Delft University of Technology
Design Synthesis Exercise

Design of a Controllable Inflatable Aeroshell
Baseline Report

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Preface

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Dear Reader,

This Baseline report is part of the deliverables for the Baseline Review (BR). It reviews the functions and requirements and introduces initial concepts for the design of an inflatable guidable aeroshell suitable for manned space flight. This project follows in the wake of a number of investigations performed by the National Aeronautics and Space Administration (NASA) on the viability of such a vehicle for entry and re-entry of atmospheres. Key driver for a new solution to (re-)entry using a heat shield is a reduction of mass as compared to conventional solutions, greatly benefiting launch costs for missions to explore and inhabit extraterrestrial environments, such as Mars.

Design Synthesis Exercise Group 02

Summary

To fulfill the need to carry human payload to Mars one solution is a Controllable Inflatable Aeroshell (CIA). The project has to demonstrate the feasibility of such a controllable inflatable aeroshell. The design of this product can be considered as complex. The first steps in simplifying the design process have been summarized in the Project Plan (PP). The next steps include a literature study to familiarize with the subject. This literature study consists of an analysis on past solutions alongside with the investigation of the primary disciplines involved in the design of a re-entry mission. Then a functional and requirements analysis is performed to generate design options for the different mission aspects of the re-entry vehicle. This baseline report also comprises the market analysis, budget breakdown, risk assessment and an approach with respect to sustainable development. Also a tool is developed and presented to make early predictions for the astrodynamic characteristics.

To arrive at the concepts shown in a Design Option Tree (DOT), it is advised to start with a functional analysis. In this analysis a Functional Flow Diagram (FFD) is made to show the logical order of functions the re-entry vehicle must perform. The functions can also be categorized in a Functional Breakdown Structure (FBS). Then the requirements are analyzed. This is done using a requirements discovery tree which shows how different subsystem requirements flow down from the top level requirements and constraints. The list of requirements is summarized in Appendix A.

A market analysis is performed which aims to minimize the risk of selecting a wrong combination of function and technology for the customer. A Strengths, Weaknesses, Opportunities and Threats (SWOT) analysis provides an overview of the primary characteristics of the proposed Controllable Inflatable Aeroshell (CIA). The design process is exposed to several risks, primarily schedule overruns and insufficient technical performance. In order to identify, analyze and manage risks a risk mitigation plan is made, which uses risk mapping as a qualitative method and Technical Resource Budgeting (TRB) as a quantitative method. Within this mission, sustainability will not have the highest priority in the design process since the total impact of a single interplanetary mission on the environment is relatively small. It will, however, play a role when faced with design choices. If budgets and technical performance allows, a more sustainable design is the preferred one.

After this the initial design concepts are generated. For this a Design Option Tree (DOT) is used. First a DOT for each of the following aspects is constructed: the trajectory, the shape and the control. In the separate trees the fully unfeasible options are eliminated. Then feasible combination of the options in the three trees are sought.

This baseline report is concluded with the feasible combinations of the concept-options. Future work includes narrowing down the design options to five concepts. In the Mid-Term Review (MTR) these concepts will be reviewed and a trade-off process will help with the selection of the final concept that shall be designed with more detail for the Final Review (FR).

Acronyms

BC Ballistic Coefficient.

BR Baseline Review.

CFD Computational Fluid Dynamics.

CG Center of Gravity.

CIA Controllable Inflatable Aeroshell.

DOT Design Option Tree.

FBS Functional Breakdown Structure.

FEM Finite Element Model.

FFD Functional Flow Diagram.

FR Final Review.

HIAD Hypersonic Inflatable Aerodynamic Decelerator.

IRVE Inflatable Re-entry Vehicle Experiment.

IRVE-3 Inflatable Re-entry Vehicle Experiment-3.

IRVE-II Inflatable Re-entry Vehicle Experiment-II.

MHD MagnetoHydroDynamics.

MTR Mid-Term Review.

NASA National Aeronautics and Space Administration.

OBS Organizational Breakdown Structure.

PP Project Plan.

RCA Resource Contingency Allowance.

RDT Requirements Discovery Tree.

SE Systems Engineering.

SPFs Single Points of Failure.

SWOT Strengths, Weaknesses, Opportunities and Threats.

TAEM Terminal Area Energy Management.

TBD To be Determined.

TCS Thermal Control System.

THOR Terrestrial HIAD Orbital Reentry.

TPM Technical Performance Measurement.

TPS Thermal Protection System.

U.S. United States.

List of symbols

A	$[m^2]$	Area	19
C_D	$[-]$	Drag coefficient	19
D	$[N]$	Drag	7
E	$[J]$	Energy	8
KE	$[J]$	Kinetic Energy	8
L	$[N]$	Lift	7
M	$[-]$	Mach number	7
PE	$[J]$	Potential Energy	8
U	$[J]$	Internal Energy	8
V	$[m/s]$	Velocity	19, 23

Contents

List of Figures

List of Tables

1 Introduction

There is a need for a feasible and cost-effective vehicle that can transport human payload to the surface of planetary surfaces such as Mars. One solution for this is an inflatable aeroshell, stowable within conventional launcher configurations in undeployed condition. The design of a controllable inflatable aeroshell can be considered as complex, featuring interaction between many different disciplines. To reduce the complexity of the design problem step have been made in the Project Plan (PP). As part of the Baseline Review (BR), this baseline report reviews the functions, requirements and initial concepts.

The purpose of this report is to show the current progress of the group in the design process. After the PP, where the specific approach of the group to the technical and management aspects of the design project has been defined, in this report the functional and requirement analysis are used to come up with feasible design options. Again Systems Engineering tools are used to help with this phase. For the functional analysis FFD and FBS have been used. The Requirements Discovery Tree (RDT) forms the basis of the requirements analysis and several DOTs have been made to find feasible design options.

First the results of a literature review are presented in chapter 2. This includes a overview of the past mission and a overview of all the currently available and used technologies for the relevant departments from the Organizational Breakdown Structure (OBS). Preliminary tool are developed for a trajectory which is discussed in chapter 3. Chapter 4 performs a functional analysis and chapter 5 performs a requirement analysis. A budget breakdown is discussed in chapter 6. After that a market analysis and a risk assessment is performed in chapter 7 and 8 respectively. The approach with respect to sustainable development is stipulated in chapter 9. Finally the design options are discussed in given in the form DOT in chapter 10.

2 Literature Review

This chapter provides an overview of findings of literature research on the topic of (re-)entry vehicles. Firstly, past solutions and their development have been investigated for the dual purpose of reference material to be used for preliminary sizing and design and the acquisition of knowledge on mission level. Secondly, each of the primary disciplines involved in the design of a re-entry mission is investigated, namely structural design, TPS, aerodynamical design, orbital mechanics, atmospheric modelling and control. These disciplines are investigated in terms of their application and new technologies (e.g. the materials typically used for thermal protection) as well as for methods used in their respective analysis and design (e.g. the use of Finite Element Model (FEM) for structural analysis).

This chapter is structured as follows. The first section gives an overview of past re-entry vehicles; the second section gives an overview of past and ongoing investigations in the use of inflatable aeroshells; subsequent sections focus on the primary distinguishable disciplines involved.

2.1 Overview of present and past (re-)entry vehicles

This section gives an overview of (re-)entry vehicles, primarily to obtain a set of reference vehicles to aid design and sizing of the payload capsule in later stages and additionally to review solutions used in the past to perform (re-)entry for human spaceflight. At this point no structure supporting the deceleration has been chosen yet, as such only payload capsule size parameters are considered. It can be argued, based on human payload constraints, that the attached payload capsule for this mission will have similar characteristics. Table 2 displays some characteristics related to the payload capsule which can be used as indicative values [1].

[1]Principal values from:
URL: http://www.nasa.gov/sites/default/files/167718main_early_years.pdf, Accessed: 28 April 2015
URL: http://www.braeunig.us/space/ Accessed: 28 April 2015
URL:http://wsn.spaceflight.esa.int/docs/Factsheets/35%20Soyuz%20LR.pdf Accessed: 28 April 2015
URL: http://www.lpi.usra.edu/lunar/constellation/orion/factsheet.pdf Accessed: 28 April 2015

Table 2: Reference missions for payload module sizing.

Mission	Apollo	Soyuz TMA	Shenzhou	Gemini	Mercury	Orion
Years [yr]	1964-1975	2010-2014	1999-	1959-1963	1959-1963	Future
Reentry module mass [kg]	5806	2900	3240	3402	1118	8777
Habitable volume [m^3]	6.17	3.5	6.0	2.55	1.9	11
Diameter [m]	3.9	2.17	2.52	2.3	1.9	5
Length [m]	3.5	2.24	2.5	3.4	5.2	Unknown
Crew size (max) [$persons$]	3	3	3	2	1	6

It must be noted that Table 2 displays typical values only to be used as first indicative values. For example the diameter is typically a maximum value since no single value can be supplied due to the cone like shape of most reentry vehicles. Moreover these designs include the size and mass of the deceleration mission of which the latter typically includes a heavy duty heat shield. Most re-entry vehicle base designs were used multiple times with minor design changes and a single externally communicated design name. As such the values in the table above should be used with proper care as indicative values only. Habitable volume estimation also depends on the mission duration and may be considerable. A study on the estimation of these parameters is given by [1]. Although this study focuses on a lunar mission it still underlines many of the important aspects with respect to payload module sizing which are applicable for Mars missions as well. It may as such prove a proper foundation for payload module sizing.

From the above mentioned reference missions in Table 2 especially the future Orion mission, currently being designed, is of great interest. The Orion crew exploration vehicle is planned to go to the Moon, Mars and further in the solar system. Being designed for similar missions distances using present day technologies the Orion mission can as such be considered as the primary reference payload appended with the other reference payloads.

2.2 Review of inflatable aeroshell technology

This section gives an overview of inflatable aeroshell technology. This serves to provide an overview of the current state-of-the-art of inflatable aeroshell technology and applications, its potential advantages over traditional rigid aeroshells and a collection of literature to use for further investigation.

2.2.1 Advantages of inflatable aeroshells

Inflatable aeroshell systems provide the following advantages with respect to traditional rigid aeroshells: [2, 3]

1. A lower weight is typically achieved, as investigated by Ref.[4];
2. An unconstrained inflatable diameter, by the launch vehicle fairing, allows use of larger aerodynamic decelerators. As a result a lower Ballistic Coefficient (BC) is achievable;
3. A smaller aeroshell volume fraction is required by a lack of need to use a backshell to protect the payload from aft side heating (in contrast to rigid aeroshells);
4. Effective cocooning of the payload by a rigid aeroshell diminishes accessibility. Adding access ports requires the use of additional verification and validation of thermal control design [5].
5. Heat is trapped by a rigid aeroshell cocooning the payload, causing potential interference with on-board payload thermal requirements.

