The Mars Society

Inspiration Mars International Student Design Competition

University of Glasgow, Scotland, United Kingdom

SPECTRE MARS
**Team Description:**

The team consists of eight Undergraduate Aero-Engineering students from the Student Aerospace Engineering Society (SAES), University of Glasgow, Scotland, United Kingdom.

The complete list is given below:

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<th>Member</th>
<th>Project Job Title</th>
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<tr>
<td>Ahmad, Taseer</td>
<td>Team Leader Manager Organiser</td>
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</tr>
<tr>
<td>Ammar, Ahmed</td>
<td>Launch System Structures</td>
<td><a href="mailto:2060575A@student.gla.ac.uk">2060575A@student.gla.ac.uk</a></td>
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<tr>
<td>Kamara, Ahmed</td>
<td>Thermal Protection System Power Requirements</td>
<td><a href="mailto:2109433K@student.gla.ac.uk">2109433K@student.gla.ac.uk</a></td>
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<td>Lim, Gabriel</td>
<td>Vehicle Health Management</td>
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<td>Magowan, Caitlin</td>
<td>Propulsion</td>
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<td>Tse, Yee Cheung</td>
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</table>
1. Abstract

The team’s design concept (SPECTRE MARS) is a partially re-usable, two stage launch vehicle that consists of the NASA SLS and Orion spacecraft with 28 Merlin 1D engines, 27 for the first stage and one for the second. Two cores of nine engines each, from the specified numbers of engines, will be used in the booster cores.

Fuel tanks and piping system will be using liquid oxygen as the fuel and liquid kerosene as the oxidizer. The Reaction Control System chosen for the project is Lynx RCS which uses a bi-propellant fuel.

Thermal Protection System chosen for the project is ASTRA ULTIMATE which is still under testing but the results have been very promising. This system, upon completion, will be the best such system up to date.

The Avionics for the project will consist of on board computers (Mongoose V), Remote Terminal Units (RTU), Command and Data Handling (C & DH), On board data buses and a Navigation and Guidance System (DSN Deep Space System)

The advanced vehicle health management system will comply by all the rules of an integrated vehicle health management system. To strengthen this network, extra monitoring devices have been added to the project. This system will also be extended to handle any problems in the engine.

According to the design requirements, the spacecraft is manned. Thus requires systems on board to cater for the humans and provide enough food to last for the length of whole mission. Enough data has been provided for the human effect on the mission.

The trajectory of the mission is the most suitable for 2018 and includes a free-return to earth, reducing the fuel required and thus reducing the cost of the mission. Existing launch facilities in French Guyana will be used for ground systems, again saving mission costs. The power requirements for the whole mission have also been covered.

There is a great expectation for humans to land on Mars in the next few decades, and this mission may be the stepping stone to do so.
2. Trajectory

Earth -> Low Earth Orbit (LEO) -> Geostationary Transfer Orbit (GTO) -> Earth Escape velocity (C3=0) -> Mars Transfer Orbit -> Earth Escape velocity (C3=0) -> GTO -> LEO -> Earth

figure 2a

The velocity at exit from the Earth sphere of influence is $v_\infty$, where $v_\infty$ is the velocity of the vehicle at the sphere of influence with respect to the coordinate of the planet.

$$v_\infty = v_{pt} - v_e$$

$$= 32.74 - 19.78$$

$$= 2.96 km/s$$

The characteristic energy $C_3 = v_\infty^2$, the characteristic energy is used to define energy requirement for the departure from the planet’s sphere of influence.

figure 2b Mars flyby mission trajectory
In this case, $C_3 = 8.76 \text{km}^2/\text{s}^2$ and the energy of the escape hyperbolic route will be

$$\frac{C_3}{2} = 4.38 \text{km}^2/\text{s}^2.$$ 

Assuming that there is an injection burn at an altitude of 300km with respect to the earth, the burn out velocity $v_{bo}$ required will be:

$$v_{bo} = \sqrt{2\left(\frac{\mu_e}{R_e + h} + \frac{C_3}{2}\right)} = 11.32 \text{km/s}$$

$R_e=$radius of the earth at equator=6378km

$h=$altitude of injection burn=300km

$\mu_e=$gravitational parameter of the earth=$GM=3.9860064 \times 10^5 \text{km}^3/\text{s}^2$

$\frac{C_3}{2} =$Energy to escape the hyberbola=$4.83 \text{km}^2/\text{s}^2$.

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**Figure 2c** flyby mission return trajectory

When the distance between the spacecraft and Mars is at the minimum, the velocity is 11.32 km/s.

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**Figure 2d** forces on the spacecraft when it is orbiting the earth
By equating the centripetal force and gravitational force acting on the spacecraft, the minimum velocity for a spacecraft to stay at the earth orbit of altitude h will be

\[ v_o = \sqrt{\frac{\mu_e}{R_e + h}} \]

For the spacecraft to stay at an altitude of 100km, \( v_o = 7.84 \) km/s. If the spacecraft is orbiting at a velocity less than \( v_o \), then it will be attracted by the gravitational force and re-enter the earth. Therefore, the spacecraft will need to decelerate from 11.32 km/s to 7.84km/s when it returns from Mars to Earth's orbit.

Further trajectory calculations have been included in the fuel tanks and piping system for the project.
3. Aerodynamics and Structure

3.1 Space Launch System

3.1.1. Introduction
The Space Launch Systems (SLS), opening new doors for science and human exploration beyond Earth’s orbit is an advanced heavy lift launch vehicle. Unique vantage points in space are to be explored and, to cross current limits, we need means that are affordable, sustainable and safe. The SLS gives us this opportunity. The SLS will carry the Orion Multi-purpose crew vehicle. The Orion spacecraft can carry two to four astronauts and is designed flexibly to support any type of mission be it deep space exploration or service missions to the International Space Station (ISS), which were earlier carried out by the retired space shuttle program.

The SLS includes the crew and service modules and also the advanced Launch Abort system which significantly increases the safety of the crew. The SLS is flexible and evolves to perform any crew and cargo missions as needed. With multiple lift vehicle configurations the astronauts aboard the Orion spacecraft can not only conduct deep space explorations further into the solar system but also possible find resources to develop new technologies and answers to our position and role in the universe.

3.1.2. Capabilities
The SLS configuration proposed for this specific mission is the “Block I”. It is a 70 metric ton launch vehicle which can lift more than 154,000 pounds (approximately 70,000kg) and provides 10 percent more thrust than the Saturn V rocket. This configuration focuses on the use of proven hardware, tools and manufacturing technologies. This will not only reduce the cost of the development and operation but also increase safety and endurance of the flight and its systems. This heavy lift vehicle configuration is 321 feet tall (approximately 98 meters) which is slightly shorter than the Saturn V and provides 8.4 million pounds (3.8 million kg) of thrust at liftoff and weighs around 5.5 million pounds (2.5 million kg).

This configuration of the SLS consists of 2 booster cores consisting of 9 Merlin engines each, 9 Merlin engines making up the core stage of the SLS. Further up is the interim cryogenic
propulsion stage, and then comes the Orion spacecraft docked into the Launch Abort system.

3.1.3. Design and Development

3.1.3.1. Core Stage
The core stage of the SLS mainly consists of five parts: the engine section, a kerosene tank, an inter tank, a liquid oxygen tank and a forward skit. It measures more than 200 feet (61 meters) in length, having a diameter of 27.6 feet (8.4 meters). The empty core weighs more than 85,275 kg and is built out of Aluminium 2219. The core carries the liquid hydrogen as a fuel and liquid oxygen as an oxidiser to feed 9 Merlin engines. The Product design review (PDR) confirms the design falls within the required budget constraints. The core stage is being manufactured and assembled at Michaud Assembly facility in New Orleans.

3.1.3.2. Booster cores
This particular configuration of the SLS will use two booster cores consisting of 9 Merlin engines each.

3.1.3.3. Upper stage / Inter-stage
This inter-stage consists of one Merlin engine whose sole purpose is to power the second stage to a desired altitude and set the spacecraft in orbit to Mars.

3.1.4. Assembled Rocket
The assembled rocket can remain at the launch pad for a minimum of 180 days and in stacked configuration for at least 200 days.

3.1.5. Program Costs
NASA predicts the program to cost about $3 billion yearly with total development costs adding up to $35 billion. In order to get the SLS ready for its 2017 test launch, the cost estimates to be about $18 billion with the rocket costing about 10 billion dollars, the Crew capsule to cost about 6 billion dollars and the launch pad and ground facilities for the SLS costing $2 billion.
3.2. Orion (Spacecraft)

3.2.1. Introduction
Orion Multi-Purpose Crew Vehicle (MPCV) is a next generation deep space exploration vehicle to be launched aboard the Space Launch Systems (SLS). It will execute long duration missions to nearby asteroids and mars and return crew members to earth. The spacecraft is capable of carrying two to four astronauts. It also supports crew and cargo missions to the International Space Station (ISS).

