Miniaturisation of a Hydrogen Peroxide Thruster

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Summary

A continuing demand exists to develop the capabilities of nanosatellites. A key element limiting the range of nanosatellite applications is the accommodation of a propulsion system. Assessment of present micropropulsion developments revealed that few would conform to the nanosatellite constraints. This research addressed this need and considered the miniaturisation of a monopropellant thruster together with the factors that affected it. Hydrogen peroxide was selected as the propellant for the thruster as it is considered to be a Green propellant, non-toxic and non-carcinogenic.

The detailed miniaturisation of the monopropellant thruster focused upon two major components: the decomposition chamber and the exhaust nozzle. An analysis was conducted of available empirical data to determine the optimal configuration of a decomposition chamber in terms of the geometry of the chamber as well as the morphology of the catalyst within it. Experimental tests allowed the effect of catalyst bed length to be investigated as a function of decomposition chamber diameter. The results indicate that an optimal mass flow rate exists for each length of catalyst bed and a shorter bed is preferred due to thermal characteristics.

The performance of the exhaust nozzle was characterised using numerical modelling. Two different aspects of the nozzle were considered: the scale of the nozzle and the geometry of the nozzle in terms of throat profile and expansion half-angle. Nine different thrust levels were considered, which covered the range 500 – 1 mN and the results indicate that the rate of boundary layer development is logarithmic. A thrust magnitude of 50 mN was found to be a critical point, beyond which the rate of boundary layer development within the nozzle accelerates. Consideration of different model geometries revealed that the thrust generated by the nozzle improved when a sharp throat geometry was employed.

Keywords: Satellite Propulsion, Hydrogen Peroxide, Micropropulsion, Nanosatellite
A continuing demand exists to develop the capabilities of nanosatellites. A key element limiting the range of nanosatellite applications is the accommodation of a propulsion system. This research investigated this need and considered the miniaturisation of a monopropellant thruster.

A literature review considered all aspects of micropropulsion together with enabling technologies. Assessment of present micropropulsion developments revealed that few would conform to the nanosatellite constraints. In addition the complexities associated with the miniaturisation of a propulsion system such as the modification of fluid flow, were highlighted.

A review of the possible applications of a propulsion enhanced nanosatellite resulted in the creation of an inspection mission scenario. Assessment of present micropropulsion developments revealed none could fulfil the mission requirements, but a miniaturised chemical propulsion could. This led to the initiation of research to miniaturise a monopropellant thruster that would meet the mission requirements within the platform constraints. Hydrogen peroxide was selected as the propellant as it is considered to be a Green, non-toxic and non-carcinogenic propellant.

The effect of scaling on the thermal characteristics of the thruster was evaluated using numerical models, which considered the effect of chamber wall thickness. It was concluded that a thin walled chamber should be combined with a heat-shield to allow radiated heat to be reflected back towards the decomposition chamber.

The options available for the manufacture of a micropropulsion system were considered with respect to machining accuracy, materials and cost. There are two main options: Micro-Electro-Mechanical Systems (MEMS) technologies or micro conventional precision machining methods. It was concluded that at present the use of the latter was preferred as the level of machining accuracy is higher and conventional materials can be used.

Following these analyses the detailed miniaturisation of the monopropellant thruster began, with a focus upon two major components: the decomposition chamber and the exhaust nozzle.

The use of hydrogen peroxide as a rocket monopropellant was prevalent in the 1960’s. Since then its use has waned in favour of other monopropellants such as hydrazine, which exhibit higher performance and improved storage characteristics. At that time significant research was conducted into the performance of hydrogen peroxide, but its use for low thrust applications was not considered. An analysis of available empirical data was conducted to determine the optimal configuration of a decomposition chamber in terms of the geometry of the decomposition chamber as well as the morphology of the catalyst bed. Two different catalyst morphologies were considered: a monolithic catalyst bed and a compressed powder catalyst bed.
The monolithic morphology was based upon a ceramic foam substrate coated with a manganese oxide catalyst. Overall it generated good decomposition characteristics, but suffered from severe internal structural degradation. A compressed silver powder catalyst generated excellent decomposition characteristics and enabled the effect of catalyst bed length to be investigated as a function of decomposition chamber diameter. The results from these tests indicate that a compressed silver powder catalyst bed is a suitable alternative to silver gauzes for use in small diameter decomposition chambers. In addition the results showed that an optimal mass flow rate exists for each length of catalyst bed and a shorter bed is preferred due to thermal characteristics.

The performance of the exhaust nozzle is critical to the overall performance of the thruster, but the presence of a boundary layer within it causes a performance deficit. The development of a boundary layer within an exhaust nozzle was considered numerically using the commercial computational fluid dynamics software Fluent®. Two different aspects of the nozzle were considered: the scale of the nozzle and the geometry of the nozzle in terms of throat profile and expansion half-angle. Nine different thrust levels were considered, which covered the range 500 – 1 mN and the results indicate that the rate of boundary layer development is logarithmic in nature. A thrust magnitude of 50 mN was found to correspond to a critical point, below which the rate of boundary layer development accelerates. Consideration of different nozzle geometries revealed that the thrust generated by the nozzle improves when a sharp throat geometry is used.

In summary this research has considered the effects of miniaturisation on the performance of a hydrogen peroxide thruster. This work has addressed the gap in published data relating to hydrogen peroxide thrusters and contributed the state of the art in the following ways:

- Successful miniaturisation of a decomposition chamber capable of stably and repeatedly decomposing a flow of hydrogen peroxide at a rate of 0.5 g s⁻¹ through an uninsulated catalyst bed 6.7 mm in diameter and 5 mm in length.
- Identification of a trend in the performance of a decomposition chamber according to the geometry of the catalyst bed and the propellant mass flow rate.
- Identification of a trend in the development of a boundary layer within a thruster nozzle that suggests a logarithmic rate of growth.
- Identification of a critical point relating to nozzle size, which corresponds to the size of nozzle that is the smallest that may be considered without boundary layer analysis.

One international conference paper has been presented on this work and one paper has been submitted to a refereed journal for publication. A second conference paper is currently in progress. The US Air Force European Office of Aerospace Research and Development funded a research contract to investigate the miniaturisation of a hydrogen peroxide thruster and two reports have been submitted on this work.
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Nomenclature

\( a \)  
Semi-Major Axis of Orbit

\( a_t \)  
Semi-Major Axis of Transfer Orbit

\( A \)  
Frontal Area

\( A^* \)  
Cross-sectional Area of Throat

\( A_c \)  
Cross-sectional Area of Catalyst

\( A_e \)  
Cross-sectional Area at Nozzle Exit

\( B \)  
Ballistic Coefficient

\( c \)  
Characteristic Exhaust Velocity

\( c_d \)  
Drag Coefficient

\( c_f \)  
Thrust Coefficient

\( c_p \)  
Specific Heat at Constant Pressure

\( \eta \)  
Proportionality Constant

\( d \)  
Number of Days in Transit Between Satellites

\( d^* \)  
Throat Diameter

\( e \)  
Eccentricity of Orbit

\( E \)  
Orbital Energy

\( E \)  
Total Energy

\( E_a \)  
Activation Energy

\( F \)  
Force Vector

\( g \)  
Acceleration due to Gravity

\( g_0 \)  
Acceleration due to Gravity at the Earth’s Surface

\( h \)  
Enthalpy

\( h_1 \)  
Enthalpy at Position 1

\( h_2 \)  
Enthalpy at Position 2

\( h_c \)  
Enthalpy in Chamber

\( h_e \)  
Enthalpy at Exit

\( H_2O_2 \)  
Hydrogen Peroxide

\( i \)  
Inclination of Orbit

\( I \)  
Impulse

\( I \)  
Spacecraft Inertia

\( I \)  
Turbulence Intensity

\( I_{sp} \)  
Specific Impulse

\( J_{22} \)  
First Zonal Harmonic

\( k \)  
Thermal Conductivity

\( k \)  
Turbulence Kinetic Energy

\( Kn \)  
Knudsen Number

\( l \)  
Turbulence Length Scale

\( L \)  
Characteristic Length

\( L \)  
Thruster Moment Arm

\( L \)  
Tube Diameter

\( L_n \)  
Nozzle Length

\( L^f \)  
Loading Factor
Nomenclature

\( m \quad \text{Spacecraft Mass} \)

\( \dot{m} \quad \text{Mass Flow Rate} \)

\( m_0 \quad \text{Initial Spacecraft Mass} \)

\( m_p \quad \text{Propellant Mass} \)

\( \eta \quad \text{Molecular Mass} \)

\( n \quad \text{Normal Vector} \)

\( n \quad \text{Number of Orbits to Complete Maneuvre} \)

\( p_o \quad \text{Ambient Pressure} \)

\( p_c \quad \text{Chamber Pressure} \)

\( p_e \quad \text{Exit Pressure} \)

\( p_i/p_e \quad \text{Pressure Ratio} \)

\( q \quad \text{Heat Flux} \)

\( r \quad \text{Orbit Radius} \)

\( r \quad \text{Spacecraft Radius} \)

\( r^* \quad \text{Throat Radius} \)

\( r_i \quad \text{Initial Orbit Radius} \)

\( r_f \quad \text{Final Orbit Radius} \)

\( R \quad \text{Local Gas Constant} \)

\( \bar{R} \quad \text{Universal Gas Constant} \)

\( Re \quad \text{Reynolds Number} \)

\( Re_c \quad \text{Critical Reynolds Number} \)

\( S \quad \text{Stationary Control Surface} \)

\( t \quad \text{Time} \)

\( T \quad \text{Time Period of Orbit} \)

\( T_e \quad \text{Chamber Temperature} \)

\( T \quad \text{Thrust} \)

\( u \quad \text{x-Direction Velocity} \)

\( u^+ \quad \text{Boundary Layer Velocity Distribution} \)

\( u_{rms} \quad \text{Root Mean Square of the Velocity Fluctuations} \)

\( u_c \quad \text{Velocity in Chamber} \)

\( u_e \quad \text{Exit Velocity} \)

\( u_e \quad \text{Velocity at Exit} \)

\( u_{eff} \quad \text{Effective Exhaust Velocity} \)

\( u_{w} \quad \text{Freestream Velocity} \)

\( u_w \quad \text{Velocity at Wall} \)

\( u_f \quad \text{Friction Velocity} \)

\( V \quad \text{Orbital Velocity} \)

\( V \quad \text{Velocity Vector} \)

\( v \quad \text{y-Direction Velocity} \)

\( V_{TO} \quad \text{Orbital Velocity at Insertion Point into Transfer Orbit} \)

\( x \quad \text{Total Number of Inspections} \)

\( y^* \quad \text{Characteristic Wall Coordinate} \)

\( \Delta V \quad \text{Change in Velocity} \)

\( \Delta V_{ins} \quad \text{Change in Velocity Required to Complete Maneuvre} \)

\( \Delta V_{TOL} \quad \text{Total Change in Velocity Required} \)
<table>
<thead>
<tr>
<th>Symbol</th>
<th>Definition</th>
</tr>
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<tbody>
<tr>
<td>$\alpha$</td>
<td>Expansion Half-Angle</td>
</tr>
<tr>
<td>$\alpha_{e}$</td>
<td>Effective Expansion Half-Angle</td>
</tr>
<tr>
<td>$\gamma$</td>
<td>Ratio of Specific Heats</td>
</tr>
<tr>
<td>$\delta$</td>
<td>Boundary Layer Thickness</td>
</tr>
<tr>
<td>$\varepsilon$</td>
<td>Turbulent Dissipation Rate</td>
</tr>
<tr>
<td>$\eta$</td>
<td>Decomposition Efficiency</td>
</tr>
<tr>
<td>$\theta$</td>
<td>Angle of Movement</td>
</tr>
<tr>
<td>$\dot{\theta}$</td>
<td>Angular Velocity</td>
</tr>
<tr>
<td>$\ddot{\theta}$</td>
<td>Angular Acceleration</td>
</tr>
<tr>
<td>$\mu$</td>
<td>Earth's Gravitational Constant</td>
</tr>
<tr>
<td>$\mu$</td>
<td>Kinematic Viscosity</td>
</tr>
<tr>
<td>$\nu$</td>
<td>Dynamic Viscosity</td>
</tr>
<tr>
<td>$\nu$</td>
<td>True Anomaly</td>
</tr>
<tr>
<td>$\rho$</td>
<td>Atmospheric Density</td>
</tr>
<tr>
<td>$\rho$</td>
<td>Density</td>
</tr>
<tr>
<td>$\tau$</td>
<td>Time Averaged Shear Stress</td>
</tr>
<tr>
<td>$\phi$</td>
<td>Flow Variable</td>
</tr>
<tr>
<td>$\psi$</td>
<td>Phase Angle Separation</td>
</tr>
<tr>
<td>$\omega$</td>
<td>Angular Velocity</td>
</tr>
<tr>
<td>$\omega$</td>
<td>Argument of Perigee</td>
</tr>
<tr>
<td>$\Omega$</td>
<td>Right Ascension of Ascending Node</td>
</tr>
</tbody>
</table>

<table>
<thead>
<tr>
<th>Acronym</th>
<th>Description</th>
</tr>
</thead>
<tbody>
<tr>
<td>ADN</td>
<td>Ammoniumdinitramide</td>
</tr>
<tr>
<td>ASTC</td>
<td>Angstrom Space Technology Centre</td>
</tr>
<tr>
<td>CFD</td>
<td>Computational Fluid Dynamics</td>
</tr>
<tr>
<td>CINCH</td>
<td>Competitive Impulse Non Carcinogenic Hypergol</td>
</tr>
<tr>
<td>DMAZ</td>
<td>Dimethylaminoethylaxide</td>
</tr>
<tr>
<td>DSMC</td>
<td>Direct Simulation Monte Carlo (method)</td>
</tr>
<tr>
<td>EADS</td>
<td>European Aeronautic Defence and Space (company)</td>
</tr>
<tr>
<td>ECAPS</td>
<td>ECological Advanced Propulsion Systems</td>
</tr>
<tr>
<td>EDL</td>
<td>Electric Discharge Layer</td>
</tr>
<tr>
<td>GAP</td>
<td>Glycidyl Azide Polymer</td>
</tr>
<tr>
<td>GPS</td>
<td>Global Positioning System</td>
</tr>
<tr>
<td>HAN</td>
<td>Hydroxyl Ammonium Nitrate</td>
</tr>
<tr>
<td>HNF</td>
<td>Hydrazinium Nitroformate</td>
</tr>
<tr>
<td>LED</td>
<td>Light Emitting Diode</td>
</tr>
<tr>
<td>LEO</td>
<td>Low Earth Orbit</td>
</tr>
<tr>
<td>LISA</td>
<td>Laser Interferometer Space Antenna</td>
</tr>
<tr>
<td>MEMS</td>
<td>Micro Electro Mechanical Systems</td>
</tr>
<tr>
<td>MiniAERCam</td>
<td>Miniature Autonomous Extravehicular Robotic Camera</td>
</tr>
<tr>
<td>NASA</td>
<td>National Aeronautics and Space Administration</td>
</tr>
<tr>
<td>NEAR</td>
<td>Near Earth Asteroid Rendezvous</td>
</tr>
<tr>
<td>PROBA</td>
<td>PProject for OnBoard Autonomy</td>
</tr>
<tr>
<td>SNAP</td>
<td>Surrey Nanosatellite Applications Platform</td>
</tr>
<tr>
<td>SSTL</td>
<td>Surrey Satellite Technology Limited</td>
</tr>
</tbody>
</table>
1 Introduction

The need to extend the on-orbit capabilities of nanosatellites has led to the requirement for a miniaturised propulsion system. The mass, power and volume constraints placed on this system by the host nanosatellite platform lead to the selection of a chemical propulsion system. In order to optimise the thrust available and minimise complexity a monopropellant system is a favourable solution.

1.1 Motivation for Research

A continuing demand exists to develop the capabilities of micro-nanosatellites. A key element that limits the range of applications for which a micro-nanosatellite is suitable, is the propulsion system. A micro-nanosatellite requires a propulsion system that is capable of producing low magnitude thrusts with repeated accuracy. For the propulsion system to be accommodated onboard the micro-nanosatellite it should also be miniaturised so as not to impact significantly on the satellite mass, power and volume. Assessment of the available propulsion systems that may be appropriate for use on a micro-nanosatellite revealed a need for a low mass, low power propulsion system. This research addressed this need and considered the miniaturisation of a monopropellant thruster together with the factors that affected it.

1.2 Scope of Research

The miniaturisation of any complete system is a complex task and is usually facilitated if approached in stages as this ensures the integrity of the system is not compromised. In terms of a propulsion system, the range of aspects to be considered include performance related factors in addition to manufacturing and assembly issues. To reduce the magnitude of the task, this research focused upon the decomposition chamber and nozzle components of a propulsion system and considered the optimisation of them as they were miniaturised in stages.

The sources of performance losses in a conventional sized thruster will still occur in a miniaturised one, however additional losses are also likely to be present. The losses were considered from main two perspectives, fluid flow and thermal.

Classical propulsion design methods are primarily based on various flow assumptions, which may not apply at micro scales. For example effects such as viscous losses, surface tension, surface charging, surface roughness, low Reynolds number and compressibility, which can be ignored, or
simply factored at larger scales, may become of critical importance. The development of a fluid flow in a small channel was considered to develop an understanding of which the critical processes are and how they result in losses.

The thrust produced by any chemical thruster is a direct function of the efficiency of the combustion/decomposition process used to create the exhaust gases. The thermal characteristics of the thruster directly influence the temperature of the exhaust gases, therefore any thermal losses should be minimised. The reduced dimensions of a miniaturised thruster lead to an increased surface area to volume ratio, thereby allowing greater heat transfer. This effect can be both an advantage and a disadvantage. As the temperature inside the combustion/decomposition chamber of a thruster increases, cooling of the chamber walls may be required to prevent them melting. Nevertheless if this effect is too great the reaction may be quenched. The effect of geometry on the thermal characteristics of the thruster was addressed to minimise thermal losses and optimise the resultant performance.

The practicalities of creating a miniaturised propulsion system were considered in terms of the technology available to manufacture the system. The growth of the electronics industry has led to significant developments in silicon processing methods. These can be used to create complex microstructures, which have a high strength to weight ratio. This Micro Electro-Mechanical Systems (MEMS) technology can potentially be used to great advantage within micro-/nanosatellite development. The miniaturisation of computational modules will allow increased data processing capabilities to be accommodated on board a micro-/nanosatellites, however the threshold for use in a propulsion system remains unclear. The alternative to using MEMS technology to fabricate a miniaturised propulsion system is to use micro conventional machining methods. This approach utilises conventional methods to machine components from conventional materials using miniaturised tooling. There are advantages to both of these manufacturing methods and the applicability of both methods was considered with respect to a miniaturised propulsion system.

The impact of all these considerations onto the design and development of a micropropulsion system is complex. For example, the predicted thermal characteristics of the propulsion system will be altered, however the effect of the use of a different structural material may also need to be quantified. To assess the effects of miniaturisation on a propulsion system it was decided that two key components would be focused upon. The decomposition chamber and the exhaust nozzle.

The design of a decomposition chamber is usually based upon empirical relations, which consider the geometry and morphology of the catalyst bed. The scale of the thruster under development here prevented these relations from being applicable as the dimensions suggested were inappropriate in terms of thermal losses and catalyst material accommodation. An empirical
approach was adopted and the length of catalyst bed required for a given density of catalyst material and diameter of decomposition chamber was investigated. This allowed the scaling of a decomposition chamber to be considered as a function of the chamber diameter.

The design of an exhaust nozzle is based upon isentropic relations to predict the expansion of the flow. The presence of any factors that reduce the expansion of the flow will reduce the observed performance of the nozzle. The primary source of losses in a fluid flow is a boundary layer, which acts as a momentum sink, reducing the freestream velocity of the flow. The nozzle of a miniaturised thruster will be small to accommodate the low mass flow rate efficiently. This will lead to a low Reynolds number and a thick boundary layer. The presence of a boundary layer within a nozzle was investigated numerically to determine the effectiveness of a nozzle at small scales.

1.3 Aims and Objectives

1.3.1 Aims

In view of the needs and constraints of a nanosatellite platform the aims of this research effort have been determined as:

i. Evaluate the constraints placed upon a propulsion system that may be accommodated onboard a nanosatellite.

ii. Assess possible missions scenarios for a propulsion-enhanced nanosatellite and select a candidate mission.

iii. Ascertain what type of propulsion system would be most appropriate to meet these mission requirements.

iv. Consider the miniaturisation of a conventional small monopropellant propulsion system to an extent that would allow accommodation on board a nanosatellite platform.

v. Characterise the performance of a monopropellant decomposition chamber as it is miniaturised to meet the mission requirements.

vi. Characterise the performance of a rocket nozzle as it is miniaturised to meet the mission requirements.

1.3.2 Objectives

The work required to achieve these aims has been translated into the following objectives:
Chapter 1: Introduction

i. Select a suitable nanosatellite platform that would benefit from the addition of a propulsion system.

ii. Analyse the possible missions that this platform could perform and develop a set of realistic mission requirements.

iii. Conduct a trade-off of propulsion system technologies including present developments to find a system that meets the mission requirements within the specified constraints.

iv. Assess the effects of miniaturisation on the overall performance of the propulsion system in terms of thermal characteristics, performance losses and manufacturing capabilities.

v. Develop, test and characterise a series of decomposition chambers that evaluate the performance of a monopropellant thruster at progressively reducing thrust levels.

vi. Create and solve a series of numerical models that simulate the flow of exhaust gases through a rocket nozzle to assess the possible sources of performance losses as the dimensions of the nozzle are reduced to accommodate the low mass flow rate.

1.4 Novel Aspects Achieved

The research presented has addressed the gap in published data relating to hydrogen peroxide thrusters and contributes to the state of the art in the following ways:

- Successful miniaturisation of a decomposition chamber capable of stably and repeatedly decomposing a flow of hydrogen peroxide at a rate of 0.5 g/s through an uninsulated catalyst bed 6.7 mm in diameter and 5 mm in length.

- Identification of a trend in the performance of a decomposition chamber according to the diameter of the chamber and mass flow rate of propellant.

- Identification of a trend in the development of a boundary layer within a thruster nozzle that suggests a logarithmic rate of growth.

- Identification of a critical point relating to nozzle size, which corresponds to the size of nozzle that is the smallest that may be considered without boundary layer analysis.

One international conference paper has been presented on this work and one paper has been submitted to a refereed journal for publication. A second conference paper is currently in progress. The US Air Force European Office of Aerospace Research and Development funded a research contract to investigate the miniaturisation of a hydrogen peroxide thruster and two reports have been submitted on this work.
1.5 Outline of Chapters

The remaining thesis consists of 6 chapters. Chapter 2 contains the literature review, which summarises the recent literature on the various subjects pertinent to this research. Chapter 3 begins with the selection of a nanosatellite platform that would benefit from the addition of a propulsion system. This is followed by the development of a mission scenario for the platform and the assessment of the capabilities of various types of propulsion system against the mission requirements. The detailed assessment of the factors affecting miniaturisation is the focus of chapter 4, with particular reference to fluid flow and heat transfer. The decomposition chamber is the subject of chapter 5. The development of two different types of catalyst bed is discussed together with testing data. Chapter 6 is concerned with the numerical modelling of the exhaust nozzle. The specification of the model geometries used is discussed together with the results generated. Chapter 7 concludes the thesis with a summary of the work completed and how this may be applied for use onboard a nanosatellite platform. An outline of the future work required to develop a flight ready system propulsion system is also presented.
2 Literature Review

This chapter will provide an overview of the literature relevant to the remainder of the thesis. First the possible applications of a propulsion-enhanced micro-/nanosatellite are evaluated. This is followed by a summary of the different types of propulsion systems together with present developments that may be suitable for use onboard a micro-/nanosatellite. The different green monopropellants that are available will then be discussed followed by an evaluation of the available literature relating to the development of a catalyst bed for a hydrogen peroxide thruster. The specific factors affecting the miniaturisation of a propulsion system are considered with respect to current and recent literature. Finally present research relating to the miniaturisation of a nozzle is summarised.

2.1 Mission Scenarios for Micro-/nanosatellites

The range of mission applications for satellites grows on a daily basis. For a given mission, the satellite used is unique due to the particular constraints or requirements imposed. In general the payload determines the size of satellite required. For example a nanosatellite weighing less than 1 kg would not be suitable for use as a communications satellite as there would be insufficient power available to transmit the signals back to Earth.

The addition of a propulsion system to any platform increases the flexibility of the satellite, but at the expense of mass and power. The need for and size of a propulsion system is determined from the mission requirements. For example the required mission lifetime may necessitate a propulsion system to overcome atmospheric drag forces, particularly in low altitude orbits. The accommodation of a propulsion system onboard a micro/nanosatellite is frequently challenging, as these satellites are usually volume limited to fit inside a particular launch vehicle. The applications of a propulsion-enhanced microsatellite have been investigated in many studies [Deutsch'03, Jackson'98, Ketsdever'00, Saunders'03, Sweeving'99]. The overall conclusion from these studies is that demonstration missions are always necessary to provide the basis for further analyses prior to mission implementation.

A review was conducted of the various proposed and achieved applications for small satellites with onboard propulsion requirements. The literature revealed three key mission areas where micro-/nanosatellites were of particular interest: formation flying missions; science missions and rendezvous and inspection missions.


2.1.1 Formation Flying Missions

Formation flying is a mission concept that is based upon the idea of two or more satellites being placed in to particular orbits such that they maintain their position relative to each other. Simple formations may consist of two satellites in the same orbit, separated by a fixed angle. In contrast complex formations may require several satellites, in different orbits, creating a set pattern relative to a fixed centre point.

At present the motivation for many formation flying mission concepts is to achieve the goal of a distributed satellite. This is where the task of a single expensive satellite is accomplished through use of many smaller satellites. There are two advantages to this approach. The first is in terms of cost. Individually the small satellites are less expensive than a single satellite, both in terms of manufacture and launch. This allows additional redundancy to be included in the formation enabling satellite fatality to be accommodated at minimal cost. The second advantage to a distributed satellite is that it is able to provide a capability that is unique due to the array of satellites involved. Maintenance and realignment of the formation will require a responsive propulsion system that is capable of delivering low magnitude, high accuracy thrusts.

The NASA Space Technology 5 (ST5) mission illustrates the need for a propulsion system onboard a miniaturised formation flying platform. These spacecraft have a total mass of 25 kg and have been designed for a three month long mission [Harris'03]. They will be required to manoeuvre on orbit to create the required spatial pattern and will utilise a cold gas propulsion system to achieve this.

Another mission concept that sought to tackle some of the demonstration aspects of a formation flying mission was PROBA-2.25 [Liddle'04]. This mission was designed around a platform called SNAP-2, which was capable of accommodating several technology demonstration components including a micropulsion system as well as suitable navigation systems. The satellite mass was estimated to be 10 kg with a mission lifetime of approximately 1 year.

2.1.2 Science Missions

There are many different sub-categories that may be made within the heading of science missions, including astronomy, earth observation, deep space exploration and interplanetary probes. The possible applications of an advanced microsatellite have been studied with earth observation and deep space missions the preferred options [Alderson'03]. The platform considered had a mass of 120 kg and a large power budget, therefore an electric propulsion was under consideration, as this offered the longest mission lifetime.
The Laser Interferometer Space Antenna (LISA) mission seeks to directly detect gravitational waves [Gianolio'04]. It aims to achieve this through the measurement of any change in distance between “proof masses” which are located inside the spacecraft. The formation consists of three spacecraft, each with a total mass of approximately 460 kg. They will be orientated in a triangular formation with a separation between them of \(5 \times 10^6\) km. In order to detect the gravity waves, the spacecraft must follow the proof masses precisely, which requires a high accuracy propulsion system. To achieve this an electric propulsion system has been selected.

Within the field of Earth observation the PROBA (PRoject for On-Board Autonomy) mission demonstrated that a satellite could successfully image the planet autonomously [Bermyn'00]. The satellite had a launch mass of approximately 100 kg and carries six technological payloads, but has no propulsion system. The successor to PROBA, PROBA-2 is currently in the final phases of manufacture and is due to be launched in 2006. This platform includes a cold gas propulsion system, which enhances the capabilities of the spacecraft.

Various interplanetary missions are currently in development such as the BepiColombo mission to Mercury [vanCasteren'05]. This mission is set to discover more about the hottest planet in the solar system and answer key questions relating to its composition and evolution. The mission consists of two spacecraft: a planetary orbiter and a magnetospheric orbiter. The planetary orbiter will focus upon the questions relating to the composition of the planet, while the magnetospheric orbiter will analyse magnetospheric particles. This mission provides an example of where a picosatellite lander could provide invaluable additional data. The addition of a lander component was initially considered for this mission, but the additional mass dictated the need for an additional launch.

### 2.1.3 Rendezvous and Inspection Missions

A rendezvous mission requires one satellite to meet with another in orbit. The target satellite may be another artificial satellite or another object such as an asteroid. The payload of the mission satellite will frequently include a range instruments to enable various measurements to be taken in the vicinity of the target satellite. Interest in the composition of near Earth objects has led to the development of several missions, such as Giotto [Reinhard'82], Deep Space 1 [Rayman'00] and Rosetta [Berner'02].

The Near Earth Asteroid Rendezvous (NEAR) Shoemaker mission to Eros demonstrated the wealth of information that can be gathered from this type of mission [Prockter'02]. The spacecraft orbited the asteroid and landed on the surface to make comprehensive scientific measurements of the asteroid. This mission collected scientific information however the same mission approach could be applied to a rendezvous with any object. Following the success of the
NEAR mission, many other missions to near-Earth objects are under consideration and development.

