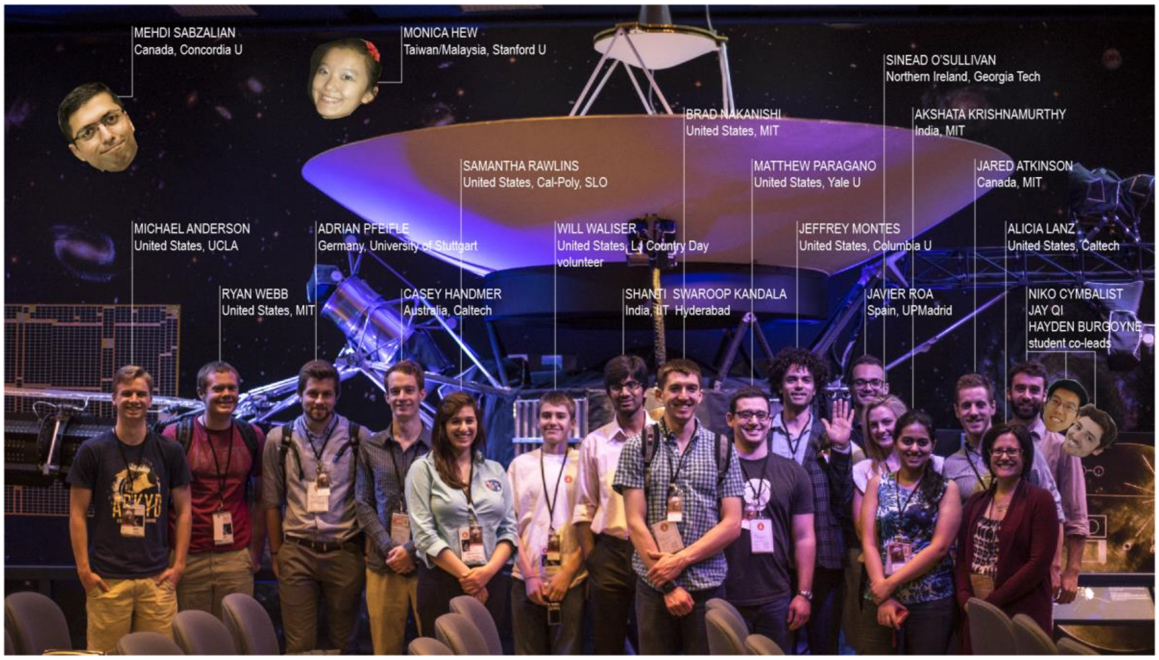
Orbiting Asteroid for Strategic In-situ Supplies

A Crewed Mission for Deep Space Asteroid Exploration

March 27, 2015



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Executive Summary

Orbiting Asteroid for Strategic In-situ Supplies (OASIS) is an innovative, frontier opening crewed mission to a robotically captured asteroid. In 2024, three astronauts will fly on SLS to a Distant Retrograde Lunar Orbit. Together with the Multi-Purpose Docking Vehicle and science payload they will dock with the Asteroid Redirect Vehicle and a deep space Labitat sent via a separate Falcon Heavy launch and ballistic trajectory. Following on from the first Asteroid Redirect Crewed Mission (ARCM), over 25 days in lunar orbit they will execute a series of space-rock walks to characterize and extract asteroid resources. OASIS is a versatile test bed of Flexible Path crewed space exploration technology as well as a generational advance in asteroid mining and asteroid planetary defense.

List of Acronyms

NEA - Near Earth Asteroid
LEO - Low Earth Orbit
MEO - Medium Earth Orbit
GTO - Geostationary Transfer Orbit
TLI - Trans-Lunar Injection

DRO - Distant Retrograde Orbit
FRT - Free Return Trajectory
PLF - Payload Fairing
GNC - Guidance, Navigation and Control
ECLSS - Environmental Control and Life Support System
ARV - Asteroid Redirect Vehicle
ARM - Asteroid Redirect Mission
ARCM - Asteroid Redirect Crewed Mission
MPDM - Multi-Purpose Docking Module
MPCV - Multi-Purpose Crew Vehicle (Orion)
ISRU - In Situ Resource Utilization
OASIS - Orbiting Asteroid for Strategic In-situ Supplies
SLS - Space Launch System
EUS - Exploration Upper Stage
D4H - Delta IV Heavy
FH - Falcon Heavy
SAFER - Simplified Aid For EVA Rescue
EMU - Extravehicular Mobility Unit
EVA - Extravehicular Activity

Table of Contents

Introduction - Why a Near Earth Asteroid mission?

Problem statement

Assume that ARM successfully returns 500 metric tons of C-type asteroid material in 2024 to a distant retrograde lunar orbit with mean radius of 61,500km.

In 5 days, each team is challenged to design a mission to land humans on an asteroid brought back to lunar orbit, extract the asteroid's resources and demonstrate their use. The launch date of the mission may be no later than January 1st, 2028.

How can this human mission be used to transform this rock in space into an oasis for future explorers? Give this mission context in the broader framework of exploration by defining how the resources can be used, but focus on designing the process of unlocking this asteroid's resources and making them available for future use. Which resources should be extracted? How should they be used? How will they be stored? How will they be utilized in future missions?

Each team's final proposed mission will be judged through a **written proposal**, a **mission fact sheet**, and an **oral presentation**. Teams will be judged primarily on the following:

- The technical merit and level of detail of their proposed mission design.
- The creativity of their proposed mission objectives and design.
- Their ability to communicate the proposed design clearly and coherently in all three final deliverables.

Suggested team organization can be found in Section 6, and suggested project milestones are in Section 7.

A resources packet and copies of some relevant references will be provided electronically to both teams. This packet will be contain reviews of subsystem topics as well as some additional resources to help guide your design process.

Inspiration and Context

Asteroids represent some of the most primitive bodies of the solar system. There are hundreds of thousands of asteroids leftover from the formation of our solar system some 4.6 billion years ago (GSFC 1996). The number of known asteroids has skyrocketed in the last two decades, and their numbers only continue to increase (Chamberlin 2015). In recent years, asteroids have been viewed through a number of lenses; to some, their potentially destructive power is a large concern for planetary defense, and occurrences like the Tunguska event in 1908 and Chelyabinsk meteor in 2013 serve as poignant reminders of this (Phillips 2008, Sample 2013). The ability to redirect asteroids on a collision course with target Earth would be a major milestone for the space programs of the world. To many others, asteroids represent an unprecedented economic opportunity of seemingly infinite resources. The one thing that is clear is that asteroids will prove to be an important part of our understanding and exploration of the solar system.

The asteroid redirect mission aims to improve our understanding of these ancient formations. At this point in time, there are a number of missions that have visited asteroids, and the Hayabusa mission launched in 2003 was the first to return a sample from the surface of an asteroid (JAXA 2008). Hayabusa-2 is in transit for yet another sample return, and the OSIRIS-REx mission is in active development to return a sample from the surface of Bennu (Clark 2013). These missions are crucial steps to exploring the asteroids, but there still remain hundreds of thousands of more and untold numbers of undiscovered asteroids; we have only quite literally scratched the surface. The asteroid redirect mission will answer fundamental questions about our cosmic neighborhood.

Developing a concept for a medium duration crewed exploration of a captured near Earth asteroid at the Keck Institute for Space Studies was apt given the concept's birth there in 2014. With steadily diminishing launch costs and exploding robotic capability it is apparent that the industrial exploitation of space resources is only a matter of time. Likewise, a complete reckoning of asteroid impact threats to our civilization has greatly benefited from automated discovery systems and telescopes during the last decade. Between now and the future is a technological gap that can only be spanned by daring, innovative missions to first retrieve an asteroid from deep space and then thoroughly explore with human eyes and hands. NASA's recognized history of science and technology development place it at the forefront of this opening frontier and it is within this context that this first mission is planned, to meet an asteroid and extract its resources.

OASIS and the Flexible Path

In 2009, the Review of U.S. Human Spaceflight Committee outlined a number of different paths for the future of human spaceflight in the United States (Augustine 2009). One such proposed plan was the so-called "flexible path," in which destinations like the Lagrange points or near-Earth objects could be explored for immense scientific return on the development pathway to Mars (Foust 2010). The asteroid redirect mission answers this call, and the OASIS mission in particular combines significant scientific return with immense potential for growth as we learn how to live and operate in deep space. The OASIS mission addresses the fundamental goals outlined in the 2014 NASA Strategic Plan to "expand the frontiers of knowledge, capability, and opportunity in space" and expand the human presence into the solar system through international collaboration (NASA 2014).

In-Situ Resource Utilization (ISRU) of extraterrestrial matter is the antidote to Tsiolkovsky's rocket equation and the key to breaking the tyranny of Earth's gravitational well. Many planned deep space missions would greatly benefit from the option to refuel an injection stage in a Lunar Distant Retrograde Orbit. The development of this game changing capability is a key component of NASA's flexible path for space exploration. Technologies agnostic about destination can be leveraged under a much wider variety of future missions, some of which cannot be dreamt of under any alternative paradigm. Whether the future of crewed space exploration leads to the Moon, Mars, or elsewhere, this Orbiting Asteroid for Strategic In-situ Supplies (OASIS) mission blazes the trail of possibility. With OASIS, we as a species take yet another step out into the solar system.

Mission Overview

Mission Statement

The mission will demonstrate the capability for humans to live and work autonomously in deep space through the creation of a long-term platform and in-situ resource utilization of an asteroid.

Assumptions

- Successful delivery of asteroid to DRO
- There exists a docking mechanism to attach to ARM
- Platform can utilize ARM solar power generation
- Astronauts have access to all points on asteroid surface
- Communication between astronauts during EVA is possible throughout all mission phases

Mission Objectives

Table 1: Primary science objectives

Primary Science Objectives
1. Characterize the internal structure and composition of asteroid
2. Characterize space environment around asteroid
3. Extract, process, and demonstrate use of resources
4. Contribute pioneering human health and behavioral data in deep space
5. Demonstrate human autonomous decision making in a deep space environment

Table 2: Secondary science objectives

Secondary Science Objectives
1. Advance our understanding of the origin and evolution of the solar system
2. Improve current asteroid classification scheme
3. Demonstrate planetary defense capability via manipulation

Table 3: Engineering objectives

Engineering Objectives
1. Reach Lunar Distant Retrograde Orbit with manned mission. Return crew successfully from a high energy, high altitude orbit using Orion capsule.
2. Demonstrate the ability for a crew of 3 to survive in a deep space environment for an amount of time greater than the support lifetime of the Orion capsule.
3. Demonstrate the ability for a crew to autonomously plan a subset of their tasks as a safe proving ground for crew operations in deep space environments (autonomy from mission control for long light-time missions such as a mission to Mars)
4. Record physiological and psychological data from crew and compare to ISS data (compare effects of LEO to LDRO on human body and psyche).
5. Demonstrate the capability of tool /spacesuit designers to create a versatile and ergonomic tool set for exploring new terrain with a view to design tools for people to interact with a new environment, applicable to a future Moon/Mars mission.

System Requirements

<u>Mission Success Verification Metrics</u>

Table 4: Scientific success criteria - minimum

Scientific Success Criteria – Minimum
1. Drill and sample subsurface asteroid material to a minimum depth of 1 m
2. Return samples to Earth with proper storage
3. Excavate 1 kg of material
4. Record radiation levels and cosmic ray exposure once per day in the human living quarters for the duration of the docked portion of the mission
5. Characterize the space environment for a single asteroid location
6. Process, utilize, and generate a specified amount of one specified resource
7. Successful completion of one experiment selected by astronaut from a predetermined list

8. Monitor physiological data once per day for the duration of the docked portion of the mission

Table 5: Scientific success criteria - full

Scientific Success Criteria – Full
1. Deployment and successful acquisition of 3D seismic tomographic array
2. Successful retaining of seismic array as long-term microseismic monitoring system
3. Drill and sample subsurface asteroid material to a minimum depth of 5 m
4. Return samples to Earth with appropriate storage
5. Excavate 3 kg of material
6. Record radiation levels and cosmic ray exposure once per day in the human living quarters for the duration of the docked portion of the mission
7. Characterize the space environment for six asteroid locations
8. Process utilize and generate a specified amount of all specified resources
9. Successful completion of one independent exercise for the duration of one hour
10. Monitor physiological data once per day for the duration of the docked portion of the mission

Mission Architecture

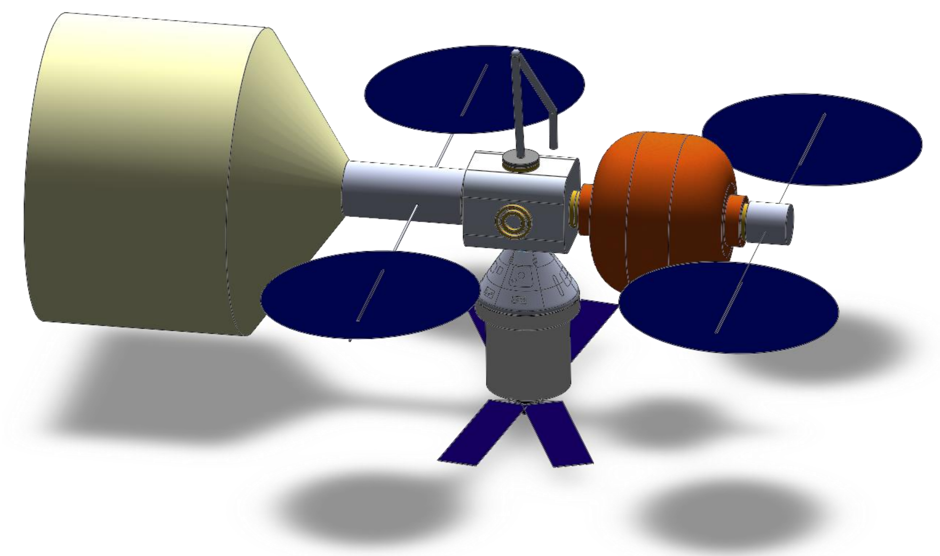


Figure 1: OASIS in the assembled configuration

The OASIS mission is designed to provide three astronauts with a laboratory and long duration habitat in deep space. The driving idea behind the mission architecture is to bring the cargo and crew safely to the Asteroid Redirect Mission at the lowest possible cost. This architecture achieves this goal with only two launches, one with a low-energy ballistic transfer trajectory and the other with a flyby trajectory.

Programmatics

Budget

Table 6: Proposed budget for OASIS

Subsystem	Cost
Launch 1 – Falcon Heavy	$135 M
Launch 2 – SLS	$1.5 B
Science Payload	$200 M
Habitat	$40 M
Multi-Purpose Docking Module	$25 M
Orion	$840 M
Propulsion	$55 M
Communications	$151.8 M
Guidance, Navigation and Control	$113 M
Power	$400 M
Total	**$3.50 B**

Estimates produced using Advanced Missions Cost Model in 2015 USD

Risk

In the design of this mission, risk of death and serious injury to the astronauts are the major mission drivers. Musculoskeletal degradation presents potential long term health risks. In comparison with missions to ISS, the mission to lunar distant retrograde orbit (DRO) increases the abort times from hours to days. A key risk in managing the dynamics and orientation of the station is the composition of the asteroid itself. Prior to any operations that may compromise the structural integrity of the asteroid; the asteroid will be thoroughly characterized so that any effect can be offset by the RCS of the station. Being the first multi component space station to be assembled at a location beyond Low Earth Orbit, there are inherent risks to crew and mission. Based on the stated scientific objectives related directly to the asteroid, there are two broad categories of risk: risk to desired outcomes due to the characteristics of the asteroid itself, and risks due to equipment failure. In the event that the contents or structure of the asteroid are ideal, there are built in descope options that can still accomplish partial completion of the objectives. In the event of equipment failure, the wide suite of experiments on the asteroid ensures that a useful minimum amount of data can be collected.

Planetary Protection

Planetary protection is the practice of protecting solar system bodies, such as planets, moons, comets, and asteroids, from contamination by Earth life, and protecting Earth from possible life forms that may be returned from other solar system bodies. Proper care and action must be taken into consideration during the design process to secure the safety and science success. (http://planetaryprotection.nasa.gov/overview) This mission will address proper compliance with both the forward and backward contamination as defined by NASA in the following aspects: (http://www.spacedaily.com/news/life-01zg.html)

1. Aseptic spacecraft assembly: The use of aseptic spacecraft assembly for any segment that can potentially have contact with the asteroid
2. Aseptic Robotic Arms and Equipment: stringent cleanliness sterilization required on all parts that has potential direct contact with the asteroid or parts of asteroid
3. Sample Return Containment: Sterilized and robotic operable storage containers
4. Curation planning and protocol

Mission Architecture of OASIS

Overview

The Orbiting Asteroid for Strategic In-Situ Supplies mission is designed to provide crew with a laboratory and long duration habitat in deep space. OASIS' mission architecture will deliver the cargo and crew safely to the Asteroid Redirect Mission (ARM) at the lowest possible cost. The safety of the crew is the highest priority. As such, many abort scenarios are implemented at the end of each systems check. The crew can abandon the mission at any phase and return to Earth.

Approach

The mission architecture was designed based on different NASA mission architectures and maneuvers. OASIS has different modules in its design that will help the mission reach its objectives. These modules are shown in Fig. 1 and are listed below:

1. The Asteroid Redirect Mission (ARM), securing the asteroid.
2. Multi-Purpose Docking Module (MPDM), linking ARM to the Labitat.
3. The Labitat and its propulsion module, shown in bright orange.
4. Orion and Orion Capsule, docked to the MPDM on the shade side.

In the list above, the modules are written in docking order. ARM is assumed to be in a Lunar Distant Retrograde Orbit. MPDM is equipped with an air lock to allow astronauts to perform Extravehicular Activities (EVA). MPDM is also equipped with a CanadarmX to help astronauts during EVA and future robotic activities. As such, the MPDM is preferred to be as close as possible to ARM to minimize the distance traveled by astronauts during EVAs and provide a better reach for CanadarmX. For the reasons mentioned, MPDM is placed between the ARM and the Labitat. The Labitat is the habitation and laboratory module for astronauts in their deep space mission. To achieve the considered configuration two options were considered to launch all modules (except ARM):

1. Three launches (2 cargo, 1 crew):
   a) First: Launch the MPDM, equipped with a guidance, navigation & control, power and communications systems.
   b) Second: Launch Labitat, equipped with a guidance, navigation and control, power and communications systems.
   c) Third: Launch Orion with Orion Service Module.

2. Two launches (1 cargo, 1 crew):
   a) Launch Labitat first, equipped with a guidance, navigation & control, power and communications systems.
   b) Launch Orion, Orion Service Module, and MPDM.

To reduce launch costs while leveraging the capability of SLS Block 1B, Option 2 was chosen. However, the team had to make sure the configuration of OASIS module assembly are respected to meet mission objectives. The following two options were considered to achieve the proposed order of modules.

Option A:
a) Launch Labitat and dock to ARM.
b) Launch Orion, Orion Service Module, and MPDM.
c) Undock Labitat.
d) Dock Orion, Orion Service Module, and MPDM.
e) Dock Labitat.

Option B:
a) Launch Labitat and insert in a parking Lunar DRO.
b) Launch Orion, Orion Service Module, and MPDM.
c) Dock Orion, Orion Service Module, and MPDM to ARM.
d) Dock Labitat to the rest.

Option B was chosen for this mission as there is a smaller risk to the mission and to the crew. In this option, there will a few maneuvers that have been tried and tested in previous NASA missions and the International Space Station. The detailed mission architecture is explained in the following phases.

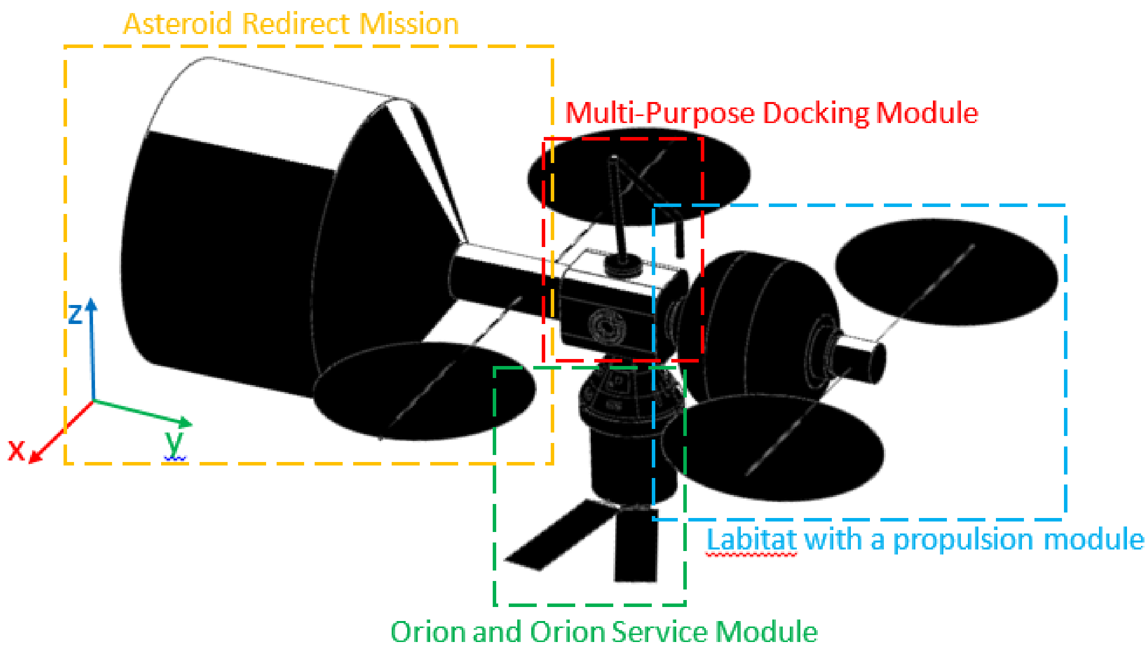


Figure 2: Spacecraft reference coordinates

Phase 1

This is the phase where the main cargo for this mission is launched. The Labitat and its propulsion module will be launched with a Falcon Heavy launch vehicle in March 2024, in a configuration as shown in Fig. 1a. As soon as the cargo is launched, it is placed in a 300 km circular parking orbit to allow the mission control to verify the readiness of spacecraft systems. The cargo is kept in this orbit for 1 to 2 orbits. After this stage, it will do a trans-lunar injection (TLI) burn that will set it in a ballistic (low-energy) trajectory towards the moon. Another systems check is performed. The Labitat is then inflated and the solar arrays are deployed as shown in Fig. 1b, followed by another systems check. The inflation is performed early in the mission to give the ground crew ample time to make a decision if anything goes wrong, either to abort the mission or to continue the mission with a decreased capacity. After about 100 days, the Labitat will be inserted into a parking Lunar Distant Retrograde Orbit (DRO). Keeping the cargo in a parking orbit will allow the crew to dock to the ARM first, without the need to re-arrange the modules on OASIS. Once a final systems check is performed by the ground crew, Phase 2 can be started.

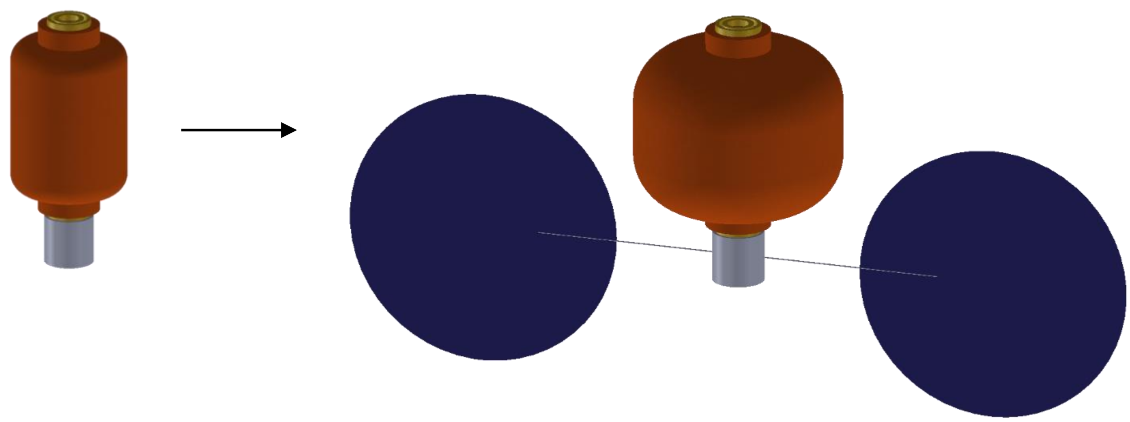


Figure 3: a) Launch configuration of the Labitat with the propulsion module. b) Inflated Labitat and deployed solar array

Phase 2

The crew is launched in this phase. Orion, Orion Service Module and the Multi-Purpose Docking Module (MPDM) will be launched on a Space Launch System Block 1B in September 2024. This launch time is 2 months over the time it takes for the Labitat to reach its parking orbit. Since the cargo's launch window is once a month, this launch time accounts for 2 delays in the cargo's launch.

Right after the crew's launch, the upper stage of the rocket is inserted into an 1806 km by 185 km elliptical parking orbit to perform a spacecraft systems readiness check. The spacecraft will be in this parking orbit for 2 orbits. If everything is nominal, the spacecraft will undergo a trans-lunar injection (TLI) burn towards the moon in a flyby outbound trajectory. A transposition, docking and extraction (TD&E) maneuver is then performed to dock Orion and Orion Service Module to MPDM's docking hatch in +y direction. This maneuver was used safely in the Apollo program to dock the Command/Service Module to the Lunar Module. Also, docking Orion to this side of the MPDM will make the final docking to the ARM safer and without a problem with the momentum of the spacecraft. Once the docking maneuver has been completed, a systems check will be performed by the ground. After 8.5 days, Orion and MPDM will be inserted into a Lunar Distant Retrograde Orbit (DRO). The ground crew will perform a final systems check before initiating the docking procedure to the ARM. After the crew has docked to ARM, there will be a final systems check for this phase before going to Phase 3.

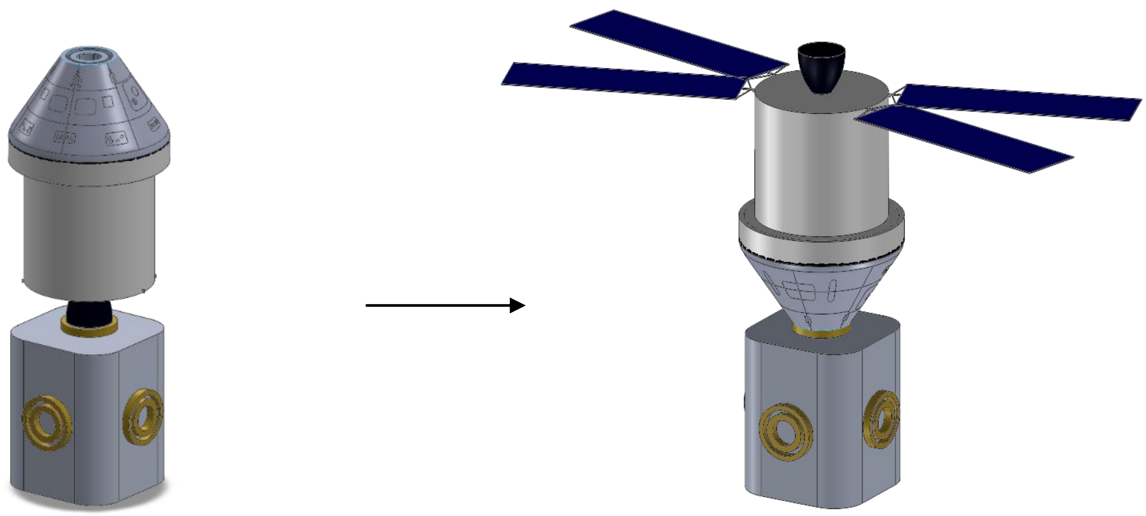


Figure 4: a) Launch configuration of Orion, Orion Service Module and Multi-Purpose Docking Module. b) Orion docked to the MPDM after a TD&E maneuver

Phase 3

This is the final phase before starting the nominal operations of the mission. CanadarmX will be used to unberth Orion from the hatch in the +y direction of the MPDM and berth it to the hatch in the –z direction of the MPDM to make way for the Labitat. This is similar to an operation that is performed on board of the International Space Station to capture and berth the SpaceX Dragon capsule using Canadarm2 (Caron, 2012). Before the start of this operation, the crew stays in the Orion capsule for continuous life support from the spacecraft. In case of emergency or an accident, the crew can still abort the mission and head back to Earth. Once Orion is successfully berthed to the hatch in the –z direction of the MPDM and systems check is performed, the Labitat can start its approach trajectory towards OASIS using the propulsion module. The Labitat will dock to MPDM's docking port in the +y direction. A final systems check is performed before starting the nominal operations phase of this mission.

