Suborbital Satellite TO SPACE

Design Synthesis Exercise:
Final Report

DSE Group 15

Technische Universiteit Delft

TU Delft

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by

DSE Group 15

June 28, 2016

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Preface

We are a group of nine bachelor students studying Aerospace Engineering at Delft University of Technology. For our Final Bachelor project, the Design Synthesis Exercise (DSE), we were given the assignment to design a payload for the DART research rocket. The assignment was commissioned by T-Minus Engineering B.V., a company specialised in designing, building and operating rocket systems and other space vehicles for research purposes, in conjunction with Delft University of Technology. This is the final report describing the payload configuration and the entire design process the team has gone through.

The payload designed will determine the reached altitude of the DART, shoot high-definition video footage and register the kinematic behaviour of the DART rocket during launch. The obtained data will be delivered to the user before the payload disappears behind the horizon.

The report includes, but is not limited to; a description of the mission outline, the satellite requirements, design trade-offs and the satellite costs. Readers who are especially interested in the satellite configuration can find the details in Chapter 5. The payload had to be designed extremely small, the volume available was a cylinder with a length of 250mm and a radius of 15 mm. To visualise the space we had available, an aluminium bar was made representing the total volume. The small slender tube quickly gave us the inspiration for the name of the payload, TubeSat.

We would like to thank T-Minus Engineering B.V. for giving us the opportunity to design the payload of their DART rocket and for answering all of our questions. We are also very grateful for the help of our tutor Chris Verhoeven and our coaches Wim Simons and Lucia Azzini, who have guided us through the design process. A special thanks goes out to Erwin Mooij of the TU Delft, for giving advice on re-entry and high altitude descent, Kees Sudmeijer of the TU Delft and MSc student Wouter Dubois, for giving advice on structural integrity, and last but not least Gerard Janssen of the TU Delft and BSc student Nils von Storch, for giving advice on communications.

Delft, June 28, 2016

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Contents

Nomenclature

List of Symbols

Symbol	Description	Unit
Roman		
a	Satellite radius	$[m]$
a	Acceleration	$[m/s^2]$
a_i	Acceleration at time t	$[m/s^2]$
a_{Para}	Acceleration of parachute	$[m/s^2]$
a_{Sat}	Acceleration of satellite	$[m/s^2]$
a_x	Acceleration in x-direction	$[m/s^2]$
a_y	Acceleration in y-direction	$[m/s^2]$
a_1	Acceleration of parachute	$[m/s^2]$
a_2	Acceleration of satellite	$[m/s^2]$
A	Area through which heat energy is transferred	$[m^2]$
A	Cross-sectional area	$[m^2]$
A	Face area of satellite dish	$[m^2]$
$A_{adhesive}$	Contact area between the quartz lens and the adhesive	$[mm^3]$
A_{Para}	Frontal area of parachute	$[m^2]$
$A_{pressure}$	Area of quartz on which vacuum acts	$[mm^3]$
A_{Sat}	Frontal area of satellite	$[m^2]$
A_1	Area of Hyperflo wall in top view	$[m^2]$
A_2	Area of Hyperflo mesh	$[m^2]$
A_3	Area of Hyperflo centre disk	$[m^2]$
AoV	Angle of view	[°]
c	Damping coefficient	$[-]$
c	Specific heat coefficient	$[-]$
c	Speed of light	$[m/s]$
c_{cr}	Critical damping coefficient	$[-]$
C_d	Two-dimensional drag coefficient	$[-]$
C_D	Drag coefficient	$[-]$
$C_{D_{Para}}$	Drag coefficient of parachute	$[-]$
$C_{D_{Sat}}$	Drag coefficient of satellite	$[-]$
$(C_{D_0})_{Friction}$	Friction drag coefficient	$[-]$
$(C_{D_0})_{Base}$	Base drag coefficient	$[-]$
$(C_{D_0})_{Wave}$	Wave drag coefficient	$[-]$
d	Distance between transmitter and receiver	$[m]$
d	Diameter	$[m]$
d_i	Diameter of hyperflo roof	$[m]$
d_p	Inlet diameter of hyperflo	$[m]$
E	Young's modulus	[Pa]
E_{ADCS}	Energy used by Attitude Determination & Control System	$[J]$
E_b	Received energy per bit	$[J]$
E_{DCS}	Energy used by Descent Control System	$[J]$
E_{Ejec}	Energy used by Ejection System	$[J]$
$E_{Emb_{i.j}}$	Energy used by Embedded System at $i.jV$	$[J]$
E_{Mar}	Margin of energy consumption	$[J]$
$E_{Pow_{supmar}}$	Energy provided by Power System as margin	$[J]$
$E_{Pow_{sup_{i.j}}}$	Energy provided by Power System at $i.jV$	$[J]$
$E_{Pow_{us}}$	Energy used by Power System	$[J]$
E_{Rec}	Energy used by Recovery System	$[J]$
E_{Sat}	Energy of suborbital satellite	$[J]$
$E_{Sat_{sup}}$	Energy provided by suborbital satellite	$[J]$
$E_{Sat_{us}}$	Energy used by suborbital satellite	$[J]$
E_{Scie}	Energy used by Scientific Payload	$[J]$
E_{Struc}	Energy used by Structural System	$[J]$
$E_{Trans_{i.j}}$	Energy used by Transmission System at $i.jV$	$[J]$
$E_{Vid_{i.j}}$	Energy used by Video System at $i.jV$	$[J]$
f	Focal length	$[m]$
f	Frequency	$[Hz]$
F	Force	$[N]$
F_{cable}	Cable force	$[N]$
F_{d_1}	Atmospheric drag force on parachute	$[N]$
F_{d_2}	Atmospheric drag force on satellite	$[N]$
F_D	Atmospheric drag force	$[N]$
F_g	Gravity force	$[N]$
F_{g_1}	Gravity force on parachute	$[N]$
F_{g_2}	Gravity force on satellite	$[N]$
F_{max}	Maximum force in yo-yo rope	$[N]$
F_T	Thrust force	$[N]$
F_x	Force in x-direction	$[N]$
F_y	Force in y-direction	$[N]$
g	Gravitational acceleration at Earth's surface	$[m/s^2]$
g_i	Gravitational acceleration at a radius of r_i from Earth's centre	$[m/s^2]$

Symbol	Description	Unit	Symbol	Description	Unit
g_0	Standard gravitational acceleration	$[m/s^2]$	P_{ADCS}	Power used by Attitude Determination & Control System	$[W]$
G	Ratio used to calculate yo-yo mass and rope length	$[-]$	$P_{ambient}$	Ambient air pressure at sea-level	$[Pa]$
G_A	Gain of satellite dish	$[-]$	P_{DCS}	Power used by Descent Control System	$[W]$
G_{AR}	Gain of receiving antenna	$[-]$			
G_{AT}	Gain of transmitting antenna	$[-]$	P_{Ejec}	Power used by Ejection System	$[W]$
h	Height	$[m]$	P_{Mar}	Margin of power consumption	$[W]$
h_i	Altitude at time t	$[m]$	$P_{Pow_{supmar}}$	Power provided by Power System as margin	$[W]$
h_{i+1}	Altitude at time $t+\Delta t$	$[m]$			
I	Area moment of inertia	$[m^4]$	$P_{Pow_{sup_{i.j}}}$	Power provided by Power System at $i.jV$	$[W]$
I	Moment of inertia of satellite including yo-yo's but excluding Steiner term of yo-yo's	$[kg \cdot m^2]$	$P_{Pow_{us}}$	Power used by Power System	$[W]$
			P_{Rec}	Power used by Recovery System	$[W]$
k	Constant	$[kg/m]$			
k	Thermal conductivity of insulation material	$[W/(mK)]$	P_{Sat}	Power of suborbital satellite	$[W]$
			$P_{Sat_{sup}}$	Power provided by suborbital satellite	$[W]$
k	Spring constant	$[N/m]$			
k_i	Spring constant of mass i	$[N/m]$	$P_{Sat_{us}}$	Power used by suborbital satellite	$[W]$
$k_{eq_{series}}$	Equivalent spring constant of a series of masses	$[N/m]$			
			P_{Scie}	Power used by Scientific Payload	$[W]$
l	Length of satellite	$[m]$			
l	Length of yo-yo rope	$[m]$	P_{Struc}	Power used by Structural System	$[W]$
l_i	Height of hyperflo	$[m]$			
l_n	Length of satellite nose	$[m]$	P_t	Power of transmitted signal	$[W]$
L	Length of discretized beam	$[m]$	$P_{t_{req}}$	Required power of transmitted signal	$[W]$
L	Thickness of insulation material	$[m]$			
L_a	Atmospheric loss factor due to rain	$[-]$	$P_{Trans_{i.j}}$	Power used by Transmission System at $i.jV$	$[W]$
L_{fs}	Free space loss	$[-]$	$P_{Vid_{i.j}}$	Power used by Video System at $i.jV$	$[W]$
L_l	Loss factor from transmitter to antenna	$[-]$			
			q	Dynamic pressure	$[Pa]$
L_{pr}	Antenna pointing loss	$[-]$	q_c	Convective heat flux	$[W/m^2]$
L_r	Loss factor from antenna to receiver	$[-]$	q_{rad}	Radiative heat flux	$[W/m^2]$
			Q	Heat energy	$[W/s]$
m	Total satellite mass	$[kg]$	Q	First moment of area	$[m^3]$
m	Mass of yo-yo's	$[kg]$	r	Radius	$[m]$
m_{ADCS}	Mass of Attitude Determination & Control System	$[kg]$	r	Ratio of final spin rate over initial spin rate of satellite	$[-]$
m_{DCS}	Mass of Descent Control System	$[kg]$	r_i	Radius from Earth's centre	$[m]$
m_{Ejec}	Mass of Ejection System	$[kg]$	r_3	Radius of Hyperflo centre disk	$[m]$
m_{Emb}	Mass of Embedded System	$[kg]$	R_b	Data rate	$[bit/s]$
m_{Mar}	Suborbital satellite mass margin	$[kg]$	R_e	Earth radius	$[m]$
m_{Para}	Mass of parachute	$[kg]$	R_N	Radius of curvature of the structure at stagnation point	$[m]$
m_{Pow}	Mass of Power System	$[kg]$			
m_{Rec}	Mass of Recovery System	$[kg]$	s	Horizontal distance	$[m]$
m_{Sat}	Suborbital satellite mass	$[kg]$	s_i	Horizontal distance covered at time t	$[m]$
m_{Scie}	Mass of Scientific Payload	$[kg]$			
m_{Struc}	Mass of Structural System	$[kg]$	s_{i+1}	Horizontal distance covered at time $t+\Delta t$	$[m]$
m_{Trans}	Mass of Transmission System	$[kg]$			
m_{Vid}	Mass of Video System	$[kg]$	S_o	Total cloth area of hyperflo	$[m^2]$
m_1	Parachute mass	$[kg]$	S_p	Maximum projected frontal area of hyperflo	$[m^2]$
m_2	Satellite mass	$[kg]$			
mp	Mesh porosity	$[-]$	SNR	Signal-to-noise ratio	$[dB]$
M	Mach number	$[-]$	t	Thickness	$[m]$
M	Mass of heated object	$[kg]$	t	Time	$[s]$
N_0	Noise spectral density	$[J]$	t_f	Flight time	$[s]$
p	Rotational rate about longitudinal axis	$[rad/s]$	t_{fall}	Fall time of satellite	$[s]$
			T	Kinetic energy	$[J]$

Symbol	Description	Unit	Symbol	Description	Unit
T	Temperature	$[K]$	$\dot{\gamma}$	Angular velocity of yo-yo rope	$[rad/s]$
T_C	Cold temperature on one side of insulation material	$[K]$	Δt	Discrete time step	$[s]$
			$\Delta X_{h,wind}$	Horizontal wind displacement	$[m]$
T_H	Hot temperature on one side of insulation material	$[K]$	ΔX_i	Flight path deviation	$[m]$
			ϵ	Ejection angle	[°]
T_{sys}	Noise temperature of the system	$[K]$	ϵ	Emissivity	$[-]$
T_w	Wall temperature in stagnation point	$[K]$	ζ	Damping ratio	$[-]$
			η	Efficiency of satellite dish	$[-]$
U	Voltage	$[V]$	θ	Launch angle	[°]
v	Velocity	$[m/s]$	θ	Angle between the horizontal and attachment point of yo-yo rope	$[rad]$
v_i	Falling velocity at time t	$[m/s]$			
v_{i+1}	Falling velocity at time $t+\Delta t$	$[m/s]$			
v_x	Horizontal velocity	$[m/s]$	θ_H	Horizontal scanning angle	[°]
$v_{x,i}$	Horizontal velocity at time t	$[m/s]$	$\theta_{P/2}$	Half power angle	[°]
$v_{x,i+1}$	Horizontal velocity at time $t+\Delta t$	$[m/s]$	θ_T	Pitch angle deviation as unit of thrust misalignment δ_T	$[rad]$
v_y	Vertical velocity	$[m/s]$			
$v_{y,i}$	Vertical velocity at time t	$[m/s]$	θ_V	Vertical scanning angle	[°]
$v_{y,i+1}$	Vertical velocity at time $t+\Delta t$	$[m/s]$	θ_w	Pitch angle deviation as unit of wind velocity V_w	$[rad]$
v_o	Initial velocity	$[m/s]$			
V	Shear force	$[N]$	θ_0	Pitch angle	[°]
V_c	Orbital velocity	$[m/s]$	λ	Wave length	$[m]$
V_w	Horizontal wind velocity	$[m/s]$	ρ	Air density	$[kg/m^3]$
w	Width	$[m]$	σ_i	Normal stress in i-direction	$[Pa]$
W_d	Natural damping frequency	$[Hz]$	τ	Shear stress	$[Pa]$
W_n	Natural frequency	$[Hz]$	τ_{ij}	Shear stress in ij-plane	$[Pa]$
Y	Yield strength	$[Pa]$	τ_{min}	Shear stress to be resisted	$[Pa]$
Greek			$\dot{\phi}$	Spin rate of satellite	$[rad/s]$
α_h	Horizontal angle of camera	[°]	$\dot{\phi}_f$	Final spin rate of satellite	$[rad/s]$
α_v	Vertical angle of camera	[°]	$\dot{\phi}_i$	Initial spin rate of satellite	$[rad/s]$
δ_T	Thrust misalignment	$[rad]$	ω_{final}	Final angular rate of satellite	$[rad/s]$
γ	Angle between the horizontal and yo-yo rope	$[rad]$	ω_i	Initial angular rate of satellite	$[rad/s]$
			$\omega_{initial}$	Initial angular rate of satellite	$[rad/s]$

Constants

Symbol	Description	Value of constant	Unit
k	Boltzmann Constant	$1.38064852 \cdot 10^{-23}$ / -228.601	$[J/K]$ / $[dBJ/K]$
ρ_0	ISA sea-level air density	1.225	$[kg/m^3]$
σ	Stefan-Boltzmann constant	$5.670367 \cdot 10^{-8}$	$[kg/(s^3 \cdot K^4)]$

List of Abbreviations

Abbreviation	Meaning	Abbreviation	Meaning
AIAA	American Institute of Aeronautics and Astronautics	ASK	Amplitude-Shift Keying
		Atm	Atmosphere
ADC	Analog-to-Digital Converter	BER	Bit Error Rate
ADCS	Attitude Determination & Control System	Boos	Booster
		BPSK	Binary Phase-Shift Keying
ADR	European Agreement concerning the International Carriage of Dangerous Goods by Road	BtB	Board-to-Board
		Budg	Engineering budget
		BW	Bandwidth
AIM	Aeronomy of Ice in the Mesosphere	CAD	Computer-Aided Design
AMRL	Aeronautical and Maritime Research Laboratory	CFD	Computational Fluid Dynamics
		cg	Centre of gravity
AoV	Angle of View	Comm	Communications

Abbreviation	Meaning
COTS	Commercial Off-The-Shelf
cp	Centre of pressure
C2C	Cradle to Cradle
DC	Direct Current
DCS	Descent Control System
Des	Design
DISC	Dart Initiated ShuttleCock
DID	Dart in Dart
DLR	German Aerospace Centre
DSE	Design Synthesis Exercise
Ejec	Ejection System
Emb	Embedded System
EOL	End-Of-Life
ESA	European Space Agency
ESC	Esrange Space Centre
ESS	Electric Storage System
FBS	Functional Breakdown Structure
FEC	Forward Error Correction
FEM	Finite Element Method
FFBD	Functional Flow Block Diagram
FM	Frequency Modulation
FPV	First Person View
GNSS	Global Navigation Satellite System
GPLD	General Purpose Low Drag
GPS	Global Positioning System
Grnd	Ground station
GS	Ground station
HD	High-definition
HPBW	Half-Power Beam Width
ISA	International Standard Atmosphere
I^{2}C	Inter-Integrated Circuit
LAT	Low-Altitude Test
LEO	Low-Earth Orbit
LHCD	Left-Hand Circular Polarisation
LNA	Low-Noise Amplifier
MAHRS	Miniature Attitude Heading Reference System
Mat	Materials
MC	Monte Carlo
MEMS	Microelectromechanical systems
MLTI	Mesosphere and Lower Thermosphere/Ionosphere
NASA	National Aeronautics and Space Administration
NAv	Not Available
NOAA	National Oceanic and Atmospheric Administration
NOL	Naval Ordnance Laboratory
NTIS	National Technical Information Service
OEM	Original equipment manufacturer
Op	Operations
OQPSK	Offset Quadrature Phase-Shift Keying
OSIRIS	Optical Spectrograph and InfraRed Imager System
PaSoS	Parachute Suborbital Satellite
PCB	Printed Circuit Board
Perf	Performance
PLL	Phase-Locked Loop
PM	Phase Modulation
PMC	Polar Mesospheric Cloud
Pow	Power System
Pro	Production
QAM	Quadrature Amplitude Modulation
QPSK	Quadrature Phase-Shift Keying
RAMS	Reliability, Availability, Maintainability & Safety
RHCP	Right-Hand Circular Polarisation
ROHS	Restriction of Hazardous Substances
Safe	Safety & Reliability
SD	Secure Digital
SLRD	Super Loki Robin Dart
SMR	Sub-Millimetre Radiometer
SNR	Signal-to-Noise Ratio
SOQPSK	Shaped Offset Quadrature Phase-Shift Keying
SPI	Serial Peripheral Interface
SPS	Satellite Pointing System
SR	Set-Reset
SSC	Swedish Space Corporation
Struc	Structural System
Sub	Subsystem
Sust	Sustainability
Sys	System
S^3	Samara Suborbital Satellite
TAS	True airspeed
TIMED	Thermosphere, Ionosphere, Mesosphere Energetics and Dynamics
TME	T-Minus Engineering B.V.
Tra	Transport
Trans	Transmission System
TU	University of Technology
UART	Universal Asynchronous Receiver/-Transmitter
UHF	Ultra High Frequency
US	United States
USAF	United States Air Force
USB	Universal Serial Bus
VALID	Verifiable, Achievable, Logical, Integral, Definitive
VCO	Voltage Controlled Oscillator
Vid	Video System
WIP	Work In Progress
WSD	Weapons Systems Division
WTT	Wind Tunnel Test

Abstract

T-Minus Engineering B.V., in conjunction with Delft University of Technology (TU Delft), has commissioned an assignment for the Bachelor Final project, called the Design Synthesis Exercise (DSE), of the Faculty of Aerospace Engineering of the TU Delft. The assignment's objective is to design and build a very small, lightweight payload for the T-Minus DART research rocket, which will be able to validate the reached altitude, shoot high-definition video footage and record the behaviour of the rocket during ascent.

The volume reserved for the payload, which is also called suborbital satellite, inside the rocket has a cylindrical shape with a length of 250mm and a diameter of 30mm. Fitting everything inside this small tube was the main challenge of the project, but also gave rise to the name of the satellite: the TubeSat. Besides the volume other challenges were encountered, the main ones being: video quality, antenna pointing, transmission interference, the high heat generation and the high g-loads.

During the project, the Scrum method was used for planning. Using this method, the entire team was responsible for planning and carrying out the plan. Scrum is an Agile method that allows sudden problems or changes to be incorporated into the planning immediately.

The Tubesat consists of a main body and a drag device, called a hyperflo. The main body is mostly made of steel, but has a ceramic nose which will act as a heat shield. The nose also has a see-through section made of quartz glass which will allow a clear view for the camera. These materials also allow the data to be transmitted without interference. The Hyperflo is made of steel, its main function is to ensure that during descent the antenna of the TubeSat will point towards Earth.

To make sure the video is of high quality the TubeSat is de-spun after ejection using two yo-yo's. With a mass of 12 grams each, they will automatically deploy due to centrifugal forces and will de-spin the TubeSat to stand still. Moreover, to cope with vibrations, the camera is enclosed in Sorbothane, a high performance shock absorbing material. The reached altitude will be validated using a GPS antenna module. The kinematic behaviour of the rocket during ascent and the TubeSat during descent is recorded using two accelerometers, a gyroscope and a magnetometer.

The video and the behavioural data are transmitted over the S-Band, using a transmitter which has an output power of 330 mW. The data will be transmitted over a bandwidth of 8 MHz using QPSK modulation. A patch antenna is used to transmit the data using right-hand circular polarisation. To make the TubeSat more shock resistant, the inside has been filled with resin. This will prevent the wiring and the electronics from breaking due to vibrations.

1 Introduction

The majority of atmospheric measurements are conducted using sounding balloons which reach altitudes up to 50 km. Low-Earth Orbit (LEO) satellites can fly as low as 120 km before atmospheric drag makes sustained orbit impossible. This leaves the region between 50 and 120 km altitude relatively uncharted, due to which it has gained the nickname "Ignorosphere". An attractive solution to this problem is the use of sounding rockets. However, sounding rockets are often relatively expensive. T-Minus Engineering B.V. (from hereon referred to as TME) is a space engineering company manufacturing low-cost, relatively small sounding rockets with a very short set-up time. Their DART research rocket consists of a solid rocket booster with a so-called passive 'dart' mounted on top. When launched, the solid rocket booster burns for approximately 5 seconds, accelerating the dart to Mach 5.2 at 5 km altitude, after which it separates from the dart due to a difference in drag. The dart then coasts up to its apogee at 120 km altitude, where it ejects the payload before falling back to Earth.

The first DART rocket will be launched in 2017 as a proof of concept. The payload (from hereon referred to as 'TubeSat', or simply 'satellite') of this first launch will determine its reached altitude and measure the kinematic behaviour of the DART during launch. It will also shoot high-definition video footage which will be used for promotional purposes. Additionally, scientific measurements can be performed which will allow for analysis of the atmosphere between 50 and 120 km altitude. Designing the TubeSat is done with a sustainable mindset, so a method/system is designed to retrieve it.

The DSE team will design a satellite that can fulfil these functions. The mission statement is as follows: *"The DSE team will design a satellite for the T-Minus DART research rocket that will validate the height of apogee, conduct scientific measurements and provide 30 seconds of high-definition video footage of its deployment. Additionally, the team will develop a method/system to retrieve the satellite."*. At the moment an ejection system is already in place for the satellite, however, this system takes up half of the dart's volume. Reducing this volume would significantly increase the space for useful equipment. Thus, the additional objective is: *"reducing the size of the ejection system"*.

This final report is set up as follows: in Chapter 2 the mission outline is defined, from which the requirements on the system are derived in Chapter 3. In Chapter 4 satellite concepts are described. In Chapter 5 the chosen concept is worked out in detail. The trajectory of the satellite is discussed in Chapter 6. The communications, including a ground station, are elaborated on in Chapter 7. A preliminary design of the electronics is given in Chapter 8. A structural analysis is done in Chapter 9. Sensitivity analyses of the trajectory simulation and communications simulation are performed in Chapter 10, after which these simulations, as well as the product, are verified and validated in Chapter 11. The production and the costs of the satellite are outlined in Chapters 12 and 13, respectively. A market analysis is performed in Chapter 14. The risk management is elaborated on in Chapter 15. After that, in Chapter 16, the operations & logistics side of the mission is discussed. In Chapter 17 the RAMS characteristics are identified. A planning for the continuation of the development of the TubeSat is displayed in Chapter 19. The sustainable development strategy applied during satellite design is explained in Chapter 18. Finally, conclusions are drawn and recommendations are given in Chapters 20 and 21, respectively.

2
Mission

To be able to design the TubeSat, the mission has to be known. In Section 2.1 the mission outline is given. From this, the functions to be performed during the mission are derived in Section 2.2. From the mission functions, the satellite subsystems are determined in Section 2.3.

2.1. Mission Outline

As stated in the introduction of this report the mission is *"to design a satellite for the T-Minus DART research rocket that will validate the height of apogee, provide 30 seconds of high-definition video footage of its deployment and possibly conduct scientific measurements."* During the mission, the TubeSat is launched using the DART research rocket made by TME. This rocket has a booster that thrusts the dart to Mach 5.2 at 5 km altitude. After separation of the booster the dart coasts up to 120 km altitude. At apogee, the TubeSat validates the altitude and is ejected from the dart to re-enter the atmosphere on its own. Right before ejection the TubeSat starts shooting video footage of its deployment and descent and it sends all the recorded data down to the ground station. A timeline of the mission is provided in Table 2.1. In this table DART rocket refers to the entire rocket system including booster and dart refers to the passive dart only. In Figure 2.1 a visual representation of the flight can be seen.

Table 2.1: Mission timeline.

Time [s]	Action
0	- *DART rocket* is launched. - *DART rocket* starts providing real-time position data to the ground station. - *TubeSat* starts measuring kinematic behaviour.
5	- *DART rocket* has reached Mach 5.2 at approximately 5 km altitude. - Due to drag separation the *booster* separates from the dart.
5 - 150	- *Dart* coasts up to an altitude of 120 km.
148	- *TubeSat* starts shooting video footage.
150	- *Dart* reaches apogee at 120 km altitude. - *TubeSat* is ejected out of the dart. - *TubeSat* validates the altitude reached by the dart. - *TubeSat* starts transferring data. - *Dart* stops providing real-time position data to the ground station.
155	- *Hyperflo* is deployed. - *Yo-yo de-spin mechanism* is passively deployed.
160	- *Hyperflo* is fully deployed.
178	- *TubeSat* stops shooting video footage.
179	- *Recovery system* is turned on.
380	- *TubeSat* impacts the ground or water.

Performing the kinematic behaviour measurements and transmitting the data generated on board to the ground station will not be stopped, but continue until the TubeSat impacts the ground or the water. The recovery system will be turned off when the TubeSat has been found or when the power supply has run out.

Conducting scientific measurements has not been taken up in the mission outline. The reason for this can be read in Section 4.1.

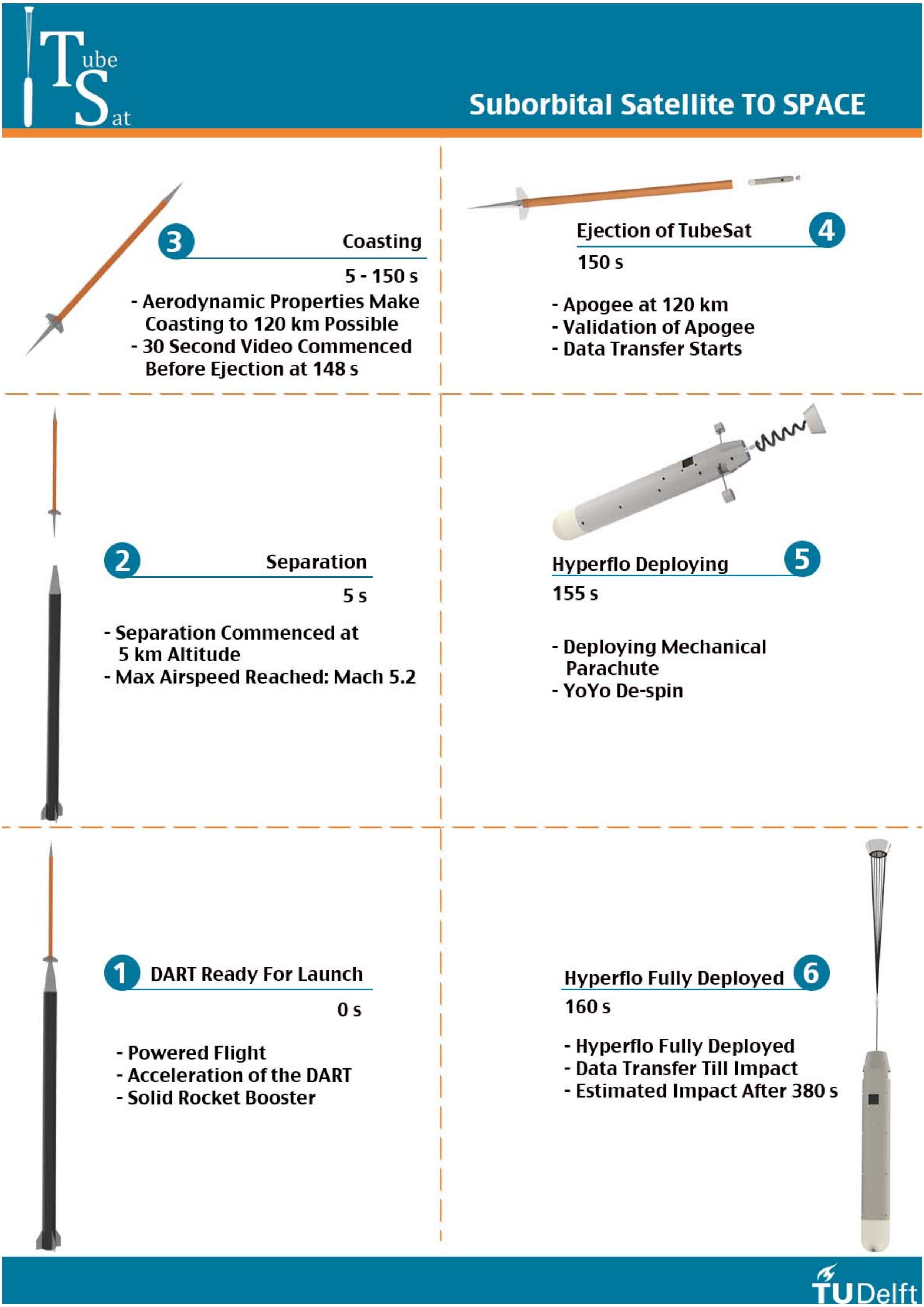


Figure 2.1: Infographics of mission outline.

2.2. Mission Functions

To successfully design the TubeSat, all mission functions and their order must be identified. This is done with the help of a functional breakdown structure (FBS) and a functional flow block diagram (FFBD).

The FBS is displayed in Figure 2.2. In this figure, the functions performed by the TubeSat are the functions in bold and italic font in the green boxes. The functions that are essential for mission success, but which are not performed by the TubeSat, are the functions shown in normal font in the yellow boxes.

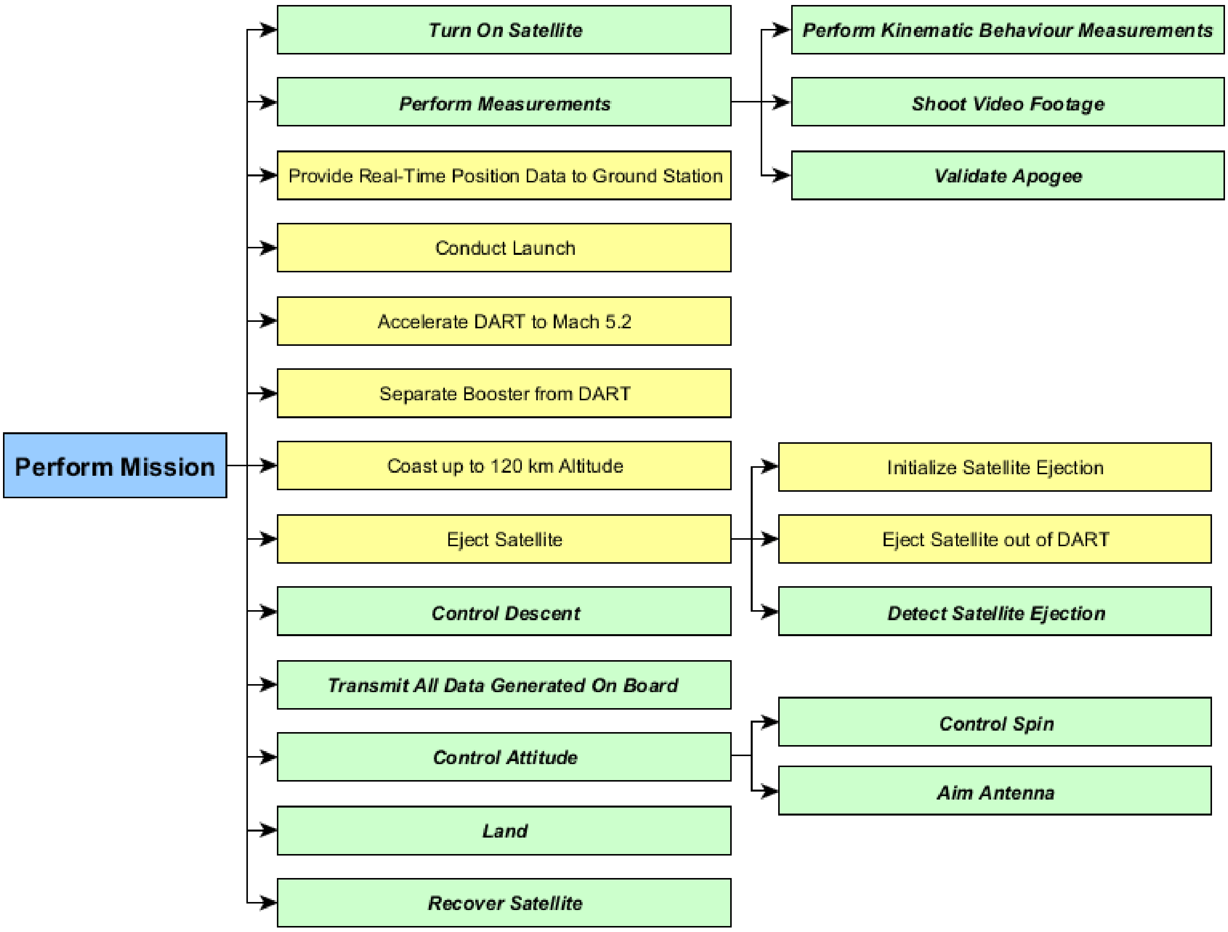


Figure 2.2: Functional breakdown structure of the mission.

The FFBD is displayed in Figure 2.3. In this figure, the functions performed by the TubeSat are the single-numbered functions in bold and italic font in the green boxes. The functions that are essential for mission success, but which are not performed by the TubeSat are the unnumbered functions shown in normal font in the yellow boxes. The function 'Start Attitude Control' is subdivided into multiple tasks. These tasks are the two-numbered functions in italics in the pink boxes.

As can be seen in Figure 2.3, multiple functions are started, but never stopped. Performing kinematic behaviour measurements and transferring data can be done until the TubeSat hits the ground or water. Controlling the attitude and descent is also necessary until landing or impact. The recovery system will be turned off when the TubeSat has been found or when the power supply has run out.

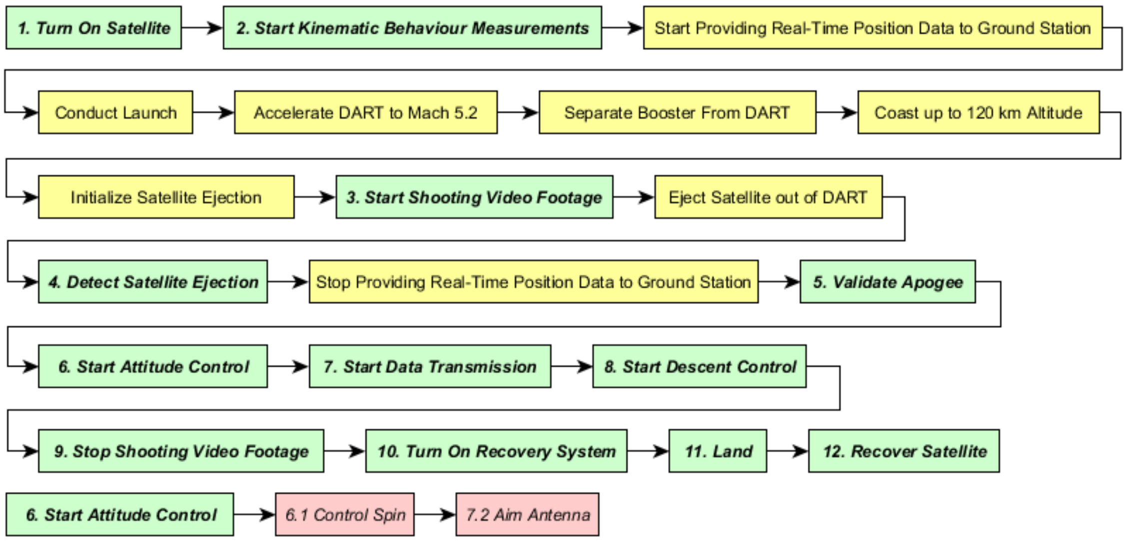


Figure 2.3: Functional flow block diagram of the mission.

2.3. Satellite Subsystems

Every satellite is built up of several subsystems. The subsystems are derived from the functions the satellite shall be able to perform. The 9 subsystems of the TubeSat are elaborated on below.

Structural System
The structural system is tasked with providing a structural backbone to the complete system. This means all components are mounted to it and the structural system has to cope with the forces exerted by all the components. The system is also tasked with protecting the other subsystems against launch loads and the atmospheric conditions during flight, such as the re-entry heat.

Power System
The power system provides power to all other subsystems which require power. The power system will mainly consist of a power supply. However, some electrical components, for example converters or resistors, might be needed for a proper power distribution.

Embedded System
The embedded system initiates and actuates the other subsystems. It also processes all data generated on board of the TubeSat.

Attitude Determination & Control System
The Attitude Determination & Control System (ADCS) is responsible for the attitude of the TubeSat. It controls the spin rate and aims the antenna to make data transmission possible. The ADCS also determines the attitude and position of the satellite. Additionally, it measures the kinematics of the DART rocket during ascent.

Ejection System
The ejection system ejects the TubeSat out of the dart at apogee. The ejection initiates several functions, such as apogee validation and shooting of the video. TME has already designed and manufactured a dual-spring ejection system, but redesign of the ejection system is allowed. This might be necessary if not enough space is available in the dart for a satellite comprising all functions mentioned in this chapter.

Video System
The video system will provide TME with a 30-second high-definition (HD) video of space. It will also capture the deployment of the TubeSat. The system mainly consists of a camera module, but might also include an image processor if the camera module does not compress the video footage.

Descent Control System
The Descent Control System (DCS) is responsible for controlling the descent of the TubeSat. In case the free fall time is not enough to transmit all data generated on board, the DCS shall extend the fall time. The system shall also make sure that the antenna is pointed in the right direction to make data transmission possible.

Transmission System
The transmission system transmits all data generated during the mission, i.e. video, kinematic behaviour measurements, height of apogee and scientific measurements, to the ground station.

Recovery System
The recovery system makes recovery of the TubeSat more likely to be successful.

3 Satellite Requirements

To be able to design the TubeSat, requirements have to be established beforehand. The functional requirements are listed in Section 3.1. Requirements on sustainability are defined in Section 3.2 and in Section 3.3 the subsystem requirements are listed. The requirements which were not met in the end are listed in Section 3.4. Finally, the technical budgets are displayed in Section 3.5.

3.1. Functional Requirements

The functional requirements below follow from the mission objectives and system functions. Only the requirements that are relevant for the TubeSat are included, meaning requirements concerning the launch of the rocket are not part of this project. In the identifiers of the requirements abbreviations are used. These abbreviations are explained in Table 3.1. The abbreviations are listed in their order of appearance.

The satellite does not actively control its attitude, as discussed in Chapter 4. Therefore the attitude measurements are purely needed to understand the kinematic behaviour of the DART rocket during launch and the satellite during its ejection and descent.

Table 3.1: Abbreviations used in the identifiers of the functional requirements.

Abbreviation	Explanation
DART	Used to indicate the project
Sys	System
Perf	Performance
Budg	Engineering budget
Des	Design

Abbreviation	Explanation
Safe	Safety & Reliability
Comm	Communications
Cost	Costs
Grnd	Ground station

- Performance:
  - **DART-Sys-Perf-1**
    The TubeSat shall send a 30-second video during descent.
  - **DART-Sys-Perf-2**
    The link budget of the mission shall be sufficient to fulfil requirement **DART-Sys-Perf-1**.
  - **DART-Sys-Perf-3**
    The TubeSat shall measure its apogee with an accuracy of $50m$.
  - **DART-Sys-Perf-4**
    The TubeSat shall measure the accelerations along all axes (XYZ) experienced by the dart during flight with an absolute accuracy of $1g$.
  - **DART-Sys-Perf-5**
    The TubeSat shall measure the accelerations along all axes (XYZ) experienced during descent with an absolute accuracy of $0.01g$.
  - **DART-Sys-Perf-6**
    The video shall have a resolution of $1080p$.
  - **DART-Sys-Perf-7**
    The video shall have a frame rate of $30fps$.
  - **DART-Sys-Perf-8**
    The TubeSat shall measure the velocities of the dart along all axes (XYZ) with an absolute accuracy of $10m/s$.
  - **DART-Sys-Perf-9**
    The TubeSat shall measure the position of the dart in three dimensions (XYZ) with an absolute accuracy of $50m$.
  - **DART-Sys-Perf-10**
    The TubeSat shall measure the rotational rates of the dart along all axes (XYZ) with an absolute accuracy of $0.35rad/s$.

- **DART-Sys-Perf-11**
  The TubeSat shall measure the attitude of the dart along all axes (XYZ) with an absolute accuracy of $0.2rad$.
- **DART-Sys-Perf-12**
  The TubeSat shall have sufficient data storage capacity to store all the data produced during the mission.
- **DART-Sys-Perf-13**
  The data shall be compressed before transmission.
- **DART-Sys-Perf-14**
  The data compression shall use an algorithm that is publicly available.
- **DART-Sys-Perf-15**
  The TubeSat shall be retrievable after its mission.

* Engineering Budgets:
  - **DART-Sys-Budg-1**
    The maximum length of the satellite and the ejection system combined shall be $420mm$.
  - **DART-Sys-Budg-2**
    The TubeSat shall have a maximum diameter of $30mm$.
  - **DART-Sys-Budg-3**
    The mass budget of the TubeSat shall be $1.2kg$.
  - **DART-Sys-Budg-4**
    The TubeSat shall be able to withstand a $100g$ (XYZ) environment rocket flight with associated vibration levels.
  - **DART-Sys-Budg-5**
    The TubeSat shall be able to withstand a thermal environment range of -60 to $+120°C$.
  - **DART-Sys-Budg-6**
    The TubeSat shall be entirely stand-alone.
  - **DART-Sys-Budg-7**
    The design of the TubeSat shall be completed before 23 June 2016.

* Design:
  - **DART-Sys-Des-1**
    The design of the TubeSat shall be physically producible.

* Safety and Reliability:
  - **DART-Sys-Safe-1**
    The TubeSat shall not endanger people on the ground.
  - **DART-Sys-Safe-2**
    The TubeSat shall not endanger assets on the ground.
  - **DART-Sys-Safe-3**
    The TubeSat shall not endanger the sounding rocket.
  - **DART-Sys-Safe-4**
    A test plan shall be created, including tests that can be done at lower altitudes (< 2 km) on a military site in the Netherlands to verify the operational capability of the TubeSat.
  - **DART-Sys-Safe-5**
    The TubeSat shall be protected against lightning strike.
  - **DART-Sys-Safe-6**
    The TubeSat shall include a mechanical or electromechanical SAFE/ARM switch that disables all systems including the ejection system.
  - **DART-Sys-Safe-7**
    The SAFE/ARM switch shall be operated remotely outside of the danger zone.

* Communications:
  - **DART-Sys-Comm-1**
    The down-link frequency shall be within the frequency domain: 2.025 - 2.3 GHz, 2.5 - 2.67 GHz (S-Band), 399.9 - 403 MHz, 432 - 438 MHz, 460 - 470 MHz (UHF-Band)[1].
  - **DART-Sys-Comm-2**
    The up-link frequency shall be within the frequency domain: 2.025 - 2.3 GHz, 2.5 - 2.67 GHz (S-Band)[1].

* Costs:
  - **DART-Sys-Cost-1**
    COTS components shall be used if available.

* Ground Station:
  - **DART-Sys-Grnd-1**
    A ground station shall be designed.
  - **DART-Sys-Grnd-2**
    The ground station shall have an antenna.

The accuracies in requirements **DART-Sys-Perf-3** to **DART-Sys-Perf-5** and **DART-Sys-Perf-8** to **DART-Sys-Perf-11** are based on a number of sensors that are currently available.
The quality of the video is partly defined by requirements **DART-Sys-Perf-5** and **DART-Sys-Perf-6**. More requirements related to video quality are defined in Section 3.3.

3.2. Sustainability Requirements

Sustainability requirements for the TubeSat are listed below. In the identifiers of the requirements, abbreviations are used. These abbreviations are explained in Table 3.2. The abbreviations are listed in their order of appearance.

Table 3.2: Abbreviations used in the identifiers of the sustainability requirements.

Abbreviation	Explanation
Sust	Sustainability
Mat	Materials
Pro	Production
EOL	End-Of-Life
Tra	Transport

- Materials:
  - **DART-Sys-Sust-Mat-1**
    The TubeSat shall use components and materials that comply with environmental standards according to 'Restriction of Hazardous Substances' [2].
  - **DART-Sys-Sust-Mat-2**
    The material with the lowest adverse environmental impact shall be used if multiple materials fulfil the required specifications.
- Production:
  - **DART-Sys-Sust-Pro-1**
    The production method with the lowest ecological footprint shall be used if multiple production methods are able to accomplish the required result.
  - **DART-Sys-Sust-Pro-2**
    Subtractive manufacturing methods shall not be used if other manufacturing methods are also able to accomplish the required manufacturing task.
  - **DART-Sys-Sust-Pro-3**
    The COTS components shall comply with the sustainability requirements for the satellite.
  - **DART-Sys-Sust-Pro-4**
    The COTS components from the companies with the best environmental strategy shall be chosen.
  - **DART-Sys-Sust-Pro-5**
    Waste materials which are contaminated shall be cleaned before thrown in the designated containers.
  - **DART-Sys-Sust-Pro-6**
    Waste materials collection by or waste materials delivery to the appropriate recycling or reuse facility shall be arranged.
  - **DART-Sys-Sust-Pro-7**
    Disassembly of components which can be reused shall be possible.
  - **DART-Sys-Sust-Pro-8**
    For recycling, disassembly of components made from different materials shall be possible.
- End-Of-Life:
  - **DART-Sys-Sust-EOL-1**
    The satellite shall be designed such that after recovery, all the parts of the satellite shall be either reused, recycled or be biodegradable.
  - **DART-Sys-Sust-EOL-2**
    A recovery method shall be designed.
  - **DART-Sys-Sust-EOL-3**
    The recovery method shall be designed such that it complies with the sustainability requirements for the satellite.

[1] URL http://www.sscspace.com/Products-Services/satellitemanagementservices/datahandling [cited 28 April 2016]
[2] URL http://www.rohsguide.com/ [cited 27 April 2016]

- Transport:
  - **DART-Sys-Sust-Tra-1**
    Transportation shall be chosen such that, depending on the distance travelled, the vehicle with the lowest environmental impact[3] is used.
  - **DART-Sys-Sust-Tra-2**
    For the transport of materials and COTS components, the method with the lowest environmental impact shall be chosen.

3.3. Subsystem Requirements

Subsystem requirements for the TubeSat are listed below. The convention that will be used for the requirement identifiers is as follows: the identifier starts with (DART), then the subsystem the requirement is applied to (e.g. Embedded System), followed by the subsystem from which the requirement is derived (e.g. Video System) and finally a numeral (e.g. DART-Emb-Vid-4). The abbreviations used in the requirement identifiers are explained in Table 3.3. The abbreviations are listed in their order of appearance.

Table 3.3: Abbreviations used in the identifiers of the subsystem requirements.

Abbreviation	Explanation
Struc	Structural System
Pow	Power System
Emb	Embedded System
ADCS	Attitude Determination & Control System
Ejec	Ejection System
Vid	Video System
DCS	Descent Control System

Abbreviation	Explanation
Trans	Transmission System
Boos	Booster
Op	Operations
Atm	Atmosphere
Sys	System
Sub	Subsystem
Safe	Safety

- **Structural System:**
  In contrast to the Midterm Report [24] the requirements for the stiffness have become obsolete as no component required it.
  - **DART-Struc-Boos-1**
    The structural system shall withstand the launch loads of $100g$ in any direction.
  - **DART-Struc-DART-1**
    The structural system and the ejection system combined shall not exceed a length of $420mm$.
  - **DART-Struc-DART-2**
    The structural system shall not exceed a diameter of $30mm$.
  - **DART-Struc-Atm-1**
    The structural system shall withstand the atmospheric flight conditions.
  - **DART-Struc-Emb-1**
    The structural system shall hold the forces exerted by the embedded system.
  - **DART-Struc-Emb-2**
    The structural system shall provide a mounting for all embedded system components.
  - **DART-Struc-Pow-1**
    The structural system shall hold the forces exerted by the power system.
  - **DART-Struc-Pow-2**
    The structural system shall provide a mounting for all power system components.
  - **DART-Struc-ADCS-1**
    The structural system shall hold the forces exerted by the ADCS.
  - **DART-Struc-ADCS-2**
    The structural system shall provide a mounting for all ADCS components.
  - **DART-Struc-DCS-1**
    The structural system shall hold the forces exerted by the DCS.
  - **DART-Struc-DCS-2**
    The structural system shall provide a mounting for all DCS components.
  - **DART-Struc-Ejec-1**
    The structural system shall hold the forces exerted by the ejection system.
  - **DART-Struc-Ejec-2**
    The structural system shall provide a mounting for the ejection system.
  - **DART-Struc-Trans-1**
    The structural system shall hold the forces exerted by the transmission system.

[3] URL abstracts.aetransport.org/paper/download/id/1632 [cited 9 May 2016]

- **DART-Struc-Trans-2**
  The structural system shall provide a mounting for all transmission system components.
- **DART-Struc-Trans-3**
  The structural system shall not interfere with the transmission signal.
- **DART-Struc-Vid-1**
  The structural system shall hold the forces exerted by the video system.
- **DART-Struc-Vid-2**
  The structural system shall provide a mounting for all video system components.
- **DART-Struc-Rec-1**
  The structural system shall hold the forces exerted by the recovery system.
- **DART-Struc-Rec-2**
  The structural system shall provide a mounting for all recovery system components.
- **DART-Struc-Sys-1**
  The structural system shall have a mass of $0.700kg$ as stated in the mass budget (m_{Struc}).
- **DART-Struc-Sys-2**
  The structural system shall insulate the inside of the satellite such that the temperature does not surpass $85°C$.

- **Power System:**
  - **DART-Pow-Sys-1**
    The power system shall provide potential differences of $1.1V$, $1.2V$, $1.8V$, $2.5V$, $2.8V$, $3.3V$ and $3.6V$.
  - **DART-Pow-Sys-2**
    The power system shall provide $9909.5mW$ in total at $3.6V$ as stated in the power budget ($P_{Pow_{sup_{3.6}}}$).
  - **DART-Pow-Sys-3**
    The power system shall provide $550mW$ at $1.1V$ as stated in the power budget ($P_{Pow_{sup_{1.1}}}$).
  - **DART-Pow-Sys-4**
    The power system shall provide $233.9mW$ at $1.2V$ as stated in the power budget ($P_{Pow_{sup_{1.2}}}$).
  - **DART-Pow-Sys-5**
    The power system shall provide $269mW$ at $1.8V$ as stated in the power budget ($P_{Pow_{sup_{1.8}}}$).
  - **DART-Pow-Sys-6**
    The power system shall provide $25mW$ at $2.5V$ as stated in the power budget ($P_{Pow_{sup_{2.5}}}$).
  - **DART-Pow-Sys-7**
    The power system shall provide $868mW$ at $2.8V$ as stated in the power budget ($P_{Pow_{sup_{2.8}}}$).
  - **DART-Pow-Sys-8**
    The power system shall provide $6428.6mW$ at $3.3V$ as stated in the power budget ($P_{Pow_{sup_{3.3}}}$).
  - **DART-Pow-Sys-9**
    The power system shall provide a power margin of $495.5mW$ as stated in the power budget ($P_{Pow_{supmar}}$).
  - **DART-Pow-Sys-10**
    The power system shall consume no more than $1039.5mW$ as stated in the power budget ($P_{Pow_{us}}$).
  - **DART-Pow-Sys-11**
    The power system shall provide $2086.8J$ in total at $3.6V$ as stated in the energy budget ($E_{Pow_{sup_{3.6}}}$).
  - **DART-Pow-Sys-12**
    The power system shall provide $17.0J$ at $1.1V$ as stated in the energy budget ($E_{Pow_{sup_{1.1}}}$).
  - **DART-Pow-Sys-13**
    The power system shall provide $46.2J$ at $1.2V$ as stated in the energy budget ($E_{Pow_{sup_{1.2}}}$).
  - **DART-Pow-Sys-14**
    The power system shall provide $8.2J$ at $1.8V$ as stated in the energy budget ($E_{Pow_{sup_{1.8}}}$).
  - **DART-Pow-Sys-15**
    The power system shall provide $0.8J$ at $2.5V$ as stated in the energy budget ($E_{Pow_{sup_{2.5}}}$).
  - **DART-Pow-Sys-16**
    The power system shall provide $26.0J$ at $2.8V$ as stated in the energy budget ($E_{Pow_{sup_{2.8}}}$).
  - **DART-Pow-Sys-17**
    The power system shall provide $1678.1J$ at $3.3V$ as stated in the energy budget ($E_{Pow_{sup_{3.3}}}$).
  - **DART-Pow-Sys-18**
    The power system shall provide an energy margin of $104.3J$ as stated in the energy budget ($E_{Pow_{supmar}}$).
  - **DART-Pow-Sys-19**
    The power system shall consume no more than $206.2J$ as stated in the energy budget ($E_{Pow_{us}}$).
  - **DART-Pow-Sys-20**
    The power system shall have a mass of $0.009kg$ as stated in the mass budget (m_{Pow}).
  - **DART-Pow-Safe-1**
    The mechanical installation and electrical connection of the energy storage system (ESS) shall occur as late as possible.

  - **DART-Pow-Safe-2**
    The voltages between the contacts of the satellite, the contacts of the energy storage system and the ground shall be checked before installation of the energy storage system.
  - **DART-Pow-Safe-3**
    The charging of the energy storage system shall occur as late as possible.
  - **DART-Pow-Safe-4**
    The power system shall include a mechanical or electromechanical SAFE/ARM switch disconnecting any mechanical system that can cause harm.

- **Embedded System:**
  - **DART-Emb-Boos-1**
    The embedded system shall detect the separation of the booster from the dart.
  - **DART-Emb-Pow-1**
    The embedded system shall use potential differences of $1.2V$ and $3.3V$.
  - **DART-Emb-Pow-2**
    The embedded system shall not use more than $1.5mW$ at $1.2V$ as stated in the power budget ($P_{Emb_{1.2}}$).
  - **DART-Emb-Pow-3**
    The embedded system shall not use more than $1419mW$ at $3.3V$ as stated in the power budget ($P_{Emb_{3.3}}$).
  - **DART-Emb-Pow-4**
    The embedded system shall not use more than $0.3J$ at $1.2V$ as stated in the energy budget ($E_{Emb_{1.2}}$).
  - **DART-Emb-Pow-5**
    The embedded system shall not use more than $536.4J$ at $3.3V$ as stated in the energy budget ($E_{Emb_{3.3}}$).
  - **DART-Emb-ADCS-1**
    The embedded system shall initiate the ADCS.
  - **DART-Emb-ADCS-2**
    The embedded system shall process the attitude and position data coming from the ADCS.
  - **DART-Emb-ADCS-3**
    The embedded system shall actuate the ADCS.
  - **DART-Emb-Ejec-1**
    The embedded system shall detect the ejection of the TubeSat.
  - **DART-Emb-Vid-1**
    The embedded system shall initiate the shooting of the video footage.
  - **DART-Emb-Vid-2**
    The embedded system shall terminate the shooting of the video footage.
  - **DART-Emb-Vid-3**
    The embedded system shall process the video footage coming from the video system.
  - **DART-Emb-DCS-1**
    The embedded system shall initiate the DCS.
  - **DART-Emb-DCS-2**
    The embedded system shall process the descent data coming from the DCS.
  - **DART-Emb-Trans-1**
    The embedded system shall send all processed data to the transmission system.
  - **DART-Emb-Trans-2**
    The embedded system shall, in case of an active link with the ground station, check if all processed data is transmitted.
  - **DART-Emb-Rec-1**
    The embedded system shall, in case of a recovery system with electronic components, initiate the recovery system.
  - **DART-Emb-Sys-1**
    The embedded system shall have sufficient data storage capacity to store all the data produced during the mission.
  - **DART-Emb-Sys-2**
    The embedded system shall have a mass of $0.011kg$ as stated in the mass budget (m_{Emb}).
  - **DART-Emb-Safe-1**
    The embedded system shall include a mechanical or electromechanical SAFE/ARM switch that disables all systems in the TubeSat and the ejection system.
  - **DART-Emb-Safe-2**
    The embedded system shall only switch from SAFE to ARM when no personnel is present in the designated danger zone.
  - **DART-Emb-Safe-3**
    The embedded system shall be initiated to switch from SAFE to ARM remotely outside of the designated danger zone.
  - **DART-Emb-Safe-4**
    When the SAFE/ARM switch fails the satellite shall be in SAFE.