Of these reasons, the first two prevail for the current mission in view of constraints by the launch vehicle, limiting entry vehicle mass and diameter. A way of handling the diameter constraints, one in the launcher fairing shroud and a more relaxed one after decoupling of launcher and entry vehicle, is utilizing deployment mechanisms. These may be either mechanical or inflatable. Comparison of these two concepts yields the following characteristics in favour of inflatable structures [2]:

1. Inflatables have a high reliability of deployment due to a self-correcting system;
2. Use of thin materials obtaining strength from inflation gas pressure reduces the weight required for inflatable systems compared to mechanical structures;
3. Packaging efficiency is higher for inflatable structures than for mechanical erectable structures;
4. Loads are absorbed over a large surface area for inflatable structures, as opposed to typical load concentrations in mechanical systems. Load concentrations require local addition of weight, typically resulting in a heavier structure;
5. Inflatables have a typically lower production cost;
6. Easily adapted to concave shapes with symmetric and curved surfaces;
7. Favorable dynamics due to the nearly constant inflation pressure induced force restoring encountered surface distortions;
8. A favorable thermal response by radiation exchange over a large area.

2.2.2 Investigation of inflatable aeroshell technology

The aforementioned reasons, primarily lower weight and high packaging efficiency, have been key drivers in past and ongoing research in the use of inflatable structures for use in aerodynamic deceleration during (re-)entry. Primary contributor is National Aeronautics and Space Administration (NASA), specifically NASA Langley Research Center, responsible for a series of tests on the feasibility and use of Hypersonic Inflatable Aerodynamic Decelerator (HIAD) concepts for entry and re-entry [3, 6–8]. A brief discussion on these tests follows. Little information on IRVE-I [3] is present, hence it is not included in the following discussion.

Inflatable Re-entry Vehicle Experiment (IRVE) II, launched in 2009, successfully met its objectives, namely: "to demonstrate inflation and re-entry survivability, assess the thermal and drag performance of the re-entry vehicle and to collect flight data for comparison with analysis and design techniques used in vehicle development" [6, p.1]. IRVE-II consisted of a rigid, cylindrical centerbody with a deployable, conical inflatable aeroshell of a so-called stacked toroid configuration [9, 10].

IRVE-III, launched in 2012, proceeded with its primary aim to demonstrate the offset of the vehicle center of gravity on the lift-to-drag ratio of the vehicle and demonstrate survability with flight-relevant heating [11], while the configuration was similar to IRVE-II and subsystems were altered predominantly in the following manners [11]:

- Support straps were added to inter- and intraconnect toroids and centerbody;
- The TPS was upgraded by use of a multi-layer system, consisting of two layers of Nextel fabric, Pyrogel insulation and a Kapton/Kevlar thin film gas barrier, in place of the Nextel fabric used in IRVE-II [7];
- A heater was addded to the pressure tank system used to inflate the stracked toroid structure;
- The addition of a Center of Gravity (CG) offset mechanism to alter the lifting behavior of the vehicle and thereby steer it.

IRVE-III succeeded in its goals, demonstrating the feasibility of a CG offset for vehicle steering [7].

Terrestrial HIAD Orbital Reentry (THOR), planned for launch in 2016, features a more blunt aeroshell with a half-cone angle of 70 instead of 60 degrees, to analyze stability and drag differences with previous [3, 6, 7] configurations [8]. In addition, it features Zylon instead of the Kevlar fibres used for IRVE-III, allowing a thinner TPS layup of a different composition. In terms of TPS, silicon carbide fabric over carbon felt insulation is used instead of Nextel fabric over Pyrogel insulation [8].

In addition to these flight tests, ground tests has been pursued in parallel to further technology developments for HIAD application [9].

Some of the most important characteristics of the re-entry vehicles during these missions are displayed in Table 3.

2.3 Structures in (re-)entry vehicles

Applications and technologies, within the field of structures and materials, as applied in (re-)entry vehicles are investigated, with an emphasis on inflatable aeroshells. These are investigated firstly for inflatable structures and secondly for non-inflatable structures.

2.3.1 Inflatable structures

The inflatable aeroshell system implemented in the IRVE satellites mainly consists of four sub-elements: an inflatable bladder, containing a pressurized medium, a structural restraint,

Table 3: Comparison of recent HIAD missions

Mission parameter	Unit	IRVE-2 [6]	IRVE-3 [7, 8]	THOR (predicted) [8]
Launch date	[-]	17-08-2009	23-07-2012	2016
Mass	[kg]	124.6	280	315
Shell diameter	[m]	2.93	2.93	3.7
Shell angle	[deg]	60	60	70
Apogee	[km]	218	469	200-250
Peak dynamic pressure	[Pa]	1180	Unknown	Unknown
Peak stagnation heating	$[\frac{W}{cm^2}]$	2.2	14.4	65
Peak temperature	[C]	100	378	Unknown
Peak Mach Number	[-]	6.2	Unknown	Unknown
Maximum deceleration	[g]	8.5	20.2	8-10

gas barrier and a thermal protection layer [3]. In addition, this bladder may be further subdivided into isolated volumes to provide avoid Single Points of Failure (SPFs). After flow initiation with pyrotechnic valves the gas flows from the storage tank to the inflatable bladder. This flow is protected by gas valves to prevent backflow from the bladders. [3]

In terms of the inflation process, the pressure and gas used for inflation are variable. The IRVE satellites featured nitrogen gas (and subliming powders), with an operating pressure of 3000 psi for IRVE-4 [12]. An alternative to the use of nitrogen gas is hydrazine, typically capable of delivering lower weight and volume, at the expense of handling, safety and cost [13]. Estimating the required minimum pressure can be done using references [14, 15].

An alternative to using a pressurised gas is the use of the local atmosphere, so-called ram-air inflation. This can be achieved by outfitting the aeroshell with inlets that allows high dynamic pressure freestream air to inflate it. This does, however, limit deployment to within the atmosphere, where the dynamic pressure is adequately high. Extending inflation possibilities can be done by using a hybrid, therefore using both an internal gas source and ram-air inflation[9].

In addition to pure inflation, rigidization may be applied. Rigidization stiffens the structure after inflation, a process that may be performed by multiple techniques. These techniques are described in references [13, 16], for example using fibres impregnated with a resin that cures at a certain temperature.

The structural load is carried by the bladder walls and may in addition be partly carried by support straps (radial direction) and gores (circumferential direction). Calculating the loads for inflatable structures can be done using membrane theory, as per Ref.[17]. Structural analysis and testing as applied to an inflatable aeroshell is described in Ref.[18].

Materials used for HIAD are typically textile materials, such as Kevlar, Vectran and Zylon. For example, IRVE-II and IRVE-III used Kevlar fibres, with a silicone coating [11]. THOR replaced the Kevlar by Zylon [2] fibres for their retainment of (usable) strength at high temperature (400 degrees Celsius versus 250 degrees Celsius) [8]. An overview of materials used

[2] Pro Fiber Zylon Technical Information.

in membrane structures in space is given by Ref.[16]. Figure 1 presents an overview of the material lay-up used for the bladder walls of IRVE-II.

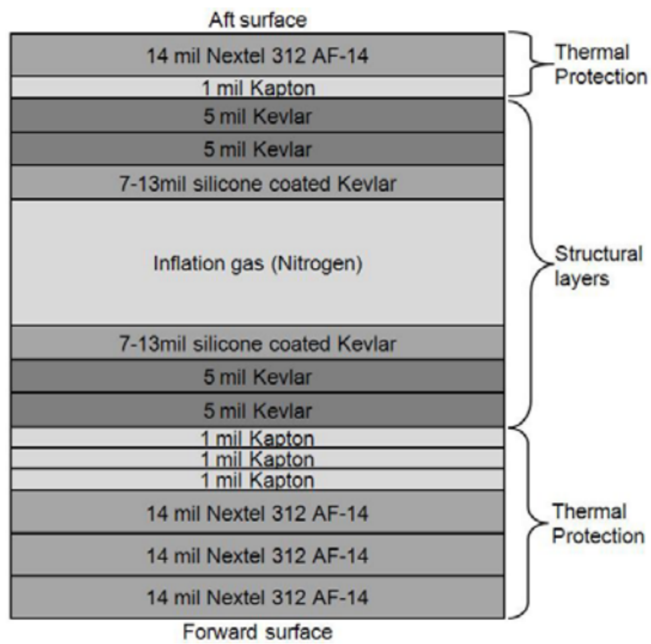

Figure 1: Example bladder wall material lay-up for inflatable shell [6, p.2].

A number of configurations exists for an aerodynamic decelerator [9]. While the discussion on aeroshell shape is limited to Chapter 10, only references for specific (structural) analysis are hereafter given. For a tension cone Ref. [19] provides an entry for structural analysis and design; for an isotensoid Ref. [20]; for a stacked toroid configuration the IRVE satellites provide good reference material. It should be noted that there is a trend towards the use of FEM in structural analysis and design, but that in view of current team member knowledge and experience this is not a feasible alternative.

2.3.2 Non-inflatable structures

Non-inflatable structures can be either deployable or non-deployable. Conventional solutions, such as the Apollo or Soyuz capsule, employ a non-deployable heat shield. An advantage of a deployable heat shield is effecting a larger surface area for aerodynamic deceleration. It must be noted that if for this instance linear actuators are used their reliability is typically lower with linear mechanisms being prone to cocking and jamming [21, p.683]. In most cases non-inflatable structures are supplemented by retropropulsive means. Non-inflatable structures can be evaluated with more conventional methods such as in Ref.[22]. The material for non-inflatable structures is to a large extent determined by the TPS.

URL: `http://www.toyobo-global.com/seihin/kc/pbo/Technical_Information_2005.pdf`. Accessed 24 April 2015

2.4 Aerodynamics

Hypersonic aerodynamics is a complex field of study. Ground testing is difficult, as the high temperatures experienced during hypersonic flight may damage testing facilities [23, 24]. For the solution of the full flow field around a body in hypersonic flow, Computational Fluid Dynamics (CFD) solution is required. This solution requires significant knowledge to implement, is computationally expensive to obtain and difficult to validate [23, 24]. A useful engineering method for determining the aerodynamic characteristics of an arbitrary body in hypersonic flow is the (modified) Newtonian theory. It provides an acceptable approximation of the pressure distribution on the side of the body exposed to the free stream. From this pressure distribution aerodynamic coefficients can be determined. As long as the body drag is dominated by the pressure drag, the modified Newtonian method provides a good approximation of the body drag coefficient [23–25]. Bodies with a high ballistic coefficient experience higher peak heat loads. This has led to the development of blunt (re-)entry vehicles[24, 26]. A method for determining an optimal shape for a (re-)entry vehicle is presented in [26]. A non zero $\frac{L}{D}$ ratio allows downrange and cross range control of a reentry vehicle [26]. Hypersonic flows show an interesting characteristic known as Mach number independence; the aerodynamic coefficients of a body in hypersonic flow ($M>5$) are independent of Mach number [24, 27, 28]. No analytical engineering method exists for supersonic flow around a blunt body. An implementation of a time marching finite difference scheme is proposed in references [27] and [23] to describe the flow field around an arbitrary body in supersonic flow.