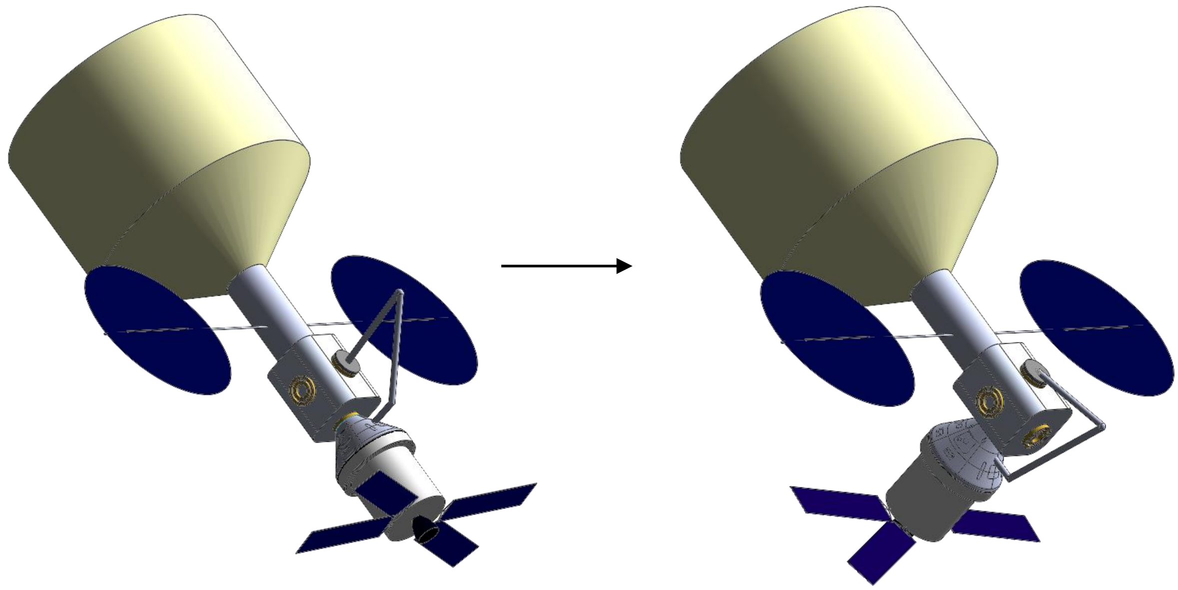


Figure 5: Orion's original docking location on the MPDM. b) CanadarmX captures Orion, unberths it and berths it to a new a docking port

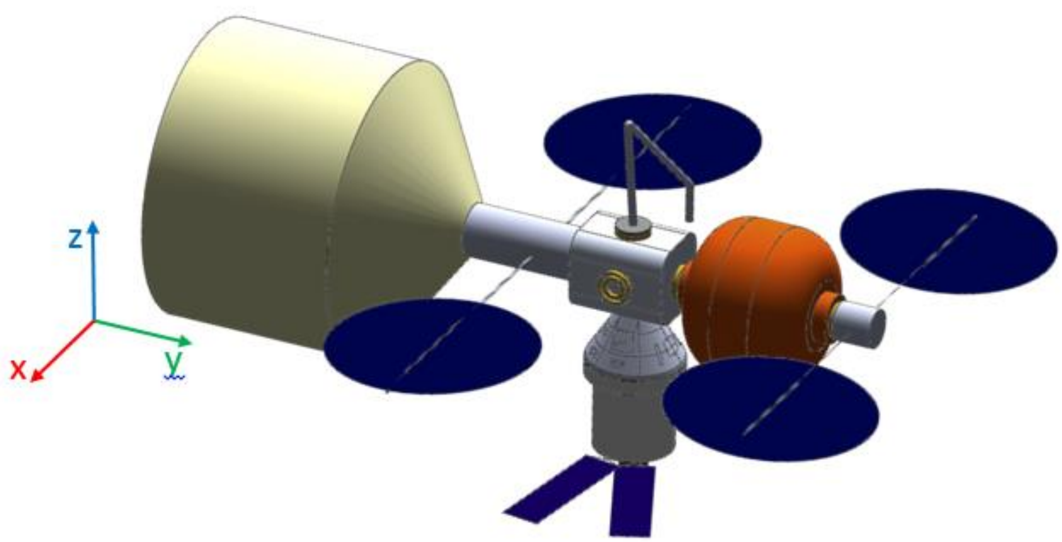


Figure 6: OASIS final configuration

Phase 4

During this stage, the crew can safely move to the Labitat and start the nominal operation of the mission. Activities performed in the nominal phase include the start of the science experiments. Seven extra-vehicular activities are to be performed in the nominal phase of the mission. The minimum duration of the mission is 42 days (22 days for science experiments, 3 days of margin, and 17 days for travel time). If everything is nominal, the mission duration could be extended to 3 months. The crew can undock and head back to Earth in case of an emergency at any time during the nominal phase.

Phase 5

This is the final phase of this mission. This phase can be initiated either after Phase 4 or at any time during the mission, when the crew has to abandon OASIS due to an emergency in case of an accident. Orion will undock from the docking port in the –z direction. A spacecraft systems readiness check will be performed by the crew and ground. Orion will do a trans-Earth injection (TEI) maneuver to go into a flyby return trajectory. After 8.5 days, Orion will enter the atmosphere and start its descent. At the moment of the splashdown in the Pacific Ocean, Phase 5 and the mission is concluded.

Figure 7: Orion EFT-1 splashdown (Photo credit NASA)

Planetary Protection

Planetary protection (PP) is a guiding philosophy in the design of an interplanetary mission. (http://www.sciencedirect.com/science/article/pii/0273117789902275) The main objective of PP is to protect solar system bodies, such as plants, moons, comets, and asteroids, from contamination by Earth life, and protecting Earth from possible life forms that may be returned from other solar system bodies. Planetary protection reflects both the unknown nature of the space environment and the interest of the scientific community to preserve the pristine nature of celestial bodies until they can be studied in detail (http://www.wired.com/2013/10/spraying-bugs-on-mars-1964). Proper care and action must be taken into consideration during the design process to secure the safety and science success. (http://planetaryprotection.nasa.gov/overview) Two major types of contamination in the planetary protection context: the forward and backward contamination:

- Forward contamination: transfer of biological organisms from Earth to another celestial body.
- Backward contamination: transfer of biological organisms from another celestial body to Earth.

According to the NASA Office of planetary protection, a space mission to a type-C asteroid is under the Committee on Space Research's (COSPAR) Category II of interplanetary contamination threat.

(http://www.nap.edu/openbook.php?record_id=11381&page=13) It stated that "… only a remote chance that contamination carried by a spacecraft could jeopardize future exploration. In this case we define "remote chance" as "the absence of niches (places where terrestrial microorganisms could proliferate) and/or a very low likelihood of transfer to those place…"
(http://www.gwu.edu/~spi/COSPAR_OP_PP_Workshop_final_Aug2009.pdf)

Since the OASIS mission involved both robotics and human operation in the proximity of an asteroid, proper planetary protection policy must be complied to prevent potential (although it is classified as a remote chance of threat.) planetary contamination between the asteroid and the manned mission. A crewed mission to the asteroid with surface operation, like in our OASIS mission, may challenge current planetary protection policies. Both forward and backward contamination will be considered in our situation since we need to preserve the science integrity of the pristine asteroid material, and also save guard the human safety. The PP philosophy will be carried out throughout the entire design process of the mission. The following aspects will be considered to help the practice of planetary protection on OASIS mission.

Aseptic spacecraft assembly: The use of aseptic spacecraft assembly for any segment that can potentially have contact with the asteroid

- Moderate cleanliness sterilization required on the OASIS Labitat and Orion module, since these parts will not be in direct contact with the asteroid or any EVA activities.
- Stringent cleanliness sterilization must applied for the Multi-purpose docking module, which will be interfacing with the astronauts during EVA, tools, and sample containers. The benefit of the Multi-purpose docking module is that it will be remaining in the DRO with ARRM module, thus the major interface between the asteroid and the mission. Thus, we don't need any additional decontamination procedure when Orion module return to Earth since Orion is not directly in contact with any EVA or asteroid activities.

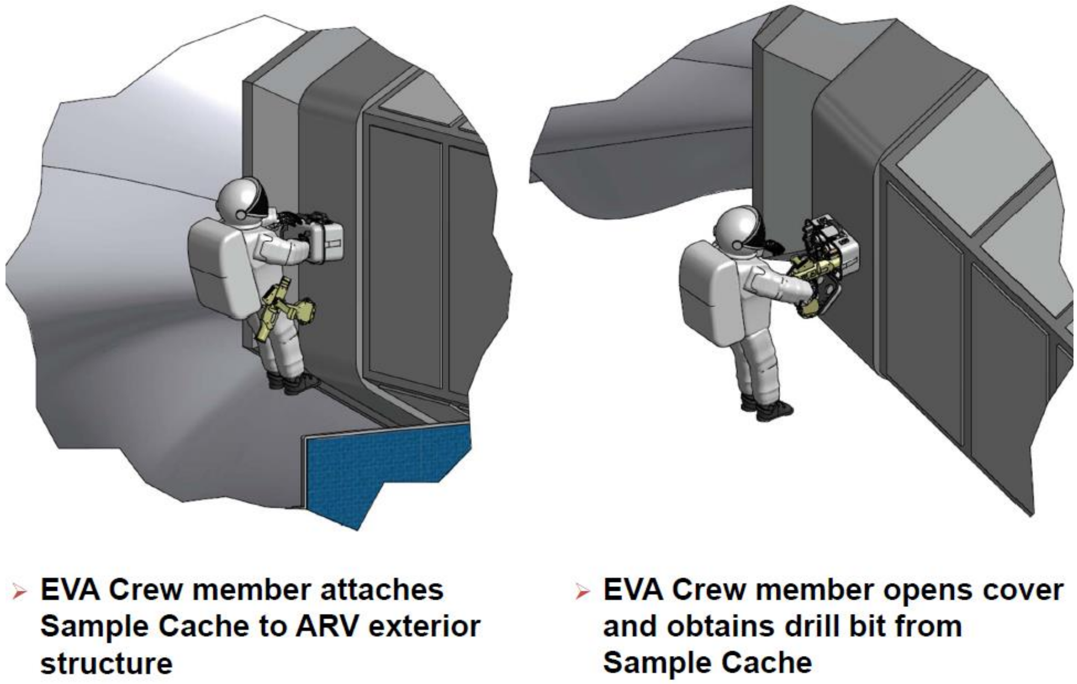


Figure X. The external drop box design on the Multi-purpose docking module for planetary protection purpose.

Aseptic Robotic Arms and Equipment : stringent cleanliness sterilization required on all parts that has potential direct contact with the asteroid or parts of asteroid

Sample Return Containment : Sterilized and robotic operable storage containers

- Double layer storage containers: The inner container unit will be in direct contact with astronauts and robotic arms during the EVA. Then, the inner container unit will be placed into the sample retrieval drop box from the outer surface of the Multi-purpose docking module. Within this drop box, a sterilized robotic arm will complete a stringent cleaning on the exterior of the inner container, and then place it into a sealed container (the outer container).
- Stringent sterilization is required for all unit that has direct / potential direct contact with the asteroid material
- All utilization processing module will be pre-designed such that they will require minimal direct human operation. If any human assist operation is needed, we will be using a dual-layer glove box to avoid possible cross contamination.

Curation planning and protocol

- Stringent sterilization is required for all ground transportation and handling of the material
- All operation upon the returned samples and tools must subjected to stringent sterilization at all time
- Long term storage and monitoring of the recovered sample condition is needed

Launch Dates

Launch 1 of Falcon Heavy with Labitat is scheduled for March 30, 2024. Launch periods for ballistic trajectories recur on a monthly schedule.

Launch 2 of SLS, MPDM and Orion is scheduled for late July 2024. Launch periods for direct trajectories recur with a period of 10.55 days, equivalent to the orbital period of the stated DRO.

Launch Vehicle

Launch 1 is performed with a Falcon Heavy. Falcon Heavy's payload capability was determined with a physical model based on uprated core specifications announced in early March 2015.

Launch 2 is performed with an SLS Block 1B, incorporating the Exploration Upper Stage. Its payload capability to the desired injection orbit was determined with an analogous physical model and calibrated by design reference.

On orbit propulsion is performed with standard hypergolic (UDMH/N2H4) bipropellant thrusters.

Mission Duration

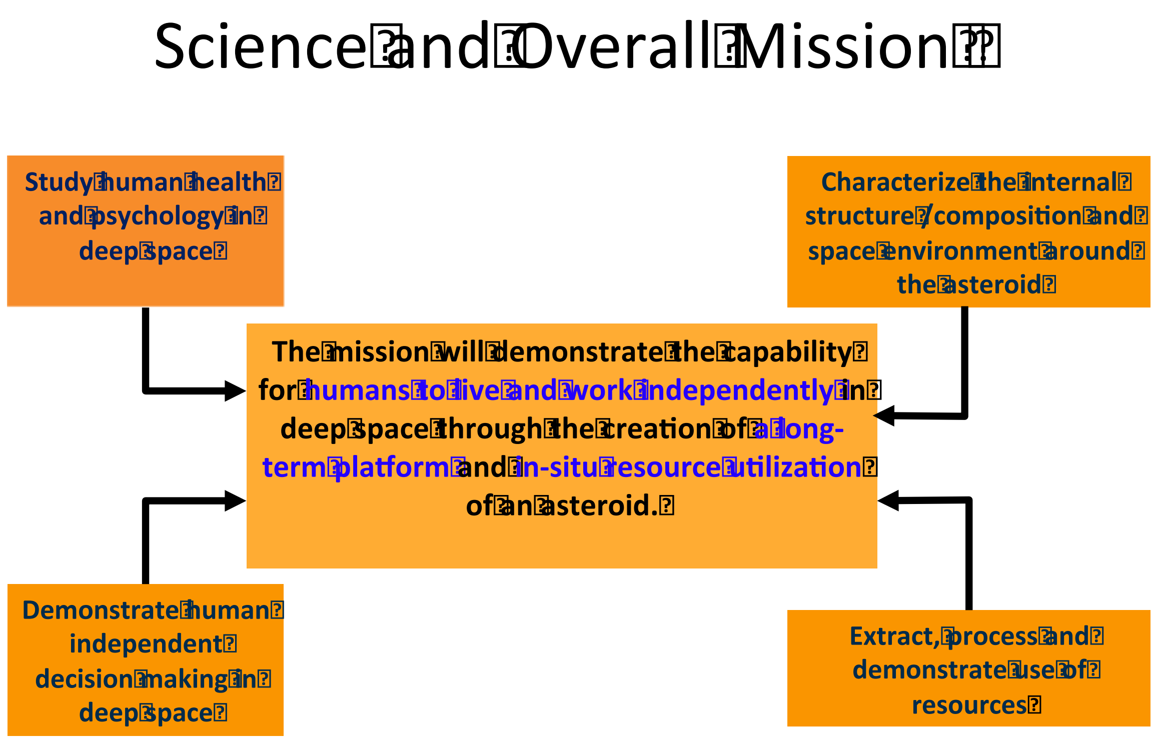
Uncrewed Labitat ballistic transfer orbits take around 100 days to reach orbital insertion.

The crewed portion of the flight will take 8.5 days for direct flyby transfer outbound, 25 days on orbit (2.5 revolutions of the DRO), and 8.5 days for return. 25 days on station covers the scientific objective of 22 days of work. The total crewed duration is 42 days or 6 weeks.

Science Overview

Exploration of near earth objects (NEOs) is essential to further our understanding of the formation of planetary bodies, the chemical and physical history of the solar system, and conditions that are capable of sustaining life (NASA Science Plan, 2014). No previous missions to NEOs have provided information on their internal structure, composition, and geology. The science objectives provide thorough insights into just one of the millions of asteroids predicted to exist in the solar system.

The Oasis Mission will advance our understanding of humans and asteroids in deep space. Humans will spend a minimum of 14 days in deep space, the longest in history, aboard a habitat where they will live and work in closely monitored conditions. Samples collected from the asteroid will advance our understanding of the structure and composition of the asteroid, as well as the asteroid space environment. In addition, in situ resource utilization of the asteroid will be demonstrated.



Flow chart of mission statements into high level science objectives.

Primary Science Objectives

Science and technology objectives involve investigation, collection, and utilization of asteroid material. Science products and required instrumentation are summarized in the following figures and tables.

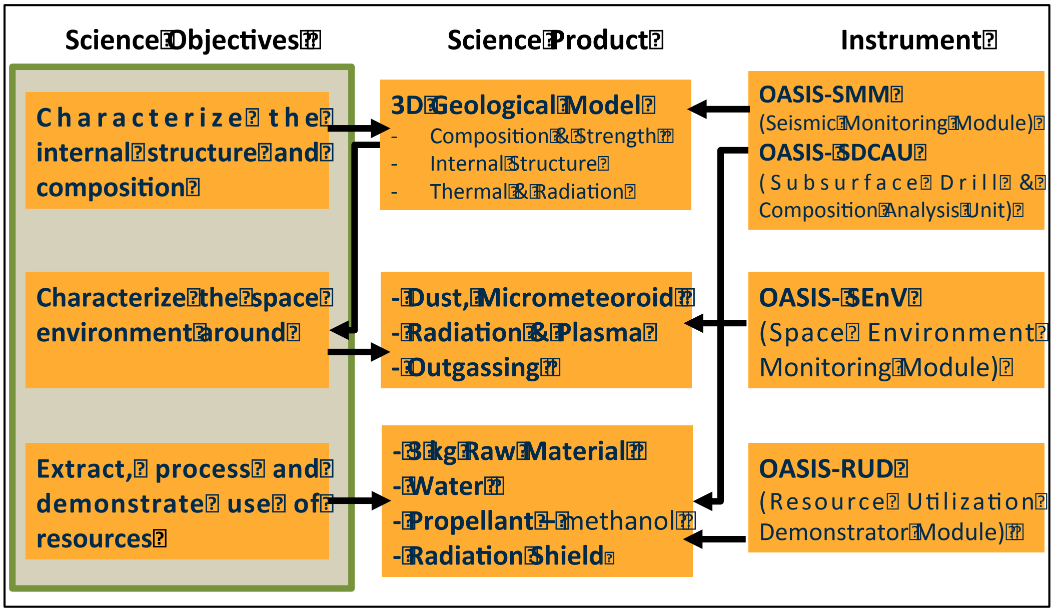


Figure 2: Flow chart indicating science objective flow into products and instrumentation

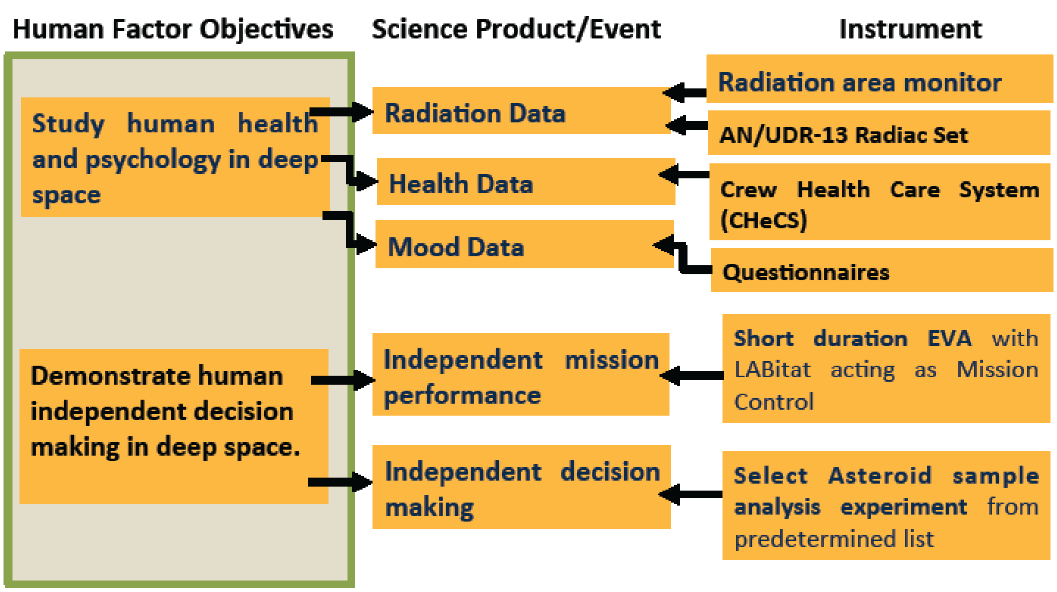


Figure 8: Flow chart indicating human science objective flow into products and instrumentation

The primary science objectives are summarized in the following table:

Table 7: Primary science objectives

1. Characterize the internal structure and composition of asteroid
2. Characterize space environment around asteroid
3. Extract, process, and demonstrate use of resources
4. Contribute pioneering human health and behavioral data in deep space
5. Demonstrate human autonomous decision making in a deep space environment

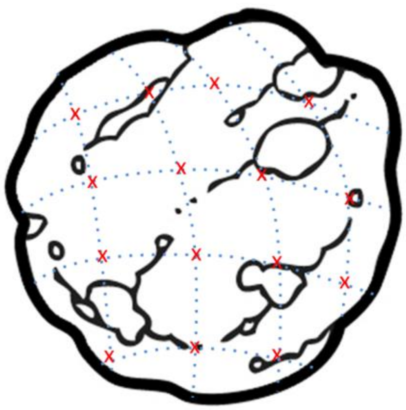


Figure X. Surface Mapping and 3D Sample Collection Location

Secondary Science Objectives

The secondary science objectives are summarized in the following table:

Table 8: Secondary science objectives

1. Advance our understanding of the origin and evolution of the solar system
2. Improve current asteroid classification scheme
3. Demonstrate planetary defense capability via manipulation

Science Traceability Matrix

An abbreviated version of the science traceability matrix for the primary science objectives is shown in the table below. A more detailed table is included as an Appendix.

Table X. The Science Traceability Matrix

Science Objectives	Measurement Objectives	Instruments	Weight (kg)	Power (W)	Cost ($M)
Characterize internal structure and composition of the asteroid	Measure the acoustic response (seismic) of the interior	Seismometers	50	100	20
	Drill 1 m borehole	Drill + sensors	10	200	5
	Obtain core sample	Coring bit			
	Determine mineralogy and water form (µm penetration)	IR Spectrometer	1	50	15
	Determine lithology and water content (cm penetration)	Neutron Spectrometer Gamma Ray Spectrometer	1	50	15
	Determine detailed mineralogy	XRF	1	50	15
Characterize space environment around the asteroid	Surface dust environment	Dust Detector	<1	<1	<1
	Plasma and magnetic field	Plasma Monitor Magnetometer	<0.1	1	1
	Radiation and space weather impact	Radiation Access Detector	1.5	4	1
ISRU	Demonstrate feasibility of deep drill technology	Deep drill	10	200	10
	Processing and demonstration of resource utilization	Custom asteroid processing unit	240	1000	44
Human Health and Behavior	Monitor radiation	RAM	0.5	5	<1
		AN/UDR-13 Radiac Set	<0.01	-	<1
	Training effectiveness	Optical cameras mounted in living space	25	1	4
		Optical cameras mounted to external surface of vehicle		1	
	Psychology and Group Dynamics	Optical cameras mounted in living space	NA	NA	NA
		Paper questionnaires	NA	NA	NA
	Suit Testing	Heart rate monitor in suit Blood pressure monitor in suit	Included in human factors requirements		
	Health Monitoring	CHeCS	Included in human factors requirements		
Human Decision Making	Successful completion of a task selected by the astronaut	Table of predetermined tasks	NA	NA	hyub

Minimum Science Mission Success Criteria

The minimum success criteria are summarized in the following table:

Table 9: Success Criteria – Minimum

1. Drill and sample subsurface asteroid material to a minimum depth of 1 m
2. Return samples to Earth with proper storage
3. Excavate 1 kg of material
4. Record radiation levels and cosmic ray exposure once per day in the human living quarters for the duration of the docked portion of the mission
5. Characterize the space environment for a single asteroid location
6. Process, utilize, and generate a specified amount of 1 specified resources
7. Successful completion of 1 experiment selected by the astronaut from a predetermined list
8. Monitor physiological data once per day for the duration of the docked portion of the mission

Full Science Mission Success Criteria

The full science mission success criteria are summarized in the following table:

Table 10: Success Criteria – Full

1. Deployment and successful acquisition of 3D seismic array
2. Successful retaining of seismic array as long-term microseismic monitoring system
3. Drill and sample subsurface asteroid material to a minimum depth of 5 m
4. Return samples to Earth with proper storage
5. Excavate 3 kg of material
6. Record radiation levels and cosmic ray exposure once per day in the human living quarters for the duration of the docked portion of the mission
7. Characterize the space environment for 6 asteroid locations
8. Process utilize and generate a specified amount of all specified resources
9. Successful completion of 1 independent exercise for the duration of 1 hour
10. Monitor physiological data once per day for the duration of the docked portion of the mission

Scientific Instrumentation

OASIS Subsurface Drill & Composition Analysis Unit (OASIS-SDCAU)

The SDCAU (Subsurface Drill and Composition Analysis Unit) module consists of 4 remote sensing instruments designed to provide information on the elemental, chemical, and lithological composition of the asteroid subsurface, as well as the water content and its form. They will be packaged inside a single drill tool capable of descending into a 5 m borehole. (Fig. 1)

Drill

The rotary-percussive drill consists of a drill bit and accompanying drill string, into which the remote sensing instruments are packaged. Based on Honeybee Robotics' Icebreaker (McKay et al, 2013) and Deep Drill (http://www.planetary.org/explore/projects/planetary-deep-drill/), the drill will be extensively field-tested and at an advanced TRL by the mission date. Including both a regular drill bit and a coring bit, the drill can achieve a depth of up to 5 m in 5 hr when not coring, and can safely and securely return samples to the operating astronaut. The drill bit is 3 cm in diameter and can obtain core samples up to 30 cm in length. In total, 3 kg of pristine asteroid material will be obtained.

Near-IR Spectrometer

The Near-IR spectrometer is a common instrument used in multiple past missions (Curiosity, OSIRIS-Rex, Hayabusa 2, etc) that operates in the 0.5 to 3.5 μm range. It provides information on the absorbance of infrared light of chemical compounds within the upper μm of the sample being analyzed. Individual compounds can have very distinct absorption characteristics, the most important for our mission being the strong absorbtion band at 3.1 μm, indicating the presence of water and its form (free or bound in a hydrated mineral). The tool itself will take spot measurements at 5 cm intervals within the borehole.

X-Ray Fluorescence

X-Ray fluorescence, an instrument used on Curiosity (http://msl-scicorner.jpl.nasa.gov/Instruments/CheMin/), is a tool that gives detailed elemental composition of the substrate under investigation via secondary emission of X-rays from an "excited" material (Thompson et al, 2008). The information it provides will be used to determine the elemental composition of the subsurface to a very detailed degree.

