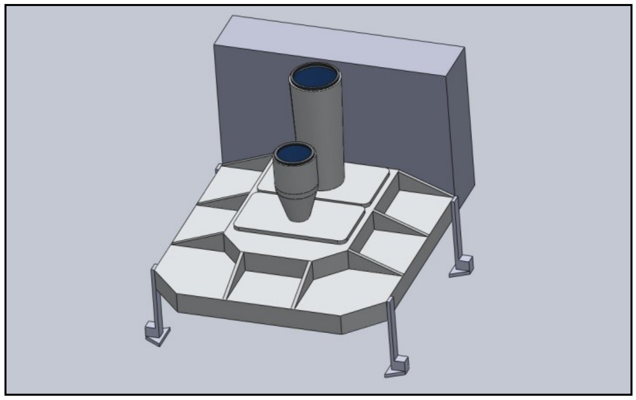


Figure 40 Laser Altimeter

I.2.3 Vector Scalar Helium Magnetometer:

This instruments purpose is to measure Jupiter's magnetic field and Europa's induced field. This magnetometer is an optically pumped magnetometer capable of operating in either a vector or scalar mode. In the vector mode, three voltages are generated proportional to the three mutually orthogonal magnetic field vector components, whereas in the scalar mode the Larmor frequency proportional to the magnitude of the magnetic field is measured. The vector fields are measured to an accuracy of 0.5%. In the scalar mode, the magnetic field is measures to precisely 0.01% or better. The Cassini mission is the first space mission where one magnetometer having a dual vector or scalar capability has been flown. This instrument is placed at the end of the boom.

Table 33 Vector Scalar Helium Magnetometer Specifications

Vector Scalar Helium Magnetometer Specifications
Axial Alignment Orthogonality better than +/- 1°
Field Measurement Range +100 mT = +/- 10V
Accuracy + 0.75% of full scale (0.5% typical)
Linearity -+ 0.015 % of full scale
Sensitivity 100 mV/nT
Scale Factor Temperature Shift 0.007% full scale/ °Celsius
Noise <12pT RMS/ Hz @ 1 HZ (< pT Option available)
Output Ripple 3 mV peak to peak @ 2nd harmonic
Analog Output @ Zero Field + 0.025 V
Zero Shift with Temperature 0.6 nT/ °Celsius
Susceptibility to Perming + 8 nT shift with + 5 Gauss applied
Output Impedance 332W + 5%
Frequency Response 3dB@ >500Hz(to > 4KHz wideband)
Overload Recovery + 5 Gauss slew< 2 milliseconds
EMI Designed to meet CEO1, CEO3, REO2, CS01, CSO2, CSO6, RSO1, RSO2, RSO3
Random Vibration > 20G RMS 20 Hz to KHz
Temperature Range -55° to +85° Celsius operating
Acceleration >60G
Weight 100 grams
Size 3.51cm x 3.23cm x8.26 cm
Connector 9 pin male "D" type , female mating connector supplied
Input Voltage Options 15 to 34 VDC @ 25mA

Mass (kg):	234	Power (V):	11-16 or 20-32	Volume (in^3):	11	Data Rate (bit/s):	9600
		Thermal (°C):	-20 - 60				

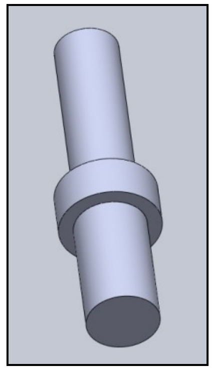


Figure 41 Vector Scalar Magnetometer

I.2.4 Dual Technique Fluxgate Magnetometer:

The purpose of this magnetometer is 9 to measure the magnetic field of the spacecraft. It was used on Cassini's mission around Saturn. It measured Saturn's magnetic field as well as that of the Titan. However, our mission uses this system to measure the atmosphere around the spacecraft as well as the spacecraft itself. It is located at the center of the 10 meter boom.

- Mass (current best estimate) = 3.00 kg
- Average Operating Power (current best estimate) = 3.10 W
- Average Data Rate (current best estimate) = 3.60 kilobits/s

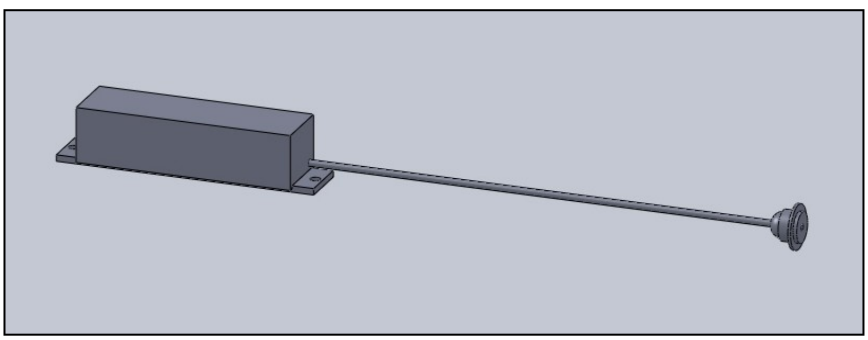


Figure 42 Dual Technique Fluxgate Magnetometer

I.2.5 ATK Coilable Boom:

The purpose of this boom is to deploy the Vector Scalar Helium Magnetometer and fluxgate magnetometer. The fiberglass boom is 10meters long. The fluxgate magnetometer is placed at the center of the boom (5m). The particular deployment method chosen has been used on Cassini, Galileo and UARS.

I.2.6 Lanyard Deployed Method:

- Most common method of deployment
- Stowed strain energy drive deployment
- Tip-mounted lanyard controls the deployment via damper or motor
- Boom tip rotates during deployment
- Least expensive method of controlled deployment

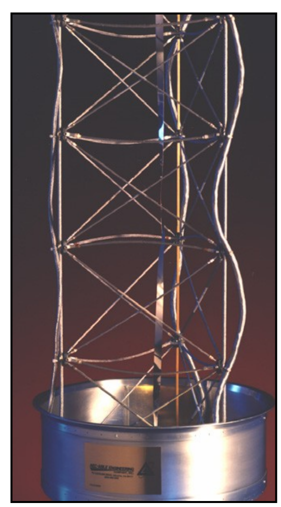


Figure 43 Lanyard Deployment System

After deciding on the instruments that is used, the team created diagrams of Europa on Satellite tool kit. These diagrams were used to see what orbits should be used for the spacecraft to view the most cross points (points where the altimeter should measure flexing) and how often the payload should take altimeter measurements. From these diagrams and skype sessions with Dr. Robert Poppalardo, the team found that a 94 degree inclined orbit with a cone angle of the sensor of 30 degrees for the altimeter is the most efficient. SOMA also used Solid works to create CAD drawings of the instruments that is used on the payload.

By studying magnetic fields and flexing, the Lee High team hopes to obtain information that gives scientists an increased amount of information about the behavior of Europa's ocean and induced magnetic field. This consequently brings researchers one step closer to finding out if there is really life on Europa.

J. Appendices

J.1 Table of Proposed Participants

Table 34 Proposed Participants

Name of Organization	Role of Organization	Total Cost of Organization
University of Alabama in Huntsville Huntsville, AL	Major partner that is the central hub for communications and design of the orbiter	*TBD
College of Charleston Charleston, SC	Major partner in charge of deciding Science Objectives and Science equipment	*TBD
ESTACA Paris, France	Major partner in charge of Communications, Computer Data System, and Lander	*TBD
InSPIRESS Level II Huntsville, AL	Minor partner in charge of developing Impact Device	*TBD
InSPIRESS Level I Huntsville, AL	Minor partner overseeing additional science objectives of Europa	*TBD

Table 35 Orbiter Cost Analysis

Variable Description		**Variable Units**
Enter Spacecraft Bus + Instruments Total Dry Mass	**2010**	KG
Enter Spacecraft Total Power Generation Capacity (LEO Equivalent)	**32**	W LEO equivalent flux
Enter Design Life in Months	**113.4**	Months
Enter Number of Science Organizations	**1.0**	Count (Enter zero for projects with no science or science organization involvement)
Enter Apogee Class	**4.0**	LEO=1, HEO/GEO=2, beyond GEO=3, Planetary=4
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
Enter Test Requirements Class	**2.5**	Less than average testing=1, Average=2, More than average=3, Extensive=4, Very extensive=5
Enter Requirements Stability Class	**3.0**	Very low volatility=1, Low=2, Average=3, High=4, Very high volatility=5
Enter Funding Stability Class	**2.0**	Stable funding=1, Some instability=2, Significant instability=3
Enter Team Experience Class [Derived from Price Model; used with permission from Price Systems LLP]	**4.0**	Extensive experience=1, Better than average=2, Average (mixed esperience)=3, Unfamiliar=4 [Ref: Price Model]
Enter Formulation Study Class	**1.0**	Formulation study (1=Major, 2=Nominal, 3=Minor)
Enter New Design Percent	**50%**	Simple mod=30%, Extensive mod=70% (average), New=100%

Variable Description		Variable Units
Enter ATP Date Expressed as Years Since 1960	**51**	Years elapsed since 1960
Regression Model Result	$223.2	DDT&E + TFU (Phases C/D/E) in Millions of 2004 Dollars including fee, excluding full cost
Enter Technology Readiness Level (TRL) Penalty Factor	**5.0**	Refer to NASA TRL scale (TRL 6 is nominal)
Enter Platform Factor [Derived from Price Model; used with permission from Price Systems LLP)	**2.20**	Platform factor (Airborne Military=1.8, Unmanned Earth Orbital=2.0, Unmanned Planetary=2.2, Manned Earth Orbital=2.5, Manned Planetary=2.7) [Ref: Price Mdoel]
Enter Functional Complexity Factor	**To Be Added Later**	To be added later
Subtotal (Non Full Cost Subtotal)	**$368.0**	Subtotal (Millions of 2004 Dollars including fee)
Calculated Size of the Government Project Office (Project Office Only--Excludes Government Functional Line/Laboratory Labor)	**62.5**	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)	**62.5**	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
Calculated Project Phase C/D Schedule Duration (Excludes O&S Phase E)	**54**	Months
Enter Override of Calculated Phase C/D Schedule Duration (or leave zero to accept calculated duration)	**58**	Months
Final Estimate of the Project Phase C/D Schedule Duration	**58**	Months
Calculated Cost of the Government Project Office	**$79.5**	Millions of 2004 Dollars
Government Service Pool Use Intenstiy Factor	**4**	1=Minimum use of service pools, 2=Less than average, 3=Average, 4=More than average, 5=Significantly more than average

Variable Description		Variable Units
Calculated Cost of Government Service Pool Use	**$44.2**	
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	**$44.2**	
Subtotal (2004$)	**$491.6**	
Ground System	**$44.2**	
Enter Override of Calculated Ground System Cost	**$0.0**	
Final Estimate of the Cost of Ground System	**$44.2**	
Subtotal (2004$)	**$535.9**	
Enter Launch Services Cost	**$0.0**	
Enter Cost Reserves	**$241.1**	
Total (2004$)	**$777.0**	
FY2010$	$896.94	

Table 36 Lander Cost Analysis

Variable Description		Variable Units
Enter Spacecraft Bus + Instruments Total Dry Mass	**63**	KG
Enter Spacecraft Total Power Generation Capacity (LEO Equivalent)	**3.77**	W LEO equivalent flux
Enter Design Life in Months	**113.4**	Months
Enter Number of Science Organizations	**1.0**	Count (Enter zero for projects with no science or science organization involvement)
Enter Apogee Class	**4.0**	LEO=1, HEO/GEO=2, beyond GEO=3, Planetary=4
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
Enter Test Requirements Class	**2.0**	Less than average testing=1, Average=2, More than average=3, Extensive=4, Very extensive=5
Enter Requirements Stability Class	**3.0**	Very low volatility=1, Low=2, Average=3, High=4, Very high volatility=5

Variable Description		Variable Units
Enter Funding Stability Class	**2.0**	Stable funding=1, Some instability=2, Significant instability=3
Enter Team Experience Class [Derived from Price Model; used with permission from Price Systems LLP]	**4.0**	Extensive experience=1, Better than average=2, Average (mixed esperience)=3, Unfamiliar=4 [Ref: Price Model]
Enter Formulation Study Class	**1.0**	Formulation study (1=Major, 2=Nominal, 3=Minor)
Enter New Design Percent	**50%**	Simple mod=30%, Extensive mod=70% (average), New=100%
Enter ATP Date Expressed as Years Since 1960	**51**	Years elapsed since 1960
Regression Model Result	$53.4	DDT&E + TFU (Phases C/D/E) in Millions of 2004 Dollars including fee, excluding full cost
Enter Technology Readiness Level (TRL) Penalty Factor	**6.0**	Refer to NASA TRL scale (TRL 6 is nominal)
Enter Platform Factor [Derived from Price Model; used with permission from Price Systems LLP)	**2.20**	Platform factor (Airborne Military=1.8, Unmanned Earth Orbital=2.0, Unmanned Planetary=2.2, Manned Earth Orbital=2.5, Manned Planetary=2.7) [Ref: Price Mdoel]
Enter Functional Complexity Factor	**To Be Added Later**	To be added later
Subtotal (Non Full Cost Subtotal)	**$67.7**	Subtotal (Millions of 2004 Dollars including fee)
Calculated Size of the Government Project Office (Project Office Only--Excludes Government Functional Line/Laboratory Labor)	**19.1**	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)	**19.1**	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
Calculated Project Phase C/D Schedule Duration (Excludes O&S Phase E)	**23**	Months
Enter Override of Calculated Phase C/D Schedule Duration (or leave zero to accept calculated duration)	**58**	Months

Variable Description		Variable Units
Final Estimate of the Project Phase C/D Schedule Duration	**58**	Months
Calculated Cost of the Government Project Office	**$10.1**	Millions of 2004 Dollars
Government Service Pool Use Intenstiy Factor	**4**	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	**$8.1**	
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	**$8.1**	
Subtotal (2004$)	**$85.9**	
Ground System	**$7.7**	
Enter Override of Calculated Ground System Cost	**$0.0**	
Final Estimate of the Cost of Ground System	**$7.7**	
Subtotal (2004$)	**$93.7**	
Enter Launch Services Cost	**$0.0**	
Enter Cost Reserves	**$42.1**	
Total (2004$)	**$135.8**	
FY2010$	$156.78	

J.2 Letters of Commitment

Date: April 25, 2011
From: Europa Mission Design Team
Re: Letter of Commitment

The members of Europa Mission Design Team hereby acknowledge that as the design team, they take on full responsibility of all identified requirements specified within the following proposal that meets the requirements within the Announcement of Opportunity, Discovery Program NNH10ZDA007O, and they intend to carry out all responsibilities specified within the proposed project pertaining directly to the UAH design team. The design team ensures that the following proposal prepared by the design team adheres to the requirements and uses factual information relevant to completing the following proposed project. The team understands that the extent of participation required of UAH is considered in the review of this proposal in determining the merits of this proposal.

_______________________/__________
C. Seth Webb, UAH IPT, Project Manager

_______________________/__________
Joseph Roth, UAH IPT Chief Engineer

_______________________/__________
Bethany Clem, UAH IPT Lead Systems Engineer

_______________________/__________
Daniel Myrick, UAH IPT Engineer

_______________________/__________
Jason Mann, UAH IPT Engineer

_______________________/__________
Jason Wilborn, UAH IPT Engineer

_______________________/__________
Josh Clifton, UAH IPT Engineer

_______________________/__________
Justin Greene, UAH IPT Engineer

_______________________/__________
Phillip Webster, UAH IPT Engineer

_______________________/__________
Zac Snodgrass, UAH IPT Engineer

April 24, 2011

Seth Webb
Project Manager
University of Alabama in Huntsville
Mechanical and Aerospace Engineering Dept.
N274 Technology Hall
Huntsville, AL 35899

Dear Mr. Webb,

The University of Alabama in Huntsville is pleased to formally acknowledge your team's design for an Europa Extraterrestrial Life Survey (EELS) mission as part of NASA's Discovery Program. We believe, should your design be selected, the science gained from this mission not only provides a greater understanding of our solar system, but helps 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,

Michael P.J. Benfield, Ph.D.
Europa Extraterrestrial Life Survey Mission Manager
The University of Alabama in Huntsville

J.3 Resumes

Bethany Lynn Clem
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clemb@uah.edu

10350 US Highway 72 Lot N
Athens, AL 35611

CITIZENSHIP U. S.