- **Attitude Determination & Control System (ADCS):**
  - **DART-ADCS-Boos-1**
    The ADCS shall register the attitude and position of the booster until separation of the booster from the dart.
  - **DART-ADCS-DART-1**
    The ADCS shall register the attitude and position of the dart until ejection of the TubeSat out of the dart.
  - **DART-ADCS-Atm-1**
    The ADCS shall withstand the atmospheric flight conditions.
  - **DART-ADCS-Pow-1**
    The ADCS shall use a potential difference of $3.3V$.
  - **DART-ADCS-Pow-2**
    The ADCS shall not use more than $181.2mW$ as stated in the power budget (P_{ADCS}).
  - **DART-ADCS-Pow-3**
    The ADCS shall not use more than $43.1J$ as stated in the energy budget (E_{ADCS}).
  - **DART-ADCS-Vid-1**
    The ADCS shall provide the camera pointing desired by the video system.
  - **DART-ADCS-Vid-2**
    The ADCS shall limit the maximum rotational rate p along the longitudinal axis to $0.35rad/s$.
  - **DART-ADCS-Vid-3**
    The ADCS shall limit the vibration to a frequency f of at most $2Hz$ during filming.
  - **DART-ADCS-Trans-1**
    The ADCS shall provide the antenna pointing desired by the transmission system.
  - **DART-ADCS-Sys-1**
    The ADCS shall measure the apogee of the TubeSat with an absolute accuracy of $50m$.
  - **DART-ADCS-Sys-2**
    The ADCS shall measure the accelerations along all axes (XYZ) experienced by the dart during flight with an absolute accuracy of $1g$.
  - **DART-ADCS-Sys-3**
    The ADCS shall measure the accelerations along all axes (XYZ) experienced during descent with an absolute accuracy of $0.01g$.
  - **DART-ADCS-Sys-4**
    The ADCS shall measure the velocities of the TubeSat with an absolute accuracy of $10m/s$.
  - **DART-ADCS-Sys-5**
    The ADCS shall measure the position of the TubeSat with an absolute accuracy of $50m$.
  - **DART-ADCS-Sys-6**
    The ADCS shall measure the rotational rates of the TubeSat in any direction during flight with an absolute accuracy of $0.35rad/s$.
  - **DART-ADCS-Sys-7**
    The ADCS shall measure the attitude of the TubeSat in flight with an absolute accuracy of $0.2rad$.
  - **DART-ADCS-Sys-8**
    The ADCS shall have a mass of $0.017kg$ as stated in the mass budget (m_{ADCS}).
  - **DART-ADCS-Sys-9**
    The ADCS shall not introduce loads greater than $100g$.

- **Ejection System:**
  - **DART-Ejec-DART-1**
    The ejection system shall not exert any force on the satellite when it is outside the dart.
  - **DART-Ejec-DART-2**
    The ejection system shall not exceed a diameter of $30mm$.
  - **DART-Ejec-Pow-1**
    The ejection system shall use a potential difference of $0V$.
  - **DART-Ejec-Pow-2**
    The ejection system shall not use more than $0mW$ as stated in the power budget (P_{Ejec}).
  - **DART-Ejec-Pow-3**
    The ejection system shall not use more than $0J$ as stated in the energy budget (E_{Ejec}).
  - **DART-Ejec-Sys-1**
    The ejection system shall eject the TubeSat at the apogee of the dart.
  - **DART-Ejec-Sys-2**
    The ejection system shall not accelerate the TubeSat with more than $100g$ during ejection.
  - **DART-Ejec-Sys-3**
    The ejection system shall have a mass of $0.015kg$ as stated in the mass budget (m_{Ejec}).

- **Video System:**
  - **DART-Vid-DART-1**
    The camera system shall film the ejection of the TubeSat out of the dart.
  - **DART-Vid-Op-1**
    The video system shall adhere to the security regulations of the launch site from which the DART research rocket is launched.
  - **DART-Vid-Atm-1**
    The video system shall withstand the atmospheric flight conditions.
  - **DART-Vid-Pow-1**
    The video system shall use voltages of $1.1V$, $1.2V$, $1.8V$, $2.5V$ and $2.8V$.
  - **DART-Vid-Pow-2**
    The video system shall not use more than $550mW$ at $1.1V$ as stated in the power budget ($P_{Vid_{1.1}}$).
  - **DART-Vid-Pow-3**
    The video system shall not use more than $36mW$ at $1.2V$ as stated in the power budget ($P_{Vid_{1.2}}$).
  - **DART-Vid-Pow-4**
    The video system shall not use more than $269mW$ at $1.8V$ as stated in the power budget ($P_{Vid_{1.8}}$).
  - **DART-Vid-Pow-5**
    The video system shall not use more than $25mW$ at $2.5V$ as stated in the power budget ($P_{Vid_{2.5}}$).
  - **DART-Vid-Pow-6**
    The video system shall not use more than $868mW$ at $2.8V$ as stated in the power budget ($P_{Vid_{2.8}}$).
  - **DART-Vid-Pow-7**
    The video system shall not use more than $17.0J$ at $1.1V$ as stated in the energy budget ($E_{Vid_{1.1}}$).
  - **DART-Vid-Pow-8**
    The video system shall not use more than $1.1J$ at $1.2V$ as stated in the energy budget ($E_{Vid_{1.2}}$).
  - **DART-Vid-Pow-9**
    The video system shall not use more than $8.2J$ at $1.8V$ as stated in the energy budget ($E_{Vid_{1.8}}$).
  - **DART-Vid-Pow-10**
    The video system shall not use more than $0.8J$ at $2.5V$ as stated in the energy budget ($E_{Vid_{2.5}}$).
  - **DART-Vid-Pow-11**
    The video system shall not use more than $26.0J$ at $2.8V$ as stated in the energy budget ($E_{Vid_{2.8}}$).
  - **DART-Vid-ADCS-1**
    The video system shall be able to function with the attitude provided by the ADCS.
  - **DART-Vid-DCS-1**
    The video system shall be able to function with the stability provided by the DCS.
  - **DART-Vid-Sys-1**
    The video system shall shoot a video during the descent of the TubeSat.
  - **DART-Vid-Sys-2**
    The video shall have a length of 30 seconds.
  - **DART-Vid-Sys-3**
    The video shall have a resolution of $1080p$.
  - **DART-Vid-Sys-4**
    The video shall have a frame rate of $30fps$.
  - **DART-Vid-Sys-5**
    The video system shall have a mass of $0.002kg$ as stated in the mass budget (m_{Vid}).
  - **DART-Vid-Sys-6**
    The video system shall compress the data produced during the mission before the data is transmitted by the transmission system.
  - **DART-Vid-Sys-7**
    The video system shall use an algorithm for data compression that is publicly available.

- **Descent Control System (DCS):**
  - **DART-DCS-Pow-1**
    The DCS shall use a potential difference of $0V$.
  - **DART-DCS-Pow-2**
    The DCS shall not use more than $0mW$ as stated in the power budget (P_{DCS}).
  - **DART-DCS-Pow-3**
    The DCS shall not use more than $0J$ as stated in the energy budget (E_{DCS}).
  - **DART-DCS-Vid-1**
    The DCS shall provide the stability control desired by the video system.
  - **DART-DCS-Vid-2**
    The DCS shall provide a pointing accuracy of 35°.
  - **DART-DCS-Trans-1**
    The DCS shall provide the descent time control desired by the transmission system.

  - **DART-DCS-Sys-1**
    The DCS shall have a mass of $0.020kg$ as stated in the mass budget (m_{DCS}).
  - **DART-DCS-Sys-2**
    The DCS shall not introduce loads greater than $100g$.

- **Transmission System:**
  - **DART-Trans-DART-1**
    The transmission signal shall be strong enough to withstand the interference introduced by the dart.
  - **DART-Trans-Op-1**
    The transmission system shall be able to fulfil **DART-Trans-Sys-1** with the available receiver.
  - **DART-Trans-Op-2**
    The transmission system shall adhere to the security regulations of the launch site from which the DART research rocket is launched.
  - **DART-Trans-Struc-1**
    The transmission signal shall be strong enough to overcome the interference introduced by the structural system.
  - **DART-Trans-Pow-1**
    The transmission system shall use potential differences of $1.2V$ and $3.3V$.
  - **DART-Trans-Pow-2**
    The transmission system shall not use more than $196.4mW$ at $1.2V$ as stated in the power budget ($P_{Trans_{1.2}}$).
  - **DART-Trans-Pow-3**
    The transmission system shall not use more than $4747.5mW$ at $3.3V$ as stated in the power budget ($P_{Trans_{3.3}}$).
  - **DART-Trans-Pow-4**
    The transmission system shall not use more than $44.8J$ at $1.2V$ as stated in the energy budget ($E_{Trans_{1.2}}$).
  - **DART-Trans-Pow-5**
    The transmission system shall not use more than $1082.4J$ at $3.3V$ as stated in the energy budget ($E_{Trans_{3.3}}$).
  - **DART-Trans-Sys-1**
    The transmission system shall transmit all data produced during the mission and compressed by the video system to the receiver at least three times.
  - **DART-Trans-Sys-2**
    The transmission system shall have a link budget sufficient to fulfil **DART-Trans-Sys-1**.
  - **DART-Trans-Sys-3**
    The down-link frequency shall be within the frequency domain: 2.025 - 2.3 GHz, 2.5 - 2.67 GHz (S-Band), 399.9 - 403 MHz, 432 - 438 MHz, 460 - 470 MHz (UHF-Band)[1].
  - **DART-Trans-Sys-4**
    The up-link frequency shall be within the frequency domain: 2.025 - 2.3 GHz, 2.5 - 2.67 GHz (S-Band)[1].
  - **DART-Trans-Sys-5**
    The transmission system shall include a ground station.
  - **DART-Trans-Sys-6**
    The ground station shall have an antenna.
  - **DART-Trans-Sys-7**
    The transmission system shall have a mass of $0.004kg$ as stated in the mass budget (m_{Trans}).

- **Recovery System:**
  - **DART-Rec-Pow-1**
    The recovery system shall use a potential difference of $3.3V$.
  - **DART-Rec-Pow-2**
    The recovery system shall not use more than $16.2J$ as stated in the energy budget (E_{Rec}).
  - **DART-Rec-Pow-3**
    The recovery system shall not use more than $80.9mW$ as stated in the power budget (P_{Rec}).
  - **DART-Rec-Sys-1**
    The recovery system shall have a mass of $0.002kg$ as stated in the mass budget (m_{Rec}).

In addition to the requirements stated above, the following requirements from Section 3.1 apply to all subsystems.

  - **DART-Sub-Sys-1**
    The subsystem shall be able to withstand a $100g$ (XYZ) environment rocket flight with associated vibration levels.
  - **DART-Sub-Sys-2**
    The subsystem shall be able to withstand a thermal environment range of -60 to $+120°C$.
  - **DART-Sub-Sys-3**
    The design of the subsystem shall be physically producible.
  - **DART-Sub-Sys-4**
    The subsystem shall not endanger assets or people on the ground.

- **DART-Sub-Sys-5**
  The subsystem shall not endanger the sounding rocket.
- **DART-Sub-Sys-6**
  The design of the subsystem shall be completed before 23 June 2016.
- **DART-Sub-Sys-7**
  COTS components shall be used in the subsystem if they are available.

Requirements **DART-ADCS-Vid-1** to **DART-ADCS-Vid-4** and **DART-DCS-Vid-1** to **DART-DCS-Vid-4** were defined to make sure that the video will be of good quality and pleasant to watch.

3.4. Compliance to Requirements

Not all requirements have been met. The requirements and the reason for noncompliance are given in Table 3.4.

Table 3.4: Requirements noncompliance table.

Requirement	Reason for noncompliance
DART-Sys-Budg-3	See Chapter 10
DART-Sys-Des-1	Electrical design could not be done
DART-Sys-Safe-6	Not applicable
DART-Sys-Safe-7	Not applicable
DART-Sys-Sust-Mat-1	No compliant component found
DART-Sys-Sust-EOL-1	Low probability of retrieval
DART-Pow-Safe-4	Not applicable
DART-Emb-ADCS-3	No electronically activated ADCS
DART-Emb-DCS-1	No electronically activated DCS
DART-Emb-DCS-2	No electronically activated DCS
DART-Emb-Safe-2	Not applicable
DART-Emb-Safe-3	Not applicable
DART-Emb-Safe-4	Not applicable

3.5. Technical Budgets

To make sure that the final satellite design is within the set limits, margins and contingencies are used in the earlier stages of the design. Below, the final technical budgets of the mass, power and energy are tabulated.

Mass Budget

The mass budget before ejection is presented in Table 3.5. The mass shown as the ejection system mass is that of the sabots. The ADCS mass resembles the mass of the yo-yo's, the DCS mass resembles the mass of the Hyperflo and the recovery system mass resembles the mass of the beacon.

Table 3.5: Mass budget.

Element	Symbol	Mass m $[g]$	% of m_{Sat}
Satellite	m_{Sat}	787	100
Structural system	m_{Struc}	700	88.9
Power system	m_{Pow}	9	1.2
Embedded system	m_{Emb}	11	1.4
ADCS	m_{ADCS}	24	3.0
Ejection system	m_{Ejec}	15	1.9
Video system	m_{Vid}	2	0.3
DCS	m_{DCS}	20	2.5
Scientific payload	m_{Scie}	0	0
Transmission system	m_{Trans}	4	0.5
Recovery system	m_{Rec}	2	0.3
Margin	m_{Mar}	0	0

Power Budget

The power budget is presented in Table 3.6. The power budget of the power system has been split up into power supply and power consumption. The power supply of the power system is set equal to 100% of the satellite's total power, because the power system has to provide all power used by the other subsystems during the mission. The power supply percentage of the power system is therefore equal to the power percentages of all other subsystems together. Power supply is indicated by positive values, whereas power consumption is indicated by negative values. In the subsystem requirements in Section 3.3 absolute values are used.

Table 3.6: Power budget.

Element	Supply/Usage	Voltage U $[V]$	Symbol	Power P $[mW]$	% of P_{Sat}
Satellite	Supply	1.1, 1.2, 1.8, 2.5, 2.8, 3.3, 3.6	$P_{Sat_{sup}}$	9909.5	100
	Usage	1.1, 1.2, 1.8, 2.5, 2.8, 3.3, 3.6	$P_{Sat_{us}}$	-9909.5	-100
Structural system	Usage	0	P_{Struc}	0	0
Power system	Supply	1.1	$P_{Pow_{sup_{1.1}}}$	550	5.6
		1.2	$P_{Pow_{sup_{1.2}}}$	233.9	2.4
		1.8	$P_{Pow_{sup_{1.8}}}$	269	2.7
		2.5	$P_{Pow_{sup_{2.5}}}$	25	0.2
		2.8	$P_{Pow_{sup_{2.8}}}$	868	8.8
		3.3	$P_{Pow_{sup_{3.3}}}$	6428.6	64.8
		1.1, 1.2, 1.8, 2.5, 2.8, 3.3, 3.6 (Margin)	$P_{Pow_{supmar}}$	495.5	5
		3.6 (For use)	$P_{Pow_{us}}$	1039.5	10.5
		3.6 (Total)	$P_{Pow_{sup_{3.6}}}$	9909.5	100
	Usage	3.6	$P_{Pow_{us}}$	-1039.5	-10.5
Embedded system	Usage	1.2	$P_{Emb_{1.2}}$	-1.5	≈ 0
		3.3	$P_{Emb_{3.3}}$	-1419	-14.3
ADCS	Usage	3.3	P_{ADCS}	-181.2	-1.8
Ejection system	Usage	0	P_{Ejec}	0	0
Video system	Usage	1.1	$P_{Vid_{1.1}}$	-550	- 5.6
		1.2	$P_{Vid_{1.2}}$	-36	-0.4
		1.8	$P_{Vid_{1.8}}$	-269	-2.7
		2.5	$P_{Vid_{2.5}}$	-25	-0.2
		2.8	$P_{Vid_{2.8}}$	-868	-8.8
DCS	Usage	0	P_{DCS}	0	0
Scientific payload	Usage	0	P_{Scie}	0	0
Transmission system	Usage	1.2	$P_{Trans_{1.2}}$	-196.4	-2.0
		3.3	$P_{Trans_{3.3}}$	-4747.5	-47.9
Recovery system	Usage	3.3	P_{Rec}	-80.9	-0.8
Margin	Usage	1.1, 1.2, 1.8, 2.5, 2.8, 3.3, 3.6	P_{Mar}	-495.5	-5.0

Energy Budget

The energy budget is presented in Table 3.7. The energy budget of the power system has been split up into energy supply and energy consumption. The energy supply of the power system is set equal to 100% of the satellite's energy, because the power system has to provide all energy used by the other subsystems during the mission. The energy supply percentage of the power system will therefore be equal to the energy percentage of all other subsystems together. Energy supply is indicated by positive values, whereas energy consumption is indicated by negative values. In the subsystem requirements in Section 3.3 absolute values are used.

In Table 3.7 the energy consumed by the recovery system is the energy used by the beacon to transmit until satellite impact. Of course, energy still needs to be provided to the recovery system after impact in order to be able to continue transmitting.

Table 3.7: Energy budget.

Element	Supply/Usage	Voltage U [V]		Symbol	Energy E [J]	% of E_{Sat}
Satellite	Supply	1.1, 1.2, 1.8, 2.5, 2.8, 3.3, 3.6		$E_{Sat_{sup}}$	2086.8	100
	Usage	1.1, 1.2, 1.8, 2.5, 2.8, 3.3, 3.6		$E_{Sat_{us}}$	-2086.8	-100
Structural system	Usage	0		E_{Struc}	0	0
Power system	Supply	1.1		$E_{Pow_{sup_{1.1}}}$	17.0	0.8
		1.2		$E_{Pow_{sup_{1.2}}}$	46.2	2.2
		1.8		$E_{Pow_{sup_{1.8}}}$	8.2	0.4
		2.5		$E_{Pow_{sup_{2.5}}}$	0.8	≈ 0
		2.8		$E_{Pow_{sup_{2.8}}}$	26.0	1.3
		3.3		$E_{Pow_{sup_{3.3}}}$	1678.1	80.4
		1.1, 1.2, 1.8, 2.5, 2.8, 3.3, 3.6 (Margin)		$E_{Pow_{sup_{mar}}}$	104.3	5.0
		3.6	(For use)	$E_{Pow_{us}}$	206.2	9.9
			(Total)	$E_{Pow_{sup_{3.6}}}$	2086.8	100
	Usage	3.6		$E_{Pow_{us}}$	-206.2	-9.9
Embedded system	Usage	1.2		$E_{Emb_{1.2}}$	-0.3	≈ 0
		3.3		$E_{Emb_{3.3}}$	-536.4	-25.7
ADCS	Usage	3.3		E_{ADCS}	-43.1	-2.1
Ejection system	Usage	0		E_{Ejec}	0	0
Video system	Usage	1.1		$E_{Vid_{1.1}}$	-17.0	-0.8
		1.2		$E_{Vid_{1.2}}$	-1.1	-0.1
		1.8		$E_{Vid_{1.8}}$	-8.2	-0.4
		2.5		$E_{Vid_{2.5}}$	-0.8	≈ 0
		2.8		$E_{Vid_{2.8}}$	-26.0	-1.3
DCS	Usage	0		E_{DCS}	0	0
Scientific payload	Usage	0		E_{Scie}	0	0
Transmission system	Usage	1.2		$E_{Trans_{1.2}}$	-44.8	-2.1
		3.3		$E_{Trans_{3.3}}$	-1082.4	-51.8
Recovery system	Usage	3.3		E_{Rec}	-16.2	-0.8
Margin	Usage	1.1, 1.2, 1.8, 2.5, 2.8, 3.3, 3.6		E_{Mar}	-104.3	-5.0

4
Design Trade-Offs

The development of the design followed a divergent convergent process, as shown in Figure 4.1. The first phase was to discover as many options as possible for all functions of the satellite, after which the preferred options per function were selected. As the descent method was the main driver of the design, the next phase was to create preliminary options and select the preferred one. These two phases were discussed in detail in the Midterm Report [24], and are summarised and updated in Sections 4.1 and 4.2. Then, Section 4.3 presents the selection of the final descent control system.

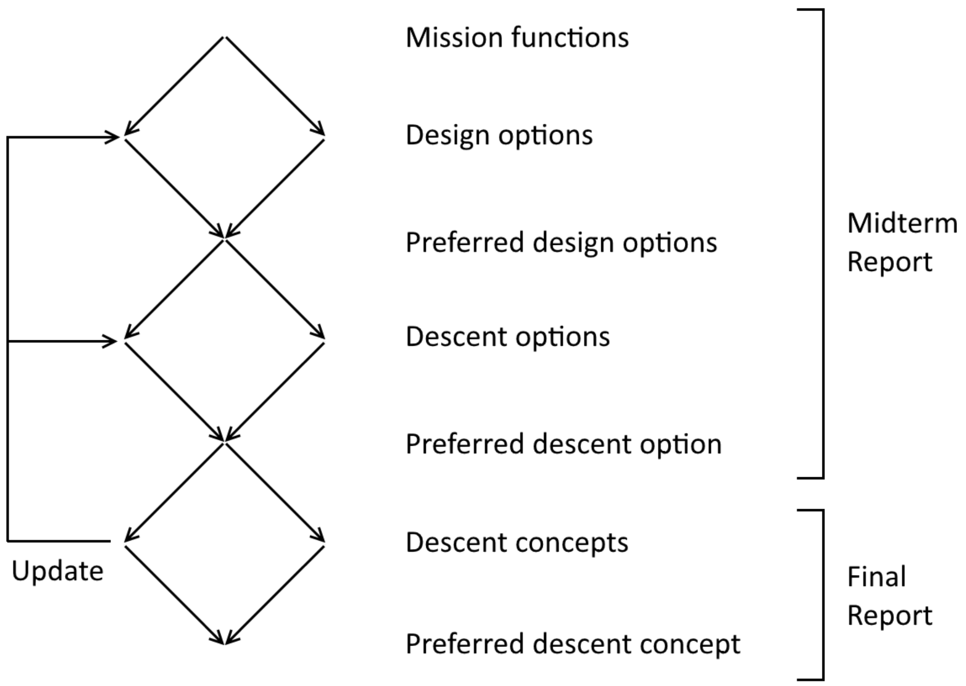


Figure 4.1: Design process.

4.1. Design Options

In order to fulfil all functions the TubeSat needs to perform, as defined in Section 2.2, a number of options were considered, as shown in Figure 4.2. The design options that were immediately considered non-feasible are not shown for brevity. Reasons for non-feasibility include: not possible to fit, too complex or not proven.

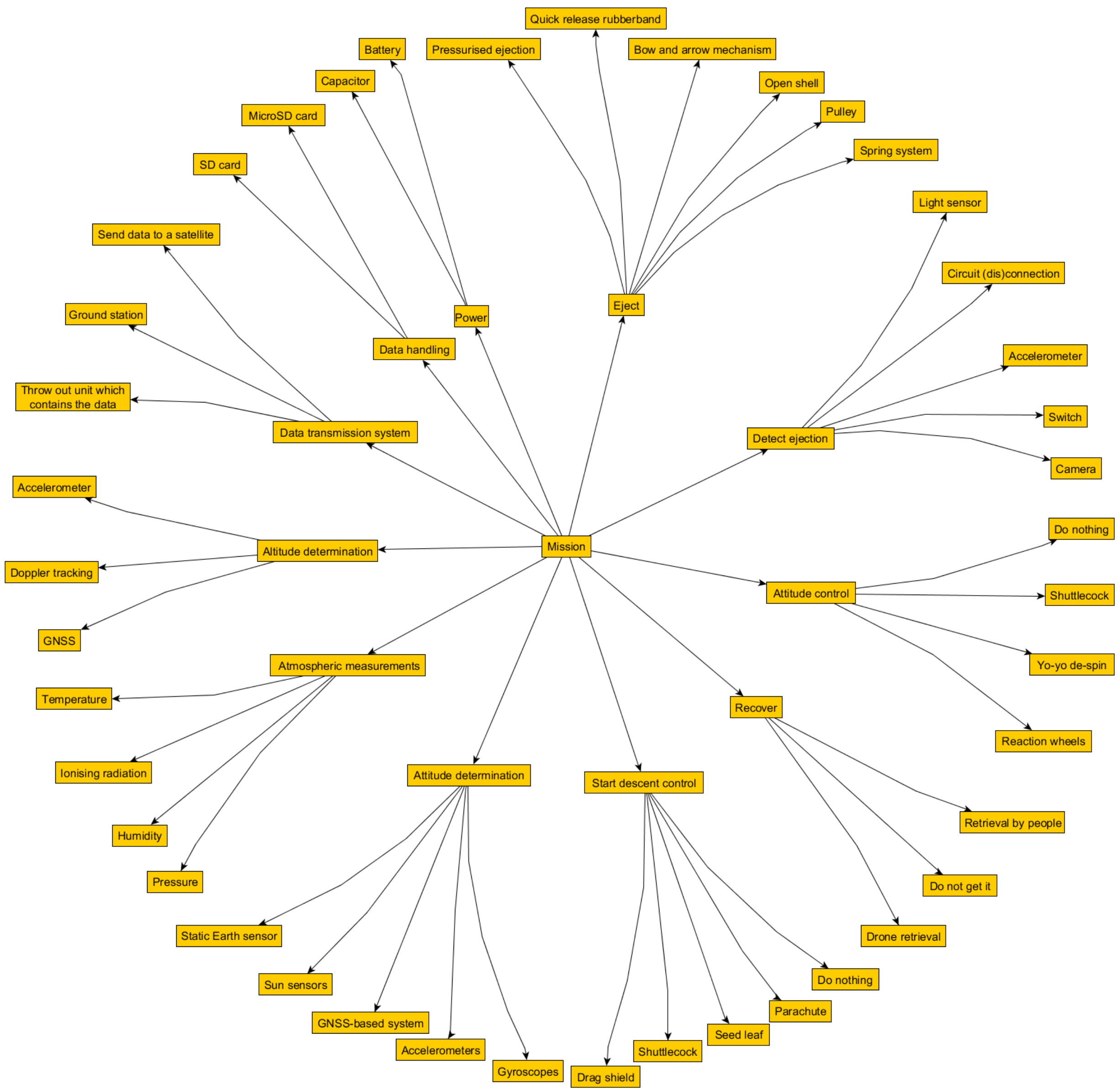


Figure 4.2: Design option tree.

The final decisions for the different functions of the TubeSat are given below. The filtering process can be read in the Midterm Report [24].

Power System
To provide power, it was decided to use batteries. From all commercially available components that were found to fit inside, batteries have the highest energy density. Also, as they are much smaller it is easier to increase or decrease the number of batteries used to adapt to what is required as the system changes. If necessary, they can also be distributed throughout the satellite to make them fit.

Satellite Ejection
For the satellite ejection, it was decided to use a timer-initiated spring system based on the design by TME, because that one has been proven and is ready for use. Developing an entirely new system would require more time to design and test the system. Since it was possible to fit all components of the design in the available space, it was not necessary to develop a different system. However, a small alteration was made, which is discussed in detail in Chapter 5.

Detection of Satellite Ejection
A breakable circuit is used to detect the ejection of the satellite out of the dart, because it is the simplest and smallest option. Although a light sensor is also simple and quite small, this component will give a signal as soon

as the dart cone is pushed off, not necessarily being the moment the satellite is ejected.

Altitude Determination
A Global Navigation Satellite System (GNSS) receiver is the preferred option for validation of the altitude. It is a widely used system that provides sufficient accuracy and is quick in use. The Global Positioning System (GPS) is preferred because it is the most common and complete option at the moment.

Attitude/Position Determination
To determine the behaviour of the dart during launch, the TubeSat determines the attitude and the position. The preferred option is a Miniature Attitude Heading Reference System (MAHRS), which consists of gyroscopes, accelerometers and magnetometers. Drift from integration is compensated by using Earth's gravity and magnetic fields as reference. However, an commercially off-the-shelf (COTS) MAHRS was found to be troublesome to integrate with the electronics because of the connector. Therefore, instead of buying a ready-to-use one, it was decided to buy and connect the components separately to obtain the same function. The data can then be sent back to the ground station to be interpreted later, as the satellite does not need it for its functions.

Attitude Control
As shown in Section 5.4, it was decided not to use active attitude control. However, to eliminate the spinning that may be left from the spin stabilisation, a yo-yo de-spin system is used, to be deployed right after ejection. The yo-yo de-spin system uses a pair of masses attached to ropes that are rolled up inside the TubeSat. After ejection, any rotation that is present will push the masses outward. Increasing the moment of inertia of the system, the satellite gets rid of its rotation. This is to avoid having to do data post-processing, if possible at all, which could lower the quality of the video.

Additionally, the Hyperflo drag device used for descent control, is in fact also responsible for attitude control, as it keeps the antenna in the satellite pointing downward to allow for communication with the ground station. More details about the Hyperflo can be found in Section 6.3.

Descent Control
The control method of the satellite descent was decided after more extensive trade-offs, which are discussed in Sections 4.2 and 4.3. The details will be discussed there. It was decided to use a drag device that does not necessarily produce a significant amount of drag, but will provide stability during satellite descent. The principle of providing stability is explained in Section 6.3. The device is shaped like a hyperflo parachute, but is made of rigid material and has a diameter which is equal to the satellite diameter.

Data Storage
For data storage, a microSD card is used. They are small and provide ample storage capacity and reading and writing speed.

Data Transmission
For the transmission of the data, a ground station will be used to send the data to.

Scientific Measurements
It was decided not to perform any additional scientific measurements, because the best position to put sensors is at the position of the camera, pointing out into the atmosphere. If, for example, a temperature sensor is located inside the satellite, it will measure the temperature of the inside of the satellite, which is influenced by the air that was already inside and the heat produced by the electronics. Moreover, it measures the temperature behind the shock wave, therefore not measuring the true air temperature. Note that the rear of the satellite is unavailable because of the drag device. What could be done is using the measurement devices already on board such. The accelerometers can be used to estimate and examine the atmosphere's density. The magnetometers can be used to check the earth's magnetic field. For future missions, scientific equipment can replace the camera, for more specific equipment.

Satellite Recovery
For the recovery of the TubeSat a combination of different methods is chosen. Firstly, the last known GPS location will be used for a very first estimate. Secondly, the TubeSat will be equipped with an additional transmitter(beacon) [18] which will send out a signal every few seconds. It was chosen to use an additional transmitter mainly because the main transmitter is very likely to break due to it being located in the nose of the TubeSat. This additional transmitter will be placed in the backside of the satellite in order to increase the chance of survival. Once the initial search area is known this signal can be used to locate the TubeSat from a distance of 500 meters (when obstructed by light vegetation) up to several kilometres (with clear line of sight[1]). This transmitter has a compatible receiver [17] which can be used by the search team. Although it is not sure if these components will survive the landing, they will greatly decrease the search time if they do.

SAFE/ARM switch
Note that the satellite does not have a SAFE/ARM switch. This is not required, because it is not a dangerous

[1] http://www.instructables.com/id/433-MHz-UHF-lost-model-radio-beacon/ [cited 27 June 2016]

system: it falls in the category of class B systems, which is explained in Chapter 16.

Overview
Table 4.1 lists the options that were chosen for all functions.

Table 4.1: Chosen design options.

Function	Design option
Power system	Battery
Satellite ejection	Spring system
Detection of satellite ejection	Circuit
Altitude determination	GPS receiver
Attitude/Position determination	Accelerometers, gyroscopes, magnetometers
Attitude control	Yo-yo de-spin system, hyperflo
Satellite descent control	Hyperflo
Data storage	MicroSD card
Data transmission	Ground station
Scientific measurements	-
Satellite recovery	Tracking device, people retrieval

4.2. Descent Options

First, a summary of the created descent options is given in the following list.

1. **PaSoS - Parachute Suborbital Satellite**
   This option uses reaction wheels to point the camera, after which a parachute controls the descent.
2. **DID - Dart in Dart**
   In this option, three parachutes will be used to slow down satellite descent. Additionally, the nose cone of the dart will be kept on for extra storage space and to act as a heat shield.
3. **DiLiH - Drop it Like it's Hot**
   This option does not use anything to control the satellite descent. Instead, the extra room available in the satellite is used to focus more on the transmission system.
4. **SPS - Satellite Pointing System**
   This system uses a parachute for the satellite descent as does the PaSoS. However, instead of reaction wheels to point the camera, the ejection is timed to occur when the dart is under the desired angle, using trajectory calculations made in advance. Note that the difference between the SPS and the PaSoS has become irrelevant after the decision was made not to use active pointing, as explained in Section 5.4.
5. **S^3 - Samara Suborbital Satellite**
   In this option the satellite will unfold a part of itself to obtain the shape of a Samara/maple seed leaf. This will slow down the satellite descent, as with its natural counterpart.
6. **DISC - Dart Initiated ShuttleCock**
   In this option the satellite claps out feathers of steel after its ejection to take a shuttlecock shape. This points the satellite in a certain orientation and slows it down slightly.

From the trade-off, it was decided to work out PaSoS, because it uses a parachute system which is a proven system and is easily produced. Also, it takes up an acceptable amount of space. It beats the SPS because that one has no active pointing. This could cause problems when deploying the parachute, because at the moment the required dynamic pressure is reached, the TubeSat may be in an unfavourable orientation which cannot be corrected. The S^3 would require more research to investigate its behaviour in practice. The DISC would take up quite some room due to the feathers around. These also require an unfolding mechanism which makes the design more complicated. Additionally, although the design resembles a shuttlecock, using feathers that clap out will leave openings in between them, and it is uncertain how that affects the performance. Some options had unacceptable issues: the DiLiH would become unstable and start tumbling during descent, which makes communicating with a ground station impossible. The DID may also have communications problems, because of the metal nose that is in front. Also, fitting three parachutes was later found to be most likely impossible.

4.3. Descent Concept Update

The design option with a parachute was revised considering two problems: from practical experience, TME found it difficult to fit a parachute in the small volume that is available. Secondly, the parachute was found to be proven to deploy successfully when opening at Mach 3, which could give problems with the opening shock. Although parachute systems were found that could fit inside the available volume without any problems, that did not take away the second issue. It was then decided to use a **mechanically deployed drag device**, that would separate immediately at satellite ejection. This drag device is not primarily used for slowing down, but for **stabilising** the TubeSat during descent, as discussed in Section 6.3. This decision was made based on the following findings:

- From the link budget presented in Chapter 7, it was found that descending without additional drag device would even provide enough time to transmit all data at least three times when the link budget is designed properly. Prerequisite is that the antenna is pointing down stably.
- In Section 6.2 it is shown that the heating will not cause the electronics to fail.
- Active attitude control was found to be no longer needed, as shown in Section 5.4. Reaction wheels were therefore omitted. However, this could cause a problem for parachute deployment, since satellite orientation at the moment of deployment may not be favourable and cannot be corrected.

4.3.1. Concepts

A number of concepts were created to function as drag device, all of which are suspended behind the TubeSat.

ShuttleChute

This drag device is shaped like a shuttlecock, and is inside the dart with the blades folded. When ejected, a spring system pulls the blades open after which they are locked in position, like an umbrella system. This is shown in Figure 4.3. Unfolding the blades creates large gaps in between them, which could be closed by a flexible material that upon folding is stored within the ShuttleChute.

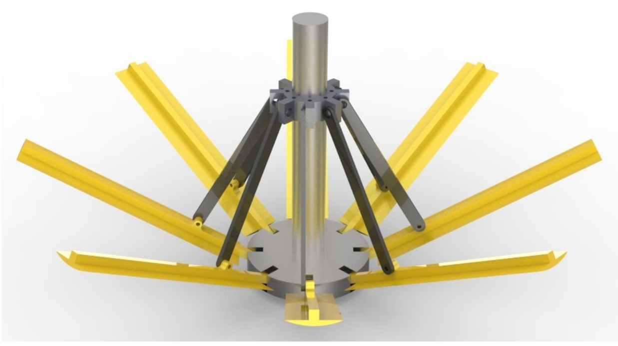


Figure 4.3: ShuttleChute concept render.

Advantage:
- Shuttlecock shape in itself is already stable.

Disadvantages:
- Mechanically complex, high failure risk.
- Many hinges, which are weak points prone to failure.
- Uses a lot of volume.

Veggie Steamer

The Veggie Steamer is a concept which is similar to the ShuttleChute, but uses the folding mechanism of a foldable vegetable steamer, as shown in Figure 4.4. The blades are interlocked in closed position and a different spring system pushes the blades open.

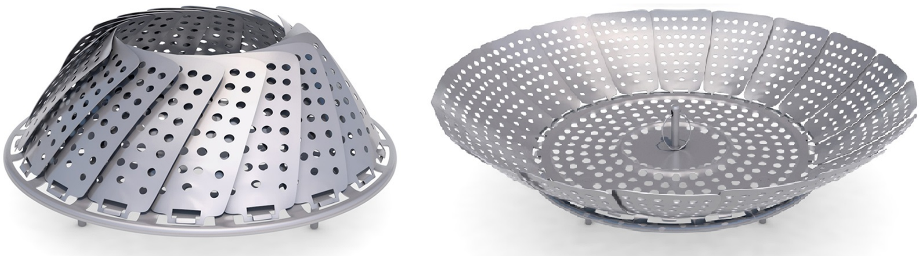


Figure 4.4: Foldable vegetable steamer mechanism.[2]

Advantage:
- No openings in between the blades.

Disadvantages:
- Mechanically complex, high failure risk.
- Many hinges, which are weak points prone to failure.

Reversed Veggie Steamer

The Reversed Veggie Steamer uses the same mechanism as the Veggie Steamer shown in Figure 4.4. The difference is that it is mounted upside down, like a rigid parachute.

Advantages:
- No openings in between the blades.
- Usable room inside the device.
- Higher drag.

Disadvantages:
- Mechanically complex, high failure risk.
- Many hinges, which are weak points prone to failure.

Hyperflo

The Hyperflo concept is based on the hyperflo parachute, which was developed for supersonic speeds and tested to be effective both subsonic as well as supersonic up to at least Mach 4.65 [50]. The hyperflo parachute is shaped like the cap of a plastic bottle, where the sides have a low porosity and the top has a high porosity. It does not need any deployment and already has the final shape a hyperflo parachute would have, by using a rigid material. The difference is however that the Hyperflo can at most have the same diameter as the TubeSat. A render is shown in Figure 4.5. (Please note that in this report Hyperflo is used to indicate the drag device in this project, and hyperflo refers to the parachute in general.)

Figure 4.5: Hyperflo concept render.

Advantages:
- Simple, no mechanisms.
- Usable room inside the device.
- Stable.

Disadvantage:
- Relative size may cause low effectiveness.

4.3.2. Concept Trade-Off

Trade-off Criteria

In the following list the criteria for the concept trade-off are given:

- **Volume**
  Although each leftover concept should be viable with regard to volume, a design with an easier fit would be desirable.

[2] The picture shows the "Stainless Steel Collapsible Vegetable Steamer" from `http://sunsella.com/products/stainless-vegetable-steamer/` [cited 3 June 2016]

- **Producibility**
  Easier production will lead to a shorter production time, which is desirable.
- **Performance**
  A higher performance is desirable, but this might go hand in hand with complexity.
- **Complexity**
  The higher the mechanical complexity of the deployment, the higher the risk of deployment failure, and/or weak points in the structure. This is undesired.

In the trade-off, favourable properties score highest. A high score for complexity thus means that it is not very complex and as such preferred.

Criteria Weight Factors

Weight factors for the criteria have been determined based on the relative importance in relation to each other. The weight factors including their reasoning can be seen in Table 4.2. Criteria with a higher weight factor, have a higher relative importance. Note that the weight factors are not used as multipliers, they are only there to determine better concepts in case of stalemates.

Table 4.2: Weight factors of trade-off criteria.

Criteria	Weight Factor	Reasons
Volume	3	Volume is one of the most important restrictions on the design, as the design asks for extreme compactness. This criteria will make or break the design, if the components cannot be fit, the mission cannot be performed.
Producibility	1	All of the concepts cannot be readily bought off-the-shelf and will have to be produced by TME themselves. The costs will not be high, but it will require time to design, produce and assemble the concept. However, this criteria is less important than volume, because it does not lead to mission failure.
Performance	2	A better performance will lead to a better mission outcome.
Complexity	3	High complexity gives a high failure risk of the mechanism. This could then make communication impossible due to instability, leading to mission failure. Therefore this criterion also has the highest weight.

Trade-Off

The trade-off of the concepts is given in Table 4.3, in which the widths of the columns represent their weights. The trade-off was done with a variation of the majority rule. Instead of giving points to the concepts, they were given grades for effectiveness per criteria, see Table 4.4. The concept with the most high scores is the best. If there were multiple concepts with the same amount of high scores, they were discussed by the group.

From the trade-off table it can be seen that the Hyperflo wins on three out of four criteria and is therefore the preferred concept. The concepts are ranked as follows:

1. Hyperflo
2. ShuttleChute
3. Reversed Veggie Steamer
4. Veggie Steamer

Because criteria and their weights are arbitrary, a sensitivity analysis for the trade-off was performed to check to what extent the decision was influenced by these subjective choices. First, it was checked what would happen if the scores were different. If all scores of the Hyperflo were lowered by one, it would have the highest score for two criteria, and still be the concept with the most highest scores. Because the scores were given based on argumentation, it was decided to be reasonable that the relative scores would not differ by more than one level.

The next step was varying the weights. However, it was immediately clear that it would not make any difference using this trade-off method, since the Hyperflo won the majority of the criteria.

Therefore, thirdly, also the trade-off method is checked. As another method, the four scores could be given grades 1 to 4 and then multiplied by the weights. However, the Hyperflo only has a lower score on performance. Therefore, the Hyperflo could only be beaten by the ShuttleChute by increasing the weight of the performance criterion to 9. This would give the Hyperflo a score of 46 and the ShuttleChute a score of 47. Although one could argue that performance should have the highest weight, it was deemed unreasonable that its weight would be three times as high as the highest weight of the rest. Therefore, the decision of choosing the Hyperflo was accepted.

Table 4.3: Descent concept trade-off table.

Concept \ Criteria	Volume	Producibility	Performance	Complexity
ShuttleChute	*yellow* Mechanism, and additional room to store cables.	*yellow* Challenge to design for all loads, many small parts.	*green* Stable in itself.	*red* Complex mechanism, weak points, small moving parts.
Veggie Steamer	*yellow* Mechanism, and additional room to store cables.	*blue* Slightly simpler than ShuttleChute.	*yellow* Not tested, needs some additional research.	*red* Complex mechanism, weak points, small moving parts.
Reversed Veggie Steamer	*yellow* Mechanism, and additional room to store cables.	*blue* Slightly simpler than ShuttleChute.	*blue* Not tested, needs some additional research, but basically a rigid parachute.	*red* Complex mechanism, weak points, small moving parts.
Hyperflo	*green* No mechanisms, room inside can be used to store cables.	*green* Easy to produce.	*yellow* Hyperflo proven, but not tested for this device-forebody diameter ratio.	*green* Simple design, no mechanisms.

Table 4.4: Legend of descent concept trade-off table.

Colour	Meaning
green	Excellent
blue	Good
yellow	Correctable
red	Bad

5
Design Details

This chapter discusses the details of the satellite design more in-depth. First, Section 5.1 shows an overview of all components that are used in the satellite. Then Section 5.2 explains how all components are fitted inside the outer shell of the satellite, of which the design is presented in Section 5.3. Next, the field of view of the camera is discussed in Section 5.4. The design of the ejection system of the satellite is then discussed in Section 5.5. Also, as the yo-yo de-spin system and the hyperflo have to be produced in-house, they also have to be designed. Their designs are shown in Sections 5.6 and 5.7, respectively. Finally, there are a number of issues to take care of at satellite ejection, so Section 5.8 describes how they are accounted for in the design, and what is expected to happen during ejection. In Section 5.9 recommendations are made on the detailed design of the satellite.

5.1. Overview of TubeSat Design

This section gives a short overview of all components inside the dart. At the rear end of the dart, the ejection system is located, as designed by TME and slightly shortened, as discussed in Chapter 4. On top of that, there is some protection for the satellite, discussed in more detail in Section 5.8. The actual satellite, with the Hyperflo, is behind that. An overview can be seen in Figure 5.1.

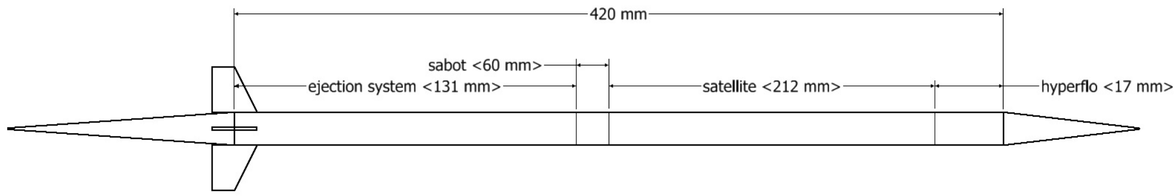


Figure 5.1: Overview of components inside the dart (not to scale).

5.2. Inner Layout

The inner layout of the satellite can be seen in Figure 5.2.

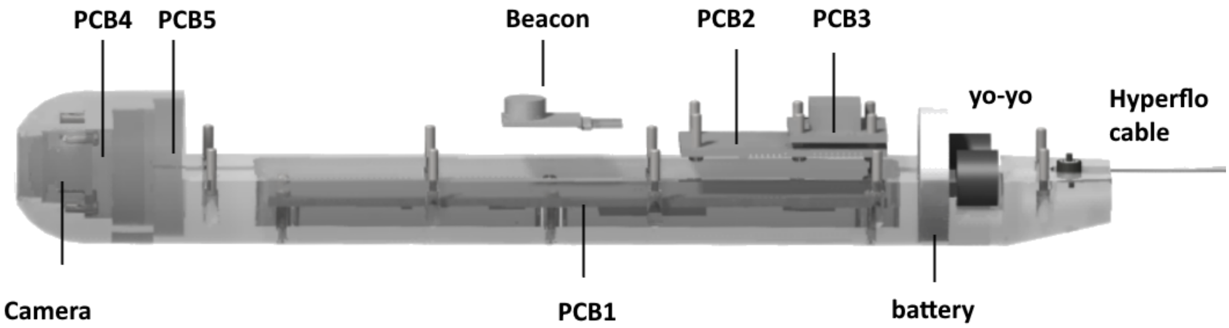


Figure 5.2: Overview of the layout of the satellite's inside.

As can be seen in Figure 5.2, the satellite (excluding the Hyperflo) is divided into four parts. At the front, there is a nose compartment where the camera and the antenna are placed. The camera is in the front, to look outside, and is attached to PCB4. The antenna is attached to PCB5 and is also placed in the front compartment. This is because the rest of the electronics are inside a Faraday cage to be protected from atmospheric electrical discharges. However, the antenna needs to be outside of the cage to be able to transmit the data.

The second compartment is where all electronics are located. PCB1 is the largest of the five PCBs and runs over almost the entire length of the compartment. Two smaller ones, PCB2 and PCB3, are to the side of it (on top of it on the figure), near the back of this compartment. To the front of the compartment the beacon module is located. PCB5, round shaped, is at the front end. PCB4 is not actually in this compartment, but attached directly to the camera. All of the electronic components are attached to these PCBs and are discussed in more detail in Chapter 8. Note that the PCB closest to the side wall (PCB3 in Figure 5.2) has the GPS chip attached on it, and there is a hole in the satellite wall in which the GPS chip fits exactly, such that the signals are not blocked. The same is done for the beacon system and an additional lead weight is added to make up for the loss of steel. At the rear end the battery is located.

This second compartment will also be filled with resin, which has a number of functions. First, it protects the electronics from vibrations, but also the small wires from vibrating loose. It is advantageous in terms of heating, in the sense that it works as a large heat sink: the aerodynamic heating will need to heat up all the resin, keeping the temperatures low for the electronics. On the other hand however, the electronics themselves also dissipate power, and it could give the problem that the heat generated cannot escape due to the resin, and still cause overheating. This is something that still needs to be investigated and can be done in the following way: assume that the entire volume is filled with resin and find the temperature increase by using the heat capacity of the resin and the total power of all electrical components. A better estimate can be retrieved when also including the aerodynamic heating.

Behind the second part, the yo-yo de-spin and Hyperflo attachment systems, which are parts three and four, respectively, are located. These are discussed in more detail in the next section.

5.3. Outer Shell

In this section the components of the outer shell are discussed, which includes the general layout, the nose cone and the rear. They will be presented in respective order.

5.3.1. General Layout

Figure 5.3 shows the outer shell. As can be seen there is a nose section and a body section, which in turn will be produced from two halves. The design of the nose cone is discussed in more detail in the next subsection. The body section has cavities on the inside for the electronics, with holes for attachment. At the rear end of the body, there are holes in the side wall for the yo-yo de-spin mechanism and a cavity for the cable of the Hyperflo.

The material chosen for the two halves of the TubeSat is steel. This material was chosen because of its combination of density, availability, cost and producibility. It is one of the more dense metals, which is not too expensive, easy to attain and producible. Between the outer shell and the dart, lubrication[1] will be used. Since both are made of steel, their friction coefficient is relatively high, which increases the risk of not ejecting properly.

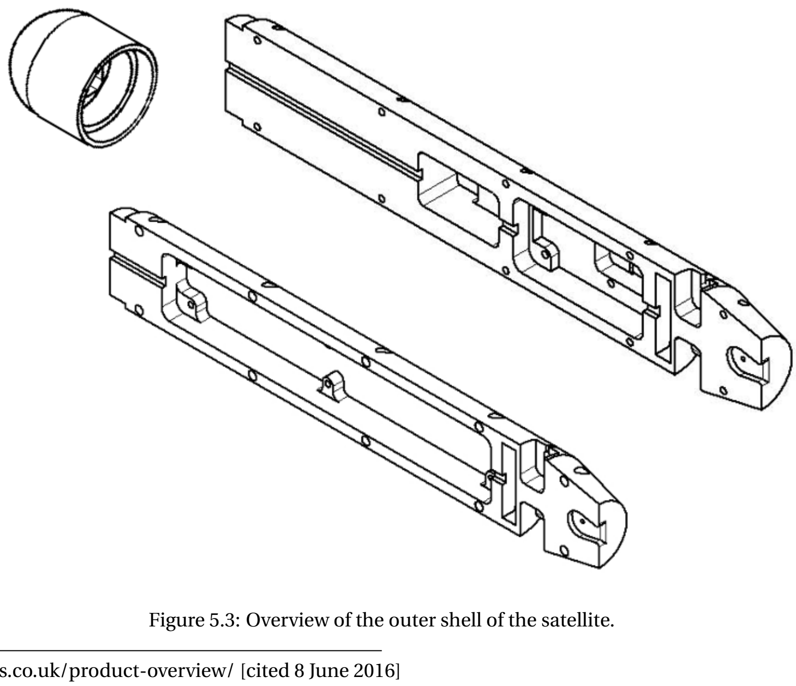


Figure 5.3: Overview of the outer shell of the satellite.

[1] http://antifrictioncoatings.co.uk/product-overview/ [cited 8 June 2016]

5.3.2. Nose Cone

For the design of the nose cone 5 factors were to be considered:

- Shape
- Interference with the transmission system
- Brittleness/Hardness of the material (launch forces)
- Angle of view of the camera
- Insulation against high temperatures due to re-entry

Shape
The shape of the nose cone is the only part of the design of the nose cone that can influence the aerodynamics of the entire satellite. According to [25], a hemispherical shape at the tip of a ballistic missile is advantageous for the heat transfer at the stagnation point. This is due to the bow shock that is created in front of the nose cone. The air loses energy over the shock wave. Part of this energy is transformed into heat, which does not reach the satellite.

Communication Interference
Due to the electric magnetic interference of metal, the nose cone needed to be made of a different material. Due to the attained temperatures at the stagnation point it was definitely not an option to place the transmission system at the front of the satellite. Note that even if the system could have been placed at the front of the satellite, the camera has priority as it is vital for the success of the mission.

Brittleness/Hardness
The material used for the nose cone needs to be able to withstand the forces due to launch. This means that according to the requirements, it will need to be able to withstand $100g$.

Angle of View of the Camera
To keep the required angle of view of the camera the material of which the nose cone is made, needs to be see-through but should not refract light at a different angle on the camera. Therefore the arced area in Figure 5.4 will have to be see-through, the determination of the angle of view can be found in Section 5.4.

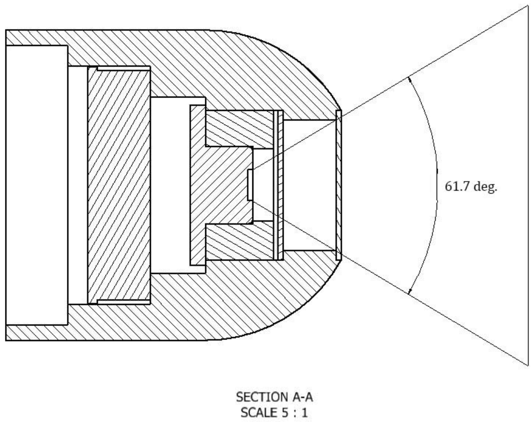


Figure 5.4: Angle of view of the camera.

Insulation
To make sure that the satellite is operable during the free fall a heat reduction system is needed, as a maximum temperature of 1500°C, determined in the simulation, is way too high for the electronics. To reduce the temperature to a 'safe' temperature of 85°C it was decided to use insulation.

Material Selection
It was decided to use a ceramic called Zirconia[2]. Although ceramics are brittle in nature, the assumption was made that the protective blocks, discussed in the next section, would be able to keep it unharmed during the launch. The ceramic was chosen because it does not conduct, is able to withstand the temperature and does not refract light.

To keep the required angle of view of the camera and to insulate the satellite sufficiently, it was decided that two flat quartz lenses were going to be used to create an air barrier/insulation. Quartz can withstand the heat and can be made to be see-through and air is a very good insulator.

See Section 6 for the calculations of the insulation.

[2]http://www.bce-special-ceramics.com/highperformanceceramics/zirconia/ [cited 7 June 2016]

Atmospheric Pressure
During the entire trajectory the nose will face large differences of pressure on the quartz lenses. The highest the pressure will be is during the descent because the quartz is the stagnation point. The geometry of the nosecone makes sure that the lens cannot move backwards. It can however pop out due to the vacuum at apogee.

The quartz lenses will be installed using an adhesive and will have sea level pressure in between the lenses. When considering that adhesives are best loaded in shear then the maximum shear load for the adhesive can be calculated using Equation 5.1.

$$\tau = P_{ambient}\frac{A_{pressure}}{A_{adhesive}} = 101kPa\frac{7.25mm^2 \times \pi}{0.5mm \times 14.5mm \times \pi} = 735kPa \tag{5.1}$$

When considering that the diameter of the quartz as $14.5mm$ and the thickness as $0.5mm$ then the shear stress is only $735kPa$. This is what most industrial adhesives can easily do.

Nose Cone Design

The final design of the nose cone can be seen in Figure 5.5.

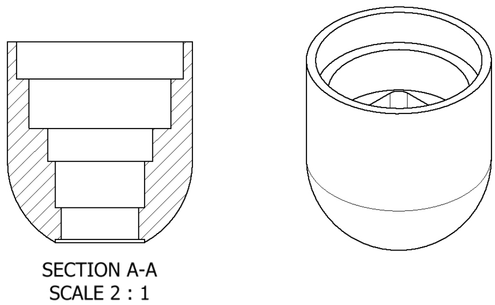


Figure 5.5: Nose cone design.

5.3.3. Rear Geometry

For the design of the rear, a number of considerations played a role. These are; the wake created which influences the Hyperflo and positioning of the Hyperflo on the rear. To investigate in detail the optimum shape of the rear to minimise the wake was considered outside the scope of this project. Instead, the assumption was made that the negative influence of the wake can be overcome by placing the Hyperflo sufficiently to the back, which is discussed in detail in Section 5.7. Also, it is assumed that giving the rear a boattail is more appropriate than just a flat rear, without exactly determining the effects of the shape. Therefore, the design of the rear was primarily driven by the positioning of the Hyperflo, and the space that was available.

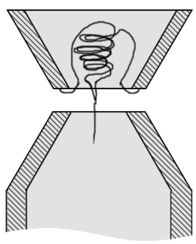


Figure 5.6: Shape of the satellite's rear, with Hyperflo positioned on top.

One possibility was to taper the rear such that the Hyperflo fits around the rear end, which saves space. However, with the space available it would only have been possible to stick the tail halfway inside the Hyperflo. This would give the risk that the Hyperflo and the rear get stuck to each other when the satellite is ejected from the DART. Therefore, it was decided to just position the Hyperflo on top of the rear, and make them the same diameter. A schematic view is shown in Figure 5.6. This provides enough room for the hole where the single cable for the Hyperflo comes out of the satellite. In addition, it also allows for the separation system that was designed for the

Hyperflo. The tapering starts immediately behind the yo-yo de-spin mechanism, to avoid having too steep of an angle.

To keep the Hyperflo from touching the main body of the satellite small blocks made of Sorbothane (elastic material), also known as sabots, were designed to be placed between the eight cables, so seven sabots in total. These sabots will also be used for the ejection of the Hyperflo, see Section 5.8.

Inside the rear of the satellite, a cavity is created to store the Hyperflo cable, as can be seen in Figure 5.7. In the cavity there is a bearing around which the cable is rolled up, and unrolls when the Hyperflo separates.

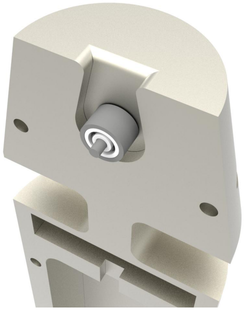


Figure 5.7: Cut-open view of the cavity in the satellite's rear for storing the Hyperflo cable.

5.4. Camera Field of View

T-Minus Engineering requires that the satellite will make a 30 second HD-video from space. This video should show that the satellite is ejected from the DART. Therefore the video starts recording just before the ejection. So then it shows how the satellite is ejected, shows the DART coasting in space, and finally the border between Earth and space. Since the available volume is limited, calculations were made to check whether a pointing system is necessary.

First a (not to scale) overview of the situation was made, as can be seen in Figure 5.8. With trigonometry it was found that the Earth's edge is 11° under the Earth surface tangential at 120 km altitude.

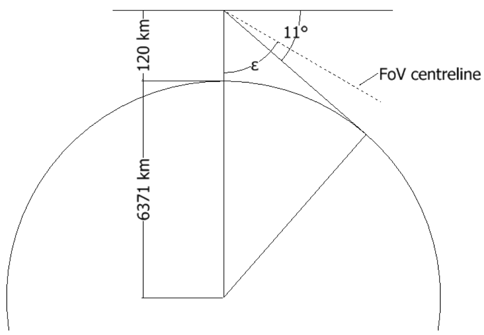


Figure 5.8: An overview of the view angles at apogee (drawing not to scale).

