Design, manufacturing and characterisation of a water fed CubeSat micro-resistojet

Dondersteen

R.M.A. Poyck
Design, manufacturing and characterisation of a water fed CubeSat micro-resistojet

Dondersteen

By

ing. R.M.A. Poyck

In partial fulfilment of the requirements for the degree of

Master of Science
in Aerospace Engineering
Chair of Space System Engineering

At the Delft University of Technology,
to be defended publicly on Thursday 07/08/2014 at 14:00

Daily supervisor: Dr. A. Cervone TU Delft
Thesis committee: Dr. J. Guo TU Delft
Dr. Gregory Pandraud, Dimes / TU Delft

An electronic version of this thesis is available at http://repository.tudelft.nl/.
Abstract

To further the applicability of nano-satellites in large scale relevant missions, it is necessary that they have their own propulsion system for manoeuvring. Multiple nanosatellites flying in formation or even a constellation, with a distributed network of instruments can obtain simultaneous, distributed and guided measurements, which are not possible with conventional single-spacecraft architectures. A cubesat propulsion system technology demonstration has been shown on the Delffi-N3Xt satellite. Demonstrating cold gas generators, which generate self-pressurising nitrogen propellant from solid material. Now comes the time for an operational demonstration of powerful, innovative and effective propulsion for adequate manoeuvrability potential. This operational demonstration will be undertaken by the DelFFi mission, by having two nano-satellites demonstrating formation flying capabilities, such as inter-satellite distance maintenance. The DelFFi satellites will operate in a network of 50 nano-satellites, called the QB50 mission, with the goal to take multi-point in-situ measurements of the lower thermosphere and research re-entry.

The gas generator based cold gas system have however been shown to be unable to produce the necessary total impulse within the stringent mass and volume constraints. The formation flying objective requires a new high performance propulsion system, which is designed during this master thesis. In order to design this propulsion system a number of tests were conducted on the candidate components and assemblies. After a survey of all commonly mentioned reaction engines, a resistojet was found to best meet the needs of this mission. Mainly because resistojets are relatively uncomplicated, do not require combustion and have a relatively low power consumption per amount of impulse generated. The best propellant for resistojets, selected from a large number of possibilities, was found to be water. Water was chosen because it is liquid around room temperature, It has a high potential impulse bit per unit of mass, it is benign, cheap and readily available.

Based on the requirements on the propulsion system the following necessary design parameters were found: a water propellant mass of 50 [g], a nitrogen pressurant mass of 0.2 [g], a storage pressure of 4.5 [bar] to 2.5 [bar], a nozzle throat diameter of 25 [μm], a nozzle area ratio of 20 [-] and a final propellant heating temperature of 500 [°C]. The resulting propulsion system performance is: a total velocity increase ΔV of 21.01 [m/s] (with a satellite mass of 3 [kg]), a total thrusting time of 17 [h] and 56 [min], a power consumption of 6.8 [W] to 3.7 [W] and a thrust force of 1.4 [mN] to 0.8 [mN].

A resistojet propulsion system requires pressure to transport the propellant from the storage tank to and through the thruster. Because pre-pressurisation is generally not allowed in CubeSats, gas generators were found to be the best option. These gas generators can be electrically initialised in orbit, where they produce gas from a solid. Gas generators used for airbag initiation in the automotive industry were found to be unsuitable for this mission because of the high shock pressure of possibly up to 95 [bar]. Additionally, three out of four gas generators that were tested malfunctioned. The reason for this malfunctioning is still unknown. For the DelFFi mission a waiver was granted, allowing pressurisation before launch. Therefore pre-pressurisation is considered to be the best option for propellant pressurisation.
The most innovative part of this thesis is the development of the MEMS micro-thrusters called Dondersteen. These consist of an integrated fluidic inlet channel, heating chamber and rocket nozzle, which are all etched in silicon. The propellant is heated using heating elements made out of the very strong material silicon carbide. These elements are suspended in the middle of the fluid flow in order to maximise the heat transfer to the propellant, and minimise the heat loss to the surroundings. The novelty not only lies in the geometry, but also in the manufacturing process. Hereby the fluidic channels and suspended heating element are both etched out of the silicon carbide covered silicon wafer in the same two etching steps. The developed thruster consists of the following three distinct sections: inlet, heating chamber and nozzle. Multiple designs were made for each of these sections. The production process was chosen such that any combination of these sections can be etched behind one another. This creates the flexibility of being able to produce and test multiple thruster layouts, in order to find the one which best fits the requirements. Different heating chamber geometries were designed and tested. These enable the assessment of the geometrical influence on heat loss, pressure drop and propellant heating capabilities.

Both the manufacturing process and the resulting heating chambers show great potential. The heating elements have been shown to be robust during the rough testing process. The propellant channels have very favourable rounded edges which reduce pressure and heat losses, while maximising the heating contact and propellant mixing. Due to the experimental nature of the manufacturing process the fluidic channel dept was found to be too large and the nozzle throat diameters were too large. The performed analysis of these discrepancies will be used to improve the manufacturing process. The tested resistance values of the resistive heater modules were found to be 200 to 600 times larger than designed. This increases the required input voltage from 5 [V] to the range from 70 [V] to 120 [V]. For the current testing phase this is not a problem since these supplies are available. The flight models will however have to be redesigned in order to comply with the 5 [V] requirement of the satellite power supply. The main reason of this discrepancy was found to be a calculation error. Some adjustments have already been listed with which this resistance can be decreased by a factor of 144. Further design efforts will have to increase this factor to obtain the real design value of the resistance.

There are some tasks that still have to be performed by a succeeding master student to consolidate the work done in this master thesis to a propulsion system flight model. The developed thrusters have to be performance tested with propellant in multiple operating conditions. The results of these tests lead the final redesign of the developed thrusters. The propellant storage system needs to be built and tested. Finally the complete in flight propulsion systems has to be defined, built and integrated into the satellites.
## Contents

Abstract .............................................................................................................. 5  
Contents ............................................................................................................. 7  
Acknowledgements ............................................................................................ 13  
Nomenclature ..................................................................................................... 15  
  Latin .................................................................................................................. 15  
  Greek ................................................................................................................ 16  
Glossary ............................................................................................................. 16  
Acronyms ........................................................................................................... 17  
List of Figures ..................................................................................................... 17  
List of tables ...................................................................................................... 21  
1 Introduction ..................................................................................................... 23  
  1.1 Research motivation .................................................................................... 23  
  1.2 Objectives .................................................................................................. 23  
  1.3 Outline of this thesis report ........................................................................ 24  
2 Propulsion system selection ......................................................................... 25  
  2.1 Propulsion system requirements ................................................................ 25  
  2.2 Propulsion system type options .................................................................. 26  
  2.3 Initial propulsion system selection ............................................................... 27  
  2.4 Resistojet propulsion system layout ............................................................. 31  
  2.5 Propellant selection .................................................................................... 32  
3 Performance calculations ............................................................................ 37  
  3.1 Propellant storage and pressurisation ......................................................... 37  
  3.2 Feed system ................................................................................................ 39  
  3.3 Heating chamber ......................................................................................... 43  
  3.4 Rocket nozzle ............................................................................................... 45  
  3.5 Calculation of design parameter to meet the requirements ......................... 51  
4 Propellant storage system ............................................................................ 61  
  4.1 Propellant storage geometry options .......................................................... 61  
  4.2 Propellant storage ....................................................................................... 65  
  4.3 Requirements on the pressurisation system ................................................. 69  
  4.4 Pressurant selection .................................................................................... 69  
5 Automotive gas generator tests .................................................................. 73
<table>
<thead>
<tr>
<th>Section</th>
<th>Title</th>
<th>Page</th>
</tr>
</thead>
<tbody>
<tr>
<td>5.1</td>
<td>Introduction of the tested gas generators</td>
<td>73</td>
</tr>
<tr>
<td>5.2</td>
<td>Requirement verification methodology</td>
<td>75</td>
</tr>
<tr>
<td>5.3</td>
<td>Test setup</td>
<td>76</td>
</tr>
<tr>
<td>5.4</td>
<td>Test procedure</td>
<td>79</td>
</tr>
<tr>
<td>5.5</td>
<td>Organisational and safety aspects</td>
<td>82</td>
</tr>
<tr>
<td>5.6</td>
<td>Hot gas generator test results</td>
<td>84</td>
</tr>
<tr>
<td>5.7</td>
<td>Recalculation of performance estimations</td>
<td>94</td>
</tr>
<tr>
<td>5.8</td>
<td>Conclusions on the gas generator tests</td>
<td>96</td>
</tr>
<tr>
<td>5.9</td>
<td>Recommendations for future testing efforts</td>
<td>97</td>
</tr>
<tr>
<td>6</td>
<td>Feed system</td>
<td>99</td>
</tr>
<tr>
<td>6.1</td>
<td>Propellant storage tank interface</td>
<td>99</td>
</tr>
<tr>
<td>6.2</td>
<td>Piping</td>
<td>99</td>
</tr>
<tr>
<td>6.3</td>
<td>Propellant control valve</td>
<td>100</td>
</tr>
<tr>
<td>6.4</td>
<td>Fluidic interconnect to the thruster</td>
<td>101</td>
</tr>
<tr>
<td>7</td>
<td>Thruster</td>
<td>103</td>
</tr>
<tr>
<td>7.1</td>
<td>Previous TU Delft resistojets</td>
<td>103</td>
</tr>
<tr>
<td>7.2</td>
<td>Thruster design methodology</td>
<td>107</td>
</tr>
<tr>
<td>7.3</td>
<td>Propellant inlet section</td>
<td>110</td>
</tr>
<tr>
<td>7.4</td>
<td>Propellant heating section</td>
<td>111</td>
</tr>
<tr>
<td>7.5</td>
<td>Nozzle section</td>
<td>132</td>
</tr>
<tr>
<td>7.6</td>
<td>Heat transfer</td>
<td>133</td>
</tr>
<tr>
<td>7.7</td>
<td>Manufacturing</td>
<td>134</td>
</tr>
<tr>
<td>7.8</td>
<td>Printed Circuit Board (PCB) mounting</td>
<td>137</td>
</tr>
<tr>
<td>8</td>
<td>Thruster testing</td>
<td>141</td>
</tr>
<tr>
<td>8.1</td>
<td>Necessary parameters</td>
<td>141</td>
</tr>
<tr>
<td>8.2</td>
<td>Testing methodology</td>
<td>142</td>
</tr>
<tr>
<td>8.3</td>
<td>Test equipment</td>
<td>143</td>
</tr>
<tr>
<td>8.4</td>
<td>Test execution and results</td>
<td>144</td>
</tr>
<tr>
<td>9</td>
<td>Future work to finalise the DelFFi propulsion system</td>
<td>153</td>
</tr>
<tr>
<td>9.1</td>
<td>Finalising the thrusters</td>
<td>153</td>
</tr>
<tr>
<td>9.2</td>
<td>Testing equipment</td>
<td>153</td>
</tr>
<tr>
<td>9.3</td>
<td>Thruster performance tests</td>
<td>154</td>
</tr>
<tr>
<td>9.4</td>
<td>Thruster design selection and alteration</td>
<td>154</td>
</tr>
<tr>
<td>9.5</td>
<td>Propellant storage system investigations</td>
<td>155</td>
</tr>
</tbody>
</table>
9.6 Propulsion system flight model design manufacturing and integration........................................................................155
10 Conclusions.....................................................................................................................................................................157
11 Recommendations........................................................................................................................................................159
  11.1 Recommendations for future research..................................................................................................................159
  11.2 Recommendations for future students..................................................................................................................159
  11.3 Recommendations for thesis supervision..............................................................................................................160
12 Bibliography......................................................................................................................................................................161
Appendix A - Propulsion system concept trade-off .........................................................................................................167
Appendix B - Fluidic properties .........................................................................................................................................168
Appendix C - Nozzle optimisation relations ....................................................................................................................169
Appendix D - Automotive gas generator test supplementary data ..................................................................................170
Appendix E - Manufacturing information ........................................................................................................................174
Appendix F - Component trade-off listings .......................................................................................................................176
Appendix G - Microscope images for geometry validation ............................................................................................179
Appendix H - Attempts at calculating the heat transfer of the nozzles ...........................................................................179
“Gaat het niet zoals het moet, dan moet het maar zoals het gaat”
(If something doesn't go as it should, it shall be as it goes)

- Author unknown -

“Wat men moet leren doen, leert men door het te doen”
(Anything one needs to learn to do, is learned by doing it)

- Aristotle -
Acknowledgements

My first thanks goes to my dear Louise, who has supported me throughout the process. Even though at too many occasions I had to favour university work, due to constantly imminent deadlines.

Furthermore I owe my gratitude to a number of people who have aided me towards the completion of this thesis. Most notably my supervisor Dr. Angelo Cervone, giving valuable advice when necessary, while granting me the freedom to act on my own accord. My second supervisor ir. Barry Zandbergen, with his relentless interest and enthusiasm, giving the right tips to help me form my own judgement. Delft propulsion system colleagues Quirino Bellini and Ivan Krusharev for their humour and their views. Dr. Henk van Zeijl for his seemingly endless stream of innovative MEMS manufacturing ideas and his help in generating design possibilities. Admirable was the last day of production, where he worked from sunrise to sunset on the next day in an effort to finish the thruster manufacturing. The men at the workshop of DEMO, always ready for discussions on production possibilities, and a help in the manufacturing of the test equipment. Jos van Driel with his swift help with sensor equipment and necessary software. Nuno, who we all know as the ruler of the cleanroom and test equipment, always ready to help out students with their novice electrical questions. Erwin van Dooren from airbagbank, who was kind enough to discuss and demonstrate the possibilities of using airbag initiators in satellites.

And a final special thanks goes to all my colleagues in the master graduation rooms. Amidst occasional chaos and disappointment they have always been able to lighten the spirit and put things into perspective.
## Nomenclature

<table>
<thead>
<tr>
<th>Latin Symbol</th>
<th>Unit</th>
<th>Description</th>
</tr>
</thead>
<tbody>
<tr>
<td>( a )</td>
<td>( [m/s^2] )</td>
<td>Acceleration</td>
</tr>
<tr>
<td>( A )</td>
<td>( [m^2] )</td>
<td>Cross-sectional area</td>
</tr>
<tr>
<td>( c )</td>
<td>( [m] )</td>
<td>Chord length of an aerofoil shape</td>
</tr>
<tr>
<td>( c^* )</td>
<td>( [m/s] )</td>
<td>Characteristic velocity</td>
</tr>
<tr>
<td>( C_p )</td>
<td>( [J/kg/K] )</td>
<td>Heat capacity of the fluid at constant pressure</td>
</tr>
<tr>
<td>( C_v )</td>
<td>( [J/kg/K] )</td>
<td>Heat capacity of the fluid at constant volume</td>
</tr>
<tr>
<td>( C_v )</td>
<td>[-]</td>
<td>Flow coefficient</td>
</tr>
<tr>
<td>( D )</td>
<td>( [m] )</td>
<td>Diameter</td>
</tr>
<tr>
<td>( f )</td>
<td>[-]</td>
<td>Fanning friction factor</td>
</tr>
<tr>
<td>( F )</td>
<td>( [N] )</td>
<td>(Thrust) force</td>
</tr>
<tr>
<td>( g_0 )</td>
<td>( [m/s^2] )</td>
<td>Gravitational acceleration at standard sea level, 9.81 ( [m/s^2] )</td>
</tr>
<tr>
<td>( G )</td>
<td>[-]</td>
<td>Specific gravity</td>
</tr>
<tr>
<td>( h )</td>
<td>( [m] )</td>
<td>Height</td>
</tr>
<tr>
<td>( H^* )</td>
<td>( [J/mol] )</td>
<td>Enthalpy</td>
</tr>
<tr>
<td>( I )</td>
<td>( [A] )</td>
<td>Electrical current</td>
</tr>
<tr>
<td>( I_{sp} )</td>
<td>( [s] )</td>
<td>Specific impulse</td>
</tr>
<tr>
<td>( Kn )</td>
<td>[-]</td>
<td>Knudsen number</td>
</tr>
<tr>
<td>( L )</td>
<td>( [m] )</td>
<td>Length</td>
</tr>
<tr>
<td>( m )</td>
<td>( [kg] )</td>
<td>Mass</td>
</tr>
<tr>
<td>( \dot{m} )</td>
<td>( [kg/s] )</td>
<td>Mass flow</td>
</tr>
<tr>
<td>( M )</td>
<td>[-]</td>
<td>Mach number of the flow</td>
</tr>
<tr>
<td>( \dot{M} )</td>
<td>( [kg/mol] )</td>
<td>Molar mass</td>
</tr>
<tr>
<td>( n )</td>
<td>( [mol] )</td>
<td>Amount of pressurant</td>
</tr>
<tr>
<td>( p )</td>
<td>( [Pa] ) (( \text{bar} ) where indicated)</td>
<td>Pressure</td>
</tr>
<tr>
<td>( p )</td>
<td>( [m] )</td>
<td>Periphery or inner wall length of a cross-section</td>
</tr>
<tr>
<td>( P )</td>
<td>( [W] )</td>
<td>Power</td>
</tr>
<tr>
<td>( r )</td>
<td>( [m] )</td>
<td>Radius</td>
</tr>
<tr>
<td>( R )</td>
<td>( [J/kg/K] )</td>
<td>Specific gas constant</td>
</tr>
<tr>
<td>( R )</td>
<td>( [\Omega] )</td>
<td>Electrical resistance</td>
</tr>
<tr>
<td>( R_A )</td>
<td>( [J/K/mol] )</td>
<td>Universal gas constant</td>
</tr>
<tr>
<td>( Re )</td>
<td>[-]</td>
<td>Reynolds number</td>
</tr>
<tr>
<td>( t )</td>
<td>( [K]/1000 )</td>
<td>Temperature</td>
</tr>
<tr>
<td>( t )</td>
<td>( [m] )</td>
<td>Thickness</td>
</tr>
<tr>
<td>( T )</td>
<td>( [K] )</td>
<td>Temperature</td>
</tr>
<tr>
<td>( U )</td>
<td>( [m/s] )</td>
<td>Velocity</td>
</tr>
<tr>
<td>( V )</td>
<td>( [m^3] )</td>
<td>Volume</td>
</tr>
</tbody>
</table>
Design, manufacturing and characterisation of a water fed CubeSat micro-resistojet

<table>
<thead>
<tr>
<th>Symbol</th>
<th>Unit</th>
<th>Description</th>
</tr>
</thead>
<tbody>
<tr>
<td>w</td>
<td>[m]</td>
<td>Width</td>
</tr>
<tr>
<td>x</td>
<td>[m]</td>
<td>Position/distance compared to a reference point</td>
</tr>
<tr>
<td>y</td>
<td>[m]</td>
<td>Length</td>
</tr>
</tbody>
</table>

**Greek Symbols**

<table>
<thead>
<tr>
<th>Symbol</th>
<th>Unit</th>
<th>Description</th>
</tr>
</thead>
<tbody>
<tr>
<td>α</td>
<td>[K]</td>
<td>The Temperature Coefficient of Resistance (TCR)</td>
</tr>
<tr>
<td>α</td>
<td>[-]</td>
<td>Aspect ratio</td>
</tr>
<tr>
<td>Δ</td>
<td>[-]</td>
<td>Aspect ratio</td>
</tr>
<tr>
<td>γ</td>
<td>[-]</td>
<td>Increment or change</td>
</tr>
<tr>
<td>Γ</td>
<td>[-]</td>
<td>Ratio of specific heats</td>
</tr>
<tr>
<td>ζ</td>
<td>[-]</td>
<td>Pressure loss coefficient</td>
</tr>
<tr>
<td>η</td>
<td>[-]</td>
<td>Efficiency</td>
</tr>
<tr>
<td>λ</td>
<td>[m]</td>
<td>Molecular mean free path between collisions</td>
</tr>
<tr>
<td>μ</td>
<td>[kg/s/m]</td>
<td>Dynamic (absolute) viscosity</td>
</tr>
<tr>
<td>ν</td>
<td>[m²/s]</td>
<td>Kinematic viscosity</td>
</tr>
<tr>
<td>ρ</td>
<td>[kg/m³]</td>
<td>Density</td>
</tr>
<tr>
<td>ρ</td>
<td>[Ω·m]</td>
<td>Resistivity</td>
</tr>
<tr>
<td>σ</td>
<td>[Pa]</td>
<td>Stress</td>
</tr>
<tr>
<td>φ</td>
<td>[-]</td>
<td>Volume fraction</td>
</tr>
</tbody>
</table>

**Glossary**

**Aspect ratio**
Ratio of the length of a suspended heating element over the width of this element. The higher the aspect ratio, the thinner the suspended heater is.

**Bond wires**
Ultra fine, highly conductive and nonreactive electrical wires. Made to make electrical connections to very small electrical components.

**Choked flow**
A compressible flow effect, in a narrower section of a flow channel, whereby the mass-flow is at a maximum for a given downstream pressure.

**CubeSat**
Nano-satellite built up out of 10 [cm] cubes, called 1U, each having a maximum mass 1.33 [kg].

**Delta-V or ΔV**
The total velocity increase that a propulsion system can deliver to the spacecraft in which it is integrated during its entire lifetime. Could also be considered to be the total impulse the propulsion system can deliver, while taking the satellite mass into consideration.

**Dondersteen**
The name of the MEMS thruster developed for this master thesis and described in this document. This Dutch word means “Thunderstone” in English. Chosen because the thruster is made mainly out of silicon, the main component in stones. And because it is powered by electricity.

**Enthalpy**
The thermodynamic state or potential of a system, which depends on its internal energy, pressure and volume.

**Flight model**
The complete hardware package of a satellite subsystem that will be used in the actual satellite.

**Knudsen number**
The ratio between molecular mean free path length and the length scale of the problem. In this report used to assess if gas continuum hypotheses are valid.

**Micro-satellite**
Satellite with a mass in the range of 10 [kg] to 100 [kg].
<table>
<thead>
<tr>
<th>Term</th>
<th>Definition</th>
</tr>
</thead>
<tbody>
<tr>
<td>Module</td>
<td>A section of the thruster, either an inlet, a heating section or a nozzle section. Each section or module can have different geometries and dimensions. Multiple of these modules are placed behind each other to form a complete thruster.</td>
</tr>
<tr>
<td>Nano-satellite</td>
<td>Satellite with a mass in the range of 1 [kg] to 10 [kg].</td>
</tr>
<tr>
<td>Plenum</td>
<td>A settling chamber where gas can be stored at a high pressure before being expelled through a rocket nozzle.</td>
</tr>
<tr>
<td>Pressurant</td>
<td>The gas used for the pressurisation. Sometimes used generically to indicate a part that pressurises a fluid, e.g. a spring.</td>
</tr>
<tr>
<td>Rarefaction</td>
<td>A density below the value where idealised gas theory can still be assumed to be valid, due to the limitation of intermolecular effects.</td>
</tr>
<tr>
<td>Resistojet</td>
<td>A thruster which produces thrust by electrically heating the propellant, by means of a resistive element, and accelerating it through a converging-diverging nozzle.</td>
</tr>
<tr>
<td>Specific impulse</td>
<td>A measure of the propellant usage efficiency of a rocket engine. It represents the ratio between the force and the propellant mass flow rate. The higher the specific impulse, the less propellant is needed for a certain total impulse of the propulsion system.</td>
</tr>
<tr>
<td>T³μPS</td>
<td>The cold gas propulsion system used on the Delfi-n3Xt mission, developed in a cooperation between TU Delft and research institute TNO. This propulsion system used novel gas generators, in order to pressurise the system only after the insertion into orbit.</td>
</tr>
<tr>
<td>Throat</td>
<td>Narrowest section of the rocket nozzle where the flow becomes sonic, when choked conditions are attained.</td>
</tr>
<tr>
<td>Viscosity</td>
<td>A measure of the resistance to deformation of a fluid.</td>
</tr>
</tbody>
</table>

**Acronyms**

<table>
<thead>
<tr>
<th>Acronym</th>
<th>Description</th>
</tr>
</thead>
<tbody>
<tr>
<td>DAQ</td>
<td>Data AcQuisiton device (by National Instruments).</td>
</tr>
<tr>
<td>DASM</td>
<td>Delft Aerospace Structures and Materials Laboratory.</td>
</tr>
<tr>
<td>DEMO</td>
<td>Dienst Elektronische en Mechanische Ontwikkeling. Which is Dutch for the electronic and mechanical support division at the TU Delft.</td>
</tr>
<tr>
<td>Dimes</td>
<td>Delft Institute for Microsystems and Nanoelectronics, the manufacturer of the Dondersteen MEMS thruster described in this thesis.</td>
</tr>
<tr>
<td>DUR-1</td>
<td>Delft University Resistojet - 1.</td>
</tr>
<tr>
<td>MEMS</td>
<td>Micro Electro-Mechanical System.</td>
</tr>
<tr>
<td>NI</td>
<td>National Instruments.</td>
</tr>
<tr>
<td>NovAM</td>
<td>Novel Aerospace Materials, a chair at the faculty of aerospace engineering at the TU Delft.</td>
</tr>
<tr>
<td>PCB</td>
<td>Printed Circuit Board.</td>
</tr>
<tr>
<td>SEM</td>
<td>Scanning Electron Microscope.</td>
</tr>
<tr>
<td>SiC</td>
<td>Silicon Carbide.</td>
</tr>
<tr>
<td>SSE</td>
<td>Space Systems Engineering, the chair under which this master thesis was executed at the TU Delft.</td>
</tr>
<tr>
<td>TU Delft</td>
<td>Delft University of Technology (Dutch: Technische Universiteit Delft).</td>
</tr>
</tbody>
</table>

**List of Figures**

**Figure 2.1** - Schematic representation of a typical monopropellant rocket engine [12] ........................................... 27
**Figure 2.2** – A typical cold gas thruster [13] ................................................................................................................. 28
Design, manufacturing and characterisation of a water fed CubeSat micro-resistojet - 28-07-2014

**Figure 6.1 - Longitudinal cross-section of the Lee Company IEP series solenoid valve [56]**, which will be used in the Delft propulsion system flight model. ................................................................. 100

**Figure 6.2 - The Lee Company EPSV fluid flow control valve [57]** ................................................................. 101

**Figure 6.3 - Nitrogen fed MEMS resistojet by Tittu Varghese Mathew, showing the needle bonded to the thruster [31]** ................................................................. 101

**Figure 6.4 - Both sides of a Luer-Lock connection [58]** ................................................................. 102

**Figure 6.5 - Luer-Lock connector, connected to the needle [31]** ................................................................. 102

**Figure 7.1 - The central components of the DUR-1 [59]** ................................................................. 104

**Figure 7.2 - Heater module vacuum tube and radiation shielding of the DUR-1.1 [60]** ................................................................. 104

**Figure 7.3 - Heating section of the DUR-1.2 [60]** ................................................................. 105

**Figure 7.4 - Final design of the water fed TU Delft resistojet DUR-H₂O [61]** ................................................................. 105

**Figure 7.5 - The heating element and alumina blanket of the DUR-H₂O after testing, some discoloration due to the burning of the thermocouple outside insulation is indicated [61]** ................................................................. 105

**Figure 7.6 - Outside view of the nitrogen fed MEMS micro-resistojet [31]** ................................................................. 107

**Figure 7.7 - Cross-section of the flow channel of one of the nitrogen fed micro-resistojet geometries [31]** ................................................................. 107

**Figure 7.8 - Schematic depictions of the top view of all sections of the Dondersteen thruster sections, the intended propellant flow direction is from left to right.** ................................................................. 107

**Figure 7.9 - Example thruster inlet module.** ................................................................. 108

**Figure 7.10 - Example heating chamber module.** ................................................................. 108

**Figure 7.11 - Example nozzle module.** ................................................................. 108

**Figure 7.12 - Example of multiple thruster modules forming a complete thruster, a combination of one inlet module, six heating chamber modules and one nozzle module. The white boundaries in this image are only to indicate the different modules which were used. In reality the modules are etched close enough together, that no boundaries are formed.** ................................................................. 108

**Figure 7.13 - A titanium nitride (TiN) heating element suspended in a flow channel on top of a low stress, thin silicon nitride membrane.** ................................................................. 109

**Figure 7.14 - A schematic representation of the intended layers for the Dondersteen resistojet. From conductive heating element at the centre of the thruster (shown on the bottom), to the resin encasing on the outside (shown on top). This stratification is thus mirrored in the layer of the heating element.** ................................................................. 110

**Figure 7.15 - Top view of the flow velocity magnitude through the empty inlet of a diamond pillar thruster. Flow direction from left to right. Red indicates high velocity, blue indicates a low velocity.** ................................................................. 111

**Figure 7.16 - Top view of the flow velocity streamlines through the empty inlet area of a thruster with circular shaped fins. Flow direction from left to right. Each line represents the flow path of a particle starting at the initial position on the left.** ................................................................. 111

**Figure 7.17 - Top view of the final inlet design.** ................................................................. 111

**Figure 7.18 - Top view of a propellant heating chamber design with straight flow channels. Flow direction from bottom to top.** ................................................................. 113

**Figure 7.19 - Top view of a propellant heating chamber design with semi-circular serpentine flow channels. Flow direction from bottom to top.** ................................................................. 113

**Figure 7.20 - Top view of a propellant heating chamber design with triangular serpentine flow channels. Flow direction from bottom to top.** ................................................................. 113

**Figure 7.21 - Top view of a propellant heating chamber design with sinusoidal serpentine flow channels. Flow direction from bottom to top.** ................................................................. 113

**Figure 7.22 - Top view of a propellant heating chamber design with alternating rectangular pillars in a single flow channel. Flow direction from bottom to top.** ................................................................. 113

**Figure 7.23 - Top view of a propellant heating chamber design with alternating circular pillars in a single flow channel. Flow direction from bottom to top.** ................................................................. 113

**Figure 7.24 - Top view of a propellant heating chamber design with alternating diamond or rhombus pillars in a single flow channel. Flow direction from bottom to top.** ................................................................. 113
FIGURE 7.25 - Top view of a propellant heating chamber design with alternating aerofoil pillars in a single flow channel. Flow direction from bottom to top ................................................................. 113
FIGURE 7.26 - Two-phase flow types [62] ........................................................................................................ 114
FIGURE 7.27 - Flow velocity magnitude through a semi-circular serpentine channel. Isometric view. Red indicates high velocity, blue indicates low velocity. Flow direction from bottom to top ................................................................. 115
FIGURE 7.28 - Flow velocity magnitude through a triangular serpentine channel. Top view. Red indicates high velocity, blue indicates low velocity. Flow direction from bottom to top ................................................................. 115
FIGURE 7.29 - Propellant flow patterns, though a heating section with non-alternating rectangular fins. On top, the velocity magnitude where black is low velocity flow and white is high velocity flow. On bottom velocity streamlines along the flow. Flow direction from left to right. Top side view ........................................................................ 116
FIGURE 7.30 - Propellant flow patterns, though a heating section with alternating rectangular fins. On top, the velocity magnitude where black is low velocity flow and white is high velocity flow. On bottom velocity streamlines along the flow. Flow direction from left to right. Top side view ........................................................................ 116
FIGURE 7.31 - Propellant flow patterns, though a heating section with alternating circular fins. On top, the velocity magnitude where blue is low velocity flow and white is high velocity flow. On bottom velocity streamlines along the flow. Flow direction from left to right. Top side view ........................................................................ 117
FIGURE 7.32 - Propellant flow patterns, though a heating section with alternating diamond (rhombus) shaped fins. On top, the velocity magnitude where black is low velocity flow and white is high velocity flow. On bottom velocity streamlines along the flow. Flow direction from left to right. Top side view ........................................................................ 117
FIGURE 7.33 - Longitudinal cross-section of one heating module with diamond shaped pillars. Including the suspended heating elements and the bond surfaces on the side. The indicated gaps show the physical separation between different heating elements ............................................. 119
FIGURE 7.34 - MEMS heater with diamond shaped fins [63]. x is the pillar length, m is the minimum distance between pillars, h is the maximum distance between pillars, l is the width of the pillars ................................................................. 120
FIGURE 7.35 - Simplified heater element for half a pillar ................................................................................... 121
FIGURE 7.36 - Performance of a diamond fin micro-evaporator [63] .................................................................. 124
FIGURE 7.37 - Modular section power consumption - Rows connected in parallel ........................................ 125
FIGURE 7.38 - Modular section power consumption - Rows connected in series ........................................ 125
FIGURE 7.39 - Heating channel etching indication with diamond shaped fins - Nominal design - 1 [W] power consumption ................................................................. 126
FIGURE 7.40 - Heating channel heating layer with diamond shaped fins - Nominal design - 1 [W] power consumption ................................................................................................. 126
FIGURE 7.41 - Semi-circular heating channel design example ............................................................................. 127
FIGURE 7.42 - A section with 1 wave across the entire width of the heating chamber ........................................ 128
FIGURE 7.43 - Channel volume fractions for a range of semi-circle outer radii, assuming 160 [mm] total wave length ................................................................................................. 129
FIGURE 7.44 - Top side view of the designed heating elements for the semi-circular serpentine heating chamber geometry ................................................................................................. 129
FIGURE 7.45 - Optical microscope image of the top view of one module of the nominal semi-circular serpentine channel design. Including the electrical connection pads to the sides and the suspended heating elements. Flow direction is from bottom to top ......................................................................................... 130
FIGURE 7.46 - Optical microscope image of the top view of one module of the large semi-circular serpentine channel design. Including the electrical connection pads to the sides and the suspended heating elements. Flow direction is from bottom to top ......................................................................................... 130
FIGURE 7.47 - Divergent nozzle sections showing a comparison in length for conical and bell shaped nozzles, with the same throat diameter and exit angle ........................................................................... 133
FIGURE 7.48 - Shape of the flow channel of the final design of the optimised bell nozzle design ............................ 133
FIGURE 7.49 - Shape of the flow channel of the final design of the sharp edge conical nozzle design ..................... 133
FIGURE 7.50 - Pattern which was used for the nominal diamond shape design. In black is the section that is etched, in white is the section that is covered by photoresist ......................................................................................... 135
Design, manufacturing and characterisation of a water fed CubeSat micro-resistojet - 28-07-2014

**Figure 7.51 - SEM image to clearly illustrate the under-etching process below the heating elements, in order to suspend them in the flow channel. This is an image of the nominal semi-circular flow channel design.**

**Figure 7.52 - Test wafer filled with the Dondersteen micro-thruster designs, used for geometrical and electrical characterisation.**

**Figure 7.53 - Schematic top view of the PCB design.**

**Figure 7.54 - Three dimensional representation of the thruster mounted on the PCB.**

**Figure 8.1 - SEM image of the nominal thruster design throat area - Straight walls, chamfered corners and ridge forming can be observed in the flow channel.**

**Figure 8.2 - SEM image of the nominal diamond shaped heater element design - Under-etching of the diamond elements, chamfered corners and ridge forming can be observed in the flow channel.**

**Figure 8.3 - SEM image of the inlet - Straight walls and chamfered corners can be observed in the flow channel.**

**Figure 8.4 - SEM image of the nominal serpentine shaped heater chamber design - Under-etching of the heating elements, chamfered corners and ridge forming can be observed in the flow channel.**

**Figure 8.5 - Bell nozzle throat depth measurement - Top surface.**

**Figure 8.6 - Bell nozzle throat depth measurement - Flow channel ridge.**

**Figure 8.7 - Bell nozzle throat depth measurement - Flow channel bottom.**

**Figure 12.1 - Development of the thermal boundary layer to thermally fully developed flow through a tube.**

---

**List of tables**

**Table 2.1 - Relevant properties of common substances which are liquid around room temperature, ordered by molar mass. Gathered in order to assess potential propellants. Most properties collected on the NIST database [23] and using [24].**

**Table 3.1 - Critical Reynolds numbers in flow channels with different geometries. The aspect ratio α is the ratio between channel height and channel width of the flow channel α = \( \frac{H}{w} \).**

**Table 3.2 - The assumptions made in the propulsion system performance calculations.**

**Table 3.3 - Final selection of the design parameters for the propulsion system.**

**Table 5.1 - Intended requirement compliance information gathering.**

**Table 5.2 - Actual requirement compliance information gathering.**

**Table 7.1 - Characteristic dimensions and performance of a water fed micro-evaporator [63], the meanings of the abbreviations are given in Figure 7.34.**

**Table 7.2 - Most important design and performance parameters of the heater modules.**

**Table 8.1 - Comparison between the design values of the thruster geometry and the actually produced values as checked with the optical microscopes.**

**Table 8.2 - Measurement results of the channel depth in specific areas, multiple versions were measured where multiple dimensions are listed. Channel depth and ridge height from the same channel are listed in the same column.**

**Table 8.3 - The summary of the cold electrical resistance tests on the different SiC heating elements of the Dondersteen thruster. All resistances have been measured on at least 4 different modules, and most of the values are an average of multiple measurements on the same module.**

**Table 12.1 - Critical Reynolds number for rectangular channels, depending on aspect ratio [31] [32].**
1 Introduction

This chapter will first describe the need for the conducted research in section 1.1. This will be followed by the statement of the goal of this research in section 1.2. Finally the global outline of this thesis documentation will be given in section 1.3.

1.1 Research motivation

Previous decades have shown ever increasing use of the standardised nano-satellite (1 [kg] to 10 [kg]) concept, called CubeSats. These CubeSats consist of building blocks of 10 [cm] cubes, called 1U, with a maximum mass of 1.33 [kg]. One CubeSat can comprise one or more of these building blocks. These fixed dimensions are meant to bring some standardisation to small satellite development and the commercially available satellite components. Therefore Cubesats set off as a relatively cheap and accessible platform for hands-on experience in space hardware at learning institutes. Ever increasingly, companies have also seen the benefits of the cubesats, especially for technology testing and demonstration [1]. To further the applicability of nano-satellites in large scale relevant missions, it is necessary that they have their own propulsion system for manoeuvring. Multiple nanosatellites flying in formation or even a constellation, with a distributed network of instruments can obtain simultaneous, distributed and guided measurements, which are not possible with conventional single-spacecraft architectures [2]. For example, dynamic phenomena in the Earth’s magnetosphere can be observed. A cubesat propulsion system technology demonstration has been shown on the Delfi-N3Xt satellite [3]. Demonstrating cold gas generators, which generate self-pressureising nitrogen propellant from solid material. Cold gas thrusters are usually only used for attitude control manoeuvres, where simplicity is more important than high performance [4]. Now comes the time for an operational demonstration, with powerful, innovative and effective propulsion for adequate manoeuvrability potential.

1.2 Objectives

This operational demonstration will be undertaken by the DelFFi mission, by having two nanosatellites demonstrating formation flying capabilities, such as inter-satellite distance maintenance [5]. The formation flying objective requires a new high performance propulsion system. This system will be partially based on experiences with the TμPS flown on the Delfi-N3Xt [6]. The DelFFi satellites will operate in the QB50 framework. This is a network of 50 nano-satellites, with the goal to take multi-point in-situ measurements of the lower thermosphere and research re-entry [7].

The aim of this master thesis is:

Design a propulsion system to meet the propulsive requirements of the two DelFFi mission formation flying satellites.
The objectives of the thesis form a subdivision of the research framework. These sub goals of the thesis have evolved with the change in the method of execution of the main objective. These objectives are the design, manufacturing and testing of:

- A propellant storage system
- An engineering model of a resistojet thruster
- A pressurisation system

A rivalling thruster is being developed by Mr. I. Krusharev, using different manufacturing techniques and design philosophies. The thruster designed by the author is produced using Micro Electro-Mechanical System (MEMS) manufacturing, in cooperation with research institute Dimes. Mr. Krusharev focuses on commercially available products and traditional mechanical manufacturing technique. Mr. Krusharev therefore collaborates mainly with the mechanical production workshops of the university. Both of these thrusters are required to be able to operate on the same propellant storage and feed system. This enabled the author and Mr. Krusharev to cooperate in the development of this propellant system. And it makes it possible to eventually chose the best performing thruster design, to be integrated in the final flight model. It was decided that the responsibility of the propellant storage and pressurisation lies with the author. Mr. Krusharev is on the other hand responsible for the propellant feed valve and the necessary drive electronics.

1.3 Outline of this thesis report
This thesis documentation will commence to state the overall system requirements and specifications, which are relevant to the propulsion system, in chapter 2. After which a selection of which type of propulsion system best befits this mission will follow. This chapter will also describe the general layout and components of the selected propulsion system. All the necessary design parameters of the propulsion system will then be calculated and the components will be selected and described. This will commence with the calculation of the necessary performance of each propulsion system section in chapter 3. This will be done by calculating the performance needed from each section, in order to fulfil the overall propulsion system requirements.

The subsequent chapters will describe the individual sections of the propulsion system. The necessary calculations are documented, design options are outlined and the final selection is described and explained. This will start with the description of the propellant storage system in chapter 4. Which is followed by a slight excursion in chapter 5. Here the extensive tests are documented, which were performed in order to test one of the pressurisation system options, described in chapter 4. Chapter 6 will briefly describe the necessary propellant feed system. The most important and intricate part of the propulsion system is the resistojet thruster itself. All the integrated sections in this thruster will be elaborated upon in chapter 7. In chapter 8 the documentation on the necessary test campaign of the newly developed thruster follows. This chapter will also describes the analysis of the results of the tests conducted thus far. Finally chapter 9 will fulfil the vital task of describing the future tasks that still need to be completed in order to go from the current prototype to the final flight model of the propulsion system.
2 Propulsion system selection

The requirements on the propulsion system performance, geometry and interfaces were already set before the start of this thesis. These requirements are described in section 2.1 below. The first step in the propulsion system design is to select the type of propulsion system. This selection will greatly influence the necessary performance calculations. In this chapter, the propulsion system selection will be explained in section 2.2. After which the lay-out and components of the selected propulsion system are described in section 2.4. Finally the selection of the optimal propellant for this propulsion system is shown in section 2.5.