After the launch of a spacecraft if any unforeseen events occur that result in spacecraft damage there is little that can be done. Some events may be impossible to rectify, for example if the main computer crashes as a result of space radiation. Others events may be dealt with if a servicing satellite were available, which could dock with the spacecraft. It may be possible to either repair the problem or provide additional information as to its exact nature. For this to be possible the target satellite needs to be designed to enable the necessary manipulations. A direct rendezvous may not always be possible, dependent upon the damage to the target satellite. Additional information may be provided instead if the servicing satellite is able to image the exterior of the target satellite, thereby detailing damage to a solar panel for example.

The Surrey Nanosatellite Applications Platform (SNAP) was a 6.5 kg spacecraft, which successfully demonstrated that rendezvous with a larger satellite is possible [Gibbon'00]. A butane cold gas propulsion system allowed SNAP to manoeuvre close to the target satellite and image it, although differences in initial orbit altitude prevented completion of the rendezvous.

The concept of a nanosatellite capable of imaging another spacecraft is under further development at the NASA Johnson Space Centre in the form of the Miniature Autonomous Extravehicular Robotic Camera (Mini AERCam) [Fredrickson'04]. This spacecraft weighs approximately 5 kg and is a free-flying spherical package with a diameter of approximately 19 cm. It is designed to image the exterior of the international space station through use of its imaging payload combined with an LED array, thereby reducing the need for inspection by an astronaut. The propulsion system consists of twelve cold gas thrusters, which operate with a xenon propellant providing the spacecraft with six degree of freedom control. This allows the spacecraft to rotate, translate and hold position in addition to point to point manoeuvring. The key difference between this spacecraft and most others is that while the length of an excursion is limited by the propellant mass available, the total propellant is unlimited. This allows an infinite number of excursions to be completed as after each excursion the spacecraft returns to the space station, allowing the propellant to be replenished.

2.1.4 Summary of Mission Scenarios

The literature relating to three different mission areas for which a micro-/nanosatellite is suitable has been summarised. The formation flying missions require a propulsion system to maintain and realign the formation pattern. The science missions require a propulsion system to perform on orbit manoeuvres allowing images or samples to be collected. Finally the rendezvous and inspection missions require a propulsion system to manoeuvre close to the target satellite and
gather the required information. From this literature it is clear that for a micro-nanosatellite to achieve a mission profile based upon one of these areas a suitable propulsion system is required. The different types of propulsion system that are available to a satellite are now reviewed to allow their capabilities to be assessed with respect the constraints of a micro-nanosatellite.

2.2 Types of Propulsion System

The method of thrust generation differentiates different propulsion system. The various categories of propulsion systems are briefly outlined below together with examples of current developments of miniaturised versions in each area. The parameter of specific impulse is used to characterise these systems, which is defined as the change in momentum generated per unit of mass propellant. Overall this parameter is a measure of fuel efficiency and will be discussed further in chapter 4. The characteristic performance values given for these systems are taken from [Sutton'01].

2.2.1 Cold Gas & Resistojet Propulsion

Cold gas thrusters and resistojets are considered to be the two simplest types of propulsion system. In the case of the cold gas thruster, a gas is stored at high pressure in an appropriate tank. It is released through a valve into a chamber, where it is expanded then expelled through a nozzle, creating a thrust. The resistojet works in a similar manner although the propellant is heated to produce a higher magnitude thrust. These systems are favoured for use on small satellites as the complexity is low and the system is reliable. The valve actuation time, dribble volume and nozzle size determine the smallest thrusts that can be produced. The heat up time of the resistojet further restricts the response time of the thruster. In general, the specific impulse of a cold gas thruster is low at approximately 80 s, although this depends on other system parameters such as chamber pressure. The addition of electrical power allows the specific impulse of a resistojet to be slightly higher dependent on the power level used.

There are various developments related to micro cold gas thrusters. The Ängstrom Space Technology Centre (ÄSTC) in Sweden has developed a micro cold gas thruster system [Kohler'02]. Based upon Micro Electro Mechanical Systems (MEMS) technologies, the system is very compact and able to be contained within a small pod. It is designed to be the primary propulsion system on board a nanosatellite, or perform station keeping duties on a larger satellite. It is capable of producing thrusts in the range 0.1 - 10 mN, with a power requirement of 0.2 W. The mass of the microthruster component is approximately 150 g, excluding the control electronics and propellant storage.
Developments in small resistojets include the low-power resistojet at Surrey Satellite Technology Limited (SSTL) [Gibbon'03], which utilises a power of 13.7 W to generate a thrust of 100 mN at a specific impulse of 99 s in a unit weighing 160 g. This system utilises butane as the propellant, however changing this to xenon allows the capabilities of the system to be extended. This is because the density of xenon is higher, allowing more propellant to be stored within the same volume [Coxhill'05].

2.2.2 Chemical Propulsion

Chemical propulsion systems utilise an exothermic chemical reaction to produce hot gases, which are expelled through a nozzle thus creating thrust. The specific impulse associated with chemical systems is significantly higher than for the cold gas systems. This is due to the higher energy of the exhaust products. The following sections discuss the various different types of chemical propulsion.

2.2.2.1 Solid Chemical Propulsion

Solid propulsion systems are the simplest of the chemical systems as no valves or propellant feed systems are required. Once the propellant grain is loaded into the thrust chamber it is ready for use. It may be stored in this condition for many years before use, however once ignited, it is not possible to control or stop the thrust. Typically solid propulsion systems are used for high impulse manoeuvres such as insertion into a geostationary transfer orbit. Recent development in other types of propulsion have reduced the use of solid propulsion overall.

Research into micro solid propulsion systems has led to the development of a "digital propulsion" system [Rossi'01]. The system consists of an array of cells, each approximately 1 mm across, containing solid propellant. The cells are then sandwiched between an igniter panel and a containing diaphragm, which bursts when the propellant is ignited. The simplicity of this system is possible due to the use of a solid propellant but due to the change in the thrust vector caused by the firing of each cell, the applications of this system are limited. In addition the cells need to be located on external surfaces, which may limit space for solar cells.

2.2.2.2 Monopropellant Chemical Propulsion

A monopropellant propulsion system offers a more flexibility than a solid propulsion system, with less complexity than a bi-propellant system. A liquid propellant is fed through a catalyst bed, which causes the propellant to decompose into hot exhaust gases, which are then expanded through a nozzle. Hydrazine is frequently used in these systems as it has good long term storage characteristics and generates a high specific impulse. Monopropellant thrusters are frequently used as part of the attitude control system onboard large satellites.
Miniaturised monopropellant system developments include a MEMS based thruster utilising hydrogen peroxide propellant at the NASA Goddard Space Flight Centre [Hitt'01, Kujawa'03a]. This is designed to produce a thrust in the range 10 - 100 μN and uses a silver catalyst, which is supported on pillars etched into the decomposition chamber. The total mass of the system is 100 g, with a thruster component mass of 5 g.

### 2.2.2.3 Bi-Propellant Chemical Propulsion

Bi-propellant propulsion systems utilise a combination of fuel and oxidiser, which is burnt to create hot exhaust gases, which are then expanded using a nozzle. The presence of two propellants allows the performance of the system, in terms of efficiency and thrust, to be significantly increased although the system complexity is also amplified. The combination of propellants is selected to optimise the molecular mass of the exhaust products and the energy of the reaction produced, with the optimal combination being liquid hydrogen fuel with liquid oxygen oxidiser. Use of these propellants brings associated long-term storage issues, which makes them unsuitable for use onboard a spacecraft.

Research into a micro bi-propellant rocket engine, which produces 15 N of thrust from a system that weighs 15 g, has been underway for many years at MIT [Mehra'00, London'00, London'01a, London'01b]. The design utilises MEMS technologies to create a slender “rocket chip” which contains numerous microscale components, such as pumps as well as cooling channels within the nozzle. It operates using ethanol and oxygen and produces a specific impulse of 150 s.

In addition research into a micro bi-propellant rocket engine is progressing in Europe [Scharlemann'05]. This system has a design thrust of 1 N and total design mass of 190 g (excluding electronics). At present all of the components have been tested individually and the process of integrating them together is the current task. This engine operates using ethanol and hydrogen peroxide and is designed to produce a specific impulse of 230 s.

### 2.2.3 Electric Propulsion

There are various types of electric propulsion, including hall-effect thrusters, colloid thrusters, pulsed plasma thrusters (PPT) and Field-Effect-Electric-Propulsion (FEEP). The general principle employed is that an ionised propellant is accelerated to a high velocity using an electric potential. The ions are then expelled creating a thrust. It is the particular method used to generate the particles that differentiates the types of propulsion. The particles themselves are very small, which means that although a high specific impulse is generated, the thrust level is low. The key disadvantage to electrical propulsion systems is the power requirement, which is high due to the large potential required. Electric propulsion systems are the preferred option for formation flying satellites because they can produce precise small thrusts with the required repeated accuracy.
addition electric propulsion systems are now frequently used for station keeping manoeuvres on board large communications satellites.

The European Space Agency (ESA) mission LISA (Laser Interferometer Space Antenna) requires the position of the satellites involved to be maintained to a very high accuracy. This dictates the use of these electric propulsion systems, onboard large satellites with high power capabilities. These missions have led to the development of systems including the Indium FEEP system by the Austrian Research Centres (ARCS) [Tajmar'04]. This system produces a thrust level in the range 0.1 – 100 μN, with a power requirement of 13 W and a mass of 600 g for the thruster component. The low thrust of these devices leads to the need for long burns to achieve the required delta-V. This results in the need for testing lasting many thousands of hours of operation to ensure reliability, which makes the development of these thrusters very expensive.

Other electric propulsion systems that are under development to meet low thrust requirements include colloid thrusters at Busek [Hruby'01]. The complete system, including sufficient propellant for 3000 hours of operation weighs 2.5 kg at present. The system encompasses 57 individual emitters, which are aligned with an accelerator grid to facilitate the creation of a stable cone of propellant. It produces a thrust range of 20 – 189 μN, with a stability of 0.1 μN and an associated specific impulse of 400 s, but requires a power of 6 W. Other developments are also underway to miniaturise PPTs [Rayburn'00] and ion propulsion [Zeuner'03] systems. The power levels of these systems range from 12.5 W for the PPT to 350 W for the ion thruster.

### 2.2.4 Summary of Types of Propulsion System

Several different types of propulsion system have been introduced. The relative advantages and disadvantages are summarised in Table 2-1 to provide a general overview of the performance in each case. The thrust and specific impulse data given in each case is characteristic of the capabilities of a micropropulsion system, based upon the literature reviewed.

<table>
<thead>
<tr>
<th>Type of Propulsion</th>
<th>Thrust Range</th>
<th>$I_{sp}$</th>
<th>Advantages</th>
<th>Disadvantages</th>
</tr>
</thead>
<tbody>
<tr>
<td>Cold Gas Thrusters</td>
<td>1 – 50 mN</td>
<td>40 – 80 s</td>
<td>Simplicity</td>
<td>Low performance</td>
</tr>
<tr>
<td>Resistojet</td>
<td>50 – 250 mN</td>
<td>75 – 120 s</td>
<td>Increased performance</td>
<td>Low delta-V capability</td>
</tr>
<tr>
<td>Solid Chemical</td>
<td>1 – 5 mN</td>
<td>200 – 300 s</td>
<td>Simplicity, High delta-V capability</td>
<td></td>
</tr>
<tr>
<td>Liquid Monopropellant</td>
<td>1 – 500 mN</td>
<td>100-250 s</td>
<td>Start-stop capability</td>
<td>Propellant tanks &amp; feed system required</td>
</tr>
<tr>
<td>Bipropellant</td>
<td>0.5 – 15 N</td>
<td>200 – 400 s</td>
<td>Increased performance, High delta-V capability</td>
<td>Two propellant tanks &amp; feed systems required</td>
</tr>
<tr>
<td>Colloid</td>
<td>10 – 200 N</td>
<td>400 s</td>
<td>Thrust resolution</td>
<td>High power requirement</td>
</tr>
<tr>
<td>PPT</td>
<td>0.1 – 0.2 mN</td>
<td>500 s</td>
<td>Simplicity</td>
<td>High power requirement</td>
</tr>
<tr>
<td>FEEP</td>
<td>0.1 – 100 μN</td>
<td>1600 – 8000 s</td>
<td>High specific impulse</td>
<td>High power requirement</td>
</tr>
<tr>
<td>Ion</td>
<td>0.1 – 50 mN</td>
<td>2500 – 4000 s</td>
<td>High specific impulse</td>
<td>High power requirement</td>
</tr>
</tbody>
</table>

Table 2-1: Summary of Propulsion Systems
2.3 Green Propellants

In order to ensure the requirements placed on the spacecraft propulsion system are fulfilled efficiently, the propellant should be carefully selected. There are five key categories of properties that require consideration, as listed below:

- **Chemical Properties**  
  - catalyst considerations, combustion products, viscosity, combustion temperature
- **Performance**  
  - specific impulse, density specific impulse, performance under free-fall conditions
- **Storage & Handling Requirements**  
  - materials compatibility, toxicity, storage pressure
- **Associated System Complexity**  
  - technology required, mass, experience, availability
- **Programmatic**  
  - cost, delivery, licences (export, ITAR)

The balance of propellant properties for a launcher propulsion system differs from those for an onboard thruster system. Some characteristics such as handling requirements and catalyst compatibility are equally important in each case. Other properties such as the long-term storage characteristics of a propellant are of differing importance. For a launcher, where the propellant may only be in the tanks for a few hours it is of little concern. In comparison it is of great importance to an onboard thruster system, where it may remain in situ for several years. The advent of microsatellites has led to the need to consider the volume consumed by the propulsion system. The use of a density specific impulse allows the density of a propellant to be factored into its specific impulse. This allows the volume of propellant required to fulfil a given set of mission requirements to be calculated and minimised dependent upon the propellant selected.

Research has been carried out into novel propellants both for launchers and onboard thrusters. This research has mainly focused upon derivatives of hydrocarbon fuels, such as Quadracyclane and Dimethylaminoethylazide (DMAZ). In comparison to the standard RP-1 (kerosene) propellant, these have a higher performance, particularly in terms of specific impulse and density. They are mostly used in bi-propellant thrusters, where high thrust is required. In the case of a monopropellant system, hydrazine ($N_2H_4$) is the standard choice. Once loaded into propellant tanks, hydrazine may be safely stored for several years and has an acceptable specific impulse, together with a short ignition delay. There are various safety hazards associated with this propellant however, as it is highly toxic and carcinogenic. It would be preferable for hydrazine to be replaced with a propellant that exhibits similar or improved performance characteristics, while being safer to handle.
The health of the workforce handling propellants is of increasing concern to regulatory bodies. As a result of this interest in "Green propellants" is increasing. The definition of a Green propellant is not yet completely clear. In some cases, the propellant should be non-toxic, whereas others it should be low toxicity, although a known carcinogen is not considered Green. There are various safety criteria that apply to any propellant. It should be non volatile, have a low sensitivity to shocks, temperature or fire, be not easily detonable, long-term storable and compatible with standard materials. In addition a Green propellant should be environmentally benign throughout the life cycle, which includes the exhaust products. Use of Green propellants is likely to become more prevalent in the near future and an evaluation of various Green propellants that may be suitable for onboard a micro-/nanosatellite is presented below. The values of specific impulse that are quoted are for vacuum operation, at optimum expansion.

2.3.1 Hydrogen Peroxide

Hydrogen peroxide (H$_2$O$_2$) was used as a rocket propellant during the 1950’s and 1960’s and much research into its properties was carried out at that time [Constantine’67, McCormick’67]. Since then however, its use has diminished in favour of higher performance propellants such as hydrazine, which have better storage characteristics. In contrast to hydrazine it is non-carcinogenic and non-toxic.

It is usually stored as a solution with water, and has many applications, such as bleach at ~3% concentration with water and cleaning in the electronics industry at 30-32% concentration with water. Typically concentrations in excess of 70% are used for rocket propulsion, although concentrations up to 98% are available.

The hydrogen peroxide molecule occurs naturally but is unstable and will inevitably decompose into water and oxygen over time. The rate at which this occurs is a function of the solution strength and storage conditions. Over time some decomposition should be expected, therefore some form of venting should be present in a propellant tank to prevent a pressure build up. Materials compatibility is well known and the materials used for a storage container in addition to the thruster should be carefully selected to prevent any unforeseen decomposition [Constantine’67]. A self-venting propellant tank suitable for use onboard a small satellite was developed to a demonstration stage as part of a previous PhD research programme [Coxhill’02]. The tank was sealed for 18 days and no appreciable increase in pressure was observed. The conclusion from this work was that the tank would be suitable for use with hydrogen peroxide for up to a year without excessive pressure build up.

The catalysis of hydrogen peroxide has been studied extensively and the strongest catalysts are known to be silver and manganese oxides. Any organic compound acts as a catalyst, so strict
handling procedures are necessary to prevent contact with skin, as this may result in severe burns. The energy released by the exothermic decomposition is significant (98 kJmol⁻¹) and this can be exploited to facilitate thermal ignition in a bi-propellant system for example with kerosene. The adiabatic decomposition temperature depends on the solution strength, for 90% H₂O₂ this is approximately 1010 K, with a specific impulse of 160 s.

2.3.2 Nitrous Oxide

Nitrous oxide (N₂O) has been researched with respect to rocket propulsion [Zakirov'01a, Zakirov'01b] but it has yet to be widely utilised in a monopropellant thruster due to long-term catalyst instabilities. Decomposition of the propellant can be achieved through a preheated catalyst bed of the commercial catalyst S-405. This consists of beads of ceramic substrate, impregnated with an iridium compound. The high temperature of the reaction (>1000 K) limits the lifetime of the catalyst as the ceramic substrate breaks down. Various novel catalysts have been proposed and researched, although none have yet been entirely successful. The alternative is to employ thermal decomposition. The disadvantage to this is that the temperature range required to achieve this is approximately 860 - 1120 K, which requires the use of exotic materials. There are no strict handling requirements for this propellant and it is compatible with common structural materials. It is widely available at relatively low cost and produces no harmful combustion products. The specific impulse achievable is significantly higher than that for hydrogen peroxide at approximately 200 s, due to the higher temperature of decomposition. In relation to storage it should be noted that nitrous oxide must be stored in gaseous form, hence the density specific impulse is low.

2.3.3 Hydroxyl Ammonium Nitrate (HAN)

HAN is a synthetic propellant that is oxygen rich, in combination with hydrogen and nitrogen (NH₂OH·NO₃) [Sutton'01]. It can be created in both solid and liquid forms and several catalysts have been successfully tested with it. In liquid form it is clear, colourless and odourless, and may be formed into aqueous solutions. The concentration of the solution influences the boiling point, viscosity and decomposition temperature, which varies from 1400 K to 1850 K. It is corrosive and toxic, but it is not known to be carcinogenic. It is expensive to obtain and there are materials compatibility issues with it as well as storage complications. The specific impulse of the propellant varies with water content but it remains high at 200-265 s.
2.3.4 Hydrazinium Nitroformate (HNF)

HNF ($\text{N}_2\text{H}_5\text{C(NO}_2\text{)}_3$) is a very energetic oxidiser that can be used in both bi-propellant and monopropellant rocket thrusters [Fick'01]. It was developed as an alternative to current oxidisers and produces high performance and clean combustion products. It is expensive to obtain as only small quantities are manufactured. It may be used in pure form, or combined with other substances, such as Glycidyl Azide Polymer (GAP), dependent upon the required performance and application. It is a new propellant and few catalysts have been found that will sustain steady decomposition. The temperature achieved upon decomposition is between 1500 K and 1950 K and the predicted specific impulse remains high at approximately 250 s.

2.3.5 Ammoniumdinitramide (ADN)

ADN ($\text{NH}_4\text{N(NO}_2\text{)}_2$) is a synthetic propellant that has been developed to compete with monopropellants such as hydrazine [Anflo'02]. It is highly hygroscopic making it possible to create solutions with water of up to approximately 70% concentration. It is frequently used in combination with other substances such as water and glycerol to maintain a suitable phase and density, although other combinations are possible. It has a significantly lower toxicity than hydrazine, as well as other synthetic propellants including HNF and HAN, and it is considerably more environmentally benign than hydrazine. The decomposition temperature is in the region of 2000 K, which necessitates the use of exotic materials and the specific impulse is approximately 250 s under steady state conditions.

This propellant is currently the focus of research by ECological Advanced Propulsion Systems (ECAPS), which is owned by the Swedish Space Corporation and the Volvo Aerospace Corporation. An experimental 1 N thruster has been developed by ECAPS and is currently in its third design iteration.

2.3.6 Dimethylaminoethylazide (DMAZ)

DMAZ ($\text{C}_4\text{H}_{10}\text{N}_4$) is also known as CINCH (Competitive Impulse Non-Carcinogenic Hypergol). It is one of a family of amine azide propellants, which have been developed as substitutes for hydrazine. They are non-carcinogenic and in particular have lower freezing points than hydrazine [Mellor'04]. DMAZ freezes at approximately 204 K, whereas hydrazine freezes at 275 K. This is of particular interest for missions where the temperature of the propellant is a concern. DMAZ is decomposed using the same commercial catalyst that is used with hydrazine, S-405. The reaction generates temperatures of approximately 1300 K. It is usually used in a gelled form to facilitate storage and other additives may be included to improve the density or specific impulse. It is flammable and materials compatibility is currently under research. At present it is only produced
in small quantities so is expensive to obtain. In addition to use as a monopropellant DMAZ may be used as a bi-propellant and is hypergolic with hydrogen peroxide. Used as a monopropellant its specific impulse remains comparable with the other synthetic propellants at 245 s.

2.3.7 Summary of Green Propellants

The preceding sections have introduced six monopropellants that are considered to be Green. The performance and available knowledge of the propellants varies together with cost and availability. In order to select the propellant most appropriate for use on board a micro-/nanosatellite a critical trade off is carried out in chapter 3, following the derivation of the mission requirements. The propellant selected is hydrogen peroxide and the implications of this on the design of a catalyst bed are now discussed.

2.4 Previous and Present Catalytic Bed Designs for Hydrogen Peroxide

The structure and arrangement of a catalytic bed for the decomposition of hydrogen peroxide has been the focus of research for many years. Over this time the composition of the catalytic bed has typically been based upon silver screens, however various other arrangements have been considered [Davis'60]. This section will review some of the historical data and present developments associated with the design of a catalyst bed for a hydrogen peroxide thruster.

2.4.1 Early Catalyst Development

Hydrogen peroxide has been in use as a rocket propellant for several decades. In many cases use of this propellant diminished as hydrazine became more widely available. Limited use has prevailed particularly for control systems of spacecraft such as the Russian Progress and Soyuz [Iarochenko'01]. Overall the catalyst beds used for these systems are based upon silver gauzes, or silver plated gauzes, however start-up difficulties arise when the temperature of the bed is less than 0 °C. As a result of this other options were developed with better cold start capabilities including potassium permanganate and cobalt oxide catalysts.

The optimal arrangement of silver screen catalyst packs was investigated extensively and numerous documents summarise the conclusions [Davis'60, Willis'60, Runckel'63]. Various factors including optimal compression ratios and pack life were considered and design parameters created. Concepts such as a “loading factor”, which is the mass flow rate of propellant per unit area of catalyst, were developed to enable the erosion of the pack to be directly assessed. Different configurations of pack were discussed for use in different environments. For example, if a pack was to operate at low temperature, it was recommended that a silver scroll preceded the silver gauzes. This creates a highly tortuous flow path for the propellant thereby increasing the
time spent in contact with the catalyst. The studies make various recommendations for the design of catalyst packs. The utility of these with respect to the parameters under consideration here will be discussed in more detail in chapter 5.

### 2.4.2 Recent Advances in Catalysts

Historically silver is the preferred catalyst for use in hydrogen peroxide thrusters. One major disadvantage to using silver was discussed above with respect to cold-start capability. In addition the maximum working temperature of silver is comparable to the adiabatic decomposition temperature (1010 K) of hydrogen peroxide observed at high concentrations. This possibility of silver screens fusing is exacerbated in an oxygen rich environment [Davis'60], which leads to interest in alternative catalysts. Use of manganese oxide catalysts has been studied for use with very high concentrations of hydrogen peroxide [Kappenstein'02]. The results indicate a high catalytic activity, but the investigation was conducted using a constant volume and a constant pressure reactor and not a thruster.

The effect of hydrogen peroxide stabilisers on the performance of catalysts is a particular concern. The case of supported catalysts was considered using a two reactors, one operating at constant pressure and the other at constant volume [Pirault-Roy'02]. Several different catalysts supported on alumina were tested including manganese oxides, silver, platinum/tin and iridium. Overall it was found that the introduction of stabilisers caused a decrease in activity, particularly in the case of the manganese oxide catalyst.

The morphology of the catalyst bed is known to have an impact on the performance observed. Development work at Aerojet has shown that a monolithic catalyst bed consisting of a stack of etched silver platelets can create a highly tortuous flow path, while minimising pressure drop [Ponzo'03]. The design enables a high surface area per unit volume to be created together with a stable flow area, which results in stable and repeatable performance. The length of the monolithic bed is reduced to approximately 12.7 mm, in comparison to greater than 50.8 mm for a traditional silver screen catalyst bed operating at the same flow rate. Overall the performance of the bed is significantly improved through use of the monolithic design.

The comparative performance of catalyst beds consisting of compressed silver gauzes, alumina supported silver and alumina supported manganese oxides have also been investigated [Eloirdi'01]. It was found that the supported silver performed well in tests with a batch reactor, but the silver gauzes performed best in the thruster firings. The reason for the difference in performance is thought to be a function of the pressure drop through the catalyst, although this remains unconfirmed.
Chapter 2: Literature Review

2.4.3 Summary of Catalyst Bed Requirements

The literature has revealed that silver remains the preferred catalyst material for the decomposition of hydrogen peroxide. The optimum morphology for a catalyst bed is dependent upon the operating conditions of the thruster and requires investigation for the particular requirements. The development of a catalyst bed for a monopropellant propulsion system will be discussed in further detail in chapter 5. Here the discussion of literature will continue with the consideration of fluid flow through small channels.

2.5 Fluid Flow at Microscale

The miniaturisation of a propulsion system will require miniaturisation of the feed system to supply the propellant to the thruster. Consideration of key aspects of fluid flow at microscale is therefore necessary to ensure the system remains practicable.

Fluid flow through small channels has been of interest for many years. As the capabilities of manufacturing techniques extend further the modifications in the flow that are observed as a result of the small dimensions become of increased interest. Research has been conducted both experimentally and numerically with combinations of the two allowing verification of the results generated. Different aspects of fluid flow are considered in the following sections.

2.5.1 Reynolds Number Effects

The Reynolds number, $Re$, is a non-dimensional number that is used to characterise fluid flow. Three types of flow are defined: laminar, turbulent and transitional. A low Reynolds number indicates a laminar flow, while a high Reynolds number indicates a turbulent flow. The critical Reynolds number characterises the point where a flow transitions from laminar to turbulent. The definition of the Reynolds number will be considered in chapter 4. Many research efforts into flows through micro channels have considered the critical Reynolds number. Mala and Li found that for channels with widths ranging from 50 - 250 μm the onset of transition was comparable with that for conventional theory at a Reynolds number of approximately 2200 [Mala’99]. In contrast Hsieh et al found that for a channel with a width of 200 μm the onset of transition was at a Reynolds number of approximately 240 [Hsieh’04]. Water was used as the working fluid in both cases to provide an incompressible medium. The difference in the results illustrates the level of uncertainty that surrounds the experimental research.
2.5.2 Flow Regimes

In addition to the Reynolds number other non-dimensional numbers exist to describe the behaviour of a fluid. The Knudsen number, $Kn$, is used to describe four different flow regimes, continuum, slip, transitional and free-molecular. These are discussed in more detail in chapter 4.

The behaviour of fluids within these various flow regimes has been considered both numerically and experimentally. Arkilic et al considered a gaseous flow through a channel with a width of 52.25 μm, which placed the flow into the transitional flow regime [Arkilic'97]. The experimental results corresponded well with theoretical predictions that included a slip boundary condition. In addition to slip characteristics, rarefaction effects were observed and validated numerically. Arya et al further developed this research by considering the tangential momentum accommodation coefficient, which governs the degree of slip observed at the surface [Arya'03].

Other studies have considered the use of the Direct Simulation Monte Carlo (DSMC) method, which allows computation of the fluid into the free molecular regime, allowing evaluation of a flow following exit from a nozzle [Alexeenko'03].

Overall the literature reveals that special consideration is required for channels when the characteristic dimension reduces below approximately 250 μm, although this is highly dependent upon the fluid and flow conditions under consideration.

2.5.3 Boundary Layer Effects

A fluid flowing over a solid surface will result in the generation of a boundary layer. The rate at which this grows is a function of various properties of the flow, including the Reynolds number and the fluid viscosity. The development of a boundary layer will be considered in more detail in chapter 4.

A boundary layer is of particular interest in a small channel and the flow of both liquids and gases have been considered experimentally in micro channels [Pfahler'91]. For liquids it was found that as the size of the channel decreased the results showed deviations from the theoretical predictions. For a given pressure drop an enhanced flow rate was observed, indicating an artificial narrowing of the channels. This demonstrates that effects of a boundary layer within a small channel are likely to be magnified. This finding is further supported by an investigation into the pressure gradient observed in a micro channel [Mala'99]. It was observed that a significant increase in the pressure gradient was required to generate a given flow rate as the diameter of the tube reduced to 50 μm. In addition the material used to fabricate the tube was investigated. It was found that to generate the same flow rate a steel tube required a lower pressure gradient than a fused silica tube.
2.5.4 Effects of Surface Forces

Surface forces are a group of forces, including surface tension and surface wetting, which hold a fluid in a particular shape or location. In particular, surface tension is of interest in microchannels as it may prevent the filling of a channel. The literature revealed that flow hindrance and blockage exist in the smallest channels. These were successfully predicted by inclusion of the effect of negative surface tension in the numerical models used [Kim'02]. The research considered channels 40 μm thick, ranging in width from 100 μm to 500 μm and observed flow blockage in the smallest channels.