TECHNICAL SKILLS Certified ISO 9001:2008 internal auditor

EDUCATION

The University of Alabama in Huntsville **Huntsville, AL**
Bachelor of Science in Engineering with a concentration in Industrial and Systems Engineering
GPA: 2.82/4.0 in Major, Expected graduation May 2011

- Lead Systems Engineer for Integrated Product Teams (IPT) 2010 – 2011
- Courses: Work Design, Operations Research, Human Factors Psychology, Systems Engineering Analysis

WORK EXPERIENCE

May 2005 – Present **General Electric Appliances** **Decatur, AL**
ISO 9001:2008 Auditor/Internal Audit Coordinator and Document Control Specialist

- Manage over 200 workstation instructions used to establish elements for each workstation. Manage over 50 quality documents that reflect procedures, tests, inspections that affect the quality of our product. Update documents to reflect changes made by Engineering, LEAN, and Industrial Engineers. Create documents for new processes. Developing skills in Microsoft Office Suite.
- Schedule audits. Assign auditors specific tasks. Schedule closing meetings. Create presentations and present findings at staff meetings. Maintain records. Develop skills in scheduling, time management, and record keeping.
- Audit specific areas and clauses.
- Perform analysis on defective electrical components. Send defective parts back to the supplier. Communicate with suppliers. Develop skills in communication, domestic and international shipments via FedEx, consistency, and recognizing trends.
- Use Microsoft Excel to create weekly supplier defect reports for particular suppliers.
- Write scripts for videos to be used as a reference for customers.

Josh Clifton
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Harvest, AL 35749

CITIZENSHIP U. S.

TECHNICAL SKILLS

Software: Solid Edge, Mathcad, MATLAB, Solar Map, Leica Application Suite, Leica Stereo Explorer, Vision 32, TomoVIEW, Patran, Nastran
Equipment: stylus and laser profilometer, optical interferometer, phase grading laser interferometer, shearography, phased array ultrasonics, microscopy, NIMS Level 1 certified using mill and lathe, can use most precision metrology equipment

EDUCATION

The University of Alabama in Huntsville **Huntsville, AL**
Bachelor of Science in Engineering with a concentration in Mechanical Engineering
GPA: 3.6/4.0, Expected graduation August 2011

WORK EXPERIENCE

May 2009 – Present **NASA** **Huntsville, AL**
NASA Co-op
- Work with the tribology team using precision metrology equipment to study failures and anomalies in flight hardware with a focus on precision rolling element bearings.
- Work with the non-destructive evaluation team to develop and use various methods to detect flaws in composite and non-ferrous materials.

Jan 2009 – May 2009 **NASA** **Huntsville, AL**
Undergraduate Space Research Program Intern
- Worked with the tribology team using precision metrology equipment to study failures and anomalies in flight hardware with a focus on precision rolling element bearings.

Jan 2000 – May 2003 **Turner Machine** **Athens, AL**
Precision Machine Co-op
- Performed precision machining and quality inspections.

HONORS AND AWARDS

NASA Group Achievement Award, Undergraduate Space Research Program Intern, President’s and Dean’s List every semester while attending The University of Alabama in Huntsville

Andrew J. Faustmann
(256)-348-9201
afaust7809@gmail.com

7809 Cadillac Drive
Huntsville, AL 35802

CITIZENSHIP	U. S.
TECHNICAL SKILLS	Adobe Acrobat X Pro, Microsoft Office Suite
EDUCATION	**The University of Alabama in Huntsville** — **Huntsville, AL** Bachelors of Arts in English Expected graduation August 2012 • Technical Editor for the multidisciplinary Integrated Product Team **Calhoun Community College** — **Huntsville, AL** General Education Requirements GPA: 3.6/4.0, August 2008 – December 2010 • Script Editor for World Literature Plays
WORK EXPERIENCE	**July 2008 – Present** — **Wilmer & Lee, P.A.** — **Huntsville, AL** Office Clerk/Process Server/Courier • Assist over thirty lawyers with preparing and filing of client's case files within the firm's database, while helping to enhance the firm's relationship with clients. • Serve those who need to be called into court to testify for cases in which the firm is covering. • Record estate closings, deposit payoffs for estate closings, distribute and gather documents to and from clients, and file subpoenas in order to process serve.
HONORS AND AWARDS	Dean's List while attending Calhoun Community College, Ranger of the Year 2000-2002 and 2004
AFFILIATIONS	Guatemala Missions, Casa Angelina Orphanage, LifeSouth Blood Drives, Clinton Elementary School, and Me Fail English?

Justin Greene
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jag0007@uah.edu

Current Address
1762 Paramount Drive Apt 7C
Huntsville, Al, 35806

Permanent Address
229 Prairie Drive
Madison, Al, 35758

CITIZENSHIP U. S.

TECHNICAL SKILLS Patran, Nastran, SolidWorks, Solid Edge, NX, Microsoft Word, Microsoft Excel, Microsoft PowerPoint, Pages, Numbers, Keynote

EDUCATION

The University of Alabama in Huntsville — **Huntsville, AL**
Bachelor of Science in Engineering with a concentration in Aerospace Engineering
GPA: 3.31/4.0 (3.41/4.0 in major), Expected graduation May 2012

WORK EXPERIENCE

May 2008 – Dec 2010 — **Raytheon** — **Huntsville, Al**
Tech Student/Intern

- Interned over the summer, from May 2008 to Aug 2008, working on the RAID program with a focus on Workplace Organization and Foreign Object Elimination.
- Interned between Dec 2008 and Jan 2009 working on the Patriot IDS satellite program with a focus on Earned Value Management.
- Interned between May 2009 and Aug 2009, as well as Dec 2009 to Jan 2010, working on Patriot Software validation and verification.

May 2005 – Aug 2008 — **Regal/Touchstar Cinemas** — **Huntsville, Al**
Manager

- Built, started, and maintained movies, general theater activities, and general maintenance on projectors.
- Opened and closed operations, scheduled movies, trained employees, and met a pre-established revenue during the general workday.

CLEARANCE Secret Clearance, granted by Raytheon Company LLC in May 2009

HONORS AND AWARDS The University of Alabama in Huntsville Academic Excellence Scholarship Aug 2007 – Present

AFFILIATIONS Formula Society of Automotive Engineers – Assistant Team Lead, American Society of Mechanical Engineers

Jason Mann
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jem0004@uah.edu

117 Laredo Circle
Huntsville, AL 35811

CITIZENSHIP U. S.

TECHNICAL SKILLS Microsoft Office Suite, Mathcad, MATLAB, Solid Edge, NX, Patran, Nastran

EDUCATION

The University Of Alabama in Huntsville **Huntsville, AL**
Bachelor of Science in Engineering with a concentration in Mechanical Engineering
GPA: 3.13/4.0 in major, Expected graduation May 2011
Calhoun Community College **Huntsville, AL**
Associates of Science Degree: Mathematics, 2006

WORK EXPERIENCE

Aug 2006 – Present **Wyle Labs/CAS Group** **Huntsville, AL**
Engineering Technician

- Support the RAID Product Office Logistics division.
- Aid the development and sustainment of the logistics database supporting the CONUS and OCONUS equipment tracking, inventorying, and assigning of FLIR Star Safire III, T3000, MSTAR, TQG, and RAID tower systems for the BETSS-C, GBOSS, and CERBERUS programs.
- Assist with test equipment design and assembly.
- Develop reports for procurement of assets, location awareness, and weekly briefings on equipment received from CONUS and OCONUS.
- Work with FLIR and Raytheon to obtain RMA's and spare equipment for Star Safire III, T3000, and RAID tower assets that need repair or replacement and track it to the end of the cycle.
- Plan missions to OEF and OIF and recommend available equipment.
- Organize and develop plans for transportation of equipment for CONUS and OCONUS based movements.
- Organize and file supporting documents (DD1149, DD3161, and DD2062) to maintain a current log of equipment accountability.

CLEARANCE Secret – Sept 2002

Daniel Bradley Myrick
(256)-415-2128
myrickd@uah.edu

2229 County Road 25
Killen, AL 35645

CITIZENSHIP	U. S.
TECHNICAL SKILLS	Patran, Nastran, Mathcad, FANUC CNC Milling Machine setup/programming, screw machine
EDUCATION	**The University of Alabama in Huntsville** **Huntsville, AL** Bachelor of Science in Engineering with a concentration in Mechanical Engineering GPA: 2.51/4.0, Expected graduation August 2011 • SAE Mini Baja
WORK EXPERIENCE	**Aug 2006 – Jun 2009** **Delphi Saginaw Steering Systems** **Decatur, AL** • Operated screw machine. • Setup and Operated Siarto Rotating Pallet Machine. • Operated, tooled, and setup Chiron CNC Milling Machine.

Joseph Roth
Phone (256)-536-5858; Cell (256)-503-9278
jpr0001@email.uah.edu

1302 Highland Avenue
Huntsville, AL 35801

CITIZENSHIP U. S.

TECHNICAL SKILLS C#, Java, Basic, Solid Edge, SolidWorks, Eclipse, Microsoft Visual Studios, MATLAB, Mathcad

EDUCATION **The University of Alabama in Huntsville** **Huntsville, AL**
Bachelor of Science in Engineering with a concentration in Aerospace Engineering
GPA: 3.57/4.0 (3.65/4.0 in major), Expected graduation May 2011

WORK EXPERIENCE

May 2008 – Mar 2009 Davidson Technologies, Inc. Huntsville, AL
Intern
- Tested and verified software programs.
- Utilized Java and C# programming languages to edit existing code and write new code to correct bugs and create new features within the program.
- Created and updated various HTML Help files for the program. Updated the user's manual for each release of the program.

May 2006 – Oct 2006 The University of Alabama in Huntsville Huntsville, AL
Research Aide
- Researched and evaluated optical instrumentation options for unmanned aerial vehicles.
- Edited and ran software simulations.
- Edited presentation files to be shown to visiting organizations.

HONORS AND AWARDS Lee High School Valedictorian class of 2006, Received Rensselaer Polytechnic Mathematics and Science Award

Daisy Smith
(256)-533-9745
dms0021@uah.edu

3210 Bluecrest St SW
Huntsville, AL 35805

CITIZENSHIP	U. S.
TECHNICAL SKILLS	Microsoft Office Suite, Computer Information Systems, Introduction to C Programming, Rhetoric and Composition, Speech Communication
EDUCATION	**The University of Alabama in Huntsville** — **Huntsville, AL** Bachelor of Arts in English Minor: Technical Writing, GPA: 3.7/4.0, Expected graduation May 2013 • Technical Editor for the multidisciplinary Integrated Product Team • Technical Writer of the Spaying or Neutering Your Pet brochure • Technical Writer of the Animal Related Job Opportunities guide
WORK EXPERIENCE	**Nov 2005 – Jun 2007** — **State Farm Insurance** — **Huntsville, AL** Licensed Insurance Agent, LSA4 • Solicited new insurance policies. • Maintained existing policies. • Handled claims. **Sep 1997 – Sep 2005** — **Wal-Mart Stores, Inc** — **Huntsville, AL** Department Manager • Supervised multiple associates. • Ordered and stocked merchandise.
HONORS AND AWARDS	Transfer Scholarship to The University of Alabama in Huntsville, The President's List, Who's Who Among Students in American Junior Colleges, Certificate of Achievement for Essay Writing on the Collegiate Assessment of Academic Proficiency (CAAP), Most Outstanding Student – Philosophy, Nomination for Top Ten Students, Certificate of Appreciation for Volunteer Work at the Academy of Science and Foreign Languages
AFFILIATIONS	Phi Theta Kappa, Sigma Kappa Delta, Ridgecrest Elementary School, Academy for Science and Foreign Languages, Children's Miracle Network, Relay for Life, Toys 4 Tots, Tools for School, Downtown Rescue Mission, Breaking Free Mission, Huntsville Public Library, Children's Hospital, United Way, LifeSouth, John Dau Foundation, American Cancer Society, Animal Services, Arbor Day Foundation

Zachary Snodgrass
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zds0001@uah.edu

5116 Riverview Drive
Huntsville, AL 35803

CITIZENSHIP U. S.

TECHNICAL SKILLS Microsoft Office Suite 2003 and 2007, SolidWorks, Windows Movie Maker, Adobe Photoshop, MATLAB, Mathcad, 3-D printer experience

EDUCATION
The University of Alabama Huntsville — **Huntsville, AL**
Bachelor of Science in Engineering with a concentration in Aerospace Engineering
GPA: 3.28/4.0, Expected graduation May 2012

WORK EXPERIENCE
Aug 2010 – Present — **Carrabba's Italian Grill** — **Huntsville, AL**
Host/Carside
- Welcome and seat customers, serve carry-out orders, assist servers, and help customers.
- Utilize people skills, multitasking, and good work ethic.

May 2005 – Jan 2010 — **U. S. Army** — **Huntsville, AL**
Engineering Aide
- Worked on 2.75 rocket, Tow, and Javelin systems.
- Used layers of composites to build helicopter doors to fit on the CH-47 Chinook.
- Used the program SolidWorks to design 3-D models and in some cases used 3-D printers to manufacture equipment based on CAD models.

CLEARANCE Secret, granted by the U. S. Army in 2005

HONORS AND AWARDS Future Business Leaders of America (FBLA), Second place in Science and Engineering Apprentice Program (SEAP) presentation completion

Christopher Seth Webb
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csw0002@uah.edu

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Huntsville, AL 35803

CITIZENSHIP	U. S.
TECHNICAL SKILLS	MATLAB, Microsoft Word, Microsoft Excel, Mathcad, Patran, Adobe Photoshop, Solid Edge
EDUCATION	**The University of Alabama in Huntsville** **Huntsville, AL** Bachelor of Science in Engineering with a concentration in Mechanical Engineering GPA: 3.0/4.0, Expected graduation December 2011
WORK EXPERIENCE	**Sep 2009 – Present** **Limited Brands** **Huntsville, AL** Sales Associate • Process shipments. • Increase sales.
HONORS AND AWARDS	Academic Excellence Scholarship , Third place in Science and Engineering Apprentice Program (SEAP) presentation completion
AFFILIATIONS	Treasurer of Alpha Lambda Delta (Freshman Honor Society) in 2007, Third place in Science and Engineering Apprentice Program (SEAP) presentation completion

Phillip Dean Webster
(256)-227-1462
phillip.webster@uah.edu

Current Address	Permanent Address
2465 Mobile Drive	1222 Byron Avenue
Huntsville, AL 35805	Decatur, AL 35601

CITIZENSHIP U. S.

TECHNICAL SKILLS Microsoft Office Suite, AutoCAD, Solid Edge, Patran/Nastran, Mathcad, MATLAB, NIMMS Certification

EDUCATION

The University of Alabama in Huntsville **Huntsville, AL**
Bachelor of Science in Engineering with a concentration in Mechanical Engineering
GPA: 3.1/4.0, Expected graduation December 2011
- Completed the MAE 490 Introduction to Engineering Design Senior Project

Calhoun Community College **Decatur, AL**
Associate in Applied Mathematics
GPA: 3.0/4.0, graduate May 2007

WORK EXPERIENCE

May 2007 – Present **The University of Alabama in Huntsville** **Huntsville, AL**
Student Assistant III
- Manage technical equipment such as laptops and projectors.
- Refurbish returned classroom materials.

Summer 2009/2010 **U. S. Department of Defense** **Ansbach, Germany**
General Engineer
- Developed and implemented a project management database for the Garrison's Public Works Engineering Division.
- Developed, designed, planned, and implemented a project to alter interior walls for the construction of a reception area.

Jul 2007 –Oct 2007 **EDO-CAS** **Huntsville, AL**
- Assembled, maintained, and repaired manual and electric wheelchairs.

HONORS AND AWARDS Eagle Scout, Vigil Honor, Decatur Elks Club 655 Scout of the Year for 2001, Rotary International Youth Leadership Award

AFFILIATIONS American Society of Mechanical Engineering, Boys Scouts of America

Jason Wilborn
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Scottsboro, AL, 35768

CITIZENSHIP	U. S.
TECHNICAL SKILLS	Solid Edge, SolidWorks, Patran, Nastran, LabVIEW, Mathcad, MATLAB, Microsoft Word, Microsoft Excel, Microsoft PowerPoint
EDUCATION	**The University of Alabama in Huntsville** — **Huntsville, AL** Bachelor of Science in Engineering with a concentration in Mechanical Engineering GPA: 3.15/4.0 in major, Expected graduation May 2011
WORK EXPERIENCE	**Aug 2009 – Present** — **U. S. Army AMRDEC** — **Redstone Arsenal, AL** Cooperative Education Student • Filament winding. • Hand lay-up of composites. • Solid modeling for structural analysis.
CLEARANCE	Secret, August 2009, U. S. Army
PUBLICATIONS	Owens, A. T., Busby, D. I., Wilborn, J. C., Roberts, J. K. “Through Thickness Pins for Improved Thermal Conductivity in Filament Wound Composite Laminates.” Army Technical Report AMR-XX-08-XX. Submitted for publication January 2011.