2.1 Propulsion system requirements

The requirements on the propulsion system were derived from the requirements on the formation flying manoeuvres. These requirements were then documented by the propulsion system responsible Dr. Cervone and ir. Zandbergen, and approved by the project leader Dr. Guo [8]. The most critical requirements, which will have the biggest impact on the thruster design are:

- **PROP-PERF-100**: The total Delta-V provided by the propulsion system shall be at least 15 [m/s].
- **PROP-PERF-200**: The thrust provided by the propulsion system shall be above 0.5 [mN].
- **PROP-PERF-205**: The thrust provided by the propulsion system shall be below 9.5 [mN].
- **PROP-PERF-400**: The propulsion system shall have a lifetime of at least 1 year under operational conditions in space.
- **PROP-PERF-410**: The propulsion system shall be able to withstand at least 5000 on-off cycles without losing its capability to meet any other performance or system requirement.
- **PROP-SYST-100**: The total wet mass of the propulsion system at launch shall be not higher than 459 [g].
- **PROP-SYST-200**: The total size of the propulsion system shall be within 90 [mm] x 90 [mm] x 80 [mm].
- **PROP-SYST-310**: The peak power consumption of the propulsion system during ignition or heating shall not be higher than 10 [W].
- **PROP-SYST-320**: The total energy consumption of the propulsion system shall be no more than 100 [kJ] per day (i.e. 1.1574 [W] daily average).
- **PROP-SYST-410**: The nominal geometrical axis of the nozzle shall be perpendicular to the plane of one of the smaller faces (100 [mm] x 100 [mm]) of the satellite.
- **PROP-SYST-500 to PROP-SYST-560**: The propulsion system shall be able to withstand the launch loads.
- **PROP-SYST-600**: The internal pressure of all propulsion system components shall not be higher than 10 [bar].
- **PROP-SYST-610**: The propulsion system shall not include any pyrotechnic devices.
- **PROP-SYST-620**: The propellant(s) used by the propulsion system shall not be hazardous for the operators or the other satellite sub-systems.
• **PROP-SYST-710**: The thermal interface between the propulsion system and the satellite shall maintain a temperature range between -20 [°C] and 80 [°C] for the propulsion system components during all the mission phases when propulsion system operations are required.

• **PROP-SYST-720**: The propulsion system shall be electrically connected to the satellite power sub-system through the standard cubesat I²C interface.

In addition to this, there are also two design guidelines which are not in the mentioned requirements document.

- The resistojet housing shall fit within the centre of the antenna board, which is in between the propulsion system and the exterior bottom panel of the satellite. The cut-out in this antenna board, and subsequently the bottom panel, has maximum dimensions of 2 [cm] by 2 [cm].
- The propulsion system shall be able to operate with a power supply voltage of 5 [V].

Next to these propulsion system specific requirements, the system shall also abide by the standard CubeSat requirements. These requirements are stated in the CubeSat design specification document [9]. Any one of these specifications is only to be disregarded when an official waiver has been granted. The procedure of requesting such a waiver is described in the same document.

### 2.2 Propulsion system type options

In essence the purpose of a propulsion system is to propel matter in a direction opposite to the direction in which a vessel is desired to travel. It is this expulsion that will induce an oppositely directed reaction force on the vessel, as stated in Newton’s third law of motion. These kind of engines are therefore called reaction engines, and they encompass virtually every common propulsion system used by spacecraft.

A general propulsion system has the following components:

- Propellant storage
- Propellant pressurisation system
- Propellant feed system
- Reaction or heating chamber
- A rocket nozzle

From Newton’s second law of motion \( F = m \cdot a \), we know that for the same amount of expellant mass, the reaction force is higher, when the acceleration is higher. Thus, the higher the velocity of the expellant is, the higher the thrust force. There are many different propulsion system concepts. A literature study was conducted as a prelude to this master thesis [10]. The purpose was to identify the best miniaturisation opportunities for reaction engines in the nearby future. The requirements of the DelFFi mission where used as a reference for nano satellite propulsion system capabilities. In the literature study, five main groups of reaction engines were identified, according to their method of thrust production. Within these groups a total of 29 individual thruster designs were evaluated. Of these 29 concepts, eight were found to be applicable for CubeSat propulsion. A more in-depth trade-off was therefore conducted on these remaining concepts. The final trade-off grading for these
concepts is shown in Appendix A. The top three most applicable propulsion system concepts and their trade-off scores are:

1. Cold gas 414
2. Resistojet 413
3. Mono-propellant chemical 412

These three lie too close together to say which concept is best for this mission. This will have to be evaluated in more detail. The fourth highest scoring concept has a score of 381, which is clearly distinguishable from the top three scoring concepts.

2.3 Initial propulsion system selection

In order to select the most applicable propulsion system for the DelFFi mission satellites, the CubeSat requirements are taken into account. These requirements are negotiable, but have generally been defined for good reasons. The following CubeSat requirements have significant influence on propulsion system development [9]:

- 2.1.3 Pyrotechnics shall not be permitted.
- 2.1.4 No pressure vessels over 1.2 standard atmosphere shall be permitted.
- 2.1.4.1 Pressure vessels shall have a factor of safety no less than 4.
- 2.1.5 Total stored chemical energy shall not exceed 100 [W·h] (360 [kJ]).
- 2.1.6 No hazardous materials shall be used on a CubeSat.

In the following subsections, the different propulsion system options will be discussed. After an elimination of all non-optional systems, only the final propulsion system remains.

2.3.1 Chemical monopropellant propulsion system

In such a propulsion system one propellant is led over a catalyst, where it decomposes exothermally, producing a gas at high temperature and pressure [11]. The basic components are shown in Figure 2.1. There is no mixing of propellants or high electrical power requirements and necessary components and development are relatively low cost. This makes it one of the simplest propulsion systems, while still yielding a good performance of 165 [s] to 244 [s] specific impulse. The fact that the propellant is liquid at room temperature also greatly simplifies the storage pressure regulation.

Figure 2.1 - Schematic representation of a typical monopropellant rocket engine [12]

The main problem for monopropellant systems is the propellant handling and storage [13]. The most common monopropellant is hydrazine (N₂H₄). This is highly toxic, corrosive and the freezing point is relatively high at around 2 [°C]. Another monopropellant is hydrogen peroxide (H₂O₂). This is not toxic, it is however corrosive, harmful and a strong oxidant. Hydrogen peroxide also auto-
decomposes and is catalysed to some degree by most materials propellant tanks are usually made of. Therefore the propellant storage needs some sort of venting mechanism. These propellant storage and handling problems can be mitigated by using alternative fuels, such as HydroxylAmmonium Nitrate (HAN), researched by NASA [14]. This has a low melting point at -20 [°C], is less hazardous, and more stable during storage. It does however require a minimum catalyst bed temperature of 316 [°C]. Another problem encountered in monopropellant engines is catalyst bed degradation. However because of smaller mass flow rate, micro-satellites (10-100 [kg]) will likely be subject to less degradation [11].

It can therefore be concluded that the simplicity, low cost and relatively high performance of the monopropellant thrusters are favourable for university projects. This development however necessitates more safety facilities, procedures, expertise and experience than which is available at the chair at this moment. Therefore it cannot be selected as the propulsion system for the DelFFi mission satellites.

2.3.2 Cold gas thruster
Cold gas propulsion systems works by leading a flow of pressurised gas through a controlled valve, after which it is accelerated in a nozzle to produce thrust. Figure 2.2 below, represents a typical cold gas thruster layout [10]. A gas is stored under high pressure, which is allowed to escape through a nozzle, whereby a thrust force is exerted on the thruster. It consists of a gas storage container, gas filling valve V1, filters F1 & F2, an optional ordnance valve, a pressure regulator, a pressure relief and a series of thrusters, with each their own filter F3 and valve V3. These systems have been used since the beginning of spacecraft propulsion. They are therefore very well tested and documented, and inheritance is abundant. Even at the TU Delft Aerospace faculty, the Delfi-n3Xt has already successfully demonstrated such a system called the T³μPS, shown in Figure 2.3.

![Figure 2.2 – A typical cold gas thruster [13]](image-url)
As stated before, cold gas propulsion systems are mainly used in applications, where simplicity and reliability is more important than performance. In the DelFFi mission however, a considerable amount of total impulse is required from the propulsion system. In a cold gas thruster there is no temperature rise due to combustion or heating of some kind. This means that the vessel normally has to constantly be at the same pressure as the required chamber pressure. Traditionally this pressurised gas was carried in high pressure storage tanks, which were pre-pressurised before launch. In CubeSats pre-pressurisation above 1.2 [atm] is however not allowed [16]. Therefore pressurisation has to be done dynamically in orbit. For example by using cold gas generators [17]. As suggested by Dr. Cervone, a cold gas thruster is a feasible option for the DelFFi satellites, when improvements such as a resistojet are in place [18].

The initial design of the DelFFi mission propulsion systems was a scaled up version of the one used in the Delfi-n3Xt. This involved increasing the size of the gas generators, shown in Figure 2.4, from 18.7 [mm] x 6.7 [mm] [19], to about 80 [mm] x 20 [mm]. The propellant load of each gas generator will then grow from 0.3 [g], to 11.4 [g]. This will in its turn result in an increase of nitrogen gas production from 0.125 [g] to 4.77 [g]. When increasing the amount of gas produced is increased by, the plenum internal volume has to grow with the same factor, when the same settling pressure is desired. This will then have to grow from 28 [cm³] to 1067 [cm³]. This volume is equivalent to a cube with inner sides of 10.2 [cm]. This plenum volume alone would not even fit the allocated DelFFi propulsion system. The internal plenum volume could be as little as half the size, which would double the settling pressure and increase the mass of the plenum. Even still it would not fit the propulsion system volume budget, together with the gas generators, valve, nozzle and drive electronics.
Together with colleague Mr. I. Krusharev, it was therefore decided that a pure cold gas thruster, based on the T3μPS gas generators, cannot meet the requirements on the DelFFi propulsion system. The specific impulse of the thruster can be increased by adding a resistojet, to elevate the temperature of the nitrogen before expulsion. This will result in a lower propellant usage and thus a lower required number of gas generators. It will however increase the complexity and power consumption of the propulsion system. With a resistojet there is also required volume increase, due to the resistojet and its drive electronics. Therefore it was found that, even with the resistojet, the CGG based propulsion system would have great difficulty to fulfil all requirements on the DelFFi propulsion system. In the course of the master thesis it became clear that pre-pressurisation of propellant vessels is allowed for this mission. In addition to this, the cost estimates that were supplied by the manufacturer were not within the budget of the DelFFi propulsion system. Therefore the performance of a cold gas thruster with a pressurised gas can be reconsidered.

2.3.3 Electrolysis propulsion system

In an effort to find alternative methods for small satellite propulsion, the author has undertaken a separate investigation into the feasibility of electrolysis propulsion systems [20]. The layout of the components in that proposed system are shown in Figure A.1 in Appendix A. This propulsion system has liquid water propellant stored at atmospheric pressure. This water is electrolysed to form hydrogen ($H_2$) and oxygen ($O_2$), whenever there is electrical power available. These reaction products are stored until thrusting is required. After possibly being heated, the oxygen is expelled, through a nozzle in order to produce thrust. The hydrogen is partially used to further pressurise the water, and partially expelled as fuel when no more pressurisation is required.

The key advantage of this propulsion system is its flexibility. This is due to the fact that the energy consumption and thrust manoeuvres are partially decoupled. Therefore it can produce the pressurised gas by electrolysis, whenever there is available excess electrical power. And it can release this pressurised gas, in order to produce thrust, at any required moment. Since it operates on gas alone, there is no risk of incomplete combustion or vaporisation, which may lead to performance reduction.

The propulsion system has been calculated, to require 6.5 times the electrical energy input of an equivalent resistojet, in order to produce the same amount of thrust. These calculations however only consider the theoretical optimal behaviour. The efficiency of the resistojet is assumed to be lower than that of an electrolysis propulsion system. Therefore it is assumed that this difference will, in reality, be less pronounced. The biggest disadvantage of the proposed technology is on the other hand the technology readiness level. This propulsion system would still require a great deal of research, which is not feasible within the timeframe of the DelFFi mission. Therefore this design option has to be discarded as well.

2.3.4 Resistojet propulsion system

Resistojet propulsion is based on the heating of a liquid or gaseous substance in order to accelerate and expel it in gaseous phase. As discussed in the previous subsection, pre-pressurisation is in principle forbidden by the CubeSat requirements. Therefore the only source of gaseous propellants for resistojets, is a gas generator. The performance of the examined CGGs developed by TNO, was found to be too low to meet DelFFi propulsion system requirements. Therefore the resistojet design
takes a liquid propellant into account. Pressure is still required, in order to feed the propellant to the propulsion system. Thus CGGs can still be integrated with the resistojet system.

At the SSE chair there is a large amount of experience in resistojet development, as will be shown in section 7.1. Preliminary calculations of a resistojet propulsion system using water as a propellant were performed. These showed that such a system can indeed fulfil the DelFFi propulsion system requirements, with respectable margins for error. It was therefore decided to conduct this master thesis, to develop a propulsion system for the DelFFi mission satellites using a resistojet. Section 2.4 will describe the possible layout of such a propulsion system. After which section 2.5 is devoted to showing which propellant is most suitable for such a propulsion system.

2.4 Resistojet propulsion system layout

A resistojet based propulsion system has a number of necessary components. There has to be a storage system for the propellant. This propellant has to be delivered to the feed system by an overpressure delivered by the pressurisation system. The flow through the feed system has to be controlled by a valve. The flow then enters the resistojet heating chamber through the propellant dispersing inlet. This inlet divides the flow across the heating chamber entrance area. Within the heating chamber, the propellant is vaporised and heated to the desired final temperature. The gaseous propellant is then expelled through a converging-diverging nozzle in order to increase the exhaust velocity.

There are also a couple of support components necessary for the operation and monitoring of the propulsion system. There has to be a fill valve to insert the propellant and possibly the pressurant into the storage tank. A temperature sensor is required, to monitor the temperature of the resistojet heating chamber. It is also desirable to have a pressure sensor in the propellant storage. The monitoring of this pressure will aid in the assessment of the operation and effectiveness of the propulsion system. The propulsion system has to be operated and monitored by electronics on a Printed Circuit Board (PCB). All the mentioned propulsion system sections will be discussed individually in the coming chapters of this report. In Figure 2.5 a schematic overview of a proposed propulsion system for the DelFFi mission satellites is shown. This is only meant as an overview of the relevant components and their relation to each other. It is not meant to be a correct structural drawing of the final propulsion system.
The numbers added to this image indicate the following parts of the propulsion system:

1. Resistojet nozzle exit
2. Mounting bracket to exterior CubeSat bottom panel
3. Resistojet housing
4. Propellant valve
5. Propellant tank lid, attached to internal mounting rods
6. Propellant tank including pressurisation
7. Spacing for electronic components on the PCB
8. Printed Circuit Board (PCB), which attaches to internal mounting rods

Not included in Figure 2.5 are the necessary sensors for monitoring the system, the propellant fill valve and a possible pressurant fill valve. Contrary to the design in the figure, the most optimal design would have the resistojet in the centre of the propellant storage system. The propellant could then be in a toroidal tank around the thruster itself. In this configuration the principal heat loss of the thruster can be absorbed by the stored propellant. This will help in the effort of keeping the water propellant from freezing. The resistojet itself could then even be used as a heater to melt the water propellant, when it freezes. The elevated temperature of the stored propellant will also result in a lower necessary heat input for vaporisation in the thruster. It will therefore decrease the power consumption of the thruster, by using the waste heat of the thruster. A toroidal propellant tank shape is relatively complex. This shape will also possibly increase the complexity of propellant pressurisation and collection, which will be discussed later in this document.

2.5 Propellant selection
As has been documented in subsection 2.3.4, it is very beneficial for the resistojet to have a propellant, which is liquid around room temperature. This is the reason why both melting temperature and boiling temperature are important selection parameters for potential propellants. With a liquid propellant, the most important function of the heating element of the resistojet, is to
vaporise the propellant. Only when the propellant is gaseous throughout the rocket nozzle, can thrust be produced most efficiently. Therefore it is important to know to which temperature the propellant has to be heated in order to vaporise it, i.e. again the boiling temperature. But it is mainly important to know what the energy requirement is, for the vaporisation of the propellant. This energy depends partially on the energy to raise the temperature of the propellant to the boiling temperature, which in turn depends on the heat capacity of the substance. The main constituent of the required vaporisation energy, is the energy needed for the molecules to break free from the liquid as a gas [21]. This so called latent heat of vaporisation, has to be added to overcome the surface resistance of the liquid. This parameter is therefore also an important performance indicator for propellant selection.

As will be shown in equation 3-24, the thrust produced by a resistojet depends on a number of different factors. In order to compare the effect of using different propellants, the behaviour is idealised. When it is assumed that the nozzle is able to ideally expand the flow to a vacuum, the thrust in equation 3-24, only depends on the propellant mass flow and exhaust velocity. In the ideal case of adiabatic expansion of the accelerated flow to vacuum conditions, the exhaust velocity equals the limiting velocity \( U_L \)[22]:

\[
U_L = \sqrt{2 \cdot \frac{\gamma R_A}{\gamma - 1} \cdot \frac{T_c}{M}}
\]

Where:
- \( U_L \) Limiting velocity [m/s]
- \( \gamma \) Ratio of specific heats [-]
- \( R_A \) Universal gas constant 8.314510 [J/mol/K] [4]
- \( \dot{M} \) Molar mass [g/mol]
- \( T_c \) Heating/combustion chamber temperature [K]

The universal gas constant is the same for every substance. The height of the limiting exhaust velocity, for a certain combustion chamber temperature, therefore depends on the following ratio:

\[
\frac{\gamma}{M(\gamma - 1)}
\]

The ratio of specific heats is difficult to define, as it depends on the temperature of the substance. The difference in ratio of specific heats between different substances is very small compared to the difference in molar mass. Therefore the maximum exhaust velocity of the substances depends mainly on their molar mass. The lower the molar mass, the higher the exhaust velocity. The higher the exhaust velocity, the higher the thrust for a certain mass flow, as will be shown in chapter 3. In this case idealisation of the behaviour is done to simplify the equations. Even when the behaviour is not ideal or when it is operating in non-vacuum conditions, this relation between molar mass and exhaust velocity still holds.

Since the DelFFi mission is mainly conducted by students, the safety of the potential propellants also plays a big role. As students do not always have the time for lengthy safety courses, and generally lack experience with dangerous substances. Also because the mission is of a highly experimental
nature, with many uncertainties and therefore potential risks. All the above mentioned propellant selection criteria have been investigated for many common substances, which are liquid at room temperature. The data has been collected from various sources, and is summarised in Table 2.1. Since the molar mass is one of the main efficiency drivers, from most substance types, e.g. hydrocarbons, only the ones with the lowest molar mass are shown in the table.

Table 2.1 - Relevant properties of common substances which are liquid around room temperature, ordered by molar mass. Gathered in order to assess potential propellants. Most properties collected on the NIST database [23] and on [24].

<table>
<thead>
<tr>
<th></th>
<th></th>
<th></th>
<th></th>
<th></th>
<th></th>
<th></th>
<th></th>
</tr>
</thead>
<tbody>
<tr>
<td>Inorganic base</td>
<td>Ammonia</td>
<td>17</td>
<td>195</td>
<td>240</td>
<td>23.4</td>
<td>Pressurising, Corrosive, Toxic</td>
<td></td>
</tr>
<tr>
<td>Oxygen hydride</td>
<td>Water</td>
<td>18</td>
<td>273</td>
<td>373</td>
<td>44.0</td>
<td>Benign</td>
<td></td>
</tr>
<tr>
<td>Inorganic acid</td>
<td>Hydrogen cyanide</td>
<td>27</td>
<td>260</td>
<td>299</td>
<td>25.2</td>
<td>Toxic, flammable</td>
<td></td>
</tr>
<tr>
<td>Alcohol</td>
<td>Methanol</td>
<td>32</td>
<td>176</td>
<td>338</td>
<td>35.2</td>
<td>Flammable, toxic, harmful</td>
<td></td>
</tr>
<tr>
<td>Peroxide</td>
<td>Hydrogen peroxide</td>
<td>34</td>
<td>273</td>
<td>423</td>
<td>51.6</td>
<td>Oxidant, corrosive, Harmful</td>
<td></td>
</tr>
<tr>
<td>Mineral acid</td>
<td>Hydrochloric acid</td>
<td>36</td>
<td>247</td>
<td>321</td>
<td>9.1</td>
<td>Corrosive, hazardous</td>
<td></td>
</tr>
<tr>
<td>Alcohol</td>
<td>Ethanol</td>
<td>46</td>
<td>159</td>
<td>352</td>
<td>38.6</td>
<td>Flammable, harmful</td>
<td></td>
</tr>
<tr>
<td>Ketone</td>
<td>Acetone</td>
<td>58</td>
<td>179</td>
<td>329</td>
<td>29.1</td>
<td>Flammable, hazardous</td>
<td></td>
</tr>
<tr>
<td>Carboxylic acid</td>
<td>Acetic acid</td>
<td>60</td>
<td>290</td>
<td>391</td>
<td>23.7</td>
<td>Flammable, hazardous</td>
<td></td>
</tr>
<tr>
<td>Mineral acid</td>
<td>Nitric acid</td>
<td>63</td>
<td>232</td>
<td>356</td>
<td>39.1</td>
<td>Oxidant, corrosive</td>
<td></td>
</tr>
<tr>
<td>Hydrocarbon</td>
<td>Pentane</td>
<td>72</td>
<td>144</td>
<td>309</td>
<td>26.2</td>
<td>Flammable, harmful</td>
<td></td>
</tr>
<tr>
<td>Ester</td>
<td>Ethyl formate</td>
<td>74</td>
<td>193</td>
<td>327</td>
<td>29.9</td>
<td>Flammable, hazardous</td>
<td></td>
</tr>
<tr>
<td>Aromatic hydrocarbon</td>
<td>Toluene</td>
<td>92</td>
<td>178</td>
<td>384</td>
<td>38.0</td>
<td>Flammable, harmful</td>
<td></td>
</tr>
<tr>
<td>Monoterpene</td>
<td>Limonene</td>
<td>136</td>
<td>199</td>
<td>451</td>
<td>39.5</td>
<td>Flammable, harmful</td>
<td></td>
</tr>
<tr>
<td>Monoterpene</td>
<td>Myrcene</td>
<td>136</td>
<td>263</td>
<td>440</td>
<td>-</td>
<td>Flammable, harmful</td>
<td></td>
</tr>
<tr>
<td>Monoterpene</td>
<td>Pinene</td>
<td>136</td>
<td>213</td>
<td>430</td>
<td>46.6</td>
<td>Flammable</td>
<td></td>
</tr>
<tr>
<td>Unsaturated ether</td>
<td>Anethole</td>
<td>148</td>
<td>293</td>
<td>507</td>
<td>45.5</td>
<td>Benign</td>
<td></td>
</tr>
<tr>
<td>Halogen</td>
<td>Bromine</td>
<td>160</td>
<td>266</td>
<td>332</td>
<td>30.9</td>
<td>Corrosive, toxic</td>
<td></td>
</tr>
<tr>
<td>Metal</td>
<td>Mercury</td>
<td>201</td>
<td>234</td>
<td>630</td>
<td>59.1</td>
<td>Toxic</td>
<td></td>
</tr>
</tbody>
</table>
Ammonia has the lowest molar mass of all substances mentioned in Table 2.1. The liquid phase is between -78°C and -33°C. It is therefore not liquid at room temperature. Ammonia is however very soluble in water, and can thus be added in small quantities to the water propellant in order to increase the performance. This mixture is generally called ammonium hydroxide. The mixing will also solve the problem of the high melting temperature of water. Thereby decreasing the risk of propellant freezing during cold periods. It will however increase the corrosiveness and toxicity of the water.

As can be seen from Table 2.1, water is the substance with the lowest molar mass, which is still liquid at room temperature. It is therefore capable of producing the highest total impulse for a given amount of fuel. One disadvantage of using water as resistojet propellant, is that the melting temperature is relatively high. Therefore there is the risk of propellant freezing. As said before, this might be circumvented by adding ammonia to the water. A second disadvantage is that it has a rather high enthalpy of vaporisation. It will therefore require relatively large amounts of energy to vaporise the liquid.

Hydrogen cyanide is the substance with the second lowest molar mass in the list. It will therefore not be able to attain the exhaust velocities and thus total impulse of water. The advantage is that the enthalpy of vaporisation is significantly lower. It will therefore require less energy to vaporise. The liquid temperature range is also more favourable, because it does not pose the risk of propellant freezing. Hydrogen cyanide is however extremely toxic and is therefore not suitable for student handling and highly experimental projects.

Alcohols have a lower heat requirement and a more favourable temperature range than water. They are also readily available and cheap. The lightest one, methanol, is very toxic, even though the molar mass is already almost double that of water. Therefore water is still favoured above alcohols as resistojet propellant.

Hydrogen peroxide still has a relatively low molar mass, and a good liquid temperature range. It is however very difficult to handle and to store. This is due to its instability, corrosiveness and harmfulness. In the presence of a proper catalyst and/or temperature, it will decompose. This produces its own heat, and exhaust products with a lower molar mass. It would however then form a chemical monopropellant thruster, which was discarded in subsection 2.3.2.

Hydrochloric acid deserves a special mention because of its very low enthalpy of vaporisation. It therefore requires the least amount of energy input to vaporise it, out of all here listed liquids. It does however have a significantly high molar mass compared to water. Out of all substances listed here hydrochloric acid is, in addition to that, very hazardous to work with. Low concentrations of <10% of the hydrochloric acid, can be introduced in the water propellant to lower the melting point down to -20°C. Again reducing the risk of propellant freezing. But this will increase the average molar mass.

Hydrocarbons are readily obtainable, very cheap and well documented. However the simple hydrocarbons with the lowest molar mass are all gaseous at room temperature. The lightest hydrocarbon which is liquid at room temperature is Pentane (C₅H₁₂). This still has a relatively low
boiling point and a high molar mass. Hydrocarbons are also toxic and flammable. This makes them clearly unsuitable as resistojet propellant.

Essential oils are usually terpenes, and all have a high molar mass, due to the large molecular structure with 10-40 carbon atoms [25]. Their liquid phase temperature range does not pose a risk of either boiling or solidifying during storage. They however have a large enthalpy of vaporisation, and the molar masses are too large to be effective as resistojet propellant.

Bromine and Mercury are the only elements that are liquid at standard room temperature and pressure. They are however the liquids with the highest molar mass in this list. Therefore they are not desirable as propellant in resistojets.

To conclude this propellant selection, water is found to be the best resistojet propellant, due to its low molar mass alone. In addition to this, it is also the most benign and readily available, cheap substance in this investigation. Other substances can be added in order to reduce the melting temperature. This reduces the chance on propellant freezing, during cold operational periods. The specific heat was not taken into consideration in this investigation. In future efforts this might be considered together with the enthalpy of vaporisation, in order to form a view on which fuel would be most energy efficient to use.
3 Performance calculations

In order to meet the requirements mentioned in section 2.1, several different free design parameters have to be weighed against each other. These parameters can be designed in order to, together, yield the necessary performance. They also need to individually meet their own specific requirements. These free design values for the water resistojet are:

- Mass of the water propellant
- Mass of the pressurant
- Storage volume
- Electrical power input
- Heat loss in the system
- Pressure loss in the system
- Final propellant temperature
- Nozzle dimensions

The performance of the system is assessed by iterating over different design parameters, until the requirements are met. The behaviour of the propulsion system is simulated using equations for partially idealised situations. For these calculations, the system starts at the initial propellant and pressurant mass. At which point all the flow parameters from propellant storage to expulsion are calculated. Time steps are taken from a full tank to an empty tank. At each time step, all flow parameters are calculated throughout the system. This data is then finally integrated over time to obtain the total performance. In the following sections of this chapter, the calculation of the flow properties of each section of the propulsion system will be described.

3.1 Propellant storage and pressurisation

The stored volume of the propellant is calculated in order to calculate the volume, and thus pressure, of the pressurant. The stored propellant mass of the current time step, is the stored propellant mass of the last step, minus the calculated mass flow of the last step, i.e.

\[ m_{prop} = m_{prop,\text{-}1} - \dot{m}_{i-1} \cdot \Delta t \]

Where:

- \( m_{prop} \) The total stored propellant mass [kg]
- \( i \) The current time step [-]
- \( \dot{m} \) Mass flow [kg/s]
- \( \Delta t \) The duration of the last step [s]

If this calculation shows that the propellant supply has been exhausted, i.e. \( m_{prop} < 0 \), the calculations stop and display the total performance values. The total stored propellant volume is the propellant mass divided by its density:

\[ V_{prop} = \frac{m_{prop}}{\rho_{prop}} \]
Where:

\[ V_{\text{prop}} \]  Propellant volume [m\(^3\)]

\[ \rho_{\text{prop}} \]  Density of the propellant at the system temperature [kg/m\(^3\)]

The new pressurant volume, i.e. the volume that contains the gas used for pressurisation, is the total storage volume minus the total stored propellant volume.

\[ V_{\text{pres}} = V_{\text{stor}} - V_{\text{prop}} \]

Where:

\[ V_{\text{stor}} \]  Is the total storage volume [m\(^3\)]

The pressurant for the current time step can be calculated using the ideal gas law, shown in equation 3-1 below.

\[ p_{\text{pres}} = \frac{n \cdot R_A \cdot T_{\text{pres}}}{V_{\text{pres}}} \]  \hspace{1cm} (3-1)

Where:

\[ n \]  Amount of pressurant [mol]

\[ p_{\text{pres}} \]  Pressurant pressure [Pa]

\[ R_A \]  Universal gas constant [J/K/mol]

\[ T_{\text{pres}} \]  Pressurant temperature [K]

Equation 3-1 only holds for hypothetical gasses where molecules are negligible in size and have no intermolecular forces [22]. Due to the relatively low temperature, pressure and flow rate of the pressurant gas, this assumption is valid.

As will be explained in chapter 6, there is some difficulty in extracting the propellant from the tank, in weightless conditions. Therefore it is highly likely that not all propellant can be extracted from the tank during the lifetime of the propulsion system. The percentage of the total initial amount of propellant, that can actually be expelled from the propellant storage is called the “expulsion efficiency”. This expulsion efficiency is in the order of 95% to 99%, for normal liquid propulsion systems [22]. The propulsion system described in this master thesis is however of a more experimental nature. It is also mostly designed by relatively inexperienced engineers. Therefore an expulsion efficiency of only 90% was assumed for the calculations in this thesis.
3.2 Feed system

The drop in total pressure due to the feeding system piping needs to be calculated. The exact layout of the piping system is not known, but it will be approximated. The ideal feed system would have the following sequential sections, also shown in Figure 2.5:

- Propellant tank exit in the centre of the face which faces the nozzle location
- Straight connection piping
- Propellant control valve
- Fluidic connection piece to the thruster

These sections would ideally all be straight, smooth and have the exact same diameter. This would lead to a system with only the losses due to a relatively short section of straight, smooth piping. Such a loss will later be shown to be negligible. Only the flow constriction loss of the flow from the propellant tank to the feed system will induce some extra losses.

3.2.1 Feed system geometry estimation

The assumption of having the same diameter everywhere is valid. Because piping is readily available from multiple, even relatively local, suppliers in any required diameter [26]. The valve diameter will be the leading dimension in this selection, since there is only a limited selection of applicable valves. In reality the positioning of the piping will most likely not be optimal. Instead it will require some bends in order to comply with geometrical restrictions. The final feed system piping will not have one continuous smooth surface. Especially at the interfaces between different feed system components, there will be a disturbances in the wall. In order to anticipate the non-ideal geometry, for the following calculations, the propellant tank exit is assumed to be on the side of the tank and valve is mounted horizontally to the bottom of the tank. All sections are assumed to be connected with flexible tubing. This geometry has five 90° angles and 4 section transitions and, given the propulsion system dimensions, 17 [cm] of piping.

For the propulsion system engineering model tests, the Lee company INX0511400A A solenoid valve is used for propellant supply control. This valve is selected for its similarity to the valve which will be used on the actual satellites. Therefore the inside diameter of 0.89 [mm] [27] [28], of this engineering model valve is used as a reference for flow calculations.

3.2.2 The pressure drop due to friction in the piping

In order to calculate the pressure drop across a section of feed system, a number of flow parameters first have to be estimated and calculated. Finally, these parameters can be inserted in equation 3-6 to yield the answer. As will be shown later in this chapter a mass flow of \( \dot{m} = 2 \) [mg/s] will never be exceeded in the designed propulsion system. Therefore a mass flow of \( \dot{m} = 2 \) [mg/s] is taken as an indication and a water density of \( \rho = 1000 \) [kg/m\(^3\)] is assumed. The flow velocity is then calculated as follows:

\[
U = \frac{\dot{m}}{\rho \cdot A}
\] 3-2
Design, manufacturing and characterisation of a water fed CubeSat micro-resistojet

Where:

- $U$ Flow velocity [m/s]
- $\dot{m}$ Mass flow [kg/s]
- $\rho$ Density of the flow [kg/m$^3$]
- $A$ Cross-sectional area of the flow channel [m$^2$]

In the given channel this corresponds to a propellant flow velocity of $U = 3.21 \cdot 10^{-3}$ [m/s]. The kinematic viscosity is a measure of a fluids resistance to deformation, which is calculated using equation 3-3.

$$\nu = \frac{\mu}{\rho}$$

Where:

- $\nu$ Kinematic viscosity [m$^2$/s]
- $\mu$ Dynamic (absolute) viscosity [kg/s/m]

The dynamic viscosity for water at room temperature (25 [°C]) is $\mu = 891 \cdot 10^{-6}$ [kg/s/m] [29]. One of the most indicative flow parameters used throughout the aerospace research spectrum, is the Reynolds number. This dimensionless quantity aids in the prediction of flow behaviour. The Reynolds number based on tube diameter is usually defined as shown in equation 3-4 below.

$$Re = \frac{U \cdot D_h}{\nu}$$

Where:

- $Re$ Reynolds number [-]
- $U$ Flow velocity [m/s]
- $D_h$ Hydraulic diameter of the tube [m]

Turbulence in the flow will induce height convectional losses, due to the chaotic behaviour. The transition between regimes occurs at critical Reynolds numbers. Typical critical Reynolds numbers, depending on the flow channel geometry can be found in Table 3.1.

**Table 3.1 - Critical Reynolds numbers in flow channels with different geometries. The aspect ratio $\alpha$ is the ratio between channel height and channel width of the flow channel $\alpha = \frac{h_{ch}}{w_{ch}}$.**

<table>
<thead>
<tr>
<th>Flow channel type</th>
<th>Critical Reynolds number</th>
</tr>
</thead>
<tbody>
<tr>
<td></td>
<td>Laminar</td>
</tr>
<tr>
<td>Straight tube [30]</td>
<td>&lt;2300</td>
</tr>
<tr>
<td>Rectangular channel $\alpha \leq 0.2$ [31][32]</td>
<td>2500</td>
</tr>
<tr>
<td>Rectangular channel $0.2 &lt; \alpha \leq 1.0$ [31][32]</td>
<td>$2500 - 375 \cdot (\alpha - 0.2)$</td>
</tr>
<tr>
<td>Rectangular channel $1.0 &lt; \alpha &lt; 5.0$ [31][32]</td>
<td>$2500 - 75 \cdot (\alpha - 1.0)$</td>
</tr>
<tr>
<td>Rectangular channel $\alpha \geq 5.0$ [31][32]</td>
<td>2500</td>
</tr>
<tr>
<td>Straight pipe [22]</td>
<td>&lt;2320</td>
</tr>
</tbody>
</table>
The hydraulic diameter $D_h$ in case of a circular tube is simply its inner diameter. The resulting Reynolds number is $3.21 \ [\cdot]$. This, as can be seen in Table 3.1, means that the flow is certainly laminar. The friction factor for laminar flow, in a channel with a circular cross-section is approximated by equation 3-5 [31].

$$f = \frac{16}{\text{Re}} \quad 3-5$$

Where:

$f$ \quad Fanning friction factor \ [-]

Which can be calculated to be $f = 4.98 \ [-]$. Using all previously calculated flow properties, the pressure drop across a straight tube is calculated using the following formula [31]:

$$\Delta p = 2 \cdot \rho \cdot U^2 \cdot f \cdot \frac{L}{D_h} \quad 3-6$$

Where:

$\Delta p$ \quad Pressure drop \ [Pa]
$L$ \quad Tube length \ [m]

The pressure drop of $90^\circ$ bends can be calculated using equation 3-6, with an equivalent length of $57$ times the inner diameter [31]. This equivalent length adds to the estimated $17 \ [\text{cm}]$ of piping. Therefore the total equivalent length of the feed system becomes $0.424 \ [\text{m}]$. Using equation 3-6 with all the calculated flow parameters, results in the following pressure drop:

$$\Delta p = 2 \cdot 1000 \cdot (3.21 \cdot 10^{-1})^2 \cdot 4.98 \cdot \frac{0.424}{0.89 \cdot 10^{-3}} = 49.0 \ [\text{Pa}]$$

This can clearly be seen to be negligible. Even if the propellant is pressurised only to atmospheric pressure of $1 \ [\text{bar}]$, this is still only $0.049\%$ pressure loss.

### 3.2.3 The pressure drop due to the propellant control valve

For the propulsion system engineering model tests, the Lee company INKX0511400A A solenoid valve is used for propellant supply control. This valve is stated to have a flow restriction of $4750 \ [\text{Lohms}]$, which is equal to $C_v = 0.004 \ [-] \ [33]$. In this case, $C_v$ is defined to be the flow coefficient which indicates the efficiency of the valve to allow fluid flow. Using this flow coefficient the pressure drop across the valve can be calculated using equation 3-7 [34].

$$\Delta p = G_{f} \left( \frac{q}{N_i \cdot C_v} \right)^2 \quad 3-7$$

Where:

$G_f$ \quad Liquid specific gravity, water $= 1.0 \ [-]$
$q$ \quad Flow rate \ [L/min]
Later in this chapter, a mass flow rate of 2 [mg/s] is shown to be above the highest value which will be encountered in operation. This mass flow rate results in a volume flow rate of $120 \times 10^{-6}$ [L/min]. This yields a pressure drop of $4.33 \times 10^{-6}$ [bar]. This pressure drop across the valve can thus be seen to be negligible, compared to the feed system pressure in the order of 5 [bar] to 7 [bar].

### 3.2.4 The pressure drop due to piping transitions

When there is a sudden contraction or expansion in a flow channel, this will introduce flow separation and turbulence. This turbulence will cause a loss in the total pressure, shown in equation 3-8 below [22].

\[
\Delta p = \zeta \cdot \frac{1}{2} \cdot \rho \cdot U^2
\]

Where:
- $\Delta p$: Pressure drop [Pa]
- $\zeta$: Pressure loss coefficient [-]

For a suddenly contracting flow, the pressure loss coefficient is [22]:

\[
\zeta_{\text{contr}} = 0.5 \left( 1 - \frac{A_2}{A_1} \right)^{\frac{3}{2}}
\]

Where:
- $A_2$: Secondary, smaller piping cross-sectional area [m$^2$]
- $A_1$: Primary, larger piping cross-sectional area [m$^2$]

For suddenly expanding flow, the pressure loss coefficient is [22]:

\[
\zeta_{\text{exp}} = \left( 1 - \frac{A_1}{A_2} \right)^{2}
\]

Where:
- $A_1$: Primary, smaller piping, cross-sectional area [m$^2$]
- $A_2$: Secondary, larger piping, cross-sectional area [m$^2$]

To simulate the transitions between the different feed system sections, they are represented by sudden diameter changes. To have an order of magnitude pressure loss estimation, an increase in diameter of a factor two is assumed. After which it immediately returns back to the general piping diameter. This agrees with the transition for the valve fittings [28]. The pressure drop of one such a transition will then be:
Due to the very low flow velocities and modest diameter change, this pressure drop is clearly negligible. Even for the estimated four section transitions, this amounts to a mere 0.020 [Pa] pressure drop. This is many orders of magnitude lower than the operating pressure of the propulsion system, which is in the order of 5 [bar] as will be shown later in this chapter.

3.3 Heating chamber

The heating chamber is divided into two sections. The first section is the vaporisation section, including the liquid phase heating and vaporisation of the propellant. The second section is the vaporised propellant heating section, where the then gaseous propellant is heated to its desired final temperature. There is no physical difference between these two sections, as both occur sequentially in the same chamber. The calculations are merely separated due to the differences in flow phenomena. Calculations on each section individually will be shown in the following two subsections.