Gamma Ray Spectrometer

Widely used in the oil, gas, and mining industries, as well as on multiple past missions (NEAR, Dawn, MESSENGER, Curiosity), the gamma ray spectrometer provides lithologic characterization of the subsurface based on detection of natural gamma radiation. Therefore, it is most sensitive to the presence of radioactive material such as K, Th, and U.

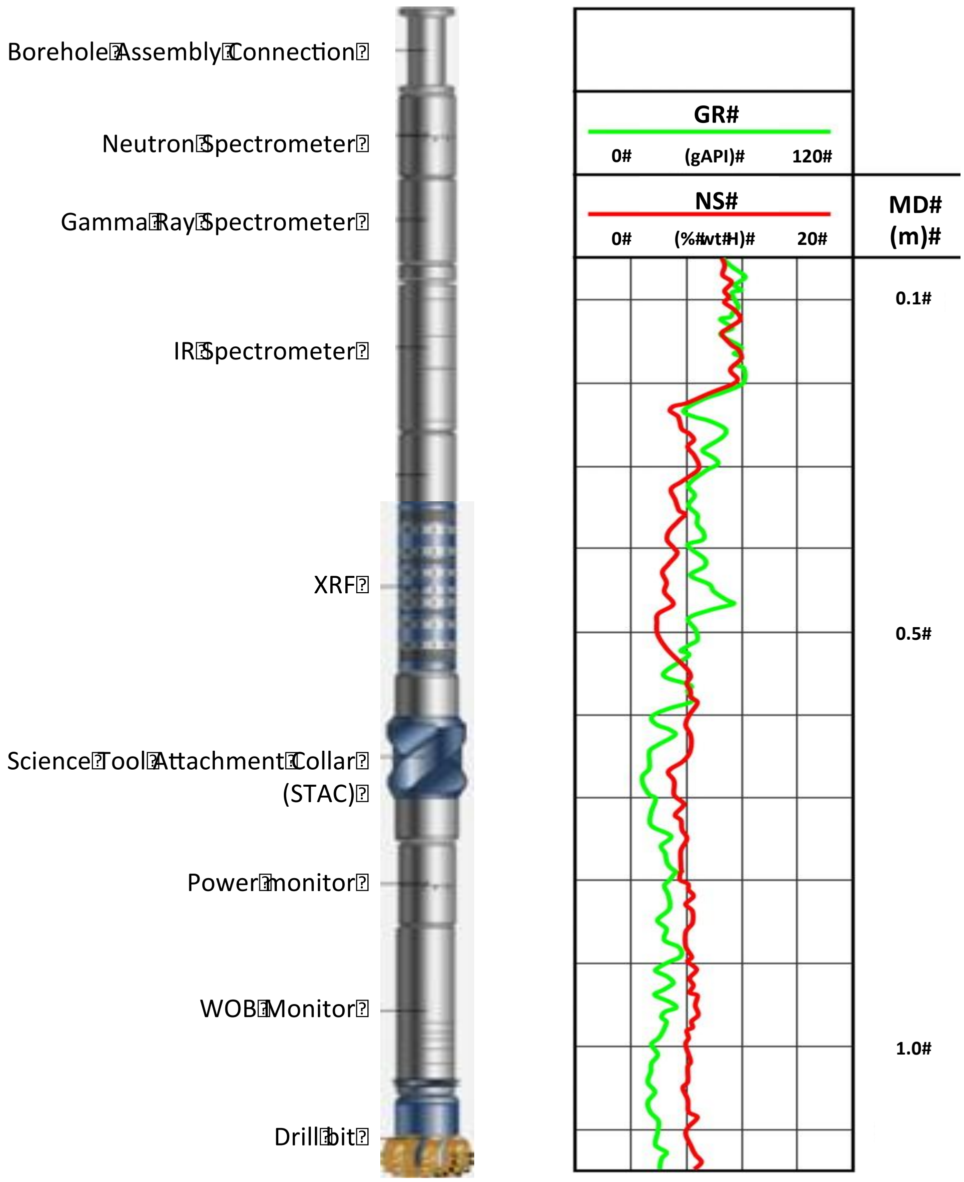


Figure 9: A generalized schematic of the Deep Drill and its remote sensing package, the SDCAU (L). An example of a potential borehole log showing the gamma and neutron response of a substrate at depth (R). Photos modified from: publications.iodp.org (L) and pgc.lyellcollection.org (R)

Neutron Spectrometer

Another workhorse of the remote sensing missions (Dawn, MESSENGER, Curiosity), the neutron spectrometer measures the hydrogen content of the substrate and can provide a proxy for overall water content. Used in combination, these instruments can form a detailed analysis of the subsurface of the asteroid, to a degree not yet achieved by previous missions.

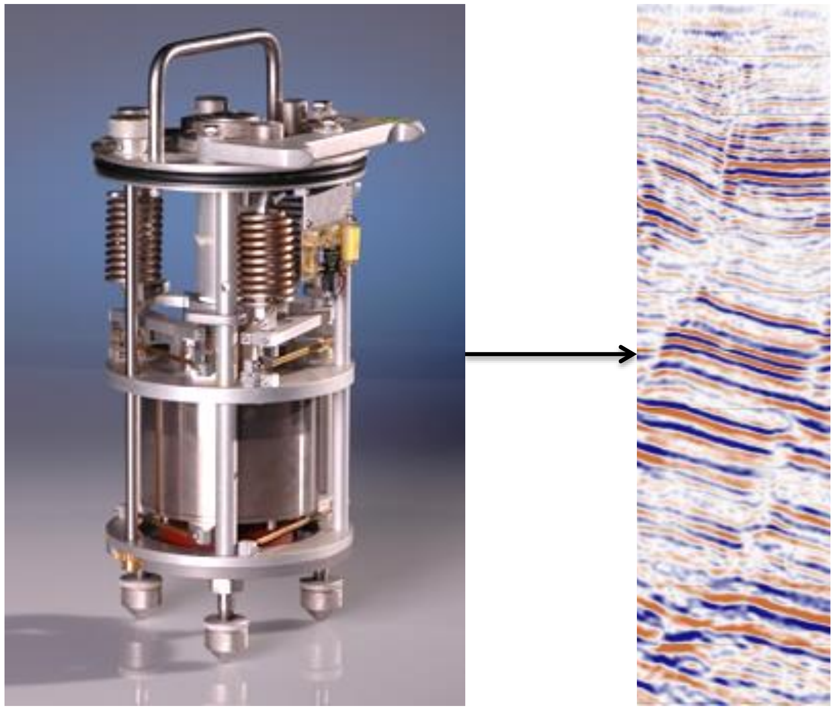


Figure 10: An example geophone and resulting seismic data. Photo credit: bgr.bund.de (L) and sub-surfrocks.co.uk (R)

OASIS Seismic Monitoring Module (OASIS – SMM)

The Seismic Monitoring Module is a seismic array (Fig.2) designed specifically to sound the interior of the asteroid using terrestrial geophysical seismic acquisition techniques. A 1 kHz source will provide the acoustic energy that enters the asteroid and is reflect off internal surfaces or interfaces (due to changes in density or seismic velocity). These reflected waves are collected by up to 10 receivers placed at the surface and processed to provide a 3D image of the interior of the asteroid (Fig. 2), coupled with the information collected by the SDCAU, the interior composition will be well constrained and a full 3D geological model can be obtained.

After the seismic sounding has been performed, the seismometers provide an addition, critical function. They will be left to record all seismic activity occurring on or in the asteroid during ISRU operations, more specifically during the excavation phase. Any significant movement or stress change within the asteroid body will be detected and can be used to warn the attending astronauts of danger. This can provide a necessary safeguard against damage to either crew or equipment.

OASIS-Space Environment Monitoring Module (OASIS-SEnV)

Module Objectives

The OASIS-Space Environment Monitoring Module (OASIS-SEnV) is the long term space environment characterization module for the captured asteroid. The main objective of the module is to characterize the space environment around the asteroid, such as dust, plasma, vacuum, and radiation environment parameters. Some of the space environment measurements can also help the understanding of the electrical properties of the asteroid. The module can be installed on the surface of the asteroid by human astronaut or by robotic arms.

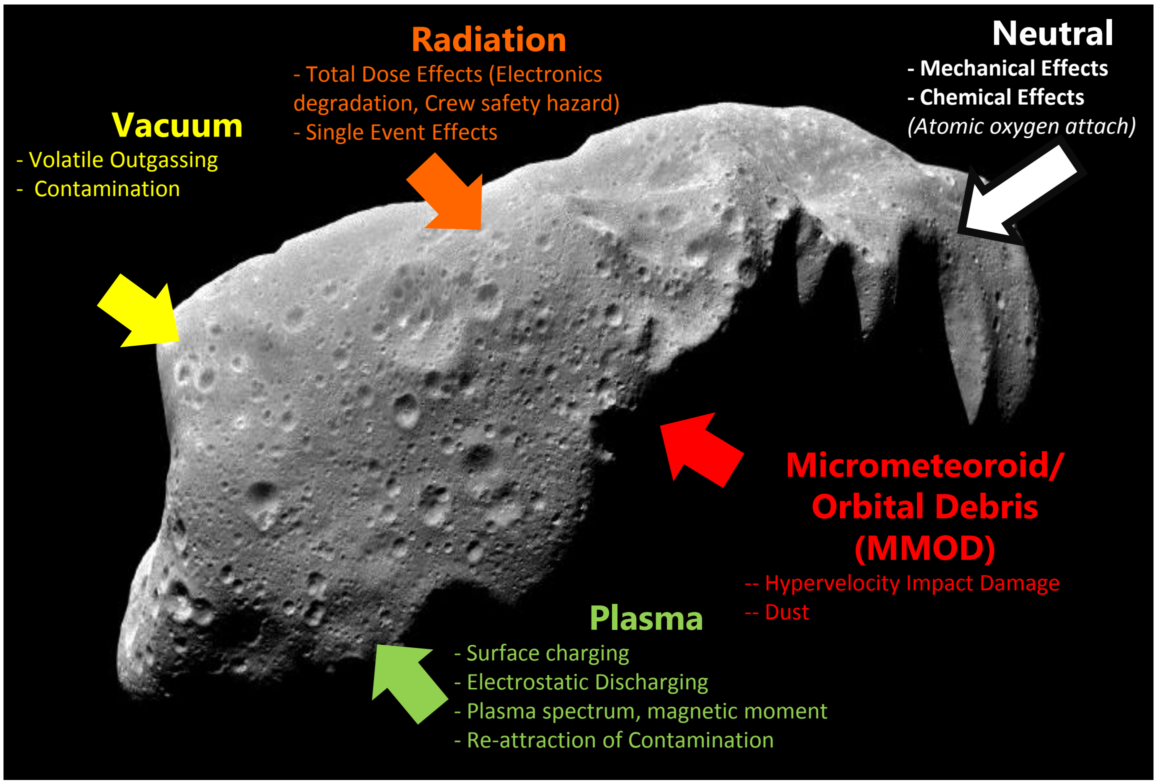


Figure 11: Space Environment around an Asteroid

The space environment studies analyze the physical environment in space and its effects on space systems and astronauts. To achieve safe and successful manned mission or unmanned space mission, proper characterization of the space environment is essential. Space environment consists of neutrals, plasma, vacuum, radiation, and micrometeoroids/orbital debris environment parameters. They corresponds to various hazards (ESA, 2014). The space environment and hazard around an asteroid is demonstrated in Fig.1. In the vacuum environment, the potential hazards are volatile outgassing from the surface of the asteroid and contamination of the surface from materials in vacuum. Volatile outgassing can cause condensation onto optical instrument operating on the asteroid, which often leads to the reduction of optics image quality and impede the science outcome of many instruments. The outgassing condensation can also backward contaminate the astronaut during the EVA via condensation on the spacesuit.

The radiation environment around the asteroid is another vital parameter. Space radiation comes from ionizing radiation which exists in the form of high-energy, charged particles. There are three major natural sources of radiation in space: trapped radiation, galactic cosmic radiation (GCR), and solar particle events (SPE). The GCR Galactic cosmic radiation comes from outside the solar system, and consists of ionized atoms. Solar particle events are injections of energetic electrons, protons, alpha particles, and heavier particles from the Sun into interplanetary space. (http://srag-nt.jsc.nasa.gov/spaceradiation/what/what.cfm) The Sun produces a constant stream of high speed particles, i.e. the Solar Wind, due to the surface activity on the Sun. The charged particles of the solar wind and GCR cannot easily penetrate the Earth's magnetic field, and thus the Earth's magnetic field protects the spacecraft in orbits around Earth. Except for the Apollo missions, NASA's manned spaceflight missions have stayed well below the altitude of the Van Allen belts. (http://srag-nt.jsc.nasa.gov/spaceradiation/what/what.cfm) Since the asteroid at DRO doesn't have standing magnetic field around it, the radiation sources of interest are GCR and SPE. They should be properly characterize to ensure the EVA operation safety on the surface. Additionally, these radiation can cause electronic degradation on the robotics and surface sensor suites on the asteroid. Thus, it is important to characterize the long term radiation environment for future development on the asteroid. On the side, the radiation from the Sun and from the asteroid's surface can contribute to the Yarkovsky effect. It is due to a force acting on a rotating body in space caused by the anisotropic emission of thermal photon momentum due to thermal gradient on the asteroid surface (http://en.wikipedia.org/wiki/Yarkovsky_effect).

The neutral environment study the neutral particles impact on the asteroid. The main effects are chemical effects, and the mechanical effects of neutral impacts, such as dust environment. The chemical effects can arise from possible volatile re-attachment on the surface, such as atomic oxygen attachment, and causing surface contamination of the sensor instrument. The mechanical impact, especially from dust environment, can cause EVA safety issue and back contaminate the astronaut spacesuit. Therefore, the characterization of the neutral environment should be done properly to ensure astronaut safety. The dust can not only be from the asteroid itself, but can also be from other sources in space. Since the asteroid is in the DRO, the only concern will be (micro-) meteoroid, which are a small particle of rock in space and weighing less than a gram. The man-made debris, i.e. the orbit debris, is not as relevant to our problem as we are far away from the cluster of the near Earth orbits. The only man-made debris concern might arise is the generation of debris from the resource extraction process in the mission. However, considering the mass is diffusing/traveling at relatively low relative speed with respect to the asteroid, the debris impact damages will not be a major safety concern for the astronaut and the habitat.

The **plasma environment** and the magnetic field (if any) around the asteroid is also important as it can have coupling effect with the solar wind, and impact the radiation safety on the surface of the asteroid (Lee et al., 2013). To complete in-situ monitoring of the plasma (electrons and ions), their composition, distribution, temperature, density, flow velocity, and the magnetic field will be necessary to complete the study of plasma around the asteroid. The study of the plasma environment can help the understanding of the coupling processes of asteroid dust, gas, and plasma as well as its interaction with the solar wind; also, a good understanding of the overall physics and chemistry of the asteroid can be extracted. The asteroid's electrical properties of the crust, the remnant magnetization, surface charging, and surface modification due to solar wing interaction can be characterized. (http://sci.esa.int/rosetta/35061-instruments/?fbodylongid=1644)

OASIS-SEnV Summary

Science Objectives: Characterize space environment around the Asteroid

Measurement/Monitoring of surface **Dust environment**
Instrument: Surface Dust Impact Monitor (sDIM)
Product: Surface map of dust mass and velocity (3D); and long term continuous measurement
Significance: By monitoring the surface dust activity, we will be able to improve the EVA safety and avoid possible sample contamination.

Measurement/ Monitoring of surface **Radiation environment**
Instrument: Surface Radiation Access Detector (sRAD)
Product: surface map of GCR and SEP for the asteroid, total radiation dosage

Measurement/ Monitoring of **Plasma Environment**
Instrument: Surface Magnetometer and Plasma Monitor (sMAP)
Product: plasma (ion/electron) and magnetic moment map

Measurement of **Vacuum Environment**
Instrument: Inductive Coupled Plasma Mass Spectrometer (ICP-MS)
Product: composition and relative content of outgassing volatile speciation; surface map of the volatile.
Will be described in OASIS- Resource Utilization Demonstrator Module (OASIS-RUD).

Table 11: OASIS-SEnV Module Instrumentation Breakdown
(Mass and power are listed per unit)

Measurement	Instrument	Science Product	Power (W)	Mass (kg)
Dust	sDIM	Dust mass and velocity	0.170	0.405
Radiation	sRAD	GCR and SEP radiation dosage	0.035	1
Plasma and magnetic field	sMAP	plasma spectrum; magnetic moment	1.5	4.2
Vacuum	ICP-MS	outgassing volatile	100	25

Module sub-Instrumentation

Two of the module will be distributed via robotic arm during the preliminary robotic survey to characterize the surface space environment condition prior to EVA activities to secure the astronaut safety and science mission success. During the main science operation, we will distribute an additional 10 modules to establish a surface space environment sensor map, which will offer continuous space environment condition data via a RF link to the habitat, then to the ground control.

Surface Dust Impact Monitor (sDIM)

The sDIM instrument provide continuous dust velocity and mass measurement on the surface. sDIM measures the dust and ice particles escaping from the surface of the asteroid. The particles are detected when impacting on the surface of the cube-shaped sensor. The sDIM senses the falling particles by an arrangement based on the acoustic impact monitoring principle. The demonstrative figure of sDIM is shown in Fig.3. There are three active pizeo active faces on the side of the cube.

We will be using the Dust Impact Monitor (DIM) from the Rosetta's lander Philae of the European Space Agency, which granted this payload great flight heritage as it is a flight proven functioning dust monitor for a similar application (on a comet). The sDIM sensor capability is shown in Table.II based on the sensor performance of the current DIM.

Science Outcome

Based on the C-type asteroid composition (Nelson et al), the ice particle, possible hydrated mineral, and other volatile components as well as dust can be detected by the sDIM. When the volatile substances sublimate from the asteroid surface, dust and other particles with a diameter between a few micrometers and centimeter can rise up due to solar radiation. (http://sci.esa.int/rosetta/35061-instruments/?fbodylongid=1644) The particles' initial velocity is not large enough to enable them to leave the asteroid gravity field, and thus falling back to the surface.

DIM is intended to monitor volatiles released from the surface of the asteroid, but do not have sufficient velocity to escape the gravity of the asteroid.

Table 12: sDIM sensor capability

Sensor Area	70 cm^2
Particle velocity	0.25 - 2 m/s
Particle radius	0.2 - 5 mm
Energy at impact	2×10^{-9} - 2×10^{-5} J
Particle mass	3×10^{-8} - 5×10^{-4} kg

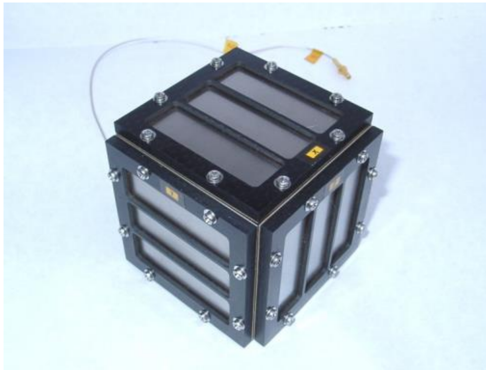


Figure 12: Dust Impact Monitor from the DIM on Rosetta mission

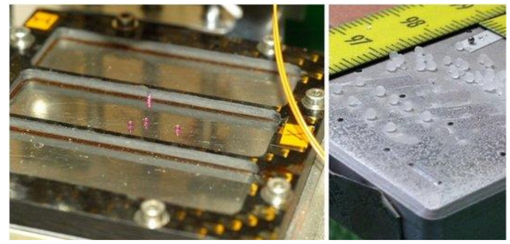


Figure 13: The instrument's sensors are calibrated with tiny high-precision spheres made of ruby (left) and ice

Justification of Instrument Selection

Reliability and Reduce Science Risk -- The flight heritage of the DIM has shown a mature technology level (TRL) to do single point dust environment monitoring on a similar environment, i.e. on a comet. Thus, the selection of such instrument will provide great reliability and reduce science mission risk of measurement failure. This also reduces the technology development cost.
Improve crew EVA safety – the characterization of surface dust condition will enable a space environment monitoring/alert system for the astronaut during or prior to the EVA.
Light weight and Portable Device – The design feature a light weight and portable design, which enable easy deployment by the astronaut or via a robotic arm.

Surface Radiation Access Detector (sRAD)

The Surface Radiation Assessment Detector (sRAD) is an energetic particle analyzer derived from the existing Mars Science Lab – curiosity rover’s Radiation Assessment Detector (RAD). It is designed to characterize the full spectrum of energetic particle radiation at the surface of the asteroid, including galactic cosmic rays (GCRs), solar energetic particles (SEPs), secondary neutrons and other particles created both in the atmosphere and in the asteroid regolith. In addition to surface condition monitoring, the sRAD can also be enable during the cruise phase of the mission to monitor the radiation levels around the route. The example surface radiation dosage time evolution result is shown in Fig.X. This information is essential for the future establishment of long term habitat on/near the asteroid.

Science outcome

- Radiation dose on the surface of Mars, which can be used for human future missions
- Full high energy particle spectrum for the Martian surface: protons, energetic ions of various elements, neutrons, and gamma rays
- Both direct radiation from space, and secondary radiation due to the interaction of space radiation with the asteroid surface rocks and soils.

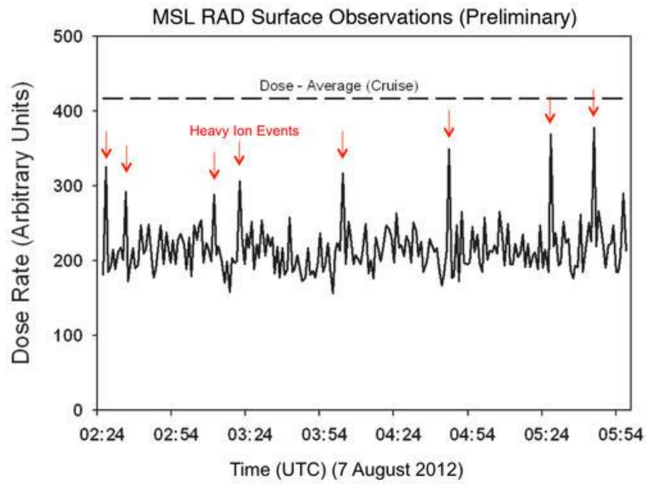


Figure X. Example MSL-RAD Surface observation. (http://mars.jpl.nasa.gov/msl/mission/instruments/radiationdetectors/rad/)

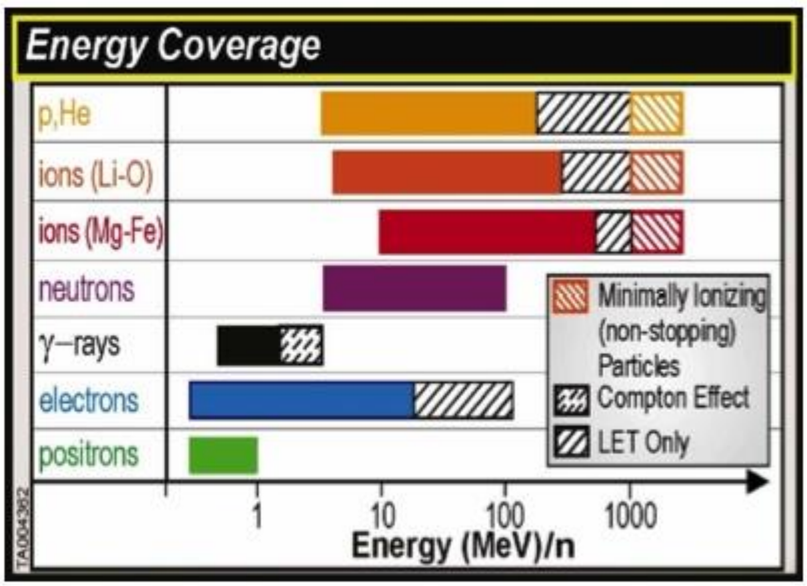


Figure Y. Energy Coverage Range of MSL-RAD. (http://mars.jpl.nasa.gov/msl/mission/instruments/radiationdetectors/rad/)

<u>Justification of Instrument Selection</u>

Reliability and Reduce Science Risk – The RAD detector is one of the first instrument set to Mars specifically to prepare for future human exploration (http://mars.jpl.nasa.gov/msl/mission/instruments/radiationdetectors/rad/**)**. It is on the Mars Science Laboratory (MSL), and has been successfully measure and identify all high-energy radiation on the Mars surface. The RAD has shown a mature technology level (TRL) to measure long term surface radiation spectrum.

Improve crew EVA safety – the characterization of surface dust condition will enable a space environment monitoring/alert system for the astronaut during or prior to the EVA.
Long term environment monitoring indicator – the RAD has been proven to be capable of long term radiation level monitoring, which is essential for exploration habitation establishment.

Surface Magnetometer and Plasma Monitor (sMAP)

The Surface Magnetometer and Plasma Monitor (sMAP) is a multi-sensor experiment, which measures the magnetic moment and the plasma condition near the surface of the asteroid. It is derived from the Rosetta Lander Magnetometer and Plasma Monitor, which is the first instrument to ever operate a magnetic sensor from within a plasma sensor. (http://sci.esa.int/rosetta/31445-instruments/?fbodylongid=901) The monitor consists of a fluxgate magnetometer, an electrostatic analyzer (plasma sensor) with integrated Faraday cup which measures ions and electors, and localized pressure sensor. It is capable of performing measurements between +/- 2000nT with a resolution of 10 pT.

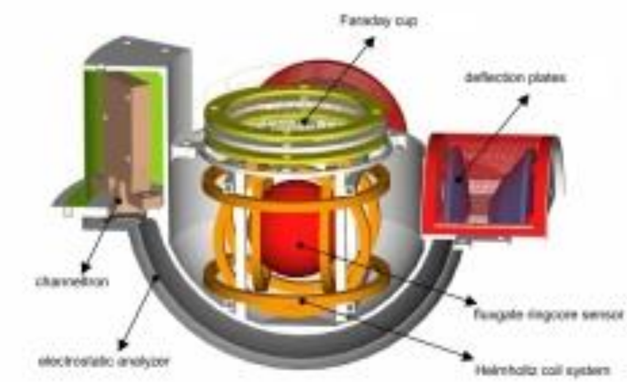


Figure P. Lander MAgnetometer and Plasma Monitor
(http://www.igep.tu-bs.de/forschung/weltraumphysik/projekte/rosetta/romap_en.html)

Justification of Instrument Selection

Reliability and Reduce Science Risk -- The flight heritage of the sMAP has shown a mature technology level (TRL) to do single point plasma environment monitoring on a similar environment, i.e. on a comet. Thus, the selection of such instrument will provide great reliability and reduce science mission risk of measurement failure. This also reduces the technology development cost.
Improve crew EVA safety – the characterization of surface plasma c will enable a space environment monitoring/alert system for the astronaut during or prior to the EVA, and hazardous surface charging situations.
Light weight and Portable Device – The design feature a light weight and portable design, which enable easy deployment by the astronaut or via a robotic arm.

OASIS-Resource Utilization Module (OASIS-RUD)

Under the assumption that the asteroid would be a C1-type asteroid (see table below), the utilization experiments are designed to show extraction and purification of water, generation of oxygen for breathing or as an oxidizer, use of regolith as radiation shield, and production of rocket fuel (methanol or methane). These ISRU end-uses were selected because they are either essential consumable products (water, fuel, oxygen) or represent a large investment in mass to put into orbit (e.g. radiation shielding). For long duration missions to Mars, the in-situ production of fuel permits greater payload delivery to the surface if return fuel does not need to be brought along, saving tons of fuel and oxidizer. Methanol/oxygen rockets have a long history, most famously on the German V2 rockets.