Nathan Johnstone Towles
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Augusta, SC 29841
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EDUCATION

B.S.Physics Expected December 2011 College of Charleston, Charleston, SC

COURSE WORK

Math
Calculus I-III, Differential Equations

Physics
General Physics I-II, Modern Physics I-II, Experimental Physics, Classical Mechanics, Nuclear Physics, Fluid Dynamics, Quantum Mechanics, Electricity and Magnetism

Astronomy
Planetary Astronomy, Stellar Astronomy and Astrophysics, NASA Space Mission Project Design and Application

COMPUTER EXPERIENCE

Operating Systems
Experienced Windows OS and Mac OS user. Moderately experienced Ubuntu Linux user

Research and Technical Software
Mathematica, Spectrasuite, Adobe CS4 Suite, LaTeX, Inkscape, The Sky 6, CCD Soft, Bisque TCS, AutomaDome, Microsoft Office Suite, iLife 2011 Suite, TPoint, Final Cut Studio Suite

Programming
Moderate experience in MPython, HTML, and Java Script

RESEARCH EXPERIENCE

Student Research
Current research consists of taking fluorescence spectral measurements of tissue phantoms, which contain known quantities of an FDA-approved photosensitizing drug (photofrin). The data is used to calibrate a quantification method that I helped develop in the summer of 2009 and for which a patent application has been filed with myself listed as an inventor. The spectral method will then be used to quantify a new experimental photo-sensitizer, HPPH. (January 2011-Current)

Principal Investigator
Operating as a PI for an Europa Explorer mission. This is an academic exercise created to be part of a competitive undergraduate NASA STEM development program with Marshall Space Flight Center, funded through NASAs SMD. The project consists of teams of scientists from the College of Charleston who are integrating with teams of engineers from the University of Alabama in Huntsville to develop and design a mission concept to accomplish goals and objectives outlined in the Discovery Announcement of Opportunity. The final products of the project is a formal grant proposal strictly adhering to the guidelines in the AO and a presentation of the mission concept to an external review panel, consisting of NASA employees and members from the Aerospace industry. (August 2010 -Current)

Co-Principal Investigator

Operated as a Co-PI for an Asteroid Sample Return mission. This was an academic exercise created to be part of a competitive undergraduate NASA STEM development program with Marshall Space Flight Center, funded through NASAs SMD. The project consisted of teams of scientists from the College of Charleston who integrated with teams of engineers from the University of Alabama in Huntsville to develop and design a mission concept to accomplish goals and objectives outlined in the New Frontiers Announcement of Opportunity. The final products of the project were a formal grant proposal strictly adhering to the guidelines in the AO and a presentation of the mission concept to an external review panel, consisting of NASA employees and members from the Aerospace industry. (May 2010 -August 2010)

Principal Investigator
Operated as a PI for an Io Observer mission. This was an academic exercise created to be part of a competitive undergraduate NASA STEM development program with Marshall Space Flight Center, funded through NASAs SMD. The project consisted of teams of scientists from the College of Charleston who integrated with teams of engineers from the University of Alabama in Huntsville to develop and design a mission concept to accomplish goals and objectives outlined in the New Frontiers Announcement of Opportunity. The final products of the project were a formal grant proposal strictly adhering to the guidelines in the AO and a presentation of the mission concept to an external review panel, consisting of NASA employees and members from the Aerospace industry. (September 2009 -May 2010)

Student Research
Developed a non-invasive method to quantify photo-sensitizer in vivo using fluorescence spectroscopy. Worked in collaboration with the Mayo Clinic in Jacksonville towards an optimized light dosimetry of photo-dynamic therapy treatments for patients suffering from Barretts esophagus. Funded through a NIH AREA Grant. (May 2009 -December 2009)

INSTRUMENTATION EXPERIENCE

Fiber Optic Fluorescence and Reflectance Spectroscopy Photometer Spectrofluorimeter 16-inch DFM Telescope at College of Charleston Confocal Microscope Scanning Electron Microscope Three-Axis Hall Probe

WORK EXPERIENCE

Apple Retail
Currently working at the store in Charleston, SC as a sales specialist in technology. Work responsibilities include disseminating general knowledge to customers, operating call in support line, restocking products, helping set up new products, trouble shooting purchased products errors, teaching informational workshops, managing/creating employees' daily schedules, and closing the store at night. (August 2010 - Current)

Student Helper
Monthly Astronomy Night open house facilitator at the College of Charleston. Responsibilities include introducing the local community to the observatory, the 16" DFM telescope, and giving brief science and astronomy based presentations. (September 2009 -Current)

Observatory Technician
Worked at the College of Charleston Observatory to integrate the new computer system with observational instruments in order to completely automate the control systems for the 16" DFM telescope and its peripherals. (September 2009-Current)

Student Intern
The internship was at the University of Alabama in Huntsville to design the Integrated Product Team classes for Fall 2010 and Spring 2011 semesters. Also functioned as mentor to local Alabama high school students who were competing to design planetary science mission concepts as an academic exercise. (May 2010 -September 2010)

Physics and Math Tutor
Math and General Physics tutor at the College of Charleston. (September 2009-May 2010)

PATENTS

Medical Patent
Submitted in conjunction with Mayo Clinic for an analytical algorithm produced for a non-invasive method to quantify photo-sensitizer in vivo. The patent is in the final processing stages, with myself listed as an inventor. (Fall 2009 -Current)

COMMUNICATION SKILLS

Fluent in English. Moderate proficiency in Spanish.

David K. Weiss

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Education
College: Junior and member of class of 2012, College Of Charleston. Charleston, SC
Declared Major: Geology. Relevant completed/current classes include: Environmental Geology, Earth History, Planetary Geology, Geochemistry, Mineralogy, Chemistry, Integrated Project Design & Application, NASA Mission Project Design, Calculus, Statistics, Structural Geology, Intro to Physics I, Intro to Physics II, Global Climate Change, Petrology, Introduction to Carbonate Environments.
High School: Graduate Winston Churchill High School, Potomac, MD, 2008
Research Experience
Introduction to Carbonate Environments Winter '11

Conducted research on carbonate environments on the island of San Salvador, Bahamas.
Research into marine, reef environment, karst topography and cave formation.

NASA Mission Project Design Academic Years 2010-2011
2009-2010
Science Team Mission Leader (PI) on the Philoctetes/Okyrhoe Integrated Asteroid Surveyor (POIAS) Mission in the 2010 NASA Mission Project Design Class. Following the 2010 NASA Announcement of Opportunity, I led the College of Charleston science team and directly coordinated the mission with our engineering team from the University of Alabama, Huntsville. The missions designed in this program were then presented before a NASA review board in the Spring of 2010 in Huntsville, Alabama. Additionally, the product of this is a published poster presentation through both the University of Alabama, Huntsville's Spring 2010 poster session and the College of Charleston's Spring 2010 poster session.

Science Team co-investigator (Co-I) on the Inner Crustal Europa Seismic and Spectral Surveyor (ICESS) mission following the 2010 Discovery Class Mission Announcement of Opportunity in the College of Charleston's NASA Space Mission Design class. I was an integral member of the science team and directly coordinated the mission with our engineering team from the University of Alabama, Huntsville. The missions designed in this program were then presented before a NASA review board in the Spring of 2011 in Huntsville, Alabama. Additionally, the product of this is a published poster presentation through both the University of Alabama, Huntsville's Spring 2010 poster session and the College of Charleston's Spring 2011 poster session.

Soil Geochemistry Spring '10
Analyzed and compared pollutant levels in the waters of urban and suburban retention ponds across the Charleston, South Carolina area and was published in the College of Charleston's Spring 2010 poster session.

Work Experience
South Carolina Space Grant Consortium, Charleston SC Spring '11

Co-Instructor
Co-taught a professional development course, “Fly Me to the Moon”, on Lunar and Planetary Science for elementary through high school teachers in the South Carolina school system through the South Carolina Space Grant Consortium.
Calleva, Poolesville, MD Summer ‘09
Summer ‘10
Outdoor Adventure Camp Instructor/Camp Counselor/Explorers Instructor
Counselor directing and managing groups of 12 kids for white water rafting, fishing, and a farming program.
Instructed and managed groups of younger kids in rock climbing in the Explorers program.
OAC Counselor managing large groups of kids from other camps that outsourced to our program in addition to employees from outsourcing businesses and college students, instructing them in white water kayaking, canoeing, tubing, caving, rock climbing, white water rafting, and a ropes course.
Bullis Day Camp, Potomac, MD Summer ‘08
Counselor
Supervised a group of 10 boys, between the ages of 4 and 12. Directed games, activities, served as life guard, and led fieldtrips.
Strosnider's Hardware, Potomac MD Summer 07
Sales/Cashier Summer, Fall ‘06
Provided consultative sales support to customers to address their hardware problems, needs, and provided advice and insight to work towards a solution.
Served as a cashier in high volume, high pressure environment.
Grill assembly and delivery and other customer support activities
Giant Food, Potomac, MD Summer, ‘05
Cashier/Bagger

Inventory restocking, cashier, and bagging.
Community Work Experience
Har Shalom Synagouge Fall ‘03 to spring ‘04
Hebrew School Teacher Aid 1996-1998
Teacher aid during Hebrew school.
One-on-one tutoring for students in need of extra help.

Skills and Certifications

Intermediate experience with Scanning Electron Microscope (SEM)
Novice at CAD
Proficient at Microsoft Word and Excel
CPR Certified
First Aid trained

Interests and Extracurricular

Six years of wilderness backpacking experience.
Member of the College of Charleston Boxing Club
Founder and Vice President of the College of Charleston Mixed Martial Arts Club
Four year high school varsity wrestler
Fix it mentality backed by strong interest/hobbyist in mechanical projects.

BIOGRAPHICAL SKETCH Hannah - Kate Fowler

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Department of Physics and Astronomy
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Email: hannahkatefowler@gmail.com
Charleston, SC 29424-0001

EDUCATION:
B.S. Business Administration and Minor in Astronomy (Expected May 2012), from College of Charleston

WORK EXPERIENCE:
Office Manager for Solidearth Inc. Summer 2005, 2006, 2007, 2008
Sales Representative for Bloom Inc. Fall 2007 - Spring 2009
Sales Representative for OXETTE USA. Fall 2010
Sales Representative for Pandora. Spring 2011

APPOINTMENTS:
Co-Investigator on ICESSS-O Mission to Europa.

SELECTED RECENT PUBLICATIONS:
1. "The ICESSS-O Europa Experiment", N. Towles, D. Weiss, H.-K. Fowler (2011), *College of Charleston Poster Session*, URL.
2. "An Excellent Experiment to Look at Cool Stuff on Europa", N. Towles, D. Weiss, H.-K. Fowler (2011), *Huntsville Engineering IPT Open House Poster Session*, URL.

SCIENTIFIC/TECHNICAL/MANAGEMENT PERFORMANCE ON RELEVANT PRIOR RESEARCH EFFORTS:
Hannah-Kate Fowler currently serves as the financial expert for the ICESSS-O mission to Europa. She has experience with Mac-OSX, Microsoft Office,

COLLABORATORS:
Nathan Towles, College of Charleston
David Weiss, College of Charleston
Jon Hakkila, College of Charleston
Cassandra Runyon, College of Charleston
Joseph Roth, University of Alabama in Huntsville
Kareem Garriga, University of Alabama in Huntsville

J.4 Master Equipment List (MEL)

MASTER EQUIPMENT LIST - MISSION Europa												
Launch Vehicle		# OF UNITS			FLIGHT HARDWARE MASSES			FLIGHT HARDWARE POWER			OTHER COMPONENT INFORMATION	
Subsystem Component	Unit Mass, Current Best Estimate (CBE)	Flight Units	Flight Spares	EMs & Proto-types	Total Mass, kg CBE	Contin-gency %	Total Mass w/ Contin-gency	Total Power, W CBE	Contin-gency %	Total Power w/ Contin-gency	Description (Vendor, Part #, Heritage Basis)	Other characteristics/issues (volume, other component-specific information)
Orbiter					0.0		0.0			0.0		
Structure	350.00	1	0		350.0		0.0			0.0		
Lander					0.0		0.0			0.0		
Structure	30.50	1	0		30.5		0.0			0.0		
Thermal	88.00	1			88.0		0.0			0.0		
ACS	90.00	1			90.0		0.0			0.0		
Star Tracker	29.00	2			58.0		0.0	10.7				
Reaction Wheel	8.50	4			34.0		0.0	22.0				
SSIRU	7.10	1						38.0				
Power	132.00	1			132.0		0.0			0.0		
ASRG					0.0		0.0			0.0		
Propulsion	183.50	1			183.5		0.0			0.0		
AMBR Engine					0.0		0.0			0.0		
Comm & CDS	168.00	1			168.0		0.0			0.0		
Ka Band Translator					0.0		0.0			0.0		
SCS750ATM Super Computer					0.0		0.0			0.0		
Radiation Shielding	154.50	1			154.5		0.0			0.0		
Fuel	2392.80	1	0		2392.8		0.0			0.0		
Integration					0.0		0.0			0.0		
Total Mass/Power					**3681.3**		**0.0**	**70.7**		**0.0**		

Figure 44 Master Equipment List

Ground Systems Payload		# OF UNITS			FLIGHT HARDWARE MASSES			FLIGHT HARDWARE POWER			OTHER COMPONENT INFORMATION	
Subsystem Component	Unit Mass, Current Best Estimate (CBE) (kg)	Flight Units	Flight Spares	EMs & Proto-types	Total Mass, kg CBE	Contin-gency %	Total Mass w/ Contin-gency	Total Power, W CBE	Contin-gency %	Total Power w/ Contin-gency	Description (Vendor, Part #, Heritage Basis)	Other characteristics/issues (volume, other component-specific information)
Athena Microscopic Imager		1			0.0		0.0			0.0		
Gas Chromatograph		1			0.0		0.0			0.0		
Cassini Radio Science		1			0.0		0.0			0.0		
Mass Spectrometer		1			0.0		0.0			0.0		
Drill		1			0.0		0.0			0.0		
Fluxgate Magnetospheric Imager		2			0.0		0.0			0.0		
Total Mass/Power					**0.0**		**0.0**	**0.0**		**0.0**		

Figure 45 Ground Systems Payload

Space Craft Payload		# OF UNITS			FLIGHT HARDWARE MASSES			FLIGHT HARDWARE POWER			OTHER COMPONENT INFORMATION	
Subsystem Component	Unit Mass, Current Best Estimate (CBE) (kg)	Flight Units	Flight Spares	EMs & Proto-types	Total Mass, kg CBE	Contin-gency %	Total Mass w/ Contin-gency	Total Power, W CBE	Contin-gency %	Total Power w/ Contin-gency	Description (Vendor, Part #, Heritage Basis)	Other characteristics/issues (volume, other component-specific information)
Europa Orbiter Radar Sounder	10.00	1			10.0		0.0	100.0		0.0		Resolution:66 m(r)by 13 km(d) at 100km FOV: 22 deg. Data Rate: 2 mbit/sec.
Magnetospheric Imager	28.11	1			28.1		0.0	20.0		0.0	Cassini	FOV: CHEMS: 160, INCA: 120, LEMMS: 15/30 deg. Data Rate: 7 kilobits/s
Radio Science	14.38	1			14.4		0.0	80.7		0.0	Cassini	Data Rate: 0.9 kilobitsper second direct to earth
Radio & Plasma Wave Science	6.80	1			6.8		0.0	7.0		0.0	Cassini	Data Rate: 0.9 kilobits per second
THEMIS	11.20	1			11.2		0.0	14.0		0.0		Resolution: Infrared = 100m/pixel; Visual = 18m/pixel FOV: Infrared: 4.6 x 3.5 degrees; Visual: 2.9 x 2.9 degrees. Temperature: 245-270 K Data Rate: 0.6 megabits per second
Ion & Neutral Mass Spectrometer	10.29	1			10.3		0.0	27.7		0.0	Cassini	FOV: 8.6 degrees Data Rate: 1.5 kilobits/s (average data rate)
Micro-Imager	4.20	2			8.4		0.0			0.0		
Near Infrared Mapping Spectrometer	18.00	1			18.0		0.0	12.0		0.0	Galileo	Resolution: .0125 @ wavelengths less then 1micron .0250 @ wavelengths greater than 1 micron FOV: 10 mrad x .5 mrad Temperature: 150 K Data Rate: 0.6 megabits per second
Ultraviolet Spectrometer	5.20	1			5.2		0.0			0.0	Galileo	Resolution: 0.7 nm below 190 nm and 1.3 nm above 190 nm FOV: 1 x 0.1 degree 0.4 x 0.1 degree Temperature: 263 - 303 K operational 253 - 303 K not operational Data Rate: .144 kilobits per second
Osiris NAC/WAC	24.10	1			24.1		0.0	27.00		0.0		
NAC	13.20											
WAC	9.50							20.0				
Electronics Box	1.40	1			1.4		0.0	7.0				
Penetrators		23	6		0.0		0.0			0.0		
Impactor		1	0		0.0		0.0			0.0		
LIDAR	3.56	1			3.6		0.0	22.00		0.0	Hayabusa	
Instrument								17.0				Resolution: 300 m by 133 m at 100 km FOV: 1 mrad Temperature: 283-333 K Data Rate: 3.008 kilobits per second
Heater		1			0.0		0.0	5.0		0.0		
Total Mass/Power					**141.4**		**0.0**	**359.4**		**0.0**		

Figure 46 Spacecraft Payload

J.5 Heritage

Instrument heritage was considered during risk assessments on the overall success of the mission. "Heritage refers to the original manufacturer's level of quality and reliability that is built into parts and which has been proven by (1) time in service, (2) number of units in service, (3) mean time between failure performance, and (4) number of use cycles. High-heritage products are from the original supplier, who has maintained the great majority of the original service, design, performance, and manufacturing characteristics. Low heritage products are those that (1) were not built by the original manufacturer; (2) do not have a significant history of test and usage; or (3) have had significant aspects of the original service, design, performance, or manufacturing characteristics altered." (NASA, NASA Systems Engineering Handbook 2007)

J.5.1 Launch Vehicle

"The capabilities available for heritage vehicles have been maintained and enhanced in the Atlas V" design. "The two-chamber RD-180 (Figure A.2.2.3-1) is a derivative of the four-chamber RD-170/171 engines used on Russia's Energia boosters (more than 25 flights)." "The RD-180 is flight-proven for Atlas V 400 and 500 series LV configurations (25 flights at print date) with more than 41,400 seconds in 221 firings at time of this publication." (Guide 2010)

J.5.2 Instrumentation

The Cassini mission is the first space mission where one magnetometer having a dual vector or scalar capability has been flown. The purpose of the magnetometer is to measure the magnetic field of the spacecraft. It was used on Cassini's mission around Saturn. It measured Saturn's magnetic field as well as that of the Titan. However, our mission uses this system to measure the atmosphere around the spacecraft as well as the spacecraft itself. The UVS was utilized on the Galilean Mission Voyager 1 and Voyager 2. It is configured similarly enough to consider it a full heritage. For more information on how the instrumentation is used on orbiter refer to section E. The Probe we attempted on the Mars missions but was not unsuccessful.

