Hitchhiker Space Taxi Launch & Return **Team Red**

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***Abstract*—The aim of this project is to develop and plan the launch and return system of a two seat space taxi in the form of a space capsule. The focus is to find a functional, safe and sustainable way to launch the vehicle, maneuver in space for rendezvous and docking and return to Earth. What has been found is that the capsule's engines will use the fuel HTP/Kerosene. The launch vehicle will initially be Ariane 6 which will take us to orbit from Kourou. Return and landing will be performed with parachutes and main engines in Algerian desert. With the planning of the recovery the total turnover time for the space taxi will initially be 163 days. The safety of the mission was achieved with redundancies of systems and abort scenarios.**

I. Introduction

The space industry is rapidly evolving and although the end of ISS is closing in many more space stations are being developed. The increase of space stations brings the desire of vehicles that are able to transport humans in a safe and simple way. Hitchhiker, the space taxi is a European independent project that is set to carry two persons to and from different space stations that are in Earth's orbit. The project is divided into four project groups: overall coordination, human aspects, vehicle design and launch and return. From the other groups certain parameters are set and the mission for this project group is to with these parameters research how the space taxi can be propelled which includes what fuel and engines to use, how to launch the Hitchhiker into orbit and from where to do this. Also research the system for the landing sequence and lastly how the turnover will be done. During this research safety, environmental impacts and certain political aspects will be considered and govern the choices.

From an initial discussion of if the taxi should be a capsule or space plane the choice was that the vehicle will be a capsule and the following parameters were set:

- A dry mass of 4420 kg
- A radius of 2.09 m
- A height of 3.50 m

As the information of the future space stations are limited estimations will be done on the currently operational ISS and its orbit. The orbit and inclination of ISS will be the target for Hitchhiker but with reserves to reach a higher orbit or another inclination if necessary. Other assumptions and estimations of i.e. ΔV losses for the launch will be made with previous data from NASA, SpaceX and Roskosmos and stated throughout the report.

The outline of this report is to present the research done in chronological order of how it was made. Starting with what fuel was used which lead to the decision of engines. With the knowledge of engines and fuel the launch, orbit and docking was researched which lead related to the later subject of landing. The research ended with the turnover which brings the circle to a close and the report is concluded with a conclusion.

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A. Fuel

Regarding the fuel decision multiple options were explored.

First the choice of what types of engine and propulsion system should be incorporated in the capsule had to be made.

The different engine types are the following:

- Solid
- Hybrid
- Liquid

Solid engines: A solid engine has many benefits the main point being simplicity. It is usually a straight forward deign of a combustion chamber filled with a solid propellants and a nozzle for directing the thrust. [1] This makes the manufacturing and maintenance of the engine more effective which leads to cost reduction. The simplicity also makes the system more reliable and less prone to complications due to ignition errors.

Another important accept is its stability. Solid propellants are often stable and can be stored for longer periods. Often the fuel and oxidizer are contained in the same molecule or in a homogeneous mixture. However solid fuel propellants tent to have quite low ISP and be very toxic. Therefore it was decided that the choice should be between liquids and hybrids.

Hybrids: Hybrids are more complex in their design and can have either liquid or gaseous oxidizer and a solid fuel. They tend to have higher ISP than solids but lower than liquids.[2] The benefits of hybrids are many, Throttleability being an important aspect. This means that the flow rate of the oxidizer can be adjusted, allowing for more precise maneuvering. Regarding fuel there is both toxic and non toxic options.

Liquids: Liquid engines are the most complex design but they are very commonly used typically in missions requiring higher ISP. Liquids are the most efficient compared too the other engine types. Many bi-liquid engines can be throttled, and here there is also both toxic and non toxic options. [3]

Having considered all the options solid were excluded earlier in the process. The choice between hybrids and liquids were mainly based on what type of fuel was most beneficial for our mission.

The constraining factors were discussed in the group and a decision was made that the fuel needed to fill these criteria below.

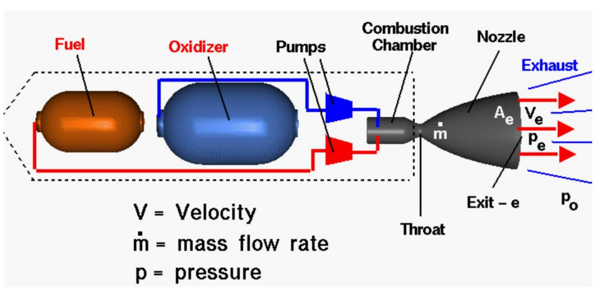

Fig. 1. Liquid engine, [4]

1) Non-toxic
2) Cost effective
3) Storagability
4) Usable for the main system and the maneuvering thrusters

Amongst the different fuel/oxidizer combination for hybrids, N_2O and paraffin was chosen. Both N_2O and paraffin are common materials, which means they are quite cheap. It is used in many student rocketry organization due to being non toxic and easily manageable. N_2O is self pressurizing which means that it needs to be kept at low temperatures or high pressures to not vaporize.