Methane/oxygen rockets are also fairly common, being used on SpaceX Raptor engines, NASA's Project Morpheus lander, and many Russian RD-series engines. Team Voyager is remaining agnositc with regards to ultimate fuel use. We believe methane is easier to produce and separate, but more difficult to use because it would need to be liquefied. Various techniques were considered for the utilization experiments, e.g. pervaporation, filtration, pyrolysis, milling, RWGS, the Sabatier process, solvent extraction, electrolysis etc. Further discussion of these methods along with a more detailed description can be found in Use of Extraterrestrial Resources for Human Space Missions to the Moon or Mars, D. Rapp, Chapter 2, 2013. One example flow sheet for producing the targeted materials mentioned previously from the asteroid is detailed in the figure below.

A concept image of the OASIS-RUD module for producing water, oxygen and methanol from the sampled asteroid material. After inserting the asteroid sample into the asteroid sample chamber, an astronaut could control the various processing steps detailed in a flow sheet below while monitoring process variables from the monitoring interface. The water, oxygen and methanol produced would be collected from the product lines and stored for further Earth-based analysis and testing.

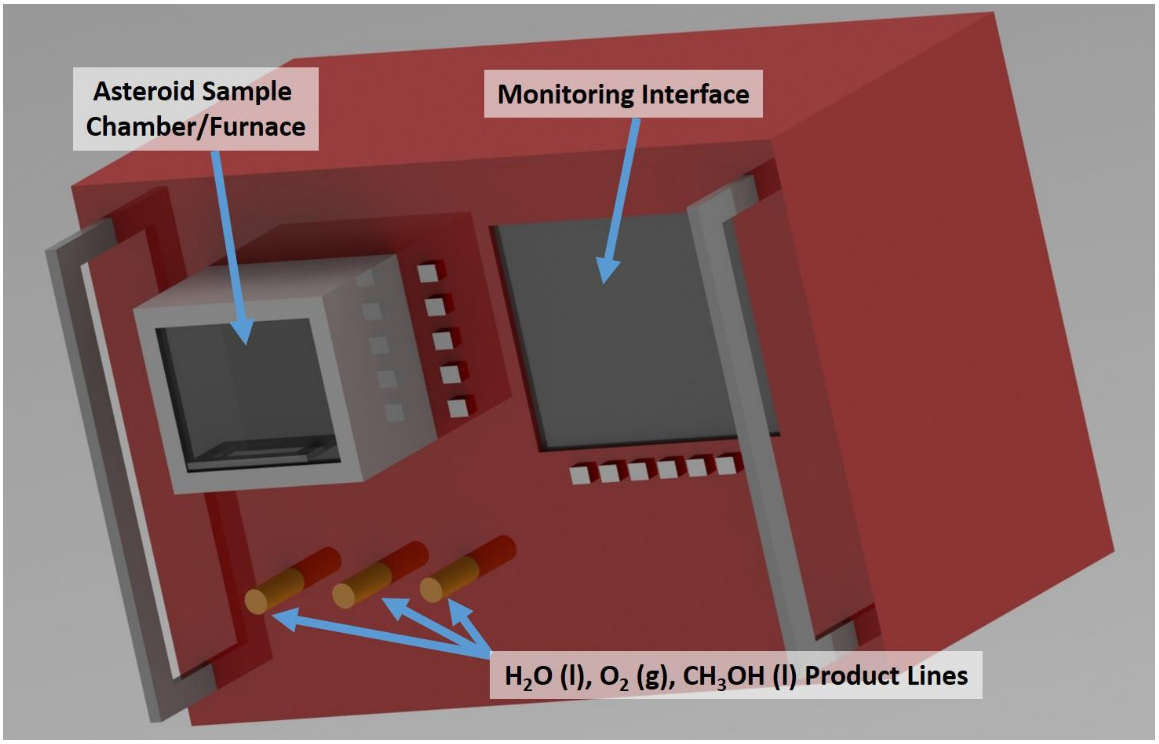


Figure 14: Conceptual Image of the OASA-RUD

The raw material extracted from the asteroid is first milled to a fine powder and heated to approximately 600°C to remove all volatiles including organics, water, ammonia, etc. followed by condensation. The remaining refractory material is pressed into a mold and sintered at approximately 1000°C to form a refractory brick, which can be used to test for radiation shielding effectiveness in situ. The condensed volatiles are fed into a centrifuge, where the various liquids present can be separated. After passing through the filter and activated carbon, the product water is collected. Furthermore, the water can be decomposed to oxygen and hydrogen gas by electrolysis. The oxygen gas is collected. The hydrogen is fed with carbon dioxide (fed from the ECLSS) to RWGS reactor to produce water and methanol, which can be separated by pervaporation. An ICP-MS could be utilized in the unit to track composition of the various intermediates and products throughout processing. This ICP-MS can also be functionalized for preliminary asteroid material analysis on board the science module.

In the end, it was decided that development of a custom utilization experimental module would be necessary, where experiments could be carried out in single, user-friendly platform—the OASIS-RUD. A 100g sample of the asteroid could be loaded into the module in a manner that would minimize risk of astronaut exposure, e.g. in a protective container. Once the sample is placed into the OASIS-RUD, all processing steps could be executed by the astronaut without further opportunity of exposure. The research and development of such a platform would be necessary for this mission. Total funding for this effort is estimated to be $100 M. The estimates for the masses, peak power, volume, and approximate cost (including R&D) are shown in the table below for the various specific components that might be included in the OASIS-RUD.

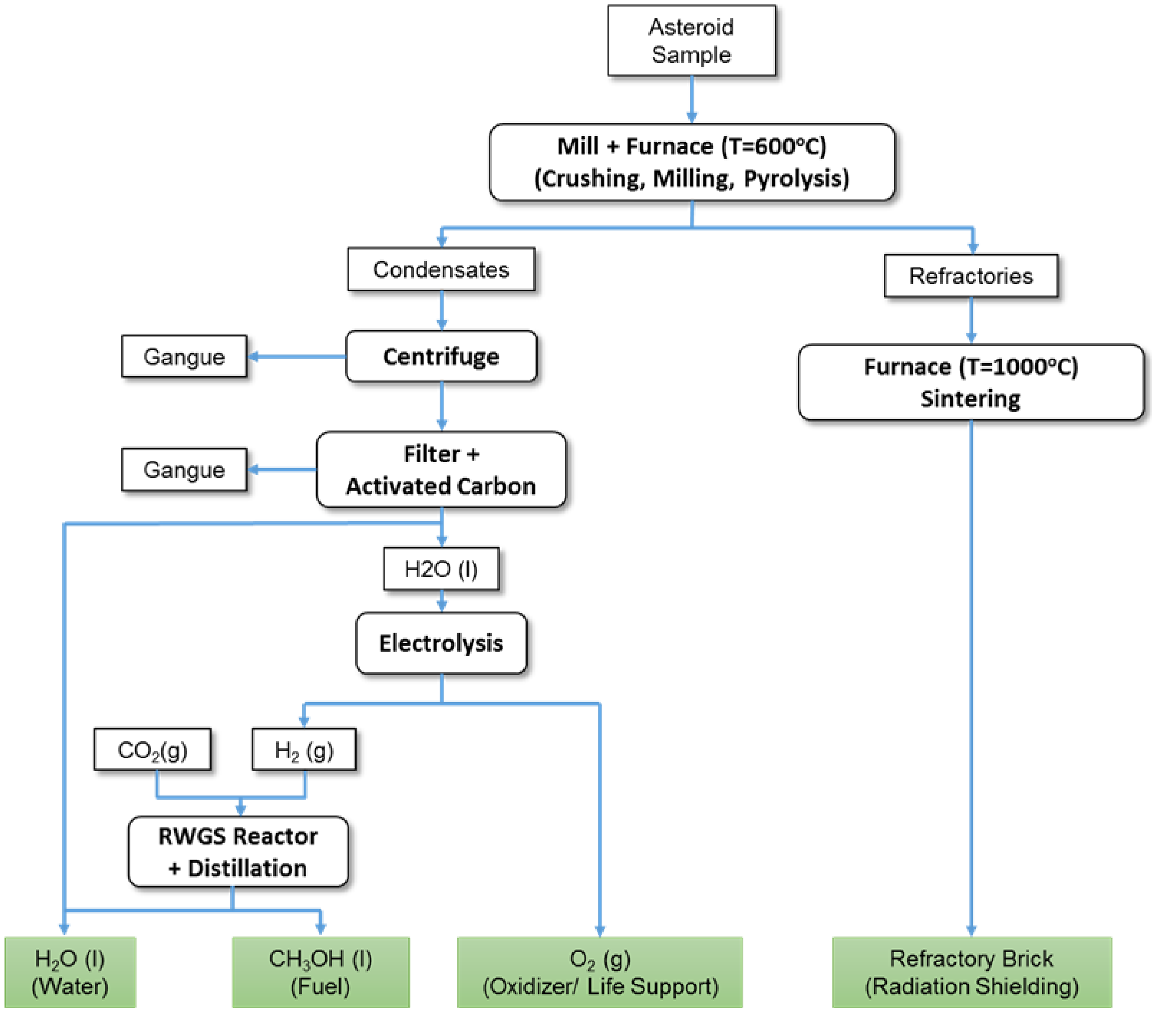


Figure 15: Flow sheet detailing the essential processing steps for the production of water, fuel, oxidizer, and radiation shielding from the asteroid

Table 13: OASIS-RUD Instrument Breakdown

Item	**Mass (kg)**	**Peak Power (W)**	**Volume (m^3)**	**Cost ($M)**
Grinder	1	100	0.004	0.5
Vacuum Heater	10	1000	0.01	1
Condenser & Centrifuge	10	150	0.005	3
Filter	1	0	0.001	0.1
Reverse Osmosis	5	10	0.001	0.5

Electrolysis	20	150	0.01	10
Oxygen Combustor	1	0	0.001	1
Condenser & Centrifuge	10	150	0.005	3
Sabatier (Methane) or RWGS + H2 (Methanol)	100	500	0.1	20
Pervaporation + Condenser + Centrifuge	30	250	0.05	4
Miscellaneous Valves, Pumps	50	50	0.05	1
TOTAL	238		0.237	44.1

Figure 16: Left-to-right: UTC Aerospace's Sabatier Reactor, Oxygen Generator/Water Processor, and a pervaporation test system by Pervatech BV

<u>Other Ideas Considered for Utilization Experiments</u>

A number of other ideas were considered for producing essential materials and products for space exploration from the asteroid material (see table with C1-type asteroid composition above). These products included:

- Iron (structural) by molten oxide electrolysis
- Silicon (solar panels) by carbothermic reduction followed by hydration and pyrolysis

These ideas were abandoned from the mission because they were not considered relevant to the aims of the mission statement. Additionally, it is thought that metal production could significantly be simplified by executing in the gravity field provided by a terrestrial body rather than in the microgravity environment present directly at the asteroid.

Human Science

Instrumentation for human science health and psychology monitoring does not involve any unique experiments. Standard life support and health systems as per the ISS will collect data that will be shared for research.

Proximity Operations

Proximity operations aboard OASIS are intended to achieve mission objectives of asteroid exploration, in-situ resource utilization, and proof of concept for longer duration human habitation in deep space.

Science and Technology Operations

Science and technology objectives of the OASIS Mission center on characterization and investigation of the asteroid. A C-type asteroid has the composition provided in Mazanek (2014). For a mass estimate of 500 metric tons and a density of 3 g/cm^3 (Brill, 2002), the resulting diameter is 3.5 m.

Table Y. The Composition Breakdown for the Asteroid (Britt et al)

Table 2.1 *Mean bulk chemical compositions of the chondrite groups (weight %)*

	carbonaceous chondrites							ordinary				enstatite	
	CI	CM	CO	CV	CK	CR	CH	H	L	LL	R	EL	EH
Si	10.5	12.9	15.9	15.6	15.1	15.3	13.3	16.9	18.5	18.9	15.8	18.6	16.7
Ti	0.042	0.058	0.078	0.098	0.13	0.11	0.017	0.060	0.063	0.062	0.05	0.058	0.045
Al	0.86	1.18	1.43	1.75	1.61	1.27	1.06	1.13	1.22	1.19	1.07	1.05	0.81
Cr	0.265	0.305	0.355	0.360	0.366	0.375	0.343	0.366	0.388	0.374	0.362	0.305	0.315
Fe	18.2	21.0	24.8	23.5	23.6	24.0	40.4	27.5	21.5	18.5	24.25	22.0	29.0
Mn	0.190	0.170	0.165	0.145	0.146	0.170	0.106	0.232	0.257	0.262	0.228	0.163	0.220
Mg	9.7	11.7	14.5	14.5	14.8	13.9	12.3	14.0	14.9	15.3	12.9	14.1	10.6
Ca	0.92	1.27	1.58	1.90	1.72	1.38	1.14	1.25	1.31	1.30	1.20	1.01	0.85
Na	0.49	0.41	0.41	0.33	0.319	0.323	0.182	0.64	0.70	0.70	0.659	0.580	0.680
K	0.056	0.040	0.035	0.031	0.029	0.030	0.021	0.078	0.083	0.079	0.068	0.074	0.080
P	0.102	0.090	0.104	0.099	0.043	0.122	~0.1	0.108	0.095	0.085	~0.07	0.117	0.200
Ni	1.07	1.20	1.40	1.34	1.27	1.36	2.45	1.60	1.20	1.02	1.44	1.30	1.75
Co	0.051	0.058	0.069	0.066	0.064	0.067	0.115	0.081	0.059	0.049	0.070	0.067	0.084
S	5.9	3.3	2.0	2.2	1.58	1.31	0.25	2.0	2.2	2.3	4.07	3.3	5.8
H_2O	18.0	12.6	0.6	2.5	0.8	5.7	0.1	–	–	–	–	–	–
C	3.2	2.2	0.45	0.56	~0.1	1.44	~0.8	0.11	0.09	0.12	0.058	0.36	0.40
O	30.0	32.0	36.5	34.8	35.5	31.2	25.6	35.7	37.7	40.0	37.3	31.0	28.0
Fe^0/Fe^{tot}	0.0	0.0	0.1	0.1	0.0	0.43	0.85	0.6	0.3	0.15	0.0	0.74	0.65
Ir(ppb)	460	595	735	760	767	642	1030	760	490	360	614	525	565
Au(ppb)	144	165	184	144	136	139	202	215	162	140	183	225	330

CI, CM, CO, CV, H, L, LL, EH and EL groups, data from Wasson and Kallemeyn (1988).
CK group, mainly from Kallemeyn *et al.* (1991). H_2O and O from Mason and Wiik (1992a); their Si/Mg ratio was used with 14.8 wt% Mg to calculate the Si content.
CR group, mainly from Kallemeyn *et al.* (1994). H_2O, C and O from Mason and Wiik (1992b); their Si/Mg ratio was used with 13.9 wt% Mg to calculate the Si content.
CH group: Analysis of Allan Hills 85085 (Wasson and Kallemeyn, 1990). The Si/Mg ratio in bulk silicate (Scott, 1988) was used with 12.3 wt% Mg to calculate the Si content. Carbon content: Grady and Pillinger (1993).
R group: Schultze *et al.* (1994). The Si/Mg ratio from the composition used by Bischoff *et al.* (1994, p. 273) to calculate production rates of cosmogenic nuclides was used with 12.9 wt% Mg to calculate the Si content.

Table 14: Outline of science operations along with duration and projected timeline

DAY	DURATION	OVERVIEW
1	3	Surface Access Preparation (3 days) (only for NASA option A selection)
5	3	Preliminary robotic survey of asteroid structure and environment
9	1	EVA #1 Deep drilling and core sample
11	1	EVA #2 Deep drilling and core sample
13	1	EVA #3 Core sample

15	1	EVA #4 Core sample
17	1	EVA #5 Core sample
19	1	EVA #6 Core sample
21	1	EVA #7 EVA with simulated Communications delay
22	1	Mission Closure Checkout

Survey of structural integrity and space environment parameters (DAY 1 - 3)
This initial and highly important survey will be performed after the docking of the Orion to the MPDM, ensuring that no more major disruptions to the asteroid occur. The intent is to characterize the space environment around the asteroid (radiation, cosmic rays, dust, etc.) and the integrity of the asteroid surface via small scale sampling (or indenting), to ensure the absolute safety of the crew. It will be a gating procedure, determining whether the first EVA will occur.

Since the safety of the crew is of the utmost importance, it will be performed robotically using the CanadarmX. The arm will place two SEnV modules as far apart on the asteroid surface as possible, and the suite will begin to record plasma, radiation, and dust concentrations while performing a small (10 cm) drill or indentation test. In addition, one SEnV will contain a seismic source while the other (on the opposite side of the asteroid) contains a receiver. A single pulse will be sourced and recorded by these instruments, the traveltime and magnitude of the arriving wave will indicate coherency of the asteroid.

Characterization and ISRU
EVA #1:
The first EVA will involve the drilling of one deep drill test, to a total of 5 m. In the upper 1 m, a core sample of 3 cm radius and 30 cm length will be acquired, and the drill will be anchored using JPL microspine technology in the case of competent substrate and harpoon anchoring in loose substrate.

Following completion of the borehole, the drill bit will be removed and the SDCAU suite placed in the borehole for the compositional analysis. The tool contains a Near-IR Spectrometer and an XRF for thermal, chemical, elemental, and water form analysis, a Gamma Ray Spectrometer for lithology determination, and a Neutron Spectrometer for water content characterization.

Once the borehole has been successfully logged, an additional seismometer and SEnV unit are placed, and detailed photo documentation occurs.

EVA #2:
The second EVA will be a repeat of the first, only at a new location, preferably as far from the first as possible.

EVA #3-7:

Additional EVAs will involve shallow (~1 m) core sample collection (up to 3 kg) at various places around the asteroid, preferably spaced in order to sample as much of the surface variation as possible.

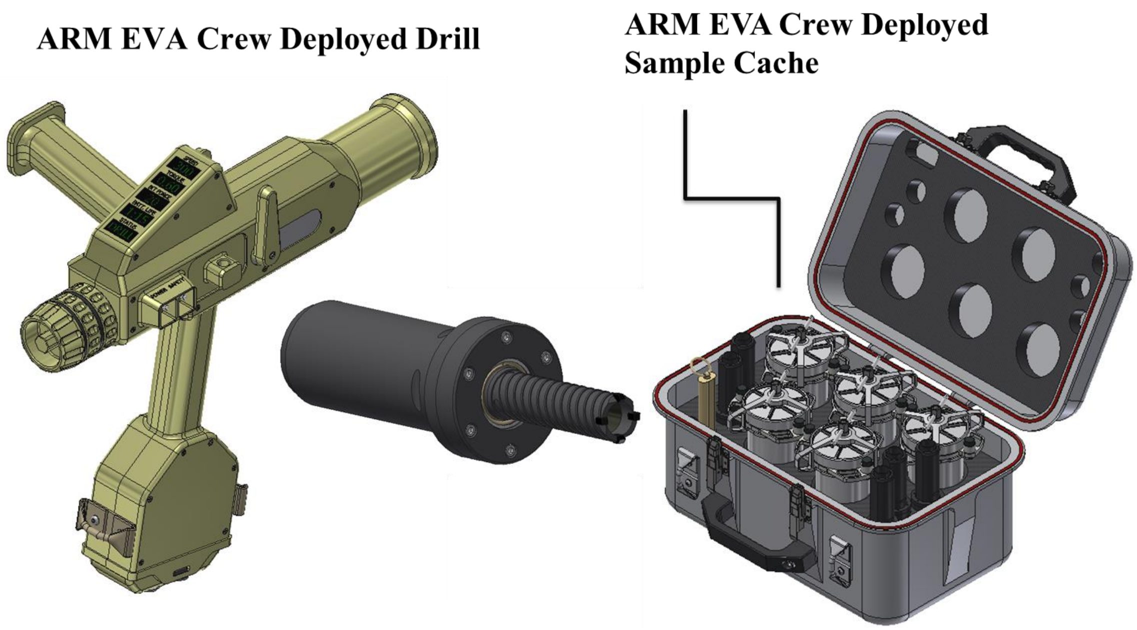


Figure P. Proposed EVA Drill Tool (courtesy of HoneyBee Inc)

Daily monitoring of the physical health of astronauts will occur per NASA-STD-3001, VOLUME 1, Revision A w/Change 1 (2015). In addition to tracking the health and well-being of each astronaut on an individual basis, this data is intended to contribute to our understanding of human physiology and health in deep space and advance the fields of bioastronautics and space medicine (Williams, 2011). Analysis of the data collected by Crew Health Care System (CHeCS)/Integrated Medical System will be looked at in the aggregate to track biomarkers over the duration of the mission, as has been done for the ISS (https://www.nasa.gov/mission_pages/station/research/experiments/1025.html).

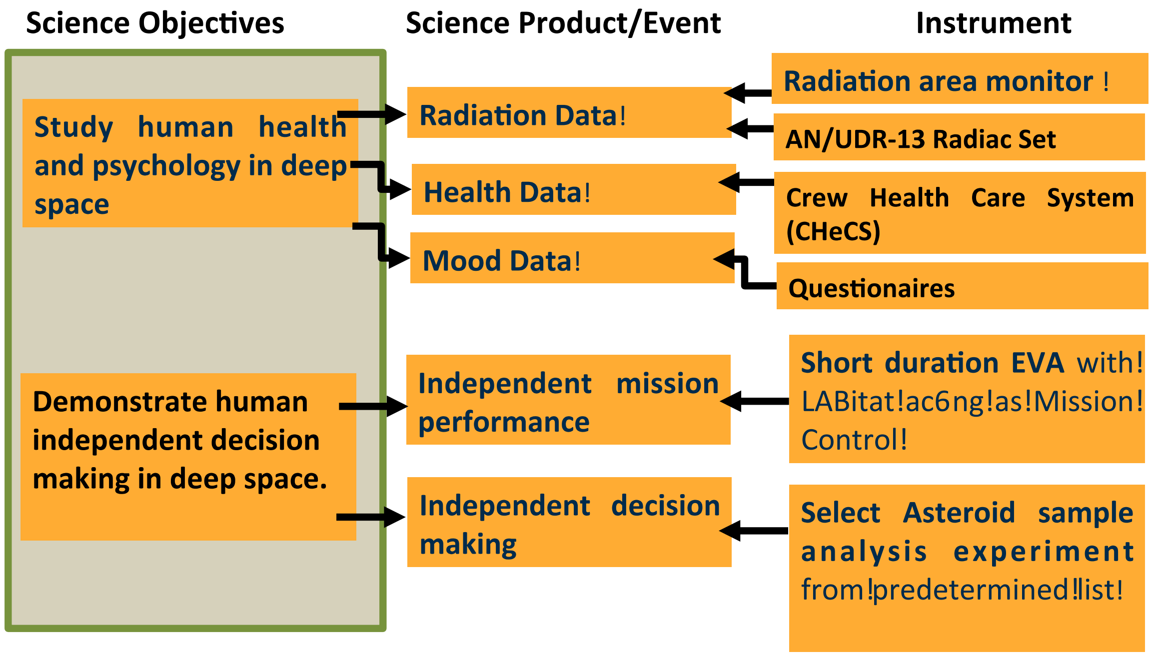


Figure 17: Flow chart indicating human science objective flow into products and instrumentation

From this data, health researchers will be able to identify risks and/or benefits to human health from exposure to deep space for a duration of at least 14 days. Especially of interest will be the comparison of health markers in deep space compared to the well-studied evolution of identical health markers in different individuals aboard the International Space Station.

Weekly mood questionnaires facilitate similar data collection for psychological and behavioral monitoring and study (Kelly, 1992). The effect of the small size of the habitat (e.g. compared to the ISS) on perceived well-being is a particular area deserving of study. As humans move toward longer duration space missions, incremental increases in length of mission duration provide important observational milestones, as we do not yet understand the long term implications of isolation and confinement in a small group space. Video recordings of crew interactions will be archived for later study, particularly of group dynamics.

Equally important to establishing psychological and physiological baselines for deep space missions is the development of trust in humans as independent actors in space through a series of low-risk incremental scenarios. Currently, crew actions in space are highly scripted and closely monitored by Mission Control (http://www.esa.int/Our_Activities/Human_Spaceflight/Astronauts/Daily_life). However, communication delays continue to increase with travel distance from Earth and at some point become incompatible with effective real time communication. In order to prepare for this eventuality, large future risk can be mitigated at the present time with a series of lower-risk activities designed to train the crew to operate in a reduced-communication environment.

One important aspect of mitigating risk is establishing proof of rational decision making in such an environment. A low-risk first step toward this goal is requiring a crew member to make a selection from a predetermined list. In this mission, a crew member will need to select an experiment from a predetermined list of previously successful experiments to carry out on a sample obtained from the asteroid. Successful completion of the experiment will serve as a positive indicator of the ability of crew to make independent decisions.

Figure 18: Astronaut aboard the ISS monitored by a CHeCS subsystem while exercising. Photo courtesy of NASA

Crew will continue establishing low-risk proof of independent operations capability by carrying out an EVA of limited duration (e.g. 2 hours) with simulated communication separation from Earth Mission Control. With Earth Mission Control in a listen-only mode, the OASIS module will act as Mission Control for two crew members carrying out a predetermined, straight forward EVA. This would be the longest time humans went without communication with Earth in human history, a significant achievement made in a controlled and relatively low-risk manner. To lower risk, in the event of an anomaly or emergency, Earth Mission Control can resume two-way communication and resolve the incident in the manner applicable to a standard EVA setting. Historically, Apollo astronauts spent 45 minutes in a communications break out on the dark side of the moon (NASA EP-66).

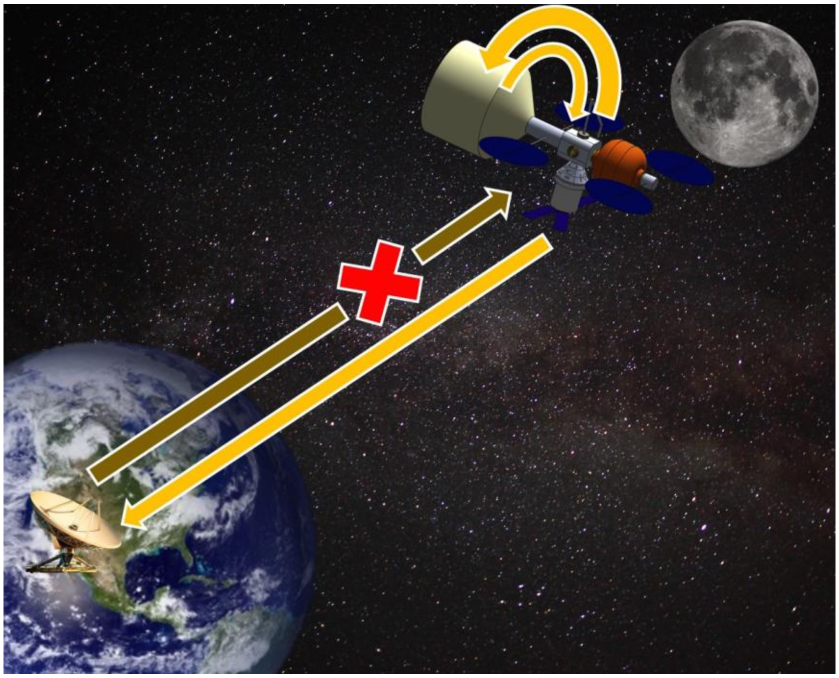


Figure 19: Schematic representation of communication structure during independent operations capability EVA

The Oasis module operates as Mission Control with local two way communication to the crew members on EVA. Communication with Earth Mission Control is one way for the short duration, with the ability to resume full communication in the event of any anomaly. Taken together, these objectives represent significant advances in human knowledge and autonomy along an incremental path toward independent human operations in deep space.

