Asteroid Mining
Space Vehicle Report
Red Team

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***Abstract*—This article presents the procedure and results of the Space Vehicle group for the project "Asteroid Mining Mission". It was developed by 5 students at KTH Royal Institute of Technology as part of the course SD2905 - Fundamentals of Human Spaceflight given by Christer FUGLESANG. The group objective was to conceptually design the spacecrafts (i.e. mining station, crew and cargo spacecrafts) required for this Phase 0 study. The study presents the requirements, assumptions, and methods employed in order to perform mission analysis (selection of trajectories and launchers, concept of operations) and to design the subsystems of these spacecrafts, including performing mass, power, and link budgets. This conceptual study does not only consider functional considerations but also includes operational, technological (i.e. Technology Readiness Levels (TRL)) and financial aspects.**

***Index Terms*—Aerospace, KTH, MSc., masters programme, rocket, space, human spaceflight, student**

I. INTRODUCTION

This study aims at designing the vehicles to bring material and humans to the asteroid, and back with minerals. Such a mission consist in a campaign of missions, both robotic and crewed, in order to deploy mining equipment, enable humans to start the mining operations and the logistic chain to function in autonomy. This study will answer questions such as: what is the optimal configuration of spacecrafts (types, sizes, numbers...) to reach the mission objectives? What will the system architecture of the space vehicles be? Which mass, power and link budgets?

A. Requirements

The requirements set in place for this project can be split into functional and non-functional. In general, the functional requirements are what the results shall do, or high level attributes, and non-functional describe lower level attributes.

Functional:

- The mission shall launch by 2030.
- The target asteroid shall be Kamo'oalewa ($2016H0_3$).
- The mission shall include astronauts.

Non-Functional:

- Mission cost should be minimised.
- TRL should be extrapolated reasonably.
- Human duration in space should be minimised in regards to radiation exposure and consumable needs.

B. Assumptions

Several assumptions were made to design the whole mission:

- An exploration of the asteroid had already been done, thus the composition of the small body is known.
- The trajectories were computed with a tool that assumes impulsive maneuvers and that uses a Lambert solver to calculate the transfer orbits.
- A general mass breakdown by National Aeronautics and Space Administration (NASA) was used to help the design of each subsystem.
- The European Space Agency (ESA) margin philosophy [3] was used for implementing mass margins at equipment, system and propellant level, as well as for ΔV margins in mission analysis. It is consistent with European Cooperation for Space Standardisation (ECSS) standards.

C. Structure of Report

This report firstly presents the methods used in mission analysis and the space vehicle design phases, going through each subsystem. Then, the results of these phases are presented next, before being discussed. A conclusion wraps up the work and invites to further analysis, in a later phase. Eventually the division of work is showed and the references listed.

II. METHODS

A. Trajectory Analysis

The design of the spacecraft and the selection of the launch vehicle started with the choice of the trajectory for the given mission. A preliminary analysis of the trajectory was made, comparing different options for the starting orbit. A direct transfer from low Earth orbit (LEO), geostationary transfer orbit (GTO), Lagrangian point L1 and an interplanetary transfer were evaluated. The design of the trajectory aim at bringing the highest possible payload to the asteroid, therefore the starting orbit for the transfer was picked following this criteria.
The mission layout was divided in three phases:

- Phase I: send mining equipment
- Phase II: send humans
- Phase III: send cargo ships

Each phase has peculiar requirements, thus three different trajectories had to be evaluated. Computing the trajectory with basic orbital mechanics was a hard task because of the variation of the asteroid orbit around the Sun and because of its relative distance from Earth, which varies with time. Therefore, an online tool by NASA was used to compute the mission design for the direct transfers. Thanks to the Trajectory Browser [1], the launch dates, ΔV and transfer time were computed. This tool, listing pre-calculated trajectories, is not able to compute low thrust transfers, nor gives any indications about the required ΔV for such a transfer. The Horizons Web-Interface [2] by NASA Jet Propulsion Laboratory was used to estimate the required change in velocity. The transfer time for the low thrust trajectory was calculated with the following equation 1.

$$\Delta t = \frac{Mass * \Delta V}{Thrust} \tag{1}$$

B. Launcher Trade Study

Selecting the launch vehicles for the three phases was not an easy task. Firstly, the payload mass that the different spacecrafts are supposed to bring to the asteroid was taken into account. The mining station needed to bring all the equipment for the mining operations, thus this mass was provided by the Logistics group. For the crew, the Human Aspects group supplied the mass for the life consumables and the mass for the Environmental Control and Life Support System (ECLSS). Their calculations were also affected by the length of the mission, which was decided during the trajectory analysis. This is an example of how the design of this mission was an iterative process. The cargo spacecrafts did not have to bring any payload to the asteroid, but they did have a payload to return to Earth, which contained the mined minerals. This value was once again provided by the Logistics Group. Secondly, the structural mass of the different spacecrafts was considered. This quantity depended on the several subsystems of each vehicle, that are discussed later in the report. The payload and the structure represent the dry mass of a spacecraft, thus to find the total wet mass that the launcher needs to bring to the parking orbit the propellant had to be evaluated. For all spacecraft, the propellant mass was calculated using the Tsiolkovsky rocket equation 2.

$$\Delta V = I_{sp} * g_0 * \ln \frac{m_{DRY} + m_{propellant}}{m_{DRY}} \tag{2}$$

where I_{sp} is the specific impulse of the selected engine and g_0 is the gravitational acceleration at Earth's surface. The ΔV for each phase was given by the selected trajectory. Regarding the crew vehicle, the propellant mass to return to Earth from the asteroid was also carried from the beginning of the mission, so it was added to the total wet mass that the launcher needed to lift to the parking orbit. Finally, given the total mass of each spacecraft, several launchers were considered, as shown in Figure 1.

Launcher	Payload capacity (kg)	Launch cost (M$)	kg/M$
SLS 2	130 000	2 000	65
Delta IV Heavy	28 790	475	61
Falcon 9	22 800	62	368
Falcon Heavy	63 800	150	425
Atlas V	18 800	153	123
Ariane64	21 650	134	162

Figure 1: Launchers payload capacity to LEO and cost

The selection of the launchers was based both on the payload capacity, on the cost per launch, and the launch site latitude compared to the declination launch asymptote (DLA).

C. Space Vehicle Design

The spacecraft design depended mainly on the mission duration, which affected the resources and materials and thus masses the other teams were required to bring. In order to keep this organised and to ensure that the total mass and power were within the chosen launcher constraints, a detailed spreadsheet was created to include every subsystem component within its respected spacecraft– cargo, crew, and mining station. The subsystems are broken down as followed in this section. Within each subsystem, a component was listed with its number of units, unit mass, and a "design maturity mass margin" (equipment margin) as a factor of 5, 10, and 20%. These margins added a percentage to the mass based on the amount of modifications the documented unit mass required, with 5% being "off-the-shelf." The dry mass of the vehicle was totaled from the subsystems and a 5% harness was added to estimate the wiring and hardware mass, then an additional 20% system mass margin. Lastly, the propellant mass was given a 2% residuals margin and then added to the dry mass for the total wet mass of each vehicle [3].