3.3.1 Calculations on the vaporisation section

In order to vaporise the propellant, it first needs to be heated to its boiling point at the given pressure. After which, the so called latent heat of vaporisation, has to be added to overcome the surface resistance of the liquid [21]. This heat allows the molecules to break free from the liquid as a gas. One of the most important performance drivers of the propulsion system, is the electrical power consumption. This electrical power consumption is a product of the heating power required by the propellant, and the efficiency of the heating process. In order to calculate the power which needs to be inserted into the propellant flow, the enthalpy difference before and after the heating section needs to be known. Enthalpy indicates thermodynamic state or potential of a system, which depends on its internal energy, pressure and volume. The enthalpy of the propellant is calculated using the Shomate equation from the NIST database [23].

\[
H^* = A \cdot t + B \cdot \frac{t^2}{2} + C \cdot \frac{t^3}{3} + D \cdot \frac{t^4}{4} - \frac{E}{t} + F
\]  

3-11

Where:

- \( H^* \)  The enthalpy of the fluid at the given temperature [J/mol]
- \( A \) to \( H \)  Empirical constants from the NIST database, shown in Appendix B
- \( t \)  Temperature [K/1000]

The temperature of the flow at the end of the vaporisation section, is equal to the temperature at which the pressure in the flow is equal to the vapour pressure. Which means that the temperature is high enough to allow the propellant to be vaporised under the given pressure. This boiling point temperature is found using the Antoine equation from the NIST database [23]:

\[
\Delta p = (\zeta_{\text{exp}} + \zeta_{\text{contr}}) \cdot \frac{1}{2} \cdot \rho \cdot U^2 = \left(1 - \frac{1}{4}\right)^2 + 0.5 \cdot \left(1 - \frac{1}{4}\right)^3 \cdot \frac{1}{2} \cdot 1000 \cdot (3.21 \cdot 10^{-3})^2
\]

\[
\Delta p = 4.99 \cdot 10^{-3} [\text{Pa}]
\]
\[
\log_{10}(\rho_{\text{vap}}) = A - \left( \frac{B}{T_{\text{vap}} + C} \right) \rightarrow T_{\text{vap}} = \frac{B}{A - \log_{10}(\rho_{\text{vap}})} - C \tag{3-12}
\]

Where:
- \(p_{\text{vap}}\) Vapour pressure [bar]
- \(T_{\text{vap}}\) Temperature at vapour pressure, i.e. boiling point [K]
- \(A, B, C\) Parameters as found in the NIST database, shown in Appendix B

When this vapourisation temperature or boiling point at the given pressure is known, the enthalpy at the end of the vapourisation section can be determined. This enthalpy is calculated with equation 3-11 using the coefficients for gaseous water, which are also documented in Appendix B. The power required for vapourisation is then calculated using equation 3-13 below.

\[
P_{\text{vap}} = \hat{m}_{\text{prop}} \cdot \left( H_{\text{vap,\text{out}}} - H_{\text{vap,\text{in}}} \right) \tag{3-13}
\]

Where:
- \(P_{\text{vap}}\) Power required for vapourisation [W]
- \(\hat{m}_{\text{prop}}\) Propellant mass flow [kg/s]
- \(\hat{M}_{\text{prop}}\) Propellant molar mass [kg/mol]
- \(H_{\text{vap,\text{out}}}\) Enthalpy of the gaseous propellant at vapourisation section exit [J/mol]
- \(H_{\text{vap,\text{in}}}\) Enthalpy of the liquid propellant at the vapourisation section entrance [J/mol]

For the heater efficiency calculations, it is assumed that the liquid water density change because of the heating up until the vapourisation is negligible. This means that the volume flow at the beginning of the vapourisation section is equal to the volume flow just before the actual vapourisation. The electrical power required for the propellant vapourisation section, depends on the efficiency of the heater to convert the electrical energy to thermal energy in the propellant:

\[
P_{\text{elec,\text{vap}}} = \eta_{\text{vap}} \cdot P_{\text{vap}}
\]

Where:
- \(P_{\text{elec,\text{vap}}}\) Electrical power required for the propellant vapourisation section [W]
- \(\eta_{\text{vap}}\) Vapourisation section efficiency [-]

This efficiency depends on the heat transfer or loss to the surroundings of the thruster. In these initial system level calculations, a heating efficiency of 50% was assumed to do the full calculations. It was later attempted to estimate the true heater efficiency, using heat transfer calculations, as will be shown in chapter 7. A detailed calculation proved to be too intensive for the timeframe of this research. The necessary assumptions to make a simplified approximation of the heat loss posed too much uncertainty to be valid. Eventually an approximate simplified solution has been calculate to verify the assumption.
3.3.2 Calculations on the vaporised propellant heating section

The second section in the heating chamber will heat the, now vaporised, propellant to its final desired temperature. These calculations are based on the assumption, that the desired final temperature is reached at the end of the section. As described for the vaporisation section, the enthalpy difference in the flow, between the entrance and exit of the heating section, is equal to the heat that has to be added to the flow to attain the desired thermodynamic state, i.e. final temperature. The enthalpy of the gaseous propellant flow at the start of this section, was already calculated as the enthalpy of the flow at the end of the vaporisation section. The enthalpy at the end of the heating chamber, is calculated using the above Shomate equation, shown in equation 3-11. The power required for heating is calculated using equation below.

\[
P_{\text{heat}} = \frac{m_{\text{prop}}}{M_{\text{prop}}} \left( H_{\text{heat,exit}} - H_{\text{evap,exit}} \right)
\]

Where:

- \( H_{\text{heat,exit}} \): Enthalpy of the gaseous propellant, with the desired final temperature, at gaseous propellant heating section exit [J/mol]

The electrical power required for the gaseous propellant heating section, again depends on the efficiency of the heater:

\[
P_{\text{Elec,heat}} = \eta_{\text{heat}} \cdot P_{\text{heat}}
\]

Where:

- \( P_{\text{Elec,heat}} \): Electrical power required for the gaseous propellant heating section [W]
- \( \eta_{\text{evap}} \): Heater efficiency [-]

Again a heater efficiency of 50% was assumed. Which means that 50% of the inserted electrical energy is lost to the surroundings.

3.4 Rocket nozzle

The purpose of a rocket nozzle is to accelerate the expellant, and to bring the total pressure of the flow as close to the ambient pressure as possible. The expulsion of the expellant will exert a force on the thruster, in agreement with Newton’s second law. Therefore a larger exhaust velocity leads to a larger force. This force works in the opposite direction of expelled particles’ velocity vector, as stated in Newton’s third law. It is therefore the intention that the nozzle increases the velocity of the expellant as much as possible, with the geometrical constraints imposed on it.

The nozzle chosen for the Dondersteen micro-resistojet is a converging-diverging nozzle, as shown in Figure 3.1. This nozzle type is the most commonly used type for rocket engines that are based on heated gas expulsion. It functions on the principle of the different behaviour of gas between subsonic and supersonic flow. A property of subsonic flow is that the flow speed increases when the flow channel cross-sectional area decreases. The subsonic flow is therefore first accelerated due to the contraction of the converging section. If the pressure difference between the heating chamber and the nozzle exit are high enough, the flow will be choked in the narrowest section, called the throat. This means that the flow attains sonic velocity. A property of supersonic flow is that the velocity
increases when the cross-sectional area of the flow channel increases. Therefore the flow velocity will further increase in the diverging part of the nozzle. The accelerated propellant flow will finally be expelled at the nozzle exit.

![Schematic cross-section of a converging-diverging nozzle](image)

**Figure 3.1 - Schematic cross-section of a converging-diverging nozzle - Flow direction from left to right**

The flow can theoretically be expanded until the ambient pressure is reached. It can also be expanded to a pressure even lower than the ambient pressure. This is excessive expansion is called overexpansion. Since the Dondersteen thruster will operate in vacuum conditions, this is however not possible or relevant. To expand the exhaust flow to vacuum ambient conditions, the exit has to be infinitely large, which is physically impossible. Therefore there is a point where the increase in exhaust velocity, does no longer weigh up to the increase in nozzle size. The exhaust flow will therefore always be under expanded. This under expansion reduces the efficiency of the nozzle to convert the potential and heat energy of the flow to kinetic energy.

In subsection 3.4.1 the theoretical nozzle performance will be discussed. This is followed by a prediction for the translation of this theoretical performance to the true performance, in subsection 3.4.2.

### 3.4.1 Theoretical nozzle prediction

As mentioned before, if sonic flow velocity is attained in the nozzle throat, the flow is choked. The mass flow will not increase anymore by decreasing the downstream pressure. This maximum mass flow, is called the critical mass flow rate. This limiting mass flow rate therefore defines the mass flow throughout the propulsion system. The critical mass flow rate through the throat of the nozzle is calculated using equation 3-15 [22].

$$m_{\text{crit}} = \frac{\Gamma_{\text{prop}} \cdot p_{\text{heat,ext}} \cdot A_t}{\sqrt{R \cdot T_{\text{heat,ext}}}}$$

where:
- $m_{\text{crit}}$: Critical mass flow [kg/s]
- $\Gamma_{\text{prop}}$: Vandenkerckhove parameter [-]
The specific gas constant is defined to be:

\[ R = \frac{R_A}{M} \]  

The above mentioned Vandenkerckhove constant is calculated using equation 3-17 [22].

\[ \Gamma_{\text{prop}} = \sqrt{\gamma_p \left( \frac{2}{\gamma_p + 1} \right)^{\frac{\gamma_p + 1}{2(\gamma_p - 1)}}} \]  

Where:

- \( \gamma_p \) is the ratio of specific heats \( C_p/C_v \) [-]
- \( C_p \) is the heat capacity of the fluid at constant pressure [J/kg/K]
- \( C_v \) is the heat capacity of the fluid at constant volume [J/kg/K]

If the mass flow corresponds to the critical mass flow, the flow is choked and the velocity in the throat is sonic. In that case the Mach number of the flow at the exit of the nozzle can be calculated, using equation 3-18 [22].

\[ \frac{A_e}{A_t} = \left( \frac{1}{M_e} \right) \left( \frac{2}{\gamma_p + 1} \left( 1 + \frac{\gamma_p - 1}{2} \cdot M_e \right) \right)^{\frac{\gamma_p + 1}{2(\gamma_p - 2)}} \]  

Where:

- \( A_e \) is the cross-sectional area of the exit of the nozzle \([m^2]\)
- \( M_e \) is the Mach number of the flow at the exit of the nozzle [-]

This equation cannot be solved analytically. Therefore the MATLAB’s “fsolve” function is used. This function needs a first estimate of the parameter to be solved, to be able to operate. If this estimation is too far from the value that needs to be found, there is the risk that the function finds other solutions. Therefore the exit Mach number was estimated to lie around \( M_e=4 \) [-]. This was later verified with the results. Assuming isentropic flow, the pressure ratio in the propellant flow, between the nozzle exit and nozzle throat is [35]:

\[ \frac{p_e}{p_t} = \left( 1 + \frac{\gamma_p - 1}{2} \cdot M_e^2 \right)^{\frac{\gamma_p}{2(\gamma_p - 1)}} \]  

Where:

- \( p_e \) is the pressure of the flow at the exit of the nozzle [Pa]
- \( p_t \) is the pressure in the flow at the nozzle throat [Pa]

Using previously defined parameters, the characteristic velocity is calculated using equation 3-20 [22]. This property reflects the energy level of the propellant, similar to the specific impulse. The
distinction is, that the characteristic velocity is independent of the pressure ratio between heating chamber and nozzle exit.

\[ c^* = \frac{1}{\Gamma_p} \cdot \sqrt{R \cdot T_{heat,exit}} \] (3-20)

Where
\[ c^* \quad \text{Characteristic velocity [m/s]} \]
\[ T_{heat,exit} \quad \text{The (desired) temperature at the exit of the heating chamber [K]} \]

The characteristic velocity is subsequently used in equation 3-21 [35], to calculate the specific impulse. This is done by taking the pressure ratio into account.

\[ l_{sp} = \frac{c^* \cdot \gamma_p}{g_0} \left( \frac{2}{\gamma_p - 1} \cdot \frac{2}{\gamma_p + 1} \right)^{\frac{\gamma_p - 1}{\gamma_p - 1}} \left( 1 - \frac{p_e}{p_t} \right)^{\frac{1}{2}} \] (3-21)

Where:
\[ l_{sp} \quad \text{Specific impulse [s]} \]
\[ g_0 \quad \text{Gravitational acceleration at standard sea level 9.81 [m/s}^2\text{]} \]

The definition of the specific impulse, see 3-22, is then used to calculate the exhaust velocity.

\[ U_e = l_{sp} \cdot g_0 \] (3-22)

Where:
\[ U_e \quad \text{Exhaust velocity [m/s]} \]

Since the flow in the throat is sonic, the pressure is equal to the critical pressure shown in equation 3-23 [22].

\[ p_t = p_{heat,exit} \left( \frac{2}{\gamma_p + 1} \right)^{\frac{\gamma_p - 1}{\gamma_p - 1}} \] (3-23)

Where:
\[ p_{heat,exit} \quad \text{The pressure at the end of the heating chamber [Pa]} \]

Using this throat pressure and the pressure ratio calculated with equation 3-19, the pressure in the exhaust flow at the nozzle exit can be determined. Using the now calculated parameters, the axial thrust force can be calculated using equation 3-24 [36]. In this equation, the ambient pressure is assumed to be vacuum i.e. 0 [Pa].

\[ F_{axial} = \dot{m}_p \cdot U_e + (p_e - p_a) \cdot A_e \] (3-24)

Where:
\[ F_{axial} \quad \text{The axial thrust force [N]} \]
\[ p_a \quad \text{The ambient pressure [Pa]} \]
As a last step the temperature at the nozzle exit has to be assessed. The temperature is needed in order to check for the possibility of condensation or freezing of the expellant to the nozzle. Incompletely vaporised has shown to be able to dramatically decrease the efficiency of the rocket engine. If the expellant temperature is lower than the boiling temperature at the exit pressure, condensation or solidification might have occurred.

3.4.2 True nozzle performance prediction

Due to the assumptions made for ideal rocket theory, the performance calculations do not include all effects that take place in reality. The many slight deviations from the ideal assumptions have varying effects on the actual performance of a nozzle. In order to anticipate the performance deviation from the ideal case, some flow properties first need to be defined.

The Reynolds number based on tube diameter, is usually defined as shown in the previously stated equation 3-4. The kinematic viscosity is calculated with equation 3-3. The flow velocity is calculated with equation 3-2. The area of a circle is \( A = \frac{1}{4} \pi \cdot D^2 \). Using these equations, the formula to calculate the Reynolds number, shown in equation 3-4, can be rewritten as following.

\[
Re = \frac{\dot{m}}{\rho \cdot \left( \frac{1}{4} \pi \cdot D_h^2 \right) \left( \frac{\mu}{\rho} \right)} = \frac{4 \cdot \dot{m}}{\pi \cdot D_h \cdot \mu}
\]

In the case of non-circular tubing, the equivalent hydraulic diameter \( D_h \) has to be calculated. The hydraulic diameter \( D_h \) is a measure to relate the area of any cross-section to the diameter of a circular cross-section with the same area. This parameter is defined as follows:

\[
D_h = \frac{4A}{\rho}
\]

Where:
- \( A \) Cross-sectional area of the tube [m\(^2\)]
- \( \rho \) Perimeter or inner wall length of the cross-section of the tube [m]

Federico La Torre has thoroughly investigated the flow behaviour through micro-nozzles [37]. For the calculations in this paper a flow of gaseous nitrogen (N\(_2\)), at 300 [K] and 10 [bar], in the inlet is assumed. It was found that nozzles with a thrust of larger than \( \sim 1 \) [mN], rarefaction and wall slip effects can be neglected. This assumption can be made due to the very low value of de Knudsen number. The Knudsen number is the ratio between molecular mean free path length and the characteristic length scale, of a problem in physics. In the case of gas flowing through a channel, this is used to assess if gas continuum hypotheses are valid. The characteristic length in this case is therefore the hydraulic diameter \( D_h \) [31], as shown in equation 3-27. When this Knudsen number rises above 0.01 [-], the molecular density is too low for traditional ideal continuum flow equations. The flow is then called rarefied, and special flow property calculation methodologies apply.

\[
Kn = \frac{\lambda}{D_h}
\]
Where:

\[ \text{Kn} \quad \text{Knudsen number [-]} \]
\[ \lambda \quad \text{Molecular mean free path between collisions [m]} \]

The mean free path of the gas molecules in the flow can be calculated using equation 3-28 [31].

\[
\lambda = \mu \sqrt{\frac{\pi}{2 \cdot \rho \cdot \rho}} \tag{3-28}
\]

In the research by la Torre the highest encountered Reynolds number was \(10^5\) [-], at a thrust of 1 [N] [37]. At this Reynolds number the difference between laminar flow property approximations, and models using turbulent effects, was less than 1.5%. Therefore it was concluded that the flow in all investigated nozzles was laminar. Probably because, due to the short nozzle length, the turbulence had neither time nor space to develop. In this study by la Torre throat diameters between 0.94 [μm] and 860 [mm] were modelled, with a thrust of 1 [μN] to 1 [MN] respectively. As will be shown later, the nozzles developed for the Dondersteen thruster fall well within this range.

In the throat of the nozzle, there can be a sharp angle between the converging and diverging sections, as seen in Figure 3.1. This throat angle can also be rounded off with a radius of \(R_c\), see Figure 3.2 below. This rounding is applied while retaining the same throat diameter \(R_t\). The radius of this throat edge chamfer will have an effect on the thrust efficiency of the nozzle, as can be seen in Table C.1 in Appendix C.

![Figure 3.2](image)

Figure 3.2 - Longitudinal cross-section of a conical nozzle, including its geometrical parameters [37]

As will be shown in section Fout! Verwijzingsbron niet gevonden., the thrust of the nozzles described in Table C.1 is very close to the thrust that the Dondersteen thruster is designed for. Therefore the thrust efficiency behaviour is assumed to be indicative of the behaviour expected from the Dondersteen thruster. It can therefore be concluded that it is most optimal to have a sharp edge, between the converging and diverging section of the nozzle. This means that the curvature radius \(R_c = 0\) [m].

In a study to optimise the combination of diverging and converging half angles of micro-nozzles, many different combinations were attempted [31]. The most relevant resulting data is summarised in Table C.2. In this study the combination of \(\theta_1 = 15\) [°] and \(\theta_2 = 20\) [°] yields the highest efficiency. With this ratio, the converging angle is relatively low, which generally has a value in the region of 30 [°] [22]. The diverging half angle is relatively high, compared to range of 12 [°] to 18 [°], normally considered optimal. When deviating from the conical shape, the long nozzle length due to these low angles can be decreased. The length of the nozzle can be reduced to about 75% of the conical length, while having the same area ratio and thrust coefficient \(C_F\), using a bell shaped nozzle [22].
With most MEMS manufacturing methods, it is difficult to obtain walls that are smooth compared to the small overall dimensions of the system. Effects of a wavy nozzle surface roughness with an amplitude of 4% - 8% of the throat diameter, perpendicular to the flow direction was investigated [37]. This lead to the formation of weak shocks in the divergent area of the nozzle. These shocks induce performance reductions up to 18%. Increasing the surface roughness even more, will lead to stronger shocks, further decreasing the nozzle performance. Roughness waves parallel to the flow direction, were on the other hand not found to induce a significant nozzle performance decrease.

The etching process used for the production of the Dondersteen micro-thruster, generally has a sub-micrometer dimensional accuracy. The throat diameter itself is in the order of 25 \(\mu\text{m}\), as will be shown in section 7.5 of this report. This could therefore cause a maximum surface roughness with an amplitude of 4% of the throat diameter. It is therefore possible, that some weak shocks waves are formed in the nozzle area. However the roughness is relatively uniform, in contrary to the very pronounced wavy surface roughness in the simulations by la Torre. And since the roughness is also generally lower, than the maximum of 1 \(\mu\text{m}\), the losses due to this roughness are expected to be limited.

The general efficiency that MEMS nozzles can achieve can be seen in the results of the research by la Torre [37]. As seen in the data in Appendix C, the nozzle efficiency can easily lie in the region of 90%. This efficiency indicates the ratio between ideal theoretical thrust of the nozzle, and the thrust which is actually delivered. This is also called the nozzle quality. The nozzle efficiency is therefore assumed to be 90% for these calculations.

The conclusions derived from the above mentioned geometrically dependent nozzle efficiency, will be described in section 7.5.

### 3.5 Calculation of design parameter to meet the requirements

In this section all the preceding calculations and assumptions are used to produce the top level design values of the propulsion system. In Table 3.2 below, all the assumptions made in these calculations are summarised.

**Table 3.2 - The assumptions made in the propulsion system performance calculations**

<table>
<thead>
<tr>
<th>Parameter</th>
<th>Assumption</th>
</tr>
</thead>
<tbody>
<tr>
<td>System temperature</td>
<td>293.15 [K]</td>
</tr>
<tr>
<td>MEMS thruster flow channel height</td>
<td>100 [\mu\text{m}]</td>
</tr>
<tr>
<td>Total satellite mass</td>
<td>3.0 [kg]</td>
</tr>
<tr>
<td>Pressurant gas</td>
<td>Nitrogen</td>
</tr>
<tr>
<td>Percentage of electrical input power lost to the surroundings while heating</td>
<td>50%</td>
</tr>
<tr>
<td>Pressure loss in the heating chamber</td>
<td>20%</td>
</tr>
<tr>
<td>Expulsion efficiency</td>
<td>90%</td>
</tr>
<tr>
<td>Nozzle efficiency/quality</td>
<td>90%</td>
</tr>
</tbody>
</table>
Since all calculations have been translated to a Matlab file, the influence of different parameters on the system performance can be evaluated. Parameters under investigation are:

- Initial pressurant mass
- Total storage volume
- Initial propellant mass
- Throat diameter
- Final vaporised propellant temperature
- Area ratio

The total system performance characteristics of interest are:

- Total ΔV
- Thrust
- Pressurant pressure
- Electrical power consumption
- Knudsen number in the nozzle throat
- Reynolds number in the nozzle throat

3.5.1 Calculation stability analysis

First of all, the stability of the calculations has to be assessed. In order to obtain the total impulse or ΔV of the propulsion system, a numerical integration is executed over the time it takes to use all propellant. This emptying of the propellant tank is therefore done in a number of discrete steps. It needs to be assessed which number of emptying steps is the minimum for a stable, repeatable and reliable integration. In agreement with Mr. Krusharev, and following simple calculations, the initial propellant mass has been set to 50 [g]. In the calculations to view the effect of initial the number of integration steps, the following design choices have been made: initial pressurant mass of 0.3 [g], total storage volume of 100 [ml], throat diameter of 25 [μm], final vaporised propellant temperature 500 [°C], area ratio of 20 [-]. The effect of the number of integration steps, on the resulting total ΔV after the calculations, is shown in Figure 3.3.
The deviation between calculations at a low number of steps is relatively large. The behaviour stabilises relatively quickly around 100 steps. When changing the design variables, this stabilisation duration will be different. Therefore a minimal of 300 iteration steps are chosen for a reliable output of the calculations. In this case the calculated total $\Delta V$ only fluctuates with 0.24% at 300 steps. This will therefore not further decrease the inaccuracies of the calculations.

### 3.5.2 Effect of initial pressurant mass variations

For the calculations to show the effect of initial pressurant mass the following design choices have been made: total storage volume of 100 [ml], throat diameter of 25 [$\mu$m], final vaporised propellant temperature 500 [°C], area ratio of 20 [-]. The resulting performance parameters for a relevant range of the initial pressurant mass are shown in Figure 3.4.
These graphs show the effect that the total \( \Delta V \) of the system is only limited above 0.1 \( [g] \) of initial pressurant mass. This curve seems to oscillate, which is an artefact of the accuracy of the measurements. The oscillation has an amplitude of less than 0.5\%, which is acceptable. Due to the higher mass flow, the thrust and power consumption do increase with increasing pressurant mass. It can therefore be concluded that the pressurant mass has to be chosen, such that the force and \( \Delta V \) requirements are met, for the entire operational range. Further increasing the pressurant mass will lead to a larger power consumption and larger pressures.

3.5.3 Effect of nozzle throat diameter variations
In the calculations to study the effect of the nozzle throat diameter, the following design choices have been made: Initial pressurant mass of 0.3 \( [g] \), Total storage volume of 100 \( [ml] \), final vaporised propellant temperature 500 \( [^\circ C] \), Area ratio of 20 [-]. The graphs of the relevant parameters are shown in Figure 3.5.
As is to be expected, the total $\Delta V$ remains constant with increasing throat diameter and is therefore not shown. Due to the increased critical mass flow rate, the power and thrust increase linearly. The nozzle throat will therefore be sized to provide the necessary thrust level throughout the service range. The diameter will not be increased much further, in order to maintain an acceptable power consumption.

### 3.5.4 Effect of final propellant temperature variations

In the calculations to view the effect of the final vaporised propellant temperature, the following design choices have been made: Initial pressurant mass of 0.3 [g], Total storage volume of 100 [ml], nozzle throat diameter of 25 [μm] and an area ratio of 20 [-]. The results of the calculation of the behaviour of the relevant performance parameters are shown in Figure 3.6.
As the temperature increases, the critical mass flow rate decreases. This decreased mass flow rate results in a slightly lower power consumption. The power consumption increases again at higher temperatures. This is due to the increased prominence of the propellant heat capacity to heat it to higher temperatures, compared to the enthalpy of vaporisation, which dominates the power consumption at lower temperatures. This change in prominence is shown in the bottom right hand corner of Figure 3.6. This graph also shows that the absolute electrical power consumption variation is nearly negligible. The maximum thrust force with increasing propellant temperature is constant and therefore not shown. Due to the increased exhaust velocity, and thus specific impulse of the thruster, the total $\Delta V$ increases with increasing temperature. It is therefore desirable to have the highest final temperature possible, in order to maximise the $\Delta V$. Due to the restrictions of the heater material and the surroundings the maximum propellant temperature has been set to 500 $[^\circ C]$, as will be discussed in chapter 7.

### 3.5.5 Effect of nozzle area ratio variations

In the study to view the effect of the nozzle area ratio, the following design choices have been made: Initial pressurant mass of 0.3 [g], Total storage volume of 100 [ml], nozzle throat diameter of 25 [μm] and an final vaporised propellant temperature of 500 $[^\circ C]$. The result of this nozzle area ratio variation on the affected performance parameters can be seen in Figure 3.7.
The mass flow rate and power consumptions remain constant with increasing nozzle area ratios and are therefore not shown. The thrust however increases due to the increased exhaust velocity, and therefore so does the total $\Delta V$. This increase in thrust force with increasing area converges quite rapidly to the theoretical maximum. The theoretical maximum thrust is the condition that the nozzle exit pressure equals the ambient pressure, which in this case is vacuum i.e. 0 [Pa]. At higher area ratios, there is only a slight thrust force increase with an increase in the area ratio. This increase of the area ratio will however lead to an increase in the nozzle length. It will also lead to an even greater increase in the nozzle wall area, which increases the heat loss, and decreases efficiency. An area ratio of 10 can be seen to more than double the total performance of the system and is without a doubt beneficial. Above an area ratio of 20, the increased theoretical performance no longer weighs up to the mentioned disadvantages. Therefore an area ratio of 20 is selected for the thrusters developed during this thesis.

### 3.5.6 Final selected system performance

By using the results of the previous subsections, the final design has been chosen. This was done by balancing all the parameters, to yield a result that can meet all requirements. The final design parameters are presented in Table 3.3.

<table>
<thead>
<tr>
<th>Parameter</th>
<th>Chosen final value</th>
</tr>
</thead>
<tbody>
<tr>
<td>Initial pressurant mass</td>
<td>0.2 [g]</td>
</tr>
<tr>
<td>Total storage volume</td>
<td>100 [ml]</td>
</tr>
<tr>
<td>Initial propellant mass</td>
<td>50 [g]</td>
</tr>
<tr>
<td>Throat diameter</td>
<td>25 [μm]</td>
</tr>
<tr>
<td>Final vaporised propellant temperature</td>
<td>500 [°C]</td>
</tr>
<tr>
<td>Area ratio</td>
<td>20 [-]</td>
</tr>
</tbody>
</table>

The calculations mentioned earlier in this chapter have been performed with these design values. The resulting magnitude of the relevant performance parameters, for the time it takes to empty the propellant tank, is shown in Figure 3.8.
The final total velocity increase or ΔV supplied by the propulsion system for these design choices is 21.01 [m/s]. The total operational time of the system is 64,573 [s], i.e. 17 [h] and 56 [min]. For the requirement of 5000 on-off cycles, this leads to a thrust time of 13 [s] per cycle. In terms of energy consumption and orbital availability that is a very acceptable duration. The pressure, electrical power consumption, thrust force and total ΔV abide to the requirements mentioned in chapter 2. These calculations already include relatively conservative efficiency for all components. The actual performance of the thruster is therefore not expected to be much lower than what is seen here. The Reynolds number indicates that, according to the work by la Torre [37], the flow through the nozzle will be laminar. The Knudsen number is far below the value for rarefied flows. Therefore the simplifications and assumptions, made to calculate the flow behaviour through the nozzle, hold for
this case. Therefore all design parameters combine into a propulsion system which performed as required.
4 Propellant storage system

During this thesis a preliminary design of the necessary propellant feed system was made, which is described in this chapter. In section 4.1, some of the propellant storage system design options will be discussed. The first purpose of the propellant storage system is, to durably and safely store the propellant, which will be described in section 4.2. Secondly it shall also apply a force to the propellant, in order to supply it to the thruster via the feed system. This will be described in sections 4.3 and 4.4. The feed system itself will be discussed in chapter 6.

4.1 Propellant storage geometry options

There are a number of, mainly geometric, decisions have to be made for the propellant storage tank. The most important decision is, with which mechanism the propellant is pressurised and fed from the tank to the feed system. In most ground based propellant feed systems, this is usually gravity assisted. The pressurant is separated from the propellant due to the gravity, which pulls the liquid propellant down towards the tank exit. The same method is used for the test setup of the engineering model of the thruster, which will be described in chapter 8. Due to the lack of gravity in orbit, the propellant has to be fed by some other mechanism. The following options are possible:

- Mixed propellant and pressurant
- Piston fed
- Bladder or membrane pressurisation
- Propellant heating
- Propeller or screw
- Rotating tank

These options will be discussed in the subsequent subsections. After which a preliminary selection is conducted in section 4.2.

4.1.1 Mixed propellant and pressurant

In this design no physical separation between propellant and pressurant is used in the propellant tank. This means that it arbitrarily feeds pressurant or propellant to the valve. This design has the risk of losing part, or all, of the pressurant, before the propellant is completely expelled. This leads to a loss in feed pressure and thus a loss in eventual thrust force. Probably some sort of a propellant collection device must be in place, to prevent too much of the gas from escaping before the water is completely expelled. This can be a sponge or porous metal near the tank exit. These materials capture the water when the movement of the structure causes the water to move past the collector. The water is then held into place by capillary forces, filling the porous structure of the collector. When the valve is opened, the pressurant first has to force the propellant out of the cavities in the collector. If the valve is closed before the propellant in the collector is exhausted, none of the pressurant will be lost in the process. Before the next opening of the valve, the collector can saturate itself with propellant again.

4.1.2 Piston fed

In this design option a piston or plunger inside a cylinder, called the barrel, forces the propellant towards the valve propellant tank exit. It therefore resembles a syringe in operation. The most
important design choice in such a “syringe” propellant tank, is how to exert pressure on the plunger. This can be done using a pressurant gas, as will be shown in subsection 4.1.7. Other methods include using a spring or a (piezo) electric actuator. Linear electrical drives come in many shapes and sizes. One example of such an actuator is shown in Figure 4.1. This actuator has a diameter of 16 [mm] to 40 [mm] and a stroke of 50 [mm] to 200 [mm]. It therefore fits the geometrical constraints of a CubeSat.

There are some difficulties with a piston fed system. Firstly there is a risk of leakage at the moving seal on top of the plunger. The system complexity and risk will be increased due to a number of moving parts and possible electronic drive circuits. An optional spring loaded system would impose pre-pressurisation problems on the system. This storage system would therefore necessitate elaborate design and testing activities. For this reason, this is not a preferred propellant storage system option.

4.1.3 Bladder or membrane pressurisation
In this design the propellant is separated from the pressurant using a bladder or a membrane. The bladder/membrane is pressurised with pressurant gas on the outside, thereby pressurising the propellant inside and forcing it through the exit opening. This confinement of the propellant is also beneficial in the reduction of propellant sloshing, which could otherwise induce disturbances on the satellite. The bladder/membrane prevents the pressurant from escaping, thereby the maximum possible pressure is maintained, up until all of the propellant is expelled. To thereafter expel the pressurant, extra features such as an extra valve on the pressurant side will be required. Due to the far lower mass of the pressurant, the expulsion of the pressurant will not weigh up to the added mass of the additional feed system and is therefore not necessary. Due to its simplicity and therefore reliability, this is a preferred option for the DelFFi mission propulsion system.

4.1.4 Propellant heating
By heating the propellant, the vapour pressure will increase first, which will increase the pressure inside the storage vessel. At a certain temperature the propellant will start boiling and become gaseous, which will increase the pressure due to the constant volume. In this method it would be beneficial to have the gaseous propellant escape the propellant tank. Because then there will be no problem with incomplete vaporisation in the heating chamber of the thruster.

The problem with such a system is, that it costs a large amount of energy to heat the entire propellant mass. Especially since it has to be heated to a temperature where the vapour pressure is sufficient, for the required propulsion system pressure. Due to the relatively large size of the propellant tank, there will be large heat losses to the surroundings. An additional problem is how to separate the liquid and the gaseous propellant. If a liquid/gaseous mix is supplied to the thruster, the
performance of the system will fluctuate. It is thus very clear that this design option is not possible for the DelFFi propulsion system.

4.1.5 Propeller or screw
In this design there is a liquid and gas mix inside the propellant storage chamber, with a propeller or screw inducing a flow in direction of the valve opening. The heavier liquid water would thereby be forced more towards the walls, and thus the valve opening, while the gas circles in the centre of the tank. The propeller could also function as a centrifugal pump, whereby the whole storage volume can be reached. Due to the necessary power consumption and moving parts, this design would increase the complexity and risk of the propulsion system. The constantly rotating parts would also introduce a mechanical moment to the structure. This in turn would have to be counteracted by the attitude control subsystem. Such a system would also require tests under weightless conditions, which cannot be conducted at the faculty. Due to these problems, a propeller or screw actuated propellant storage and pressurisation system is not an option for the DelFFi propulsion system.

4.1.6 Rotating tank
When a filled propellant tank is rotating it will force the heavier liquid propellant to the sides of the tank, where one or multiple collection holes will lead the propellant to the valve. Due to the lack of gravity this rotating tank requires some radial fins, in order to guide the propellant to the sides. It therefore uses the same operational principle as a centrifugal pump. This type of system has all the same issues as mentioned for the propeller or screw propellant storage system, described in subsection 4.1.5. These problems will be amplified due to the fact that the whole tank is rotating, instead of only a propeller. In addition to this, it will be difficult to connect the propellant collection and filling connection and the necessary sensors, to a rotating tank. This design option is therefore clearly not suitable for the DelFFi propulsion system.

4.1.7 Commercial-off-the-shelf (COTS) solution
A COTS solution for a propellant storage and pressurisation system is the SKF single point automatic lubricator, shown in Figure 4.2.

![Figure 4.2- A cross-sectional view of the SKF single point automatic lubricator, electrolysis powered lubricant feed system](image)

A. Time setting slot
Allows easy installation and accurate adjustment of lubrication flow

B. Gas cell
Generates pressure to enable lubricant dispensing

C. Easy grip top cover
Facilitates easy and quick fitting

D. Piston
Special piston shape helps ensure optimum emptying of lubricator

E. Lubricant container
Transparent lubricant container allows visual inspection of dispense rate

F. SKF Lubricant
Filled with a wide range of high quality SKF lubricants

[39]
The single point automatic lubricator is a plastic cylinder with a piston which drives the lubricant out. The piston is in its turn driven by pressurised hydrogen, which is dynamically produced using electrolysis. This process is self containing using a coin cell battery. The system is designed to contain the hydrogen (H₂), which is the smallest molecule, for up to a year of operation. It should therefore have no trouble containing the water without leakages. The properties of this automatic lubricator connect very well to the needs of the DelFFi satellite propulsion system [40]:

- Grease (propellant) capacity: 60 [ml] or 125 [ml]
- Emptying time: 1-12 months
- Temperature range: -20[°C] to 60 [°C]
- Maximum pressure (BOL): 5 [bar]
- Wet mass: 130 [g] or 200 [g]
- Price: €25 to €35

Assuming a grease density of 1 [kg/l], the dry mass is around 70 [g] or 75 [g] depending on the capacity. That fits within the DelFFi propulsion system mass budget. Due to the relatively low price compared to normal spacecraft hardware, it is easy to acquire one lubricator to perform tests with. When taking it into consideration for the real flight hardware, SKF has to be contacted if they are willing to cooperate. They can then be asked if it is possible to supply the part without the grease inside. The following important preliminary tests have to be performed:

- Leak tightness and operational characteristics when used in combination with water
- The possibility of interfacing with a feed system
- The possibility of actuation in space (the lubricator is meant to be manually actuated and controlled)
- The behaviour under discontinuous outflow. Because the lubricator is meant to produce a constant outflow of grease, and the thruster operates in pulses.

It did not fit within the timeframe of this thesis to contact SKF for possibilities of cooperation, or to test the part for applicability as a propellant feed system.
4.2 Propellant storage

The most important requirements that can be identified for the propellant storage tank are:

- It shall not react with:
  - The pressurant
  - The propellant
  - The manufacturing and storage environment
- It shall withstand the anticipated pressure loads and environmental loads.
  - Possible shock loads of pressurant initiation
  - Launch loads
- It shall withstand temperatures between -20 [°C] and 80[°C].
- The weight shall be less than 100 [g].
- The pressurisation or thermal cycles expansion of the volume shall stay within the geometrical constraints of the propulsion system.

Due its simplicity and reliability, the propellant tank is chosen to be a bladder containing 50 [mg] of water. This bladder separates the water inside from the pressurant outside the bladder. For the bladder mass estimation a rubber bulb, usually used in medical applications, is used as an example [41]. This has a diameter of 54.5 [mm], which gives a slightly larger volume than necessary. The mass of this bladder is 25 [g] - 35 [g] and the price is €0.74 in large volumes. A more reasonable price for smaller numbers is probably €6.- [42]. The bladder is surrounded by a spherical stainless steel tank, with an internal volume of 100 [ml], which contains the pressurant and the propellant filled bladder.

![Figure 4.3 - A rubber bulb, normally used for medical applications, envisioned to be a possible propellant storage bladder [42].](image)

The optimal shape for an internally or externally pressurised vessel is a sphere. This shape will result in the smallest wall thickness for a vessel that can sustain a certain pressure, and therefore the lowest mass. The thickness that a spherical vessel shall have in order to contain the design pressure, will therefore serve as a benchmark for the necessary storage system wall thickness. The stress in the wall of a thin-walled spherical pressure vessel is [43]:

$$\sigma = \frac{p \cdot r}{2t}$$

Where:

- $\sigma$ Stress in the wall [Pa]
- $p$ Internal pressure [Pa]

Design, manufacturing and characterisation of a water fed CubeSat micro-resistojet - 28-07-2014

65
\[ r \quad \text{Radius [m]} \]
\[ t \quad \text{Wall thickness [m]} \]

The mass of a thin-walled spherical pressure vessel is approximated by:

\[ m = A \cdot t \cdot \rho = 4\pi \cdot r^2 \cdot t \cdot \rho \quad \text{(4-2)} \]

Where:
- \( m \): Mass of the pressure vessel [kg]
- \( A \): Surface area of the vessel [m]
- \( \rho \): Density of the vessel material [kg/m³]

Combining equation 4-1 and 4-2, the minimum mass required for a spherical pressure vessel, in order for it not to yield under the necessary overpressure is:

\[ m = \frac{2\pi \cdot r^3 \cdot \rho \cdot \rho}{\sigma_y} \quad \text{(4-3)} \]

Where:
- \( \sigma_y \): Yield stress of the vessel material [Pa]

Since the radius \( r \) is dependent on the necessary internal capacity, this value is fixed. The internal pressure \( p \) is also already determined, in order to have the right performance of the propulsion system. According to the theory of material selection by Ashby [43], the material index that needs to be optimised for this problem will therefore be \( \sigma_y / \rho \). The higher the value of this material index is, the better the given material suits the design of the propellant tank. The graphical method for finding the best material for the propellant storage tank is shown in Figure 4.4.

**Figure 4.4** - Material selection chart for the optimisation of the ratio between density and strength. The farther the materials in the search region are from the dotted line, the better they would serve as propellant tank materials [43].
In Figure 4.4, the region above the blue dotted line is where the materials have the highest material index. The best materials for the propellant tank would either be Carbon Fibre Reinforced Polymer (CFRP) or high strength ceramics. Due to the manufacturing process of high strength ceramics, it would be very difficult and time-consuming, i.e. expensive, to make the propellant tank out of this material. At the chair of Novel Aerospace Materials (NovAM), in the aerospace faculty of the TU Delft, there is an active and renowned specialisation in composite structures for aerospace products. It would therefore be very interesting to discuss if it is possible to make such a small propellant tank out of CFRP and how this could be manufactured. NovAM might even be interested in cooperating on the development and manufacturing of this tank, in order to advance and test their own capabilities with the materials.

Other options are the standard high performance aerospace metals, magnesium and titanium alloys. These are however difficult and expensive to produce. The price of production of these metals has to be weighed up to the reduced cost, and only limited reduced performance, when using high strength aluminium alloys. Therefore if there is a high strength aluminium, which can be manufactured to the necessary design, that would be the preferred material.

A very important consideration in this propellant tank design, is that H2O propellant freezing must be taken into account. Especially during storage, launch and initial operational mode of the satellite, when the thermal control subsystem is not working completely yet. Water has a relatively high melting point of 273.15 [K] and the ambient temperature during these phases easily drops below this temperature. When containing the liquid propellant in the flexible bladder, this problem is mitigated. This is because the bladder has the capacity to expand and contract when necessary.

Up to this point, it was assumed that the propellant storage vessel will be spherical. From a manufacturing and integration point of view, this is however not the most optimal shape. Due to the stacked squares build-up of a CubeSat, the most logical shape would be a square based prism. The preliminary design, worked out during this thesis, features a five sided body of the tank. This body is closed off with a flat lid, which is fastened by a number of screws. The envisioned body and lid can be seen in Figure 4.5 and Figure 4.6 respectively. The leak tightness of this design is guaranteed by equipping the body of the tank with a groove, along the whole interface plane with the lid. This groove is not shown in Figure 4.5. This groove can then accommodate an o-ring. This o-ring seals the gap between the body and lid, when both are pressed against each other by the fastening screws.
In the tank lid in Figure 4.6, an inner row of holes is visible for attachment to the tank body. Four other holes in the corners of the lid are present to fasten the propellant tank to the internal mounting rods of the satellite. Smaller holes are present to accommodate the electrical feed through for the pressure sensor. There are also two threaded cylinders positioned on top of the lid. The central one is the interface for connecting the propellant storage bladder on the one side, and the feed system on the other. The off centre cylinder is the interface for the pressurant fill valve.