Engineering

Asteroid Location

The problem statement specifies a distant lunar retrograde orbit (DRO) with a mean orbital radius of 61500km. This orbit is substantially smaller than orbits typically discussed in the literature, passing within the L1 and L2 Earth Moon Lagrange points. This orbit is 30% faster than the usual case, requiring fresh calculations of insertion and departure Δv. Orbital optimization calculations were performed with two custom designed software packages to ensure that the design orbit met the requirement of 100 years of passive stability. The orbit is in the plane of the Earth Moon system, and its orbital periapsis is 56,550km from the Moon.

Launch

1. Vehicle Selection

In order to accommodate sending the mass of the Orion capsule, habitat, and assorted subsystems into the Distant Retrograde Orbit, more than one launch will be required. It is understood that use of the Orion capsule requires at least one launch vehicle to be NASA's SLS launch vehicle, of which the Block 1B was selected for its increased mass capability. A trade study was conducted for selection of a second vehicle to transport the remaining payload. Comparisons of Δv to payload mass capability for varying launch vehicles were made and can be found in Figure 1.

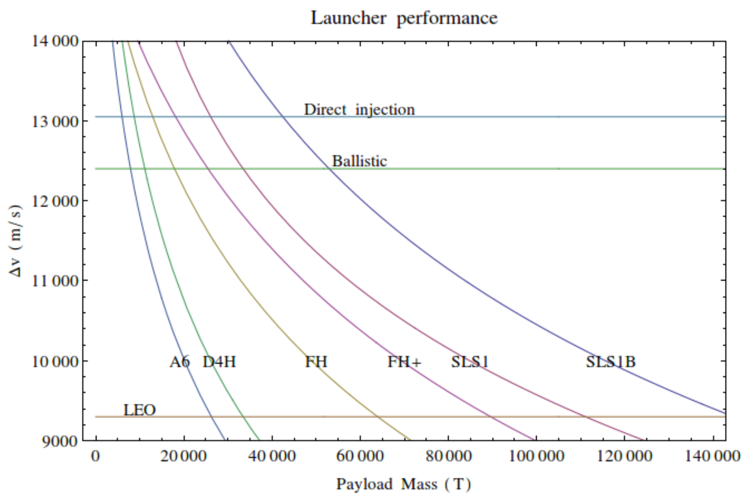


Figure 20: Launch Vehicle Performance Comparisons

Falcon Heavy capability was estimated taking into account recently announced F9 core upgrades. The size of the second stage had to be estimated assuming a nominal mission to GTO, subject to launch thrust constraints. Falcon Heavy upgrade (FH+) is an alternate vacuum stage that utilizes the high Isp (383s) Raptor engine and methalox fuel. SLS and the other remaining launch vehicle capacities were estimated based on publicly available shuttle parameters as seen in Appendix A. From this, the Falcon Heavy was selected as the second launch vehicle due to the low mass capability of the Arianne 6 and the Delta IV Heavy and the cost and launch restrictions of the SLS1 and SLS1B.

2. Launch Operations

The launch windows for the mission depend on the type of transfer orbit to the lunar DRO. For the case of a flyby outbound trajectory the launch windows are defined by the phasing in the DRO. The period of the DRO is 10.5 days, resulting in an optimum launch opportunity every time the spacecraft completes one revolution on the DRO. The ballistic trajectory depends on the Sun-Earth-Moon system position, which repeats every ~28 days. Delays in the launch date can be corrected during the trajectory with additional Δv expenses. There exists a linear relation between the launch date delay and the additional Δv required for the maneuver. Belbruno and Carrico (2000) estimate the extra Δv requirements in 15 m/s per day from nominal launch.

Launch opportunities to ballistic orbits occur on a monthly basis (Belbruno and Carrico, 2000). The estimated launch date is March 30th, 2024, which is the time of closest approach obtained in the simulation of the ballistic trajectory with the full force model that was fully developed using MATLAB and FORTRAN. From this program, the nominal arrival date for the cargo was found to be July 8th. In order to anticipate possible launch delays for the cargo, the crew launch is scheduled for September 10th, 2024. This allows for a total of two failed launch attempts before having to reschedule the SLS launch. The crew arrival date is expected for September 18th, 2024. Launch window estimates in relation to the Moon's inclination can be found in Figure 2.

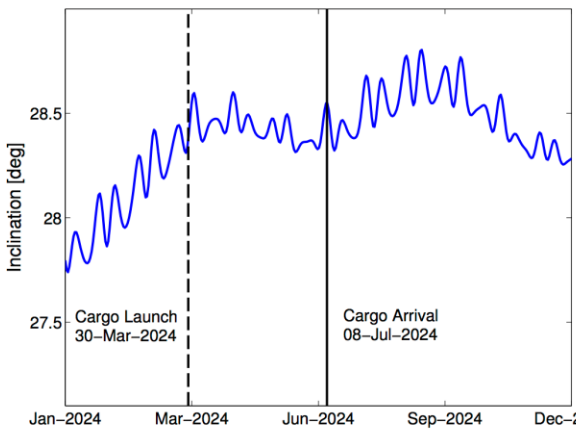


Figure 21: Inclination of the moon and launch schedule

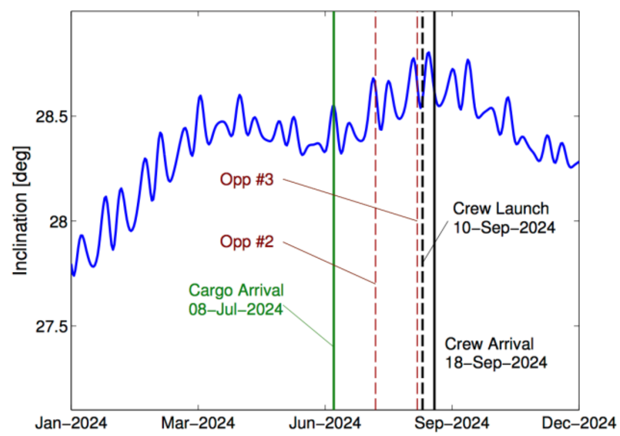


Figure 22: Crew launch

In order to minimize Δv requirements for inclination change, both missions will be inserted into a ~28 degree inclination orbit towards the Moon with SLS launching from Kennedy Space Center, and the Falcon Heavy launching from SpaceX's commercial spaceport in Boca Chica Beach, Texas due to each facility being designed for the respective launch vehicle. The periodic changes in the inclination of the Moon are less than 0.5 degrees during the transfer trajectory. Such variations may require corrections of about 10 m/s on the trajectory.

3. Launch Configuration

3.1. First Launch

The first launch with the Falcon Heavy will bring the inflatable habitat, a propulsion module, solar panels, radiators, arm, airlock and communication equipment in a ballistic trajectory to the DRO of the asteroid. The Falcon Heavy can bring 18.5T of payload mass in a ballistic transfer orbit to a DRO. With a mass of 18.5T for the above mentioned payload the, Falcon Heavy is able to perform the injection into the ballistic transfer orbit.

3.2. Second Launch

The second launch with the SLS Block 1B will bring the Orion together with the node and the science rack into an indirect flyby transfer orbit to the DRO of the asteroid. A trade was conducted in order to effectively consider the possibility of solely using the Orion Service Module's propulsion system to propel both the Orion capsule and the additional payload into the DRO. Unfortunately, using the payload initial mass of 34.5T, the Orion Service Module's propulsion Isp of 316s, and the required Δv's for injection, the Δv required to insert the full SLS payload results in a remaining Δv capability of approximately 350 m/s. This value will be later shown to be close to half of the required Δv for even the most optimal return trajectory of Orion.

In order to successfully avoid this dilemma, a propulsion system of 18 Draco thrusters from SpaceX's Dragon capsule was attached to the node with 3T of propellant. The SLS Block 1B is capable of injecting 37.8T of payload into a transfer orbit to the DRO of the asteroid, however inclusion of the Draco propulsion system and required propellant only increases the SLS total payload mass to 36.5T. The Draco thrusters will be used to perform the powered flyby maneuver, as they provide a Δv of 252 m/s, which is the order of magnitude of Δv that is needed for the powered flyby. Due to the 3T of Draco propellant being used for the powered flyby maneuver, the Orion capsule's propulsion is still needed during approach. As is such, prior to insertion into the flyby trajectory, the launch configuration performs an Apollo-like docking maneuver with the node undocking flipping around and docking again to the Orion capsule so that the Orion's propulsion system can be used. Nevertheless, despite using Orion's fuel during the DRO approach, approximately 750 m/s of Δv remains for the return of Orion to Earth.

Transit

1. Transfers to Lunar Distant Retrograde Orbits

The mission is divided into two main phases from the perspective of the trajectory design: the cargo phase, and the crewed phase. The main constraint for the crewed phase is the time of flight, driven by the life support capabilities of the Orion capsule. For the cargo phase, however, alternative transfer methods which minimize the-Δv budget are explored.

1.1. Cargo Phase

1.1.1. Ballistic trajectories

Parker et al. (2015) located ballistic transfer trajectories to DROs in the Earth-Moon system, considering of the gravitational perturbations from the major bodies in the Solar System. Candidate orbits are those which remain stable for 100 years when propagated forward in time, and depart from the DRO towards the Earth when propagated backwards. The interior ballistic transfer originates from an Earth centered elliptic orbit with an apogee of 330,000 km. After a series of flybys about the Moon, the spacecraft enters the DRO without additional insertion maneuvers. The exterior ballistic transfer starts with an orbit arc which approaches L1 in the Earth-Sun system. The probe then returns towards the Earth and encounters the Moon. After decreasing its energy with a single lunar flyby, the spacecraft enters the DRO. Belbruno and Carrico (2000) constructed an exterior ballistic trajectory which departs from a 300 km LEO orbit, and requires a finite burn of 3,160 m/s to injection. Both the HITEN and GRAIL missions have successfully verified the use of a ballistic trajectory into lunar orbit, though they were designed to exploit the properties of exterior ballistic trajectories. While either transfer has roughly equivalent Δv requirements, interior ballistic DRO injections are preferred to minimize the distance to the Earth.

In order to simulate the dynamics of the ballistic trajectories, a set of Monte Carlo simulations were created that included gravitational perturbations from the DE430 JPL ephemerides: all the planets, the Sun, the Moon, Pluto, and the four major asteroids (Ceres, Vesta, Pallas, and Hygiea), and can be seen in Figure 3.

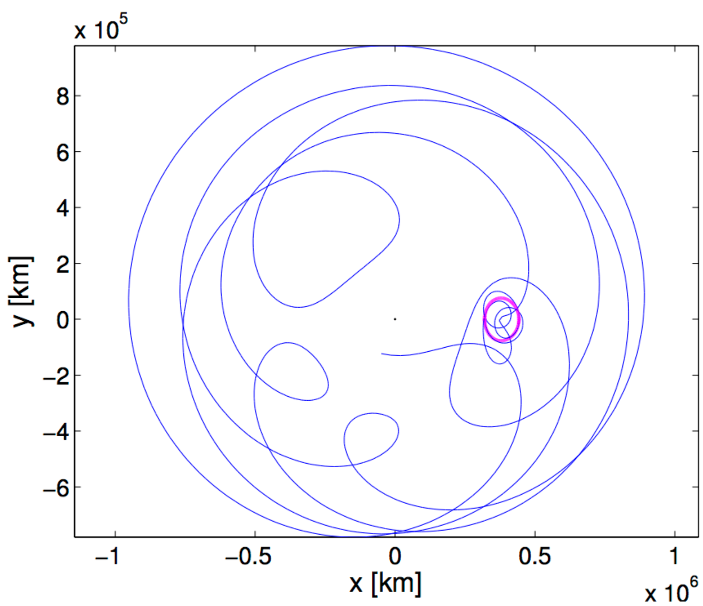


Figure 23: Ballistic trajectory computed from full ephemeris model

The pink line sketches the ideal DRO computed from the CRTBP. The minimum distance to the Earth is 85,000 km. Date of closest approach is 30-Mar-2024

1.1.2. <u>Low-thrust</u>

In contrast to the ballistic trajectory, solar electric propulsion (SEP) systems were considered for cargo transfer. A SEP derived from the ARRM stage was considered in order to take advantage of its high technology readiness level (TRL). Such a system, producing 5N of continuous thrust from 50kW of solar power would consume 4.5T of Xenon propellent in 100 days of use. The remaining mass was estimated at 1.5T (tank + solar panel + electrical switching + bus), for a total booster mass of 6T. With Falcon Heavy capable of delivering 18.5T to a ballistic insertion orbit, the tradeoff point occurs if SEP can deliver 24.5T. Falcon Heavy can deliver 11700m/s to 24.5T, a deficit of 700m/s compared to the baseline ballistic orbit. The SEP stage can deliver 3000m/s of Δv over 100 days. As it is unable to exploit the Oberth effect, SEP would be required to deliver roughly 3300m/s of Δv to achieve DRO insertion. SEP was thus deemed uncompetitive in this instance. For larger payload masses, longer time scales and/or more powerful SEP systems, continuous propulsion is a viable proposition. Transfers to lunar orbits via low-thrust trajectories take about 1.5 yrs.

1.1.3. Orbit Selection

Ballistic trajectories allow for a significant reduction in-the total Δv required for the transfer trajectory, being limited to the injection total Δv the launcher upper stage can provide. If the total mass to be delivered exceeds the launcher capabilities, low thrust trajectories may be considered. For the considered mission the required payload mass can be injected into a ballistic trajectory by the upper stage of Falcon Heavy, therefore low-thrust electric propulsion is discarded due to its cost and longer mission duration. The optimized values from Belbruno and Carrico (2000) are considered for designing the baseline trajectory.

1.2. Crewed Phase

1.2.1. Free-return trajectory

A transfer trajectory via a free-return trajectory to the Moon is explored, as can be seen in Figure 4. The spacecraft departs from a 463 km circular orbit and is injected into a free-return trajectory to the Moon, which requires 3,093 m/s. The insertion to the DRO decomposes in two finite burns maneuvers. First, a negative Δv maneuver of 233 m/s reduces the velocity close to the escape value. The final Δv required for insertion into the DRO is 447 m/s. Free-return trajectories were exploited by Apollo (in particular Apollo 13) for post trans lunar injection (TLI) safety.

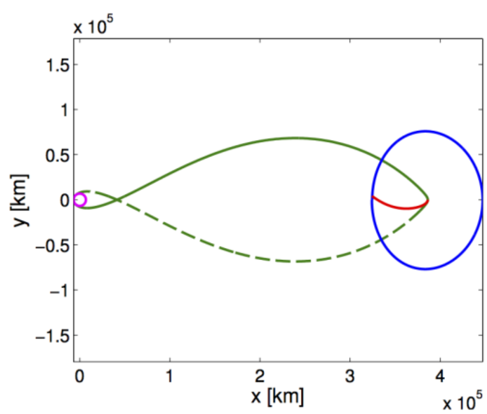


Figure 24: Free-return transfer trajectory to a lunar DRO

1.2.2. Flyby outbound trajectory

Variations in the imparted Δv lead to higher flyby distances about the Moon, higher flight times and lower Δv requirements. In particular, Condon and Williams (2014) report an optimal indirect transfer trajectory that requires 2859 m/s of Δv for an insertion from a 1806 x 40.7 km Earth orbit. After 6.5 days the spacecraft performs a 241 m/s maneuver, which leads to a lunar flyby to reduce the energy of the probe. A final burn of 242 m/s is required to complete the insertion into the DRO. The total rendezvous and phasing Δv budget is estimated to be ~11 m/s. The escape trajectory (also a lunar flyby) is divided in two impulses of 146 m/s and 497 m/s, respectively. The total time until insertion is about 8.5 days. Parker et al. (2015) consider a more optimistic value for the insertion maneuver, with a Δv of 350 m/s, however the Δv of 480 m/s is used in order to produce conservative results.

1.2.3. Direct outbound trajectory

The simplest way to arrive to the DRO consists in designing a geocentric ellipse with apoapsis past the orbit of the moon,-such that the spacecraft is in the limit of the sphere of influence of the Moon. At this point, a single burn of about 600 m/s is applied for injection into the DRO. Such orbits are constructed by increasing the amount of Δv imparted for the injection. A summary of the Δv requirements for the three trajectories can be seen in Table 1.

Table 15: Time of flight and Δv budget for transfers to DRO from a 185-300 km LEO orbit

Orbit type (origin)	ΔV [m/s]	Time of flight [days]
Direct transfer (200 km)	3700	~6.5
Flyby outbound trajectory (185x1806 km starting orbit)	3370 (of which 480 for flyby and insertion)	~8.5
Free-return	3750 (of which 600 for flyby and insertion)	~4.5
Ballistic transfer (300 km)	~3100 (inserting LEO)	~100

1.2.4. Orbit Selection

Due to constraints on the total available Δv, the direct transfer and the free-return trajectories are discarded. The flyby outbound trajectory described by Condon and Williams (2014) is reproduced to optimize the total Δv budget of the transfer maneuver. The return trajectories are analyzed in detail further in this paper.

2. Lunar Distant Retrograde Orbits

The problem statement specifies a distant retrograde orbit with a *mean* radius of 61500km. This is substantially (~10%) smaller than typical DROs referenced in literature, necessitating new calculations. The following figure shows the family of DROs in the range C[2.92,2.95], where C is the Jacobian constant:

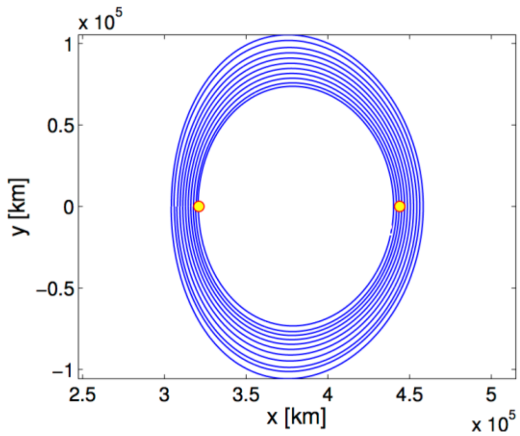


Figure 25: Family of periodic DROs for C ∈ [2.92, 2.95] in the synodic reference frame

Calculations were performed with a 4th order symplectic integrator and restricted 3-body interaction model. In particular, the periapsis is at 56,550km, within the L1 distance of 61,300km. The resulting orbital period is 10.55 days, and ΔV requirements for direct insertion (with or without flyby) are higher. Tangential orbital velocity in a non-rotating frame is 1369km/s, and 351m/s in a co-rotating frame.

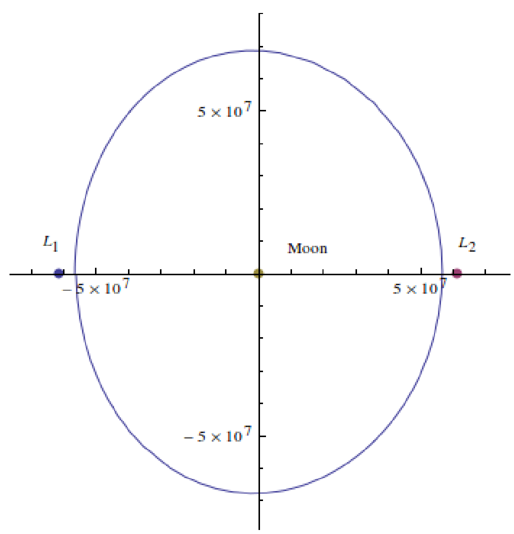


Figure 26: Target DRO

The stability of this orbit was verified for over 70 years, but eventually entered a region of chaotic 3 body transfers. The following figure shows a corotating frame of 100 years of orbits - the last 30 become chaotic. It is important to note that despite the orbit becoming chaotic, it never reaches Earth.

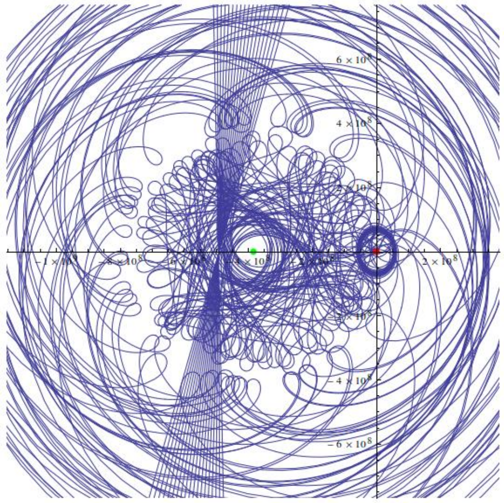


Figure 27: Co-rotating frame of 100 years of orbits - the last 30 become chaotic

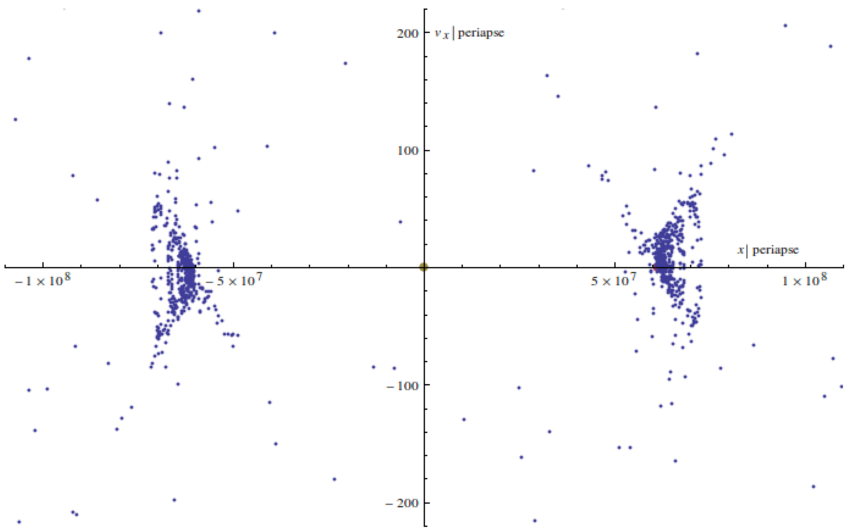


Figure 28: Poincare phase recurrence diagram

The parameter space was then mapped to reproduce Poincare phase recurrence diagram, and can be seen below.

3. Delta V Summary

A full breakdown of payload components, types of propellant, Δv distributions, and their justifications for the launch vehicles and payload propulsion systems can be seen in the table below.

	Falcon Heavy	**SLS**	**Justification**
Earth to LEO (LV)	9.3 km/s	9.3 km/s	Understood requirement

LEO to DRO transfer (LV)	3.1 km/s	3.37 km/s	FH: Ballistic SLS: Free-Return-Flyby
Total LV Δv	12.4 km/s	12.67 km/s	FH: Justification for 18.5T Payload SLS: Justification for 37.8T Payload
Payload Propulsion	MMH/N_2O_4	Orion & Draco system: MMH/N_2O_4	High Isp, reliable, and storable.
Payload Prop. Max Δv	.150 km/s	Orion: 1.1 km/s Draco: .252 km/s	FH: could need a lot less if only being used for corrections and docking Orion: Assuming additional prop in service module Draco: Resulting Δv from including 3T of propellant
Payload Insertion correction	0.02 km/s	0.02 km/s	Due to utilizing currently unproven systems, these numbers were used to be conservative
Targeting, Rendezvous, Orbit trim maneuvers	0.015 km/s	0.015 km/s	A very small amount of Δv is required in order to maintain proper trajectory during transit.
Insertion	0 km/s	.480 km/s	FH: Ballistic trajectory Orion: Flyby-injection,without free return
Phasing	0.05 km/s	0.05 km/s	Assuming launch window estimates are correct and do not require additional phasing
DRO Orbit Corrections	0 km/s	0 km/s	The DRO is assumed to be stable, else the ARM propulsion is able to be used to maintain the DRO.
Departure	0 km/s	.750 km/s	FH: Not returning. Orion: Minimum requirement to return. In abort case scenario, crew will inhabit orion and use Orion's remaining ECLSS capabilities until a
Total Payload Δv Required	0.04 km/s	1.270 km/s	FH: Extra Δv was allocated to ensure arrival of the cargo. Orion + Draco: The draco additional propulsion allows for the safe return of the crew with additional fuel for a 10% margin.

4. Orbit Anomaly Considerations

4.1. <u>Free-Return Trajectory</u>

The free-return trajectory allows for simple abort options. In the case that the main engine fails to complete the first stage of the insertion maneuver, the spacecraft will return to the Earth through a ballistic trajectory. While the dynamics of the system guarantees the safety of the crew, the free-return trajectories require a Δv-that exceeds the present capabilities of the mission.

4.2.Abort Options During Transit

Williams, 2014, closely studied the abort options for the trajectory used for this mission. Abort options are considered for critical Orion-maneuvers during the mission:

- failed outbound flyby
- failed DRO insertion

A failed maneuver indicates that no part of the maneuver was performed by the main engine. In the case of a main engine failure, the set of auxiliary (AUX) engines is used to return the Orion spacecraft to Earth in minimum time using the available propellant remaining. It is found that the AUX engine set has the performance needed to successfully complete the abort contingency mission for each of the cases studied.

In any case before aborting the mission, the Orion capsule has to undock from the inflatable habitat to provide enough Δv to return safely back to Earth. In case of a failed outbound flyby, Orion returns with two burns of the AUX engines back to Earth. The first burn takes place 15 minutes after the failed outbound flyby, which is enough time to relocate all of the astronauts back into the Orion capsule and undock from the inflatable habitat. The second burn positions Orion on a trajectory that will return the crew to Earth in 11.51 days via a moon flyby back to Earth.

If the DRO insertion fails, two AUX maneuvers are required to abort the mission. The first AUX maneuvers takes place 2.67 days after the failed DRO insertion. The second AUX maneuver is a powered moon flyby that brings Orion back to Earth after 18.32 days. All of these scenarios will successfully return the crew back to Earth, as they will be utilizing Orion's ECLSS system for survival.

4.3.Abort options while in the DRO

4.3.1. Direct Return

The parametric analysis on the return trajectories from the DRO allows for determination of the possible abort scenarios. A window of minimum Δv for the return trajectory has been located, which spans 21 hours.

Due to restrictions on the Δv remaining after insertion and rendezvous, certain locations in the DRO do not allow for a direct return trajectory to be performed immediately. To counteract this, in the case that a direct return trajectory need to be pursued during a period where the required Δv is more than remains, the Orion capsule will undock from the habitat, and remain in the DRO until a direct return can be performed. The calculated direct return trajectories can be found in the figure below.