1) Propulsion: After the trajectory analysis, it was decided to have different propulsion systems for each phase as the requirements were not the same. For the mining station it was chosen to opt for electrical propulsion using SPT 140 Hall thrusters [4]. This choice was made to have a high thrust based on a limit of electrical power available. This electrical thruster uses Xenon as propellant and comes with a price of 850 $/kg [5]. Considering the use of the thruster, it was possible to choose some components of the propulsion subsystem considering what had been done with the SPT 100-D previously. Therefore, it was decided to use a Xenon Flow controller [6] with the power processing unit [7], and their quantities considering the mission's needs.
As for the crew and the cargo mission, it was determined to use chemical propulsion. Here, it was the I_{sp} that was the decision factor which lead to the use the Vinci engine based on the high thrust available, even if it has at high volume [8]. The engine can be used at full throttle for the crew vehicle because it is very heavy, reaching a maximum acceleration of $1/3\ g_0$, which is sustainable by the humans. For the cargo spacecraft, the engine is throttle down to 67 % [9] on the way forward, when there is no payload, in order to have an acceleration below 1 g_0, thus the solar arrays will not get damage. As for the propellant, it uses a mixture ratio of 5.8

between liquid oxygen and liquid hydrogen (LOX/LH2). The cost for this propellant is that LOX cost at 0.21 $/kg and LH2 cost at 5.5 $/kg based on Saturn V values. All the data for the propulsion thruster can be find in the Table I

Table I: Propulsion system for each phase

Parameters	Mining station	Crew vehicle	Cargo
Type of thruster	SPT-140D[19]	Vinci engine[1]	Vinci engine[1]
Thrust (total)	3.287 N	180kN	180kN
Cost per unit	6M$	25-30M$	25-30M$

2) Power: To design the power systems of the 3 spacecrafts, a first estimation was performed using the estimated required power for mining (to design the station) and the Orion and ATV power systems (respectively 11.2 kW and 4.8 kW) as reference for the crew and cargo spacecrafts [10] [11]. The ISS power system (80 kW average) was also studied to provide an order of magnitude to compare with. They served as scaling references. For the second, more accurate, estimation, a power budget was established from the product tree that was realized for the mass budgets. Hence, for each subsystem a power assessment was made: using the known power value of components or using an analog value from existing systems (e.g ISS, Orion, ATV, etc.). The Logistics group also provided the peak power for the mining equipment on the station (61 kW), while the electric propulsion design also stood for a major power driver (power being a bottleneck for the number of engines).

Once the required power identified, the design methodology used in [12] was used to calculate:

- The end of life (EOL) generated power, assuming a margin of 5% and a regulation efficiency of 0.93 (accounting for a power loss of 7% in the shunts)
- The beginning of life (BOL) generated power, assuming losses due to cell interaction with protons and electrons (18%), meteroids and UV rays (3%), failures (4%) and calibration errors (2%)
- The surface of solar arrays was then deduced, using GaAs solar cells from Alat (state-of-the-art of commercial lightweight efficient cells) and a power/mass ratio of 300 W/kg). The Conventional 3J Cells were selected [13].
- Eventually, the total mass of the solar panels were calculated.

The last step was to design the batteries for each spacecraft, especially the mining station. For the latter, a design value for the energy rating (energy to be stored by batteries) of 120 kWh was estimated considering the fact that the peak power is 61 kW and that charging cycles consist of 30 minutes (rotation period on the asteroid, i.e. duration when solar arrays do not produce much or at all).

3) Communications: In order to specify the communication architecture, the first step was to define the mission objectives and requirements. These were provided by the Coordination Group and consisted in:

- The crew should be able to communicate with Earth under all situations
- Video communication shall be an option in the nominal mode, with the crew
- The mining station and cargos shall be able to continuously share their telemetry and housekeeping data.

From these top-level requirements, the data rate could be firstly estimated: video transmission was assumed having a data rate of 10 s of Mbps [14]. For the calculations of the link budgets, a data rate of $10\ \mathrm{Mbps}$ was considered for the communication link of the crew vehicle. The data rate for the housekeeping and telemetry was taken as $8000\ \mathrm{bps}$ (approx. obtained from [14]). For all the communication links, a Bit Error Rate (BER) of about 10^{-5} has been targeted. In order to minimise the E_b/N_o ratio, a concatenated convolution and Reed-Solomon (interleave depth of 5) modulation scheme was considered for the data transfer.

In order to achieve the desired data rate for the crew vehicle, a high frequency transmission band will be required, given the fact that the transmission distance would range between 14 million to 40 million km. Therefore, the crew vehicle would communicate in the Ka frequency band. This will not only ensure high bandwidth but also support beyond line of sight requirements [15]. Since the housekeeping and telemetry requires a much lower data rate, communication in X band would suffice for the mining and cargo vehicle. Lower frequencies for this have been ruled out, due to the high transmission distance. For the link budget calculations, approximate figures for the frequencies in Ka and X band have been considered. The exact value would be decided through the approval of the International Telecommunications Union (ITU).

The results of the link budget and antenna dimensioning are presented in the Results section.

4) Structure: For the estimation of the structure, the calculation was based mainly on the mass of the fuselage on NASA estimation of 21.7% of the total mass. Then to increase the estimation, there were discussions with the other teams to know their expectations to have with the structure of the spacecraft of each phase. For the mining station, the mass of the anchoring system and the landing gears had to be included in the calculation, given by the Logistics team. Considering the crew spacecraft, it was also needed to estimate a heat shield to protect them for the reentry. This will be more explained in the heat shield design and the spacecraft atmospheric entry design parts. Also, the airlock had to be considered here and raw estimate of 10 tons had been found for the whole system. Besides, as it was considered that the airlock would take part in the fuselage mass estimation, the estimation of the mass of the rest of the fuselage was changed to half of the previous estimation so it would fit more the design of the configuration presented in the spacecraft configuration part. As for the cargo, the same estimation for the fuselage was made but it was also needed to consider the re-entry of the material and so it needed to add two heat shields for the precious metals.

Furthermore, for each phase tanks were considered to be in the structure part and so they have been designed considering the propellant. For the xenon for example, by checking companies providing xenon tanks for xenon thrusters, estimations for the mass and the volume needed were made. The estimates were given by MT aerospace [16] and the model S-XTA 120 was chosen but which is still under development but can be considered finished before the launch date. And so considering the mass of propellant already estimated a total mass based on the number of tanks for the Phase 1 was calculated. The LOX/LH2 tanks were designed based on the data from the space shuttle external, knowing the mixture ratio of 5.8. And so the tank weight and volume were scaled down [17] for the spacecraft mass budget.

5) Thermal: For the Thermal Control System (TCS) of the spacecrafts, the approach was to use similar technologies that are currently used on the ISS and other (un)crewed space vehicles, while leveraging on the advantages of future technologies (e.g lightweight radiators). The TCS architecture of all 3 vehicles is similar, while being more complex for the crew spacecraft due to the life support system to be considered.

The conceptual TCS architecture consist in a passive system reducing the need for cooling, and an active one. The passive one include Multi-Layer Insulation, coating and selected couplings, cold plates, heat exchangers, while the active one mostly consist in closed-loops fluid circuits (single phase) using ammonia, with tubing, pumps, tanks, valves, heaters, sensors, etc.

Both collect, transport and reject heat to space, by radiative transfer thanks to deployed or body mounted radiators.

The heat collection is ensured with cold plates and heat exchangers, the transport by lines, pumps and valves, and the rejection by radiators.

Depending on the spacecraft environment and mode, the active system can regulate the heat rejection.

To conceptually design the TCS, reference values were used from documentation, such as the design value of 30.2 kg/kW for dimensioning radiators [18].

A more detailed analysis considering the thermal environment, beginning and end of life configurations, as well as a thermal exchange model would be needed in a later phase of the project.

6) Environmental Control and Life Support System: The ECLSS is dealt by the dedicated team. The corresponding masses have been considered in the study.

7) Atmospheric entry: The atmospheric entry is a very critical moment which can become difficult to handle when the entry vehicle is heavy or when it is manned. Actually, the main issue is to cancel orbital velocity in order to land safely. The kinetic energy is fully transformed into heat that as to be evacuated from the vessel otherwise the mission fails.

a) Requirements: The crew vehicle and the cargo transporting respectively humans and precious metals need to be recovered on Earth and satisfy some requirements listed in Table II.

Table II: Atmospheric entry requirements

Parameter	Crew vehicle	Cargo
S/C mass	32.9 t	50.024 t
Maximum acceleration	8 g	/
Maximum entry duration	6h	/

b) Recovery strategy: Three strategies can be employed for the entry : the ballistic entry, the skip-glide and the aerobreaking maneuvers.

Ballistic entry is the easiest one and consists in a one-straightforward entry but has major drawbacks such as the enormous accelerations and heat generation.

The aerobreaking consists in performing several dozens of high altitude and low incidence ballistic entries without slowing down as much as needed to touchdown. After each atmospheric entry the vessel is slowed down by several meters per second. The main advantage of that maneuver is that it can slow down the vessel to low earth orbit without any heat shield. That maneuver can last up to several months and because of that it can not be used for the crew vehicle. That maneuver is not chosen for the cargo vehicle neither because for the final entry from low earth orbit, a heat shield is needed in any case, and because the vessel is so heavy that the maneuver would take too much time to be performed.