When one overlays the camera's angle of view, the minimum and maximum ejection angle ϵ can be found. This is the angle between the middle of the field of view and the nadir direction. The ejection angle is found with Equation 5.2.

$$\epsilon = 90 - 11 \pm \frac{AoV}{2} \tag{5.2}$$

where ϵ is the ejection angle in ° and AoV is the angle of view in °.

When a camera has an asymmetric field of view the smallest angle should be used. The chosen camera has a focal length of 2.99 mm and the sensor is 3.6 by 2.7 mm^3. With Equations 5.3 and 5.4 the horizontal and vertical angles of view can be calculated. These equations are derived from Figure 5.9.

[3]URL http://www.st.com/content/st_com/en/products/imaging-and-photonics-solutions/imaging-modules/vb6955cm.html [cited 6 June 2016]

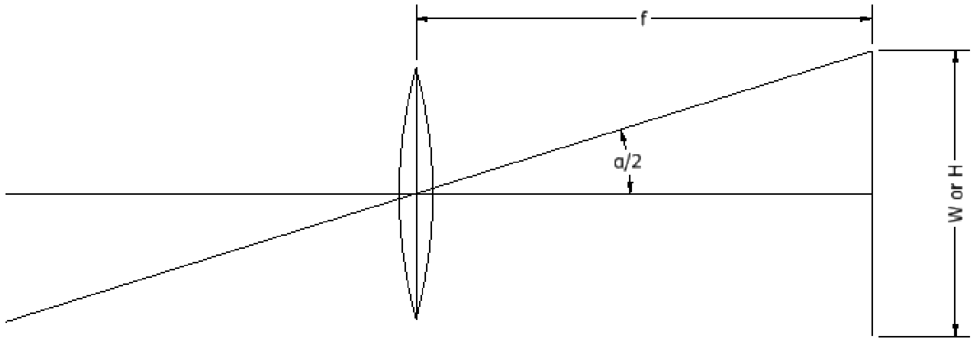


Figure 5.9: An overview of the lens-sensor system.

$$\alpha_h = 2 * \text{atan}\left(\frac{w}{2f}\right) \tag{5.3}$$

$$\alpha_v = 2 * \text{atan}\left(\frac{h}{2f}\right) \tag{5.4}$$

where α_h is the angle of view in horizontal direction in °, α_v is the angle of view in vertical direction in °, f is the focal length in mm, w is the width of the sensor in mm and h is the height of the sensor in mm.

When the values above are fed into Equations 5.3 and 5.4, it is found that the smallest angle of view is 51°. Which leads to a required ejection angle between 28° and 130°. The angle at which the DART is going to eject the satellite is between these angles according to T-Minus Engineering, so it is not needed to build an active pointing subsystem in the satellite. However a yo-yo de-spin system will be used to slow the rotation of the satellite to make the images clearer.

5.5. Ejection System

The current system uses a stiff spring, the impulse spring, to push loose the friction fitted nose cone, after which a long soft spring, expulsion spring, pushes the payload away gently. However, a small change was made in the design. It was decided to shorten the length of the longer spring, such that it does not extend outside of the DART. This way, the push is only guided, using the inner of the DART, and the satellite will not tumble due to centre of gravity (cg) offsets. A schematic overview of the ejection system is shown in Figure 5.10. In Subfigure (a) the design by TME is shown. The impulse spring is 55mm compressed, and the expulsion spring 65mm. The length that was left for the satellite was 250mm, so in total this is 370mm. The equilibrium lengths of the spring are 120mm for the impulse spring and 500mm for the expulsion spring, as shown in Subfigure (b). It can be seen that there is only 250mm of length inside the dart for the expulsion spring. It was decided to reduce the equilibrium length of the expulsion spring to 200mm, to also account for overshoot of the springs. This decreases the compressed length of the expulsion spring to 26mm, meaning that an additional 39mm is available for the satellite, totalling to 289mm. The properties of the spring are shown in Table 5.1. The total length of the ejection system including actuation is 131mm.

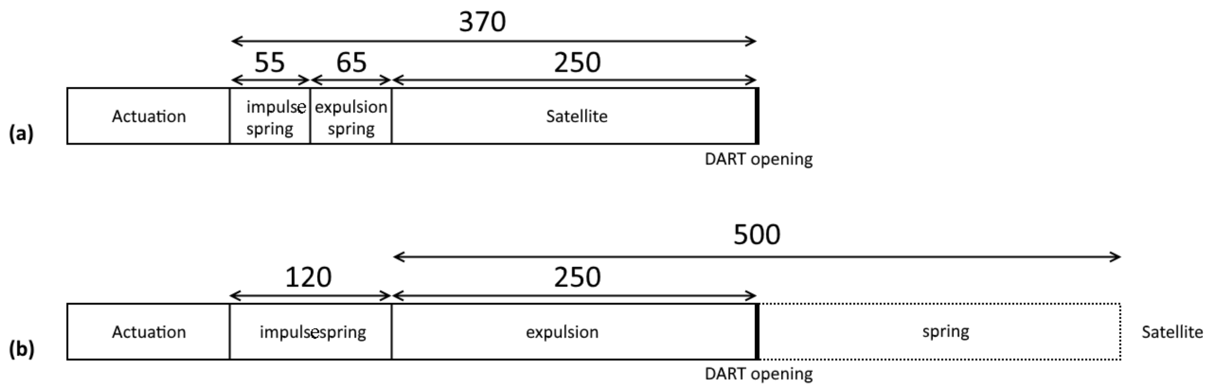


Figure 5.10: Spring dimensions in TMEs design.

Now that the spring does not extend outside the DART, there is also a risk of the opposite, that it fails to push out the satellite. To reduce this risk, this is another reason to use lubrication[4] between the inner of the DART and the outer shell of the satellite.

Table 5.1: Properties of the ejection system springs.

Spring	Spring constant [N/mm]	Length [mm]	Compressed length [mm]
Impulse spring	1.38	120	55
Expulsion spring	0.45	200	26

5.6. Yo-Yo De-Spin Mechanism

In this section the yo-yo de-spin mechanism will be shown. The masses of the yo-yo's and the maximum force in the ropes will be calculated. After choosing a material, also the dimensions of the yo-yo's will be determined. The calculations made for the yo-yo de-spin mechanism are based on [21].

The parameters of interest are the ratio $\frac{\omega_{final}}{\omega_{initial}}$, the mass of the yo-yo's, the length of the wires and the force in the wires.

Looking at Figure 5.11 the total kinetic energy of the de-spin situation is given by Equation 5.5.

$$T = \frac{1}{2} I\dot{\phi}^2 + \frac{1}{2} m \left[a^2\dot{\phi}^2 + 2al \cdot \cos(\theta - \gamma)\dot{\phi}\dot{\gamma} + l^2\dot{\gamma}^2 \right] \tag{5.5}$$

where:

- T is the total kinetic energy in J.
- I is the moment of inertia of the satellite including yo-yo's but excluding the Steiner term of the yo-yo's in $kg \cdot m^2$.
- $\dot{\phi}$ is the spin rate of the satellite in rad/s.
- m is the total mass of the yo-yo's in kg.
- a is the satellite radius in m.
- l is the length of the yo-yo rope in m.
- θ is the angle between the horizontal and the attachment point of the yo-yo rope in rad.
- γ is the angle between the horizontal and the yo-yo rope in rad.
- $\dot{\gamma}$ is the angular velocity of the yo-yo rope in rad/s.

Note that in this case $\dot{\theta} = \dot{\phi}$, because the rope length of the yo-yo's is constant.

The main point of interest is the point where the yo-yo's will release. The most reliable and widely used method for this is to simply have the rope 'slip off' ones $\theta - \gamma = 0$. A sketch of this method can be seen in Figure 5.12.

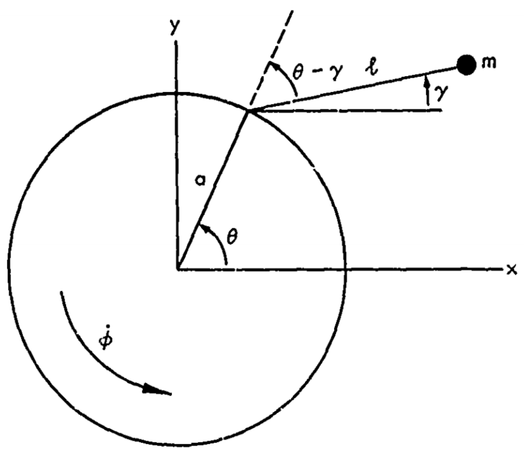


Figure 5.11: Sketch of yo-yo de-spin situation [21].

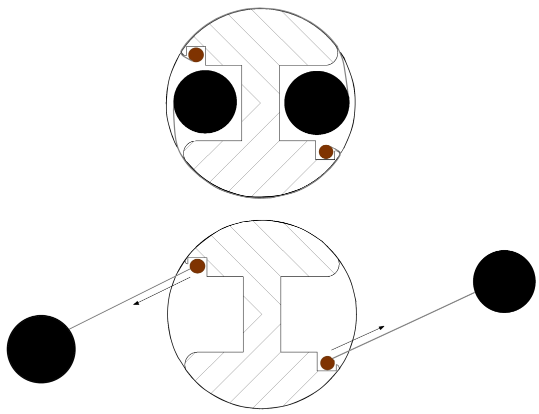


Figure 5.12: Sketch of yo-yo deployment method.

[4]http://antifrictioncoatings.co.uk/product-overview/ [cited 8 June 2016]

With $\theta - \gamma = 0$ the conservation of energy is shown by Equation 5.6.

$$\frac{1}{2}I\dot{\phi}_f^2 + \frac{1}{2}m\left(a\dot{\phi}_f + l\dot{\gamma}\right)^2 = \text{constant} = \frac{1}{2}\left(I + ma^2\right)\dot{\phi_i}^2 \tag{5.6}$$

where:

- $\dot{\phi}_f$ is the final spin rate of the satellite in rad/s.
- $\dot{\phi}_i$ is the initial spin rate of the satellite in rad/s.

The conservation of momentum is shown by Equation 5.7.

$$I\dot{\phi}_f + m(l+a)\left(a\dot{\phi}_f + l\dot{\gamma}\right) = \text{constant} = \left(I + ma^2\right)\dot{\phi}_i \tag{5.7}$$

Combining Equations 5.6 and 5.7 results in:

$$\frac{l}{a} + 1 = \frac{\frac{(1-r)I}{ma^2} + 1}{\sqrt{\frac{(1-r^2)I}{ma^2} + 1}} \tag{5.8}$$

where:

- r is the ratio $\frac{\dot{\phi}_f}{\dot{\phi}_i}$.

Rearranging Equation 5.8 leads to Equation 5.9.

$$r = \frac{\frac{(G+1)^2}{l/a+1)^2} - (G+1)}{G} \tag{5.9}$$

where:

- G is $(1-r)I/ma^2$.

Using Equation 5.9 the ratio r can be calculated for a given rope length and the total yo-yo mass. First look at the rope length. Wrapping the rope around the satellite can cause complications when unwinding, certainly when two yo-yo's would be used, a good value for the length of the rope therefore would be half of the circumference of the satellite. The yo-yo's would never be able to entangle. Moreover, as they both release at opposite sides they cannot collide. This eliminates the risk of one of the yo-yo's bouncing back to the satellite and damaging it.

Besides the mass of the yo-yo's the ratio r also depends on G which again depends on r, I and m. The wanted ratio for r is 0 as a total stop of rotation would be ideal. Thus an iterative process was used to find the needed mass of the yo-yo's, an r of zero and arbitrary yo-yo mass was filled in to find the I and G. Then the corresponding r was found. The yo-yo mass m was varied until the final value for r resulted in ± 0. The final yo-yo mass found is $24g$.

According to [21] the maximum force in the wire occurs at $\theta - \gamma = 0$, since that is when the angular velocity of the rope $\dot{\gamma}$ is maximum. The maximum force is given by Equation 5.10.

$$F_{max} = m\left(a\dot{\phi}^2 + l\dot{\gamma}^2\right) = m\dot{\phi_i}^2 l\left(\frac{ar^2}{l} + \frac{\dot{\gamma}^2}{\dot{\phi_i}^2}\right) \tag{5.10}$$

where:

- F_{max} is the maximum force occurring in the yo-yo rope in N.

Rewriting Equation 5.10 using the parameters G, r, l and a leads to Equation 5.11.

$$F_{max} = m\omega_i^2 l\left[\frac{ar^2}{l} + \left(\frac{G+1-r(l/a+1)}{(l/a+1)\,l/a}\right)^2\right] \tag{5.11}$$

where:

- ω_i is the initial spin rate of the satellite in rad/s.

As this is fully dependent on the initial spin rate of the satellite a precise estimate cannot be made. However, when checking the magnitude of the force for an extremely high spin rate of $100Hz$ the total force in the wire is only $0.20N$. Also it has to be kept in mind this is a single yo-yo on a single rope. Using two yo-yo's would half the maximum force making it even smaller.

The choice between one or two yo-yo's has to be made the choice is based mainly on previous mission involving yo-yo's. The common practise is to use two yo-yo's on either side of the satellite so that the forces acting on the satellite through the wires counter each other. Therefore two yo-yo's will be used on the Tubesat.

During the final design phase a volume of only $12x12x9mm$ (width x depth x height) was available for each yo-yo. As there needs to be space for the yo-yo to 'roll' out of the hole the maximum dimensions were set to be $11x12x8mm$. With a circular yo-yo with a diameter of $11mm$ this comes to a maximum volume of $0.69cm^3$. The mass of one yo-yo has to be at least $12g$. This gives a needed density of $13.5g/cm^3$. Not many materials have the density needed, some examples are gold and platinum. Eventually tungsten carbide was found which is often used in industries for cutting tools thus widely available and not as expensive as other heavy materials. It has a density of $15.6g/cm^3$ which brings the final dimensions of each yo-yo to a diameter of $11mm$ and a height of $8mm$.

The final parameters of each yo-yo are listed below:
- Mass of $12g$.
- Diameter of $11mm$.
- Height of $8mm$.
- Length of rope of $47mm$.
- Maximum force of $0.10N$ (in one rope).

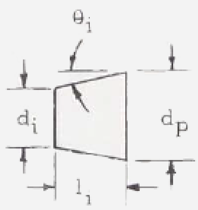
5.7. Hyperflo

The design of the Hyperflo is based on information from [50] and [51]. These papers elaborate on research that was performed to investigate the influence of certain parameters on the performance of the hyperflo parachute. These parameters include parachute geometry, porosity distribution, location of parachute relative to forebody, roof design and they were tested at varying Mach numbers up to 4.65. The general features that characterise a hyperflo parachute compared to a conventional one are [51, p. 8]:

1. Lower exit-to-inlet area ratio
2. At the inlet non-porous material is used
3. This results in a much lower total porosity, although the roof has a high porosity.

I Modified Guide Surface Ribless Types

Parachute Model	Suspension Line Length	θ_i (deg)	d_i/d_p	l_i/d_p	Roof Type	S_o/S_p (2)
A	1 d_m	20	0.7	0.41	(1)	2.64
C	1 d_m	20	0.5	0.69	(1)	5.45
D	1 d_m	20	0.35	0.89	(1)	4.09
E	2 d_m	11	0.5	1.29	(1)	5.85



(1) Standard guide surface ribless

(2) Ratio of total cloth area to maximum projected frontal area

Figure 5.13: Schematic overview of hyperflo geometries in the Phase I tests [51].

The overall geometry of the Hyperflo is based on design A, described in [51, p. 13]. An overview of this is given in Figure 5.13. This design has favourable properties for inlet inflation, roof inflation, drag and stability around suspension point. However, it features instability around its centre of gravity. Alternatively, design D could have been considered, as it has favourable characteristics for all of these points except for drag generation. Since the main purpose of the Hyperflo is to point the antenna downward as stable as possible, stability is the most important parameter. However, this issue due to the overall geometry could be repaired and is best done by reducing the effective area of the roof [50, p. 5]. It was found that the most effective way was by introducing a highly porous mesh around the periphery of the roof [51, p. 16]. Using a centre vent caused structural failure, and using an annular vent without a mesh caused canopy instabilities. The mesh proved to provide a satisfactory structural connection. [51, p. 17]. The best performing models had a high porosity mesh roof (15-35%) [50, p. 5], for the Hyperflo 25% was chosen.

The next design consideration is how far to suspend the Hyperflo behind the forebody. The turbulent wake created by the forebody can have a large impact on drag and stability of a trailing device. To avoid this effect, a rule of thumb is a distance of 8 to 9 times the forebody diameter [41]. The research in [51] also describes a test that investigates the effect of separation distance. From the tests it appeared the optimum canopy locations were 6 to 8 calibres behind the forebody [51, p. 25]. The close-in positions performed better with respect to stability, drag and inflation, although all tested positions performed well. Also, [32, p. 5-30] mentions "more than four times — preferably six — the forebody diameter". Because in this case the Hyperflo only has the same diameter as the forebody, it was decided to position the Hyperflo 8 calibres aft of the satellite. The considerations that were posed about only using the minimum needed distance to save weight are not relevant here.

Finally, the number of suspension lines used is 8. There should be enough to provide enough stability, but the number is limited due to the small volume, as well as the small area that is available to attach the lines to the canopy. From [32]: "The number of suspension lines should be a multiple of four, better of eight." Therefore it was chosen to use 8 suspension lines. Designing to carry the loads was not a main consideration. From the simulation in Section 6.2, the expected maximum tension was only 19 N. Therefore, it is expected that the cables have no problems carrying the load, even when including dynamic loads.

These suspension lines will be stored inside the Hyperflo, but in order not to cram everything tightly, part will also be stored inside the satellite. Only a single line is attached and stored in the satellite. The storage of this line was discussed in Section 5.2. This line covers a quarter of the distance, the eight attached to the Hyperflo cover the other three-quarters. At the location where they meet, they are connected using an eye-eye swivel. This is so that rotation of the Hyperflo will not cause the suspension lines to entangle.

As a final note it should be mentioned that all research is based on a situation where the hyperflo parachute has a larger diameter than the forebody, whereas this Hyperflo is only as wide as the satellite. In [50], though not clearly investigated, there were indications found that a parachute-to-forebody ratio of 3.33 had a slightly reduced performance compared to a ratio of 2.5. This could mean that a ratio of 1 would not have problems, however it is of course not reasonable to draw any conclusions here. It is therefore recommended to perform tests to see whether this ratio gives problems to the functioning of the Hyperflo.

Dimensioning
With all the theory behind the design known, it is time to put actual numbers to the design. An overview can be seen in Figure 5.15. As the Hyperflo should be as large as possible, the diameter of the roof is $29mm$. Using the ratio of $\frac{d_i}{d_p} = 0.7$, the inlet diameter is $20.3mm$; using the ratio of $\frac{l_i}{d_p} = 0.41$, the height is $11.9mm$. The thickness of the walls is $3mm$; eight equidistant holes go all the way through the walls, through which the suspension lines run and are attached.

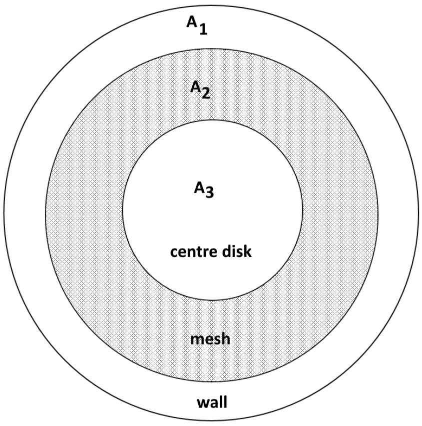


Figure 5.14: Top view of the Hyperflo roof.

For the design of the roof, a total roof porosity of 25% is used. The radius of the centre disk can be calculated using Equation 5.12, with A_i the area of the respective roof region and mp the mesh porosity. The different areas are as shown in Figure 5.14. The left hand side is the fraction of porous area to total roof area, and the right hand side is the resulting roof porosity. Expanding the expressions for the areas results in Equation 5.13 and it can be solved for the radius of the centre disk r_3. Assuming mesh porosity of 0.8 results in radius of the centre disk $r_3 = 8.5mm$.

$$\frac{mp \cdot A_2}{A_2 + A_3} = roofporosity \tag{5.12}$$

$$\frac{\pi\left(\frac{23}{2}\right)^2 - \pi r_3^2}{\pi\left(\frac{23}{2}\right)^2} = 0.25 \tag{5.13}$$

With a separation distance of 8 calibres, the Hyperflo will be 240mm behind the satellite. The swivel will be at 60mm behind the satellite. The effective length of the single cable is this 60mm minus the length added due to the swivel. The other eight cables each have an effective length of 180mm. Of course, the actual lengths will be longer for attachment.

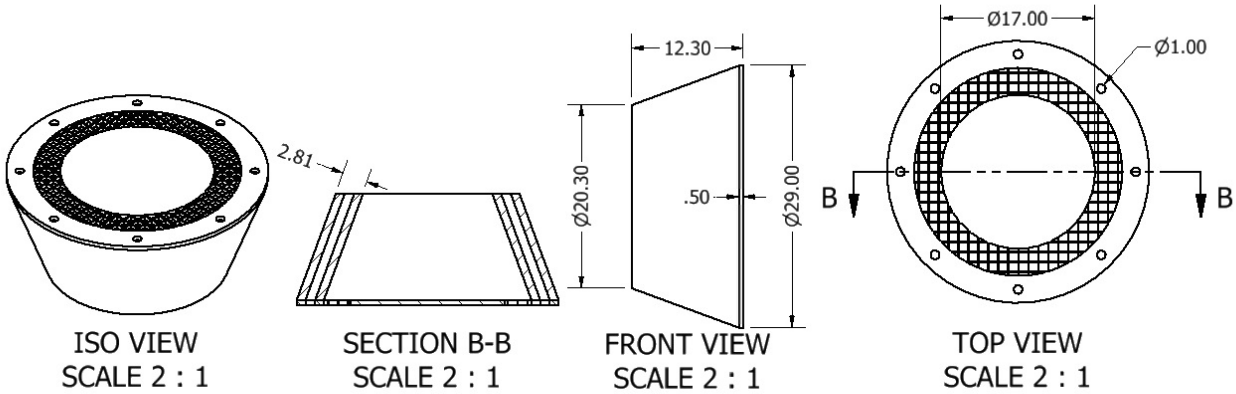


Figure 5.15: Technical drawing Hyperflo.

Material Selection
The material of the Hyperflo has to be able to withstand high temperatures due to aerodynamic heating. At the same time, it must give the Hyperflo a lower ballistic coefficient than the satellite. Because of the shape of the Hyperflo designed to provide drag, and the low mass compared to the forebody due to the size, it is assumed that the requirement on ballistic coefficient will be met without trouble. Therefore steel was chosen, to be able to withstand the temperatures.

5.8. Ejection

The overall ejection of the suborbital satellite will be discussed in this section, this includes the ejection from the DART and the ejection of the Hyperflo.

5.8.1. Ejection from the DART

The overview of the satellite, before launch can be seen in Figure 5.16.

Figure 5.16: Overview of TubeSat in the dart before launch.

From Figure 5.16, it can be seen that the nose cone of the satellite will be protected by a Sorbothane sabot. This sabot will have a thin ($2mm$) plate of steel in front of it to protect it from the spring at ejection. Note that all the sabots (including the ones protecting the Hyperflo) will be coated in anti-friction coating[5], to make the ejection as smooth as possible.

The spring designed by TME will eject the nose cone and push the TubeSat out of the dart.

5.8.2. Hyperflo Ejection

The spring system, to eject the satellite from the DART, pushes the nose cone loose, after which the satellite is pushed away. However, it is desired to also separate the Hyperflo from the satellite. Therefore, small blocks made of elastic material will be placed between the Hyperflo and the rear of the satellite, in between the 8 cables, so 7 blocks in total. They have a height 3mm more than what is available inside the DART. Then, when everything is loaded, they will be forced into compression. At ejection of the satellite, they spring back in un-deformed shape, pushing off the Hyperflo and the nose cone. A second important function of these blocks is that they provide for some room between the Hyperflo and the satellite. This protects the attachments of the cables to the Hyperflo, which otherwise have the risk that the cables snap.

In the design of the elastic blocks, it was also considered that they should not get stuck themselves in either the Hyperflo or the satellite. Therefore, they are designed in a T-shape, as shown in Figure 5.17. The figure also shows how the flanges will be deformed when forced between the Hyperflo and the satellite. When they are allowed to spring back, these flanges exert a force that pushes the T-shaped block outward, away from the satellite. The exact dimensions are shown in Chapter 12.

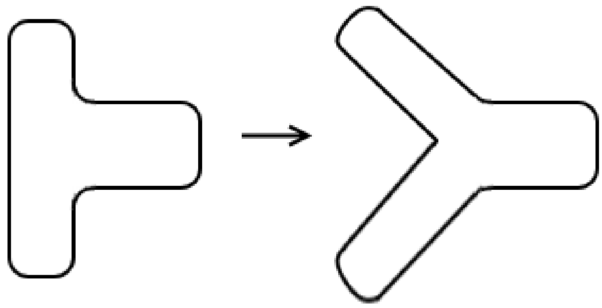


Figure 5.17: Hyperflo separation blocks.

As soon as the Hyperflo separates from the main body it will start to deploy the cables that attach it to the satellite. Due to the lower ballistic coefficient of the Hyperflo the lines will slowly deploy. This will continue until the Hyperflo is fully deployed.

5.9. Recommendations

The following lists the recommendations for further research of the topics in this chapter:

- The resin inside the satellite could give the problem that the heat dissipated by the components cannot escape, and cause overheating from the inside. This is something that still needs to be investigated and can be done the following way: Assume that the entire volume is filled with resin, and find the temperature increase by using the heat capacity of the resin and the total power of all the electric components. A better estimate can be retrieved when also including the aerodynamic heating.
- It is recommend to test whether the Hyperflo performs as expected, as it has a smaller drag-device to fore-body diameter ratio than usual. Performing a CFD analysis is an option since it may be difficult to test in real circumstances.
- A value for mesh porosity was just assumed without any reference. What still needs to be done is decide how the mesh exactly looks like, and what the porosity of the mesh really is. Then, the final radius of the centre disk and the total roof porosity can be found using Equation 5.12.

[5]http://antifrictioncoatings.co.uk/product-overview/ [cited 8 June 2016]

6 Trajectory Simulation

To estimate crucial design aspects like fall time and heat development, a simulation of the trajectory is made. This simulation consists of two parts: launch and re-entry. The launch simulation is used to calculate the apogee altitude and impact area. Additionally, it is a means of verifying the feasibility of a mission to 120 km altitude given the current booster/dart configuration. The re-entry simulation is the main focus of this chapter, and is used to calculate the fall time and (internal) temperature development. As such, it is an indispensable tool in the design of e.g. the external shape and the communication system.

This chapter is structured as follows. Section 6.1 covers the launch simulation. Section 6.2 covers the re-entry simulation. Section 6.3 covers stability. Section 6.4 states the assumptions made in this chapter. Section 6.5 states the recommendations for further research and development of the trajectory simulation.

6.1. Launch Simulation

This section is dedicated to explaining the launch simulation model. It covers the flight phases, dynamics model, drag model, wind influence, MATLAB code block diagram and simulation results. The atmospheric model and gravity model are explained in Section 6.2.

Flight Phases

The launch simulation distinguishes three phases of flight. During the first phase the rocket is guided by launch rails. It is characterised by acceleration in the direction of the launch rails, without influence of wind. The second phase commences the instant that the rocket leaves the launch rails and lasts until booster burnout. At the start of this phase the rocket changes direction into the wind, as described in the 'Wind Influence' subsection. The third and last phase begins at burnout and lasts until apogee.

Dynamics Model

The dynamics model is based on a simple approach: find the acceleration for every time step, and from that derive the distance travelled in both the horizontal and vertical direction (from hereon referred to as x- and y-direction). Based on the approach, this subsection is structured as follows. Force equilibrium is set up in x- and y- direction. From force equilibrium, Newton's second law is used to find the acceleration. Piecewise time integration then yields the velocity components. Finally, using the acceleration and velocity results from previous steps, a basic dynamic equation is used to find the distance travelled in x- and y-direction.

Acceleration & Velocity

The sum of the forces acting in x- and y-direction are given by Equations 6.1 and 6.2.

$$\sum F_x = \sum m a_x = (F_T - F_D)\cdot cos\theta = \left(F_T - \frac{1}{2}\rho v^2 C_D A\right)\cdot cos\theta \tag{6.1}$$

$$\sum F_y = \sum m a_y = (F_T - F_D)\cdot sin\theta - F_g = \left(F_T - \frac{1}{2}\rho v^2 C_D A\right)\cdot sin\theta - mg \tag{6.2}$$

where F_T, F_D and F_g are the thrust force, drag force and force of gravity, respectively, all in N. θ is the launch angle with respect to the ground. F_T is a fixed number (it varies with atmospheric conditions, but this is not considered here). F_D consists of the air density ρ in $kg\cdot m^{-3}$, velocity v in $m\cdot s^{-1}$, drag coefficient C_D and the frontal area A in m^2. F_g consists of the mass m in kg and the gravitational acceleration g in $m\cdot s^{-2}$. Equations 6.1 and 6.2 can be rewritten into Equations 6.3 and 6.4 to obtain the acceleration at any instant.

$$\sum a_x = \frac{\sum F_x}{m} = \frac{(F_T - F_D) \cdot cos\theta}{m} = \left(\frac{F_T}{m} - \frac{\frac{1}{2}\rho v^2 C_D A}{m}\right) \cdot cos\theta \quad (6.3)$$

$$\sum a_y = \frac{\sum F_y}{m} = \frac{(F_T - F_D) \cdot sin\theta - F_g}{m} = \left(\frac{F_T}{m} - \frac{\frac{1}{2}\rho v^2 C_D A}{m}\right) \cdot sin\theta - g \quad (6.4)$$

Note that the thrust acceleration term is only present during the boosted flight phase. More about this can be read in the 'Flight Phases' subsection. The velocity components are easily obtained by integrating Equations 6.3 and 6.4 over the time period during which the acceleration has acted.

$$v_x(t) = \int_0^t \left[\left(\frac{F_T}{m} - \frac{\frac{1}{2}\rho v^2 C_D A}{m}\right) \cdot cos\theta\right] dt \quad (6.5)$$

$$v_y(t) = \int_0^t \left[\left(\frac{F_T}{m} - \frac{\frac{1}{2}\rho v^2 C_D A}{m}\right) \cdot sin\theta - g\right] dt \quad (6.6)$$

$$v(t) = \sqrt{v_x(t)^2 + v_y(t)^2} \quad (6.7)$$

Equation 6.7 combines Equations 6.5 and 6.6, and is an analytical expression for the total velocity as a function of time. In the launch simulation, the time integration from Equations 6.5 and 6.6 is performed in a discrete number of steps. This is shown by Equation 6.8. The continuous time domain is split up into $\frac{t}{\Delta t}$ discrete time steps Δt.

$$v_{i+1} = v_i + \frac{dv_i}{dt}\Delta t \quad (6.8)$$

where v_i and v_{i+1} are the velocities in $m \cdot s^{-1}$ at times t and $(t + \Delta t)$, respectively, $\frac{dv_i}{dt}$ is the acceleration in $m \cdot s^{-2}$ at time t and Δt is the time step in s. Defining dv = a_i Δt, Equation 6.8 can be rewritten as Equation 6.9.

$$v_{i+1} = v_i + dv \quad (6.9)$$

which is an expression that a computer can easily work with. Using this numerical integration scheme, the x- and y- components of the velocity at time step $i+1$ are given by Equations 6.10 and 6.11.

$$v_{y,i+1} = v_{y,i} + dv_y \quad (6.10)$$

$$v_{x,i+1} = v_{x,i} + dv_x \quad (6.11)$$

Distance

The basic equation for the distance travelled by an object over a time period t, given an acceleration a acting on it and an initial velocity v_0, is given by Equation 6.12.

$$s = v_0 t + \frac{1}{2}at^2 \quad (6.12)$$

Splitting the continuous time domain into discrete time steps Δt, and identifying the horizontal and vertical travel as s and h, respectively, leads to Equations 6.13 and 6.14.

$$ds = v_{x,i}\Delta t + \frac{1}{2}\frac{dv_{x,i}}{dt}dt^2 = v_{x,i}\Delta t + \frac{1}{2}dv_x\Delta t \quad (6.13)$$

$$dh = v_{y,i}\Delta t + \frac{1}{2}\frac{dv_{y,i}}{dt}dt^2 = v_{y,i}\Delta t + \frac{1}{2}dv_y\Delta t \quad (6.14)$$

where ds and dh are the distances travelled in x- and y-direction over time step Δt, both in m. Using Equations 6.13 and 6.14, the explicit expressions in Equations 6.15 and 6.16 are obtained.

$$s_{i+1} = s_i + ds \tag{6.15}$$

$$h_{i+1} = h_i + dh \tag{6.16}$$

The basic dynamics model of the simulation has now been covered. More in-depth aspects will be covered in the subsequent sections.

Drag Model

The drag model used in the launch simulation is based on drag data provided by TME. Separate drag data is available for the boosted- and coasting configurations. For both configurations the drag coefficient is given as a function of the Mach number. Cubic spline interpolation is used to connect data points.

Wind Influence

One of the benefits of having a launch simulation, as opposed to just having a re-entry simulation, is the possibility of calculating the horizontal travel over the entire trajectory. This leads to much more accurate estimations of the impact area than would be the case for an isolated re-entry simulation. Estimating the horizontal travel is more than simply a matter of integrating the horizontal velocity over the flight time. Wind, manufacturing error and Earth rotation are all factors that contribute to deviation from the predicted trajectory. Of these factors, only the influence of wind is treated here. Text in this section is partially taken and adapted from [60]. Concepts and equations presented are derived for (sounding) rockets.

Due to horizontal wind a rocket will get an angle of attack and change its trajectory. There are two simplified approaches to account for the horizontal displacement as a result of wind influence; the rocket is either assumed to be pitch stabilised (constant pitch leading to transverse force resulting in sideways motion) or infinitely statically stable (angle of attack always zero by rotating rocket nose into the wind). Both will be briefly discussed here.

Under the pitch stabilised assumption the wind is a force that "pushes" the rocket sideways, without actually changing its orientation with respect to the wind. As such, the horizontal wind displacement follows directly from $\Delta X_{h,wind} = V_w t_f$, where V_w is the horizontal wind velocity and t_f is the flight time.

A somewhat more refined approach is the infinite static stability assumption. The rocket is assumed to have an infinitely large static stability (or infinitesimally small mass product of inertia) and as such the momentary angle of attack will always be equal to zero. Note that this approach only works for rockets that are given an initial velocity at launch (e.g. through guidance rails or launch tower), since otherwise the rocket would immediately re-orient itself towards the wind direction and fly away horizontally. Also note that the trajectory displacement resulting from this approach is considerably larger than from the pitch stabilised approach. At the end of the powered flight, the rocket will have acquired a horizontal velocity with respect to the wind that is maintained during the coasting flight that follows. For example, if a sounding rocket is launched vertically from a launch tower, reaching a velocity of 50 m/s at the end of the tower and meeting a horizontal wind velocity of 10 m/s, the rocket will continue its flight under an angle with respect to the horizontal wind of $90° - \arctan(10/50) = 78.7°$.

The launch simulation uses the infinite static stability assumption to calculate the angle under which the rocket leaves the launch rails.

Code Block Diagram

Figure 6.1 shows a code block diagram of the MATLAB launch simulation code. The block diagram is divided into 5 'lanes': import, variables, boosted, coasting and results. The 'import' and 'results' lanes are located outside of the calculations loop. The 'variables' loop consists of variables that are (in)directly related to the acceleration and are updated every iteration. The 'boosted' and 'coasting' lanes consist of calculating the distance travelled in horizontal and vertical directions for the boosted and coasting dart configurations, respectively.

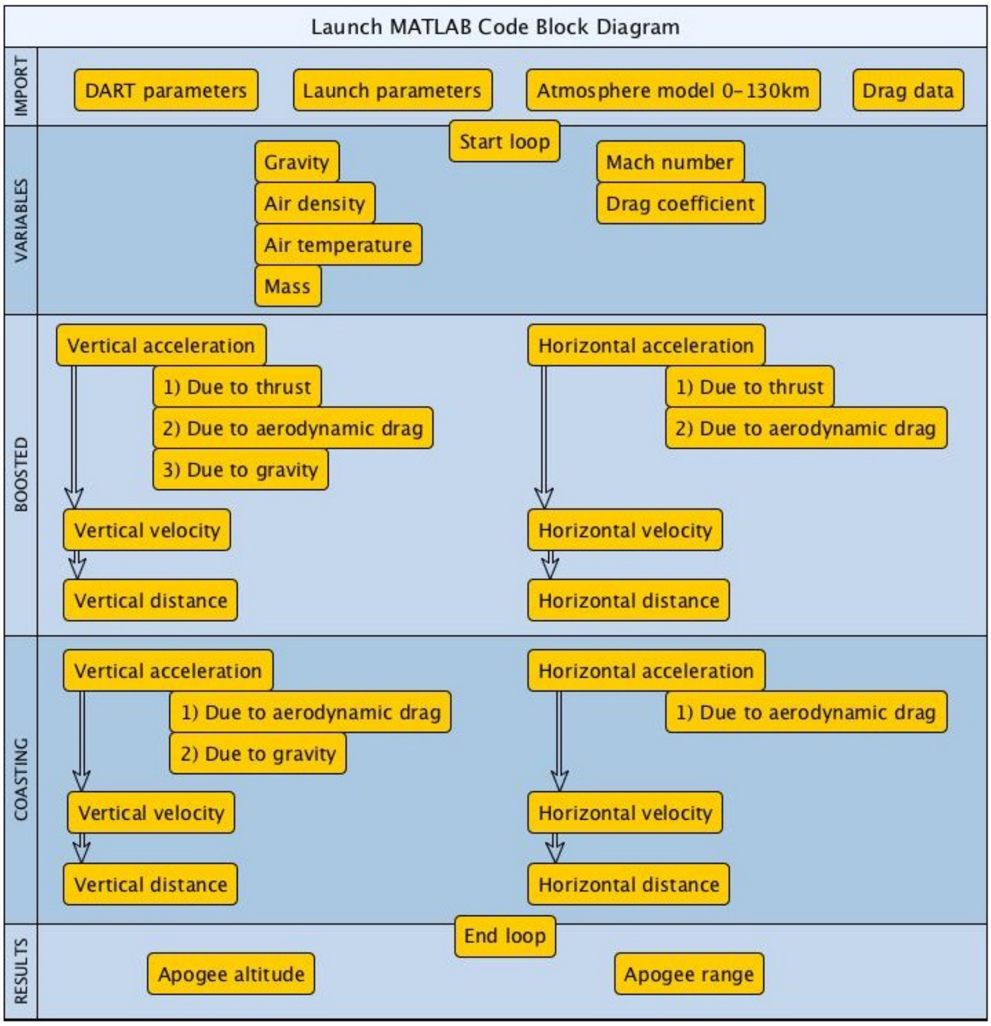


Figure 6.1: A block diagram of the MATLAB launch simulation code.

Results

An overview of the input and output parameters of the launch simulation is given in Table 6.1.

Table 6.1: Input and output parameters of launch simulation.

Input parameter		**Unit**	**Output parameter**		**Unit**
Rocket parameters			Phase 1		
Booster thrust	9967	N	Tower exit velocity	48.1	m/s
Burn time	5	s	Total flight time	0.165	s
System mass	(29.6+3.75)	kg	Phase 2		
			Tower exit angle	87.9	deg
Launch parameters			Burnout Mach number	5.96	[-]
Launch angle	88	deg	Burnout altitude	4250	m
Wind velocity	0	m/s	Total flight time	5.0	s
Tower length	4	m	Phase 3		
			Apogee altitude	132.1	km
			Total flight time	168.3	s
			Horizontal range		
			Total	19.4	km

Some comments about the results in Table 6.1:

- The altitude reached at apogee is 132.1 km. This is a 10.1% deviation from the intended 120 km. However, TME uses a conservatory factor of 10% on the drag data when calculating apogee. When also using this approach, the simulated apogee altitude equals 119 km as can be seen in Section 11.2. This shows that the launch simulation, and the apogee range associated with it, is accurate.
- The total horizontal travel is calculated as twice the horizontal travel during the launch trajectory. Since the re-entry is unpowered, it is likely that the actual horizontal travel will be lower. As such, 19.4 km is a conservative estimate of the impact area radius.

Figures 6.2-6.6 show the results in graphical format.

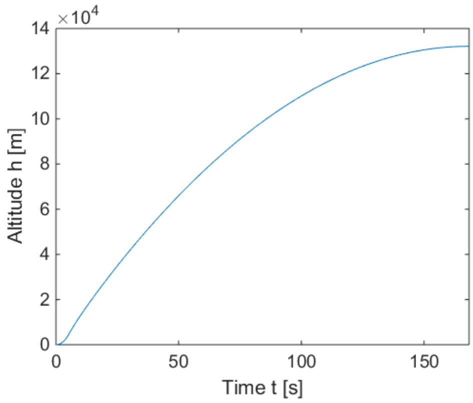


Figure 6.2: A plot of the altitude versus time during ascent.

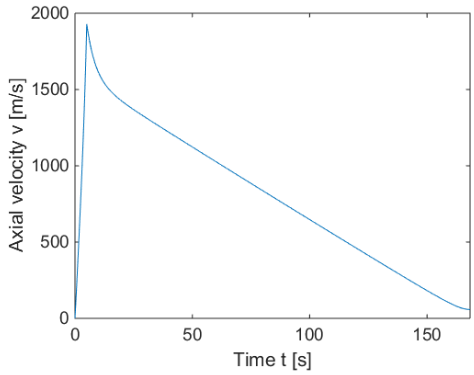


Figure 6.3: A plot of the axial velocity versus time during ascent.

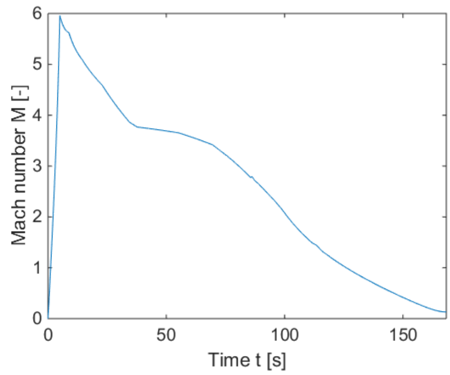


Figure 6.4: A plot of the Mach number versus time during ascent.

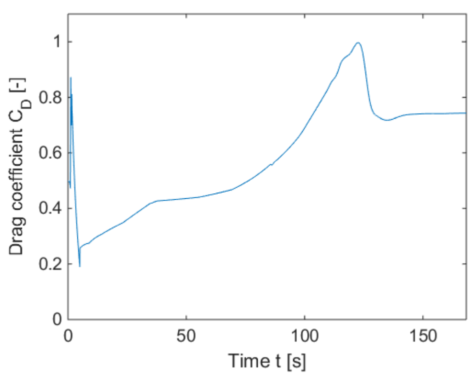


Figure 6.5: A plot of the drag coefficient versus time during ascent.

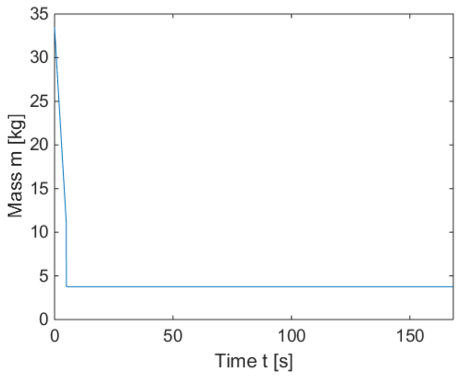


Figure 6.6: A plot of the rocket mass versus time during ascent.

6.2. Re-entry Simulation

After ejection from the DART, TubeSat begins its atmospheric re-entry. In this phase of flight, three factors are of special interest: fall time, attitude and heat development. The fall time is a decisive factor in determining how much time can be spent gathering data before transmission needs to be initiated. The attitude determines if a telecommunications link with the ground station can be established and if so, how efficient the data transfer process will be. Finally, heat development has a major influence on the design of the external shape and insulation, as well as the necessity and type of drag devices.

This section is dedicated to explaining the re-entry simulation model. It treats the dynamics model, atmospheric model, drag model, gravity model, heat model and satellite-parachute interaction, in addition to presenting a MATLAB code block diagram and simulation results.

Dynamics Model

The basic dynamics model of the re-entry simulation is the same as that of the launch simulation, and will not be repeated here.

Atmospheric Model

During re-entry, the TubeSat passes through several layers of the atmosphere: the thermosphere, mesosphere, stratosphere and troposphere. Atmospheric properties differ considerably per layer. Nowadays, there are many models around, each with their own focus and underlying mathematical model.

The atmospheric model used for trajectory simulations is the National Oceanic and Atmospheric Administration (NOAA) U.S. Standard Atmosphere 1976[1] [42]. Resulting from cooperation between the NOAA, NASA and USAF and published in 1976, this model describes the variation of atmospheric parameters such as temperature, pressure and air density up to altitudes of 1000km. In the trajectory simulation data points are connected using cubic splines. Some irregularities associated with cubic spline interpolation of the NOAA US76 model are discussed in the 'Results' subsection.

Drag Model

There are a number of problems associated with estimating the aerodynamic drag force. Not only does it vary with the Mach number, but it also strongly depends on the orientation of the satellite. Additionally, there is the problem of interaction between TubeSat and the hyperflo parachute, as can be read in Chapter 4. Consequently, an accurate estimation of the drag behaviour can only be made by CFD simulations and wind tunnel tests. However, to be able to proceed in the design process without unnecessary iterations an approximation of the drag behaviour must be made.

A rough estimate of the drag behaviour the system can be made by superposition; the total drag force is assumed to be the sum of the TubeSat and hyperflo parachute drag forces. The TubeSat drag coefficient is estimated using [22]. The hyperflo drag coefficient is found from [50]. Variation of drag coefficients with Mach number is included in the model. The effect of satellite orientation on the drag coefficient is not accounted for; this is a complex interaction that is part of more advanced design stages.

[22] presents simple, physics-based, closed-form analytical expressions for calculating the drag coefficient of slender bodies (missiles) for a variety of nose shapes. The drag coefficient is split up into three main contributors: wave drag, base drag and friction drag. Superposition is then used to find the total drag coefficient.

For M>1, the wave drag coefficient is calculated using Equation 6.17.

$$(C_{D_0})_{wave} = \frac{3.6}{\frac{l_n}{d}(M-1)+3} \tag{6.17}$$

where l_n is the length of the nose in m, d is the diameter in m and M is the Mach number. As such, the wave drag coefficient is a function of the nose fineness ratio $\frac{l_n}{d}$. For subsonic Mach numbers the wave drag coefficient is 0.

For M>1, the base drag coefficient is calculated using Equation 6.18.

$$(C_{D_0})_{base} = \frac{0.25}{M} \tag{6.18}$$

and for subsonic Mach numbers the base drag coefficient is calculated using Equation 6.19.

$$(C_{D_0})_{base} = 0.12 + 0.13M^2 \tag{6.19}$$

[1] Atmospheric model suggested by Dr. Ir. E. Mooij.

Finally, for all Mach numbers the friction drag coefficient is calculated using Equation 6.20.

$$(C_{D_0})_{friction} = 0.053\frac{l}{d}(\frac{M}{ql})^{0.2} \tag{6.20}$$

where l is the body length in m, d is the diameter in m and q is the dynamic pressure in Pa.

Some comments can be made about Equations 6.17-6.20:

- Assumptions are that the variation in the free stream speed of sound and viscosity with altitude is relatively small, and a turbulent boundary layer. Given the large altitude variations in the DART mission profile, this means that the predicted drag coefficient is inaccurate for a considerable part of the descent.
- In the simulation, the friction drag component is neglected above 20 kilometres altitude. This is done because the friction drag equation, Equation 6.20, has the air density in its denominator. As such, this drag term becomes extremely large for altitudes above ± 20 kilometres.
- Supersonic drag is dominated by the drag due to the shock wave on the nose.
- Body base drag can be a major contributor to the total drag during coasting flight, because of the low pressure in the base.
- Skin friction drag is a major contributor to subsonic drag.

[50] gives a relation between the drag coefficient and Mach number of hyperflo parachutes of varying mesh porosity and configuration. This relation is in essence a best-fit curve to data acquired from free flight tests and sled tests. To include the hyperflo drag characteristics in the simulation, data is read off from Figure 6.7 and data points are connected using cubic splines.

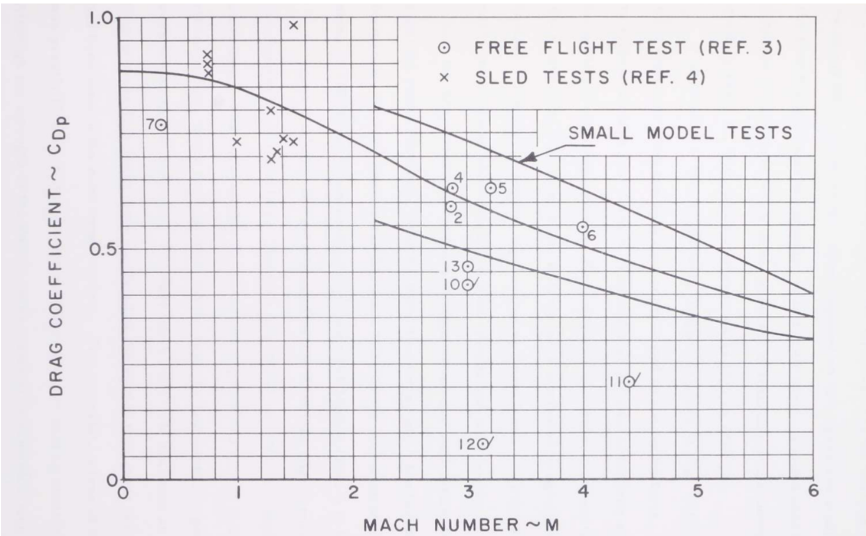


Figure 6.7: Variation of hyperflo drag coefficient with Mach number [50].

It will be necessary to test the final system in a high speed wind tunnel to obtain accurate aerodynamic data. This and more recommendations can be found in Section 6.5 and Chapter 21.

Gravity Model

The gravity model has been covered in the launch simulation section, and will not be repeated here.

Heat Model

As a consequence of the high velocities reached during re-entry, the heat development must be investigated. This is done using Chapman's Equation [40], as shown in Equation 6.21.

$$q_c = \frac{111 \cdot 10^8}{\sqrt{R_N}} \frac{\rho}{\rho_0}^{0.5} \frac{v}{V_C}^{3.15} \tag{6.21}$$

where q_c is the convective heat flux in W/m^2, R_N is the radius of curvature of the structure at stagnation point in m, ρ is the air density in kg/m^3, ρ_0 is the International Standard Atmosphere sea level air density in kg/m^3, v is the velocity in m/s and V_C is the orbital velocity in m/s.

Here, the orbital velocity V_c is assumed constant, as the altitude differences are negligible compared to the Earth's radius. Assuming thermal equilibrium, Equation 6.21 can be set equal to Equation 6.22 to find the wall temperature.

$$q_{rad} = \sigma \cdot \epsilon \cdot T_w^4 \tag{6.22}$$

where q_{rad} is the radiative heat flux in W/m^2, σ is the Stefan-Boltzmann constant $5.670367 \cdot 10^{-8}$ in $kgs^{-3}K^{-4}$, ϵ is the emissivity and T_w is the wall temperature in the stagnation point in K.

The value of ϵ is taken 0.92. This is the emissivity of glass[2]. The exact materials to be used are not yet known, but glass is used as an estimation, because there needs to be a transparent part in the structure for the camera to look out of.

Additionally, the conductive heat flux from Equation 6.21 is integrated to find the total heat energy that needs to be dealt with, as shown in Equation 6.23. This is important to know for designing how much energy a heat shield needs to cope with.

$$Q = \int_0^{t_{fall}} q_c \mathrm{d}t \tag{6.23}$$

where Q is the heat energy in W/s and t_{fall} is the fall time in s. For the conduction of the heat energy through the suborbital satellite, Equation 6.24 was used [26].

$$Q = kA\frac{T_H - T_C}{L} \tag{6.24}$$

where:

- Q is the heat energy in W/s.
- k is the thermal conductivity of the insulating material in $W/(m \cdot K)$.
- A is the area through which the heat energy is transferred in m^2.
- T_H and T_C are the hot and cold temperatures on either side of the insulation material in K.
- L is the thickness of the insulation material in m.

Equation 6.24 is then used in combination with the equation for specific heat [26], see Equation 6.25.

$$dT = \frac{Q}{c \cdot M} \tag{6.25}$$

where:

- c is the specific heat coefficient.
- M is the mass of the object heated in kg.
- dT is the temperature change in K.
- Q is the heat energy in W/s.

The temperature determined from Equation 6.23 is used to determine the heat energy transferred into the satellite. Note that Equations 6.24 and 6.25 can be used multiple times to determine the temperature in the TubeSat when multiple layers of insulation are used.

Satellite-Parachute Interaction

Given the presence of a stability device (hyperflo parachute), attached to the TubeSat by cables, the satellite-parachute interaction needs to be taken into account. A simple interaction model is set up by assuming the TubeSat and hyperflo parachute to be two individual point masses, on which three forces act: a gravity force (which pulls both down), a drag force (which pushes both up) and a cable force. The assumption is made that immediately after ejection the system of satellite plus parachute falls straight down, and the cables connecting the two are in stretched position throughout the fall. Given that the TubeSat has a higher ballistic coefficient, the cable force will pull up on the TubeSat and down on the parachute. Using Equations 6.26, 6.27 and 6.28, taking subscripts 1 for the parachute and 2 for the TubeSat, the cable force is calculated.

[2]URL http://support.fluke.com/find-sales/Download/Asset/3038318_6251_ENG_A_W.PDF [cited 18 May 2016]

$$a_1 = a_2 \tag{6.26}$$

$$\frac{F_{g_1} - F_{d_1} + F_{cable}}{m_1} = \frac{F_{g_2} - F_{d_2} - F_{cable}}{m_2} \tag{6.27}$$

$$F_{cable} = \frac{1}{\frac{1}{m_1} + \frac{1}{m_2}} \cdot (\frac{F_{g_2} - F_{d_2}}{m_2} + \frac{F_{d_1} - F_{g_1}}{m_1}) \tag{6.28}$$

For cable forces greater than zero, Equation 6.28 is added to the acceleration equations of the TubeSat and parachute, Equation **??** from Section 6.2. This can be seen in Equations 6.29 and 6.30 for the parachute and satellite, respectively.

$$a_{para} = g - \frac{\rho v^2 C_{D_{Para}} A_{Para}}{2m_{Para}} + \frac{F_{cable}}{m_{Para}} \tag{6.29}$$

$$a_{Sat} = g - \frac{\rho v^2 C_{D_{Sat}} A_{Sat}}{2m_{Sat}} - \frac{F_{cable}}{m_{Sat}} \tag{6.30}$$

As stated before, under normal circumstances (i.e. satellite has greater ballistic coefficient than parachute) the cable tension makes the parachute acceleration greater while making the satellite acceleration smaller.

Code Block Diagram

Figure 6.8 shows a code block diagram of the MATLAB re-entry simulation code. The block diagram is divided into 4 'lanes': import, variables, acceleration-velocity-altitude and results. The 'import' and 'results' lanes are located outside of the calculations loop. The 'variables' loop consists of variables that are (in)directly related to the acceleration and are updated every iteration. The 'acceleration-velocity-altitude' lane consists of calculating the distance travelled in horizontal and vertical directions.

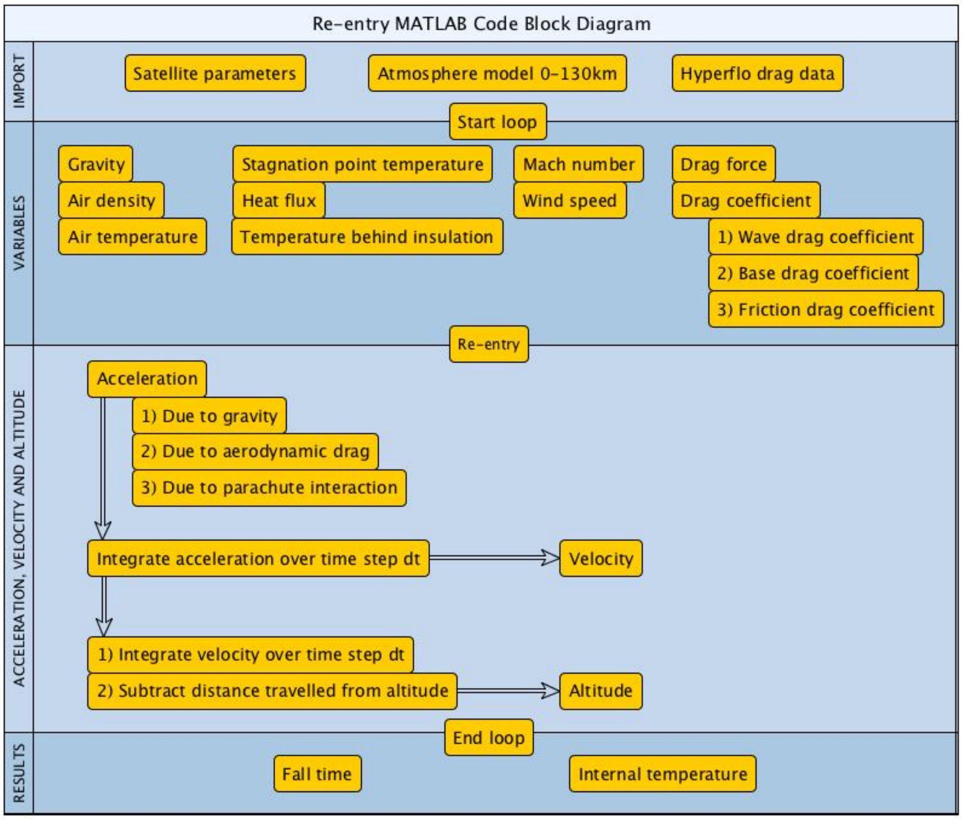


Figure 6.8: A block diagram of the MATLAB re-entry code.