If a residual charge remains in the structural material of the channel this can cause an interfacial Electric Double Layer (EDL) to develop. Dependent upon the composition of the fluid this can lead to pockets of stationary fluid within the channel, due to electrostatic attraction. The result of this is to cause the apparent viscosity of the fluid within this layer to increase to be several times higher than the bulk viscosity of the fluid [Mala'97].

2.5.5 Surface Roughness Effects

The roughness of the channel will affect the efficiency with which the fluid passes through it. This may be characterised through use of a friction factor, a concept that was developed in the early 1900's by Nikuradse in relation to flows through pipes. The work considered the pressure drop of a fluid flowing through pipes of various roughnesses. The conclusion from this work was that if the internal relative surface roughness is less than 5%, the roughness effect on laminar flow characteristics could be ignored. The relative surface roughness is the ratio of the average roughness height to the diameter of the tube.

Numerous research efforts have investigated the effect of the friction factor on the pressure drop observed in flows through microchannels [Brutin'03, Ma'97, Mala'99, Pfahler'91, Peiyi'83]. The conclusions from these investigations have varied significantly. The work by Peiyi and Little considered the friction factor in a microchannel of depth 130-200 μm [Peiyi'83]. They concluded that the friction factor was influenced by the surface roughness of the channel even in laminar flow states. These conclusions were supported by the work of Mala and Li, which found that the pressure drop observed along the length of the tube and corresponding friction factor was greater than predicted by theory [Mala'99]. In comparison Pfahler et al found that tests with channels of depth 0.5-50 μm generated results in good general agreement with theory for the larger cases, but deviations were observed for the smaller cases [Pfahler'91]. These tests utilised different fluids from those used in the previous research, which may provide an explanation for the difference in the behaviour observed. Overall the conclusion from these papers is that frictional losses in
excess of those predicted by theory may be expected in flows through microchannels. The effect of surface roughness will be discussed in more detail in chapter 4.

The effect of channel roughness on both fluid flow and heat transfer in small tubes was investigated experimentally [Kandlikar'01]. Tubes with diameters of 1.062 mm and 0.622 mm were used and it was found that the presence of surface roughness increased heat transfer and pressure drop characteristics.

### 2.5.6 Thermal Effects

The thermal characteristics of a fluid flow are modified as the dimension of the channel changes due to the non-linear relation between the rate at which the volume and surface area reduce. The effect of this on heat transfer has been investigated extensively. A review of the work completed found that the heat transfer effects related to the surface area dominated those related to the volume [Guo'03].

All of the surface forces considered here will cause increased losses as the dimensions of a thruster reduce. Recognition of these effects may allow their presence to be predicted and accounted for resulting in a better understanding of the flow environment.

### 2.5.7 Summary of Microscale Fluid Flow Effects

The preceding sections have considered the effect of various flow phenomena on flows in microscale channels. The literature has revealed that greater consideration should be given to these effects at microscale. Overall the results are conflicting and it remains impossible to determine a particular channel dimension where these effects become critical. The presence of these losses leads to concerns that the performance of a micro nozzle will be severely reduced. Research considering the performance of a micro nozzle is now reviewed.

### 2.6 Previous and Present Micro Nozzle Research

The nozzle of a rocket engine allows the hot gases produced by the combustion reaction to be accelerated. It consists of a convergent section, which accelerates subsonic flow followed by a divergent section, which accelerates supersonic flow. The point where the change in geometry from convergent to divergent occurs is called the throat. If the nozzle is designed and operating correctly it is here that the flow will become supersonic. The behaviour of small-scale nozzles has been of interest for many years with both experimental and numerical studies conducted. There are two key areas that appear to be the focus of most studies: the particular geometry of the nozzle and the Reynolds number of the flow.
2.6.1 Micro Nozzle Geometry

The geometry of a nozzle is described by a series of parameters, including the diameter of the throat and exit as well as shape of the nozzle itself. The method of designing a nozzle will be discussed in more detail in chapter 6, with reference to the theory required.

The effect of the geometry on the flow behaviour has been investigated by many parties, with different nozzles of interest in each case [Bayt'97, Bayt'98, Choudhuri'00, Choudhuri'01a, Choudhuri'01b, Hussaini'96, Ivanov'99, Kim'94, Kujawa'03a, Kujawa'03b, Wang'04, Yang'04]. Overall the throat width of the nozzles considered in these papers is in the range 20 - 90 μm or 300 - 380 μm. The type of geometry used varies and in many cases is the subject of the study, although in general it is found to contribute little to the performance overall. Hussaini concluded that the contoured nozzle produced the optimal results, if the exit curvature was allowed to be negative [Hussaini'96]. In this study the presence of a significant boundary layer was noted to cause considerable modifications to the flow profiles observed. This is supported by work by Kim, who numerically investigated resistojet nozzles with different divergence angles, but a fixed exit area [Kim'94]. This resulted in nozzles of different lengths and it was found that the shortest produced the highest specific impulse but in addition generated the highest divergence losses, due to the high expansion angle. The profile of the throat itself was investigated, using a sharp throat, with a width of 27 μm [Ivanov'99]. The results indicate that the boundary layer initiated at the throat, although this is difficult to identify. A specific impulse efficiency was defined as the ratio of the actual specific impulse to that predicted and a value of approximately 85% was calculated together with a thrust of 5 - 13 mN. This work also evaluated the accuracy of different numerical techniques in terms of the vacuum specific impulse predicted. It was concluded that the Navier-Stokes equations over-predicted the vacuum specific impulse, while the DSMC method provided a more realistic estimate. This is because the Navier-Stokes method is unable to compute the flow exterior to the nozzle as it is no longer in the continuum. The DSMC method is able to compute this region and therefore is able to correctly simulate the flow within the nozzle when operating in a vacuum environment.

Another consideration in terms of nozzle geometry is whether it is 2D or 3D, i.e. is the exit profile of the nozzle square or circular. The interest in MEMS based devices has led to the development of MEMS nozzles, which are 2D and inherently flat. The modification of the nozzle profile causes significant increases in losses, indicating that a 2D simulation is not sufficient as it does not account for the out of plane losses. In the case of a 2D MEMS nozzle these may be considerable due to a boundary layer building up on the top and bottom surfaces. These effects have been observed to cause the exit area to reduce by up to 47% dependent upon the Reynolds number [Bayt'98]. The studies completed by Kujawa et al confirm this, where the flow was
found to be under expanded [Kujawa'03a, Kujawa'03b]. This indicates the presence of a thick boundary layer and the need for a higher divergence angle nozzle. In addition these studies considered the effect of temperature on the flow and found that thermal losses significantly impacted performance. These findings are further supported by two other studies, which consider the flow within a flat micronozzle [Alexeenko'00, Alexeenko'03]. These works focus upon numerical simulations, which are compared with previous findings based upon an axi-symmetric conical nozzle. It was again found that the thrust as well as the specific impulse was significantly lower for the flat nozzle in comparison to the conical nozzle.

From these studies it becomes clear that the performance of the nozzle is severely degraded by viscous losses. The utility of the nozzle itself then becomes questionable and the use of an orifice as a compromise between efficiency and complexity becomes of interest. The performance of an orifice in comparison to a de Laval nozzle was considered for Reynolds numbers of 1300 and 130 [Jamison'02a, Jamison'02b]. The throat diameter was 1 mm and experiments utilised cold flows of various gases to produce thrusts in the range 5 - 500 μN. The results indicated that viscous effects were present throughout the operational test range, although in the case of the orifice the losses were much reduced. This work was extended to investigate the effect of the orifice thickness both experimentally and numerically [Lilly'04]. This study revealed that a thicker orifice produced higher propulsive efficiencies, although the smoothness of the edge of the orifice did not appear to affect the results. Another later study investigated nozzle efficiency at low Reynolds numbers and compared results from numerical simulations with experimental data [Ketsdever'05]. This study focused upon flows in the transitional regime and investigated the effect of geometry and its influence on the resultant specific impulse. The propulsive efficiency of an orifice was compared with that of various de Laval nozzles, where the half-angle of the diverging section was varied from 20° to 40° in 5° steps. Overall it was found that while the nozzles incurred significant viscous losses, a 2% increase in specific impulse was observed for each successive increase in divergence angle. The numerical results showed that viscous effects dominate a flow where the Reynolds number is 60. In addition experimental results indicated that the thrust generated by an orifice was greater than that generated by a nozzle for a cold flow of helium or nitrogen with a Reynolds number less than 100. In terms of specific impulse however the nozzle still performed better than the orifice throughout the range investigated.

### 2.6.2 Low Reynolds Number Flows

The Reynolds number of a flow through a small nozzle will be inherently low due to the dimensions. The effect of this on performance is of interest for both cold and hot flows. Experimental investigations have revealed that for a cold flow with a Reynolds number of less
than 1000 the boundary layer may fill the nozzle entirely [Grisnik'87]. This is to be expected due to the severe viscous effects observed in the other papers that focused upon the geometry.

From these papers it can be concluded that the geometry of the nozzle, in particular the divergence angle has a significant impact on the performance of the nozzle. Viscous effects should be expected to perturb the flow predicted by isentropic theory and a 3D simulation is necessary to account for out of plane effects. In addition it can be seen that these papers account for a small range of nozzle sizes, therefore it is difficult to evaluate theoretically how the viscous effects develop with reducing dimensions.

### 2.6.3 Summary of Micro Nozzle Research

The literature has revealed that the development of a boundary layer within a micro nozzle is a significant consideration. The range of nozzle sizes considered by these papers is limited to those of particular interest at that time. As a result there is no correlation of the development of a boundary layer with respect to the nozzle size as the dimensions reduce. The flow regime of the fluid is found to be a key consideration as not all numerical methods will predict the flow behaviour satisfactorily if the continuum regime is not maintained.

### 2.7 Conclusions from the Literature Review

This chapter has reviewed the available literature that is relevant to this research. The range of missions that a micro-/nanosatellite may be capable of was reviewed, with respect to three mission areas: formation flying; science and inspection and rendezvous. This review will be extended in chapter 3 with the selection of a satellite platform and the derivation of a possible mission scenario. The different types of propulsion system that are available together with typical performance parameters were summarised in Table 2-1. Examples of current developments that may be appropriate for use onboard a micro-/nanosatellite were reviewed. The suitability of these systems will be discussed further following the development of the mission requirements in chapter 3. In addition various green propellants were introduced and discussed with respect to performance, availability and heritage. Following the selection of a propulsion system, a critical trade-off of these propellants will be summarised to select a suitable propellant.

Research into catalyst beds for the decomposition of hydrogen peroxide was discussed with reference to conventional designs and present research. These arrangements will be considered in more detail in chapter 5 with reference to the miniaturisation of a decomposition chamber. Finally the flow of a fluid through microchannels and nozzles was considered with particular reference to the development of a boundary layer. This will be considered in more detail in chapter 6 with reference to the successive miniaturisation of an exhaust nozzle.
3 Mission Analysis & Propulsion System Selection

Micro-/nanosatellites are increasingly being called upon to perform more and more complicated and demanding missions. In order to facilitate the successful completion of these ambitious missions, a propulsion system that can produce low thrusts with repeatable high accuracy is required. This chapter will introduce a nanosatellite development that would provide a suitable platform for a micropropulsion system. The addition of a propulsion system would greatly enhance the capabilities of the satellite, hence the development of a suitable mission is discussed and the associated delta-V requirements are calculated. The capabilities of different propulsion systems are then evaluated with respect to these requirements, based upon the constraints that the nanosatellite platform imposes. Finally, the performance of some current micropropulsion system developments is considered with reference to these requirements to determine whether any of these would be appropriate.

3.1 Nanosatellites and Propulsion Requirements

An artificial satellite may be classified in various ways. Possible classifications include the satellite application, type of orbit, size, or cost. Classification through mass is useful as it enables a direct correlation between the predicted size of the spacecraft and the cost of the launcher required to transfer it into orbit. Table 3-1 lists three different categories of satellite according to their wet mass, which includes propellant mass, power, and characteristic overall dimension [Mueller'00]. The potential capabilities of a satellite of mass in the range 1 to 15 kg is of interest in this discussion.

<table>
<thead>
<tr>
<th>Class of Satellite</th>
<th>Complete Mass (kg)</th>
<th>Spacecraft Power (W)</th>
<th>Spacecraft Dimension (m)</th>
</tr>
</thead>
<tbody>
<tr>
<td>Microsatellite</td>
<td>10-100</td>
<td>10-100</td>
<td>0.3-2</td>
</tr>
<tr>
<td>Nanosatellite</td>
<td>1-10</td>
<td>1-10</td>
<td>0.1-0.3</td>
</tr>
<tr>
<td>Picosatellite</td>
<td>&lt; 1</td>
<td>&lt; 1</td>
<td>&lt; 0.1</td>
</tr>
</tbody>
</table>

Table 3-1: Classification of Satellites

The addition of a propulsion system to a micro-/nanosatellite would extend both the range and lifetime of missions possible with the platform. The mass, power, and volume constraints of a micro-/nanosatellite dictate the use of a micropropulsion system. The definition of a micropropulsion system is not always clear, as the classifying parameter is not always specified. If the thrust magnitude is the defining criterion then an electrical propulsion system, such as the
ion thruster developed by EADS is suitable [Zeuner'03]. This system produces thrust levels of 0.3 – 41 mN, but the mass is 1.8 kg with an associated power requirement of 350 W. This type of thruster is suitable for use onboard a large spacecraft with very fine pointing requirements, such as the Laser Interferometer Space Antenna (LISA) mission [Gianolio'04]. It is unsuitable for use onboard a micro-/nanosatellite where the propulsion system mass budget is typically 1-10 kg, and power budget 10-100 W. In addition to producing low thrust, a micropropulsion system for micro-/nanosatellites should have proportionally low mass and power characteristics, thereby fitting within the constraints imposed by the micro-/nanosatellite platform.

3.1.1 Platform Requirements

In order to ensure the propulsion system is designed within realistic physical constraints a suitable platform is required. The chosen platform is PalmSat, a concept currently under development at the Surrey Space Centre, illustrated in Figure 3-1. The addition of a propulsion system to PalmSat would greatly increase the range of missions that it could fulfil. In particular, it would be able to maintain its orbit thereby significantly extending its mission lifetime. Other manoeuvres would also be possible such as rendezvous with other satellites, although initial technology demonstration would be necessary first.

![Figure 3-1: The PalmSat Concept](image_url)

The shape of PalmSat is a regular hexagonal prism, measuring 100 mm across and 60 mm in height. The total mass of the satellite is 1 kg with an associated average power of 3 W. Based upon the assumption that the propulsion system may account for 10% of the total mass and 25%
Chapter 3: Mission Analysis & Propulsion System Selection

of the power, this places the following constraints onto the propulsion system design [Underwood '04]:

- **Propulsion system mass:** < 100 g
- **Propulsion system power requirement:** < 1 W
- **Propulsion system dimensions:** 100 mm diameter, 60 mm height

### 3.1.2 Satellite Propulsion

Throughout the course of space history thousands of spacecraft have been launched. Examples of mission that have had onboard propulsion include interplanetary missions such as the Viking and Cassini-Huygens missions, geostationary communications satellites, such as Intelsat and low earth orbiting (LEO) science missions, such as Envisat and the UoSat series of satellites.

The variety of possible mission specifications inevitably dictates different propulsion subsystem requirements. For spacecraft requiring a propulsion system, its successful operation is often critical to the mission success. The need to include a propulsion system is derived from the mission requirements. Two main areas of requirements relating to the need for a propulsion subsystem can be identified.

- **On Orbit Maintenance** — including active station keeping and attitude control
- **On Orbit Manoeuvring** — including orbit raising, plane changes and de-orbiting

A particular point to note from these requirements is the need for de-orbiting. As a result of the increased number of satellites launched in recent years legislation will soon come into effect stating that all spacecraft should be capable of de-orbiting at the end of life. If the spacecraft is in LEO the atmospheric drag forces may be sufficient to slowly reduce the orbital velocity, causing re-entry to the Earth’s atmosphere. If the spacecraft is in a higher altitude orbit an impulse will be required to initiate the return of the spacecraft to Earth. This impulse will be provided by a propulsion system, thereby leading to an increased number of satellites requiring a propulsion system.

### 3.2 Mission Profile

Various different types of mission scenario were considered in chapter 2 and three key areas were identified: formation flying; science and rendezvous and inspection. A detailed assessment of each case was carried out, with PalmSat as the platform. A simple inspection mission was selected as a candidate mission, requiring a micropropulsion system. It was assumed that a PalmSat would be placed into the same orbit as a constellation of six satellites, with a mission lifetime of one year. The mission is designed to provide remote inspection of the constellation.
saturates to inspect for damage. This would allow a health-check to be carried out remotely on
the constellation and in addition it would demonstrate some of the fundamental aspects of a
formation flying mission. Some of the complications associated with operating a satellite within a
formation together with key manoeuvres that are required to complete such a mission are now
discussed. Simple calculations are also performed to provide an estimation of the delta-V
required to achieve this mission. The numbers quoted have been calculated using simple models
allowing for only major perturbation effects, which will be discussed later.

3.2.1 Station-Keeping

Once a satellite has been placed into its desired orbit, it is usually necessary for it to remain
approximately in that orbit. Left unattended, various perturbing forces will act upon the
spacecraft altering the orbital elements, eventually causing the spacecraft to re-enter the Earth’s
atmosphere. The magnitude of the perturbing forces can be predicted from the location of the
satellite, therefore the impulse required to maintain position can be calculated. There are several
forces that cause modifications in the orbit of a spacecraft. Magnetic field disturbances, solar
radiation pressure, atmospheric drag and the effects of the Earth’s aspherical geopotential all
cause perturbations in the position of a spacecraft. The magnitude of influence of each of these
effects on the spacecraft varies. For a nanosatellite in LEO, the two main perturbations requiring
consideration will be effects of the Earth’s aspherical geopotential and atmospheric drag. The
effect of various perturbing forces will now be evaluated using equations from [Larson’92].

3.2.1.1 Atmospheric Drag Compensation

To evaluate the magnitude of this perturbation on the satellite, the first consideration is the
ballistic coefficient, \( B \), given by Equation 3-1, where \( c_d \) is the drag coefficient, \( A \) is the frontal
area and \( m \), the mass of the satellite. The drag coefficient is dependent upon the configuration of
the spacecraft and typically has a value of between 2 and 4. As a small satellite is under
consideration, a value of 2 is used for this analysis. The satellite has a frontal area of 0.024 m\(^2\),
based upon the dimensions stated previously, combined with an additional height of 60 mm for
the propulsion system, and a mass of 1 kg. These combine to give a ballistic coefficient for
PalmSat of 20.83 kgm\(^{-2}\).

\[
B = \frac{m}{c_d A}
\]

Equation 3-1: Ballistic Coefficient
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The rate at which the orbital altitude of the satellite will degrade can then be calculated using Equation 3-2, where $\rho$ is the atmospheric density at the orbital altitude, in kgm$^{-3}$ and $a$, the semi-major axis of the orbit (the orbit radius in the case of a circular orbit), in m.

$$\Delta a = -2\pi \frac{\rho a^2}{B}$$

Equation 3-2: Rate of Altitude Degradation per Orbit

Assuming the satellite is placed into a circular orbit at an altitude of 600 km, where the atmospheric density is $9.89 \times 10^{-14}$ kgm$^{-3}$, a loss of altitude of 1.45 m per orbit will result, which corresponds to 21.63 m per day. The delta-V required to maintain this orbit can be calculated from Equation 3-3, where $V$ is the orbital velocity, in ms$^{-1}$. Based upon the above figures for the orbit specification, this necessitates a delta-V of $7.87 \times 10^{-4}$ ms$^{-1}$ per orbit, or 0.082 ms$^{-1}$ per month. This is an average value that will vary with seasons and the solar cycle, which cause the structure of the atmosphere to change slightly.

$$\Delta V = \pi \frac{\rho a V}{B}$$

Equation 3-3: Delta-V Required to Maintain Orbit per Orbit

3.2.1.2 Earth’s Aspherical Geopotential Effects

The position of a satellite in orbit about the Earth is defined by six orbital elements: semi-major axis, $a$, eccentricity, $e$, inclination, $i$, Right Ascension of the Ascending Node, $\Omega$, argument of perigee, $\omega$ and true anomaly, $v$. The other perturbation that is of significance to PalmSat in LEO is that of the Earth’s aspherical geopotential, in particular that of the first zonal harmonic, $J_2$.

This causes secular perturbations in two of the six elements that specify the orbit of a satellite. The Right Ascension of the Ascending Node is caused to regress and the argument of perigee is caused to precess, resulting in the orbit plane precessing relative to an inertial coordinate frame.

The magnitude of this effect is dependent upon a number of factors including the orbital inclination, eccentricity and semi-major axis. The magnitude of the effects will be the same regardless of the size of the satellite, therefore these effects are usually predicted and monitored, but not corrected for.

3.2.1.3 Relative Perturbation Effects

When considering the position of one satellite relative to another the prediction of all these perturbation effects become more complex. For example, the effect of atmospheric drag may not affect each satellite equally. An average drag force is therefore predicted and corrected for periodically. GPS data can then be used to determine the actual on orbit position of the satellite and any necessary corrections can be carried out retrospectively.
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The magnitude of the relative effects of other perturbations on adjacent satellites will be dependent on the magnitude of separation between them. Assuming that any separation between the satellites is small then if altitude separations exist first order Keplerian effects will dominate the perturbations due to differences in orbital velocity. In contrast if a difference in the orbital inclination is present then secular effects due to the Earth’s aspherical geopotential will become evident. For the purposes of this analysis it is assumed that the separation between adjacent satellites is small and any drift in position will be corrected for retrospectively.

3.2.2 Attitude Control

The attitude of a satellite is affected by external forces in addition to operations on board the spacecraft itself. The level of impulse required to carry out attitude control manoeuvres is generally low, significantly lower than orbit changing manoeuvres. For a nanosatellite, with a mass of 1 kg, impulses of the order of μNs can be expected. The attitude may be controlled using one of a number of systems, which may be based upon thrusters, magnetorquers and momentum wheels or a gravity gradient boom. The magnitude of torque available from a thruster based attitude control system is specified at the design stage. The magnitude of the impulse is then determined by the firing time. For very fine pointing requirements the size of torques required by the satellite may be too small for a thruster system. In this case a system based upon momentum wheels or control moment gyros may be used. These systems transfer momentum to effect a change in attitude but still require a means of generating the momentum. This is usually achieved using thrusters operated in short bursts to generate or dump the required momentum.

Dependent upon the mission scenario and the associated pointing requirements, the delta-V requirements placed on the attitude control system vary. Frequently a window is specified for satellite attitude, with a correction taking place when the limit of this window is reached. The minimum impulse bit capability of a propulsion system will dictate whether it is capable of meeting the spacecraft attitude control requirements. At this stage it is assumed that the propulsion system under consideration will be capable of a minimum impulse bit that will be appropriate. In order to generate requirements for the attitude control system the time taken to recover the initial orientation will therefore be considered. It shall be assumed that a satellite must maintain pointing to within ±5° of its initial orientation. When this limit is reached the initial orientation should be recovered and the resultant drift cancelled within 60 s. The frequency of the attitude control manoeuvre will be dependent upon the external forces incident upon the spacecraft.

In order to calculate the delta-V required per manoeuvre, first the inertia of the spacecraft should be considered. The dimensions of PalmSat create a structure that is approximately cylindrical, where the length is comparable with the diameter. In addition PalmSat may be considered to be a
homogeneous mass, therefore the inertia of the spacecraft may be considered centrally without the need for considering the effect of the three principal axes. The inertia of the spacecraft, \( I \), may be calculated using Equation 3-4, where \( m \) is the spacecraft mass in kg and \( r \) is the radius of the spacecraft in m.

\[
I = mr^2
\]

**Equation 3-4: Definition of Inertia**

Using this equation, assuming that the mass is 1 kg and the radius is 80 mm, it is possible to calculate that the inertia of PalmSat is approximately \( 6.4 \times 10^4 \) kgm\(^2\). The manoeuvre itself is now considered. The requirements selected state that the initial position of the satellite should be recovered within 60 s.

Assuming the use of thrusters to complete this manoeuvre, one of two approaches may be used. Either the spacecraft is under constant torque, accelerating for the first 30 s then decelerating for the second 30 s, or two short impulsive thrusts are used in combination with a coast phase. The first method requires very small accurate thrust levels in order to create the smooth thrust profile required. The second method allows the use of a slightly higher thrust level, to create the start and stop impulses. The impulse style approach is considered and it is assumed that the manoeuvre profile will consist of a 5 s accelerating burn, followed by a 50 s coast and a 5 s decelerating burn. The angular velocity, \( \dot{\theta} \), of the spacecraft during the coast phase should be \( 1.745 \times 10^3 \) rads\(^{-1}\). The angular acceleration, \( \ddot{\theta} \), required to achieve the necessary angular velocity within 5 s is then \( 3.491 \times 10^4 \) rads\(^2\).

The thrust required to achieve this angular acceleration is calculated using Equation 3-5, where \( I \) is the principal moment of inertia of the spacecraft in kgm\(^2\) and \( L \) is the moment arm of the thruster in m.

\[
F = \frac{I \ddot{\theta}}{L}
\]

**Equation 3-5: Thrust Required to Initiate Spacecraft Rotation**

### 3.2.3 On-Orbit Manoeuvres

There are various manoeuvres that a satellite may be required to perform whilst in orbit. One in particular is a change in altitude, which may be approached in a variety of ways. The most
propellant efficient method of altitude change is to use a Hohmann transfer. This involves placing the satellite into a transfer ellipse, where the two ends are tangential to the two orbital altitudes, as illustrated in Figure 3-2a. The delta-V required to achieve this manoeuvre is given in Equation 3-6, where \( r_1 \) is the initial orbit altitude, \( r_2 \) the final orbit altitude and \( a_i \) is the semi-major axis of the transfer orbit. Using a Hohmann transfer a total delta-V of 26.9 ms\(^{-1}\) is required to increase the orbital altitude of a satellite from 600 km to 650 km.

\[
\Delta V = \sqrt{\mu} \left[ \left( \frac{2}{r_1} - \frac{1}{a_i} \right) \frac{1}{2} - \left( \frac{1}{r_1} \right) \frac{1}{2} + \left( \frac{2}{r_2} - \frac{1}{a_i} \right) \frac{1}{2} - \left( \frac{1}{r_2} \right) \frac{1}{2} \right]
\]

Equation 3-6: Delta-V Required for a Hohmann Transfer to Change Orbital Altitude

An alternative approach is to use a one-tangent-burn, which enables a faster transfer between the two orbit altitudes. This manoeuvre also uses a transfer orbit, but it is arranged such that it is tangential to the initial orbit and intersects the final orbit, as shown in Figure 3-2b. The angular difference between the flight path of the transfer ellipse and the final orbit is then corrected with the second burn. An infinite number of possible transfer orbits exist and the one used is determined by specifying either the size of the transfer orbit, the angular change of the transfer or the time of flight required. The larger the transfer ellipse used, the greater the angular difference and the shorter the time of flight.

Finally a spiral transfer may be used. This involves a constant thrust throughout the manoeuvre as the satellite slowly approaches the target altitude. This type of approach is of particular interest for missions where a large altitude change is required, but time is less of a constraint allowing the delta-V requirement to be minimised. For an altitude change of 50 km this approach will not significantly reduce the delta-V required.
In the case of PalmSat it is expected that the launch vehicle would place the satellite into the correct orbit thereby minimising the need for any significant orbit transfer manoeuvres. In order to transit around the orbit small manoeuvres would be necessary and these are now considered.

### 3.2.4 Changing Position

The chosen mission requires PalmSat to move around a constellation of satellites. It is assumed that there are six, evenly spaced, satellites in the constellation and the angle separating the satellites is fixed at 60°. The movement of PalmSat around the constellation can be considered to be a formation flying case. The simplest formation features two satellites travelling around the same orbit, with a small phase separation, ψ. This is illustrated by Figure 3-3, where satellite 1 may be considered to be the PalmSat inspector and satellite 2 the next constellation satellite to undergo inspection. In this case the phase angle will be the same as the separation angle between adjacent satellites and therefore fixed at 60°.

![Figure 3-3: Phase Angle Separation](image)

A small change in the orbital altitude of the inspector satellite will cause it to drift relative to the others. Returning the inspector to the same altitude as the rest of the constellation will then cancel this drift. Figure 3-4 illustrates this in terms of the relative motion, where the black satellites are the constellation satellites in a fixed pattern, and the red satellite is the inspector satellite.

The motion may be either forwards or backwards relative to the formation, dependent upon whether an accelerating or decelerating force is applied to the inspector. For a given altitude change the delta-V required is the same whether the orbit altitude is increasing or decreasing. At a lower altitude the atmospheric drag effects are greater and the delta-V per km required to raise the orbit is greater than that required to reduce the altitude [Larson'92]. It follows that raising the
orbit of the inspector, causing it to move backwards around the constellation, is the most efficient approach.

The speed with which this relative motion occurs is dependent upon the altitude difference between the satellites. The larger the altitude difference, the faster the manoeuvre is completed due to the greater difference in relative velocity. The delta-V required to effect this change however is also larger, hence a trade-off is necessary to optimise the time taken for the manoeuvre and the delta-V required.