Finally, skip glide entry is a kind of mix between the two strategies: the recovery is done performing an atmospheric entry and re-entry. The first entry is done at very high altitude and at low incidence angle. Low drag gently slow down the vessel while its proper lift and Earth's curvature is bending upward the altitude trajectory until the vessel literally bounces on the atmosphere. After the atmospheric exit, the spacecraft slowed down enough to have a suborbital trajectory. Then a re-entry is performed to finalize the recovery. For that maneuver, a heat shield needs to be used to protect the vessel against heat. That strategy seems the most adapted one for our purpose. Figure 2 illustrates the concept.

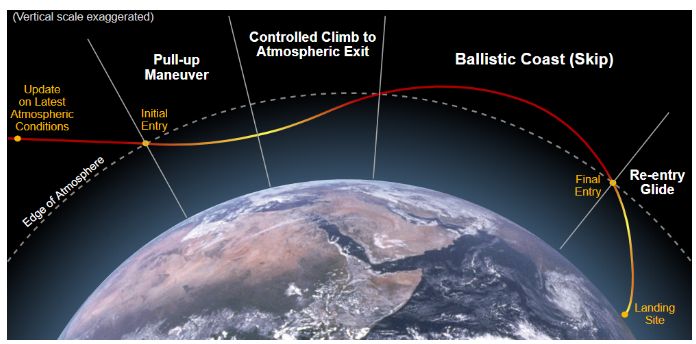

Figure 2: Skip re-entry strategy [19]

Because all kinetic energy is transformed into heat, the lighter is the spacecraft, the lighter will be the thermal protection. Thus, spacecrafts are split into lighter parts before entry and only the parts with interest are recovered : the precious metal tanks in the cargo, and the crew, which is located in the airlock (command module) of the manned spacecraft. The modules are separated with pyrotechnic charges because it is highly reliable. Figure 9 shows the spacecraft configuration for atmospheric entry.

c) Trajectory analysis: Equations of motion [20] were used to compute trajectory into the atmosphere . The entry altitude was set as commonly to 122 km, entry velocity to 11.9 km/s and the entry incidence angle was tuned for both spacecrafts. Then, heat load was computed also with thermal equations [20].

d) Heat shield design: The heat shield is made of a low density ablative material which has been tested in laboratory under different hypersonic flow conditions. An ablation rate $\dot{r}$ can be extrapolated from the data [21] as a function of stagnation heat load with an exponential model (see appendix C). The heat equations gave the heat load with respect to time. Heat load on the heat shield was considered constant equal to the maximum heat load during a characteristic time t_c. That duration was determined such that the total heat received by the shield was not changed from the real case with fluctuating heat flux, as summarized in equation 3.

$$\int q(t)dt = q_{max} \times t_c \tag{3}$$

With that definition the heat shield thickness δ was computed with equation 4 considering a security margin s of 20%.

$$\delta = s \times \dot{r}(q_{max}) \times t_c \tag{4}$$

III. Results

A. Trajectories

In this section the chosen trajectories are presented, and are later discussed in section IV of this report. For all transfers, the selected starting orbit is a LEO at an altitude of 200 km

1) Mining Station: The characteristic parameters of the transfer of the mining station are summarized in Table III.

Table III: Mining station trajectories results

Type of transfer	Low thrust
ΔV	6.03 km/s
Δt	4.6 years

The mining station does not come back to Earth, therefore there is no return trip trajectory.

2) Crew spacecraft: The manned spacecraft follows a direct transfer, since the travel time was minimized. After some months on the asteroid the crew returns to Earth. The mission design is shown in Figure 3.

Trajectory Itinerary

	Date	ΔV	
Earth Departure	Jan-09-2032	3.65 km/s	C3 = 9.6 km^2/s^2 DLA = 36°
112-day transfer			
Asteroid Arrival	Apr-30-2032	3 km/s	
112-day stay			
Asteroid Departure	Aug-20-2032	1.89 km/s	
80-day transfer			
Earth reentry	Nov-08-2032	-	12.24 km/s reentry
304-day total mission		4.9 km/s 8.55 km/s	post-injection ΔV total ΔV

Figure 3: Crew mission layout

3) Cargo spacecraft: The two cargo ships follow a similar direct trajectory. Both spacecraft return to Earth to transfer the minerals. The mission design for the first cargo sent to the asteroid is shown in Figure 4.
The layout for the second cargo can be seen in Figure 5.

Trajectory Itinerary

	Date	ΔV	
Earth Departure	Jun-06-2032	4.08 km/s	C3 = 19.4 km^2/s^2 DLA = -23°
310-day transfer			
Asteroid Arrival	Apr-12-2033	1.27 km/s	
30-day stay			
Asteroid Departure	May-12-2033	425 m/s	
200-day transfer			
Earth reentry	Nov-28-2033	-	11.97 km/s reentry
1.48-yr total mission		1.7 km/s 5.77 km/s	post-injection ΔV total ΔV

Figure 4: First Cargo mission layout

Trajectory Itinerary

	Date	ΔV	
Earth Departure	Nov-28-2032	3.84 km/s	C3 = 13.9 km^2/s^2 DLA = 27°
185-day transfer			
Asteroid Arrival	Jun-01-2033	1.67 km/s	
100-day stay			
Asteroid Departure	Sep-09-2033	439 m/s	
260-day transfer			
Earth reentry	May-27-2034	-	11.86 km/s reentry
1.49-yr total mission		2.11 km/s 5.95 km/s	post-injection ΔV total ΔV

Figure 5: Second cargo mission layout

B. Launcher Trade Study

1) Mining Station: The station has a very large wet mass, therefore it was decided to use two launchers. One launcher brings the dry mass into LEO, while the propellant is brought with the second launch. The station is then refueled in orbit, and then departs from Earth.
Selected launch vehicles:

- SpaceX Falcon Heavy for the dry mass
- SpaceX Falcon 9 for the propellant mass

2) Crew spacecraft: The crew vehicle has a much lower dry mass than the mining station, but it performs a direct transfer, thus the propellant mass is significantly higher. In addition, the spacecraft also contains the propellant for the return trip. For these reasons, two launchers were needed in this case as well. The dry mass an the propellant for the way forward are brought with the first launch, while the second launcher lifts the return propellant into LEO. The crew vehicle is then refueled in orbit.
Selected launch vehicles:

- SpaceX Falcon Heavy for the dry mass and the way forward propellant
- SpaceX Falcon 9 for the return propellant

3) Cargo spacecraft: The cargo vehicles do not have an actual payload on the way forward, therefore the dry mass to lift to LEO is only made of the structural mass. In addition, the return propellant is produced on the asteroid, making the wet mass that the launcher has to bring into LEO quite low. For this reason, one launch per cargo is sufficient. Both cargo vehicles are launched with SpaceX Falcon 9 rocket.

C. Spacecraft Configurations

The configuration shown in Figure 6 shows how the crew and cargo spacecraft dock onto the mining station anchored on the asteroid. Because the mining station isn't pressurized, the crew vehicle must dock onto it via the airlock positioned above the fuel tanks aft of the fuselage, hence the perpendicular position. The crew vehicle does not deploy its solar arrays during operations on the asteroid and instead uses power generated from the solar arrays on the mining station.

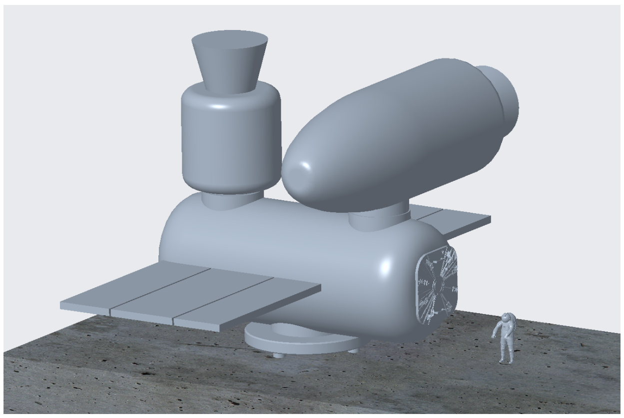

Figure 6: Spacecraft docking configuration

The mining station is initially anchored onto the asteroid using a docking ring and is the only module that is physically docked directly to the asteroid. The mining station dimensions are 5m in diameter and 13m in length to match the payload fairing of the Falcon Heavy which it is launched with. Figure 7 shows the relative size of the station with an astronaut standing beside, however much of the equipment used in operations is excluded from the schematic. The logistics team provides further explanation, but essentially, the collected metals are condensed and stored in containers as they wait to be loaded into the cargo vehicle through the docking port. The vehicle contains two cargo bays each with a heat shield for separate reentry, and propellant tanks for LH2 and LOX.