Results

The simulation results, as well as some irregularities in the simulation graphs, will now be discussed.

Simulation Results

Figure 6.9 shows a plot of the altitude versus the time. Ground impact occurs 228 seconds after apogee. There is a notable change in descent velocity around $20km$ and $160s$, which corresponds to the dramatic increase in air density at that altitude seen in Figure 6.13. The increase in air density also leads to peaks in dynamic pressure, drag force and stagnation point temperature at that altitude, see Figures 6.14, 6.12 and 6.15, respectively. Additionally, the increase in air density leads to a rapid deceleration, which shows in the peaks in Mach number and vertical velocity, see Figures 6.11 and 6.10, respectively. An overview of some key numbers can be found in Table 6.2.

The insulation of the nose cone consists of two parts: a Zirconia shell and a quartz/air barrier. The temperature behind both insulation parts can be seen in Figure 6.16, the left graph shows the temperature behind the insulation of the zirconia shell, the right graph shows the temperature behind the quartz/air insulation. According to requirement **DART-Struc-Sys-2** the temperature in the TubeSat is not supposed to exceed 85°C. However, this limit is exceeded as can be seen in the zirconia graph in Figure 6.16, which is likely due to the assumption that the stagnation point temperature is the same temperature all over the frontal area, see Section 6.4. This assumption was made to make computations for the insulation easier, but in real life the temperature will be lower on the zirconia shell as the temperature decreases due to thermal radiation. Therefore the temperature behind the zirconia insulation will be lower then in the graph. Note that the max temperature is reached after 155 seconds, but the data will have been sent once after about 60 seconds. Note that the temperature used for the insulation calculations were assumed to be worse case.

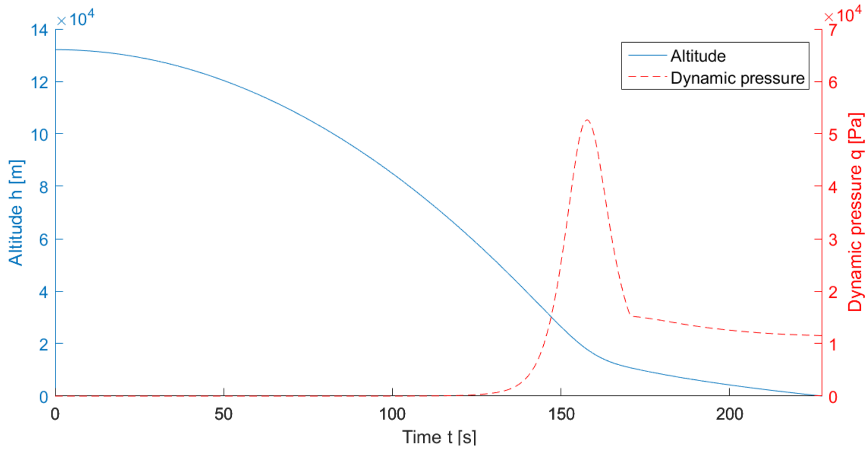


Figure 6.9: A plot of the altitude versus time during descent.

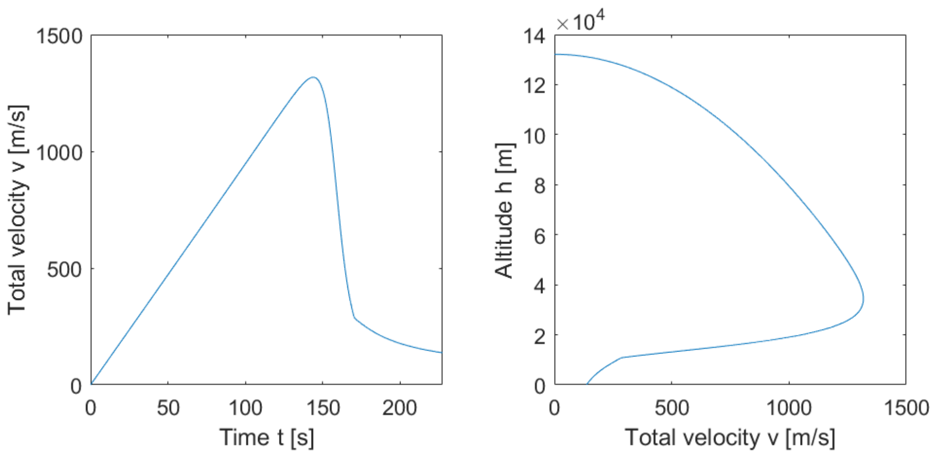


Figure 6.10: Plots of the velocity versus time and altitude during descent.

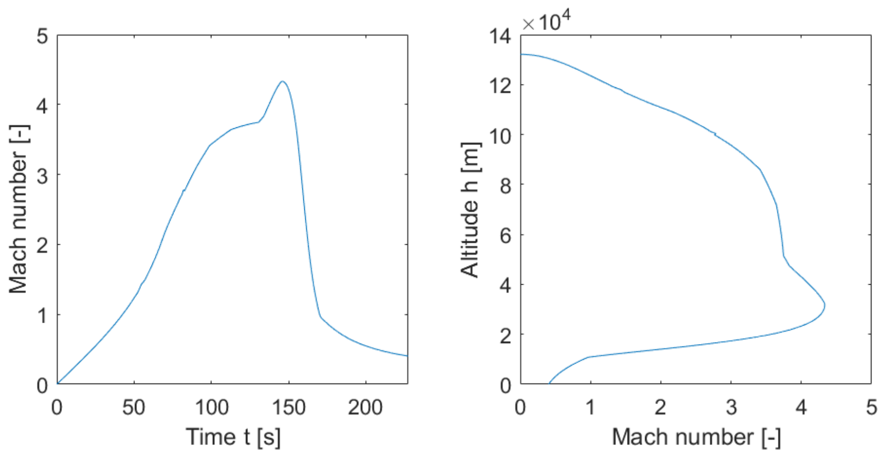


Figure 6.11: Plots of the Mach number versus time and altitude during descent.

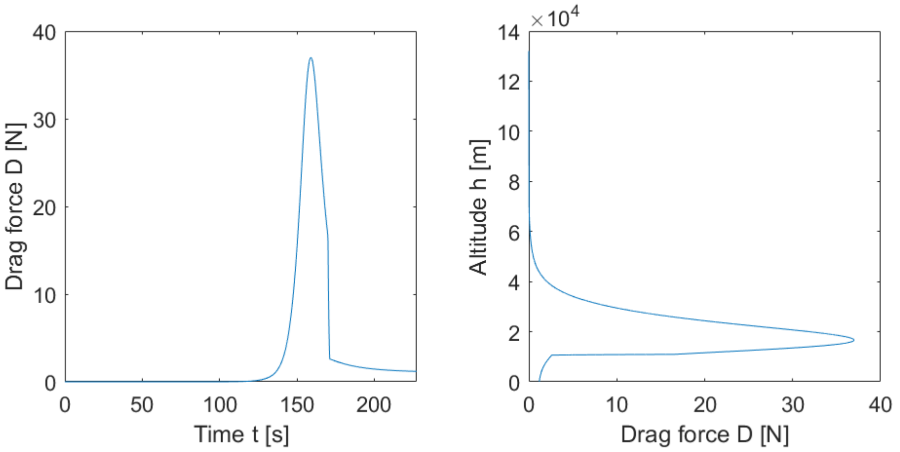


Figure 6.12: Plots of the satellite drag force versus time and altitude during descent.

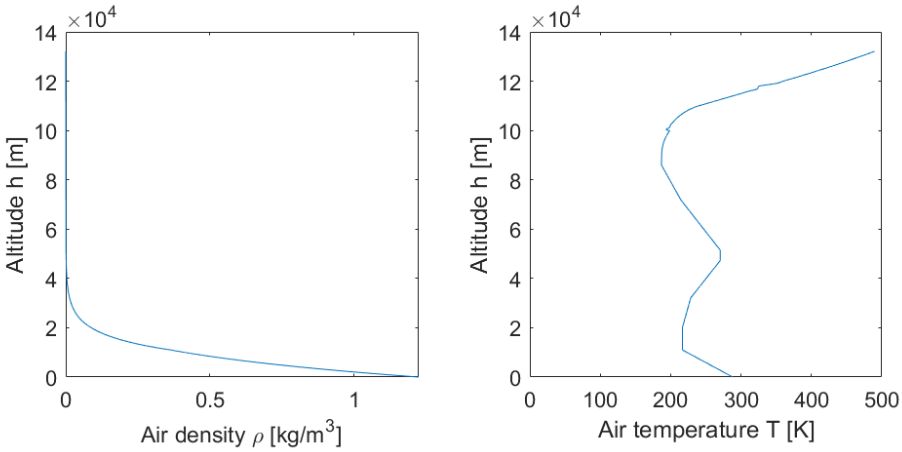


Figure 6.13: Plots of the air density and air temperature versus altitude.

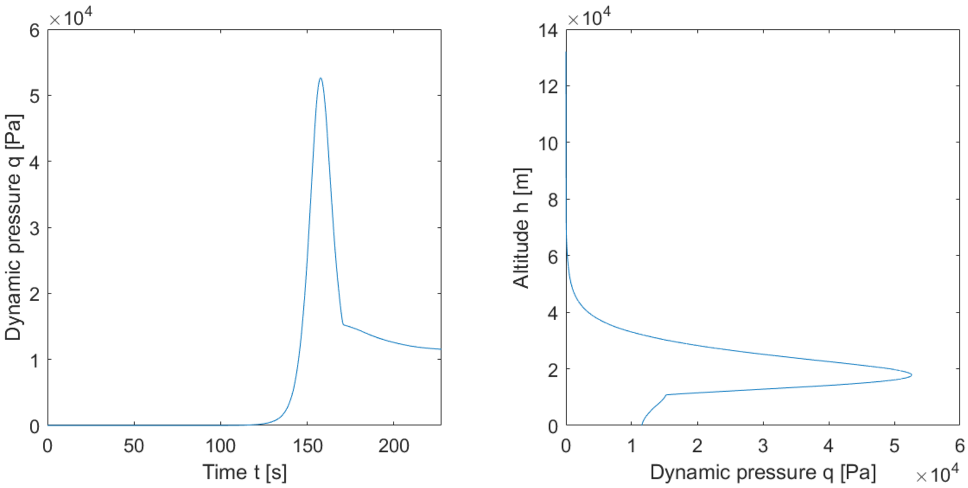


Figure 6.14: Plots of the dynamic pressure versus time and altitude.

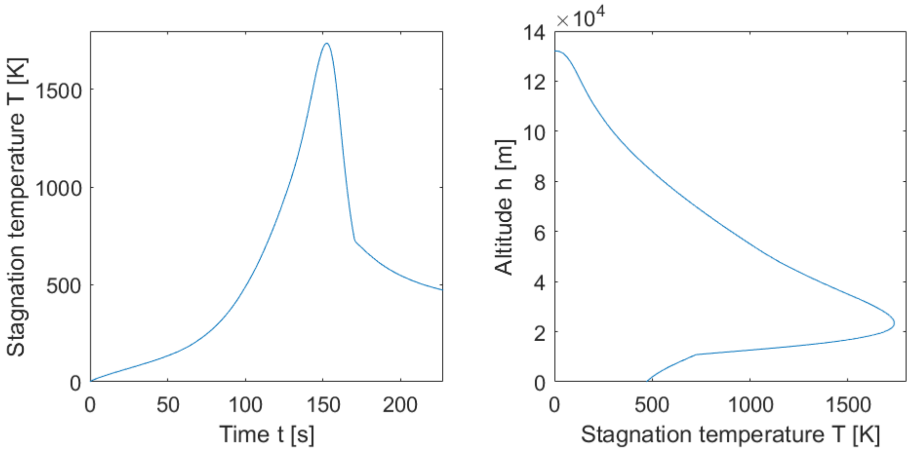


Figure 6.15: Plots of the stagnation point temperature versus time and altitude during descent.

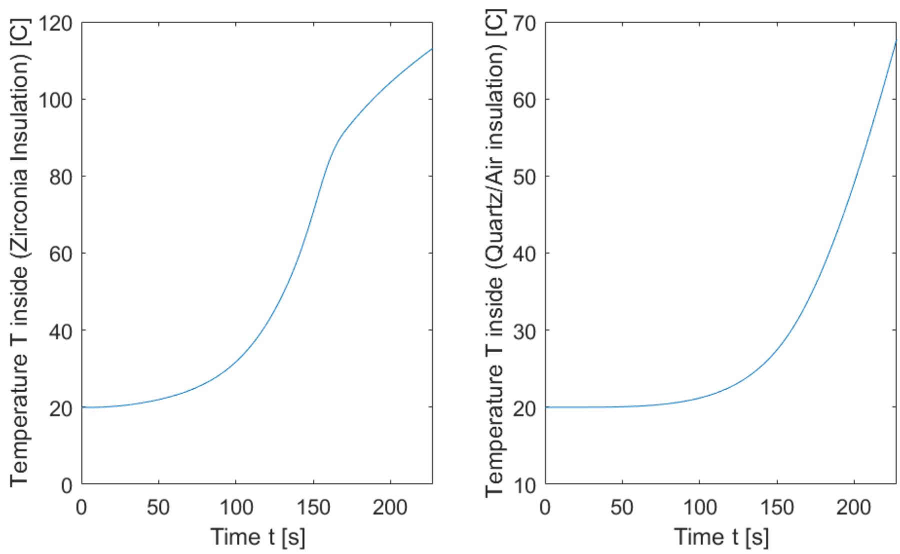


Figure 6.16: Plots of the temperature behind the insulation versus time during descent.

Table 6.2: An overview of the maximum magnitudes of descent parameters.

Parameter	**Value**	**Unit**
Fall time	228	s
Mach number	4.36	[-]
Velocity	1325	m/s
Stagnation point temperature	1746	K
Dynamic pressure	53374	N/m^2
Drag force	37.4	N

Irregularities

There are a few of irregularities in the result graphs. In the Mach graphs, Figure 6.11, there is a 'peak atop a peak' between 20 and 50 km. In the air temperature graph, Figure 6.13, there is an irregularity in the graph at 100 km altitude. Finally, in the dynamic pressure graphs, Figure 6.14, there is a sudden change around 160 seconds. These irregularities can be explained as follows.

The 'peak atop a peak' in the Mach graph corresponds to entering the stratosphere from the mesosphere, and the sudden change in temperature gradient that is associated with that. From Figure 6.13 it can readily be seen that between 20 and 50 altitude the air temperature drops quite rapidly. Such a drop in air temperature corresponds to a lower speed of sound, which explains the increase in Mach number. The irregularity in the air temperature graph at around 100 km altitude is explained by the data set that the graph is based on. From 0-100 km altitude, [42] gives atmospheric parameters such as the air density, temperature and pressure in steps of 100 meters. However, for altitudes above 100km the atmospheric parameters are given in steps of 1000 meters. Cubic splines are used to interpolate between data points, and this change in step size is the cause of the 'wobble' in Figure 6.13. The sudden change in the dynamic pressure graphs at 160 seconds is caused by the drag model. The drag coefficient of the satellite is the sum of three components: wave drag, friction drag and base drag. For subsonic Mach numbers the wave drag, which is a significant part of the drag coefficient, disappears. The drag force then becomes significantly lower (Figure 6.12) and the satellite decelerates much slower (Figure 6.10). As a consequence of the latter, the dynamic pressure doesn't drop rapidly after ± 160 seconds, but slowly reduces further until impact.

6.3. Stability

During the descent, stability is a very important consideration. The TubeSat needs to point down to allow for communication. Due to the restricted time of the DSE project, no simulation of the time response was made. Instead, an assumption is made on the behaviour of the system in terms of stability. It is expected that the system is kept stable similar to what happens with a parachute system. This was the most important reason why a design was chosen that had something trailing behind, instead of for example only dropping a shuttlecock. A sketch of possible situations is shown in Figure 6.17.

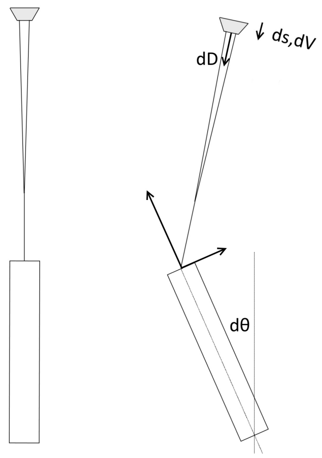


Figure 6.17: Tubesat and Hyperflo under a disturbance.

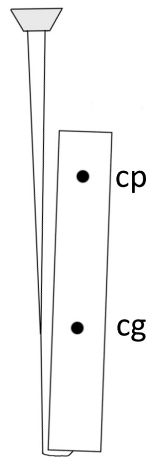


Figure 6.18: Tubesat with a cg higher than cp has risk of flipping around.

The working principle is based on the assumption that it works like a regular parachute, which is proven to keep the forebody stable and not tumbling. The drag device has a lower ballistic coefficient than the satellite. Therefore it falls slower than the satellite and pulls it vertical. When there is a disturbance $d\theta$, the direction of the tensile cable force tends to counteract the disturbance and pulls the satellite back into straight position. Also, the disturbance pulls the drag device an additional distance ds, which increases its velocity. This in turn increases the drag on the drag device. Therefore, the larger the deviation, the larger the counteracting force, which prevents the satellite from tumbling. Damping of this oscillatory motion is achieved due to the parachute lagging the forebody in this motion.

It can however still go wrong if the cg is above the centre of pressure (cp) of the satellite, which may lead to the situation as shown in Figure 6.18. In such a case, communications is again impossible. Therefore, the satellite was designed to have the cg as close to the nose as possible.

6.4. Assumptions

This section presents the assumptions used for the re-entry simulation.

The following list contains the assumptions and simplifications made in the simulation. The effects of assumptions are discussed as well. The list is ordered, in the sense that assumptions with a major effect on the results are listed on top.

- The trajectory simulations are two-dimensional.
- The satellite is assumed to be oriented nose down throughout the descent. In reality the angle of attack will change constantly due to atmospheric (side) winds, dynamic instability of the TubeSat, and non-symmetric

distribution of the satellite mass. The zero-lift assumption is a source of error for the descent time, heating and communications.

- The stagnation point temperature of the TubeSat was assumed to be present on the full frontal surface. This is a conservative assumption, as it assumes higher internal temperatures. Consequently, if the internal temperature (after insulation layers) is acceptable in the simulation, it will also be acceptable in reality.
- The atmospheric model used (NOAA US 1976) does not account for seasonal and latitudinal weather variations. In reality there could be variations in atmospheric parameters from hour to hour, and from location to location. Not incorporating these variations may introduce significant error in the model. Some parameters subject to seasonal and latitudinal change are the gravitational acceleration, air density and air temperature. System parameters like booster thrust level and aerodynamic drag are very sensitive to changing atmospheric conditions.
- The drag coefficient of the TubeSat is found from superposition of three components: wave drag, friction drag and base drag. These components, all of which are Mach number dependent, are found from missile literature [22]. The Mach-C_d relations used in [22] are based on data from a variety of supersonic missiles, and are not valid at high altitudes. As a consequence the drag calculations are wrong for a significant part of the flight envelope.
- The drag coefficient of the hyperflo parachute is found from literature (see Figure 6.7). The Mach-C_d relation is found from free fall and sled tests, none of which account for the fact that the parachute is located in the wake of an object. Since the diameter of the parachute is the same as that of the object in front of it, i.e. the TubeSat, it cannot be assumed that the hyperflo is not affected by wake flow. Consequently, error is introduced in the simulation by simply assuming the Mach-C_d relation from [50] to be true.
- For the conduction of heat energy, the assumption was made that the nose cone of the satellite is a flat panel. In reality the nose cone has a semi-sphere shape, which concentrates the heat energy in the centre, as opposed to a linear heat energy distribution for a flat panel. As such, the internal temperature will be higher in reality than under the flat panel assumption.
- The cables connecting the satellite to the parachute are assumed to be fully extended throughout the descent. In reality this is not the case, since it will take a while for aerodynamic drag to separate the two objects. This assumption does not account for 'shocks' in the cables due to the cables reaching their fully extended position after a relative acceleration, and as such the tension force in the cables will be larger in reality than in the simulation.
- The gravity field of Earth is not modelled: only altitude variations from a baseline value (9.8 $\frac{m}{s^2}$ at sea level) are included. Errors due to gravity offsets are relatively small, but still affect the fall time to a certain extent.
- Adiabatic heat transfer is assumed. Consequently the internal temperature due to heat transfer in the simulation will be higher than in reality.

6.5. Recommendations

The following list contains the recommendations for further research. The effects of assumptions are discussed as well. The list is ordered, in the sense that recommendations that are expected to have a major impact on the results are listed on top.

- The launch trajectory should be simulated in 3D. Although the current two-dimensional approach gives a general idea of the system behaviour, it is not possible to make accurate predictions about a real flight until three-dimensional aspects are accounted for.
- The aerodynamic behaviour of the design should be analysed using CFD or wind tunnel tests. The latter may also be used to validate CFD results.
- The complete aerodynamic behaviour as found from CFD/WTT should be implemented in the simulation. This includes dynamic stability, angles of attack changes, shifting of the centre of pressure with Mach number and satellite-parachute interaction. Including the full aerodynamic behaviour in the simulation allows the re-entry to be simulated under any set of circumstances such as, for example, a completely different launch site.
- If any instability is found, it should be researched whether lowering the cg of the satellite body has a beneficial effect. If so, alterations to the design must be considered, such as changing materials used to move the cg.
- The thermal behaviour of the design should be analysed using FEM. Schmidt's method for temperature development may be used [30].
- The impact area model should be improved. Attempts have been made to estimate the horizontal travel, but no useful results have been found (see Section 6.1). The model should include the effect of Earth rotation and wind speeds that vary with altitude.
- Seasonal, latitudinal and altitude variations of atmospheric (weather) conditions should be included in the model.
- Earth's gravity field at the launch site location should be modelled.

7
Communications

Transmitting the data is of utmost importance, since there is a chance the TubeSat will not be retrieved. Therefore a lot of research has been put in to the transmission system. A general layout of the transmission system is displayed in Section 7.1. The governing equations and procedure for the transmission system design are explained in Section 7.2. The program that has been written for the analysis of the link budget is explained in Section 7.3. The order of the data transmitted is explained in Section 7.4.The design of the transmission system using the available equipment at ESC is discussed in Section 7.5. The transmission system using a stand-alone system is elaborated on in Section 7.6. Conclusions are drawn and a trade-off is done between the stand-alone system and the ground station available at ESC in Section 7.7. Finally, recommendations for improvement and further research are listed in Section 7.8.

7.1. General Layout Communication System

The general lay-out of the communication system, which includes the transmitter and the receiver, is given in Figure 7.1 [11].

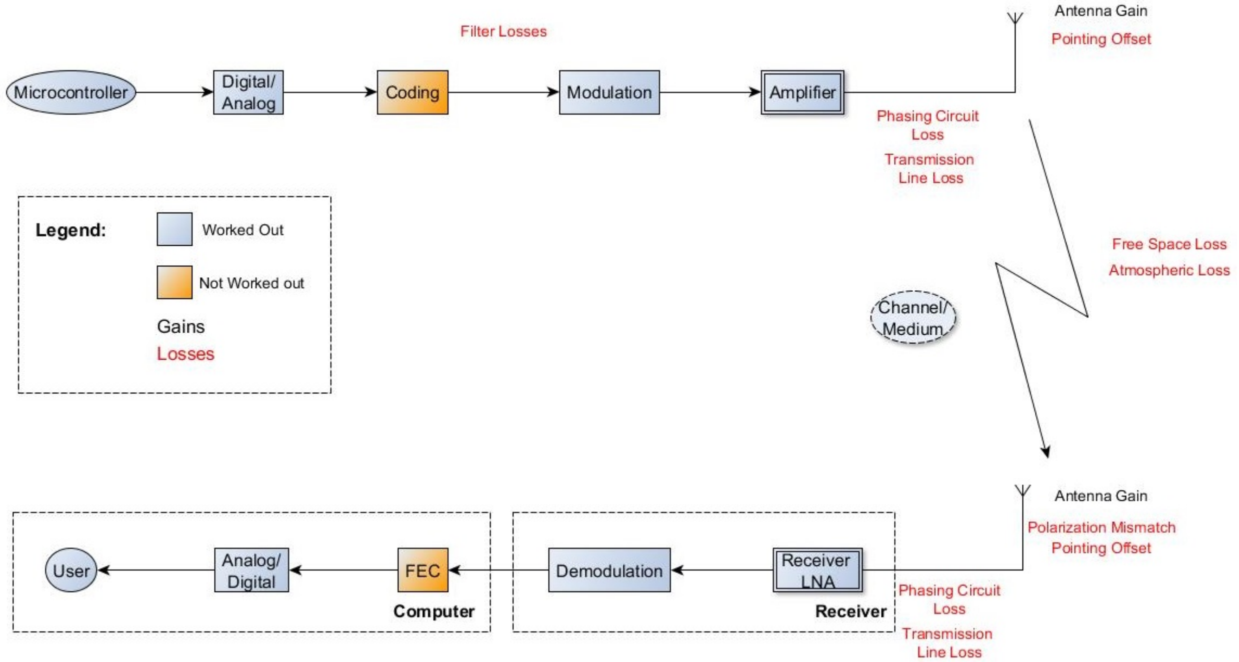


Figure 7.1: The general lay-out of a communication system.

As can be seen, no research has been done into encoding of the data, since as of now this has not been required. Moreover, the following parameters have not been analysed:

- Filter losses
- Transmission line losses
- Phasing Circuit losses

since not enough is known about the circuitry or the compatibility of equipment yet.
Two possible solutions for the transmission system have been worked out. The first one is a system that fully

depends on the equipment available at the launch site. In this analysis ESC is taken as reference using their User's Handbook [20]. The second one is a fully stand-alone system, which does not depend on any equipment provided by the launch site. This is an advantage, since this increases the freedom of the design of the communications system. Moreover, the stand-alone system will be made up of COTS components, which can decrease the cost of the operation when multiple launches have to be conducted.

7.2. Governing Equation and Procedure

In order to analyse the communication link References [14, 34] were used. The analysis of the communication link is performed using Equation 7.1.

$$SNR = \frac{E_b}{N_0} = \frac{P_t \cdot G_{AT} \cdot L_l \cdot G_{AR} \cdot L_r \cdot L_{pr}}{L_{fs} \cdot k \cdot T_{sys} \cdot R_b} \tag{7.1}$$

where:

- SNR is the signal-to-noise ratio.
- E_b is the received energy per bit in J.
- N_0 is the noise spectral density in J.
- P_t is the power of the transmitted signal in W.
- G_{AT} is the gain of the transmitting antenna.
- G_{AR} is the gain of the receiving antenna.
- L_l is the loss factor from transmitter to antenna.
- L_r is the loss factor from antenna to receiver.
- L_{pr} is the antenna pointing loss.
- k is the Boltzmann constant in $m^2 kg s^{-2} K^{-1}$.
- L_{fs} is the free space loss.
- T_{sys} is the noise temperature of the system in K.
- R_b is the data rate in bps.

Converting Equation 7.1 in decibels will result in equation 7.2.

$$SNR = E_b - N_0 = P_t + G_{AT} + L_l + G_{AR} + L_r + L_{pr} - L_{fs} - k - T_{sys} - R_b \tag{7.2}$$

The SNR influences the bit error rate (BER). Naturally, it is desired that the BER is as low as possible. The BER shows the probability that a single bit is sent wrongly. The BER also depends on the way the data will be coded/-modulated for sending.

For the design of the telecommunications of the TubeSat a number of steps are to be followed, using the equations mentioned before. The steps are as follows:

1. Determine the data generated due to the video and measurements.
2. Determine the descent time that is available to send the data to the ground station.
3. Determine the required data rate to transmit all the data.
4. Assume a BER which is sufficient enough to get a good video.
5. Choose a modulation. This will be based on the required SNR to get the desired BER.
6. Determine the required SNR for that specific modulation.
7. Choose the equipment that will be used in the transmission system.
8. Determine the values of all the parameters required to set up the link budget.
9. Check whether the link will provide a sufficient signal to noise ratio.

In the following sections steps 1 to 5 will be explained. Moreover, the parameters for the link budget which will be the same for both ground stations will be listed.

Data generated

In order to determine the data generated, the data rate and the operation time of each component is taken into account. These components can be found in Chapter 8. The results are shown in Table 7.1

Table 7.1: Data generated by the satellite.

Component	Data Rate (kbps)	Time (s)	Data Generated (MB)
Camera	5000	30	18.75
GPS	9.6	220	0.28
Gyroscope	2.0	390	0.10
Magnetic Sensor	0.4	390	0.02
Accelerometer (High g's)	25.6	390	1.25
Accelerometer (Low g's)	2.4	390	0.12

Everything has to be sent at least once, which comes down to 21 MB of data. For redundancy the data is required to be sent at least 3 times thus the total amount of data adds up to 63 MB.

Descent time

The descent time available of the current design will approximately be 230 seconds. Naturally, the data cannot be sent immediately, since it will take approximately 50 seconds to make the data transmission ready. So, only 180 seconds will be left to transmit the data. In the first 50 seconds, the 30-second video will be recorded.

Required BER and SNR

As the mission is to deliver a high quality video, the main design requirement for the communication link is the Bit Error Rate (BER). This determines the amount of errors in the transmitted data and thus the final quality of the video. The BER should not exceed 10^{-4} in order to maintain the quality of the video [31, 48]. However in the calculations made to set up the link budget a BER of 10^{-5} is used to make sure the quality of the video will not be diminished.

The required SNR will depend on the type of modulation used. BPSK and QPSK have the lowest required SNR of 9.6 dB in order to achieve a BER of at most 10^{-5} [14]. An additional minimum link margin of 3 dB is added to take into account uncertainties [34]. So, a SNR of at least 12.6 dB is required to send the video without having its quality decreased too much. Both the stand-alone system and the ESC equipment will need a SNR of at least 12.6 dB, since it is assumed both will use at least QPSK or BPSK modulation.

Parameters for the Link Budget

In this section, the parameters that will be the same for both ground stations are listed.

Pointing offset loss for the receiving antenna

Since a tracking system will be used, it is possible to fix the maximum pointing offset loss. The tracking equipment will make sure that the TubeSat will always be visible in the half power beam width, which will result in a maximum pointing offset loss of 3 dB.

Free Space Loss

The free space loss includes the losses due to the distance between the transmitter and the receiver and the atmospheric attenuation. The free space loss is given by Equation 7.3.

$$L_{fs} = 20 \cdot \log \frac{4\pi d}{\lambda} \qquad [dB] + L_a \tag{7.3}$$

where d is the absolute distance between the transmitter and the receiver in m and λ is the wavelength in m. L_a is the atmospheric loss due to rain. This is a constant which depends on the carrier frequency [34]. The free space loss will naturally change as the TubeSat will move along its trajectory. This has been taken into account in the calculation of the SNR.

Boltzmann constant

The Boltzmann constant will have a value of $1.3806 \cdot 10^{-23} m^2 kg s^{-2} K^{-1}$, which is approximately $-228.601 dB$ when converted.

7.3. Link Budget Analysis Program

A flow diagram of the Link Budget Analysis Program is shown in Figure 7.2.

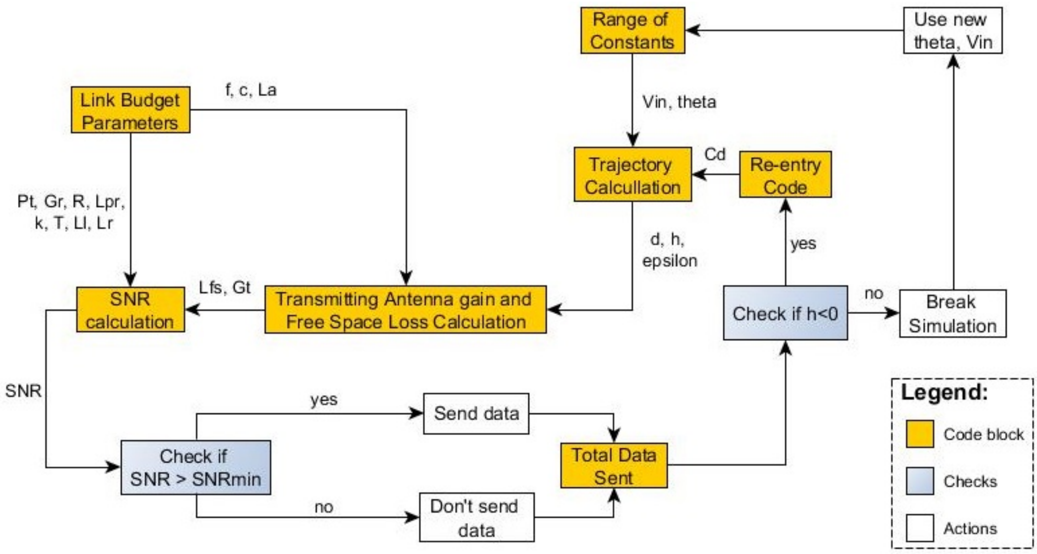


Figure 7.2: Flow diagram of the link budget analysis program.

The following assumptions have been made for the program:

1. **Ejection at apogee.**
   For the simulation, it is assumed that the TubeSat will only eject at apogee. It does not account for the possibility that the TubeSat will be ejected right before or right after apogee.
2. **Pointing loss of receiver is equal to 3 dB**
   It is assumed that the receiver is able to keep the TubeSat within the half power beam width, which will lead to an additional loss of at most 3 dB.
3. **Atmospheric loss of 1 dB included.**
   This is assumed based on the frequency of the signal [34]. Assuming this will lead to some discrepancies in the SNR. But there errors will be at most 1 dB [34].
4. **Minimum link margin of 3 dB**
   An additional link margin of 3 dB was added to account for unforeseen losses[34].
5. **Radiation pattern for typical patch antenna**
   The radiation pattern of a typical patch antenna is used for the analysis of the transmitting antenna gain for a given angle with respect to the ground station. So it has to be kept in mind that if the radiation pattern of the actual antenna is very different that this will have to be implemented in the code.
6. **The TubeSat will always be pointing downwards.**
   The TubeSat might tumble during descent or have a different orientation. However, using the hyperlfo will ensure that the TubeSat will be pointing downwards during descent. Therefore the discrepancies due to this assumption will be negligible.
7. **Short wires in the satellite**
   Assuming short wires in the TubeSat implies there are no big losses from the transmitter to the antenna. Since the TubeSat is very small the wires will be very short and therefore the losses will be negligible.
8. **Perfect impedance matching**
   The circuitry should be designed to have nearly perfect impedance matching, if this is not the case, extra losses should be accounted for.
9. **No interference of objects in front**
   In reality, a camera is in front of the antenna. This will lead to interference due to the circuitry of the camera. This will lead to errors in the SNR. However, because not many metals are present in the camera these losses are assumed to be very small.
10. **Instant change of modulation**
    For the stand-alone system, Wi-Fi is used for transmission. For the Wi-Fi system the change in modulation was implemented instantly when the required SNR was reached. In reality this will most likely take several seconds, this should be tested and then implemented.

The code uses the trajectory model in order to obtain the position of the satellite during descent. Thus also includes all the assumptions made in the trajectory simulation. The sensitivity analysis of the simulation, which can be found in section 10.2, revealed that the most influential parameter on the total amount of data sent was the launch angle. Because the launch angle will change based on the wind direction on the launch day, the chance of this happening is quit high. Therefore the viability of the communication link is checked for every launch angle (80° - 88°).

7.4. Data Transmission

It is useful to send the data as many times as possible. One of the main issues can be the tracking of the signal by the ground station. To ensure the tracking of the signal, the TubeSat could send different data initially to elongate the time the Ground Station would have to track the TubeSat and lock on the signal. The data transmitted by the TubeSat is listed below in its order:

1. Send confirmation that the recording of the video has been started.
2. Send confirmation of the ejection of the satellite.
3. Send an 'empty' signal.
4. Send confirmation that the video recording has been finished.
5. Send message that the video will be sent.
6. Start sending the video.
7. Send message that the video has been sent.
8. Send message that the kinematic behaviour measurements will be sent.
9. Send the kinematic behaviour measurements.
10. Send confirmation that the kinematic behaviour measurements have been sent.
11. Repeat steps 7 to 9 two more times.
12. Repeat steps 4 to 10 until impact on the ground.

Steps 1 to 3 will take about 50 seconds, which is assumed to be enough time for the tracking of the satellite. It is important that the TubeSat keeps transmitting during the first 50 seconds, otherwise the TubeSat might might not be tracked in time. The remaining steps will be performed in about 180 seconds. The scientific measurements will be sent more often, for they cannot be checked visually, as is the case for the video.

7.5. Using Available Equipment

In this section the link budget using the ESC ground station is discussed. First, the available equipment that will be used at ESC is listed and the choices are explained. Secondly, the communications equipment on the TubeSat is listed and explained. At last, the results of the link budget are discussed.

Esrange Space Centre Ground Station

ESC has a couple of tracking antennas, which are shown in Table 7.2. All of them are able to receive a right hand circular polarization (RHCP). Some of them can also receive in left hand circular polarization (LHCP).

Table 7.2: The available tracking equipment at ESC.

Tracking equipment	Frequency [MHz]	Gain [dBi]	G/T [dB/K]	Beam width [°]	Tracking speed [°/s]	Tracking acceleration [°/s^2]
P-band	215-260 295-405	20 14	- -	18 28	5 -	5 -
L/S-band	1440-1790 (L) 2200-2400 (S)	-	5.0(L) 8.0(S)	4 and 30 (3dB)	20	20
L/S-band (DLR)	1440-1790 (L) 2200-2400 (S)	-	14 (S)	2.4(L) 1.8(S)	30	30
S/X-band	2200-2400 (S) 8025-8400 (X)	-	23(S) 33(X)	0.70(S) 0.18(X)	4	10

Moreover, ESC has a frequency plan for telemetry and data-video transmission, as shown in Table 7.3. In order to perform the mission, the system should comply with this frequency plan. The following technical data must be provided at least two months in advance [20]:

- Frequency
- Output power of the transmitter
- Bandwidth

Table 7.3: The frequency plan of ESC for telemetry and data-video transmission.

Frequency band [MHz]	Maximum output power [W]	Maximum bandwidth [MHz]
137-141	10	0.025
227-256	10	0.5
400-405	2	0.5
1425-1525	10	5
2200-2400	10	10

From tables 7.2 and 7.3, it can be seen that only the S-band can completely be used for video transmission. Moreover, the S-band will have the highest available bandwidth and the highest available output power. For these reasons, the data will be transmitted over the S-band.

Still, there are three tracking antennas left to choose from. For now, it is assumed that the 'L/S-Band' tracking equipment is used, since this has the largest beam width. Having a larger beam width is desirable, since that will lead to the largest field of view. Would this tracking equipment prove to be insufficient, one could still select the 'DLR L/S-Band' or the 'S/X-band-tracking equipment.

After the tracking equipment has been chosen, a receiver has to be chosen. Naturally, a receiver with either a BPSK or QPSK demodulation scheme has to be selected, since that would require the lowest SNR [14]. ESC has the following receivers that are able to demodulate a QPSK signal:

- **Microdyne Mod. 700 MR**
  - FM,PM,BPSK,QPSK demodulation.
- **Diversity Combiner, Microdyne 1620-PCB**
  - FM, PM, BPSK, QPSK demodulation.
  - Simultaneous Pre- and Post-Detection.
- **Cortex RTR receiver and combiner**
  - FM, PM, BPSK, QPSK, OQPSK, SOQPSK demodulation.
  - RHCP/LHCP receiver.

Since the data will be transmitted in a RHCP, the 'Cortex RTR receiver and combiner' is the most suitable one for the mission.

Equipment on the TubeSat

The components used on the TubeSat should be compatible with the chosen equipment on ESC. This sets the following requirements on the satellite:

- **ES-Comm-1:** The TubeSat shall be able to transmit inside the S-Band.
- **ES-Comm-2:** The TubeSat shall be able to do BPSK/QPSK modulation.
- **ES-Comm-3:** The TubeSat shall be able to send the data in a RHCP.
- **ES-Comm-4:** The TubeSat shall be able to transmit with a sufficient gain and transmission power to acquire a sufficient link budget.

With these requirements, the following equipment can be used for the mission:

- **Micro S-Band Transmitter** [1]
  - 20 mm in diameter
  - Maximum output power of 0.33 W
  - Frequency range: 2.205 to 2.295 GHz
  - Standard centre frequency: 2.255 GHz
- **Two-Bit S-Band Antenna** [2]
  - 22 mm in diameter and 5.1 mm in height
  - Centre frequency: 2.205 to 2.295 GHz
  - Standard centre frequency: 2.255 GHz
- **AD9157 Digital-to-Analog Converter** [5]
  - 12 x 12 mm
- **ADRF6720-27 Quadrature Modulator** [4]
  - Output frequency range of 400 to 3000 MHz.
  - 6 x 6 mm
  - Baseband bandwidth of 1000 MHz.

[1] URL http://www.syntronics.net/s-band-transmitters.html [cited 3 June 2016]
[2] URL http://www.syntronics.net/cookie-s-band-antennas.html [cited 3 June 2016]

Polarisation
The polarisation of the transmitting and the receiving antenna should be matched to have optimal reception. If this is not the case, there will be extra losses which have to be taken into account. The polarisation of the patch antenna is selectable [43]. A linearly polarised transmitting antenna would lead to a loss of 3 dB[3]. For now, it is assumed that the patch antenna will be RHCP, so there will be no losses due to polarisation mismatch.

Modulation
With the digital-to-analog converter (AD9157) and the quadrature modulator (ADRF6720-27), it would be possible to use *QPSK modulation*.

Frequency
For now it is not known which centre frequency will be used. It is assumed that a frequency of 2.290 GHz is used, since this will have the highest free space loss [14].

Data rate
When QPSK is used and a bandwidth of 10 MHz is available, a maximum data rate of 10 Mbps can be achieved [14]. For now, a data rate of 8 Mbps is assumed, since already a concept exists which is able to do this [29] and this will require a smaller bandwidth.

Transmitting antenna gain
Furthermore, the patch antenna has a gain of about 5 to 7 dB[4]. For now, a maximum gain of $5dB$ is assumed, since this is the lowest value.

Pointing offset of the transmitting antenna.
The gain of the antenna will change when it is viewed from the side. This pointing offset loss is taken into account in the calculation using the radiation pattern of a typical patch antenna [43].

Final specifications of the downlink transmission system.

In short, the specifications of the transmission system of the TubeSat are as follows:

- QPSK modulation.
- Output power of 0.33 W.
- A data rate of 8 Mbps.
- Right hand circular Polarisation
- Transmission bandwidth of 8 MHz.
- Transmitting gain of at 5 dB.

Final Link Budget

Now that all the parameters of the link budget are known, the SNR can be calculated and evaluated. First, the results of the link budget are discussed. Second, the order of the transmission of the data is explained.

Final link budget

Now that all the relevant parameters of the link budget are known, the link budget for the downlink can be set up. In Table 7.4, the values of all the parameters are given. Some of them are also listed as variable, since they are dependent on the trajectory and the distance.

Table 7.4: Calculation of signal-to-noise ratio for the downlink.

Parameter	Value	Unit	Comments
$P_{t_{req}}$	-4.8	dBW	Maximum transmission power of 0.33 W.
G_{AT}	var.	dB	Gain of antenna depends on the radiation pattern.
L_l	0	dB	Negligible due to the short wires.
G_{AR}/T_{sys}	8	$dB \cdot K^{-1}$	Includes the losses of the receiver.
L_{pr}	-3	dB	
L_{fs}	var.	dB	Depends on the trajectory.
k	-228.601	$dBJ \cdot K^{-1}$	
R_b	69	$dBbit \cdot s^{-1}$	A fixed data rate of 8 Mbps.

Using all these values, the SNR can be calculated. The results are shown in Figure 7.3, which also shows the minimum required SNR required for QPSK modulation.

[3] URL http://www.antenna-theory.com/basics/polarization.php [cited 8 June 2016]
[4] URL http://www.antenna-theory.com/antennas/patches/antenna.php [cited 3 June 2016]

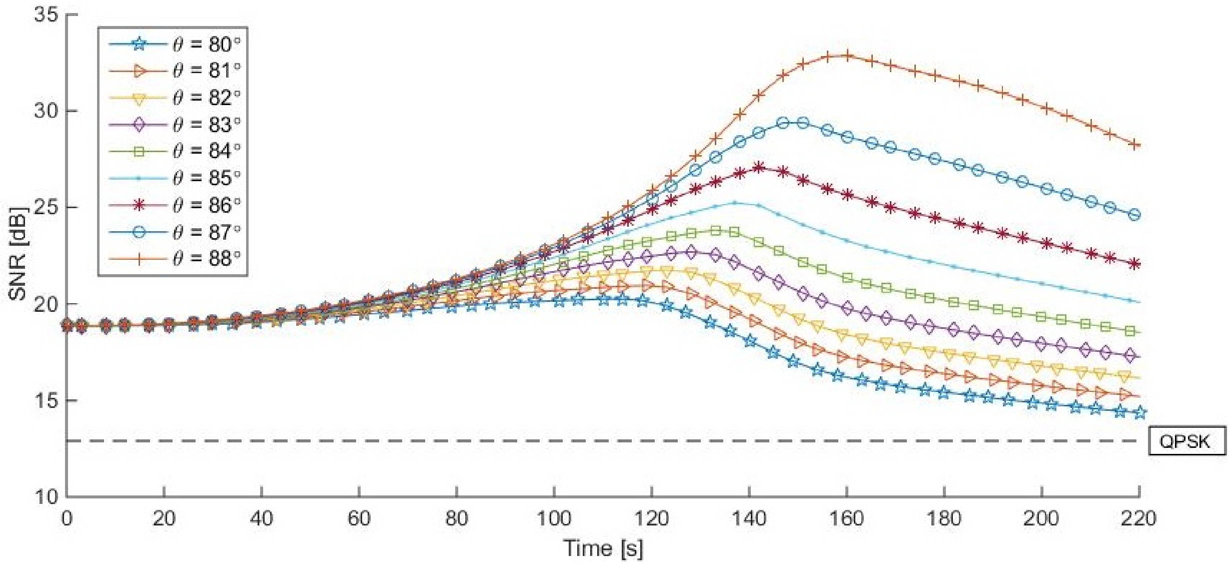


Figure 7.3: The SNR over time for different launch angles for the transmission at ESC.

As can be seen, during the whole descent, the SNR is more than the required 12.6 dB. This means that the data can be sent all the time after ejection. However, since the video will be 30 seconds and it will take around 50 seconds for tracking, only about 180 seconds will be left to transmit the data. With a data rate of 8 Mbps, 180 MB of data can be transmitted. Since more than enough data can be sent, one could also consider lowering the data rate in order to increase the SNR in case the remaining losses will make the SNR insufficient.

7.6. Stand-Alone System

In this section a design for an independent transmission system will be discussed. This design will involve requirements and options for the ground station and the corresponding equipment needed for the suborbital satellite. Firstly, the frequency choice will be presented, this determines what kind of equipment will be needed and what antenna size will be required. Afterwards the needed equipment for the TubeSat will be presented as this determines the final requirements of the ground-station.

Frequency

The frequency determines what kind of equipment the TubeSat and ground station would need, therefore this choice is critical. Firstly, and most importantly the transmission system of the TubeSat has to fit inside of the satellite. Besides this, other important requirements are the need for COTS and a free to use frequency.

The frequency choice has a large influence on the total system, mainly the length of the antenna and the free space loss. Also not all frequencies can be used, some require licenses or special permission. But the largest influence it has is on the amount of COTS products available. The amount of commercially available equipment for a frequency differs a lot from band to band. First, it was checked which frequency bands are available for everyone[5].

The following frequency bands were analysed:

- 14-14.35 MHz
- 28-29.7 MHz
- 144-146 MHz
- 435-438 MHz
- 1.24-1.30 GHz
- 2.4-2.45 GHz
- 3.4-3.5 GHz
- 10-10.45 GHz

Only the Wi-Fi band (2.4-2.45 GHz) and the band used for FPV drones (1.24-1.3GHz) had COTS transmitters, which were able to fit.

[5]URL https://www.fmv.se/Global/Verksamhet/Frekvensf%C3%B6rvaltning/REF%205_1_%20ECC%20Frekvensplan_ERCREP025_110621.pdf [cited 6 June 2016]

In order to choose between the two bands, a trade-off was done. The criteria and the weights are shown in Table 7.5.

Table 7.5: Weight factors of trade-off criteria.

Criteria	Weight Factor	Reasons
Compatibility	5	If it is not compatible with the rest of the system extra converters will have to be used which will increase the size.
Transmission Power	3	A higher transmission power will increase the quality of the signal.
COTS	2	The availability of COTS components will decrease the costs of the communications system significantly.

Table 7.6: Frequency trade-off table (Freq. = Frequency, Crit. = Criteria).

Crit. / Freq.	Compatibility	Transmission Power	availbility COTS antennas
1.24-1.30 GHz	*yellow* No full modules available.	*green* High transmission power available.	*yellow* The ones found did not have high gains.
2.4-2.45 GHz	*green* Full modules available.	*yellow* Only up to 100 mW available[6].	*green* A lot of the satellite antennas are available.

Table 7.7: Legend of frequency trade-off table.

Colour	Meaning
green	Excellent
blue	Good
yellow	Correctable
red	Unacceptable

As can be seen from Table 7.6, the 2.4-2.45 GHz is better to choose to transmit the data.

Satellite Equipment

The best transmitter found in the 2.4GHz band was the xbee WiFi module[7], with the following properties:

- 3.40x2.20x0.30 cm
- 40 mW transmission power
- PCB antenna with 0.6 dB gain, which is nearly omnidirectional

The WiFi module is able to achieve several data rates which depends on the standard used. For now it is assumed that the 802.11g standard is used. The data rate will also depend on the used modulation [16]:

- BPSK (6 Mbps)
- QPSK (12 Mbps)
- 16-QAM (24 Mbps)
- 64-QAM (48 Mbps)

The required SNR for these different modulations in shown in the final link budget results.

[7] URL http://www.digi.com/products/xbee-rf-solutions/modules/xbee-wi-fi#specifications [cited 6 June 2016]

Ground Station

In this part, the minimum needed gain of the satellite dish will be calculated. Then a compliant COTS dish will be presented and the requirements for the pointing system will be calculated.

Satellite Dish
The required gain of the receiving antenna depends on the minimum SNR needed. The minimum needed SNR is defined as follows: *The SNR should be sufficient to send at least 63 MB of data for the lowest analysed launch angle.* From the link budget simulation a G/T of at least 5 was needed to accomplish a total data sent of 63 MB for a launch angle of 82°. Now to acquire the needed gain the noise temperature of the ground station is needed, which is estimated to be 135 K [34]. This gives a required gain of 26.3 dB. However, as for the WiFi modules the SNR determines at which modulation the data can be sent and therefore a higher SNR can greatly increase the data rate. Thus for the design of the ground station a more powerful antenna is used. The specifications[8] are listed below:

- Gain of 30 dB
- Diameter of 1.8m
- BW of 4.5°

Pointing System
In order to receive the data, it is required that the antenna on the ground station is able to point towards the TubeSat with sufficient accuracy. As was discussed in Section 7.2, it was assumed that the pointing offset loss of the receiver would be at most 3 dB. This means that the receiving antenna is able to keep the antenna within its half-power beam width (HPBW).

The following aspects should be taken into account for the tracking:

- HPBW ($\theta_{1/2}$) of the receiving antenna.
- Angular speed of the TubeSat with respect to the receiving antenna.
- Achievable tracking speed at the ground station.
- Area that has to be scanned by the antenna.
- Time it takes to scan the whole area.

Angular speed of the satellite
After simulation of the trajectory of the satellite, the maximum radial speed of the TubeSat with respect to the ground station was found to be approximately $1.5°/s$. This is the minimum speed at which the pointing system should be able to move.

Achievable tracking speed
The achievable tracking speed of the ground station is dependent on the mechanical system, of which the design is beyond the scope of the project. For now, it is assumed that the highest achievable tracking speed is equal to $20°/s$ which is equal to the tracking speed of the L/S-Band Tracking system [20].

Area that has to be scanned
Side winds and the rotation of the earth will change the direction with respect to the initial 'horizontal' launch angle. Because the great uncertainty of wind a large deviation of 20° is assumed. So the horizontal scanner will have to cover twice this, so $\theta_H = 40°$. The vertical angle range it will have to cover, depends on two extreme scenarios:

1. The dart will follow an almost perfect ballistic trajectory and reach a height of 140 km while only covering 10 km of horizontal distance.
2. The dart will be launched at a very low angle of 80° travelling a horizontal distance of 50km and only reaching 100km height

These two extremes give a vertical scanning angle of $\theta_V = 22.5°$. These angles are visualised in Figures 7.4 and 7.5. With these two angles one can imagine a single 2 dimensional plane that will have to be scanned by the dish. The area of this plane will of course depend on the distance between the ground station and the satellite. The greatest distance of the two extremes will be chosen, which is 140 km, since this will lead to the largest area. This gives and area of 57 (vertical) by 117 (horizontal) km that will have to be scanned by the small dish, which is shown in Figure 7.5.

[8] URL http://www.alibaba.com/product-detail/2-4GHz-Antenna-High-Gain-Wifi_60053489404.html?spm=a2700.7724838.0.0.UdPHyg [cited 6 June 2016]

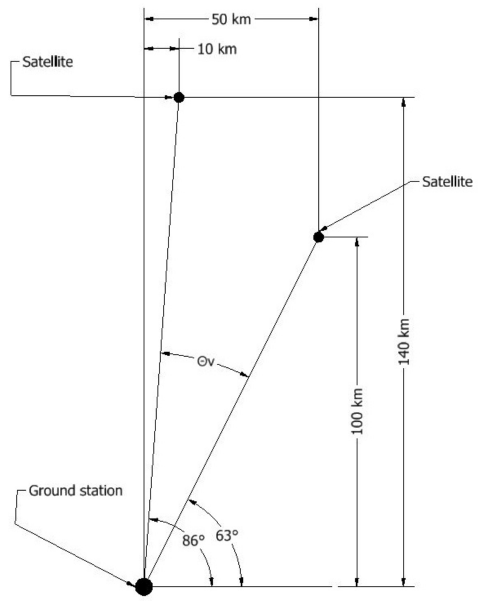


Figure 7.4: Vertical scanning angle sketch.

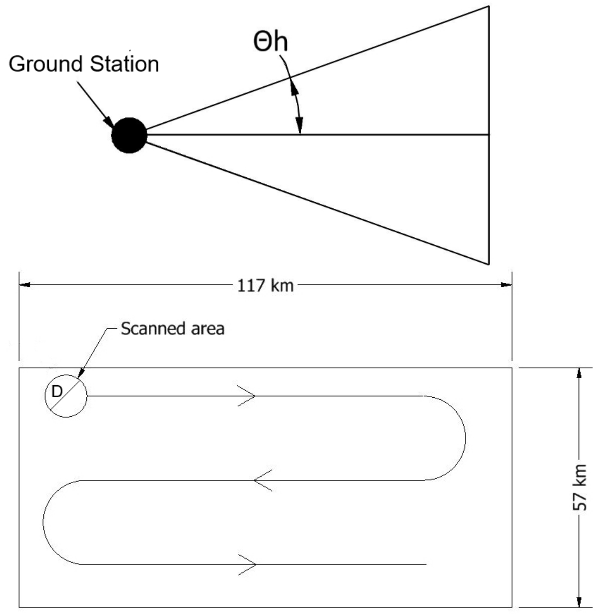


Figure 7.5: Horizontal scanning angle sketch and 2D plane that needs to be scanned.

Time to scan the area
The area that the dish covers depends on the distance from the plane. The smallest possible distance is chosen, since this is the worst case. This is a distance of 110 km. The coverage of the dish is also shown in Figure 7.5. Using the beam width of the antenna and the distance between the satellite and the ground station, diameter of the covered area can be calculated, which is approximately 9 km.

When the diameter, the area which has to be scanned, and the tracking speed is known, the time it will take to scan the whole area can be determined. Of course, it is possible that not the whole area needs to be scanned to find the satellite. However, for the analysis the worst case is taken. Using all the necessary parameters, it was found that the 30 dB dish antenna would need approximately 14 seconds to scan the whole area.

The TubeSat will start transmitting useful data after 50 seconds. It then would seem that a tracking speed of 14 seconds would be sufficient. However, the tracking speed of the antenna system was taken to be $20°/s$, which is equal to the one at ESC. Since it may not be feasible to acquire this tracking speed, it may take even longer to scan the entire area, which may be problematic.

If it would be needed to shorten the time to scan the whole area, an antenna with a smaller gain may be used to scan the area and give a first approximation of the TubeSat's location. The bigger dish would then only have to scan a smaller area. The smaller antenna would then not be able to fully receive the signal, but it would be able to detect it. To be able to detect a signal the SNR of the link should be larger than zero, because the power of the signal will then exceed the power of the noise.

Using Equation 7.2, the minimum required gain of the receiving antenna is determined. The values of the parameters for this calculation are shown in Table 7.8.

Table 7.8: Values for the minimum antenna gain calculation.

Parameter	Value (dB)	Comments
SNR	0	Required to be higher than zero.
P_t	-14	40 mW of transmission power.
$G_A T$	0	Nearly isotropic antenna .
L_l	0	Assumed zero.
L_r	0	Assumed zero.
L_{pr}	-3	Satellite will remain in the 3dB beam width of the receiving antenna.
L_{fs}	142	Worst case: distance of 140 km.
k	-228.6	
T_{sys}	21.30	Standard value for a 2.4 GHz communications system [34].
R_b	64	Data rate of 2.25 Mbps.

This resulted in a required antenna gain of at least 16.3 dB. When looking at COTS antennas, the chosen antenna[9] had a gain of 20 dB, which was the lowest gain that would be sufficient.

The chosen antenna has a HPBW of 18° both in the vertical and horizontal plane. With this HPBW scanning the area with a tracking speed of 20°/*s*, the area will be scanned in 4 seconds, which is four times faster than with the larger dish.

Receiver

In order to receive the WiFi signal, a simple WiFi receiver would be sufficient. The 'PowerWiFi Outdoor WiFi USB Antenna'[10] can be used.

Final Lay-out of the ground station.

Figure 7.6 shows the final lay-out of the ground station using WiFi.