This option was firstly explored. Typically hybrids are not used in missions carrying high payload mass due to generally not being able to be as effective as bi-liquids. Increasing research is being conducted in to the field due to hybrids having a couple of advantages over bi-Liquids. Handling two fluids required more plumbing and a more complex design which generally tend to make the engine more expensive. Hybrids have liquid or gaseous oxidizer while the fuel is solid, this means that the design tends to be less complex compared to liquids.

A company was found that is exploring using hybrids for space missions carrying higher payloads. HyImpulse designs their own hybrids and values from their engines was taken to try to approximate if it could work for our payload mass.

To be able to do these calculations we made assumptions regarding the mass of the capsule and the acceleration. The assumptions were made based on data from the crew dragon mission which will be presented later in this report. Doing this calculation it was decided that 8 Hyimpulse engines would be sufficient for launch/landing stages which is feasible for the design. [5]

However the purpose of an incorporated propulsion system in the capsule was also maneuvering abilities in orbit. Therefor the next step was to explore if N_2O could be used as a cold gas thruster and it was seen that it could not produce sufficient thrust. In the light of this information the idea of a hybrid did not seem feasible hence reaching the decision of having a bi-liquid engine.

There are many possible combinations of oxidizer/ fuel and many were considered, but looking through the options it narrowed down to two combinations.

LOX/LH2: Liquid oxygen have many benefits. It is quite cheap compared to other propellants and is quite commonly used in many missions, like Space shuttle. [6] It has non toxic byproducts and also one of the highest specific impulses which makes it a really efficient fuel option. [7]

Both LOX and LH2 (Liquid hydrogen) is cryogenic, which means they have really low boiling points. Liquid oxygen has a boiling point of -183C. This means that it needs to be kept insulated to maintain the low temperature. This poses quite a challenge when it comes to handling and storagebility. [8]

Regarding the constraining factors this combination was quite promising, It is non toxic, relatively cheap, high ISP (285s) but regarding storagebility it is quite difficult.

Hydrogen peroxide/Kerosine: The hydrogen peroxide/kerosine option is not a commonly used fuel but it was used in the gamma 2 engine that will be referenced later on.

Looking through all possible storable options seen in the figure below hydrogen peroxide/ kerosine was decided to be the best one. It is a cost efficient fuel/oxidizer combination and can be easily purchased. The performance rate is not as high as the Liquid oxygen/hydrogen option but it is sufficient, it has an ISP of 265s which is enough for our purposes. Main factor is that this fuel/oxidiser combination was the only one of the storable options that was non toxic, which was a constraint that was highly considered in the decision making. It is also hybergolic which means that it has a self ignition ability, which also reduces the complexity of the system and it can be easily reignited if any issues occur regarding ignition. One other important factor is that hydrogen peroxide can be used as a monopropellant for the orbital maneuvering, something the hybrid option could not provide. [9]

There is some issues regarding handling. If hydrogen peroxide comes in contact with the wrong materials it can lead to rapid decomposition, which means that it will require proper ventilation.

Considering all the factors this was the fuel that filled all our requirements and deemed to be the most appropriate choice.

Table 1. Properties of environmentally safe fuels

Fuel	K_{m0}	K_m	ρ_{fu}, kg/m^3	I_{un}, s	ρ_{ox}, kg/m	ρ_{cc}, kg/m^3	Δv, m/s
$O_{2cc} + CH_{4cc}$	4.000	3.600	834.7	401.60	1140	425	6620
$O_{2cc} + C_{10}H_{20}$	3.400	3.060	1040.0	372.10	1140	820	6134
$O_{2cc} + C_2H_6O$	2.087	1.873	987.6	315.96	1140	790	5009
$H_2O_2 + C_{10}H_{20}$	7.285	6.500	1294.0	336.88	1420	820	5553
$H_2O_2 + CH_4$	8.500	7.650	1024.0	344.50	1420	425	5678
$H_2O_2 + C_2H_6O$	4.434	4.000	1225.0	327.80	1420	790	5009
$N_2O_4 + (CH_3)_2N_2H_2$	3.060	2.754	1187.0	346.72	1442	790	5636

Here K_{m0} is the stoichiometric ratio of the fuel; K_m is the ratio of the fuel components; ρ_{fu} is the fuel density; I_{un} is the unit pulse; ρ_{ox} is the oxidant density; ρ_{cc} is the density of the combustible component; and Δv is the increment in the spacecraft speed.

Fig. 2. Storable options

B. Engines

Since design of a complete propulsion system is not the objective of this project it was decided to find similar already developed engines and use them as a baseline.