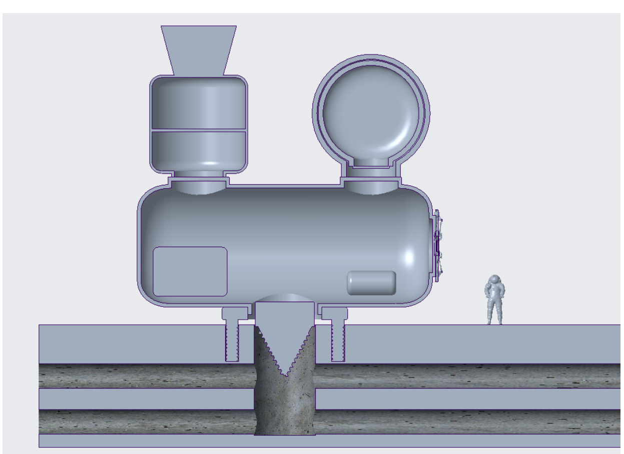

Figure 7: Mining station configuration

The section view shown in Figure 8 shows the relative positioning of the modules within the crew vehicle. As previously stated, the airlock docks with its respective port on the mining station in a perpendicular orientation. The propellant tanks are pictured along with the Vinci engine. The nose cone area of the craft is about 100 m^3 to accommodate the minimum habitat volume required by the three astronauts (75 m^3) with an additional volume reserved for equipment etc. To reduce mass on reentry, a heat shield is added to the airlock for use as a common module on reentry while the life support systems and entire nose cone are discarded prior. It is assumed that enough communications are included within the airlock, as well as seats and safety equipment for the astronauts to control the module and land properly.

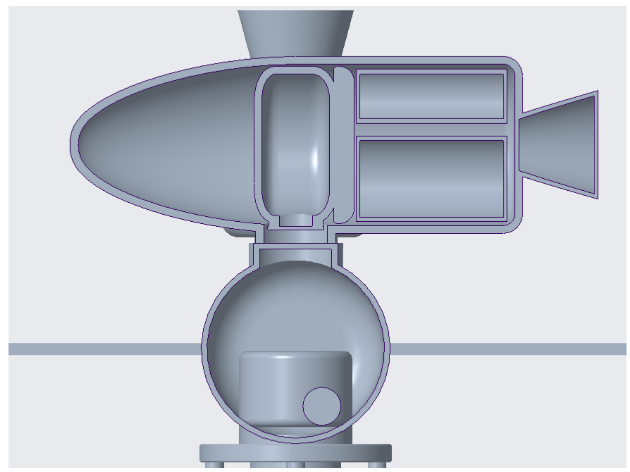

Figure 8: Crew vehicle configuration

D. Spacecraft Mass Budget

Adding mass margins on each equipment and harness, the nominal dry mass was obtained. Adding then system margins results in the total dry mass budgets for each spacecraft in Table IV. Here, it can be seen that the cargo vehicles have the same total dry mass but in Table V, the propellant masses differ. This is because of the slight difference in trajectory to the asteroid. It is also important to reiterate that the crew vehicle is the only vehicle which launches with its return propellant – the cargo vehicles utilise the propellant created on the asteroid (4 529 kg for the first cargo and 4 672 kg for the second cargo), and the mining station does not return.

Table IV: Dry mass budget for each spacecraft [kg]

Spacecraft	Nominal Dry Mass	Total Dry Mass
Mining Station	53 096	63 715
Cargo 1	10 349	12 418
Cargo 2	10 349	12 418
Crew	28 272	33 927

Table V: Wet mass budget for each spacecraft [kg]

Spacecraft	Propellant Mass	incl. Residuals	Total Wet Mass
Mining Station	16 770	17 100	80 815
Cargo 1	8 575	8 746	21 164
Cargo 2	8 707	8 881	21 299
Crew	37 944	38 703	72 630

E. Spacecraft Power Budget

The required power for the mining station, cargo and crew spacecrafts were respectively estimated to 65 kW, 8 kW and 20 kW.

The results of the power generation requirements and solar array dimensioning (area and mass) are showed in the Table VI.

Table VI: Power budget and solar array sizing

	Mining Station	Cargo S/C	Crew S/C
Required power [kW]	65	8	20
EOL Power [kW]	73	9	23
BOL Power [kW]	98	12	30
Solar Arrays Area [m^2]	306	38	94
Solar Arrays Mass [kg]	327	40	101

For batteries, Li-ion batteries of 5 kWh each were selected, and their number sized according to the spacecraft energy rating calculation: 120 kWh for the mining station, 10 kWh for both cargo and crew spacecrafts. Thus, 24 batteries (1 139 kg, equipment margin included) were selected for the station and 17 batteries (788 kg, equipment margin included) for both cargo and crew spacecrafts. Besides, 2 Power Distribution and Control Units (PDCU) were added, for redundancy.

F. Spacecraft Communication Budget

From the data rates explicited in the Methods section and the parameters described in VII, an estimated link budget was designed and the antenna diameters and transmission powers were obtained.

Table VII: Link Budget Parameters

Parameter	Value	Unit	Comment
Transmission distance	3.5×10^{10}	m	
Frequencies:	1.8×10^{10}	Hz	Ka-downlink
	8.4×10^{9}	Hz	X-downlink
Data rates:	10	Mbps	video
	8 000	bps	telemetry & housekeeping
Ground Station gain	74.16	dB	
Tx Antenna efficiency	0.7		assumed
T_s (Noise temp)	338	K	assumed
Other losses	3	dB	approx. [14]
Minimum E_b/N_o	2.4	dB	selected BER & modulation
Link Margin	3	dB	considered

Three types of antenna were proposed for the mission.

- High Gain Antenna (HGA): 1 HGA of Cassegrain type as the primary communication antenna with Earth. The HGA on the crew vehicle communicates in the Ka band (10 Mbps downlink, 100 W) and in the X band (8 Kbps downlink, 30W) for the mining station and cargo.
- Medium Gain Antenna (MGA): 2 MGAs, either Horn or Patch, to be used in the safe mode, in case the HGA is not available. They communicate in the X band (8 Kbps downlink, 30W).
- Low Gain Antenna (LGA): 3 dipole antennas to be used as a LGA in case of emergencies and Launch and Early Orbit phase (LEOP). Since, the LGA would be transmitting in a broad beam, it would still be able to communicate with the Earth, in case the spacecraft fails to point itself or for any other such emergencies. They communicate in the X band (8 Kbps downlink, 30W).

G. Spacecraft atmospheric entry design

Results of the atmospheric entry computations are summarized in Table VIII.

Table VIII: Heat shield design results

Parameter	Cargo	Crew	Unit
Vessel mass	24 555	1 200	kg
Vessel diameter	5	5	m
Vessel Lift/Drag	0.2	0.5	/
Vessel C_D	1	1	/
Entry incidence	6	7	°
Maximum G-Force	3.22	7.4	g
Entry duration	432	3.56	hours
Max. Heat load q_{max}	2297	2 100	W/cm^2
Max. Temperature T_{max}	2 309	2 300	°C
t_c	60.5	35	s
$\dot{r}$	2.88	2.10	mm/s
δ	21	9	cm
Heat shield mass	1 110	470	kg

The heat shields needs structural support to fix it. That structure increases the mass. The final mass budget retained for the heat protections is :

- 600 kg for crew mission
- 2 500 kg (2 times 1 250 kg) for the cargo

After atmospheric entry, vessels are recovered with parachute which allow soft landing especially for the manned vehicle. Landing is planned to happen on ground and not on the ocean like for Apollo mission because cargo spacecrafts, full of metals would have a high risk of sinking which cannot be taken. Landing site will be Utah desert (USA) like for Mars Sample Return mission (ESA). It will be used for both crew and cargo vessels because two different landing sites would add operation complexity.