![Figure 3-4: Relative Motion of a Phase Shift Manoeuvre](image)

First let it be assumed that the transfer manoeuvre occurs within the space of one orbit. The time period of this transfer orbit is then the same as the time taken for the next constellation satellite to arrive where the inspector had been initially, i.e. for the constellation satellite to travel through an angle of \((2\pi - \psi)\). If the transfer manoeuvre takes place over a longer time period then the time period of the orbit is calculated based upon the time taken for the constellation satellite to pass through an angle of \((2\pi - \psi/n)\), where \(n\) is the number of orbits taken for completion of the manoeuvre.

The time period of the orbit, \(T\), required for the inspector satellite is calculated from Kepler's third law given in Equation 3-7. Here \(\theta\), is the angle through which the constellation satellite has travelled in the time taken for PalmSat to orbit once in radians and \(\omega\), is the angular velocity of PalmSat, in \(\text{rads}^{-1}\). In addition Equation 3-7 states the relation of the orbital time period to the semi-major axis of the orbit, \(a\) in \(\text{m}\) and the Earth’s gravitational constant, \(\mu\), which has the value of \(3.986 \times 10^{14} \text{ m}^3\text{s}^{-2}\). This is then rearranged to give Equation 3-8.
Equation 3-7: Orbital Time Period

\[ T = \frac{\theta}{\omega} = 2\pi \sqrt{\frac{a^3}{\mu}} \]

Equation 3-8: Orbit Semi-Major Axis

\[ a = \sqrt{\frac{\mu T^2}{4\pi^2}} \]

Equation 3-9: Orbital Energy

\[ E = \frac{\mu}{2a} = \frac{1}{2} V^2 - \frac{\mu}{r} \]

Equation 3-10: Orbital Velocity

\[ V = \sqrt{2\left(E + \frac{\mu}{r}\right)} \]

Knowledge of the semi-major axis of the orbit then enables calculation of the orbital energy, \( E \), using Equation 3-9. Then through rearrangement of the Vis-Viva integral, also given in Equation 3-9, the orbital velocity can be found using Equation 3-10, where \( r \) is the orbit radius. The delta-V required to enter and exit the transfer orbit can then be calculated. Using these equations it is possible to trade-off the time taken to complete the phase-shift manoeuvre between adjacent satellites against the delta-V required in each scenario. This is shown in Table 3-2, where the angle between adjacent satellites is set to 60°.

<table>
<thead>
<tr>
<th>Number of Days in Transit</th>
<th>d</th>
<th>-</th>
<th>-</th>
<th>-</th>
<th>1</th>
<th>7</th>
<th>14</th>
</tr>
</thead>
<tbody>
<tr>
<td>Number of Orbits</td>
<td>n</td>
<td>1</td>
<td>4</td>
<td>10</td>
<td>14.9</td>
<td>104.3</td>
<td>208.6</td>
</tr>
<tr>
<td>Angle for Target to Move per Transfer Orbit (rad)</td>
<td>( \theta )</td>
<td>5.236</td>
<td>6.021</td>
<td>6.178</td>
<td>6.213</td>
<td>6.273</td>
<td>6.278</td>
</tr>
<tr>
<td>Time Period of Transfer Orbit (s)</td>
<td>( T )</td>
<td>4834.366</td>
<td>5559.520</td>
<td>5704.551</td>
<td>5736.348</td>
<td>5791.969</td>
<td>5796.604</td>
</tr>
<tr>
<td>Semi-Major Axis of Transfer Orbit (km)</td>
<td>( a )</td>
<td>6179.484</td>
<td>6782.931</td>
<td>6900.388</td>
<td>6926.006</td>
<td>6970.704</td>
<td>6974.423</td>
</tr>
<tr>
<td>Orbital Velocity at Insertion Point in Transfer Orbit (km/s)</td>
<td>( V_{\text{tr}} )</td>
<td>8.031</td>
<td>7.666</td>
<td>7.600</td>
<td>7.586</td>
<td>7.562</td>
<td>7.560</td>
</tr>
<tr>
<td>Delta-V to Enter/Exit Transfer Orbit (ms^-1)</td>
<td>( \Delta V )</td>
<td>473.565</td>
<td>107.984</td>
<td>42.461</td>
<td>28.392</td>
<td>4.030</td>
<td>2.014</td>
</tr>
<tr>
<td>Total Delta-V per Manoeuvre (ms^-1)</td>
<td>( \Delta V_{\text{total}} )</td>
<td>947.129</td>
<td>215.968</td>
<td>84.922</td>
<td>56.784</td>
<td>8.060</td>
<td>4.028</td>
</tr>
</tbody>
</table>

**Table 3-2: Length of Phase Shift Manoeuvre Transfer Data**

It is immediately obvious that the length of time available for the transit manoeuvre has a significant impact on the delta-V required. The relation is exponential hence the magnitude of the delta-V savings reduces as the transfer time increases.

It is assumed that there are six satellites in the constellation, therefore PalmSat will be required to make a minimum of 6 phase shift manoeuvres in the mission lifetime. It is assumed that PalmSat remains in the vicinity of each constellation satellite for one week each time, and the mission lifetime is one year. This means that if one inspection were made per satellite a total of...
53.83 days would be available for each transfer manoeuvre. Increasing the frequency of
inspection would have a direct impact on the utility of the inspection itself. The same calculation
is therefore repeated, with the number of days available for the transit dictated by the number of
inspections required over the total mission lifetime. Table 3-3 presents the data relating to a range
of inspection frequencies and includes an estimate of the total mission delta-V required for each
case considered.

<table>
<thead>
<tr>
<th>Total Number of Inspections</th>
<th>x</th>
<th>6</th>
<th>12</th>
<th>18</th>
<th>24</th>
<th>30</th>
</tr>
</thead>
<tbody>
<tr>
<td>Number of Days in Transit</td>
<td>d</td>
<td>53.83</td>
<td>23.42</td>
<td>13.28</td>
<td>8.21</td>
<td>5.17</td>
</tr>
<tr>
<td>Number of Orbits</td>
<td>n</td>
<td>802.067</td>
<td>348.958</td>
<td>197.872</td>
<td>122.329</td>
<td>77.033</td>
</tr>
<tr>
<td>Angle for Target to Move per Transfer Orbit (rad)</td>
<td>θ</td>
<td>6.282</td>
<td>6.280</td>
<td>6.278</td>
<td>6.275</td>
<td>6.270</td>
</tr>
<tr>
<td>Time Period of Transfer Orbit (s)</td>
<td>T</td>
<td>5800.033</td>
<td>5798.468</td>
<td>5796.352</td>
<td>5793.335</td>
<td>5788.687</td>
</tr>
<tr>
<td>Semi-Major Axis of Transfer Orbit (km)</td>
<td>a</td>
<td>6977.173</td>
<td>6975.918</td>
<td>6974.221</td>
<td>6971.800</td>
<td>6968.071</td>
</tr>
<tr>
<td>Orbital Velocity at Insertion Point in Transfer Orbit (km/s)</td>
<td>V</td>
<td>7.558</td>
<td>7.559</td>
<td>7.560</td>
<td>7.561</td>
<td>7.563</td>
</tr>
<tr>
<td>Delta-V to Enter/Exit Transfer Orbit (m²/s²)</td>
<td>ΔV</td>
<td>0.524</td>
<td>1.204</td>
<td>2.123</td>
<td>3.436</td>
<td>5.459</td>
</tr>
<tr>
<td>Total Delta-V per Manoeuvre (m/s²)</td>
<td>ΔV_{Total}</td>
<td>1.047</td>
<td>2.407</td>
<td>4.246</td>
<td>6.871</td>
<td>10.917</td>
</tr>
<tr>
<td>Total Mission Delta-V (m/s²)</td>
<td>ΔV_{Total}</td>
<td>6.283</td>
<td>28.887</td>
<td>76.434</td>
<td>164.905</td>
<td>327.512</td>
</tr>
</tbody>
</table>

Table 3-3: Effect of Number of Inspections on Propellant Mass for Manoeuvres

3.2.5 Inspection

The primary application of PalmSat in the proposed mission is to inspect another satellite.
Satellites are designed with onboard backup systems and fault diagnosis capabilities, although
correction capacity is limited. If, for any reason, the spacecraft then becomes damaged beyond
these capabilities, or cannot diagnose a fault, there is little that can be done from the ground. A
solution to this is to utilise a nanosatellite that would be capable of imaging other satellites and
determining if they are in working order. At a minimum this could diagnose a problem, even if its
solution were not yet feasible and possibly determine whether a replacement satellite is required.
An additional direct application for the nanosatellite is for it to provide a reference point to check
the calibration of sensors used by larger satellites for attitude control.

For the present scenario the PalmSat would be required to transit between adjacent satellites using
the phase shift manoeuvre previously described. Upon arrival at the target constellation satellite
some level of inspection is then required. There is little that PalmSat could see by simply flying
along a little distance away from the target satellite in the constellation. The satellite under
inspection could be commanded to rotate, such that the various panels were presented for
inspection. However, the initial orientation of the constellation satellite would need to then be
regained in what may be a highly complex manoeuvre. Alternatively the PalmSat nanosatellite
could be commanded to move into a sub-orbit, such that it then circumnavigated the constellation
satellite. The simplest way to do this is to utilise a Hill’s orbit, which can be described using
Hill’s equations [Clohessy’60, Prussing’93]. These are summarised in Equation 3-11, where $\omega$ is the angular velocity of the spacecraft in rad/s and $x$, $y$ and $z$ are positions, with the dots representing the first and second time derivatives.

$$\ddot{x} - 2\omega \dot{y} - 3\omega^2 x = 0$$
$$\ddot{y} + 2\omega \dot{x} = 0$$
$$\ddot{z} + \omega^2 z = 0$$

**Equation 3-11: Hill’s Equations**

Hill’s equations describe the motion of a satellite as seen by an observer in a circular orbit. By assuming the constellation satellite is in a perfectly circular orbit, the motion of the PalmSat nanosatellite around it can be considered. For these equations the $x$-direction is considered to be perpendicular to the orbit track in the radial direction, the $y$-direction along the orbit track in the direction of satellite motion and the $z$-direction normal to both the $x$- and $y$-directions across the orbit track. This is illustrated by Figure 3-5.

These equations may be solved either numerically or analytically. The analytic solutions are given in Equation 3-12, where the motion is seen to be a function of $\omega t$, the frequency of oscillation of the satellites. There are six constants of integration present: $A$, $W$ and $Z$ are initial position constants in the $x$-, $y$-, and $z$-directions respectively, relative to the target satellite. $\epsilon$, $b$ and $\delta$ are additional constants that cause modifications to the shape, and drift of the resulting ellipse that forms the natural solution. $\epsilon$ and $\delta$ are both phase terms, which cause the position of the PalmSat nanosatellite to be shifted relative to the target. If $b$ is non-zero it causes an offset in radial coordinate to be introduced, which results in the satellites drifting apart.

$$x = A \cos(\omega t + \epsilon) + b$$
$$y = W - \frac{1}{2} b(\omega t + \epsilon) - 2A \sin(\omega t + \epsilon)$$
$$z = Z \cos(\omega t + \delta)$$

**Equation 3-12: Analytical Solution to Hill’s Equations**
Inspection of the equation reveals that the $x$- and $y$-equations are coupled, meaning that motion in both directions must be considered simultaneously. The $z$-equation however is decoupled, which allows cross-track motion to be considered or ignored as required. The result of this is that the PalmSat nanosatellite may be placed into a sub-orbit around the constellation satellite with motion in the $x$- and $y$- directions (radial and along-track) only, allowing inspection of four of the six sides of the spacecraft [Wie’98]. The natural solution to Hill’s equations results in an elliptical orbit that is twice as long as it is wide. Figure 3-6 illustrates the trajectory of the inspection sub-orbit relative to the target satellite.

![Figure 3-6: Schematic of Inspection Sub-orbit](image)

Equation 3-12 allows the position of the PalmSat nanosatellite to be calculated relative to the target satellite at any given instant in time, while PalmSat is in the sub-orbit. The velocity of PalmSat, relative to the target satellite, while it is in the inspection sub-orbit is calculated from the first time derivative of the equations given in Equation 3-12, as shown in Equation 3-13.

$$
\dot{x} = -A \dot{\omega} \sin(\alpha x + \epsilon) \\
\dot{y} = -\frac{1}{2} b \dot{\omega} - 2A \dot{\omega} \cos(\alpha x + \epsilon) \\
\dot{z} = -\omega Z \sin(\alpha x + \delta)
$$

Equation 3-13: Velocity of PalmSat Relative to Target in a Hill’s Orbit

In order to place the PalmSat nanosatellite into a Hill’s orbit, a delta-V is required. To calculate the magnitude of the delta-V the velocity of PalmSat relative to the target satellite, immediately before and after the manoeuvre is considered. It is assumed that the manoeuvre is impulsive, which allows PalmSat to remain in the same position, but gain velocity. To simplify the analysis it is assumed that when this manoeuvre occurs, PalmSat is located at exactly the same orbit altitude as the target satellite, allowing $b$ to be zero. Finally it is assumed that the distance separating PalmSat and the target satellite must not be less than 100 m at any time. The natural solution to Hill’s equations result in an ellipse, the size of which is specified by the constants $A$ and $W$, as indicated in Figure 3-6. The other constant present in the $x$-, $y$-, $\dot{x}$- and $\dot{y}$- equations is the phase term $\epsilon$, which causes the position of PalmSat at the on-set of oscillation to be moved. To allow PalmSat to follow the sub-orbit trajectory shown in Figure 3-6, $\epsilon$ is set to be $\pi/2$.  

3-14
These assumptions allow the initial velocity of PalmSat, relative to the target satellite to be zero in both the $x$- and $y$- directions. Following the impulsive manoeuvre the velocity of PalmSat, relative to the target satellite, will be $\dot{x} \text{ ms}^{-1}$ in the $x$- direction and $\dot{y} \text{ ms}^{-1}$ in the $y$-direction. Substituting the value of $\pi/2$ for $\varepsilon$ in Equation 3-13 and using trigonometric identities results in Equation 3-14, which describes the velocity of PalmSat at any time following the manoeuvre.

$$
\begin{align*}
\dot{x} &= -A \omega \cos \alpha \\
\dot{y} &= 2A \omega \sin \alpha 
\end{align*}
$$

*Equation 3-14: Velocity of PalmSat in Hill's Orbit*

The delta-V required to place PalmSat into an inspection ellipse about the target satellite is the calculated using the vector sum of Equation 3-14, when $t = 0$. This results in Equation 3-15 as the $\dot{y}$-equation becomes zero and the cosine component of the $\dot{x}$-equation becomes 1.

$$
\Delta V = A \omega
$$

*Equation 3-15: Delta-V Requirement to Enter a Hill's Orbit*

The magnitude of $A$ determines the amplitude of the oscillation, which results in the sub-orbit of PalmSat about the target satellite. It follows that the larger the inspection ellipse, the greater the delta-V required to enter it. Based upon the previous assumptions, that PalmSat should remain a minimum distance of 100 m from the target satellite, $A = 100$ m. The delta-V requirement to enter and exit each inspection ellipse is then calculated to be 0.106 ms$^{-1}$ and 0.212 ms$^{-1}$ per inspection manoeuvre. Hill’s equations do not account for any orbit perturbations, however the duration of the inspection will be short, therefore it is assumed that these effects would be minimal.

### 3.3 Summary of Delta-V Requirements

A simple inspection mission is considered for PalmSat. The various assumptions used to develop the delta-V requirements for this mission are summarised in Table 3-4.

<table>
<thead>
<tr>
<th>Assumption</th>
<th>Value</th>
</tr>
</thead>
<tbody>
<tr>
<td>Orbit Altitude (km)</td>
<td>600</td>
</tr>
<tr>
<td>PalmSat Mass (kg)</td>
<td>1</td>
</tr>
<tr>
<td>Number of Satellites in Constellation</td>
<td>6</td>
</tr>
<tr>
<td>Mission Duration (years)</td>
<td>1</td>
</tr>
</tbody>
</table>

*Table 3-4: Mission Assumptions*

The delta-V requirements to maintain the desired orbit and complete the various manoeuvres considered are summarised in Table 3-5. The required frequency of the different manoeuvres is dependent upon various factors. The atmospheric drag correction will be conducted retrospectively dependent upon altitude degradation, however one manoeuvre per month is
expected. The attitude of the satellite will be affected by changes in the space environment as well as other forces. The number of inspections required will dictate the frequency of the phase shift and inspection manoeuvres. The maximum number of inspections possible will be discussed in section 3.4.2.

<table>
<thead>
<tr>
<th>Manoeuvre</th>
<th>Delta-V (ms⁻¹)</th>
</tr>
</thead>
<tbody>
<tr>
<td>Atmospheric Drag Compensation</td>
<td>0.082 per month</td>
</tr>
<tr>
<td>Attitude Control</td>
<td>$1.1 \times 10^7 \times 2$ per manoeuvre</td>
</tr>
<tr>
<td>Phase Shift</td>
<td>$1.047 \times 10.917$ per manoeuvre</td>
</tr>
<tr>
<td>Inspection Orbit</td>
<td>$0.108 \times 2$ per manoeuvre</td>
</tr>
</tbody>
</table>

Table 3-5: Inspection Mission Delta-V Requirements

3.4 Propulsion System Selection and Sizing

The delta-V required to effect each of the manoeuvres needed to achieve the mission specified was determined in the preceding sections. Various different types of propulsion system are available to meet these requirements. The typical performance of the different categories of propulsion system will now be considered to determine that which is most appropriate. This will also enable the calculation of representative thrust levels for each manoeuvre considered. In addition the present developments will be evaluated against the requirements to determine whether any of these systems may be appropriate. This is followed by the selection of an appropriate propellant for the chosen system.

3.4.1 Summary of Propulsion Systems

The various different methods of rocket propulsion were summarised in chapter 2 together with indications as to their performance. The chemical systems overall have a high thrust to weight ratio with a reasonable specific impulse, whereas the electrical systems have a low thrust to weight ratio with a very high specific impulse. These characteristics are illustrated by Figure 3-7, which shows the approximate position of each type of propulsion system according to their typical power consumption and thrust to weight ratio.
The differences in the performance of the systems are clear, with the groups covering much of the chart area. The first point to note is that the electrical based systems, including the resistojets are all located towards the top of the chart as they require high power levels to operate. The capabilities of PalmSat outlined in section 3.1.1 stated that a power of 1 W was available for the propulsion system. It is therefore unlikely that it would be possible to accommodate an electrical propulsion system onboard PalmSat. Based upon these assumptions, this leaves cold gas and chemical propulsion systems as the only available options.

### 3.4.2 Derivation of Thrust requirements

The next stage is to determine the thrust level required to accomplish each of the necessary manoeuvres within the time required. A delta-V is a change in velocity, therefore it follows that the magnitude of thrust, $T$, required to complete the manoeuvre may be calculated from Equation 3-16, which is derived from Newton’s second law of motion, $F=ma$.

$$\tau = m \frac{ΔV}{Δt}$$

**Equation 3-16: Calculation of Thrust Requirement**

For a given manoeuvre the delta-V, $ΔV$, required is fixed, therefore the key parameter in Equation 3-16 is the time taken to complete the manoeuvre, $Δt$. Ideally the time taken should be as short as possible, however there are practical limitations to this in terms of the thruster response time. These limitations include aspects such as valve actuation time, which would typically be of
the order of $5 \times 10^3$ s [SmithP'04]. Other factors such as catalyst bed heating time, if a monopropellant system were in use, may be significantly longer, of the order of seconds. The result of these effects is a transient thrust profile, which reduces the accuracy of the delta-V generated. To alleviate these effects a burn time that is longer than the predicted transients is required. The constraints imposed by PalmSat have led to the need for a chemical or cold gas propulsion system. It is therefore assumed that to alleviate the possibility of entirely transient thrust profiles a minimum burn time of 5 s is necessary.

The parameter that limits the delta-V available is the total mass of propellant available. For a chemical propulsion system the propellant mass, $m_p$, in kg is calculated using Equation 3-17, which is derived from the rocket equation [Larson'92]. Here $m_s$ is the initial mass of the spacecraft in kg, $\Delta V$ is total delta-V, in $\text{m s}^{-1}$ for all the manoeuvres and $I_p$ is the specific impulse of the propellant being used in s. For this analysis is assumed that the specific impulse is 120 s, to be indicative of a monopropellant chemical system.

$$m_p = m_s \left(1 - \exp\left(-\frac{\Delta V}{g I_p}\right)\right)$$

Equation 3-17: Propellant Mass

The initial specification for the propulsion system dictated the need for the total wet mass of the system to not exceed 100 g. In general the propellant will account for approximately 70-80% of a propulsion system mass, however as the propulsion system reduces in size this fraction is likely to be modified. Assuming that the allowable propellant mass is 50% of the complete propulsion system mass, 50 g of propellant can be accommodated. If the propulsion system were operating with a specific impulse of 120 s, this would correspond to a total delta-V of 60 $\text{m s}^{-1}$.

The delta-V required to overcome atmospheric drag per month was calculated to be 0.082 $\text{m s}^{-1}$. If a burn time of 5 s is assumed, then a thrust level of 16.4 mN is required. If the correction is applied every two months this would lead to the required thrust level increasing to 32.8 mN. In addition the propellant mass required to overcome atmospheric drag is $6.97 \times 10^{-5}$ kg per month, or 0.836 g for the complete one year mission.

The attitude control manoeuvre considered required two delta-V impulses to start and stop the rotation. Each impulse required a delta-V of $1.1 \times 10^4$ $\text{m s}^{-1}$ over a time of 5 s, to create the 22 $\mu\text{N}$ torque required. Due to the low thrust level required for these manoeuvres, for the purposes of this analysis it is assumed that a momentum wheel will be used for the attitude control system.

The mission scenario described required PalmSat to transit around the constellation of six satellites and inspect each one. The total number of inspections that may be completed during the mission lifetime of one year is dependent upon the delta-V necessary to achieve them. The total
delta-V required to complete a given number of inspections was calculated and is presented in Table 3-3. The greater the number of inspections achieved, the shorter the transit time between adjacent satellites, therefore the greater the delta-V required to complete the manoeuvre. The parameter that limits the amount of delta-V available will be the propellant mass available for the manoeuvre. The total mass of propellant required to complete the phase-shift manoeuvres is presented in Table 3-6, for five different inspection frequencies.

<table>
<thead>
<tr>
<th>Total Number of Inspections</th>
<th>x</th>
<th>6</th>
<th>12</th>
<th>18</th>
<th>24</th>
<th>30</th>
</tr>
</thead>
<tbody>
<tr>
<td>Delta-V to Enter/Exit Transfer Orbit (ms⁻¹)</td>
<td>aV</td>
<td>0.524</td>
<td>1.204</td>
<td>2.123</td>
<td>3.436</td>
<td>5.459</td>
</tr>
<tr>
<td>Total Mission Delta-V (ms⁻¹)</td>
<td>AVₜₒₜ</td>
<td>6.283</td>
<td>28.887</td>
<td>76.434</td>
<td>164.905</td>
<td>327.512</td>
</tr>
<tr>
<td>Propellant Mass for All Phase Shift Manoeuvres (g)</td>
<td>mₚ</td>
<td>5.32</td>
<td>24.24</td>
<td>62.87</td>
<td>130.71</td>
<td>242.86</td>
</tr>
</tbody>
</table>

Table 3-6: Propellant Mass Requirements for Phase Shift Manoeuvres

The total propellant mass is limited to 50 g, therefore the maximum number of inspections possible is two per constellation satellite, giving a total delta-V of 28.89 ms⁻¹ and a propellant mass of 24.24 g. For insertion into each transfer orbit a delta-V of 1.204 ms⁻¹ is required. Consideration of the minimum firing time of 5 s allows a maximum thrust level of 241 mN. The delta-V requirement for these manoeuvres is significantly higher than for any of the others considered, which results in a higher thrust requirement. Increasing the burn time will reduce the thrust required. For example a burn time of 12 s corresponds to a thrust level of 100 mN.

The phase shift manoeuvre calculations indicated that twelve inspection manoeuvres would be required. Insertion into each inspection sub-orbit requires a delta-V of 0.108 ms⁻¹, which if achieved in the minimum firing time of 5 s, would require a thrust of 21.6 mN. Alternatively a thrust of 20 mN could be used if a firing time of 5.4 s was acceptable. The total mission delta-V for this manoeuvre would be 2.59 ms⁻¹, which leads to the requirement for 1.08 g of propellant for the complete mission.

### 3.4.3 Summary of Propulsion System Requirements

An estimate of the thrust required to perform each of the manoeuvres in a single burn, within an appropriate firing time was calculated in the previous section. In addition an estimate of the propellant mass required to perform these manoeuvres for the duration of the mission was given. The assumptions used to calculate these values are summarised in Table 3-7. In addition the thrust required to generate the necessary delta-V for each manoeuvre is given in Table 3-8 together with the time taken to complete the burn and the total propellant mass required.

<table>
<thead>
<tr>
<th>Assumption</th>
<th>Value</th>
</tr>
</thead>
<tbody>
<tr>
<td>PalmSat Propellant Mass (g)</td>
<td>50</td>
</tr>
<tr>
<td>Minimum Burn Time (s)</td>
<td>5</td>
</tr>
</tbody>
</table>

Table 3-7: Thrust Derivation Assumptions
Table 3-8: Propulsion System Thrust Requirements

From this summary it is clear that a propulsion system capable of producing two thrust levels is required, one at approximately 10 - 20 mN and another at approximately 100 mN. In addition it is possible to calculate that the total propellant mass required to complete all the necessary manoeuvres for the entire mission duration is 26.16 g. From this estimation it becomes clear that all of the required manoeuvres are possible including an appropriate propellant margin within the 50 g limit.

The miniaturisation of a propulsion system will inherently modify the performance characteristics observed, therefore the present developments will now be considered with reference to the mission requirements derived.

3.4.4 Summary of Present Developments

Examples of some current research efforts into micropropulsion systems were discussed in chapter 2. Table 3-9 summarises the key data from eleven different development projects, where the (M) subscript refers to a MEMS based system, and the bracketed entries correspond to the theoretical predictions. In addition data relating to a hydrazine monopropellant thruster is included to provide a comparison with a current industry standard thruster that produces a similar thrust level. It should be noted that the mass data used here is that of the thruster component only, with no associated systems or propellant masses. This is to enable direct comparisons with the mission requirements and strict platform constraints under consideration. The power and thrust to weight ratio data for each thruster are plotted in Figure 3-8 to demonstrate the breadth of capability of the different propulsion systems.

Table 3-9: Summary of Micropropulsion Developments
Chapter 3: Mission Analysis & Propulsion System Selection

It can be seen from Figure 3-8 that the electrical propulsion systems again dominate the top left corner of the chart, with high power requirements and low thrust to weight ratios. The chemical systems continue to dominate the bottom right corner, with inherently higher thrust to weight ratios and lower power requirements. The specific impulse of the different thrusters is not considered at this stage, as the thruster is strictly limited in mass and power yet the specific impulse should be as high as possible within these constraints.

The mission requirements are superimposed on to the chart. The three key manoeuvres are considered, the higher thrust phase shift manoeuvre and the two lower thrust inspection and atmospheric drag correction manoeuvres. The platform constraints restrict the power available to the thruster to 1 W and it is assumed that this will be fully utilised. The thrust to weight ratio for each manoeuvre is based upon the thrust requirement given in Table 3-8 together with the assumption that the design thruster mass is 25 g.

It can be seen that these requirements are located towards the bottom right corner of the chart, indicating the need for a high thrust to weight ratio and a low power requirement. The mission requirements fall between the monopropellant and bi-propellant capabilities, but above those of the cold gas thruster in terms of thrust to weight ratio and far away from the electrical propulsion systems in terms of both power and thrust to weight ratio.
Chapter 3: Mission Analysis & Propulsion System Selection

The only system presently under development that approaches the mission requirements is the hydrogen peroxide monopropellant thruster. This system is manufactured using MEMS techniques and the results from testing demonstrate the complications with using silicon as a structural material [Hitt'01]. Initially the decomposition chamber contained pillars, etched from the silicon substrate, which were coated with silver. The decomposition reaction revealed that the bond between the silicon and the silver was poor, resulting in poor decomposition. In addition the silicon surrounding the decomposition chamber acted as a heat sink, drawing heat away from the reaction. Further research into low thrust, low mass monopropellant developments revealed that there is little research currently underway in this area.

3.4.5 Propulsion System Selection

The data presented demonstrates that the mission requirements severely restrict the available options for a propulsion system. The low power budget prevents the use of an electrical system, as even though a system of capacitors may be sufficient to generate the required potential, the associated mass would be prohibitive.

The very low power requirement of the cold gas system is encouraging and this system may be able to accommodate the low thrust inspection and drag compensation manoeuvres. The thrust required for the phase shift manoeuvre however, would be in excess of its capabilities without the use of high pressures. The resistoljets offer a performance gain, at the expense of power, but not one that is significant enough.

This leaves the chemical systems, with the bi-propellant system offering the highest potential thrust to weight ratio, for a power only slightly more than the cold gas system. The associated complexity of this system must be considered though, as the values quoted do not appear to be sustainable long term. The monopropellant hydrogen peroxide system is operating in the appropriate power range, although the thrust to weight ratio is too low at this stage of development. The thrust to weight ratio offered by the hydrazine monopropellant thruster is suitable for the requirements set out, but the operating power required is more than that available.

Overall, while some of the developments reviewed approach the requirements set out none are directly appropriate for use onboard PalmSat. It was therefore decided that a new system should be developed, specifically for PalmSat, with the known platform constraints in mind. The above analysis has indicated that a chemical propulsion system is necessary to meet the mission requirements. To minimise system complexity a monopropellant system was selected for further analysis and development.
3.4.6 Propellant Selection

In order to select the most appropriate propellant, various properties of different propellants were critically compared. Six different propellants were considered, however for various reasons hydrazine was not included, although it would have been the optimal selection. First, the handling requirements would have precluded use in a university research environment and in addition this research aimed to use a Green propellant.