If calculations prove that this structure is not capable of containing the required pressure, it can be adapted to feature support posts. An example of such a support post in a simple propellant tank design is shown in Figure 4.7. This has to be compatible with the parts which separate the propellant and pressurant. In case of a propellant storage bladder, this is more difficult. Alternatively, the propellant tank can also be made cylindrical. A cylindrical pressure vessel is a good compromise between rigidity under internal pressure and easy CubeSat integration.

Figure 4.5 - Preliminary design of the square prism shaped propellant tank body

Figure 4.6 - Preliminary design of the square prism shaped propellant tank lid

Figure 4.7 - JPL Micro-Inspector Multifunctional Tank Design featuring Liquid and Plenum Tank [44]
4.3 Requirements on the pressurisation system

The pressure supply is inextricably bound to the propellant storage tank choice. A trade-off between having an optimal tank design and a functional pressure supply has been made. The following options are possible for pressure supplies:

- No pressure supply
- A mechanical pump
- A pressurant gas
- A spring

To be applicable to the DelFFi propulsion system, the pressurisation system must abide by the following requirements:

1. Any chemical reaction and resulting heat shall not damage the rest of the propulsion system.
2. The pressurant shall not corrode the storage system.
3. The pressure variation due to volume change must be well characterised. This is needed in order to know the pressure variation during the emptying of the propulsion system.
4. Possible residue of the pressurant system shall not hinder the operation of the propellant feed system.
5. Possible electrical power consumption for initiation shall not be higher than 9 [W], for no more than 10 [min] per orbit. (This takes the propulsion system maximum of 10 [W] into account and leaves some room for the most vital electronics on the PCB)
6. The mass shall not exceed 75 [g].
7. It shall be able to withstand the maximum acceptable launch loads.
8. It shall be able to operate throughout the operational temperature range of -20 [°C] to 80 [°C].
9. A possible initial shock pressure shall not damage the surrounding structure, to a point where it will hinder its operation.
10. The pressure shall be kept between the minimum and maximum design pressures throughout the operational temperature range.
11. The volume of the pressurant and its support structure shall be smaller than 100 [ml].
12. The pressurisation system shall work on 5 [V] or it shall include a voltage converter.

In section 4.4, these requirements will be applied to the design options, in order to yield a final design choice.

4.4 Pressurant selection

One of the most conventional pressurisation methods is the use of a pressurant gas. Cold gas propulsion systems serve as examples of such a pressurisation source, since they are solely based on pressurised gas. Valve leakage is one of the most critical issues of cold gas thrusters. Loss rates of 10% of the propellant throughout a mission are not unusual [45]. This is due to the subjection to high pressures. The leakage risk increases when valves have undergone many cycles, because of the increased likelihood of contaminant deposition on the valve seat [44]. During the long storage period, the launch procedure and early operations of the satellite this leakage could pose a serious performance risk.
In addition to this risk, the CubeSat requirement that pressurised vessels are only allowed up to 1.2 [bar] during launch [9], was still enforced at the beginning of this thesis. Therefore the cold gas generators designed by TNO for the T³μPS were envisioned to be used to pressurise the system in orbit. These generators produce cold gas in a controlled gradual way, without significant shock waves. TNO was the developer and producer of these gas generators. In the mean while, a spin-off company called CGG technologies, has been set up to bring the development to the market. The Chief Technical Officer of the company, ir. Berry Sanders, has been contacted for collaboration. This has however not lead to a possible agreement yet. Therefore alternative methods had to be found.

4.4.1 Commercially available gas generators

Gas generators for airbags generally use nontoxic, clean-burning, low temperature, high reliability and mass produced solid propellants. All these properties make them very applicable for micro satellite space use. Alkali azides (\(\text{NaN}_3\) or \(\text{KN}_3\)) propellants have been used a lot, because they decompose smoothly at 300 \(^{\circ}\text{C}\) [46].

Airbag inflators are generally quite large compared to the dimensions of a CubeSat. Cold gas curtain inflators from Key Safety Systems for example, are 30.0 [mm] to 33.3 [mm] in diameter and 100 [mm] to 328 [mm] in length [47]. They have a mass of 500 [g] to 700 [g] and produce 1.2 [mol] to 3.6 [mol] of helium. One of the smallest airbag inflators is used for side- and knee airbag applications. An example of such a gas generator from Key Safety Systems, has a diameter of 30 [mm] and a length of 160 [mm] [48]. It releases 0.92 [mol] of a gas mixture of 50% argon and 50% helium and has a mass of 260 [g]. These show that the gas generators generally produce too much gas, are too heavy and relatively large. There are however very small generators, which are generally used only to initiate the main stage or to actuate seatbelt pretensioners.

4.4.2 Automotive gas generator usage feasibility calculations

To calculate the feasibility of using small automotive gas generators, a representative generator was used for performance calculations. The pressure is supplied by gaseous nitrogen pressurant coming from a gas generator, which is initiated in orbit, when the pressurisation is required. The pressurant section of the spherical tank is not yet filled with pressurant during propulsion system integration. Therefore this volume will be taken by air at normal atmospheric pressure. Using the density of air at standard conditions of 1.225 [kg/m\(^3\)], the pressurant volume will contain 49[mg] of air. Since air is made up out of close to 80% nitrogen, it is assumed that it has approximately the same properties as nitrogen. The mass of the air is simply added to the mass of the pressurant for the calculations.

In order to estimate the mass generated by the gas generator, the “SPECIAL DEVICES INC” FGI (OCH) initiator, with a total output of 260 [mg] nitrogen [49], is used. This gas generator is depicted in Figure 4.8. The maximum diameter is 11.1 [mm] and the total length including the connection pins is 19.3 [mm]. The price of this gas generator is indicated to be $ 1.48 per generator for large quantities. A more reasonable price of € 25.- from Dutch suppliers can be assumed for smaller quantities.
When the generator is initiated, the gas is released in the pressurant chamber with a volume of 50 [ml], which will settle to room temperature (20 [°C]). Using the ideal gas law (equation 3-1) the system is calculated to have a final stabilised pressure of 4.5 [bar]. The energy consumption of this exact generator is not stated. Many times larger generators have however been seen to require around 6 [W] for up to 2 [s]. This smaller generator will therefore certainly not consume more power. This one-time initiation power consumption easily fits within the 10 [W] budget for the propulsion system.

Therefore it can be concluded that the automotive gas generators meet the pressurisation, size, mass and power consumption requirements for the DelFFi propulsion system. Therefore the research can proceed into a physical testing stage, to assess if the gas generators meet the other propulsion system requirements as well.

4.4.3 Airbag gas generator availability and testing

A local scrapyard was visited first to gather material for preliminary tests. They have many airbag modules in a crate at €25,- a piece. These airbag modules There are however the following problems with:

- The available models are mainly steering wheel airbags, which are too big and do not the right shape for CubeSat application.
- All modules are mixed up and badly searchable.
- All modules are different, so there is no reliability in reproducibility.
- Only entire modules are present, which have to first be taken apart. This is not desirable without professional supervision, since they contain explosives.

Therefore it was decided not to test any of the airbag modules that are available at the scrapyard.

There are a number of companies which manufacture airbag gas generators. One of them is Autoliv, which advertises with “micro gas generators”. These are designed to be used in seat belt pretentioners. E-mails were sent to both the French and Dutch representatives of Autoliv. The French representatives never responded to requests. The Dutch representative called “Van Oerle Alberton BV” provided contact details of R&D&E Director Global Webbing, Mr. S. Valkenburg. He warned about the many regulations the inflators impose on storage, handling and volume. Mr. Valkenburg then referred to Senior Director - Inflator Global Development Mr. T. Bertacini. Mr. Bertacini however
unfortunately never answered my request on the possibilities of obtaining the micro gas generators from Autoliv.

A local car dealer explained that only specialised companies are allowed to handle and sell airbag parts. This is partially due to the fact that they contain explosives, and partially because airbags are human safety systems. One such a company is Airbagbank.eu, which specialises in airbag manufacturing. This company was able to extensively discuss the different automotive gas generator possibilities. The most applicable gas generator type was then provided and tested. This whole procedure and its results are described in chapter 5. The conclusion is that the tested gas generators cannot be used on the DelFFi satellites propulsion system. The main reason for this is the extremely high shock pressure and the questionable reliability. It can therefore be concluded, that either pre-pressurisation or cooperation with “CGG technologies”, is needed to pressurise the propulsion system.
5 Automotive gas generator tests

This chapter specifies the results of the tests conducted on airbag initiator gas generators. The gas generators are under investigation to assess if they could be utilised to pressurise the liquid water propellant for the DelFFi propulsion system, as indicated in chapter 4. Due to the fact that the gas generators were acquired from a vendor, and not the manufacturer, the performance is not specified. Even if the performance would have been known, the performance would still need to be validated for propellant pressurisation. Some of the operational parameters, which are important to satellite use, are never specified by the suppliers. Before the tests, the performance was only estimated using data from similar gas generators. An explanation of the test item, and the need for it is explained in sections 5.1 and 5.2. The test setup and procedure are discussed in sections 5.3, 5.4 and 5.5, after which the test results are shown in section 5.6. The assumptions made for the preliminary calculations are verified and adjusted to fit the test results in section 5.7. This last chapter also includes calculations on the performance of the gas generators, when used for pressurisation of the DelFFi propulsion system. Unfortunately three out of the four obtained gas generators malfunctioned. Therefore not all necessary tests could be conducted and not all conclusions can be drawn.

5.1 Introduction of the tested gas generators

As specified in chapter 2, when this investigation was started pre-pressurisation before launch is forbidden in the standard cubesat requirements. Therefore in-orbit pressurisation is required. At this moment it is clear that the use of pre-pressurisation is allowed for this mission. One of the options for this pressurisation is a gas generator. After a survey of the generally available gas generators, the automotive industry seems to use gas generators yielding the right amount of gas. These gas generators have an appropriate size and mass, are produced in very high quantities and are therefore cheap and reliable. Four such gas generators were obtained from the airbag manufacturer Airbagbank.eu, for €25.- each. These have been used to test their performance for possible usage in satellite pressurisation. Airbagbank specialist E. van Dooren specified that all these gas generators are very similar. The obtained gas generators were said to be very representative for the average behaviour of the product group. For electrical actuation these gas generators require a specialised plug which costs €22.50- each. To contain the gas generator, a standardised casing was provided, to interface the gas generator with the test facilities.

The obtained hot gas generators that have been characterised are shown in Figure 5.1. These gas generators are simple metal casings containing a solid chemical explosive. This solid explosive is ignited using an imbedded resistive heater element, which is connected to a DC power supply with two electrical pins. Due to the simple nature of the heating element both pins can be connected to either the positive or the negative supply terminal. After ignition the intentionally weakened top of the gas generator will rupture, releasing the hot gas.
A technical drawing of the gas generator, including approximate relevant dimensions in [mm] can be seen in Figure D.1 in Appendix D. The cross-section of a gas generator after the explosion is shown in Figure 5.2 below.

In Figure 5.2 the following internal sections within the gas generator can clearly be distinguished:

1. Electrical connection pin
2. Electrical and thermal insulation
3. Resistive heating element
4. Primary charge cavity
5. Primary charge metal casing
6. Primary charge coating
7. Secondary/main charge cavity
8. Secondary/main charge metal casing
9. Burst section
As can be seen from the cross-section in Figure 5.2 the gas generator works in the following stages:

- The resistive heating element is heated using electrical power.
- The heat activates the primary charge and produces hot gas.
- The hot gas breaks through the coating through two holes in the primary metal casing.
- The hot gas ignites the secondary, or main, charge.
- The main charge reaction produces hot pressurised gas which breaks through the burst section of the outside casing.

In the automotive industry these gas generators are used for either seatbelt pre-tensioners or for airbag initiators. For a seatbelt pre-tensioner, the hot expanding gas is used to drive a piston inside a cylinder. This piston will drive a steel cable fastened to the seat belt; in effect tightening it around the user. When used as an airbag initiator, the hot gas is used to activate a second stage. Either by rupturing a tank filled with pressurised gas, or to ignite a larger solid explosive charge. Mainly the gas produced by these second stages will fill the airbags, whereas the smaller gas generator is only used for initiation.

5.2 Requirement verification methodology

Not all these requirements on the DelFFi pressurisation system, mentioned in section 4.3, could have and have been checked in the test procedure. But the relevant performance had to be characterised, in order verify the compliance to final design requirements. The test procedure includes a preparation phase and the detonation of all four acquired gas generators. The information needed for verification of different requirements was intended to be gathered across different tests. The requirement information that was intended to be gathered in each test is shown in Table 5.1 below. The reason behind the mentioned information gathering, will be explained in more detail in the description of the test setup and procedure, in section 5.3 and 5.4.

Table 5.1 - Intended requirement compliance information gathering

<table>
<thead>
<tr>
<th>Requirement</th>
<th>Preparation</th>
<th>GG 1</th>
<th>GG 2</th>
<th>GG 3</th>
<th>GG 4</th>
</tr>
</thead>
<tbody>
<tr>
<td>1 - Combustion temperature</td>
<td>Yes</td>
<td>Yes</td>
<td>Yes</td>
<td>Yes</td>
<td>Yes</td>
</tr>
<tr>
<td>2 - Corrosion</td>
<td>Yes</td>
<td>Yes</td>
<td>Yes</td>
<td>Yes</td>
<td>Yes</td>
</tr>
<tr>
<td>3 - Volume change</td>
<td>Yes</td>
<td>Yes</td>
<td>Optionally</td>
<td>Optionally</td>
<td></td>
</tr>
<tr>
<td>4 - residue</td>
<td>Yes</td>
<td>Yes</td>
<td>Yes</td>
<td>Yes</td>
<td>Yes</td>
</tr>
<tr>
<td>5 - Power</td>
<td>Yes</td>
<td>Yes</td>
<td>Yes</td>
<td>Yes</td>
<td>Yes</td>
</tr>
<tr>
<td>6 - Mass</td>
<td>Partially</td>
<td></td>
<td></td>
<td></td>
<td></td>
</tr>
<tr>
<td>7 - Launch loads</td>
<td></td>
<td>Partially</td>
<td>Partially</td>
<td>partially</td>
<td>partially</td>
</tr>
<tr>
<td>8 - Operational temperature</td>
<td>Partially</td>
<td>Partially</td>
<td>partially</td>
<td>partially</td>
<td>partially</td>
</tr>
<tr>
<td>9 - Shock pressure</td>
<td>Yes</td>
<td>Yes</td>
<td>Yes</td>
<td>Yes</td>
<td>Yes</td>
</tr>
<tr>
<td>10 - Temperature change</td>
<td>Partially</td>
<td>Partially</td>
<td>Optionally</td>
<td>Optionally</td>
<td>partially</td>
</tr>
<tr>
<td>11 - Volume</td>
<td>Partially</td>
<td></td>
<td></td>
<td></td>
<td></td>
</tr>
<tr>
<td>12 - Voltage</td>
<td>Optionally</td>
<td></td>
<td></td>
<td></td>
<td></td>
</tr>
</tbody>
</table>

Since three out of four gas generators did not properly ignite, the information gathering in Table 5.1 could unfortunately not be realised. The details of this will be elaborated upon in the following sections.
5.3 Test setup
The tests consisted of the initiation or rather detonation of the gas generators, after which the released gas was captured. This gas should have been subsequently subjected to a number of tests to characterise its behaviour under different conditions. In order to do this, the gas was lead into a test chamber, to which the necessary sensors were connected. The required hardware for these tests is discussed in the subsections below. The connections between all pieces of hardware is shown at the end of the chapter in Figure 5.6. The tests to which this chamber should have been subjected will be described in section 5.4.

5.3.1 Test chamber
The test chamber is a 3 [mm] thick aluminium square prism shaped pressure vessel, specifically built by the author for this problem. Other available vessels could not be used, because the connections were incompatible with the necessary equipment. In addition to that, due to the explosive nature of the gas generators, the safety of existing vessels could not be guaranteed. The volume of this pressure vessel has been tested to be 1002 [ml]. This was done by filling the chamber with water and determining the mass of the water needed to completely fill the chamber. The assumption was made that the density of water is 1 [g/ml]. Inevitably some of the water was still clinging to the test chamber, or was lost during transferring the water to the measurement cup. In order to assess the gas pressure variation due to volume change, half of the chamber was filled with water in some of the tests, as will be described in subsection 5.4.3. The pressure vessel is capped by 4 [mm] clad aluminium 7075 plates, held together by 6 [mm] diameter steel rods. The cap on one side incorporates three holes to which the pressure sensor, thermocouple and gas generator holder, can easily be connected. On the bottom side there is a hole for the valve to pass through. Initially a bicycle tire Dunlop valve was put in place to characterise the behaviour of the test chamber under pressure. For the actual testing, a different valve was used to extract the water in the volume alteration test described below. For the sealing between the test chamber and end caps, 1 [mm] thick rubber sheet is used. The same rubber was cut to size and utilised to seal the holes where the pressure sensor, gas generator and valve pass through the structure.

The final assembled configuration of the test chamber with the pressure test valve is shown in Figure 5.3. The mentioned sensors and valves are visible in this figure as well.

Figure 5.3 - The test chamber including sensors and gas generator inlet on the left, and test valve on the right

The test chamber including the water release valve can be seen in Figure 5.4. This valve assembly adds an additional 12 [ml] of volume to the total internal test chamber volume.
5.3.2 Temperature measurement

Temperature measurements are important to characterise the behaviour of the pressurant gas. First of all, the initial temperature of the chemical reaction needs to be monitored. This is done in order to assess whether it would potentially harm the surrounding structure. Secondly the temperature effect on the pressure needs to be known. Optimally a k-type thermocouple would be used near the exhaust of the gas generator. This thermocouple is capable of measuring and withstanding temperatures from -50 °C to 1100 °C. After the failure of many leakage tests, as will be explained in subsection 5.6.1, the thermocouple had to be placed on the outside of the test chamber. It could therefore only measure the complete system temperature. The thermocouple output was transferred to the computer using the National Instruments NI 9211 data collector together with the NI USB-9162 converter. This data collected at the maximum frequency of 4.5 [Hz] by the LabVIEW program.

There was also an external thermometer to measure the ambient temperature of the room. This aided calibration of the thermocouple and determination of when the system has stabilised close to the room temperature. This stabilisation is important in order to be sure that effects which are not important to the measurement are stabilised. Variables such as thermal expansion of the test chamber.

To characterise the behaviour of the gas across the temperature range, a temperature controlled hot air blower was used to elevate the temperature. The distributed hot air flow can heat the chamber as evenly as possible. This air blower can heat the air to up to 600 °C, so the necessary 80 °C internal temperature was easily attainable.

5.3.3 Power consumption measurement

The voltage to the gas generator was regulated using a power supply set to 12 [V], because they are designed to be driven by car batteries. A multimeter was used to check if the output voltage of the power supply was indeed exactly as required. Therefore without relying solely on the settings on the power supply. The current should have been recorded by measuring the voltage across a shunt resistor, of which the resistance has been measured to be 33.2 [Ω]. This voltage was monitored by a National Instruments (NI) Data AcQuisiton device (DAQ) voltmeter, as schematically shown in Figure 5.5 below.
The DAQ that will be used is the NI USB-6008, which has a sampling frequency of 10 [kHz], a maximum voltage (drop) of 10 [V], the absolute accuracy at full scale is 14.7 [mV] at 25 [°C] and maximally 138 [mV] across the temperature range. Using Ohm’s law, the current across the shunt resistor, and thus the gas generator, can be calculated when the voltage difference is known.

\[ I = \frac{V}{R} \]

Where:

- \( I \) Electrical current [A]
- \( V \) Electrical potential difference [V]
- \( R \) Electrical resistance [Ω]

The voltage across the gas generator is the total potential difference, from the power source, minus the voltage measured across the shunt resistor. The power consumption by the gas generator can then be calculated using Joule’s first law. However the setup was unbalanced and the measuring circuit took too much voltage away from the gas generator for it to be initiated. Additionally, the other difficulties encountered in gas generator testing were more severe and required all attention. The capturing of the voltage drop using LabVIEW also did not work properly. Therefore this measurement was not conducted during this test campaign. If full characterisation is needed, this test has to be conducted in the future.

5.3.4 Pressure measurements

The pressure measurement is the primary reason for gas generator testing. The other measurements are only instrumental to this goal. The pressure was measured using a Feteris components DMP 331 pressure sensor with a range of 0 [bar] to 40 [bar]. The sensor was placed very close to the exhaust of the gas generator, in order to capture the initial shock pressure after the chemical reaction has occurred. The data from the sensor was transported to computer using the NI USB-6008 DAQ, as with the power consumption measurements. This was chosen to sample a frequency of 3.3 [Hz] using the LabVIEW program.

The final positioning of all hardware as it was tested is schematically represented in Figure 5.6 below.
5.4 Test procedure

To fully characterise the behaviour of the gas from the gas generator, the test procedure consists of the following five different stages:

1. Characterisation and calibration
2. Initiation stage
3. Volume alteration stage
4. Temperature alteration stage
5. Residue collection and disassembly

Each of these stages is described in the consecutive subsection below. As mentioned, four gas generators have been acquired for testing. All stages should have been conducted for at least two gas generators. If these two tests would have been successful, the gas characterisation would have been complete enough for a first order investigation. The remaining two gas generators could have been used to supplement the tests where something went wrong, to do additional tests, or to assess the initial pressure after initialisation. However, as stated before, three gas generators malfunctioned.

5.4.1 Characterisation and calibration

Before starting the actual tests on the gas generators the test setup itself needs to be properly characterised. The most important test is to see if there is excessive leakage when the test volume is under pressure. Large leakages lead to a misleadingly low stabilisation pressure. The volume and temperature alteration steps take some time to properly conduct. If too much of the gas escapes...
during this time, the gas cannot be properly characterised. Therefore it was first assessed by using a dunlop valve instead of the water release valve. A pump capable of going up to 7 [bar] was used to pressurise the test chamber. Due to the pressurisation of the air being pumped in, the temperature of the gas inside the test chamber rose above ambient temperature. After stopping with pumping, the temperature soon settled back to room temperature. The pressure and temperature sensors were activated during the pumping and the subsequent settling time. The pressure sensor was used to monitor the pressure rise due to pumping, the pressure drop due to temperature stabilisation, and the possible pressure drop due to leakage. The temperature sensor was used to monitor when the temperature settled back to around room temperature.

After the leakage tests, the test chamber was fitted with the water release valve, and filled with 500 [ml] of water. This finished setup is shown in Figure 5.7. After the whole setup was been built up it was left untouched for a minimum of 15 [min] in order for it to attain environmental temperature and pressure. After which all the measurement equipment were switched on and checked in a dry run, without the gas release. This procedure determines if they are rightly calibrated to environmental conditions and if the test volume attained room temperature again.

![Figure 5.7 - The test setup just before gas generator initiation. Pressure sensor and gas generator interface on top, water release valve on the bottom.](image)

### 5.4.2 Initiation stage

Due to the explosive nature of the chemical reaction in the gas generator, the initiation phase is extremely short. During and directly after chemical reaction, the released gas will have a high temperature, which should have been monitored. But as previously said, the thermocouple could not be inserted into the volume leak free, and was therefore attached to the outside of the chamber wall. The sensor measurements and recording were started before the initiation of the gas generator.

The gas generators are designed to be initiated by using 12 [V] of a car battery. In the DelFFi satellites however, a 5 [V] power supply is used for the propulsion system. This supply can be converted on the Printed Circuit Board (PCB) to deliver 12 [V]. However this conversion would cost power, volume and
mass. Therefore, when the essential tests were completed, it would have been favourable to also test of the gas generators can operate with 5 [V]. The generators are initiated using a simple bi-directional electrical resistive element. According to Joule’s first law, they should therefore also function at a lower voltage, with a higher current.

After capturing the initiation phase, the system was left to rest until the internal temperature settles close to room temperature. Only then the actual pressurant pressure can be determined. When this settling pressure was measured, the following test stages commenced.

5.4.3 Volume alteration stage
In order to characterise the gas pressure behaviour due to volume alterations, half of the chamber of the pressure vessel was filled with water. This water was released in measured quantities, to increase the volume in a stepwise manner. During these steps the pressure within the system was be carefully measured to document the pressure variation due to volume increase.

All sensors are situated on top of the pressure vessel, above the water. The water release valve is positioned in the bottom of the test chamber. Gravity therefore assures that only water comes out when opening the valve, and all the pressurant remains inside. Small quantities of water can be released using a manual release valve. The water was collected in a measuring cup, and the release valve was closed after the release of 50 [ml] of water every step. These water release steps are repeated until 50 [ml] water remains in the pressure vessel. This remainder is left in, in order not to run the risk of releasing the pressurant gas together with the water.

5.4.4 Temperature alteration stage
The operational range of the DelfFii satellite is -20 [°C] to 80 [°C]. There are however operational difficulties of cooling the entire connected test system to -20 [°C]. Therefore the characterisation with the range from room temperature to 80 [°C] was chosen to be sufficient. This behaviour would have been extrapolated downward in order to cover the entire range. Therefore the system was heated to 80 [°C] using a hot air blower, while documenting both pressure and temperature.

The cooling of the system can be achieved by using the liquid nitrogen which is available in the manufacturing hall of the faculty. If the gas generators are selected as an option for the DelfFii propulsion system, the operation in temperatures down to -20 [°C] will also have to be physically tested.

5.4.5 Residue collection and disassembly
After the tests were finished, the gas and remaining water was released through the water release valve and the system was partially disassembled. After this disassembly, the parts close to the gas generator exit were assessed for damage and corrosion. The residue of the gas generator was collected as much as possible. This in order to be able to decide if the residue could pose a hazard to the operation of the pressurisation system or propellant storage.

5.4.6 Expected results
In order to anticipate the behaviour of the gas generators and the possible implications to the test setup, the estimated results were first theoretically calculated. The expected properties of the gas generated is determined by comparing the gas generators to similar airbag initiators, of which the information is given [50]. Using their volume to chemical mass ratio, the chemical mass of the gas
generator is estimated to be 660 [mg]. The test containment volume has been designed to have an internal volume of 1 [l]. Of this volume, 50% was filled with water, in order to conduct the volume change tests mentioned in subsection 5.4.3. This leaves 500 [ml] of volume for the gas. Assuming an air density of 1.225 [kg/m³], and the molar mass of air to be 0.0287 [kg/mol], the amount of air present in this volume is 12.8 [mmol]. For further calculations, the air, as well as the gas produced by the gas generator are assumed to be 100% nitrogen.

The highest possible pressure is achieved when 100% of the chemical mass is transformed into nitrogen gas. The maximum temperature of this gas is expected to be in the order of the flame temperature of fireworks, which is 2000 [K] [51]. Since all gas is assumed to be nitrogen, the average temperature is simply determined as shown in equation 5-2.

\[ T_{avg} = \frac{T_{gen} \cdot m_{gen} + T_{air} \cdot m_{air}}{m_{gen} + m_{air}} \]  

5-2

Where:

- \( T_{avg} \): Average internal temperature [K]
- \( T_{gen} \): Temperature of the gas from the gas generator [K]
- \( m_{gen} \): Mass of the gas from the gas generator [kg]
- \( T_{air} \): Temperature of air in the test chamber [K]
- \( m_{air} \): Mass of air in the test chamber [kg]

Using the ideal gas law, shown in equation 3-1, at this average temperature of 1176 [K] the pressure inside the chamber will reach 8.6 [bar]. However in the more likely low pressure case, an estimated 10% of the chemical mass is transformed into nitrogen gas. The maximum flame temperature of the reaction is assumed to be a low 373.15 [K]. Using the ideal gas law, the pressure in the chamber in this case is calculated to be 1.2 [bar]. In reality the value will be somewhere in between, but probably closer to the lower bound. If the pressure is too low, the sensitivity of the pressure sensor will not be high enough to get accurate data. Therefore the first test will be performed without water, in order to make sure it will be able to withstand the maximum pressure. If the pressure during this test is found to be very low, the amount of water in subsequent tests will be increased, in order to increase the accuracy of the measurements.

The pressure is expected to settle at a room temperature of 298.15 [K]. The settling pressures can again be calculated using the ideal gas law. With the estimated gas mass of the highest pressure estimations, the settling pressure will be 2.19 [bar]. With the estimated gas mass of the lowest pressure estimations, the settling pressure will be 1.17 [bar]. Again the real value of the settling pressure will lie in between these two values. This is because the low pressure case is too conservative and it is physically impossible for 100% of the explosive to be converted to nitrogen gas, as assumed in the high pressure case.

5.5 Organisational and safety aspects

The responsibility for setting up, testing, data collection, data processing and analysis of these gas generator tests lied with the author of this thesis. Designated test engineer Quirino Bellini would
have aided in the test operations. Partially to gain experience into hardware testing and partially to check if the setup and measuring methods are correct. However due to the failure to set up the current measuring circuit, all tests were performed solely by the author.

Due to the fact that explosives were tested, safety had to be considered thoroughly. With the high pressure and high temperature which were encountered, all components had to be able to withstand the loads at all times. The rods have been calculated to be able to withstand the tensile force exerted by the end plates. The walls were calculated not to yield under the internal pressure. These necessary safety calculations are shown in the subsections below.

5.5.1 Structural integrity of the test chamber walls
The end plates of the test chamber are made out of clad 7075 aluminium with a thickness of 4 [mm]. The square prism which makes up the walls of the test chamber is made out of aluminium with a thickness of 3 [mm]. The square shape was chosen due to the availability of the material, a cylindrical has less tendency to bulge under pressure. The rods holding the endplates together have a diameter of 6 [mm]. The internal dimensions of the test chamber are 7.5 [cm] x 7.5 [cm] x 20 [cm]. Therefore the longest side face is the weakest spot of the walls. The stress is approximated by the stress in a clamped plate, as shown in Figure 5.8. Using simplified Roark’s formulas, the maximum stress in sheet material subjected to a uniform pressure can be calculated using equation 5-3 [52].

\[
\sigma_{\text{max}} = \frac{p \cdot b^2}{2t^2 \left( 0.623 \left( \frac{b}{a} \right)^6 + 1 \right)}
\]

Where:

- \(\sigma_{\text{max}}\) Maximum stress [N/m²]
- \(p\) Uniform compressive surface pressure on plate [Pa]
- \(b\) Minor length of rectangular plate [m]
- \(t\) Plate thickness [m]
- \(a\) Major length of rectangular plate [m]

![Figure 5.8 - Rectangular Flat Plate, uniform load, edge clamped [52]](image)

When the maximum stress is taken to be the yield stress of the material, the pressure will correspond to the maximum pressure that the vessel can hold before yielding. The yield strength for aluminium 2024 is 324 [MPa] [53], and for aluminium 7075 is 503 [MPa] [53]. However, since the type of aluminium is not known, a lower value of 200 [MPa] is used, to be on the safe side. Using equation 5-3, the maximum pressure before yielding in that case is 202 [bar] of overpressure.
Therefore, the maximum pressure of 8.6 [bar], is containable with a safety factor of 26.6 on yielding of the structure. Of course the side plate is not really totally clamped, and there are weak points at the edges etc. This safety factor is however adequately high, that there will be no threat of the walls yielding under the given pressure.

5.5.2 Structural integrity of the structural rods
The rods also had to be theoretically assessed to handle the loads of pressurisation, as they keep the end-caps together. The diameter of the rods is 6 [mm], thus the area is 28.27 [$\text{mm}^2$]. The rod material is probably stainless steel with a yield strength of 215 [MPa] [53]. The internal area of the end caps is 7.5 [cm] x 7.5 [cm]. The internal overpressure exerts a force on the end plates on both sides, which are held together by four rods. Therefore the tensile force in the rods, due to the pressure will be two times the overpressure force on one end, divided by four. Using the yield strength of the material, the maximum internal pressure in the test chamber before the rods will yield, is expressed by equation 5-4.

$$p_{in} = \frac{2 \cdot \sigma_r \cdot A_r}{A_{end}}$$

Where:
- $p_{in}$: Internal test chamber overpressure [Pa]
- $\sigma_r$: Yield strength of the rod material [Pa]
- $A_r$: The cross-sectional area of the rod [$\text{m}^2$]
- $A_{end}$: The internal area of the end plate [$\text{m}^2$]

This will result in a maximum overpressure of 21.6 [bar]. Which means that there is a safety factor of 2.8, with respect to the maximum theoretical pressure that could possibly be encountered in the tests. This calculation does not take the decrease in rod diameter due to the screw thread into account. However, since the worst case is assumed which in reality will never occur, and there still is a high safety factor, the rods will certainly be able to handle all realistic test conditions.

5.5.3 Heating test integrity
For the highest possible pressure case, the average gas temperature in the test chamber is calculated to be 1176 [K]. The maximum possible pressure is therefore subsequently calculated to be the pressure of the gas at this temperature. During the heating tests, the average temperature only went up to 353.15 [K], and the body of water, restricting the volume, was already mostly gone. This heating temperature is also nowhere near the maximum operating temperature of any of the structural materials. Therefore it can be concluded, that the pressure under heating would not pose a threat to the structural integrity of the test setup, and therefore the people around it.

5.6 Hot gas generator test results
Due to the sequential nature of the steps of the test procedure, the results are also segmented. The following subsections within this section will describe the results of each individual part of the testing campaign.
5.6.1 Characterisation and calibration

The first step in the test phase was the characterisation of the test volume itself. To this end, the Dunlop valve was installed, to connect a pressurisation pump. During the first pressurisation, it was immediately clear that the test chamber had a large leak rate. There was a clearly audible leakage around the thermocouple entrance. During this first pressurisation test the thermocouple was fed through a 1.5 [mm] hole in the sensor-side end cap. It was fixed and sealed using Bosch hot-melt adhesive. This leakage was shown to be unacceptably large, since most of the air immediately escaped and there was almost no pressure build up at all. The sensors were therefore not activated to characterise the leak rate. The leakage was caused by the outer material of the thermocouple, made of a very sleek flexible plastic. There are only very few adhesives which have a proper adhesion to those kind of materials.

The hot-melt glue was then removed completely. As a second attempt, the thermocouple was secured and sealed using cyanoacrylate adhesive. Due to the long drying time of larger amounts of this adhesive, it was left for 3 days to dry. After this drying period the leakage of the test chamber was characterised using the pressure and temperature sensors. The pressure variation observed during the test period is shown in Figure 5.9.

![Figure 5.9 - Pressurisation test using cyanoacrylate adhesive as a sealant](image)

It is visible that the leakage is unacceptably high for the longer duration tests. With a pressure drop of 1 [bar] in only 2 minutes, the settling behaviour cannot be properly characterised. The volume and temperature alteration tests take a significant amount of time. With this amount of leakage, the gas would escape before proper characterisation could be achieved. The pressure drop due to leakage could also interfere with the pressure drop and rise due to characterisation procedures. The leakage itself is probably caused by air bubbles in the adhesive, due to the excessive shrinkage which was observed while solidifying.

Following the second-glue test “Siliconen kit - bouw”, a silicon sealant, was used. To be sure of proper solidification and optimal sealing the sealant was left to set for 2 days. The chamber was again pressurised and left to settle to determine the leakage rate. The result of one of these tests is shown in Figure 5.10 below.
The leakage rate can be seen to be slightly lower, with 1 [bar] of pressure loss in 3 minutes. However as stated before, this is still not acceptable for the pressurisation durations encountered in this test campaign.

Next, an interlocking Swagelok threaded fitting was used together with a byrolock. Rubber was placed between the two parts, after which the thermocouple was lead through. When tightening the Swagelok components the rubber effectively clamped the thermocouple. This chamber setup was tested under pressure, the result of which are shown in Figure 5.11.

A pressure drop of 1 [bar] in 2 minutes was observed, which is higher than the previous method with the silicone sealant. For the before mentioned reasons, this leakage rate would again be too large for the necessary gas generator testing. Test quickly concluded that the leakage yet again originated from the thermocouple feed-through. After careful deliberation it has been decided to move the
thermocouple to the outside wall of the test chamber. This means that the initial temperature of the chemical reaction will not be properly measured. However the average temperature behaviour of the test chamber is well characterised, since the entire body of the chamber is made out of metal. The good thermal conduction will ensure a relatively homogeneous temperature within the entire structure.

In this last pressurisation test shown in Figure 5.11, the pressurisation valve was also opened in the end, to determine if there was any drift of the sensor. As can be seen in the data, during this test, there has been 0.00 [bar] drift, which is to say none. This was later found to be the case during all tests.

The thermocouple was thus bonded to the outside wall of the test chamber. The spot on the wall was chosen to be where the hot gas from the gas generator would impact the wall on the inside. The hole previously made for the thermocouple was closed with a nut and a bolt with washers, and rubber sealing on both sides. The test volume was then again pressurised with the sensors initiated. The pressure and temperature in time can be seen in Figure 5.12 and Figure 5.13 respectively.

![Figure 5.12 - Pressurisation test with external thermocouple - Pressure variation](image-url)
As can be seen in the temperature variation in Figure 5.13, the test chamber was actually still settling to environmental temperature. However this 0.6 °C difference does not affect the measurement significantly. At 120 [s] the pressure sufficiently stabilised at 3.676 [bar]. At 400 [s] the pressure dropped down to 3.576 [bar]. This means a drop of 0.1 [bar] in 280 [s]. As can be seen in the tendency of the graph, this leak rate also decreases with decreasing overpressure. For the application of testing the gas generators, this leakage rate is acceptable, even though not desirable.

The leakage test included a submersion in water, to determine where the leakage is situated. The leakage was thereby seen to be concentrated at the gas generator adapter. However since this is a specialised component, which has already been isolated as much as possible with rubber sealing, there is not much else to be done. In subsequent tests the seal was closed with even more force, in order to minimise the gas loss.

In the future, the pressure dependent leakage rate can be determined, for example in [mg/s/Pa]. This will help in the calculation of the influence of leakage on the readings of the tests.

### 5.6.2 Shock load test

The first gas generator was used without any water in the test volume. This was to determine the initial shock pressure and the final settling pressure, to determine how the future tests should be executed. The initial temperature shock due to the chemical reaction was also monitored to assess potential hazard to the surrounding structure. The pressure and temperature variations of this test can be seen in Figure 5.14 and Figure 5.15 respectively.
Both the shock pressure stage and the stabilisation pressure stage cannot be clearly distinguished from these images. Therefore these stages are separated and analysed individually. Firstly the shock pressure stage is separated from the other data in Figure 5.16.
As can be seen from the indicated data points, the sampling frequency of the pressure sensor was not high enough to unambiguously capture the shock pressure. A curve has been fitted to match the data points. However in reality some of the points could be part of an initial oscillation and probably none of the data points represents the absolute maximum shock pressure. The experiment should therefore be executed multiple times to ensure that the behaviour is repeatable. In this first test the maximum shock pressure was measured to be 10.413 [bar]

The sensor itself could have influenced the readout. Possibly some oscillatory behaviour occurred within the mechanics of the sensor, due to the sudden significant increase in pressure. This could have lead to an unrealistically high reading if the sensor mechanism overshoots, or an unrealistically low reading if there was some internal friction. However, the reading seems to look relatively clean, and oscillation free. Additionally, the maximum measured value is only at 26% of the sensor range. Therefore the reading is assumed to be sufficiently reliable.

The shock temperature stage is separated from the whole temperature data range in Figure 5.17 below. Due to the placement of the temperature sensor on the outside of the test chamber, the response time depends on the mass of the entire system. The duration of this phase is therefore significantly larger than that of the shock pressure.
Before gas generator initiation the temperature had stabilised at 23.7 [°C]. The temperature rise of the chamber can be seen to have a maximum at 39.05 [°C]. This indicates a temperature rise of only 15.4 [°C]. This maximum temperature is very low compared to the maximum operating temperature of e.g. aluminium at 150 [°C] to 200 [°C] or of steel at 200 [°C] to 700 [°C] [43].

5.6.3 Settling pressure
As seen Figure 5.14, more than 10 [min] after the gas generator initiation the pressure was sufficiently stabilised. The final part of the pressure stabilisation phase can be seen in Figure 5.18 below.

The only jumps in the data points which can be observed is due to the accuracy of the pressure sensor, with a resolution of 0.02 [bar]. The average pressure measured during this time is 1.344 [bar].
5.6.4 **Volume and temperature alteration behaviour**

Unfortunately the tests that were meant to characterise the volume and temperature alteration behaviour of the released gas all malfunctioned. The initiation stage of these generators described in section 5.1 ignited, but it somehow failed to ignite the main stage. The pressure generated by the combustion of the initial stage was however enough to fracture the burst section of the gas generator. The explosive pellets of the main charge were thereby simply expelled from the generator without being ignited. The actual pressures and temperatures that were recorded during the malfunctional gas generator tests are added in Appendix D for completeness. These only indicate the system behaviour of the first stage. The amount of gas from this primary charge is too low to successfully analyse the behaviour of the expelled gas. The small amount of gas is not sufficiently discernible within the large amount of air present in the test chamber before initiation.

The temperature alteration is however partially assessed with the first gas generator test. If the initial pressure estimation and the final settling pressure estimations are both accurate with the findings, the estimations on gas behaviour are verified. This will be shown in subsection 5.7.1.

5.6.5 **Residue collection and disassembly**

For the first test, the gas generator reacted i.e. exploded properly. When the structure was taken apart no structural damage was spotted to the affected parts. Even the rubber sealing, which has the lowest maximum service temperature, showed no sign of damage. Some residue and condensation was however deposited on all interior parts, as seen in Figure 5.19.