Firstly, we needed to specify the required parameters. For the main engines these are:

- Use of HTP and Kerosene
- Enough thrust for on pad abort scenario
- Flight proven design
- Easy and fast to develop

For the maneuvering engines these properties were required:

- Commercially available
- HTP monopropellant
- Thrust proportional to thrust of Crew Dragon maneuvering engines
- Capable of pulsed operation

The thrust for on pad abort scenario was based on Crew Dragon capsule. The necessary acceleration of 3,5 G needs to be maintained for up to 8 seconds. Since the weight of the capsule was not known, a conservative guess of 75% of Crew Dragons 12500 kg was used. Later in project we arrived to total weight of 6000 kg. If we would work on real project, following calculations would be iterated upon. The thrust was calculated using Newton's second law of motion. 1

$$F = m * a = 9375 * 3,5 * 9,81 \approx 320000N \tag{1}$$

This means that in 4 engine configuration each needs to produce 80 kN of thrust.

Based on calculated parameters a Gamma 2 engine was chosen as a baseline. It was developed for British Black Arrow rocket upper stage in 1969. It was based on Gamma 201 and 301 engines from Black Knight rocket. This engine was cheap and quick to develop, thanks to use of open oxidiser cycle. To drive the turbopumps HTP is catalytically decomposed and resulting steam drives a turbine in a fuel pump. This leads into much lower temperatures and thus much lower costs than in conventionally used cycles. As can be seen on Fig. 3 , the Gamma 2 engine used one set of turbopumps and 2 sets of chambers and nozzles. [10]

Main parameters are [10]:

- Thrust: 64,6 kN
- Isp: 265 s
- Size: 1,4 x 1,2 x 0,7 m (w x h x d)
- Weight: 173 kg

Based on these values, dimensions of Super Draco engines, and taking into account more than 50 years of technological progress, we estimated these parameters for our engine [11]:

- Thrust: 80 kN
- Isp: 265 s
- Size: 0,8 x 1,2 x 0,4 m (w x h x d)
- Weight: 110 kg

Fig. 3. Gamma 2 Engine [12]

According to requirements, we searched for commercial solution for our maneuvering thrusters. We arrived to 220 N Monopropellant thrusters from Nammo. They are currently in use on upper stage of European Vega-C rocket. First flight was performed in 2022, so these thrusters can be taken as already flight proven hardware. For full attitude control 16 thrusters will be used. Parameters:

- Thrust: 220 N
- Isp: 130 s
- Size: ø 160 mm x 168 mm
- Weight: 1,48 kg

C. Orbit

The selected orbit for this case study is the International Space Station (ISS) orbit, characterized by readily available data as follows:

- altitude ≈400km $(h_p = 416km,\ h_a = 422km)$
- orbital period $T \approx 93min$
- inclination $i = 51.6°$

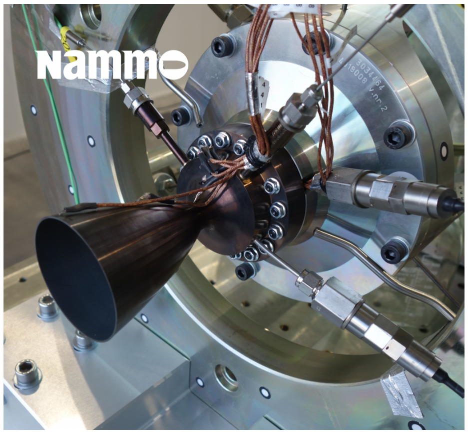

Fig. 4. Nammo Monopropellant thruster [11]

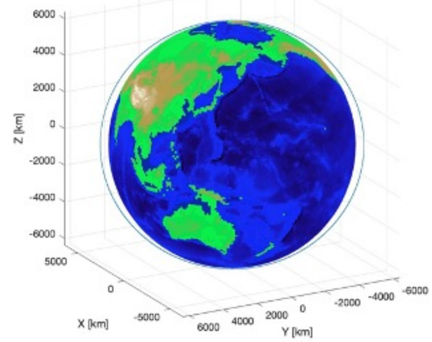

Fig. 5. ISS orbit

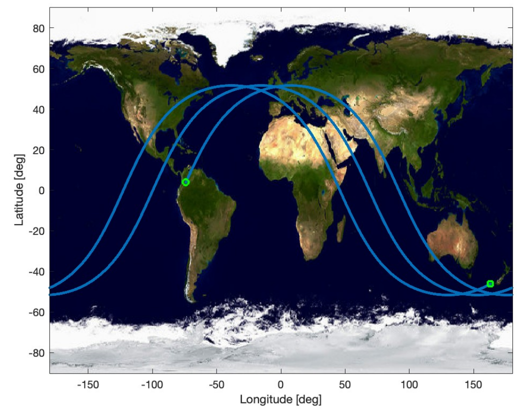

Fig. 6. ISS groundtracks

- eccentricity $e = 0.0007415$
- RAAN $\Omega = 50.1°$
- argument of periapsis $\omega = 336.3°$