The Orion consists of four functional modules: the launch abort system, the crew module, the service module and the spacecraft adapter. The Launch Abort system functions as the emergency escape system during the launches, the crew module transports the crew and the cargo, the service module supports the propulsion and electrical needs and the spacecraft adapter is a structural transition which docks the spacecraft onto the SLS.

3.2.2. Crew module
The design of the crew module is derived from the earlier Apollo crew module. It features a 57.5 degree frustum shape. This particular shape benefits to reduce the area required for thermal protection and also provides 10 time safety during re-entry and deep space manoeuvrability. The diameter and length of the CM are about 5 meters and 3 meters respectively and it weighs about 8500 kg. The major difference between the CM and Apollo CM is that, the Orion CM is twice the size and carries at least four crew members.

The Orion crew module also features a glass cockpit inspired from the Boeing 787. It features the advanced digital controls and computerized systems including features like the auto dock system. During emergency situations the crew is allowed to take over the controls of docking. Among the new improved features is the waste management facility which is described as a “unisex camping style toilet”, better preferred than the plastic “Apollo Bags”.

During re-entry the Orion crew module deploys recovery parachutes and features a splash down landing into the ocean. The parachutes being used are similar the ones used in the shuttle SRB’s and are made of Nomex. Nomex will also be used on areas with a possibility of critical heating like the bay doors.
3.2.3. ATV based service module
The ATV-based service module will be featured in the future deep exploration missions into the space. It provides the crew Module with propulsion and electrical needs. It also provides the astronauts with supplies of gas and water. The service module features solar panels in the form of x-rays and provides basic power needs. The four solar panels consist of 40Ah rechargeable batteries and have a span of 22.3 meters. The power generated by these solar panels is 3800 watts. The crew module will re-enter the earth’s atmosphere at 11 kilometres per hour, the fastest ever.

3.6.4 Launch Abort systems (LAS)
The LAS fairing consists of a lightweight composite structure similar to a graphite tennis racquet weighing 3000 pounds (1360kg). The various parts of the Orion Launch Abort system from the top are the nose cone, altitude control motor, canard section, Jettison motor, inter-stage, abort motor and adapter cone which docks onto the crew module. Between the crew module and the adapter cone, there are several boost protective covers made of fiberglass to protect the CM from aerodynamic and impact stresses.

The abort motor consists of four nozzles at a cant angle of 30 degrees and provides a thrust of 2.25kN per nozzle. The altitude motors consist of 8 nozzles at cant angle of 90 degrees providing a thrust of 11kN per nozzle. Lastly the Jettison motor consists of 4 nozzles at a Cant angle of 35 degrees providing a thrust of 48kN per nozzle. The whole system can carry 5468 pounds (2480kg) of fuel.
4. Propulsion

4.1 Introduction
The choice of the engine was made by considering the cost, technical quality and simplicity as main objectives. The desire was to choose an engine with a strong research background in recent years and is ready or will be ready before the required date, 2018, which will increase the project feasibility. Therefore, the market research conducted was concentrated on projects that will have finished their development soon.

4.2 Engines considered
All the permissible kinds of engines for this competition were considered during the research stage of this project, chemical, solar electric and air breathing.

4.2.1 Solar Electric
Solar electric propulsion has never been tested, however, from earlier research it was known there is an engine designed by Ad-astra, the VASIMR engine. The VASIMR engine is currently in development and is due to undergo testing at the International Space Station in 2016.
It is a nuclear electric engine which uses ionised plasma to produce thrust. There is potential that solar panels could be fitted to provide the electricity to ionise the plasma instead of radio waves. Although the VASIMR engine does not produce as much thrust as other engines, such as chemical engines, it is more efficient.

Through further research and information from Ad-asta we decided attaching solar panels to the VASIMR engine would not produce enough thrust to power the rocket anywhere near the speed that is required to reach Mars in a reasonable time frame, although in the future this might be a possibility for other applications, it will not be feasible by 2018.

4.2.2. Air Breathing
Another type of engine considered was the SABRE engine, an air breathing engine, currently under development by Reaction Engines Limited. It is being designed to provide the capability for the launch vehicle to travel single stage to orbit and produces approximately 1960N of thrust at sea level.

Because it is air breathing, it does not require an oxidiser such as liquid oxygen as long as it is in the earth’s atmosphere. If it is above the earth’s atmosphere it can use an oxidiser to burn with the fuel in the same way as a conventional rocket engine. This reduces the overall mass of the launch vehicle. The air would have to be compressed a lot before it entered into the combustion chamber, producing a lot of unwanted heat in the process. This is overcome
using the engines cooling system to cool the air significantly before it is pressurised and injected into the combustion chamber, preventing the engine from melting. It includes technological advances such as frost control, which would be needed due to low temperature the air will be cooled to, and light-weight technology.

The programme is definitely receiving funding for development up until 2016, but it is unknown how long the programme will take to complete and when the engine will be ready for use. For this reason it was decided not to use this engine in our project.

4.2.3. Chemical
Chemical propulsion is by far the most common form of propulsion for rocket engines. The F-1 engine, developed by Rocketdyne is the most powerful single chambered rocket engine to ever be produced. These engines were used on the Saturn V, the launch vehicle for the Apollo programme, the programme that enabled the first human landing on the moon. The first F-1 engine started development in the late 1950’s and continued development during the 1960’s whilst the original version of the engine was being used on Saturn V. The new developments resulted in a new engine specification, the F-1A, intended for use on future Saturn V launch vehicles; however, Saturn V production stopped and the F-1A engine configuration was never used. There have been many proposals to develop new expendable boosters based on the F-1 rocket engine but none as of yet have gone past the drawing board. There is potential in the future for an F-1B engine however it will most likely not be ready for 2018.
Merlin 1D Rocket Engine

4.3.1 Characteristics and performance
Merlin 1D is the chosen engine for the flyby Mars mission for both the lower stage, to provide the initial boost to the launch vehicle to enable it to enter into low earth orbit, and the upper stage, to transfer the vehicle past the geosynchronous transfer orbit. It was developed by SpaceX and began testing in 2011. It boasts the highest thrust to weight ratio of any rocket engine and is the most efficient booster engine ever produced and the best liquid rocket engine in history. It passed a rigorous testing programme in 2013 to ensure the engine was mission ready in which it performed extremely well, exceeding the industry standard and is a suitable engine for use on a manned-mission in terms of the safety of the engine.

Performance table:

<table>
<thead>
<tr>
<th>Characteristics</th>
<th>Values</th>
</tr>
</thead>
<tbody>
<tr>
<td>Sea level thrust</td>
<td>6.77MN</td>
</tr>
<tr>
<td>Burn time</td>
<td>165s</td>
</tr>
<tr>
<td>Specific impulse</td>
<td>263s (2.58km/s)</td>
</tr>
<tr>
<td>Chamber pressure</td>
<td>7.0MPa</td>
</tr>
<tr>
<td>Dry engine weight</td>
<td>8400kg</td>
</tr>
<tr>
<td>Burnout engine weight</td>
<td>9150kg</td>
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</table>
It is the engine currently used on the SpaceX developed Falcon 9 v1.1 and is intended for use on their Falcon Heavy launch vehicle, currently under development. SpaceX are hoping this new Merlin 1D engine, replacing the previous Merlin 1C engine, will enable their Falcon 9 and Falcon Heavy vehicles to become fully reusable.

<table>
<thead>
<tr>
<th>Performance Table:</th>
</tr>
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<tbody>
<tr>
<td>Thrust (vac.)</td>
</tr>
<tr>
<td>Thrust (SL)</td>
</tr>
<tr>
<td>Thrust-to-weight ratio</td>
</tr>
<tr>
<td>Chamber pressure</td>
</tr>
<tr>
<td>Isp (vac.)</td>
</tr>
<tr>
<td>Isp (SL)</td>
</tr>
</tbody>
</table>

4.3.2. How it works?
Merlin 1D is a liquid fuel engine, using RP-1 and liquid oxygen as propellants in a gas generator power cycle, fed dual impeller turbo-pump. The hydraulic actuators are supplied with the high pressure fluid, which then recycles in the low pressure inlet. The turbo-pump also controls the rolling motion of the exhaust nozzle of the second stage Merlin Vacuum engine.