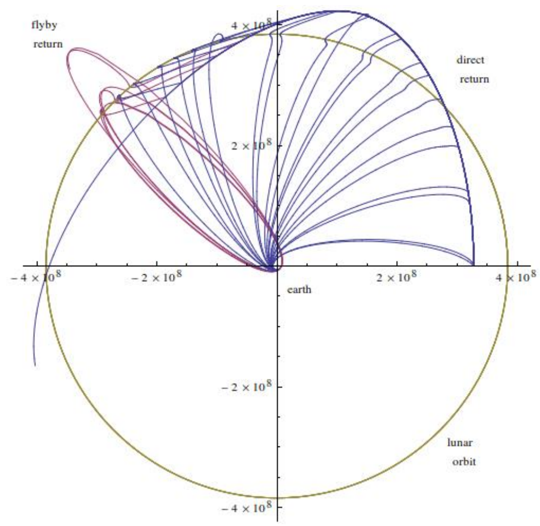


Figure 29: Direct return trajectories in a non-rotating frame

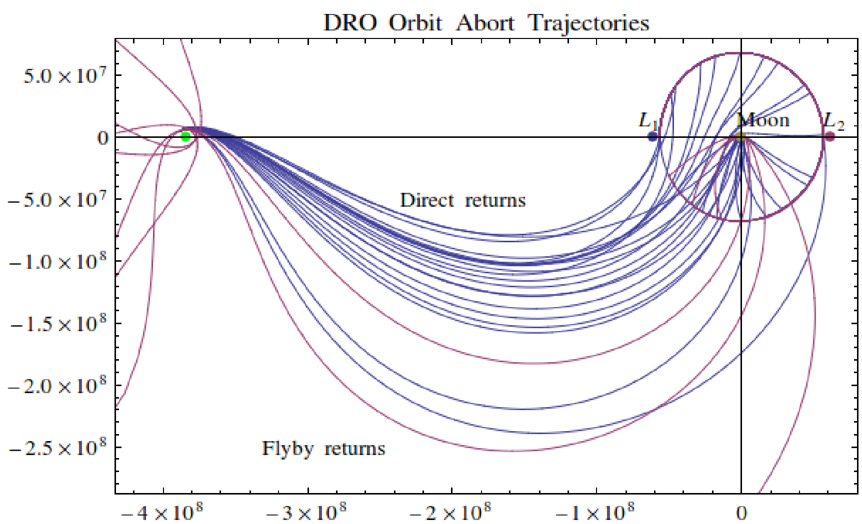


Figure 30: Direct return trajectories in a rotating frame

4.3.2. **<u>Lunar Flyby</u>**

Lunar flyby orbits were computed from 15 evenly spaced points in the DRO. Roughly 25% (corresponding to days 8-10.5 of the DRO) of them are useful for powered flybys to get back to Earth. These scenarios will be treated similar to that of the direct return trajectories, where the Orion capsule will detach from the habitat and wait in the DRO until return is possible. Fortunately, the Orion capsule will have enough remaining ECLSS to allow for the safe return of the crew.

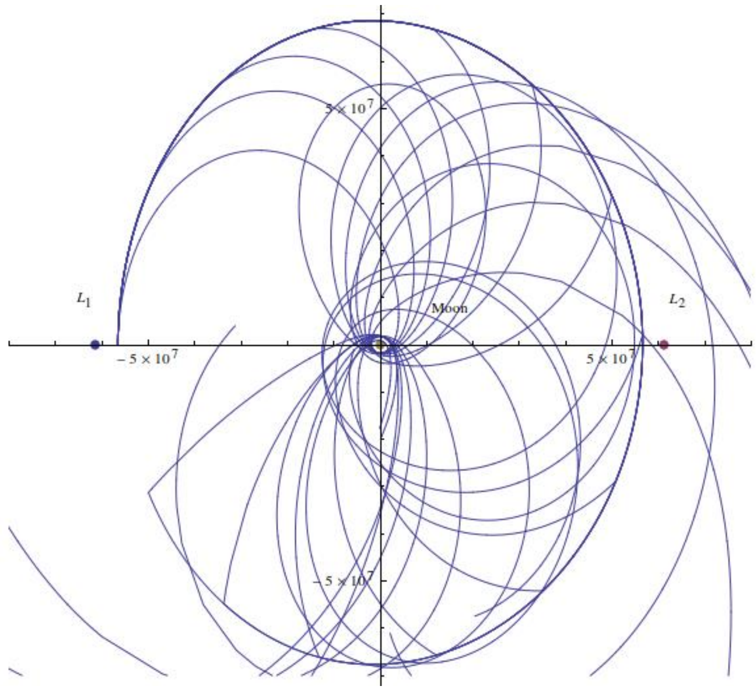


Figure 31: Lunar flyby orbits from 15 evenly spaced points in the DRO

Blue direct return orbits as seen in Figure 10 below, were found with MCMC optimization. Similarly, red powered flyby orbits were found with two parameter linear optimization. Brown direct return orbits were computed analytically using Mathematica, illustrating a limitation of the MCMC procedure, in which a local minima precluded automated discovery of the optimal tangential direct return trajectory. Given an abort Δv budget of 750m/s, immediate aborts are available for 60% of the DRO without modifying the Orion mass profile.

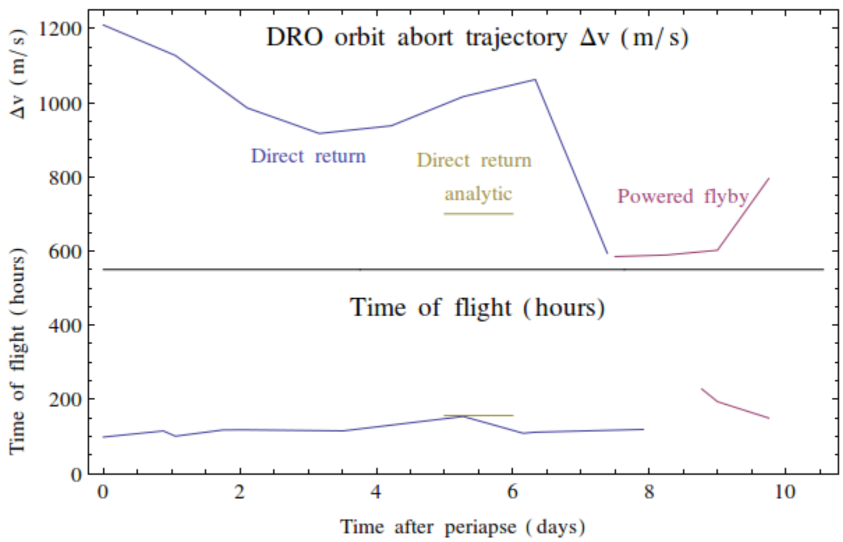


Figure 32: Δv requirements for mission abort at any phase of the DRO

The following scheme shows the failure flow diagram for the maneuver.

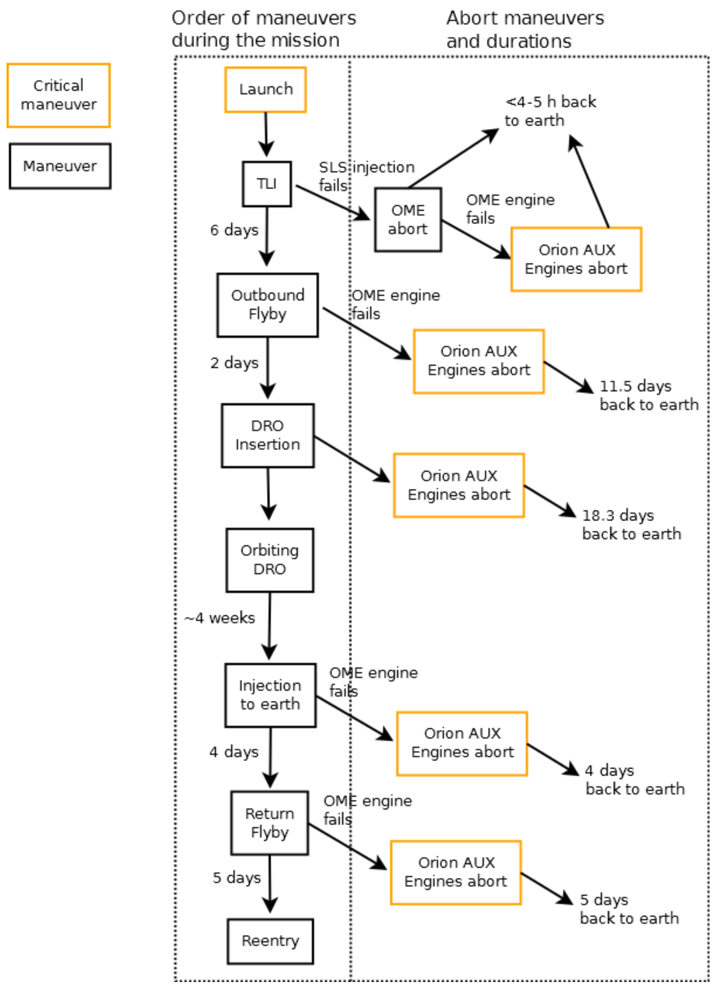


Figure 33: Overview of the different abort scenarios considered

Entry/Reentry, Descent, Landing

5. Entry/Reentry, Descent, Landing

To approximate a possible reentry trajectory for the Orion capsule, the tool ASTOS from Astos Solutions was used. ASTOS is a software package to simulate and optimize launch and reentry trajectories. There is no official data available for the aerodynamics of Orion, therefore so Apollo data for drag and lift coefficients (Ernest J. Hillje, 1969) were taken due to Apollo’s similar shape.

Multiple reentry trajectories were found depending on the initial conditions (reentry angle/perigee altitude at Earth). The figures below show the results for the reentry trajectory with a reentry angle of 5°. This trajectory results in a maximum deceleration factor of 7.4 g and a re-entry time of approximately 24 minutes.

Return launch windows are set by position of the oceans in Earth and location within the DRO. Due to an order of magnitude difference in the two, return window selection may readily assure a water landing, as can be seen in Figure 346.

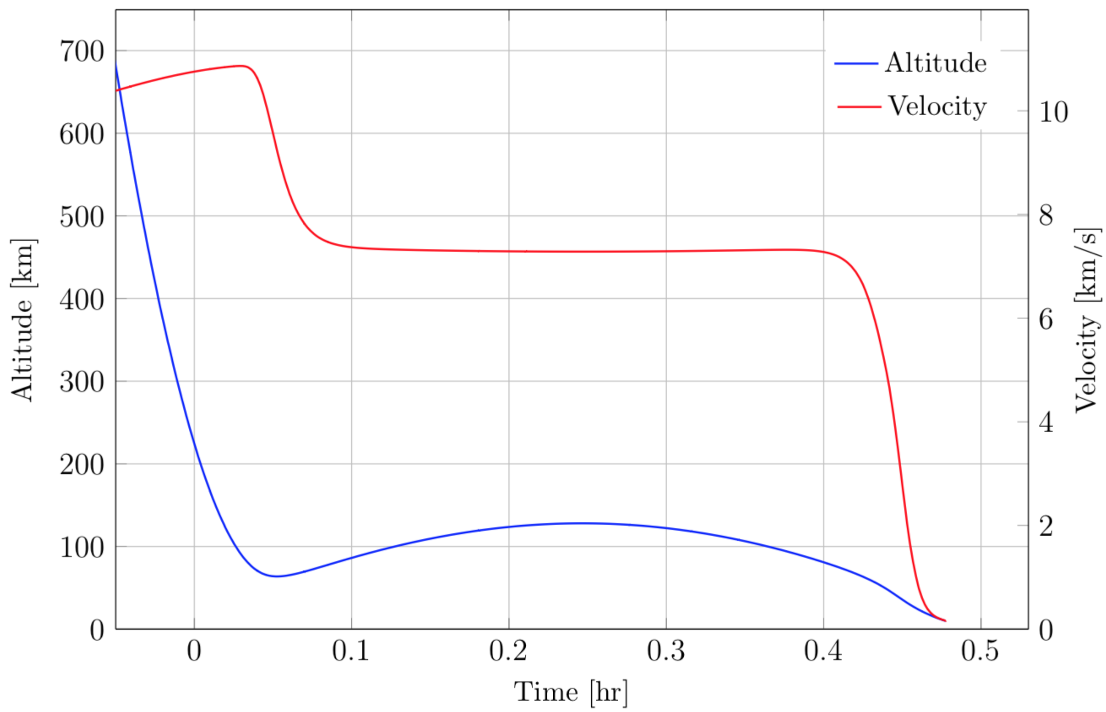


Figure 34: Altitude and velocity over time for a reentry trajectory with a reentry angle of 5°

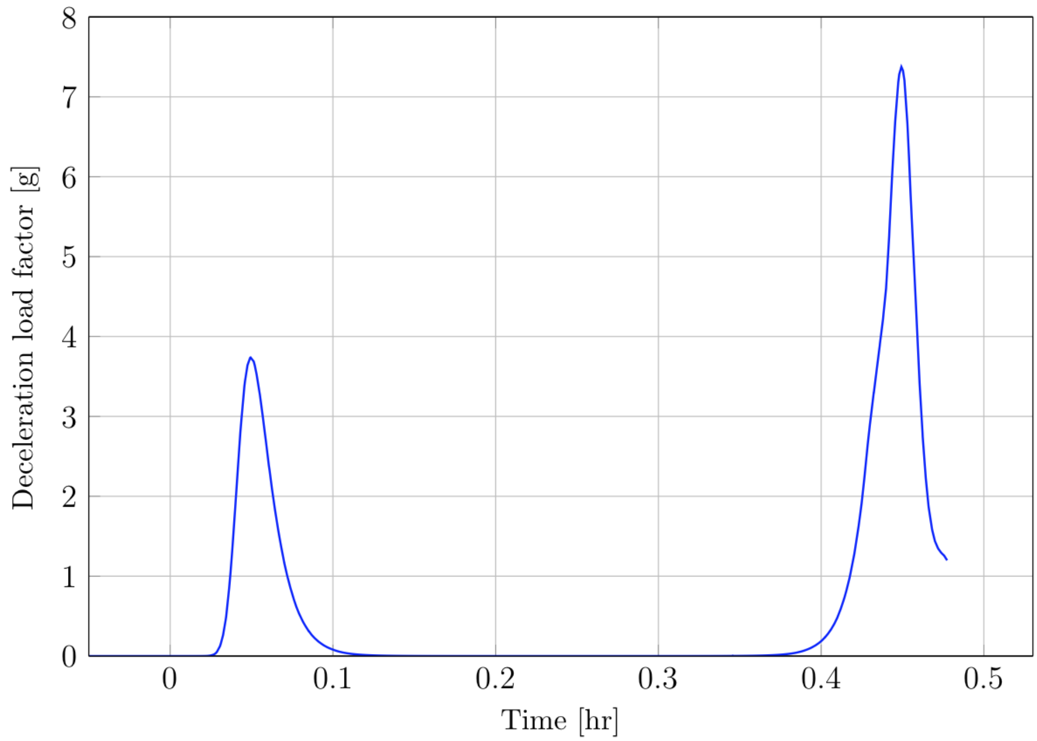


Figure 35: Deceleration load factor over time for reentry angle of 5°

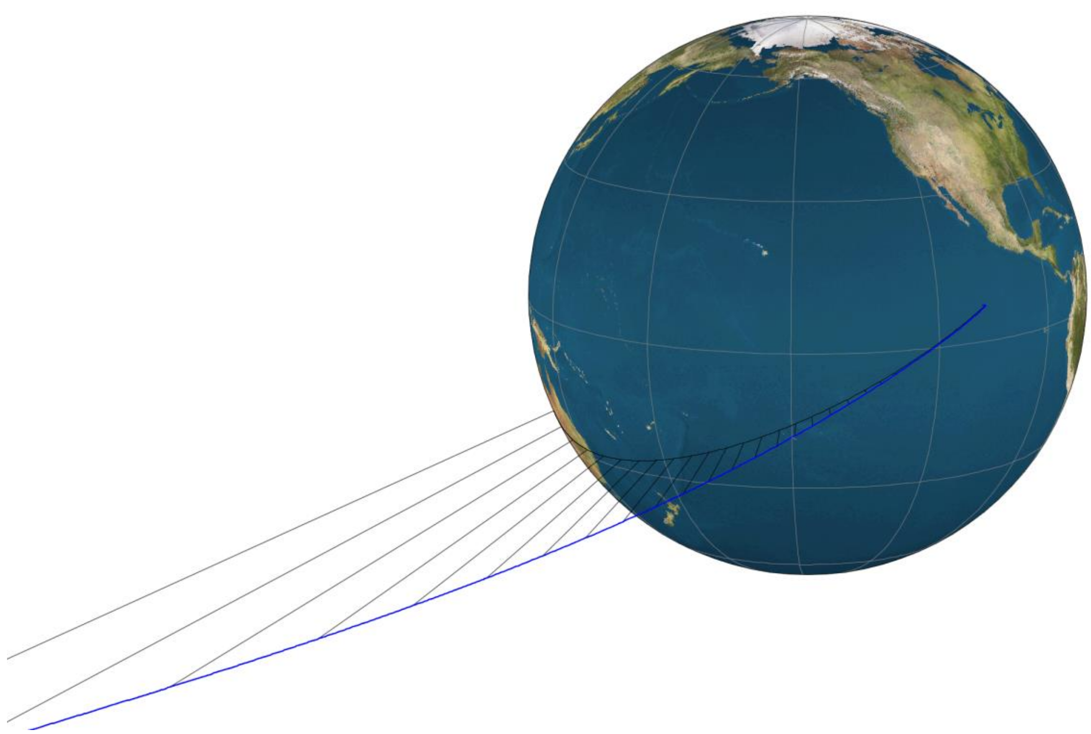


Figure 36: 3D-Visualization of the reentry trajectory

Habitat

The architecture of NASA's TransHab module as continued by Bigelow Aerospace is the baseline for OASIS's *Labitat*, a versatile module intended to support extended-duration missions in deep space on a flexible path to Mars. As its name suggests, it acts as both the primary habitat and pressurized experimental platform for the mission. The choice to use Bigelow's BA 330 as a point of departure is due to its favorable mass-to-weight ratio, the potential for using structural core (albeit modified) as an emergency radiation shelter, and for its spatial potential. By the proposed launch date of March 30th, 2024, inflatable habitats will have a proven history in LEO such as the BEAM module by Bigelow Aerospace launching to the International Space Station in 2015 building on the extended testing of Genesis I and II. Labitat attempts to address its situation in deep space via modifications that increase crew protection to the higher radiation environment and improve on the interior architecture of the Bigelow BA 330 with extended duration living in mind.

The Labitat consists of a central aluminum and composite core that serves a multitude of purposes:

- Structural backbone of the module
- A Coronal Mass Ejection (CME)-rated shielded area which doubles as the sleeping zone
- Surface for the mounting of all needed storage and utilities (science racks, bathroom, food preparation area, communication/workstation, individual workstations, exercise equipment, ECLSS, and food and water storage)
- Four mechanical chases running conduit for power, data, life support and connection to other modules

During launch and transit, the consumables are stored inside the inner diameter of the core and within its sleep corridors.

Fully expanded, the outer expandable shell is 0.46 meters thick, and designed as layers of Nextel, open-cell foam, Kevlar, Combitherm, and Nomex. According to Bigelow's claims, this wall section is equal to the International Space Station in radiation shielding and superior to it ballistically. Labitat's total habitable volume is 201.46 cubic meters which comes out to 67.16 cubic meters per crew member. The interior architecture, or the organization of interior spaces that support human activities and needs, is modified significantly from the baseline.

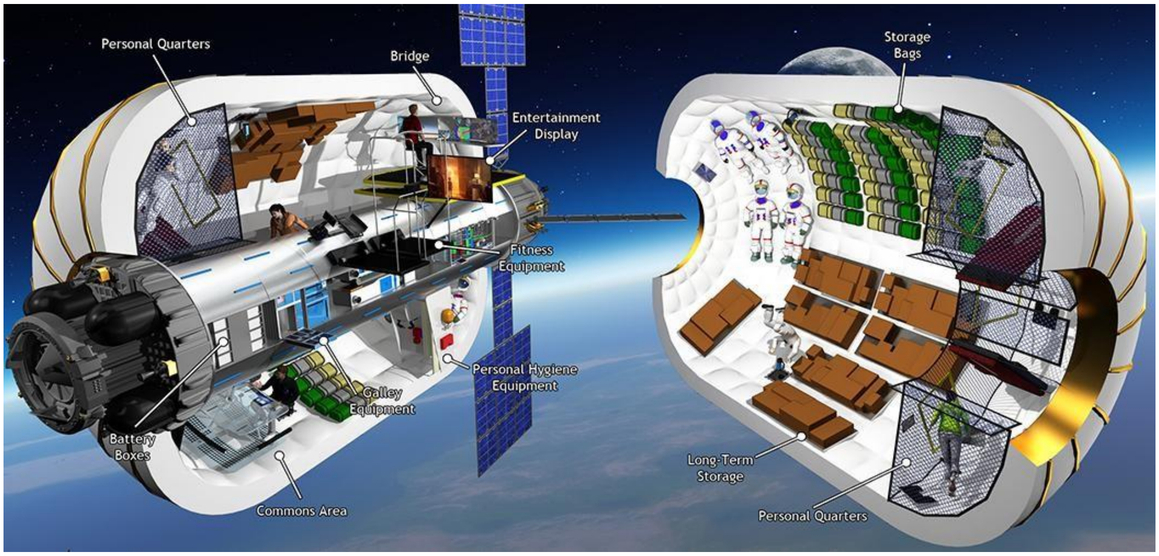


Figure 37: A section perspective of Bigelow Aerospace's BA 330 module (source: Bigelow Aerospace)

In the Bigelow Aerospace BA 330, the large curving inner surface of the wall is covered in storage bags and various screens partitioning off spaces. This is a missed opportunity given the unique topology of the torus: moving around the core, roughly following the path of the overall "donut" shape, one is able to experience two distinct things at once

1. The experience of wrapping around a surface that is receding away like a horizon (the core as a ground)
2. The experience of being contained or wrapped around (the interior wall surface as a sky)

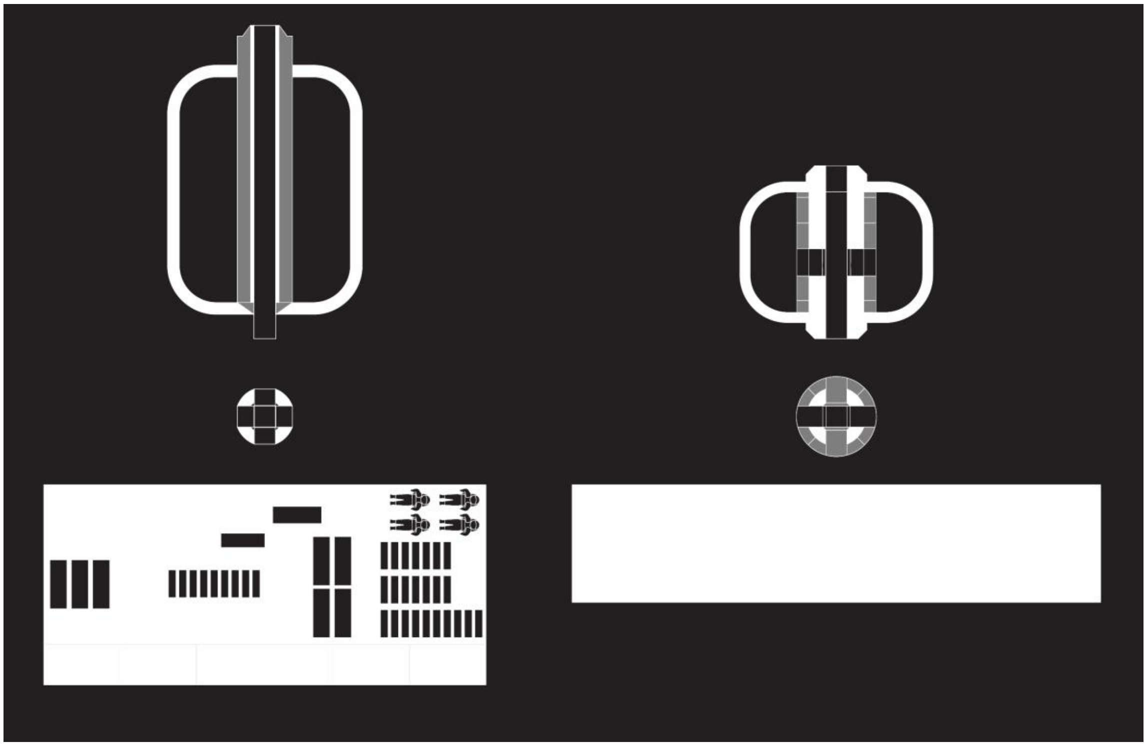


Figure 38: Comparison of core and wall utilization of Bigelow BA 330 (left) and Labitat (right)

One has the option to experience, in a single space, two distinct "ups" while theoretically being able to move endlessly in two distinct directions: one, where "up" is in the direction of the enclosure and another where "up" is in the direction of the core (much like a hamster in a hamster wheel). The possibility of endless movement made available by a toroidal space is one that Bigelow has not exploited.

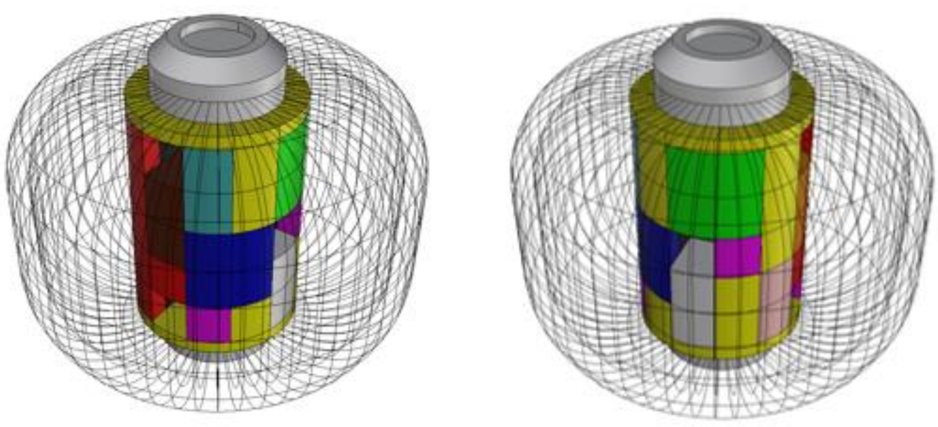


Figure 39: Color-coded diagram of Labitat's utility zone, which frees up the interior wall from objects

In the BA 330, the massive surface of the interior wall is almost completely covered in storage bins, ruining the potential of a large open field for play and general variety and duration of movement. In response, Labitat is a design that keeps the interior wall surface unobstructed by objects, leaving only space free for movement. The operational and programmatic spatial requirements are thus condensed around the core into an organized grid of 1x1 meter cubbies and racks of various depth. The depth of each rack depends on whether the there is a mechanical chassis/structural base directly behind it or not. There are 14 racks with a volume of 1.22 cubic meters, 40 with a volume of 0.39 cubic meters, and 24 with a volume of 0.36 cubic meters making for a total of 41.32 cubic meters of flexible, re-purposable utility volume. With each crew member allocated personal storage and a personal workstation and the remainder being for collective mission needs including generous space for scientific experimentation, Labitat is well set up to be both an extended duration Habitat and a deep space laboratory suiting the needs of a crew of three.