These parameters have been used by various countries using different launch vehicles, indicating their reliability for future applications without necessitating additional adjustments. Moreover, this choice is based on the expectation that these parameters will remain relevant beyond the retirement of the ISS. As a backup for this, Axiom intends to initiate its station's development in this orbit after, firstly, connecting to the current ISS and then using it for their new station. The selected orbit can be seen in Figure (8) and its ground tracks in Figure (6). It is possible to acknowledge that the ISS completes just above 15 orbits a day and the ground-track advances westward by an angle $\Delta\lambda$ equal to the Earth's rotation during one orbital period T of the satellite:

$$\Delta\lambda = T \cdot \omega_E \tag{2}$$

Therefore, ISS is in a good position for launch approximately every 3 days, allowing to reach this launch frequency in the future. This data also supports the decision of making a direct launch, which limits the launch windows, albeit there would be a downside in terms of recovery and refurbishment time (of which further details will be explained below). While this decision is tentative and subject to revision, it provides a robust starting point for assessing the necessary velocity costs calculations for reaching Low Earth Orbit (LEO), with potential applicability to other orbits. Furthermore, the fuel mass calculations accommodate inclination modifications, facilitating the navigation to any future LEO station that may be developed in future.

D. Launch site

Having selected the target orbit, it was possible to asses the feasibility of launch in different sites. It was decided to reach the ISS orbit directly without making any inclination adjustment, while ensuring enough fuel to perform attitude change, as needed. Consequently, the latitude of the launch site needs to be lower than the inclination to be reached:

$$\phi < i \tag{3}$$

This first requirement leads to the exclusion of certain launch sites, since the case at hand implies an inclination of $51.6°$. Then the "Slingshot effect" was taken into consideration in order to choose which site could maximize the boost from Earth's rotation and have the biggest gain in the ΔV cost for launch. Initially a first estimation was carried out using:

$$\begin{aligned}\Delta V_{1st-approx} &= v_{E,rot} \cdot cos(i) \\ &= (460 \cdot cos(\phi))cos(i) = 288.83 \text{ m/s}\end{aligned}$$

After that, the model was updated to take into consideration the azimuth of every site

$$\beta_{inertial} = \arcsin\left(\frac{\cos(i)}{\cos(\phi)}\right), \tag{4}$$

where i is the desired inclination and ϕ is the launch site latitude. Also, the rotation of the earth is taken into account as shown in the system below to finally reach the ΔV_{gain} for each launch site.

$$\begin{cases} v_{rotx} = v_{orb} sin(\beta_{inertial}) - v_{eqrot} \cos\phi \\ v_{roty} = v \cos\beta_{inertial} \end{cases} \tag{5}$$

$$\Delta V_{gain} = v_{\text{orb}} - v_{\text{rot}}, \tag{6}$$

Once the above mentioned calculation is made, it was possible to gather the information in Table I [13]

TABLE I
LAUNCH SITE COMPARISONS

Launch site	Latitude [°]	Azimuth [°]	ΔV_{gain}	i achievable [°]
Kourou	5.2	35.8	279.9	5-100
Baikonur	45.9	62.1	287.3	49-99
Kennedy	28.6	42.7	283.1	28-62
Vandenberg	34.6	47.1	284.5	51-145
Naro space center	34.4	46.9	284.5	
El Hierro	27.6	-	-	75-105 / 121-152

Therefore, the above Table shows that the majority of the launch sites which have been preliminarly taken into account could actually be a potential candidate. In fact, most of them allow for a direct launch into the selected orbit and would provide a gain in ΔV cost of around ≈ 285 m/s, which is not far from the first estimation taken into account above.

Since the cost difference between different sites which had been preliminarly identified as potential candidates is very small, the ultimate decision for selecting the launch spot was a commercial one: the launch site which has been ultimately chosen was Kourou, French Guyana, to the extent Kourou is an oversea department of France and the study wants to commercially focus on the European market. This is because the American market is already a mature one, with a number of well established private companies which - at the point of our space taxi being operative - would have already had decades of experience and would therefore be likely preferred over a newly established operator. The European market, instead, has more space for a new entrant private company which could work closely with ESA and it could therefore allow for better commercial opportunities and better project financing.

E. Launch

Included in planning the launch of the vehicle the first thing calculated was the ΔV needed to achieve the targeted orbit. When this and the mass of the vehicle was known the choice of which launch vehicle to use was considered. This followed by safety futures such as abort scenarios and lastly the rendezvous and docking was researched and decided.