![Residue and condensation inside the test chamber](image)

**Figure 5.19 - Residue and condensation inside the test chamber**

The absolute amount of residue was rather small, as can be seen in Figure 5.20. The residue that was captured has a mass of only 71.3 [mg]. It seemed to mainly consist of the primary charge coating shown in Figure 5.2.
The residue of this generator will therefore not pose a large threat to a possible propulsion system. It will only have to be kept from entering the small fuel lines and valves. This can be achieved by separating the pressurant from the propellant by means of a piston, membrane or bladder. Or by filtering the propellant before exiting the tank.

After the first successful test, all three subsequent gas generator did not fully ignite. This resulted in the expulsion of the largest part of the actual propellant grain, which is ignited to generate the gas. The malfunctioning was possibly due to the large clamping force on the gas generators. This had to be done slightly different to the automotive integration standard, to integrate it with the test chamber, and to improve long duration leak tightness. An alternative explanation is that gas generators might have come into contact with the water, which was used to dynamically alter the test chamber volume. From a reliability point of view, this is disastrous, but it grants some insight into the explosive that is used.

The residue collected from the three failed tests can be seen in Figure 5.21. This consists of green multi-tubular explosive grains, similar to those seen in gun propellants. The diameter is 2 [mm] and the lengths are 1 [mm] to 3 [mm] and they have seven circular perforations. These perforations are common in solid propellant grains, and ensure a progressive burn rate. Which means that the area, where the chemical reaction can take place, increases during the burn time of the grain. A schematic representation of the propellant grain can be seen in Figure 5.22. Next to those grains, some residue from the initial stage coating, shown in Figure 5.2, is also present.
The total mass of the residue that was captured of all three failed tests is 2,590 [mg], i.e. an average of 863 [mg] each. However, some of the propellant already ignited to cause the bursting of the gas generator, not all residue is actually propellant and that not all residue could be captured. Therefore the actual explosive charge per gas generator is estimated to be at least 900 [mg].

5.7 Recalculation of performance estimations
In this section, the assumptions in the calculations are adapted to fit the findings of the tests. These adapted calculations will then be used to estimate the performance of the gas generators, when they would be used as for the DelFFi mission propulsion system pressurisation.

5.7.1 Pressure calculation adjustments
Due to the fact that the three malfunctioning gas generators expelled the primary chemical charge, it was possible to analyse this propellant. The material of these grains is similar to smokeless gunpowder as used in the defence industry. Simple smokeless gunpowder is a single base propellant containing nitrocellulose, \( C_6H_7(NO_2)_3O_5 \). This nitrocellulose has a molar mass of 297 [g/mol]. Most probably there are some additives to aid in safety, performance or manufacturing, but the ones used in the gas generators cannot be estimated. Therefore the assumption is made that the grains are solely based on nitrocellulose.

When introduced to heat, the decomposition reaction shown in Figure 5.23 takes place.

\[
1 \cdot C_6H_7(NO_2)_3O_5 \rightarrow 1.5 \cdot CO_2 + 4.5 \cdot CO + 3.5 \cdot H_2O + 1.5 \cdot N_2
\]
As stated in subsection 5.6.5, there is approximately 900 [mg] of nitrocellulose in each gas generator. Assuming 100% of the explosive charge combusts, this leads to the production of 37.04 [mmol] of hot gas. The internal test chamber volume is 1002 [ml]. Assuming an air density of 1.225 [kg/m³], and the molar mass of air to be 0.0287 [kg/mol], the amount of air present in this volume is 42.77 [mmol]. The molar heat capacity of all species in the gas from the generator, and in the air, is similar [23]. Therefore the average temperature of the gas in the test chamber can be calculated using equation 5-2. Since it is known that the shock pressure must be at least 10.413 [bar], the flame temperature of the reaction can be determined. Using the ideal gas law shown in equation 3-1, the gas must have a temperature of 3517 [K]. This is relatively high, since the flame temperature of nitrocellulose is generally in the range of 2400 [K] and 3100 [K].

However, in reality, the hot gas and the present air will not mix instantaneously. The gas will actually be a lot hotter near the gas generator, than in the rest of the testing chamber. This will locally increase the pressure, potentially also causing a pressure wave. When the gasses mix, the temperature is averaged, and the pressure drops, as seen in Figure 5.16. It is therefore probable that the actual flame temperature was indeed in between 2400 [K] and 3100 [K].

After cooling down, the water will condensate, and thus the remaining molar mass of the gasses will be:

\[
\frac{1.5 \cdot (12 + 2 \cdot 16) + 4.5 \cdot (12 + 16) + 3.5 \cdot (2 \cdot 1 + 16) + 1.5 \cdot (2 \cdot 14)}{1.5 + 4.5 + 3.5 + 1.5} = 31.2 [g/mol]
\]

Due to the condensation of the water, the amount of gas from the gas generator is decreased by:

\[
\frac{1.5 + 4.5 + 1.5}{1.5 + 4.5 + 3.5 + 1.5} \cdot 100\% = 68\%
\]

As mentioned in subsection 5.6.1 the leakage rate was found to be 0.1 [bar] in 280 [s]. This leakage rate will go down with lower pressure, and the test chamber was actually more leak tight during the gas generator tests. But due to the lack of an overpressure dependent leakage rate the worst case leakage during the time of the test is assumed to be:

\[
0.1 [\text{bar}] / 280 [\text{s}] \cdot 620 [\text{s}] = 0.221 [\text{bar}]
\]

Taking all of this into account in the ideal gas law shown in equation 3-1, the settling pressure would be 1.394 [bar]. This is close to the actually measured 1.344 [bar]. The reason why in reality it is a little bit lower could be that not all the explosive charge fully reacted, or that it contained other constituents next to nitrocellulose. The leakage rate can also have been higher than previously measured due to the high shock pressure.
5.7.2  Application in the DelFFi propulsion system

When the generators are utilised as pressurant gas supplies for the DelFFi mission satellite propulsion system the expansion volume will be smaller than encountered in the successful test. In order to calculate the temperature shock on the propellant storage tank the heat capacity of the gas is neglected. This is acceptable because its volumetric heat capacity is orders of magnitude lower than that of the aluminium walls. When the chamber volume is decreased to the 100 [ml] required for the satellites, the wall volume will decrease to around 25% of that of the test chamber. As shown in subsection 5.6.2, the temperature of the test chamber rose with 15.35 [K], during the gas generator test. Assuming a similar volumetric heat capacity to the test chamber, the temperature of the entire propellant tank would increase by 61.5 [K]. When starting at room temperature, this shock temperature is within the service range of the intended wall materials [53].

Using the properties of the released gas deduced in subsection 5.7.1, the pressures can be determined using the pressurant volume of 100 [ml]. Using the ideal gas law, the shock pressure would then be 95 [bar]. However, the ideal gas law doesn’t hold anymore at these high pressures and temperatures. Therefore the actual pressure peak will be lower. The shock load would again be of only very small duration, in the order of 0.1 [s] as seen from Figure 5.16. For the settling pressure calculation, it is assumed that the volume is built such that there will be no initial leakage. The settling pressure at room temperature will then be 6.49 [bar].

The temperature rise of the propellant storage system due to the combustion process is thus acceptable. However the very high estimated shock pressure would require a relatively heavy tank. Despite of this high shock pressure the final settling pressure will still not be the required 10 [bar] for the DelFFi propulsion system.

5.8  Conclusions on the gas generator tests

In this test campaign four automotive gas generators were tested, for possibility of being the pressurant supply for the DelFFi satellites propulsion system. Three of these gas generators malfunctioned during testing, by expelling the main propellant without igniting it. This is possibly due to the larger clamping force on the gas generators, in order to integrate them with the test chamber. The supplier, Airbagbank.eu was also perplexed by the low reliability and some additional tests will be conducted using their own test facilities. Judging from the appearance and behaviour, the gas generators probably contain nitrocellulose grains. Each gas generator holds 900 [mg] of propellant, with a molar mass of 297 [g/mol]. These grains are ignited using an electrical resistive heating element. Near to 100% of this propellant is converted into hot gas, with a molar mass of 27 [g/mol], upon initiation. The products of reaction are mainly O2, CO2, CO and H2O. When cooling down, 68% of the expellant amount remains gaseous with a molar mass of 31.2 [g/mol]. The gas generators have the following very promising properties for application in the DelFFi propulsion system:

- Pressurisation is achieved within a few milliseconds.
- Electrical power consumption is below 12 [W] with a duration in the order of milliseconds.
- Most of the expellant is gaseous and relatively benign, when cooled down to room temperature.
- They have a small mass of only 13 [g] with a gas production of about 0.6 [g].
- There is only a small amount of solid residue present after combustion, in the order of 70 [mg].
There are however also the following major problems which make them unsuitable for the given mission:

- For a pressurant volume of 100 [ml], the shock pressure is estimated to rise to a possible maximum of 95 [bar]. The settling pressure will then be 6.49 [bar], which is in range for the DeFFi mission propulsion system.
- The reliability was found to be low, with 3 out of 4 gas generators malfunctioning.
- The flame temperature is estimated to be on the higher end of the 2400 [K] to 3100 [K] range, which is hard to contain in the small volume.
- A voltage converter might be required convert the 5 [V] satellite power supply to the 12 [V] for which the gas generator was built.

When the shock pressure, flame temperature and reliability issues can be solved, the automotive gas generators could prove to be effective gas generators for pressurisation purposes. For now however, these gas generators have proven to be incompatible with the DeFFi propulsion system requirements.

As shown in section 5.2, a number of requirements were meant to have been checked for compliance to the DeFFi propulsion system. Some of these envisioned requirement checks were actually not carried out due to the malfunctioning of the gas generators. In Table 5.2 below, the requirements compliance table, shown in Table 5.1, is depicted once more. This time it indicates which requirements were actually checked and which were not.

<table>
<thead>
<tr>
<th>Requirement</th>
<th>Preparation</th>
<th>GG 1</th>
<th>GG 2</th>
<th>GG 3</th>
<th>GG 4</th>
</tr>
</thead>
<tbody>
<tr>
<td>1 - Combustion temp.</td>
<td>yes</td>
<td>failed</td>
<td>failed</td>
<td>failed</td>
<td></td>
</tr>
<tr>
<td>2 - Corrosion</td>
<td>yes</td>
<td>partially</td>
<td>partially</td>
<td>partially</td>
<td></td>
</tr>
<tr>
<td>3 - Volume change</td>
<td>no</td>
<td>failed</td>
<td>failed</td>
<td>no</td>
<td></td>
</tr>
<tr>
<td>4 - residue</td>
<td>yes</td>
<td>yes</td>
<td>yes</td>
<td>yes</td>
<td></td>
</tr>
<tr>
<td>5 - Power</td>
<td>partially</td>
<td>partially</td>
<td>partially</td>
<td>partially</td>
<td></td>
</tr>
<tr>
<td>6 - Mass</td>
<td>Partially</td>
<td>N.A.</td>
<td>N.A.</td>
<td>N.A.</td>
<td>N.A.</td>
</tr>
<tr>
<td>7 - Launch loads</td>
<td>N.A.</td>
<td>N.A.</td>
<td>N.A.</td>
<td>N.A.</td>
<td>N.A.</td>
</tr>
<tr>
<td>8 - Operational temp.</td>
<td>partially</td>
<td>partially</td>
<td>partially</td>
<td>partially</td>
<td></td>
</tr>
<tr>
<td>9 - Shock pressure</td>
<td>yes</td>
<td>failed</td>
<td>failed</td>
<td>failed</td>
<td></td>
</tr>
<tr>
<td>10 - Temperature change</td>
<td>Partially</td>
<td>failed</td>
<td>failed</td>
<td>failed</td>
<td></td>
</tr>
<tr>
<td>11 - Volume</td>
<td>Partially</td>
<td>no</td>
<td>no</td>
<td>no</td>
<td></td>
</tr>
</tbody>
</table>

5.9 Recommendations for future testing efforts

If the gas generators are to be further assessed for spacecraft pressurisation, at least the following steps need to be taken:

- It needs to be found out why three out of four gas generators failed, as they should be very reliable.
  - Possible causes are the clamping forces, causing warping or premature damaging of the burst section of the generators.
Another cause could be the possible contact with the water, which was used for decreasing the test chamber internal volume.

- More gas generators shall be tested using the method described above, to fully characterise the expelled gas.
- The mass of the gas generators needs to be measured accurately, before and after detonation. This will help in estimating the amount of propellant in each generator.
- Different types of gas generators could be tested.
6 Feed system

This chapter will briefly discuss the feed system of the propulsion system. Due to the fact that the selection of the feed system was mainly conducted by Mr. Krusharev, this is mainly added to indicate the necessary interfaces and to have a complete view of the propulsion system. The feed system is minimised as far as possible, and will only be instrumental to the propulsion system. The valve which regulates the propellant flow is the main component of the feed system. Furthermore there is the interfacing to the propellant tank on one side, and the thruster on the other and piping in between where necessary. These sections of the feed system will be discussed in the subsequent sections of this chapter.

6.1 Propellant storage tank interface

As described in chapter 4, the interface between the propellant storage tank and the feed system is a protruding threaded cylinder. Since the propulsion system sketch in Figure 2.5 is only meant to be schematic, the actual layout is slightly different. The main difference is that in the sketch, the suspended valve will not by a structurally sound design. It will neither either be easily integrated with the rest of the satellite structure. The biggest difference in reality will be that the valve will actually be placed and fastened horizontally on the propellant tank lid. The valve will be connected to the side of the cylindrical fluidic interface of the propellant tank. If possible, this is directly welded to the valve inlet. Otherwise a small piece of rigid piping will be necessary in between.

In parallel to this, there is also a connection to connect the pressurant fill valve on top of the tank. This filling has to be done, as close to the launch of the satellite as possible, in order to minimise the possible leakage. Once the pressurant is filled into the storage tank, the connection can be welded shut, after which the fill valve can be removed. The welding will minimise the risk of pressurant leakage. And the removal of the pressurant fill valve will decrease the mass of the propulsion system during launch. In this case, there also have to be no compromises on the fill valve selection. Any best applicable valve can be chosen, since it will be removed afterwards.

6.2 Piping

As said before, the valve will be connected to the propellant tank interface horizontally. Ideally, the valve inlet will be directly welded to this interface. This would then also minimise the risk of leakage. If direct welding is not possible, a section of rigid piping will be welded between the two points. The thruster section of the propulsion system will actually be mounted to the bottom panel of the satellite, as shown in Figure 2.5. Therefore the valve and the thruster will be connected using a flexible hose, which is screwed in place. This flexible hose will ease the integration, and allow for misalignment of the propulsion system components.

For the rigid piping the industrially manufactured capillary tubes by “Salomon’s metalen” can be used [26]. Here a whole range of orifice size and wall thicknesses can be found for RVS capillary tubing. For the propulsion system, the size which best fits the valve is chosen. This is tubing with an outside diameter of 2.4 [mm] and an inside diameter of 1.6 [mm]. The inside diameter therefore connects
well to the inside diameter of the valve which is 1.59 [mm], as will be described in section 6.3. The wall thickness of 0.4 [mm] is relatively large for such piping. This large wall thickness will ensure that there is enough welding connection.

Swagelok flexible tubing will be used to connect the valve and the thruster propellant flows. Swagelok produces a great number of flexible hose types. The tubing must be chosen such that it will withstand the pressure and temperature requirements. The connection has to be chosen such that it best fits the valve which is used. Therefore the same Swagelok flexible piping which is used in the engineering model feed system is envisioned for use in the flight model. This piping will be elaborated upon in the master thesis documentation by Mr. Krusharev.

### 6.3 Propellant control valve

The propellant flow control valve that will be used for the flight model was chosen by Mr. Krusharev to be a Lee company, IEP series solenoid valve. The inner diameter of the interface tubing of this valve is 1/16” i.e. 1.59 [mm].

In Figure 6.1 below, a cross-section of a solenoid valve is shown. This example is in fact the Lee company IEP series valve, which will be used in the DelFFi mission flight model of the propulsion system. As can be seen, this valve closes the flow using a plunger which presses a seal to the outlet port. This plunger is moved, to open the flow path, by running electricity though the coil. Contamination in the propellant flow can cause improper sealing and therefore leakages. In order to prevent contamination of the seal, a filter will be placed in front of the valve. All valves of the Lee Company have the recommended filtration levels stated in the product charts. For the IEP series solenoid valve filtration down to 10 [μm] is recommended [56].

![Cross-section of a solenoid valve](image)

**Figure 6.1 - Longitudinal cross-section of the Lee company IEP series solenoid valve [56], which will be used in the DelFFi propulsion system flight model.**

The feed valve is chosen for its similarity to the valve that was used in T³μPS on the Delfi-n3Xt satellite. This was the Lee company Extended Performance Solenoid Valve (EPSV) depicted in Figure 6.2 below. These have a piping diameter of 1.6 [mm], a power consumption of 0.5 [W], a total length, including piping of 33 [mm] and a maximum diameter of 6.4 [mm]. The mass of the valve is under 6 [g]. Due to the fact that The Lee Company EPSV valve has already proven itself in spacecraft propulsion in the T³μPS, from a reliability point of view it is favourable to use the same type, or one that closely resembles this flight proven one.
6.4 Fluidic interconnect to the thruster

This section will describe the necessary connection between the flexible tubing coming from the valve to the resistojet thruster. As will be explained in chapter 7, the thruster will have an inlet channel with a square cross-section. The sides of this square are 400 [$\mu$m] long. This has been designed such that a needle, with an external diameter of 300 [$\mu$m] can be inserted. This needle is then fixed in place and sealed by adding high temperature adhesive sealant. This adhesive is added in between the channel wall and the needle outer surface. This method was first shown by Tittu Varghese Mathew, for his nitrogen fed MEMS resistojet [31]. This connection is shown in Figure 6.3.

![Figure 6.2 - The Lee Company EPSV fluid flow control valve [57]](image)

![Figure 6.3 - Nitrogen fed MEMS resistojet by Tittu Varghese Mathew, showing the needle bonded to the thruster [31]](image)

For the production process of the MEMS resistojet it is important that the inlet channel width is as small as possible. This thruster and its manufacturing process will be described in more detail in chapter 7. If the width of this channel is not equal to the depth of the heating section flow channel, an extra etching step has to be executed. The possible depth of the inlet channel is also limited to the maximum etching that can be done, while maintaining structural integrity of the silicon piece. Therefore a survey of generally available needles was conducted. The results of this survey are shown in Table F.1 in Appendix F. The smallest needles available have an outside diameter of 0.3 [mm]. Since multiple suppliers of these needles are available, the chair preferred supplier called Farnell was chosen. The selected needle is called “WELLER - KDS3012P - DISPENSING NEEDLE, GA30, ID 0.15MM”. It has an inner flow channel diameter of 0.15 [mm].
The selected needle has a Luer-Lock connection interface. This is an internationally used standardised leak tight connection for pressurised needles. The connection has a male cone on one side, and a female cone on the other side. The male cone is slid into the female cone, and fastened into place by a screw thread on the outside of the needle. This connection is best visualised in Figure 6.4.

![Figure 6.4 - Both sides of a Luer-Lock connection [58]](image)

To connect it with the flexible hose, two connection pieces have been acquired. One connector is the Luer-Lock connector shown in Figure 6.5. The second connector connects this Luer-Lock connection to the standard Swagelok flexible hose connection.

![Figure 6.5 - Luer-Lock connector, connected to the needle [31]](image)
7 Thruster

This chapter will deal with the most extensive and novel part of the propulsion system, the resistojet thruster itself. During the research, a number of thruster geometries were assessed. The ones which are perceived to be most optimal for the DelFFi propulsion system resistojet have been produced. This production method consists of etching the necessary channels in the centre of a silicon wafer. Part of this channel is designated as the heating chamber. Suspended in the middle of the heating chamber is a series of silicon carbide resistive heating elements. This heating element vaporises and heats the propellant flow, after which it is accelerated and expelled through a nozzle. This MEMS resistojet has been given the name “Dondersteen”, which is Dutch for “thunderstone”. This name was chosen because thunderstones were once thought to be produced by lightning, i.e. electricity, which is what powers a resistojet. Secondly the thruster is mainly made out of silicon, which is the main compound in most stone.

The manufacturing of the thruster was sponsored by the Delft Institute for Microsystems and Nanoelectronics (Dimes). The design process was therefore conducted in cooperation between the author and the Dimes representatives Dr Henk van Zeijl and Bruno Morana. The design methodology, materials and accuracies were all chosen to fit the capacities of Dimes, while meeting the requirements of the DelFFi propulsion system.

In section 7.1 some of the previous work done on resistojets at the TU Delft is summarised. This is followed by the method by which the design and subsequent manufacturing were undertaken in section 7.2. The individual sections of the thrusters are dealt with next. The inlet in section 7.3, the heating part in section 7.4 and the nozzle in section 7.5. The extensive heat transfer calculations were conducted, in an effort to verify the heat loss assumptions made in chapter 3. These heat transfer calculations are shown in section 7.6. The thruster manufacturing process is then explained in section 7.7. Finally the integration of the thruster on the electronic circuit is described in section 7.8.

7.1 Previous TU Delft resistojets

At the TU Delft there have already been multiple projects to gain knowhow about resistojet. The following subsections will shortly summarise their setup and conclusions.

7.1.1 The first nitrogen fed resistojet DUR-1

The first resistojet was called the Delft University Resistojet - 1 (DUR-1) [59] and is shown in Figure 7.1. This was constructed with a modular build-up, in order to test various heaters, insulation and nozzles. As a heater, a helically shaped steel tube was used. Compared to a straight hollow tube, this reduces the length while the exhaust temperature is the same. Nitrogen propellant was, used at a pressure of 2.75 [bar] to 3.5 [bar]. This propellant was heated to up to 470 [K], by having a heating temperature of 675 [K]. The main conclusion of this research is that the specific impulse increased by around 60%, by applying heat to the propellant. The best nozzle used in this design has a throat diameter of 0.91 [mm] and an expansion ratio of 4.36. The scale of this resistojet however, is far away from a cubesat scale. It has not been optimised for low mass or volume, and has a maximum
power consumption of 300 [W] during the tests. An additional problem was the low efficiency of the heating process of only 36%.

![Diagram of central components of DUR-1](image)

**Figure 7.1 - The central components of the DUR-1 [59]**

### 7.1.2 Improvement on the nitrogen fed resistojet DUR-1.1 and DUR-1.2

The second resistojet development at the TU Delft is the advancement of the DUR-1 to the DUR-1.1 and DUR-1.2 [60]. These are the continuation of the research done on the DUR-1, made more applicable to micro-satellites. Again nitrogen was used as a propellant. This time with a feed pressure of 2.6 [bar] and a chamber pressure of 2 [bar]. The thrust level was tested to be up to 50 [mN], with a specific impulse of 142 [s]. The maximum heating chamber temperature was 1000 [K], with a power consumption of 35 [W]. The major improvement of this resistojet compared to the previous version, is the heating efficiency of 80%. This is more than 2 times the efficiency of the DUR-1. This has been achieved by encasing it in a purposely built vacuum tube and surrounding it with radiation heat shields. This is encasement shown in Figure 7.2. The DUR-1.2 features a slight addition to the previously used helix shape, as shown in Figure 7.3. This reduces the size necessary for the thruster to a diameter of 33 [mm] and a length of 27 [mm]. The performance characteristics of this thruster are however still too high, for application in the DelFFi nano-satellites.

![Image of heater module vacuum tube and radiation shielding of DUR-1.1](image)

**Figure 7.2 - heater module vacuum tube and radiation shielding of the DUR-1.1 [60]**
7.1.3 The water fed resistojet DUR-H$_2$O

After the research on the DUR-1.2, the next development was the water fed resistojet DUR-H$_2$O [61], shown in Figure 7.4. This research was conducted to gain insight into the use of liquid propellant in a thruster. The research was also meant to give the first experience with boiling flow. This design featured a helically coiled steel tube as a heating chamber, similar to the previous DUR designs. The inner diameter of this tube is 1.3 [mm], with a total length of 0.70 [m] and a coil diameter of 25 [mm]. The DUR-H$_2$O reached a chamber temperature of 1000 [K], at a chamber pressure of 2 [bar]. The nozzle throat diameter of 0.4 [mm], resulted in a critical mass flow rate of 24 [mg/s]. The expected thrust for this resistojet was 19 [mN] with an $I_{sp}$ of 79 [s].

After the application of insulation as shown in Figure 7.5, a thermal efficiency of 86% was obtained. During this research, the theoretical approximation of the heat loss at stable conditions was proven to be accurate. The pressure drop was proven to be substantially less significant than theoretical predictions indicated. The major point of caution found during this research was the very low nozzle efficiency. This was observed to be due to condensation in the nozzle. The lesson for future liquid fed resistojets is, to make sure that the conditions in the nozzle are designed such that the expellant is gaseous throughout the length of the nozzle.

7.1.4 A MEMS micro-resistojet

After the large scale resistojet models were completed, the next evolution was to use Micro Electro-Mechanical Systems (MEMS) manufacturing techniques. This enabled the creation of an integrated, very small scale and precise thruster. The outside structure of this thruster is shown in Figure 7.6. A
needle forms the fluidic interconnect with the propellant feed system. The propellant is inserted into the thruster via this needle, which is bonded inside the thruster inlet using epoxy adhesive. The propellant is then guided through the straight rectangular heating channel, where it is heated to an elevated temperature. This heat comes from an aluminium electrical resistive heating layer, which is deposited on top of the channel. This heat is then conducted to the flow via the silicon body and channel walls of the thruster. Finally, the propellant is accelerated and expelled via a converging diverging nozzle. This internal channel structure is clearly depicted in Figure 7.7. Several chamber geometries were tested, with different heating chamber geometries and different nozzle sizes.

Heating chambers with 150 [μm] and 50 [μm] channel width were tested, in both single and multiple parallel channels concepts. All channels had a length of 2 [cm], with a uniform etching depth of 50 [μm]. Nozzle throat diameters of 5 [μm] and 10 [μm] were tested. Valuable experience was also gained in the necessary test setup for MEMS thrusters. Calculations on the result of this research showed a lower nozzle efficiency for lower flow Reynolds numbers in the nozzle throat. This indicates the prominence of viscous losses in micro fluidics. The three mechanisms of heat loss were studied for this thruster: free convection, radiation and conduction. The conduction through the structure proved to be the most prominent of the three.

The power consumption of this device is less than 3 [W], in order to heat the propellant with a mass flow rate of 1 [mg/s], to a temperature of 350 [°C]. The discharge factor, i.e. the ratio between experimental and theoretical mass flow rate, of the nozzle was found to be 50% to 80%, across the range of mass flow rates. The heating temperature in this design was limited by the low maximum operating temperature of the aluminium heater and epoxy glue. When these materials are replaced by materials with a larger temperature range, the maximum temperature of the flow can be substantially increased. The propellant heating efficiency was found to be around 13%, which is very low compared to reference resistojets. This loss was mainly due to the high conductive losses within the silicon structure and the metallic fluidic interconnect to the feed system. There was also no radiative heat shield in place, even though the heating element was positioned on top of the channel. This meant that the heating element lost a lot of its heat, even before it could reach the propellant flow. During the testing the input current was manually controlled to achieve a certain heater temperature. From this experience it was advised to develop a PID controller to control this current in future projects.
7.2 Thruster design methodology
The general structure of the resistojet thruster that was designed is shown in Figure 7.8 below.
The numbers depicted in Figure 7.8 indicate the following sections of the thruster.

1. Propellant inlet
2. Divergent inlet section
3. Heating section
4. Nozzle convergent area
5. Nozzle throat
6. Nozzle diverging area

For testing purposes, it is desirable to vary the dimensions and geometries of some of these sections, in order to optimise the performance. The manufacturing process only allows for a square window of 22 [mm] by 22 [mm] to fit all designs on. This is the size of the mask with which the etching process is executed. Different sections on this masking window can be used sequentially, to build up thrusters with the desired composition of sections. Therefore the overall design of the thruster has been divided into three different sections. The inlet, the heating area and the nozzle. Each of these sections, from now on called “modules” can have multiple designs. Any number of modules can then be placed behind one another to form the necessary thruster cross-section. This is best explained with images. Figure 7.9, Figure 7.10 and Figure 7.11 show simplified examples of what a thruster module can look like. As shown in Figure 7.12 multiple of these modules can then be etched one behind the other, to form the desired thruster design. The actual masking window including the thruster modules is shown in Figure E.1 in Appendix E.

![Example thruster inlet module](image1)

![Example heating chamber module](image2)

![Example nozzle module](image3)

![Example of multiple thruster modules forming a complete thruster](image4)

The different thruster modules will be discussed in the three subsequent sections of this chapter.

The previous TU Delft MEMS resistojet had an aluminium heating element. It became immediately clear that for the new design an aluminium heater is not preferred, due to its low melting point. The concept of a new type of heating chamber evolved from one of the heating elements that Dimes has produced before, shown in Figure 7.13. It consists of a titanium nitride (TiN) heating element...
suspended in a flow channel on top of a silicon nitride membrane. Such an element is suspended in the centre of the flow channel. That positioning maximises the contact with the fluid flow and minimises the heat loss from the heating element to the surroundings. The resistive element can also be made of platinum, which a maximum operating temperature of 800 °C. An advantage of platinum is that it is the least reactive metal, ensuring that oxidation will not take place due to the water.

![Figure 7.13 - A titanium nitride (TiN) heating element suspended in a flow channel on top of a low stress, thin silicon nitride membrane](image)

Mr. Morana recently developed silicon carbide (SiC) resistive heaters at the Dimes facilities. SiC is a very promising material for resistive heating elements due to its inertness, high strength, relatively low density and high maximum operating temperature. As can be seen in Figure 4.4 it is one of the materials with the highest strength to density ratio. This means that a SiC heater would not need to be placed on top of a membrane to be placed in the flow. The heater element itself can be the membrane.

The necessary propellant flow channels are etched out in silicon, because this is the easiest and cheapest production method for these kinds of structures. Due to the highly developed and standardised method of processing silicon by etching, the accuracy is also high. The inaccuracy of this process at Dimes is below 10% of the channel depth. Another option for the channel manufacturing is sand blasting the shape into glass. This would be desirable because glass has a very low heat conductivity and would minimise heat loss through the structure. This method was however discarded by Dimes due to the high labour intensiveness and cost.

A new development, called greyscale lithography, could have been used for approximating 3-D shapes. This would enable the production of truly conically shaped nozzles and rounded flow channels. This method deposits the photoresist layer with different thicknesses. In the plasma etching phase, the photoresist layers will then subside after different exposure times, thereby attaining a differential etching depth. Due to the still very experimental phase of this technology this was not chosen for this time-critical design and production.

The thrusters were designed to be enclosed on both sides with an anodically bonded layer of glass for conductive insulation. A reflecting metal layer would be applied to the outside of this glass layer for it to work as a radiation shield as well. Finally the thruster would be encased in an approximately 1 [cm] layer of epoxy bonding compound, to give some extra rigidity, insulation and protection. All the intended layers of the thruster are indicated in Figure 7.14, from the heater in the centre to the outside resin encasing. The industrial silicon insulator compound is readily available at Dimes and
inexpensive. This final layer of epoxy bonding compound was later reconsidered. This compound is actually designed to conduct heat very well, and would therefore be very unsuitable to insulate the thruster. A proper material for this encasing still needs to be found. Since the production process was rushed the external layers are still unsure.

![Figure 7.14 - A schematic representation of the intended layers for the dondersteen resistojet. From conductive heating element at the centre of the thruster (shown on the bottom), to the resin encasing on the outside (shown on top). This stratification is thus mirrored in the layer of the heating element.](image)

### 7.3 Propellant inlet section

This first section of the thruster serves as the interface between the feed system and the thruster. It includes a long straight channel for inserting the fluidic interconnect needle and a diverging section to spread the propellant flow across the width of the heating section. The need for an injector structure in the diverging section will be examined in this chapter. Such an injector is put in place to homogeneously spread the propellant across the entire width. The choice of using an injector at all depends on the need to evenly spread the propellant over the heater chamber entrance. The following injector concepts can be thought of:

- No obstructions
- Fins
- A wall with small orifices
- Flow channels

The more constricting the concepts are, the higher the pressure drop is. The injector will have to be designed such that the pressure drop is minimal, while having a homogenous propellant distribution over the heating chamber entrance.

The flow pattern in a MEMS thruster was qualitatively evaluated using the COMSOL software. The mass flow was chosen to be 1 [mg/s], as is necessary for the thruster. In the inlet of the thruster the water propellant is still liquid and therefore the flow velocity is 0.021 [m/s], which is relatively low. Therefore laminar flow can be assumed. The flow behaviour was analysed for the thruster geometry with both circular and diamond shaped fins. Since the flow velocity in the inlet and heating chamber width are the same for all varieties, this behaviour is representative for all designs. In the Figure 7.15 and Figure 7.16 the resulting flow velocity patterns are shown for two heating chamber geometries.
As can be seen in Figure 7.15 the colour of the flow at the entrance of the heating section, where the pillars are, is the same from side to side. This indicates that the magnitude of the flow velocity across the inlet is the same. In this case the flow velocity in the centre of the heater channel is no more than 9% larger than that near the sides of the heating chamber. Therefore the empty inlet section is well capable of dividing the flow without additional structures or channels. The same conclusion can be drawn from Figure 7.16. Here it can be seen that all the streamlines, and thus the water propellant, are homogeneously distributed across the width of the thruster heating section entrance. Therefore there is no need for an injector structure to distribute the water propellant flow across the width of the heating section.

The inlet section starts with a 1 [cm] long straight section, with a square cross-section with 400 [μm] sides, to insert the fluidic interconnect. It will then feature a divergent section with a divergent half angle of 45°, to convert the inlet width to the 3000 [μm] of the heating section. This geometry is shown in Figure 7.17. Since the transition from the small inlet diameter, to the larger vaporisation section is gradual, there is no significant pressure loss due to the expansion.

7.4 Propellant heating section

This section of the thruster is the part where the propellant needs to be vaporised and brought to an elevated temperature. The design efforts on this section focus on maximising the heat transfer to the
propellant, while minimising the pressure drop and heat loss to the surroundings. As calculated in chapter 3, the propellant mass flow needs a certain power input in order to be vaporised and heated to the desired temperature. When the heat transfer from the resistive heating elements to the propellant flow is maximised, the length is minimised, aiding in a low heat loss to the surroundings.

In all design efforts of the heating section in this report a SiC resistive heating element is suspended in the propellant flow channel. The geometry of both the flow channel and the heating elements are the design parameters. In the following subsections the different design options for this thruster section are first described. After which a selection is made from the most promising geometries, based on performance estimations. These geometries are the ones which are taken in production. Therefore the reasoning for the dimensions within the heating chamber designs will be discussed. At the end of this section the estimated performance of the heater sections is calculated.

7.4.1 Propellant heating chamber geometry options

There are many possibilities for the design of the resistive heater element. The main geometry design considerations, in no order of importance, are:

- Mechanical load tolerance
- Redundancy
- Heating element contact surface
- Flow mixing properties
- Manufacturability

In these heating sections a balance must be found between the optimal heating element designs, and the flow channel geometry. The flow channels must be designed such that the flow vaporisation and heating is as homogeneous as possible. The heating elements need to have enough contact area with the flow to conduct the necessary amount of heat, while maintaining structural integrity. The heaters need to be sized such that they produce the right amount of resistance at the operating temperature, in order to dissipate the set amount of power at the fixed potential difference of 5 [V].

When using conventional machining techniques, there are only a limited number of options for the propellant flow channel geometry, depending on the product availability. However, since the heating chamber will be a MEMS structure, the geometry and internal features are totally up to the designer and only bounded by the precision constraints of the MEMS manufacturing. Infinite possibilities exist to design the structures inside the heating chamber section. But the most logical choices can be separated in the following main categories:

- No structures, just a simple open rectangular cross-section, as shown in Figure 7.8
- Channels or straight fins parallel to the flow direction, shown in Figure 7.18
- Winding or serpentine channels or fins, shown in Figure 7.19, Figure 7.20 and Figure 7.21
- Free standing pillars or fins, shown in Figure 7.22, Figure 7.23, Figure 7.24 and Figure 7.25

The open rectangular cross-section in Figure 7.8 can also be seen as a straight flow channel design, as seen in Figure 7.18, but with only one channel. Therefore two main heating chamber geometry groups can be identified. One group has flow channels, which are shown in the images below. Of course, all kinds of differently shaped channel geometries can be thought of. However in the interest of time and common sense, only the most basic line types where chosen for this comparison.
The second group of heating chamber geometries has differently shaped, free standing pillar or fin structures in the heating chamber. These are shown in the images below.

The shape of the aerofoil pillars are defined using the equations for the symmetrical 4-digit NACA aerofoil:

\[
y_1 = \frac{t}{0.2} \left[ 0.2969 \sqrt{\frac{x}{c}} - 0.1260 \left( \frac{x}{c} \right) - 0.3516 \left( \frac{x}{c} \right)^2 + 0.2843 \left( \frac{x}{c} \right)^3 - 0.1015 \left( \frac{x}{c} \right)^4 \right]
\]
Where:

\[ y_t \quad \text{Half thickness of the chord to surface [m]} \]
\[ t \quad \text{Maximum thickness as a fraction of the chord length [-]} \]
\[ c \quad \text{Chord length [m]} \]
\[ x \quad \text{Position along the chord from 0 to c [m]} \]

Using this formula, the thickness at the trailing edge does not totally converge to zero. This is problematic when using this equation in the computer modelling. Therefore the last coefficient is changed from \(-0.1015\) to \(-0.1036\). This has the smallest effect on the overall geometry, while making the thickness converge to zero at the trailing edge.

### 7.4.2 Propellant heating chamber geometry selection

In order to prevent the possibility of propellant reaching the nozzle without being vaporised, the flow has to be forced to make contact with the heating elements. This incomplete vaporisation is also avoided by properly mixing the flow. In the flow in straight channels as seen in Figure 7.18, flow stratification might occur as is shown in Figure 7.26. The straight channel is the simplest possible heating chamber geometry. However, especially in the absence of gravity, it the fluid flow stratifies in the middle of the flow, it could continue a straight path to the exit, without contacting the heaters or the hot walls. It can then pass through the heating chamber without properly heating up or even without vaporisation. When there are obstructions or winding channels, the liquid is forced to mix with the other flow or contact the surface because the movement in a straight line is broken. As many problems with incomplete vaporisation were encountered in the past, the straight channel design is therefore discarded.

![Figure 7.26 - Two-phase flow types [62]](image)

The partially liquid flow shown in Figure 7.26 would potentially severely reduce the performance of the thruster. If the internal heat transfer within the flow is therefore not fast enough, the geometry of the heater chamber has to be optimised to increase the contact area between the hot chamber walls and the flow. This can be done by having winding serpentine channels, roughing up the channel walls or placing obstructions (fins) in the flow.
In the section where the propellant is gaseous, it needs to be ensured that there is no part of the geometry which could act as a throat nozzle. This would make the flow sonic in this section and lead to large pressure losses. Therefore the narrowest section in the heater must be at least 3 times larger than nozzle throat [37]. This will ensure that the flow speed will not become sonic before the throat.

For the sinusoidal serpentine heating channels shown in Figure 7.21, this might become an issue, due to the large decrease in width in the narrow sections of this flow. The greatly fluctuating channel width will also result in cycles of large flow acceleration and deceleration. This will always be accompanied by losses in the flow. Due to the complexity of this channel, it was unfortunately not possible to model the flow properties through it, during the scope of this thesis. Therefore the sinusoidal serpentine channel design is also not further evaluated.

In order to evaluate the differences in flow properties in the triangular and the semi-circle serpentine design, the flow through these sections was modelled using the COMSOL software. The result of which is shown in the two figures below. The mass flow rates through these sections were chosen to be in the range of 1 [mg/s] across all heating channels combined. As was shown in chapter 3, that is the maximum expected mass flow rate for the Dondersteen thruster. The working fluid was chosen to be liquid water at room temperature.

As can be seen from the images above, the flow pattern through both channels is quite different. The Semi-circular serpentine channel in Figure 7.27 shows a very smooth velocity profile, with relatively small boundary layers on the sides. The triangular serpentine channel in Figure 7.28 shows a far more irregular flow pattern. The flow can be seen to stagnate in the corners of the channels, increasing the velocity in the centre of that section. It can also be seen that there is a far larger boundary layer of stagnated flow on the inside triangle of the flow. This is seen by the fact that the blue layer, of low velocity flow, is thicker on one side of the channel than on the other. Therefore the losses in the flow of the semi-circular design are assumed to be lower, and the heat transfer more optimal. The semi-circular channel design is therefore the flow channel design which will be produced for the thruster tests. To further verify this assumption, this can also be compared to the flow around aerodynamic bodies. Where the semi-circular serpentine resemble the flow around a cylinder, and the triangle...
shape resembles the flow around an angled cube or rhombus. For the same Reynolds number of the flow, the drag coefficient of the flow around a rhombus is about 1.5 or 2 times as high as that around a cylinder. Therefore the angular channel design is expected to induce more drag.

As a first analysis of the heating module designs with the pillar elements in the flow channel, the necessarily for alternating pillars was assessed. In previous research, designs have also included non-alternating pillar designs. The flow velocity patterns through both alternating and non-alternating designs are assessed using the COMSOL software. The result of which is shown in the images below.

In Figure 7.29 the flow can clearly be seen to stagnate behind the pillars. The flow velocity magnitude also shows that the flow velocity is very irregular. The flow constantly decelerates where there are no pillars, and accelerates in between pillars. As can be seen from the streamlines, virtually no particles flow in the space behind the pillars, resulting in a lot of useless space. Some of the streamlines are also virtually straight from beginning to end. This means that flow stratification and incomplete vaporisation can occur. In Figure 7.30 some flow stagnation can be seen in front of and behind the pillars. However the velocity magnitude profile between the pillars is very uniform. This means that there are no phased flow accelerations and decelerations. In the streamline figure it is also visible that the channel is a lot more optimally used. Additionally, none of the streamlines passes through in a straight line, which avoids stratification.