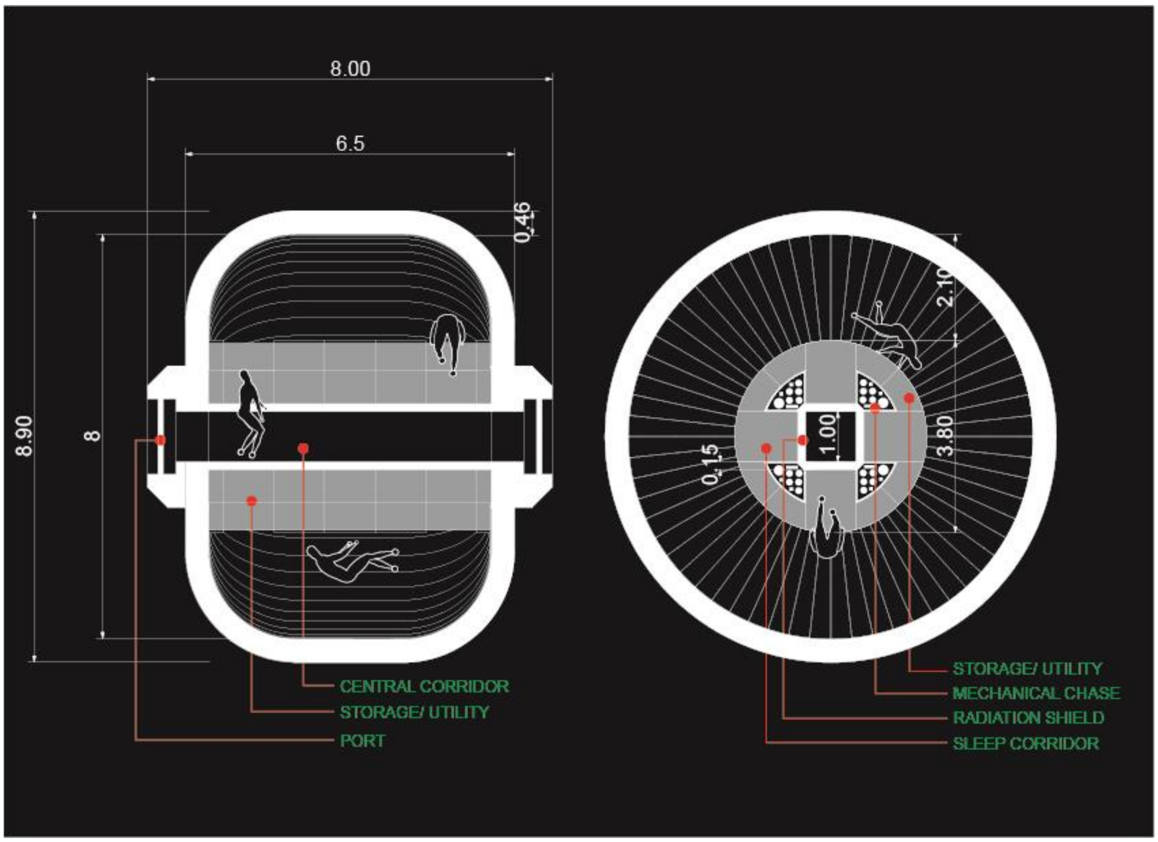


Figure 40: Longitudinal and transverse sections of Labitat

The core of Labitat has potential to be used as a radiation shelter in a deep space mission unprotected from the Earth's magnetosphere. In addition to the inflated 0.46m wall, equivalent to 15g/cm^2 shielding, a 15cm dedicated radiation shield is provided around the central corridor of the core. That in turn is surround by all the utilities mentioned above, including tanks used to store water and waste, estimated to be equivalent to 50g/cm^2 shielding as elaborated in the Human Factors - Radiation Hazards section.

Longitudinally, a 1x1 meter corridor with circular cross section runs the full length of Labitat, providing access to the two ports on each end. Although OASIS immediately requires only one port to interface with the Multi-Purpose Docking Vehicle (MPDM) in the direction of the ARV (Asteroid Redirect Vehicle), having two ports makes Labitat far more versatile as hardware along NASA's flexible path to Mars where larger volumes for larger crews will be required.

One of the major goals of Labitat is to organize habitable space based on separating programmatic extremes. On one end of the spectrum is sleep, which wanted to be a private, confined and decluttered space, and the other is hard science, which tends to be loud, full of instrumentation and shared. These two activities are the extreme conditions of the typical notions of "living quarters" and "laboratory". Rather than partitioning off an already confined space to separate these two programs and in the process losing any hybrid in-between uses, Labitat works with its two main integrated structurally integrated zones in a productive way. Cutting through the core transversely are three sleep corridors, with one allocated to each crew member. The three sleep corridors are equally spaced along the central core with the middle one rotated 90 degrees relative to the others since not doing so would constitute too much loss of shielding in that particular direction. Given the mass and geometric launch vehicle restrictions, three human bodies could not fit end-to-end inside the corridor while maintaining access for each crew member individually. Thus, the zone where crew members (assuming 8 hours of sleep/day) will spend 30% of their lives is also the area of greatest radiation protection in Labitat. Additionally, it is the area of Labitat that is tightest and most confined spatially, an environment that astronauts prefer for sleeping. Thus, sleep occurs in the safest and most comfortable part of Labitat, an area also isolated from potentially loud or hazardous scientific and life-support machinery.

Labitat is positioned as a critical part of OASIS since it supports both the immediate mission of supporting extended-duration human experimentation on an asteroid in a lunar Distant Retrograde Orbit and the larger vision of deep space exploration on a pathway to Mars. Regardless of the availability of in-situ resource utility on the first manned mission to the asteroid, a large Labitat benefits subsequent missions that feature extended human living and a versatile, re-programmable spacecraft that excels equally as a habitat and as a platform for experimentation.

Guidance, Navigation, and Control
Overview

The OASIS mission Guidance, Navigation, and Control (GN&C) systems are responsible for determining the trajectory, position, and attitude of the various OASIS spacecraft and controlling the vehicles. GN&C systems are critical to the OASIS mission given the necessary rendezvous and docking maneuvers as well as general station-keeping of a long-duration habitat.

The hardware required for guidance, navigation, and control is separated into two distinct categories: sensors and actuators. Sensors provide feedback on environmental conditions. These inputs are processed by an onboard flight computer, which then outputs commands to the vehicle actuators to correct or maintain the spacecraft attitude.

Approach:

The OASIS GN&C systems must be designed to support two key phases of the mission. The first phase consists primarily of transit to the distant retrograde orbit and the docking of OASIS modules. This system must be highly responsive for maintaining pointing accuracy during propulsive maneuvers and operations in close proximity to other spacecraft. The second phase consists primarily of station keeping of the docked modules. It is important for this assembly to use a non-propellant-based control system for this deep space science platform to be extensible and reusable. The OASIS mission has adopted GN&C systems which are currently in use or development, and will undoubtedly continue to be improved by 2024.

Asteroid Redirect Robotic Mission:

The Asteroid Redirect Robotic Mission spacecraft concept employs a system of five 10-kW Hall thrusters along with four clusters of hydrazine RCS thrusters for propulsion and attitude control (Brophy 2012). Once the Multi-Purpose Docking Module has been docked to the ARRM spacecraft, the Hall thrusters will be disabled and the remaining RCS propellant dedicated to future control maneuvers of OASIS.

Orion Multi-Purpose Crew Vehicle:

Orion possesses a large array of guidance, navigation, and control hardware, including but not limited to star trackers, GPS, and low-thrust RCS (Mamich).

One sensor of particular interest for other components of the OASIS mission is the Sensor Test for Orion Relation Navigation Risk Mitigation (STORRM) instrument. STORRM is an innovative sensor suite developed for Orion and flight-tested on STS-134. This suite combines a high definition camera and LIDAR-based Vision Navigation Sensor (VNS) to provide improved docking accuracy (Ball Aerospace). The use of STORRM has been proposed for the navigation system of OSIRIS-Rex (Chow 2011), and should be well-characterized by the time it has been integrated into Orion for its OASIS flight.

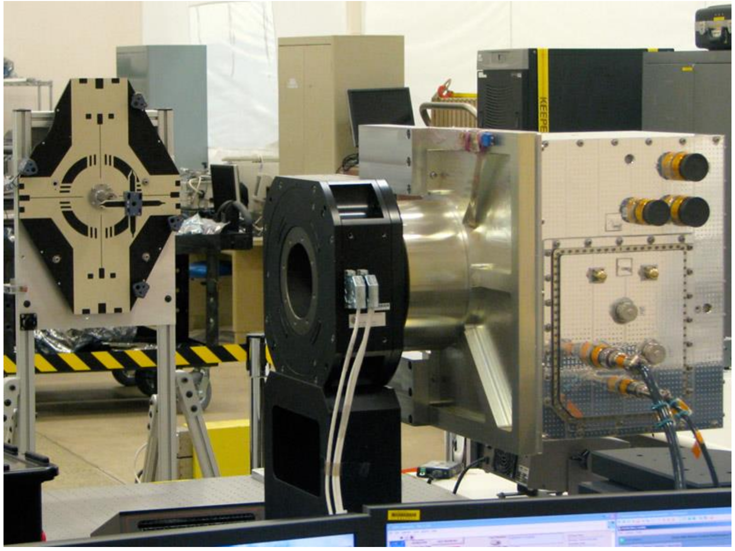


Figure 41: Lab test of the STORRM system. Credit: NASA

Multi-Purpose Docking Module:

The Multi-Purpose Docking Module is to be attached to Orion during launch. The module will be dependent on Orion's GN&C system in transit to the distant retrograde orbit. Once docked to the ARRM spacecraft, the RCS system onboard the ARRM spacecraft will be used to control the whole assembly.

Labitat Module:

The Labitat module is slated to be launched separately, and is intended to be the workhorse of the GN&C system. A number of different hardware choices are available for this module. The Labitat will be equipped with STORRM, a GPS receiver, inertial measurement units, and star tracker and sun sensor suites for attitude and orbit determination. The sun sensors will also be highly utilized to precisely point the Labitat's solar arrays at the sun.

Once docked to the other OASIS spacecraft at the DRO, the Labitat module will take over all station keeping operations. In its position at the end of the length of the docked spacecraft assembly, the Labitat module is in an optimal position to induce control torques on the vehicle assembly.

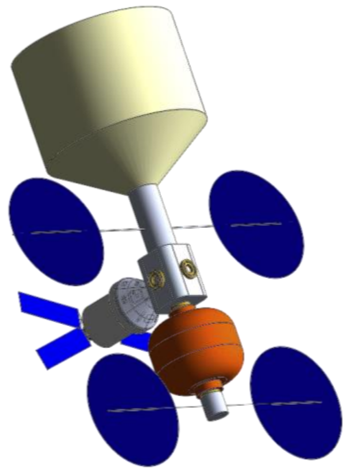


Figure 42: The full assembly of OASIS modules; Labitat at the bottom edge

The Labitat control hardware is modeled after that of the International Space Station, which uses a thruster-based attitude control system provided by Russia and a non-propulsion-based system provided by the United States (Boeing 2011, Gurrisi 2010). The Labitat will control its attitude primarily with four control moment gyroscopes. An RCS system will be used for desaturation of the CMGs or contingency operations. This RCS system will be disabled while astronauts are performing extravehicular activities.

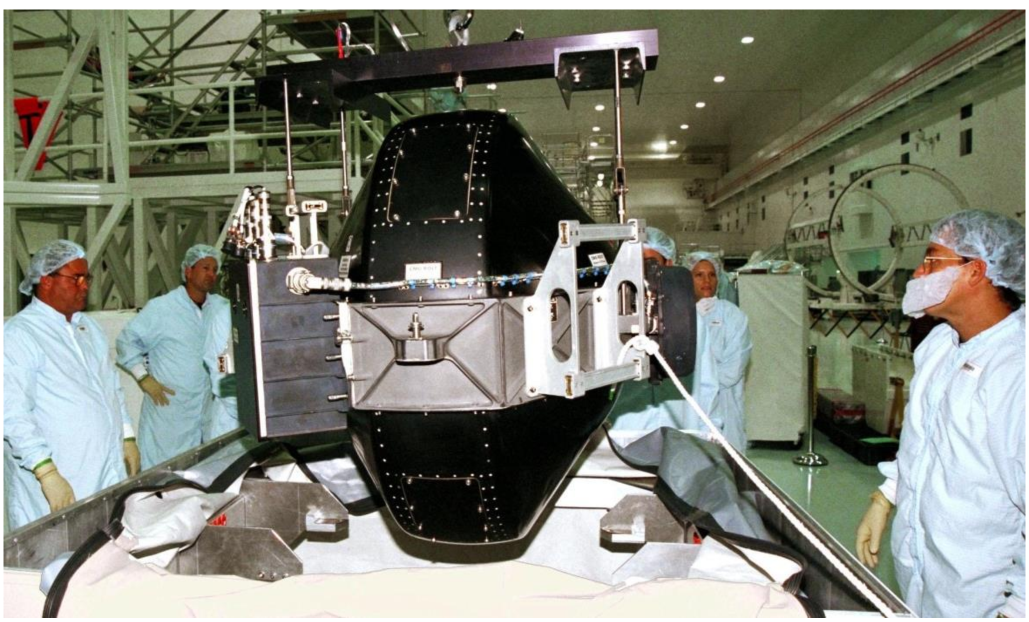


Figure 43: Control Moment Gyroscope unit for the ISS. Credit: NASA

The budgeted mass and estimated cost of each notable component is tabulated below:

Table 16: Mass budget of the GNC Subsystem

Component Name	Quantity	Mass (kg)	Total (kg)
RCS Clusters + Propellant	-	500	500
Control Moment Gyroscope	4	272.155	1088.62
		Total:	1588.62

Table 17: Cost estimate of the GNC Subsystem

Component Name	Quantity	Cost ($M)	Total ($M)
RCS Clusters + Propellant	-	20	26.58
Control Moment Gyroscope	4	7	28
STORRM (Spaceflight101 STORRM)	1	10	10
Remaining Sensors	-	15	15
		Total:	79.58

The simplified cost calculation for the RCS system is based on the notional cost of the ARRM spacecraft RCS system and assumes the cost scales linearly with propellant mass. The Labitat has been allocated 400kg of propellant and 100kg in thruster components, whereas the ARRM spacecraft may employ 900kg of propellant at a total cost of $59.8M(Nealon 2011).

Summary:

The Guidance, Navigation, and Control systems onboard the OASIS mission spacecraft are designed to control the vehicles in transit to the DRO as well as a general station keeping for long-duration human habitation. The GN&C systems selected for use are currently in use or development, and many already have significant flight experience. This system is expected to be highly capable for safe spacecraft rendezvous and docking in 2024, and will help lead the way for future missions.

Power
Overview

The Power systems for the OASIS mission generate, store, and distribute power across vehicle subsystems. These systems utilize high efficiency solar arrays and a number of Lithium-ion batteries. The requested power allocations for each major mission activity are included below.

Asteroid Redirect Robotic Mission:

The ARRM spacecraft concept has two deployable 10.7m diameter solar arrays with 33% efficient solar cells. The circular solar array is representative of the MegaFlex arrays proposed by ATK Space Systems in 2012. A rectangular array, Mega-ROSA, was also proposed by Deployable Space Systems (NASA 2012).

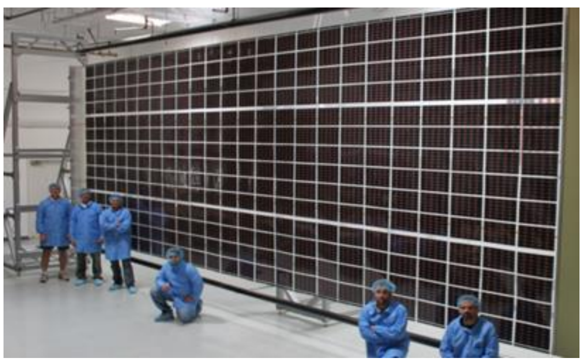


Figure 44: Mega-Rosa Array (Credit: NASA)

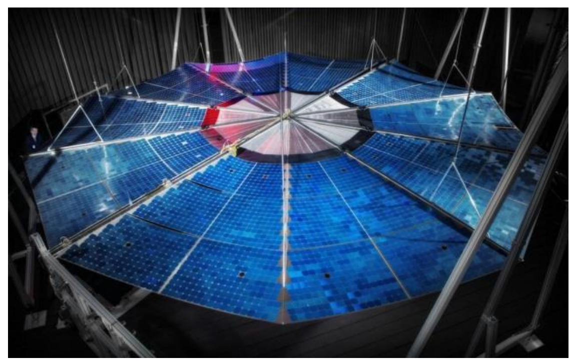


Figure 45: MegaFlex Array (Credit: ATK)

At end of life, it is estimated that this power system can provide 41.2kW. An onboard rechargeable Lithium-ion battery could provide 392Whr assuming a 15% depth of discharge (Brophy 2012).

Orion Multi-Purpose Crew Vehicle:

Orion uses six Lithium-ion batteries to provide power to the rest of the spacecraft. Each battery has a mass of 44.8kg and a capacity of 30Ah. The service module developed by ESA for Orion is designed with 30% efficient solar cells and the four array elements together nominally generate 11.1kW (Spaceflight101 Orion).

Multi-Purpose Docking Module:

The Multi-Purpose Docking Module is not equipped with any solar panels and must have sufficient battery capacity to survive the short-duration flight with Orion to the DRO. The ARRM spacecraft does not require nearly as much power when its electric propulsion system is no longer in use; the ARRM spacecraft concept dedicates 40kW of power to the electric propulsion system and only 1,200W to the remainder of the vehicle (Brophy 2012). Once attached to the ARRM spacecraft, the docking module will be powered entirely by excess power generation from ARRM through an electrical dock.

Labitat Module:

The Labitat Module has been equipped with deployable solar arrays sized to produce the remaining power required to complete mission objectives. The Labitat module requires 10kW for science operations over the course of the full mission, and an additional 15kW for the spacecraft bus and miscellaneous uses. With a 33% solar cell efficiency and 25% line item margin, two MegaFlex solar panels with a 9.5m diameter provide 25kW over the mission duration and make the Labitat self-sufficient. The module will be complemented with Lithium-ion batteries of similar performance to the ARRM spacecraft concept.

Power Budget:

The requested power allocations for each major mission activity and power generation capabilities are included below:

Table 18: Power Consumption

Purpose	**Total Power Requirement (kW)**
Science Operations	10
Docking Module Bus System	10
Labitat Module Bus System	15
ARRM Bus System	1.2
Miscellaneous	10
Total	46.2

Table 19: Power Generation

Source	**Total Power Generated (kW)**
ARRM Array	41.2
Labitat Module Array	25
Total	66.2

The power generation capabilities of the ARRM spacecraft concept array and the Labitat Module Array exceed the power consumption of all draws by 20kW. The two 10.7m circular arrays on the ARRM and two 9.5m circular arrays on the Labitat together generate over half of the 110kW produced by the complete power system of the International Space Station (NASA Glenn 2011).

Subsystem Mass:

The Asteroid Retrieval Feasibility Study estimates the full mass of the electrical power subsystem of the ARRM vehicle concept in the table below:

Table 20: Predicted ARRM Electrical Power Subsystem Mass

Component	Mass (Kg)
Solar Arrays	854.2
Power Cable and Harness Subsystem	90.0
Power Management & Distribution	120.8
Battery System	24.6
Total	1089.6

These mass values are sized for two 10.7m diameter solar arrays, which are by far the largest mass contributions to the subsystem. To estimate the mass of the power subsystem onboard the Labitat module with 9.5m diameter arrays, we can estimate all subsystem components scale by the ratio of the solar panel diameters squared. This gives a scale factor of 0.788 between subsystem components of the ARRM and Labitat module. The estimated mass of the Labitat module is given in the table below:

Table 21: Predicted Labitat Electrical Power Subsystem Mass

Component	Mass (Kg)
Solar Arrays	673.1
Power Cable and Harness Subsystem	70.92
Power Management & Distribution	95.19
Battery System	19.38
Total	852.59

Subsystem Cost:

The Asteroid Retrieval Feasibility Study also estimates the complete cost of the Electrical Power Subsystem in fiscal year 2012 dollars, and all development and hardware costs sum to $504.8M. Applying the same scale factor determined in the subsystem mass calculations, this places the power system cost of the OASIS modules at approximately $397.78M (Brophy 2012).

Communications
Overview

The OASIS mission Communications system is used to transmit telemetry, science data, and voice communications to the ground and between spacecraft. The Communications system of OASIS is primarily modeled after the existing or proposed systems for the ARRM vehicle concept and the Orion Multi-Purpose Crew Vehicle.

Approach:

The ARRM concept vehicle and Orion capsule served as baseline ideas of our approach, as both spacecraft are designed with deep space in mind. The ARRM concept vehicle's communication system is designed for a maximum communication distance of approximately 2AU.

Multi-Purpose Docking Module:

The Multi-Purpose Docking Module is launched with Orion, and so it does not contain an independent long distance, high data rate communication system. The electrical connection between the docking module and ARRM allows the docking module to interface with ARRM's long distance systems. The module does have a radio for UHF communication for nearby communication or emergency situations.

Labitat Module:

The primary communication system for the Labitat module is an improved version of the Lunar-Laser Space Terminal (LLST) that was successfully tested onboard the LADEE mission in 2013. This system uses an infrared laser transmitter to achieve world-record (Lincoln Laboratory 2013) downlink rates on the order of 622Mbps and uplink rates of 20Mbps from the Moon to Earth. All the components of the LLST system, namely the optical and modem modules and the controller electronics, have a combined mass of 30kg (Menrad 2014). Laser communications is an area of tremendous potential for future space missions, and will greatly increase the capability and fidelity of deep space communication.

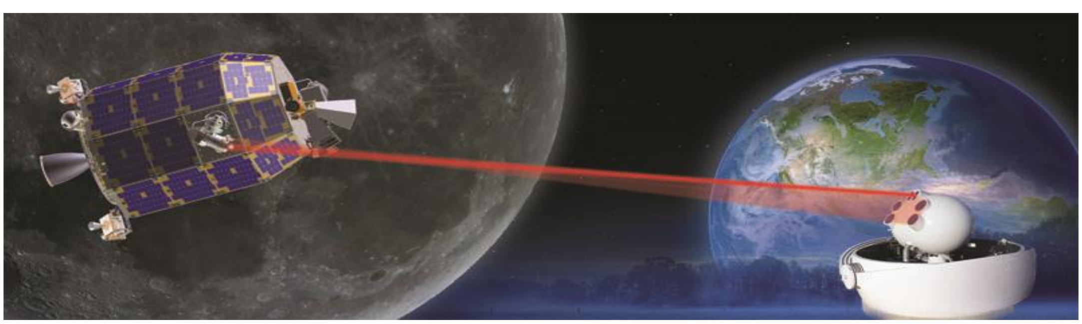


Figure 46: Figure: LADEE demonstrated high-rate, two-way laser communications in 2013. Credit: NASA

The secondary communication systems for the Labitat module are modeled after the ARRM system. A deployable reflect array transmits data on the Ka-band with 50Mbps downlink and 20Mbps uplink (Eutelsat, Keesey 2015), and an omnidirectional antenna transmitting on the X-band provides safe mode communications (Brophy 2012).

Subsystem Mass:

The full Ka-band/X-band communication system configuration proposed for ARRM has an approximate mass of 76.9kg. Combined with LADEE's 30kg LLST, the total mass of the Labitat communication system is 106.9kg.

Subsystem Cost:

The proposed cost of the Communications and Tracking system for the ARRM vehicle is given in the Asteroid Retrieval Feasibility Study as $86.8M. The LLST system development costs total approximately $65M (Chaplain 2013) for a combined Labitat communications cost of $151.8M.

Summary:

The Communication system of the OASIS mission utilizes promising new technologies for deep space communications. The docking module employs the communication system of the ARRM vehicle concept, and Labitat module combines laser communication technologies with systems already proposed for the ARRM. These systems will pave the way for high data rates from deep space and improve the scientific return of the OASIS mission.

Structural Design and Layout

The platform consists of the asteroid and ARM vehicle, 3 modules, 3 auxiliary structures, and the docked Orion crew module and service module. The three modules are the Multi-Purpose Docking Module, Labitat, and Labitat Bus Module, and the 3 auxiliary structures are the Candarm 3 robotic arm, a Kibo-inspired exposed rack for experiments and scientific tool storage, and an airlock.

Table 22: Mass budget

Equipment	Mass (in metric tons)
Orion	26
Habitat	12
Science	1
Node	6
Arm	0.4
Airlock	0.4
Power	1
Propulsion + GNC	2.1 (0.5+1.6)
Avionics	0.5 (0.25 0.25)
Thermal	2.1 (0.6+1.5)
Communications	0.2
Spares	0.5
ECLSS	1.1

Considering the constraints on the number of launches for the overall mission and the full capacity payload masses of the launch vehicles, two launches, one each on Falcon Heavy and SLS 1B would be able to deliver the complete mass to the asteroid brought back into the DRO. The full capacity of the Falcon Heavy launch vehicle is 18.5 tons and that of the SLS 1B is 37 tons. The first launch would be Falcon Heavy and the second would be the SLS 1B. The respective payloads equipments for each launch are tabulated below.

First launch is with the Falcon Heavy, which has a maximum payload capacity of 18.5 metric tons. The launch mass on Falcon Heavy for the current mission is expected to be 18.3 metric tons, the breakdown of which is given below.

Table 23: Mass budget for launch 1

Equipment	Mass (in metric tons)
Labitat	12
Power	1
Propulsion + GNC	2.1 (0.5+1.6)
Avionics	0.5
Thermal	0.6
Communications	0.2

ECLSS	1.1
Arm	0.4
Airlock	0.4

Second launch is with the SLS Block 1B, which has a maximum payload capacity of 37 metric tons. The launch mass on SLS Block 1B for the current mission is expected to be 35 metric tons, the breakdown of which is given below.

Table 24: Mass budget for launch 2

Equipment	Mass (in metric tons)
Orion	26
Science	1
Node	6
Thermal	1.5
Spares	0.5

The ARM vehicle is required to support an aft facing docking port, which will dock to the Multi-Purpose Docking Module. This module will serve a node, with connection for the airlock, Exposed Experiment Platform, Orion, Labitat, and Canadarm3. In addition to these purposes, an additional docking face remains unallocated at this time to allow for resupply of consumables or possible future expansion of this research facility.

The Labitat, as its name suggests, serves both as the primary habitat and laboratory for the crew. To maximize volume for a given mass, this is an inflatable module. By the proposed launch date of this mission, inflatable habitats will have a proven history in LEO, such as the BEAM module by Bigelow Aerospace at the ISS (http://www.nasa.gov/content/new-expandable-addition-on-space-station-to-gather-critical-data-for-future-space-habitat/).

The Labitat consists of a central aluminum and composite core that serves several purposes: structural backbone of the module, surface for installation of racks or storage, location for sleeping compartments and restroom storage, conduit for power and tubing, and its interior serves as an emergency radiation shelter. During launch and transit, consumables are stored inside the inner diameter of the core.

The outer expandable shell is 0.46 meters thick, and designed as layers of Nextel, open-cell foam, Kevlar, Combitherm, and Nomex. This design is based on references and draws inspiration from NASA’s Transhab Concept (Transhab 2007).

The Labitat bus houses propulsion for course-correction of the Labitat during its ballistic trajectory to the DRO and for station keeping, GNC and avionics, ACS, radiators and heat exchangers for the Thermal Control System, and additional solar panels to augment the panels from the ARM. The bus is placed on the aft of the Labitat on the same center axis as the Labitat and MPDM, positioning it as the station element farthest from the asteroid. This provides a greater moment arm for ACS positioning, and places thrusters far away from scientific samples to minimize contamination from impinging thruster plumes.

Thermal Control

The mission architecture for thermal control is driven nearly entirely by cold-cases. Due to the large surface area of the habitat/Orion module, lower crew number, and very cold radiative environment (some sink temperatures can be well below 100K), the radiators are extraordinarily effective, rejecting >250 W/m^2 (Human Spaceflight: Mission Analysis and Design, p519). Since radiator area is fixed, and normally designed for the highest mission heat loads and hottest environments, these radiators become sources of undesirable heat leak during cold environments. However, transient or consumable methods of rejecting heat (sublimation, boilers) are not recommended due to the long mission duration.

The thermal condition may be augmented by tapping into the ARM's 50 kW-class solar panels for supplementary heating during cold thermal periods without significant mass penalty (0.7 kg/kW). This method is infinitely preferred over the alternative radiator modification of louvers, which in a dusty environment could fail open or closed. The syzygy of the earth-moon-sun system may introduce periodic cold conditions which may not be corrected by thermal augmentation, but additional study is required to confirm the feasibility of the present design.

The component sizing was carried out with numbers suggested in Human Spaceflight: Mission Analysis and Design for Active Thermal Control. The thermal design power (TDP) which must be dissipated from the ATCS system is assumed to 6 kW for the habitat module and 2 kW for the docking module. The surface area of the modules is given as 300 m2 and 200 m2, respectively, and the inflatable habitat module is assumed to have insulation built in. For completeness, a value for multi-layer insulation (MLI) of 2 kg/m2 is included for the habitat module but is not considered to be part of the thermal system, as it is bookkept under structure.