1) Total ΔV for achieving orbit: The velocity to stay in an circular orbit is calculated from setting the gravitational acceleration equal to the centrifugal force and thus yields the equation

$$v = \sqrt{\frac{GM}{r}}.$$

With an altitude of 400 km the velocity is calculated to be 7670 m/s. Furthermore the losses due to drag and gravitational pull were taken into consideration as Eq (7) and are estimated to be around 2000 m/s

$$\Delta V_{\text{total}} = \Delta V_{\text{orbit}} + \Delta V_{\text{air}} + \Delta V_{\text{gravity}} \tag{7}$$

Thus, with the ΔV_{gain} from Kourou previously discussed, the final total ΔV needed to reach orbit is 9400 m/s.

2) Launch vehicle: In order for the vehicle to reach the orbit, different launcher were considered. Since, as explained above, the project is focussing on Europe and the ESA, the ultimate decision landed on Ariane 6. Ariane is an expendable launch system which is designed with two stages that are both powered by hydrolox (liquid hydrogen - liquid oxygen) engines. There are two possible variants of this system:

- Ariane 62, with 2 P120 solid booster
- Ariane 64, with 4 P120 solid booster

The 62 configuration can launch up to 10350 kg of payload into LEO orbit, therefore this variant is the one selected for the case study, since the payload estimated for the vehicle is approximately 6500 kg The first stage (LLPM) is powered by the Vulcain 2.1 and contains approximately 140 tons of propellant. In addition, two P120 solid rocket booster, each containing 142 tons of propellant and delivers 4650 kN of thrust, are part of the first stage. For this stage, knowing the initial mass (480808 kg), the final mass and the Isp of the engine, it is possible to estimate the first ΔV from the following equation:

$$\Delta V_1 = I_{sp} \cdot g_0 \cdot ln(\frac{m_0}{m_f}) \approx 3815\text{m/s} \tag{8}$$

The upper stage (UPLM) uses the Vinci engine, powered by LOX/LH2, it carries 31 tons and delivers 180 kN of thrust. For this stage it is also possible to calculate the ΔV in order to reach the overall cost to insert the orbit.

$$\Delta V_2 \approx 5600\text{m/s} \tag{9}$$

3) Abort scenarios: A crucial aspect of planning a space mission is of course considering different offnominal cases. It is one of the most important aspects. Offnominal cases in all stages where discussed, this includes the launch stage, in orbit and landing. The offnominal cases discussed in this section is regarding the launch stage.

We divided the launch abort in different sections:

1) Pre liftof:If there is any issues on the launch pad there must an emergency escape plan. There has been a case where a fire started on the launch pad. This could be done by a Pad abort test.
2) Post liftof: If the system is already ignited and the liftoff cannot be canceled a few options are possible depending on the velocity. Mainly there must be a redirection of the trajectory
    a) Emergency landing: Here the there must be an option to land on a precalculated location in the ocean. Our capsule will be adapted so it can survive such an abort plan.
    b) If there is enough velocity a common abort plan is to first orbit the earth once, and the land at the landing site about 90 minutes after liftoff.
    c) If there is a even higher velocity it is possible to go to a lover orbit, this is classified to be around at an altitude of around 190 km.

4) Rendezvous and Docking: In order to reach the ISS various manoeuvres are necessary. The launcher burn would put the vehicle in a lower orbit than the ISS, where the rendezvous operation would start using the vehicle engine and maneuvering thrusters. The vehicle would be positioned in a 10 km orbit lower than the ISS and it would be behind the station. Afterwards, a first co-elliptic burn can be performed in order to reach another coplanar orbit of 2.5 km below the ISS. At this stage, different checks can be performed to ensure the correct positioning of the vehicle relative to the ISS. In case of not optimal positioning with respect to the station, it would be possible to also wait on this orbit, which, since it is lower than the ISS, would have a shorter period and make the vehicle "catch up" w.r.t the station. Then, a second co-elliptic burn would bring the vehicle in the ISS orbit where the docking can start. In this case study, the docking is considered autonomous after the last break maneuver made by the solid thrusters tin order o be in close proximity to the station. A lambert solver was used to calculate and plot the different orbit and the transfers between them, using as inputs the keplerian parameters of the initial and final orbits. The computed costs for the rendezvous are as follows:

- $\Delta V_1 = 60\ m/s$
- $\Delta V_2 = 40\ m/s$

Therefore the overall strategy can be summarized as shown in Figure (ref) with two transfer orbit in order to reach the ISS final orbit. In Figure (ref) the second transfer can be seen more clearly.