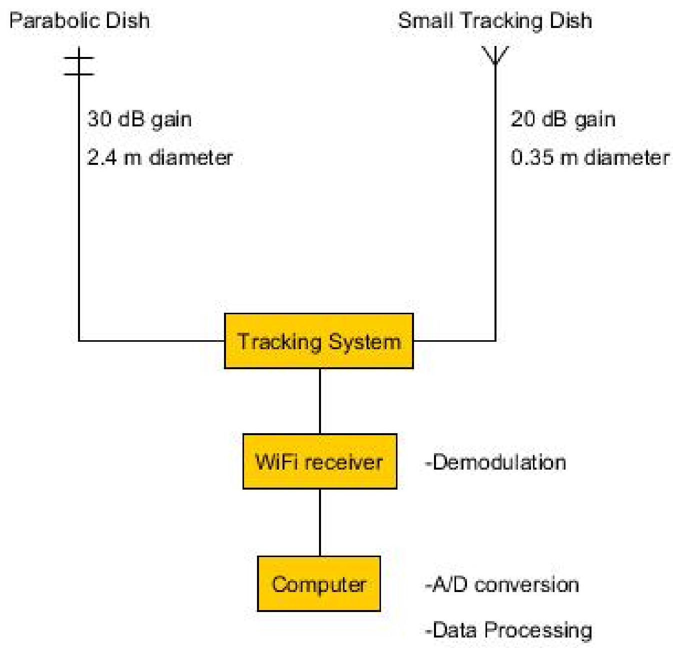


Figure 7.6: Lay-out of the WiFi ground station.

Final link budget

In this section, the final results for the link budget of the stand alone system will be shown and discussed.

In order to send with a higher data rate, a more complex modulation is needed which requires a higher SNR. The needed SNR for each modulation [34] and the SNR over time for every launch angle is shown in 7.7. The worst scenario occurs for the lower launch angles of 80° to 82°. They can only send data using QPSK modulation (as can be seen in Figure 7.7) and reach a total sent data of **175 MB**. This is more than enough to fulfil the requirement of 63 MB, besides this there is also an extra link margin of 1.4 dB, adding this to the standard of 3 dB, this results in a final link margin of 4.4 dB.

[9]https://www.wifi-shop24.com/24-GHz-WiFi-Directional-Panel-Antenna-Outdoor-20dBi [cited 15 June 2016]
[10]https://www.electro-sat.nl/powerwifi-outdoor-wifi-usb-antenna.html [cited 13 June 2016]

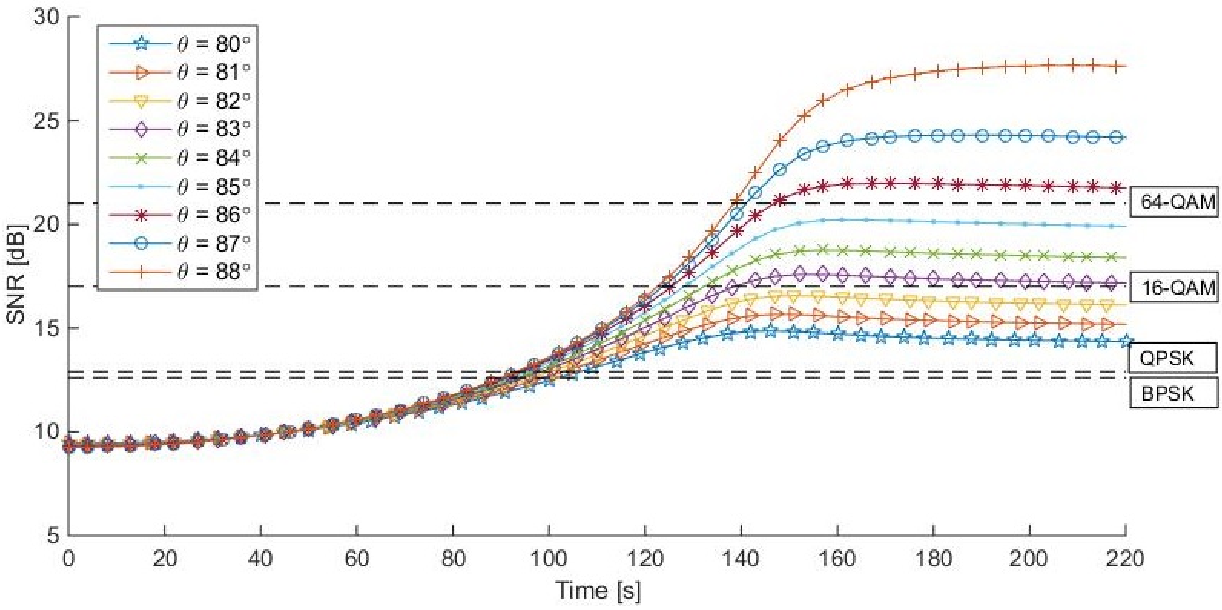


Figure 7.7: The SNR over time for different launch angles for WiFi.

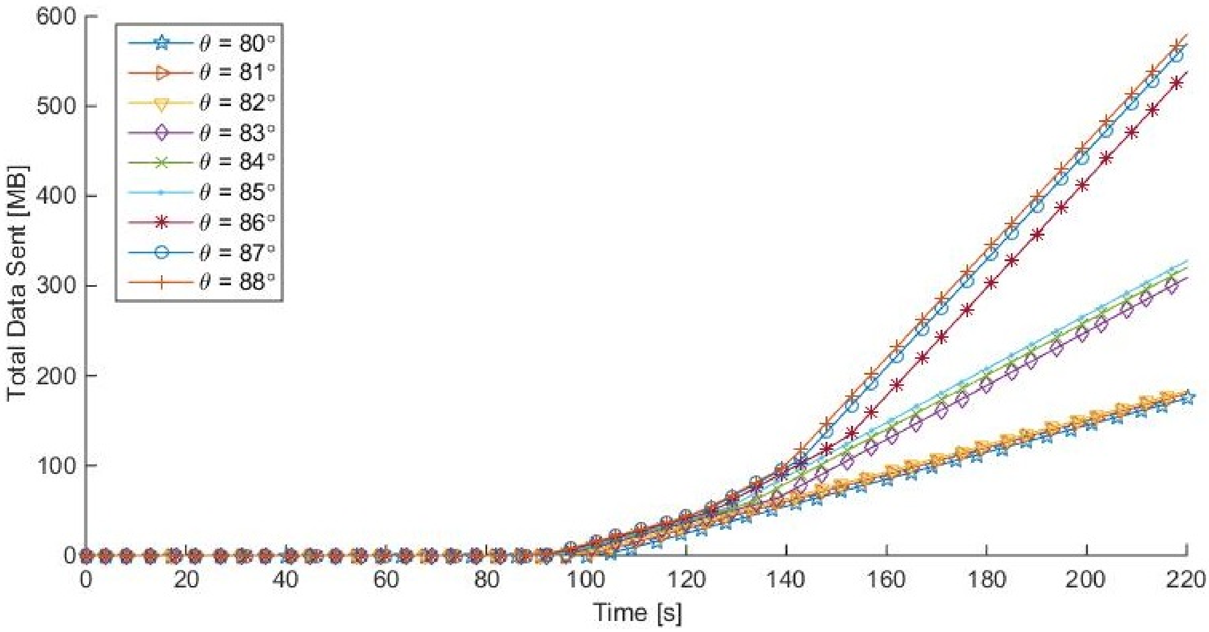


Figure 7.8: The amount of data sent over time for different launch angles for WiFi.

7.7. Trade-Off

Currently there are two options for the ground station, one using the ground station at ESC and one using a stand-alone one. In this section a trade-off is made between the ground station at ESC and the stand-alone system. Firstly, the advantages and the disadvantages of the ESC ground station an the stand-alone system are listed. Secondly, the trade-off criteria are listed. Finally, the trade-off is done and the communication flow diagram is shown.

Advantages and Disadvantages of the Two Options

Esrange Space Centre Ground Station

- **Advantages**
  - All the equipment is already there.
  - The tracking system has already been proven.
  - All the equipment is compatible with one another.
  - There are licenses for non-public bands, which will eliminate the chance on interference.
  - A very high transmission power is allowed.
  - Signal-to-noise ratio is sufficient during the whole mission.
- **Disadvantages**
  - Transmission system has to be compatible with the available equipment, which limits the freedom of the design.

Stand-Alone System

- **Advantages**
  - More freedom in the design of the transmission system.
  - The ground station can be made with COTS components, which will reduce the cost significantly.
  - The launch can be conducted at any location, since the stand-alone system can be placed everywhere.
- **Disadvantages**
  - Transmission is done via WiFi, which is a public band, which increases the chance of interference.
  - The tracking system has to be tested and validated.
  - The whole ground segment has not been proven to work.
  - There is a limitation transmission power.
  - All the equipment of the ground station has to be transported to the launch site.
  - Really sensitive to the launch angle.

Trade-Off Criteria and Weights

Table 7.9 shows the criteria, their weights and the reasoning.

Table 7.9: Weight factors of trade-off criteria.

Criteria	Weight Factor	Reasons
Link margin	4	The link margin is along with risk the most important aspect of the communications, as this indicates the quality of the signal. Since not every loss has been analysed yet, a higher link margin is desirable.
Feasibility	3	The feasibility of the ground station will be of utmost importance for success of the mission. If it is not possible to use one of the ground stations, it is not possible to transmit the data.
Compatibility	2	A good compatibility between the communications system of the TubeSat and the ground station is crucial for the transmission of the data, however, the design is made to be compatible. Therefore, this criteria has a lower weight.
Risk	4	The risk of using the ground station should be considered carefully, since this will have a big impact on the success of the mission.

Trade-Off and Communications Flow Diagram

Table 7.10 shows the trade-off of the two ground stations. Table 7.11 shows the legend of the trade-off table.

Table 7.10: Ground Station trade-off table.

Option \ Criteria	Link Margin	Feasibility	Compatibility	Risk
ESC ground station	*green* Additional link Margin of at least 1.8 dB throughout the whole mission.	*green* Already fully equipped and available.	*blue* Designed for compatibility.	*blue* ESC equipment has been proven to work.
Stand-alone system	*blue* Additional link margin of 1.4 dB in worst case angle.	*yellow* Ground station has yet to be produced and tested.	*blue* Designed for compatibility.	*yellow* Equipment has not been proven to work.

Table 7.11: Legend of Communications trade-off table.

Colour	Meaning
green	Excellent
blue	Good
yellow	Correctable
red	Unacceptable

As is apparent from the trade-off, the ESC ground station is the better option. Therefore, the design with the ESC ground station will be worked out from this point on. Figure 7.9 shows the communications flow diagram with the ESC ground station including all the components which will be used. At each component, it is listed what is does. Moreover, properties of the signal are also listed, such as carrier frequency, data rate, signal power and the type of data.

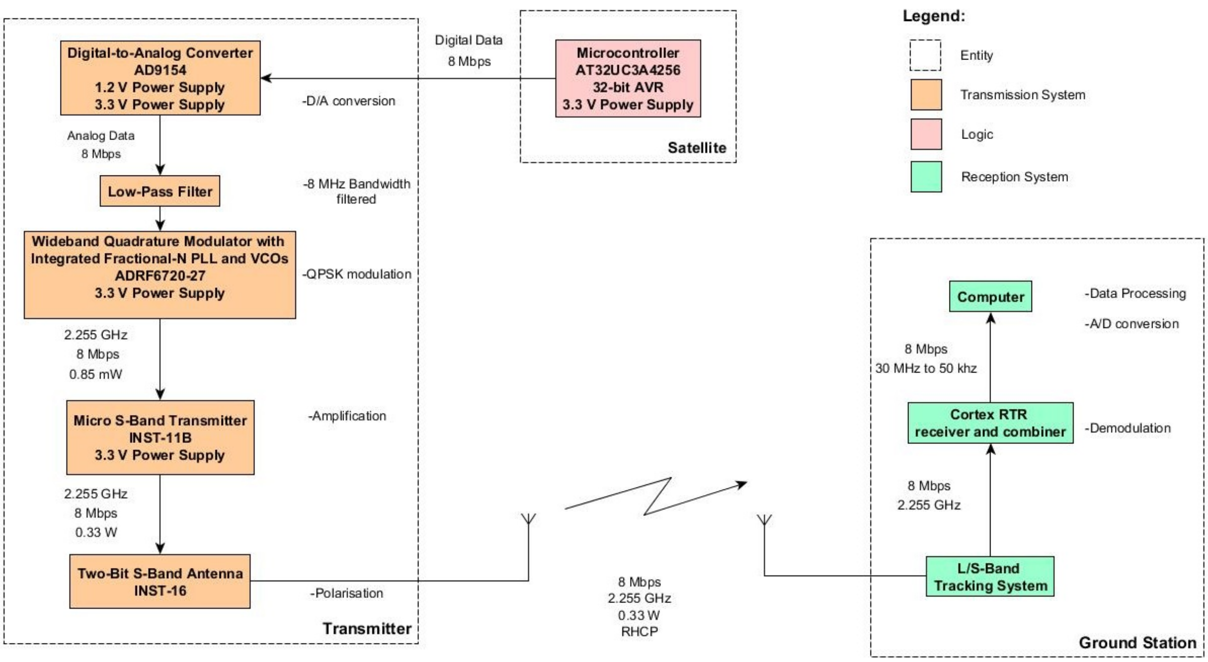


Figure 7.9: The communications flow diagram with the ESC ground station.

7.8. Recommendations

In this section a couple of recommendations are listed to further work out and improve the ground station. Recommendation for both options are listed, since it might be used in the future when multiple launches are required.

Recommendations for the ESC Ground Station

- **Coding Schemes**
  Encoding the data before sending has the advantage of forward error correction, i.e. finding and correcting errors in the received data. When this is done, a lower SNR is required to get a sufficient BER. For example, one could use Reed-Solomon Coding, due to which only a SNR of 2.7 dB is required for a BER of 10^{-5}. Reed-Solomon Coding is commonly used in broadcasting [13].
- **Analysis of remaining losses**
  As of now, the exact lay-out of the circuitry has not been determined. Therefore, several losses, phasing circuit loss, the filter loss and the transmission line losses, could not have been determined yet.
- **Impedance Matching of the antenna.**
  Impedance mismatching of the antenna could lead to some losses [10].
- **Determine circuitry for the low pass filter.**
  A low pass filter has to be used to get the correct bandwidth for the transmission. However, the filter itself has not been designed yet.
- **Check the compatibility of the components.**
  The AD9157 and the ADRF720-27 have been made to be compatible with each other. However, not much is known about the compatibility between the modulator and the transmitter. Their compatibility has to be checked.
- **Check the feeding of the patch antenna.**
  The polarisation of a patch antenna depends on the way the information is fed to the antenna. It is uncertain if it is possible to get RHCP with the Two-Bit S-Band Antenna. If this is not the case, additional losses have to be taken into account for the polarisation mismatch.
- **Altering the data rate**
  If the remaining losses are so much such that the required SNR is not achieved, one could also look into reducing the data rate, since then the SNR would increase.
- **Different antenna**
  At the moment, the patch antenna is behind the camera module. For now, it was assumed that the interference was not that large, but this may not be the case. Therefore, it is recommended to look at a different kind of antenna. The antenna can be mounted on the hyperflofor instance. Moreover, a ring feed antenna may be used [46], which can be mounted around the lens of the camera. This would increase the performance.

Recommendations for the Stand-Alone System

- **Quickness of connection**
  The time it takes for the WiFi module to connect with the ground station will have to be researched.
- **The stability of the connection**
  The influence of the speed and rotation of the TubeSat on the stability of the WiFi signal has to be investigated. Of course WiFi is not designed to cope with these kinds of conditions, the protocol might not be sufficient for the mission.
- **Modulation change**
  The WiFi module should automatically switch modulation once there is enough SNR to do so. The speed it does this with should be checked, as in the simulation it is assumed to be done instantly.
- **Compatibility**
  The system should be checked as a whole. All the components should work together properly, this should be thoroughly checked.
- **Research the 1.3 GHz band in case WiFi will not work**
  As this has no power restriction it can be very beneficial to use instead of the WiFi band. The 1.3 GHz is often used for live-stream video, directly connecting the camera to the transmitter. For a mission requiring live-stream video this could be very interesting.

8 Electronics

The TubeSat largely consists of electronics, as the main objectives of the mission are to shoot a video of space and perform measurements. As the team members are not electronics engineers, the team is providing TME with a more general electronics design. The detailed design will be left to the electronics engineer(s) of TME. Important decisions related to the electronics were made in cooperation with Eric Smit, the electronics engineer from TME.

An overview of the electrical components used is given in Section 8.1. The electrical block diagram is shown in Section 8.2. The data handling and software block diagrams are displayed in Sections 8.3 and 8.4, respectively. Finally, recommendations on the electronics are given in Section 8.5.

8.1. Electrical Components

For the design of the electronics, first, components which are essential for accomplishing the mission objective have to be determined. A small camera module is required to shoot the video footage of space. To validate the apogee of the DART and to determine the position of the TubeSat a GPS antenna module is required. Moreover, the following components are needed to measure the DART and TubeSat kinematics: low g's accelerometer, high g's accelerometer, magnetic sensor and gyroscope.

The first step was to search for COTS components which are small enough to fit in the satellite, because one of the requirements is that only COTS components should be used if possible. Most of the components found, require a supply voltage of $3.3V$ and are compatible with either a SPI, I^2C or UART interface. However, the camera found initially is not compatible with any of these interfaces. The advantage is that this camera module is one of few cameras that have a resolution of $1080p$ and a frame rate of 30 frames per second and is including a video compression module. The disadvantage is that the camera module requires three connection cables. Together with the relatively large size of the camera itself, this makes the camera too big to fit in the TubeSat and leave enough space for the other components.

Therefore, a new camera had to be found. The new camera found has a frame rate of 33 frames per second at a resolution of $1080p$. The disadvantage is that the camera module does not include a compression module. Therefore, an image processor had to be found. Another disadvantage is that the operating voltage of the camera module and the image processor is not equal to $3.3V$. However, this could be solved by using DC-DC converters, because they can be relatively small.

To let the electrical components work properly, a microcontroller is required. An Atmel microcontroller would be a good choice, because they are relatively easy to program[1]. The first criterion to choose the right Atmel microcontroller was the presence of the right interfaces in the required amount. The second criterion was the operating voltage of the microcontroller; the preferred operating voltage was $3.3V$. After all components had been found, they had to be integrated on printed circuit boards (PCBs) together with the data transmission system components determined in Chapter 7. Knowledge of electronics is required to be able to properly integrate the components. Therefore, it has only been taken into account that a significant amount of space on the PCBs is dedicated to wiring and connecting the components. The dimensions of the PCBs are shown in Figure 8.1 in Section 8.2.

The beacon module is powered by the battery and is activated by the microcontroller. The implementation should make sure that once activated it will continue working until the battery dies. Special care should be taken that the attachment of the wiring survives the crash landing.

Two functions though have not been integrated yet on the PCBs: turning on and turning off of the satellite. The first is taken care of by a kinetic switch[2] linked to an SR latch so that the launch forces turn on the satellite. To prevent the battery to catch fire due to an under voltage -and set on fire a forest- a battery supervisor should cut the power just before the lower voltage limit of the battery.

[1] This advice was given by Eric Smit, the electronics engineer of TME, during a meeting on May 30, 2016.
[2] URL http://www.circoraerospace.com/pdf/CIRCOR-KineticSwitch-brochure-2014.pdf [cited 7 June 2016]

Furthermore, the ejection detection plugs straight into the microcontroller. A two core wire is attached to the PCB, here it connects to a resistor and one of the voltage supplies to make a circuit. The analog-to-digital converter (ADC) of the microcontroller is used to measure the voltage over the resistor. Once the TubeSat is ejected from the dart, the wire is broken so the circuit will be open. At this point the ADC will read $0V$.

The electronics are integrated on 5 different PCBs. The electrical components which can be found on the PCBs are listed in Table 8.1.

Table 8.1: Overview of electrical components (unspec. = unspecified).

PCB	Component	Component number	Brand	Dimensions (l x w x t or d x t) [mm]	Mass [g]	Ref.
PCB 1	Microcontroller	AT32UC3A4256	Atmel	7 x 7 x 1	1.3	[6]
	Image processor	STV0991	STMicroelectronics	10.0 x 10.0 x 1.12	unspec.	[53]
	MicroSD connector	DM3AT	Hirose Electric Group	21.55 x 13.85 x 1.68	unspec.	[27]
	Modulator	ADRF6720-27	Analog Devices	6 x 6 x 0.75	unspec.	[4]
	Digital-to-Analog Converter	AD9154	Analog Devices	12 x 11.75 x 0.85	unspec.	[5]
	DC-DC converter 3.6 V - 3.3 V	TS3310ITD1022	Silicon Labs	2 x 2 x 0.8	unspec.	[49]
	DC-DC converter 3.6 V - 2.8 V	TS3310ITD1022	Silicon Labs	2 x 2 x 0.8	unspec.	[49]
	DC-DC converter 3.6 V - 2.5 V	TS3310ITD1022	Silicon Labs	2 x 2 x 0.8	unspec.	[49]
	DC-DC converter 3.6 V - 1.8 V	TS3310ITD1022	Silicon Labs	2 x 2 x 0.8	unspec.	[49]
	DC-DC converter 3.6 V - 1.2 V	LTC3250-1.2	Linear Technology	3 x 3 x 1	unspec.	[36]
	Resistor	unspec.	unspec.	unspec.	unspec.	-
PCB 2	3-Axis MEMS accelerometer low g's	ADXL362	Analog Devices	3 x 3.25 x 1.06	0.018	[3]
	3-Axis MEMS accelerometer high g's	ADXL375	Analog Devices	3 x 5 x 1	0.030	[2]
	3-Axis MEMS gyroscope	ITG-3701	InvenSense	3 x 3 x 0.75	unspec.	[28]
	3-Axis MEMS magnetic sensor	MMC3316xMT	Memsic	2.0 x 2.0 x 1.0	unspec.	[39]
PCB 3	GPS antenna module	ORG1410	OriginGPS	10 x 10 x 5.8	2.5	[44]
PCB 4	5.0 MP camera module	VB6955CM	STMicroelectronics	7.5 x 7.5 x 4.6	unspec.	[54]
PCB 5	Micro S-band transmitter	INST-11B	Syntronics	20.3 x 2	unspec.	7.5[3]
	S-band antenna	INST-16	Syntronics	20.3 x 5.1	unspec.	7.5[3]
-	Battery	TL-2450	Tadiran Lithium Batteries	24 x 5.6	8.8	[55]
	Transmitter (beacon)	DRA887TX	Dorji Applied Technology	23.3 x 12.0 x 5.9	unspec.	[18]

In Table 8.1 the resistor is not specified. A simple resistor is needed to convert a voltage of $1.2V$ to $1.1V$ as can be seen in Figure 8.1 in Section 8.2. The converters used to supply a voltage of $3.3V$, $2.8V$, $2.5V$ or $1.8V$ all have

[3]This is the number of the section in which the footnotes referring to the components on the website of the manufacturer are stated.

the same component number. This is because this converter has an input voltage range of $0.9V$ to $5.0V$ and it can output a voltage of either $1.8V$, $2.1V$, $2.5V$, $2.8V$, $3V$, $3.3V$, $4.1V$ or $5V$[4]. Since it can only output one of these voltages, 4 converters of this type are needed. The battery will not be mounted on a PCB.

An additional note for the GPS receiver has to be made. It takes time for the receiver to acquire lock, and this only starts after ejection, as the GPS signals cannot penetrate through the metallic dart. This means that the highest altitude registered may be lower than the actual apogee. The combination of accelerometers, gyroscope and magnetometer for the registration of the behaviour of the DART rocket will be used to back-calculate the actual apogee. All of the data must be timestamped with an accurate on-board clock. In addition, measures are taken to make acquisition as quickly as possible. From the datasheet [44] of the chosen GPS receiver, it was found that with a cold start, the acquisition time is <35s. By providing position, velocity, time, and satellite ephemeris data, a hot start can be achieved, which results in an acquisition time of less than one second. Position, velocity and time data can be provided by simulating the trajectory before launch. The orbits of the GPS satellites can also be simulated and the ephemeris data can be found from this.

8.2. Electrical Block Diagram

The electrical equipment of the product is shown in the electrical block diagram, Figure 8.1. This diagram shows the interactions between the electrical components, such as power consumptions and voltage conversions.

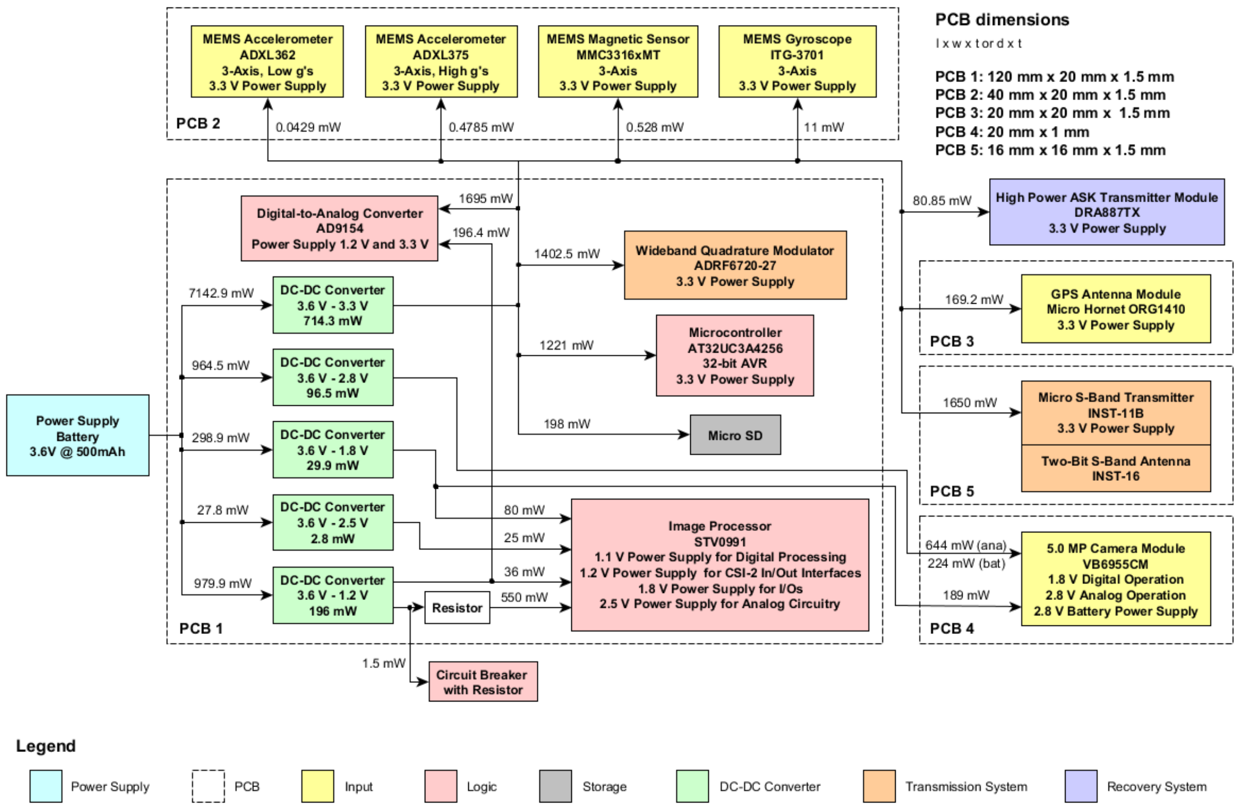


Figure 8.1: Electrical block diagram (ana = analog, bat = battery).

As can be seen from Figure 8.1, for each component the supply voltage(s) and power consumption are specified. Five different DC-DC converters are used to accommodate for all the voltage requirements of the electronics. The power mentioned in front of the DC-DC converters is the total power consumed by the converters itself and the electrical components behind the converters. The power consumed by the converters is specified in the converter blocks. The converter used to supply a voltage of $3.3V$, $2.8V$, $2.5V$ or $1.8V$ has a maximum efficiency of 92%[4]. Therefore, the assumption was made that the efficiency of the converter equals 90%. The converter used to supply a voltage of $1.2V$ has an efficiency of around 81%[5]. The assumption was made that the converter efficiency equals 80%.

[4]URL http://www.silabs.com/Support%20Documents/TechnicalDocs/TS331x.pdf [cited 8 June 2016]
[5]URL http://cds.linear.com/docs/en/datasheet/3250fa.pdf [cited 8 June 2016]

It must be mentioned, that not all the components are being used the entire flight, as can be seen on the active status timeline, Figure 8.2. For most of the components, the brand and component number are specified, as can be seen both in Figure 8.1 and Table 8.1. Only one battery is used, as the battery chosen provides enough energy for all electrical components, as can be read from Figure 8.3. The transmitter and digital-to-analog converter use the largest amount of power. Other components which use a significant part of the power available are the microcontroller and the camera. The image processor and the camera have more than one power supply arrow in Figure 8.1; this is due to the different voltages required within the component.

The microSD card has not been specified, because those cards are readily available and the requirements on the card are met by numerous cards. The card should have a memory size of at least 250 Mb taking into account a safety margin. The write and read speed should be at least 6 Mbps taking into account a safety margin.

The timeline, Figure 8.2, shows an overview of when the electrical components are active. As it can be observed, the camera is filming for only 30 seconds and the image processor is on for a second longer. The transmitter starts transmitting right after ejection. Most of the other components are active the entire flight. The DC-DC converters are shown to be active the entire flight, however, it depends on the components supplied by a specific converter if the converter is in use. This is not shown on the timeline, but it is taken into account in the power and energy consumption shown in Figure 8.3.

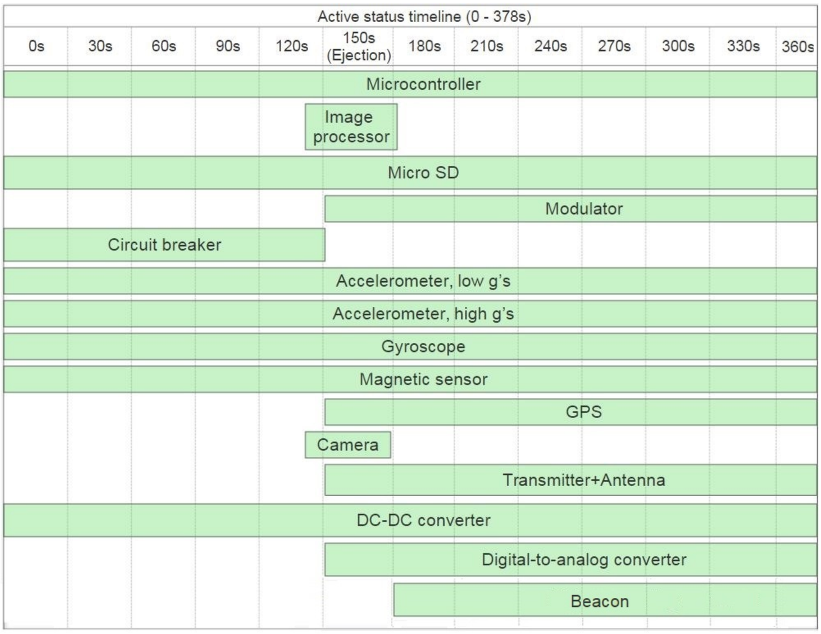


Figure 8.2: Active status timeline of electrical components.

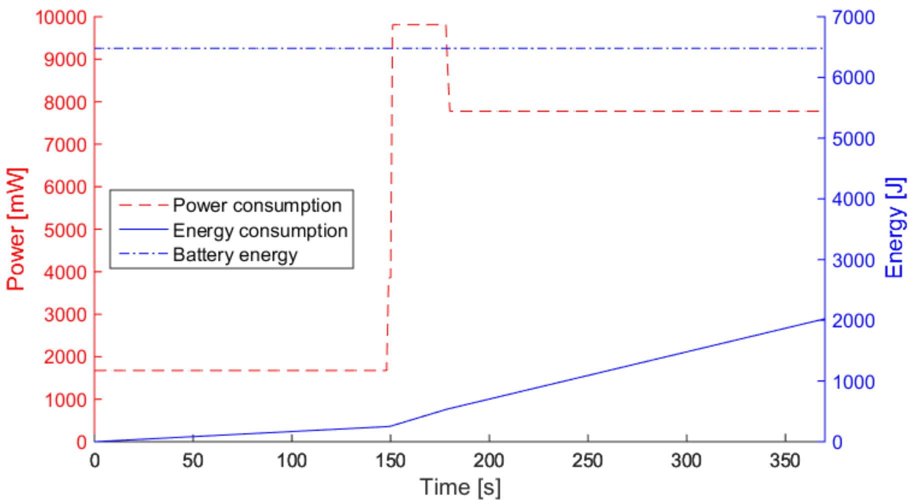


Figure 8.3: Power and energy consumption as a function of time.

Power and energy consumption as a function of time are depicted by Figure 8.3. For both the power and energy consumption a circuit efficiency of 95% is assumed to account for wiring and connection losses. In Figure 8.3, the power consumption can be read from the left vertical axis, whereas the energy consumption and battery energy can be read from the right vertical axis. It can easily be seen that the battery energy is higher than the energy consumption. Therefore, it is clear that only one battery is enough to power the satellite. As not all components are active the entire flight, the power consumption with respect to time shows when the highest drain on the battery occurs. As expected, this is during shooting of the video footage and the transmission. With the energy left in the battery after satellite impact, the beacon will be able to transmit for approximately another 15 hours.

8.3. Data Handling Block Diagram

The data handling between the electrical components is shown in Figure 8.4. The interfaces, buses and data rates between the components are given. More details about what kind of data is being send from one component to another are shown in Figure 8.5. The analog-to-digital converter shown in the diagram is integrated in the microcontroller.

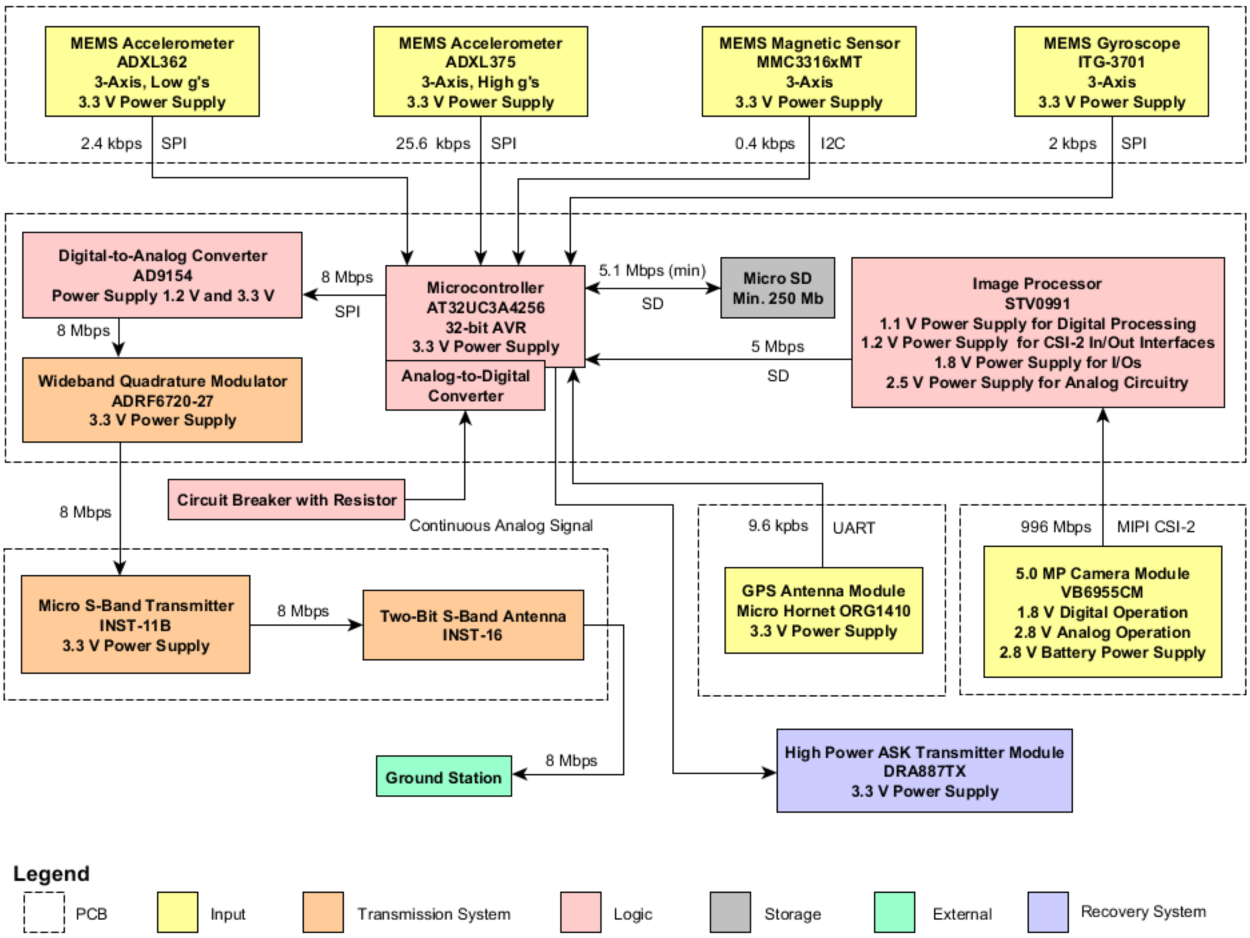


Figure 8.4: Data handling block diagram.

8.4. Software Block Diagram

The software block diagram is shown in Figure 8.5. It illustrates the connection between the components by describing the data send between components. It can clearly be seen that all the data are first send to the micro-controller and from there to the microSD card and the transmission system, i.e. modulator, transmitter and the antenna, from which it is send to the ground station.

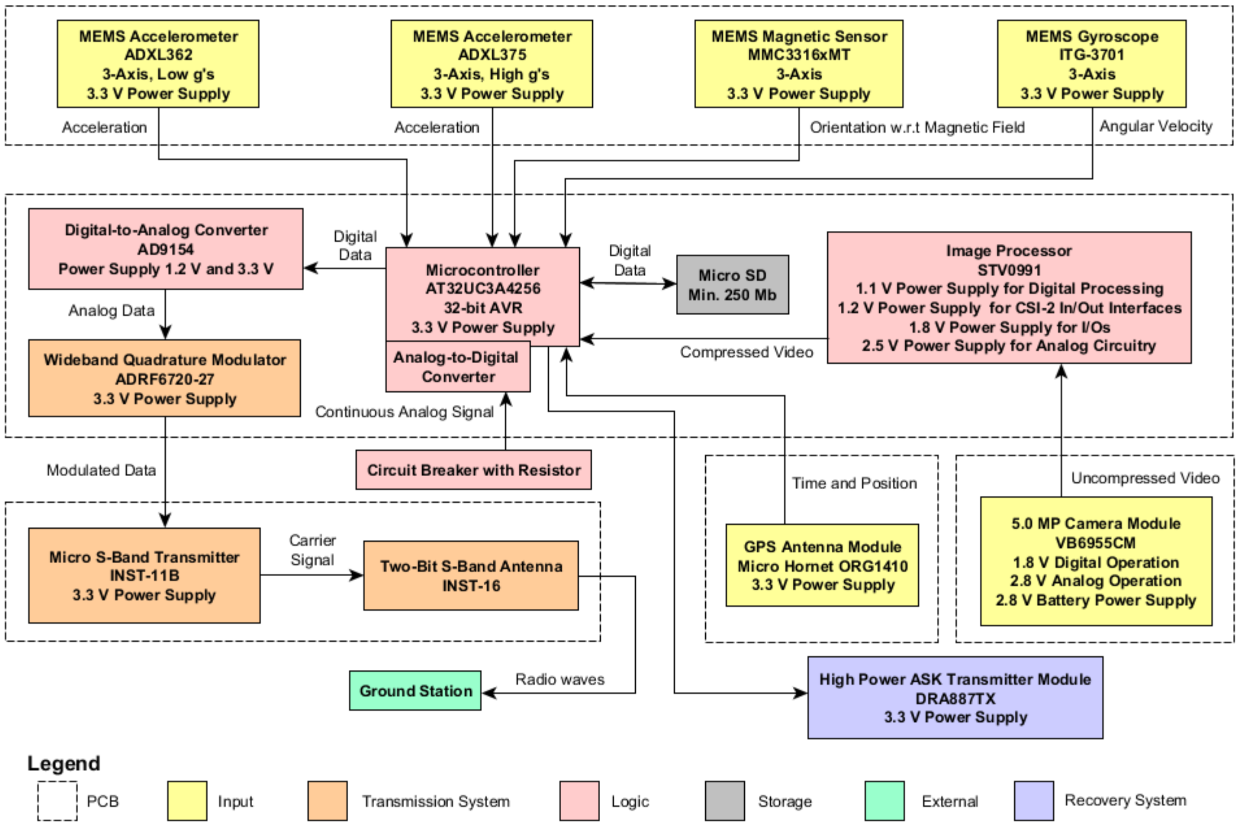


Figure 8.5: Software block diagram.

8.5. Recommendations

Below, some recommendations are given to finish the design of the electronics.

- As already mentioned, since the team members are not electronics engineers, the detail design of the electronics still needs to be performed.
- An electronics engineer should design the kinetic switch and SR latch for turning on and turning off the satellite.
- The main structure, made out of metal, will protect the internal components against lightning strike. Yet the antenna and the camera are located outside this protection. They will need to have surge protection to make sure that no excessive amount of current will flow into the other electronic components. Designing this is however outside of the expertise of the team members, so it is recommended that an electronics engineer does this.
- It is also recommended that an electronics engineer checks the chosen components for their performance and for compatibility and checks whether certain electrical components need to be added for the electronics to perform successfully.
- An electronics engineer should also look into the influence of ionisation.
- An electronics engineer should estimate how much the components will heat up.
- An electronics engineer should test the electrical components for shock. Especially (soldered) connections are very prone to shocks; these have to be checked extensively.
- An electronics engineer could also check whether some components are oversized. In that case, the engineer can replace those components by more correctly sized ones, which are less expensive.
- An electronics engineer has more knowledge about component performance. The engineer could check whether the chosen components have a good efficiency or whether components with a higher efficiency are available.
- Optionally, an electronics engineer could look into whether scientific measurement equipment can still be added. The battery can deliver enough energy to support them. However, they have not been implemented by the team. The reason for that is explained in Section 4.1.
- It is recommended that the unused energy in the battery and the unused storage capacity of the microSD card is used to shoot extra video footage during satellite descent to obtain more data about the descent trajectory until impact. This video footage can be stored on the microSD card without transmitting it to the ground station. Since there is a high probability that the microSD card survives the impact, this video footage can be valuable extra information if the satellite is retrieved. However, due to this extra energy consumption the amount of time the beacon will be able to transmit after satellite impact reduces.

9 Vibrational & Structural Analysis

In this chapter the structural and vibrational analyses of the TubeSat are discussed. The chapter is structured as follows: a vibrational analysis is presented in Section 9.1. Then the structural analyses of the individual parts are discussed in Section 9.2. This chapter is concluded with assumptions and recommendations in Sections 9.3 and 9.4, respectively.

9.1. Vibrational Analysis

The vibrations that the TubeSat will experience during its mission will be caused by the rocket launch. These self-excited vibrations are caused by the burning of propellant in the rocket engine. Variations in the engine thrust caused by irregular burning of the propellant will start vibrations. These kind of vibrations are called thrust oscillations or Pogo oscillations, because the vibrations can be compared by the longitudinal movement of a pogo stick. If the vibrations happen to match the natural frequency of the rocket ($\frac{W_a}{W_n} = 1$), then dangerous oscillations might occur in the TubeSat, which might cause it to break.

To determine the natural frequency, W_n, of the TubeSat it needs to be modelled. In this case the most vibration sensitive part of the TubeSat is the nose cone, as it is made of a brittle material. Therefore it is separately modelled as well. In Figure 9.1 a discretized drawing of the TubeSat model can be seen.

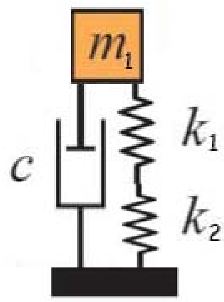


Figure 9.1: Discretization of the TubeSat.

The assumption was made to discretize the TubeSat and the nose cone as an axial beam, see Section 9.3. The spring constant of the body and the nose cone, and their equivalent spring constant are then determined using Equations 9.1 and 9.2.

$$k = \frac{EA}{L} \quad (9.1) \qquad k_{eq_{series}} = \frac{1}{\frac{1}{k_1} + \frac{1}{k_2}} \quad (9.2)$$

Then the natural frequency can be determined using Equation 9.3.

$$W_n = \sqrt{\frac{k}{m}} \quad (9.3)$$

As the TubeSat will be damped by the Sabot, the natural frequency equation changes to that of a forced damped free vibration, shown by Equations 9.4, 9.5 and 9.6.

$$W_d = W_n\sqrt{1-\zeta^2} \quad (9.4) \qquad \zeta = \frac{c}{c_{cr}} \quad (9.5) \qquad c_{cr} = 2\sqrt{km} \quad (9.6)$$

where:

- k is the spring constant in N/m.
- k_{eq} is the equivalent spring constant in N/m.
- E is the Young's modulus in Pa.
- A is the cross-sectional area in m^2.
- L is the length of the satellite in m.
- W_n is the natural frequency in Hz.
- W_d is the natural damping frequency in Hz.
- ζ is the damping ratio.
- m is the satellite mass in kg.
- c is the damping coefficient of the damper.
- c_{cr} is the critical damping coefficient.

The actual calculations for the vibrational analysis were not done due to the fact that the damper coefficient of the sabot was unknown. This coefficient was not found in the data sheet provided by Sorbothane [52]. Therefore it was impossible to determine the natural damping frequency.

Once the coefficient has been provided, vibrational data is needed from a test launch to be compared to the natural damping frequency. If the ratio between the two is not equal to 1, the TubeSat should not break. If it is it will need to be redesigned, or damped with different material. When the ratio is not equal to 1, the TubeSat can be tested in a shaker.

9.2. Structural Analysis of Individual Parts

In this section structural analyses of the rollout system and the Hyperflo are presented, as these parts of the TubeSat are the only parts that experience structural forces during descent.

9.2.1. Rollout System

The rollout system of the Hyperflo will only experience the force exerted on it by the line that is attached to the Hyperflo. The maximum force that is transferred through the line was determined to be 19 N, through the simulation, see Chapter 6. Note that the lines have been tested for 54.4 kg[1] and will therefore not fail due to the applied force.

With this force the maximum shear stress on the steel pin ($d = 1mm$) can be determined and checked for failure. Using the 19 N as shear force in combination with Equation 9.7, the shear stress can be determined.

$$\tau = \frac{VQ}{It} = \frac{19 \cdot 8.3 \cdot 10^{-11}}{4.9 \cdot 10^{-14} \cdot 0.5 \cdot 10^{-3}} = 64.4MPa \tag{9.7}$$

where Q is the first moment of area in m^3, I is the moment of inertia in m^4, t is the thickness in m and V is the shear force in N.

How to obtain Q and I is shown in Equations 9.8 and 9.9, respectively.

$$Q = \int y dA = 8.3 \cdot 10^{-11} m^3 \tag{9.8}$$

$$I = \frac{\pi d^4}{64} = 4.9 \cdot 10^{-14} m^4 \tag{9.9}$$

where d is the diameter of the pin in m.

In Figure 9.2 the area and y-distance from the centre to the yellow dot, which is the cg of the top half of the circle $= \frac{4r}{3\pi}$, used for the first moment of inertia calculation, can be seen.

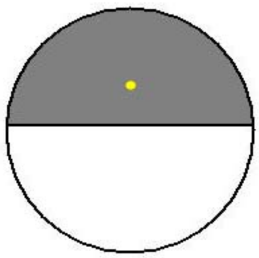


Figure 9.2: Explanation first moment of area.

Then using the Von Mises yield criterion formula, Equation 9.10, the yield strength can be determined.

[1] http://www.berkley-fishing.com/berkley-terminal-tackle-leaders-berkley-steelon/berkley-steelon-nylon-coated-wire/1285569.html

$$Y = \sqrt{\frac{1}{2}[(\sigma_x - \sigma_y)^2 + (\sigma_y - \sigma_z)^2 + (\sigma_z - \sigma_x)^2] + 3\tau_{xy}^2 + 3\tau_{yz}^2 + 3\tau_{xz}^2} \tag{9.10}$$

The formula for the yield criterion can be simplified by removing all the forces which are not present, this leads to Equation 9.11.

$$Y = \sqrt{3\tau^2} = \sqrt{3 \cdot (64.4 \cdot 10^6)^2} = 111.5MPa \tag{9.11}$$

The determined yield force is lower than that of steel, which is 250 MPa. As this yield criterion was determined for the worst case scenario, it is safe to assume that the rollout system will not fail.

9.2.2. Hyperflo

In this section the structural analysis of both the top and the bottom of the Hyperflo are discussed.

Bottom of the Hyperflo

The Hyperflo is attached to the other side of the line that is attached to the rollout system. Therefore to be in equilibrium it should also experience 19 N. From Figure 9.3 it can be seen that the force needs to be split into multiple vectors to determine the force F_2, exerted on the Hyperflo. Note that the Hyperflo is attached with eight lines.

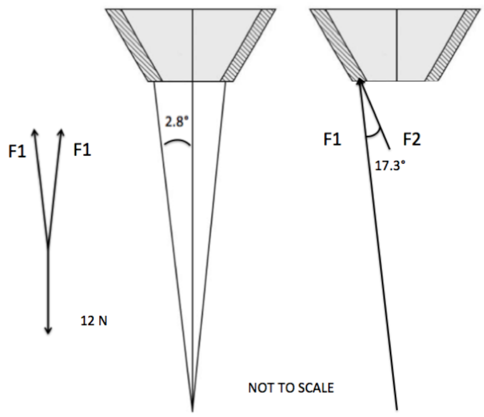


Figure 9.3: Hyperflo line angle.

To determine F_1 we decompose the 19 N tension into eight separate equal forces. This can be seen in Equation 9.12.

$$F_1 = \frac{19}{8 \cdot \cos(2.8)} = 2.4N \tag{9.12}$$

If one looks at the right side of Figure 9.3 the equation to determine F_2 is equal to Equation 9.13.

$$F_2 = \frac{F_1}{\cos(17.3)} = 2.5N \tag{9.13}$$

This means that the shear stress, that the adhesive along the length that the line is inserted (6.3 mm) into the Hyperflo (per line), should be able the stress found using Equation 9.14.

$$\tau_{min} = \frac{F_2}{\pi \cdot d \cdot l} = \frac{2.5}{\pi \cdot 1 \cdot 6.3} = 0.13N/mm^2 \tag{9.14}$$

The epoxy instant mix of Loctite[2] is able to withstand this shear stress (max $3200psi = 22N/mm^2$) and the temperatures experienced during the ballistic trajectory.

[2]http://www.loctiteproducts.com/tds/EPXY_5MIN_tds.pdf [cited 14 June 2016]

Top of the Hyperflo
The maximum drag force that is experienced during the ballistic trajectory is 20.6 N. This was determined by the re-entry simulation, as can be seen in Chapter 6.

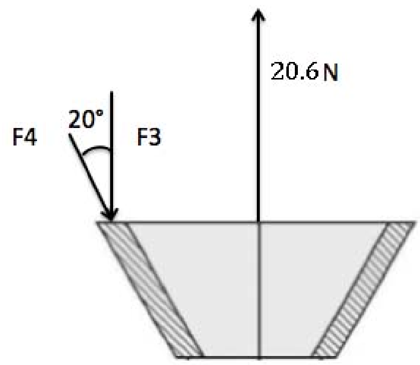


Figure 9.4: Hyperflo top forces.

To determine the force exerted on the steel pin, F_4, the drag force experienced by the Hyperflo needs to be decomposed. This is done using Equation 9.15.

$$F_3 = \frac{20.6}{8} = 2.6N \tag{9.15}$$

Then F_4 can be determined using Equation 9.16.

$$F_4 = F_3 \cdot \cos(20) = 2.4N \tag{9.16}$$

The shear stress, along half the length of the wall of the Hyperflo, experienced by the adhesive is determined using Equation 9.17.

$$\tau_{min} = \frac{F_4}{\pi \cdot d \cdot l} = \frac{2.4}{\pi \cdot 1 \cdot 6.3} = 0.12N/mm^2 \tag{9.17}$$

Again the epoxy instant mix of Loctite is more than sufficient to deal with the shear stresses experienced during the ballistic trajectory.

9.3. Assumptions

The assumptions made in this chapter will be presented in this section.

- The spring constant of the TubeSat body and nose cone are based on the assumption that they are modelled as axial beams. In general this does not change a lot in preliminary vibrational analysis, as this assumption is also used in wing discretisation.
- If the part can take the maximum force, it will be able to withstand any other force experienced during the trajectory. As the part can take the maximum force that it will experience during the trajectory it won't fail for any other (lower) force.

9.4. Recommendations

- The eigenfrequencies of both individual components and the system as a whole should be accounted for. As an example, ESA requires individual components to have a first natural frequency of at least 75Hz [19].
- Find the damping ratio of the sabots, from Sorbothane.
- Once the TubeSat has been designed to withstand the forces of the rocket launch, it should be tested in a shaker.
- Look for a less expensive adhesive, as the adhesive used for the hyperflo and the isolation is over-designed. Although this might be difficult due to the extreme temperature requirement.
- Test the rollout system and the Hyperflo, with a small scale launch (up to a lower height).

10
Sensitivity Analysis

Sensitivity analyses have been performed to study the effect of varying launch conditions. In this chapter the sensitivity analyses of the trajectory- and communications simulations are discussed. Section 10.1 contains an explanation of the trajectory simulation Monte Carlo analysis. Section 10.2 an explanation of the Monte Carlo analysis performed on the link budget calculations. A sensitivity analysis has also been performed for the trade-off of the descent concept. This was discussed at the end of Subsection 4.3.2 and is not repeated in this chapter.

10.1. Trajectory Simulation

In the design process of the DART payload, separate simulation software has been developed for the launch- and re-entry trajectory. A sensitivity study by Monte Carlo analysis is performed on the individual simulations. In the following subsections the input- and output parameters, as well as the Monte Carlo analysis results and recommendations, are discussed.

Launch Trajectory Parameters

In the sensitivity analysis of the launch trajectory the parameters under study are: apogee altitude, apogee range. Furthermore, the following input parameters are varied: booster thrust, burn time, dart mass, air density scaling factor, wind speed, launch angle. Not included in this study are, most significantly, the variation of air temperature, manufacturing errors that cause variations in frontal area and drag coefficient, aerodynamic heating, variations of propellant- and booster mass.

The sensitivity analysis is performed on a relatively coarse grid to save computation time; unless stated otherwise, every input parameter is varied over three values: the expected value ± an offset. Input parameter ranges are argued in the following.

Booster thrust
Booster thrust is heavily affected by the surrounding air temperature. As such, it is important to account for variations in booster thrust. The booster thrust is ranged from 9500 N to 10500 N. Note that this implies using a 'block thrust' model, which is not a very accurate approach; see Chapter 11: Figure 11.3 for an accurate sea level thrust graph of the Super Loki sounding rocket.

Burn time
The apogee altitude of 133.3 km relies on a very specific set of booster conditions. Among these conditions is the burn time, which directly influences the specific impulse, and as such can make a large difference in the final apogee altitude. For the scope of this sensitivity analysis, it is ranged from 4.5 s to 5.5 s.

Dart mass
The expected mass of the dart is 3.75 kg. This number is based on CAD model weight analysis of the TubeSat and an empty dart weight of 3.0 kg. If the mass deviates from the expected value it will most likely be higher, since the TubeSat might be filled with resin to counter vibrations, see Chapter 9. Additionally, wiring could add weight as the design is not yet completely worked out. In the sensitivity analysis the mass is varied from 3.6 – 4.0kg.

Air density scaling factor
Air density changes with pressure and temperature. Warmer climates will have a lower-than-average air density, whereas colder climates can have a higher-than-average air density (average is taken as 1.225 $\frac{kg}{m^3}$). To account for different launch climate conditions, the air density is scaled by a factor k. This is a somewhat erroneous approach, as there is no linear relation between the temperature at sea level and at e.g. 100 km altitude. The scaling factor is ranged from 0.9 to 1.1 to account for both warmer and colder launch climates.

Wind speed
The effect of wind is an important consideration in any launch. As many rockets tend to 'turn into the wind', wind can deviate the trajectory significantly, and as such cause serious delays on launch days. In this sensitivity

analysis the ground wind speed is ranged from 0 to 10 m/s. This range is based on [8]. On the Beaufort scale 0, 5 and 10 m/s represent no wind, a gentle breeze and a fresh breeze, respectively.[1]

Launch angle
The launch angle of a sounding rocket depends mostly on atmospheric conditions such as the wind speed, but other aspects like the expected apogee range can be taken into account as well. Based on indications by TME, the launch angle is ranged from 75° to 88° with respect to the horizontal.

Monte Carlo Results: Launch Trajectory

The results of the Monte Carlo analysis of the launch trajectory are presented in Figure 10.1. Critical data points are those with a low apogee altitude, and those with a large apogee range. In the figure, these points are located in the bottom and to the right of the data set, respectively. The lowest apogee altitude reached is 49.7 km, and the largest apogee range 163.8 km. More information about these critical data points can be found in Table 10.1.

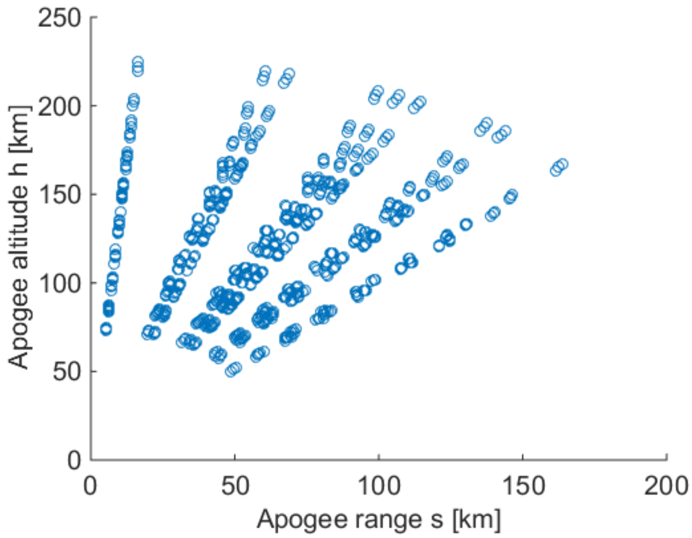


Figure 10.1: A scatter plot of the apogee altitude and -range for all parameter combinations.