The configuration chosen for this project has been adopted from the Falcon Heavy rocket. This means there are three cores, each with 9 Merlin 1D engines, using the unique cross-feed propellant system. The image by Nick Kaloterakis shows the arrangement of the engines in one core.

Configuration of a single core with 9 engines.
Three cores make the first stage- the side ones are the boosters, which work together with the center core at lift off, producing 17,615 kN of thrust. The central core throttles back until the boosters separate and then returns to operating on full power, until it separates too. All the cores are predicted to have spindly metallic legs that will unfold on return to the Earth and a set of retro rockets to slower its decent, according to Blanchard (2014). This makes all the 27 first stage engines reusable. The second stage consist of 1 more Merlin 1D engine, but the thrust in vacuum is significantly higher- 801 kN, which will stay attached to the spacecraft and return to earth with it. It can be restarted as much as needed to correct the trajectory as long as there is enough fuel.

### 4.4. Cost

As stated above the Merlin 1C engine has been replaced with Merlin 1D and one of the main reasons behind this development is reusability. This means that effectively with every launch the cost of the propulsion system as a whole will drop. According to Anthony (2014), the cost per pound to launch into Earth’s orbit could go down to 500$ or less (approximately 1100$ per kg), which is one twentieth of the price nowadays of non-reusable rockets.

SpaceX comments on how the price of the engine will be reduced:

“Improved Manufacturability: Simplified design to use lower cost manufacturing techniques. Reduced touch labor and parts count. Increased in-house production at SpaceX.” The latter means avoiding any payment to subcontractors, which is decreasing the overall price of the product.

Also this private company’s policy is not to advance technology, but to apply it in a way that offers the lowest launch cost.

However, the price of the engine is not revealed by the company and it is really an approximation. It is expected it will be cheaper than Merlin 1C, which is speculated to cost a bit more than $1 million. Moreover, according to Pultanova (2014), the current price of the
launch of the Falcon 9 (which uses 10 Merlin 1D engines) is $54m and if the reusable concept is fully functional, it can drop $200,000.

All those arguments support the statement that Merlin 1D is a really good choice for the propulsion system for this mission as it is likely one of the cheapest available options.

First test fire of the Merlin 1D engine at SpaceX's Rocket Development Facility in McGregor, Texas.

4.5. Development of Merlin 1 rocket engine.
Merlin’s development starts from Merlin 1A, producing 340kN thrust, used in 2006 and 2007 for the first stage of Falcon 1. Then it was developed further and Merlin 1B had more powerful turbo-pump and more thrust. However, it was not even used before the progress continued and the new Merlin 1C, using a regeneratively cooled nozzle and combustion chamber, was produced. It was used in the third and fourth Falcon 1 mission and the first five Falcon 9 flights until 2013. Figure 2 shows the change done to adapt Merlin 1C engine to the structure of Falcon 9 and explains the reduction in weight, compared with the previous two versions.
Comparison of the structure between Merlin 1B and Merlin 1C. The red circle shows the part removed from the engine design.

The next advance was the creation of Merlin 1D engine. Tom Mueller, propulsion engineering vice president, says that it is designed so that it has the ability to throttle between 70% and 100%. Many other vital characteristics have been improved and the comparison between Merlin 1C and Merlin 1D rocket engines is shown in Table 2.

<table>
<thead>
<tr>
<th>Performance comparison of Merlin 1C and Merlin 1D</th>
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<tbody>
<tr>
<td><strong>Merlin 1C</strong></td>
</tr>
<tr>
<td>Thrust in Vacuum</td>
</tr>
<tr>
<td>Thrust to Weight Ratio</td>
</tr>
<tr>
<td>Chamber Pressure</td>
</tr>
<tr>
<td>Nozzle Expansion Ratio</td>
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</table>

Table 3
In an interview SpaceX comments the improvements of Merlin 1D:

- Increased reliability:
- Simplified design by eliminating components and sub-assemblies.
- Increased fatigue life.
- Increased chamber and nozzle thermal margins.

The next development will be Merlin 2 rocket engine. However, until now this is just a concept and all the resources are concentrated in the production of Merlin 1D engines.

4.6. Conclusion
Rocket engine has been deemed as one of the cheapest on the market or in development at the moment. It keeps our design simple as the same type of engine can be used for both the first and second stage of the mission whilst still producing more than enough thrust. The manufacturing of the engine is cheaper because of engines simple design whilst keeping the technical quality and the rate of failure to a very high standard.
5. Fuel Tank and piping system

5.1. Introduction
The propellant tank contributes a lot to the total weight of the launch vehicle. For any extra weight in the propellant tank will decrease the possible weight carried in the payload. Therefore, the fuel tank must be designed to be light weight. Also, the tank needs to be able to sustain high stress. The tank is subjected to static and dynamic loads.

The static load includes hydrostatic pressure, overpressure and payload weight.

The hydrostatic pressure is formed by the state of the fluid inside the tank.

Overpressure is sometimes used for cavitation in the fuel pump.

Payload weight is the weight acted by the spacecraft.

The dynamic load can be divided into inertia load due to vehicle acceleration, control load, which is the bending of structure caused by thrust vectoring and aerodynamic load due to the vehicle shape and external aerodynamic forces.

5.2. Fuel and oxidizer Description
The following fuel and oxidizer have been selected:

<table>
<thead>
<tr>
<th>Name</th>
<th>type</th>
<th>Boiling pt. (K)</th>
<th>Freezing pt.(K)</th>
<th>Density(kg/m³)</th>
<th>Molecular Weight</th>
</tr>
</thead>
<tbody>
<tr>
<td>Liquid Oxygen(LOX)</td>
<td>Oxidizer</td>
<td>90</td>
<td>54</td>
<td>1140</td>
<td>32.0</td>
</tr>
<tr>
<td>Liquid kerosene(RP-1)</td>
<td>Fuel</td>
<td>420</td>
<td>200</td>
<td>810</td>
<td>170.33</td>
</tr>
</tbody>
</table>

The optimum oxidizer to fuel ratio is 2.56. And the temperature of combustion is 3670K.

The temperature to store the oxidizer will be 70K while the temperature to store the fuel is 298K.

Fuel tank structure analysis has been provided in figure 5a.

Figure 5a circular and spherical shell with forces due to internal pressure; axial force on a cylindrical shell with spherical cap under axial loading pressure.
From figure 5a, we found that the allowable thickness $t_s$ of the spherical shell tank with pressure $P$, radius $R_s$, and material yield stress $\sigma_y$ is given as follow:

$$t_s = \frac{1}{2} R_s \frac{P}{\sigma_y}$$

For the cylinder part of the shell tank, the thickness $t_c$ is given by:

$$t_c = R_c \frac{P}{\sigma_y}$$

Therefore, the thickness of the circular cylindrical tank shell is twice the thickness of the sphere cape. $t_c = 2t_s$

For the weight of the sphere and cylinder shell.

$$W_s = 2\pi R^3 \frac{P}{\sigma_y/\rho} \text{ for the sphere shell and }$$

$$W_c = 2\pi R^3 \left(\frac{L}{R_c} - 1\right) \frac{P}{\sigma_y/\rho} \text{ for the cylinder.}$$

Where $\rho$ is the weight density of the material used in the tank shell.

The weight of the cylinder part of the tank made

$$W_c = \rho_t g \int_0^L \pi D dx \quad \cdots\cdots (1)$$
The pressure at any point $x$ is $p + ng \rho_p x$.

The minimum thickness of wall of tank, would be modified as:

$$\frac{t}{D} = \frac{p + ng \rho_p x}{2\sigma_y} \quad (2)$$

Substituting (2) into (1) gives

$$W_c = \pi D^2 \rho_t g \left( \frac{pL}{2\sigma_y} + \frac{\rho_p ng l^2}{4\sigma_y} \right)$$

The minimum thickness of the upper and lower spherical caps are given as following:

$$\frac{t_{su}}{D} = \frac{P}{4\sigma_y}$$

$$\frac{t_{sl}}{D} = \frac{P + \rho_p ng l^2}{4\sigma_y}$$

The minimum total weight of the tank:

$$W_t = \pi D^2 \rho_t g \left( L + \frac{D}{2} \right) \left( \frac{pL}{2\sigma_y} + \frac{\rho_p ng l^2}{4\sigma_y} \right)$$
The weight of the propellant contained in the tank:

\[ W_p = \rho_p g (L + \frac{D}{2}) \frac{\pi D^2}{4} \]

The ratio of minimum tank weight to propellant weight is therefore:

\[ \frac{W_t}{W_p} = \frac{3 \rho_t 2 + \frac{D}{L} (p + \frac{1}{2} \rho_p n g L)}{\sigma_y \rho_p 3 + \frac{D}{L}} \]

5.3. Material

The material used for the fuel tank is aluminum-lithium (Al-Li) alloy. The Al-Li alloy is a material that is stronger and lighter than aluminum. The commercial aluminum alloy contains 2.45wt% of lithium. The use of Al-Li alloy has already been used in many different aspects in the aerospace industry. The material has been used in the external fuel tank of the space shuttle, Atlas V and Delta IV EELV rockets.