Table 3-10 summarises data relating to the six different green propellants introduced in chapter 5 and ranks them according to their performance in six categories, where (1) indicates good performance and (6) indicates poor performance. Decomposition temperature is included for two reasons, first to minimise the cost of materials by eliminating the need for exotic materials. Second, if a MEMS approach were to be adopted the decomposition temperature should not exceed the maximum working temperature of silicon of approximately 1500 K. The specific impulse category identifies the efficiency of the propellant, whereas the density specific impulse category allows consideration of the physical storage characteristics of the propellant. For the purposes of this development, it is critical that the catalysis of the propellant is well understood. The catalysis category is therefore included to rank each propellant according to the degree of understanding associated with it. Storage and handling requirements are considered to ensure the risk from utilising the propellant can be minimised through strict adherence to them. The cost of the propellant is included to provide an approximate indication of the current pricing situation, as a strict budget is associated with any program of research.

From Table 3-10 it can be seen that both hydrogen peroxide and nitrous oxide score equally. Both propellants perform exceptionally in all categories apart from the two relating to specific impulse. Based upon the density specific impulse figures it can be seen that hydrogen peroxide has a considerably superior storage density and for this reason it is ranked first. The long term storage characteristics of hydrogen peroxide are known to be complex, however it is surmised that within the mission lifetime of one year, this should be of less concern. Overall the synthetic propellants have higher performance characteristics but the current lack of detailed understanding of their decomposition characteristics and high cost makes them less attractive for this research. From this comparison, hydrogen peroxide is chosen to be the propellant for this research.

<table>
<thead>
<tr>
<th>Propellant</th>
<th>Decomposition Temperature (K)</th>
<th>Specific Impulse (s)</th>
<th>Density Specific Impulse (kgm^-3)</th>
<th>Catalysis</th>
<th>Storage &amp; Handling Requirements</th>
<th>Cost</th>
<th>Overall Ranking</th>
</tr>
</thead>
<tbody>
<tr>
<td>H2O2</td>
<td>1010</td>
<td>(1) 160 (6)</td>
<td>224000 (5)</td>
<td>Known</td>
<td>Known</td>
<td>Low</td>
<td>1 (15)</td>
</tr>
<tr>
<td>N2O4</td>
<td>1100</td>
<td>(1) 200 (5)</td>
<td>149000 (6)</td>
<td>Known</td>
<td>Known</td>
<td>Low</td>
<td>2 (15)</td>
</tr>
<tr>
<td>HAN</td>
<td>1400</td>
<td>(3) 250 (1)</td>
<td>310200 (2)</td>
<td>Known</td>
<td>Un known</td>
<td>High</td>
<td>3 (16)</td>
</tr>
<tr>
<td>DMAMZ</td>
<td>1300</td>
<td>(3) 245 (4)</td>
<td>265727 (4)</td>
<td>Known</td>
<td>Partially known</td>
<td>High</td>
<td>4 (18)</td>
</tr>
<tr>
<td>ADN</td>
<td>2000</td>
<td>(6) 250 (1)</td>
<td>325000 (1)</td>
<td>Partially known</td>
<td>Un known</td>
<td>High</td>
<td>5 (19)</td>
</tr>
<tr>
<td>HNF</td>
<td>1600</td>
<td>(5) 250 (1)</td>
<td>286000 (3)</td>
<td>Un known</td>
<td>Partially known</td>
<td>High</td>
<td>6 (21)</td>
</tr>
</tbody>
</table>
3.5 Summary of Approach

The utility of a propulsion-enhanced PalmSat has been demonstrated and an appropriate mission scenario has been identified. Consideration of the characteristics of different types of propulsion systems has led to the selection of a monopropellant system. As yet, no successful monopropellant development is available to provide the necessary thrusts for the proposed mission scenario within the mass, power and volume constraints. A review of the available literature relating to the monopropellant thruster development at GSFC [Hitt'01, Kujawa'03a] has revealed many unforeseen complications in the development of a microthruster. As a result of this, the remainder of this research will consider the miniaturisation of a monopropellant thruster in stages. This will allow the various effects of miniaturisation to be identified and encompassed within the design. The thrust levels required by the mission described will dictate the extent of miniaturisation required. The final system is unlikely to conform to standard design strategies, however it will enable the successful generation of thrust on a small scale. A summary of the thruster parameters dictated by the platform and mission requirements are given in Table 3-11.

<table>
<thead>
<tr>
<th>Parameter</th>
<th>Value</th>
</tr>
</thead>
<tbody>
<tr>
<td>Thrust Levels Required</td>
<td>10 mN; 100 mN</td>
</tr>
<tr>
<td>Thruster Mass</td>
<td>25 g</td>
</tr>
<tr>
<td>Propellant</td>
<td>Hydrogen Peroxide</td>
</tr>
</tbody>
</table>

Table 3-11: Summary of Thruster Parameters
Chapter 4: Fundamentals & Complexities of Miniaturisation

The development of an appropriate mission scenario and evaluation of propulsion systems has led to the need for a new monopropellant microthruster. In order to successfully miniaturise a propulsion system there are numerous characteristics that require investigation to determine how the behaviour of the thruster may be modified due to the small dimensions. The theory used to describe the operation of a thruster is discussed in section 4.1. The different manufacturing techniques that are available for the manufacture of a microthruster are considered in section 4.2. This allows the constraints placed on the fabrication of the system to be considered in terms of machining accuracy and materials used. This is followed by an introduction to computational fluid dynamics in section 4.3, which describes the numerical methods used to predict a fluid flow. An evaluation of the various effects that may influence the generation of thrust is then presented, in sections 4.5 and 4.6 with a focus on the thermal and fluidic characteristics of the thruster. It should be noted that this discussion is limited to the thruster component itself consisting of the decomposition chamber and exhaust nozzle.

4.1 Essential Rocket Theory

In order to explore in detail how various effects may influence the performance of a propulsion system as it is miniaturised, it is necessary to understand the key parameters that are used to determine and quantify it. This section will summarise the derivation of some aspects of rocket theory for reference in later sections. The equations used are sourced from two references [Humble'95, Hill'92].

Newton's second law of motion dictates that a force vector, \( \mathbf{F} \) applied to a system of mass, \( m \) results in a change in momentum as shown in Equation 4-1, where \( \mathbf{v} \) is the resultant velocity vector of the system.

\[
\mathbf{F} = \frac{d}{dt} (m\mathbf{v})
\]

Equation 4-1: Newton's Second Law of Motion

If the resultant velocity vector of the system is unsteady then the total momentum of the system can be defined through the use of a control mass. This is shown in Equation 4-2, where \( \rho \) is the
density and the subscript "cm" denotes the control mass. Here the density and resultant velocity vector are evaluated for each system volume increment, $dV$.

$$F = \frac{d}{dt} \int_{\text{cm}} \rho v \, dV$$

**Equation 4-2: Control Mass Definition of Newton's Second Law of Motion**

In the case of a propulsion system it is the flow through the engine with respect to thrust that is of interest, therefore it is more convenient to define a control volume instead of a control mass. All the relevant dimensions are included within this control volume, with a control surface required for the exit flow. The momentum equation for a control volume is defined in Equation 4-3, where $n$ is the outward normal vector at any point and the subscripts "cv" and "cs" denote the control volume and control surface respectively.

$$\sum F = \frac{d}{dt} \int_{\text{cv}} \rho v \, dV + \int_{\text{cs}} \rho (v \cdot n) \, dA$$

**Equation 4-3: Control Volume Definition of the Momentum Equation**

Inspection of this equation reveals that the first integral is the vector sum of the momenta of all parts of the control volume at any one instant of time. In addition the second integral is the net rate of flow of momentum out of the control surface.

Now consider the thrust of a stationary rocket, which is illustrated in Figure 4-1. For simplicity it is assumed that the force is one dimensional, with a steady exit velocity, $u_e$ and propellant mass flow rate, $m$. The stationary control surface, $S$, is aligned such that it intersects the exhaust flow at the nozzle exit plane perpendicularly. Flow through this control surface will cause a positive thrust force to act in the opposite direction to $u_e$. The reaction to this positive thrust force is then shown in Figure 4-1, which acts on the control volume.

![Figure 4-1: Rocket in Static Firing](image-url)
The momentum equation for the control volume described is slightly modified from that given in Equation 4-3 as the second integral is defined in terms of the mass flow rate of the exit flow. For simplicity only the forces in the direction parallel to the flow are considered and this is aligned with the x-direction shown in Figure 4-1. The momentum equation for this control volume is given in Equation 4-4, where $u$ refers to the velocity in the x-direction only.

$$\sum F_x = \frac{d}{dt} \left( \int_{\text{in}}^{\text{out}} p u \, dV + \int_{\text{in}}^{\text{out}} \mu \, d\eta \right)$$

Equation 4-4: Momentum Equation for a Rocket Engine

When the propellant is within the propellant tanks it may be considered to be stationary. In addition when the propellant is in the combustion chamber the flow may be considered to be steady, which allows the elimination of the time derivative term. Equation 4-4 then reduces to the form shown in Equation 4-5. The geometry of the rocket engine, together with the control surface definition dictates that any flow exiting the control volume does so via the exit nozzle, where the flow may be assumed to be steady with a magnitude $u_e$. This allows the final reduction on the right hand side of Equation 4-5.

$$\sum F_x = \int_{\text{in}}^{\text{out}} u \, d\eta = n u_e$$

Equation 4-5: Simplified Momentum Equation

If the flow exiting the control volume may be considered to be a continuum, then it is necessary to consider the pressure either side of the stationary control surface. The pressure just inside the exit plane is denoted by $p_i$ and the exterior pressure by $p_e$. The pressure on the control surface may be considered to be uniformly $p_e$ except in the plane of the exit flow. The sum of forces incident on the control volume can then be expressed as shown in Equation 4-6, where $A_e$ is the cross-sectional exit area of the nozzle.

$$\sum F_x = \sum A_e (p_i - p_e) = T$$

Equation 4-6: Sum of Forces on the Control Surface

From this it can then be seen that the thrust is a result of a pressure distribution over the interior and exterior surfaces. Combining Equation 4-5 and Equation 4-6 and rearranging allows the thrust to be defined in terms of momentum and the pressure terms as shown in Equation 4-7.

$$T = n u_e + (p_e - p_i) A_e$$

Equation 4-7: Thrust Equation

If the pressure in the exhaust exit plane is the same as the ambient pressure, then the flow is correctly or optimally expanded and corresponds to the maximum thrust for given chamber
conditions. Under these conditions the second term on the right hand side of Equation 4-7 disappears and the thrust is solely a function of the exit velocity and propellant mass flow rate. For a rocket engine operating in the vacuum of space, perfect expansion is never reached and instead a compromise is made between the expansion achieved and the resultant mass of the nozzle.

The thrust that results from a rocket firing has been identified. The next stage is to consider the impulse, $I$, imparted to the spacecraft by the thruster firing. This is determined by the magnitude of thrust, $T$, in combination with the firing time, $t$, shown by Equation 4-8, assuming that no significant variations occurring during the firing.

$$ I = Tt $$

**Equation 4-8: Definition of Impulse**

It is more convenient to define a specific impulse, $I_{sp}$, the total impulse per unit weight of propellant and is used as a measure of how efficiently the propellant delivers thrust. This is defined in Equation 4-9, where $g_o$ is the acceleration due to gravity at the Earth’s surface and $m_p$ is the mass of propellant expelled during the firing.

$$ I_{sp} = \frac{T}{g_o m_p} = \frac{kgms^{-2} \times s}{ms^{-2} \times kg} $$

**Equation 4-9: Definition of Specific Impulse**

The dimensional analysis is included in Equation 4-9 to demonstrate that the unit of specific impulse is seconds. This means that the value is independent of the amount of propellant expelled in one burn, but dependant upon the propellant itself. Equation 4-9 may be rearranged to give the thrust as a function of specific impulse as shown in Equation 4-10, where $\dot{m}$ is the mass flow rate of propellant.

$$ T = \frac{I_{sp} m_p g_o}{t} = \dot{m} I_{sp} g_o $$

**Equation 4-10: Definition of Thrust as a Function of Specific Impulse**

Assuming correct expansion of the flow, Equation 4-7 and Equation 4-10 may be compared directly, to give an ideal exhaust velocity shown in Equation 4-11. This excludes any inefficiencies and losses but allows the calculation of the maximum possible exit velocity, known as the effective exhaust velocity, $u_{eff}$.

$$ u_{eff} = I_{sp} g_o $$

**Equation 4-11: Effective Exhaust Velocity as a Function of Specific Impulse**
Chapter 4: Fundamentals & Complexities of Miniaturisation

The effective exhaust velocity is an ideal maximum exhaust velocity. In reality the exhaust velocity will be lower than this value due to inefficiencies in the expansion of the flow and thermal losses. To calculate a value of exhaust velocity that does allow for losses, an approach based upon thermodynamics is required. Consider the flow through a rocket engine, the flow enters the combustion chamber, reacts and is then exhausted through a nozzle. The flow is a continuum, therefore it is possible to relate the flow velocity at two different locations through evaluation of thermodynamic properties, such as the enthalpy, $h$. If the flow velocity in the $x$-direction is considered then the velocity at two different positions may be related using Equation 4-12, assuming that location 1 is upstream of location 2.

$$u_2 = \sqrt{2(h_1 - h_2) + u_1^2}$$

**Equation 4-12: Velocity Relation Using Enthalpy**

Assuming that location 1 is the combustion chamber, denoted by the subscript $c$, and location 2 is the exit from the nozzle, denoted by the subscript $e$. Equation 4-12 may be re-written as Equation 4-13.

$$u_e = \sqrt{2(h_e - h_c) + u_e^2}$$

**Equation 4-13: Updated Velocity Relation**

The velocity in the combustion chamber is assumed to be zero as is it significantly slower than the velocity at the exit, allowing this term to disappear. Experimental evaluation of the enthalpy of a flow is difficult, but the temperature of the flow is easily sampled. The temperature and enthalpy of a flow are related through the definition of specific heat at constant pressure, $c_p$, ($c_p = \Delta h/\Delta T$). This expression allows Equation 4-13 to be converted to be in terms of temperature and reduce to Equation 4-14.

$$u_e = \sqrt{2c_p(T_e - T_c)}$$

**Equation 4-14: Substitution of $c_p$**

The specific heat term may be redefined in terms of the local gas constant and the ratio of specific heats, $\gamma$, ($c_p = R\gamma/(\gamma - 1)$). Substitution of this term combined with the factorisation of the combustion temperature gives Equation 4-15.

$$u_e = \sqrt{\frac{2RT_e}{\gamma - 1}\left(1 - \frac{T_c}{T_e}\right)}$$

**Equation 4-15: Substitution of Gas Constant**

The local gas constant, $R$ is not always known, therefore it is more convenient to define this in terms of the universal gas constant, $R_u$ and the local molecular mass, $M$ where $R = (R_u/M)$. In

4-5
addition the temperature of the flow at the exit from the nozzle is unknown, but the exit pressure is known through the pressure ratio. A thermodynamic relation links a temperature ratio to a pressure ratio through the use of the ratio of specific heats, $\gamma$. Equation 4-16 encompasses all these alterations and provides a relation that is easily used to calculate the predicted exhaust velocity. The key parameters within this equation that may result in losses are the combustion chamber temperature, $T_c$, through thermal losses and the pressure ratio through poor flow expansion.

$$u_e = \sqrt{\frac{2\gamma R T_c}{(\gamma - 1) M}} \left[1 - \left(\frac{P_e}{P_r}\right)^{\frac{\gamma - 1}{\gamma}}\right]$$

Equation 4-16: Exit Exhaust Velocity

Following identification of the potential sources of losses, methods of fabrication available for the manufacture of a micropropulsion system are considered.

### 4.2 Methods of Fabrication

As the dimensions of the propulsion system become smaller manufacturing the components becomes increasingly difficult using conventional methods. To alleviate this alternative methods may be used. There are two main options for the fabrication of a micropropulsion system: Micro-Electro-Mechanical-Systems (MEMS) fabrication methods or micro conventional precision machining techniques. MEMS techniques are derived from processes within the electronics industry and utilise etching methods to create a quasi-three dimensional structure from silicon wafers. Micro conventional precision machining techniques utilise miniaturised tools and machines to create components from standard materials using conventional methods.

One of the key differentiating factors between the two techniques is the accuracy of the finished component. The accuracy of any fabrication method can be characterised in two ways, absolutely or relatively. Absolute accuracy is defined with reference to the feature size under consideration, while relative accuracy is defined with reference to the part under consideration. For example in terms of house construction, a relative accuracy of $10^{-5}$ is considered excellent, i.e. 1 mm variation per m length. In contrast, micro conventional precision engineering methods are expected to achieve a relative accuracy of at least $10^{-7}$, and at present relative accuracies of $10^{-6}$ are becoming standard. When relative accuracy is used in association with MEMS processes $10^{-2}$ is considered good. It is therefore important to consider this distinction, as the accuracy of MEMS processes is usually quoted as absolute accuracy, which will appear to be significantly higher due to the feature detail that is possible.
The Taniguchi plot [Madou’98] in Figure 4-2 shows the range of application for microconventional precision machining in terms of both absolute and relative tolerances. It can be seen that the relative tolerance for a house is comparable with that for a micromachine that has been fabricated using MEMS techniques. In addition to the consideration of accuracy, there are advantages to each fabrication method and some particular aspects are summarised below.

**4.2.1 MEMS Fabrication**

The MEMS fabrication process utilises combinations of thin silicon wafers, with a thickness of approximately 0.5 mm, to create structures. There are various techniques that may be used, however in general a design for each wafer is created and transformed into a highly detailed structure through the application of a mask. Once the mask is applied to the wafer the appropriate material is etched away. It is the etching approach that enables the high feature detail associated with this technology as no physical tooling is required. Features with dimensions of the order of microns are possible, allowing high complexity components to be created. Once etched adjacent wafers are joined together using anodic bonding to create a wafer stack. The maximum number of wafers that can be joined together is approximately six, to prevent alignment issues from
arising. This approach means that while a high level of feature detail may be attained, the overall geometry of the components is essentially two-dimensional. This is a distinct advantage in some cases, as a large number of features may be contained in a small package, however if the required features are three-dimensional this is less of an advantage. In addition, the design for each mask that is created must be verified. Once this happens the mask may be re-used at any time, but it is a protracted process to complete.

A key advantage of this method is the associated low production cost. Following certification of a design, the required masks are mass produced and the necessary wafers are created using batch processing techniques. The result is a low unit cost, with the capability for large quantities of components to be produced in a short timescale. The advantages of this are clearly evident within the silicon industry, where components such as computer processor chips are manufactured on a large scale. The high setup and qualification costs are then offset against the unit cost of the thousands of components. The repeatability of the process is also a distinct benefit of this manufacturing method as once a design is qualified production of the component may be easily stopped and re-started.

The other aspect of this technology is that relating to the component integration that is possible. The capability of small feature sizes means that it is possible for numerous different functions to be combined into a single component. This utility of this has been demonstrated by the silicon industry where the boundaries of functions such as memory space within a given chip size have been extended further than it was ever thought possible.

Many of the advantages of MEMS processing techniques link directly to the electrical properties that it is possible to combine within the component. This is of less interest to a propulsion system, where it is the structural properties of a component that are important. In addition, it is unlikely that a propulsion system would require the volume of production that is associated with MEMS techniques, therefore use of the process is likely to become prohibitively expensive. Overall, while the technology may be relevant to other sub-systems onboard a spacecraft, it appears unlikely that MEMS technology is at present of much utility to a propulsion system.

4.2.2 Micro Conventional Precision Machining

The micro conventional precision machining industry offers an alternative to MEMS processing techniques. These techniques are based upon conventional methods, which utilise miniaturised tools, allowing small features to be created from conventional materials. For example micro-milling machines are currently able to create features of 50 μm, with a tolerance of 1 μm. These advances mean that in certain cases micro conventional precision machining is a viable competitor to MEMS fabrication.
When creating a component using micro conventional precision machining methods a prototype of the component is required prior to production on any scale. This allows the design to be evaluated and any design flaws to be identified and there are two primary ways in which this may be done. A direct prototype may be created using conventional manufacturing techniques, or alternatively a representative prototype from resin or plastic, may be created. Use of a representative prototype allows direct identification of any sections that may be highly complex to machine or that may require special tooling. The techniques used to create these prototypes allow a minimum feature size of 5 μm, with a tolerance of ±3 μm, thereby ensuring a high accuracy evaluation. Following qualification the design is then input into the appropriate automated machines for high tolerance fabrication. The disadvantage to using the representative prototyping approach is the additional cost and time involved. The advantage is that a higher quality design will result from the additional evaluation stages, therefore dependent upon the complexity of the component it may be a more efficient approach.

One of the key attractions of micro conventional precision machining is the high speed and low cost with which a design may reach the prototype stage, in comparison to MEMS techniques. This directly impacts the production stages, as even if the number of components required is low, the initiation cost is lower, therefore there is less initialisation cost to be amortised per component.

The previous sections within this chapter have discussed how the surface finish of a component may affect the efficiency of the flow passing through it. Use of micro conventional precision machining techniques allows the surface finish of a component to be controlled directly. This is due to the variety of machining techniques that are available in addition to the different materials that may be used, in contrast to the etching techniques associated with MEMS manufacture.

### 4.2.3 Comparison of Fabrication Methods

The main advantages and disadvantages of the two fabrication types available for machining micro scale components have been summarised in the previous two sections. From this evaluation it is clear that the chosen fabrication method influences numerous design aspects.

The critical factor when selecting a fabrication method appears to be the level of precision that is acceptable. In order to create device with smaller and smaller features, MEMS fabrication is undoubtedly the solution, however the poor relative tolerance associated with MEMS may limit its range useful applications. The required scale of production is an additional consideration. The capabilities and availability of MEMS fabrication continues to extend. As a result the possibility of small batch size runs may arise, however this may still not be suitable for the two or three components required by a propulsion system. The materials used by MEMS techniques, such as silicon, are not favoured for high temperature environments, whereas conventional materials such
as steel and other metals are, although the mass will increase. The selection of the manufacturing technique used for the fabrication of a propulsion system is likely to require evaluation according to individual requirements. For the development of a miniaturised monopropellant thruster, it was decided that the use of micro conventional precision machining techniques was more suitable.

### 4.3 Computational Fluid Dynamics

Computational Fluid Dynamics (CFD) is a method of predicting the behaviour of a fluid flow and how it interacts with its surroundings. This section will introduce some of the key concepts associated with this field. The fundamental numerical theory of fluid flow will be introduced together with the governing equations.

The commercial CFD solver, Fluent®, version 6.1 was used for all CFD work completed during this research. The numerical approach behind the finite volume method used within Fluent® will therefore be introduced together with the approximations it requires. Finally a generalised assessment of the quality of CFD will be made.

#### 4.3.1 Navier-Stokes Equations

The Navier-Stokes equations are an open set of equations that enable the prediction of fluid properties from one location to the next. They comprise of three sections, addressing continuity of mass, momentum and energy. The equations may be expressed in conservation or non-conservation form, dependent upon the situation [Anderson'95]. In the case of fluid flow through a thruster, the equations are used in conservation form. This approach considers an infinitesimal fluid element that is small enough to be negligible with regards to calculus, but contains enough fluid molecules to be regarded as a continuum. At the instant of calculation the element is assumed to be stationary within the fluid and the forces on each face of the element are expressed through equations. From these equations various flow properties may be calculated at a specific moment in time for an infinitesimal fluid element. For two-dimensions the properties calculated include density, \( \rho \); pressure, \( P \); temperature, \( T \); \( x \)-direction velocity, \( u \); and \( y \)-direction velocity, \( v \).

The conservation forms of the complete equations for a 2-dimensional unsteady viscous flow are stated in Equation 4-17, Equation 4-18 and Equation 4-19, in Cartesian Tensor notation.

\[
\frac{\partial \rho}{\partial t} + \nabla \cdot (\rho \mathbf{u}) = 0
\]

**Equation 4-17: 2-D Navier-Stokes Continuity Equation**
Chapter 4: Fundamentals & Complexities of Miniaturisation

\[
\frac{\partial (\rho u_i)}{\partial t} + \nabla \cdot (\rho u_i u_j) = -\frac{\partial P}{\partial x_i} + \frac{\partial \tau_{ij}}{\partial x_j} + \rho f_i
\]

Equation 4-18: 2-D Navier-Stokes Momentum Equation

\[
\frac{\partial (\rho E)}{\partial t} + \frac{\partial (\rho E u_j)}{\partial x_j} = \frac{\partial}{\partial x_j}\left( k \frac{\partial T}{\partial x_j} - u_j \rho \dot{u} + u_j \tau_{ij} \right) + \rho f_i u_j
\]

\[
\tau_{ij} = \nu \left( \frac{\partial u_i}{\partial x_j} + \frac{\partial u_j}{\partial x_i} \right) - \frac{2}{3} \delta_{ij} \frac{\partial u_l}{\partial x_l}
\]

Equation 4-19: 2-D Navier-Stokes Energy Equation

In the momentum and energy equations, \( \tau_{ij} \) represents the shear stress tensor and \( f_i \) represents the body forces within the fluid. In the energy equation, \( E \) represents the total energy and \( \dot{q}_i \) represents the heat flux out of the fluid element. The thermal conductivity of the fluid is represented by \( k \) and \( \delta_{ij} \) is the Kronecker delta\(^1\). Inspection of the momentum equation reveals that the left hand side contains velocity and convection components, while the right hand side contains a pressure gradient, a viscosity term and a body force term.

The interlinked nature of the Navier-Stokes equations results in few situations where they may be solved directly. Various parameters affect the complexity of the solution, including geometry and flow state, therefore instead of attempting to solve the equations directly, an iterative approach is usually adopted. The development of computer technology has enabled significant advances in this field allowing the generation of solutions for an increasing range of geometries.

### 4.3.2 Computational Methods

The development of CFD methods has enabled a significant increase in the range of cases where solutions to the Navier-Stokes equations are now possible for both laminar and turbulent flows. A range of solution methods is available, however overall they all begin from the same principle.

It is assumed that the bulk fluid may be split into infinitesimal elements, across which there is a small change in any flow property, collectively denoted by \( \phi \) [Anderson'95]. The way in which these changes are evaluated then distinguishes the different methods. There are three different approaches that may be used to model a fluid flow: finite difference methods, finite element methods and spectral methods. Each of these methods will now be briefly reviewed.

---

\(^1\) The Kronecker delta is a mathematical function of two variables, which has a value of one if they are equal and zero if they are not. i.e. \( \delta_{ij} = 1 \) if \( i = j \), and \( \delta_{ij} = 0 \) if \( i \neq j \)
4.3.2.1 Finite Difference Methods

A finite difference method describes the unknown flow variables, \( \phi \), through use of point samples at the nodes of a grid. A Taylor series expansion is used to generate finite difference estimates of \( \phi \). This results in an algebraic equation for all values of \( \phi \) at each node [Versteeg'95].

4.3.2.2 Finite Element Methods

In comparison a finite element method uses a simple piecewise function that is valid on the element to describe local variations in \( \phi \). The governing equation is then satisfied precisely by the exact solution of \( \phi \). The exact solution is found though use of approximating functions, which cause residual errors to be produced. Iteration enables minimisation of these errors thereby resulting in an acceptable solution. The number of iterations required is dependent upon the necessary solution accuracy [Versteeg'95].

4.3.2.3 Spectral Methods

This approach approximates the unknown \( \phi \) variables through use of either a truncated Taylor series or a series of Chebyshev polynomials. The key advantage of this method is that the approximations made are valid throughout the fluid domain instead of just locally. Replacement of the unknowns with an algebraic expression and evaluation of the residual errors is again used to evaluate the required coefficients for the solution of the series [Versteeg'95].

4.3.3 The Finite Volume Method

There are advantages and disadvantages to each of the approaches described. Overall however, it is difficult to link directly between the algebraic expressions used and the flow properties they represent. The finite volume method is a special case of the finite difference method and was developed to allow a direct link between the equations used and the principle of conservation that governs fluid flow. The complete flow field under consideration, known as the domain is split into finite control volumes. The governing equations of fluid flow over all the finite control volumes within the domain are then integrated. Discretisation of the flow field is then achieved through substitution of finite difference type approximations to represent the key processes of convection and diffusion. The resultant equations are iterated to provide a solution. It is the initial integration of the control volumes that distinguishes this method from other techniques as this creates the direct link between the numerical algorithm and the principle of conservation. The conservation of any flow variable can then be considered to be a balance between various processes tending to increase or decrease it as shown in Equation 4-20.
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\[
\begin{pmatrix}
\text{Rate of change of} \\
\text{\(\Phi\) in the control} \\
\text{volume with} \\
\text{respect to time}
\end{pmatrix}
\begin{pmatrix}
\text{Net flux of} \Phi \text{ due} \\
\text{to convection in} \\
\text{control volume}
\end{pmatrix}
\begin{pmatrix}
\text{Net flux of} \Phi \text{ due} \\
\text{to diffusion in} \\
\text{control volume}
\end{pmatrix}
\begin{pmatrix}
\text{Net rate of} \\
\text{creation of} \Phi \\
\text{inside control} \\
\text{volume}
\end{pmatrix}
\]

Equation 4-20: Conservation of a Flow Variable

The various transport phenomena that are observed are then represented through discretisation techniques together with their rate of change with respect to time [Versteeg'95]. This is the method used by the commercial CFD package, Fluent®.