The conclusion from this is that alternating pillars are required for a more stable flow pattern, a more efficient use of the flow channel and a more secure vaporisation. In the following figures the flow pattern investigation of circular and diamond shaped pillars is depicted.
As can be seen from Figure 7.31, the flow pattern around the circular pillars is quite irregular. There is a lot of flow stagnation before and behind the pillars. And in between the pillars there is flow acceleration. During the simulations a rhombus shaped flow stagnation region was identified around the circular pillars. In Figure 7.32 a very constant velocity flow pattern can be seen around the diamond shaped pillars. There is a slight flow deceleration in between pillar rows. As can be seen from the streamline figure, the particle flow is almost uniformly divided across the entire channel space. Comparatively very little flow stagnation is encountered in front of or behind the pillars and the observed boundary layers are a lot smaller than for the other geometries. This also gives reasoning behind why the diamond shaped pillars were found to perform best in micro evaporator structures previously investigated at Dimes [63].

Therefore it can be concluded that the diamond shape pillars create the best flow pattern. This results in small boundary layers, minimal flow stagnation and homogeneous flow velocities. Therefore these pillar designs were chosen for the production of the physical test models of the thruster.

7.4.3 Necessary electrical resistance calculations
The heat produced by a resistive element is governed by Joules law:

\[ P_{heat} = I^2 \cdot R \]  

\[ 7-2 \]

Where:

- \( P_{heat} \) Dissipated heat [W]
- \( I \) Electrical current through the resistive element [A]
- \( R \) Electrical resistance of the material [Ω]
The electrical resistance in its turn is determined by resistor geometry and resistivity [63]:

\[ R_0 = \frac{\rho \cdot L}{w \cdot h} \]  

7-3

Where:
- \( R_0 \) The electrical resistance at the reference temperature [Ω]
- \( \rho \) Resistivity of the resistive material [Ω·m]
- \( L \) Length of the resistor [m]
- \( w \) Width of the resistor [m]
- \( h \) Height of the resistor [m]

Since the resistivity of a material depends heavily on temperature, the temperature dependent resistance is calculated with the following formula [63]:

\[ R(T) = R_0 \left( 1 + \alpha \cdot \Delta T \right) \]  

7-4

Where:
- \( \alpha \) The Temperature Coefficient of Resistance (TCR) [/K]
- \( \Delta T \) The temperature rise compared to the reference temperature [K]

Unfortunately a simple mistake has been made in this part of the calculations, which has profound consequences for the resulting thrusters. The value of the resistivity of the SiC was wrongly converted from the paper by Mr. Morana. This led to a wrong design of the resistive elements, which were now designed to yield 100 times the required electrical resistance. SiC, has a resistivity of \( \rho = 1.27 \times 10^{-4} \) [Ω·m] (the calculations were made with a value of \( \rho = 1.27 \times 10^{-6} \) [Ω·m]) when doped with 15 [sccm] NH_3 [64]. The resistivity decreases to \( 0.38 \times 10^{-4} \) [Ω·m] (the calculations were made with a value of \( \rho = 0.38 \times 10^{-6} \) [Ω·m]) when doped with 55 [sccm] NH_3. The calculations which will follow that depend on the resistivity will therefore not have the correct values. The TCR of SiC is \( \alpha = -0.83 \times 10^{-3} [/K] \) when doped with 15 [sccm] NH_3 and \( \alpha = -0.5 \times 10^{-3} [/K] \) when doped with 55 [sccm] NH_3.

When connected in series, the total resistance is simply the sum of the resistances of the individual elements, as shown in equation 7-5.

\[ R_{\text{tot}} = R_1 + R_2 + \cdots + R_n \]  

7-5

When the resistive elements are placed in parallel, the reciprocal of the total resistance will be equal to the sum of the reciprocals of the individual elements, as shown in equation 7-6.

\[ \frac{1}{R_{\text{tot}}} = \frac{1}{R_1} + \frac{1}{R_2} + \cdots + \frac{1}{R_n} \]  

7-6

Therefore this simplifies to equation 7-7, when all \( n \) individual elements have identical resistance.

\[ R_{\text{tot}} = \frac{R_1}{n} \]  

7-7
7.4.4 Design and lay-out of the resistive heating element

For the prototype model all heating elements will span the entire width of the heating chamber. The suspended heating elements are laid on top of the pillar, in order to structurally support them across the heating chamber. Therefore the pillars or flow channel walls have a dual purpose. On the one hand they divide and mix the flow, and on the other, they support the heating elements. A longitudinal cross-section of such a heating chamber module is shown in Figure 7.33. All the important elements can be identified. The flow channel which is filled with the diamond shaped pillars. The heating layer on top has three different sections:

1. On the sides there are the bond surfaces, where the electrical connections are bonded to the power supply.
2. In between the pillars are the suspended heating elements, which dissipate most of the heat, and conduct it to the passing flow.
3. On top of the pillars is a wider continuation of the heating layer. This will dissipate some heat to the pillars, which in turn will heat the passing flow.

![Longitudinal cross-section of one heating module with diamond shaped pillars. Including the suspended heating elements and the bond surfaces on the side. The indicated gaps show the physical separation between different heating elements.](image)

In discussions with the manufacturers at DIMES a heater membrane thickness of 100 [nm] to 300 [nm] was said to be achievable. This was confirmed when making the heater geometries, which was designed for a SiC layer with 200 [nm] thickness. During the week of production this thickness range indication was suddenly changed to 300 [nm] to 500 [nm]. Calculations were quickly conducted to investigate how much the resistivity should be changed to accommodate for this new thickness, while maintaining the same geometry.
Due to the modular design it is easy to put multiple heating modules behind one another. Each heating module is therefore designed to produce the practical amount of 1 [W] of heating power. The heating elements will be connected to an external power source using bond pads. Based on their actual performance, multiple heating elements will be placed in parallel or in series in a multitude of different configurations. Since a new set of thrusters will be manufactured for the flight model this can incorporate these inter-element connections on the silicon wafer itself. Instead of having seven separate connections going to each heater module, this will only require one electrical connection to the control system. This will greatly reduce the time required for manufacturing. Due to the fact that fewer wires need to be bonded, and the configuration is already implemented in one production step. To increase the reliability the single electrical bond pad can still be using several bond wires. If one bond wire breaks, the others will still be able to supply the necessary electricity to the element.

The advantage of these heating elements is that they can not only be used for heating, but also as a thermistor. As told before, the resistivity of materials depends on the temperature in the relationship shown in equation 7-4. When the TCR of the material is experimentally determined, the temperature of a material can be derived from the electrical resistance. This characteristic is exactly what will be used in the thrusters to assess the temperature of the heaters. During the testing campaign this will enable the use of the unpowered heating elements as temperature sensors, in order to measure the flow temperature. For the final flight model of the propulsion system, these measurements will be used to keep the heating elements at the desired 500 [°C].

7.4.5 Diamond shaped fins heater
In the pillar flow channel design, the heating section has the same geometry as MEMS evaporators previously investigated at DIMES [63]. As the diamond shaped fins were shown to have the most stable behaviour and less problems with back-flow, these were chosen to be the nominal design case. An example of a simplified layout of a MEMS heater using diamond shaped fins is shown in Figure 7.34. This image includes some of the characteristic dimensions for these heating modules. A more accurate image of the final nominal design case was already shown in Figure 7.33.

![Figure 7.34 - MEMS heater with diamond shaped fins [63]. k is the pillar length, m is the minimum distance between pillars, h is the maximum distance between pillars, l is the width of the pillars.](image)

This heating chamber configuration consists of a rectangular chamber, with diamond (rhombus) shaped fins, of which the largest diameter is placed in parallel with the flow direction. One could already look at Figure 7.39 and Figure 7.40 for a reference. The heater layer spans between these pillars in direction perpendicular to the flow direction. The layer occupies the centre of the pillars and
is suspended between these pillars in a rectangular flat shape. Since there are multiple rows of fins placed behind one-another, the flow then also passes through the rows of these heating filaments.

To calculate the resistance across one of these resistive heating element rows, the geometry is simplified. The most basic repetitive shape is a combination of a rectangular suspended section and a trapezoidal section on the pillars, as shown in Figure 7.35.

![Figure 7.35 - Simplified heater element for half a pillar](image)

The rectangular section is comprised of half of the suspended heating element, and the corner of the diamond shaped pillar. The trapezoidal section is the remaining part of half of the diamond element. Therefore:

\[ L_r = \frac{1}{2} L_{sus} + \frac{w_s \cdot w_{fin}}{2 \cdot L_{fin}} \]

Where:
- \( L_r \): Length of the rectangular section [m]
- \( L_{sus} \): Length of the suspended element (distance between two pillars) [m]
- \( w_s \): Width of the suspended element [m]
- \( w_{fin} \): Width of the fin with respect to the flow [m]
- \( L_{fin} \): Length of the fin with respect to the flow [m]

\[ L_t = \frac{1}{2} w_{fin} - \frac{w_s \cdot w_{fin}}{2 \cdot L_{fin}} \]

Where:
- \( L_t \): Length of the trapezoidal section [m]

For the rectangular section, equation 7-3 is used to calculate the resistance of the element at the reference temperature. For the trapezoidal section, the change in width has to be taken into account. The width of the trapezoidal section from left to right, as seen in Figure 7.35, can be calculated using simple trigonometric rules.
Design, manufacturing and characterisation of a water fed CubeSat micro-resistojet

\[ w_t(x) = \left( \frac{L_{\text{fin}} - w_s}{L_t} \right) \cdot x + w_s \]  \hspace{1cm} 7-10

Where:

\[ w_t(x) \]  Width of the trapezoidal section as a function of the distance of the narrow section of the trapezoid [m]
\[ x \]  Distance from the narrowest section of the trapezoid [m]

Taking into account the variable width of the trapezoidal section equation 7-3 can be rewritten as follows:

\[
R_0 = \frac{L}{\int_0^L \frac{\rho}{w_t(x)} \cdot h \cdot dx} = \frac{L}{\int_0^L x \cdot h \cdot w_t(x) \cdot dx} = \frac{L}{\int_0^L \frac{x}{w_s} + \left( \frac{L_{\text{fin}} - w_s}{L_t} \right) \cdot x \cdot dx} \hspace{1cm} 7-11
\]

The integral can be solved using the following standard integral solving method [65].

\[
\int \frac{u \cdot du}{a + b \cdot u} \cdot du = \frac{1}{b^2} \cdot (a + b \cdot u - a \cdot \ln|a + b \cdot u|) + C \hspace{1cm} 7-12
\]

Using this relation, equation 7-11 can be written as:

\[
R_{0_{\text{trap}}} = \frac{\rho}{h} \cdot \frac{1}{\left( \frac{L_{\text{fin}} - w_s}{L_t} \right)^2} \left( \left( w_s + \frac{L_{\text{fin}} - w_s}{L_t} \cdot \ln \left| w_s + \frac{L_{\text{fin}} - w_s}{L_t} \right| \right) - \left( w_s - w_s \cdot \ln|w_s| \right) \right) \hspace{1cm} 7-13
\]

The total resistance of electrical resistors in series is the sum of the resistances of the individual resistors. Therefore the resistance of the section shown in Figure 7.35 is the sum of the resistances of the straight part and the trapezoidal part. The resistance of the heater per fin is then two times this amount. And the resistances per row of fins, is the amount of fins, times the resistance of the heater of each fin. i.e.:

\[
R_{0_{\text{row}}} = 2 \cdot \left( R_{0_{\text{trap}}} + R_{0_{\text{straight}}} \right) \cdot \text{no}_{\text{rows}} \hspace{1cm} 7-14
\]

Where:

\[ R_{0_{\text{row}}} \]  Resistance of the resistive heater for each row of fins [Ω]
\[ R_{0_{\text{straight}}} \]  Resistance of the straight part in Figure 7.35 [Ω]
\[ R_{0_{\text{trap}}} \]  Resistance of the trapezoidal part in Figure 7.35 [Ω]
\[ \text{no}_{\text{rows}} \]  Number of fins in one row [-]
The resistance at operational temperature is calculated using equation 7-4. This actual resistance, together with the known system voltage of 5 [V] is used to calculate the power consumption of the resistor using Joule’s law in equation 7-2 and Ohms law.

The results of the calculations were checked by running simulations on the same geometries with the COMSOL program. For these diamond shaped resistors, the resistance was found to be 1.65 times higher in the simulations ran on COMSOL than what the simplified calculations showed. When looking at the current density through the diamond elements it was immediately clear why this is the case. In the simplified calculations, it was assumed that the current density would be homogeneously distributed across the width of the diamond heaters. In the simulations however it showed that there was virtually no flow in the tip of the elements, and most of the current went through the centre of the diamonds following the line of the suspended heating elements. The conversion coefficient of 1.65 was therefore used to convert the simplified calculations into the, more accurate, simulated results. This conversion coefficient is only valid when the ratio between the pillar length, pillar width, heating element width and heating element length is in the region as which is used in the design shown here. To make the calculations more accurate, the final flight model design will have to be simulated with COMSOL in order to get the most exact results possible.

Heating and subsequent heat losses of the resistive heater layer electrical connection bonding area are neglected. It is assumed that since the width of this section is significantly larger than the heating elements in the flow, the losses in the sides are negligible. This section will also be kept as short as possible, in order to further minimise losses.

The pillar dimensions could not be theoretically optimised within the time schedule of this thesis. Therefore the dimensions are taken from the previously tested micro-evaporators, by Marko Mihailović [63]. These dimensions were derived within the much larger timeframe of a PhD thesis, and were shown to be effective. The dimensions used for this design are shown in Table 7.1, and are clarified in Figure 7.34. The documented performance is also shown in this table, some of which is clarified in Figure 7.36.

Table 7.1 - Characteristic dimensions and performance of a water fed micro-evaporator [63], the meanings of the abbreviations are given in Figure 7.34.

<table>
<thead>
<tr>
<th>Parameter</th>
<th>Dimension</th>
<th>Unit</th>
</tr>
</thead>
<tbody>
<tr>
<td>Fin length, k</td>
<td>160 · 10^{-6}</td>
<td>[m]</td>
</tr>
<tr>
<td>Fin width, l</td>
<td>40 · 10^{-6}</td>
<td>[m]</td>
</tr>
<tr>
<td>Maximum channel width, h</td>
<td>60 · 10^{-6}</td>
<td>[m]</td>
</tr>
<tr>
<td>Minimum channel width, m</td>
<td>20 · 10^{-6}</td>
<td>[m]</td>
</tr>
<tr>
<td>Total heater section width</td>
<td>2680 · 10^{-6}</td>
<td>[m]</td>
</tr>
<tr>
<td>Total heater section length</td>
<td>4000 · 10^{-6}</td>
<td>[m]</td>
</tr>
<tr>
<td>Side wall thickness</td>
<td>325 · 10^{-6}</td>
<td>[m]</td>
</tr>
<tr>
<td>Maximum vaporised mass flow</td>
<td>1.1 · 10^{-6} - 1.4 · 10^{-6}</td>
<td>[kg/s]</td>
</tr>
<tr>
<td>Maximum power input</td>
<td>3</td>
<td>[W]</td>
</tr>
</tbody>
</table>
The maximum vaporised mass flow rate through this reference evaporator is in the same range as the required mass flow for the MEMS resistojet. The exit temperature is a lot lower than required for the Dondersteen thruster. In the MEMS resistojet, the flow will therefore be heated in gas phase for a longer duration of time. Gaseous propellant with the same mass flow rate as the liquid propellant will need more volume. The maximum input power will also be around 1.8 to 2.3 times higher.

For these reasons it is likely to assume that the heating volume and heater contact area also have to be at least twice that of the micro-evaporator. In order to keep the design as comparable as possible, and for reasons of nozzle performance, the channel height is kept at the same 100 [μm] as for the micro-evaporator. Propellant “spitting” can occur when liquid water is propelled forward by boiling flow behind it. If the liquid is propelled with such a velocity that it cannot be vaporised anymore within the subsequent flow path, this could flow through the nozzle as a liquid. This drastically reduces the efficiency of a resistojet. In order to prevent this, the propelled liquid, needs to be forced to make contact with the heater area. Therefore the width of the heating section is kept the same as for the micro-evaporator. Only the heating chamber length, of the nominal design, is increased by at least a factor two. The heating section will therefore have a length of at least 8 [mm]. This ensures that the propellant flow path is long enough to vaporise and heat the propellant completely and relatively uniformly.

Since the length of each fin is 160 [μm], and the total length has to be 1.00 [cm], there are 57 Rows of fins in the nominal design. The heating chamber designs are built up out of sections consuming 1 [W] of electrical power. Since in the nominal design, the power consumption is 7 [W], each section will have 8 rows. This also complies with the fact that each section should have an even number of rows, since the pillars have to be alternating. This will make the total length of the heater section 8,960 [μm] long. The width of the heating section is chosen to be 3,000 [μm], for convenience, since that means that there are 50 fins per row.

Using the above mentioned equations, the power consumption in this section can be calculated as a function of the suspended heating element width. This relation is shown for the rows connected in parallel in Figure 7.37 and for the rows connected in series in Figure 7.38.
As can be seen, the width of the suspended element in parallel connection has to be around 8 [μm], to achieve a total power consumption of 1 [W]. Since the distance between the fin tips is 20 [μm] the aspect ratio of the main span is 2.5. This aspect ratio is assumed to be adequate for a structurally sound design. This has also been discussed with Mr. Morana, who is the most experienced expert on SiC heating elements available at Dimes. The width of the suspended element in series connection however has to be around 0.11 [μm], to have a total power consumption of 1 [W]. This means an aspect ratio of the main span of 182. This aspect is too high for a structurally sound design, which can withstand the necessary loads. Additionally it will most probably pose a problem for manufacturing as well. Therefore all rows in one heater section will be connected in parallel, which is also easier for connecting and testing purposes. In addition to that, from a redundancy point of view parallel connections are a lot better. Since if one fails, the others can still produce power. When the rows of heating elements would be connected in series, a failure at any point in the chain of heating elements will render the whole heating module useless.

In order to have enough space to bound the power supply cables to on the bound pads, the wall thickness is increased to 2000 [μm]. This also makes sure that the side walls will be able to cope with the required chamber pressures. The final nominal design of one heating section with diamond shaped fins is shown in Figure 7.39 and Figure 7.40.
In order to compare the effects of pillar size on the performance of the heater module, a second diamond pillars heating chamber module was designed. To give a proper base of comparison the diamond pillars were increased in size by a factor four. This should be significant enough to show some consistent difference with the nominal design. The important dimensions and performance of the two diamond pillar designs are shown in Table 7.2.
### Table 7.2 - Most important design and performance parameters of the heater modules

<table>
<thead>
<tr>
<th>Parameter</th>
<th>Diamond fins nominal</th>
<th>Diamond fins larger</th>
<th>Unit</th>
</tr>
</thead>
<tbody>
<tr>
<td>Fin length, ( k )</td>
<td>( 160 \cdot 10^{-6} )</td>
<td>( 640 \cdot 10^{-6} )</td>
<td>[m]</td>
</tr>
<tr>
<td>Fin width, ( l )</td>
<td>( 40 \cdot 10^{-6} )</td>
<td>( 160 \cdot 10^{-6} )</td>
<td>[m]</td>
</tr>
<tr>
<td>Maximum channel width, ( h )</td>
<td>( 60 \cdot 10^{-6} )</td>
<td>( 240 \cdot 10^{-6} )</td>
<td>[m]</td>
</tr>
<tr>
<td>Minimum channel width, ( m )</td>
<td>( 20 \cdot 10^{-6} )</td>
<td>( 80 \cdot 10^{-6} )</td>
<td>[m]</td>
</tr>
<tr>
<td>Total heater section width</td>
<td>( 3,000 \cdot 10^{-6} )</td>
<td>( 3,000 \cdot 10^{-6} )</td>
<td>[m]</td>
</tr>
<tr>
<td>Total heater section length</td>
<td>( 1,280 \cdot 10^{-6} )</td>
<td>( 1,280 \cdot 10^{-6} )</td>
<td>[m]</td>
</tr>
<tr>
<td>Side wall thickness</td>
<td>( 500 \cdot 10^{-6} )</td>
<td>( 500 \cdot 10^{-6} )</td>
<td>[m]</td>
</tr>
<tr>
<td>Design power input</td>
<td>( 1.023 )</td>
<td>( 1.006 )</td>
<td>[W]</td>
</tr>
</tbody>
</table>

#### 7.4.6 Semi-circle channel design

This design has channels of semi-circles in alternating directions behind one another, as shown in Figure 7.41.

![Figure 7.41 - Semi-circular heating channel design example](image-url)

The most important difference between this heating chamber design and the diamond pillar one is the flow patterns it will induce. These flow patterns will determine the pressure drop, heat transfer rate within the flow and the heat losses to the surroundings. The pressure drop and heat losses need to be as small as possible, for an as large as possible heat transfer rate. To be able to objectively assess the performance differences between both designs, the total module size and the cross-sectional channel area are chosen to be as similar as possible.

The volume fraction of the channel in the heating section of the diamond shape is can be calculated easily. The area of the diamond shaped fins is:

\[
A_{\text{fin}} = \frac{1}{2} w_{\text{fin}} \cdot L_{\text{fin}} \tag{7-15}
\]
The heating chamber area per fin is:

\[ A_{\text{sec}} = x_{\text{fin}} \cdot y_{\text{fin}} \]  \hspace{1cm} 7-16

Where:

- \( x_{\text{fin}} \) Lateral distance between the centre of the fins [m]
- \( y_{\text{fin}} \) Lateral distance between the centre of the fins [m]

The volume fraction of the channel in the semi-circular design is relatively easily defined for a section with the length of one wave. This is shown in Figure 7.42 below.

![Diagram](image)

**Figure 7.42 - A section with 1 wave across the entire width of the heating chamber**

From the geometrical properties, it can be seen that the volume fraction of the channel in such a section is:

\[ \phi_{\text{channel}} = \frac{\pi (r_{\text{out}}^2 - r_{\text{in}}^2)}{4 \cdot r_{\text{out}} (r_{\text{out}} + r_{\text{in}})} \]  \hspace{1cm} 7-17

Where:

- \( \phi_{\text{channel}} \) Volume fraction of the channels [-]
- \( r_{\text{out}} \) Outer radius of the semi-circle [m]
- \( r_{\text{in}} \) Inner radius of the semi-circle [m]

The volume fraction of the channel in the diamond shaped fin heating chamber design can be calculated to be 67%. For simplicity, the element length for the semi-circular sections is taken to be equal to the diamond fin length of 160 [µm]. If the outer radius is then chosen, the length of the inner radius follows from equation 7-17. In Figure 7.43 the resulting volume fraction of the channel is shown for a number of outer radii.
A channel volume fraction of 67% can be seen to be achieved at an outer radius of 70 [μm] and a resulting inner radius of 10 [μm]. This gives a total width per wave of 140 [μm]. For a total heater width of 3000 [μm], there will be 21 channels next to one-another. And to make the semi-circle design as comparable with the diamond fin design as possible, the length of the heating section module will also be chosen to be 1280 [μm]. This means each channel will make a total of 8 waves, per heater module.

For simplicity, the heating elements in this design were chosen to be straight rectangular beams from one side of the heating chamber to the other. They are supported throughout the heating chamber by the walls of the serpentine channels. As said before, due to the fact that the electrical potential difference is set to be 5 [V], the resistance of each modules must be 25 [Ω] in order to produce 1 [W] of electricity. This information is used in equations 7-3 and 7-4 in order to calculate the necessary heating element width. This heating element design is shown in Figure 7.44. The finally manufactured geometry is shown in Figure 7.45. The calculations of the resistance of these heating elements were also checked by simulating the current in COMSOL. In this case the simulations and the simplified equations matched 100%.
Also for this geometry a roughly four times large design was made. This was done in order to test the influence of the size of the serpentine channels on the thruster performance. The smaller serpentine channels are expected to have a larger heat transfer to the propellant flow, due to their larger contact surface. However this increased contact area is also expected to induce a larger pressure loss. The geometry for this larger serpentine design is shown in Figure 7.46.

The relevant design parameters for both semi-circle serpentine heating modules are shown in Table 8.1. In this table a comparison is made between the designed dimensions of the heating modules and the actual dimensions after production.

7.4.7 Electromigration
Small scale electronics in MEMS such as in this heater run the risk of failure due to electromigration, when operating at high current density [66]. Electromigration is the physical movement of conductive material, on an atomic-scale, due to the electrical current. The collisions between the electrons and atoms in the Ohmic conduction regime, results in a momentum exchange between the electrons and atoms. For accurate calculations on the impact of this phenomenon on the lifetime of electronic parts, both extensive analytical and numerical calculations are used. However, they quickly go a lot more into depth than is required for this design. The goal is to keep far away from the current densities where electromigration becomes an issue. Therefore the traditional black’s equation is usually used to get an order of magnitude estimate on the Mean Time To Failure (MTTF):
Design, manufacturing and characterisation of a water fed CubeSat micro-resistojet

\[ MTTF = A \frac{w \cdot t}{j^2} \cdot e^{\frac{E_a}{kT}} \]

Where:
- \( A \): Constant depending on technology features
- \( w \): Width of the conductor [m]
- \( t \): Thickness of the conductor [m]
- \( j \): Current density [A/m\(^2\)]
- \( E_a \): Activation energy [eV]
- \( k \): Boltzmann’s constant, \( 1.3806488 \cdot 10^{-23} \) [J/K]
- \( T \): Absolute temperature [K]

The necessary property is however not very well defined or known for the doped SiC which is used in the heating layers of the Dondersteen thruster. Thus to get an indication, some reference values of maximum allowable current densities need to be used. The maximum current density calculated for designs that were discussed in the previous subsections is \( 2 \cdot 10^6 \) [A/cm\(^2\)]. As a reference, aluminium can withstand a current density of up to \( 10^5 \) [A/cm\(^2\)] and copper can withstand \( 10^6 \) [A/cm\(^2\)]. This is over a period in the order of 10 years. Therefore the much harder SiC should not have problems for a relatively short duration of time at \( 2 \cdot 10^6 \) [A/cm\(^2\)], during less than one month of operation.

### 7.4.8 Pressure drop calculation

The performance parameters of the geometry including fins in the heating channel will be calculated using traditional packed bed calculations. Since actual in depth numerical or analytical calculations of such flow is not within the scope of this research. The Author assumes that especially for cylindrical pillars, the flow will resemble the flow across spherical elements in a packed bed. This assumption will of course have to be validated by the experimental data further into the research.

The pressure drop across the section will be calculated using the Ergun pressure drop correlation [62].

\[
\Delta P = 150 \cdot \frac{(1-\varepsilon)^2}{D_p^2 \cdot \varepsilon^3} \cdot \mu \cdot U_s \cdot L + 1.75 \cdot \frac{(1-\varepsilon)}{D_p \cdot \varepsilon^3} \cdot \rho \cdot U_s^2 \cdot L
\]

Where:
- \( \Delta P \): Pressure drop [Pa]
- \( \varepsilon \): Ratio between the gas volume over the total volume (packed bed porosity) [-]
- \( D_p \): Particle diameter [m]
- \( \mu \): Fluid viscosity [N·s/m\(^2\)]
- \( U_s \): Superficial or free stream fluid velocity [m/s]
- \( L \): Packed bed length [m]
- \( \rho \): Fluid density [kg/m\(^3\)]

All these parameters speak for themselves; except for the superficial fluid velocity \( U_s \). This is the velocity which the stream would have if the packed bed volume would not contain the pebbles. For the liquid phase this free stream velocity is easily calculated.
\[
U_s = \frac{\dot{m}}{A \cdot \rho}
\]

Where:
\(\dot{m}\) Propellant mass flow rate [kg/s]
\(A\) Empty channel cross-sectional area [m²]

The mass flow rate is assumed to be 1 [mg/s], as calculated in chapter 3. The porosity of the diamond shaped heating geometry is 0.67 [ - ] as shown in subsection 7.4.6. The particle diameter is assumed to be the width of the pillar, which is 40 [μm]. The viscosity of liquid water at 100 [°C] is 0.282·10⁻³ [N·s/m²]. The length of the chamber is 8.96 [mm]. The density of liquid water at 100 [°C] is 958.4 [kg/m³]. This all results in a pressure drop of for a flow through of liquid water only. The pressure drop is then calculated to be 127,569 [Pa] = 1.28 [bar]. This connects relatively well with the assumption made of 20% of the pressure, which is 0.9 [bar]. Due to the much smoother and regulated flow patterns, the proposed resistojet is however expected to witness less of a pressure drop. The same can be done for a full flow of gaseous water. If the results are then averaged, it will come even closer to the true value. Due to time constraints no further elaboration was attempted.

### 7.5 Nozzle section

Most of the characteristic nozzle dimensions were already decided in chapter 3. To reach the desired operating performance, the nozzle throat diameter must be 25 [μm]. As described in sub-section 3.4.2, possible performance decreases are dependent on geometrical design properties. Therefore the nozzle curvature, indicated in Figure 3.2 with \(R_u\) will be 0 [m]. In other words, the nozzle throat will be a sharp-edged angle between the converging and diverging sections. As shown in Appendix C, the most optimum ratio for the converging and diverging angles is 15° and 20° respectively. If the nozzle would be made with these angles in a simple conical shape, this would lead to a very long nozzle. A long nozzle would in turn lead to large heat and pressure losses, due to conduction and friction at the wall.

As stated before, all the flow channels besides the nozzle throat should have a diameter of at least three times the throat diameter. With a diameter larger than this value, no premature shockwaves or nozzle behaviour will be formed [37]. Therefore it was decided that the divergent angle would be chosen to be a steep 45°, in order to minimise the length. At a diameter of three times the throat, the converging angle will then change to 15°, in an effort to have the maximum nozzle efficiency. As mentioned in chapter 3 the length of the nozzle can be reduced to about 75% of the conical length, while having the same area ratio and thrust coefficient \(C_F\), using a bell shaped nozzle [22].

The bell shape is a simple type of parabola of which the function is derived by the theory explained by ir. Zandbergen [22]. The diverging angle behind the throat is a standard 60° [22] and the exit angle is chosen to be the mentioned optimal for MEMS nozzles of 15° [37]. This reduced length will further reduce the possible heat loss through the nozzle walls. This nozzle is therefore designed for optimum performance in a slightly experimental shape. A comparison between the diverging section of a conical nozzle and a bell nozzle, with the same exit angle and throat diameter, was made. This comparison is shown in Figure 7.47, clearly showing the smaller length of the bell shaped nozzle. The final shape of this optimised nozzle is shown in Figure 7.48 and the design parameters are summated in Table 8.1.
The primary design will however be a more conventional conical shape. By choosing a converging half-angle of 30° and a diverging half-angle of 20°, this nozzle should still be able to achieve a nozzle efficiency of up to 91.1% [37]. The disadvantage however is the longer length, with might lead to more losses in the flow. The final design of this nozzle is shown in Figure 7.49 and the design parameters are summated in Table 8.1.

### 7.6 Heat transfer

A number of attempts have been made to calculate the heat transfer of the thruster. These calculations were conducted in order to substantiate the assumptions made in the calculations of chapter 3. A number of heat transfer cases can be identified for this thruster:

- From the heating element, to the flow
- From the flow to the thruster walls
- Internal conductance within the thruster
- Conductance to the external structure
- Radiation to the surroundings of the thruster
All these attempts could not be conducted successfully within the timeframe of this thesis. The numerical calculations attempted with the aid of literature required too many assumptions. These assumptions could not be made sensibly at the time the calculations were conducted. The calculations as far as they were conducted are documented in Appendix H. Another attempt was made by using the COMSOL software. This software was actually successfully used to make some qualitative liquid flow trade-offs and to verify the resistive heating element calculations, both in this chapter. The attempts to simulate the heating of the flow were halted as they were too time-consuming. In order to produce the thrusters in time a more qualitative approach was taken and results of earlier research served as a reference.

7.7 Manufacturing

In order to assess the influence that the different design parameters have on the performance a number of different module configurations were used. How the desired combinations of different thruster modules form the test thrusters, is best shown in Table E.1 in Appendix E. The reasoning behind the selection is also shown in this table. The manufacturing had to be rushed, due to the imminent maintenance closure of the production facilities. This led to the fact that the desired combinations of thrusters were not all produced. Some of the combinations were devised by the manufacturers on the spot. The main distinction between the designs is the length. One selection of thrusters is designed to produce 7 [W] of heating power. These are the ones that are designed to the specifications of the DelFFi mission propulsion system. The other thrusters are designed to study the influence of length on the performance of the thrusters. These have a series of 14 heater modules as a heating chamber, capable of producing 14 [W] of heat. These will however not all be used to heat the propellant flow. This length and number of heating elements does facilitate the possibility of conducting elaborate tests. Hereby some of the elements can be used as temperature sensors and others as heaters. In this way the flow behaviour and heat loss can be well characterised.

The first step of the production process is to deposit a layer of SiC on a silicon wafer. Due to the fact that the silicon wafer is highly conductive an oxide layer is placed in between the silicon substrate and the SiC heater layer for electrical isolation. This layer will later be manipulated such that it becomes the suspended heating element in the flow channel. This layer was deposited by the SiC expert at Dimes, Mr. Morana. For the production of the flow channels, the masking process described in section 7.2 is used. Each thruster module has a specific place in the masking window. Each desired thruster module is then etched sequentially to build up the desired thruster design. The unique method in this fabrication is that the heating membrane is not produced separately. The SiC is already deposited on the silicon wafer, and the heating elements are etched free in the same etching steps in which the flow channels are produced.

The etching process itself is done in three stages. The first stage of the etching process is to etch away the part of the SiC layer which is not needed for the heating. This includes the separation slits between the electrical contact pads of the heating modules, shown in Figure 7.33. Also the straight heating elements which are on top of the semi-circular flow channel walls, shown in Figure 7.44, need to be etched in this step.

The second stage of etching is called Deep reactive-ion etching (DRIE). This is a highly anisotropic etching process, which means it only etches straight down from the surface. An example pattern that
is used is shown in Figure 7.50 below. The sections that need not be etched are covered with photoresist material. The wafer is then bombarded with ions. Under this bombardment the unprotected areas are excavated, and the areas protected by the photoresist are unharmed. After this etching step the resistive heating elements are still on top of a silicon ridge, so they are not suspended in the flow yet.

Figure 7.50 - Pattern which was used for the nominal diamond shape design. In black is the section that is etched, in white is the section that is covered by photoresist.

The third step of the etching process is to use isotropic etching to achieve under-etching below the heating elements, in order to make them suspended in the flow. This process excavates the silicon walls which were still supporting the heating elements. The under etching below a heating element can be seen in Figure 7.51. In this image, the inherent problem of this method can clearly be seen. The fact that a ridge of silicon remains where the under-etching fronts from both sides of the heating element meet.

Figure 7.51 - SEM image to clearly illustrate the under-etching process below the heating elements, in order to suspend them in the flow channel. This is an image of the nominal semi-circular flow channel design.

This is where the manufacturing process ended during the timeline of this master thesis. The manufacturers at Dimes were unfortunately not able to finish the thrusters before their facilities closed for major overhaul. The following steps indicate what will be done in the future, in order to finalise the Dondersteen thrusters.
After the channels have been etched, and the heating elements are suspended in above the flow channel, the thrusters can be closed. Parallel to the above mentioned process, one wafer is produced by etching with a mirror image of only the flow channels. This wafer is then placed on top of the other wafer, with the flow channels facing each other. These wafers are then anodically bonded to fix them together and seal the gap between them. The thruster is then a closed design with the heating elements suspended in the middle of the flow.

Since the thrusters are then closed, the SiC heating layer is in the centre of the silicon wafer sandwich. The electrical connections have to be made on the bond areas shown in Figure 7.33. In order to connect this heating layer to the power supply, a hole is etched on top of the silicon wafer. This etching reaches from the top of the silicon wafer down to the SiC heating layer in the centre. The holes have a square area with sides of 1 [mm]. Since the electrical bond wires cannot be bonded on SiC directly, a thin aluminium layer is first deposited on top of the exposed SiC. The bond wires can later be bonded to this aluminium layer by a special process, conducted at Dimes, which fuses the aluminium to the bond wire.

The final step in the manufacturing process there is 100 [μm] loss by cutting the thrusters apart, when diamond saws are used. The outset is however cut all thrusters out of the wafers by etching on the back of the wafer. This could potentially cut a narrower line. This etching is done in the same etching step as the opening of the electrical connection areas.

The manufacturing could unfortunately not be fully finished by Mr. van Zeijl before the closure of the Dimes production facilities. Therefore it was agreed to finalise the production of one of the wafers, to such a stage that the geometrical and electrical characterisation could already be completed. This enables vital preliminary investigations on the deviation between the design and the practice of this novel manufacturing method. This test wafer is shown in Figure 7.52.

![Figure 7.52 - Test wafer filled with the Dondersteen micro-thruster designs, used for geometrical and electrical characterisation.](image-url)
Due to the necessity of the under-etching, the heater elements cannot be wider than 9 [μm]. Otherwise the etching process will not be able to remove enough material from underneath the heating elements. Heating elements that were calculated to be wider than 9 [μm] have been divided into smaller elements of 9 [μm] wide.

In conversations with Dimes, the idea came up to do the fluidic thruster tests on the halves of the thrusters that were now actually available. To this end a glass layer would be bonded to the open side of the wafer to close the thruster structures. A number of steps would have had to be taken to make this possible. First of all, the protective oxide layer would have to be removed from the SiC, at a different etching facility on the campus. After which a specialised company would have to be ordered to produce a glass covering, with a recess for the heating elements. This is because the heating elements stick out from the silicon wafer slightly. So if a glass layer would be placed on top it would only touch the heating elements. This glass plate with recesses would then have to be accurately bonded to the silicon wafer, without the adhesive clogging any part of the flow channel. The thrusters would have had to be separated by using a diamond saw. This gives the risk of material chippings clogging the channels. A special inlet cap would have to be produced to serve as a fluidic interconnect, in order to feed the propellant flow to the thruster. This is because there is no needle that would fit the too small inlet opening. Then the SiC electrical connection pads, which would then be sandwiched between the glass layer and the silicon wafer, need to be made accessible. This could possibly be by adding holes through the glass plate which is bonded on top.

This idea was however discarded; because it would take a few months to get this all commissioned and build at external companies. Next to that there are all kinds of uncertainties if it would have worked. After which the actual tests would still have had to commence. By the time all these arrangements could have been made, the Dimes manufacturing facilities would be up and running again. At Dimes the rest of the production would only take another day. Therefore this temporary solution was seen as not worth the trouble, cost and manpower.

### 7.8 Printed Circuit Board (PCB) mounting

To connect the finished thrusters to the electrical power supply and measuring devices, bond wires are connected to the small electrical connection pads of the heating elements. These bond wires cannot span large distances, due to their small thickness. Therefore the necessary electrical circuits are printed on a PCB, on which the thruster can be mounted. The bond wires then only have to span the distance from the top of the thruster to the substrate on which it is mounted. The PCB is then designed to incorporate the possibility to connect the electrical circuit to the necessary external devices. The thruster itself is very small, thin and relatively fragile. The PCB mounting has the added advantage of adding increased structural stability to the thruster. It also provides a method to connect the thruster to the test setup.

It was first understood that he PCB would be designed, built and integrated at Dimes. However at a later stage it was said that this was not the case. The Dimes representatives said that the university’s electronic and mechanical support division, Dienst Elektronische en Mechanische Ontwikkeling (DEMO), will be able to do it. To this end, the representative Mr. B. Schelen was contacted. He explained that DEMO does not produce PCBs anymore. They are however available and willing to cooperate in the outsourcing of the production to the company Eurocircuits.
Mr. Schelen also recommended that if very high PCB temperature resistance is required, micro Electronics bv. is able to produce ceramic PCBs. The material which is normally used for PCBs will start delaminating and disintegrating at temperatures in excess of 180 [°C]. He did mention that this only happens after hours of continuous high temperature operation. Experiences with the previous MEMS resistojet has shown that temperatures up to 400 [°C] did not significantly harm the PCB material [31].

The PCB was designed to be as small and functional as possible. Mr. N. Dos Santos has been a great help in designing the eventual PCB layout shown in Figure 7.53. A three dimensional rendering of the thruster mounted on the PCB is shown in Figure 7.54.

The PCB is designed to have copper lined, circular holes in the area on which the PCB is mounted. There are in place to make sure that the PCB will not be able to absorb all of the heat coming from the thruster. The holes will reduce the heat loss by conductivity through the PCB material. The copper lining will make sure that the heat which is absorbed by the PCB is quickly transported away. This will reduce the risk of heat damage to the PCB material. Careful though has gone in to the placement of the golden bond pads, in order to make sure they can be used by both the 7 and the 14 module long thrusters. The PCB has four screw holes in the corners in order to fasten it to the test rig.
Because many different thrusters have to be tested, the connection between the PCB and the test setup has been made easily interchangeable. The easiest and cheapest way to connect it is using headers. Because some boards only require 7 connections on each side and others require 14, it would be easiest to use the 7 pin single row headers. And put on one on each side of the board for most setups and 2 headers on each side for the 14 connection thrusters. A small trade-off between optional connector pins was conducted to find the most appropriate one. The best candidates in this trade-off are shown in Table F.2 in Appendix F. The pins that were eventually selected are also shown in Figure 7.54. The “TE CONNECTIVITY / AMP - 215307-7 - SOCKET, VERTICAL, 2ROW”, is the female connector which is linked to the electrical system. This simplifies the testing of multiple thrusters sequentially. The female connector is fixed to the electrical test system and each PCB has its own male headers to connect to. When swapping different thrusters they can be disconnected and reconnected simply by clicking the connection in and out.
8 Thruster testing
The physical testing of the MEMS micro-thruster Dondersteen is conducted to validate the theoretically predicted performance. This verification consists of both static and dynamic testing. The static testing is conducted to assess how much the final product differs from the theoretical design. In the dynamic tests, the thrusters are operated in several different methods, to determine the performance of the mission relevant parameters. This chapter is only meant to give a rough overview of these tests. The actual test plan was written by the designated test engineering Quirino Bellini in cooperation with the author and the other propulsion system engineer Mr. Krusharev [67]. The reader is referred to this document for further information.