The propellant tanks use Aluminum-Lithium alloy 2195(Al-Li 2195) as the material. The material has a density of 2685kg/m³. The yield strength \( \sigma_y \) of Al-Li 2195 is 521.6 MPa. This alloy is relatively light while it has high yield strength. This alloy contains 1.0% Lithium, 4.0% Copper, 0.4% Magnesium, 0.4% Silver and 0.12% Zirconium other than Aluminum.

Applying Tsiolkovsky's rocket equation, assume there is no external force, the delta-v (\( \Delta v \)) will be:

\[ \Delta v = v_e ln \frac{m_0}{m_1} \]

Where,

\( v_e \) is the effective exhaust velocity

\( m_0 \) is the initial total mass, including propellant

\( m_1 \) is the final total mass

Delta-v (Maximum change in velocity of rocket) from Mars to earth will be 11.32-7.84= 3.48 km/s.

The final mass of the spacecraft (crew module) is 8500kg.

The crew members, equipment and essentials needed is roughly 1500 kg.

Therefore, the final payload will be 10 000 kg.
The spacecraft is using the Merlin 1-D (vacuum) in space. One single engine 4333kg.

\[ m_1 = 14333 \text{kg} \]

The specific impulse provided by this engine is 311s.

By \( v_e = g_0 I_{sp} \),

Where,

\( g_0 \) is the acceleration due to gravity at the earth surface =9.81m/s\(^2\).

Effective exhaust velocity=3051m/s.

The initial mass of spacecraft with propellant is 44843 kg.

The mass of propellant used is 44843-14333=30510kg.

### 5.4. From Earth Orbit to Mars

Assume that the earth orbit the spacecraft depart from has an altitude of 300km. The orbital speed of the spacecraft at 300km is 7.73 km/s by using the equation

\[ v_o = \sqrt{\frac{\mu_e}{R_e + h}} \]

Using the same equation,

\[ \Delta v = v_e ln \frac{m_o}{m_1}, \]

the mass of the spacecraft when departing earth orbit(300km) can be obtained.

\[ \Delta v = 11.32 - 7.73 = 3.59 \text{km/s} \]

The final mass when the spacecraft flyby Mars is 44843kg.

The effective exhaust velocity of Merlin 1D is 3051m/s.

The mass of the spacecraft leaving the earth is therefore 145450kg.

And the propellant needed 100607kg.

The total propellant mass is 30510+100607=131117 kg.

The optimum oxidizer to fuel ratio is 2.56,

the mass ratio of LOX to RP-1 is 1.407.
So if the mass of the RP-1 is 131117.

The mass of RP-1 is 28492 kg and mass of LOX is 102625 kg.

The height of the fuel tank is determined by the density of the propellant and the diameter of fuel tank.

\[ W_p = \rho_p g (L + \frac{D}{2} \frac{\pi D^2}{4}) \]

The height of the RP-1 tank is 0.8m,

The height for the liquid oxygen tank is 5.16m

The weight of both tank is given by

\[ W_t = \pi D^2 \rho_t g (L + \frac{D}{2}) \left( \frac{pL}{2\sigma_y} + \frac{\rho_p g L^2}{4\sigma_y} \right) \]

For specific gravity \( g \) is 0.

The mass of the fuel tank can be simplified as

\[ M_t = \pi D^2 \rho_t (L + \frac{D}{2}) \left( \frac{pL}{2\sigma_y} \right) \]

The fuel tank mass of RP-1 is 2811 kg

Fuel tank mass of LOX tank is 46364kg.

Therefore, the mass of the spacecraft with fuel when it enters the orbit is

\[ 145450 + 2811 + 46364 = 194625 \text{ kg}. \]
5.4. From Earth to orbit

The characteristic equation is:

\[ \sum F = m \frac{dv}{dt} + v_e \frac{dm}{dt} \]

F is the total force acting on the rocket= thrust-weight. At vacuum, the Merlin 1D engine can provide 720kN thrust. The weight of the spacecraft will be

\[(194625+9\times4333) \times 9.81=2291832 \text{ N}\]

As there are 9 engines used at this stage:

\[ F=720\times9-2291832=4188168\text{ N}. \]

The exhaust velocity will be \(9.81 \times 300 = 2943\text{ m/s}\) as the specific impulse is estimated to be 300s. And the mass flow rate of the propellant will be

\[ -\frac{dm}{dt} = \frac{F}{g_0 i_{sp}} = 244.6 \text{ kg/s} \text{ for each engine.} \]

dv is the difference in velocity from 3500m/s to 7730m/s.

\[ dv = 7730-3500 = 4230\text{ m/s}. \]

From the calculation above, it is found that the engine has to burn for 92.6s. And the mass of propellant used will be

\[ 92.6 \times 244.6 \times 9 = 203850\text{ kg}. \]

The composition of the propellant is

- 44298 kg of RP-1 and 159552 kg of LOX.

Stage 1B has a diameter of 4m.

The height of RP-1 tank is 2.35m

The height for LOX tank is 9.14m.

The mass of RP-1 tank is 12828kg

Mass of LOX tank is 127776 kg.

The mass for stage 1B: \(194625+4333\times9+203850+12828+127776=578076\text{ kg.}\)

Stage 1A has two more liquid rocket boosters that are identical to stage 1B. The thrust will be:

\[ 650\text{ kN} \times 27 = 17550\text{ kN}. \]
The weight will be:

\[(578076+4333\times 18) \times 9.81= (656070) \times 9.81=6436047N.\]

The net force is:

\[17550\text{kN}-6436047=11114\text{kN}\]

The specific impulse at sea level: 282s. So the exhaust velocity is 2766m/s. The mass flow rate of propellant is 235kg/s per engine. The engine has to run for 80.1s. So, the total mass of propellant is:

\[235\times 80.1 \times 27=508234.5\text{kg}.\]

Where 110440 kg is RP-1 and 397794.5 kg is LOX.

The height of each RP-1 tank will be 3.43m.

While the height of LOX tank is 11.9m.

The weight of RP-1 tank will be 23373kg.

The weight for LOX tank is 207577kg.

The total weight of rocket complex is

\[578076+4333\times 18+508234+(23373+207577)\times 2=1626204\text{ kg}.\]

The stages have been shown in figure 5d.

![figure 5d different stages of fuel tanks.](image-url)
6. Reaction Control Systems

The reaction control system (RCS) is a group of small rockets that are used to adjust the attitude of the spacecraft in orbit. The system consists of several small rockets that can be fired, either simultaneously or individually, allowing small adjustments in attitude to be applied whilst the spacecraft is in orbit. This can be particularly effective for manipulating the spacecraft during re-entry or for causing small, intricate movements whilst docking the craft.

6.1. Propellant Selection

Before selecting the engine a range of different engines that could use different types of fuel were looked at. Early on ion drives and Hall Effect thrusters were ruled out due to the fact they only produced a small amount of thrust that would be too weak to move our spacecraft as required. Engines that used monopropellant fuel such as Hydrazine were also considered; however these fuels were very toxic and were less efficient that the eventual choice of a bi-propellant blend of fuel. A bio-propellant gives a superior performance to monopropellant fuel and is much more efficient. In modern systems, bi-propellants can be made to be non-toxic meaning the crew is at less danger if a leak occurred.

6.2. XR-3N22 Lynx RCS Engine

The XR-3N22 Lynx RCS Engine, developed by XCOR Aerospace, is a modern RCS engine that is currently under development. The engine produces approximately 23kg of thrust, a large amount of thrust for the systems requirement. The engine uses a proprietary, bi-propellant blend fuel that is entirely non-toxic which is pressure fed into the engines. The fact that the fuel blend has similar performance to traditional toxic fuels such as NTO/MMH is important; however it is a lot safer due to the fact it is non-toxic. The fuel is also much cheaper than current fuels such as hydrazine. The exact elements used in the fuel could not be disclosed by XCOR due to ITAR regulations. The mass of a single RCS, including the fuel tank filled is 27kg, with a second engine adding only 2kg. The engine, including the valves and electronics, is approximately 23cm long with a diameter of 10cm with the tanks being the size of a standard scuba tank. The tanks are Kevlar overwound to be able to cope with the demands of high pressure.

