## Introduction

It is imperative that the winged-jetpack has a reliable and adequate propulsion system which will allow for mission requirements to be fulfilled in an efficient and safe manner. The key requirements of the propulsion system are:

* Provide a flight time greater than 30 minutes
* Able to provide a thrust of double the craft weight
* Utilize propellant which can be sourced on the Mars surface
* Be reliable, safe, and trustworthy

As outlined in the key requirements, the safety of the Astronaut operating the system is of paramount importance and will be considered, where necessary, in each design decision. The following analysis will utilize mars conditions specified in numerous sources (Table 1) (Katharina Lodders and Bruce Fedly, 1998).

Table 1 - Mars Atmospheric Conditions

|  |  |  |  |
| --- | --- | --- | --- |
| Property |  |  |  |
| Value |  |  |  |

## Problem Definition

### Design Constraints

Approximate conditions for the craft were provided and are outlined in the bullet points below. The propulsion system was required to produce thrust two times the weight of the craft, and be able to supply a lift force equal to the craft weight (for cruising/gliding conditions). Main design constraints have been provided below:

* Craft Mass: 350kg
* Craft Weight Force:
* Full thrust requirement:
* Cruise Thrust requirement:
* Combustion Chamber Radius: 250mm

### Propulsion Selection

This document isn't intended to provided a complete justification into design selection; for more information please feel free to contact the team. Literature research suggested the ideal propulsion system would be a liquid propellant rocket due to two key advantages:

* Variable thrust (Limit acceleration felt by user, assists control systems)
* Mars produced fuel/oxidiser
* Extensive literature and prior art available
* Proven use in vacuum environments
* Reliable and (relatively) safe

### Fuel/Oxidiser Selection

The fuel was selected to be carbon monoxide and oxidiser to be liquid oxygen. The main reasoning behind this decision was the fact that these substances can be produced on Mars using electrolysis and carbon dioxide (which makes up 95% of Mars’ atmosphere). Electrolysis is a technique that uses a direct electric current to force a chemical reaction that turns carbon dioxide into carbon monoxide and oxygen. During this process carbon monoxide and oxygen are produced in a two to one ratio. As the fuel and oxidiser react in the engine they will come together to produce carbon dioxide gas as the exhaust.

Such a system has been previously researched and proposed for a ‘Mars rocket vehicle’ (see Landis, G. A. and Linne, D. L. 2001). Research on this engine and fuel selection indicates it will have an expected specific impulse of 250s (under the conditions of Earth’s gravity).

### Combustion Chamber Conditions

Approximate combustion chamber conditions are required to commence the rocket design phase; namely the pressure and temperature. These conditions are usually approximated for a given fuel and oxidiser combination, however in this case no literature could be found detailing this information. “NASA CEA” can be utilized to determine approximate combustor conditions for given systems, however a lack of access meant other approximation methods were implemented.

The combustion chamber temperature can be approximated by determining the adiabatic flame temperature of the combustion process. However, for simplicity both the pressure in the chamber and temperature are assumed based on literature. The chamber pressure is taken as 2.75mPa and the chamber temperature as 3000K. Though not completely accurate, these combustion properties should provide a somewhat accurate starting point. Further design iterations will increase accuracy.

### Assumptions

The main assumptions which will be implemented to simplify the design process and allow for analytical calculations to be conducted are outlined below. It is important to note that these design assumptions has been sourced from Vincent Wheatley’s propulsion course notes (Wheatley, 2016). The theoretical analysis utilizes information from both "Rocket Propulsion Elements" and Vincent Wheatley's propulsion course notes.

* Conditions at ground level will be used for entire flight path
* Constant properties (specific heat ratio, specific gas constant)
* Steady state flow
* Propellant gasses are homogeneous
* Propellant gasses are ideal gasses
* No friction or boundary layers on nozzle walls
* Uniform flow and velocities throughout rocket
* All gasses leave the engine axially

## Performance Analysis

The specific impulse () is a function of gravity and the expected value of 250s needs to be modified to be in terms of Mars’ gravity. Let denote the exhaust velocity and gravity (in ). Then,

The thrust equation gives, , where is the cruise thrust (in ) and is the mass flow rate (). Then,

Integrating this equation and using the fact that the initial weight of the craft is 350kg gives,

where is time measured in seconds.

Assuming the mass includes of fuel and oxidiser, the time at which the fuel and oxidiser runs out can be calculated using this equation.

This is using the mass flow rate calculated at full thrust, when in reality full thrust will not be necessary for the entire flight time. More likely, 90% thrust will be necessary for take-off and landing. It was calculated that flight will be eight times more efficient when the craft is horizontal gliding, thus the rest of the flight need only use 12.5% of thrust.

Assume take-off and landing will take 2 minutes total. Then the amount of fuel used for take-off and landing can be calculated:

Then, the remaining fuel can be used for time horizontal gliding.