8.1 Necessary parameters
The dimensional parameters of the thrusters need to be obtained, in order to assess the accuracy of the manufacturing process. The actual dimensions are also necessary to recalculate the theoretical performance prediction, to draw conclusions on the difference between the theoretical and actual performance. The relevant dimensional parameters which need to be determined are:

- The roughness of the walls
- Etching depth
- Diverging half angle of the inlet
- Heating chamber dimensions
  - Width of the pillars or flow channels
  - Length of the pillars or flow channels
  - Total width of the heating chamber
- Nozzle dimensions
  - Converging half angle
  - Nozzle throat diameter
  - Nozzle diverging half angle
  - Exit diameter
- SiC heating layer thickness

The characteristics of the SiC heating elements need to be known. This in order to better understand the effect of geometry on the electrical conduction properties, and to assess the specific properties of the SiC material used. To this end, the temperature dependent electrical resistance needs to be determined.

The performance parameters, for the thruster in operation, are required to assess the differences between the theoretical and the true performance. The performance parameters which need to be known are:

- Properties of the incoming propellant at the entrance of the thruster
  - Pressure
  - Temperature
  - Mass flow rate
• Properties of the separate heating elements
  o Electrical input power
  o Temperature

All these parameters need to be known for different propellant pressure, and electrical input powers.

After the propulsion tests have been executed, the thrusters have to be inspected for malfunctioning. The PCB, thruster exterior and the heating membrane have to be inspected for defects or alterations.

8.2 Testing methodology
The tests required to determine the necessary parameters mentioned in section 8.1, can be divided in four different groups. The methodologies of executing the tests in each of these groups are described in the subsequent sub-sections.

8.2.1 Room temperature tests
The optical microscopes available at Delft Aerospace Structures and Materials Laboratory (DASML) facility were used to determine the internal dimensions mentioned in section 8.1. These have been used to check all geometric design parameters, which will be discussed in section 8.4. The resistance values have also been determined in order to predict the electrical behaviour.

8.2.2 Characterisation tests
To characterise the behaviour of the heater membranes, the vacuum oven in the TU Delft aerospace faculty clean room will be used. The oven will bring the thrusters from room temperature to an elevated temperature. The thruster inside the oven will be connected to externally placed resistance measurement device. Parallel to these resistance measurements, the temperature measurements will also be collected at the same intervals. The material with the lowest melting point of the thruster configuration is the PCB material. This PCB material has a maximum operating temperature of 150 °C. In order to leave sufficient margin to this maximum temperature, the maximum temperature inside the oven will be set to 100 °C. The resistivity of the doped SiC was previously seen to decrease monotonically and approximately linear with an increase in temperature [64]. Therefore a limited range of temperature, with several data points will already suffice for the temperature dependent resistance characterisation.

8.2.3 Performance tests
The performance parameters are then determined to assess the differences between the theoretical and the true performance. This is done in dynamical tests, where the thruster is actually operated as it was assumed to be operated for the calculations. The tests therefore have to be conducted in a vacuum oven, in order to simulate the orbital pressure conditions. The results will afterwards have to be checked if they need to be corrected for the incomplete vacuum. All parameters that need to be measured during the performance testing were mentioned in section 8.1. The parameters either have to be constant, or they have to be measured simultaneously at the same point in time. This is in order to have the right correlation of the parameters.

8.2.4 Post test inspection
The structural integrity and possible alterations in the PCB and the thruster geometry will be inspected using the microscopes mentioned in sub-section 8.2.1. These microscopes are situated in
the DASML and can magnify up to 100 times, which is sufficient to inspect for cracks, molten or burnt sections and other malfunctions. The digital cameras attached to these microscopes will be used to produce images depicting the thruster state after testing.

8.3 Test equipment
A short investigation was conducted on the necessary temperature and pressure sensors for the thruster engineering model testing. The requirements for the sensors on the engineering model:

- They shall have a maximum operating pressure of more than 7 [bar], preferably even 10 [bar] (The valve for the engineering model has a maximum operating pressure of 7 [bar], therefore the engineering model tests will not go higher than 7 [bar]. However, if the sensors used for the engineering model have a maximum operating pressure of 10 [bar] they can be reused in the flight model tests)
- They shall be vacuum compatible.
- They shall be compact (maximally about 10 [cm] by 10 [cm] in order for it to fit inside the vacuum chamber)
- They shall have an operational temperature range of 20 [°C] to 80 [°C] (The actual temperature of the propulsion system lies between -20 [°C] to 80 [°C], but the vacuum chamber can only heat, and not cool, so temperatures below room temperature are physically impossible to test in the present facilities)

The following pressure sensors are available at the chair:

- Two Omega engineering inc. PX181-200G5B pressure transducers, operational range:
  - 0 [bar] to 13.8 [bar]
  - -55 [°C] to 105 [°C]
  - They were tested to be vacuum compatible
- Unknown sensor on the T³ μPS engineering model, possibly this is the Intersema MS5407 with an operational pressure range of 0 [bar] to 7 [bar]. Most probably only a pressure sensor, without integrated temperature sensor.
- Two Kulite LE-30-125-100A, operational range:
  - -55 [°C] to 120 [°C]
  - 0 [bar] to 7 [bar]

The Omega engineering inc. PX181-200G5B has thereafter been integrated into the feed system of the test setup of the thruster engineering models. If these at any point appear to be damaged or otherwise inadequate the following sensor can be acquired from Farnell:

- HONEYWELL S&C - HSCSANN150PA2A5 (absolute pressure) or HONEYWELL S&C - HSCDANN150PG2A5
  - Pressure range 0 [bar] to 10.3 [bar]
  - Temperature range -20 [°C] to 85 [°C]
  - Price € 33.86
  - Mass 5 [g]
  - Supply voltage 5 [V]
Especially due to its low mass and correct supply voltage, this is also a pressure sensor which can be included in the flight model of the propulsion system. The necessary thermocouples for temperature measurements are already available in the vacuum oven which will be used for the tests. If these appear to be inadequate, one of the following sensors can be acquired from Farnell:

- IST INNOVATIVE SENSOR TECHNOLOGY - TSIC 306 SOP-8
  - Temperature range -50 [°C] to 150 [°C]
  - Resolution 0.1 [°C]
  - Supply voltage 3.0 [V] to 5.5 [V]
  - Price € 4.85
- VARIOHM EUROSENSOR - ETP-SP-8-23-10K3A1B Thermistor
  - With screw thread
  - Temperature range -40 [°C] to 125 [°C]
  - Price € 17.04

8.4 Test execution and results

8.4.1 Geometrical measurements results

The thickness of the SiC heater layer was tested by the professional personnel at Dimes. It was found to be 300 [nm] as was defined by the design. The other geometrical parameters where all checked using optical microscopes. Some of the imagery taken of these measurements are shown in Appendix G. All the measured parameters have been compared to their design values in Table 8.1. The computer imagery program that was linked to the optical microscope was already calibrated for all available lenses. This means that the necessary dimensions can be directly measured using the program itself, as shown in Appendix G.

Table 8.1 - Comparison between the design values of the thruster geometry and the actually produced values as checked with the optical microscopes.

<table>
<thead>
<tr>
<th>Module</th>
<th>Parameter</th>
<th>Design value</th>
<th>Actual value</th>
<th>Unit</th>
<th>Deviation [%]</th>
</tr>
</thead>
<tbody>
<tr>
<td>Inlet</td>
<td>Straight section width</td>
<td>400</td>
<td>110</td>
<td>[μm]</td>
<td>-72.50%</td>
</tr>
<tr>
<td></td>
<td>Straight section length</td>
<td>5000</td>
<td>1969</td>
<td>[μm]</td>
<td>-60.62%</td>
</tr>
<tr>
<td></td>
<td>Divergent section length</td>
<td>1500</td>
<td>1450</td>
<td>[μm]</td>
<td>-3.33%</td>
</tr>
<tr>
<td></td>
<td>Diverging angle</td>
<td>45</td>
<td>45.0</td>
<td>[*]</td>
<td>0.00%</td>
</tr>
<tr>
<td>Heating module shapes</td>
<td>Heater diamond width</td>
<td>40</td>
<td>36.6</td>
<td>[μm]</td>
<td>-8.50%</td>
</tr>
<tr>
<td>nominal</td>
<td>Pillar diamond width</td>
<td>40</td>
<td>20.1</td>
<td>[μm]</td>
<td>-49.75%</td>
</tr>
<tr>
<td></td>
<td>Heater diamond length</td>
<td>160</td>
<td>158</td>
<td>[μm]</td>
<td>-1.25%</td>
</tr>
<tr>
<td></td>
<td>Pillar diamond length</td>
<td>160</td>
<td>70.8</td>
<td>[μm]</td>
<td>-55.75%</td>
</tr>
<tr>
<td></td>
<td>Wall roughness</td>
<td>0</td>
<td>0.54</td>
<td>[μm]</td>
<td>n/a</td>
</tr>
<tr>
<td></td>
<td>Suspended heating element</td>
<td>20</td>
<td>23.6</td>
<td>[μm]</td>
<td>18.00%</td>
</tr>
<tr>
<td></td>
<td>Suspended heating element width</td>
<td>9</td>
<td>7.57</td>
<td>[μm]</td>
<td>-15.89%</td>
</tr>
<tr>
<td>Heating</td>
<td>Total chamber width</td>
<td>3000</td>
<td>3019</td>
<td>[μm]</td>
<td>0.63%</td>
</tr>
<tr>
<td></td>
<td>Heating layer separation</td>
<td>10</td>
<td>9.81</td>
<td>[μm]</td>
<td>-1.90%</td>
</tr>
<tr>
<td></td>
<td>Heater diamond width</td>
<td>160</td>
<td>158</td>
<td>[μm]</td>
<td>-1.25%</td>
</tr>
<tr>
<td>Module</td>
<td>Parameter</td>
<td>Design value</td>
<td>Actual value</td>
<td>Unit</td>
<td>Deviation [%]</td>
</tr>
<tr>
<td>-------------------------</td>
<td>------------------------------------------------</td>
<td>--------------</td>
<td>--------------</td>
<td>------</td>
<td>---------------</td>
</tr>
<tr>
<td>module diamond shapes</td>
<td>Pillar diamond width</td>
<td>160</td>
<td>143</td>
<td>[μm]</td>
<td>-10.63%</td>
</tr>
<tr>
<td>shapes large</td>
<td>Heater diamond length</td>
<td>640</td>
<td>578</td>
<td>[μm]</td>
<td>-9.69%</td>
</tr>
<tr>
<td></td>
<td>Pillar diamond length</td>
<td>640</td>
<td>508</td>
<td>[μm]</td>
<td>-20.63%</td>
</tr>
<tr>
<td></td>
<td>Wall roughness</td>
<td>0</td>
<td>0.60</td>
<td>[μm]</td>
<td>n/a</td>
</tr>
<tr>
<td></td>
<td>Suspended heating element average length</td>
<td>90</td>
<td>99.7</td>
<td>[μm]</td>
<td>10.78%</td>
</tr>
<tr>
<td></td>
<td>Suspended heating element width (total)</td>
<td>38</td>
<td>32.6</td>
<td>[μm]</td>
<td>-14.21%</td>
</tr>
<tr>
<td></td>
<td>Total chamber width</td>
<td>3000</td>
<td>3021</td>
<td>[μm]</td>
<td>0.70%</td>
</tr>
<tr>
<td></td>
<td>Heating layer separation</td>
<td>10</td>
<td>11.4</td>
<td>[μm]</td>
<td>14.00%</td>
</tr>
<tr>
<td>Heating module</td>
<td>Wall roughness</td>
<td>0</td>
<td>0.56</td>
<td>[μm]</td>
<td>n/a</td>
</tr>
<tr>
<td>Serpentines nominal</td>
<td>Channel width</td>
<td>60</td>
<td>57.5</td>
<td>[μm]</td>
<td>-4.17%</td>
</tr>
<tr>
<td></td>
<td>Inner channel curvature diameter</td>
<td>20</td>
<td>23.4</td>
<td>[μm]</td>
<td>17.00%</td>
</tr>
<tr>
<td></td>
<td>Outer channel curvature diameter</td>
<td>140</td>
<td>139</td>
<td>[μm]</td>
<td>-0.71%</td>
</tr>
<tr>
<td></td>
<td>Heating element total width</td>
<td>29</td>
<td>22.0</td>
<td>[μm]</td>
<td>-24.14%</td>
</tr>
<tr>
<td></td>
<td>Number of channels</td>
<td>21</td>
<td>21</td>
<td>[-]</td>
<td>0.00%</td>
</tr>
<tr>
<td></td>
<td>Heating layer separation</td>
<td>10</td>
<td>11.0</td>
<td>[μm]</td>
<td>10.00%</td>
</tr>
<tr>
<td>Heating module</td>
<td>Wall roughness</td>
<td>0</td>
<td>0.33</td>
<td>[μm]</td>
<td>n/a</td>
</tr>
<tr>
<td>Serpentines Large</td>
<td>Channel width</td>
<td>232</td>
<td>226</td>
<td>[μm]</td>
<td>-2.59%</td>
</tr>
<tr>
<td></td>
<td>Inner channel curvature diameter</td>
<td>88</td>
<td>77.14</td>
<td>[μm]</td>
<td>-12.34%</td>
</tr>
<tr>
<td></td>
<td>Outer channel curvature diameter</td>
<td>552</td>
<td>557</td>
<td>[μm]</td>
<td>0.91%</td>
</tr>
<tr>
<td></td>
<td>Heating element total width</td>
<td>77</td>
<td>80.8</td>
<td>[μm]</td>
<td>4.94%</td>
</tr>
<tr>
<td></td>
<td>Number of channels</td>
<td>5</td>
<td>5</td>
<td>[-]</td>
<td>0.00%</td>
</tr>
<tr>
<td></td>
<td>Heating layer separation</td>
<td>10</td>
<td>10.2</td>
<td>[μm]</td>
<td>2.00%</td>
</tr>
<tr>
<td>Nozzle sharp edge</td>
<td>Convergent angle</td>
<td>30</td>
<td>30.2</td>
<td>[*]</td>
<td>0.67%</td>
</tr>
<tr>
<td></td>
<td>Throat diameter</td>
<td>25</td>
<td>34.4</td>
<td>[μm]</td>
<td>37.60%</td>
</tr>
<tr>
<td></td>
<td>Divergent angle</td>
<td>20</td>
<td>20.5</td>
<td>[*]</td>
<td>2.50%</td>
</tr>
<tr>
<td></td>
<td>Exit diameter</td>
<td>500</td>
<td>499</td>
<td>[μm]</td>
<td>-0.20%</td>
</tr>
<tr>
<td>Nozzle bell shape</td>
<td>Convergent angle 1</td>
<td>45</td>
<td>45.3</td>
<td>[*]</td>
<td>0.67%</td>
</tr>
<tr>
<td></td>
<td>Angle change diameter</td>
<td>75</td>
<td>90.7</td>
<td>[μm]</td>
<td>20.93%</td>
</tr>
<tr>
<td></td>
<td>Convergent angle 2</td>
<td>15</td>
<td>18.3</td>
<td>[*]</td>
<td>22.00%</td>
</tr>
<tr>
<td></td>
<td>Throat diameter</td>
<td>25</td>
<td>30.2</td>
<td>[μm]</td>
<td>20.80%</td>
</tr>
<tr>
<td></td>
<td>Divergent angle</td>
<td>60</td>
<td>59.8</td>
<td>[*]</td>
<td>-0.33%</td>
</tr>
<tr>
<td></td>
<td>Exit angle</td>
<td>15</td>
<td>13</td>
<td>[*]</td>
<td>-13.33%</td>
</tr>
<tr>
<td></td>
<td>Exit diameter</td>
<td>500</td>
<td>498</td>
<td>[μm]</td>
<td>-0.40%</td>
</tr>
<tr>
<td>Nozzle Ivan Krusharev</td>
<td>Convergent angle</td>
<td>45</td>
<td>45.1</td>
<td>[*]</td>
<td>0.22%</td>
</tr>
<tr>
<td></td>
<td>Throat diameter</td>
<td>30</td>
<td>43.7</td>
<td>[μm]</td>
<td>45.67%</td>
</tr>
<tr>
<td></td>
<td>Divergent angle</td>
<td>30</td>
<td>29.9</td>
<td>[*]</td>
<td>-0.33%</td>
</tr>
<tr>
<td></td>
<td>Exit diameter</td>
<td>800</td>
<td>795</td>
<td>[μm]</td>
<td>-0.63%</td>
</tr>
</tbody>
</table>

In Table 8.1 all the design parameters have been mentioned and checked for completeness. Some of them, such as the geometrical angles, are exactly correct. Many of them are however far removed.
from their design value. The most important and most deviating values will be discussed. As mentioned in section 7-3, due to the rushing of the manufacturing process, the inlet could not be designed to specifications. The width and length of the small diamond shaped pillars deviate quite significantly from the design value. This is due to the under-etching of the isotropic etching process. This effect is actually very beneficial as it still supports the heating elements, but there is less material, so less conductivity from the pillars to the silicon structure of the thruster. The wall roughness can be seen to be well below 1[μm] for all different designs. This is exactly as specified by Dimes before production and is acceptable since it remains below 4% deviation of the most critical dimension, which is the nozzle throat. The suspended heating element length can be seen to be larger due to the under-etching of the pillars. The width of the suspended elements for most of the designs is off by a relevant amount. This is also one of the causes for the resistance of the heating elements, which is higher than it was designed to be.

Finally the most important deviation, the nozzle throats. Depending on the magnitude of the converging and diverging angle ratios, the nozzle throat diameter significantly deviates from the design values. This has a major impact on the performance of the thruster. A larger nozzle throat area leads to a larger mass flow, which in turn requires a higher electrical power input in order to vaporise it. This widening of the nozzle throat is an artefact of the experimental production process. The results from this design will help in fine-tuning the production in such a way that the nozzle throats can be made more accurate for the final flight models of the thruster.

Subsequent to these tests, the Scanning Electron Microscope (SEM) was used, in order to assess the profile of the channel walls and floor at an angle. As can be seen in the following images, side walls of the flow channels are still relatively straight. The corners of the flow channels are rounded off. And at the points where there are heating elements or small flow passages, substantial ridge formation is observed.

Figure 8.1 - SEM image of the nominal thruster design throat area - Straight walls, chamfered corners and ridge forming can be observed in the flow channel.

Figure 8.2 - SEM image of the nominal diamond shaped heater element design - Under-etching of the diamond elements, chamfered corners and ridge forming can be observed in the flow channel.
The measurements on the channel depth were assessed using an optical microscope. An optical microscope can only focus on one plane perpendicular to the viewing direction. Anything that is closer to or further away from the lens is blurred. The method of channel depth measurement is therefore very simple. The microscope is focussed on layer A of material. The platform on which the sample rests is then moved upwards or downwards until layer B comes in to focus. The height with which the platform had to be moved to shift the focus from layer A to layer B is then equal to the vertical distance between the two layers.

This is illustrated in the following figures. In Figure 8.5 the microscope is focussed on the top surface of the thruster wafer. The platform is then moved up by 70 [μm] to put the ridge in the throat into focus, as shown in Figure 8.6. Finally the platform is moved up by another 20.5 [μm] to focus on the flow channel bottom, shown in Figure 8.7. This means that this flow channel has a total depth of 90.5 [μm].

During these tests, the platform movement could be read out with an accuracy of 0.5 [μm] and the focussing was seen to be accurate to about 2 [μm]. The measurement results on the channel depth and ridge height are displayed in Table 8.2. All the channel depths were designed to be 50 [μm] deep on one side. It can clearly be seen that this is not the case.
Table 8.2 - Measurement results of the channel depth in specific areas, multiple versions were measured where multiple dimensions are listed. Channel depth and ridge height from the same channel are listed in the same column.

<table>
<thead>
<tr>
<th>Module</th>
<th>Parameter</th>
<th>Dimension [μm]</th>
</tr>
</thead>
<tbody>
<tr>
<td>Nozzle Ivan Krusharev</td>
<td>Channel depth</td>
<td>88.5</td>
</tr>
<tr>
<td></td>
<td>Ridge height</td>
<td>16</td>
</tr>
<tr>
<td>Optimised bell nozzle</td>
<td>Channel depth</td>
<td>96 90.5 89.5</td>
</tr>
<tr>
<td></td>
<td>Ridge height</td>
<td>17 20.5 15.5</td>
</tr>
<tr>
<td>Nozzle sharp edge</td>
<td>Channel depth</td>
<td>91 79 79</td>
</tr>
<tr>
<td></td>
<td>Ridge height</td>
<td>13 12 9</td>
</tr>
<tr>
<td>Heating module diamond nominal</td>
<td>Channel depth</td>
<td>78 77.5</td>
</tr>
<tr>
<td></td>
<td>Ridge height</td>
<td>13 13</td>
</tr>
<tr>
<td>Heating module diamond large</td>
<td>Channel depth</td>
<td>86 83.5</td>
</tr>
<tr>
<td></td>
<td>Ridge height</td>
<td>28.5 20</td>
</tr>
<tr>
<td>Heating module serpentine nominal</td>
<td>Channel depth</td>
<td>71.5 71</td>
</tr>
<tr>
<td></td>
<td>Ridge height</td>
<td>34.5 33</td>
</tr>
</tbody>
</table>

8.4.2 Electrical characterisation tests

During the manufacturing process a non-conductive oxide layer was placed on top of the SiC heater for protection during the etching process. Due to this layer no electrical contacts could be placed on the SiC. In order to conduct the electrical characterisation this layer first had to be removed, which was done at the faculty of applied sciences by Dr. van Zeijl. During this process the thruster cavities themselves had to be shielded, to only expose the electrical contact pads on the side. This shielding was achieved by adding Kapton tape strips on top of the thruster cavities. This shielding worked for the etching process. However during the removal of the Kapton tape, some of the heater structures were severely damaged. The heating elements with the largest aspect ratios were even completely destroyed. These were the heating elements of the designs with the large flow channels. During the clamping of the etching process the wafer was also broken into several pieces. This violent additional process is therefore expected to alter some of the characterisation results.

The characterisation tests have so far only included the electrical conductivity of the heating elements. This has been proven to be far removed from the design values. In order to test the resistance, miniature electrical probes were placed on the sides of the SiC heating elements. Different electrical potential differences where then applied between the two heater terminals. The result of these tests then forms a current-voltage characteristic, or I-V curve for the tested heating element. From this characteristic, the resistance is found using Ohm’s law. The results of these characterisation tests are shown in Table 8.3. As shown in this table, for each heating element type, at least four different tests were conducted, in order to show the divergence in the measurements.

Table 8.3 - The summary of the cold electrical resistance tests on the different SiC heating elements of the Dondersteen thruster. All resistances have been measured on at least 4 different modules, and most of the values are an average of multiple measurements on the same module

<table>
<thead>
<tr>
<th>Section</th>
<th>Parameter</th>
<th>Design value</th>
<th>Actual value</th>
<th>Unit</th>
</tr>
</thead>
<tbody>
<tr>
<td>Diamond shaped heaters nominal design</td>
<td>Electrical resistance 1</td>
<td>25</td>
<td>8,994</td>
<td>[Ohm]</td>
</tr>
<tr>
<td></td>
<td>Electrical resistance 2</td>
<td>25</td>
<td>13,300</td>
<td>[Ohm]</td>
</tr>
<tr>
<td></td>
<td>Electrical resistance 3</td>
<td>25</td>
<td>15,000</td>
<td>[Ohm]</td>
</tr>
<tr>
<td></td>
<td>Electrical resistance 4</td>
<td>25</td>
<td>11,500</td>
<td>[Ohm]</td>
</tr>
</tbody>
</table>
As can be seen from the measured values in Table 8.3, the measured values of the electrical resistance differ from the design values with 2 to 3 orders of magnitude. This is such a large deviation, that it completely alters the behaviour. As can be seen from the combination of Joule’s and Ohm’s law in equation 5-1, when the resistance increases, the voltage also has to increase in order to yield the same power dissipation. As can be seen from this formula, an increase in resistance, will lead to a necessary increase in voltage of the square root of the magnitude increase of the resistance, in order to get the same output power. Therefore the 100 to 1000 times increase in the resistance, necessitates a 10 to 32 times increase in voltage, in order to achieve the same power dissipation. For the case of these heating elements, this means that the input voltage will have to increase to 50 [V] to 160 [V] depending on the type of heating element. For the thruster tests, this might be possible, but for the flight model this is highly undesirable. As to increase the voltage to such a high level will induce a lot of electrical losses in the necessary boost- or step-up converter.

The hot characterisation procedure in order to find the TCR of the SiC heaters can also be performed on the test structures. However since there are no electrical connections bonded to the heating elements yet, this will also have to be done using probes. To this end the thermal characterisation chamber, i.e. oven, has to be equipped with miniature electrical probes on the inside. This is not present yet, and it is too time-consuming to still implement it during the course of this thesis. However due to the very high divergence of the resistance that was measured, it is very desirable to also measure this TCR, before proceeding with the production process. If this value also differs as much from the design value, this will have to be taken into consideration in future design iterations.

### 8.4.3 Analysis of test results

As explained before, the manufacturing had to be rushed, due to the imminent maintenance closure of the Dimes production facilities. Due to this rushing, the inlet could not be made to specifications. The width and depth of which were produced to a square of 100 [μm], instead of the desired 400 [μm].

<table>
<thead>
<tr>
<th>Section</th>
<th>Parameter</th>
<th>Design value</th>
<th>Actual value</th>
<th>Unit</th>
</tr>
</thead>
<tbody>
<tr>
<td><strong>Diamond shaped heaters</strong></td>
<td><strong>Electrical resistance 1</strong></td>
<td>25</td>
<td>12,500</td>
<td>[Ohm]</td>
</tr>
<tr>
<td><strong>large design</strong></td>
<td><strong>Electrical resistance 2</strong></td>
<td>25</td>
<td>17,000</td>
<td>[Ohm]</td>
</tr>
<tr>
<td></td>
<td><strong>Electrical resistance 3</strong></td>
<td>25</td>
<td>7,500</td>
<td>[Ohm]</td>
</tr>
<tr>
<td></td>
<td><strong>Electrical resistance 4</strong></td>
<td>25</td>
<td>14,500</td>
<td>[Ohm]</td>
</tr>
<tr>
<td></td>
<td><strong>Average</strong></td>
<td>25</td>
<td>12,900</td>
<td>[Ohm]</td>
</tr>
<tr>
<td><strong>Serpentine heating channels</strong></td>
<td><strong>Electrical resistance 1</strong></td>
<td>25</td>
<td>4,038</td>
<td>[Ohm]</td>
</tr>
<tr>
<td><strong>nominal</strong></td>
<td><strong>Electrical resistance 2</strong></td>
<td>25</td>
<td>3,500</td>
<td>[Ohm]</td>
</tr>
<tr>
<td></td>
<td><strong>Electrical resistance 3</strong></td>
<td>25</td>
<td>4,500</td>
<td>[Ohm]</td>
</tr>
<tr>
<td></td>
<td><strong>Electrical resistance 4</strong></td>
<td>25</td>
<td>5,500</td>
<td>[Ohm]</td>
</tr>
<tr>
<td></td>
<td><strong>Average</strong></td>
<td>25</td>
<td>4,400</td>
<td>[Ohm]</td>
</tr>
<tr>
<td><strong>Serpentine heating channels</strong></td>
<td><strong>Electrical resistance 1</strong></td>
<td>25</td>
<td>7,000</td>
<td>[Ohm]</td>
</tr>
<tr>
<td><strong>large</strong></td>
<td><strong>Electrical resistance 2</strong></td>
<td>25</td>
<td>12,000</td>
<td>[Ohm]</td>
</tr>
<tr>
<td></td>
<td><strong>Electrical resistance 3</strong></td>
<td>25</td>
<td>13,500</td>
<td>[Ohm]</td>
</tr>
<tr>
<td></td>
<td><strong>Electrical resistance 4</strong></td>
<td>25</td>
<td>15,000</td>
<td>[Ohm]</td>
</tr>
<tr>
<td></td>
<td><strong>Average</strong></td>
<td>25</td>
<td>11,900</td>
<td>[Ohm]</td>
</tr>
</tbody>
</table>
One significant problem is that the size of the throat differs substantially from the design value. This would lead to a larger critical mass flow, which in turn leads to a larger power consumption. Because the power consumption is already designed to be near the maximum set by the requirements it is not acceptable for it to become even higher. The manufacturing crew has been notified of this and will fine-tune the production process to yield more accurate results in the future. Due to the process of first anisotropic etching and subsequent isotropic etching, it was difficult to find the balance. The masks had to be intentionally made smaller, in order to cover for the sideways etching of the isentropic etching. Apparently however the mask was either not reduced in size enough, or the wafer was exposed to the isotropic etching process for too long. It could also be a combination of both. During the production process, the masks will therefore be further reduced in size, or the isotropic etching process is reduced in duration.

This increase in nozzle throat area could have been slightly diminished by the formation of a ridge in the nozzle throat, which reduces the area. This ridge relative to the bottom surface of the flow channel is due to isotropic etching process, which has less effect on narrower sections. Unfortunately the channel has also been etched too deep, which immediately nullifies the area shrinking effect of the ridge. In the end the nozzle throat is both larger in height and larger in width/diameter for all nozzle designs. This significantly increases the critical mass flow rate, and therefore the necessary electrical power input. For the nominal nozzle design, the nozzle throat channel depth is increased by a factor 1.43 and the diameter is increased by a factor 1.37. This increases the nozzle throat area by a factor 1.96. As shown in Figure 3.5, both the thrust force and the power consumption scale linearly with nozzle throat area. Both values will therefore be almost doubled. For the thrust, that is not a problem, as it still falls well within the range specified in the requirements. The power consumption was already designed to be close to the maximum allowed value. Therefore the resulting power consumption will be almost twice the maximum allowed value.

The largest problem that was immediately detected when doing the electrical resistance measurements is that the resistance is a factor 200 to 6000 off of the design value. Most of the discrepancy was quickly traced back to a design error. The resistivity that was used for the design is a factor 100 lower than that of SiC in reality. This value was used due to a mistake in copying the results from earlier research, whereby the decimal point shifted two places. For the engineering model this is not such a large problem. This will require 63 [V] to 113 [V], depending on the type of heating module, to produce the same amount of heating power. Power supplies with these voltages are readily available at the university.

The flight models however do have to operate using a voltage of 5 [V]. The heating elements therefore need to be redesigned. From the decision making process it can be concluded that the following parameters can be changed:

- The resistivity of the SiC can be reduced by a factor 4 by doping it with more NH₃
- The thickness of the silicon carbide layer can be increased by a factor 2
- The length of the heating elements can be decreased by a factor 2
- The width of the heating elements can be increased by a factor 3
- The length of the heating module can be increased by a factor 3

These changes alone would already decrease the resistance of the elements by a factor 144. In addition to this the throat area can still be reduced slightly, which will reduce the critical mass flow,
and thus the power consumption. This is however difficult because it has already proven to be
difficult to manufacture the small size of the nozzle as it was. These suggested changes will result in a
total length of the thruster of 34.6 [mm] and it will reduce the total width of the thruster to
5.5 [mm]. This still fits within the geometrical constraints of the DelfFi propulsion system.

A more drastic change would be to change or enhance the heating elements using platinum. This is
also highly temperature resistant and inert, but has a far lower resistivity.
9 Future work to finalise the DelFFi propulsion system

This chapter will describe the steps that have to be taken, from the current development stage of the propulsion system development, to the final flight model. The steps that still need to be taken can be subdivided into different categories, which are manifested as sections of this chapter. The main function of this chapter is to serve as an initial, straightforward roadmap for the succeeding student on this research. Therefore it will state where the necessary material can be found and who needs to be asked.

9.1 Finalising the thrusters

Due to the fact that the production process had to be halted, the thrusters have not been completely finished yet. The thruster prototypes are already near to completion and will be finalised at Dimes in August 2014 or September 2014 at the latest. There will then be complete, closed and separated thrusters. As mentioned before, the problem with the too small inlet channel is that no available needle will fit it. Therefore a specialised connector would have to be made and bonded to every thruster prototype. These problems have been discussed with Mr. van Zeijl. He will attempt to etch a hole in the inlet structure which is big enough for the needles. If he finds that that is not possible, he will contact the micro-machining staff at DEMO. Probably electrical discharge machining can be used to produce the necessary channel.

The thrusters will then have to be integrated with the already manufactured PCBs. These PCBs are stored by Mr. Cervone and await integration. The electrical connections called headers will first have to be soldered to the PCBs. These headers have been ordered and received; they are stored by Mr. Dos Santos. After the headers are soldered the PCBs can be given to Dimes. They will bond the thrusters to the specified area on the PCB. Dimes has specialised staff which will connect the electrical connection pads on the thruster to the golden bond pads on the PCBs using bond wires.

Finally the needles will have to be bonded to the thrusters, in order to provide the necessary fluidic connection. These needles have been ordered and received from Farnell and they are stored by Mr. Cervone. The needles will be bonded using the same method and epoxy adhesive used by Mr. Mathew [31]. This is best done using the microscope available in the cleanroom. The needle is then inserted slightly into the inlet cavity. The adhesive is applied to outside of the needle and around the inlet cavity after which the needle is fully inserted into the inlet cavity. After the hardening of the adhesive, the thruster assembly is finished and ready for testing.

9.2 Testing equipment

The generic fluidic test setup mentioned in chapter 8 was designed and built in collaboration between Mr. Bellini, Mr. Krusharev and the author. It was then characterised by Mr. Bellini and is being used by Mr. Krusharev. During the testing the thruster design by Mr. Krusharev will therefore gain valuable experience with the test setup. These experiences will be documented in his thesis documentation, which is therefore a good source of testing information.
The feed system for both Mr. Krusharev’s thruster and the Dondersteen thrusters is the same. It ends in a flexible Swagelok tube with a 062 MINSTAC interface, which needs to be connected to thruster. To that end a connection piece was acquired that connects the Luer Lock interface of the thruster inlet needle, to the Swagelok 062 MINSTAC tube interface. This connection piece is stored by Mr. Cervone. The Dondersteen engineering model thruster assembly can be connected to the thrust stand using the specially made connection plate, which is stored by Mr. Cervone. Four screws in the attachment holes of the PCB are used to connect the assembly to this plate.

The main work that still needs to be done before the testing can commence is the connection of the electrical system. The connections have to be fed through the vacuum chamber walls with the already present com port connections. This feed-through is firstly used to supply the electrical power to the heating elements. Secondly they are used to measure resistance of the heating elements in order to determine their temperature. Preferably a heating chamber power supply regulator will be developed. The power supply can then be cut off or current regulated when the temperature of the heater reaches a certain percentage above the target temperature. The power can then be switched on again when the heater reaches a temperature, a certain percentage below the target temperature.

9.3 Thruster performance tests
As described in chapter 8 the extensive characterisation test on the Dondersteen thrusters have already been conducted. Thus both all geometries and the heating element resistances are already known. The tests that still need to be conducted are the performance tests. These tests are meant to see what the relations are between the thrust level, propellant mass flow and power consumption under different operating conditions.

It preferably needs to be evaluated what the minimum impulse bit of the thruster is. When the propellant valve is opened, the propellant will fill the entire space in between the valve and the exhaust nozzle. After the valve is closed, this section is still an open system filled with propellant. It will however not have the backpressure of the propellant pressurisation system anymore. The behaviour of this phase has to be properly characterised. It could results in the expulsion of relatively large amounts of propellant, without producing the necessary amount of thrust. A propulsion system redesign might have to follow. This could be as simple as taking a certain percentage of extra fuel into account.

9.4 Thruster design selection and alteration
After the performance tests are completed, the thruster design has to be re-evaluated. The best performing or most promising thruster modules will be redesigned for the final flight model of the DelFFi propulsion system. The nozzle with the best efficiency is selected for the final thruster design. The performance tests will indicate the most best performing heating module for both heating phases. The best vaporisation module is the one which vaporises the propellant with the least amount of electrical energy input. The best gaseous propellant heating section is the one which raises the propellant to the desired final temperature with the least amount of electrical power input. The length of those sections also has to be taken into consideration to see if they also fit the
geometrical design restrictions of the DelFFi propulsion system. If both heater sections have a different optimal heating chamber module, the heating chamber will be manufactured with both types of heater modules.

Due to the novelty of the production method, the resulting geometries were found to deviate from the design values. The acquired data on the magnitude of deviations is being used at Dimes to fine-tune the manufacturing process. Especially the nozzle throat section will be investigated, so the diameter can better resemble the design value. It will also be attempted to further minimise the ridges in some of the smaller flow channels. Finally the channel depth will be reduced, in order to more closely resemble the intended 50 [μm].

The most important redesign that has to be done is to rectify the mistakes made on using the wrong resistivity. In addition to this it was found that the resistance values were even more off than this calculation error alone would have caused. This means that the resistance of the heating elements has to be reduced by a factor 200 to 600, depending on the module type. The suggestions on how this can be done are given on subsection 8.4.3. It is strongly advised to use software do a more accurate analysis of the resistance behaviour of the resistive elements. COMSOL can readily be used for this purpose. Easy tutorials are available on electricity flow simulations, it is relatively easy to do and it is not extremely computationally intensive.

Possibly some other discrepancies will be found during the performance tests, which might also necessitate a certain redesign of one or more of the thruster modules.

9.5 Propellant storage system investigations
In this thesis all options for propellant storage and pressurisation were posed, and pre-pressurisation was identified as the most suitable concept for this mission. The design of this propellant storage system needs to be finalised using calculations after which it needs to be manufactured. The biggest question which needs to be answered is how to separate the propellant from the pressurant. Suggestions have been made in chapter 4 of this document, but these need to be tested.

It needs to be investigated which pressurant gas can best be used. At this moment, nitrogen is assumed to be used, which has the advantages that it is cheap inert. There might however be other gasses which are also suitable, but have larger molecule. The larger the molecules are, the less problem there is with leakage. The largest anticipated problem for the complete propulsion system is leakage. If the leakage of a pre-pressurised system is found to be too high, collaboration options with CGG technologies need to be investigated. Chief Technical Officer of the company, ir. Berry Sanders has been found to be interested in such collaboration.

9.6 Propulsion system flight model design manufacturing and integration
After the testing campaign of all the separate propulsion system components and assemblies, the final propulsion system design has to be made. When this design layout is completed, the necessary components have to be ordered or manufactured and finally assembled. This task will end by the support in the integration of the propulsion system into the two DelFFi mission satellites.
During and after the operation of the propulsion system in orbit, the data has to be analysed to see if the propulsion system worked as expected. This will lead to the final conclusions on the functionality of the system.
Conclusions

The aim of this master thesis research was: “Design a propulsion system to meet the propulsive requirements of the two DelFFi mission formation flying satellites”. In order to design this propulsion system a number of tests were conducted on the candidate components and assemblies. After a survey of all commonly mentioned reaction engines, a resistojet was found to best meet the needs of this mission. This was mainly selected because resistojets are relatively uncomplicated, do not require combustion and have relatively low power consumption per amount of impulse generated. These advantages were even more applicable to a cold gas thruster system. A cold gas thruster based system was however shown to be incapable to meet all requirements and restrictions of the DelFFi mission propulsion system. The best propellant for resistojets, selected from a large number of possibilities, was found to be water. Water was chosen because it is liquid around room temperature, it has a high potential impulse bit per unit of mass, and it is benign, cheap and readily available.

Based on the requirements on the propulsion system the following necessary design parameters were found: a water propellant mass of 50 [g], a nitrogen pressurant mass of 0.2 [g], a storage pressure of 4.5 [bar] to 2.5 [bar], a nozzle throat diameter of 25 [μm], a nozzle area ratio of 20 [-] and a final propellant heating temperature of 500 [°C]. The resulting propulsion system performance is: a total ΔV of 21.01 [m/s], a total thrusting time of 17 [h] and 56 [min], a power consumption of 6.8 [W] to 3.7 [W] and a thrust force of 1.4 [mN] to 0.8 [mN].

A resistojet propulsion system requires pressure to transport the propellant from the storage tank to and through the thruster. Because pre-pressurisation is generally not allowed in CubeSats, gas generators were found to be the best option. These gas generators can be electrically initialised in orbit, where they produce gas from a solid. Gas generators used for airbag initiation in the automotive industry were found to be unsuitable for this mission because of the high shock pressure of possibly up to 95 [bar]. Additionally, three out of four gas generators that were tested malfunctioned. The reason for this malfunctioning is still unknown. For the DelFFi mission a waiver was granted, now allowing pressurisation before launch. Therefore pre-pressurisation is now considered to be the best option for propellant pressurisation.

The most innovative part of this thesis is the development of the MEMS micro-thrusters called Dondersteen. These consist of an integrated fluidic inlet channel, heating chamber and rocket nozzle, which are all etched in silicon. The propellant is heated using heating elements made out of the very strong material silicon carbide. These elements are suspended in the middle of the fluid flow in order to maximise the heat transfer to the propellant, and minimise the heat loss to the surroundings. The developed thruster consists of the following three distinct sections: inlet, heating chamber and nozzle. Multiple designs were made for each of these sections. The production process was chosen such that any combination of these sections can be etched behind one another. This creates the flexibility of being able to produce and test multiple thruster layouts, in order to find the one which best fits the requirements. Different heating chamber geometries were designed and tested. These will enable the assessment of the geometrical influence on heat loss, pressure drop and propellant heating capabilities. Due to the experimental nature of the manufacturing process, some of the geometrical parameters were not exactly made to specifications. Namely the channel dept was
found to be too large and the nozzle throat diameters were too large. The performed analysis of these discrepancies will be used to improve the manufacturing process.