Image 6a: XR-3N22 Lynx RCS Engine thruster undergoes testing
The cost of a single system, which is only an estimate due to the engine being in development, would be approximately $3million for a single system (engine and tank). This cost may be lower, as the aim of XCOR is to make access to space affordable.

6.3. Apollo Reaction Control System

The RCS from the Apollo missions is fairly outdated technology wise however still manages to compete with its modern day rivals. Each thruster can produce 45kg of thrust, about double the amount of the Lynx RCS. The Apollo RCS also uses a pressure fed bi-propellant as fuel much like the Lynx system. The fuel for the RCS consisted from nitrogen tetroxide as the oxidizer and monomethylhydrazine as the fuel. The RCS from the Apollo missions came in clusters of 4 thrusters each weighing 2.3kg with a titanium fuel tank weighing 3kg and carrying 31kg of fuel. Each quad assembly measured 8 feet (2.4 m) by 3 feet (0.91 m) and had its own fuel tank, oxidizer tank, helium pressurant tank, and associated valves and regulators making the system much more complex and heavier than its modern counterpart.

Image 6b: An Apollo RCS quadrant showing its 4 thrusters at right angles to each other

6.4. Selection of Engine and Fuel

In the mission the modern lynx RCS will be used due to the fact it is a much simpler design and is less bulky overall. It would also be a lot more expensive to get hold of the Apollo RCS as it is no longer in use and the cost of manufacturing a new system would be too great. The fuel that X-COR designed specifically for their system, which the mixture cannot be revealed due to regulations, will also be used in the mission as the RCS fuel.
6.5. Positioning of Systems

6.5.1. Orbiter
At least 4 thrusters would be required for attitude control; however more systems would make manoeuvring the spacecraft a lot easier. Furthermore, a back-up system would be required in case the main system failed. Our orbiter will have 12 thrusters in 2 separate systems as the standard RCS and a further 6 as a back-up plan which would still enable us to control the spacecraft’s attitude. They will be located in different locations on the spacecraft; 1 system on the front of the spacecraft, the other system at the rear, and the backup system also located at the front, as it is more important to be able to control the nose of the spacecraft.

6.5.2. Boosters
The booster only requires a short time when the RCS is needed at the top of the suborbital flight path. For this reason less fuel is required to be carried to save weight. Due to the large size of the boosters, more systems must be installed to gain the appropriate effect; therefore, 36 thrusters will be used along the length of each booster to optimise our control, giving a total of 12 systems across the 2 boosters.

6.5.3. Cost and weight

<table>
<thead>
<tr>
<th></th>
<th>Cost</th>
<th>Weight</th>
</tr>
</thead>
<tbody>
<tr>
<td>For 1 System</td>
<td>$18 million</td>
<td>38kg</td>
</tr>
<tr>
<td>Each Booster</td>
<td>$108 million</td>
<td>228kg</td>
</tr>
<tr>
<td>Spacecraft</td>
<td>$54 million</td>
<td>114kg</td>
</tr>
<tr>
<td><strong>Total</strong></td>
<td><strong>$270 million</strong></td>
<td><strong>570kg</strong></td>
</tr>
</tbody>
</table>
7. Thermal Protection System

7.1. Why do we need one?
The thermal protection system (TPS) is one of the key considerations for any space vehicle design if the vehicle is required to come back to earth or land on another planet. Its job is to ensure that the temperature rise on the outer surface of the vehicle generated by drag and pressure does not affect the inner parts of the vehicle. Its design and reliability is, therefore, crucial for the operation of the vehicle and could directly affect the success of a mission.

Another reason why the TPS design is extremely important for the whole project is that it is usually associated with composing a very significant fraction of the mass of the vehicle. Furthermore, it has been assumed that despite this being a vehicle designed for the 2018 Mars mission, this is a potential re-launch vehicle (RLV), and therefore in this case many other key issues are directly dependent on the type of TPS. Those include fuel consumption, the number of reusability cycles, maintenance and refurbishment time and cost, vehicle reliability and safety, as well as overall vehicle cost.

7.2. Current available recommendations
Extensive research of existing and developing technologies was carried out in order to find the best up to date solution for the problem. Different alternatives included several types of ablative protection materials. Even though such materials have been used in the past some of them have known disadvantages such as high density, unknown oxidation resistance, thermal expansion problems etc. Furthermore the very concept of an ablative TPS does not suit the needs of this project—that is they are not an innovative solution and by their very nature are not the correct choice for a re-launch vehicle and this would induce recurring costs. Another option that was not chosen was to use material similar to those employed in the protection of the Space Shuttle. Although these materials have good thermal properties and are much more reusable than the ablative protectors, they have many disadvantages. A key disadvantage of ceramic tile, for example, is their weight. Additionally they have been seen as very cost and time ineffective when it comes to replacement and maintenance, which also does not suit the needs of this project very well, due to it being a relatively small craft in question. Some new metallic TPS concepts had been developed and also looked at in the research. After careful consideration, the following two things were agreed on:
Due to lack of new development in highly thermally resistive materials it would not be possible, at that stage, to replace reinforced carbon-carbon (RCC) as the thermal protection material for parts of the vehicle where extreme heating is observed, such as nose and leading edges. This is unfortunate since, even though RCC has excellent thermal resistive properties—it is estimated to be able to withstand temperatures of up to 1920 K- and is already tested as a suitable material for the purpose, RCC has very high density -in the order of 1680 to 1980 kg/m$^3$. Additionally this material choice is associated with high cost and high need of maintenance.
7.3. ASTRA ULTIMATE

For the rest of the windward area of the vehicle an advanced new type of TPS was chosen, which although still in development has proved to be promising thanks to a series of tests. The system of choice is called “ULTIMATE” and was developed in Europe within the frame of the German ASTRA ULTIMATE project. What is different about this system is that it is a metallic TPS. The panel of the system consists of a load carrying ‘stand-off’ metallic casing. Within that casing there is micro fiber insulation. The outer surface is formed by a honeycomb sandwich panel with a total thickness of approximately 10 mm and cubical (or feasibly hexagonal) cells made from thin foil gauges. Panels overlap in order to seal the structure and reduce aerodynamic drag and negative weather effect, as well as to seal the panel in case of thermal expansion.

![Ultimate TPS Panel Design](image1)

These panels are envisaged as a structure that is separate from the main body of the vehicle. The interface between the panels and the substructure is to be achieved via Ω-shaped stand offs. An additional lower frame will also be included to give a rigid interface with the substructure. Figure 7a shows a sketch of the envisaged design of a single panel. The ‘ULTIMATE TPS’ is intended to be used for most of the windward area of the vehicle and is aimed to withstand temperatures in the range up to 1170 K. Extensive tests have shown that the design is capable to withstand temperatures. The HOPPER project was used to carry out tests and simulations. Boundary conditions were set to be as follows:

- **Steady state γ-TiAl upper face sheet at initial temperature:** 293K
- **Steady state Aluminum tank wall initial temperature:** 20 K
- **Transient Aluminum tank wall temperature:** 20K

Results showed that while maximum temperature during a mission of the HOPPER vehicle would be $T=1070 K$ at $t=910 s$ while maximum substructure temperature would be $T=408 K$ at $t=1400 s$, which is well within the operational temperatures of the material to be used.

![Panel Assembly Array](image2)
7.3.1. Testing
Additionally structural test were performed. Although these test were inconclusive for some parts of the panels they showed that in general the structure is sufficiently strong to withstand the loads in normal operation.

In addition to the thermal and structural test other testing has been performed to qualify the ULTIMATE TPS for space missions. These tests include:

- Heat-Emitting Ability Measurements
- Building Strength Test
- Tensile Strength Test
- Compression Test
- Panel Assembly Testing (the assembly array can be seen in Figure 7b)
- Demonstrator Vibration test

After all tests have been performed and the results summarized it is stated that the ULTIMATE TPS has proven to be reusable for up to 5 cycles while if only the thermal domain is regarded the reusability of the system goes up to 20 cycles. This proves that, although the system is still in development, it is a suitable choice for the purposes of this project in 2018.