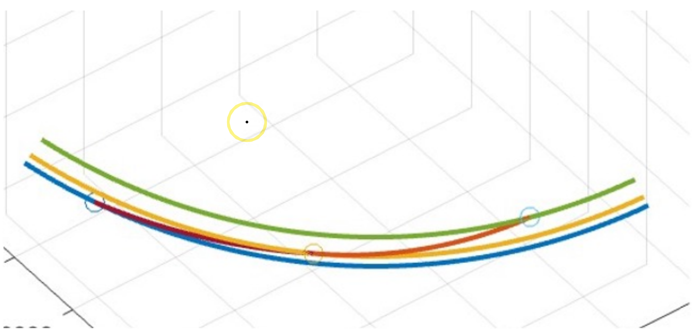

Fig. 7. Rendezvous strategy

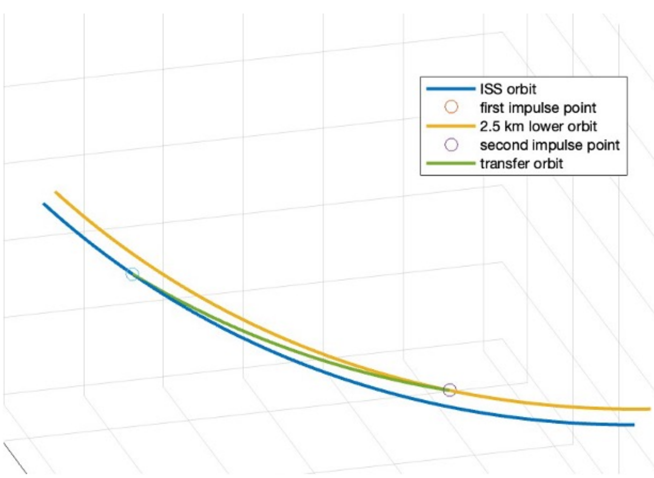

Fig. 8. 2nd coelliptic burn

To summarize, the total cost for rendezvous and docking manoeuvre is approximately 100 m/s. The possibility of having to perform multiple manoeuvre was also considered with safety margins allocated to the ΔV in order to have enough fuel even in non-nominal cases.

F. Landing

As with the other systems, the landing sequence is fully automated and is dependent on highly accurate maneuvers. The sequence is complex and uses many different systems such as both maneuver and main engines, heat shielding for aerodynamic breaking and parachutes. Due to the complexity, certain assumptions and rough estimations have been made which will be stated through out the different parts of the landing. The sequence in whole is inspired by both SpaceX's Crew Dragon and Roskosmos' Soyuz capsule.

1) Undocking and Deorbit: The first step in the landing sequence is to undock from the space station. This is done with the maneuvering thrusters and when a safe distance from the station has been achieved, the capsule will be angled in the opposite way from its orbit and a retrograde burn will be initiated. This will brake the capsule and set it in an orbit with an perigee that is inside Earth's atmosphere. The magnitude of ΔV needed for the retrograde burn is dependent on the orbit for the space station but for a station with about the same orbit as ISS a ΔV of about 100-200 m/s is needed[14]. The exact ΔV needed for this will be calculated for the specific space station and must be precise as the safety of the atmospheric entry is dependent on this.

2) Atmospheric entry and breaking: At a velocity of 7.5 km/s relative to the Earth's surface the capsule enters the atmosphere[15]. From an ISS orbit, the angle of the capsule towards the horizon must be 1.35° as a lower angle would risk the capsule to skip on Earth's atmosphere and enter a new orbit and a too high angle would create too high braking forces from the atmosphere and could be fatal for the astronauts or break the capsule. This is why as previously stated a precise calculation for the retrograde burn is needed. With this angle the atmospheric breaking will be similar to that of the Soyuz and thus we can estimate that the landing spot will be about 2500 km in horizontal distance from the atmospheric entry. The atmospheric braking will bring the capsules velocity down to 150 m/s and have a maximum g-load of about 4.5 g.[14]

3) Parachuted and soft landing: When the capsule has slowed down to a velocity of 150 m/s, an initial high speed parachute, a drogue chute, will be deployed which firstly slows down the capsule and also stabilizes it in the air. When the velocity is reduced to 60 m/s two main parachutes are deployed and brakes the capsule down to 15 m/s where they disconnect and the main engines ignites. In this phase the capsule is 50 m above the ground which gives the opportunity to scan the ground and aim for the smoothest area in close proximity. During the last 50 m the engines brings the capsule to a soft landing at 1 m/s, this will not be a problem for the crew or the capsule. The different speeds and for the different parachutes are estimations and inspired by the deployment speeds for the capsules of Orion[16], Crew Dragon[17] and Soyuz[14].