The aerothermodynamics of hypersonic flight are complex. In contrast with subsonic and supersonic aerothermodynamics during hypersonic flight also internal chemical reactions and gas composition changes occur because of the high temperatures encountered [23]. This significantly increases the computational cost of CFD methods. In order to estimate the heat flux into the vehicle body therefore the approximate method outlined in [29] [23] is proposed. This method uses the same input parameters as the modified Newtonian method that will be used for the drag estimation and is adaptable to both laminar and turbulent flow around arbitrary bodies in hypersonic and supersonic flows. From this method it follows that the local heat flux into the arbitrary body is dependent on the freestream density & velocity as well as on local body angle with respect to the undisturbed freestream flow, distance measured along the body surface from the stagnation point and local entropy ratio between the wall and total conditions [23, 29].

2.5 Thermodynamics in (re-)entry vehicles

Thermodynamics is used to ensure that components of the reentry vehicle stay within a certain temperature range. These temperature ranges are a function of the intended use of the components and material selection. This literature review is broken down in three major parts. The first briefly explains the principles needed to describe the transfer of heat within structures. The second concerns the heat shield needed for the reentry phase, which is heavily linked to the aerodynamics. This is split up in non-inflatable and inflatable heat shields. The third deals with the thermal control of the capsule itself. For example, the temperature inside the capsule should be suitable for human payloads.

2.5.1 Thermodynamic principles

At the start of the mission the reentry vehicle can be seen as a closed system that has a total energy (E) in the form of internal energy (U), kinetic energy (KE) and potential energy (PE). For successful reentry the kinetic and potential must be reduced. This can be done by transferring the energy across the boundary of the system. There are three methods to transfer the energy. Energy transfer by heat, by work and by mass flow. The latter requires the system to be open such that mass is allowed to leave the system [30]. An example of energy transfer by heat is the heating of gas near the wall of the heat shield and heating of the heat shield itself. Energy transfer by work could for example be the work done by the skin friction drag. The use of thrusters is an example of energy transfer due to mass flow as the propellant mass flows out of the open system.

Energy transfer by heat, or simply heat transfer, has three modes: conduction, convection and radiation. Conduction is the transfer of heat between particles of a material due to interactions between these particles. Convection is the transfer of heat between a solid surface and a moving fluid. Radiation is transfer of heat due to the emission of electromagnetic energy from a surface to its surroundings [30, 31]. Each of these modes can be described by governing equations as described in Ref.[32]. Radiation is given by the Stefan-Boltzmann law and conduction by Fourier's law [30, 32].

2.5.2 Thermal protection system

The thermodynamic principles can be used to design a heat shield. The design of such a heat shield, also called a Thermal Protection System (TPS), depends on several parameters. In general it all comes down to the trajectory the reentry vehicle will follow. The steeper the reentry trajectory, the larges the deceleration and the more g-loads will be endured by the (human) payload. The reentry vehicle will have a nominal trajectory with deviations. An overshoot results in a longer descent and can result in skipping off the planet, whereas an undershoot will result in a high descent rate. The larger the overshoot angle, the longer the reentry vehicle will be in atmosphere an thus creating a higher heat loading on the structure. On the other hand, for a larger undershoot angle, the heat development over time will be faster inducing a larger heat flux on the system. Also, the undershoot angle determines the trajectory with the highest dynamic pressure, which is important for structural sizing.

Not only the trajectory is important for the design, but also the shape with its bluntness. The bigger the heat shield and the larger the bluntness, the more the thermal loads can be distributed over the capsule. This will result in a thinner TPS such that less mass is needed. This can cause aerodynamic instability, however [33].

The temperature development in the structure through multiple layers of material can be determined when the heat loading and heat flux is known. An example of a method to determine the heat development is given by [34].

Heat shield for non-inflatable structures Throughout the last century several reentry attempts have been performed with different TPSs. This paragraph and the next paragraph focus on several concepts. This paragraph focuses on non-inflatable solutions and the next on inflatables.

A typical TPS example of a non-inflatable structure is the TPS of the reentry vehicle of the Apollo mission [35]. Different from inflatables, the TPS of the Apollo also has an aft shield and a crew compartment heat shield, because the front shield cannot deflect the direct flow from the sides and the aft part of the structure. An overview of the TPS lay-up is shown in figure 2. It consist of an ablative layer, which is sacrificed during reentry. The second layer is brazed stainless steel. This layer is followed by an insulator to keep the temperature difference between the outer layers and the inner layers. Last but not least, the connection to the rest of the structure is an aluminum honeycomb structure.

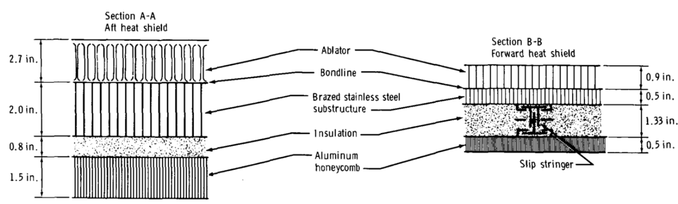

Figure 2: Example of the TPS lay-up for the Apollo reentry vehicle [35, p.5].

Heat shield for inflatable structures The usage of an inflatable structure is preferable over a non-inflatable structure for several reasons. An inflatable heat shell is not directly limited to the size of the launcher. Therefore a much larger diameter of the TPS can be obtained. This is an advantage because the heat loads on the structure can be distributed resulting in a lower front shell mass. Furthermore, the heat at the back side of the vehicle will be severely lower, such that there is no need for a back shell [3]. Therefore, less mass is needed and less effort needs to be done to verify and validate the thermal loads on payload closely located to the back shell. Although the system is very promising in terms of mass, the system does add a lot of complexity.

The TPS of the IRVE vehicles consists of several layers [12] which are shown in Figure 3 similar to the before mentioned Figure 1. The outer layer protects the structure from the direct incoming flow by distributing and absorbing or dissipating the heat. Also, it will carry the shearing loads from the flow. The second layer is an insulator and keeps the temperature behind the shield at a low level. Finally, the last Kevlar layer provides a fabric connection between the heat shield and the rest of the structure and carries most of the structural loads. The Kapton in this layer is an impermeable layer and keeps hot gases from passing this layer.

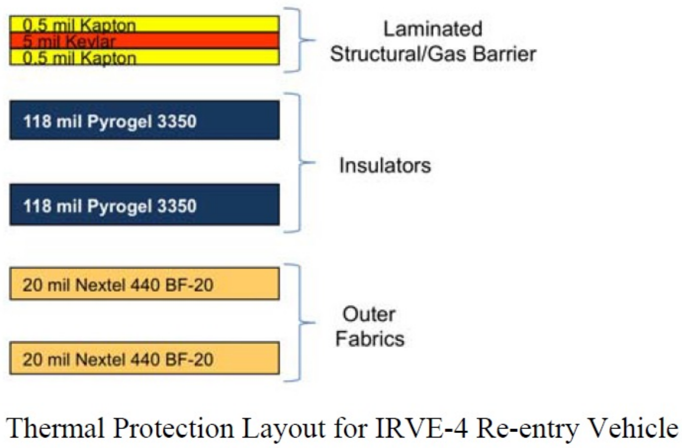

Figure 3: Example of the TPS lay-up for the IRVE-4 reentry vehicle[12, p.6].

There has not yet been an actual entry on Mars with an inflatable shield, but NASA has demonstrated a well verified and validated concept [6, 7]. Another non-verified concept is the use of ballutes, which can be applied in case of high mass transportation [36]. Due to its large size the thermal loads on the structure are small. However, the capsule, located in front of the ballute is exposed to a the free flow and needs additional thermal protection.

2.5.3 Thermal control system

For the Thermal Control System (TCS) of the crew capsule the book on Thermal Control by R. Karam is leading[31]. Karam explains that heat fluxes due to conduction and radiation can be related to temperature using certain proportionality factors that depend on physical constants, material properties, surface conditions, geometry and temperature. The purpose of thermal control is to change the proportionality factors such that the desired temperature range is reached and retained. The changing of the proportionality factors is done by proper selection of materials and configurations.

The actual TCS is divided into two parts, namely passive and active thermal control. The distinction is made whether the control component uses power or not. For example the use of insulation is a form of passive thermal control. Radiators and heaters are examples of active thermal control. Also a hot and cold case are considered to define the upper and lower limits for the temperatures. The hot case has the maximum possible influx of heat during the mission and the cold case the minimum possible influx of heat during the mission. Naturally these limits should lie within the desired temperature range. If this is not the case, the TCS should be improved [31].

Karam also describes how to perform thermal analysis using thermal models, which can be used in combination with Holman's book on heat transfer [32]. The predictions made in the thermal analysis rely on the law of conservation of energy. Both books ([31, 32]) describe how to do the analysis with computational methods, where Holman elaborates more on the use of a grid to find the thermal distributions at certain times. In this way the TCS can be verified by analysis.

2.6 Control

The control of a vehicle encompasses the dynamics model of the spacecraft, implementing a controller in that model and designing the sensors and actuators on the spacecraft that perform the actual control. This subsection focuses first on dynamics and controller design, then on the implementation in hardware.

2.6.1 Orbit

The orbit to allow an aerocapture requires a simulation of the spacecraft through both space and atmosphere. A comprehensive book on this is Computational Orbital Mechanics, [37], while a thesis on an implementation in MATLAB is found in [38]. In determining the orbit of the satellite, atmospheric properties are very important. A program developed by the NASA named "Mars-Global Reference Atmospheric Model 2010 (Mars-GRAM 2010 v1.0)" is used to determine the atmospheric properties at all longitudes, latitudes and heights [39]. Lift may be used to control the trajectory [40].

2.6.2 Flight Dynamics

The book [41] gives an introduction to flight dynamics in general, giving an introduction to stability and controllability of aerospace vehicles. However, it is focused on aircraft. A more in-depth study of stability and control of spacecraft can be found in [42, 43]. One of the problems of inflatable structures is the fact that they deform more under loads than rigid structures. The effect of this on the dynamics of the vehicle is given in [44] as well as in [10], where particularly this effect on IRVE-II is detailed. For a general introduction into control systems engineering, the book [45] is available.