7.3.2. Loading
A single TPS panel is expected to have the dimensions of 500x500 mm and 40 mm in the direction perpendicular to the protecting surface. The honeycomb panels are to have a thickness of 10 mm each. Rough estimations show that this would result in each panel having an approximate mass of 23 kg, thus contributing to the total vehicle mass at the rate of 92 kg per m² of protected area. These are rough estimations and are based on density of titanium of 4.3 g/cm³ and density of the ceramic fiber insulation of 100 kg/m³. While making these estimations however, it was acknowledged that these are somewhat average values since densities vary, especially in the case of the ceramic fiber insulation types.

7.3.3. Maintenance
Another key advantage of the chosen TPS for the orbiter is the reduced maintenance cost and time. ULTIMATE, being a metallic system has the advantage of higher ductility and flexibility of materials compared to the ceramic tiles of the Space Shuttle for example. This would most definitely result in fewer damages during re-entry. This, in addition with the relatively low panel cost which will be discussed later in this report, would result in much lower maintenance cost. Furthermore because of its construction it should be easier to inspect and manipulate each panel separately which would require much less maintenance man hours. Turn cycle would also accelerate by the aid of the health management system which would notify the crew and the maintenance team of any faults in the system even
before the vehicle has landed. Considering there would be significantly low amount of
damage and the replacement of panels should be relatively simple due to the structure of
the system, an approximation of 1000 working hours could be made for the replacement,
refurbishment and inspection of the panels.

### 7.4. Cost

Cost is of course another important aspect that needs to be taken into account. Some
research on material allowed the approximation that producing one panel of the ULTIMATE
TPS would cost somewhere between 700 to 800 USD to manufacture (including materials
cost), which adds up to a price close to 3000 USD per m² of protected area.

### 7.5. Appendix

#### 7.5.1. Maintenance Flow Chart:

```
Landing

Gathering data from Health management system

Inspection of the faulty regions as indicated by health management system

Repair or replacement of panels and further inspections of additional faults and inspection of the substructure underneath the TPS panels

Testing

Any faults

- Yes: Fixing faults
- No: Clearing for launch

Launch
```
8. Avionics

8.1. On board computers: CPU (Mongoose-V)
The CPU (Mongoose-V) is selected as the CPU of our spacecraft. The Mongoose-V is a 32-bit microprocessor developed by Synova, Inc and sponsored by the NASA Goddard space flight center. It is a radiation hardened version of the MIPS R3000 CPU. The processor offers a highly integrated solution for many spacecraft processor application like embedded instrument controllers. It incorporated on chip cache memory, on-chip peripheral functions and full hardware support for IEEE-745 floating point.

8.2. Remote Terminal Units (RTU)
The RTU offloads the On Board Computer from analogue and discrete digital data acquisition and actuators control tasks. It will controlled by the Power Conversion and Distribution Unit and On-board Computer. This unit has function such as gather telemetry from sensors and units, responsible for conditioning of analogue sensor, control actuators and sensors and power distribution.

8.3. Command and Data Handling (C &DH)
This system handle all data received and sent by the spacecraft. The data is sent through the RF transmitter and received from the receiver unit. And the space link, or communication link between the spacecraft and the ground based Deep Space Network (DSN). The basic data flow consists of Telemetry(TM) and telecommands (TC). The downlink TM data can be Spacecraft HK data, Orbit (position) data, Payload data (science data), Telecommand reception status (CLCW) or Memory dump data. While uplink TC data will be either direct command to the spacecraft for reconfiguration or application-specific command.

8.4. Onboard buses and data networks
The requirement of a data bus is summarized by the ESA as the following. The data bus should be able to acquire synchronous data frames from sensors with controlled latency, transmit synchronous to actuators with controlled latency and transfer asynchronous and isochronous data packets between nodes. It should also provide a medium access control services to node, give accurate distribution of time data and time reference pulse and provide cross-strapping mechanism.

The SpaceWire developed by the ESA is selected as the data bus for the spacecraft. The system was standardized by in 2003 in ECSS-E-ST-50-12C. The SpaceWire facilitates the construction of high-performance on-board data-handling systems and increase compatibility between data-handling sub-systems. The system is easy to implement its
interface with any digital ASIC and FPGA device. It also reduces the system integration cost and allows re-use of data handling equipment across different missions. The SpaceWire can implement 5000-8000 logic gates, which is relatively small and hence reduce integration cost. The small integrated structure, combining the use of Low Voltage Differential Signalling (LVDS) reduce power consumption. The LVDS also gives fault isolation capability to the system. Besides, the SpaceWire device is radiation tolerant, making it resistant to damage caused by ionizing radiation in outer space. Lastly, the SpaceWire can rover from rapid link failure.

8.5. Navigation and guidance

DSN-Deep Space Network

Cruising

When the spacecraft is cruising, communication with ground control will be done through the Deep Space Network (DSN). The DSN is an international antenna network that supports space explorations. The network consists of 3 ground stations, each 120 degrees apart, such that the spacecraft can communicate with ground control at any time.

DSN help engineers to track the spacecraft. This navigation service is called "tracking coverage" and it includes Doppler, ranging and "Delta DOR" (delta differential one-way ranging). Delta-DOR uses two widely separated antennas to track the transmitting signals from the spacecraft in order to measure the time delay between signals arriving at the two stations. The technique of measuring this delay is DOR. The navigation will be using the S/X-band, which ranges from 2.3-2.8 GHz.
9. Vehicle Health Management

By converting data into functional information, Vehicle Health Management (VHM) is focused on the improvement of operational performance. The VHM system complements the fleet operation by monitoring, gathering and evaluating the available data. The engineering and maintenance staff will be able to make use of this information to make timely, cost-efficient, and repeatable maintenance decisions.

9.1. Pyrometer

A pyrometer is a device used to measure temperature and radiation. To monitor the temperature of moving elements during the fly-by mission an infrared radiation pyrometer, KT 15 II manufactured by Heitronics Infrarot Messtechnik could be used. It is a non-contact temperature measurement which is capable of measurements of temperature range from -50°C and 3000°C. In addition, it has a fast response time, has a high resolution and utilizes chopped radiation method allowing it to be extremely stable.

9.2. Isotope Wear Detector

An isotope wear detector which detects radioactive materials, during the fly-by mission it could be used to detect wear and tear of bearings and rotatory seals. The Mini Rad-V will be used for the requirement, it is developed by Laurus Systems and is a sensitive yet durable vehicle mounted radiation detection system. The system offers and instantaneous alarm and operates in temperatures ranging from -23°C and 50°C.

9.3. Fiber-optic Deflectometer

The fibre-optic deflectometer is used to detect bearing loading and deflections and is an improvised version of a traditional accelerometer. The system will provide enhanced diagnosis of bearing degradation of the bearing of the engine turbopump. As the system is under development, additional information is currently unavailable.

9.4. Spectrometer

A spectrometer is used to detect non-metal wear. Tuneable Diode Laser Absorption Spectroscopy (TDLAS) is an advanced system developed by Physical Sciences Inc. Utilizing diode laser as a light source for transmission it calculates the characteristic absorption which is pre-defined, creating a lower signal which is detected by a photo-diode. The TDLAS is useful as it is able to detect very low concentration measurements (Physical Sciences Inc, 2013).
9.5. Integrated Vehicle Health Management

Integrated Vehicle Health Management (IVHM) is defined as the competency of the system to perform tests and monitoring of current and future states of on a complex vehicle or system. IVHM comprises a system that consists a number of technologies that enables automatic detection, diagnosis, prognosis and to reduce the consequences of faults detected.

The aim of a Health Management system is to detect, predict and diagnose indications of faults in order to tackle defect efficiently. In addition, it is used to analyse the effects and provide a reliable monitoring system by defining the current health of the system and maintenance based on predictions, which reduces the total cost of maintenance.

9.6. Structure

Health monitoring sensors are integrated into a vehicle to dispatch information to a data processing unit. Critical data that require immediate attention will be manipulated and processed on board while the rest of the information will be processed off board. By transferring data which are less time critical allows all the historical data for the vehicle to be compared with current performance to provide a comprehensive report on the degradation trends.

The Open Systems Architecture for Condition-Based Maintenance (OSA-CBM) is a standard architecture for moving information in a condition based maintenance system. The following are the key parts within the OSA-CBM functional blocks, which will be elaborated in the following sections.

9.6.1. Data Acquisition (DA)
9.6.2. Data Manipulation (DM)
9.6.3. State Detection (SD)
9.6.4. Health Assessment (HA)
9.6.5. Prognosis Assessment (PA)
9.6.6. Advisory Generation (AG)

9.6.1. Data Acquisition

Data collected from sensors which are connected to the data acquisition system and converted via the Analogue-to-Digital Converter (ADC) module which reads the sensors.