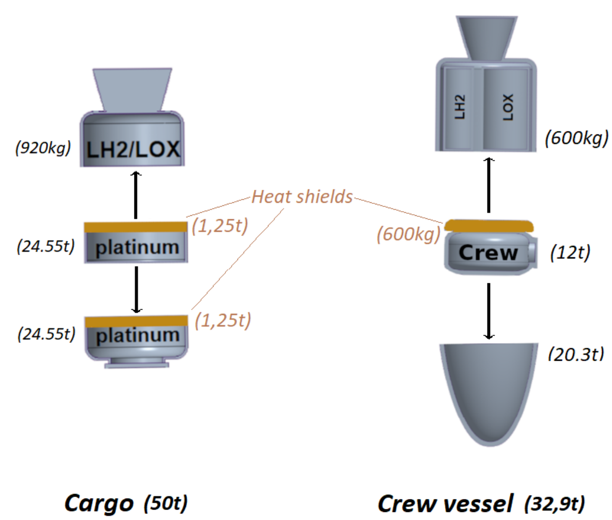

Figure 9: Spacecraft split before entry details

H. Operations

Figures 10 - 12 show the mission concept of operations on a timeline, split into the three main phases. Phase 0 is assumed to be the preceding exploratory mission which determined the composition and size of the asteroid as well as the landing site of the mission.

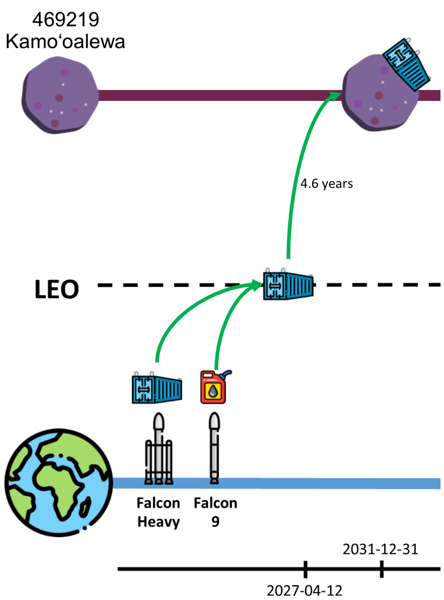

Figure 10: Phase I– Launch, assembly, and rendezvous of mining station to asteroid

Phase I, Figure 10, begins with the launch of the mining station on a Falcon Heavy into LEO where it is refueled and beings its journey to the asteroid. This spacecraft uses purely electric propulsion as time is not an issue for the equipment. The mining station autonomously rendezvous and anchors with its docking ring, sideways in respect to launch orientation, onto the asteroid awaiting the crewed efforts in starting mining operations.

Phase II, Figure 11, involves the launch of the crewed craft with a much shorter trajectory duration compared to the mining station. This spacecraft utilises both chemical and electrical propulsion in an attempt to minimise both travel time and propellant mass, using chemical on critical burns and electrical for coasting. A shorter travel time also reduces consumables required by the crew which in turn reduces mass. Once the crew vehicle docks to the mining station and begins operations, they remain on the asteroid for 112 days before ensuring the automatised operations are set and they are free to return to Earth.

In Figure 12, the continued operations make up Phase 3 after the crew has departed. Nearly a year after the crew has left, the first cargo craft arrives at the mining station to be loaded with precious metals mined on the asteroid. This sequence is repeated after the first cargo returns to Earth followed by the arrival of a second cargo. In both scenarios, the cargo uses chemical propulsion, some of which is produced on the asteroid. Phase III, and the entire mission, is concluded with the delivery of the second cargo craft to Earth on the 27^{th} of May, 2034.

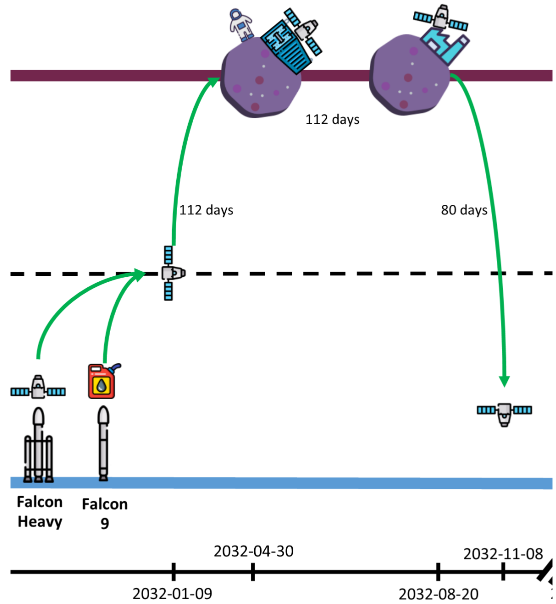

Figure 11: Phase II– Launch of crew and start of mining operations followed by return of crew to Earth

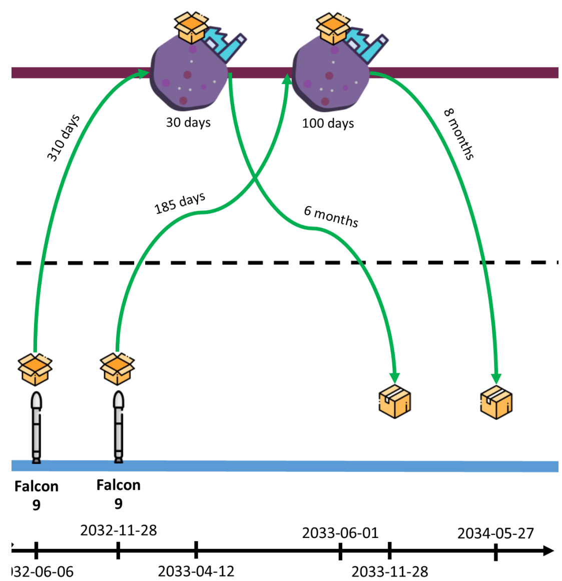

Figure 12: Phase III– Routine operations and delivery of cargo containing precious metals to Earth

The entire mission timeline can be visualised in Figure 13.

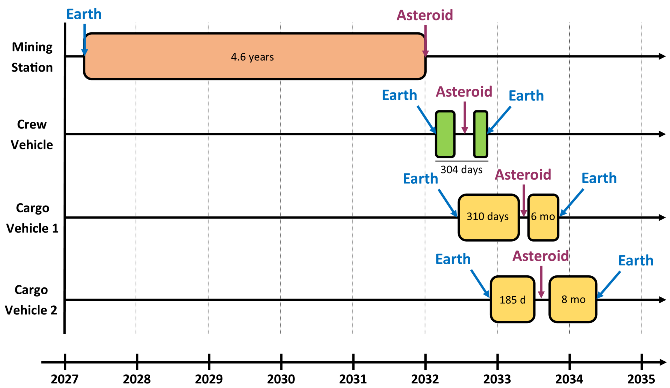

Figure 13: Mission timeline

IV. Discussion

A. Trajectory Analysis

In this section the selected trajectories are discussed. First of all, the parking orbit where each spacecraft is placed before leaving Earth's influence is a LEO at 200 km altitude. This orbit was selected because it enables the vehicles to bring as much payload as possible to the asteroid, and also because it is the one used by the Trajectory Browser tool. Then each spacecraft follows a different path to the asteroid.

The mining station is very heavy, but the requirement on the transfer time to reach the asteroid was not severe, therefore a low thrust transfer was chosen in order to save propellant mass. Therefore, the total cost of the travel was reduced. In addition, the propellant mass needed for a direct transfer would have been enormous, thus the number of launches to bring the total wet mass into LEO would have increased, making the total cost of phase one unacceptable. As stated in the results section, the station does not return to Earth once the mission is over, therefore no trajectory for the way back was designed.

The requirements on the manned vehicle made a direct transfer the only possible choice. In fact, the total mission time must be under one year in order to make it sustainable for the humans. Another requirement was to design a launch window that would give the humans enough stay time on the asteroid, so that they can set up all the mining operations. The crew spacecraft must also leave Earth only once the mining station has reached the asteroid, as decided by the Logistics group.

Regarding the cargo spacecraft, the time to reach the asteroid was not a problem since the vehicle can be launched while the mining is proceed, thus no time is lost. Instead it was desired to come back to Earth as fast as possible, in order to make the operation more profitable. It was possible to satisfy this requirement thanks to the following reasons. On the way forward, the cargo does not bring any payload, so only the structural mass has to be brought to the asteroid. Therefore, even though the ΔV is high, the low dry mass enabled the usage of a direct transfer. In addition, the return propellant is produced on the asteroid, as stated previously in the report. On the return trip instead, the cargo brings back the minerals, so the dry mass is significant. However, the return ΔV is a lot lower, which makes again a direct transfer feasible.