2.6.3 Sensing

An overview of state measurement components can be found in [21]. If no gyrometer can be feasibly implemented, the angular rate can be estimated based on vector measurements [46].

2.6.4 Actuation

Finally, the actuators will allow the spacecraft to actually follow the prescribed trajectory and orient it such that the appropriate side of the spacecraft is pointed towards the flow. Several typical ways of actuating the orientation vehicle are given in [21]. An An executive overview of the thruster system of the Orion capsule can be found in [47]. For hypersonic flows with ionized boundary layers, MagnetoHydroDynamics (MHD) can be used to control the flow and produce a moment. An overview of current state of MHD technology can be found in [48], prediction of forces can be found in [49]. An overview of retropropulsion for Mars entry is given in [50], which shows a beneficial interaction between retropropulsion and hypersonic deceleration in the atmosphere. Center of gravity offset can be used to control orientation as well: [51] uses it to provide roll control on lift-generating vehicles, while IRVE-III used it to introduce an angle of attack to generate lift [11]. At lower Mach numbers (partial) control can also be performed by bodyflaps as was for example done in the NASA SpaceShuttle missions.

3 Trajectory analysis

4 Functional analysis

The functional analysis of the design is described by a pair of Systems Engineering (SE) tools, namely a FBS and a FFD. The former is an AND-tree that gives an overview of functions to be fulfilled by the system in a categorized manner; the latter expands upon this by sequencing the events. The FFD is discussed in section 4.1; the FBS in section 4.2.

4.1 Functional Flow Diagram

The FFD is given in Figure 5. As entry commences, the sequence and procedure for functions should be initialized. This is done in step 1.0-1.4, which subsequently contain setting the on-board computer mode to its pre-programmed instructions for entry and conveying these instructions to the power, attitude control and thermal control subsystems. Hereafter the aeroshell is deployed - provided that the vehicle concept features a deployable aeroshell - either by mechanical means or by inflation in step 2.0-2.3. Monitoring of deployment takes place in 2.3, for example by use of measuring the pressure in the inflatable volume at selected locations for a tank-gas inflation system. In case activation is monitored to be incomplete, this is fed back and activation re-performed. The aerobraking manoeuvre then proceeds by controlling the vehicle along its trajectory, captured in function 3.1, while maintaining vehicle integrity by carrying structural and thermal loads without failure, captured in functions 3.2 and 3.3 respectively. Finally, the entry procedure is terminated at some point (described by a terminal velocity and altitude) after which Terminal Area Energy Management (TAEM) is activated. Hereupon other mechanisms of deceleration take over, left untreated since these are outside the scope of the mission.

4.2 Functional Breakdown Structure

The activities described in the FFD of the previous section are categorized in the FBS. The FBS is given in Figure 4.

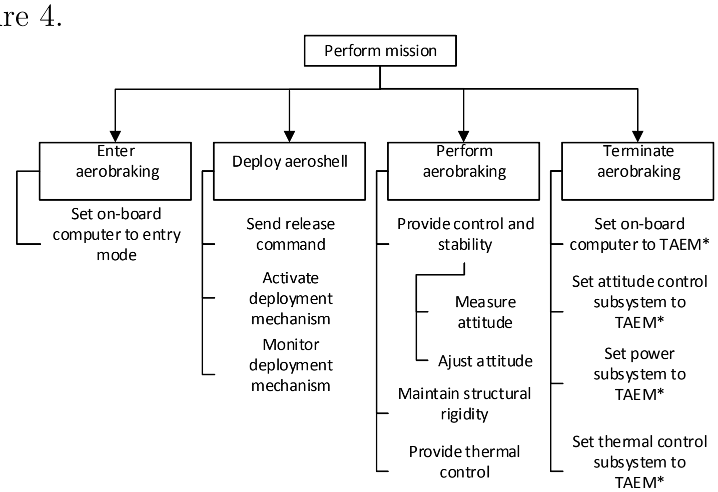

Figure 4: The FBS of the aeroshell mission

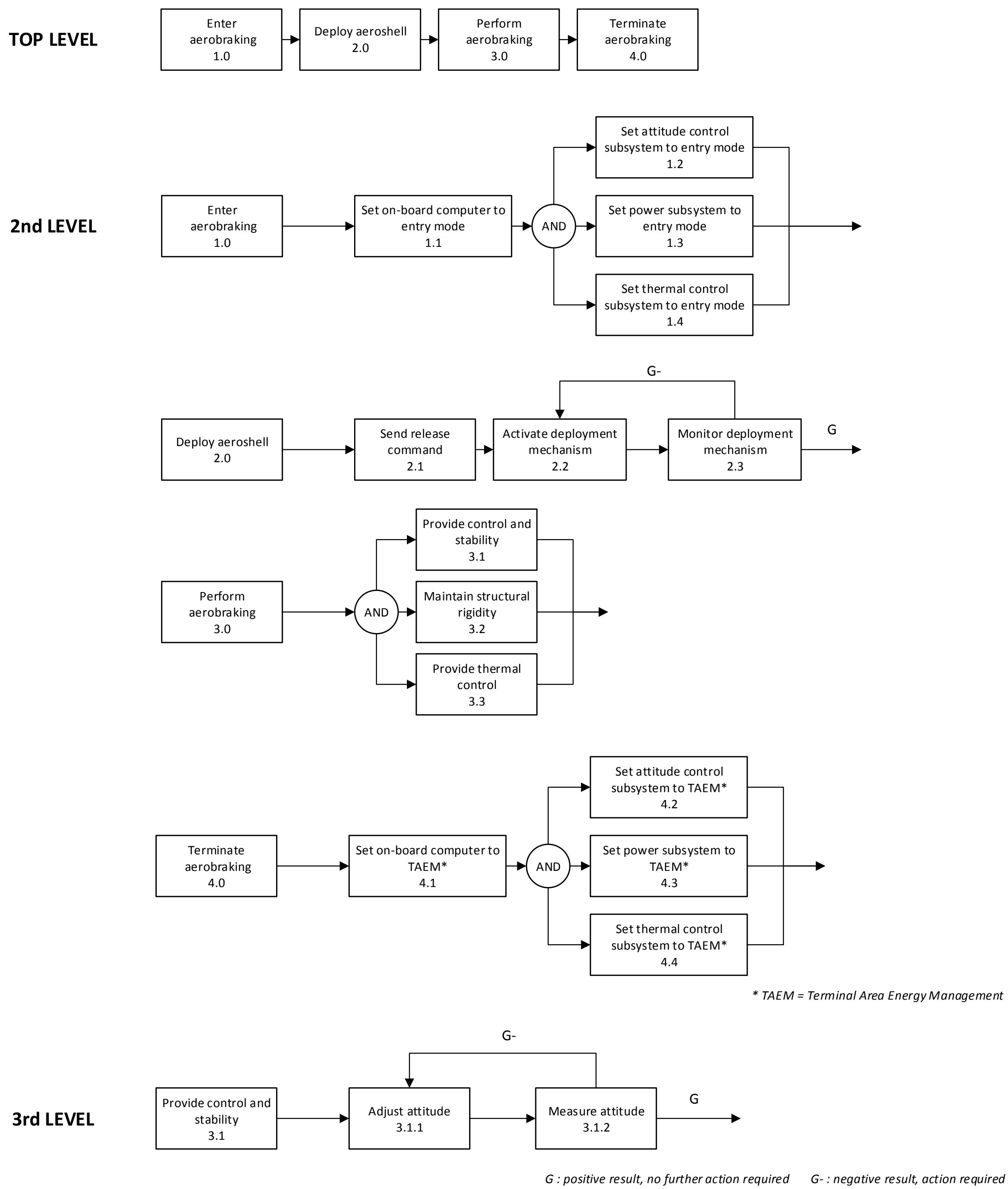

Figure 5: The FFD of the aeroshell mission

5 Requirement analysis

This section will analyze the requirements of the re-entry mission. It will start by providing a requirements discovery tree, which visualizes the way the different requirements on the re-entry vehicle flow down from the top level mission requirements and constraints. This will be followed by an analysis of the subsystem requirements. This analysis will follow from the flown down presented in the RDT. Finally, the requirements will be as precisely defined and documented as possible at this stage. For the unknown values the acronym To be Determined (TBD) will be used. The output of this final step will be the requirements used during the product design.

5.1 Requirements Discovery Tree

The overarching mission requires the system to perform a manned re-entry on Mars. Several performance requirements and mission constraints are imposed on the system. These overall requirements and constraints were detailed in the project plan [52]. From these overall requirements and constraints, subsystem requirements can be derived. Figure 6 graphically displays this requirements discovery, and provides a sample parameter which has a requirement imposed on it due to the top level requirements. Each of the subsystems' requirements will be elaborated and expanded upon in the remainder op this section.

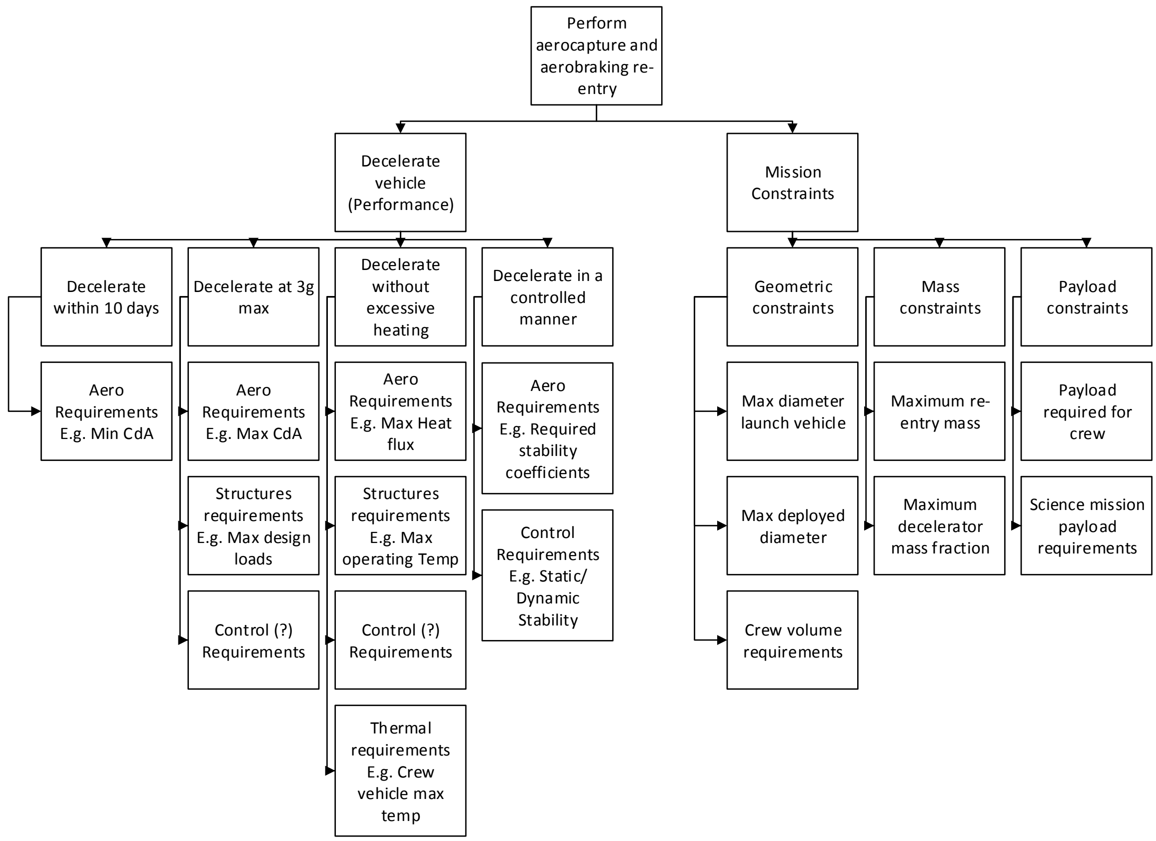

Figure 6: Requirements Discovery Tree

5.2 Top-level requirements

To provide a full overview of the requirements the top level requirements as stated in the Project Plan (PP) are restated in table 4. These top level requirements are appended with an additional requirement CIA-A09 concerning the support of human payload. A full overview of all the requirements mentioned of this chapter, including the top-level requirements, is given in appendix A.