Table 10.1: Critical scenario's for apogee altitude and -range.

Parameter	Critical apogee altitude	Critical apogee range
Apogee altitude h [km]	**49.7**	166.9
Apogee range s [km]	48.7	**163.8**
Booster thrust T_{boost} [N]	9500	10500
Burn time t_b [s]	4.5	5.5
Dart mass m [kg]	3.6	3.6
Air density scaling factor k_ρ [-]	1.1	0.9
Wind speed v_{wind} [m/s]	10	10
Launch angle θ [deg]	75	75

Some comments can be made about the launch trajectory Monte Carlo results:

- Figure 10.1 shows a clear division of the data points into 'lanes' that 'point' towards the origin. Upon closer inspection, nine 'lanes' can be distinguished, each of which corresponds to a combination of wind speed and launch angle. This makes sense, since the sensitivity analysis is performed for three wind speeds and three launch angles, giving a total of nine possible combinations. The 'lane' closest to the y-axis corresponds to a 88° launch angle and 0 m/s wind speed, whereas the 'lane' furthest from the y-axis corresponds to a 75° launch angle and 10 m/s wind speed.
- In addition to the 'lanes', Figure 10.1 shows data points accumulating in groups within their respective lanes. These groups correspond to combinations of booster thrust, burn time, dart mass and air density scaling factor. Again, a logical kind of 'grouping' occurs; for example, the group of data points located to the far

[1] Retrieved from http://www.spc.noaa.gov/faq/tornado/beaufort.html on 18-06-2016

right in Figure 10.1 all follow from parameter combinations favourable for large apogee range. One of the data points in this 'group' is described as 'critical apogee range' in Table 10.1.
- From Table 10.1, the lowest apogee altitude is 49.7 km, which corresponds to a fall time of 165.5 seconds. This scenario only occurs for the unlikely combination of low booster thrust, low burn time, thick air, high wind and small launch angle. Even so, the data communication process is not endangered (see Chapter 7).
- From Table 10.1, the largest apogee range is 163.8 km. The horizontal travel is calculated as twice the apogee range. This is a very conservative approach, as the horizontal distance component during re-entry will likely be much less due to the absence of a thrust component in the horizontal direction.
- In Figure 10.1, the highest apogee altitude is 224.7 km. This altitude is reached for a combination of high booster thrust, long burn time, thin air, zero wind speed and 88° launch angle.
- The effect of wind drift and rotation of the Earth are not taken into account in the simulation.

Re-entry Trajectory Parameters

In the sensitivity analysis of the re-entry trajectory the parameters under study are: fall time, stagnation point temperature. Furthermore, the following parameters are varied: TubeSat mass, apogee altitude, TubeSat drag coefficient scaling factor, hyperflo parachute drag coefficient scaling factor, air density scaling factor, air temperature scaling factor, emissivity, nose cone radius. These input parameters are chosen for their direct influence on the fall time and stagnation point temperature.

The sensitivity analysis is performed on a relatively coarse grid to save computation time; unless stated otherwise, every input parameter is varied over three values: the expected value ± an offset. Input parameter ranges are argued in the following.

Mass
The expected mass of the satellite is 0.75 kg. This number is based on CAD model weight analysis. If the mass deviates from the expected value it will most likely be higher, since the TubeSat might be filled with resin to counter vibrations, see Chapter 9. Additionally, wiring could add weight as the design is not yet completely worked out. In the sensitivity analysis the mass is varied from 0.4 – 1.2kg in steps of 0.1 kg.

Apogee altitude
The apogee altitude is a consequence of the launch conditions. The expected apogee altitude is 133.3 km, but the apogee will likely be lower due to 3D effects that are unaccounted for in the launch simulation. Therefore, the apogee altitude is ranged 110 km - 133.3 km.

TubeSat drag coefficient scaling factor
The TubeSat has a slender, cylindrical shape with a rounded nose. [22] describes this shape and its drag coefficient, which varies strongly with Mach number. Because of this variation, an expected value can only be given over the whole Mach range. This value is then scaled using a factor k. This scaling factor ranges from 0.9 to 1.1.

Hyperflo parachute drag coefficient scaling factor
[50] describes the variation of the drag coefficient with Mach number for a hyperflo parachute. As with the TubeSat drag coefficient, a scaling factor k is used. This scaling factor ranges from 0.9 to 1.1.

Air density scaling factor
Air density changes with pressure and temperature. Warmer climates will have a lower-than-average air density, whereas colder climates can have a higher-than-average air density (average is taken as 1.225 $\frac{kg}{m^3}$). To account for different launch climate conditions, the air density is scaled by a factor k. This is a somewhat erroneous approach, as there is no linear relation between the air density at sea level and at e.g. 100 km altitude. The scaling factor is ranged from 0.9 to 1.1 to account for both warmer and colder launch climates.

Air temperature scaling factor
The air temperature influences the speed of sound. As a consequence, on a cold day higher Mach numbers will be achieved, and vice versa. To account for different air temperature conditions, the air temperature is scaled by a factor k. This is a somewhat erroneous approach, as there is no linear relation between the air temperature at sea level and at e.g. 100 km altitude. The scaling factor is ranged from 0.9 to 1.1 to account for both warmer and colder launch climates.

Emissivity
As stated in Chapter 5, the nose cone is made out of zirconia. The emissivity of zirconia can range from 0.4 – 0.95 depending on, amongst others, the temperature of the material [61].

Nose cone radius
The nose of the TubeSat is a semi-sphere, with a slenderness ratio of 0.5. This comes down to a nose radius of 15 mm. A smaller nose cone radius increases drag unnecessarily, whereas a too slender nose takes up valuable space. The nose cone radius is therefore ranged from 10 to 20 mm.

Monte Carlo Results: Re-entry Trajectory

The results of the Monte Carlo analysis of the re-entry trajectory are presented in Figure 10.2. Critical data points are those with a high equilibrium temperature, and those with a low fall time. In the figure, these points are located to the top and to the left, respectively. The highest equilibrium temperature achieved is 2017 K, and the lowest fall time 185 seconds. More information about these critical data points can be found in Table 10.2.

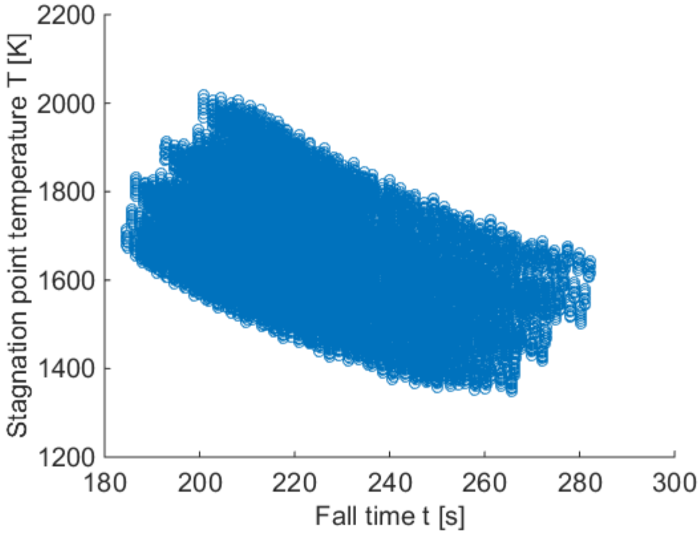


Figure 10.2: A scatter plot of the stagnation point (equilibrium) temperature and fall time for all parameter combinations.

Table 10.2: The parameter combinations most critical for fall time and equilibrium temperature.

Parameter	**Critical fall time**	**Critical stagnation temperature**
Fall time t	**184.6**	200.9
Stagnation temperature T [K]	1714	**2017**
Mass m [kg]	1.2	1.2
Apogee altitude h [km]	110	133.3
TubeSat drag coefficient scaling factor k_{cd} [-]	0.9	0.9
Hyperflo parachute drag coefficient scaling factor $k_{cd,para}$ [-]	0.9	0.9
Air density scaling factor k_ρ [-]	0.9	0.9
Air temperature scaling factor k_T [-]	0.9	0.9
Emissivity ϵ [-]	0.9	0.9
Nose cone radius r_n [m]	0.02	0.01

Some comments can be made about the re-entry trajectory Monte Carlo results:

- Although somewhat cluttered, Figure 10.2 shows a division of the data points into 'lanes'. Each 'lane' corresponds to a combination of drag coefficient scaling factors (TubeSat and hyperflo) and apogee altitude. There are 3^3 or 27 'lanes'.
- In addition to the 'lanes', Figure 10.2 shows data points accumulating in groups within their respective lanes. These groups correspond to combinations of mass, emissivity, nose cone radius and atmospheric scaling factors (air density and air temperature).
- Table 10.2 shows the data point with the lowest fall time. The combination of a high mass, low apogee altitude, low drag force and thin air is logical to produce the lowest fall time. However, one would not expect a large nose cone radius to belong to the most critical (lowest) fall time. More about this in Chapter **??**.
- Table 10.2 also shows the data point with the highest stagnation point temperature. The combination of a high mass, high apogee altitude, low drag force, thin air, low speed of sound (causing higher Mach numbers) and small nose cone radius is logical to produce the highest stagnation point temperature.
- A stagnation point temperature of 2017 K is too high for the internal instruments (see Chapter 6). However, as stated before, this temperature only occurs for a very unfortunate combination of parameters. The system needs to be re-evaluated before launch to verify that such a combination of parameters cannot occur.

Recommendations

The following recommendations are made for the trajectory sensitivity analysis:

- A more refined sensitivity analysis needs to be performed. This means both more data points, and a more accurate range of parameters. Such an analysis can be very computationally expensive, so an efficient scheme should be selected on beforehand.
- Future sensitivity analyses should not include unrealistic combinations of input parameters. For example, a 88 degree launch angle in combination with a high horizontal wind speed does not make sense. In case of high horizontal wind speeds the launch rails are pointed more into the wind.
- A more accurate calculation of the re-entry horizontal travel is required.
- The effect of wind drift and rotation of the Earth should be taken into account in the simulation.

10.2. Communications Simulation

The link budget is depended on several critical parameters namely: transmission power, gain of the transmitter, angle of launch, height of ejection and the acquired data rate. Of course there are several other factors like the gain of the ground station, losses due to atmospheric conditions and pointing losses. However as discussed in 7.3, losses due to atmospheric conditions and pointing losses have been taken in to account in a conservatively. The gain of the ground station is assumed to be correctly stated by ESC and therefore taken as a constant. In order to do a complete sensitivity analysis of the code a Monte Carlo simulation was performed, varying the critical parameters while looking at the final amount of data sent. In the following table the range with which the parameters were varied and their nominal value can be seen.

Table 10.3: Range of variation of communication parameters.

Parameter	Range		Nominal	
Transmission power	230 - 330	mW	330	mW
Added transmission antenna gain	-3 - 0	dB	0	dB
Data rate	6 - 10	$Mbps$	8	$Mbps$
Angle of launch	80 - 88	°	88	°
Height of ejection	100 - 120	km	120	km

As can be seen in the table, some parameters are only examined in a range less than the nominal value. This mainly to safe time while running the simulation since it is known that an increase in these parameters only has a positive effect on the link budget. The range of variation are explained below.

- The transmission power is mainly depended on the battery. A batteries performance tends to become less when it is not fully charged. The transmission system will not be the only subsystem needing power, this might also influence the acquired power. Therefore quit a large deviation from the nominal has been taken (30%).
- The transmission antenna gain has a very large deviation. An addition of -3 dB to the antenna gain means the output signal strength is halved. This large deviation was chosen because there will be some interference due to metal components in the satellite. Also the antenna will be placed right behind the camera. It is unsure how much this would affect the antenna. It could have a large effect on the final antenna gain, for that reason this high range was chosen.
- The data rate was varied positively and negatively compared to the nominal value. This is because a higher data rate could actually be counter productive since it results in a lower SNR. Thus a higher data rate can make the SNR lower than what is required and therefore make the data stop sending all together. The upper limit of 10Mbps was taken because this corresponds to the maximum bandwidth provided by Esrange and QPSK modulation. The offset for the lower limit was taken equal to the upper limit.
- The initial angle of launch is a very important parameter. It influences the total distance between the TubeSat and ground station which is the biggest loss in the link budget. The change of only 1° is already several extra kilometres in absolute distance. The reasoning for such a large deviation is the fact that in the simulation winds are not taken in to account. By taking this large deviation in angle it accounts for winds which might 'push' the TubeSat or dart away from the ground station during ascent and descent.
- The height of ejection is assumed to be after or at apogee, a lower altitude will mean a lower transmission time. As the dart has never actually flown to this height the risk of something going wrong is still quit high. Thus also a large range has been used for the height of ejection.

The Monte Carlo simulation was performed in the following way: all parameters were cross-varied, the combination of parameters that were unable to send at least 25 MB of data were recorded because this would mean a total failure (not being able to send the video at least once). This resulted in 3664, out of 11664, combinations which were unable to send 25 MB with a minimum SNR of 12.6 dB. To visualise the influence of each parameter a plot was made for each one, with on the y-axis the magnitude of that particular parameter and on the x-axis the combination 'number'. So when looking at the plots every x value represents one combination of parameters which was unable to send 25MB. From the 5 plots created, the height of ejection did not have any influence, the entire range of heights had equal amount of 'failure' combinations, thus not having any influence. The other plots however do not have an equal distribution over their range. They are plotted in Figure 10.3. From the four plots it can be deduced that there is one critical parameter, namely: the launch angle θ. All the plots have failures over their entire range of values except the launch angle. Therefore the launch angle must have the most influence on the link budget out of all these parameters and thus is the critical parameter. The other parameters show logical results. The power and gain have more failure combination at a lower power and at a lower gain. The data rate has more failure combinations for a higher data rate, this is because a higher data rate lowers the final SNR. This can result in a SNR less than 12.6 which results in non useful data. Therefore the higher data rates have more failure combinations.

It is very well possible that during the actual launch the launch angle will not be optimal. Because it has such a large influence on the link budget it was decided to design the communications such that the link budget closes for all the launch angles (80° - 88°, assuming nominal values for the other parameters). Thus the launch angle variation was implemented in the code, this can be seen in the graphs shown in Chapter 7.

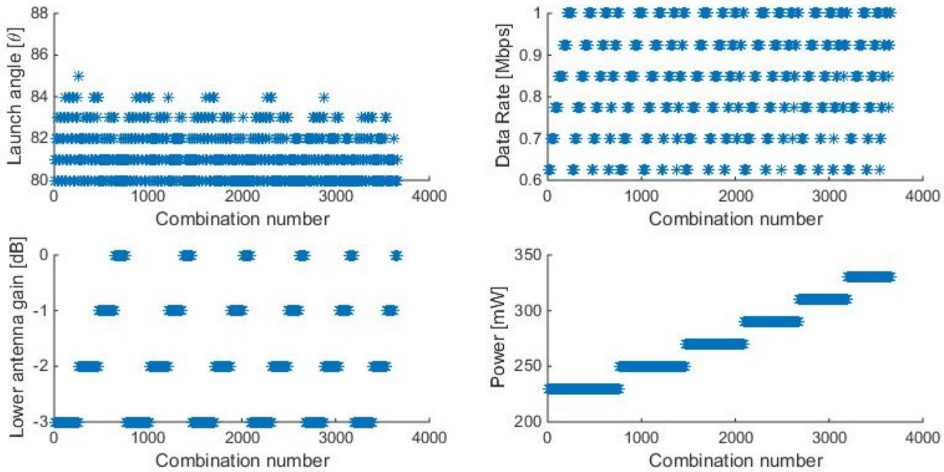


Figure 10.3: All 3664 parameter combinations of which the total data sent was less than 25 MB.

11

Verification & Validation

Verification and validation is a crucial part of any design. It helps to determine if the problem is being solved in the right way (verification), and if the right problem is being solved (validation). The V&V process will be done in a number of steps, based on [12, 23]. In Section 11.1, the process of requirements validation is explained. Section 11.2 covers the verification and validation of the trajectory simulation. Section 11.3 covers the verification and validation of the link budget. Finally, Sections 11.4-11.5 discuss the plan of attack for the satellite's verification and validation respectively.

11.1. Requirements Validation

Validation of requirements is done to check whether the requirements are correctly formulated, following the VALID criteria. VALID stands for Verifiable, Achievable, Logical, Integral, Definitive. The requirements were phrased with this in the mind, and afterwards also checked.

11.2. Trajectory: Verification & Validation

This section covers the verification and validation of the trajectory simulations.

Launch Verification

The launch verification consists of three parts: a convergence study, a predictability analysis, and comparison with TME data of the dart mass effect.

Convergence study

The launch simulation is tested for convergence by running the code for increasingly fine time steps. An overview of the results is presented in Table 11.1. The convergence study is performed for the usual booster configuration, i.e. $F_{booster}$ = 9967 N, t_b = 5 s, m_{prop} = 22.6 kg.

Table 11.1: Launch trajectory convergence study results.

Time step [s]	Apogee altitude [km]	Change wrt previous time step [%]
1.0	119.8	
0.5	126.5	+ 5.6
0.1	143.9	+ 13.8
0.05	138.1	- 4.0
0.01	133.3	- 3.5
0.005	132.7	- 0.5
0.001	132.1	- 0.4

From Table 11.1, the following observations can be made. From time step 1.0 s, the apogee altitude increases to a maximum of ± 144 km at time step 0.1 s. After this increase, the apogee altitude converges to a value of ± 132 km at time step 0.001 s.

Hence, the following conclusions can be drawn about the simulation. Apparently, for time steps larger than 0.1 the simulation is not convergent. This could be due to the fact that a coarse time grid 'skips' over important instances in the launch, such as burnout or leaving the launch rails. The simulation should only be used with a time step smaller than 0.1. In this report, all launch results are obtained for a time step of 0.01, since the apogee error is < 1.0% compared to the 'asymptotic' apogee value of 132.1 km.

Predictability analysis

The behaviour of the launch simulation is analysed. A selection of parameters is varied, the expected effect of the variations on apogee altitude is logged, and results are compared to these expectations. An overview of the varied parameters, expected effects and comply checks is given in Table 11.2.

The predictability analysis is similar, but not equal to, a sensitivity analysis (see Chapter 10). The aim of a sensitivity analysis is to find out what combination of parameters gives the least favourable result. In a predictability analysis, however, the aim is only to verify that the simulation behaves rationally, i.e. according to expectations.

In Table 11.2, the selection of parameters is based on direct effect on apogee altitude. Furthermore, scaling factors of 0.9 and 1.1 are used to check the effect of a parameter decrease and increase, respectively.

Note that launch angles higher than 90° are not possible, so rather than using a scaling factor, the launch angle is varied as 87, 88 and 89 degrees. Also, the nominal value of the wind speed is 0 m/s, so using a scaling factor is not possible there. Rather, the wind speed is varied as -1.0 and 1.0 m/s. Finally, the expected effect of dart mass is not shown. This is done because there is an optimum dart mass for apogee altitude, which will be discussed in the next subsection.

Table 11.2: Effect of parameter variations on apogee altitude.

Parameter ***Baseline***	**Scaling factor/Value**	**Expected effect**	**Apogee altitude [km]** ***133.3***	**Comply**
Air density	0.9	Increase	148.0	✓
	1.1	Decrease	119.1	✓
Thrust	0.9	Decrease	95.7	✓
	1.1	Increase	176.1	✓
Burn time	0.9	Decrease	98.2	✓
	1.1	Increase	172.7	✓
Dart mass	0.9		133.2	
	1.1		132.4	
Dart frontal area	0.9	Increase	139.5	✓
	1.1	Decrease	127.3	✓
Drag coefficient	0.9	Increase	148.0	✓
	1.1	Decrease	119.1	✓
Launch angle	87 [°]	Decrease	133.0	✓
	89[°]	Increase	133.5	✓
Wind speed	-1 [m/s]	Increase	133.6	✓
	1 [m/s]	Decrease	133.0	✓

Dart mass effect comparison

The final step in the launch simulation verification is comparing results with independent simulation results. Figure 11.1, provided by TME, shows the influence of the total dart mass (empty dart + payload) on apogee altitude. It can be seen that there is an optimal total dart mass for which the highest apogee is reached. This optimum is achieved for a total dart mass of about 3.8 kg.

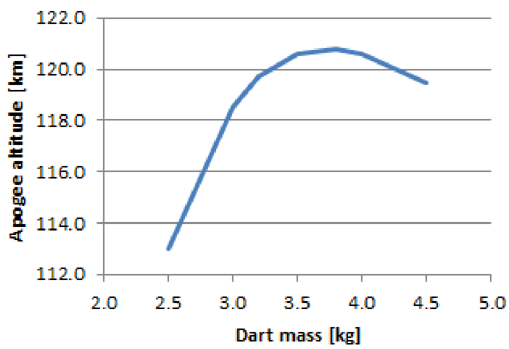


Figure 11.1: The influence of the total dart mass on apogee altitude. A courtesy of TME.

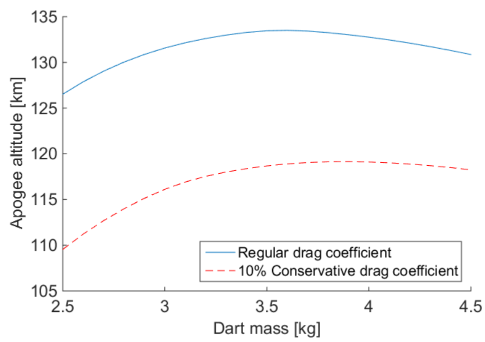


Figure 11.2: The influence of the total dart mass on apogee altitude. Launch simulation results.

Figure 11.2 ('Regular drag coefficient' curve) shows the dart mass effect, resulting from the launch simulation. Again, there is a clear optimum for the total dart mass, but here it is located at 3.6 kg.

There are two notable differences between between Figures 11.1 and 11.2: the maximum apogee altitude (121 km versus 133.5 km) and the optimal total dart mass (3.8 kg versus 3.6 kg). These differences are likely caused by the conservative approach of TME. In their simulation, margins are used on the drag coefficient of up to 10 %. This conservative approach on the drag coefficient is applied to the launch simulation as well. The '10% conservative drag coefficient' curve in Figure 11.2 is the resulting curve. It has an apogee altitude of 120 km and an optimal total dart mass of 3.9 kg, a result that resembles the TME curve much more than the 'regular drag coefficient' curve.

Launch Validation

The launch validation consists of three parts: a comparison to Super Loki flight data, a Low-Altitude Test (LAT), and finally comparison to DART launch flight data.

Comparison to Super Loki flight data

The launch simulation is validated by comparison to the Super Loki Robin Dart (SLRD) sounding rocket. SLRD is a further developed version of the post-war Loki sounding rockets, and around 9000 have been delivered between its introduction in 1968 and its retirement in 2001. It is capable of reaching altitudes of up to 120 km [7].

Table 11.3 shows the booster and dart parameters of SLRD. Note that the table shows an average value and a maximum value for the thrust. This is because the thrust varies strongly over the booster's 2.11 second burn time, see Figure 11.3.

Table 11.3: Important parameters of the Super Loki Robin Dart.

Parameter	**Value**	**Imperial unit**	**Value**	**Metric unit**
Dart diameter	1.625	in	41.28	mm
Dart mass	13.50	lb	6.124	kg
Booster diameter	4.000	in	101.6	mm
Booster mass (full)	50.87	lb	23.07	kg
Booster mass (empty)	13.36	lb	6.060	kg
Propellant mass	37.51	lb	17.01	kg
Average thrust	4021	lbf	17885	N
Maximum thrust	5524	lbf	24571	N
Burn time	2.110	s	2.110	s

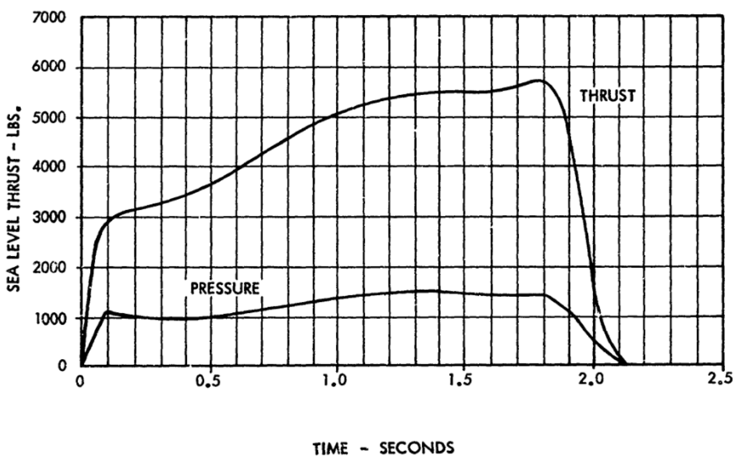


Figure 11.3: Variation of Super Loki sea level thrust with burn time.

The parameters from Table 11.3 are inserted in the launch simulation, and results are compared to SLRD flight data found from [7], see Table 11.4.

Table 11.4: Error between real ascent data and simulation results.

	Apogee altitude [km]	Apogee range [km]
Real	113.4	41.8
Simulation	111.8	40.7
Error	**1.4%**	**2.6%**

In the SLRD flight data, the apogee altitude is 372,000 ft or 113.4 km. The launch simulation yields an apogee altitude of 111,8 km. The error is 1.4%.

Low-Altitude Test (LAT)

To validate the DART launch simulation without the cost associated with a high altitude launch, i.e. to 120 km altitude, a Low-Altitude Test (LAT) should be performed on a military site in the Netherlands. More about this test in Section 11.6.

Comparison to DART launch flight data

The final step in the launch simulation validation is to compare the simulation results to DART flight data from the launch to 120 km altitude. Although this is in a way a reflective step it is a crucial one, because it is the only time that the predicted behaviour can directly be compared to the actual behaviour.

Re-entry Verification

The re-entry verification consists of three parts: a convergence study, a predictability analysis, and a comparison to widely-used fall time calculators.

Convergence study

The launch simulation is tested for convergence by running the code for increasingly fine time steps. An overview of the results is presented in Table 11.5. The convergence study is performed for a satellite mass of 0.75 kg and an initial altitude of 133.3 km.

Table 11.5: Results from convergence study of re-entry simulation.

Time step [s]	Fall time [s]	Change [%]	Stagnation Temperature [K]	Change [%]
1.0	228		1779	
0.5	227.5	- 0.2	1758	- 1.2
0.1	227.5	0	1742	- 0.9
0.05	227.5	0	1740	- 0.1
0.01	227.5	0	1738	- 0.1
0.005	227.5	0	1738	0
0.001	227.5	0	1738	0

From Table 11.5, the following observations can be made. The fall time converges immediately when moving from a 1.0 s to a 0.5 s time step. It then stays pretty much constant for all smaller time steps. Similarly, the stagnation temperature converges very quickly; a time step of 0.5 s is already within 1.2% of the 'asymptotic' stagnation temperature value.

Hence, the following conclusions can be drawn about the simulation. Apparently, the re-entry simulation is convergent for all time steps equal to or smaller than 1.0. All re-entry results are obtained for a time step of 0.1 s, since for this time step the stagnation temperature results are within one percent of the 'asymptotic' stagnation temperature value.

Simulation predictability analysis

The behaviour of the re-entry simulation is analysed. A selection of parameters is varied, the expected effect of the variations on fall time and stagnation point temperature is logged, and results are compared to these expectations. An overview of the varied parameters, expected effects and comply checks is given in Table 11.6.

In Table 11.2, the selection of parameters is based on direct effect on fall time and/or stagnation point temperature. Scaling factors of 0.9 and 1.1 are used to check the effect of a parameter decrease and increase, respectively.

Note that the effects of air temperature and nose cone radius on fall time do not comply with expectations. A decrease in air temperature increases the Mach number and thus the aerodynamic drag, so is expected to increase the fall time, and vice versa. However, the fall time is decreased. Furthermore, a decrease in nose cone radius lowers the friction drag coefficient (see Chapter 6), so is expected to decrease fall time, and vice versa. However, the fall time is increased.

These counter-intuitive effects may be explained by the following. As stated before, a decrease in nose cone radius lowers the friction drag coefficient, causing a higher acceleration and consequently a higher Mach number. It is possible that the effect of an increased Mach number is stronger than the effect of a lower friction drag coefficient. After all, the Mach number is present in the equations of all TubeSat and hyperflo drag coefficient components, while the nose cone radius is only present in the friction drag coefficient (see Chapter 6). This would explain the increase in fall time associated with a smaller nose cone radius, but does not explain the decrease in fall time for lower air temperatures.

Table 11.6: Effect of input parameter variation on fall time and equilibrium temperature.

Parameter	**Scaling factor**	**Expected effect on fall time**	**Fall time [s]**	**Comply**	**Expected effect on temperature**	**Equilibrium temperature [K]**	**Comply**
Baseline			227.5			1741.5	
Air density	0.9	Decrease	222.2	✓	Increase	1747.3	✓
	1.1	Increase	232.6	✓	Decrease	1736.2	✓
Air temperature	0.9	Increase	227.4		Increase	1750.7	✓
	1.1	Decrease	227.5		Decrease	1734.8	✓
Initial altitude	0.9	Decrease	219.4	✓	Decrease	1650.0	✓
	1.1	Increase	235.2	✓	Increase	1829.3	✓
Satellite frontal area	0.9	Decrease	225.5	✓	Increase	1759.7	✓
	1.1	Increase	229.5	✓	Decrease	1724.6	✓
Hyperflo frontal area	0.9	Decrease	224.3	✓	Increase	1751.6	✓
	1.1	Increase	230.7	✓	Decrease	1731.8	✓
Satellite drag coefficient	0.9	Decrease	225.5	✓	Increase	1759.7	✓
	1.1	Increase	229.5	✓	Decrease	1724.6	✓
Hyperflo drag coefficient	0.9	Decrease	224.3	✓	Increase	1751.6	✓
	1.1	Increase	230.7	✓	Decrease	1731.8	✓
Satellite mass	0.9	Increase	233.0	✓	Decrease	1713.6	✓
	1.1	Decrease	222.8	✓	Increase	1767.1	✓
Hyperflo mass	0.9	Increase	227.7	✓	Decrease	1740.8	✓
	1.1	Decrease	227.4	✓	Increase	1742.2	✓
Nose cone radius	0.9	Decrease	227.8		Increase	1758.9	✓
	1.1	Increase	227.2		Decrease	1726.1	✓
Emissivity	0.9	Increase	227.5	✓	Increase	1788.0	✓
	1.1	Decrease	227.5	✓	Decrease	1700.5	✓

Comparison to a widely-used fall time model

Many fall time calculation methods can be found, varying strongly in accuracy. Equation 11.1 seems to be one of the most widely used models in online fall time calculators[1,2], lecture slides and on scientific fora[3]. It gives the fall time as a function of initial height.

$$t_{fall} = \sqrt{\frac{m}{gk}}\,\text{acosh}\left(e^{\frac{hk}{m}}\right) \tag{11.1}$$

where t_{fall} is the fall time in s, m is the mass of the falling object in kg, g is the gravitational acceleration in m/s^2, k is a constant equal to $\frac{1}{2}\rho C_D A$, where ρ is the air density in kg/m^3, C_D is the drag coefficient of the falling object and A is the frontal area of the falling object in m^2, and h is the initial height in m.

The outcome of Equation 11.1 is compared to that of the re-entry simulation. To this purpose, the drag coefficient, gravitational acceleration and air density are kept constant. Input parameters are: $m = 1\ kg$, $g = 9.81\ m/s^2$, $k =$

[1] http://keisan.casio.com/exec/system/1231475371 [cited 20 May, 2016]
[2] http://hyperphysics.phy-astr.gsu.edu/hbase/mechanics/fallq.html [cited 20 May, 2016]
[3] https://www.grc.nasa.gov/www/k-12/airplane/termv.html [cited 20 May, 2016]

$(\frac{1}{2})(1.225)(0.5)(4\pi 0.1^2)$ and $h = 1000\ m$. This essentially simulates the fall of a 1 kg sphere from an altitude of 1 km. Equation 11.1 yields a fall time of t_{fall} = 63.76 s. In comparison, the re-entry simulation yields a fall time of t_{fall} = 63.8 s for a time step of 0.1 s under the same conditions. The error between the numerical and analytical results is 0.06%. Smaller time steps yield an even lower error.

Re-entry Validation

The re-entry validation consists of three parts: a comparison to the Mk.82 bomb, a Low-Altitude Test (LAT), and finally a comparison to DART launch flight data.

Comparison to Mk.82 bomb

Validation of the re-entry simulation is done by comparison to a Mk.82 bomb. This bomb is a well-documented, publicly available military application. The US-made Mark 82 (Mk.82) bomb is an unguided general purpose bomb that has been used by all branches of the US military, as well as some of its international allies.

The following scenario has been found in a report on the Mk82 bomb, released in 1997 by the Weapons Systems Division (WSD) of the Australian Aeronautical and Maritime Research Laboratory (AMRL) [33]. A 500 lb Mk82 GPLD (General Purpose Low Drag) bomb is dropped from a F/A-18 Hornet fighter aircraft at an altitude of 10,000 ft. At the moment of release, the aircraft is diving 42 degrees from the horizontal and has a true airspeed (TAS) of 630 ft/s. The fall time is approximately 16 seconds.

This bomb run result is now compared to results produced by the re-entry simulation. The Mk.82 bomb properties are retrieved from [33] and put into the model. the numerical simulation yields a fall time of t = 15.8 s for a time step of 0.1 s. The error compared to the bomb run is 1.25%. Smaller time steps yield a somewhat smaller error.

Low-Altitude Test (LAT)

To validate the re-entry simulation without the cost associated with a high altitude launch, i.e. to 120 km altitude, a Low-Altitude Test (LAT) should be performed on a military site in the Netherlands. More about this test can be read in Section 11.6.

Comparison to DART rocket launch flight data

The final step in the re-entry simulation validation is to compare simulation results to DART flight data from the launch to 120 km altitude. Although this is in a way a reflective step it is a crucial one, because it is the only time that the predicted behaviour can directly be compared to the actual behaviour.

11.3. Link Budget: Verification & Validation

In this section the link budget simulation will be verified & validated. Firstly unit verification will be shown, than the code will be validating using the data of the Delfi-n3xt.

The equation used for the link budget is:

$$SNR = \frac{E_b}{N_0} = P_t + G_{AT} + L_l + G_{AR} + L_r + L_{pr} - L_{fs} - k - T_{sys} - R_b \tag{11.2}$$

The variables should be implemented as dB. The following parameters were estimated using references: G_{AT}, L_l, G_{AR}, L_r, L_{pr}, k, T_{sys}. As these were given in dB their implementation was straight forward, thus no unit verification is needed for these inputs. The parameters left (P_t, L_{fs}, R_b) required either a calculation or a conversion. The unit verification of these parameters is shown in the following table.

Table 11.7: Unit verification

	Formula	Inputs	Analytical Answer	Simulation answer
$P_t[dBW]$	$10log_{10}(P(W))$	$P = 1W$	0 dBW	0 dBW
$R_b[dBBps]$	$10log_{10}(R(Bps))$	$R = 10\cdot 10^6 Bps$	70 dBBps	70 dBBps
$L_{fs}[dB]$	$10log_{10}((\frac{4\pi df}{c})^2)$	$d = 100.000m$ $f = 2.4GHz$ $c = 3\cdot 10^8 m/s$	140.045 dB	140.45 dB

As can be seen form the table the values match exactly thus the unit conversions were correct. As said before the validation of the code will be done using the data of the Delfi-n3xt [57]. The losses for the Delfi-n3xt were implemented in the code, the results of the code and the real answer are shown.

Table 11.8: Validation results

Parameter	Value
Transmission frequency	146 MHz
Distance	850 km
Transmission power	0.2 W
Data Rate	1200 symbols/s
Transmission losses	-14.84 dBW
Free space Loss	-141.46 dB
Other path losses	-2.3 dB
Receiver gain	11.46 dB
Receiver noise temperature	264 K
SNR from report	**26.45 dB**
SNR from code	**26.50 dB**

The error is only 0.2%, which is expected for a simple code as this. The origin of the error is suspected to be a combination of rounding errors and truncation errors.

11.4. Product Verification

After the verification of the numerical model is performed, a check is performed whether the TubeSat meets the requirements imposed on it. Depending on the requirement, different verification methods are used. These methods are described below. They are taken from [12, 23].

- **Inspection**
  Inspect the design documentation or the product to show compliance with the requirement. This method could be used for requirements on size.
- **Analysis**
  Establish, by mathematical or other analysis techniques, that the product complies with the requirement. This method could be used for requirements on the camera.
- **Demonstration**
  Establish, by operation, adjustment or reconsideration of a test article, the satellite's compliance with a requirement. This method can be used for requirements on the thermal range to be withstood by the satellite.
- **Test**
  Test the compliance of (a representative model of) the product with a requirement under representative conditions. This method can be used for requirements concerning the flight conditions.

For each requirement these verification steps are performed:

1. The objectives of the verification task are established.
2. The type of verification is determined.
3. The required inputs and outputs are determined.
4. The potential risks associated with partial fulfilment or not fulfilment of the requirements are assessed.
5. A decision on steps, if the verification fails, is made.
6. The verification activities are planned.

First, the components requirements are verified. However, as most of the components are COTS components, the verification does not need to be performed on them. This is because the components are already tested by the manufacturer, and they fulfil the specifications given. Second, the subsystem and sustainability requirements are verified. Last, the functional requirements are verified. For all the requirements, redesign is recommended in case the verification fails, however, for some of them, it is possible to perform the mission nevertheless.

- **DART-Pow-Sys-2** **DART-ADCS-Pow-1** **DART-DCS-Pow-1**
  DART-Emb-Pow-1 **DART-Vid-Pow-1** **DART-Trans-Pow-1**
  Component off-the-shelf (DC-DC converter), does not need to be tested.

- **DART-Pow-Sys-3** **DART-Pow-Sys-6** **DART-Pow-Sys-8** **DART-Pow-Sys-10**
  DART-Pow-Sys-5 **DART-Pow-Sys-7** **DART-Pow-Sys-9** **DART-Pow-Sys-12**
  DART-Rec-Pow-1 **DART-Ejec-Pow-1**
  Component off-the-shelf (battery), does not need to be tested.

- **DART-Pow-Safe-4** **DART-Emb-Safe-2** **DART-Emb-Safe-4** **DART-Sys-Safe-7**
  DART-Emb-Safe-1 **DART-Emb-Safe-3** **DART-Sys-Safe-6**
  SAFE/ARM switch is not required.

- **DART-ADCS-Sys-1** **DART-ADCS-Sys-5** **DART-Sys-Perf-4** **DART-Sys-Perf-9** **DART-ADCS-Sys-2** **DART-ADCS-Sys-6** **DART-Sys-Perf-5** **DART-Sys-Perf-10** **DART-ADCS-Sys-3** **DART-ADCS-Sys-7** **DART-Sys-Perf-8** **DART-Sys-Perf-11** **DART-ADCS-Sys-4** **DART-Sys-Perf-3**
  Components off-the-shelf (accelerometers, gyroscope, magnetic sensor and GPS antenna module), so does not need to be tested.
- **DART-Vid-Sys-7** **DART-Vid-Sys-8** **DART-Sys-Perf-13** **DART-Sys-Perf-14**
  Component off-the-shelf (image processor), so does not need to be tested.
- **DART-Vid-Sys-3** **DART-Vid-Sys-4** **DART-Sys-Perf-6** **DART-Sys-Perf-7**
  Component off-the-shelf (camera module), does not need to be tested.
- **DART-Sub-Sys-7** **DART-Sys-Cost-1**
  COTS components were chosen because of the characteristics mentioned in their data sheets. It is assumed that the components fulfil the characteristics mentioned in their data sheets, they are not tested to prove that.
- **DART-Struc-Trans-3**
  - **Main objective:** Proper installation of the transmission system
  - **Type:** Inspection, Demonstration
  - **Inputs:** Satellite, ground station
  - **Outputs:** Receive data
  - **Risks:** Might not transmit all the data - partial failure of the mission
  - **Failure:** Redesign (structural system)
  - **Plan:** Demonstrate receipt of data on the ground
- **DART-Struc-Sys-1** **DART-Emb-Sys-5** **DART-Vid-Sys-6** **DART-Trans-Sys-8** **DART-Pow-Sys-21** **DART-ADCS-Sys-9** **DART-DCS-Sys-2** **DART-Sys-Budg-3** **DART-Rec-Sys-1** **DART-Ejec-Sys-3**
  - **Main objective:** Total mass
  - **Type:** Inspection, Demonstration
  - **Inputs:** Measuring device (scales)
  - **Outputs:** The weight of each subsystem
  - **Risks:** Might not reach the desired height - partial failure
  - **Failure:** Redesign
  - **Plan:** Weigh the systems at least 3 times on a scale with an accuracy of ± 0.5 grams
- **DART-Pow-Sys-4** **DART-Pow-Sys-19** **DART-Vid-Pow-2** **DART-Vid-Pow-10** **DART-Pow-Sys-11** **DART-Pow-Sys-20** **DART-Vid-Pow-3** **DART-Vid-Pow-11** **DART-Pow-Sys-13** **DART-Emb-Pow-2** **DART-Vid-Pow-4** **DART-DCS-Pow-2** **DART-Pow-Sys-14** **DART-Emb-Pow-3** **DART-Vid-Pow-5** **DART-DCS-Pow-3** **DART-Pow-Sys-15** **DART-Emb-Pow-4** **DART-Vid-Pow-6** **DART-Trans-Pow-2** **DART-Pow-Sys-16** **DART-Emb-Pow-5** **DART-Vid-Pow-7** **DART-Trans-Pow-3** **DART-Pow-Sys-17** **DART-ADCS-Pow-2** **DART-Vid-Pow-8** **DART-Trans-Pow-4** **DART-Pow-Sys-18** **DART-ADCS-Pow-3** **DART-Vid-Pow-9** **DART-Trans-Pow-5** **DART-Rec-Pow-2** **DART-Rec-Pow-3** **DART-Ejec-Pow-2** **DART-Ejec-Pow-3**
  - **Main objective:** Power and energy consumption
  - **Type:** Analysis, Demonstration
  - **Inputs:** Mathematical technique, measuring device (voltmeter, ammeter)
  - **Outputs:** Power and energy consumption
  - **Risks:** Not enough power and/or energy
  - **Failure:** Redesign (Components)
  - **Plan:** Analyse and demonstrate, for the flight duration, that all electrical components, including wiring and connection losses, consume less power and energy than can be provided by the battery
- **DART-Pow-Safe-1** **DART-Pow-Safe-3**
  - **Main objective:** Safety
  - **Type:** Inspection, Demonstration
  - **Inputs:** TubeSat
  - **Outputs:** TubeSat with working power system
  - **Risks:** Safety risks
  - **Failure:** Redesign (electrical components, connections), accept
  - **Plan:** Demonstrate the electrical components can be connected as late as possible.

- **DART-Pow-Safe-2**
  - **Main objective:** Voltage compatibility
  - **Type:** Inspection
  - **Inputs:** TubeSat with electrical components
  - **Outputs:** Checked electrical subsystem
  - **Risks:** Failure
  - **Failure:** Redesign (electrical connections)
  - **Plan:** Inspect the voltage compatibility within the TubeSat
- **DART-Emb-Boos-1** **DART-Emb-Ejec-1** **DART-Emb-DCS-1** **DART-Emb-Rec-1**
  DART-Emb-ADCS-1 **DART-Emb-Vid-1** **DART-Emb-DCS-2** **DART-ADCS-Boos-1**
  DART-Emb-ADCS-2 **DART-Emb-Vid-2** **DART-Emb-Trans-1** **DART-ADCS-DART-1**
  DART-Emb-ADCS-3 **DART-Emb-Vid-3** **DART-Emb-Trans-2**
  - **Main objective:** Detection and processing of data
  - **Type:** Low-altitude test
  - **Inputs:** DART rocket, TubeSat
  - **Outputs:** Data
  - **Risks:** Partial failure/Failure (loss of data)
  - **Failure:** Redesign (embedded system)
  - **Plan:** Perform the low-altitude test planned in Section 11.6
- **DART-Emb-Sys-1** **DART-Sys-Perf-12**
  - **Main objective:** Data storage
  - **Type:** Analysis, Inspection
  - **Inputs:** Simulation program data generation
  - **Outputs:** Simulated amount of generated data
  - **Risks:** Partial failure/Failure (loss of data)
  - **Failure:** Redesign (Choose microSD card with sufficient memory)
  - **Plan:** Simulate the amount of data generated on board and inspect the microSD card if it has sufficient memory
- **DART-ADCS-Vid-1** **DART-ADCS-Trans-1**
  - **Main objective:** Pointing accuracy
  - **Type:** Analysis
  - **Inputs:** Simulation program, ADCS specifications, camera pointing requirement
  - **Outputs:** Pointing accuracy
  - **Risks:** Partial failure (loss of data, bad video view)
  - **Failure:** Redesign (ADCS)
  - **Plan:** Simulate the pointing accuracy required
- **DART-ADCS-Vid-2**
  - **Main objective:** Rotational rate during descent
  - **Type:** Analysis
  - **Inputs:** Simulation program, ADCS specifications
  - **Outputs:** Maximum rotational rate
  - **Risks:** Partial failure (bad video)
  - **Failure:** Redesign (ADCS)
  - **Plan:** Simulate the rotational rate achieved by the ADCS
- **DART-ADCS-Sys-10** **DART-DCS-Sys-3** **DART-Ejec-Sys-2**
  - **Main objective:** Introduction of extra loads
  - **Type:** Analysis, Low-altitude test
  - **Inputs:** Simulation program, expected launch loads and ADCS specifications (analysis), DART rocket and TubeSat (test)
  - **Outputs:** Extra loads detection
  - **Risks:** Failure (loads too high)
  - **Failure:** Redesign (ADCS)
  - **Plan:** Simulate loads introduced by ADCS, perform the low-altitude test planned in Section 11.6
- **DART-Vid-ADCS-1** **DART-Vid-DART-1** **DART-Vid-Sys-2**
  DART-Vid-DCS-1 **DART-Vid-Sys-1**
  - **Main objective:** Camera placement and operation
  - **Type:** Low-altitude test
  - **Inputs:** DART rocket, TubeSat
  - **Outputs:** Video
  - **Risks:** Failure (not proper video)

  - **Failure:** Redesign (video system)
  - **Plan:** Perform the low-altitude test planned in Section 11.6

- **DART-Vid-Op-1 DART-Trans-Op-2**
  - **Main objective:** Adhere to the security regulations
  - **Type:** Inspection
  - **Inputs:** TubeSat
  - **Outputs:** Approved TubeSat
  - **Risks:** Failure
  - **Failure:** Redesign (video system, transmission system)
  - **Plan:** Inspect whether the subsystems adhere to the security regulations

- **DART-DCS-Vid-1 DART-DCS-Vid-2 DART-DCS-Trans-1**
  - **Main objective:** Stability control and pointing accuracy
  - **Type:** Low-altitude test
  - **Inputs:** DART rocket, TubeSat
  - **Outputs:** Data of stability and pointing accuracy
  - **Risks:** Partial failure (stability problems)
  - **Failure:** Redesign (DCS)
  - **Plan:** Perform the low-altitude test planned in Section 11.6

- **DART-Trans-Sys-1 DART-Trans-Op-1**
  - **Main objective:** Check whether data generated on board can be send 3 times
  - **Type:** Analysis
  - **Inputs:** Fall time simulation, data generation simulation, transmission system specifications
  - **Outputs:** Fall time, data rate between transmission system and ground station
  - **Risks:** Partial failure (possibility of data loss)
  - **Failure:** Redesign
  - **Plan:** Simulate fall time and the time it takes to send the data generated on board to the ground station three times

- **DART-Trans-Struc-1 DART-Trans-Atm-1 DART-Sys-Perf-1 DART-Trans-DART-1 DART-Trans-Sys-2 DART-Sys-Perf-2**
  - **Main objective:** Link budget
  - **Type:** Analysis, Low-altitude test
  - **Inputs:** Transmission simulation (analysis), DART rocket and TubeSat (test)
  - **Outputs:** Strength of the transmission
  - **Risks:** Failure (loss of data)
  - **Failure:** Redesign (transmission system)
  - **Plan:** Simulate the transmission, perform the low-altitude test planned in Section 11.6

- **DART-Trans-Sys-3 DART-Trans-Sys-4 DART-Sys-Comm-1 DART-Sys-Comm-2**
  - **Main objective:** The right frequency domain
  - **Type:** Inspection, Analysis
  - **Inputs:** Transmission system, Simulation
  - **Outputs:** Frequency domain
  - **Risks:** Failure (safety, law regulations, launch site regulations)
  - **Failure:** Redesign (transmission system)
  - **Plan:** Inspect/Simulate the frequency domain

- **DART-Trans-Sys-5 DART-Trans-Sys-6 DART-Sys-Grnd-1 DART-Sys-Grnd-2**
  - **Main objective:** Ground station design
  - **Type:** Inspection
  - **Inputs:** Ground station
  - **Outputs:** Working ground station
  - **Risks:** Failure (data loss)
  - **Failure:** Redesign (ground station)
  - **Plan:** Inspect the ground station

- **DART-Ejec-Sys-1 DART-Ejec-DART-1**
  - **Main objective:** Ejection of the TubeSat, the ejection system does not extend outside of the DART
  - **Type:** Test
  - **Inputs:** DART rocket and TubeSat
  - **Outputs:** Behaviour of the ejection system
  - **Risks:** Failure (no/faulty ejection)
  - **Failure:** Redesign (ejection system)

  - **Plan:** Perform the ejection on the ground, check whether the ejection system extends outside of the DART

- **DART-Sys-Sust-EOL-2 DART-Sys-Sust-EOL-3 DART-Sys-Perf-15**
  - **Main objective:** Recovery system
  - **Type:** Inspection
  - **Inputs:** Recovery system
  - **Outputs:** Recovery system
  - **Risks:** Not recovered TubeSat
  - **Failure:** Redesign (recovery system), accept
  - **Plan:** Inspect the recovery system

- **DART-Struc-Pow-1 DART-Struc-ADCS-1 DART-Struc-Ejec-1 DART-Struc-Trans-1 DART-Struc-Emb-1 DART-Struc-Vid-1 DART-Struc-DCS-1 DART-Struc-Rec-1**
  - **Main objective:** Structural integrity
  - **Type:** Test (loads test)
  - **Inputs:** TNO testing facility[4], prototype
  - **Outputs:** Data, specifying the structural integrity
  - **Risks:** Partial failure of the mission
  - **Failure:** Redesign
  - **Plan:** Test one TubeSat prototype

- **DART-Struc-Pow-2 DART-Struc-ADCS-2 DART-Struc-Ejec-2 DART-Struc-Trans-2 DART-Struc-Emb-2 DART-Struc-Vid-2 DART-Struc-DCS-2 DART-Struc-Rec-2**
  - **Main objective:** Attachment of the components
  - **Type:** Inspection, Demonstration
  - **Inputs:** TubeSat, components
  - **Outputs:** Mounted TubeSat
  - **Risks:** Bad mounting (components fall off)
  - **Failure:** Redesign of the mounting point
  - **Plan:** Demonstrate mounting once

- **DART-Sub-Sys-2 DART-Sys-Budg-5 DART-Struc-Sys-2**
  - **Main objective:** Thermal range
  - **Type:** Test (temperature test)
  - **Inputs:** NLR testing facility[5,6], prototype
  - **Outputs:** Data about thermal range, inside temperature, insulation abilities
  - **Risks:** Failure
  - **Failure:** Redesign
  - **Plan:** Test one TubeSat prototype, also the insulation abilities

- **DART-Sub-Sys-3 DART-Sys-Des-1**
  - **Main objective:** Production of TubeSat
  - **Type:** Demonstration
  - **Inputs:** TubeSat drawings and materials, COTS components
  - **Outputs:** TubeSat
  - **Risks:** Failure
  - **Failure:** Redesign
  - **Plan:** Demonstrate that the TubeSat is possible to produce

- **DART-Sub-Sys-4 DART-Sys-Safe-1 DART-Sys-Safe-3 DART-Sub-Sys-5 DART-Sys-Safe-2**
  - **Main objective:** Ensure safety of people and assets on the ground and the sounding rocket
  - **Type:** Analysis
  - **Inputs:** Simulation of TubeSat impact area, wind conditions
  - **Outputs:** TubeSat impact area
  - **Risks:** Endangering people and assets on the ground and the sounding rocket
  - **Failure:** Redesign if necessary, otherwise change launch window
  - **Plan:** Simulate the impact area

- **DART-Sub-Sys-6 DART-Sys-Budg-7**
  - **Main objective:** Finishing the design
  - **Type:** Inspection

[4]URL https://www.tno.nl/en/collaboration/expertise/technical-sciences/structural-dynamics/ [cited 13 June 2016]
[5]URL http://www.nlr.nl/downloads/e1188-environmental-testing-of-aerospace-equip.pdf [cited 14 June 2016]
[6]URL http://labs.tudelft.nl/index.php?action=instrument&id=429 [cited 14 June 2016]

  - **Inputs:** TubeSat design process
  - **Outputs:** Finished design
  - **Risks:** Failure
  - **Failure:** Accept
  - **Plan:** Check whether design is finished on 23 June 2016.

- **DART-Sys-Sust-Mat-1 DART-Sys-Sust-Pro-1 DART-Sys-Sust-Pro-3 DART-Sys-Sust-Tra-1 DART-Sys-Sust-Mat-2 DART-Sys-Sust-Pro-2 DART-Sys-Sust-Pro-4 DART-Sys-Sust-Tra-2**
  - **Main objective:** Being sustainable
  - **Type:** Inspection
  - **Inputs:** COTS components, manufacturers, materials
  - **Outputs:** Sustainability inspection
  - **Risks:** Not sustainable
  - **Failure:** Accept
  - **Plan:** Inspect the sustainability of COTS components, materials, manufacturers

- **DART-Sys-Sust-Pro-5 DART-Sys-Sust-Pro-7 DART-Sys-Sust-EOL-1 DART-Sys-Sust-Pro-6 DART-Sys-Sust-Pro-8**
  - **Main objective:** Sustainable end-of-life
  - **Type:** Inspection
  - **Inputs:** The end-of-life design
  - **Outputs:** Sustainability inspection
  - **Risks:** Not sustainable
  - **Failure:** Accept
  - **Plan:** Inspect the sustainability of end-of-life of the TubeSat

- **DART-Sys-Budg-6**
  - **Main objective:** Stand-alone TubeSat
  - **Type:** Inspection, Demonstration
  - **Inputs:** TubeSat
  - **Outputs:** TubeSat
  - **Risks:** Failure
  - **Failure:** Redesign
  - **Plan:** Inspect and determine if the TubeSat is stand-alone

- **DART-Sys-Safe-4**
  - **Main objective:** Test the TubeSat at a low altitude for its performance and compatibility with the DART rocket
  - **Type:** Inspection
  - **Inputs:** Test plan (specified in Section 11.6)
  - **Outputs:** Test plan
  - **Risks:** Non-accepted test plan
  - **Failure:** Redesign (test)
  - **Plan:** Inspect the test plan for feasibility and compliance with regulations

- **DART-Sys-Safe-5**
  - **Main objective:** Protect TubeSat against lightning strike
  - **Type:** Analysis
  - **Inputs:** Simulation of high voltage on TubeSat
  - **Outputs:** Protection of TubeSat against lightning strike
  - **Risks:** Failure
  - **Failure:** Redesign
  - **Plan:** Analyse whether the TubeSat can withstand a lightning strike

- **DART-Struc-Boos-1 DART-ADCS-Vid-3 DART-Sub-Sys-1 DART-Sys-Budg-4**
  - **Main objective:** Indication of structural integrity of the design
  - **Type:** Test (vibration test)
  - **Inputs:** TNO testing facility[7], prototype
  - **Outputs:** Data, specifying the structural integrity
  - **Risks:** Failure of the mission (does not survive the launch)
  - **Failure:** Redesign
  - **Plan:** Test one TubeSat prototype

[7] URL https://www.tno.nl/media/1749/sd-shock-and-vibration-2012.pdf [cited 13 June 2016]

- **DART-Struc-DART-1 DART-Struc-DART-2 DART-Sys-Budg-1 DART-Sys-Budg-2 DART-Ejec-DART-2**
  - **Main objective:** Indication of the size
  - **Type:** Inspection
  - **Inputs:** Measuring device, TubeSat
  - **Outputs:** The length of the TubeSat
  - **Risks:** Failure of the mission (does not fit into the DART)
  - **Failure:** Redesign
  - **Plan:** Measure at least 3 times with a calibrator
- **DART-Struc-Atm-1 DART-ADCS-Atm-1 DART-Vid-Atm-1**
  - **Main objective:** Structural integrity
  - **Type:** Test (atmospheric conditions)
  - **Inputs:** TNO testing facility[8], prototype
  - **Outputs:** Data, specifying the structural integrity
  - **Risks:** Failure of the mission
  - **Failure:** Redesign
  - **Plan:** Test one TubeSat prototype

11.5. Product Validation

Verification of a product is checking whether all the particular requirements for a specific use of the product are met. Product validation focuses more on what needs to happen in practice than on requirements. The validation techniques are specified below [12, 23].

- **End-to-End Information System Testing**
  Show compatibility of project information systems (e.g. data, timing).
- **Mission Scenario Tests**
  Demonstrate that flight hardware and software can execute the mission under flight-like conditions (nominal & contingency) without real timeline.
- **Operations Readiness Tests**
  Demonstrate that all elements of the ground segment (e.g. software, hardware, people, facilities) accomplish the mission plan using real timeline.
- **Stress-Testing and Simulation**
  Assess system robustness to variations in performance and fault conditions.

Elaboration on each of the validation techniques is given below. The validation tests are discussed as well.

End-to-End Information System Testing

The data compatibility of all the systems will be checked. The focus is on the time it takes the data to get ready for the transmission. Turning on and off of the components, and ejection timer are tested as well.