The tested resistance values of the resistive heater modules were found to be 200 to 600 times larger than designed. This increases the required input voltage from 5 [V] to the range from 70 [V] to 120 [V]. For the current testing phase this is not a problem since these supplies are available. The flight models will however have to be redesigned in order to comply with the 5 [V] requirement of the satellite power supply. The main reason of this discrepancy was found to be a calculation error. Some adjustments have already been listed with which this resistance can be decreased by a factor of 144. Further design efforts will have to increase this factor slightly to obtain the real design value of the resistance.

There are some tasks that still have to be performed by a succeeding master student to consolidate the work done in this master thesis to a propulsion system flight model. The developed thrusters have to be performance tested in multiple operating conditions. The results of these tests lead the final redesign of the developed thrusters. The propellant storage system needs to be built and tested. Finally the complete in flight propulsion systems can be defined, built and integrated into the satellites.

Therefore it can be said that the aim of the master thesis has been fulfilled to a large extent. A design has been made of the propulsion system by weighing the different component candidates against each other based on test results. Some components however still require testing and redesign, after which the final flight version design can be produced.
11 Recommendations

This chapter will include some recommendations on different aspects of a master thesis. Section 11.1 lists a few research topics that could advance the performance of the Dondersteen thrusters. In section 11.2 some of mistakes I have made are formulated as requirements for future students, so they are warned for making the same mistakes. Finally in section 11.3 some recommendations are made to the university, to better facilitate the conduction of master thesis research.

11.1 Recommendations for future research

The inside of the thrusters can be coated with an oxide layer, to achieve more isolation than the highly conductive silicon.

For interfaculty cooperation, members of both sides need to co-operate on the design. In this research it was a too one-sided approach. Of the author working mainly on the design and Dimes staff mainly on production. A design produced more in a direct cooperation between both fields would make for a better balance between optimal manufacturing, and necessary design lay-out.

The MEMS thruster could be combined with other MEMS developments. The most mentionable is to integrate a MEMS valve into the same component. This would reduce integration problems and further minimise the size of the propulsion system. The reason that this was not developed for the Dondersteen thruster is that the technology is too immature to handle the necessary pressures for long durations of time without leakage.

11.2 Recommendations for future students

Get a proper thesis topic before the literature study. It is very important to have the proper specific background knowledge on the thesis topic. Otherwise this necessary literature knowledge gathering will cost a lot of time during the thesis. My mistake was that the literature study was conducted in a too broad range; therefore there was only a very small overlap with the thesis.

First completely finish documenting the previous phase, before going to the next. Otherwise there will never be concrete numbers and figures of what went wrong/right. And it is difficult to show what you’ve actually spent your time on. Do not only focus on project deadlines. An extensively documented master thesis research document is required. If you only follow project deadlines which often have to do with manufacturing and testing, it is very easy to do more work than befits a master thesis. And you will then also face the problem of documenting all of these steps, and cramming it into the page limit.

Do not be tempted to calculate all the necessary behaviour yourself. Some of the research takes years to perform correctly. It is sometimes better to get a right source for than to try and approach everything with crude and idealised methods. In this case I spent too much time on heat conduction calculations and trying to model the behaviour in the heating chamber. Such CFD analysis would be a separate master thesis already. The values from experiences at Dimes should have been used earlier instead.
11.3 Recommendations for thesis supervision

Databases of available materials and test equipments within the whole university should be built and maintained. It is the author’s belief that all necessary testing equipment is generally available at the university, but there is no knowledge of what is where. A database with specialities of people within the entire university would help in the same way. That way if there are any questions on a specific topic, the students will know who to turn to.

It would be beneficial to have some more general technical staff for the entire university. The meetshop is a very good example and should to my opinion surely be extended.
12 Bibliography


Design, manufacturing and characterisation of a water fed CubeSat micro-resistojet

Jet Propulsion Laboratory California Institute of Technology, Pasadena, CA, USA, 2010.


Appendix A - Propulsion system concept trade-off

Table A.1 - Grading of different concepts for the trade-off of the DelFFi mission satellites propulsion system [10]

<table>
<thead>
<tr>
<th>Property</th>
<th>Mass</th>
<th>In orbit availability</th>
<th>TRL</th>
<th>scientific value</th>
<th>Electrical power consumption</th>
<th>storability</th>
<th>propellant safety</th>
<th>Cost</th>
<th>manufacturability</th>
<th>reliability</th>
<th>total</th>
<th>ranking</th>
</tr>
</thead>
<tbody>
<tr>
<td>Cold gas</td>
<td>0</td>
<td>8</td>
<td>8</td>
<td>7</td>
<td>7</td>
<td>7</td>
<td>6</td>
<td>7</td>
<td>7</td>
<td>7</td>
<td>414</td>
<td>1</td>
</tr>
<tr>
<td>solid chemical</td>
<td>1</td>
<td>8</td>
<td>7</td>
<td>4</td>
<td>7</td>
<td>8</td>
<td>4</td>
<td>5</td>
<td>4</td>
<td>7</td>
<td>373</td>
<td>5</td>
</tr>
<tr>
<td>Mono-propellant chemical</td>
<td>3</td>
<td>9</td>
<td>6</td>
<td>4</td>
<td>8</td>
<td>4</td>
<td>5</td>
<td>6</td>
<td>7</td>
<td>8</td>
<td>412</td>
<td>3</td>
</tr>
<tr>
<td>bipropellant chemical</td>
<td>1</td>
<td>9</td>
<td>6</td>
<td>4</td>
<td>8</td>
<td>5</td>
<td>4</td>
<td>5</td>
<td>4</td>
<td>4</td>
<td>334</td>
<td>7</td>
</tr>
<tr>
<td>Resistojet</td>
<td>2</td>
<td>7</td>
<td>7</td>
<td>3</td>
<td>4</td>
<td>7</td>
<td>7</td>
<td>7</td>
<td>8</td>
<td>6</td>
<td>413</td>
<td>2</td>
</tr>
<tr>
<td>Arcjet</td>
<td>4</td>
<td>7</td>
<td>7</td>
<td>3</td>
<td>2</td>
<td>7</td>
<td>7</td>
<td>7</td>
<td>6</td>
<td>4</td>
<td>381</td>
<td>4</td>
</tr>
<tr>
<td>Colloid</td>
<td>0</td>
<td>7</td>
<td>7</td>
<td>2</td>
<td>1</td>
<td>8</td>
<td>8</td>
<td>4</td>
<td>4</td>
<td>3</td>
<td>308</td>
<td>8</td>
</tr>
<tr>
<td>Solar thermal</td>
<td>2</td>
<td>6</td>
<td>6</td>
<td>5</td>
<td>7</td>
<td>7</td>
<td>7</td>
<td>3</td>
<td>3</td>
<td>4</td>
<td>339</td>
<td>6</td>
</tr>
</tbody>
</table>
Figure A.1 - Schematic overview of components for a proposed electrolysis propulsion system, those indicated with * are optional [20]

Appendix B - Fluidic properties

Table B.1 - The parameters for the Shomate equation to calculate the enthalpy change in fluids. Including the parameter validity range between the minimum temperature ($T_{\text{min}}$) and the maximum temperature ($T_{\text{max}}$) [23].

<table>
<thead>
<tr>
<th>Parameter validity range</th>
<th>$T_{\text{min}}$ [$K$]</th>
<th>Water (liquid)</th>
<th>Water (gaseous)</th>
<th>Water (gaseous)</th>
</tr>
</thead>
<tbody>
<tr>
<td></td>
<td>274</td>
<td>374</td>
<td>1700</td>
<td></td>
</tr>
<tr>
<td></td>
<td>500</td>
<td>1700</td>
<td>6000</td>
<td></td>
</tr>
<tr>
<td>Shomate equation properties</td>
<td>A</td>
<td>-203.6060</td>
<td>30.09200</td>
<td>41.96426</td>
</tr>
<tr>
<td></td>
<td>B</td>
<td>1523.290</td>
<td>6.832514</td>
<td>8.622053</td>
</tr>
<tr>
<td></td>
<td>C</td>
<td>-3196.413</td>
<td>6.793435</td>
<td>-1.499780</td>
</tr>
<tr>
<td></td>
<td>D</td>
<td>2474.455</td>
<td>-2.534480</td>
<td>0.098119</td>
</tr>
<tr>
<td></td>
<td>E</td>
<td>3.855326</td>
<td>0.082139</td>
<td>-11.15764</td>
</tr>
<tr>
<td></td>
<td>F</td>
<td>-256.5478</td>
<td>-250.8810</td>
<td>-272.1797</td>
</tr>
<tr>
<td></td>
<td>G</td>
<td>-488.7163</td>
<td>223.3967</td>
<td>219.7809</td>
</tr>
</tbody>
</table>
Table B.2 - The parameters for the Antoine equation, to calculate the vapour pressure or the boiling temperature at a certain pressure. Different values belong to different temperature ranges between the minimum temperature $T_{\text{min}}$ and maximum temperature $T_{\text{max}}$ [23].

<table>
<thead>
<tr>
<th>Validity range</th>
<th>Antoine equation parameter</th>
</tr>
</thead>
<tbody>
<tr>
<td>$T_{\text{min}}$ [K]</td>
<td>$T_{\text{max}}$ [K]</td>
</tr>
<tr>
<td>273</td>
<td>303</td>
</tr>
<tr>
<td>304</td>
<td>333</td>
</tr>
<tr>
<td>334</td>
<td>363</td>
</tr>
<tr>
<td>379</td>
<td>573</td>
</tr>
</tbody>
</table>

Table B.3 - Properties of water at atmospheric pressure [29]

<table>
<thead>
<tr>
<th>$T$ [°C]</th>
<th>$\rho$ [kg/m$^3$]</th>
<th>$\mu$ [kg/m/s]</th>
<th>$\nu$ [m$^2$/s]</th>
<th>$k$ [W/m/K]</th>
</tr>
</thead>
<tbody>
<tr>
<td>0</td>
<td>999.9</td>
<td>0.001787</td>
<td>$1.787 \times 10^{-6}$</td>
<td>0.56</td>
</tr>
<tr>
<td>5</td>
<td>1000</td>
<td>0.001514</td>
<td>$1.514 \times 10^{-6}$</td>
<td>0.57</td>
</tr>
<tr>
<td>10</td>
<td>999.7</td>
<td>0.001304</td>
<td>$1.304 \times 10^{-6}$</td>
<td>0.58</td>
</tr>
<tr>
<td>15</td>
<td>999.1</td>
<td>0.001137</td>
<td>$1.138 \times 10^{-6}$</td>
<td>0.59</td>
</tr>
<tr>
<td>20</td>
<td>998.2</td>
<td>0.001002</td>
<td>$1.004 \times 10^{-6}$</td>
<td>0.59</td>
</tr>
<tr>
<td>25</td>
<td>997.1</td>
<td>0.000891</td>
<td>$0.894 \times 10^{-6}$</td>
<td>0.60</td>
</tr>
<tr>
<td>30</td>
<td>995.7</td>
<td>0.000798</td>
<td>$0.802 \times 10^{-6}$</td>
<td>0.61</td>
</tr>
<tr>
<td>35</td>
<td>994.1</td>
<td>0.000720</td>
<td>$0.725 \times 10^{-6}$</td>
<td>0.62</td>
</tr>
<tr>
<td>40</td>
<td>992.3</td>
<td>0.000654</td>
<td>$0.659 \times 10^{-6}$</td>
<td>0.63</td>
</tr>
<tr>
<td>50</td>
<td>988.1</td>
<td>0.000548</td>
<td>$0.554 \times 10^{-6}$</td>
<td>0.64</td>
</tr>
<tr>
<td>60</td>
<td>983.2</td>
<td>0.000467</td>
<td>$0.475 \times 10^{-6}$</td>
<td>0.65</td>
</tr>
<tr>
<td>70</td>
<td>977.8</td>
<td>0.000405</td>
<td>$0.414 \times 10^{-6}$</td>
<td>0.66</td>
</tr>
<tr>
<td>80</td>
<td>971.8</td>
<td>0.000355</td>
<td>$0.366 \times 10^{-6}$</td>
<td>0.67</td>
</tr>
<tr>
<td>90</td>
<td>965.3</td>
<td>0.000316</td>
<td>$0.327 \times 10^{-6}$</td>
<td>0.67</td>
</tr>
<tr>
<td>100</td>
<td>958.4</td>
<td>0.000283</td>
<td>$0.295 \times 10^{-6}$</td>
<td>0.68</td>
</tr>
</tbody>
</table>

Appendix C - Nozzle optimisation relations

Table C.1 - Divergent section length $L_{\text{div}}$ and thrust efficiency $\eta$, for micro-nozzles with different throat curvatures. The nominal (non-viscous) thrust is $T_{\text{ideal}} = 1.14$ [mN], and the diverging and converging angles are $\theta_1 = \theta_2 = 15^\circ$, for all designs [37].

<table>
<thead>
<tr>
<th>$R_c/R_t [-]$</th>
<th>$L_{\text{div}}$ [μm]</th>
<th>$\eta$ [%]</th>
</tr>
</thead>
<tbody>
<tr>
<td>0</td>
<td>356.71</td>
<td>92.4</td>
</tr>
<tr>
<td>1</td>
<td>360.00</td>
<td>91.3</td>
</tr>
<tr>
<td>2</td>
<td>363.29</td>
<td>90.8</td>
</tr>
<tr>
<td>4</td>
<td>369.87</td>
<td>89.8</td>
</tr>
<tr>
<td>6</td>
<td>376.46</td>
<td>89.2</td>
</tr>
<tr>
<td>8</td>
<td>383.04</td>
<td>88.6</td>
</tr>
</tbody>
</table>
Table C.2 - The thruster efficiency $\eta$ for micro-nozzles with varying combinations of convergent half-angle, $\theta_1$, and divergent half-angle, $\theta_2$. All combinations were calculated for a nominal (non-viscous) thrust of $\sim 1$ [mN] and a sharp ($R_c=0$) nozzle throat curvature [37].

<table>
<thead>
<tr>
<th>$\theta_1$ [°]</th>
<th>$\theta_2$ [°]</th>
<th>$\eta$ [%]</th>
</tr>
</thead>
<tbody>
<tr>
<td>45</td>
<td>12</td>
<td>88.4</td>
</tr>
<tr>
<td>45</td>
<td>15</td>
<td>88.4</td>
</tr>
<tr>
<td>45</td>
<td>18</td>
<td>88.4</td>
</tr>
<tr>
<td>45</td>
<td>20</td>
<td>88.7</td>
</tr>
<tr>
<td>30</td>
<td>12</td>
<td>90.3</td>
</tr>
<tr>
<td>30</td>
<td>15</td>
<td>90.8</td>
</tr>
<tr>
<td>30</td>
<td>18</td>
<td>90.8</td>
</tr>
<tr>
<td>30</td>
<td>20</td>
<td>91.1</td>
</tr>
<tr>
<td>15</td>
<td>12</td>
<td>92.4</td>
</tr>
<tr>
<td>15</td>
<td>15</td>
<td>92.5</td>
</tr>
<tr>
<td>15</td>
<td>18</td>
<td>92.7</td>
</tr>
<tr>
<td>15</td>
<td>20</td>
<td>92.8</td>
</tr>
<tr>
<td>15</td>
<td>25</td>
<td>92.4</td>
</tr>
<tr>
<td>15</td>
<td>30</td>
<td>91.4</td>
</tr>
<tr>
<td>15</td>
<td>35</td>
<td>88.9</td>
</tr>
<tr>
<td>15</td>
<td>40</td>
<td>86.5</td>
</tr>
</tbody>
</table>

**Appendix D - Automotive gas generator test supplementary data**

![Technical drawing of the gas generator exterior](image)

Figure D.1 - Technical drawing of the gas generator exterior, with approximate dimensions in [mm]
There was a failure in capturing the pressure during the third gas generator test. Therefore there is no graph of the test chamber pressure behaviour during this test. The temperature was captured successfully during this test, and is displayed in Figure D.4 below.
Figure D.4 - Gas generator test 3 - Shock Temperature

Figure D.5 - Gas generator test 4 - Shock pressure
Figure D.6 - Gas generator test 4 - Shock Temperature
In the following table the thruster modules described in this thesis are combined to form complete thrusters. The number of each module in each design is shown, including the reasoning behind why this configuration is to be tested. Explanation of the acronyms is shown in Table E.2.

### Table E.1 - Different desirable thruster module configurations which would be crucial or interesting to test

<table>
<thead>
<tr>
<th>Thruster</th>
<th>Module</th>
<th>Total length [μm]</th>
<th>Number of thrusters</th>
<th>Reason</th>
</tr>
</thead>
<tbody>
<tr>
<td>T1</td>
<td>1 7</td>
<td>1</td>
<td>3</td>
<td>Nominal design test</td>
</tr>
<tr>
<td>T2</td>
<td>1 7</td>
<td>1</td>
<td>3</td>
<td>Test the effect of enlarging the diamond shapes (decreasing the contact area)</td>
</tr>
<tr>
<td>T3</td>
<td>1 7</td>
<td>1</td>
<td>3</td>
<td>Test the difference between serpentine and diamond channels</td>
</tr>
<tr>
<td>T4</td>
<td>1 7</td>
<td>1</td>
<td>3</td>
<td>Test the effect of enlarging the serpentine channel width (decreasing the contact area)</td>
</tr>
<tr>
<td>T5</td>
<td>1 7</td>
<td>1</td>
<td>3</td>
<td>Testing the nominal design with an different shaped nozzle design</td>
</tr>
<tr>
<td>T6</td>
<td>1 14</td>
<td>1</td>
<td>3</td>
<td>Testing the effect of heating chamber length on diamond shaped pillars</td>
</tr>
<tr>
<td>T7</td>
<td>1 14</td>
<td>1</td>
<td>3</td>
<td>Testing the effect of heating chamber length on serpentine heating channels</td>
</tr>
<tr>
<td>T8</td>
<td>1</td>
<td>1</td>
<td>3</td>
<td>Separately testing the nozzle performance</td>
</tr>
<tr>
<td>T9</td>
<td>1</td>
<td>1</td>
<td>3</td>
<td>Separately testing the different nozzle shape performance</td>
</tr>
<tr>
<td>T10</td>
<td>1</td>
<td>1</td>
<td>3</td>
<td>Production of a nozzle for Ivan's thruster, to test the difference between a MEMS slit nozzle and a nozzle produced using traditional machining techniques</td>
</tr>
</tbody>
</table>
Table E.2 - Explanation of the abbreviations used in Table E.1

<table>
<thead>
<tr>
<th>Abbreviation</th>
<th>Description</th>
</tr>
</thead>
<tbody>
<tr>
<td>I</td>
<td>Inlet</td>
</tr>
<tr>
<td>D-1</td>
<td>Nominal (small) diamond fins</td>
</tr>
<tr>
<td>D-2</td>
<td>Large diamond fins</td>
</tr>
<tr>
<td>S-1</td>
<td>Nominal (small) serpentine channels</td>
</tr>
<tr>
<td>S-2</td>
<td>Large serpentine channels</td>
</tr>
<tr>
<td>N-1</td>
<td>Nominal nozzle design (45 [deg] to 20 [deg], sharp edge, 25 [μm] throat)</td>
</tr>
<tr>
<td>N-2</td>
<td>Bell shaped optimised nozzle</td>
</tr>
<tr>
<td>N-K</td>
<td>Nozzle design for Ivan Krusharev</td>
</tr>
<tr>
<td>T</td>
<td>Thruster</td>
</tr>
</tbody>
</table>
Figure E.1 - The photoresist mask that was used in the etching process of the fluidic channels and heating elements of the Dondersteen prototype thrusters.

Appendix F - Component trade-off listings
Table F.1 - Different needle options for use as the fluidic interconnect between the thruster and the feed system.

<table>
<thead>
<tr>
<th>Name</th>
<th>Type</th>
<th>Diameter [mm]</th>
<th>Length [mm]</th>
<th>Website</th>
</tr>
</thead>
<tbody>
<tr>
<td>Neojectwegwerpnaalden</td>
<td>Luer</td>
<td>0.45</td>
<td>13</td>
<td><a href="http://www.praxisdienst.nl/nl/Injectie+infusie/Naalden/Wegwerpnaalden+s%5Cndaard+maten/Wegwerpnaalden+s%5Cnpeciale+maten+0+45+x+13mm+bruin.html">http://www.praxisdienst.nl/nl/Injectie+infusie/Naalden/Wegwerpnaalden+s\ndaard+maten/Wegwerpnaalden+s\npeciale+maten+0+45+x+13mm+bruin.html</a></td>
</tr>
<tr>
<td>Neodent dentalewegwerpnaalden</td>
<td>Luer</td>
<td>0.50</td>
<td>23</td>
<td><a href="http://www.praxisdienst.nl/nl/Injectie+infusie/Naalden/Wegwerpnaalden+s%5Cndaard+maten/Neodent+dentale+w%5Cnegwerpnaalden+0+50x+23mm+oranje.html">http://www.praxisdienst.nl/nl/Injectie+infusie/Naalden/Wegwerpnaalden+s\ndaard+maten/Neodent+dentale+w\negwerpnaalden+0+50x+23mm+oranje.html</a></td>
</tr>
<tr>
<td>Name</td>
<td>Type</td>
<td>Diameter [mm]</td>
<td>Length [mm]</td>
<td>Website</td>
</tr>
<tr>
<td>-------------------------------------------</td>
<td>---------------</td>
<td>---------------</td>
<td>-------------</td>
<td>-------------------------------------------------------------------------</td>
</tr>
<tr>
<td>Ultradunne canules recht</td>
<td>Luer</td>
<td>0.40</td>
<td>18</td>
<td><a href="http://www.praxisdienst.nl/nl/Injectie+infusie/Naalden/Wegwerpnaalden+s">http://www.praxisdienst.nl/nl/Injectie+infusie/Naalden/Wegwerpnaalden+s</a>	andaard+maten/Ultradunne+canules+recht+0+40+x+18.html</td>
</tr>
<tr>
<td>Sterican canules voor de dentale anesthesie</td>
<td>Luerlock</td>
<td>0.5</td>
<td>25</td>
<td><a href="http://www.praxisdienst.nl/nl/Injectie+infusie/Naalden/Wegwerpnaalden+s">http://www.praxisdienst.nl/nl/Injectie+infusie/Naalden/Wegwerpnaalden+s</a>	andaard+maten/Sterican+canules+voor+de+dentale+anesthesie+0+5+x+25	mm+oranje.html</td>
</tr>
<tr>
<td>Sterican*-insulinenaalden</td>
<td>Luerlock</td>
<td>0.45</td>
<td>12</td>
<td><a href="http://www.praxisdienst.nl/nl/Injectie+infusie/Naalden/Wegwerpnaalden+s">http://www.praxisdienst.nl/nl/Injectie+infusie/Naalden/Wegwerpnaalden+s</a>	andaard+maten/Sterican+insulinenaalden+0+45+x+12+bruin.html</td>
</tr>
<tr>
<td>BD 30Gx 1/2&quot; - 0.3x13mm Injectienaald geel</td>
<td>?</td>
<td>0.3</td>
<td>13</td>
<td><a href="http://www.medicosupply.nl/bd-30gx-1-2-03x13mm-injectienaald-geel.html">http://www.medicosupply.nl/bd-30gx-1-2-03x13mm-injectienaald-geel.html</a></td>
</tr>
<tr>
<td>Terumo Botox spuit/naald 1ml U-100</td>
<td>Met spuit</td>
<td>0.3</td>
<td>12</td>
<td><a href="http://www.medicosupply.nl/terumo-botox-spuitspuit-naald-1ml-u-100-29g-">http://www.medicosupply.nl/terumo-botox-spuitspuit-naald-1ml-u-100-29g-</a>	03x12mm.html</td>
</tr>
<tr>
<td>LLA4656300</td>
<td>Luer lock</td>
<td>0.3</td>
<td>12</td>
<td><a href="http://www.mls.be/html/catalogus.asp?lang=nl&amp;group=4.2.2&amp;detail=1">http://www.mls.be/html/catalogus.asp?lang=nl&amp;group=4.2.2&amp;detail=1</a></td>
</tr>
<tr>
<td>MS injectienaald luer lock</td>
<td>Luer lock</td>
<td>Multiple</td>
<td>Multiple</td>
<td><a href="http://www.schippers.nl/varkens/luer-lock/ms-injectienaald-luer-lock;pgid=8d8bg5xs3eN1SROoXOCSTBuVz0000Vl699KsH;sid=qCpv8QRRUImp8fiF-bqhgRbuMQAaRS_x7l=">http://www.schippers.nl/varkens/luer-lock/ms-injectienaald-luer-lock;pgid=8d8bg5xs3eN1SROoXOCSTBuVz0000Vl699KsH;sid=qCpv8QRRUImp8fiF-bqhgRbuMQAaRS_x7l=</a></td>
</tr>
<tr>
<td>Delvo injectienaald luer lock</td>
<td>Luer lock</td>
<td>Multiple</td>
<td>Multiple</td>
<td><a href="http://www.schippers.nl/varkens/luer-lock/delvo-injectienaald-luer-lock;pgid=8d8bg5xs3eN1SROoXOCSTBuVz0000Vl699KsH;sid=qCpv8QRRUImp8fiF-bqhgRbuMQAaRS_x7l=">http://www.schippers.nl/varkens/luer-lock/delvo-injectienaald-luer-lock;pgid=8d8bg5xs3eN1SROoXOCSTBuVz0000Vl699KsH;sid=qCpv8QRRUImp8fiF-bqhgRbuMQAaRS_x7l=</a></td>
</tr>
<tr>
<td>MS Alu-hub injectienaald luer lock per 100 stuks</td>
<td>Luer lock</td>
<td>Multiple</td>
<td>Multiple</td>
<td><a href="http://www.schippers.nl/varkens/luer-lock/ms-alu-hub-injectienaald-luer-lock-per-100-stuks;pgid=8d8bg5xs3eN1SROoXOCSTBuVz0000Vl699KsH;sid=qCpv8QRRUImp8fiF-bqhgRbuMQAaRS_x7l=">http://www.schippers.nl/varkens/luer-lock/ms-alu-hub-injectienaald-luer-lock-per-100-stuks;pgid=8d8bg5xs3eN1SROoXOCSTBuVz0000Vl699KsH;sid=qCpv8QRRUImp8fiF-bqhgRbuMQAaRS_x7l=</a></td>
</tr>
<tr>
<td>WELLER - KDS3012P - DISPENSING NEEDLE, GA30, ID 0.15MM</td>
<td>Luer lock</td>
<td>0.30</td>
<td>12.7</td>
<td><a href="http://nl.farnell.com/weller/kds3012p/dispensing-needle-ga30-id-0-15mm/dp/3975678">http://nl.farnell.com/weller/kds3012p/dispensing-needle-ga30-id-0-15mm/dp/3975678</a></td>
</tr>
</tbody>
</table>

Table F.2 - Different electrical connector options between PCB and electrical power supply
<table>
<thead>
<tr>
<th>Image</th>
<th>Name</th>
<th>Price [euro]</th>
<th>Availability</th>
<th>Link</th>
</tr>
</thead>
<tbody>
<tr>
<td>![Image](TE CONNECTIVITY / AMP - 1241050-7 - CONNECTOR, HEADER, 14POS, 2ROW.png)</td>
<td>TE CONNECTIVITY / AMP - 1241050-7 - CONNECTOR, HEADER, 14POS, 2ROW</td>
<td>0.975</td>
<td>9.450 (UK)</td>
<td><a href="http://nl.farnell.com/te-connectivity-amp/1241050-7/connector-header-14pos-2row-2-54mm/dp/2399735">http://nl.farnell.com/te-connectivity-amp/1241050-7/connector-header-14pos-2row-2-54mm/dp/2399735</a></td>
</tr>
<tr>
<td>![Image](TE CONNECTIVITY / AMP - 215307-7 - SOCKET, VERTICAL, 2ROW, 14WAY.png)</td>
<td>TE CONNECTIVITY / AMP - 215307-7 - SOCKET, VERTICAL, 2ROW, 14WAY</td>
<td>0.93</td>
<td>744 (UK) 111 (EU)</td>
<td><a href="http://nl.farnell.com/te-connectivity-amp/215307-7/socket-vertical-2row-14way/dp/3419230?MER=en-me-pd-r2-acce-con">http://nl.farnell.com/te-connectivity-amp/215307-7/socket-vertical-2row-14way/dp/3419230?MER=en-me-pd-r2-acce-con</a></td>
</tr>
<tr>
<td>![Image](MOLEX - 10-89-7140 - CONNECTOR, HEADER, 14POS, 2ROW, 2.54MM.png)</td>
<td>MOLEX - 10-89-7140 - CONNECTOR, HEADER, 14POS, 2ROW, 2.54MM</td>
<td>2.52</td>
<td>149 (US)</td>
<td><a href="http://nl.farnell.com/molex/10-89-7140/connector-header-14pos-2row-2-54mm/dp/2112307">http://nl.farnell.com/molex/10-89-7140/connector-header-14pos-2row-2-54mm/dp/2112307</a></td>
</tr>
<tr>
<td>![Image](FCI - 67997-216HLF - BOARD TO BOARD, HEADER, 16POS, 2ROW.png)</td>
<td>FCI - 67997-216HLF - BOARD TO BOARD, HEADER, 16POS, 2ROW</td>
<td>0.77</td>
<td>1745 (US)</td>
<td><a href="http://nl.farnell.com/fci/67997-216hlf/board-to-board-header-16pos-2row/dp/1923919">http://nl.farnell.com/fci/67997-216hlf/board-to-board-header-16pos-2row/dp/1923919</a></td>
</tr>
</tbody>
</table>
Appendix G - Microscope images for geometry validation

Figure G.1 - Inlet - width measurement

Figure G.2 - Heating module - Large diamonds - heater and pillar width, length measurements

Figure G.3 - Heating module - Large diamonds - Sidewall roughness measurement

Appendix H - Attempts at calculating the heat transfer of the thrusters

This section describes the calculation of the heat loss for the thruster to the surrounding. The following assumptions are made:

- The heating chamber is a rectangular prism.
- The height of the heating chamber is at least an order of magnitude smaller than the length and the width.
- The resistive heating element is a flat plate in the centre of the heating chamber.

Since the heating element is suspended in the flow, nearly all the heat from the element will have to pass through the propellant. There will be conduction to the exterior via the heating membrane itself. Due to the membrane thickness of only 300 [nm] this is only minimal. The largest part of the heat loss will occur through the contact between the water and the silicon wall of the heating chamber.

For the heat loss, the following has to be calculated sequentially:
- The heat flow from the heater to the water.
- The heat flow of the water to the surrounding silicon.
- The heat flow through the silicon.
- The radiation of the silicon to its surrounding.

The heat transfer from the heater to the propellant will determine the effectiveness of the heater. The heat transfer characteristics will determine the length and geometry of the heating channel, for it to heat the propellant to the desired temperature.

The conductive heat transfer through a material is [68]:

\[ Q_{ij} = C_{ij} (T_i - T_j) \]  

Where:

- \( T \) Temperature [K]
- \( C_{ij} \) Conductive coupling between the two surfaces [W/K]

For the conduction through a part the conductive coupling can be determined by:

\[ C_{ij} = \frac{k \cdot A}{L} \]  

Where:

- \( k \) Conductivity of the medium [W/m/K] (temperature and material dependent)
- \( A \) Cross-sectional area [m\(^2\)]
- \( L \) Length of the conductive path [m]

For conduction through contacting surfaces:

\[ C_{ij} = h_c \cdot A \]  

Where:

- \( h_c \) contact conductance [W/m\(^2\)/K]
- \( A \) Contact area [m\(^2\)]
When looking at the shape dependence of $C_{ij}$ the following standard shapes are worked out as following:

![Diagram of block/flat plate and disc/radial direction]

The heater section will be approximated by this last shape of the disc in radial direction. $\alpha$ will then be $2\pi$, $d$ will be the length of the heating section and $R_1 - R_0$ is the thickness of the layer.

A cross-section of the heater section is shown in the following picture.

![Cross-section of the heating section, from the central heater to the outside casing]

Each of the heat couplings now has to be determined. First of all there is a conductive coupling from the heater to the water.

The optional glass layer has a thickness of 500 [$\mu$m]. And the metal reflector which is chosen is aluminium.

**Conductive coupling from heater to propellant**

The convective heat transfer from a surface to a fluid is shown in equation 12-4 below.

$$Q = h \cdot A \cdot (T_w - T_f)$$  \hspace{1cm} 12-4

Where:

- $h$ Convection coefficient [W/m$^2$/K]
- $A$ Contact area [m$^2$]
For the calculation of this heat transfer it is crucial to know the flow regime. Whether the flow is laminar, in transition or turbulent has a big impact on the convection coefficient $h$. The Reynolds number gives an indication in which flow regime a flow is. The Reynolds number based on tube diameter is usually defined as determined equation 12-5 below.

$$Re = \frac{U \cdot D_h}{\nu}$$  

Where:
- $Re$ Reynolds number [-]
- $U$ Flow velocity [m/s]
- $D_h$ Hydraulic diameter of the tube [m]
- $\nu$ Kinematic viscosity [m²/s]

The equivalent hydraulic diameter $D_h$ is a measure to relate the area of any cross-section to the diameter of a circular cross-section with the same area. This parameter is defined as follows:

$$D_h = \frac{4A}{p}$$

Where:
- $A$ Cross-sectional area of the tube [m²]
- $p$ Periphery or inner wall length of the cross-section of the tube [m]

The kinematic viscosity is calculated with equation 3-3.

$$\nu = \frac{\mu}{\rho}$$

Where:
- $\mu$ Dynamic (absolute) viscosity [kg/s/m]
- $\rho$ Density of the flow [kg/m³]

The flow velocity can of course be calculated as follows.

$$U = \frac{\dot{m}}{\rho \cdot A}$$

Where:
- $\dot{m}$ Mass flow [kg/s]
Since the area of a circle is \( A = \frac{1}{4} \cdot \pi \cdot D^2 \), the Reynolds number in equation 12-5 can be rewritten using equations 3-3 and 3-2.

\[
Re = \frac{\dot{m}}{\rho \cdot \left( \frac{1}{4} \cdot \pi \cdot D^2 \right) \cdot \left( \frac{\mu}{\rho} \right)} = \frac{4 \cdot \dot{m}}{\pi \cdot D_h \cdot \mu}
\]

Assuming that the rectangular flow channel has smooth walls, the critical Reynolds numbers can be approximated using the aspect ratio of the flow channel \( \alpha = \frac{h}{w_{ch}} \). The relation between the aspect ratio and critical Reynolds number can be seen in Table 12.1. For values of the aspect ratio which are not mentioned, the critical Reynolds number will be linearly interpolated from the known points.

Table 12.1 - Critical Reynolds number for rectangular channels, depending on aspect ratio [31][32]

<table>
<thead>
<tr>
<th>Aspect ratio (( \alpha ))</th>
<th>Critical Reynolds number (Re)</th>
</tr>
</thead>
<tbody>
<tr>
<td>( \leq 0.2 )</td>
<td>2500</td>
</tr>
<tr>
<td>1.0</td>
<td>2200</td>
</tr>
<tr>
<td>( \geq 5.0 )</td>
<td>2500</td>
</tr>
</tbody>
</table>

For the heat transfer calculations, the following additional assumptions are made:

- The duct is long enough for the flow to be fully developed over most of its length \( L \).
- The entering flow has a uniform temperature \( T_0 \) at \( x=0 \).

The heat will flow from the wall to the propellant with a heat flux \( q_w' \), due to the wall temperature \( T_w \). Both the heat flux and wall temperature can be functions of longitudinal direction of the duct. In the heating region, a thermal boundary layer will start close to the wall, as can be seen in Figure 12.1. This boundary layer will grow in thickness in the downstream longitudinal direction, until the flow is fully thermally developed at \( x = X_f \).
The thermal entrance region has different thermal behaviour, and therefore calculation methods than the thermally fully developed region. It is therefore necessary to see how significant this region is, in order to assess whether it needs to be considered or can be neglected. The flow entrance length, for low and high Prandtl numbers, can be estimated using equation 12-10 [29].

\[
\frac{X_T}{D_h} \approx 0.05 \cdot \text{Re}_{D_h} \cdot \text{Pr}
\]

Where:
- \(X_T\) Thermal entrance length [m]
- \(\text{Re}_{D_h}\) Reynolds number based on tube diameter [-]
- \(\text{Pr}\) Prandtl number [-]

The Prandtl number can be calculated with the following formula [62]:

\[
\text{Pr} = \frac{\mu \cdot C_p}{k_p}
\]

Where:
- \(C_p\) Heat capacity of the fluid [J/kg/K]
- \(k_p\) Conductivity of the fluid [W/m/K]

For the initial calculations, the water shown in Table B.3 in Appendix B is used. The assumption is made that the local pressure changes are negligible.

**Conductive coupling from heater to propellant assuming uniform wall heat flux**
When the local wall heat flux \( q_w \) is assumed to be constant, the Nusselt number can be used to determine the heat transfer coefficient. The definition of the Nusselt number based on hydraulic diameter is shown in equation 12-12.

\[
\text{Nu}_{D_h} = \frac{h \cdot D_h}{k}
\]

Where:
- \( \text{Nu}_{D_h} \) Nusselt number based on hydraulic diameter [-]

For laminar flow and slug flow, the Nusselt number is in the order of 1. For turbulent flow, the Nusselt number is generally in the order of 100-1000. For the case of uniform wall heat flux, with fully developed laminar flow, an empirical relation for the Nusselt number is shown in equation 12-13 below [69].

\[
\text{Nu}_{D_h} = -1.047 + 9.326 \cdot G
\]

Where:
- \( G \) A parameter given by equation 12-14 below

\[
G = \frac{1 + \left(\frac{1}{\alpha}\right)^2}{\left(1 + \frac{1}{\alpha}\right)^2}
\]

Where:
- \( \alpha \) Aspect ratio of the channel \( \alpha = \frac{h_{ch}}{w_{ch}} \)

The Reynolds number will thus first have to be calculated to show that the flow regime. After that the relevant Nusselt number can be found in the literature. Using the Nusselt number, equations 12-12, 3-26 and the known values of the conductivity \( k \) from, the convection coefficient \( h \) can be determined. The flow temperature calculations can be conducted for small segments in the longitudinal direction of the stream channel, with a length of \( dx \). Under the given assumption of constant wall heat flux, with a known total heat input of the wall section \( Q \) can be used to calculate the temperature difference between the flow temperature and wall temperature can be calculated using equation 12-4. For this small flow section it is clear that the temperature increase due to the heat flux can be calculated using equation 12-15 below.

\[
dT = C_p \cdot \dot{m}_f \cdot Q
\]

Where:
- \( \dot{m}_f \) Mass flow of the flow [kg/s]

This only holds when the length of the channel section is smaller than the flow velocity. Therefore the section length \( dx \) has to be chosen to be adequately small to ensure that \( dx < 1 \text{[s]} \cdot V \).

**Conductive coupling from heater to propellant assuming isothermal walls**
In these calculations, instead of a constant heat flux, a constant wall temperature is assumed. This results in a slightly different Nusselt number. The channel is again subdivided into small section with a length of dx. The flow temperature in the section is assumed to be equal to the flow temperature at the exit of the previous flow section. Using equation 12-12 and 12-4 the heat transfer to the channel section can be determined. Using this heat transfer, equation 12-15 can be used to calculate the flow temperature at the end of the channel section. Calculating these properties for every channel section is calculated. The sum of all these section will yield the total heat transferred and the flow temperature at the end of the channel.

For the calculations, the differentiation has to be made between “fully developed flow” and “slug flow”. The latter is a fluid flow with a Prandtl number approaching zero. For that kind of flow the viscosity is so much smaller than the thermal diffusivity, that the longitudinal velocity profile remains uniform.

**Heat loss to the surrounding**

In the actual satellite, the thruster will be suspended in a cylinder which guides it through the structure, to end up in

**Alternative method for calculating the heat transfer from heater to the flow**

For high power density and small particle size, the heat transfer in packed beds is usually calculated using the Achenbach correlation (from source: B-09 page (85)). The heat transfer coefficient can be expressed by:

\[
h_t = Nu \frac{k_b}{D_p} \tag{12-16}
\]

Where:
- \(h_t\) Heat transfer coefficient [W/m\(^2\)/K]
- \(Nu\) Nusselt number [-]
- \(k_b\) Bulk gas conductivity [W/m/K]
- \(D_p\) Packed bed particle diameter [m]

The Nusselt number can be determined by:

\[
Nu = \frac{1-\varepsilon}{\varepsilon} \left( 0.622926 \left( \frac{Re}{1-\varepsilon} \right)^{2.32} + 6.44603 \cdot 10^{-4} \left( \frac{Re}{1-\varepsilon} \right)^3 \right) Pr^{0.33} \tag{12-17}
\]

Where:
- \(\varepsilon\) Bed porosity [-]
- \(Re\) Reynolds number [-]
- \(Pr\) Prandtl number [-]

The Reynolds in its turn can then be determined using: 
Design, manufacturing and characterisation of a water fed CubeSat micro-resistojet

\[ \text{Re} = \frac{\rho \cdot D_p \cdot \nu_s}{\mu} \]  \hspace{1cm} 12-18

Where:
- \( \rho \) Fluid density [kg/m\(^3\)]
- \( D_p \) Particle diameter [m]
- \( \nu_s \) Superficial or free stream fluid velocity [m/s]
- \( \mu \) Fluid viscosity [N·s/m\(^2\)]

The Prandtl number can be calculated with the following formula:

\[ Pr = \frac{\mu \cdot C_p}{k_b} \]  \hspace{1cm} 12-19

Where:
- \( C_p \) Heat capacity of the fluid [J/kg/K]

Knowing the heat transfer coefficient, the heat transfer to the surrounding can be determined. Since the heating element is totally surrounded by the propellant, it can reasonably be assumed that all conductive heat is transferred between the heater and the silicon via the water. The radiation between the heater and the silicon though the water needs to be determined later.

Knowing the transfer coefficient, the minimal length of the heating element can be determined.