Table 4: Overview of mission top-level requirements

ID	Description
CIA-A01	The re-entry vehicle shall be able to cope with an entry velocity of seven kilometers per second.
CIA-A02	The inflated aeroshell shall have a maximum diameter of 12 meters.
CIA-A03	The system shall have a diameter not exceeding 5 meters in stowed condition
CIA-A04	The maximum entry mass of the re-entry vehicle shall be 10,000 kilograms at the start of the mission.
CIA-A05	The hypersonic deceleration system mass shall not be heavier than ten percent of the total re-entry vehicle mass.
CIA-A06	The control system shall have a maximum failure probability of 5.0e-4.
CIA-A07	The maximum allowable loads on the re-entry vehicle shall be 3 Earth g's in each axis.
CIA-A08	The re-entry vehicle shall have a maximal aerobraking duration of ten Earth days.
CIA-A09	The re-entry vehicle shall accommodate TBD humans as payload.
CIA-A10	The re-entry vehicle shall decelerate to a velocity of TBD at a height of TBD above the surface of Mars.

5.3 Aerodynamic Requirements Discovery

A number of aerodynamic requirements can be seen in figure 6. These will all be discussed here and have been summarized in table 5.

Table 5: Overview of Aerodynamic requirements

ID	Description
CIA-B01-Aero-01	The system shall produce a maximum heat flux of no more than TBD $[\frac{W}{cm^2}]$
CIA-B01-Aero-02	The system shall be stable
CIA-C01-Aero-02-01	The system shall be statically stable
CIA-C01-Aero-02-02	The system shall be dynamically stable
CIA-B07-Aero-01	The system shall have a $C_D A$ of TBD m^2

Requirement CIA-B01-Aero-01 follows from the entry velocity of 7 $[\frac{km}{s}]$. Since the heat flux subjected upon a body is proportional to V^3 [29] the highest values for the heat flux will be found during the first orbit around Mars. Because the system needs to be controllable requirement CIA-B01-Aero-02 will need to be fulfilled. An uncontrollable system may encounter loads that are higher than acceptable and that may endanger the system integrity and the lives of the crew. This requirement flows down into two parts: static and dynamic stability. These are represented in the subrequirements CIA-C01-Aero-02-01 and CIA-C01-Aero-02-02.

Requirement CIA-B07-Aero-01 is caused by the need for a limitation of the maximum allowable decelerations that occur. This can have a significant impact on the mission and mission duration, since this limits the maximum allowable drag force that can be achieved. Lastly the source of requirement CIA-B08-Aero-01 is the limitation of the deceleration duration. If the deceleration takes too long the well-being of the astronauts is affected negatively.

5.4 Structural Requirements Discovery

The vehicle structure faces a number of requirements, functional and operational. The functional requirements are stated in Table 6, the operational requirements in Table 7. These requirements are briefly discussed hereafter.

Table 6: Overview of functional requirements on structures subsystem

ID	Description
CIA-B01-Struc-01	The structure shall operate within a temperature range of [TBD,TBD] degrees Celsius
CIA-B02-Struc-02	The structure shall support deployment
CIA-B07-Struc-03	The structure shall sustain the maximum mechanical loads without failure
CIA-B09-Struc-04	The structure shall connect payload and deceleration mechanism

Requirement CIA-B01-Struc-01 follows from the aerodynamic heating as a consequence of the dissipation of kinetic energy corresponding to a velocity of 7 [km/s], as stated in requirement CIA-A01. The TPS reduces the temperature to within acceptable limits, which are translated to a range of temperature in which the structures subsystem should operate. These temperatures therefore follow from the TPS. This requirement is essential because temperature can have a substantial effect on the mechanical properties of materials as well as thermal expansion [53]. These mechanical properties are essential to meet requirement CIA-B07-Struc-03, namely to handle the structural loads induced during aerocapture and (re-)entry. These have been limited to 3g in each axis in requirement CIA-A03. In addition, if a deployment functionality (for example an inflation system) is a concept feature, deployment should be performed by the subsystem; in case such a functionality is not present, there is no deployment, hence nothing to support and the requirement is logically satisfied. This is stated in requirement CIA-B02-Struc-02. Lastly, payload and deceleration mechanism should be connected to prevent separation during re-entry and satisfy the payload constraint. This is stated in requirement CIA-B09-Struc-04.

Table 7: Overview of operational requirements on structures subsystem

ID	Description
CIA-B02-Struc-05	The structure shall have a maximum diameter not exceeding 12 [m] in deployed configuration
CIA-B03-Struc-06	The structure shall have a maximum diameter not exceeding 5 [m] in stowed configuration
CIA-B04-Struc-07	The structure shall have a mass not exceeding 350 [kg]

The operational requirements CIA-B02-Struc-05 and CIA-B03-Struc-06 follow from the geometric constraints (by launcher considerations). Requirement CIA-B04 states that the structural subsystem should respect the mass budget.

5.5 Thermal Protection Requirements Discovery

The requirements for the thermal subsystem flow down from the RDT and are listed in Table 9. The requirements are split up in two parts, the TPS and the TCS. The TPS mainly distributes the heat load and flux generated by decelerating the re-entry vehicle, whereas the TCS controls the temperature of the payload and other subsystems.

Table 8: Overview of thermal requirements

ID	Description
CIA-B01-TPS-01	The TPS shall be able to withstand the maximum heat flux of TBD $\left[\frac{W}{cm^2}\right]$.
CIA-B01-TPS-02	The TPS shall be able to withstand the maximum heat load of TBD $\left[\frac{J}{cm^2}\right]$.
CIA-B01-TCS-01	The TCS shall keep the subsystems within their operative temperature range.
CIA-B01-TCS-01-crewmodule	The TCS shall keep the crew module within a temperature range of TBD and TBD $\left[\frac{J}{cm^2}\right]$.

Requirement CIA-B01-TPS-01 follows from the trajectory the re-entry vehicle is following and is bounded by the shortest trajectory, the maximum undershoot trajectory, the vehicle can follow. During this trajectory the vehicle will see the fastest heat development. Not only the fastest heat development is important, but also the duration of deceleration. For the longest trajectory, the maximum overshoot trajectory, energy will be dissipated into the shell. The TPS should be able to cope with this heat development over time. This need is covered by requirement CIA-B01-TPS-02. The payload and subsystems are only able to withstand a certain temperature range. To keep the payload and subsystems within this range the TCS follows requirement CIA-B01-TCS-01.

5.6 Control Requirements Discovery

-Don't crash into Mars -Don't

Table 9: Overview of Control requirements

ID	Description
CIA-B06-TPS-01	The TPS shall be able to withstand the maximum heat flux of TBD $\left[\frac{W}{cm^2}\right]$.

5.7 Control Requirements Discovery

6 Budget breakdown

This section will describe the mass breakdown of the re-entry vehicle. It will be split in two parts; the first part will give a mass breakdown of the crew compartment, the second part will provide an initial mass breakdown of the hypersonic decelerator. These breakdowns are extrapolated from existing re-entry vehicles and design studies.

6.1 Crew Module

From requirements CIA-A04 and CIA-A05, it can be determined that the crew module mass is 9000kg at re-entry. This is roughly 1.6 times the mass of the Apollo Command Module [3], and is roughly equal to the Orion Multipurpose Crew Vehicle [4]. The Apollo missions housed three crew members, while the Orion will house 4 to 5 crew members. Realizing that the number of crew members carried appears to scale with the vehicle mass, and taking into account the mass of the heat shield of the Apollo and the Orion vehicles, the 9000 kg of re-entry mass is likely to be able to support a crew of six. It is assumed that each crew member weighs 85 kg, and that a mission payload of an additional 1000kg is carried within the crew module.

The remaining subsystem masses of the crew module are estimated based on either the Orion or the Apollo vehicles. Subsystem masses are scaled linearly based on the number of crew members they were designed to accommodate for. The complete mass breakdown for the crew module can be found in Table 10.

Table 10: Crew Module Mass Breakdown

Subsystem	Mass[kg]	Fraction [%]
Vehicle Structure	3000	33.3
Subsonic Re-entry System	1000	11.1
Martian Mission Payload	1000	11.1
Furnishing	600	6.7
Crew	540	6.0
Navigation Equipment	500	5.6
Electronic Equipment	500	5.6
Environmental Control	450	5.0
Batteries	450	5.0
Communication systems	300	3.3
Telemetry	200	2.2
Mass contingency	460	5.1
Total	9000	100

[3]URL: http://braeunig.us/space/specs/apollo.htm, Accessed 30 April 2015

[4]URL: http://www.spaceflight101.com/orion-spacecraft-overview.html, Accessed 30 April 2015

6.2 Hypersonic Decelerator

The mass breakdown of the hypersonic decelerator is based completely on literature, until more detailed information becomes available later in the design. The primary reference for the initial mass breakdown is a NASA design study of Martian Aerocapture missions [4]. The allocated mass fractions for the subsystems were kept similar. A ΔV budget of $150ms^{-1}$ for reaction control was suggested in the design study. This corresponds to a propellant mass of roughly 50kg per the Tsiolkovsky equation. The complete initial mass budget can be found in Table 11.