Table 25: Labitat thermal control system mass, power, volume, and cost

	LABITAT MODULE			
ITEM	**MASS (kg)**	**Power (W)**	**Volume (m^3)**	**Cost ($M)**
Heat Exchangers	18.5	0	0.0232	2
Cold plates	72	0	0.168	5
Pumps + Accumulator	28.8	138	0.102	1
Plumbing + Valves	*(15% of wetted parts)*	0	0	0.5
Instrumentation	*(5% of wetted parts)*	0	0	0.5
Fluids	*(5% of wetted parts)*	0	0	0
Radiators	424	0	1.6	1
Insulation	600	0	0	0.5
Heaters	4.2	0	0	0.1
Penalty for P,V,I,F	*135.8*			

TOTALS	**1283.3 kg**	**138.0 W**	**1.9 m³**	**$10.6M**
TOTALS (excl. insulation)	**683.3 kg**	**138.0 W**	**1.9m³**	**$10.1 M**

Table 26: MPDV thermal control system mass, power, volume, and cost

	DOCKING MODULE			
ITEM	**MASS (kg)**	**Power (W)**	**Volume (m³)**	**Cost ($M)**
Heat Exchangers	17.5	0	0.0184	2
Coldplates	24	0	0.056	5
Pumps + Accumulator	9.6	46	0.034	1
Plumbing + Valves	*(15% of wetted parts)*	0	0	0.5
Instrumentation	*(5% of wetted parts)*	0	0	0.5
Fluids	*(5% of wetted parts)*	0	0	0
Radiators	141.3	0	0.5	1
Insulation	400	0	0	0.5
Heaters	1.4	0	0	0.1
Penalty for PVIF	*48.1*			
TOTALS	**641.9 kg**	**46.0 W**	**0.6 m³**	**$10.6M**

ECLSS

While Orion will be capable of performing continuous CO2 and moisture removal, the inter-cabin air flow is poorly defined, and therefore a redundant ECLSS system will be required to ensure proper pressure control and air revitalization. Mission durations are too short to be useful for a closed-loop life support system, therefore an open-loop system will be used, which will vent excess moisture and CO2 overboard. Expected subsystems will be a solid amine swingbed, which is being used on Orion (<20 W power, ~50 kg, and <50 L volume), high pressure nitrogen and oxygen tanks for air makeup, water accumulators for water storage, and an Orion-based ARS system (0.45 m3, 60 kg, 150 W) for controlling trace contaminants. The following table details the mass required for consumables, which includes tankage and valving for a 3-person crew of 6 weeks.

Table 27: ECLSS requirements

Component	**Consumption Rate**	**Totals (kg) (6 weeks)**	**Volume (m^3)**	**Power (W)**
Food	1.8 kg/crew/day	250	0.25	0
Water (drinking)	2 kg/crew/day	284	0.4	20
Water (hygiene)	0.82 kg/crew/day	116	0.2	20
O_2	0.82 kg/crew/day	215	0.7	20
N_2	0.06 kg/crew/day	16	0.1	20
Consumables Subtotal		877 kg	1.65 m3	80 W
Amine Swingbed	-	50 kg	0.05 m3	20 W

ARS	-	60 kg	0.45 m^3	150 W
Dehumidifier	-	20 kg	0.05 m^3	150 W
Fire Detection & Suppression	-	5kg	0.01 m^3	5 W
TOTAL ECLSS (excl. thermal)		1012 kg	2.21 m^3	405 W

By depending on the ECLSS capabilities listed above for the duration of the stay at the asteroid station, the capabilities of the Orion capsule will remain unused, and it will be stored in a powered down mode. Orion will be stocked with 21 days of consumables life support capability to facilitate a swift abort and safe return to Earth from the station at any point of mission with a minimum margin of 3-4 days.

Human Factors

Crew Size and Selection

The mission calls for a crew of three astronauts. With Orion's 84 human-day ECLSS endurance, the mission can support 28 days of deep-space transit, which provides a safety buffer in case of labitat integration failure. Under nominal mission conditions, the crew can bring back a larger mass of samples.

A crew of three also reduces mass and volume requirements for the deep-space labitat (pressurizable volume 239.5m^3) while providing enough crew for 2 person EVA teams with one local CanadarmX operator and in-hab supervisor. It also reduces consumable requirements.

Crew composition will reflect the origin and purpose of the mission, with one geologist, one engineer/pilot, and one mission specialist/robotic payload operator. Crew selection will reflect representation of international partner contribution, most likely two from the US and one European.

Radiation profiles are significantly smaller than a six month ISS mission and as such present no selection restriction on age or gender. The mission calls for standard Extravehicular Mobility Unit (EMU) space suits to leverage their compatibility with CanadarmX and surface tether operation.

Crew training will leverage Johnson Space Center's expertise in astronaut training, while moving in the direction of greater astronaut independence. Future deep space missions will have a communications time delay and require operational independence. Training will leverage long duration ISS flights' experimentation with time delayed communications (such as by US astronaut Scott Kelly) to accustom both astronauts and mission control to the new operational paradigm.

While OASIS will be the first mission to rendezvous with an exogenic space object, it will leverage NASA experience with satellite retrieval and Hubble servicing missions. Safety considerations for crewed flight will be reflected in the EVA schedule and CONOPS, as detailed in the science section of this report. In brief, EVAs will utilize two independent tether systems as required, one for CanadarmX supported operations, and one for labitat and structure surface operations. Mission re-nitrification prebreathing requirements are met by the provision of oxygen masks in the radiation shielded sleeping area. EVA objectives, set by the science operations, call for rotation of EVA roles.

Physiology, Medicine, Radiation Considerations

Psychological Effects

Long term, deep space missions pose great psychological risks on astronauts in the form of sensory deprivation, isolation and confinement. The OASIS program aims to quantify and mitigate these risks in service of the overarching goal of preparing for a long-duration spaceflight to Mars.

Sensory deprivation is a key factor in this mission. A lack of the puff of gravity and rootedness can be overcome with a well-designed habitat which, like the ISS habitat, will have designated walls to give the feeling of orientation and having an “up” and “down”. Another way in which sensory deprivation manifests itself is through a loss of sense of time, as the environment will not lend itself to having clear distinctions between day and night. Again, this can be overcome by providing the astronauts with working schedules that will allow them as close to normal working hours and lifestyle patterns as back on Earth, a method taken from Antarctic overwintering analog studies by the European Space Agency.

Figure 47: Dr. Alex Kumar overwintering in Antarctica as part of an ESA Mars analog mission

Isolation is also an important consideration. With only three astronauts on board, there will be a higher requirement for crew integration. Although the astronauts will never be physically alone- a problem in itself- isolation onboard space missions is a common occurrence. Tending to initially emerge from a lack of communication with Earth, factors such as cultural, background and personality differences tend to propagate the problem further. It is important to note that, given the intercultural aspect of this long-duration mission, these effects are nearly certain. Although eliminating communication delays is not possible, the effects of isolation can be alleviated through further astronaut training and higher collaboration with partnering space agencies.

Confinement anxiety, primarily an effect of habitat design, will be addressed through the Labitat design- a living and working space designed with the psychological wellness of the astronauts at the forefront.

Physiological Effects

The physiological effects of a deep space mission are severe. The lack of gravity in space causes extreme physiological changes in the body, such as muscle atrophy, motion sickness and bone demineralization.

Muscle atrophy occurs when there is a lack of gravity and the skeletal muscle no longer needs to maintain posture as with terrestrial location- the astronauts can move freely without using the same muscles that they would ordinarily use on Earth. Muscles quickly weaken and decrease in size, with up to 20% of an astronaut's muscle being lost in as few as 5-11 days. Two hours of exercise a day using special-built exercise apparatus from NASA should mitigate against muscle atrophy, however it will not eliminate it completely.

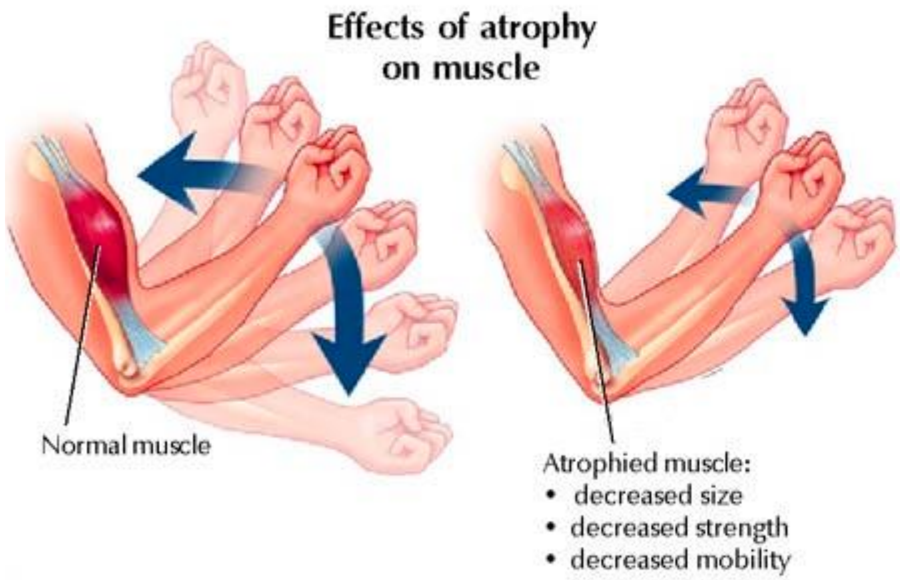


Figure 48: Effects of atrophy on muscle

Motion sickness affects around 40% of astronauts in space, and the effects can last up to three days upon arrival in space. Symptoms include nausea, vomiting, headaches and dizziness, and these effects create an operational inefficiency in the astronauts as the first 3 days in space are lost to acclimatizing. However in this mission, as they will take more than 3 days to reach their DRO, the astronauts should be fully operational by the time they arrive. Motion sickness can be aided with normal, over the counter remedies.

Microgravity induces a loss of bone density in astronauts in space, as they are no longer statically loaded by gravity. Pressure and use of the skeletal system on Earth through walking and running are absent in space. In addition to this, bone loss could be aggravated by low light levels, vitamin deficiency and high levels of carbon dioxide. To reduce these effects, astronauts will exercise for a minimum of two hours a day, as described above, as well as eating a nutritionally balanced meal with supplemental vitamins.

Medicine/Health care

Long duration space flight comes with special health care challenges. These are not expected to differ substantially from those of Apollo, MIR, or the ISS. Similar to container ships and remote Antarctic stations, health care will take the form of first aid, with a comprehensive kit of pharmaceuticals including broad spectrum antibiotics. Secure channels for communication with a mission control based Flight Surgeon are available for the standard confidential daily crew conference. In the event of severe debilitating injury or death, standard and clear operating procedures will be followed.

Health monitoring of the crew can be described by recurring measurements taken over different periods. Twice daily, each crew member will complete a quick medical check-in, consisting of a pulse-oximeter and a blood pressure test. This regime is designed to use simple, standardized medical equipment and create a database of cardiovascular profiling in the deep space environment to compare to large reserves of long-duration data in Low Earth Orbit. In addition to these daily tests, weekly blood and urine sample will be taken. The blood tests will be conducted using the same type of equipment as used on the ISS, specifically the NASA Portable Clinical Blood Analyzer. Finally, constant monitoring of the CO_2 removal rates can be used to back out average estimates for metabolic rates, and continuous radiation environment monitoring will be conducted for use in follow-up ground studies.

Radiation Considerations

Radiation is a serious hazard to crewed missions in deep space, especially with a view towards long duration. Radiation hazards in space derived from several sources which have various time and intensity characteristics.

- Cosmic radiation. GeV scale. Constant isotropic homogeneous back-ground. Dose over likely mission durations isn't likely to substantially increase risks of radiation
- Solar electromagnetic radiation. UV, visible, IR. Effective point source, radial flow, can be shielded with windows
- Solar neutrino. Point source, radial flow, color oscillations and no risk of radiation illness
- Solar neutrons. Low flux. Straight lines, not affected by magnetic fields
- Solar wind. Charged particles, mostly protons and helium nuclei. Variable flux. MeV scale corresponds to a particle velocity of 0.01c. Moderate penetrating power. Solar flares and coronal mass ejections can increase flux by factors of a million or more. Dominant component of radiation-that-needs-shielding. Flux direction complicated
- Van Allen belt radiation. Concentrated solar wind, dominant contributor in MEO, encountered during TLI flight

Different levels of shielding are necessary to protect against these types of radiation. The following figures are taken from (Letaw *et al.* 1989).
The most energetic (and thus difficult to shield) radiation is galactic cosmic rays (GCRs). The following figure shows annual dose for a given shield thickness. The key takeaway is that the dominant contribution is neutrons followed by target secondaries - showers of particles generated by a primary impact in the shield. Unshielded GCR exposure for a nominal 50 day mission is 120mS, ~10% of the NASA lifetime limit on non-prompt exposure.

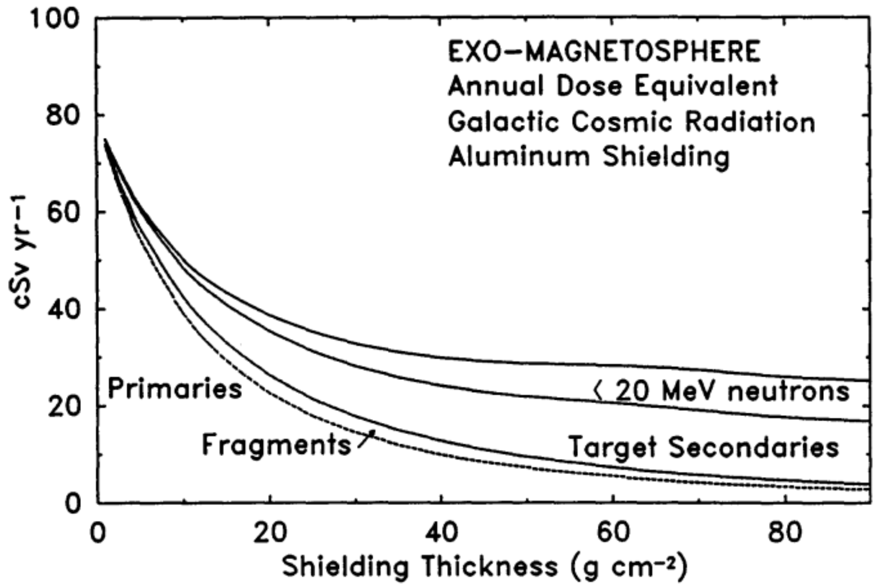


Figure 49: Annual dose equivalent from galactic cosmic radiation at zero tissue depth as a function of aluminum shielding thickness

The other key source of radiation exposure is Coronal Mass Ejections (CMEs) or solar storms. During a particularly violent event in 1972, unshielding prompt exposure exceeded 6 Sv, a universally fatal dose. Fortunately, it is possible to shield effectively against solar wind particles with low molecular mass components of the spacecraft. In the following figure, 10cm of Al shielding drops the prompt dose of radiation by more than an order of magnitude, to below symptomatic levels.

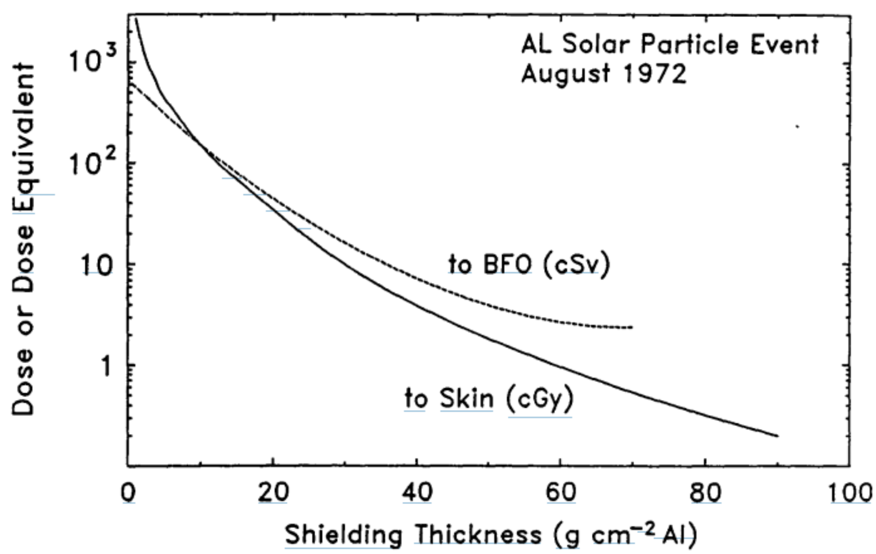


Figure 50: Dose equivalent to the blood-forming organs and skin dose as a function of aluminum thickness for the August, 1972 anomalously-large solar energetic particle event

Aluminium is not the most effective material for shielding against GCRs and high energy solar wind. Materials with high proton density are preferable, such as water, methane, and other liquid fuels. Hydrogen's low density and storage issues make it unsuitable for radiation shielding. Other shielding materials include lithium borate or liquid oxygen, both of which fall between water and Al on an efficacy per mass or per volume basis. Lead and copper are both relatively ineffective. The following figure shows an e-folding length of 15cm for water shielding against GCRs, which represents an upper bound for CME shielding efficacy.

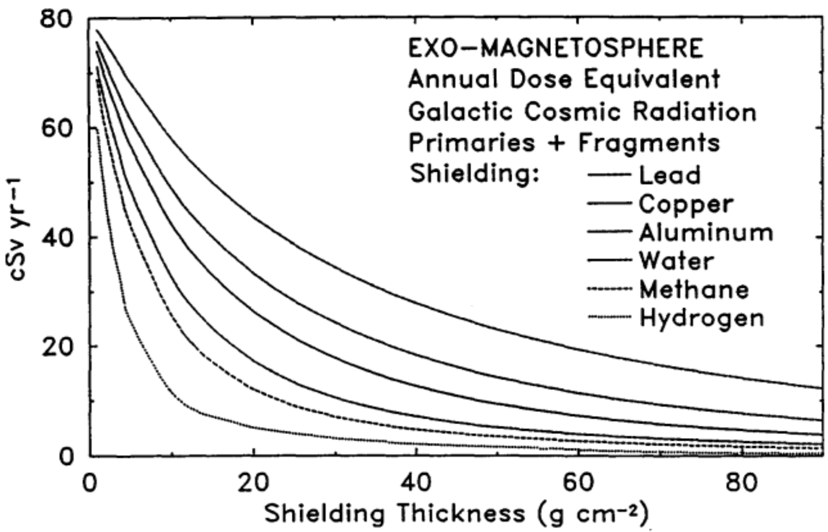


Figure 51: Annual dose equivalent from galactic cosmic radiation as a function of shielding thickness for several possible spacecraft shielding materials

Our design uses pressure vessel, micrometeoroid shielding, and thermal shielding for passive shielding throughout the crewed components of the mission. Additionally, there is a radiation shelter in the core of the habitation module which has an integrated linear density exceeding 60g/cm^2 between. This configuration minimizes crew exposure to radiation for a given shield mass, and reduces GCR flux by a factor of 20, and solar wind penetration by a factor of a million. Expected integrated dose for an exceptionally large CME is 40 mSe, five times lower than the already conservative NASA 30-day exposure limit for astronauts of any age, and smaller than the expected 25 mSe (17 (25 days in hab) + 2.7 (20 days sleep in shelter) + 5.5 (5 days on EVA)) steady state background for a mission of this length, based on Apollo 14 data scaled by increased radiation shielding.

It is worth noting that the captured asteroid does not provide any meaningful radiation protection. The gyroradius of solar wind particles is around 1000km, meaning that exposure covers 2π steradians, or a full half sky. Given uncertainty of shelter orientation with respect to the (variable) solar magnetic field, full shielding makes sense, while a ~5m asteroid 10m from the habitat will provide shielding for at most 8% of the radiative field.

Worst-case scenario radiation exposure amounts to 65 mSe, roughly half a 6 month ISS mission and less than 7% of any NASA mission lifetime limit. This radiation protection strategy reduces risk to crew from radiation exposure to levels below other sources of risk to the mission.

Policy and Outreach

Objectives;

The OASIS mission represents an exciting new time in the human exploration of outer space and an important stepping stone to the exploration of Mars or Martian moons. The technology development and science objectives achieved during this mission are critical to reducing risk to a human mission to Mars. The primary objective is to reduce the risk of funding loss due to a lack of information over the importance of this mission to the long term goal of bringing humans to Mars. The reduction of the long-term funding risk can be achieved through developing a strong legal framework for asteroid mining, engaging international partners in spacecraft production, culturing Private-Public partnerships to deliver critical hardware and encourage new industries, and educating the public on the excitement and benefits of this important stepping stone into humanity's future.

Activities to Engage Public:

NASA has made many strides in recent years to increase public awareness of spaceflight programs. Given the relative proximity to Earth and availability of high-speed laser communications, real-time communications will be possible including high definition video. Though it is not possible to know exactly what new media tools will exist in a decade's time, it is reasonable to expect outreach through one or more of the following options:

- Social media (Twitter, Instagram, etc.) @CSC2015_Voyager
- Live video interviews
- Live 3D video of on-asteroid EVA operations using virtual reality (Oculus Rift) from cameras mounted on astronaut's helmets
- Competitions to design new astronaut meals (reality TV competition in partnership with Food Network)
- Competition for high school/university students to design a science experiment on mission
- "Exploration of Mars Passport", similar to Orion EFT-1 where a small computer chip containing names of public is flown on mission
- Contest to name vehicles
- Choice of EVA suit color schemes
- Flying components for the ISRU aboard the ISS
- Astronauts speaking publicly about the OASIS mission and the science mission and steps towards Mars

Many of these components have been used in the past, Congress has canceled programs they felt were over-budget, under-progressing, or not in the national interest. Regular, demonstrated success with critical mission components (SLS Block 1B, ISRU, sensor suite, EVA procedures in the NBL, habitat and docking module) should be publicized as much as possible.

International Collaboration:

The use of international partners will allow for greater project stability through combined funding support. In return for funding contributions to support science instruments, international partners will be given payload proportions according to total funding. Through collaborative funding, governments appear hesitant to remove project funding from budgets if international partners are affected, which may cause international embarrassment due to the high-profile of the mission.

Moreover, components, vehicles, and scientific payloads will be sourced from many international corporations and governments. Non-US sourced components or vehicles include:

- Orion Service Module (ESA)
- Robotic Arm (MDA, Canada)
- Neutron Spectrometer & Gamma Ray Spectrometer (Schlumberger, Worldwide)
- XRF (FUB, Germany)
- Pervaporation Test Cell (Pervatech BV, Netherlands)

- Seismic Array (Sensor Geophysical, Canada)

At present, because of the unique nature of this mission, the US is planned to be the primary operator as well as providing all the astronauts for this mission.

Policy Considerations:

Since the relocation of the orbit of an asteroid has never before been attempted, nor the consumption and use of natural resources of an asteroid, including up to the point where the asteroid has been completely destroyed, there exists several important legal considerations which need to be considered before launching such a mission. Fortunately, present legal interpretation leads to the conclusion that the mission represents the best interests of humanity and is therefore legal.

Policy concerning the utilization of outer space, including its natural resources, is governed by the UN Outer Space Treaty (1967), which in its preamble, states that "The exploration and use of outer space, including the Moon and other celestial bodies, shall be carried out for the benefit and in the interests of all countries, irrespective of their degree of economic or scientific development, and shall be the province of all mankind." In addition, "there shall be freedom of scientific investigation in outer space, including the Moon and other celestial bodies, and States shall facilitate and encourage international cooperation in such investigation."

Article II states that "Outer space, including the Moon and other celestial bodies, is not subject to national appropriation by claim of sovereignty, by means of use or occupation, or by any other means." While this would prevent the claim of sovereignty of a nation to all asteroids in space, judicious, peaceful, transparent, scientific use of an asteroid, even leading to its eventual destruction, is legally permitted, on the basis that conflicting resolutions in treaties shall not "[lead] to a result which is manifestly absurd or unreasonable" (1969 Vienna Convention on the law of treaties, Article 32).

The OASIS mission allays fears on the legality of the mission by being a multinational mission, inclusive of programs with and without space programs, which operates for peaceful exploration and use of space, in which resources collected are shared among participating nations. Therefore, OASIS should be politically acceptable even under the most conservative interpretation of Article II of the UN Outer Space Treaty. While HR 5063 attempts to clarify the role of the US government in licensing companies the right to mine asteroids commercially, this bill is outside the scope of this mission, as well as associated legal questions.

OASIS and Governmental Process:

The OASIS mission is consistent with President's stated mission of advancing technology to travel to Mars by mid-2030s and should pass the appropriations and authorizations committees, should cost be minimized. However, it is unclear whether the 2016 Presidential election may result in a change of space exploration vision, which could impact the authorization and appropriation of such a mission. Involvement of Public-Private partnerships as well as competitive elements are politically attractive at present, and will be pursued aggressively to ensure funding.

Future Uses of OASIS

OASIS' primary mission includes three weeks of objectives on orbit. The mission hardware remains and, with a successful mission, the potential for ISRU expendable replenishing exists. The horizon of the OASIS mission includes future missions to build out the OASIS system, conduct further investigation, and leverage technology to future, larger asteroid captures in similar orbits. ISRU propellant may be leveraged for missions to the surface of Mars, the Moon, or other asteroids.

Conclusion

For centuries, humans have stared up at the night sky and wondered what life would be like on other worlds. However, we know that the journey is fraught with danger and littered with unknowns. For humans to survive and thrive in outer space, we must adopt a new paradigm in the way we conduct our space missions, from increased radiation protection to in-situ resource utilization and Earth independence. For our survival in space, we must become true "astro-nauts", prepared to sail from the safety of our homeland to explore the new world. The OASIS mission serves as a critical stepping stone on this journey, between the shores of our planet and the unexplored lands of Mars.

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Appendix A

Full Science Traceability Matrix

Science Objectives	Measurement Objectives	Instruments	Weight (kg)	Power (W)	Cost ($M)
Primary					
Characterize internal structure and composition of the asteroid	Measure the acoustic response (seismic) of the interior	Seismometers	50	100	20
	Drill 1 m borehole	Drill + sensors	10	200	5
	Obtain core sample	Coring bit			
	Determine mineralogy and water form (μm penetration)	IR Spectrometer	1	50	15
	Determine lithology and water content (cm penetration)	Neutron Spectrometer	1	50	15
		Gamma Ray Spectrometer			
	Determine detailed mineralogy	XRF	1	50	15
Characterize space environment around the asteroid	Surface dust environment	Dust Detector	<1	<1	<1
	Plasma and magnetic field	Plasma Monitor	<0.1	1	1
		Magnetometer			
	Radiation and space weather impact	Radiation Access Detector	1.5	4	1
ISRU	Demonstrate feasibility of deep drill technology	Deep drill	10	200	10
	Processing and demonstration of resource utilization	Custom asteroid processing unit	240	1000	44
Human Health and Behavior	Monitor radiation	RAM	0.5	5	<1
		AN/UDR-13 Radiac Set	<0.01	-	<1
	Training effectiveness	Optical cameras mounted in living space	25	1	4
		Optical cameras mounted to external surface of vehicle		1	
	Psychology and Group Dynamics	Optical cameras mounted in living space	NA	NA	NA
		Paper questionnaires	NA	NA	NA
	Suit Testing	Heart rate monitor in suit	Included in human factors requirements		
		Blood pressure monitor in suit			
	Health Monitoring	CHeCS	Included in human factors requirements		
Human Decision Making	Successful completion of a task selected by the astronaut	Table of predetermined tasks	NA	NA	NA
Secondary					
Advance our understanding of the origin and evolution of the solar system	Measure the acoustic response (seismic) of the interior	Seismometers	50	100	20
	Drill 1 m borehole	Drill + sensors	10	200	5
	Obtain core sample	Coring bit			
Improve our current asteroid classification scheme	Determine mineralogy and water form (μm penetration)	IR Spectrometer	1	50	15
	Determine detailed mineralogy	XRF	1	50	15
Demonstrate planetary defense capabilities	Demonstrate feasibility of deep drill technology	Deep drill	10	200	10