Table 37 Heritage

Subsystem	Proposed Design	Heritage	Prior Usage
Propulsion	AMBR	N/A	N/A
Power	ASRG	N/A	N/A
ACS	Mono Propellant Hydrazine Thrusters	Full	
Structure	Octagonal	Depends on level of Adaption	Messenger, Mariner4, etc
Communications	LGA and HGA	Depends on deployment and mounting	
Thermal	N/A	N/A	N/A

J.6 List of Abbreviations and Acronyms

AA	Associate Administrator
AIME	Asteroid-Moon Micro-Imager Experiment
AL	Alabama
AM&O	Agency Management and Operations
AMBR	Advanced Material Bipolar Rocket
AO	Announcement of Opportunity
AOR	Authorized Organizational Representative
ASRG	Advanced Stirling Radioisotope Generator
CADRe	Cost Analysis Data Requirement
CASP	Cross-Agency Support Program
CCR	Central Contractor Registry
CD-ROM	Compact Disc-Read Only Memory
CDR	Critical Design Review
CHEMS	Charge Energy Mass Spectrometer
DSN	Deep Space Network
EA	Environmental Assessment
EIS	Environmental Impact Statement
ESTACA	Ecole Supérieure des Techniques Aéronautiques et de Construction Automobile
EPO	Education Public Outreach
EORS	Europa Orbiter Radar Sounder
EVM	Earned Value Management
FAQ	Frequently Asked Questions
FY	Fiscal Year
GCMS	Gas Chromatograph and Mass Spectrometer
GDS	Ground Data System
IAT	Integration, Assembly, and Test
ICD	Interface Control Document
INCA	Ion and Neutral Camera
INMS	Ion and Neutral Mass Spectrometer
IPT	Integrated Product Team
JEO	Jupiter Europian Orbiter
JPL	Jet Propulsion Laboratory
JSC	Johnson Space Center
LEMMS	Low Energy Magnetospheric Measurement System
LEO	Lower Earth Orbit
LIDAR	Light Detection and Ranging
LOLA	Lunar Orbiter Laser Altimeter
LV	Launch Vehicle
MCR	Mission Concept Review
MEL	Master Equipment List
MEU	Main Electrical Unit
MI	Microscopic Imager
MIMI	Magnetospheric Imaging System
MIPS	Million Instructions Per Second
MMRTG	Multiple Mission Radioisotope Thermoelectric Generator
MO&DA	Mission Operations and Data Analysis
MOS	Mission Operations Services
MOU	Memorandum of Understanding
NAC	Narrow Angle Camera
NASA	National Aeronautics and Space Administration
NASA-	STD NASA-Standard
NEN	Near-Earth Network
NEPA	National Environmental Policy Act
NFS	NASA FAR Supplement
NIMS	Near Infared Mapping Spectrometer
NISN	NASA Integrated Services Network
NLSA	Nuclear Launch Safety Approval
NODIS	NASA Online Directives Information System
NOI	Notice of Intent
NPD	NASA Policy Directive
NPR	NASA Procedural Requirements
NSEH	NASA System Engineering handbook
NRA	NASA Research Announcement
NRC	National Research Council
NRP	NASA Routine Payload
NSPIRES	NASA Solicitation and Proposal Integrated Review and Evaluation System
NSS	NASA Safety Standard
NTO	Nitrogen Tetroxide
OSIRIS	Optical, Spectroscopic, and Infrared Remote Imaging System
PDF	Portable Data Format
PDR	Preliminary Design Review
PI	Principal Investigator
PM	Project Manager
POC	Point of Contact
PS	Project Scientist
PSE	Project Systems Engineer
RAT	Rock Abrasion Tool
RHU	Radioisotope Heater Unit
ROD	Record of Decision
ROM	Rough Order-of-Magnitude

RPS	Radioisotope Power System
RPWS	Radio and Plasma Wave Science
RR	Readiness Review
RSS	Radio Science Subsystem
RTG	Radioisotope Thermoelectric Generator
RY	Real Year
SC	Space Craft
SCaN	Space Communication and Navigation
SDR	System Design Review
SE	System Engineering
SEO	Science Enhancement Option
SMD	Science Mission Directorate
SN	Space Network
SOW	Scope of Work
SPICE	Spacecraft, Planet, Instrument, C-matrix, Events
SSIRU	Scalable Space Inertial Reference Unit
THEMIS	Thermal Emission Imaging System
TBD	To be decided
TMC	Technical, Management, and Cost
TRL	Technical Readiness Level
TT&C	Telemetry, Tracking, and Commanding
UAH	University of Alabama in Huntsville
URL	Uniform Resource Locator
U.S.	United States
USB	Universal Serial Bus
U.S.C.	United States Code
VEEGA	Venus Earth Earth Gravity Assist
WAC	Wide Angle Camera
WBS	Work Breakdown Structure

J.7 Reference

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J.8 NASA-Developed Technology Infusion Plan

The AMBR engine was designated as the propulsion system of choice and no alternatives were presented. The ASRG was chosen over several alternatives such as the RTG, the use of batteries, and solar arrays. However, because of the ASRG having the lowest W/kg rating, three units are needed to provide the wattage required for the mission. Despite this, the ASRGs offer greater cost savings and the lowest power degradation ratings as compared to the three remaining power systems that were defined in the power system trade study.

J.9 Description of Enabling Nature of ASRG

Four types of space craft power systems were evaluated in the Power System Trade Study. Batteries, solar panels, ASRG (Advanced Stirling Radioisotope Generator), and RTG (Radioisotope Thermal Generator) were chosen because of their prior use in space craft design. The conclusion of the trade study shows that the decision to use the three, government provided, ASRGs is the most adequate option for the Europa mission.

The FOM (Figures of Merit) used to assess the four energy sources are cost, power per unit mass, and degradation of the power system. Watts per unit mass is the FOM that is rated the highest because it would be detrimental to the mission if there wasn't enough power to operate the science equipment once the space craft reaches Europa. The second FOM is degradation of the power system because the entire system has to be able to maintain its full capabilities throughout the length of the mission. Finally, cost was the last FOM for the trade study. It was considered to have the lowest rating because the other FOM are critical to successfully fulfilling the purpose of the Europa Mission.

The ASRG is designed to comply with the planetary protection protocols. A preliminary assessment was done in 2007 to determine if there was an issue with the heat sterilization of the ASRG. An extensive analysis was not performed but radiation sterilization was found acceptable to meet the planetary protection requirements in the JEO, Section 4.4.5.1. (JEO 2009)

J.10 Calculations

J.10.1 UAHuntvsille Orbiter

The calculations begin by sizing the propellant mass in order to obtain the dry mass of the system.

Overall analysis of orbiter.

$m_{total} := 4615\,kg$ $\qquad\qquad g_0 := 9.8065\,\dfrac{m}{s^2}$

$m_{0\_orbiter} := m_{total} = 4.615 \times 10^3\,kg$

Total mass comes from NASA Launch Services

$I_{sp} := 335\,s$ $\qquad\qquad I_{sp\_ACS} := 210\,s$

$Thrust\_AMBR := 200\,lbf$ $\qquad\qquad Thrust\_AMBR = 889.6443\,N$

All of that comes from AMBR_engine.pdf

$\Delta V := 2260\,\dfrac{m}{s}$ $\qquad\qquad deltaVtoJovianTour := 914\,\dfrac{m}{s}$

based on JEO table ***Table 4.3-3. page*** 4.3-6

$$m_{prop\_orbiter} := m_{0\_orbiter} \cdot \left(1 - e^{\frac{-\Delta V}{g_0 \cdot I_{sp}}}\right)$$

$$m_{prop\_orbiter} = 2.2954 \times 10^3\,kg$$

The propellant mass for the ACS/RCS system must also be taken into account.

$$m_{prop\_toJTour} := m_{total} \cdot \left(1 - e^{\frac{-deltaVtoJovianTour}{g_0 \cdot I_{sp}}}\right) = 1.1208 \times 10^3\,kg$$

$$m_{orbiter\_at\_JT} := m_{total} - m_{prop\_toJTour} = 3.4942 \times 10^3\,kg$$

$$deltaV\_ACS\_RCS := .025 \cdot \Delta V = 56.5\,\frac{m}{s}$$

$$m_{Hydrazine\_ACSRCS} := m_{total} \cdot \left(1 - e^{\frac{-deltaV\_ACS\_RCS}{g_0 \cdot I_{sp\_ACS}}}\right) = 124.8944\,kg$$

2.5% of the total delta-V was used as the ACS/RCS delta-V, based on discussions with Dr. Turner. Additionally, the mass was checked against the mass used in the JEO, and it was more than that used in the JEO, so the team utilized the above value, confident it was a conservative value.

$m_{dry\_orbiter} := m_{total} - m_{prop\_orbiter} - m_{Hydrazine\_ACSRCS} = ?.1947 \times .0^{3} \cdot kg$

$$PMF\_orbiter := \frac{m_{prop\_orbiter} + m_{Hydrazine\_ACSRCS}}{m_{total}}$$

$PMF\_orbiter =).5245$

$m_{margin\_1} :=).3(\cdot m_{dry\_orbiter} = \,58.3968\,kg$

Saving 30% of the dry mass for margin

$m_{useable\_orbiter\_1} := m_{dry\_orbiter} - m_{margin\_1} = 1.5363 \times .0^{3}\,kg$

Next, the masses of the deployed systems were removed for book-keeping. All of these masses were decided by the team. The lander was allotted 130kg based on the small amount of science onboard the lander.

$m_{allotted\_lander} := 130\,kg$

$m_{allotted\_impactor} := 40\,kg$

$m_{allotted\_seismic} := 150\,kg$

$m_{seismic\_devices} := 3.572\,kg \cdot 29 = \;03.588\,kg$

$m_{useable\_orbiter\_real} := m_{useable\_orbiter\_1} - m_{allotted\_seismic} - m_{allotted\_lander} - m_{allotted\_impactor} = 1.2163 \times .0^{3}\,kg$

Below is a preliminary breakdown of the masses expected for each system. The percent factor used to size each system was decided upon after researching the JEO and multiple orbiter missions, and referring to Charles Brown Elements of Spacecraft Design to get an idea of what percent of total dry mass each system usually requires.

$m_{structure\_1} := 28 \cdot m_{useable\_orbiter\_real} = \,40.5526\,kg$

$m_{structure} := 350\,kg$

$m_{thermal\_1} := 063 \cdot m_{useable\_orbiter\_real} = \,'6.6243\,kg$

$m_{thermal} := 78\,kg$

$m_{ACS\_1} := 075 \cdot m_{useable\_orbiter\_real} = \,'1.2194\,kg$

$m_{acs} := 30\,kg$

$m_{power\_1} := 097 \cdot m_{useable\_orbiter\_real} = \;17.977\,kg$

$m_{power} := 115\,kg$

$m_{propulsion\_1} := 135 \cdot m_{useable\_orbiter\_real} = \;64.195\,kg$

$m_{propulsion} := 160\,kg$

$m_{comm\_1} := 063 \cdot m_{useable\_orbiter\_real} = \,'6.6243\,kg$

$m_{comm} := 78\,kg$

$m_{comp\_data\_system\_1} := 063 \cdot m_{useable\_orbiter\_real} = \,'6.6243\,kg$

$m_{CDS} := 78\,kg$

$m_{rad\_shielding\_1} := 11\% \cdot m_{useable\_orbiter\_real} = 38.6536\,kg$

$m_{rad\_shield} := 135\,kg$

$m_{science\_PL} := 131.6\,kg$ $m_{scientific\_1} := 11\% \cdot m_{useable\_orbiter\_real} = 33.7885\,kg$

A running total of the orbiter mass is kept through the equation below

$$m_{total\_orbiter} := m_{prop\_orbiter} + m_{Hydrazine\_ACSRCS} + \left(m_{allotted\_lander} + m_{allotted\_impactor} + m_{allotted\_seismic} + m_{structure\_1} + m_{thermal\_1} + m_{ACS\_1} + m_{power\_1} + m_{propulsion\_1} + m_{comm\_1} + m_{comp\_data\_system\_1} + m_{rad\_shielding\_1} + m_{scientific\_1} + 25\,kg\right) \cdot 1.43$$

The terminology "_1" is used to denote that there are the first cut values of the orbiter, based on generic percents.

A preliminary analysis of the lander, used to obtain the lander's dry mass, is below. The team used the dry mass to verify that the lander would be able to sustain the 5kg of science it is designed to deliver to the Europan surface.

$I_{sp\_lander} := 220\,s$

$h_{orbit} := 100\,km$

$g_{Europa} := 1.314\,\frac{m}{s^2}$

$Mass\_allocated\_to\_lander\_on\_orbiter := 130\,kg$

$\Delta V_{lander} := \sqrt{2 \cdot g_{Europa} \cdot h_{orbit}} = 512.6402\,\frac{m}{s}$

$m_{0\_lander} := Mass\_allocated\_to\_lander\_on\_orbiter = 130\,kg$

$\Delta V_{lander\_} := 1.488\,\frac{km}{s}$

$$m_{prop\_lander} := m_{0\_lander} \cdot \left(1 - e^{\frac{-\Delta V_{lander\_}}{g_0 \cdot I_{sp\_lander}}}\right) = 64.7762\,kg$$

$m_{science\_L} := 5\,kg$

$mass\_Landed_{lander} := m_{0\_lander} - m_{prop\_lander} = 65.2238\,kg$

$$\text{sciencepercent}_{Lander} := \frac{m_{science\_L}}{\text{mass\_Landed}_{lander}} = 0.0767$$

$$\text{mass}_{Europa} := 4.8 \cdot 10^{22}\,kg \qquad G_{grav} := 6.67428 \cdot 10^{-11}\,m^3 \cdot kg^{-1} \cdot s^{-2} \qquad G_{grav} = 6.6743 \times 10^{-11}\,\frac{m^2}{kg^2} \cdot N$$

$$\mu_{Europa} := G_{grav} \cdot \text{mass}_{Europa} = 3.2037 \times 10^{12}\,\frac{m^3}{s^2}$$

$$m_{Jup} := 1.8986 \cdot 10^{27}\,kg = 1.8986 \times 10^{27}\,kg$$

$$r_{Jup} := 66854\,km = 6.6854 \times 10^{4} \cdot km$$

These came from Wikipedia

Assuming an average orbit height of 75 km, based on the elliptical 50kmX100km altitude orbit, the radius of our orbit around Europa, assuming circular orbit becomes:

$$r_{Europa} := 1569\,km + 75\,km = 1.644 \times 10^{6}\,m$$

$$vel_{Europa} := \left(\frac{\mu_{Europa}}{r_{Europa}}\right)^{.5} = 1.396 \times 10^{3}\,\frac{m}{s}$$

The JEO shows a delta-V of 792m/s to inject into Europan orbit. The velocity of the impact device is assumed to be the velocity ICESSS is moving before this maneuver, which is its Europan orbit velocity + 792:

$$vel_{Europa} := \left(\frac{\mu_{Europa}}{r_{Europa}}\right)^{.5} = 1.396 \times 10^{3}\,\frac{m}{s}$$

$$vel_{Impactor} := \left(vel_{Europa} + 792\,\frac{m}{s}\right) = 2.188 \cdot \frac{km}{s}$$

$$\text{circumference}_{Europa} := 2 \cdot \pi \cdot r_{Europa} = 1.033 \times 10^{7}\,m \qquad \text{period}_{Europa} := \frac{\text{circumference}_{Europa}}{vel_{Europa}} = 2.055 \cdot hr$$

After determining the impactor velocity, the team looked at the velocity and angle of impact for the seismic probes, assuming they are released at 75km altitude.

$$h_{release} := 75\,km$$

$$vel_{vert} := \left(2 \cdot g_{Europa} \cdot h_{release}\right)^{.5} = 443.9595\,\frac{m}{s}$$

$$vel_{horiz} := vel_{Europa} = 1.396 \times 10^{3}\,\frac{m}{s}$$

$$vel_{total} := \left(vel_{vert}^{2} + vel_{horiz}^{2}\right)^{.5} = 1.4649\,\frac{1}{s} \cdot km$$

$$angle\_impact := asin\left(\frac{vel_{vert}}{vel_{total}}\right) = 17.642 \cdot deg$$

Next, the thrust to weight ratio is determined, to figure out the number of AMBR engines needed. This requires looking at the weight at both Jupiter and Europa

$$perijove\_at\_JOI := 5.2 r_{Jup} = 3.4764 \times 10^{8}\,m$$

From the JEO, the insertion burn for Jupiter Orbit Insertion occurs at 5.2 Jovian radii.

$$Weight_{Jupiter} := G_{grav} \cdot \frac{m_{total} \cdot m_{Jup}}{perijove\_at\_JOI^{2}} = [illegible].8389 \times 10^{3}\,N$$

Based on the JEO, the delta-V to Europa was determined, and mass for the propellant used up was removed from the total mass of the spacecraft to find weight at Europa:

$$deltaVtoEuropa := 1379\,\frac{m}{s}$$

$$m_{prop\_toEuropa} := m_{total} \cdot \left(1 - e^{\frac{-deltaVtoEuropa}{g_0 \cdot I_{sp}}}\right) = 1.582 \times 10^{3}\,kg$$

$$m_{orbiter\_at\_EIO} := m_{total} - m_{prop\_toEuropa} = 3.033 \times 10^{3}\,kg$$

$$Weight_{Europa} := G_{grav} \cdot \frac{m_{orbiter\_at\_EIO}\, m_{Europa}}{r_{Europa}^{2}} = 3.5951 \times 10^{3}\,N$$

Due to the larger weight at Jupiter, the weight Jupiter is what the team must base the propulsion system's thrust to weight ratio on.