Table 11: Hypersonic Decelerator Mass Budget

Subsystem	Mass[kg]	Fraction [%]
Thermal Protection system	500	50
Spacecraft structure	350	35
Aeroshell	200	20
Connection	150	15
Reaction Control System	150	15
System Dry Mass	100	10
Propellant Mass	50	5

7 Market Analysis

This chapter gives the results of a market analysis for a controllable inflatable aeroshell. To define the market for the product three dimensions are used: function, technology and customer. The purpose of the market analysis is a minimization of risk of selecting the wrong combination of function and technology for a selected set of customers. A Strengths, Weaknesses, Opportunities and Threats (SWOT) analysis provides a brief overview of the product characteristics. As such, the chapter commences with subsequent sections on function, technology and customer. It follows upon this with a section on the outcome of the SWOT analysis.

7.1 Customer

Prospective customers are scientific or governmental agencies on one hand and private ventures on the other hand. The former includes NASA as a leading contributor to and investigator of the use of inflatable aeroshells for (re-)entry, as explained in Chapter 2. NASA operates by order of the United States (U.S.) government, currently in pursuit of human exploration of Mars in the 2030s [5]. This has been formulated as goals in the NASA Authorization Act of 2014[6] and the U.S. National Space Policy, issued in 2010[7].

The interest expressed by the U.S. in human exploration of Mars is shared by a number of private ventures, most notably Mars One, the Inspiration Mars Foundation and SpaceX. The former two are non-profit organisations, while SpaceX is a commercial venture. Mars One has expressed its goal as the permanent human settlement on Mars with planned departure of the first non-human payload in 2020 and the first human payload in 2026[8]. The Inspiration Mars Foundation, in cooperation with NASA, seeks to transport two humans, a male and female, to Mars for planned launch in 2021[9]. SpaceX is a privately funded venture currently working in close cooperation with NASA to provide launchers for manned missions to Mars[10].

These planned missions illustrate the commercial interest in human spaceflight to Mars. Commercial interest in controllable inflatable aeroshells is directly coupled to this by the fact that these provide a cost-effective means of entry and re-entry (see Chapter 2) primarily by reduced launch costs. Along with this commercial interest, ongoing investigations by NASA provide an indication of scientific interest in this field of study. In the end, all interest is fueled by human curiosity and desire to explore and habitate extraterrestrial environments. These environments are expected to expand beyond Mars and therefore interest in (re-)entry vehicles is expected to remain.

[5]URL: `https://www.nasa.gov/content/nasas-journey-to-mars`. Accessed 28 April 2015

[6]URL: `http://science.house.gov/sites/republicans.science.house.gov/files/documents/HR%204412.pdf`. Accessed 28 April 2015

[7]URL: `https://www.whitehouse.gov/sites/default/files/national_space_policy_6-28-10.pdf`. Accessed 28 April 2015

[8]URL: `http://www.mars-one.com/`. Accessed 28 April 2015

[9]URL: `http://spacenews.com/39714inspiration-mars-sets-sights-on-venusmars-flyby-in-2021/`. Accessed 28 April 2015

[10]URL: `http://www.spacex.com/falcon9`. Accessed 28 April 2015

Direct customers are thus governmental agencies on one hand, primarily NASA, and commercial providers on the other hand.

7.2 Function

Primary prospects for the use of a controllable inflatable aeroshell are the following:

- Perform entry for manned spaceflight on Mars;
- Serve as a basis for design extrapolation to perform manned (re-)entry at other sites, for example Earth;
- Serve as a basis for design extrapolation to perform (re-)entry of unmanned spaceflight;
- Further the technology development and application of inflatable technologies in spaceflight.

A direct function or use is the first item: the controllable inflatable aeroshell provides aerodynamic deceleration (for (safe) transportation of) human payload in a cost-effective manner. The latter is effected primarily by requiring a small launcher volume and a lower weight than conventional and undeployable solutions [3, 4]. Therefore its main function can be described as performing manned entry. While the aeroshell is designed for entry on Mars, the design can be extrapolated to perform entry or re-entry on a number of sites, for example Earth. The second and third items are therefore secondary functions fulfilled, distinguished from the primary function by their indirect relation to the product. In addition, the fourth item is a secondary function.

7.3 Technology

The functionalities provided by the aeroshell, thus the deceleration provided during (re-)entry, are effected by an inflatable aeroshell as primary technology used. This may be further subdivided into an aerodynamic shape, a control system, a TPS and a supporting structure (including deployment mechanisms).

7.4 SWOT Analysis

Identification of the primary characteristics, in terms of a SWOT analysis[11], of the proposed controllable inflatable aeroshell yields Table 12.

Table 12: SWOT Analysis

Strengths (S)	**Weaknesses (W)**
+ Lightweight solution + Compact solution	- Development risk
Opportunities (O)	**Threats (T)**
+ Growing demand + Breakthrough technology/materials	- Catastrophic failure manned mission - Competing concepts (e.g. Orion)

[11] URL: `http://www.usfca.edu/fac_staff/weihrichh/docs/tows.pdf`. Accessed 28 April 2015

Strengths and weaknesses are internal to the design, while opportunities and threats are external factors. While the design retains a development risk, being a relatively new concept, this weakness can be mitigated by proper verification activities. Such activities do, however, incur additional time and costs to the design process. As such, risk remains inherent to the design.

8 Risk assessment

This section will cover the initial risk assessment that was carried out during the conceptual design phase. First a risk map was constructed, followed by an explanation of the contingency margins that will be used during the various design phases. These are based on the outcomes of the risk assessment. Section 8.1 will cover the risk map, after which section 8.2 will discuss the technical contingency allocation.

8.1 Risk map

A risk map has been constructed in order to identify which mission and design elements pose the biggest risk. From the risk map it can be seen which elements require the most attention in order to mitigate the risks they pose. The risk map is shown in table 14. The colors correspond to the amount of risk each table cell represents. The numbers in table 14 correspond to the elements shown in table 13. It is difficult however to assign technology states to some of the systems, since as of yet no definitive concepts have been selected to be used. Because of this the risk map will be updated and reviewed regularly, in order to effectively mitigate the risks present in the different design options and concepts.

Table 13: Risk map elements

Number	Element
1	Flight control system
2	Deployment system
3	Impact of launch vibrations
4	Operational mission duration
5	Structural integrity under mission loads
6	Heat resistance
7	Long space exposure
8	Communication systems

Table 14: Risk map

Feasible in theory				1,4,6,7
Working laboratory model				
Demonstrated in-flight on Earth				3,5
Derived from used technology on Mars				2
Demonstrated in-flight on Mars			8	
	Negligible	**Marginal**	**Critical**	**Catastrophic**

As can be seen from the two preceding tables there are a lot of mission elements that carry a lot of risk with them; these can be seen in the upper right corner of table 14. In order to mitigate the risks inherent with these elements more time will be allocated in analyzing those specific elements.

8.2 Technical contingency management

From the risk map of the preceding section one can see that there are many risks involved with the development of a hypersonic inflatable aeroshell. Several Technical Performance Measurement (TPM)'s will be used to evaluate the performance of the system at different stages of the design process. These TPM's follow from the top-level requirements discussed further in chapter 5. The TPM's that will be used are:

- Hypersonic deceleration system mass fraction
- Aerobraking duration
- Control system reliability

Table 15 shows the Technical Performance Measurement (TPM) factors that will be used during the various stages of the design process. The lower limits for the RCA's have been adopted from a Goddard Space Flight Center technical standard [54]. These RCA's will be used to account for the uncertainty of the analysis methods that will be used to evaluate all the concepts.

Table 15: RCA factors that will be used

Technical Performance Measurement	**Conceptual design contingency [%]**	**Preliminary design contingency [%]**
Hypersonic deceleration system mass fraction	30	20
Aerobraking duration	30	20
Control system reliability	30	20

9 Approach with respect to sustainable development

Sustainable development in engineering means that the design, production, operation and disposal of a product should be done in a sustainable way. In this case sustainable means that energy and resources are used in a manner that does not threaten the environment or the needs of future generations [55]. In this chapter the general approach with respect to the sustainable development of the controllable inflatable aeroshell is briefly discussed. Since no large advances have been made with respect to the Project Plan (PP) this approach is for now the same as in the in the original PP [52].

Even though sustainability is becoming more important in engineering, it does not have a high priority in these kind of space missions. The reason for this is that the proposed mission is a single mission and therefore its total impact will be relatively small. For example, it is acceptable that the production of the space vehicle is less sustainable than the production of one small passenger aircraft, since the aircraft is produced in large numbers whereas only one space vehicle is produced. It can therefore be said that sustainability will not be the design driver for the controllable inflatable aeroshell. Of course, sustainable methods are preferred when they do not add much costs and very unsustainable methods are to be avoided. It must be noted that even though a space mission itself may never be fully sustainable, advances towards more sustainable development can still be made. This design aims at a mass reduction with respect to more conventional designs. As such, if a mission is performed, this aeroshell design will help to increase the sustainability of the whole mission. For example, a lighter than conventional design will lower the launch emissions and is as such helping towards a more sustainable world.

Some examples of sustainable methods within the aeroshell design can however also be mentioned. In the process of producing the aeroshell unnecessary polluting methods that threaten the natural environment should be avoided. Also interplanetary forward contamination should be prevented. In this case it means that life and other forms of contamination should not be transferred from Earth to Mars. In practice this is already standard procedure guided by COSPAR as a result of article IX in the Outer Space Treaty of 1967. [56] Another, less important, form of sustainability is the avoidance of space debris in the atmospheres of Earth and Mars.

Therefore sustainability will not be a leading driver for the design of the controllable inflatable aeroshell, as long as very unsustainable methods are avoided. The latter is effected by a preference of sustainable methods in choices for production and operation of the aeroshell.

10 Design Option Structuring

This chapter gives an overview of initial design concept selection, focusing on the different missions aspects of configuration, mission profile and control mechanisms. This concept generation is done systematically by means of a Design Option Tree (DOT) for each of the aspects. Combination of the different trees then yields a number of final concepts. The next phase is eliminating clearly unfeasible or undevelopable concepts. The first section gives a DOT for each of the mission aspects in subsequent subsections, including eliminating unfeasible options. The second section proposes the generation of concepts used in the next phase of concept selection and trade-off, to take place after the BR.

10.1 Design Option Trees

This section presents the DOTs for the three aspects of vehicle configuration, mission profile and control mechanisms. These are combined in the next step of concept generation to yield a number of concepts for further selection and analysis, as described in the next section.