4.3.3.1 Turbulence Models

The various methods that may be applied to solve the governing equations for the flow variables have been discussed in the preceding sections. A turbulent flow will contain small eddies, which appear as small-scale fluctuations in time. In order to resolve these fluctuations the governing equations need to be solved using a time averaged approach. The presence of the fluctuations within the equations means that they are not closed, so a turbulence model is applied to model the fluctuations [Fluent'03].

Many different turbulence models exist and each uses a different approach, making each one suitable for different situations. Some turbulence models are designed for use with external flows and some for internal flows.

The size of the turbulent eddies within the flow are characterised by a turbulence length scale, \(l\). This is defined in Equation 4-21, where \(L\) is the diameter of the tube. In the case of a non-circular tube, this will be the hydraulic diameter, which is defined as the ratio of four times the cross-sectional area of the tube to the wetted perimeter of the cross-section.

\[l = 0.07L\]

Equation 4-21: Definition of the Turbulence Length Scale

4.3.3.2 The Standard \(k-\varepsilon\) Model

The simplest complete turbulence model is the standard \(k-\varepsilon\) model. It is a semi empirical model, based upon transport equations for the turbulence kinetic energy, \(k\) and the turbulence dissipation rate, \(\varepsilon\). The inlet value for the turbulence kinetic energy is calculated within Fluent® using Equation 4-22, where \(u_{avg}\) is the mean flow velocity and \(l\) is the turbulence intensity, defined as the ratio of the root-mean-square of the velocity fluctuations to \(u_{avg}\). The inlet turbulence dissipation rate is also calculated within Fluent® using Equation 4-23, where \(C_p\) is an empirical proportionality constant, with a value of 0.09 and \(l\) is the turbulence length scale.
4.3.3.3 The Realisable $k$-$\varepsilon$ Model

The realisable $k$-$\varepsilon$ model is a modification of the standard $k$-$\varepsilon$ model, which was developed to improve the mathematical representation of the turbulence within a flow. The term realisable relates to the normal stresses within the fluid, which must remain positive to satisfy the mathematical constraints. This is achieved by allowing $C_n$ to become a variable that is a function of the mean strain and rotation rates as well as the turbulence fields. The model has been validated extensively and was found to produce substantially better results than the standard $k$-$\varepsilon$ model for channel and boundary layer flows [Fluent'03].

4.3.4 Assessment of Solution Quality

The use of any computational process has associated risks in terms of quality, as the solution will always be a function of the inputs used. This is a particular consideration in terms of CFD as the definition of the flow environment, the class of the problem and the quality of the code used have a direct implication on the solution observed. The quality of a solution may be judged in a number of different ways. Three mathematical concepts are used to facilitate this, convergence, consistence and stability. Convergence is defined as the capability of a numerical method to approach the exact solution as the grid spacing used reduces to zero. Consistence is a measure of how well the numerical expressions represent the governing equations and stability is associated with the damping of errors as the numerical method proceeds.

Application of these principles to a CFD solution is rarely appropriate however, as round-off errors occur as the grid spacing reduces. Instead other rules based upon conservativeness, boundedness and transportiveness are used, which yield robust calculation schemes. Conservativeness dictates the global conservation of the flow variables across the entire domain, which is physically essential. Boundedness is similar to stability and requires that, for a problem without sources, the solution is bounded by minimum and maximum boundary conditions of the flow variable. Finally transportiveness accounts for the directionality of the influencing terms of diffusion and convection.
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The direct application of these properties has led to their acceptance as an alternative measure of solution quality. The iterative nature of the solution allows a measure of the overall conservation of the flow properties to be generated and monitored, called residuals. As the solution improves these reduce allowing the quality of the solution to be assessed. In addition the dependence of the solution on the grid used is assessed. If the same result is achieved with finer grid spacing the solution is found to be grid independent. A successful simulation result is therefore based upon the convergence of the iterative scheme applied and grid independence [Versteeg'95].

4.4 Scaling Effects

Equation 4-7 in the preceding discussion highlighted the strong dependence of the magnitude of thrust generated upon both the exit velocity and mass flow rate of the exhaust products. It is advantageous to utilise a small thruster to meet the range of low thrust requirements dictated by small satellites, as this allows performance to be maximised through minimising the overall system mass in addition to minimising losses. The potential effects of miniaturisation on a thruster will be considered and the sources of various predicted losses will be discussed. The effect of these losses on the thruster performance together with how they are caused and develop will be discussed in detail in later sections.

![Figure 4-3: Illustration of Scaling](image-url)

The miniaturisation of any object causes its physical dimensions to be scaled. For linear dimensions the scaling will occur at a preset rate, however for quadratic dimensions such as area and cubic dimensions such as volume this is no longer the case. The index of the dimension has a
direct impact upon its behaviour under scaling. The higher the index the faster the dimension will reduce. For example, consider the two cubes shown in Figure 4-3. The smaller cube has a characteristic dimension that is half that of the larger cube. From the surface area and volume data it is evident that the other dimensions do not scale in the same way. The surface area reduces by a factor of four and the volume by a factor of eight.

This is an example of the cube-square law, which indicates that as the order of the dimension increases the rate at which it will reduce under miniaturisation also increases. Understanding of this effect is vital when considering a propulsion system, as it will affect both the heat transfer and the fluid flow characteristics of the thruster.

### 4.5 Modification of Heat Transfer at Microscale

The heat transfer characteristics of a thruster are critical to the performance. This was demonstrated by Equation 4-16, which indicated that the exhaust velocity is directly proportional to the square root of the temperature in the decomposition chamber. The efficiency of the chemical reaction that enables the production of the hot gases is dependent of various factors, which include the geometry of the combustion chamber. In particular the length and volume of the chamber are important to ensure there is enough time for the reaction to completed before the gases are expelled. The cube-square law described in section 4.4 means that as the dimensions of the combustion chamber are reduced the associated volume reduces at a higher rate than the length. This could lead to insufficient volume being available within the chamber thereby preventing the gases expanding properly and leading to premature discharge of the exhaust gases. Consideration of this effect at the design stage will allow the appropriate volume to result, however the heat transfer characteristics also require attention.

![Figure 4-4: The Effect of Characteristic Dimension on Heat Transfer](image-url)
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The cube-square law indicates that a volume will rapidly reduce with miniaturisation, however the associated surface area reduces at a slower rate. The result is that there is less volume available to generate heat and more surface area available to transfer the heat away. To investigate this further a series of models were developed and evaluated using Fluent®. Figure 4-4 illustrates the scale of the models used, where the characteristic length is assumed to be the length of one side of the cube. The model was assumed to be a block of solid stainless steel 316, at the centre of which was located a heat source that maintained at a constant temperature of 873 K. The physical properties of the steel used are summarised in Table 4-1.

<table>
<thead>
<tr>
<th>Property</th>
<th>Value</th>
</tr>
</thead>
<tbody>
<tr>
<td>Density</td>
<td>7960 kgm⁻³</td>
</tr>
<tr>
<td>Specific Heat Capacity</td>
<td>500 Jkg⁻¹K⁻¹</td>
</tr>
<tr>
<td>Thermal Conductivity</td>
<td>16.3 Wm⁻¹K⁻¹</td>
</tr>
</tbody>
</table>

Table 4-1: Physical Properties of 316 Stainless Steel

<table>
<thead>
<tr>
<th>Characteristic Length</th>
<th>Volume</th>
<th>Surface Area</th>
<th>SA/Volume Ratio</th>
<th>Total Exterior Heat Transfer per Unit Surface Area</th>
</tr>
</thead>
<tbody>
<tr>
<td>20 mm</td>
<td>8 x 10⁻⁶ m³</td>
<td>2.4 x 10⁻³ m²</td>
<td>300</td>
<td>4.05 x 10⁶ Wm⁻²</td>
</tr>
<tr>
<td>10 mm</td>
<td>1 x 10⁻⁶ m³</td>
<td>6 x 10⁻⁴ m²</td>
<td>600</td>
<td>8.10 x 10⁶ Wm⁻²</td>
</tr>
<tr>
<td>5 mm</td>
<td>1.25 x 10⁻⁷ m³</td>
<td>1.5 x 10⁻⁴ m²</td>
<td>1200</td>
<td>1.61 x 10⁷ Wm⁻²</td>
</tr>
<tr>
<td>2.5 mm</td>
<td>1.56 x 10⁻⁸ m³</td>
<td>3.65 x 10⁻⁵ m²</td>
<td>2400</td>
<td>3.09 x 10⁸ Wm⁻²</td>
</tr>
</tbody>
</table>

Table 4-2: Heat Transfer Data

Table 4-2 summarises the results of a model created to investigate the miniaturisation of a heated cube. The emissivity, $\varepsilon$, of the exterior surface of the steel was assumed to be constant at 0.58. In reality this value will change with the temperature, however it was decided that as steady state operation was of interest a constant value was sufficient. The exterior temperature was assumed to be 4 K. Convection losses were neglected thereby simulating a space environment. From the tabulated results it is possible to see that the heat transfer through the surface increases by a factor of two as the characteristic length is halved. This is the same pattern that the surface area to volume ratio exhibits.

The implications of this become more pronounced when the heat transfer through the wall is also considered. In a conventional scale thruster the wall thickness of the chamber is negligible in comparison to the overall diameter. As the chamber reduces in size the wall thickness reaches a finite minimum dictated by physical constraints. At this point heat conduction through the wall then requires consideration, as it may no longer be considered to be insignificant.

4.5.1 Thermal Losses from a Steel Decomposition Chamber

When a decomposition chamber is considered the lengths are no longer all equal. The definition of a characteristic dimension becomes of importance at this stage to provide a measure for any miniaturisation that takes place. Either the diameter of the chamber or its length may be used as
the characteristic length. For a given reaction the propellant flow will be required to remain within the chamber for a minimum time for the reaction to be completed. This limits the minimum length of the chamber for a given propellant mass flow rate. The diameter of the chamber has no such limit associated with it, however it will influence the propellant mass flow rate, hence the chamber length. The decomposition chamber diameter is therefore selected as the characteristic dimension.

To examine the losses from a cylindrical chamber as the diameter reduced a new model was created to simulate the combustion chamber of a thruster also using Fluent®. Due to the symmetry present within a cylinder, a model utilising a quarter section of the chamber was generated to minimise computing time. A wire-frame sketch of the model geometry used is shown in Figure 4-5.

The outer section is the wall of the cylinder, assumed to be stainless steel and has a thickness maintained at 1.5 mm. The inner section is the open chamber, where the combustion would occur. For this analysis, where the steady state characteristics were of interest, it was appropriate to model this region as a solid section, which was maintained at a constant temperature of 873 K. A radiative boundary condition was applied to the exterior surface of the wall and symmetry conditions were applied to the symmetry lines. The emissivity, $\varepsilon$, of the exterior surface of the steel was assumed to be constant at 0.58 and the exterior temperature was assumed to be 4 K.

![Figure 4-5: Wire-Frame Schematic of Cylinder Geometry](image)

A series of models was created based upon this geometry, with the external diameter of the model varying from 6.5 mm to 26 mm. The steady-state solution of each model was considered and a typical conduction profile was observed. The range of models created allowed the effect of the constant wall thickness to be observed directly, as illustrated by Figure 4-6, where the total diameter of the model reduces from left to right. As the diameter of the model reduces, the wall thickness becomes larger relative to the internal diameter of the section and the temperature range observed increases. The exterior surface temperature is seen to reduce as the overall diameter reduces, indicating the wall is beginning to insulate the chamber.
Figure 4-6: Thermal Contours Through Wall Thickness

The rate at which the heat is leaving the surface of the cylinder was then considered. Figure 4-7 shows the heat flux per unit surface area as a function of external diameter. The data reveals that the heat flux across the exterior surface increases as the relative wall thickness increases. This confirms that while the external surface is at a lower temperature, the increase in the surface area to volume ratio results in additional heat loss.

Figure 4-7: External Surface Heat Flux per Unit Area with Diameter

4.5.2 Assessment of Other Materials

The results in section 4.5.1 indicate that heat transfer is a key factor when considering the miniaturisation of a decomposition chamber. The materials that the thruster itself is constructed from will have also a significant impact on the thermal characteristics. At conventional scales the maximum working temperatures of materials are considered to prevent failure in operation,
however heat loss is less of a concern. Here the thermal conductivity and heat capacity of the material are key material characteristics to consider. This is of particular importance when considering the use of MEMS technologies to miniaturise the system. The crystalline structure of silicon leads to a high thermal conductivity and in addition it has a high specific heat capacity, which allows it to store a large amount of energy within its bulk. The combination of these two effects means that the material will quickly draw heat away from the reaction, while taking a long time to reach an equilibrium point. In contrast a material such as steel has a significantly lower thermal conductivity and specific heat. This reduces the rate at which the combustion chamber accumulates energy and allows it to retain more of the energy released by the reaction.

To assess the impact of this, two additional materials were considered, silicon and Macor®. Macor® was selected as it is a machinable ceramic, which may be suitable for use either as the structural material for the decomposition chamber, or as an insulating material. For this analysis the emissivity of silicon was assumed to be 0.5 and the emissivity of Macor® to be 0.77. The physical properties of these materials are summarised in Table 4-3.

<table>
<thead>
<tr>
<th>Material</th>
<th>Macor®</th>
<th>Silicon</th>
</tr>
</thead>
<tbody>
<tr>
<td>Density</td>
<td>2520 kg/m^3</td>
<td>2330 kg/m^3</td>
</tr>
<tr>
<td>Specific Heat Capacity</td>
<td>790 J/kg°C</td>
<td>705 J/kg°C</td>
</tr>
<tr>
<td>Thermal Conductivity</td>
<td>1.46 W/m°C</td>
<td>148 W/m°C</td>
</tr>
</tbody>
</table>

Table 4-3: Physical Properties of Macor® and Silicon

The high thermal conductivity of silicon led to external surface temperature for each diameter modelled being higher than for both steel and Macor®. The total energy lost from the exterior surface of the model was predicted to be the lowest for the silicon cases as the low emissivity reduces the heat flux. The impact of different materials on the above analysis was considered and the results are summarised in Table 4-4 and plotted in Figure 4-8.

<table>
<thead>
<tr>
<th>Material</th>
<th>Steel</th>
<th>Macor®</th>
<th>Silicon</th>
</tr>
</thead>
<tbody>
<tr>
<td>Diameter (mm)</td>
<td>Tsurf (K)</td>
<td>Flux per Unit Surface Area (W/m^2)</td>
<td>Tsurf (K)</td>
</tr>
<tr>
<td>13</td>
<td>870.75</td>
<td>1.46E+06</td>
<td>847.21</td>
</tr>
<tr>
<td>10</td>
<td>870.36</td>
<td>1.92E+06</td>
<td>845.9</td>
</tr>
<tr>
<td>7</td>
<td>869.30</td>
<td>2.73E+06</td>
<td>842.87</td>
</tr>
<tr>
<td>5</td>
<td>866.60</td>
<td>3.77E+06</td>
<td>836.68</td>
</tr>
<tr>
<td>4.5</td>
<td>866.86</td>
<td>4.19E+06</td>
<td>834.87</td>
</tr>
<tr>
<td>4.25</td>
<td>866.43</td>
<td>4.43E+06</td>
<td>833.08</td>
</tr>
<tr>
<td>4</td>
<td>866.24</td>
<td>4.65E+06</td>
<td>828.07</td>
</tr>
<tr>
<td>3.75</td>
<td>863.35</td>
<td>4.97E+06</td>
<td>826.08</td>
</tr>
<tr>
<td>3.5</td>
<td>864.42</td>
<td>5.37E+06</td>
<td>822.71</td>
</tr>
</tbody>
</table>

Table 4-4: Data Summary of Heat Flux Models
Figure 4-8: External Heat Flux per Unit Surface Area Materials Comparison

The data shown indicates that the emissivity of the structural material used has a direct impact on the thermal losses incurred. This is to be expected, however the proximity of the curves indicates that the thermal conductivity and heat capacity of the materials also has a significant effect.

4.5.3 Conclusions

The thermal characteristics of a thruster are central to the resultant performance. The low mass flow rates of propellant used in a microthruster will result in less energy being released per second, therefore containment of the energy is critical. The high surface area to volume ratio of a microthruster exacerbates the thermal losses, reducing the overall performance. These models have demonstrated that any additional mass increases the thermal losses. The material to be used in the construction must have high structural integrity but be able to withstand the extreme temperature range to which it will be subjected and be capable of being machined accurately. Three different materials were investigated, stainless steel 316, Macor® and silicon. The models suggest that the high specific heat capacity of Macor® will increase thermal losses under steady state conditions. In addition the high thermal conductivity of silicon will lead to energy quickly being absorbed into the bulk material. Thus it is concluded that the chamber should be machined from a material such as stainless steel for ease of manufacture and strength, but a heat shield should be included in the design to reflect radiated heat back into the thruster.
4.6 Modification of Fluid Flow at Microscale

The efficiency with which the fluid passes through the thruster affects the performance of the thruster in terms of the resultant pressure drop and flow velocity. There are various fluid characteristics that affect this performance in addition to physical factors such as the size, shape and orientation of the pipes or chambers through which the fluid is passing. Increased interest in flows utilising small channels has led to significant research into fluid flows, both liquid and gas, within them. A review of the available literature on microflows was presented in chapter 2. This revealed that in certain circumstances external and internal forces might modify the fluid behaviour significantly and cause alterations to the predicted flow. It is therefore necessary to be aware of these effects and how they may cause the flow behaviour to change. Some of the key flow characteristics are now discussed with reference to the modifications predicted at microscale.

4.6.1 Flow Characterisation

The state of a fluid flow may be typified using the Reynolds number, $Re$. Equation 4-24 defines the Reynolds number, where $\rho$ is the fluid density; $u$, the fluid velocity in the axial-direction; $L$, a characteristic length, the diameter of the pipe for internal flows; and $\mu$ is the fluid viscosity.

$$Re = \frac{\rho u L}{\mu}$$

Equation 4-24: Definition of the Reynolds Number

This non-dimensional number, relates the inertial and viscous forces in a fluid and characterises whether a flow is likely to be laminar or turbulent. The critical Reynolds number, $Re_c$, is the point where a flow transitions from being laminar to become turbulent. In general for internal flows the critical Reynolds number is approximately 2300 [Schlichting’00]. The critical Reynolds number for microchannel flows has been the focus of research, which was summarised in chapter 2. Overall the literature is inconclusive, with different approaches finding different results. In general however, flows through microchannels are predominantly laminar as the Reynolds number is very low due to the small channel dimension. This means that an environment is created where the viscous forces within the fluid dominate the flow behaviour. The effect of this on the efficiency with which the fluid flows should therefore be considered.

4.6.2 Definition of Flow Regimes

It is useful to be able to describe a flow numerically in order to predict the behaviour that may occur. In order to do this effectively the correct functions should be used according to the flow regime. Four flow regimes exist to describe the behaviour of a fluid flow: continuum, slip flow,
transition and free-molecular flow. There are various characteristics that determine which regime is relevant, however the most frequently used is the Knudsen number, \( Kn \). The Knudsen number is defined as the ratio of the mean free path of the fluid, \( \lambda \), to the characteristic dimension of the channel, usually the height or diameter, \( l \), as shown in Equation 4-25. In addition the Knudsen number can be expressed in terms of the ratio of specific heats of the fluid, \( \gamma \), the local Mach number of the flow, \( M \) and the Reynolds number, \( Re \), is given. Figure 4-9 illustrates the location of the different regimes in relation to the Knudsen number, indicating a flow in the continuum regime has a Knudsen number less than 0.01.

\[
Kn = \frac{\lambda}{l} = \frac{\gamma M}{\sqrt{2} Re}
\]

Equation 4-25: Definition of the Knudsen Number

As the Knudsen number increases the continuum flow regime becomes invalid and entry into the slip regime is observed, followed by the transition regime then the free-molecular regime. For the flows investigated in the course of this research the Knudsen number remained in the continuum regime.

When a flow is in the continuum regime various flow properties can be assumed to be either constant, or vary slowly from one location to the next, allowing a direct prediction of the fluid behaviour. The key factor defining the continuum regime is that at the walls there is no slip between the flow and the stationary boundary, i.e. the fluid immediately adjacent to the wall is stationary. A flow in this regime can be predicted using the Navier-Stokes equations, which are introduced in section 4.3.1.

When the Knudsen number increases above 0.01, the no-slip boundary condition associated with the continuum regime is no longer valid and a slip boundary condition is required. The slip boundary condition allows the velocity of the flow immediately adjacent to the wall to be non-zero, thereby escaping discontinuities. The Navier-Stokes equations remain applicable with this modification up to a Knudsen number of approximately 0.1.

The transition flow regime encompasses flows for which the Navier-Stokes approximations are no longer appropriate. Kinetic gas theory is used to describe the flows, which allows the bulk properties of the fluid to be described based upon the assumption that the molecules travel in straight lines and exert no forces on one another except during a collision. Collisions are assumed
to be completely elastic and the walls of the container are assumed to be precisely smooth, resulting in no change in the tangential velocity during collisions with the walls. Accommodation coefficients then account for the inadequacies in these assumptions.

As the Knudsen number of the flow increases above 10 the kinetic gas theory is no longer applicable as all continuity assumptions break down and a statistical approach is required. The characteristic dimension of the channel associated with this kind of flow is now comparable with the mean free path of the fluid and few collisions occur. Flow in this regime may be likened to rarefied gas flow and complex numerical methods, such as the Direct Simulation Monte Carlo (DSMC) methods, are required to approximate to the flow. The DSMC method utilises a time step shorter than the mean collision time. This allows the molecular motion and intermolecular collisions to be split, thereby permitting the progress to be simulated in time and space.

The literature relating to the behaviour of fluid flow in microchannels was reviewed in chapter 2. Overall it was revealed that special consideration is required for channels when the characteristic dimension reduces below approximately 250 μm, although this is highly dependent upon the fluid and flow conditions under consideration.

### 4.6.3 Boundary Layer Development

The primary fluid property that affects the flow efficiency is viscosity. It is the viscosity of a fluid that causes the development of a boundary layer when a fluid is flowing past a stationary surface. The fluid immediately adjacent to the surface is assumed to be stationary, corresponding to the continuum no-slip boundary condition, $v = 0 \text{ ms}^{-1}$. The fluid velocity then increases with distance away from this surface, until the freestream velocity, $u_{\infty}$, is reached. These components are illustrated in Figure 4-10, where $u$ represents the flow velocity in the $x$-direction.

![Figure 4-10: Boundary Layer Schematic](image)

It may be assumed that all viscous effects are contained within this layer, the thickness of which, $\delta$, is dependent upon the shear stresses within the fluid, caused by the adjacent layers of fluid moving at slightly different velocities, in addition to the boundary layer state. It may be laminar or turbulent, dependent upon the flow velocity and surface roughness. The low flow rates associated with small scale flows lead to a laminar boundary layer, which is inherently thicker due
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to the transportation of regions of low velocity fluid without mixing. The effect of a boundary layer will be discussed in more detail in chapter 6, with particular reference to nozzle flows.

4.6.4 Surface Forces

Surface forces originate from the intermolecular forces within the fluid itself. At conventional scales these forces are negligible as they are mostly short range (< 1 nm), however they can lead to long range effects (> 0.1 μm) such as surface tension in a liquid. The sources and effects of surface forces have been investigated with reference to microfluidics [Ho'98, Adamson'97] and it is found that they start to dominate the flow observed, thereby dictating modifications in the equations used to describe the flow. Van der Waals forces and electrostatic forces are two of the most important types of force to be considered. Van der Waals forces are the weakest among all the forces, however they contribute to many phenomena including surface tension, wetting of surfaces and adhesion. For devices with very small channels, the adhesion of a fluid to the structure is a major complication as it can lead to a blockage of the flow.

Electrostatic forces affect charged molecules and particles and are significantly longer range than van der Waals forces. They may be used to control the movement of a fluid with charged particles, however they are frequently a problem when a residual surface charge remains, as particles will be attracted to the wall forming an immobile layer. This can cause the viscosity of the flow to appear higher than the nominal viscosity, as the immobile layer slows the flow.

4.6.5 Surface Roughness Effects

The different methods available for the fabrication of a micropropulsion system were considered in section 4.2. It is inevitable that the creation of a channel will result in a surface that is not perfectly smooth. The degree of surface roughness is dependent upon the process used, however the small dimensions of the channels lead to the roughness having more impact than at conventional scales. The friction factor may be used as a measure of surface roughness, the value of which depends on the flow state. The overall conclusions from the literature indicate that for small channels deviations from the flow predicted by standard theory should be expected. In particular it was noted that different materials had a significant impact on the flow rate observed.

4.7 Conclusions

This chapter has assessed the effect of miniaturisation on various characteristics that are important to the efficient operation of a thruster. Consideration of the different manufacturing techniques available has highlighted the benefits and drawbacks of each approach. For applications requiring high volume manufacture, MEMS processing techniques are the optimal solution. Here a single
thruster is under consideration, therefore micro conventional precision machining techniques are more appropriate.

The techniques associated with computational fluid dynamics were introduced to demonstrate the range of fluid flow situations that may be calculated using the Navier-Stokes equations. As the dimensions of the channel containing a fluid flow reduce, the flow regime will change and models based upon these equations will no longer be valid. Consideration of these limitations allows the correct modelling technique to be selected for the situation of interest.

Finally the physical effects of scaling were assessed in terms of heat transfer and fluid flow. Numerical modelling was used to demonstrate the change in thermal losses from a decomposition chamber as the internal diameter reduced but the wall thickness remained constant. The modification of a fluid flow at small scales was considered in terms of the development of a boundary layer and the effect of surface forces on a flow through a microchannel.
5 The Decomposition Chamber

This chapter is concerned with the effect of miniaturisation on the efficient decomposition of the chosen propellant, hydrogen peroxide. The particular aspects of chemistry relating to the catalysis of the reaction are discussed first followed by a review of the available literature relating to the design of a catalyst bed for a hydrogen peroxide thruster. The two catalyst bed arrangements developed during the course of this research are then introduced and discussed together with the experimental results obtained. The impact of miniaturisation on the performance of the decomposition chambers is then discussed.

5.1 The Chemistry of Catalysis

An engine generates thrust by releasing some of the chemical energy contained within a propellant and changing it into kinetic energy. A rocket engine is required to carry all propellant necessary with it, as an oxygen supply is not available. In the case of a chemical rocket engine, a reaction occurs using one, two or three propellants, which generates hot gases. The acceleration of these gases to generate thrust is the task of the exhaust nozzle, which will be discussed further in chapter 6. Here the creation of hot gases from the initial propellant is discussed.

A monopropellant rocket engine creates thrust by decomposing the single propellant into hot exhaust gases, via an exothermic reaction. The propellant stores energy in chemical bonds between the different elements, which form molecules within it. Chemical bonds may take a number of different forms, including covalent, ionic and metallic. The conversion of one substance into another requires chemical bonds to be both broken and created. Breaking bonds requires energy and making bonds releases energy.

The red trace in Figure 5-1 shows the reaction pathway as a function of energy for a generic exothermic chemical reaction. The energy of the reactants represents the energy stored within the reactants in the form of chemical bonds prior to the reaction. Initiation of a reaction requires an activation energy, $E_A$, to break the first bonds, represented by a peak in the sketch shown. A drop in the graph represents the energy released by the reaction and the energy remaining is the energy contained within the chemical bonds of the products of the reaction.

The energy released by the reaction characterised is as an exothermic reaction. The energy of the products is lower than that of the reactants, with the difference in energy being released as heat.
In the case of an endothermic reaction the energy of the products is higher than that of the reactants and heat is absorbed.

Figure 5-1: The Reaction Pathway

A chemical reaction may be accelerated through the use of a catalyst. A catalyst is defined as a substance that accelerates a chemical reaction without being consumed in the process. Introduction of a catalyst into a reaction allows the activation energy to be reduced, thereby accelerating the reaction. The green trace in Figure 5-1 illustrates the catalysed path with a reduced activation energy required to initiate the reaction.

Catalysis may be homogeneous or heterogeneous, dependent upon the phase of the catalyst compound in relation to the reactants [Atkins’97]. Homogeneous catalysis is when both the reactants and catalyst are in the same phase. Heterogeneous catalysis is when the reactants and catalyst are in different phases. In heterogeneous catalysis the catalyst is usually solid and the reactants either gaseous or liquid. The key advantage of using heterogeneous catalysis is that it is easier to contain, and therefore recycle, the catalyst material. The process of heterogeneous catalysis involves the reactants adsorbing onto the surface of the catalyst. This weakens the bonds within the reactants allowing the reaction to progress with a lower activation energy. Following the reaction the products are desorbed from the surface and exhausted. The reaction occurs on the surface of the catalyst therefore it is pertinent to optimise the available surface area. For this reason a solid catalyst is usually in the form of a fine powder or a section of highly porous foam.

5.1.1 Catalysis of Hydrogen Peroxide

Hydrogen peroxide (H₂O₂) was introduced in chapter 2 with reference to appropriate propellants for a monopropellant thruster. Hydrogen peroxide is a very similar compound to water (H₂O) and it possesses similar physical properties. It occurs naturally and is found in low concentrations in
streams and rivers, although it is an unstable compound and readily decomposes into water and oxygen. The decomposition reaction is exothermic and the temperature of the decomposition products is dependent upon the initial concentration of the hydrogen peroxide and the efficiency of the decomposition [McCormick'67].