Based on Brown Elements of Spacecraft Design, a thrust to weight of .5 ensures minimal gravity losses, whereas cases of between .3 and .5 must be analyzed to verify the assumption, and smaller thrust to weight ratios not results in negligible burn losses.

$Thrust\_to\_Weight := 5$

$$Number\_of\_Engines := \frac{Weight_{Jupiter} \cdot Thrust\_to\_Weight}{Thrust\_AMBR} = .7196$$

$Engines\_Whole := 3$ 4 engines yeilds .73 then .9898

$$Thrust\_to\_Weight\_Real\_Europa := \frac{Thrust\_AMBR \cdot Engines\_Whole}{Weight_{Europa}} = .7424$$

$$Thrust\_to\_Weight\_Real := \frac{Thrust\_AMBR \cdot Engines\_Whole}{Weight_{Jupiter}} = .5516$$

This high thrust to weight has the added benefit of still allowing us to be over 0.35 in the event of loss of an AMBR engine, allowing the mission to carry on with minimal propellant losses.

Next, the propellant tanks are sized:

$\rho_{NTO} := 1.45 \frac{gm}{cm^3}$ $MW_{NTO} := 92.01 \frac{g}{mol}$

$r_{ox\_to\_fuel} := 1.2$

$\rho_{Hydrazine} := 1.004 \frac{gm}{cm^3}$ $MW_{Hydrazine} := 32.05 \frac{g}{mol}$

$m_{prop\_orbiter} = 2.2954 \times 10^3\ kg$

$$m_{NTO} := m_{prop\_orbiter} \cdot \frac{r_{ox\_to\_fuel}}{1 + r_{ox\_to\_fuel}} = .2521 \times 10^3\ kg$$

$$m_{Hydrazine\_prop} := m_{prop\_orbiter} \cdot \frac{1}{1 + r_{ox\_to\_fuel}} = .0434 \times 10^3\ kg$$

$$Ideal\_Vol_{NTO} := \frac{m_{NTO}}{\rho_{NTO}} = .863 \cdot m^3$$

$Vol_{ullage} := 0.$

The ullage and residual values were given in AMBR_engine.pdf

$Vol_{resid} := 0.$

$$Vol_{NTO} := Ideal\_Vol_{NTO} \cdot (1 + Vol_{ullage} + Vol_{resid}) = .9153 \cdot m^3$$

$$Ideal\_Vol_{Hydrazine} := \frac{m_{Hydrazine\_ACSRCS} + m_{Hydrazine\_prop}}{\rho_{Hydrazine}} = .163\cdot m^3$$

$$Vol_{Hydrazine} := Ideal\_Vol_{Hydrazine} + Vol_{ullage} + Vol_{resid} = 1.2334\cdot m^3$$

$P_{tank} := 400\cdot psi$ $\qquad$ $SafetyFac := 1.5$

$$P_{design} := SafetyFac\cdot P_{tank} = 600\cdot psi$$

Assuming Grade V Titanium, 6Al-4V is the tank material:

$$E_{Ti} := 113.8GPa = 1.138 \times 10^{11}\ Pa$$

$$\rho_{Ti} = 0.159\cdot\frac{lb}{in^3}$$

$$\rho_{Ti} := 4420\frac{kg}{m^3}$$

Values come from
http://www.veridiam.com/pdf/DataSheetTitaniumAlloy.pdf

$$yield\_strength_{Ti} := 825MPa = 8.25 \times 10^8\ Pa$$

From
http://asm.matweb.com/search/SpecificMaterial.asp?bassnum=MTP641

$$r_{poisson} := .342$$

$shear\_modulus := 44GPa = 4.4\times 10^{10}\ Pa$ $\qquad$ $shear\_strength := 550MPa = 5.5\times 10^8\ Pa$

The below equation comes from Charles Brown Elements of Spacecraft Design

$$Mass\_tank(r, w) := 2\pi\cdot r^2\cdot(r + w)\cdot P_{design}\cdot\frac{\rho_{Ti}}{yield\_strength_{Ti}}$$

$r_{Hydrazine} := .55m$

$w_{Hydrazine} := .5m$

Given

$$Vol_{Hydrazine} = w_{Hydrazine}\cdot\pi\cdot r_{Hydrazine}^2 + \frac{4}{3}\pi\cdot r_{Hydrazine}^3$$

$w_{Hydrazine} > .4m$

Note that width is arbitrary, and can be changed. As width approaches zero, we get a sphere with lower limit on mass.

$r_{Hydrazine} > 0m$

$$\text{Hydrazine\_sizes} := \text{Minimize}\left(\text{Mass\_tank}, r_{Hydrazine}, w_{Hydrazine}\right) = \begin{pmatrix} 0.5788 \\ 0.4 \end{pmatrix} m$$

$$mass_{Hydrazine\_tank} := \text{Mass\_tank}\left(\text{Hydrazine\_sizes}_{0,0}, \text{Hydrazine\_sizes}_{1,0}\right) = 5.672\,kg$$

Below is the calculation assuming spherical tank, in order to show the absolute minimum mass required.

$$m_{Hydrazinesphere} := \frac{3}{2} \cdot P_{design} \cdot Vol_{Hydrazine} \cdot \frac{\rho_{Ti}}{yield\_strength_{Ti}} = 1.006\,kg$$

Next, the NTO tank mass is determined in the same way as above:

$r_{NTO} := 55m$ $\qquad w_{NTO} := 5m$

Given

$$Vol_{NTO} = w_{NTO} \cdot \pi \cdot r_{NTO}^{2} + \frac{4}{3} \pi \cdot r_{NTO}^{3}$$

$$w_{NTO} > 35m$$

$$r_{NTO} > 1m$$

$$\text{NTO\_sizes} := \text{Minimize}\left(\text{Mass\_tank}, r_{NTO}, w_{NTO}\right) = \begin{pmatrix} 0.5263 \\ 0.35 \end{pmatrix} m$$

$$mass_{NTO\_tank} := \text{Mass\_tank}\left(\text{NTO\_sizes}_{0,0}, \text{NTO\_sizes}_{1,0}\right) = 3.804\,kg$$

$$m_{NTOsphere} := \frac{3}{2} \cdot P_{design} \cdot Vol_{NTO} \cdot \frac{\rho_{Ti}}{yield\_strength_{Ti}} = 0.429\,kg$$

The pressurant tanks for the propulsion system are COPV tanks. Being composite materials, a stress analysis to obtain minimum thickness is beyond the scope and capabilities of this team. Instead, the mass of the pressurant tanks is based upon the mass of the pressurant tanks in the AMBR_engine.pdf document, and scaled based on the difference in propellant tank mass for the AMBR_engine.pdf and the propellant tank mass found above.

$m_{press\_tank\_pdf} := 10.5kg$ $\qquad m_{NTO\_Tank\_PDF} := 25kg$

$$mass\_pressurant\_tank := m_{press\_tank\_pdf} \cdot \frac{mass_{NTO\_tank}}{m_{NTO\_Tank\_PDF}} = 4.198\,kg$$

$$m_{pressurant\_tanks} := 2\,mass\_pressurant\_tank = 8.396\,kg$$

$$P_{press} := 31MPa = 3.1 \times 10^{7}\,Pa$$

$SafetyFactor_{press} := 1.5$

$P_{design\_pressurant} := P_{press} \cdot SafetyFactor_{press} = 65 \cdot bar$
(because 2 are needed)

Assuming that the ratio of Helium mass follows the ratio of tank mass already described above, a mass for the helium of the pressurant system is obtained. Knowing the properties of helium, assuming 300K as the temperature, the volume of the tanks is found, for CAD modeling purposes.

$Temp := 300K$

$MW_{He} := 4.0026 \frac{gm}{mol}$

$R_U := 8.31447 \frac{J}{K \cdot mol} = 0.0821 \cdot L \cdot \frac{atm}{mol \cdot K}$

$m_{He\_pdf} := 1.7kg$

$m_{He\_real} := m_{He\_pdf} \cdot \frac{mass_{NTO\_tank}}{m_{NTO\_Tank\_PDF}} = 1.2987kg$

$PV = nR_U \cdot T$

$Vol_{He} := \left(\frac{m_{He\_real}}{MW_{He}} \right) \cdot R_U \cdot \frac{Temp}{P_{press}} = 0.046 \cdot m^3$

$r_{press\_tank} := \left(\cdot \frac{Vol_{He}}{4\pi} \right)^{\frac{1}{3}} = 222.612 \cdot mm$

Next, based on the AMBR_engine.pdf, the Propellant Management Device adds 10% of the tank mass to the mass of the propulsion system:

$m_{prop\_tanks\_plus\_PMD} := 1.1 \cdot \left(mass_{NTO\_tank} + mass_{Hydrazine\_tank} \right) = 37.4246kg$

$m_{AMBR\_engines} := 3 \cdot 5.5kg = 16.5kg$

Below is a mass breakdown of the propellant system values, transducers, feed lines, etc, based on scaling the breakdown used in the AMBR_engine.pdf by a factor of 16/12 to account for 16 RCS thrusters instead of 12. The mass stack is also edited to include the true masses for the tanks.

Item	Number used	PDF total mass (kg)	Scaled Mass (kg)
Helium Fill & Drain Valve	3	0.3	0.4
Helium Filter	6	0.7	0.933333333
Pressurant Pyro Valve	7	1.4	1.866666667
Pressure Regulator	2	4.62	6.16
High Pressure Transducer	1	0.23	0.306666667
Check Valves	4	5.44	7.253333333
Low Pressure Transucer	4	0.92	1.226666667
Ground-Checkout Hand Valve	4	0.3	0.4
Propellant Pyro Valve	3	0.6	0.8
RCS Propellant ISO Valve	2	1.3	1.733333333
Propellant Fill&Drain Valve	6	0.9	1.2
Propellant Filter	3	0.9	1.2
Low Pressure Transucer	6	1.38	1.84
Feed Lines and Misc. Hardware	N/A	12	16
Total Mass		30.99	41.32

$$m_{miscellaneous\_RCS\_plus\_prop} := 41.32kg$$

$$m_{RCS\_thrusters} := 16\cdot .7kg = 1.2kg$$

$$m_{propulsion\_rcs} := m_{prop\_tanks\_plus\_PMD} + m_{He\_real} + m_{pressurant\_tanks}$$
$$+ m_{miscellaneous\_RCS\_plus\_prop} + m_{RCS\_thrusters} + m_{AMBR\_engines} = 187.1394kg$$

$m_{SIRU} := 7.1kg$ Northrop Grumman Scalalbe SIRU, internally redundant, only need one.

2 star trackers, not axially alligned, so there is redundancy. SED 16 Autonomous Star Tracker system by Sodern

$$m_{star\_trackers} := 2\cdot 2.9kg = 5.8kg$$

$$m_{extra} := m_{propulsion\_1} + m_{ACS\_1} - m_{prop\_tanks\_plus\_PMD} - m_{He\_real} - m_{pressurant\_tanks}$$
$$- m_{miscellaneous\_RCS\_plus\_prop} - m_{RCS\_thrusters} - m_{AMBR\_engines} = 68.275kg$$

$$m_{extra2} := m_{extra} - m_{SIRU} - m_{star\_trackers} = 55.375kg$$

This shows how much extra mass is available based on the assumed masses of the ACS and propulsion systems and the true masses of the systems. This actually allows for the possibility of reaction wheels being utilized to increase pointing accuracy for the vehicle.

Bellow, mass is added for 4 reaction wheels and 4 sun sensors

HR14 Constellation Series Reaction Wheels by Honeywell (8.5kg each)

$m_{rxn\_wheel} := 4 \cdot 8.5kg = 4kg$ $\qquad m_{sun\_sensor} := 4 \cdot .3kg = .2kg$

$m_{extra3} := m_{extra2} - m_{rxn\_wheel} - m_{sun\_sensor} = 0.175kg$

$m_{acs\_derived} := m_{rxn\_wheel} + m_{SIRU} + m_{star\_trackers} + m_{sun\_sensor} = 8.1kg$

Next, the Computer Data System is sized.

An engineering data rate and uplink data rate are assumed:

$dr_{uplink} := 500 \qquad dr_{engineering} := 2000$

One orbit of Europa is 2.05 hours. With this, every other 1.025 hour interval, there is no communication to Earth. Additionally, once every 3.5 days (the Europan orbit of Jupiter) there is a 2.5 hour loss of comm as Jupiter is in between Europa and Earth. Thus, the CDS system has to store enough data for all that time, assuming a worst case scenario of 1.025 hours without comm passing, then entering the 2.5 hours behind Jupiter, then coming out from behind Jupiter but being on the far side of Europa for another 1.025 hours (even though this is not how the math would work out, since it is known that C^3PO orbits every 2 hours, if is is behind Jupiter for 2.5, it does not have the full 1.025 hours on the far side of Europa after coming out from behind Jupiter.

$compression := 75\%$ Assume 75% compression, since the vast majority of the data is in image format, which can only be compressed without loss by 75%.

$bps\_all := 41 \cdot \frac{10^6}{s}$ This comes from the data rates of the instruments

Assuming that only half the instruments are on at any given time, which in all likelihood is far higher a percent that truly is able to be on, due to the low downlink capability.

$percent\_operating := 50\%$

$bps\_total := bps\_all \cdot (1 - compression) \cdot percent\_operating = 5.125 \times 10^6 \frac{1}{s}$

$total\_bits\_stored := bps\_total \cdot 4.55hr = .3948 \times 10^{10}$

$Gb\_to\_store := \frac{total\_bits\_stored}{10^9} = 3.9475$

$Number\_of\_SDRAM\_4Gb := \frac{Gb\_to\_store}{4} = 20.9869$

$Num\_SDRAM\_4Gb := 25$ 25, to allow for additional storage in case something unforeseen occurs.

$m_{SDRAM} := 11gm \cdot Num\_SDRAM\_4Gb = 0.275kg$ The 11gm mass is from pdf of product

$m_{R750} := 1.5kg$ This mass is from pdf of product

$m_{PCU} := 15gm \cdot 3 = 0.045kg$ Assume 3 power converter units, because RAD750 uses triple modular redundant processing.