B. Launcher Trade Study

As for all the other aspects of this mission, also the selection of the launch vehicles differs for the three phases.

The mining station wet mass is significant, therefore it was possible to lift it in orbit at once only by using SLS Block 2 rocket. However, the launch for this vehicle is too expensive, probably over 2 000 M$ per launch [22], thus it was decided to bring the dry mass with Falcon Heavy, and later the propellant mass with Falcon 9. This strategy implies the in orbit refueling of the station, but it was supposed that such a technology would be ready by 2030.

A Falcon Heavy rocket is sufficient to launch the crew spacecraft into LEO, except for the propellant for the return trip. A Falcon 9 rocket was then chosen to bring this remaining mass. Once again an in orbit refueling was used.

Finally, each cargo spacecraft is launched with a Falcon 9 rocket. In this case, a single launch is sufficient to launch the total mass of the vehicle.

To conclude, only SpaceX rockets have been selected, since they are their launching site has a latitude (28.5°) close to the trajectories' Declination of Launch Asymptotes (DLA), which are 36°, –23°, 26°, and because the ratio between the payload capacity to LEO and the launch cost is the best one on the market, as shown in Figure 1 of the report.

C. Space Vehicle Design

The resulting spacecraft designs satisfy the initial requirements while incorporating redundancies and margins. Assumptions made on TRL developments of current technology allowed for use of new concepts such as refueling and near closed-loop life support systems which in turn reduced launch mass. Significant mass savings were also made on the power system by selecting state-of-the-art commercial lightweight solar panels (combined with high efficiency of GaAs cells) and Li-ion batteries. Mass was also conserved by separating the cargo craft into two reentry vehicles thus reducing the heat shield thickness. By continuously taking opportunities to save launch mass, cheaper launch vehicles were available making the mission more cost-effective and potentially profitable.

D. Off-nominal Case

For the off-nominal case the case studied was a missed launch date from the asteroid (due to uncompleted work or due to reparation needed). It has been estimated, based on the launch date and data from NASA for the trajectories between the asteroid and earth, that an another possible following launch windows that could be 1 month later. But this one would need a lower ΔV and so would take around 80 days to return.

Also an other redundancy that was considered with the logistic group was to assume propellant production on the asteroid that could be used then for more direct trajectories to come back but for now this technology is still under development but may be possible before the launch date.

A last but not least redundancy to be done is to have redundant parts from cargo/station but also a smaller backup engine in case if anything happens to the main one.

E. Sustainability Aspects

Sustainability is gaining importance as a key design factor, requiring the whole lifecycle and design process to integer such considerations: especially considering operational and disposal phases. For instance, the re-entry shall integer safety requirements for collision avoidance and debris risk mitigation. Ecodesign also applies to the production phase where the choices of technologies and materials (e.g rare earth minerals as Li, GaAs for the power system) is a major factor for dealing with sustainability aspects. Indeed, in the example of materials, their supply chain can be critical and need to be traceable, assessed with environmental and social impact factor, so as to be able to make responsible choices. Sustainability aspects to be considered in the design cycle also include to abide by the Planetary Protection standards that potentially could constraint the system design (although in the case of a mining mission, this risk should have been addressed before to minimize such requirements). Other aspects include the importance of regulations and the interaction between public organizations and private groups, the importance of locally produced resources and the value of bringing back minerals to Earth. These aspects are discussed more in detail by the Coordination group.

V. Conclusion

Mining an asteroid is a complicated mission. To fulfill that goal is an incredible optimisation problem to reduce the overall system masses and to shorten as much as possible the crew mission's duration. The final purpose is to reach a point where that mission is affordable, profitable and safe enough for the company to make profit and to attract investors. Space vehicle design has been studied here and a first overall architecture has been determined. Discussions and decisions were based on leveraging existing and near-term technologies as much as possible, but also by determining the best strategies to reduce the mission's mass and ΔV without compromising the safety of the mission. Detailed mass budgets, a launch and trajectory strategy, spacecraft overall designs and an atmospheric entry strategy result from that study that was optimised to fit in the mission requirements. Eventually, while this study is still conceptual, this is a first iteration that can be the starting point for future more detailed engineering development.

VI. Division of Work

For the division of work the task were separated as showed in the following Table IX

Table IX: division of work

Task	Team members
Trajectories	Alberto
Launch Trade study	Antoine & Alberto
Vehicle design	Sara & Adrien & Antoine
Re-entry	Maxime
Presentation and Report	Everybody

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APPENDIX A
SPACECRAFT BOM

Subsystems	Component	Number units	Mass per unit (kg)	Margin 5,10,20%	Mass w/ Margin
Structure			16,273		17,536.88
Thermal			3,162		3,349.80
Communication			128		162.23
CDH			20		42.00
Propulsion			26		219.47
GNC			48		164.40
Power			389		1,534
	Solar arrays	1	327	10	360
	Batteries (Li-ion 5kWh each)	24	45.2	5	1,139
	PDCU	2	17	5	36
S/C Dry Mass					27,611
Payload			22,907		27,559
Nominal Dry Mass before harness (=without system margins)			50,568		
Harness (5% of nominal dry mass)			2,528		
Nominal Dry Mass (=without system margins)			**53,096**		
Total Dry Mass with system margins (+20%)			**63,715**		
Propellant Mass (with 2% residuals)			17,100.00		
Total Wet Mass			80,815		0.00

Figure 14: Mining Station Mass Budget

Subsystems	Component	Number units	Mass per unit (kg)	Margin 5,10,20%	Mass w/ Margin
Structure			15,441		16,372.06
Thermal			1,111		1,156.86
Communication			63		93.98
	HGA (antenna & RF parts)	1	25	5	26.25
	HGA Pointing mechanisms	1	11	5	11.55
	Medium gain antenna (MGA	2	20	5	42.00
	Low Gain Antenna (LGA)	3	1	5	1.58
	TWTA	2	3	5	6.30
	Deep space transponders	2	3	5	6.30
CDH			20		42.00
Propulsion			550		577.50
ECLSS			5,592		5,220.75
GNC			48		164.40
Power			868		916
Payload			2,031		2,383
	Humans	3	80	5	252.00
	Water to bring	1	456	10	501.60
	Food to bring	1	638	10	701.80
	Clothing to bring	1	310	5	325.50
	Packaging	1	547	10	601.70
Nominal Dry Mass before harness (=without system margins)			26,926		
Harness (5% of nominal dry mass)			1,346		Power:
Nominal Dry Mass (=without system margins)			**28,272**		
Total Dry Mass with system margins (+20%)			**33,927**		Power w/ 20%
Propellant Mass			37,931.00		
Propellant margin (residuals, 2%)			38,689.62		
Total Wet Mass			72,616.55		

Figure 15: Crew Spacecraft Mass Budget

Subsystems	Component	Number units	Mass per unit (kg)	Margin 5,10,20%	Mass w/ Margin
Structure			6,726		7,437.43
Thermal			565		622.75
Communication			128		162.23
CDH			20		42.00
Propulsion			550		577.50
GNC			48		164.40
	Star and sun trackers	6	5	5	31.50
	Pin point landing system	1	20	20	24.00
	Ring laser gyroscope	3	3	10	9.90
	Accelerometer	6	10	10	66.00
	(use HET thrusters for control)				0.00
	Reaction wheels	3	10	10	33.00
Power			807		850
Payload			50000		0
	(minerals, only for return phase budget)		50000	5	0
					0
					0
					0
Nominal Dry Mass before harness (=without system margins)			9,856		
Harness (5% of nominal dry mass)			493		
Nominal Dry Mass (=without system margins)			**10,349**		
Total Dry Mass with system margins (+20%)			**12,418**		
Propellant Mass			8,821.00		
Propellant margin (residuals, 2%)			8,997.42		
Total Wet Mass			21,415.86		