Test: Testing of all software of the TubeSat, seen in Chapter 8.4, Figure 8.5, and the transmission system

- **Plan:** The TubeSat is tested on the ground, 120 km far away from the ground station. The idea is that the whole mission, with the focus on the data transfer, is simulated. The timer starts the ejection system, after which the microcontroller is notified about the ejection. Right before ejection, the camera starts recording the 30-second video, while the accelerometers, gyroscope, magnetic sensor and GPS generate data. The data is compressed and transferred to the transmission system. Then the transmission system transmits all the data to the ground station.
- **Inputs:** The TubeSat, ground station
- **Outputs:** Compressed video, data, transmission initiation time, transfer time within the system, time between ejection initiation and notification of the microcontroller, transmission time
- **Risks:** No notification of ejection, no notification when recording started, system failure during the transfer within the components, loss of data, the data handling being too slow, transmission too slow, link budget too small
- **Failure:** Redesign (new components, power system/embedded system, transmission system)

[8] URL https://www.tno.nl/media/2461/lr-leaflet-climatic-altitude-chamber-3.pdf [cited 13 June 2016]

Mission Scenario Tests

The TubeSat is tested in different climate scenarios, low/high temperature, low pressure. A check for vibrations, and high loads is performed as well. The main goal is to test whether the complete TubeSat is able to cope with extreme environments and still perform as desired.

Test: Temperature test in autoclave

- **Plan:** To test the TubeSat for its ability too withstand very high or low temperatures, it will be placed in an autoclave. While in the autoclave, the TubeSat will be turned on. The video system will make a high-definition video of 30 seconds and the measurement devices will also gather data. The data are compressed and transferred to the transmission system. During this test, also the structural system is tested for these temperatures.
- **Inputs:** TubeSat, autoclave[5,6]
- **Outputs:** Compressed video, data, transfer time within the system
- **Risks:** The temperature is too high/low for the system/components to work
- **Failure:** Redesign to withstand higher and/or lower temperatures

Test: Climate test

- **Plan:** The TubeSat will be put in a climate chamber. The climate in the chamber is changed to simulate the climate during the fall. While in the chamber, the TubeSat will be turned on. The video system will make a high-definition video of 30 seconds and the measurement devices will also gather data. The data are compressed and transferred to the transmission system. During this test, also the structural system is tested for the climate changes.
- **Inputs:** TubeSat, climate chamber[6,8]
- **Outputs:** Compressed video, data, transfer time within the system
- **Risks:** The climate, climate changes lead to failure of systems inside of the TubeSat.
- **Failure:** Redesign to withstand climate changes

Test: Pressure test in pressure chamber

- **Plan:** The TubeSat will be put in a pressure chamber. The pressure in the chamber is slowly lowered. While in the chamber, the TubeSat will be turned on. The video system will make a high-definition video of 30 seconds and the measurement devices will also gather data. The data are compressed and transferred to the transmission system. During this test, also the structural system is tested for the pressure.
- **Inputs:** TubeSat, pressure chamber[9]
- **Outputs:** Compressed video, data, transfer time within the system
- **Risks:** The pressure is too low for the components to work properly
- **Failure:** Redesign to withstand lower pressures

Test: TubeSat natural frequency test

- **Plan:** The vibration data of the DART rocket, excluding the TubeSat, will be obtained from a test launch or from a simulation. Then the natural damping frequency of the TubeSat is determined using discretisation. A check will be performed whether the vibration frequency of the DART rocket and the natural damping frequency of the TubeSat are equal. If this is the case, the TubeSat needs to be redesigned to prevent the structure from failing during launch. Then the TubeSat will be put on a shaker to test its ability to cope with the vibrational launch loads. This will be done during the 'Vibrations test on a shaker', as specified below.
- **Inputs:** TubeSat, DART rocket/simulation
- **Outputs:** Vibration data of the DART rocket, natural damping frequency of the TubeSat
- **Risks:** Natural frequency of the TubeSat and the vibration frequency of the DART are equal
- **Failure:** Redesign of structural system

Test: Vibrations test on shaker

- **Plan:** The TubeSat will be put on a shaker. While on the shaker, it will be turned on. The video system will make a high-definition video of 30 seconds and the measurement devices will also gather data. The data are compressed and transferred to the transmission system. During this test, the structural system is tested for the vibrations.
- **Inputs:** TubeSat, vibration shaker [7]
- **Outputs:** Compressed video, data, transfer time within the system
- **Risks:** The vibrations are too high for the system to function
- **Failure:** Redesign to withstand higher vibrations

[9] URL http://www.space-airbusds.com/en/equipment/thermal-vacuum.html [cited 16 June 2016]

Operations Readiness Tests

Using real timeline, the launch is set up. It is determined if all the tasks are performed in time and if the operation manual is complete.

Test: Set-up test

- **Plan:** The entire launch system will be set up. The launch tower will be set up, the TubeSat will be put in the DART rocket and the DART rocket will be installed on the launch tower. The booster will be prepared for launch. Moreover, the equipment on the ground station will be set up correctly. It will be measured how long it takes to do this.
- **Inputs:** Launch tower, DART rocket including booster, TubeSat
- **Outputs:** Set-up time DART rocket, set up time launch tower
- **Risks:** It takes longer than expected to set up the DART rocket and the ground station for the mission.
- **Failure:** Reconstruct the operations manual to decrease required time

Stress-Testing and Simulation

As the conditions during the flight might change, such as weather conditions, variations in performance are simulated to demonstrate that the TubeSat is able to perform nevertheless.

Test: Load test

- **Plan:** The resistance of the TubeSat against the loads induced during launch and ejection will be tested. The launch loads and ejection loads will be simulated to know what loads can be expected. Then the TubeSat will be subjected to these loads. The measurement devices will perform measurements to check whether the equipment can give results to the accuracy specified in their data sheets under high loads. The top of Hyperflo is connected with pins, and these are loaded as they would be during a flight. Also the connection, cables, between the TubeSat and the Hyperflo are loaded to test if they can survive the loads during the flight.
- **Inputs:** TubeSat, testing facility[4]
- **Outputs:** The behaviour of the TubeSat under the introduced loads
- **Risks:** The TubeSat is not able to perform under the loads, the cables snap, Hyperflo breaks
- **Failure:** Redesign to withstand higher loads

Test: Hyperflo test

- **Plan:** The behaviour of the Hyperflo and the TubeSat during high speeds will be tested. The focus will be on the stabilising ability of the Hyperflo and the connection between the Hyperflo and the TubeSat.
- **Inputs:** TubeSat, wind tunnel[10]
- **Outputs:** The aerodynamic abilities of the Hyperflo and satellite
- **Risks:** The Hyperflo does not work as expected, the cables break
- **Failure:** Redesign of the descent system

Test: Wind test

- **Plan:** The behaviour of the satellite and Hyperflo when subjected to different wind speeds will be tested. The satellite will be put in a climate chamber or wind tunnel which can simulate a wide range of wind speeds. The video and the measurements will be taken, to check if the equipment can perform properly in these conditions as well.
- **Inputs:** Satellite, climate chamber[8] or wind tunnel[10]
- **Outputs:** Behaviour of satellite, equipment and the Hyperflo subjected to different wind speeds
- **Risks:** The Hyperflo and satellite cannot perform in hard or changing winds
- **Failure:** Launch in favourable wind conditions

11.6. Low-Altitude Test

A specific requirement, **DART-Sys-Safe-4**, states that a test plan shall be created, including tests that can be done at lower altitudes (< 2 km) on a military site in the Netherlands to verify the operational capability of the suborbital satellite. This low-altitude test is worked out below.

Test: Test launch to an altitude of 2 km on a military site in the Netherlands

- **Plan:** A launch is conducted at a military ground. First, the launch tower is set up at the launch location agreed upon with the Dutch military. The impact area of the DART rocket is simulated, taking into account

[10] URL http://www.lr.tudelft.nl/nl/organisatie/afdelingen/aerodynamics-wind-energy-flight-performance-and-propulsion/facilities/wind-tunnel-lab/ [cited 16 June 2016]

the current wind direction and speed. After permission of the military, the launch tower rail is put in the appropriate direction and under the appropriate angle. The booster is prepared for launch and the satellite is put into the dart. The dart including the booster is then installed on the launch tower rail. The DART rocket is launched remotely from a safe distance and the satellite is turned on. The DART will coast up to an altitude lower than 2 km. At apogee, the satellite will be ejected. The camera starts to record a 5-second HD video right before ejection, while the accelerometers, gyroscope, magnetic sensor and GPS generate data. The data is compressed and transferred to the transmission system.Then the data is transmitted to the ground station. The generation of data by the measuring devices and the transmission of data continues until the satellite hits the ground. During operation, the operations manual, Chapter 16.2, is strictly followed.

- **Inputs:** The satellite, the DART rocket, ground station, launch tower, operations manual, other facilities stated in the operations manual
- **Outputs:** 5-second video, kinematic behaviour data, transmission initiation time, data transfer time within the system, time between ejection initiation and notification of the microcontroller, data transmission time to the ground station
- **Risks:** Most important ones: launch loads or ejection loads damage the satellite, no notification of ejection, no notification when recording has started, system failure during data transfer between/within the components, loss of data, data handling being too slow, failure during the ejection, component or software failure (risks during launch and descent related to the DART rocket are not taken into account here)
- **Failure:** Redesign of the failed component(s)/subsystem

12
Production

The production of the TubeSat will be discussed in this Chapter. It will be structured as follows: In Section 12.1 the production of each individual part is discussed. Then, the assembly of all parts is discussed in Section 12.2. Note that references/footnotes were not used on purpose in this chapter, the references and footnotes can be found in Appendix A.

12.1. Production of Individual Parts

This section contains the production plans of each individual part. As COTS products are to be used during the production of the TubeSat, most of the parts will be a combination of products to be purchased. An overview of the entire TubeSat can be seen in Figure 12.1, the numbered parts can be found in Table 12.1. Note that all technical drawings in this section are in mm.

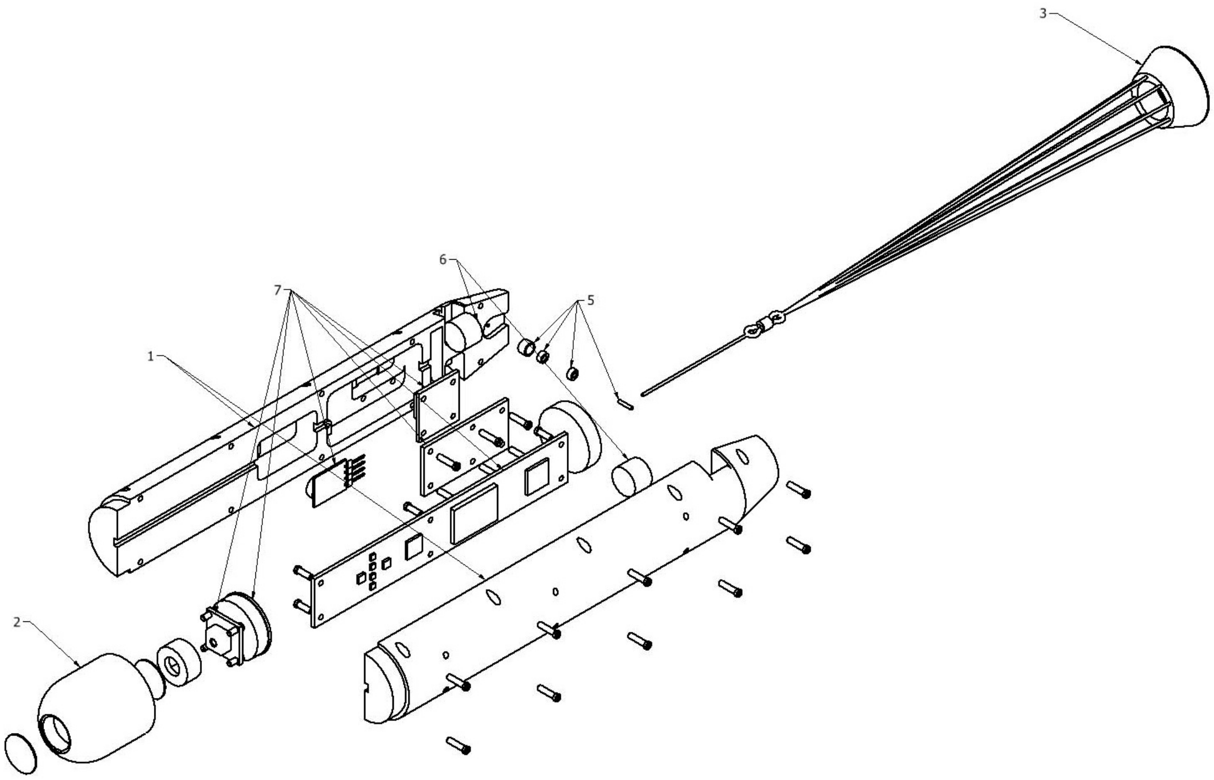


Figure 12.1: Overview TubeSat.

Table 12.1: TubeSat Parts.

Components	
1. Outer shell	5. Rollout system
2. Nose cone	6. yoyos
3. Hyperflo	7. PCBs (6 pieces)
4. Sabots (not shown)	

12.1.1. Outer Shell

The outer shell of the TubeSat will be made of steel and will be milled by TME. According to requirement **DART-Sys-Sust-Pro-2**, 'subtractive manufacturing methods shall not be used if other manufacturing methods are also able to accomplish the required manufacturing task', in this case milling was decided over casting and 3D printing because of the following reasons:

- Casting needs to be finished, because of the surface quality. This will cost more time/money.
- 3D printing cannot print the angles (90 degrees) that the model requires.

The shell will be split in two along the z-axis (both milled separately), to make the assembly of the TubeSat easier.

The left and right side of the outer shell can be seen in Figures 12.2 and 12.3.

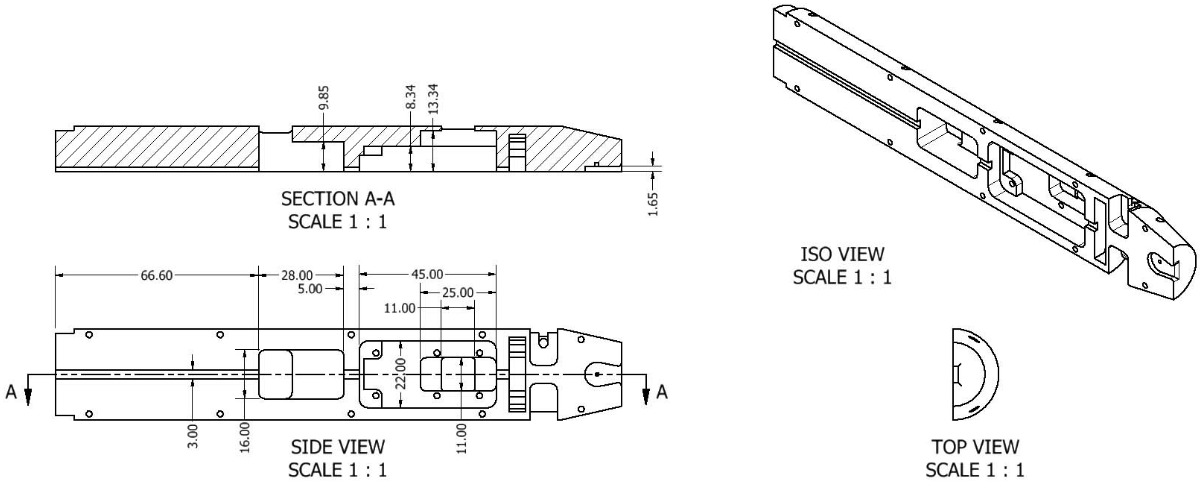


Figure 12.2: Left outer shell production.

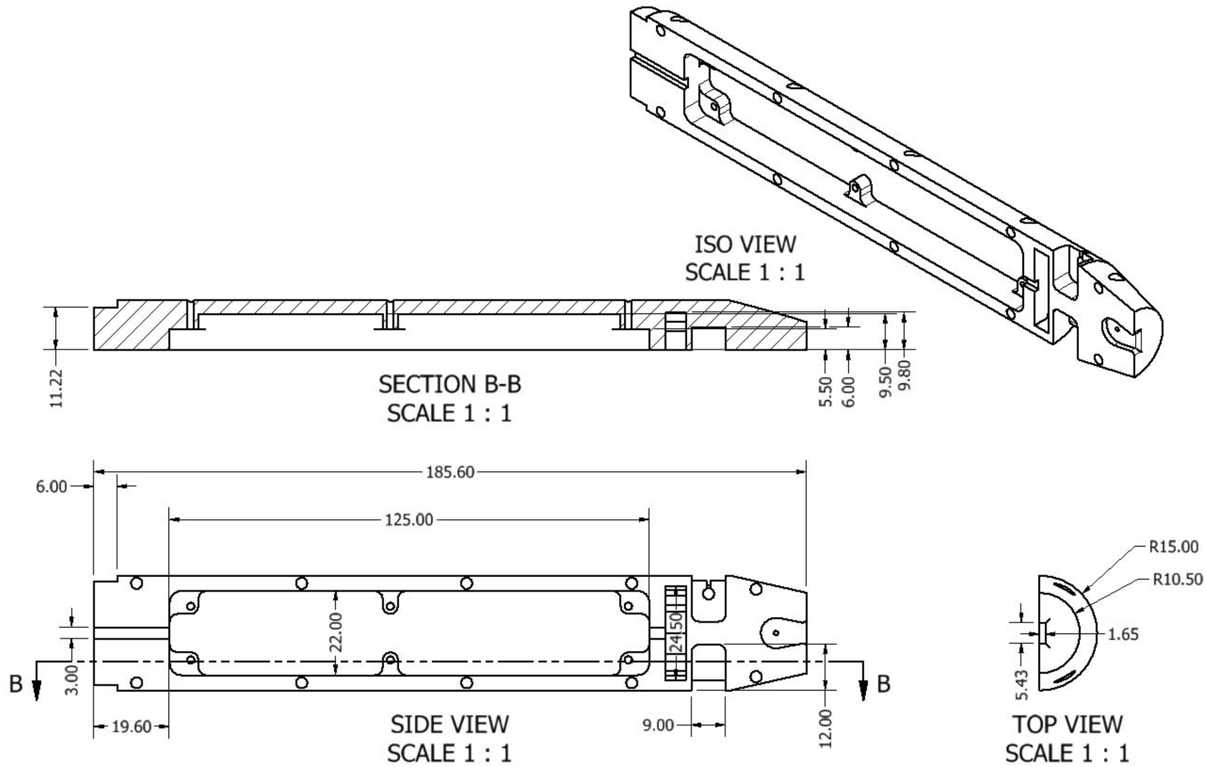


Figure 12.3: Right outer shell production.

12.1.2. Nose Cone

A visual representation of the nose cone can be seen in Figure 12.4.

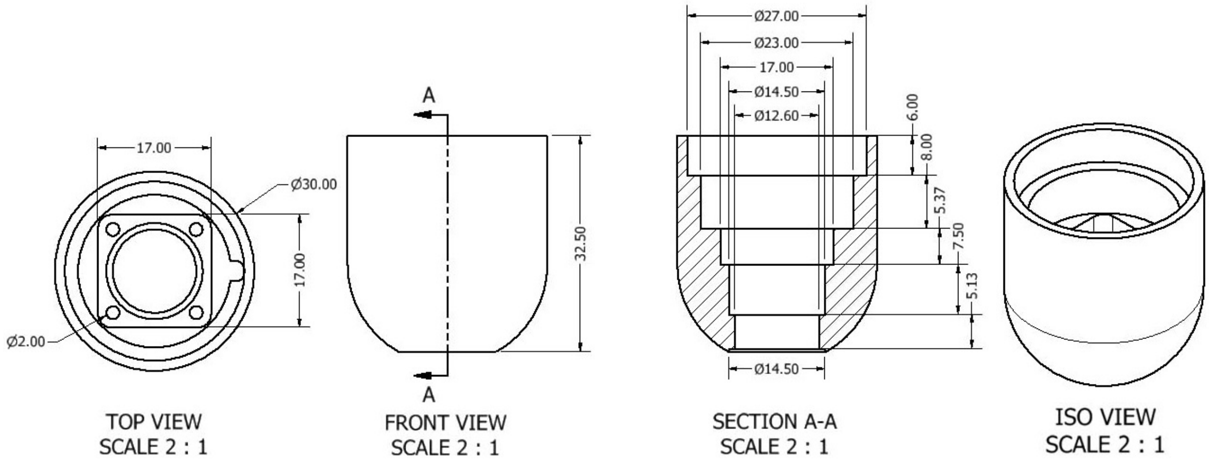


Figure 12.4: Nose cone production.

If one looks at the cross-section, one can see that the nose cone consists of a single nose cone with two lenses. These lenses will be glued to either end to create a wall of air insulation and to provide vision for the camera. The nose cone is made of zirconia and the lenses are made of quartz, these materials were chosen for their melting temperatures, which are under those experienced during the mission.

12.1.3. Hyperflo

A visual representation of the Hyperflo can be seen in Figure 12.5.

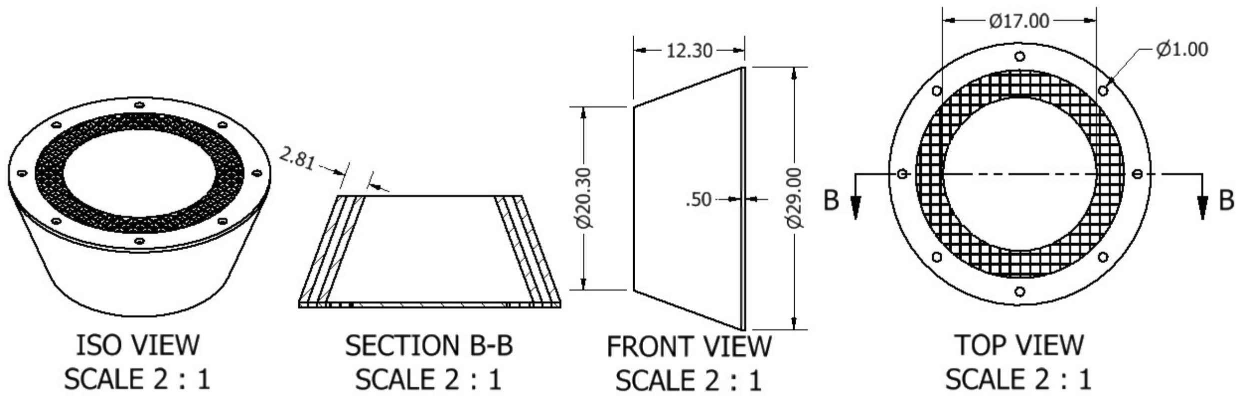


Figure 12.5: Hyperflo production.

The main body of the Hyperflo will be made of steel and be milled by TME. The top mesh structure will also be made of steel, to create the mesh a water cutter will be used. It was decided to use a water cutter to create the mesh, because of the high precision (0.006" = .15 mm[1]).

Eight holes will then be drilled at equal distances in the wall of the Hyperflo for the lines, see the bottom view of the Hyperflo. The steel pins will be inserted, and glued with adhesives, from the top to half the height of the Hyperflo. The tops of the pins will then be molten/welded to the top of the Hyperflo, to seal the holes. The lines will then be glued to the holes by placing adhesives at the bottom of the holes and pushing in the lines. The other ends of the lines will be attached to the eye eye swivel, which in turn will be attached to the rollout system. Note that the adhesives were chosen, because it is easier to glue then to create thread for bolts (for holes of 1mm diameter).

[1] http://www.teskolaser.com/waterjet_cutting.html [cited 16 June 2016]

12.1.4. Sabots

A visual representation of the sabots can be seen in Figure 12.7.

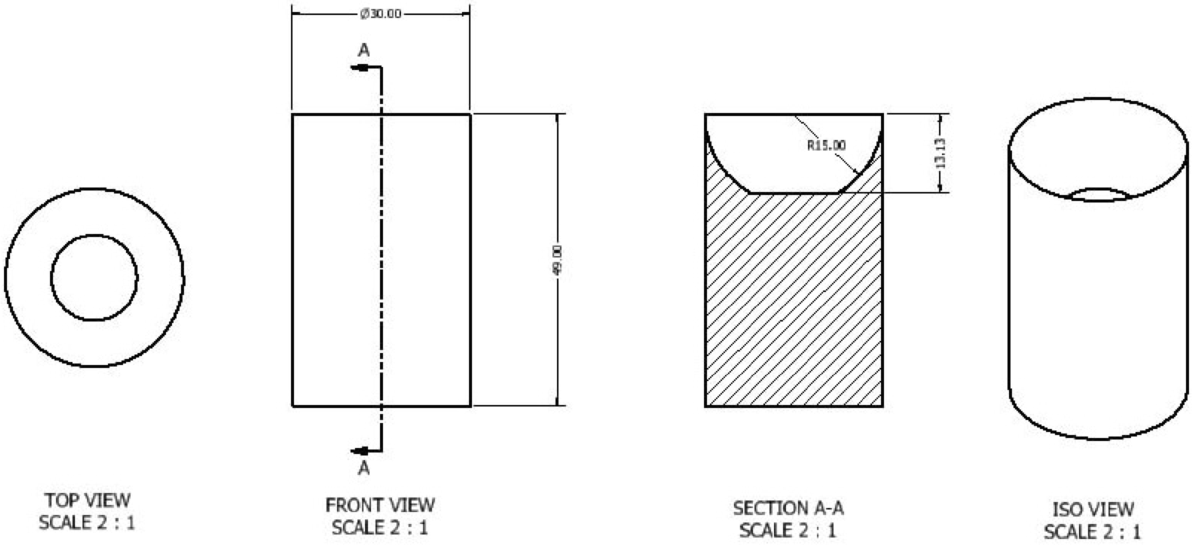


Figure 12.6: Bottom sabot production.

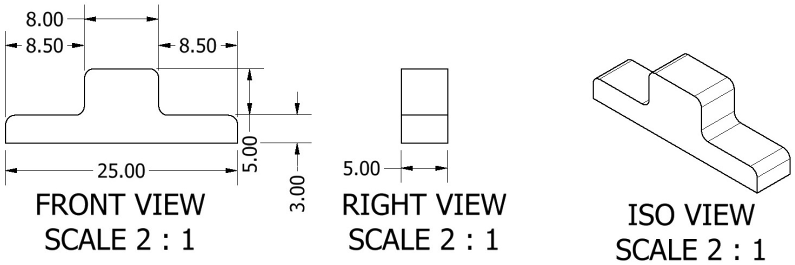


Figure 12.7: Top sabots production.

All the sabots will be made of Sorbothane and will be custom made by Sorbothane (the company has the same name as the product). Note that the top sabot will be produced eight times.

12.1.5. Rollout System

A visual representation of the rollout system can be seen in Figure 12.8.

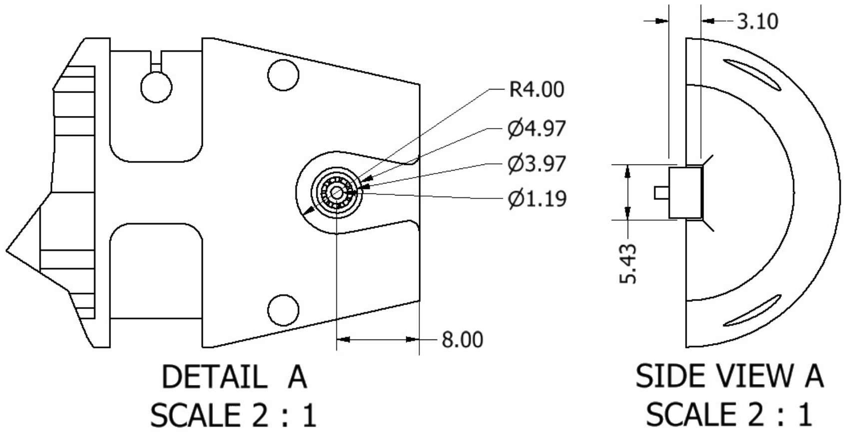


Figure 12.8: Rollout system production.

The rollout system consists of four different parts. Two bearings, which are attached to the outer shell by a steel rod, around which a sleeve is placed. The rope (shark wire) is then wound around the sleeve. All parts can be found commercial of the shelf (COTS). Note that two bearings are placed next to each other as a wider bearing could not be found.

The production of the rollout system will be pretty straightforward. A steel pin will be placed through both bearings. The sleeve will then be press fit on the bearings. The system is then ready to be used in the assembly. The structural analysis of the rollout system can be found in, Section 9.

12.1.6. Yo-Yo De-Spin Mechanism

A visual representation of the yoyo system can be seen in Figure 12.9.

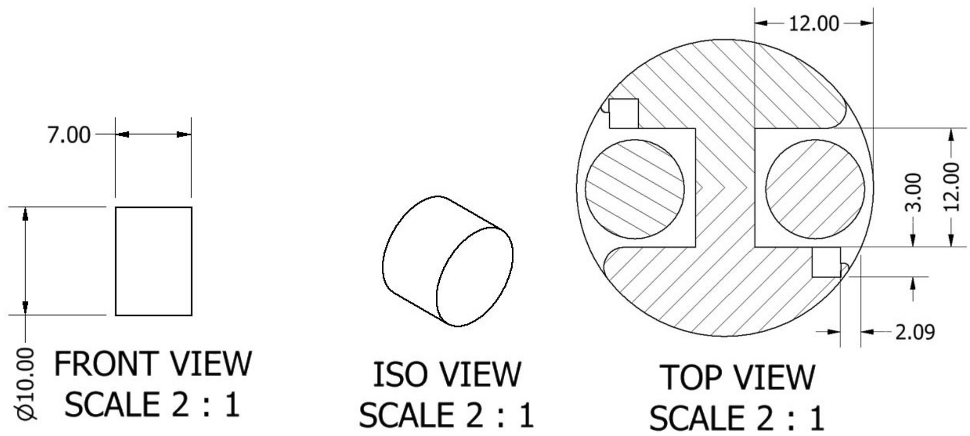


Figure 12.9: Yo-yo de-spin mechanism production.

The yo-yo's will be turned from a block of tungsten carbide. They will be attached to the TubeSat with lines which are attached to a ball, which in turn will be placed in a ball socket. Note that the ball socket connection is used, because it is used to detach the yo-yo's and dump the momentum. Once the ball is placed in the ball socket the yo-yo's are wound around the main body and placed inside their cavities.

12.1.7. PCBs

A visual representation of the PCBs can be found under the PCB Design header. Note that the battery is not included in the production of the PCB. It will however be purchased and used in the design, see Section 12.2.

General Production

The production of a PCB consists of three parts:

1. Production of the circuit board (PCB)
2. The production/purchase of the modules to be included on the PCB
3. The placement and attachment of the modules on the PCB.

After the circuit board is designed the production is outsourced to eurocircuits[2]. At the same time the modules needed for each individual PCB will be purchased from various companies, see the purchased parts section. For the placement of the modules on the PCB a pick and place machine was found. This was however deemed unnecessary as a pick and place machine is usually only used for big batches. It was decided to place the modules in person. Finally, it was decided that to attach the modules to PCB a reflow oven would be used, this was decided upon as some modules that are included in the PCB designs are too small to be soldered.

PCB Design

The TubeSat will contain 5 PCBs each containing their own set of modules, the modules placed on each PCB will be noted. An additional module is the Beacon module which is bought as a complete PCB unit.

[2] http://www.eurocircuits.com [cited 13 June 2016]

PCB 1
A visual representation of PCB 1 can be seen in Figure 12.10.

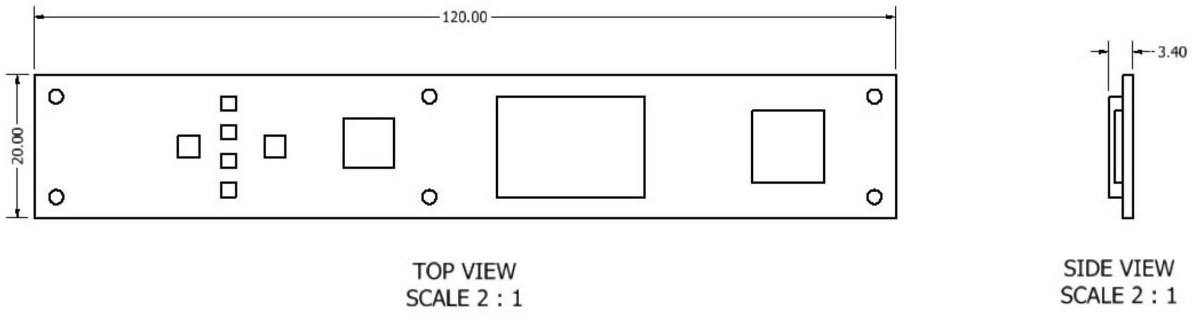


Figure 12.10: PCB 1 production.

Table 12.2 contains all the components that are to be placed on PCB 1.

Table 12.2: PCB 1 components.

Components		
Microcontroller	Modulator	DC-DC converter 3.6V to 2.5V
MicroSD connector	Resistor	DC-DC converter 3.6V to 1.8V
Digital-to-Analog Converter	DC-DC Converter 3.6V to 3.3V	DC-DC converter 3.6V to 1.2V
Image processor	DC-DC converter 3.6V to 2.8V	

PCB 2
A visual representation of PCB 2 can be seen in Figure 12.11.

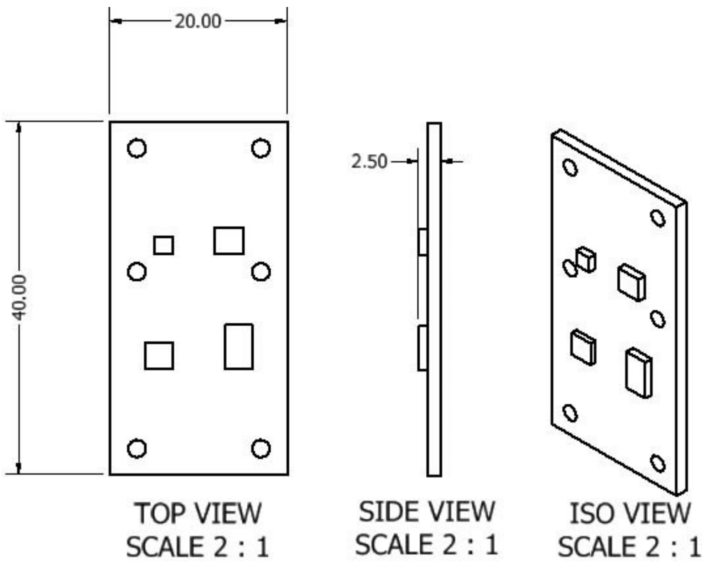


Figure 12.11: PCB 2 production.

Table 12.3 contains all the components that are to be placed on PCB 2.

Table 12.3: PCB 2 components.

Components	
3-Axis MEMS accelermometer low g's	3-Axis MEMS accelerometer high g's
3- Axis MEMS Gyroscope	3-Axis MEMS magnetic sensor

PCB 3: GPS
A visual representation of PCB 3 can be seen in Figure 12.12. The only component to be placed on the 3rd PCB is the GPS antenna module.

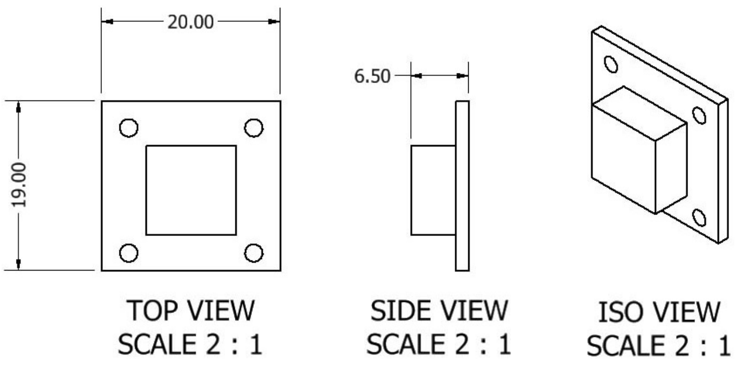


Figure 12.12: PCB 3 production.

PCB 4: Camera
A visual representation of PCB 4 can be seen in Figure 12.13.

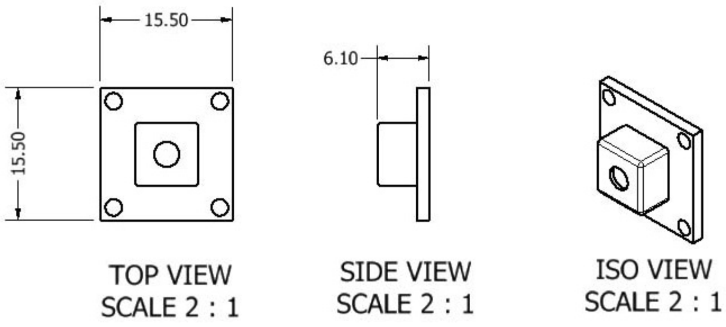


Figure 12.13: PCB 4 production.

The only component to be placed on the 4th PCB is the 5.0 megapixel camera. Note that before the camera is attached to the PCB it will be modified to be able to withstand the vibrational forces. This will be done by taking the camera module apart and placing a ring of rubber around the lens of the camera to prevent it from shattering, without impeding the view.

PCB 5: Transmitter
A visual representation of PCB 5 can be seen in Figure 12.14.

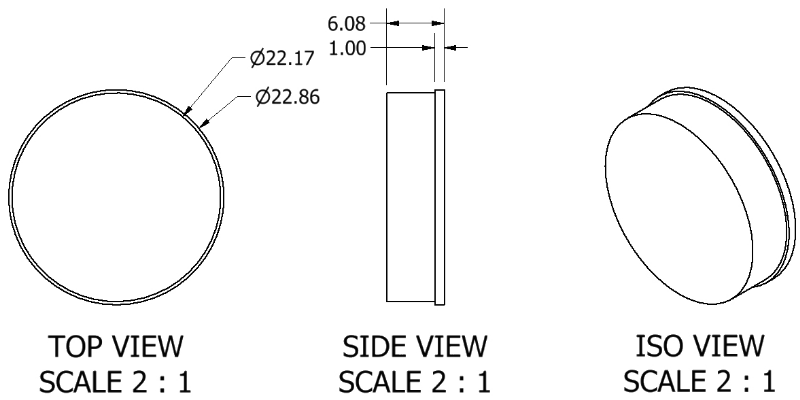


Figure 12.14: PCB 5 production.

The only component to be placed on the 5th PCB is the transmitter. Note that it does consist of a few parts which will have to be put together. The figure shows the size of the system, in reality it will look a bit different. As all the parts will be stacked without a cover.

Beacon
A visual representation of the beacon module can be seen in Figure 12.15.

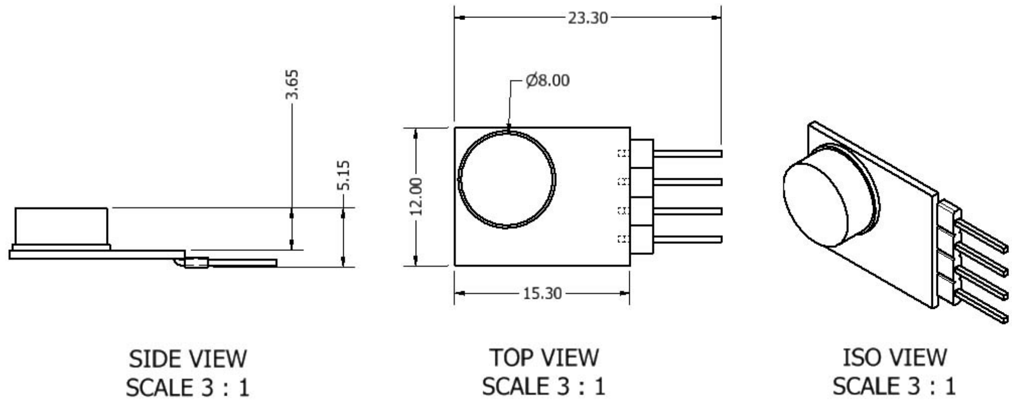


Figure 12.15: Beacon production.

The beacon system is a complete module on PCB.

12.2. Assembly

The exploded view of the TubeSat can be seen in Figure 12.16.

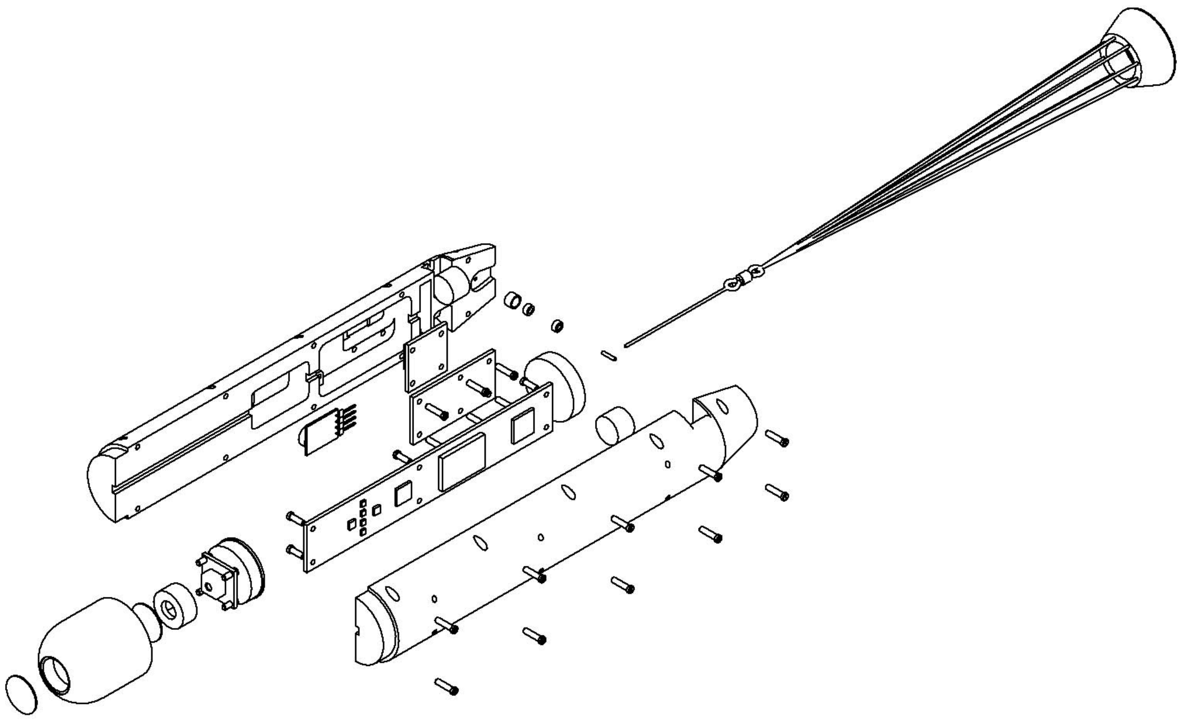


Figure 12.16: TubeSat exploded view.

From Figure 12.16 it can be seen that the assembly of the TubeSat will not be very complicated. The following procedure will be followed:

Left Half

1. The rollout system is placed into the left half of the TubeSat.
2. The rollout line is attached to the swivel and to the rollout system. It will also be rolled up immediately.
3. PCB 2, 3 and the beacon are placed and attached into the left half.

4. One of the yoyos is placed in its slot.
5. The battery is placed in its slot in the left half.
6. The wiring of PCB 2, PCB 3, the beacon and the battery is placed.

Right Half

1. PCB 1 is placed and attached in the right half.
2. The other yoyo is placed in the opposite hole and attached.
3. The wiring of PCB 1 is placed.

Nose Cone

1. The camera is placed in its sabot, and then into the nosecone.
2. The wiring of the camera is placed so it does not get in the way of the transmission system.
3. The transmission system is inserted behind the camera
4. The camera is attached to the nose cone.

Final Assembly

1. The wires between each of the parts are connected.
2. The left half of the TubeSat is attached to the right half with the bolts. Bolts were chosen over adhesives so the TubeSat can be taken apart.
3. The nose cone is attached to the main body, by gluing it.
4. The TubeSat is filled with resin[3].
5. The Hyperflo is connected to the swivel of the rollout system.
6. The lines leading from the Hyperflo are rolled into the Hyperflo itself.
7. The bottom sabot and the TubeSat are treated with anti-friction spray.
8. The bottom sabot is placed in the DART, followed by the TubeSat.
9. The ejection sabots are placed on the TubeSat, followed by the Hyperflo. Note that force will need to be exerted to compress the ejection sabots.
10. The nose cone of the DART is placed and the system is ready to be launched.

[3] http://www.polyservice.nl/Poly-Pol-PS-230-250-ml-Giethars-p-16150.html [cited 15 June 2016]

13 Costs

This chapter contains the estimated cost of the TubeSat. It is structured as follows: In Section 13.1, a table is presented with the estimated costs for each individual component. The material cost for parts produced at TME will be presented in Section 13.2. Followed by the final costs discussion in Section 13.3. Finally the recommendations will can be found in Section 13.4.

Note that references/footnotes were not used on purpose in this chapter, the references and footnotes can be found in Appendix A.

13.1. Production Costs

The production costs (single unit) of each separate component are presented in Tables 13.1 and 13.2.

Table 13.1: Production costs, single unit (Part 1).

System	Parts	COTS	Production	Costs $
Outer shell	Two halves	No	Custom made by TME	
Nose cone	Nose cone	No	Custom made by BCE	NAv.
	Quartz lens	No	Custom made by PGO	NAv.
Hyperflo	Hyperflo	No	Custom made by TME	
	Swivel	Yes	Purchased from Berkley-Fishing	2
	Rope/Line	Yes	Purchased from Berkley-Fishing	3
	Steel pin	Yes	Purchased through Aliexpress	2
Sabot	Nose	No	Custom made by Sorbothane	NAv.
	Tail 8x	No	Custom made by Sorbothane	NAv.
Yo-yo's	Yo-yo 2x	No	Custom Made by TME	NAv.
	Bearing ball 2x	Yes	Purchased from VXB	(2 pc.) 3
Ejection system	Bearing 2x	Yes	Purchased from SKF	NAv.
	Sleeve	Yes	Purchased from Fastenal	1
Power system	Battery	Yes	Purchased through Farnell	9
	DC-DC converter	Yes	Purchased through Mouser	(4 pc.) 4
	DC-DC converter	Yes	Purchased through Digi-Key	3
Embedded system	Microcontroller	Yes	Purchased from Atmel	NAv.
	MicroSD card	Yes	Purchased from Sandisk	9
	MicroSD Connector	Yes	Purchased through Farnell or Mouser	3
ADCS	Accelerometer, low g's	Yes	Purchased through Farnell or Mouser	9
	Accelerometer, high g's	Yes	Purchased through Farnell or Mouser	10
	Gyroscope	Yes	Purchased through Digi-Key	7
	Magnetic sensor	Yes	Purchased through Digi-Key	4
	GPS antenna module	Yes	Purchased through Future Electronics	30
Communications	S-Band transmitter	Yes	Purchased from Syntronics	1380
	S-Band antenna	Yes	Purchased from Syntronics	518
	BtB connector	Yes	Purchased from HRS	8
	Modulator	Yes	Purchased from Analog Devices	8
	DAC	Yes	Purchased from Analog Devices	80
			Total Costs (Current Table)	2093

Table 13.2: Production costs, single unit (Part 2).

System	Parts	COTS	Production	Costs $
Video system	Camera module	Yes	Purchased from STMicroelectronics	NAv.
	Image processor	Yes	Purchased from STMicroelectronics	NAv.
Recovery system	Transmitter	Yes	Purchased from Dorji Applied Technologies	4
	Receiver	Yes	Purchased from Dorji Applied Technologies	5
Extras	Resin	Yes	Purchased from Poly-Service	6
	Bolts	Yes	Purchased from Fabory and Jeveka	200
	Epoxy instant mix	Yes	Purchased from Loctite	NAv.
	Anti-friction coating	Yes	Purchased from LIP	NAv.
			Total Costs (Current Table)	215

13.2. Material Costs

The only parts that will be made at TME are the yoyos, the Hyperflo and the outer shells. The yoyos will be made of tungsten carbide and the outer shell and the Hyperflo of the TubeSat will be made of steel. Table 13.3 contains an estimation on the material costs.

Table 13.3: Material costs.

System	Volume $[m^3]$	Costs $
Yo-yo 2x	330mm (L) x 11.2mm (D)	NAv.
Outer shell/Hyperflo	500mm (L) x 35mm (D)	18
	Total Costs	18

13.3. Final Costs

The final material and parts costs add up to about $2326. This is however by no means the final cost for the whole product. This is because the following expenses were not taken into account or were not available:

- **Production hour cost**
  Firstly the production hours should be estimated. Then based on those hours an average wage per hour should be determined, this could be different per part. This was not done yet, because no references for accurate estimation were found and giving an estimate based on nothing would be a waste of time.
- **Unavailable costs**
  A number of companies did not state prices for their products/materials, therefore the costs for those products/materials have not been provided in the tables.

13.4. Recommendations

Below, some recommendations about the TubeSat production are made.

- **Estimation of production hours**
  The estimated production hours have not been estimated yet, since it is a new product and it is not known how long the assembly would take. This can be found by actually producing the product.
- **Estimation of cost per hours**
  This recommendation should follow the recommendation above. When the amount of production hours are determined the cost per hour for each part of the production should be determined, to get a better overview of the total costs.
- **Call the companies where the prices are not available**
  Some companies do not state the prices for the products or materials. This is yet to be investigated.

14 Market Analysis

For every product brought on the market, a market analysis has to be done up front to investigate the viability of the product. The focus of the market analysis of this project has been established in collaboration with Hein Olthof from TME. In Section 14.1 the focus of the market analysis is explained. Sections 14.2 and 14.3 elaborate on the European and Non-European market, respectively.

14.1. Focus of Market Analysis

The product designed during this project, the TubeSat, is not meant to be brought on the market. The TubeSat is being developed to validate the DART rocket. Because of the very specific dimensions of the TubeSat, it can only be used in combination with the DART rocket. However, if customers would buy the DART rocket to do research on the atmosphere between 50 and 120 km altitude, for which the DART rocket has been developed, they would have their own specific wishes regarding scientific measurement equipment. Every customer would like to measure different atmospheric parameters, which means that they would have to design their own payload or have it designed for them to fit their wishes. The DART rocket is meant for a very specific and limited market, hence the chance of a customer exactly needing the same equipment is very small. Therefore, it can be concluded that there is no market for the TubeSat itself.

Therefore the focus of the market analysis is shifted to the DART rocket itself. TME has already done an analysis for their return on investment for the DART rocket during the start-up of their company. By networking, mainly with climate and meteorology institutes, they have established that there is a sufficient market and return on investment for their project, due to the fact that the DART rocket is a low-cost and rapidly deployable rocket system. TME have also established a target price for the rocket already. Because the world of sounding rockets is a closed world in which prices are not openly discussed, it is not possible for the team to verify this target price. Also, a complete, general market analysis would not add any value to the project for TME.

For this reason, it has been decided in consultation with TME that this market analysis will focus on a part not yet performed by TME: *The market for using the DART rocket to validate data of the mesosphere and lower thermosphere obtained by Earth observation satellites*[1]. The focus will be on the European market. This choice is based on the goals of TME. They would first like to establish a market share in the European market of the sounding rocket industry before extending their market to Asia and overseas, mainly because the paperwork in order to ship rockets is extensive[2]. However, during the market analysis a lot of non-European Earth-observation missions were found which would potentially have the need for sounding rockets, these missions will also be listed. These could be used once TME is ready to broaden their market to outside Europe.

14.2. List of European Earth's Atmosphere Observation Missions

A list has been established of European Earth observation missions researching the mesosphere and lower thermosphere, which potentially have the need to validate the obtained data using the DART rocket. The agencies in charge of these missions have been contacted for a preliminary assessment of their interest in the DART rocket. However, it often takes a while to get a response from agencies as large as these. A ten-week period is therefore not long enough to properly assess their interest. No responses have been received yet from these agencies. However, these European missions are listed below to give TME a general idea of how many missions could potentially benefit from the DART rocket.

European Space Agency

The European Space Agency (ESA) has several programs running, of which one is called Copernicus. This program consists of a new family of satellites dedicated to making accurate observations of Earth. The Copernicus program consists of 6 different missions, also called Sentinels. Each of these missions will be accomplished with at least 2

[1] The mesosphere and lower thermosphere are the atmospheric layers between 50 and 120 km altitude.

[2] The information given in this paragraph as well as the suggestion for the market analysis was provided by Hein Olthof from TME during a meeting with the design team and Hein on April 25, 2016.

satellites. The data of two of these missions could potentially be validated with the DART rocket, namely the data of Sentinel-4 and -5. These two missions will be dedicated to providing information of atmospheric variables and monitoring air quality, stratospheric ozone and solar radiation. They will also be used for climate monitoring. The launch of the first satellites of Sentinel-4 and -5 is planned for 2019[3].

Swedish Space Corporation

The Swedish Space Corporation (SSC) leads an international aeronomy and astronomy minisatellite mission called Odin. The mission is a partnership between Sweden, Canada, Finland and France. SSC is responsible for the satellite design and development and the satellite operation. The main objective of the Odin mission is atmospheric research into stratospheric and mesospheric ozone and coupling of atmospheric regions[4]. Odin has also studied the formation of stars, but that mission was finished in the Spring of 2007 and is not of importance in this case. One of the main instruments for the aeronomy mission is an Optical Spectrograph and InfraRed Imager System (OSIRIS). OSIRIS scans the Earth's atmosphere between 10 and 120 km altitude[5] and produces height profiles of O_3, NO_2 and stratospheric aerosols. The other main instrument for the aeronomy mission is a sub-millimetre radiometer (SMR) which produces height profiles of other atmospheric species, N_2O among others[6].

14.3. List of Non-European Earth's Atmosphere Observation Missions

Since the list of European Earth's atmosphere observation missions researching the mesosphere and lower thermosphere is rather short, also a list of non-European missions has been established. This list can be used as a reference when TME decides to broaden their market to outside of Europe.

NASA

The National Aeronautics and Space Administration (NASA) has several missions observing the mesosphere and lower thermosphere, for which it might be possible to use the DART research rocket to validate the obtained data. Three missions are described below.

The first mission is Thermosphere, Ionosphere, Mesosphere Energetics and Dynamics (TIMED) which explores the region between 60 and 180 km altitude. The main objective of the TIMED mission is to study the energy transfer into and out of the Mesosphere and Lower Thermosphere/Ionosphere (MLTI) region and the variations in pressure, temperature and winds that are caused by this energy transfer. Four instruments are installed on TIMED in order to do all the measurements[7]. The first one is an ultraviolet spectrograph which determines the composition and temperature profiles of the MLTI region. It also measures its auroral energy inputs. The second instrument measures the heat emitted by the atmosphere, the vertical distribution of elemental constituents and the temperature profiles from the Earth's surface up to 180 km altitude. The third instrument on board is a Doppler interferometer measuring the wind in the MLTI region. This instrument determines the speed and direction of the wind. The last instrument on board consists of a spectrometer and photometers to measure solar ultraviolet radiation[56]. Although the DART rocket does not reach 180 km it could still be useful for the TIMED mission to at least validate a part of their data efficiently.

The second mission is Aeronomy of Ice in the Mesosphere (AIM). AIM studies the Polar Mesospheric Clouds (PMC's) which form an icy membrane in the mesosphere. These PMC's seem to be occurring more frequently in the recent years. AIM measures thermal, chemical and other properties of the environment in which the PMC's arise. It also studies how the PMC's are distributed and the size of the particles within them. AIM has three instruments on board in order to accomplish this: one takes pictures of the PMC's, another one measures the temperature and composition of the environment and a third one measures how much dust enters the mesosphere from meteors. The AIM measurements form the first part of studies into long-term variations in the mesosphere and their influence on global climate change[8]. However, because the AIM mission was launched in 2007 and has already been extended in 2013[9], it might be the case that the AIM mission has already been completed and the data has already been analysed when the DART research rocket becomes available on the market.

The last mission is the Earth Observing System Aura. The main objective of this mission is to determine the global distribution of temperature and the concentration of trace gases in the atmosphere up to and including the mesosphere. Aura has four instruments on board to take these measurements, of which, one has already been shutdown. The other three still measure emissions from ozone and other trace gasses[10]. Again the DART rocket could be used to validate their measurements.

[3] URL https://directory.eoportal.org/web/eoportal/satellite-missions/c-missions/copernicus-sentinel-4 [cited 28 April 2016]
[4] URL https://earth.esa.int/web/guest/missions/3rd-party-missions/current-missions/odin [cited 28 April 2016]
[5] URL http://www.snsb.se/en/Home/Space-Activities-in-Sweden/Satellites/Odin/ [cited 28 April 2016]
[6] URL http://odin-osiris.usask.ca/ [cited 28 April 2016]
[7] URL http://science.nasa.gov/missions/timed/ [cited 28 April 2016]
[8] URL http://science.nasa.gov/missions/aim/ [cited 28 April 2016]
[9] URL http://www.nasa.gov/content/goddard/aim-mission-extended [cited 28 April 2016]
[10] URL http://aura.gsfc.nasa.gov/about.html [cited 28 April 2016]

15

Risk Management

This chapter presents how the technical risks are managed. In Section 15.1, the general procedure of risk management is presented. Then in Section 15.2, the technical risks that had been identified and taken care of with the design are presented. Section 15.3 discusses the risks for the future phases of the project. Finally, Section 15.4 presents some opportunities, which are also 'risks' [45], but then with a positive impact.

15.1. Risk Management Plan

Each risk will have a unique identifier which will not be reused. A qualitative risk analysis is performed on the risks. The effects of these risk factors on product performance/quality, cost and schedule was determined. The probabilities and impacts are determined and the risks are plotted into a risk map. From this, it was determined which measures to take for which risks. A risk management strategy was developed, and the effects of this strategy were estimated, after which another risk map was drawn based on this estimation. The tests that were decided to be developed to reduce some of the risks can be found in Chapter 11.

15.1.1. Determination of Risk Probability

For creating the risk map, the probabilities of the identified risk factors were divided into a number of levels, from least to most probable. The levels are divided using clear descriptions, such that the estimations of the probabilities can be logical and have a solid ground.

For the technical risks, four levels will be used:

- **1: Demonstrated before.** Low risk probability, because the technique has been used and proven to work before, in similar conditions.
- **2: Proven technology.** Relatively low risk probability, because the technology has been used and proven, however in different fields and circumstances.
- **3: Tested on small scale.** Relatively high risk probability, the concept works, but never used in this field and/or at this scale.
- **4: New concept.** High risk probability, concept works in theory but has never been tested.