10.1.1 Configuration of aerodynamic deceleration

The configuration of the aerodynamic deceleration mechanism, given in figure 8, can be at a first level be subdivided into inflatable and non-inflatable systems. For non-inflatable systems an *AND* subdivision can be made. A blunt or pointed body can be considered and at the same time this structure may be deployable or undependable. Pointed bodies will not be considered since the attached shock will cause excessive aerodynamic heating [27] making this concept unfeasible. The blunt body may further be subdivided into lifting and non-lifting bodies. Lifting bodies are deemed high lift-to-drag ratio bodies. Geometrically symmetric bodies, with a corresponding lower lift-to-drag ratio, may also feature a lifting component by featuring a CG located away from the symmetry axis. This CG offset is sometimes also actively used for control purposes[7] as is also explained in section 10.1.3.

The second separation is the subdivision between deployable and non-deployable structures. Where as for inflatable structures deployment is inherent to the design non-inflatable structure may feature no deployment mechanism. The deployable configurations feature a change in geometry specifically for the reentry which can for example be controlled by actuators. The deployable systems are already cut off due to the more complicated (thus less reliable) deployment combined with the inherent higher weight of non-inflatable structures [4] compared to deployable inflatables.

The inflatable structures on the right hand side of Figure 8 can be subdivided in fore or aft placed inflatables or a combination thereof. This can be further broken down to an Isotensoid, tension cone and stacked configuration. Variations here off are also examined by NASA[9]. A Isotensoid configuration consists of a single inflatable covering the whole frontal surface area. In a tension cone structural rigidity is provided by only a single inflatable outer ring, which is filled in between with fabric. Finally a stacked configuration features simply multiple inflatables which are stacked to form the desired configuration.

The inflatable structures on the right hand side of Figure 8 can be subdivided in fore or aft placed inflatables or a combination thereof. Attached inflatables can be further broken down to an isotensoid, tension cone and stacked configuration. An isotensoid configuration consists of a single inflatable covering the whole frontal surface area. In a tension cone structural rigidity is provided by only a single inflatable outer ring, which is filled in between with fabric. Finally a stacked configuration features multiple inflatables which are stacked to form the desired configuration. [3, 19, 20] Trailing inflatables are derivatives of the Disk-Gap-Band parachute concept and, as their name suggests, trail the payload capsule [9]. The feasibility of a such a ballute has been investigated and a ballute is deemed feasible and effective in terms of mass for aerocapture, albeit not specifically aimed at Mars entry [57, 58].

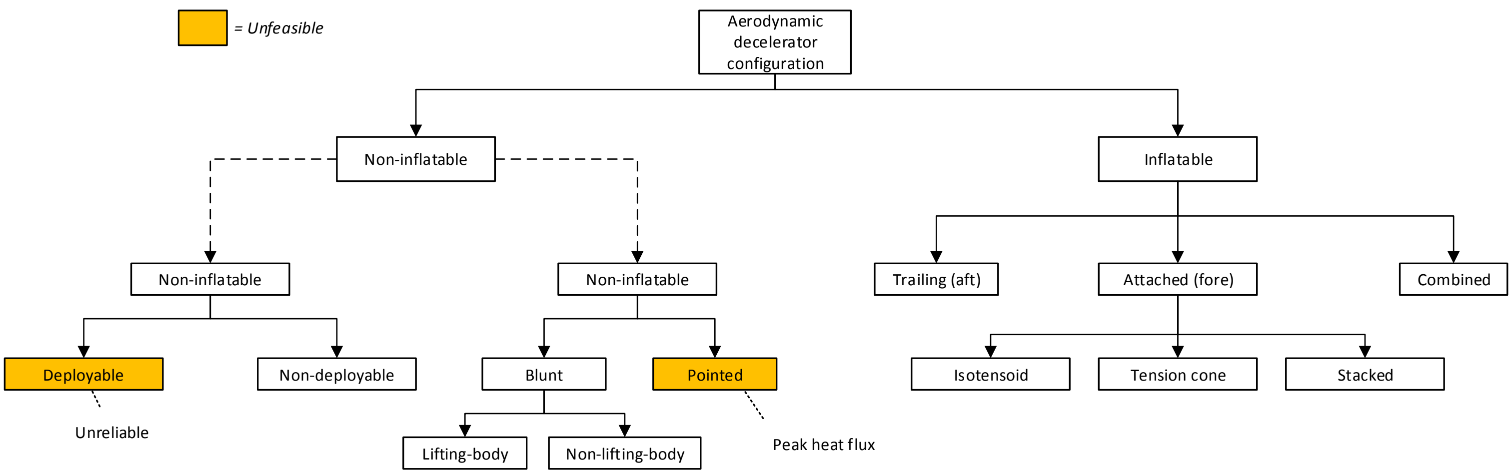

Figure 7: Design Option Tree for entry vehicle configuration

10.1.2 Mission profile

In terms of mission profile, freedom exists in the rate at which kinetic energy is dissipated. On one hand a short mission duration can be used, in the order of minutes or hours, featuring a steep descent; on the other hand a longer mission duration can be used, in the order of days or weeks (or longer), featuring a shallow descent. The rate at which kinetic energy, conforming to a velocity of 7 km/s, is dissipated places a lower bound on the mission duration. Entry in the order of minutes becomes prohibitive for the thermal and mechanical loads, which will most likely be in excess of the 3g limit imposed (requirement CIA-A07). An upper bound is placed by the requirement (CIA-A08) that aerobraking duration shall be less than ten (Earth) days. In a first estimation one can therefore distinguish mission profiles in the order of hours on one hand and days on the other hand.

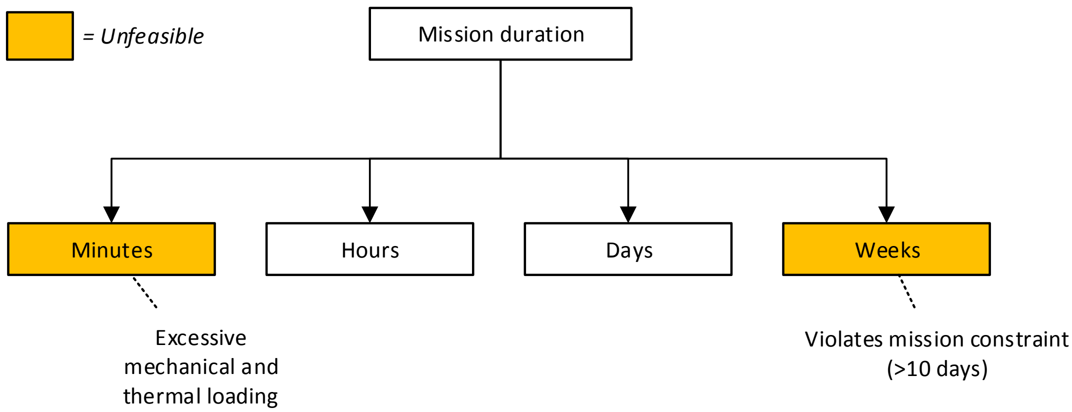

Figure 8: Design Option Tree for entry vehicle configuration

10.1.3 Control

10.2 Concept generation

This section presents a proposed strategy for the generation and selection of concepts based on the DOTs given in the previous section. The DOTs presented previously form the basis for concept selection: the feasible end-nodes of the trees are combined to yield a number of concepts. For example one of these combinations would be a rigid lifting body with long mission duration and control via aerodynamic shape alteration. From this number of concepts the unfeasible combinations are firstly eliminated. This is followed upon by eliminating the weakest concepts. The remaining concepts are then used for further analysis, to yield a select number of concepts deemed most viable to fulfill the mission need statement, as stated in Ref.[52]. After formulation of a set of trade-off criteria, the concepts are then analyzed for their performance in terms of these criteria. A review of this process takes place in the Mid-Term Review (MTR). A trade-off then yields the concept selected for further analysis and design, with the Final Review (FR) as final activity.

11 Conclusion

The goal of this technical project is to design a hypersonic decelerator that meets the requirements set by the customer. Several steps have been taken to ensure that the final product will meet said requirements.

First a literature review was conducted to identify the computational methods that will be used to analyze the different product concepts. This was done for the fields of structural engineering, aerodynamics, thermal engineering, control systems and astrodynamics. In addition to this the current states of (re-)entry vehicle technologies in general and inflatable aeroshell technologies in particular were determined. Following this a mass budget breakdown was made of the crew capsule and hypersonic decelerator. Thirdly the functional flow of the design mission was analyzed. This was done by producing a Functional Flow Diagram and Functional Breakdown Structure. Following this a requirement analysis was executed by making a Requirements Discovery Tree. The RDT consists of a requirement flowdown, documenting how system and subsystem requirements follow from the top-level requirements set by the customer.

After the requirement analysis a market investigation was conducted. This consisted of listing the potential customers and product functions, succeeded by a SWOT analysis. Following this an assessment was made of the potential risk sources encountered during the product design. In addition to this a procedure was set up to ensure that the Technical Performance Measurements of the final product will meet the demands of the customer. This will be done with the use of Resource Contingency Allowances that decrease as the product design matures. Furthermore the project approach with respect to sustainable development was described, followed by an exploration of all different design options in a Design Option Tree (DOT). This DOT contains all concepts that were envisioned, including those that will be eliminated later on in several design phases.

Following this Baseline Review (BR) the DOT will be used to conceive several conceptual designs that will be analysed and evaluated against each other during a trade-off process. The result of this trade-off will be a ranking of concepts, with the best concept being analyzed in further detail after the Mid-Term Review (MTR).

References

[1] M. Rudisill, R. Howard, B. Griffin, and Others, "Lunar Architecture Team - Phase 2 Habitat Volume Estimation : Caution When Using Analogs ," in *Earth and Space Conference 2008*, (Long Beach), NASA Langley Research Center, 2008.

[2] C. Cassapakis and M. Thomas, "Inflatable Structures Technology Development Overview," in *AIAA 1995 Space Programs and Technologies Conference*, (Huntsville, AL), AIAA, 1995.

[3] S. J. Hughes, R. A. Dillman, B. R. Starr, and Others, "Inflatable Re-entry Vehicle Experiment (IRVE) Design Overview," in *18th AIAA Aerodynamic Decelerator Systems Technology Conference and Seminar*, AIAA, 2005.

[4] A. M. D. Cianciolo, J. L. Davis, D. R. Komar, and Others, "Entry, Descent, and Landing Systems Analysis Study: Phase 1 Report," Tech. Rep. TM-2010-216720, NASA, 2010.

[5] C. Johnson, H. Tran, and H. Dill, "Stardust Backshell and Back Interface Plate Design Verification Tests in the NASA Ames Arc Jet Facilities," in *32nd Thermophysics Conference*, (Atlanta, GA), AIAA Paper 97-2483, 1997.

[6] R. A. Dillman, S. J. Hughes, R. J. Bodkin, and Others, "Flight Performance of the Inflatable Reentry Vehicle Experiment II," tech. rep., NASA, 2010.