The ACRA KAM-500 developed by Curtiss-Wright Controls has been chosen to facilitate data acquisition for the fly-by mission. Table 9a below shows the features of the data acquisition unit.
Features | Details
--- | ---
Rugged airborne data acquisition | MIL-STD environmentally qualified to operate in harsh environments
Reliable aircraft data acquisition | Driven by hardwired finite state machines
High speed flight data acquisition | High system throughput and data integrity
Compact and low power | Ideal when space and power are limited
Modular COTS hardware | 100+ plug-in modules fit all chassis
Powerful software | Intuitive, fast set-up and programming; tightly integrated with display/analysis

Table 9a: Features of the data acquisition unit

The ACRA KAM-500 allows a storage temperature of between -55°C and 105°C and an operating temperature of -40°C and 85°C. The ACRA KAM-500 requires additional heat sinks to be attached each chassis to be sufficient for the fly-by-mission. The ACC/HSK/001 is an add-on module which can be added to the ACRA KAM-500. The modification would also provide protection from radiation and convection.

9.6.2. Data Manipulation
Data manipulation is the conversion of flight data obtained to a programming language in order to allow manipulation of the data on a computer based program.

An additional ACRA KAM-500 can be used for data manipulation, with the aid of the KSM-500 KAM System Manager Tool Suite. KSM-500 KAM System Manager is range of tools designed for the ACRA KAM-500, it enables data to be manipulated, which will then generate the necessary information to the relevant display and analysis.

9.6.3. State Detection
State detection is the system that issues the status of the flight condition. The process can be carried out via an Emergency Detection System (EDS). The system will be used to detect imminent vehicle failure and to prompt the crew to initiate an abort. A reliable EDS should abide by guidelines as follows:

- Minimize parameters in order to minimize the possibility of a false abort as more parameters increase the complexity of the EDS, and potential for anomalous indications of failures leading to an abort.
- Parameters must be met in the event of a catastrophic fault which requires immediate abort.
- Minimum of two readings for an accurate level of confirmation and the reduction of false aborts.
- Detection of any problems during engine start-up to detect early degradation to initiate an early abort.
• Consider aborts which have taken place in the past.

The above mentioned guidelines can be achieved by the ATLAS EDS which has met all the necessary requirements, ground-rules as well as the assumptions. The combination of sub-systems and are classified as the “Higher-level” and “Low-level” parameters and is shown in the following:

“Higher-level” parameters:
• Pitch, Yaw and Roll Rates: Vehicle control indicator.
• Fuel Pressure: Fuel intake indicator.
• Vehicle Attitude Errors: Provides verification of vehicle rates for vehicle control indicator.
• Power Control: Provide high level indicators of power availability.
• Range Safety Officer Control: Shuts down engine to provide safe abort.
• Manual Engine Shut Down: Not required.
• Engine Performance Indicator

“Low-level” parameters:
• Liquid Oxygen (LOX) Starvation: Abort System.
• Pressurization Control Sensors: Abort System.

The availability of an existing ATLAS database which allows updates to take place ensures that the system will be able to function at its optimal state.

9.6.4. Health Assessment
Health assessment is the development of technologies to determine the nature as well as the severity of an adverse event and to determine if it will lead to a further series of failures. Other than provide fault isolation, it also provide evidence of anomalies. The aim is to allow the flight to be safer, more reliable as well as being cheaper to a certain extent.

ARINC 653 (Avionics Application Standard Software Interface) is a standard used for software within the Integrated Modular Avionics (IMA) architecture or in space applications. ARINC 653 allows integration of different systems into a single hardware, which would allow the overall robustness of the system to be improved. The ARINC 653 Health Monitoring mechanisms would also assist in the isolate of faults and prevents the fault or failure to propagate.

9.6.5. Prognosis Assessment
The prognosis assessment calculates the Remaining Usable Life (RUL) of the component based on the data processed by the health assessment and to enhance the reliability. It involves the estimation of component health from its material condition and/or its
functional behaviour and the trending and prediction of this health with usage. Prognosis assessment also involves scientific analysis that use past performance data to predict the future equipment behaviour as well as the RUL.

Analogue telemetry could be used for prognosis assessment, an analogue telemetry interface is included in circuits and mechanisms, and therefore the information produced reflects the data for equipment performance. Equipment with predictive algorithms that utilize equipment analogue telemetry could be used to measure equipment performance (ATP) to identify any equipment that would fail prematurely. Predictive algorithms break down the telemetry behaviour which will illustrate the presence of accelerated aging which may be present.

9.6.6. Advisory Generation

Advisory generation is the Decision Support System (DSS) of IVHM. By utilizing the information obtained from the health assessment and prognosis assessment, health reports are generated. The goal of advisory generation is to calculate algorithms of the functionality. Appropriate advice will then be issued, should the functionality fall below 100%.

The RAD750 6U-160 could be used for advisory generation; it is a single-board computer which utilized the PowerPC RAD 750 microprocessor. The microprocessor is radiation hardened and has an operational range of up to 200MHz. The rail temperature range of the RAD750 6U-160 is between -28°C and 70°C and weights 1220 grams.

9.7. Engine Management

9.7.1. Problems

Table 9 below shows a list of errors which are expected to occur in the engines:

Table 9b: Errors expected to occur in the engines

<table>
<thead>
<tr>
<th>Type of Problem</th>
<th>Explanation</th>
</tr>
</thead>
<tbody>
<tr>
<td>Bolt Torque Relaxation</td>
<td>Loosening of container due to extreme vibration</td>
</tr>
<tr>
<td>Coolant Passage Split and Leakage</td>
<td>Exposure of thrust chambers in high pressure, temperature and vibration during the operation, reducing the maximum kinetic energy production</td>
</tr>
<tr>
<td>Tube Cracks</td>
<td>Caused by hot liquids and gases</td>
</tr>
<tr>
<td>High Torque</td>
<td>Caused by high friction, as a result from high temperature and pressure, between static and rotating parts which can eventually result in complete failure</td>
</tr>
<tr>
<td>Turbine Blades Cracking</td>
<td>Occur because of localized heating of turbine caused during harsh conditions resulting in lower efficiency and balance</td>
</tr>
<tr>
<td>Condition</td>
<td>Description</td>
</tr>
<tr>
<td>----------------------------</td>
<td>-----------------------------------------------------------------------------</td>
</tr>
<tr>
<td>Bellow Failure</td>
<td>Fatigue of the bellows caused by excessive transient energy and velocity</td>
</tr>
<tr>
<td>Electrical Connections Failure</td>
<td>Due to faulty circuits as a result of excessive vibration</td>
</tr>
<tr>
<td>Bearing Damage</td>
<td>Caused by high pressure on the bearing during flight</td>
</tr>
<tr>
<td>Tube Fracture</td>
<td>Occurs because of fatigue as a result of high vibration</td>
</tr>
<tr>
<td>Turbo-pump Leakage</td>
<td>Occurs because of harsh conditions or material damage</td>
</tr>
<tr>
<td>Lube Pressure Damage</td>
<td>Occurs due to contamination which obstructs the flow</td>
</tr>
<tr>
<td>Valve Failure</td>
<td>Mainly caused by friction and contamination</td>
</tr>
<tr>
<td>Internal Valve Leakage</td>
<td>Resulted from harsh conditions, contamination and friction</td>
</tr>
<tr>
<td>Regular Discrepancies</td>
<td>A combination of regular errors as a result of contamination</td>
</tr>
<tr>
<td>Hydraulic Control Assembly Failure</td>
<td>Occurs as a result from high pressure</td>
</tr>
</tbody>
</table>

**9.7.2. Solution**

Most of these problems could be detected and prevent on the ground when performing the diagnosis and prognosis assessment. Alterations could also be made or improved to the design or material used for the shuttle.

During the fly-by mission Emergency Detection System (EDS) would be able to transmit data obtained from the sensors to the ACRA KAM-500. The data is then constantly sent back to the ground system on a predetermined time interval. Officers will use the data provided to provide advices on the relevant actions to carry out. On the shuttle, advices provided from the RAD750 6U-160 could also be used. Further improvement in technology might allow robots designed to assist and solve the problem during the course of the mission.