The decomposition reaction of hydrogen peroxide may consist of up to twenty intermediate steps. Equation 5-1 represents the global reaction and illustrates that the exhaust products are water and oxygen.

\[
2\text{H}_2\text{O}_2 \rightarrow 2\text{H}_2\text{O} + \text{O}_2 + \text{heat}
\]

**Equation 5-1: Decomposition of Hydrogen Peroxide**

There are many known catalysts for the decomposition of hydrogen peroxide. Low concentrations of hydrogen peroxide may be catalysed using a small piece of liver or potato, which contain the enzyme, catalase. The decomposition reaction is rapid, but the heat released quickly causes the enzyme to denature and stop working. At high concentrations a catalyst with a longer lifetime is required and interest moves primarily to silver and manganese oxides. Historically silver, in the form of compressed gauzes, is the preferred catalyst for the decomposition of hydrogen peroxide in a conventional thruster. This is primarily due to availability and because the gauzes themselves are easy to handle thereby facilitating installation and compression within the decomposition chamber. Other possible forms of silver for use as a catalyst include silver powder and silver coated pellets. The primary alternative to the use of a silver catalyst is a manganese oxide catalyst, which will produce a faster reaction, enabling a rapid start up characteristic [Kappenstein'05, Eloirdi'01]. It may be used as either a solution or a solid catalyst supported on another surface. Use of this catalyst in solution form requires the design of injectors to spray the catalyst onto the propellant, which adds complexity to the system. It is difficult to create solid manganese oxide in any form other than a fine powder. For use as a catalyst it is therefore supported on a substrate, which provides the required structural strength.

### 5.2 Performance Characterisation

In order to compare the performance of a monopropellant thruster using different catalysts a characterisation parameter is needed. This should be a characteristic that is independent of the conditions downstream of the decomposition chamber. The characteristic used is the characteristic exhaust velocity, \( c^* \), which is defined in Equation 5-2 as a function of the stagnation pressure in the decomposition chamber, \( p_0 \), in Pa, the throat area, \( A^* \), in \( m^2 \) and the mass flow rate of propellant, \( \dot{m} \), in kg s\(^{-1}\). This definition is a function of parameters that may be measured through experiment.
Chapter 5: The Decomposition Chamber

Equation 5-2: Definition of the Characteristic Exhaust Velocity

Alternatively, a definition of $c^*$ that is based upon the propellant characteristics may be derived [Hill'92]. This is given in Equation 5-3, where $\gamma$ is the ratio of specific heats of the propellant, $R$ is the universal gas constant, $T_c$ is the temperature in the decomposition chamber and $M$ is the propellant molecular mass.

$$ c^* = \frac{\rho_c A^*}{\rho} $$

Equation 5-3: Alternative Definition of the Characteristic Exhaust Velocity

Both of the definitions given for $c^*$ demonstrate that it is a function of the propellant composition and the conditions within the decomposition chamber only. This makes it a suitable parameter for evaluating the efficiency of the catalyst bed. The design of a catalyst bed is now considered.

5.3 Recommendations for Catalyst Pack Design

The dimensions of the catalyst pack depend on various inputs, including the propellant concentration, operating pressure and allowable pressure drop. Standard operating conditions are assumed to be a chamber pressure of approximately 20 bar and temperature of 10 °C and above. An estimate of the required catalyst pack size may be made using empirical guidelines, which assume that the catalyst is in the form of silver gauzes [Davis'60].

The first stage is to make an estimate of the mass flow rate required for a given thrust level, using Equation 5-4 [Davis'60]. Here $\dot{m}$ is the mass flow rate of propellant, in kg/s, $T$ is the required thrust, in N, $I_{sp}$ the specific impulse of the propellant, in s, $\eta$ is the decomposition efficiency and $g$ is the acceleration due to gravity, in m/s$^2$.

$$ \dot{m} = \frac{T}{I_{sp} \eta g} $$

Equation 5-4: Mass Flow Rate Calculation

Based upon this equation, given that the ideal specific impulse of hydrogen peroxide is approximately 160 s, assuming an efficiency of 0.75 and a required thrust of 100 mN the mass flow rate is calculated to be $8.495 \times 10^{-5}$ kg/s.

The next characteristic to consider is the required cross-sectional area of the catalyst pack. This is related to the mass flow rate through the catalyst pack via an empirical parameter called the
loading factor, $L_f$. The loading factor gives an indication as to the mass flow rate of propellant that a given cross-sectional area of catalyst is capable of decomposing. The literature suggests that for “small diameter” packs a loading factor of $59 - 117 \text{ kg} \cdot \text{m}^{-2}$ should be used [Davis'60], although no indication is given of the bounding diameters to which this applies. The diameter of the catalyst pack is then calculated using Equation 5-5 [Davis'60], where $A_r$ is the cross-sectional area of the pack, in m$^2$ and $L_f$ is the loading factor. Based upon the calculated propellant flow rate applying a $L_f$ of $59 \text{ kg} \cdot \text{m}^{-2}$ gives a required pack diameter of 1.35 mm, while a $L_f$ of $117 \text{ kg} \cdot \text{m}^{-2}$ gives a required pack diameter of 0.96 mm.

$$A_r = \frac{n \pi}{L_f}$$

Equation 5-5: Cross-Sectional Area of the Catalyst Calculation

Finally the length of the catalyst pack is considered. From previous studies it is known that while the length of the catalyst bed has little effect on the start up characteristics, it has a direct influence on extent of decomposition observed for a given propellant flow rate [Willis'60]. For a given diameter of catalyst bed, a longer length will enable a higher flow rate of propellant to be completely decomposed. This implies that for a given mass flow rate there is an optimum catalyst bed length. A minimum length for the catalyst bed is suggested to be 31.75 mm to prevent incomplete decomposition, although the minimum diameter to which this relates is unknown [McCormick'67]. Other research has demonstrated that a catalyst pack length of 20 mm was optimal for a 12.7 mm chamber diameter, when filled with 100 compressed silver gauzes [Coxhill'02]. This suggests that the length of the catalyst bed should be investigated to determine the optimum size for a given diameter.

In summary this analysis has suggested that for a nominal thrust level of 100 mN a propellant mass flow rate of $8.495 \times 10^{-5} \text{ kg} \cdot \text{s}^{-1}$ is necessary. In addition, to decompose this flow rate of propellant a catalyst pack of diameter 0.96 – 1.35 mm and length 31.75 mm is required.

### 5.4 Assessment of Recommendations

The dimensions suggested by the design process outlined in section 5.3, result in a catalyst pack that is very long relative to its small diameter. There are three main points that would require attention if a pack were to be developed based upon these figures, which will now be discussed.

#### 5.4.1 Handling and Alignment Issues

The analysis above was based upon the assumption that silver gauzes were being used for the catalyst. For the thrust level of 100 mN considered, the recommended diameter of the gauzes was
0.96 – 1.35 mm dependent upon loading factor. As discussed earlier in section 5.1.1, silver gauzes are the preferred solution for conventional hydrogen peroxide thrusters as they are easy to handle. In these cases the diameter of the gauzes is of the order of centimetres. With the dimensions under discussion, the significantly reduced diameter of the gauzes required makes handling of them prohibitive. The gauge of wire and type of weave used for the gauzes would require careful selection to ensure structural integrity of the individual gauzes without excessive pressure drop once compressed. Further, the length of the catalyst bed required leads to potential alignment problems of the gauzes within the chamber. As the discs are inserted into the chamber it is possible that some will rotate prior to compression, leading to regions of varying density.

5.4.2 Thermal Characteristics

The thermal characteristics of a decomposition chamber were discussed in chapter 4. The conclusions from the analysis performed were that as the surface area to volume ratio of a decomposition chamber increases, the thermal losses from the chamber increase. The surface area to volume ratio of a long thin decomposition chamber would be very high and would therefore exacerbate thermal losses. The application of insulation to a decomposition chamber was also considered in chapter 4 and the conclusion was that any additional mass in the chamber structure would also add to thermal losses. From these considerations it is clear that a shorter pack with a larger diameter would improve the thermal characteristics of the decomposition chamber. In addition a heat shield should be considered in the design to minimise radiative thermal losses.

5.4.3 Physical Accommodation of Length

The analysis in section 5.3 indicated that a minimum length of the catalyst bed should be 31.75 mm. When the catalyst bed is integrated into a thruster assembly, the complete length would be significantly longer than this initial value. The accommodation of this length onboard the nanosatellite platform selected would be possible, but complex and would reduce the range alignment positions possible. From this perspective a shorter catalyst bed would be preferred, as the possibility of accommodating more than one thruster then arises.

5.4.4 Conclusions from Assessment of Catalyst Pack Sizing

The three preceding sections have discussed key aspects relating to the proposed sizing of the catalyst pack for a 100 mN thruster. From this discussion it would appear that the design guidelines considered are not appropriate for the sizing of the catalyst pack in this thruster. This analysis in combination with the literature review reveals that the physical property appearing to dominate the efficiency of the reaction is the surface area of the catalyst available. To address
this, two different avenues of research were pursued with respect to the catalyst structure: a monolithic catalyst bed and a compressed powder bed.

The overall aim of this research is to miniaturise the propulsion system to produce low magnitude thrust within a small system. Evaluation of the available literature has indicated that a reduced diameter catalyst pack will be required to accommodate the low mass flow rates associated with this objective. The aim of the remainder of this section is to develop an understanding of how the required geometry and morphology of the catalyst are modified as the magnitude of thrust decreases. To investigate the validity of the dimensions suggested by the empirical analysis, a larger catalyst pack diameter was selected for initial characterisation. The nominal diameter of the initial catalyst bed investigated was set to be 9.0 mm for both types of catalyst bed. Further reductions in the diameter of the compressed powder catalyst bed were also achieved. The following sections outline the approaches used in each case together with the results obtained.

5.5 Development of a Monolithic Catalyst Bed

Use of a monolithic catalyst bed allows scope for the catalyst material to be supported, therefore facilitating the use of a catalyst such as manganese oxides. This section will first evaluate the different monolithic supports available before summarising the steps taken to develop and test a monolithic catalyst bed based upon a manganese oxide compound.

5.5.1 Substrate Selection

A substrate for a monolithic catalyst should possess a number of qualities:

- High Specific Surface Area
- High Structural Integrity
- Low Pressure Drop

Each of these qualities is equally important. For example, if the specific surface area is high, but the structural integrity is low then the lifetime of the catalyst will be limited. In this scenario a lower specific surface area may be acceptable if this enables an increase in the structural integrity. There are various media that may provide a useful solution. The supporting structure may be created in a method similar to that used to form a foam or artificially machined. Using MEMS manufacturing techniques it would be possible to develop an artificially porous structure. An example of this is shown in Figure 5-2, where the honeycomb structure on the left of the image is designed to form the substrate for a catalyst bed. The catalyst is deposited on the internal walls of each cell. The disadvantage of this approach is the bed length limitation of approximately 3 mm due to problems relating to bonding wafers. The structure is strong and the pressure drop will be
low, but the total specific surface area will also be low due to the short length. The Ångstrom Space Technology Centre, Sweden developed these wafers as part of the collaborative work under European Space Agency contract no. 17091/03/NL/Sfe.

![Etched Silicon Wafer](image)

A similar approach applied to a different substrate material allows this length limit to be removed. Researchers at the University of Poitiers, are developing a catalyst bed based upon a mullite block, illustrated in Figure 5-3 [Kappenstein'05]. The block contains several 1 mm width channels, the inside wall of which are coated with catalyst.

![Mullite Monolith](image)

Figure 5-3: Mullite Monolith [Kappenstein'05]

The geometry of the channels results in a low specific surface area. To improve this, several layers of catalyst are deposited to create a rough surface and maximise the number of possible active sites for decomposition. The regular structure of the block leads to a high strength, but this also increases the possibility of viscous losses leading to an increased pressure drop.

A further option is to utilise a ceramic foam monolith. Various methods may be used to create a ceramic foam, with the most common creating a positive image of a host structure. A piece of polyurethane foam is saturated with a ceramic slurry, which is then solidified before the host
structure is burnt away. The result is a reticulated structure that has large pores connected through windows and separated by columns of ceramic [Richardson‘00]. An example of a section of foam created using this method is shown in Figure 5-4, which illustrates significant variation in both pore sizes and the low structural density. The diameters of the pores in this image are in the range 0.04 – 1.5 mm.

![Reticulated Ceramic Foam](image)

**Figure 5-4: Reticulated Ceramic Foam [Richardson‘00]**

A foam manufactured in this way is suitable for applications where the stresses on the structure are low, for example in filters. The poor structural integrity of this type of foam makes it unsuitable for use as a catalyst substrate in the high pressure environment involved. In principle a ceramic is an ideal material for a catalyst substrate as it can withstand the high temperature and the oxidising environment better than most other materials. The success of a ceramic foam as a catalyst substrate is dependent upon the structural integrity of the material. The fraction of solid material present within a given volume of foam directly influences the resultant structural integrity. The high pressure present within a decomposition chamber leads to the need for a high density foam for this application.

The percentage of solid material present within a section of foam i.e. the foam density, influences both the structural integrity and the specific surface area. It is the manufacturing process that limits the maximum foam density achieved. The Dytech Corporation Ltd uses an alternative production technique to that described above and foams with a significantly higher density result [SmithR ‘04]. The density of the foam and average pore size can be selected during manufacture thereby allowing an optimal combination of properties for the desired application to be used. The manufacture of the foam is conducted in six stages:

i. **Form a Dispersion**

   Ceramic powder is mixed with a liquid carrier and a polymerisable monomeric material.

ii. **Agitate to Introduce Gas**

   A surfactant is added to the mixture, which is then agitated to create bubbles of gas.

5-9
iii.  Allow to Polymerise

Once the required density is achieved an initiator is added to allow the monomer to polymerise within a mould.

iv.  Dry

The foam is removed from the mould and dried slowly at first at room temperature before being force dried in an oven.

v.  Shape

At this stage the ceramic is called “green” ceramic and it is easily machined into the desired shape.

vi.  Fire in a furnace

The second drying phase in the furnace ensures the bubbles formed in the foam microstructure are solidified, resulting in the creation of a structure that contains pores with solid walls, instead of adjoining columns.

Windows within the pore walls cause the structure to be open-celled, while maintaining the superior strength and density. An example of the resulting high density structure of the foam is shown in Figure 5-5, where the foam density is 15% and the average pore diameter is 150 μm.

![Figure 5-5: SEM Image of a 15% Density Foam](image.jpg)

The image was taken using a Scanning Electron Microscope and illustrates the open cell nature of the foam through the multitude of windows in the cell walls. The additional material present within this structure is immediately obvious in contrast to that shown in Figure 5-4. It is this added material in particular that increases the structural integrity of the material and raises the specific surface area.

This material was chosen as the substrate for the monolithic catalyst due to the favourable combination of properties it contains. Several different densities of this type of foam were
considered as substrates for the catalyst to allow an assessment of the trade-off between structural integrity and specific surface area. Figure 5-6 illustrates the visual differences in the range of foams used, where from left to right the foam densities are 10%, 15%, 20%, 23%, 27% respectively.

![Figure 5-6: Range of Foam Density](image)

**5.5.2 Catalyst Selection and Preparation**

The primary materials suitable for use as a catalyst for hydrogen peroxide in a monopropellant thruster were considered in section 5.1.1. It was found that silver and manganese oxides were the most suitable catalysts for this application. The low viscosity and high activity of a manganese oxide solution made it a favourable option for deposition within the porous substrate.

Various methods of preparation were investigated to ensure an even and thorough coating could be deposited throughout. The most successful method consisted of seven stages:

i. **Clean Substrate**
   
   Loose particulate was removed in an ultrasonic bath.

ii. **Dry Substrate**
   
   The foam substrate was dried using a drying cabinet.

iii. **Primary Coat of Potassium Permanganate**
   
   Deposition of the catalyst on the foam was achieved by suspending one end of the foam cylinder and submerging in a saturated potassium permanganate solution. Capillary action drew the solution into the pores. The speed of this process is illustrated in Figure 5-7, where the time is in seconds. In this case a section of 20% density foam, 9 mm in diameter and 30 mm in length is being coated. For lower density foams the rate of deposition increases.

iv. **Dry Potassium Permanganate onto Substrate Surface**
   
   The foam was dried in a controlled environment at 4 °C.

v. **Secondary Coat of Potassium Permanganate**
vi. **Dry Potassium Permanganate onto Substrate Surface**

vii. **Heat Substrate to 900 °C at a rate of 3 °C per minute**

Heating to 900 °C is necessary to decompose the water-soluble potassium permanganate solution into insoluble manganese oxides and soluble potassium deposits, which were then removed.

![Saturation of an Alumina Foam with Potassium Permanganate Solution](image)

**Figure 5-7: Saturation of an Alumina Foam with Potassium Permanganate Solution**

The critical stage of the process was found to be step four, drying the catalyst onto the surface of the substrate. The rate at which the catalyst material dried was found to significantly affect the quality of catalyst deposition. Figure 5-8 shows two different ceramic foams of the same density that were dried in different conditions. The foam in the left image was dried at room temperature, while the foam in the right image was dried in a temperature controlled environment at 4 °C.

![Comparison of Manganese Oxide Deposition at 100x Magnification](image)

**Figure 5-8: Comparison of Manganese Oxide Deposition at 100x Magnification**

The images were taken using an optical microscope combined with a digital camera eyepiece. The thin dark band around the edges of the foam in the left hand image is coupled with a very light central region, indicating that most of the catalyst has deposited at the edge of the foam with almost none remaining in the interior. Little variation in colour is observed in the right hand image, indicating an even covering of catalyst. It is thought that the catalyst solution was drawn back towards the surface of the foam during the drying process at room temperature as a result of capillary forces. Drying at the colder temperature where the solution remains more viscous throughout the drying phase appears to eliminate this effect. A sample of the manganese oxide
deposit was analysed using X-Ray Diffraction and the results indicate that the compound was predominantly Mn$_3$O$_4$.

An alternative deposition method that was tried utilised flash vaporisation to dry the catalyst faster than it could draw out. This appeared to achieve reasonable results, however the deposition remained uneven throughout the catalyst as shown by Figure 5-9, where the two ends of the block exhibit a significantly lighter colour indicating a lack of catalyst material. The squares are 5 mm across for scale.

![Figure 5-9: Flash Vaporisation Deposition](image)

Initially three different densities of foam (10%, 15%, 20%) were prepared for testing using the first deposition method described. Following initial tests higher density foams were also prepared due to structural issues, which will be discussed later.

### 5.5.3 Catalyst Testing Preparation

The next phase was to prepare the catalyst coated cylinders for testing with hydrogen peroxide by securing them within a suitable chamber, formed of a section of stainless steel tube. A layer of magnesium oxide based ceramic adhesive (Aremco Ceramabond™ 571) was deposited onto the surface of the foam, to ensure that the exterior pores were sealed. Before the adhesive dried completely it was moulded using a pre-cut section of steel tube to ensure it would fit into the required section of tube. The adhesive coated foam was then left to air-dry for an hour before it was placed in the furnace at 93 °C for two hours to cure. The second stage was to bond the sealed foam into a sleeve using the additional ceramic adhesive. Figure 5-10 shows an axial cross-section of a finished test section. For scale the squares are 5 mm.

![Figure 5-10: Section of Catalyst Bonded into Sleeve](image)

It was essential that the bond between the sleeve and the catalyst was sufficient to prevent any propellant from escaping the catalyst and to prevent the catalyst from moving within the casing.
Before the test pieces were exposed to hydrogen peroxide, pressure drop tests were performed on each piece to evaluate the bond. The mass flow rate of nitrogen, at a given pressure, through each of the test pieces was monitored to determine the quality of the bond. The objective of this was to establish that the bond was sufficient by a good correlation of results between different samples for a given density. This also ensured that any test pieces with poor bonds between the foam and the sleeve were quickly identified, as a result of the unusually low pressure drops observed. An example of the testing data for 20% density foam is shown in Figure 5-11. In this example the data from four different catalysts assemblies is presented. The first two digits indicate the density of the foam and the remainder of the characters in the designation identifies the particular catalyst batch and number. It is clear that three test pieces have bonded successfully and one has not, which results in the higher mass flow rate. The error bars placed on to the plot account for the errors associated with the mass flow rate and pressure data, 5% error for the pressure transducer and 2% for the mass measurement.

A linear trendline was created for each data series and it fell within the error bars in all cases. All of the test pieces that passed the pressure drop test were then ready to be tested with hydrogen peroxide.

5.6 Development of a Compressed Powder Catalyst Bed

The key aspect of the monolithic catalyst bed development was the deposition of the catalyst material onto the substrate. In contrast the key aspect of the compressed powder catalyst bed
development was the design of a chamber suitable to hold the correct volume of catalyst material. This section will first discuss the different materials that were available for use as the catalyst material, then outline the design used to test the performance of the material selected.

5.6.1 Catalyst Material Selection

The monolithic catalyst bed used manganese oxides, which are difficult to generate as an unsupported solid as discussed previously. For a compressed powder bed the option of using ceramic beads to support a catalyst was available but the literature revealed only limited success with this approach [Eloirdi'01]. The alternative was to use a coarse powder, where the size of the particles was selected such that the surface area was sufficient to maintain decomposition without the pressure drop becoming prohibitive. It was decided that the most practical method of achieving this was to use a powder of solid silver. The size of the particles used was selected based upon the pore size of the ceramic foam used previously. Two different grades of powder were selected, one coarser than the other to enable an investigation as to the influence of surface area and pressure drop at this scale. The particle size range of the coarser powder was in the range 680 - 1600 μm, while the finer powder had a particle size of 150 - 200 μm.

5.6.2 Design of a Catalyst Test Chamber

The key consideration in the design of the chamber to contain the silver powder was the flow path of the propellant. It was essential that the propellant would be forced through the catalyst itself and not bypass around it. To achieve this the powder should be under compression thereby presenting a fixed impediment for the propellant to negotiate.

Figure 5-12 shows a sketch of the final configuration of the design. It consists of three key sections, the chamber itself, a plunger section and a compression device. The chamber has a nominal internal diameter of 9.7 mm, which is comparable with the diameter of the monolithic catalysts used. The plunger section accommodates two o-rings, 7 mm internal diameter and 1 mm cross-section. These create a seal between it and the chamber wall.

The length of the various components was set to allow a variation in the catalyst pack length from 0 - 40 mm to be investigated. This placed some constraints on the materials that were suitable for the manufacture of the chamber due to the depth of the bores required. Two holes were drilled in the wall of the chamber downstream of the catalyst. This allowed a short section of tube to be located in each, enabling a pressure and temperature measurement to be taken. At the end of the chamber a hole was created and threaded with an M5 fine thread. The throat was created using a brass insert, which was screwed into the thread created and contained a 1 mm diameter hole. The easy removal of this component facilitated the unloading of the catalyst after firing.
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Figure 5-12: CAD Model of Powder Test Chamber

A pleated stainless steel filter disc, manufactured by Microfiltrex was located either end of the catalyst material to contain it and enable compression. The mesh size chosen for the filter was 60 μm to ensure no loss of powder with minimal pressure drop.

To reduce the diameter of the catalyst within the chamber a series of collars was created to fit inside the chamber. Two sets of collars were used to create two additional chambers. The wall thickness of the first collars was 1 mm, causing a reduction in diameter of 2 mm to 7.7 mm. The wall thickness of the second collars was 1.5 mm, causing a reduction in diameter to 6.7 mm. The filter discs were located either end of the collar to maintain the flow path through the catalyst.

5.6.3 Catalyst Testing Preparation

The pressure drop through a 9.7 mm diameter catalyst pack was evaluated for a given length of powder using a set up similar to that described previously for the monolithic chamber. The silver powder was weighed into discrete amounts to enable evaluation of the resultant pack length and density. The two different grades of powder used corresponded to two different pack densities. Different lengths of pack were then investigated to determine the change in the pressure drop observed. The results from the tests are shown in Figure 5-13 and Figure 5-14.

The lengths of the catalyst beds that resulted from the masses of powder selected varied, as the packing density assumed initially was too high. The shape of the powder grains prevented a close packing density, hence the lengths of catalyst bed that resulted were longer than predicted.
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Pressure Drop Tests - Coarse Silver Powder

Figure 5-13: Pressure Drop Test Data for the Coarse Powder

Pressure Drop Tests - Fine Silver Powder

Figure 5-14: Pressure Drop Test Data for the Fine Powder

Prior to testing the length of each catalyst pack was measured using Vernier Calipers to validate the estimated packing density, which is defined as the ratio of the volume of silver present to the total volume available. The shorter lengths of the fine powder packs indicated a higher packing density and increasing the pack length of the fine powder resulted in an increase in the pressure drop as expected.
There is little difference in the data collected for the coarse powder for the different lengths of pack investigated. This indicates that the pressure drop through the pack is very low and is consistent with the voids between the powder particles being relatively large.

This data allowed the approximate packing density of each silver powder to be calculated. For the coarse powder this was estimated to be 0.41 and for the fine powder 0.47. From these figures the mass of powder required for use in each of the chambers was calculated. The data corresponding to the different lengths of catalyst tested in the 9.7 mm, 7.7 mm and 6.7 mm diameter chambers is given in Table 5-1, Table 5-2 and Table 5-3 respectively.

<table>
<thead>
<tr>
<th>Length</th>
<th>Fine Powder</th>
<th>Coarse Powder</th>
</tr>
</thead>
<tbody>
<tr>
<td>5.0 mm</td>
<td>1.825 g</td>
<td>1.590 g</td>
</tr>
<tr>
<td>7.5 mm</td>
<td>2.738 g</td>
<td>2.385 g</td>
</tr>
<tr>
<td>10.0 mm</td>
<td>3.650 g</td>
<td>3.180 g</td>
</tr>
<tr>
<td>12.5 mm</td>
<td>4.563 g</td>
<td>3.975 g</td>
</tr>
</tbody>
</table>

Table 5-1: Mass of Catalyst Required for a Given Length in the 9.7 mm Diameter Chamber

<table>
<thead>
<tr>
<th>Length</th>
<th>Fine Powder</th>
<th>Coarse Powder</th>
</tr>
</thead>
<tbody>
<tr>
<td>5.0 mm</td>
<td>1.140 g</td>
<td>0.995 g</td>
</tr>
<tr>
<td>7.5 mm</td>
<td>1.710 g</td>
<td>1.493 g</td>
</tr>
<tr>
<td>10.0 mm</td>
<td>2.280 g</td>
<td>1.990 g</td>
</tr>
</tbody>
</table>

Table 5-2: Mass of Catalyst Required for a Given Length in the 7.7 mm Diameter Chamber

<table>
<thead>
<tr>
<th>Length</th>
<th>Fine Powder</th>
<th>Coarse Powder</th>
</tr>
</thead>
<tbody>
<tr>
<td>5.0 mm</td>
<td>0.861 g</td>
<td>0.751 g</td>
</tr>
<tr>
<td>7.5 mm</td>
<td>1.292 g</td>
<td>1.127 g</td>
</tr>
<tr>
<td>10.0 mm</td>
<td>1.723 g</td>
<td>1.503 g</td>
</tr>
</tbody>
</table>

Table 5-3: Mass of Catalyst Required for a Given Length in the 6.7 mm Diameter Chamber

5.7 Testing Set-up

All testing with hydrogen peroxide was conducted at the SSTL propulsion test site at the Westcott Venture Park, Westcott, Bucks. The handling procedures outlined in the “Hydrogen Peroxide Handbook” were followed at all times and full protective clothing was worn when in proximity to the propellant [Constantine'67].

The test-rig used to characterise the performance of both catalysts was based upon a set-up used in previous work [Coxhill'02]. It consisted of three sections; the propellant feed system, the catalyst test bed and the data acquisition set up. A general schematic of the layout of the propellant feed system and catalyst test bed is shown in Figure 5-15. Not all instrumentation was present for the tests with the monolithic catalyst. The presence of a variable choke allowed the propellant mass flow rate to be adjusted to optimise the decomposition reaction.
5.7.1 Monolithic Catalyst Test Set-up

The ½ inch steel tubes containing the catalyst were attached to the ¼ inch propellant feed system pipes using Swagelok® reducing union fittings, as shown in Figure 5-16. This allowed the test piece to be changed easily, while maintaining a good seal. The image also shows the location of the thermocouple downstream of the catalyst and the 1mm diameter choke used.

The performance of a monopropellant thruster may be characterised by the extent of decomposition achieved and the smoothness of the reaction. Initially a single temperature measurement was recorded then, as the behaviour of the catalyst became better understood, mass flow data was also recorded.
5.7.2 Compressed Powder Catalyst Test Set-up

The design of the test chamber used for the compressed powder catalyst tests was discussed in section 5.6.2 and is shown in Figure 5-17. This was also connected to the propellant feed system through use of Swagelok® fittings.

![Figure 5-17: Compressed Powder Test Set-up](image)

The two short sections of tube located in the side of the end section of the chamber were used to locate instrumentation. One contained a thermocouple and the other ended in a fitting allowing a pressure transducer to be located at the end of it. The fixed length of the tube between the pressure transducer and the flow allowed the small error in the data recorded to remain constant. The thermocouple was brazed into position with the tip located in the centre of the exhaust flow.

The catalyst was loaded into the chamber immediately prior to testing. A filter was inserted into the chamber first, followed by the silver powder. The second filter disc was carefully inserted to ensure it remained normal to the flow direction. The O-rings were moistened using a little distilled water and the plunger section was inserted into the chamber. Finally the compression section was screwed into place.

5.7.3 Instrumentation

The instrumentation system used to record data was originally developed for previous work [Coxhill'02]. The output signals from the sensors used were converted from a current to a voltage via a data acquisition card (DAC), manufactured by National Instruments and logged by a computer. Each signal was registered to a particular channel of the sixteen available. The current from the sensors was in the range 4 – 20 mA, which was converted to a voltage in the range 0.88 – 4.4 V through use of a 220 Ω resistor. This voltage was then fitted to the 0 – 5 V range of the card and detected by an analogue to digital converter (ADC) located within the computer. The
software used to monitor the card was the National Instruments LabVIEW® system and both the calibration and acquisition programs written for the previous research were used for this research.