Figure 16: Cargo Mass Budget

Appendix B
Spacecraft Mission Layout

SELECTED OPTION: ELECTRIC PROPULSION				**FALCON HEAVY**		
				S/C wet mass (kg)	**63800**	
Isp EP (s)	1967					
DeltaV EP (km/s)	6,03	LEO TO ASTEROID				
				Propellant mass (kg)	17100	
				S/C Wet mass in LEO (kg)	**80815**	
				Structure mass (kg)	**41556**	
				Payload mass (kg)	**22159**	
Total electric thrust (N)	3,287			Total dry mass (kg)	**63715**	
# thrusters SPT 140	19					**Earth Departure: Apr-12-2027**
Single thruster thrust (N)	0,173			Transfer time (years)	**4,7**	**Asteroid Arrival: Dec-31-2031**
Elec_Power consumption (kW)	57					
LAUNCHER SELECTION : 1 FALCON HEAVY FOR DRY MASS. 1 FALCON 9 FOR PROPELLANT. LAUNCHES COST = 230 $M						

Figure 17: Mining station mission specifics

SELECTED OPTION: CHEMICAL PROPULSION				**FALCON HEAVY**		
				S/C wet mass (kg)	**63800**	
Isp CP (s)	465					
DeltaV forward (km/s)	6,65	LEO TO ASTEROID				
DeltaV back (km/s)	1,89	ASTEROID TO GROUND		Propellant way forward (kg)	26551	
				Return propellant (kg)	11393	
VINCI ENGINE THRUST (N)	180000			Total propellant mass (kg)	38703	
				S/C Wet mass in LEO (kg)	**72630**	
LAUNCHER SELECTION				Structure mass (kg)	**31544**	
	1 FALCON HEAVY FOR DRY + PROPELLANT			Payload mass way forward (kg)	**2383**	
	1 FALCON 9 FOR REMAINING PROPELLANT			Payload mass way back (kg)	**1383**	
	TOTAL LAUNCH COST = 230 $M			Total dry mass (kg)	**33927**	
THROTTLE PERCENTAGE	1			Transfer time way forward (days)	112	**Earth Departure: Jan-09-2032**
FINAL ACCELERATION (m/s^2)	2,66	**OK FOR HUMANS**		Stay time at the asteroid (days)	112	**Asteroid Arrival: Apr-30-2032**
				Transfer time way back (days)	80	**Asteroid Departure: Aug-20-2032**
				Total mission time (days)	**304**	**Earth reentry: Nov-08-2032**

Figure 18: Crew vehicle mission specifics

SELECTED OPTION: CHEMICAL PROPULSION				**FALCON 9**		
				Wet mass capacity (kg)	**22800**	
Isp CP (s)	465					
DeltaV Forward (km/s)	5,35	LEO TO ASTEROID				
DeltaV Back (km/s)	0,425	ASTEROID TO GROUND		Propellant Way forward (kg)	8746	
				S/C Wet mass in LEO (kg)	**21164**	
VINCI ENGINE THRUST (N)	120600			Return propellant (kg)	4529	
				Structure mass (kg)	**12418**	
LAUNCHER SELECTION : 1 FALCON 9. LAUNCH COST = 80 $M				Payload mass (kg)	**37500**	
THROTTLE PERCENTAGE WAY FORWARD	0,67			Transfer time way forward (days)	310	**Earth Departure: Jun-06-2032**
ACCELERATION WAY FORWARD (m/s^2)	9,71	**OK FOR ARRAYS**		Stay time at the asteroid (days)	30	**Asteroid Arrival: Apr-12-2033**
ACCELERATION WAY BACK (m/s^2)	3,61	**OK FOR ARRAYS**		Transfer time way back (days)	200	**Asteroid Departure: May-12-2033**
				Total mission time (years)	**1,5**	**Earth Reentry: Nov-28-2033**

Figure 19: Cargo 1 mission specifics

SELECTED OPTION: CHEMICAL PROPULSION			**FALCON 9**		
			Wet mass capacity (kg)	**22800**	
Isp CP (s)	465				
DeltaV Forward (km/s)	5,51	LEO TO ASTEROID			
DeltaV Back (km/s)	0,439	ASTEROID TO GROUND	Propellant Way forward (kg)	8881	
			S/C Wet mass in LEO (kg)	**21299**	
VINCI ENGINE THRUST (N)	120600		Return propellant (kg)	4672	
			Structure mass (kg)	**12418**	
LAUNCHER SELECTION : 1 FALCON 9. LAUNCH COST = 80 $M			Payload mass (kg)	**37500**	
THROTTLE PERCENTAGE WAY FORWARD	0,67		Transfer time way forward (days)	185	**Earth Departure: Nov-28-2032**
ACCELERATION WAY FORWARD (m/s^2)	9,71	**OK FOR ARRAYS**	Stay time at the asteroid (days)	100	**Asteroid Arrival: Jun-01-2033**
ACCELERATION WAY BACK (m/s^2)	3,61	**OK FOR ARRAYS**	Transfer time way back (days)	260	**Asteroid Departure: Sep-09-2033**
			Total mission time (years)	**1,5**	**Earth Reentry: May-27-2034**