This with of fuel the aircraft can travel for a total of 5.65 minutes on full thrust and 32.85 minutes comprised of 2 minutes of 90% thrust used in take-off and landing, and 30.85 minutes of 12.5% thrust during horizontal gliding.

### Propellant properties

First the specific gas constant for the propellant will be calculated. Taking the universal gas constant and diving by the weighted average of the molecular weight will achieve this. Note that the weighted average was calculated on a molar basis for combustion products. An value of 283.46 was calculated.

The specific heat ratio would also be calculated using a similar weighted average method. However, the specific heat ratios for both oxygen and carbon monoxide are 1.4. Therefore a value of 1.4 will be utilized for the following analysis.

## Rocket Design

### Nozzle Throat

The main aim of the initial design is to propose some rough dimensions for the nozzle and the combustion chamber. The throat area is calculated using relationships outlined in "Rocket Propulsion Elements" (Sutton, 1992). The "nozzle coefficient" ( is calculated using a back pressure equal to the atmospheric pressure (perfect expansion) and the thrust correction factor () is assumed to be 98%. A throat diameter of 18.94mm is calculated.

### Nozzle Exit

The exit area of the nozzle will be calculated using a derivation from isentropic flow relations outline in Wheatley's propulsion notes (Wheatley, 2016). Again, analysing the nozzle with perfect expansion allows the pressure at the exit to equal the back pressure (600 Pa). Multiplying by the throat area and substituting in the specific heat ratio of 1.4 yields an exit area of 0.031313 m^2 and an exit diameter of 200 mm. An expansion ratio of 110 is not out of the ordinary and is reasonable for preliminary design iterations.

### Combustion Chamber

The volume of the combustor can be approximated by multiplying the throat area by a characteristic length (Sutton, 1992). Sutton defines the characteristic length as "the length that a rocket of the same volume would have if it were a straight tube and had no converging section" and recommends a particular range based on propellant. A characteristic length of 40 is utilized for this analysis. Once a recommended chamber volume is produced the desired combustor radius can be used to determine the corresponding combustor length. A chamber diameter of 250mm is sufficed and an acceptable length of 228.57mm is supplied. A combustor length within the same order of magnitude (similar) to the diameter of the combustion chamber is expected. This will integrate into the craft nicely as well.

Radius supplied as 125mm or 0.125m:

Figure 1, adapted from Wheatley's notes, shows a summary of dimensions and general schematic.

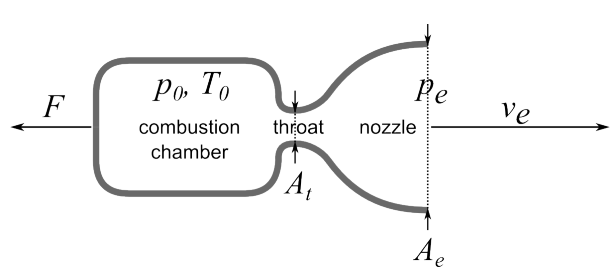
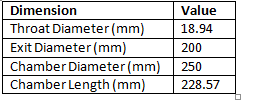
 

Figure 1 - Rocket Design Schematic (Wheatley, 2016)

## Limitations & Recommendations

### Mass Flow Check

In specific impulse calculations Earth's gravity is sometimes utilized as a method of normalising. The method in which the specific impulse used in this analysis was determined should be researched to ensure the accuracy of our mass flow rate calculation. Alternatively, the mass flow rate at the throat of the rocket could be calculated (Wheatley, 2016). A steady state assumption would mean the mass flow at the throat is equivalent to the mass flow throughout the entire rocket system. Isentropic flow relations for the throat density and velocity can be implemented to do this. The mass flow utilized in our analysis didn't differ much from the mass flow rate at the throat, thus out analysis holds. The assumption that the flow would be choked at the throat is important for this analysis. Isentropic relations for density and velocity at the throat:

### Pressure Differential Stress

Low pressure outside the rocket casing will cause a large pressure differential. Materials that can survive these high stresses must be evaluated. Numerous rockets have flown in vacuum in the past, so it's evident that these stresses can be survived. A literature review of spacecraft would easily indicate which materials to use.

### Injector Design

The injector system design must cater for the low atmospheric pressure in which the rockets will be operating. "Rocket Propulsion Elements" provides an in-depth guide to injector selector and design. Further analysis would continue here.

### Combustion Chamber Temperature

The combustion chamber temperature can be approximated through an adiabatic temperature calculation. This analysis isn't very involved, however for simplicity of calculations a combustion chamber temperature assumption was made. For future design iterations a more exact chamber temperature would be calculated this way; further down the line computational fluid dynamics or the NASA CEA would be implemented to find an exact value.

## Bibliography

Landis, Geoffrey A., and Diane L. Linne (2001). ‘Mars rocket vehicles using in situ propellants’ Journal of Spacecraft and Rockets. 38(5): 730 – 735.\

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