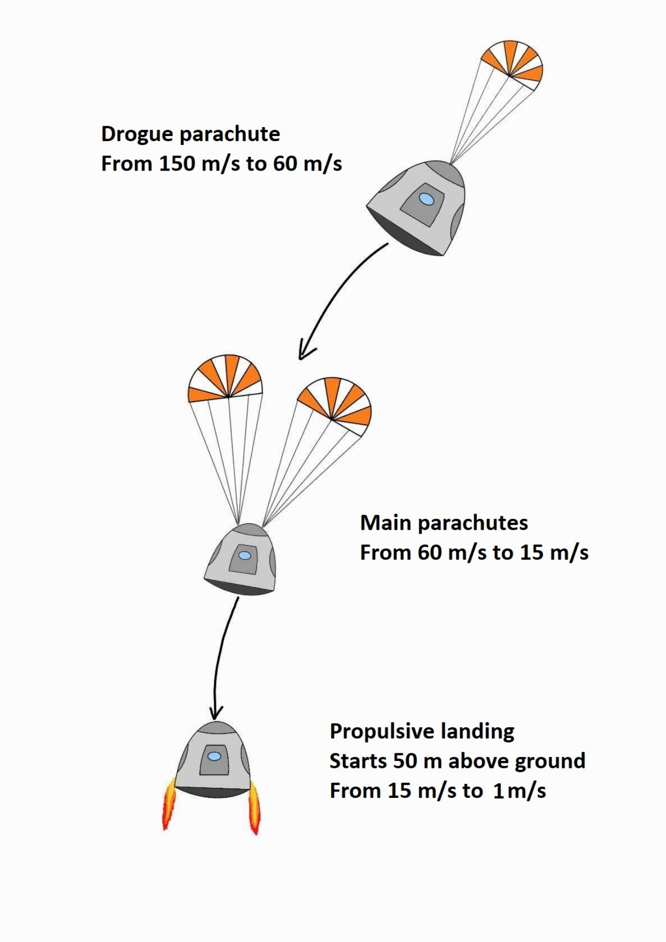

Fig. 9. Landing sequence of Hitchhiker

4) Landing site: The first discussion of the landing was if it would be on water or land. The choice was to land on land since it enables easier recovery and refurbishment as saltwater destroys the heat shield. The landing site is set to be in the Algerian desert at around 31 N, 0 W. The accuracy to land at that spot is a circle with radius 25 km which is similar to that of Soyuz's landing accuracy[14]. The reasons to this site is firstly that there are no buildings or populated areas in proximity and secondly it is relatively close to Europe which is important for the turnover time. With research and by learning from every landing the goal is to eventually bring the radius for accuracy down to a few meters and at that stage the landing site can be chosen more arbitrarily. One possibility would be to land on a sea platform closer to Europe, this would bring down the turnover time and simplify the recovery.

5) Non-nominal landing cases: There are three different non-nominal cases to consider for the landing. The first one is that one of the parachutes does not deploy properly. In this case the vehicle will continue its decent as in the nominal case but use the engines with a larger throttle setting. This will need more fuel than in the nominal case but as the main engines are used for both the landing and the abort scenarios there will be more than enough fuel from the abort scenario to slow down the capsule enough for a touchdown that will not be lethal for the crew. Also note that due to the specific abort scenarios the available thrust from the main engines are high for our mass and thus with a high impulse will be able to slow us down from high speeds, although with the cost of exposing the crew to high g-forces.

The second non-nominal case is if one of the engines do not start. In this case the engine on the opposite side will shut of in order to have symmetry with one engine on either side of the capsule and as previously stated, we have high thrust available which enable us to land smoothly with only two engines.

The third non-nominal case is if all of the engines do not start. This will result in a hard landing of about 15 m/s where only the parachutes with the help from maneuvering engines are used to slow down the capsule. The landing will result in the destruction of the heat shield but the velocity will be low enough to not risk the lives of the crew.

G. Turnover

As was stated previously, our landing site is in the Algerian desert. This can pose significant challenges for crew, payload, and capsule safe recovery. The main objectives are quick arrival to landing spot, short transport time to refurbishment facility in France and low costs. All locations named in this chapter can be found in Fig. 11

First step was to find suitable staging location for our ground crew. It needs to be as close as possible to the landing location. It needs to be easily accessible at least by roads. Infrastructure for both winged and rotary aircraft operations are big advantage. All of this should be already in place to save costs. After scanning of surrounding areas using Google Maps and Mapy.cz small airstrip was found at 30.7672778N, 0.7065089E. It is part of neighbouring oil and gas infrastructure and thus still in use. With paved runway 1700 metres long and road connecting to coast of Algeria it is an ideal spot. Short term accommodation can also be arranged in nearby housing for oil workers.