$m_{CDS\_cabling} := 20kg$ SEDS MIL-STD-1773 Fiber Optic Data Bus (or SEDS 1773 bus)

From Brown Elements of SC design:

$m_{EngineeringDataProcessor} := 10kg$ $m_{ScienceDataProcessor} := 15kg$

Although a large percent of the above mass (the cabling mass accounts for over 40% of the total mass) is just an assumed mass, the team feels confident that these masses are conservative estimates, if not over estimates, based on the computer data system mass in the JEO report.

$$m_{CDS\_final} := m_{CDS\_cabling} + m_{SDRAM} + m_{R750} + m_{PCU} + m_{ScienceDataProcessor} + m_{EngineeringDataProcessor} = 46.82\,kg$$

From brows ESCD

$p_{misc} := 20W$ Telecom interface, data flow control, command processing, signal conditioning

$p_{R750} := 10W$

$p_{eng\_data\_processor} := 5W$

$num_{instruments\_orbiter} := 14$

$p_{science\_processor} := 2W + 1W \cdot num_{instruments\_orbiter} = 16W$

$$power_{CDS} := p_{misc} + p_{R750} + p_{eng\_data\_processor} + p_{science\_processor} = 51W$$

The Telecommunications system is sized below:

First, the maximum necessary comm. rate is established, this is twice the science data rate (due to seeing Earth roughly half of the orbit period) plus the comm. rate needed to beam back the accumulated data from the eclipse period noted in the Computer Data System calculations.

$$Comm\_ordinary\_rate := 2 \cdot bps\_total \cdot (1 - compression) = 2.5625 \times 10^{6} \frac{1}{s}$$

$$Comm\_additional\_rate := \frac{total\_bits\_stored}{.5day} \cdot (1 - compression) = 4.8581 \times 10^{5} \frac{1}{s}$$

$$Comm\_rate := Comm\_ordinary\_rate + Comm\_additional\_rate = 3.0483 \times 10^{6} \frac{1}{s}$$

$diam_a := 3m$ diameter of the high gain antenna is 3 meters

Based on Charles Brown Elements of SC design, the below equation estimates mass for a parabolic antenna based on diameter.

$$m_{HGA} := 2.89 \frac{kg}{m^2} \cdot diam_a^{\,2} + 6.11 \frac{kg}{m} diam_a - 2.59kg = 41.75kg$$

The following masses for components of the telecommunication system also come from brown Elements of SC design pg 489.

$m_{TWTAs} := 6.2kg$

$m_{transponders} := 7.6kg$

$m_{LGAs} := 2.1kg \cdot 2 = 4.2kg$

$m_{MGA} := 2.1kg$

A cabling mass is assumed at 5kg, which seems reasonable and conservative based on the Magellan telecommunication system which had a total mass of 101kg and only 7.8kg of cabling.

$m_{cable} := 5kg$

$$m_{telecom} := m_{HGA} + m_{TWTAs} + m_{transponders} + m_{LGAs} + m_{MGA} + m_{cable} = 66.85kg$$

The thermal system is the next system to be sized:

The below values are the values of the spacecraft. The absorbtivity and emmisivity are for aluminum from Charles Brown Elements of Spacecraft Design. The height and diameter are based on fitting the orbiter into the payload fairing.

$\varepsilon_{sc} := 034$ $\alpha_{sc} := 2$ $d_{sc} := 4.35m$

$h_s := 4.888m$

$$r_{sc} := \frac{d_{sc}}{2}$$

$area_{sc} := 2 \cdot \pi \cdot r_{sc}^{\,2} + 2 \cdot \pi \cdot d_{sc} \cdot h_s = 163.3215\, m^2$ $area_{sc} = 4\pi \cdot r_{sphere}^{\,2}$

$r_{sphere} := 5m$

Given

$r_{sc\_theo} := \mathrm{Find}(sphere) = 1.6051\,m$

The thermal analysis is a baseline analysis to obtain the hot and cold temperature extremes of the spacecraft, assuming a sphere with the same surface area as the spacecraft. The spacecraft's area is calculated assuming the spacecraft is a cylinder. The equations for calculating the hot and cold temperatures, the thermal information for the planets, as well as their radii and local solar radiation come from Elements of Spacecraft Design by Brown.

$G_{s\_V} := 2630\frac{W}{m^2}$ $\quad G_{s\_J} := 47\frac{W}{m^2}$ $\quad IR\_emm_V := 153\frac{W}{m^2}$ $\quad IR\_emm_J := 13.6\frac{W}{m^2}$

The terminology used here denotes values at Venus with a V subscript, values at Jupiter with a J subscript, and values of the spacecraft with a SC subscript. G_s indicates the solar radiation constant. Infrared radiation emission from Europa is assumed negligible.

$R_V := 6051.8\,km$ $\quad R_J := 71492\,km$ $\quad a_V := .8$ $\quad a_J := .343$

The heights of the spacecraft from each planet are based on the JEO trajectory data.

$h_{sc\_V} := 20266\,km$ $\quad h_{sc\_J} := 5709000\,km$

Viewing factors are calculated:

$$F_{sc\_V} := .5\left[1 - \left[\frac{\left(h_{sc\_V}^2 + 2\cdot h_{sc\_V}\cdot R_V\right)^{.5}}{h_{sc\_V} + R_V}\right]\right] = 0.0134$$

$$F_{sc\_J} := .5\left[1 - \left[\frac{\left(h_{sc\_J}^2 + 2\cdot h_{sc\_J}\cdot R_J\right)^{.5}}{h_{sc\_J} + R_J}\right]\right] = 2.3238\times 10^{-?}$$

$K_{a\_V} := .9$

$\sigma_{sc} := 5.67\cdot 10^{-8}\frac{W}{m^2 K^4}$

$ASRG_{heat} := 3\cdot .69\cdot 230W = 476.1W$ $\qquad heat_{science} := 50W$

$Q_W := ASRG_{heat} + heat_{science} = 526.1W$ $\qquad$ This represents internally generated heat

$$T_{max} := \left[\frac{\frac{G_{s\_V}\cdot\alpha_{sc}}{4} + IR\_emm_V\cdot\varepsilon_{sc}\cdot F_{sc\_V} + G_{s\_V}\cdot a_V\cdot\alpha_{sc}\cdot K_{a\_V}\cdot F_{sc\_V} + \frac{Q_W}{\pi\cdot\left(2\cdot r_{sc\_theo}\right)^2}}{\sigma_{sc}\cdot\varepsilon_{sc}}\right]^{\frac{1}{4}} = 518.9937\,K$$

$$T_{min} := \left[\frac{q\_emm_J \cdot \varepsilon_{sc} \cdot F_{sc\_J} + \dfrac{Q_W}{\pi \cdot \left(D_{sc\_theo}\right)^2}}{\sigma_{sc} \cdot \varepsilon_{sc}}\right]^{\frac{1}{4}} = 202.198\ K$$

Next, a maximum temperature calculation is performed assuming a thermal shield is designed to reject solar radiation at Venus, causing the spacecraft to experience heat input from only the albedo of Venus, the IR emission of Venus, and the internal heat generation:

$$T_{shield} := \left[\frac{q\_emm_V \cdot \varepsilon_{sc} \cdot F_{sc\_V} + G_{s\_V} \cdot a_V \cdot \alpha_{sc} \cdot K_{a\_V} \cdot F_{sc\_V} + \dfrac{Q_W}{\pi \cdot \left(D_{sc\_theo}\right)^2}}{\sigma_{sc} \cdot \varepsilon_{sc}}\right]^{\frac{1}{4}} = 256.6586\ K$$

Next, the scientific instrumentation thermal requirements are taken into account. Most of the equipment must be hotter than the cold temperature of the spacecraft at Jupiter and Europa.

$$k_{MLI} := 0.0004 \frac{W}{m \cdot K} = 4 \times 10^{-4} \frac{m \cdot kg}{K \cdot s^3}$$

$t_{layer} := 7 \frac{mm}{30} = 0.2333 \cdot mm$ This is from http://tdserver1.fnal.gov/nicol/lhc_irq_cryostat/ch_darve/public/publi/ICEC19_MLI.pdf

$layers := 18$

$t_{MLI} := t_{layer} \cdot layers = 4.2 \cdot mm$ $T_{science} := 270K$

$T_{sc} := T_{min} = 202.198K$

$A_{fac} := 1.2$ This A_{fac} factor adjusts for instruments not being placed right beside one another, which increases the shielding material to shield all of the instruments which need to be within the same temperature environment.

$$A_{sci\_shield} := 2.85 \cdot A_{fac} \cdot m^2 = 3.42 m^2$$

$$q_{Heat} := k_{MLI} \cdot A_{sci\_shield} \cdot \frac{\left(T_{science} - T_{sc}\right)}{t_{MLI}} = 22.084 W$$

This is the heat leak from the scientific instrumentation. Assuming a heater efficiency of 85%, power requirement to replace the heat is:

$Eff_{heater} := 8\!\!:$

$$W_{thermal\_power} := \frac{q_{Heat}}{Eff_{heater}} = \text{:}5.9813W$$

$$\rho_{MLI} := 06\frac{kg}{m^2}$$

$$m_{MLI} := \rho_{MLI} \cdot A_{sci\_shield} \cdot A_{fac} = \text{)}.2462kg$$

$$\rho_{heaters} := 2\frac{kg}{m^2} \qquad A_{heaters} := 1.095m^2$$

Using strip heater mass information from Brown Elements of Spacecraft Design, and assuming the heaters cover the back of the scientific instrumentation to obtain an area for the strip heaters.

$$m_{heaters} := \rho_{heaters} \cdot A_{heaters} \cdot A_{fac} = \text{:}.628kg$$

Due to need to conserve power, see what trade-off exists between heat pipes vs heaters. Heat pipes need more mass in all likelihood but are free on power.

$performance := 5080W \cdot cm$ 1.27cm diam pipe

$distance_{pipe} := 4m$

$$W_{delivered} := \frac{performance}{distance_{pipe}} = 2.7W$$

$$pipes_{needed} := \frac{q_{Heat}}{W_{delivered}} = .7389$$

$$mass_{per\_length} := .33\frac{kg}{m}$$

$length_{real} := 2 \cdot distance_{pipe} = \text{:}m$

$m_{pipes} := mass_{per\_length} \cdot length_{real} = \text{:}.64kg$

An extra .02 kg of mass saves 26 W of power. Clearly heat pipes are a good decision.

Louvers is used for the ASRGs. Again, the source of the mass information for the louvers is Charles Brown, Elements of Spacecraft Design.

$$\rho_{louvers} := 7.3\frac{kg}{m^2}$$

$A_{ASRG\_louvers} := .145m^2 \cdot 3$ Louvers are on the side of each ASRG, so multiply area by 3

$$m_{louvers} := \rho_{louvers} \cdot A_{ASRG\_louvers} \cdot A_{fac} = \text{:}.8106kg$$

$m_{thermostats} := 0.9kg$

$m_{shield} := 50kg$ Mass for the thermal shield at Venus. This shield still needs to be designed, the mass is based on the masses of other heat shields for VEEGAs.

$m_{thermal\_subsystem} := m_{MLI} + m_{pipes} + m_{louvers} + m_{thermostats} + m_{shield} = 56.7868kg$

Seismic Probe Deployment Device is the next system analyzed.

Phase-A study needs be conducted to determine if the use of a thermal knife is better than utilization of pyrotechnic deployment. The team has designed the seismic probes with thermal knife in mind, as non-explosive actuators (NEA) usually have larger masses than pyrotechnic actuators, offering a more conservative mass estimate. The benefits to avoiding pyrotechnic ejection of the seismic probes are that it avoids possible chemical contamination of scientific instruments on the orbiter and it has lower functional shock, lowering its physical impact on the rest of the orbiter. Overall, this lowers failure possibilities. Since this mission is deploying so many seismic probes, it is desirable to avoid the risk of functional shock causing unforeseen problems.

The only disadvantage of the utilization of a thermal knife, other than slightly higher mass, is that this ejection system does not actuate instantaneously like explosive actuators. The thermal knife requires a small amount of time to melt the cord material. This is not a large issue, as the seismic devices are designed to fall within large area "quadrants" of Europa, and the error in release time is overshadowed by the error introduced by the probes entering at a low angle and skipping along the surface until they come to rest.

$diam_{pen} := 136mm$ $max\_length_{pen} := 126mm$ $m_{pen} := 3.57kg$

Based on the size of the seismic probes, a rough size of the spring used to eject them can be calculated, both its free length and compressed length inside the canister.

$l_{free} := 5in$ $l_{compressed} := 2.25in = 57.15mm$

Picking 2.5m/s for the eject velocity, which is enough to send it away from the spacecraft in a timely matter, and entering a guess value for the spring constant:

$k_s := 1\frac{N}{m}$ $vel_{spring} := 2.5\frac{m}{s}$

Given

$$\frac{1}{2}\cdot k_s\cdot \Delta x^2 = \frac{1}{2}m_{pen}\cdot vel_{spring}^2$$

$$k_{needed} := Find(k_s) = 14.0432\cdot\frac{lbf}{in}$$

A real spring with similar properties

http://www.diamondwire.com/compression_form.aspx

DWC-135NO-19

$h_{solid} := 1.786in$ $\qquad$ $diam_{wire} := 135in = .429 \cdot mm$

$d_o := 1.58in = 0.13$ $\qquad$ $\rho_{steel} := 7850\frac{kg}{3}$ $\qquad$ $k_{real} := 14.1\,\frac{lbf}{in}$

$$m_{spring} := h_{solid} \cdot \pi \cdot \left[\left(\frac{d_o}{2}\right)^2 - \left(\frac{d_o - 2 diam_{wire}}{2}\right)^2\right] \cdot \rho_{spr\_steel} = .1408 \cdot kg$$

$force_{spring} := k_{real} \cdot \Delta x = 2.87 \cdot lbf$

The mass of the thermal knife is from instrument reference:

$m_{thermal\_knife} := 250gm$

Next, a canister is designed to hold the spring, seismic probe, and thermal knife.

$t_{canister} := 125in = .17 \cdot mm$ $\qquad$ $\rho_{Al} := 2712\frac{kg}{m^3}$

$l\_extra_{pen} := 1in$ $\qquad$ Some extra room, for growth, or wiring

$$h_{canister} := max\_length_{pen} + compressed + 2.5t_{canister} + l\_extra_{pen} = .523 \cdot in$$

2.5 $t_{canister}$ is for the 3 cross-sectional elements (one at top and bottom, and one that is in the center, sitting atop the spring) where the cylinder is not hollow. 2 of them (the one at bottom and the middle) are the same thickness as the canister, and must be able to tolerate the 53lb spring force shown in the calculations above. The top one is just a cover, and has no real force on it. It isnt monolithic with the canister, and is only weakly attached. It is designed to be pushed out of the way by the seismic probe, and will probably not even be half of the thickness of the rest of the canister. It may not be of the same material either. But for this analysis, in order to be conservative, it is assumed to be metal, and is half the thickness of the rest of the canister.

$clearance_{pen} := 4mm$

$$inner\_diam_{canister} := diam_{pen} + clearance_{pen} = .511 \cdot in$$

$$m_{canister\_real} := h_{canister} \cdot \pi \cdot \left[\left(\frac{inner\_diam_{canister} + 2 \cdot t_{canister}}{2}\right)^2 - \left(\frac{inner\_diam_{canister}}{2}\right)^2\right] \rho_{Al} + 2.5 \cdot t_{canister} \cdot \pi \cdot \left(\frac{inner\_diam_{canister}}{2}\right)^2 \cdot \rho_{Al} = 1.1698\,kg$$

$$m_{seismic\_real} := m_{seismic\_devices} + 29 \cdot m_{thermal\_knife} + m_{spring} + m_{canister\_real} = 48.8465\,kg$$

A brief analysis to obtain the diameter of the impactor follows, for the sake of knowing how large it is maximum, in order to leave space for it in the spacecraft. The density is aluminum in order to get the maximum space. The ball will probably be steal, having an 8 inch radius.

$$m_{impactor} := 35kg$$

$$\rho_{Al} = 2.712\times 10^{3}\ \frac{kg}{m^{3}}$$

$$vol_{impactor} := \frac{m_{impactor}}{\rho_{Al}} = 0.0129\ m^{3}$$

$$r_{impactor} := 0.5m$$

Given

$$vol_{impactor} = \frac{4}{3}\pi \cdot r_{impactor}^{3}$$

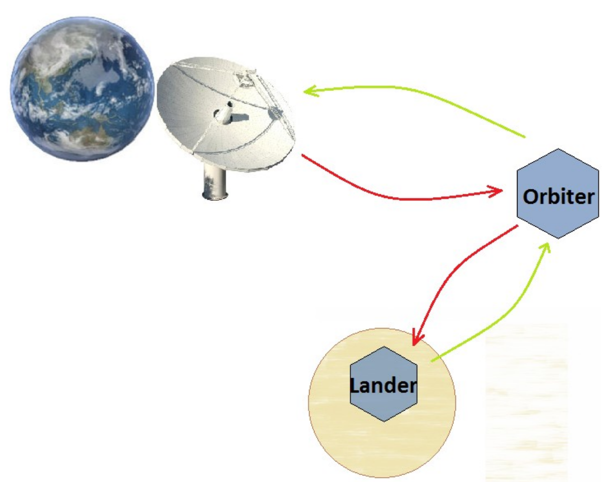
$$r_{impactor\_real} := Find(r_{impactor}) = 0.1455m$$

$$d_{impactor\_real} := 2r_{impactor\_real} = 0.291m$$

$$d_{impactor\_real} = 1.457\ in$$

J.10.2 ESTACA's Telecommunication's Calculations

In this part, we will report all our studies and explanations that allowed us to design the telecommunication system with the necessary equipments.



For the Europa mission, telecommunication works like on the scheme above. In one hand, the information about Europa has to be transmitted to the earth. The green arrows stand for this first way of telecommunication. In the other hand, the Earth has to be able to give Earth orders. The red arrows stand for this first way of telecommunication. In our project we will design equipments of the telecommunication system of the Lander. This system has to be able to communicate information with the Orbiter.