**9.8. Cost**

Table 9c below shows the estimated cost of the systems:

Table 9c: Estimate cost of operating the systems:

<table>
<thead>
<tr>
<th>Influencing Factors</th>
<th>Amount (USD)</th>
</tr>
</thead>
<tbody>
<tr>
<td>Monthly cost of labour in French Guiana</td>
<td>1,400.00</td>
</tr>
<tr>
<td>Total cost of systems</td>
<td>112,000,000.00</td>
</tr>
</tbody>
</table>

The average monthly wage for labour in French Guiana is $1, 400. A total of $240, 000 will be set aside to all aspects of labour. In addition, the total cost of the systems is about $112 million, which brings the total cost for VHM to come to total of $112, 240,000.
10. Human Factors
There are many effects that a manned spaceflight to mars will have on the human body which makes the entire mission very complex. The human body is very sensitive and putting it in such a harsh environment for a period of greater than 500 days brings many dangers and complications to the mission.

10.1. Food
Since this is a manned mission, the humans need to maintain the crucial elements they require for life. Food is an element that is difficult to implement in this mission as the spacecraft cannot simply be restocked like the ISS. This means that the crew must carry all the food which they will require for the entirety of the mission. Based on the food intakes of current astronauts, 230kg of food would be required for the 500 day trip to mars. This not only adds weight to the spacecraft, but also takes up valuable space in the ship which could be used for other systems or experiments. The astronauts’ food must also be nutritious to maintain their health and a range of flavours would be preferable. A solution to this issue would be for astronauts to grow their own food in the spaceship during the mission. Although this technology is only under development, this would allow the crew to grow fresh fruit and vegetables such as carrots, tomatoes and strawberries which would give them fresh tastes and keep them healthy.

10.2. Human Health
The crew’s health is also crucial during the mission and must be maintained to ensure they can carry out their daily tasks and missions. Only basic medical supplies could be taken on the mission so only the healthiest and most able astronauts would be selected. The effect of radiation from space would also damage the long term health of the astronauts if nothing was implemented to prevent this. To limit the astronauts’ radiation exposure we have included a strong protection to radiation on the outside shield of the spacecraft which is explained later in the report.

10.3. Psychological Impact
Having two humans isolated in a small spacecraft for over 500 days brings many psychological challenges to the mission. With no real-time communication with anyone else other than the other crew member, the astronauts could suffer from mental health issues which could severely compromise the mission. Furthermore, these issues would not be helped due to the lack of mental health services available as psychologists will be far away on earth. Being locked in a confined space could be enough to drive someone insane, so stringent testing would need to be implemented and only the mentally strongest people could be selected for the mission.
11. Ground Systems

From the start of looking at ground systems it became obvious that an existing spaceport had to be used due to the fact creating a new spaceport, with all the required facilities would far exceed our budget. For this reason, we considered the three most popular and largest spaceports in the world: Baikonur in Kazakhstan, Kourou in French Guiana and Cape Canaveral, Florida, USA.

![Image 11a: Location of possible ground systems](image)

When choosing which location to select as the main base for ground systems multiple important factors were considered. It was required that the spaceport had multiple, high quality, facilities such as a launch facility, vehicle maintenance/construction facility as well as locations to house mission control and astronaut preparation areas. The sites climate was also considered as poor weather conditions frequently delay launches which not only waste money, but can affect the time we have to reach the appropriate trajectory. The launch inclination was also looked at as these effect the possible trajectories available to use. The latitude of the launch location was also a critically important factor as the closer to the equator the spaceport is, the easier it is to reach orbit due to the fact the orbital speed is fastest at the equator, caused by the earth rotation being fastest at the equator.

<table>
<thead>
<tr>
<th></th>
<th>Baikonur</th>
<th>Cape Canaveral</th>
<th>Kourou</th>
</tr>
</thead>
<tbody>
<tr>
<td>Latitude</td>
<td>46°</td>
<td>28°</td>
<td>5°</td>
</tr>
<tr>
<td>Launch Inclination</td>
<td>49°-99°</td>
<td>28°-57°</td>
<td>5°-100°</td>
</tr>
<tr>
<td>Facilities</td>
<td>Everything needed</td>
<td>Everything needed</td>
<td>No sufficient runway</td>
</tr>
<tr>
<td>Climate</td>
<td>Huge temperature variations</td>
<td>Risk of hurricanes and thunderstorms</td>
<td>Stable climate</td>
</tr>
</tbody>
</table>

*Table 11b: Ground system location comparison*
For the launch of the mission Kourou in French Guiana will be used as the location for main hub containing launch facilities and other ground systems. Kourou is the preferred option as it is closest to the equator, making it a lot easier to reach a low earth orbit, allowing the spacecraft to weigh more that it would if it was being launched from the other spaceports. Kourou also has a very stable climate which creates ideal conditions for launch as there are fewer worries of launches being postponed. Moreover, in the past it has been proven that a poor climate can catastrophically affect certain components of the spacecraft. Kourou also has numerous excellent facilities that could be made use of, including:

- Four launch pads
- A final assembly building
- Administration Centre
- Satellite tracking station
- Weather station
- Control Centre (Soyuz)
- Residential area in town

Despite these fantastic facilities, Kourou lacks a sufficient runway for a carrier vehicle. Even with this setback, it does not affect the mission due to the fact the launch will take place from a Launchpad and carrier vehicle is not required. This makes Kourou the ideal location for the missions’ ground systems and launch location.
12. Power Requirements

12.1. Why is power important?
It is important to consider the power requirements as they have a significant effect on the total initial mass of the spacecraft, especially with chemical based non-renewable power sources. Also related is that the power systems need to be able to function during a great variance of load conditions. The innovations in batteries, auxiliary power units and fuel cells in recent years have made huge steps forward and provided many new exciting opportunities within engineering projects.

12.2. Power and Energy Requirements
Energy will be needed for the adequate function of many systems on the Mars Flyby spacecraft. These Systems include: Avionics Systems; IVHM Systems; Reaction Control Systems and Human energy consumption.

12.3. Avionics
The Avionics Systems will include predominantly the navigation system, whilst at particular times during the flight it will also be involved in sending commands to alter the states of other systems. For the Navigation part it will not withdraw any more than 100W and this is the same for the systems controls. Totalling up to 200W but averaging out to a rough maximum average of 110W with a 10% safety margin this will mean that a 24V battery would be needed to provide a current of 5 Amps. A 24V rechargeable battery will be enough to keep this system running.

12.4. Integrated Vehicle Health Management
There are a number of known systems that will take part in the IVHM of the spacecraft, all of the following will be in operation for a; modules that are part of the KAM-500 will require a maximum of 3W at any one time and there will be a subsystem containing 13 on these modules, there will be two such sub-systems hence bringing the maximum for that type of system to 78 Watts. The PowerPC RAD750® microprocessor (encompassed in a ‘6U CompactPCI’ single-board computer) will dissipate a maximum power of 14W at any time during the mission. The Infrared KT15 IIP Series Infrared Radiation Pyrometer will dissipate no more than 36mW. The Mini Rad – V (a radiation detection system) is a small, almost passive device generally will not use any more than 0.5W, but when alarming it may use up to 5W and this will be averaged to a maximum average of 0.55W for the whole mission (assuming that the system spends much less than 100th of the time of the mission sounding any alarms). There is also a project in development called ATLAS and it is deals a major part
in the IHM including abort procedures and this system is estimated to take up 100W putting the total power requirement for the IVHM 192.586W and with a 5% safety margin this puts the total estimated consumption rate at 202.2158W and will be rounded down to 202W. This system can also be powered by another 24V rechargeable standard aircraft battery.

12.5. Reaction Control
The reaction control system acts more as a correcting element than a basis element of propulsion, it’s more of an instantaneous source of short bursts of propulsion. It’s main purpose is to put the spacecraft back on the correct trajectory if it diverts from the one planned, for this reason it is noted that it will not require a great overall amount of energy through the flight and a 24 Volt Aircraft battery will suffice.

12.6. Human Effects
The energy requirement that were taken into consideration were of 4 main types, “air” generating, water generating, toilet & ‘shower’ usage and internal thermal control. Temperature control will be maintained by three radiators (apart from the thermal insulation if the spacecraft) on average will take roughly 4.9kW of power throughout the mission. Water recycling will be carried out by a NASA’s water recovery system which also a 1kW source of power; this machine will be working roughly a quarter of the time so an average of 250W will be needed. 1 kW will be needed for the air production. The toilet and shower systems will be in use for a rough time of 3 hours per day and will also be 1kW to run each, hence an average of 1kW x (3hrs/24hrs) = 125W so in total the Human consumption on the Fly By mission will be 6.275kW or 76,000kWh for the whole mission. For this 16 panels (at 1.650 x 0.99 metres) of solar panels (1kW/m^2 at Earth down to 432W/m^2 at mars) will be deployed to transfer the energy required and produce 6.912kW of energy creating a 10% safety margin. The excess may be used to recharge the rechargeable batteries.