After landing the main objective is to find the capsule as fast as possible. This can be best achieved by use of helicopters. They combine fast travel time, necessary payload capacity and ability to land almost anywhere. For our purposes helicopters from the heavy lift category were considered. In the end two Boeing H-47 Chinooks were chosen. They have a two rotor configuration, maximum speed of 302 km/h, maximal operating range of 306 km, useful payload of 12,565 kg and each can carry up to 55 persons. [18] It is long proven platform capable of carrying oversized cargo underneath while still offering large transport compartment for ground crew and all of its equipment. This purchase of two Chinooks will cost 6 million USD.[19] The estimated time of arrival to landing spot is 30 minutes. This can be further lowered by taking off before capsule touchdown.

Upon arrival to the landing site the ground crew starts with opening of the capsule. The passengers are assisted in getting out of the capsule and are checked by medical professional. The returning cargo (if any is present) is also taken out. Afterwards the priority is to get both the passengers and cargo out of the dessert and to their destination as fast as

Fig. 10. Boeing H-47 Chinook performing lifting operation [20]

possible. There are two possibilities. Either the passengers are flown 70 km to the staging area, from where they can be transported using private aircraft. Or they can be flown to a Mécheria, city 280 km away with international airport. From there commercial aviation can be used to get quickly to anywhere in Europe. This all should take roughly 2 hours.

When the first helicopter leaves with passengers and cargo, remaining crew starts securing the capsule and preparing it for lift. Structural integrity is checked, especially in the remaining hydrogen peroxide systems. To assure additional safety excess fuel and oxidiser can be transferred out of the capsule. After return of the first helicopter a lifting operation is performed. Capsule is attached under one of the Chinooks and lifted. In this manner it is transported to Mécheria. All of this should take approximately 5,5 hours from capsule touchdown.

When the capsule arrives to Mécheria, it is loaded onto custom truck trailer. From here it continues on road and according to European regulations it needs to be classified as an oversized cargo.[21] It first travels to a port in coastal city Oran. There it can be loaded on ferry and shipped to Almería, Spain. Afterwards it continues its journey thru Spain all the way to refurbishment facility located near Bordeaux, France. Because of the complications associated with being oversized cargo, this whole transport will take approximately 2 days. This time will slightly decrease in the future thanks to route familiarity and optimization. It can be further reduced if the helicopter in use proves to be capable enough to perform the lifting operation not only to Méchria, but further to Oran or even mainland Europe. Lastly, even faster turnaround speeds can be achieved by using cargo aircraft. Thanks to a large diameter it was determined that only real civilian option in this category is Antonov An-124 Ruslan. [22]

II. CONCLUSION

To complete this project many different approaches were taken to reach our conclusions. We did a research based project with educated assumptions to be able to calculate different approximations of different parameters.

Regarding the *fuel*, a bi-liquid propulsion system was chosen with a hydrogen peroxide/kerosine oxidizer/fuel combination. This because of the cost effectiveness, non toxicity and storagebility of the fuel.

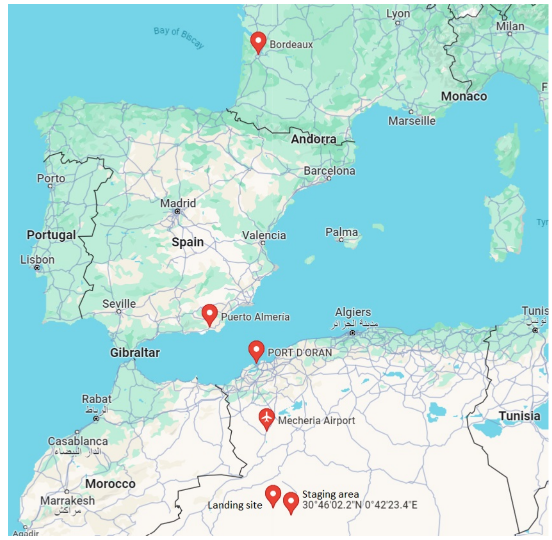

Fig. 11. Map of transport route

The *main engine* parameters were based on British Gamma 2 engines. The main requirements came from the on pad abort scenario. To keep the cost and complexity low, commercial maneuvering engines using monopropellant HTP were chosen. With further development, more accurate parameters and dimensions can be calculated and specified.

The selected *orbit* being the ISS, was considered a good starting point, however, future expansion of this study to be able to reach any other station could be performed and interstation travel could also be implemented.

Offnominal cases were considered regarding the launch stage and abort procedure for pre/post liftoff were constructed. Further research for the *landing sequence* could contain the structural analysis of the vehicle and the heat shield if it would be able to handle a touchdown of 1 m/s without breaking. Precise calculations on what speeds and where in the atmosphere to enter would also ensure that this kind of landing would be possible with this kind of accuracy.

The *turnover* time was based on rough estimations, especially for the loading and truck transport. Deeper analysis will be needed to specify these times more accurately. Further alternatives can also be explored.

To conclude the aim for the project was to create a launch/landing for a space taxi. Doing research of similar projects and doing our own approximated calculations we've reached results that are feasible for such a project.

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