[7] R. a. Dillman, S. J. Hughes, and R. J. Bodkin, "Flight Performance of the Inflatable Reentry Vehicle Experiment 3," tech. rep., NASA, 2012.

[8] R. Dillman, S. Hughes, J. Dinonno, and Others, "Planned Flight Of The Terrestrial Hiad Orbital Reentry (THOR)," tech. rep., NASA, 2014.

[9] B. Smith, on P, C. L. Tanner, and Others, "A Historical Review of Inflatable Aerodynamic Decelerator Technology Development," tech. rep., Georgia Institute of Technology, NASA Langley, 2010.

[10] D. M. Bose, M. D. Toniolo, F. M. Cheatwood, and Others, "Impact of Vehicle Flexibility on IRVE-II Flight Dynamics," 2011.

[11] R. A. Dillman, V. T. Gsell, and E. L. Bowden, "Attitude Control Performance Of IRVE-3," tech. rep., NASA, 2012.

[12] D. K. Litton, D. M. Bose, F. M. Cheatwood, and Others, "Inflatable Re-entry Vehicle Experiment (IRVE-4) Overview," tech. rep., NASA, 2011.

[13] Freel, R. , G. Bilyeu, and Others, "Inflatable deployable space structures technology summary," in *49th International Astronautical Congress*, 1998.

[14] J. a. Samareh, "Estimating Mass of Inflatable Aerodynamic Decelerators Using Dimensionless Parameters," tech. rep., NASA Langley, 2011.

[15] G. Brown, "Estimating Minimum Inflation Pressure for Inflatable Aerodynamic Decelerators," in *20th AIAA Aerodynamic Decelerator Systems Technology Conference and Seminar*, no. May, 2009.

[16] C. H. Jenkins, *Gossamer Spacecraft: Membrane And Inflatable Structures Technology For Space Applications*. Reston, Virginia: American Institute of Aeronautics and Astronautics, 2001.

[17] W. Young and R. Budynas, *Roarks's Formulas for Stress and Strain*. New York: McGraw-Hill, 7 ed., 2002.

[18] M. C. Lindell, S. J. Hughes, M. Dixon, and Others, "Structural Analysis and Testing of the Inflatable Re-entry Vehicle Experiment (IRVE)," tech. rep., American Institute of Aeronautics and Astronautics, 2006.

[19] K. Yamada, Y. Kimura, D. Akita, and Others, "Study on Low-ballistic-coefficient Atmospheric-entry Technology Using Flexible Aeroshell," *Transactions of the Japan Society for Aeronautical and Space Sciences, Space Technology Japan*, vol. 7, no. ists26, 2009.

[20] B. Smith, on P., I. G. Clark, and Others, "Thermal, structural, and inflation modeling of an isotensoid supersonic inflatable aerodynamic decelerator," *IEEE Aerospace Conference Proceedings*, 2011.

[21] J. R. Wertz, D. F. Everett, and J. J. Puschell, *Space Mission Engineering: The New SMAD*. Hawthorne: Microsim Inc., 1 ed., 2011.

[22] T. Megson, *Aircraft Structures for Engineering Students*. Oxford: Elsevier, 5 ed., 2012.

[23] J. D. Anderson Jr., *Hypersonic and High-Temperature Gas Dynamics*. Reston, Virginia: American Institute of Aeronautics and Astronautics, Inc., second ed., 2006.

[24] J. J. Bertin, *Hypersonic Aerothermodynamics*. AIAA, 2nd editio ed., 1994.

[25] J. J. Bertin and R. M. Cummings, "Critical Hypersonic Aerothermodynamic Phenomena*," *Annual Review of Fluid Mechanics*, vol. 38, no. 1, pp. 129–157, 2006.

[26] J. E. Theisinger and R. D. Braun, "Multi-Objective Hypersonic Entry Aeroshell Shape Optimization," *Journal of Spacecraft and Rockets*, vol. 46, no. 5, pp. 957–966, 2009.

[27] J. D. Anderson Jr., *Fundamentals of Aerodynamics*. McGraw-Hill Education, fourth ed., 2007.

[28] B. R. Hollis, K. T. Berger, S. A. Berry, and Others, "Entry , Descent , and Landing Aerothermodynamics : NASA Langley Experimental Capabilities and Contributions," pp. 1–37.

[29] M. E. Tauber, G. P. Menees, and H. G. Adelman, "Aerothermodynamics of Transatmospheric Vehicles," in *AIAA/ASME 4th Joint Thermophysics and Heat Transfer Conference*, 1986.

[30] Y. A. Çengel and M. A. Boles, *Thermodynamics: An Engineering Approach*. McGraw-Hill, 6th ed., 2010.

[31] R. D. Karam, *Satellite Thermal Control for Systems Engineers*. AIAA, 1998.

[32] J. P. Holman, *Heat Transfer*. New York: McGraw-Hill, 9th ed., 2002.

[33] D. Smoot and F. Price, “3rd AerosDace Sciences Meeting,” *Engineering*, no. 66.

[34] K. Daryabeigi, “Thermal Analysis and Design Optimization of Multilayer Insulation for Reentry Aerodynamic Heating,” *Journal of Spacecraft and Rockets*, vol. 39, no. 4, pp. 509–514, 2002.

[35] J. E. Pavlosky and L. G. St.leger, “Apollo experience report: Thermal protection subsystem,” no. January 1974, 1974.

[36] J. L. Hall, “Presented at the 4,” no. September, pp. 2–6, 2001.

[37] C. Weiland, *Computational Space Flight Mechanics*. Springer, 1 ed., 2004.

[38] Z. V. Leszczynski, *Modeling, simulation and visualization of aerocapture.pdf*. PhD thesis, Naval Postgraduate School, 1998.

[39] C. G. Justus and D. L. Johnson, “Mars Global Reference Atmospheric Model 2001 Version (Mars-GRAM 2001): Users Guide,” 2001.

[40] R. Esmaelzadeh, “Atmospheric Re-Entry Control for Low Lift/Drag Vehicles,” in *The 9th Iranian Aerospace Society Conference*, Feb. 2010.

[41] J. Mulder, W. van Staveren, J. van der Vaart, and Others, *Flight Dynamics Lecture Notes*. Delft: TU Delft, march 2013 ed., 2013.

[42] F. D. Steketee, “Dynamic Stability Of Space Vehicles,” in *Nasa Cr-945*, no. November 1967, 1967.

[43] D. Ito, J. Georgie, J. Valasek, and Others, “Reentry Vehicle Flight Controls Design Guidelines: Dynamic Inversion,” no. March, 2002.

[44] E. Axdahl, J. R. Cruz, M. Schoenenberger, and Others, “Flight Dynamics of an Aeroshell Using an Attached Inflatable Aerodynamic Decelerator,” in *20th AIAA Aerodynamic Decelerator Systems Technology Conference and Seminar*, no. AIAA-2009-2963, pp. 1–17, 2009.

[45] N. S. Nise, *Control Systems Engineering*. Wiley, 6 ed., 2011.

[46] R. Azor, I. Y. Bar-Itzhack, and R. R. Harman, “Satellite Angular Rate Estimation from Vector Measurements,” *Journal of Guidance, Control, and Dynamics*, vol. 21, no. 3, pp. 450–457, 1998.

[47] D. S. Jones, S. J. Brooks, M. W. Barnes, and Others, “Executive Summary of Propulsion on the Orion Abort Flight-Test Vehicles,” Dec. 2012.

[48] E. M. Braun, F. K. Lu, and D. R. Wilson, “Experimental research in aerodynamic control with electric and electromagnetic fields,” *Progress in Aerospace Sciences*, vol. 45, pp. 30–49, Jan. 2009.

[49] M. Kawamura, Y. Nagata, H. Katsurayama, and Others, “Magnetoaerodynamic Force on a Magnetized Body in a Partially Ionized Flow,” *Journal of Spacecraft and Rockets*, vol. 50, no. 2, pp. 347–351, 2013.

[50] A. Korzun and R. Braun, “Performance Characterization of Supersonic Retropropulsion for Application to High Mass Mars Entry, Descent, and Landing,” in *AIAA Atmospheric Flight Mechanics Conference*, pp. 1–29, 2009.

[51] T. Petsopoulos, F. J. Regan, and J. Barlow, “Moving-Mass Roll Control System for Fixed-Trim Re-Entry Vehicle,” *Journal of Spacecraft and Rockets*, vol. 33, no. 1, pp. 54–60, 1996.

[52] S. Balasooriyan, B. van Dongen, D. Hage, and Others, “Design Synthesis Exercise Project Plan,” Tech. Rep. 01, Delft University of Technology, Delft, 2015.

[53] W. D. Callister and D. G. Rethwisch, *Materials science and engineering: an introduction*. New York: Wiley, 7 ed., 2007.

[54] Goddard Space Flight Center, “Rules for the Design , Development , and Operation of Flight Systems Director of Applied,” tech. rep., National Aeronautics and Space Administration, Greenbelt, Maryland, 2013.

[55] M. H. Masud, M. Akhtaruzzaman, M. S. Bari, and Others, “Engineers’ Obligations towards Sustainable Environment,” in *2nd International Conference on Engineering Professional Ethics and Education*, (Kuala Lumpur), International Islamic University Malaysia, 2011.

[56] United Nations, “United Nations Treaties and Principles on Outer Space,” 2008.

[57] R. R. Rohrschneider and R. D. Braun, “Survey of Ballute Technology for Aerocapture,” *Journal of Spacecraft and Rockets*, vol. 44, no. 1, pp. 10–23, 2007.

[58] K. L. Miller, D. Gulick, J. Lewis, and Others, “Trailing Ballute Aerocapture : Concept and Feasibility Assessment,” *AIAA*, no. July, 2003.

A Requirements overview

Table 16: Overview of mission top-level requirements

ID	Description
CIA-A01	The re-entry vehicle shall be able to cope with an entry velocity of seven kilometers per second.
CIA-A02	The inflated aeroshell shall have a maximum diameter of 12 meters.
CIA-A03	The system shall have a diameter not exceeding 5 meters in stowed condition
CIA-A04	The maximum entry mass of the re-entry vehicle shall be 10,000 kilograms at the start of the mission.
CIA-A05	The hypersonic deceleration system mass shall not be heavier than ten percent of the total re-entry vehicle mass.
CIA-A06	The control system shall have a maximum failure probability of 5.0e-4.
CIA-A07	The maximum allowable loads on the re-entry vehicle shall be 3 Earth g's in each axis.
CIA-A08	The re-entry vehicle shall have a maximal aerobraking duration of ten Earth days.
CIA-A09	The re-entry vehicle shall accommodate TBD humans as payload.

?? Other req. tables if finished.