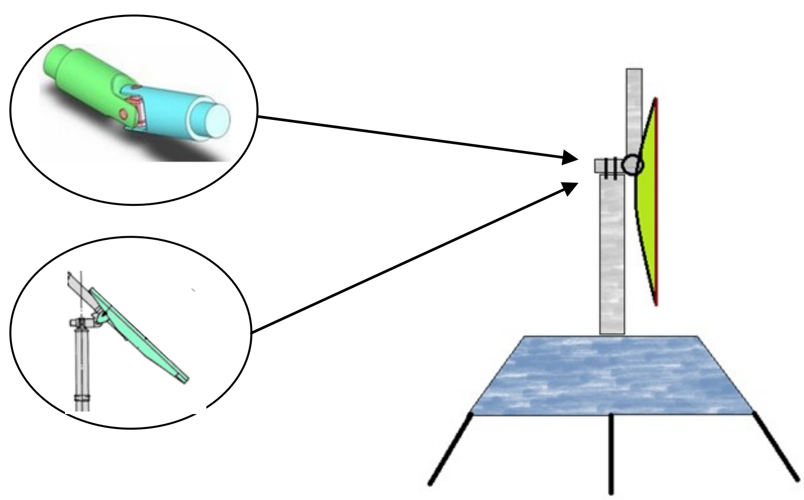
J.10.2.1 Frequency choice

We decided to work with the S band which range from 2GHz to 4GHz. Therefore, the data transmitted from the Lander to the Orbiter are spread from 2GHz to 3GHz. The band S is convenient because its reception by the other stations is simple. Moreover, its emission can be done in every direction.

J.10.2.3 Kind of antenna choice

Hertzian beams and satellites communication need narrow beam in order to communicate the energy in the direction of a single receptor. Thus the necessary power is lower. Likewise the reception antenna receives only beams came from the space area of the transmitter. To obtain a narrow beam (so an antenna with a high directivity), the wave length must be little (so a high antenna). Consequently, the choice of the antenna depends on a compromise.

That’s why; firstly we thought to use an antenna with parabolic reflector, thus its increase the gain of the antenna. Also, we would to mount the antenna on a swivel, so it can talk to the Orbiter even when it isn't directly overhead. For the swivel we thought of a system which can rotate in azimuth direction and latitude direction like a gimbal. We can see a scheme of the Lander with examples of rotating system below.



Secondly, we change our point of view. We think that an **omnidirectional antenna** instead of the parabolic reflector is more convenient. Indeed this kind of antenna doesn't need to be mounted on a swivel system. So it's less heavy and less bulky. By using an omnidirectional antenna, we are sure cover the entire area of the Orbiter during the twelve hours. Moreover it enables to reduce the mass of the Lander.

In order to achieve its mission, the Lander is equipped of an **UHF antenna** (Ultra High Frequencies). The frequency is contained between 300MHz and 3000MHz. Thus the main frequency to communicate is 2.5 GHz (compromise between the S band and the UHF antenna).

We decide to use a **quarter wave monopole antenna**. It is a single element antenna fed at one end, that behaves as a dipole antenna. It is formed by a conductor λ/4 in length. It is fed in the lower end, which is near a conductive surface which works as a reflector. The current in the reflected image has the same direction and phase as the current in the real antenna. The quarter-wave conductor and its image together form a half-wave dipole that radiates only in the upper half of space. In this upper side of space the emitted field has the same amplitude of the field radiated by a half-wave dipole fed with the same current. Therefore, the total emitted power is one-half the emitted power of a half-wave dipole fed with the same current. As the current is the same, the radiation resistance (real part of series impedance) is one-half of the series impedance of a half-wave dipole. As the reactive part is also divided by 2, the impedance of a quarter wave antenna is Ohms.

J.10.2.3 Environmental conditions

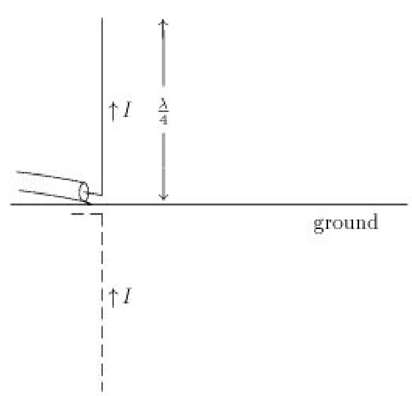
This antenna will work with difficult environmental conditions with probably important thermodynamic constraints. So we need to use a material with a low dilatation coefficient in order to prevent an irreversible damage of the material from happening.

Moreover during the launch, the Lander is subjected to acoustic noise and vibrations. It is possible that resonance frequencies of the antenna and the satellite are nearby. So it is necessary that the space equipments undergo analyses and very complete mechanic tests. This analyze enable to determine the necessary structure recess. The antenna results from a compromise between the robustness and the mass. That’s why; we decide to use Aluminum for the antenna. Aluminum is remarkable for the metal's low density and for its ability to resist corrosion due to the phenomenon of passivation. It's also a very good conductor. Structural components made from aluminum and its alloys are vital to the aerospace industry.

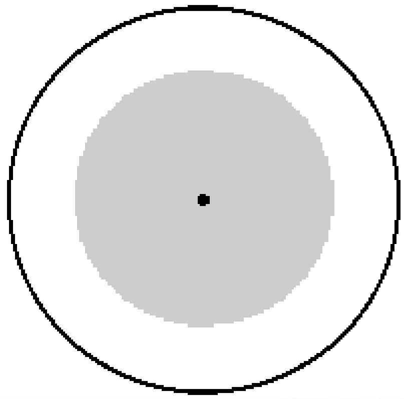
J.10.2.4 Sizing of the antenna

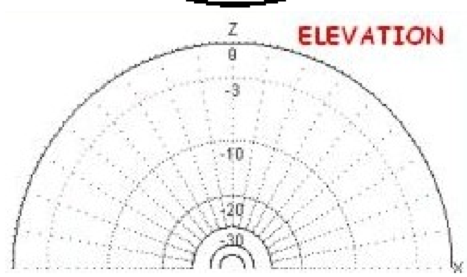
The Orbiter has an elliptical trajectory around Europa with a periapsis of 50 km and an apoapsis of 100 km.
We work with a frequency of = 2.5 GHz.
The speed of sound in the weightlessness is = 3e8 m/s.
The wave length is .

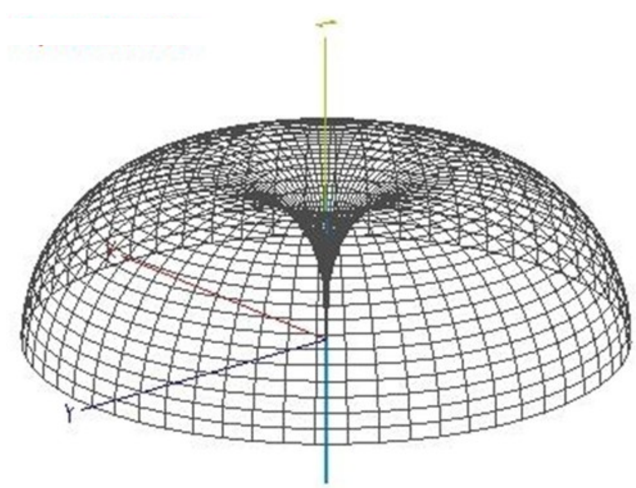
The length of the antenna in order to obtain the maximal radiated power is *λ/4* = 3 cm.
The voluminous mass of the aluminum is ρ = 2700kg/m^3 and we estimate the mass of the antenna some of milligrams.

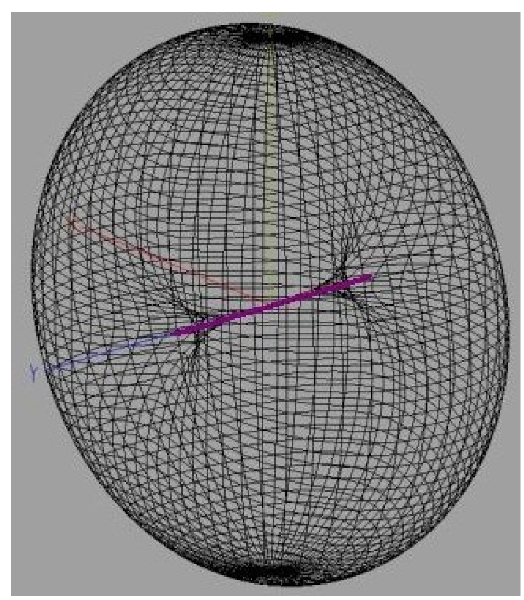


The radiation pattern is a half sphere. An omnidirectional antenna is an antenna that radiate uniformly in every directions of a horizontal plane. The power radiated decrease with the elevation angle or below the plane. The radiation diagram has the following shape:









Radiation diagram for the vertical in 3D

Radiation diagram for all directions in 3D

The complex impedance the antenna is 36+i21 with a resistance of *Ra* = 36 Ohms and an electrical reactance of 21 Ohms.

The gain *Ga* and the length *L* of the antenna is linked by the relation with *Z* the impedance in the weightlessness so a gain of *Ga* = 2.06 dB.

The output of this kind of antenna is excellent, about 90 %.

The bandwidth is equal to 10 % of the frequency, so 250 MHz.

The radiation power is calculate with this relation:

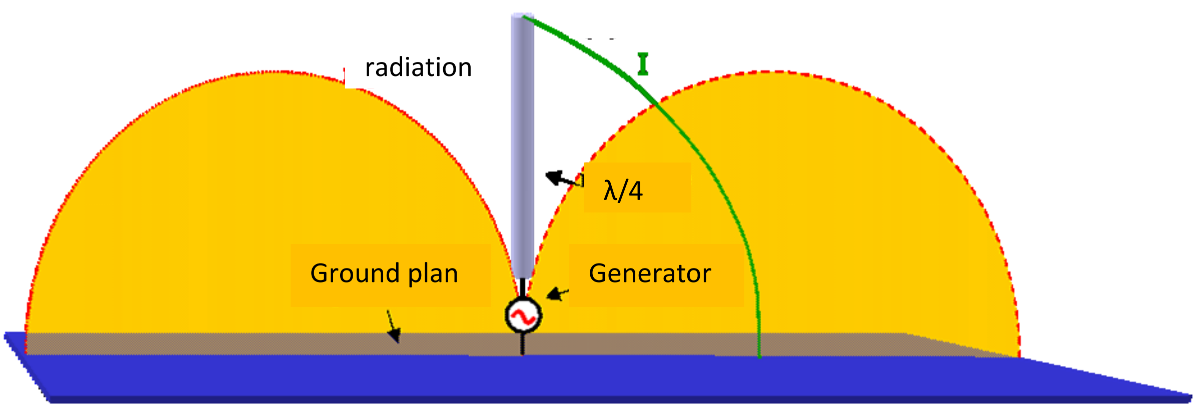
With *Pr*: power radiation of the reception antenna
Pe: power radiation of the emission antenna
Gr: gain of the reception antenna
Ge: gain of the emission antenna

We consider that the power of the antenna of the Orbiter is 20 W (this value is chosen with the informations in american team documents).

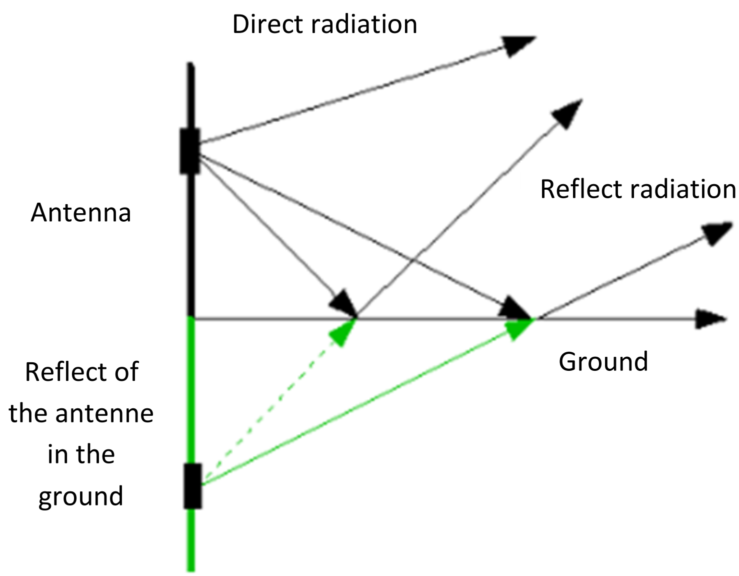
Thus we can calculate the power of the antenna of Lander: P = 2 567.9 W.

J.10.2.5 Ground plane

Our antenna had radiated strand which length is λ/4. So we need the natural ground (the surface of the ground must be infinitely great in front of the wave length; the ground is infinite, homogeneous and perfectly conductor) or an artificial ground in order to create a reaction that assimilate our antenna to a λ/2 wave. Conductors are disposed in radial way on the base of the antenna. Conductors perform like a mass plan. This king of antenna has very low impedance at its feet that is the feeding source (36Ω) and a very high one at its head.



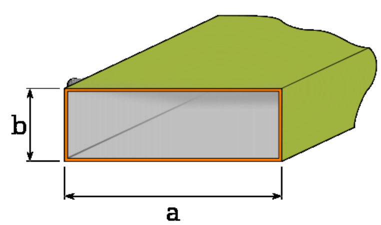
The supplying has to be done by an adaptation intermediate system. Ground properties are extremely important for the antenna performances. Indeed, the reflected rays have to be reflected and not absorbed. This is the reason why an artificial ground plane is most of the time more profitable. However, mechanically, it is more difficult to install.



For our project we will use the upper surface of the Lander as the mass plane. Indeed this surface is totally plane. It is a square that measure one meter long, so it is really longer than the height of the antenna. The material is aluminium. So it is a good conductor.

J.10.2.6 Waveguide

For wave in the order of centimeter, we have to use waveguide. A waveguide is a hollow metal pipe used to carry radio waves. It is used as connecting microwave transmitters and receivers to their antennas. It consists of a hollow metallic conductor.



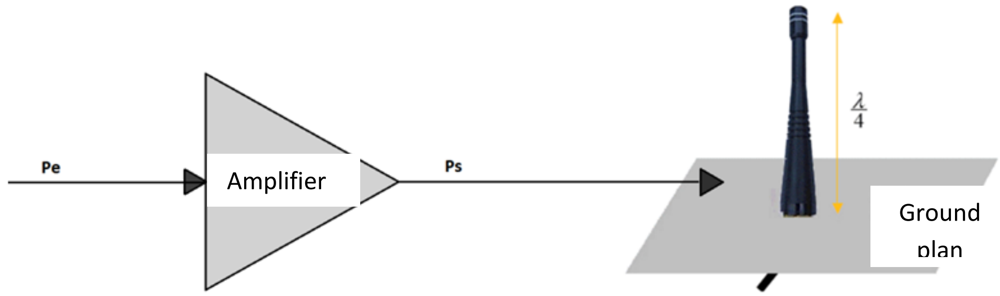
The dimensions of the hollow metallic waveguide determine which wavelengths it can support, and in which modes. Frequencies below the guide's cutoff frequency will not propagate.

Waveguide name			**Frequency Band Name**	**Recommended Frequency Band of operation (GHz)**	**Cutoff frequency of lowest order mode (GHz)**	**Cutoff frequency of next mode (GHz)**	**Inner dimensions of waveguide opening (inch)**
EIA	**RCSC**	**IEC**http://en.wikipedia.org/wiki/International_Electrotechnical_Commission					
WR 340	WG 9A	R26	S band	2.20 — 3.30	1.736	3.471	3.400 × 1.700

Thus we use the waveguide IEC R26. Its size is 8.636*4.318 cm.

J.10.2.7 Amplifier

The amplifier is linked to the antenna by the waveguide.
The signal has to be amplified in order to be receiving by the antenna of the orbiter. Indeed, the power at the end of the antenna is too low. The losses are due to the distance between the orbiter and the Lander, this distance can be up to 100km. So we have to work out the gain of the amplifier. This gain is chosen functions of the entry power of the antenna and the distance to go to the orbiter.



Considering these losses, we need to provide 2567.9W to cover the distance between the Lander and the orbiter, and 40,3W for the equipments supplying.
Besides our antenna without amplification deliver 118W.

So we need an amplification factor (A_p) such as:

$$A_p = \quad = \underline{\qquad} = 22.1$$

However there are some losses in current lines and guide. Also, in case of adding of equipment or in case of raising of the orbiter orbit, we want to gross-up the amplification factor to 23.

So we choose the power ratio following: .

We work out the gain of the amplifier: $X_{db} = 10 \times \log_{10} \quad = 10 \times \log_{10} \quad = 13.6$ dB

With X_{db}: number of decibel
: Power at the end
: Entry power
A: amplification factor

So the **gain of the amplifier** is **13.6 dB**.

We notice that with simplifications, we can assimilate the HF amplifier to an operational amplifier working as linear load. By the way, we can calculate the amplification factor for the voltage and for the power like that:

Amplification factor for the voltage: $A_v = \quad = 1+$

With: $X_{db} = 20 \times \log_{10}$

Knowing the voltage ratio, it is easy to estimate the value of the resistances.