Figure 20: Cargo 2 mission specifics

Appendix C
Ablation rate for heat shield design

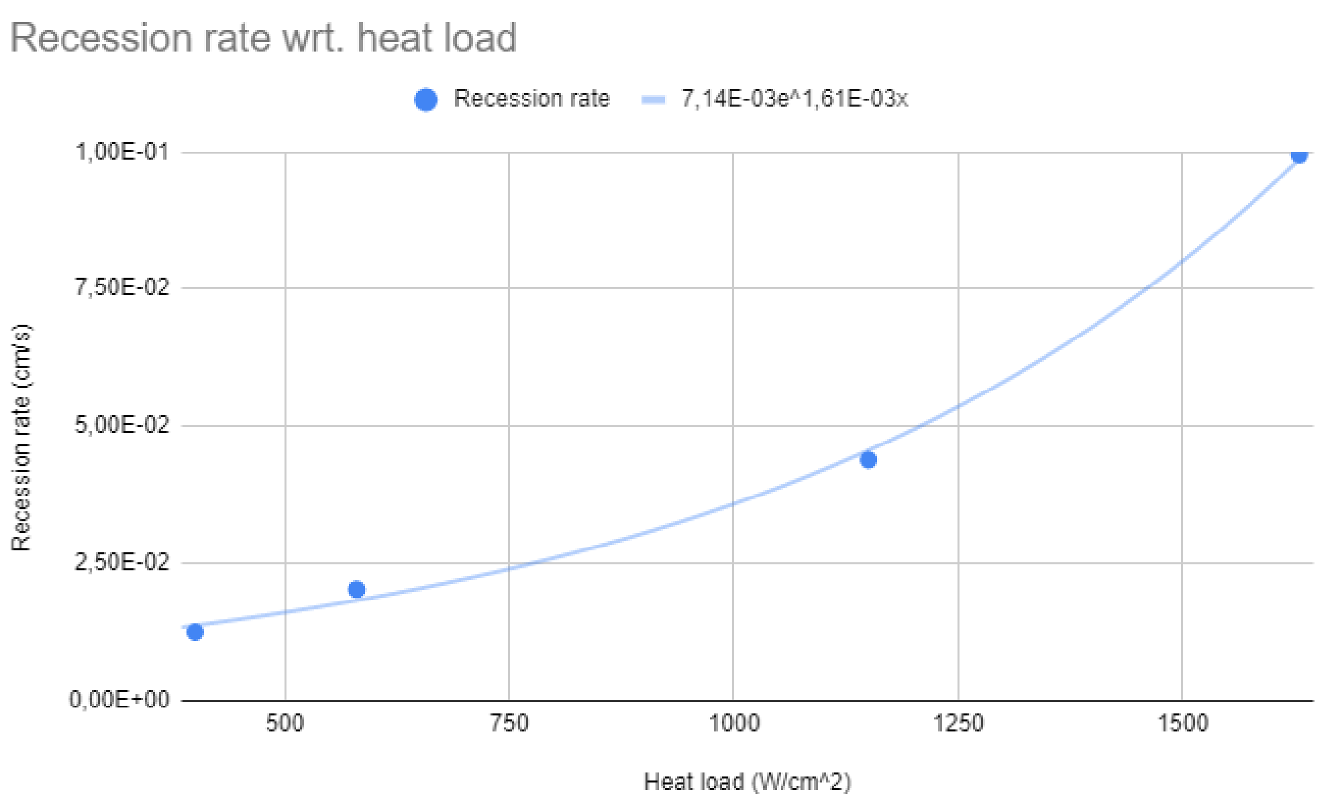

Figure 21: Ablation rate of low density ablative material

APPENDIX D
RE-ENTRY MATLAB CODE

A. Crew vehicle trajectory design

```
clear all
close all

%% Time resolution
dt = 1;
T = [0];

%% Vessel characteristics:
m = 12e3;
D = 5;
A = pi*D^2/4;
AoM = A/m;

LoD = 0.5;
CD = 1;
rn = 10;
eps = 0.8;

%% Constants:
C1 = 1.83e-8;
C2 = 3;
C3 = 4.736e4;
b = 1.22;
h1 = 260000; % Earth freestream enthalpy
const = struct('g',@g,'C1',C1,'C2',C2,'C3',C3,'a',@a,'b',b,'rho',@rho,'f',@f,'CD',CD,
    'LoD',LoD,'AoM',AoM,'dt',dt);
sigma = 5.67e-12;

%% Initial conditions:
V_Entry = 11900;
Gamma_Entry = -7 * pi/180;
H_Entry = 122e3;

U = [V_Entry ; Gamma_Entry ; H_Entry];

%% resolution
i = 1;
while U(3,i)>0 && U(3,i)<100000e3
    T = [T T(i)+dt];
    U = [U RK2(U(:,i),@dF,const)];
    i = i+1;
end
N = i;
V = U(1,:);
Gamma = U(2,:);
H = U(3,:);

%% G-Force
G_Force = [0];
for i=1:N-1
    acc = (V(i+1)-V(i))/dt;
    g = acc*sqrt(1+LoD^2)/9.81;
    G_Force = [G_Force g];
end
```

```
%% Thermal aspects*
qc = zeros(1,N);
qs = zeros(1,N);
qr = zeros(1,N);
Tw = zeros(1,N);
hw = zeros(1,N);
ht = zeros(1,N);
A = zeros(1,N);
for i=1:N
    ht(i) = 1/2*V(i)^2 + h1;
    qs(i) = C1*sqrt(rho(H(i))/rn)*V(i)^C2;
    Tw(i) = (qs(i) / (sigma * eps))^0.25;
    hw(i) = 940 * Tw(i) + 0.1043 * Tw(i)^2;
    qc(i) = qs(i) * (1-hw/ht);
    A(i) = a(V(i),rho(H(i)));
    if (H(i)<72000 && H(i)>40000 && V(i)>9000 && V(i)<16000)
        qr(i) = C3 * rn^A(i) * rho(H(i))^b * f(V(i));
    else
        qr(i) = 0;
    end
end

qt = qr + qc;
Qt = dt*sum(qt);
qt_max = max(qt);
Ablation_time = Qt/qt_max
Ablation_heat_load = qt_max
f = 1;
figure(f)
hold on
plot(T,qc)
plot(T,qr)
plot(T,(qc+qr))
legend('Qc','Qr','Qc + Qr')
xlabel('Time (s)')
ylabel('Heat flux W/cm^2')
hold off

f = f+1;
figure(f)
plot(T,Tw-273)
xlabel('Time (s)')
ylabel('Wall temperature ( C )')
hold off

f = f+1;
figure(f)
plot(T,H/1000)
xlabel('Time (s)')
ylabel('Altitude (km)')

f = f+1;
figure(f)
plot(T,G_Force)
xlabel('Time (s)')
ylabel('G Force (g)')
```

```

f = f+1;
figure(f)
plot(T,V)
xlabel('Time (s)')
ylabel('Velocity (m/s)')
```

B. Crew vehicle trajectory design

```
clear all
close all

%% Time resolution
dt = 5;
T = [0];

%% Vessel characteristics:
m = 24555;
D = 5;
A = pi*D^2/4;
AoM = A/m;

LoD = 0.2;
CD = 1;
rn = 10;
eps = 0.8;

%% Constants:
C1 = 1.83e-8;
C2 = 3;
C3 = 4.736e4;
b = 1.22;
h1 = 260000; % Earth freestream enthalpy
const = struct('g',@g,'C1',C1,'C2',C2,'C3',C3,'a',@a,'b',b,'rho',@rho,'f',@f,'CD',CD,
    'LoD',LoD,'AoM',AoM,'dt',dt);
sigma = 5.67e-12;

%% Initial conditions:
V_Entry = 11900;
Gamma_Entry = -6 * pi/180;
H_Entry = 122e3;

U = [V_Entry ; Gamma_Entry ; H_Entry];

%% resolution
i = 1;
while U(3,i)>0 && U(3,i)<200000e3
    T = [T T(i)+dt];
    U = [U RK2(U(:,i),@dF,const)];
    i = i+1;
end
N = i;
V = U(1,:);
Gamma = U(2,:);
H = U(3,:);

%% G-Force
G_Force = [0];
for i=1:N-1
```

```
    acc = (V(i+1)-V(i))/dt;
    g = acc*sqrt(1+LoD^2)/9.81;
    G_Force = [G_Force g];
end


%% Thermal aspects*
qc = zeros(1,N);
qs = zeros(1,N);
qr = zeros(1,N);
Tw = zeros(1,N);
hw = zeros(1,N);
ht = zeros(1,N);
A = zeros(1,N);
for i=1:N
    ht(i) = 1/2*V(i)^2 + h1;
    qs(i) = C1*sqrt(rho(H(i))/rn)*V(i)^C2;
    Tw(i) = (qs(i) / (sigma * eps))^0.25;
    hw(i) = 940 * Tw(i) + 0.1043 * Tw(i)^2;
    qc(i) = qs(i) * (1-hw/ht);
    A(i) = a(V(i),rho(H(i)));
    if (H(i)<72000 && H(i)>40000 && V(i)>9000 && V(i)<16000)
        qr(i) = C3 * rn^A(i) * rho(H(i))^b * f(V(i));
    else
        qr(i) = 0;
    end
end

qt = qr + qc;
Qt = dt*sum(qt);
qt_max = max(qt);
Ablation_time = Qt/qt_max
Ablation_heat_load = qt_max
f = 1;
figure(f)
hold on
plot(T,qc)
plot(T,qr)
plot(T,(qc+qr))
legend('Qc','Qr','Qc + Qr')
xlabel('Time (s)')
ylabel('Heat flux W/cm^2')
hold off

f = f+1;
figure(f)
plot(T,Tw-273)
xlabel('Time (s)')
ylabel('Wall temperature ( C )')
hold off

f = f+1;
figure(f)
plot(T,H/1000)
xlabel('Time (s)')
ylabel('Altitude (km)')

f = f+1;
```

```
figure(f)
plot(T,G_Force)
xlabel('Time (s)')
ylabel('G Force (g)')

f = f+1;
figure(f)
plot(T,V)
xlabel('Time (s)')
ylabel('Velocity (m/s)')
```

C. *Functions used*

```
function y = a(V,rho)

y = 1.072e6 * V^(-1.88) * rho ^(-0.325);

end
```

```
function y = f(V)
VE = 1000*[16 15.5 15 14.5 14 13.5 13 12.5 12 11.5 11 10.75 10.5 10.25 10 9.75 9.5
    9.25 9 0];
fE = [2040 1780 1550 1313 1065 850 660 495 359 238 151 115 81 55 35 19.5 9.7 4.3 1.5
    0];
y = interp1(VE,fE,V);
end
```

```
function dU = dF(U,const)
V = U(1);
Gamma = U(2);
H = U(3);
rho = const.rho(H);
g = const.g(H);
R = H + 6371e3;
dV = -rho/2 * const.CD * const.AoM * V^2 - g * sin(Gamma);
dGamma = rho/2 * const.CD * const.AoM * V * const.LoD - (g - V^2/R) * cos(Gamma)/V;
dH = V * sin(Gamma);
dU = [dV ; dGamma ; dH];
end
```

```
function y=g(h)
g0 = 9.81;
y = g0/(1+h/6371e3)^2;
end
```

```
function y = rho(h)
y = 1.225 * exp(-h/8.4345/1000);
end
```

```
function Up = RK2(U,F,const)
dt = const.dt;
k1 = F(U,const);
k2 = F(U + dt/2 * k1,const);
k3 = F(U + dt/2 * k2,const);
k4 = F(U + dt * k3,const);
Up = U + dt/6 * (k1 + 2*k2 + 2*k3 + k4);
end
```