15.1.2. Determination of Risk Impacts

There are four levels of severity for the risks:

- **1: Negligible.**
- **2: Marginal.**
- **3: Critical.**
- **4: Failure.**

15.1.3. Risk Management Actions

After having assessed the different risks, one of the following possible actions were/will be taken:

- **Accept** : When very tight on resources, risks with negligible impact may be accepted.
- **Mitigate** : Actions will be taken to reduce risk impact and/or probability.
- **Watch** : No actions taken yet, but resources are used to monitor.

15.2. Risks Accounted for in Design

The following lists the risks were identified at the start of/during the project. A description of how they were taken into account during the design, or what policy was followed is discussed as well.

- **RT-01.** There is a risk that the dart does not reach the required altitude. This is a **failure** for the **performance**. Some test have been done to an altitude of around 2km, so the probability of failure is **2**.
  Accept: Managing this risk is the responsibility of TME and was outside the scope of this project.

- **RT-02.** The accelerations encountered in this mission are very high. A lot of care had to be taken to design for this. An additional complication was using COTS components which may or may not be suitable for these circumstances. Should this risk become reality, the impact on the **performance** will be a **failure**, and the probability of this is **3**.
  Mitigate: During design, only components that were proven to withstand accelerations of at least 100g were selected. This will decrease the **probability** of this risk.

- **RT-03.** There is a risk that at a certain moment the choice has to be made to use more expensive, proven parts for some subsystems. Since the plan is to use COTS components only, there will be a **critical** impact on **cost** and the probability that this will happen is **3**.
  Accept: If for a certain purpose there are no COTS components that meet all the requirements, there is no choice but to use more expensive components that do fit the requirements. Using components that are unsure to survive has a high risk of failure, meaning that what seemed like money saved would in fact be losing all money. Though that risk would still be an option in case using more expensive components is not viable or is impossible with the available budget. This risk is a particular important one to communicate to the client.
  During the design, all electronic components chosen were COTS. There are some components that are not COTS, these are components that need to be produced either by TME or outsourced, such as the structure and Hyperflo. However, these have been designed then, and their costs are not high.

- **RT-04.** Using COTS components instead of own production makes one dependant on the manufacturer. This can have a **critical** impact on the **schedule**. But since it can be assumed that companies that do not keep to their delivery times are less likely to sustain in the market, the probability is estimated **1**.
  Mitigate: When choosing components, shops were chosen that are well known. This will reduce the risk **probability**, but the effect is not very large, as it was already very low.

- **RT-05.** If the TubeSat does not fit, this will simply be a **failure** in **performance**, and possibly also for **cost** and **schedule**, since you have to go back and redesign. The probability estimated is **3**, because from experience from previous projects, it is known that a system can easily become larger than you think.
  Mitigate: Contingency management of the volume budget to reduce the risk **probability**.

- **RT-06.** The probability that the TubeSat is not retrieved is estimated to be very high, **4**, but since the requirements state that all relevant data must be transmitted to the operator during descent, the impact on the **performance** is **negligible**.
  Accept: From the customer's needs, the main objective is to retrieve the data of the TubeSat, which have to be transmitted to a ground station. Because of this, the risk of not being able to recover the TubeSat was accepted.

- **RT-07.** The use of CAD is a risk factor, because it lets you fit everything perfectly in any space. Combined with the small size creates the risk of designing a system that cannot be produced. This is a **failure** in terms of performance, with a probability of **3**.
  Mitigate: The systems engineer keeps track of all design steps and this will be an explicit task, to make sure the system is producible. This reduces the risk **probability**.

- **RT-08.** In this case, the use of CAD is a risk factor, because it lets you fit everything perfectly in any space. Combined with the small size creates the risk of designing a system that cannot be produced. This has a **critical** impact on **cost** and **schedule**, with a probability of **3**.
  RT-08: Mitigate. The systems engineer keeps track of all design steps and this will be an explicit task, to make sure the system is producible. This reduces the risk **probability**.

- **RT-09.** There is a risk that the TubeSat is not able to cope with the high temperatures encountered during descent. This will be a mission **failure**, with a probability of **3**, since thermal protection systems do exist, but have never been done at this scale.
  RT-09: Mitigate. Research was performed on the temperatures that need to be coped with. The design was made such that there is insulation between the stagnation point and the electronics behind. This reduces the risk **probability**.

- **RT-10.** Electrical discharges pose a risk to electronic systems. If this happens it leads to a **failure** impact on the mission performance. The probability is estimated to be **2**.

Mitigate: The TubeSat was designed to protect the electronics, by placing them inside a metal shell. There is still a connection to the antenna however that is outside of this. Therefore, a surge protection is put between the antenna and the rest of the electronics. This reduces the **impact** of this risk.

- **RT-11.** Because of the limited time for transmitting the data, there is a risk that not all data is transmitted in time. This is not a complete failure, but has a **critical** effect on the performance. The probability is estimated to be **2**.
  Mitigate: Use contingency management for the link budget, and consider drag devices to increase link time. These reduce the **probability** of the risk.
  During the design it was however found that there is plenty of time to send all the data required, so in the end, it did not even matter and the probability is lowered to **1** without actively designing for descent time increase.

- **RT-12.** There is a risk that the sensors do not record properly during the mission, which will lead to mission **failure**. These are things that have been used in other circumstances, so the probability of this risk is **2**.
  Mitigate: The **probability** of this risk is reduced by designing tests of these subsystems.

- **RT-13.** Insufficient power supply during the mission will lead to **failure** of the mission. The probability is estimated to be **3**, because of the small scale in this mission.
  Mitigate: Use contingency management of the power budget. The **probability** of this risk was further reduced by designing tests of these subsystems.

- **RT-14.** As the TubeSat may not be retrieved, there is a risk that it can harm the environment. If this is the case, that goes against the requirements of the mission, thus it is a **failure**. The probability is estimated to be **2**.
  Mitigate: Use a sustainable design approach to reduce the **impact** of this risk.

- **RT-15.** The ejection system had been tested, but has been changed slightly. Therefore, there is a risk that it does not work as desired: due to the shorter spring, there is the risk that the TubeSat is not pushed out entirely, and stays stuck in the DART. This added on by the fact that both the outer shell of the satellite and the DART are made of steel. Not ejecting will mean **failure**, because filming and communications will be impossible. The probability is only **2**, because it is only slightly adapted from a proven system.
  Mitigate: Therefore, to reduce the **probability**, lubrication is used between the TubeSat and the DART. Additionally, a test will be performed to check that the TubeSat is ejected successfully.

- **RT-16.** The data received from the TubeSat may contain errors, and certain chunks could be rendered not useful as a result. This is a risk with a **marginal** impact on **performance**, but with a probability of **1**, because a properly designed link has a very low probability of errors.
  Accept: No action required, because the risk is very low, it was decided not to add a FEC because the SNR was sufficient during the whole descent.

- **RT-17.** A prerequisite however for the statement in **RT-16** is that the TubeSat must point the antenna down. There is the risk that the TubeSat becomes unstable, starts tumbling, making communication impossible. This is a **failure**, with a probability of **3**.
  Mitigate: To reduce the **probability** of this risk, the chosen design has a drag device stabilising it, instead of a design with drag/stability devices on the TubeSat itself. This makes it harder to tumble. Also, the data will be sent continuously, multiple times.

- **RT-18.** Due to the shape of the nosecone, there is a risk that the expulsion spring gets stuck around the nosecone. The TubeSat may not eject and this will be a **failure** with a probability of **2**.
  Mitigate: In the design, a plate is put between the spring and the TubeSat, which makes the risk non-existent.

- **RT-19.** The brittle material of the nosecone has a risk of shattering due to the vibrations, with an expected probability of **4**. This has a **marginal** effect on the performance, because the rest of the structure remains intact and the components can still function, only there is no protection against the aerodynamic loads and heating. For the camera it does not matter a lot because it only has to film for 30 seconds. The antenna is protected behind the camera, and heating takes a while anyway, so only a small decrease in transmission time could occur if this happens.
  Mitigate: The **probability** of this risk is reduced by enveloping the nosecone in sorbothane.

- **RT-20.** There is a risk that the lens of the camera shatters due to the vibrations. This is something that has happened before [1], therefore the probability is estimated **4**. The impact would be **failure**, because the mission objective of taking a video will not be accomplished.
  Mitigate: The **probability** of this risk is reduced by putting a rubber ring around the lens to damp the vibrations.

Figure 15.1 shows these risks with their impacts and probabilities plotted in a risk map, before and after the measures taken.

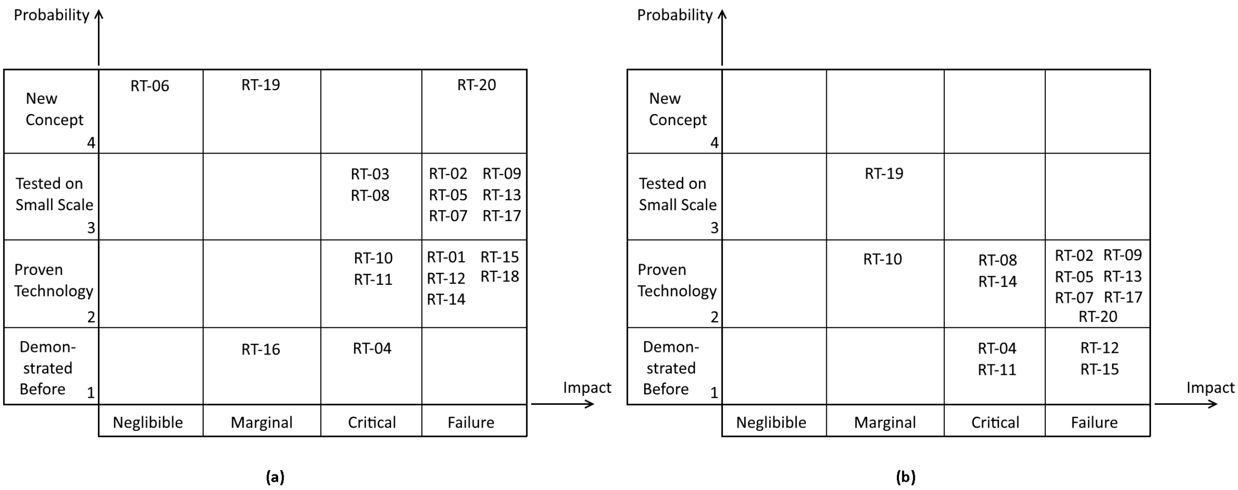


Figure 15.1: Risk map of technical risks before (a) and after (b) conducting risk management strategy.

15.3. Risks for Future Phases

The risks discussed in this section have been identified, but the measures still have to be performed. These too are plotted in a risk map, including the expected effects if the measures are taken, see Figure 15.2.

- **RTF-01.** The ejection system had been tested, but has been changed slightly. Therefore, there is a risk that it does not work as desired: the spring could still give the TubeSat a push outside of the DART. This only has a **marginal** impact, because the Hyperflo will stabilise it eventually. The probability is only **2**, because it is only slightly adapted from a proven system.
  Mitigate: Therefore, to reduce the **probability**, a test will be performed to check that the spring does not extend to outside of the DART.

- **RTF-02.** When the cg of the satellite is not low enough, there is a risk that although it falls stably, the body is flipped around: descending with the nose upward. This would be a **failure**, because communications will be impossible, and has a probability of **2**.
  Mitigate: To mitigate this risk, a detailed simulation (wind tunnel test is unlikely) must be performed to analyse the behaviour of the entire system, not being modelled as point masses. If instability is observed, alterations should be made to the design to move the cg location, until the behaviour is satisfactory. In the end, this reduced the **probability** of this risk.

- **RTF-03.** During transport, the satellite may get damaged. In particular impacts on the nosecone and the Hyperflo cables that can be cut. Depending on the severity, this could render the satellite unfit for launch, causing a **failure**. The probability is estimated **2**.
  Mitigate: To reduce the **probability** and **impact** of this risk, it is recommended to design a transport jig, which could be as simple as a box with soft fillings.

[1] Based on experience from Ir. K.J. Sudmeijer of TU Delft.

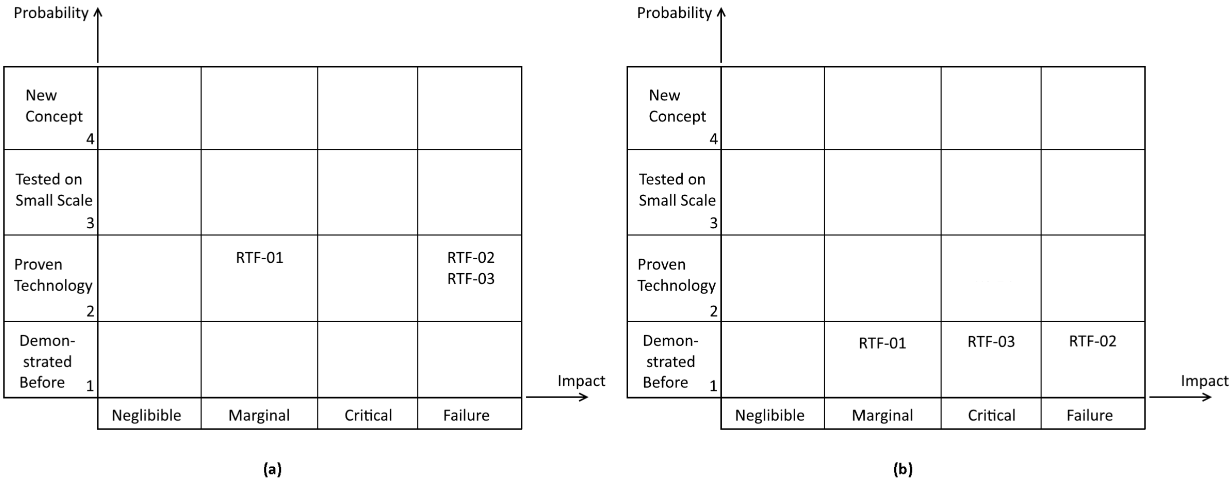


Figure 15.2: Risk map of future technical risks to be addressed before (a) and after (b) conducting risk management strategy.

15.4. Opportunities

Two opportunities have been identified and are listed below. These of course do not have to be mitigated, but could bring beneficial effects to the mission/project when they are taken.

- Launching from Esrange Kiruna, Sweden was a highly likely option, according to TME. With this latitude and the planned altitude, it is possible to launch the satellite to film footage with the light of an aurora borealis surrounding it. This makes the video extra spectacular and increase the promotional value.
- Most of the COTS components listed are sold in dollars, buying them with a higher Euro rate, if/when that happens, increases the return on investment.

16
Operations & Logistics

In order to perform the mission of the TubeSat, the DART rocket system has to be launched. Thus an operation plan is made, not only for the TubeSat, but also for the rocket launch. This plan is based on the safety regulations of the launch site, therefore these are discussed in Section 16.1. The operational plan itself can be found in Section 16.2 and in Section 16.3 one can find a contingency plan for the operations. After the operations have been discussed, the logistics are addressed, in particular the transport regulations in Section 16.4 and the required facilities in Section 16.5.

16.1. Launch Site Safety Regulations

This section addresses the safety regulations that are implemented at the launch site. Since TME indicated that the launch will probably take place at the Esrange Space Centre (ESC) near Kiruna, Sweden, the safety regulations of that launch site are used in this section[58]. Since the ESC is located in Sweden the safety regulations are based on Swedish law and Swedish safety and security regulations. All operations should comply with these legislation's. It is possible to get a waiver for safety requirements that cannot be satisfied, but this requires a lot more work because it requires proof that this particular system will not decrease the safety of personnel and facilities. Both the ground safety and the flight safety are discussed below.

16.1.1. Ground Safety

During ground operations at the launch site, so from the moment the system enters the launch site until the system leaves the launchpad, the ground safety policy of the ESC is active. In this case the system consists of the DART, the booster, and TubeSat. The ESC uses the following methods to protect personnel and property from harm by systems on site:

- Identify all the known hazards associated with the project.
- Implement safety criteria.
- Minimise exposure of personnel to hazardous systems.
- Establish safe operating procedures.
- Plan for contingencies.

The **identification of the hazards** can already be started during the design of the system. Hazardous systems are categorised as a class A or a class B system. A class A system meets three criteria: (1) initiation of the system can lead to events which can result in injury to or even death of personnel or damage to property, (2) enough potential energy exists to initiate the device; and (3) no safety devices control the energy output. Class B systems are most likely not initiated by themselves and cannot injure personnel or damage property when initiated at the wrong moment. All systems are assumed to be a class A system until it is proven to be a class B system. Hazardous systems include, but are not limited to, electronics, pressurised and chemical systems. For the TubeSat the power, embedded, ejection are identified as hazardous. A SAFE/ARM switch is needed to keep the ejection system from causing harm, therefore there needs to be a SAFE/ARM switch inside of the DART. The power and embedded system are class B hazardous systems and they do not require further safety measures inside of the satellite.

The used **safety criteria** depend on the category of the system. The criteria for a class A system are stricter than those for a class B system. The criteria for both class A and B systems include design and operations considerations. These considerations will be used during the design of the TubeSat and the writing of the operational plan. However, in this part, these considerations will not be elaborated on. In Section 3.3 the actual requirements set by the regulations are stated.

To **minimise exposure of personnel to hazardous systems** only active essential personnel is allowed to be in the danger areas. These areas are the launch pad, explosive handling areas, and other areas in which hazardous systems are located. There is a minimum of two persons and a maximum of twelve persons per area. This number includes the safety officer and further personnel designated to the specific danger area. Additionally, the amount of time the personnel spends in danger areas should be limited, so an efficient task execution is needed.

A **safe operating procedure** complies with all regulations stated in the Safety Manual. A first version of this procedure is outlined in Section 16.2.

16.1.2. Flight Safety

Flight safety is used to make sure the impact of the rocket and payload is in the designated area(s). To predict whether nothing will land outside this area, the flight of the rocket has to be simulated before launch to calculate the trajectory of the rocket and debris. In this simulation actual wind measurements made on the launch day should be included to calculate the orientation of the launch tower. The regulations on flight safety also dictate the communications to the local population in the period before the launch and the warning signals on launch day. Furthermore, real-time position data must be provided. This can be done using a ground-based system or an on-board system. When using a ground-based system it must be made sure this system works up till apogee.

16.2. Operations Manual

In this section not only the manual for the operation of the TubeSat but for the the entire launcher is given. This manual is based on a general rocket-satellite combination, defined as a class A system. Although it is set up for a launch at ESC the procedure will not change significantly for an other launch site.

One should keep in mind that at ESC, all the technical data of the communications including the frequency, output power and bandwidth has to be submitted two months in advance in order to comply with their frequency plan. This will also be the case for different launch sites.

Below, the manual is given as a list of chronological steps from the arrival on the launch base until the landing of the TubeSat:

1. Install launch tower on launchpad. This is only the initial installation, making sure the tower is stable and can be operated.
2. Perform wind measurements. These measurements should be taken at multiple altitudes, for example with a small weather balloon. The launch should be aborted when winds are stronger than the maximum allowed wind.
3. Perform a trajectory simulation using actual wind data. The simulation should be done for multiple launch tower headings to find the best tower settings. The best settings are defined by a combination of two requirements: (1) all debris should land in the ESC impact area and (2) the highest possible apogee, while still meeting requirement (1), should be reached.
4. Adjust the tower heading to the one found in the simulation.
5. Check all ground systems for proper operation. When all systems work properly, proceed to the next step, otherwise hold the procedure until faults are fixed.
6. Check all satellite systems for proper operation. When all system work properly, proceed to the next step, otherwise hold the procedure until faults are fixed.
7. Install TubeSat in the DART rocket.
8. Install the rocket engine in the booster.
9. Assemble booster-dart system.
10. Install ignition into the engine.
11. Install the rocket into the launch tower.
12. Connect ignition system to the tower.
13. Clear launch site.
14. Ignite rocket.
15. Track rocket.
16. Stop tracking when TubeSat falls behind horizon and signal is lost.
17. Retrieve TubeSat from landing location.

From this operations manual a training plan can be derived. This plan is then used to train personnel that is going to work with the system. Training is needed to make sure operations are conducted in a proper and safe way.

16.3. Contingency Plan

The reliability of the TubeSat and the DART rocket cannot be 100%, because errors and failures might occur. To ensure the safety of the payload, but mainly of the people that might be effected by the DART system and TubeSat, a contingency plan is determined. The contingency plan shall identify the risks and their impact, but mainly safety measures, which will be introduced to make sure that no personnel is harmed. The mission continuation after failure is also covered in the contingency plan.

16.3.1. Operational Risks and Impact

To define a contingency plan, all the risks and their impact need to be identified first. The most important risks to negate are the ones that could harm people. Furthermore, the risks that harm the objectives of the mission are added as well.

1. **Operation of all ground system(s)**
   Without the ground systems a launch is not allowed.
2. **Failure of ground system(s) during the mission**
   Loss of communication will be a loss of scientific data. A loss of tracking provides uncertainty about when and where the booster, dart and TubeSat will land.
3. **Errors in the wind measurements and simulation**
   This could mean that the booster, dart and TubeSat could land outside the secured ground.
4. **Electrical shortage during installation of the battery**
   During the assembly of the TubeSat and the insulation of the battery the following could happen.
   - **Spontaneous combustion of the batteries**
   - **Melting of the cables**
   - **Breakage of the circuits**
   - **False activation of subsystems due to electrical shortage**
   - **Mechanic gets electric shock**
5. **Transport Damage**
   During transport damages might occur. These can cause issues during transport or in later stages.
   - **False activation of subsystems due to transport damage**
   - **Subsystems are activated during the flight**
   - **Other kind of damage from transport**
6. **Activation of the ejection system during the installation in the DART**
   This can cause harm to the personnel handling the TubeSat.
7. **Failure of the ejection system**
   Whether activated too early, too late or never this will cause the descent time of the TubeSat to change. Either it will fall from a wrong altitude or the descent velocity will get too high.
8. **The booster explodes**
   This will stop the mission immediately.
9. **The booster does not perform as expected**
   This will lead to a lower ejection altitude or even no lift off.
10. **Recovery system is activated during assembly**
    This can cause harm to the mechanics assembling the TubeSat.
11. **Recovery system is activated inside DART**
    This might cause a failure of the ejection thus failing the mission.
12. **TubeSat lands in a non retrievable zone**
    - **It lands in a location from which retrieval is dangerous**
    - **The landing location is not legally accessible**
      This location could for example be a military zone.
13. **TubeSat landing causes destruction of property**
    This could be coupled to the miscalculation of the wind, but when the TubeSat lands outside of the landing zone it could hit and damage something.

16.3.2. Safety Measures

After all the operational risks and their impacts are determined, safety measures are defined. The numbers specify for which risk the safety measure accounts.

1,2 Redundancy
To ensure that the ground system works without critical failures, it should be designed redundantly.

3 Safety Margins
When simulating the mission and taking the weather measurements, safety margins need to be taken into account in case of any errors, or faulty instruments. These margins need to be established using completed missions on the launch site and data from lower altitude DART launches.

4,6,10 Assembly Jig
An assembly jig should be used during the production of the TubeSat. This jig should make sure that if during assembly a subsystem is activated it can do no harm to the mechanics. In this jig the final functionality check can be done. When the TubeSat gets installed into the DART a similar (or the same) jig should be used. When moving an assembled TubeSat or DART around on the launch site it should remain inside this jig. Only when installing the DART onto the booster on the launch platform it can be taken out of the jig.

4 Electrical Safety
To prevent electrical damages different safety measures should be taken. Grounding the jig, high voltage gloves, or fire redundant clothing are some of them. Also special care has to be taken to prevent any electrostatic discharges that could render the electronics useless.

4,5 Spare Components
A lot of risks could be solved by having spare components, so if a component breaks/is faulty, it could just be replaced by a working one.

4,5,7 Spare TubeSat
When a problem is discovered during a very late stage or when the TubeSat gets damaged/destroyed it needs a quick replacement.

5 Transport Safety
Transportation can be a big risk, however, using special casing for the TubeSat and its parts will ensure that the parts do not move/get damaged during transport. For electrical components antistatic casing should be used.

7,8,9,11 Second DART System
Have a second DART system available for when there is not enough time to find the source of a problem and fix it.

12 Experts
When ever there is a recovery that needs people to in to a dangerous area to retrieve the TubeSat. It could mean that experts are needed. For instance divers when it is in the ocean or mountaineers when it is on a glacier.

12 Permit
If the TubeSat lands in a restricted zone permission to retrieve it is required.

13 Insurance
The TubeSat could hit someones property. An insurance can settle the damages.

13 Slow descent speed
If the descent speed is low enough the impact of landing will not cause any damages.

16.3.3. Continuation of Mission

Implementing the safety features will lead to two results. First, if a part fails before the launch there is the opportunity to replace components or the complete TubeSat. This will take time and might cause a delay in launch time. This could in term mean that different rockets in the same campaign can also be delayed or even cancelled. Second, if a problem occurs while in flight, this will most likely cause the TubeSat to fail the mission. Most importantly, no one should be hurt while handling the TubeSat or the DART rocket.

16.4. Transport Regulations

In this section the transport regulations are discussed. First, the transport of the booster will be discussed. Then, the transport of the DART and the payload are discussed.

The booster is classified as an ADR[1]class 1 explosive. To transport this explosive the driver should have a certificate to show knowledge of the materials. In addition, the vehicle used to transport the material should have a certificate to show it is safe. Since TME does not have a qualified driver or vehicle, they cannot transport the booster by themselves and will need a different company to do this for them. The same is the case for potential reserve fuel for the booster.

The DART and TubeSat can be transported by TME themselves, since these are not dangerous items. When travelling by road, the only requirement is that the TubeSat and DART fit in the vehicle. When travelling by air, the DART especially may need some special packaging in order to be transported safely; the sharp edges may damage other luggage.

[1]The European Agreement concerning the International Carriage of Dangerous Goods by Road (ADR) is a treaty of the United Nations that governs transnational transport of hazardous materials.

16.5. Facilities

In this section, the required facilities are listed and elaborated on. The facilities are divided in primary and secondary ones. Primary being the ones that are directly needed to perform the mission, and secondary being those which are not directly required to perform the mission. First, the primary facilities are discussed, after which the secondary ones are discussed.

16.5.1. Primary Facilities

Primary facilities are defined to be facilities which are directly needed to perform the missions.

The following primary facilities are supplied by the Esrange Space Centre:

Impact Area This is the area where the DART rocket is allowed to land/crash. It it important that the DART rocket does not endanger anyone during the complete mission.

Communication on GS There needs to be communication possible between people, for instance when the ground station is large.

Safety Measures Safety measures should be taken in order to keep people safe in case the mission fails.

Antennas for telemetry These are needed to receive the transmitted data.

Tracking devices These will be important to retrieve the TubeSat.

Transportation on GS If the ground station is large, vehicles might be needed to reduce commuting time (e.g. carts, bikes).

Assembly Halls There should be a place where the entire rocket can be assembled.

Carrier Frequency This is the frequency over which the data will be send.

The following primary facilities are supplied by T-Minus Engineering B.V.:

Launch Tower The complete DART rocket will be placed on a launch tower to be able to launch it with the desired launch angle and in the desired direction.

Booster This is the part of the system that initialises the mission and actively gives impulse to the DART and the TubeSat to reach the desired altitude.

The DART This is the part of the system which transports the TubeSat to apogee and ejects it there.

The TubeSat This is the part of the system that performs position and scientific measurements and shoots video footage.

Workshops Repairs or replacements might be needed and a space should be available to perform these.

16.5.2. Secondary Facilities

Secondary facilities are defined to be facilities that are not directly related to the mission performance.

The following secondary facilities are supplied by the Esrange Space Centre:

Catering A mission might take up days, so there needs to be food and drinks for the ground staff.

Safety Equipment In case of emergency, there should be safety equipment available (e.g. jackets, fire extinguishers).

Recreation There might be large intervals between different mission phases, some things to pass time are provided (e.g. billiards, books).

Accommodation The mission may take up weeks, so accommodation is needed.

Office Space Office space is required for meetings. The space also includes office equipment.

The following secondary facilities are supplied by T-Minus Engineering B.V.:

Spare Parts When specific components are damaged they can be replaced by spare parts (e.g. fins, pumps, antennas). The spare parts should be provided by TME.

Spare Booster In case the nominal booster is not sufficient to perform the mission, a spare booster should be used instead.

Spare DART In case the nominal DART is not sufficient to perform the mission, a spare DART should be used instead.

Spare TubeSat In case the nominal TubeSat is not sufficient to perform the mission, a spare TubeSat should be used instead.

Reserve Propellant Enough propellant should be available to perform the mission. To ensure this, reserve propellant should be provided.

17 RAMS Characteristics

RAMS stands for Reliability, Availability, Maintainability and Safety. These characteristics are important for the operation of the system. They represent; the probability that the system can be operated without failures for a certain amount of time (reliability), the probability that the system is ready for use (availability), what parts should be checked and how they should be checked (maintainability), and what measures should be taken to make sure neither people nor property are harmed (safety). These characteristics can be found below.

Reliability
In systems engineering reliability is defined as: *"Probability that a system will perform in a satisfactory manner for a given period of time when used under specified operating conditions."* [15] The TubeSat works in a satisfactory manner when it accomplishes at least the following criteria [24]:

- Determine apogee within 50m accuracy.
- A clear, 1080p, 30-second video of Earth and space.
- Transmit video data back to the ground station.

To keep the probability that the system reaches this goals as high as possible it is designed with high quality components and materials. The attachment of the electronics is made in way that the PCBs are fixed to the body. By filling the cavities with resin it is made sure that the components are completely fixed with respect to each other so the connections will not break.

Availability
As stated before, the availability is the probability that the system is ready for use when required. COTS products are most of the time quickly available when needed, this is why, where possible, COTS components are used. However, some parts, like the body and the PCBs, are not available in COTS versions since these are parts that are not used in other systems. PCBs are available within a week after ordering, so this is not a real problem for the availability. Also the production of the body can be done within a week. So one can make a TubeSat in approximately one or two weeks. As can be read in Section 17 there is no scheduled maintenance that can compromise the availability. The limiting factor of the availability is the launch site. These should be reserved multiple months before a launch which makes it impossible to launch the system a week after ordering.

Maintainability
The maintainability tells how easy, accurate and safe it is to perform maintenance on a system. The TubeSat is designed for a one time use, which means that there is no planned maintenance. Although during the short life time of the TubeSat there might still be a need for repairs. During transport there is always a risk that the system gets damaged in way that compromises the proper functioning of the system, it that case repairs should be made. It might also happen that a subsystem fails before launch, in which case the part should be replaced.

To repair or replace parts that are on the inside of the TubeSat after the production one has to first remove the nosecone. To do this the adhesive between the nosecone and the TubeSat's body should be removed. This can be done with chemicals or in a mechanical way. The camera is then to be removed by removing the bolts that attach it to the body. After this the bolts that hold the two body parts together can be removed and the body can be opened. When a part on a PCB fails, one should remove the resin and PCB(s) from the body. The body can be prepared for re-use, but it is not possible to remove all resin from the PCB, therefore a new PCB is needed.

Overall the maintainability is not good for this system. A lot of parts are fixed in a way that takes a lot of time to remove. In case of the PCBs it is even not possible to repair them because of the resin. When one considers that the system is designed for a one time use this is no problem as long as it is handled properly before launch.

Safety
To make sure the TubeSat is safe to use, multiple steps have been taken. The design is made without sharp cutting edges and exposed electrical wiring, also there are no moving parts which can harm people. To make sure the TubeSat is handled in a safe way, people who handle the TubeSat should be knowledgeable of it's operational manual. A operational manual is defined in Section 16.2, a contingency plan tackling the risks for personnel can be found in Section 16.3.

18
Sustainable Development Strategy

Over the years, sustainability has become an important factor in the (life cycle) design of aircraft and satellites. Therefore, during the design of the TubeSat sustainability will also be taken into account. Sustainable development is defined as: *'Meeting the needs and aspirations of the present and not compromising this opportunity for future generations, to meet ones own need'* [9].

18.1. Strategy Trade-Off

When discussing the sustainability of the TubeSat, there are three main divisions of its life cycle:

1. **Design, Procurement and Manufacturing**:
   The sustainability of the design and production. In this phase of life, the designer is encouraged to choose design and production techniques and materials which are eco-friendly. Materials should be sustainable, preferably recyclable and have an eco-friendly acquisition process. The production techniques should be environmentally friendly. Low weight of the TubeSat is also favourable.
2. **Life**:
   The functional life of the TubeSat. The TubeSat should be designed such that during the lifetime it has reduced negative impact on the environment. This is a short phase in the lifetime of the TubeSat. The TubeSat should be of high quality and reliability, and should produce no waste. Use of renewable energy or eco-friendly batteries is also preferred. Recovery and reuse of the TubeSat prolongs this phase of the life cycle.
3. **End-of-life**:
   Sustainable end-of-life of the TubeSat. It should be designed what happens to the TubeSat after it is no longer in use, before the project enters the production phase. The TubeSat is designed for both end-of-life scenarios, either it is retrieved or not retrieved. For the first case, design for reuse or recycling should be considered. For the second case, the design should focus on a TubeSat which will be the least harmful to the environment if not retrieved, for example by using non-toxic materials.

A trade-off is made between three sustainability strategies: Eco-design, Bio-mimicry and Cradle to Cradle (C2C). The criteria used are Feasibility, Cost, Requirements, Design/Construction, Life, End of life. For each criterion there are three grades: Good (represented by green colour), Neutral (represented by yellow colour) and Insufficient (represented by red colour).

Eco-design is defined as: *'Not a specific method or tool, but rather a way of better design through analysing and synthesising in order to reduce environmental impacts throughout the product's life cycle.'* [35]. This means that Eco-design takes into account all life phases of a product, from the extraction of raw materials all the way until the disposal and possible recycling of the product. *'**Bio-mimicry** is the imitation of the models, systems, and elements of nature for the purpose of solving complex human problems'* [59]. It is taking design lessons from nature, such as evolving to survive or be resource efficient. The product should be life-like and should ensure that the product both performs really well, and fits in a sustainable system. *'**Cradle to Cradle** models human industry on nature's processes viewing materials as nutrients circulating in healthy, safe metabolisms. It suggests that industry must protect and enrich ecosystems and nature's biological metabolism while also maintaining a safe, productive technical metabolism for the high-quality use and circulation of organic and technical nutrients'* [38]. In Cradle to Cradle strategy every waste is a resource for something else. The products should fit their biological and cultural context and should use renewable energy from local sources.

For feasibility of the sustainable strategy, Eco design is the best, as its main goal is to reduce negative environmental impact. This is quite doable even when using Commercial Off-The-Shelf components. On the other hand, one of the main points in C2C is "Less bad is not good"[47]. C2C requires to leave no ecological footprint, which is almost impossible to achieve, partially because the chance of not retrieving the TubeSat is significant. Trying to design a sustainable TubeSat is more costly and therefore, all of the strategies are neutral. All strategies are equally capable of fulfilling the requirements for sustainability. For life, none of the strategies stand out, neither positively nor negatively. As already mentioned, the change of not retrieving the TubeSat is significant and therefore, in end of life, C2C is the worst. Not retrieving the TubeSat will leave a significant ecological footprint.

Table 18.1: Sustainability strategy trade-off (O = Option, Crit = Criterion).

O \ Crit	Feasibility	Cost	Requirements	Design/ Construction	Life	End-of-life
Ecodesign	*green* achievable goal	*yellow* sustainable TubeSat™ is costly	*green* capable of fulfilling requirements	*green* focus on ecological design	*green* equally good	*yellow* not the main focus, but acceptable
Biomimicry	*yellow* harder to achieve the goal	*yellow* sustainable TubeSat™ is costly	*green* capable of fulfilling requirements	*yellow* more difficult to design, but possible	*green* equally good	*yellow* not the focus, but acceptable
Cradle to Cradle	*red* significant chance of not retrieving TubeSat™	*yellow* sustainable TubeSat™ is costly	*green* capable of fulfilling requirements	*red* parts will have ecological footprint	*green* equally good	*red* will have ecological footprint

Table 18.2: Legend of sustainability strategy trade-off.

Colour	Meaning
green	Good
yellow	Neutral
red	Insufficient

Based on the sustainability strategy trade-off it has been decided to use the strategy Ecodesign during the design process. From the trade-off it was concluded that this strategy is the most realistic and feasible one for this design project.

18.2. 'The Ten Golden Rules'

'The Ten Golden Rules' is a tool which can be used to implement Ecodesign. The tool has been developed by a professor at the Royal Institute of Technology (KTH) in Stockholm. *'The Ten Golden Rules were developed by Luttropp[1] as the "lowest common denominator"of ten of the most common issues that must be addressed in Ecodesign.'* 'The Ten Golden Rules' are defined as follows [24, 37]:

1. Do not use toxic substances and utilise closed loops for necessary but toxic ones.
2. Minimise energy and resource consumption in the production phase and transport through improved housekeeping.
3. Use structural features and high quality materials to minimise weight in products if such choices do not interfere with necessary flexibility, impact strength or other functional priorities.
4. Minimise energy and resource consumption in the usage phase.
5. Promote repair and upgrading for system dependent products.
6. Promote long-life.
7. Invest in better materials, surface treatments or structural arrangements to protect products from dirt, corrosion and wear, thereby ensuring reduced maintenance and longer product life.
8. Prearrange upgrading, repair and recycling.
9. Promote upgrading, repair and recycling using simple materials.
10. Use few joining elements.

[1] Luttrop is the professor at the Royal Institute of Technology who developed the 'Ten Golden Rules'.

18.3. Implementation of 'The Ten Golden Rules'

As stated in the requirements, COTS components are used in of the TubeSat. Finding ones which are small enough to fit inside of the TubeSat was a challenge. Therefore, the size was the main driving force when choosing them. This means that the choices were limited, and not all the rules were possible to fulfil. Each rule is discussed below.

1. Not all the components comply with 'Restriction of Hazardous Substances', as some of the components are from other countries than European Union, e.g. micro controller, and for some of the components it was not exactly specified which materials are part of the product. Materials used for the shell of the TubeSat are steel, zirconia and the lenses are made of quartz. The Hyperflo is made of steel, with shark wire as a cable. The yo-yo's are made of tungsten carbide. The inside of the satellite is filled with resin. In Section 5.3 the reasoning behind the choice of material is given. Some of the materials are recyclable, e.g. steel or tungsten carbide. However, when choosing the material, sustainability was not as important as material properties.

2. Most of the companies do not specify their sustainable strategy, way of transportation or all the production methods they use. The main manufacturing method for the satellite is milling, however laser-cutting and drilling are used as well. Even though, the requirement was to not use subtractive manufacturing methods, these ones were the only suitable options, as explained in Chapter 12. The transporting method is explained in Section 16.4. Depending on the distance, different vehicle will be chosen. However, car/truck is the most probable option, as this is the easiest way to transport the DART and the TubeSat.

3 The mass of the TubeSat is bit more than $0.787 kg$, as can be seen in Section 3.5 in Table 3.5. Therefore, it can be concluded that this rule has been fulfilled, as the mass is minimised. As can be seen in Section 11.2, Figure 11.1, lowering the weight, until some point, is not a problem when trying to reach 120km.

4. Even though using a capacitor is more sustainable, the size is too big. Therefore a battery is chosen. The battery provides more energy than required, as can be seen in Chapter 8, Figure 8.3, however due to the sizing and the energy needed, it is the best fit.

5.,6. During the descent of the TubeSat, the descent control is used more for the stability, to transmit effectively, than to slow down, as there is enough time to transmit. This means that it will most likely not be reusable. However, it still might be recycled, because recovery system is designed, Section 4.1. By knowing the last GPS coordinates, and using the siganl from a beacon the TubeSat can be retrieved, either by a team or by rewarding hikers.

7. The best possible materials and COTS components were chosen to withstand all the possible, loads and temperature ranges.

8.,9. As already mentioned, the product should be recyclable. However, some of the connections might not be the best for dis-assembly, such as the COTS components, or the assembly done with glue.

10. The TubeSat has a lot of different components which need to be fixed and connected inside. However, the descend method chosen, Section 4.3 is quite simple, so it does not add extra mechanical joints, only in the Hyperflo.

From this overview it can be seen that not all the rules are fulfilled and some are fulfilled partiality. Some decisions were made during the design process which stirred the design away from sustainability. In any design there is always a trade off between the sustainability and the best performance design. High loads and temperatures needed to be taken into account, while being able to fit all the components into the required size and this made it less sustainable.

18.4. Contribution to Sustainability

The TubeSat is designed to be sustainable according to 'The Ten Golden Rules'. However, it is not possible to fulfil all the rules entirely, as explained in the Section 18.3. These rules, nevertheless, make the design more sustainable. They promote use of reusable materials, and recycling. Therefore, the TubeSat is more sustainable than it would be without using these Golden rules.

The TubeSat's main purpose is to make a video at the apogee of the DART, and in later redesign even taking different scientific measurements. This contributes to the understanding of the "Ignorosphere", which is the region between 50-100 km and mainly unexplored part of the atmosphere. More knowledge might lead to important discoveries within this layer. Mainly noctilucent clouds, as they could be linked with climate change[2] [1], could bring a lot of new information about climate change and sustainability.

As mentioned the DART is launched to 120 km to the unexplored part of the atmosphere, as no balloons or satellites could go there. The possibility is to use sound rockets. However, these are usually too expensive for simple

[2] URL http://www.bloomberg.com/news/articles/2015-06-09/at-space-s-edge-swirls-of-blue-show-atmosphere-changing [cited 9 June 2016]

and frequent missions. On the other hand the DART is a system for fast and low-cost probing of the upper atmosphere. If we compare the sustainability of the DART and the sounding rocket, it it clear that the DART is more sustainable. First, the sounding rocket is much bigger, meaning more materials, longer production. And second, because it is bigger, its heavier, so more fuel to get it into the same height, which means more emissions. Therefore it can be concluded that DART is more sustainable than sounding rockets.

18.5. Recommendations

Recommendations about how to make the TubeSat even more sustainable are discussed in this section. As have been said not all of 'The Ten Golden Rules' have been fulfilled. Nevertheless, it is recommended to follow their lead.

- It is advised to contact the suppliers and find out all the missing information about sustainability, such as sustainable strategy, materials and production methods used.
- With the advance in technologies, smaller, more sustainable COTS components could be chosen.
- The descent system could be redesigned such that the descent is slower. Which could prolong the life and make the TubeSat possible to reuse.
- If the TubeSat is possible to reuse, it should be redesigned as such. More active recovery could be designed to increase the probability of retrieving the TubeSat as well. However, recovery might also have negative impact on the environment, therefore more research is needed.
- Better connecting techniques, to promote disassembly, should be chosen.

19 Future Planning

Until now all the material presented has been developed for TME. However, from now on TME will have to continue the development of the TubeSat. This chapter will discuss the most apparent tasks in Section 19.1. But there will be a large portion of work that cannot be foreseen. Section 19.2 will discuss the traditional method for project management and the to be delivered Gantt chart. In Section 19.3 an alternative philosophy of project management is introduced and the implementations of this Agile -Scrum and Kanban- are discussed. Finally, in Section 19.4 a recommendation is made on how TME can continue from this report and plan the future of the TubeSat and the company.

19.1. Workload

After the assignment of the design has been finished TME will have to take over. The first logical step will be reading this report. Afterwards one or several transfer meetings need to be planned. In these meetings TME has the opportunity to ask the team or specific team members questions they have about the design. Additionally, there is space to have a discussion about the approach taken by the team. From this moment TME will have to continue by themselves although the team will most likely still be available for questions.

Major parts that still need to be properly designed are the embedded and power systems. As this is the most uncertain part it can still have a large impact on the entire design of the TubeSat. At the same time, the mechanical designers should become familiar with the design so they can implement changes needed by the electronics. Next, the components that are normally only provided as OEM components need to be sourced as they are not normally sold as just one piece. Towards the end of this phase, the verification and validation plan has to reviewed and the tests have to be planned.

At this point the subsystems can be produced and the verification and validation can start. After all the subsystems have been verified the entire TubeSat can be assembled and validated. Meanwhile the test launch should be planned and eventually this test should occur before the end of 2017.

With the lessons learnt from the test launch the TubeSat can be redesigned. Depending on the time it takes to redesign the TubeSat a date can be set and the actual launch can be prepared. Before the launch actually takes place, TME should invite the team that did the initial design of the TubeSat to the launch. When this is all arranged the actual launch can take place.

After the launch the video should be marketed to the entire world. Finally, the TubeSat can be redesigned to do the first scientific mission for a customer.

19.2. Waterfall

In any project three aspects need to be managed: quality, resources and time. With a project management a plan can be made on how to execute the project on time, within budget and with the desired quality.

Traditionally that is done using a waterfall method. The name indicates that all the steps are done in a somewhat linear fashion cascading from one stage to the next. These stages include: mission analysis, setting up requirements, breaking up into subsystems, designing subsystems and the verification and validation. Characteristically all the value is only presented at the end of the project. Also it is required that everything about the project is known in advance [1].

In this method there is a clear function for someone taking control of all the planning and making sure the entire team sticks to the plan. The planning is mostly done in advance and from the stage to stage deadlines get shifted around when a project is ahead of or behind the schedule. The person in charge of the plan will have to communicate with the team members to get the information he needs to maintain the plan. Also he will have to

[1] URL https://www.youtube.com/watch?v=Kc4fTryN8Nw [cited 16 June, 2016]

communicate the plan to the team. This will lead to meetings. Also the team might get the sense that they are forced to follow the plan that at times might get detached from the reality the team is working in.

The waterfall approach is commonly visualised by a Gantt chart. This type of chart is useful to understand inter-dependencies of different tasks. In Figure 19.1 the Gantt chart for the workload of Section 19.1 can be seen.

Figure 19.1: A Gantt chart representing the future workload for finishing the TubeSat.

However, there are some limitations to the Gantt chart. On a day to day scale a lot will change. The person in charge of planning will have to change the chart and is most likely the only person that really knows the plan. This poses a large risk for the planning and the overall view of work.

19.3. Agile

There are three issues within the waterfall method that cause the planning to detach from reality. First, the requirements will never be complete especially at the start of the project. Secondly, the customer will never know what he wants. Finally, the team does not support the planning and at times will want to go back and improve work that was supposed to be finished.

Within the software development industry efforts have been made to change the linear waterfall method in to a more agile planning. In 2001 the manifesto for agile software development was introduced. This manifesto calls for valuing:

- **Individuals and interactions** over processes and tools
- **Working software** over comprehensive documentation
- **Customer collaboration** over contract negotiation
- **Responding to change** over following a plan

Thus trying to find a solution using the strength of the team and not blindly following a predetermined method and trying to deliver a product as soon as possible. This will provide more insight into the project and will filter inherent issues out much sooner. Show the customer the product, a customer will find it easier to indicate what he does not want. Learn from the product and continue developing it to increase the quality of the product by incorporating the lessons learnt from the mistakes.

Compared to waterfall project management Agile provides a base on which continued value can be delivered. Also most of the Agile implementations spread the planning and the responsibility of the plan over the entire team. Maybe even more important, Agile will allow requirements to be changed throughout the project. This will help to get closer to what a customer needs and will also allow the team to incorporate new ideas or use the latest technology that got released during the project. Also, agility increases with every team member being capable of doing most tasks.

There are two popular methods which implement Agile. These methods are Scrum, Subsection 19.3.1, and Kanban, Subsection 19.3.2. Both methods focus on making the team more effective and allow the team to become predictably effective.

19.3.1. Scrum

Scrum is a project management method that originates from the software development industry. More recently it has been used to implement Agile and has found greater use outside of software. In contrast to the waterfall method, Scrum has clear intervals at which value is presented. Furthermore it involves the entire team in planning and achieving the goals, which are set to be achieved in shorter amounts of time.

There are three clear roles in Scrum. First, there is the product owner. This is either the client or the manager that manages multiple teams. Second, there is the development team. And last, each team has one member that guides the team, who is called the Scrum master.

The following aspects make up the general flow of using Scrum. The design team of the TubeSat has been able to implement the first four.[2]:

- **Backlog**
  All tasks that will have to be performed by the team will be on the backlog. During a project tasks are taken off and broken down in smaller activities, and new tasks are added. The order indicates the chronological order in which the tasks need to be performed.
- **Sprint**
  A sprint is a set amount of time, normally between two or four weeks. At the start of the sprint the team decides what tasks from the backlog should be performed. The amount of work that is left at the end of the sprint is an indication on how well a team can estimate the work needed for a task. For this relatively short project, the team has implemented one-week sprints.
- **Scrum board**
  For every sprint a new Scrum board needs to be created. In general, it is a table with the main goals in the first column, followed by 'to do', 'being done' and 'finished' columns. Each task is placed in the first column and has a horizontal swim lane in which all the team members place sticky notes with activities that need to be done, are being done, or are finished. The order of the urgency of the deliverables is displayed by the order of the swim lanes with the most urgent deliverables placed at the top. The Scrum board can be seen as a live planning. The Scrum board of the design team can be seen in Figure 19.2.
- **Daily Scrum meeting**
  Each morning the work day starts with a general meeting in front of the Scrum board in which every team member states what he/she has done on the previous day and what he/she is planning to do on the present day. This provides the opportunity for the rest of the team to re-allocate work force and make sure the swim lanes with the greatest importance get finished first. The sticky notes are also moved to their respective new column as determined by the team member working on the task.
- **Demonstration**
  During the demonstration the team shows the product owner what they came up with. This is the moment at which the product owner can give feedback. This feedback can directly be implemented in the next sprint.
- **Retrospective**
  After each sprint the team looks back at the sprint during the retrospective. This discussion is not focused on actual technical details but more on process. This is a good moment to review how the team works together and how decisions are made.

Scrum has however two prerequisites to apply it effectively. Everyone has to work at the same times and everybody has to be critical about the progress that is made. This is why the daily meeting is important because everybody has to show the progress or issues found the previous day. The way in which the issues get tackled by the group is how Scrum is Agile.

19.3.2. Kanban

Compared to Scrum, Kanban has very little rules. However the very little rules still allow a group to work effectively even if they do not work at the same times or an equal amount of time. The origin of Kanban can be found in the Toyota Motor Corporation to plan the Lean or Just-in-Time Manufacturing.

Just like Scrum, Kanban uses a board to visualise the planning. On this board several essential columns are [3]:

- **Backlog:** all assignments that need to be performed are in here.
- **Breakdown:** the assignments get broken down into separate tasks.
- **In progress:** the tasks are taken from the breakdown and get done.
- **Verify:** after finishing the task a verification has to be performed to make sure the result is useful.
- **Done:** after a task has been verified it waits with the other tasks until the entire assignment is delivered.

[2] URL https://www.youtube.com/watch?v=XU0llRltyFM [cited 20 April, 2016]
[3] https://www.youtube.com/watch?v=CD0y-aU1sXo [cited 16 June, 2016]

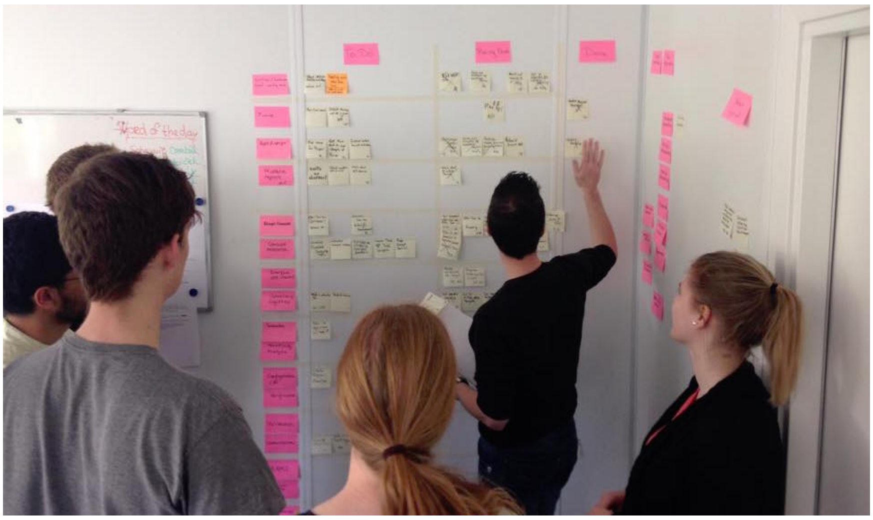


Figure 19.2: The actual Scrum board of the design team in use during a stand-up meeting.

The tasks -unlike Scrum- can continuously move around after work on them has been done. The task are ordered in columns with the most urgent ones on top. There is only one additional rule that applies to the breakdown. This is the Work In Progress (WIP) limit. This rule limits the amount of tasks or assignments that are allowed to be in the respective columns. Whenever a task is stagnating it would most likely cause the column upstream to fill up and the column downstream to empty. Team members will notice that the task is lagging and can swarm to get it done.

Kanban is very adaptive and can be tailored to the needs of the team as long as the changes are incremental. For instance the WIP limits can be tuned, but also other elements can be added as long as they add value to the team. This could for example be elements like a daily meeting to discuss the issues everyone is facing or a retrospective every once in a while or roles can be assigned. It is not needed but there could be someone responsible to keep track of the progress or a chief engineer can be assigned. The last one can do the verification signing off on the completed designs. This could be very useful if a team does not work at the same times or the same amount of time.

19.4. Recommendations

The following recommendations are made for the planning:

- Do plan to have the transfer meetings.
- Try to incorporate Agile in the planning.
- Try out Kanban as a planning method.

20 Conclusions

In order for TME to showcase the capabilities of the DART research rocket, DSE Group 15 was tasked with designing the TubeSat. The TubeSat will provide 30 seconds of high-definition video footage of its deployment, validate the height of apogee and measure the kinematic behaviour of the DART rocket during launch and the TubeSat during descent. Additionally, a retrieval method for the TubeSat has been designed and the ejection system designed by TME has been adapted.

First of all, the 30-second video turned out to be the most complex part of the design. However, a camera system was found which would be able to make the required video and is small enough to fit comfortably. The yo-yo de-spin mechanism has been implemented to make sure the video will be of good quality and pleasant to watch. Less post-processing will be required. Two yo-yo's made of tungsten carbide have been fitted in the TubeSat.

The real challenge started with the transmission of the data. This was solved by configuring a complete transmission system and using the ground based system available at the most likely launch sites. Both systems use S-band frequencies. To determine the transmission time available a descent simulation was performed. With this simulation it was found that no additional drag device was needed to increase descent time.

However, a hyperflo drag system, made of steel, is used to point the TubeSat to make sure that the antenna points towards the ground station. Unconventionally, the Hyperflo has the same diameter as the TubeSat to keep the device simple.

Probably the simplest target to fulfil was to validate the height of apogee. This was simply done by incorporating a GPS antenna module. However, this chip needs to have its CoCom limits removed.

The decision has been made not to do any scientific atmospheric measurements as the location of the camera would be the ideal location for measurement instruments. In an actual scientific mission the TubeSat could be redesigned to include a scientific payload. Still, there are two accelerometers, a gyroscope and a magnetic sensor included to map the kinematic behaviour of the DART rocket and the descent of the TubeSat.

For the recovery system a beacon has been implemented in the TubeSat. It will send out a signal every few seconds. Once the initial search area is known this signal can be used to locate the TubeSat. The receiver placed inside the TubeSat has a compatible receiver which can be used by the search team.

The ejection system requires a change in the spring system. The smaller spring system will increase the space for the TubeSat slightly. More importantly it means that the elongated spring does not extend outside of the dart. Extending outside of the dart could cause the TubeSat to tumble uncontrollably. Additionally, the smaller ejection system provides space for a sabot to protect the satellite nose cone during ejection out of the DART rocket.

All the components that together form the TubeSat are held together by the structural system. This system is designed to cope with the $100g$ loads experienced during launch. Also, the structure provides insulation to the high temperatures caused by the re-entry. The structure is mainly made of steel, however, the nose cone is made of ceramics, since during re-entry higher temperatures are expected to occur at the nose. Due to the insulation the electronics will not experience a temperature higher than $85°C$.

Although all the electronic components have been picked, a detailed electronics design has not been performed. Since the design team consists of only aerospace engineers the experience is simply not available. TME will have to do the electronics design themselves.

21 Recommendations

Due to the limited time available there are various topics that have not been designed into detail. These details do however, impact the performance of the mission and need to be further investigated. Recommendations for specific chapters have already been made, so that the TubeSat performs as desired. The biggest recommendations are given below.

As stated in the Chapter 20, the communications system has been the biggest challenge to design. At this point, there is a complete system made out of separate components, but it will take an electrical engineer to make it work. Furthermore, there could be a crucial component missing or the losses in the transmission might not be accurate or mapped properly.

Another large part of the work done on the design is the trajectory simulation. This simulation does not include the effect of wind and especially not the change in wind with altitude. The rotation of the Earth and the location of the launch are not part of the simulation either. These could however, have a large effect and are thus very interesting to include in the simulation.

Just as important for the communication system as for the trajectory simulation are the aerodynamic characteristics of the TubeSat. These include finding out the drag characteristics of the satellite in combination with the hyperflo, but also finding the stability characteristics to make sure that the antenna is pointing towards the ground station. Either wind tunnel testing or CFD research could be executed to find this out.

Two main issues have been dealt with, but are very dependent on the characteristics of the materials used. First, there is the Sorbothane that will absorb vibrations to protect vulnerable components. For this material the exact specifications have to be found. Also a vibration test should be performed on the entire TubeSat. Finally, the thermal behaviour of the satellite has to be tested to make sure that the components can withstand the temperatures.

Within the electronic part of design a lot uncertainties remain and these can only be solved by an electrical engineer. The lack of engineers with knowledge of electronics meant that no detailed electronics design has been made. Only the components that should be able to perform all functions are chosen in such a manner that they should work together. This could mean that the microcontroller that is chosen will work but might be oversized.

Finally, we would like to recommend to TME to use an Agile planning method. We found that Scrum worked for us but it might be better for TME to use Kanban.

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A

Production References

The references to products discussed in chapter 12 are presented in this appendix. The product headings are structured as followed: Part, Company.

Nose Cone

Ceramic Nose, BCE
http://www.bce-special-ceramics.com

Quartz Glass, PGO
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