Amplification factor for the power: **A_p =** = = (1+)
= **23**

J.10.2.8 Battery

We decide to use a rechargeable lithium battery: the VES 100, because its weight is low (810 g) and it provides 118 W/kg that is a necessary power for the antenna.

You can find the characteristics of its battery here:

Cell electrical characteristics	
Nominal voltage	3.6 V
Nominal capacity at C/1.5 rate at 4.1 V/3 V & 20°C	27 Ah
Maximum discharge current at 25°C	100 A (Continuous ~2 s pulse)
Specific energy (minimum guaranteed)	118 Wh/Kg
Energy density	230 Wh/l
Cell mechanical characteristics	
Diameter (max)	53 mm
Height (max)	185 mm
Mass (max)	0.81 kg
Cell operating conditions	
Lower voltage limit for discharge	Continuous (0°C to +45°C) 2.57 V
Charging method	Constant current/constant voltage (CCCV)
Charging voltage (max)	4.1 V
Recommended continuous charge current	GEO/MEO C/10 LEO (20 % DOD) C/5
Operating temperature	Charge +10°C to +35°C Discharge 0°C to +40°C
Storage and transportation temperature	-40°C to +65°C

NB: VES 100 are sold only in modules or batteries.

J.10.2.9 Interface Antenna/structure

The environment close to an antenna is not always released. The metal objects located at a distance about the wavelength is able to produce an effect of shade in the direction considered, if their dimension is about the wavelength or more.

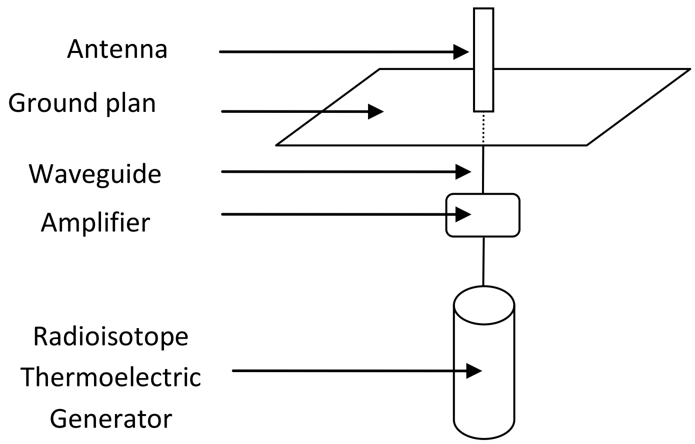
Disturbances of the operation of the antenna is able to appear by the presence of conducting bodies, in the immediate environment of the antenna. In general, the frequency of resonance of an antenna depends on the capacity of the antenna compared to its environment. Thus, if a conducting body is close to the end of the antenna, we will observe a reduction of the frequency of resonance. If its body has large dimensions and connected on the ground or the mass, we has a decrease of the resistance of radiation, because the lines of electric field will join the mass by a short way, instead of spreading itself in space. Moreover, the frequency of resonance of an antenna depends on the inductance of the parts. That's why, for a quarter wave antenna, the conductors near the top of the antenna will not have the same effect that if they are close to the base of the antenna.

In our study, we chose to reduce the obstruction of the antenna by maintaining it relatively close to a metal plan (ground plane). So we have to take account of these problems.

In order to supplement this preliminary draft, we have to simulate these phenomena of antenna on structures. Unfortunately, due to time constraints, this study could not be doing.

J.10.2.10 Conclusion

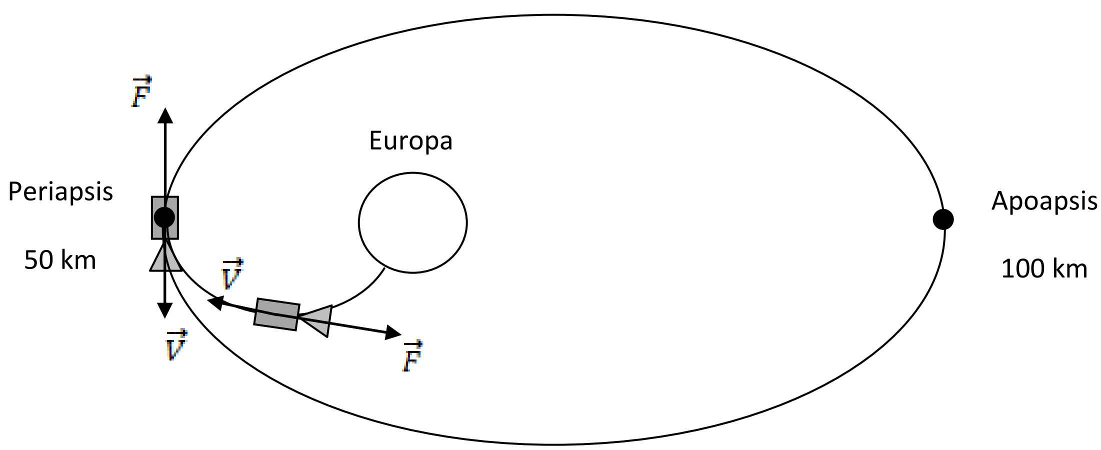
Eventually, for this part the equation is to transmit information to the orbiter considering the losses and with the necessary power. In order to achieve this mission, we use a quarter wave monopole antenna (UHF), an electrical amplifier of 13.6 dB, a rechargeable lithium battery delivering 118 W and a waveguide between the amplifier and the antenna.



J.10.3 ESTACA's Trajectory and Propulsion Calculations

J.10.3.1 Landing trajectory

In order to slow down Lander in an optimal way, this one must be always directed so as the thrust vector is aligned and in the opposite direction than the speed vector. The loss of speed of the Lander has the effect of change the trajectory and thus, a change of orientation of the vector speed. The nozzles used for the orientation must constantly correct the attitude of this one. Furthermore, if the place of landing is not contained in the initial plan of the orbit, it is necessary to make a first operation to change the slope.
To minimize the speed of Lander acquired during the free-fall, the beginning of the operation has to be made at periapsis of the orbit, which is at the distance of 50 km of Europa's surface (cf. diagram below).



The expression of the conservation of energy allows deducting the speed on an elliptic orbit.

With

Where 2a is the length of the major axis of the elliptic orbit, R = 1561 km the equatorial radius of Europa, z the height of the Lander, and respectively the altitude of the periapsis and the apoapsis.
We can thus calculate the speed in the periapsis (speed purely tangential), note down :

By spotting the landed by a point M, in a system of polar coordinate of origin O Europa's center, we can write the position, the speed and the acceleration in the following way:

The forces acting on the Lander are the weight and the thrust :

Where the total thrust of the engine.
And
The Newton's second law gives the following relation :

We can deduct the equations of the movement of the Lander :

By resolving these equations we can determine the trajectory and the thrust to adopt (by playing on the module of).

J.10.3.2 Simplifying of the problem
The problem can be simplified by considering two operations:

The first one to cancel the orbit speed (by considering this one instantaneous so that the orientation and the altitude of the Lander remain unchanged).

The second to cancel the speed acquired during the free-fall

We shall consider the Lander as an isolated system subjected only to the gravity of Europa which we will consider uniform. The aerodynamic forces due to the atmosphere of Europa can be neglected because of the low pressure atmospheric pressure which is Pa at the surface. For a vertical fall, without thrust, from the periapsis of the initial orbit, the equation of the movement can be written :

We can then integrate two times this equation with null initial velocity and an initial altitude equals to the periapsis altitude :

Thus we can calculate the duration of a fall to arrive at a given altitude and the velocity which would have the Lander at this altitude and this time. Before impacting the ground at z = 0, the time of fall and the final velocity of the Lander would be :

For : ⇒

The total velocity which it is necessary to take into account to cancel the speed of the Lander is the sum of the velocity on orbit with the velocity acquired during the free-fall.

For an operation made from the periapsis we obtain:

The total mass of the Lander hold back for the mission is:

Thus, we can deduct the propellants mass used to slow down the Lander from a velocity thanks to Tsiolkovski formula:

This can be written after simplification:

By using the propellant MMH (Monométhylhydrazine) for the reducer and some Dinitrogen tetroxide for the oxidizer with an ISP of 314, we obtain the propellant mass following :

The mass of scientific instruments used for the mission is . This result in a dry mass of structure calculated below:

To arrive with a velocity close to zero on Europa's surface, the Lander thrust must be sufficient enough so that the duration for cancelling its initial orbit velocity is lower or equal to the time take to arrive at the surface.

Thus, to determine the order of thrust used to slow down the Lander, we shall look for the thrust which will allow canceling the speed of the Lander for duration equals to the time of free fall. This means overestimating the necessary thrust because during the real operation, the trajectory is partially elliptic and the weight contribution to the acceleration is lower than its contribution for a vertical fall.

The deceleration due to the thrust of the Lander can be written thanks to the Newton’s second law in the following way:

Where q is the mass flow rate of the engine, and are always on the same axis and of opposite direction. We will consider thrust constant and that the mass of the Lander decreases because of the ejection of gases.

The projection of the previous equation onto the axis carrying and can be written :

We can then integrate this relation by taking the initial velocity at time t = 0 :

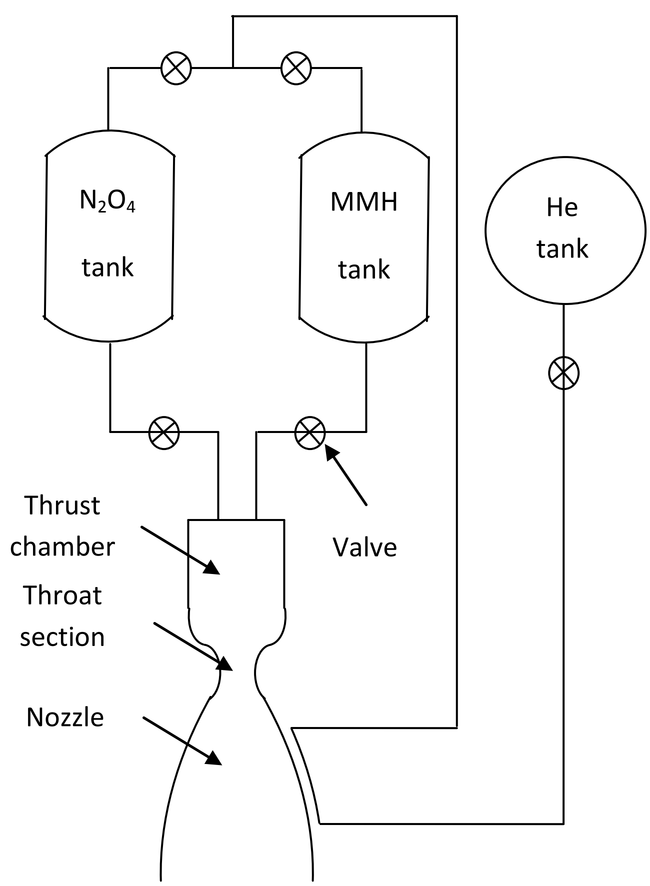
We obtain:

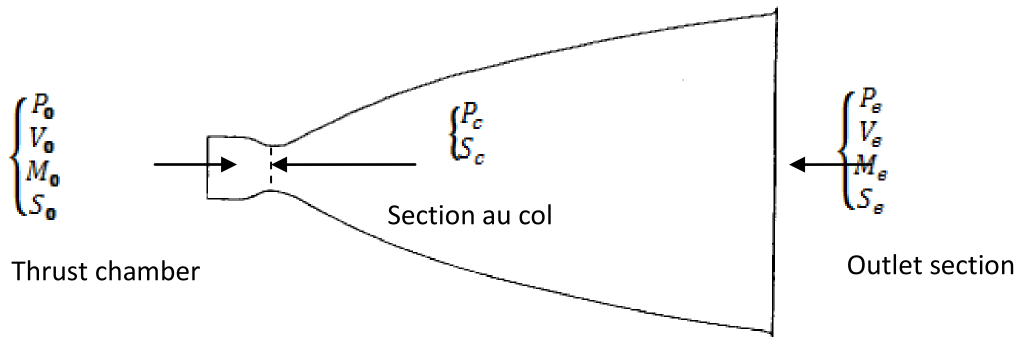
Thus, with a mass flow rate , the thrust that we must use in order of having a null velocity at time is :

J.10.3.3 Sizing of the propellant system

The architecture chooses is an engine feed by two propellant tanks pressurized by a tank of helium. The mass flow rates are command by solenoid valve (cf. plan below). We increase the pressure of helium by increasing its temperature in the contact with the nozzle before injects it in the propellant tanks.

During the propelled operations, two solenoid valves are opened in the MMH N2O4 ratio and the others permitted to adjust the mass flow rate of helium to keep the pressure constant in the propellant tanks.





The table below give all the principle size of the engine and the characteristic of flow calculate for the thrust, ISP and propellant previously established.

Propellant table			Engine table		
	1,24	-	P_e/P_0	0,0013	-
R	8,3145	J/mol K	P_0	1000000	Pa
M_{gas}	21	g/mol	P_e	1334,0037	-
r	395,9272	J/kg K	S_e/S_c	50,0000	-
T_0	3240	K	η_{cf}	0,975	-
ISP	314	s	Cf_{th}	1,7940	-
$M_{propellant}$	57,1	kg/s	Cf_r	1,7491	-
Rm	1,73	-	η_{c*}	0,99	-
M_{ox}	36,1842	kg/s	F	645	N
M_{fuel}	20,9158	kg/s	q	0,2094	kg/s
ρ_{ox}	1400	kg/m^3	C^*_r	1778,8473	m/s
ρ_{fuel}	874	kg/m^3	S_c	0,0004	m^2
g_{earth}	9,81	m/s^{-2}	S_e	0,0186	m^2
			D_c	0,0218	m
			D_e	0,1540	m
			$V_{chamber}$	100	m/s
			V_{sound}	1261,220548	m/s
			M_0	0,079288274	-
			S_0/S_c	7,458221806	-
			S_0	0,002778018	m^2
			D_0	0,059473372	m

J.10.3.3 Sizing of ergols tanks

The size of the lander must not be in excess of a one meter length cube. That's why the tanks shape can't be spherical because it would be larger than 0.5 m of diameter per tank. Thus, we decided to use cylinder-shaped tanks, for the propellant, with spherical cap shape for bottom and ball-shaped tanks for the helium (as you can see on the diagram of the previous part).
We will consider a margin of 5 % on the tanks volume to cover the dead volume, the gaseous sky, the losses thermodynamics, the inexhaustible as well as the volume occupied by equipments.

There is one diameter of optimal bottom which allows limiting its mass. The brief delay of this project doesn't make possible the calculus of this optimum. That's why, we will take the ratio from the development of ARIANE 5 EPC:

The volume of the spherical cap bottom is given by:

With h the height of the cap. We can write a relation between the diameter and the radius of the spherical cap:

By fixing the value of , we can deduct from it, via the ratio and calculate h and the volume . Then we can deduct the height of cylinder :

The total height of the propellant tank is:

The size of propellant tank based on is given by the table below :

Propellant tanks table		
V_{ox}	0,0258	m^3
V_{tank}	0,0239	m^3
$V_{tank\ ox}$	0,0271	m^3
$V_{tank\ fuel}$	0,0251	m^3
$D_{cyl\ ox}$	0,3	m
$R_{cap\ ox}$	0,1667	m
h_{ox}	0,0940	m
$V_{cap\ ox}$	0,0038	m
$H_{cyl\ ox}$	0,2776	m
$H_{tank\ ox}$	0,4656	m
$D_{cyl\ fuel}$	0,3	m
$R_{cap\ fuel}$	0,1667	m
h_{fuel}	0,0940	m
$V_{cap\ fuel}$	0,0038	m
$H_{cyl\ fuel}$	0,2492	m
$H_{tank\ fuel}$	0,4372	m

To establish the tank pressure, we must know pressure drop of our system (loss in the pipes, the solenoid valves, the injectors…). We will not calculate it, however we will take a value similar to the engine LEROS 1 of Snecma, which presents characteristics close to our engine. We will neglect the pressure due to the acceleration, because of the little height of propellant. The tank pressure which we will keep is :

J.10.3.4 Sizing of helium tank

The Helium mass necessary to pressurize our tanks can be calculated with the formula which follows, usually used in preliminary studies:

With the volume to be pressurized, the pressure of pressurization and the final pressure in the tanks.

The tanks used to contain the liquid helium are too heavy, that's why we will base ourselves on the pressurization system of the H10 stage of ARIANE 4 designed by AIR LIQUIDE. Thus, we will use some cold helium at , under a pressure of 220 bars. We will take (order of magnitude given by the literature) because the calculus of the pressure evolution is too long for our study. After having calculated the mass of helium thanks to the previous formula, we can determine the volume of helium. The data are recapitulated in the following table :

Helium table		
	1,65	**-**
ρ_{He}	**0,1785**	**kg/m^3**
r	**2077**	**J/kg/K**
P	**1500000**	**Pa**
P_g	**3000000**	**Pa**
T_0	**100**	**K**
P_0	**22000000**	**Pa**
m_0	**0,3467**	**kg**
V_{He}	**0,003273203**	**m^3**
$D_{tank\ He}$	**0,184214906**	**m**