The flow rate of the propellant was initially measured using a Micro Motion® model D coriolis flow meter, which utilises coriolis forces to measure the flow rate. The fluid passes through a vibrating pipe, generating coriolis forces, causing a phase shift in the sensor signals that are proportional to the flow rate. The key advantage of this technique was that the flow measurement is entirely independent of the pressure, viscosity, density and temperature of the fluid. The coriolis flow meter used was accurate to within 1% of the scale reading for a flow rate as low as approximately 0.8 gs⁻¹ and to 13% of the scale reading for a flow rate of approximately 0.4 gs⁻¹.

As the propellant flow rate reduced below this a different set up was required to monitor the flow rate. A propellant tank was placed onto a set of digital scales that was linked directly to the computer. This logged the mass reading at a rate of 1 Hz. Plotting this data with respect to time allowed the average mass flow rate to be calculated. When testing the set up a steady mass flow rate of 0.07 gs⁻¹ was achieved.

The temperature of the exhaust products was measured using an insulated K-type thermocouple placed downstream of the catalyst. This type of thermocouple will operate up to a temperature of 1000 °C, so was ideal for placement in the exhaust flow, where the maximum temperature of the decomposition products was predicted to be approximately 600 °C. The output signal from the thermocouple was amplified using a temperature transmitter amplifier. These amplifiers are cold junction compensated such that when a 24 V supply is supplied they output a signal in the range 4 – 20 mA. Calibration of the output current was accomplished using a thermocouple simulator.

The static pressure was measured using pressure transducers, manufactured by GEMS Ltd®. The transducers contain strain gauges, which are bonded to a diaphragm. Application of pressure moves the diaphragm introducing a strain in the gauges. This is measured as an electrical resistance, the magnitude of which is proportional to the pressure present. Two pressure transducers were used in these tests, one rated for operation between 0 – 6 bar and the other for operation in the range 0 – 25 bar. Both transducers used were calibrated in the propulsion laboratory on campus using the commercially calibrated pressure panel.

5.8 Testing Results

This section will review the results generated through the testing of both types of catalyst. The variations investigated and trends observed will be discussed.
5.8.1 Monolithic Catalyst Bed Results

Several samples of each density of catalyst were tested using the set up described above. The data recorded were repeatable and generated temperatures in excess of 500 °C. The initial tests were conducted using the lower density foams, which had the highest internal surface area. An example of the temperature data recorded for the monolithic catalyst bed is shown in Figure 5-18, for a 15% density foam.

![Temperature Profile for 15% Density Foam Catalyst Bed](image)

Figure 5-18: Temperature Profile for a 15% Density Foam

The initial temperature rise to 300 °C took place in two stages with a small plateau appearing at approximately 175 °C. This is thought to correspond to the boiling temperature of hydrogen peroxide and appears on all temperature traces when the initial temperature is below approximately 150 °C. The temperature rise from 300 °C to 500 °C occurred at a significantly slower rate, which is thought to be due to the thermal characteristics of the setup as no insulation was present.

The shape of this temperature profile implies that the flow through the catalyst is not smooth. The rough shape of the profile indicates that the decomposition process accelerates and slows in quick succession. The inlet conditions to the catalyst were varied in an attempt to reduce the severity of these oscillations however little change was observed. The coriolis mass flow meter was added to the setup to provide additional information about the flow environment. Data was recorded for test runs with 20% density foam catalysts, an example of which is shown in Figure 5-19, where the temperature trace is shown in blue and the mass flow data in red. In general an initial peak in the mass flow rate data is expected as the void upstream of the catalyst is filled, following this
peak the flow should become smooth. This is not the case for the data shown, where an oscillatory pattern is evident throughout indicating unsteady flow within the thruster.

Inspection of the data reveals a link between the time period of the oscillations in both traces. One cycle for the mass flow rate data includes a sharp peak followed by a flat section. The temperature data exhibits a saw-tooth profile but maintains a steady climb overall. There are many factors that may contribute to this instability, including the energetic nature of the decomposition reaction and the void immediately upstream of the catalyst.

Inspection of the data reveals a link between the time period of the oscillations in both traces. One cycle for the mass flow rate data includes a sharp peak followed by a flat section. The temperature data exhibits a saw-tooth profile but maintains a steady climb overall. There are many factors that may contribute to this instability, including the energetic nature of the decomposition reaction and the void immediately upstream of the catalyst.

![Temperature and Mass Flow Rate Data for a 20% Density Foam Catalyst Bed](image)

**Figure 5-19: Temperature & Mass Flow Rate Data for a 20% Density Foam**

Previous work with hydrogen peroxide highlighted the need to minimise the void volume immediately upstream of the catalyst pack [McCormick'67]. The flat sections in the mass flow rate data correspond to reversed flow. The mass flow meter used can not register this, but movement of the flow pipes themselves indicated movement of the propellant. The shape of the temperature profile supports this interpretation, as the peaks in the mass flow rate corresponds to the troughs in the temperature profile. It appears that the high flow rate reduced the temperature, but filled the catalyst. When the flow reversed the propellant left in the catalyst decomposed, causing the temperature to rise again. The frequency of these oscillations is high initially and reduces as the temperature of the exhaust products increases, however the exact reason for this remains unknown.

The presence of these oscillations raised concerns about the useful lifetime of the ceramic foam as a catalyst. Following testing of the lower densities of foam a significant amount of powder was
found to be loose at the upstream end of the catalyst. To evaluate the extent of the internal damage, pressure drop tests were conducted on the test pieces after firing.

![Pressure Drop Tests](image)

**Figure 5-20: Pressure Drop Tests Following Firing**

The effect of the firing on the internal structure of the catalyst itself was immediately evident and an example of the pressure drop data before and after testing is shown in Figure 5-20. In the image 10F is a 10% density foam, 15F a 15% density and 20C a 20% density foam. The firing clearly affected all of the foam catalysts as the gradient and location of each trend line has moved, although not all in the same way. The pressure drop through the 10% density foam reduced, while for the other two foams shown it increased.

![Filled Section Catalysts](image)

**Figure 5-21: Filled Section Catalysts**
It was decided that a visual inspection of the interior of the foam was required in order to investigate the reasons for these contrasting results. This was achieved by filling the catalyst with a low viscosity adhesive, Loctite® 420, which allowed them to be sectioned for visual inspection. To ensure the adhesive was present throughout the foam a soft vacuum was created at one end of the catalyst using a water pump. The three foams corresponding to the data above are shown in Figure 5-21, where the flow direction during testing was from left to right and the squares are 5 mm across for scale.

The images reveal the extent of degradation that occurred within the foam itself. In the case of the 10% density foam, a significant cavity is present at the upstream end of the catalyst, indicating a very energetic reaction has taken place. The removal of material, combined with severe degradation of the remaining internal structure supports the reduced pressure drop observed for this catalyst. Inspection of the 15% density foam reveals that approximately 40% of the initial catalyst material has broken down. The total operation time for the 15% density foam was appreciably longer than that of the 10% density foam, leading to the greater degradation. However the pressure drop increased, not decreased as would have been expected if this additional loss of material were the only explanation. In addition to the void, loose material can be observed at the upstream end of the foam. This material will cause the pressure drop to increase above the initial value as it artificially increases the density of the structure in this region. This theory is supported by Figure 5-22, which shows an enlarged section of the 20% density foam upstream of the catalyst. The 20% density case also exhibited an increased pressure drop after firing. In this image it is possible to identify substrate particles embedded within the adhesive. While these are no longer compressed at the upstream face of the catalyst, their presence supports the theory outlined above. In addition the smaller amount of material lost in the 20% density case corresponds with the lower increase in pressure drop observed.

This inspection revealed the fragility of the low density foams. Catalysts based upon a higher density substrate (23%, 27%) were therefore created to investigate the impact of the stronger structure on the performance. The results from testing these foams were very encouraging with
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respect to the pressure oscillations previously observed, however the maximum temperature achieved was disappointing. An example of the data generated using the 23% and 27% density foams with respect to the 15% density foam shown before is given in Figure 5-23.

The green trace corresponding to the 23% density foam is significantly smoother than that of the 15% density case, and indicates a considerably smoother decomposition characteristic. Following the previous tests when good temperature characteristics were generated using the manganese oxide catalyst, the low performance was unexpected.

![Temperature Profile](image)

**Figure 5-23: Temperature Profiles for Higher Density Foams**

It had been necessary to change the method of catalyst deposition at this time from the initial method described in section 5.5.2 to the flash vaporisation method. The chilled environment used to dry the catalyst onto the surface of the foam suffered a change that was undeterminable. The result of that change was to increase the drying rate, resulting in deposition similar to that observed when the catalyst dried at room temperature. As a result the flash vaporisation technique described in section 5.5.2 was developed with outwardly good results. Following the test runs conducted, where low decomposition temperatures were observed the method of catalyst decomposition was considered as a source of the poor performance.

Internal inspection of the catalysts led to the discovery of the uneven deposition that was shown in Figure 5-9. Prior to the tests the foam catalyst beds were simply broken at the centre to inspect the quality of deposition. The dark colour at that point indicated a good catalyst loading, but the axial inspection conducted subsequently revealed the uneven nature of the deposition overall. The lower catalyst loading is believed to cause the reduction in temperature achieved.
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The improved decomposition characteristic demonstrates that the increased pressure drop associated with the higher density foam is favourable. To investigate this further pressure drop tests were again conducted on the catalysts after firing. The data for the foams corresponding to the temperature data given previously is shown in Figure 5-24, where the 27A2 trace relates to a 27% density foam and the 23A1 trace to a 23% density foam.

It can be seen that the pressure drop for the higher density foams has increased, although to a lesser extent than observed previously. The total firing duration will influence these results, as the substrate material is breaking down and presumably this will occur over time. Two tests runs of 60 s were conducted with each of the higher density foams. Previously three test runs of 60 s were performed and in the case of the 15% density foam catalyst approximately eight test runs were conducted.

![Figure 5-24: Higher Density Foams Pressure Drop Tests Following Firing](image)

5.8.2 Compressed Powder Catalyst Bed Results

The two different grades of silver powder were tested in the chamber described previously. The length and diameter of the catalyst bed was varied to determine if an optimum length existed for each chamber diameter tested. Three diameters of chamber were used: 9.7 mm; 7.7 mm; 6.7 mm. The results from the testing of each will now be discussed.

5.8.2.1 Test Results from the 9.7 mm Diameter Decomposition Chamber

The first set of tests was conducted using the chamber described in section 5.6.2, with a catalyst pack diameter of 9.7 mm. The coarse powder was tested initially and an example of the data
recorded is shown in Figure 5-25, where the length of the bed tested was 10 mm. The temperature profile is shown in blue, the chamber pressure in green and the mass flow rate of propellant in red. For this test the propellant was pressurised to 4 bar. The initial peak in the mass flow rate data trace is to be expected as the propellant fills the catalyst bed. The unsteady nature of the data following this peak was undesired and indicated the need for an additional flow restriction upstream of the catalyst to provide greater control over the mass flow rate of propellant.

This was initially achieved through the use of a Lee Visco Jet, which is a fixed flow restriction device. The impact of this on performance is shown in Figure 5-26. The presence of the flow restriction causes the mass flow rate of propellant to be reduced, allowing the decomposition to proceed smoothly with an excellent temperature profile resulting.
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A significant oscillation remained in the pressure trace indicating that an instability was still present. To provide additional control over the propellant flow rate through the catalyst pack, the Visco Jet was replaced with a needle valve immediately upstream of the pack. This further improved the pressure and mass flow rate data as shown in Figure 5-27.
Following these adjustments to the test rig, the remainder of the testing proceeded. Four different lengths of catalyst were tested with the coarse grade of powder in the 9.7 mm diameter chamber and a sample of the data collected in each case is presented in Table 5-4. The pressure and mass flow rate data given in each case is an average value for a particular test run. Overall the coarse powder demonstrated a good decomposition characteristic, although the decomposition temperature achieved remained below 500 °C. The value of $c^*$ was calculated for each length of catalyst tested using Equation 5-2, assuming a throat diameter of 1 mm.

<table>
<thead>
<tr>
<th>Performance Characteristic</th>
<th>5 mm Catalyst Bed</th>
<th>7.5 mm Catalyst Bed</th>
<th>10 mm Catalyst Bed</th>
<th>12.5 mm Catalyst Bed</th>
</tr>
</thead>
<tbody>
<tr>
<td>Chamber Pressure, $p_c$</td>
<td>4.12 bar</td>
<td>4.55 bar</td>
<td>4.20 bar</td>
<td>6.15 bar</td>
</tr>
<tr>
<td>Mass Flow Rate, $m$</td>
<td>0.41 g s$^{-1}$</td>
<td>0.41 g s$^{-1}$</td>
<td>0.39 g s$^{-1}$</td>
<td>0.59 g s$^{-1}$</td>
</tr>
<tr>
<td>Decomposition Temperature, $T_e$</td>
<td>383 °C</td>
<td>456 °C</td>
<td>450 °C</td>
<td>423 °C</td>
</tr>
<tr>
<td>Characteristic Exhaust Velocity, $c^*$</td>
<td>799.69 ms$^{-1}$</td>
<td>883.15 ms$^{-1}$</td>
<td>857.02 ms$^{-1}$</td>
<td>829.53 ms$^{-1}$</td>
</tr>
</tbody>
</table>

Table 5-4: Data Summary of Coarse Silver Powder Catalyst Tests in 9.7 mm Diameter Chamber

Inspection of the data reveals that all of the tests produce values of $c^*$ that are significantly higher than the theoretical maximum. For hydrogen peroxide at a concentration of 85%, this is approximately 701 ms$^{-1}$. Inefficiencies in the measurements taken will contribute to this and in addition no allowance is made for the presence of a boundary layer within the throat of the thruster. The potential magnitude of this effect is discussed in more detail in chapter 6. Here, as the throat used for each test is the same, while the magnitude of $c^*$ is high, the relative magnitude of $c^*$ for the different configurations remains of interest. The data presented for the 5 mm length case is an estimate, indicated by the text in italics. This is because the data recorded contained significant variations in the pressure data recorded for each test. The pressure oscillations observed in the 7.5 mm length case were the smallest in magnitude, indicating good decomposition, which is reflected in the highest value of $c^*$ calculated.

Following tests with the coarse powder the fine silver powder was tested. The smaller particle size leads to an increased pressure drop through the pack as the void space is reduced. An example of the data from a test run with a 10 mm length catalyst is shown in Figure 5-28. The temperature data trace shows a good decomposition characteristic, with a temperature of 450 °C achieved within the data sampling period of 60 s. Overall the performance of the finer powder in the 9.7 mm diameter chamber appears comparable with that observed for the coarse silver powder. The key difference is an increase in the amplitude of the oscillations recorded in the pressure and mass flow rate data traces. The reason for this change in the oscillations observed is unclear. The higher density of the catalyst bed as a result of the smaller particle size may require a higher chamber pressure to eliminate the oscillation, however the increased mass flow rate may quench the decomposition reaction. Two different lengths of catalyst bed were tested with the fine silver powder and a sample of the data collected is given in Table 5-5. The figures given are average values for a given test run.
10 mm Fine Silver Powder Catalyst Test Data

- Mass Flow Rate
- Chamber Pressure
- Temperature

Figure 5-28: 10 mm Fine Silver Powder Catalyst Test Data

<table>
<thead>
<tr>
<th>Performance Characteristic</th>
<th>7.5 mm Catalyst Bed</th>
<th>10 mm Catalyst Bed</th>
</tr>
</thead>
<tbody>
<tr>
<td>Chamber Pressure, $p_c$</td>
<td>3.86 bar</td>
<td>3.49 bar</td>
</tr>
<tr>
<td>Mass Flow Rate, $m$</td>
<td>0.35 gs$^{-1}$</td>
<td>0.33 gs$^{-1}$</td>
</tr>
<tr>
<td>Decomposition Temperature, $T_r$</td>
<td>460 °C</td>
<td>455 °C</td>
</tr>
<tr>
<td>Characteristic Exhaust Velocity, $c^*$</td>
<td>877.66 ms$^{-1}$</td>
<td>841.62 ms$^{-1}$</td>
</tr>
</tbody>
</table>

Table 5-5: Data Summary of Fine Silver Powder Catalyst Tests in the 9.7 mm Diameter Chamber

The effect of the oscillations on the performance of the catalyst bed is evident in the calculated value of $c^*$, which is lower than in the previous tests. The extent of the oscillations observed in combination with a limited supply of propellant led to the decision to cancel additional tests with this catalyst in the 9.7 mm chamber and focus on smaller diameter chambers.

5.8.2.2 Test Results from the 7.7 mm Diameter Decomposition Chamber

The internal diameter of the decomposition chamber was reduced to 7.7 mm through the use of collars, with a wall thickness of 1 mm. The tests for the 7.7 mm diameter chamber were conducted using three different lengths of collar: 5 mm, 7.5 mm and 10 mm. Both grades of powder were tested with all three lengths of catalyst bed and the performance observed was strikingly different for the two cases.

The coarse powder exhibited a poor decomposition characteristic overall, with the temperature reaching a maximum of 380 °C for the 10 mm length bed. The excessive instability in pressure trace prevented a value $c^*$ to be calculated in all but one case. A summary of the data collected is presented in Table 5-6.
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Table 5-6: Data Summary of Coarse Silver Powder Catalyst Tests in 7.7 mm Diameter Chamber

<table>
<thead>
<tr>
<th>Performance Characteristic</th>
<th>5 mm Catalyst Bed</th>
<th>7.5 mm Catalyst Bed</th>
<th>10 mm Catalyst Bed</th>
</tr>
</thead>
<tbody>
<tr>
<td>Chamber Pressure, $p_0$</td>
<td>unsteady</td>
<td>2.55 bar</td>
<td>unsteady</td>
</tr>
<tr>
<td>Mass Flow Rate, $m_0$</td>
<td>0.46 gs$^{-1}$</td>
<td>0.33 gs$^{-1}$</td>
<td>0.463 gs$^{-1}$</td>
</tr>
<tr>
<td>Decomposition Temperature, $T_c$</td>
<td>300 °C</td>
<td>340 °C</td>
<td>380 °C</td>
</tr>
<tr>
<td>Characteristic Exhaust Velocity, $c^*$</td>
<td>-</td>
<td>614.94 ms$^{-1}$</td>
<td>-</td>
</tr>
</tbody>
</table>

In contrast, the tests using the fine powder generated significantly improved results with temperatures in excess of 500 °C achieved, although severe oscillations remained in the pressure data. An example of the data generated is shown in Figure 5-29. The mass flow rate trace is not included in this chart as it was recorded using the mass balance arrangement described in section 5.7.3 and an average value was calculated. A summary of the data collected for these tests is given in Table 5-7, with an average figure given for the chamber pressure.

Table 5-7: Data Summary of Fine Silver Powder Catalyst Tests in 7.7 mm Diameter Chamber

<table>
<thead>
<tr>
<th>Performance Characteristic</th>
<th>5 mm Catalyst Bed</th>
<th>7.5 mm Catalyst Bed</th>
<th>10 mm Catalyst Bed</th>
</tr>
</thead>
<tbody>
<tr>
<td>Chamber Pressure, $p_0$</td>
<td>4.9 bar</td>
<td>4.5 bar</td>
<td>4.6 bar</td>
</tr>
<tr>
<td>Mass Flow Rate, $m_0$</td>
<td>0.43 gs$^{-1}$</td>
<td>0.41 gs$^{-1}$</td>
<td>0.44 gs$^{-1}$</td>
</tr>
<tr>
<td>Decomposition Temperature, $T_c$</td>
<td>505 °C</td>
<td>485 °C</td>
<td>490 °C</td>
</tr>
<tr>
<td>Characteristic Exhaust Velocity, $c^*$</td>
<td>906.35 ms$^{-1}$</td>
<td>873.44 ms$^{-1}$</td>
<td>831.98 ms$^{-1}$</td>
</tr>
</tbody>
</table>

Figure 5-29: 5 mm Fine Silver Powder Catalyst Data in the 7.7 mm Diameter Chamber

5.8.2.3 Test Results for the 6.7 mm Diameter Decomposition Chamber

Due to the instabilities observed using the coarse powder in the 7.7 mm diameter decomposition chamber, it was decided that only the fine powder would be tested in the 6.7 mm diameter decomposition chamber. Collars with a wall thickness of 1.5 mm were created in the same three
lengths were created to create catalyst beds of length 5 mm, 7.5 mm and 10 mm. Overall the severe pressure oscillations observed in previous cases remained present. The temperature profiles recorded were encouraging, with temperatures approaching 500 °C registered.

Table 5-8 provides a summary of the data recorded for the tests with the 6.7 mm diameter chamber. The data for the 5 mm catalyst bed is in italics to indicate that the values are estimated. Overall the calculated values of $c^*$ indicate a good decomposition characteristic, however the pressure oscillations remain.

<table>
<thead>
<tr>
<th>Performance Characteristic</th>
<th>5 mm Catalyst Bed</th>
<th>7.5 mm Catalyst Bed</th>
<th>10 mm Catalyst Bed</th>
</tr>
</thead>
<tbody>
<tr>
<td>Chamber Pressure, $p_f$</td>
<td>$7 \text{ bar}$</td>
<td>$3.6 \text{ bar}$</td>
<td>$4.1 \text{ bar}$</td>
</tr>
<tr>
<td>Mass Flow Rate, $m$</td>
<td>$0.6 \text{ g}^{-1}$</td>
<td>$0.33 \text{ g}^{-1}$</td>
<td>$0.38 \text{ g}^{-1}$</td>
</tr>
<tr>
<td>Decomposition Temperature, $T_r$</td>
<td>$497 \degree \text{C}$</td>
<td>$444 \degree \text{C}$</td>
<td>$453 \degree \text{C}$</td>
</tr>
<tr>
<td>Characteristic Exhaust Velocity, $c^*$</td>
<td>$928.44 \text{ ms}^{-1}$</td>
<td>$968.15 \text{ ms}^{-1}$</td>
<td>$858.63 \text{ ms}^{-1}$</td>
</tr>
</tbody>
</table>

Table 5-8: Data Summary of Fine Silver Powder Catalyst Tests in 6.7 mm Diameter Chamber

The data shown indicates that the propellant mass flow rate for tests with the 5 mm catalyst bed was higher than those used previously. The reason for this was that the decomposition reaction appeared to be significantly smoother. Analysis of all the data recorded indicated that improved performance would result if a higher mass flow rate of propellant were used.

5.8.2.4 Results for High Mass Flow Rate Tests

In order to generate higher mass flow rates, the chamber pressure was also increased, which required a change of pressure transducer. The 0 - 6 bar transducer was replaced with a similar device that was capable of operating in the range 0 - 25 bar. In addition due to the higher propellant mass flow rates to be used the coriolis mass flow meter was again used for these tests.

Tests were conducted with a selection of the catalyst bed sizes used previously. There were two aims to these tests: first to establish whether the performance of the catalyst beds improved at a higher mass flow rate and second to find the point at which the mass flow rate became too high for the decomposition reaction to be sustained.

The last catalyst bed to be tested previously was the smallest bed used, at 5 mm length and 6.7 mm diameter. This catalyst bed was the first to be tested at a range of increased mass flow rates. Figure 5-30 shows the decomposition temperature achieved for each mass flow rate of propellant tested using this catalyst bed. Superimposed onto this chart is an estimated trendline, which indicates an approximately logarithmic relation between the propellant mass flow rate and decomposition temperature achieved.
Figure 5-30: The Effect of Mass Flow Rate on Decomposition Temperature

From Figure 5-30 it can be concluded that an increase in the mass flow rate of propellant does result in an improved decomposition temperature. The logarithmic nature of the relation indicates that significant improvements may be achieved with small increases in mass flow rate. Above a mass flow rate of approximately 1.25 gs\(^{-1}\) these improvements reduce dramatically. In addition the pressure to which the propellant should be pressurised to achieve flow rates in excess of 1.25 gs\(^{-1}\) starts to become prohibitive for a small satellite. The data presented shows that this catalyst bed was tested with propellant mass flow rates in excess of 4 gs\(^{-1}\). Prior to each test the decomposition chamber assembly was cooled to below 100 °C and the mass flow rate was set. This resulted in a realistic starting environment for the catalyst bed and each time it responded quickly. As the mass flow rate of propellant increased the response time also increased, however for mass flow rates above 1 gs\(^{-1}\) the decomposition temperature recorded exceeded 500 °C within 25 s of operation. This would be further optimised if the thermal characteristics of the decomposition chamber were enhanced.

The second aim of these tests was to determine the highest propellant mass flow rate that the catalyst bed could accommodate. The literature revealed that a silver gauze catalyst bed would “wash out” if the propellant mass flow rate were too high [Coxhill'02]. It was assumed that the same phenomena would occur with a compressed powder catalyst bed, however the data indicates that this is not the case. Within the range of propellant mass flow rates tested no “wash out” occurred, indicating that a the catalyst bed would accommodate a wide range of operations.

In addition to an improvement in the decomposition profile significant improvements were observed in the pressure and mass flow rate traces as the chamber pressure increased above 7 bar.
and the mass flow rate above 0.7 gs$^{-1}$. This is demonstrated by Figure 5-31, which shows the test data for the second test run at a propellant mass flow rate of 0.7 gs$^{-1}$.

![Figure 5-31: 5 mm Fine Silver Powder Catalyst Data in the 6.7 mm Diameter Chamber](image)

Following the success of the tests conducted with the 5mm long catalyst bed in the 6.7 mm diameter chamber it was decided that a smaller catalyst bed should be tested. The ratio of dimensions of the 6.7 mm diameter, 5 mm length pack appeared to work well, therefore it was decided to maintain this length to diameter (L:D) ratio and miniaturise the catalyst as far as possible within the constraints of the rig. This resulted in a catalyst bed that was 3.3 mm in diameter and 2.5 mm in length. The collar used to contain the catalyst material for this bed is shown in Figure 5-32, where the squares are 5 mm. The small diameter of the catalyst bed required each end of the collar to be contoured to allow the filter disc to be accommodated properly.

![Figure 5-32: Collar For the 3.3 mm Diameter, 2.5 mm Long Catalyst Bed](image)
The cross-sectional area of this catalyst bed was just 32% of the total cross-sectional area of the decomposition chamber. The tests revealed that a catalyst bed of this size is capable of decomposing hydrogen peroxide. Severe instabilities were present in the data recorded and the maximum decomposition temperature achieved was approximately 210 °C. It is thought that the performance of this catalyst bed would improve dramatically with additional control of the propellant mass flow rate and optimisation of the test set-up. One aspect that would require significant attention is the wall thickness of the chamber. The thickness of the steel chamber wall surrounding this catalyst bed was 4.7 mm, which will result in significant thermal losses.

To build further on the data collated from the 5mm long catalyst bed in the 6.7 mm diameter chamber it was decided that the length to diameter ratio should be maintained for additional tests with the other diameters of decomposition chamber used. Tests were conducted using 9.7 mm and 7.7 mm diameter catalyst beds, with an associated length of 7.3 mm and 5.75 mm respectively. Each catalyst bed was tested at a range of mass flow rates between 0.4 gs⁻¹ and 1.5 gs⁻¹. Tests were limited to this range so as to preserve propellant, as little additional gain was observed in the previous tests above a propellant mass flow rate of 1.25 gs⁻¹. The data recorded demonstrates a similar trend to that observed previously, with an increase in the decomposition temperature as a result of a higher propellant mass flow rate. The results for these tests are displayed in Figure 5-33 and a trendline has been superimposed over the data. As the chamber pressure and propellant mass flow rate increased the instabilities observed in the pressure and mass flow rate data were minimised once again.

![Figure 5-33: Test Data for 9.7 mm and 7.7 mm Diameter Catalyst Beds](image-url)
The results from these tests confirm the previous conclusion that it is possible for a compressed powder catalyst bed to sustain operation at a higher mass flow rate than initially thought likely. The propellant mass flow rate that caused "wash out" was not found in the range tested here. Due to the increased mass of silver in both catalyst beds in comparison to the previous test it is thought that this point would be in excess of the mass flow rates tested previously.

5.8.2.5 Evaluation of the Catalyst Material

The lifetime of a catalyst bed is critical to the complete operation of the thruster. The monolithic catalyst bed experienced severe degradation during the decomposition reaction, causing catalyst material to be lost. In comparison the temperature and pressure within the decomposition chamber of the compressed powder catalyst bed caused the silver to sinter. The result is a porous block of silver particles, which fit precisely within the chamber. The extent of the sintering that occurred was dependent upon the decomposition temperature achieved. An example of the catalyst bed that results is shown in Figure 5-34.

![Figure 5-34: Coarse Silver Powder Following Firing](image_url)

The squares in the left hand image are 5 mm across and the right hand image is taken through a microscope to illustrate the fusing of the particles. When silver screens are used as the catalyst material in a hydrogen peroxide thruster the resultant catalyst bed usually becomes narrower than the chamber diameter. This is because the thermal expansion coefficient of silver is approximately twice that of stainless steel. As the silver heats up, the gauzes become squashed against the wall of the chamber and upon cooling shrink away. In the tests carried out here, in general the sintered block of silver powder had to be forcibly removed from the chamber. These two effects combined lead to the generation of a stable catalyst bed, the performance of which may be reliably predicted, as the internal structure will not change.

The other noticeable change in the catalyst material after firing is the colour. When removed from the casing the catalyst material appears white in colour instead of silver. The reason for this is a roughening of the surface, which was revealed through use of a microscope as shown in Figure 5-35. In the image the particle appears a silver colour again, indicating that the surface roughness causes a diffuse reflection of light resulting in a white colour.
Extended firing of a compressed silver powder catalyst bed results in a bed that appears a dull grey colour. The two images in Figure 5-36 show a catalyst bed following removal from the decomposition chamber. It had been tested for a total duration in excess of 15 minutes, which is significantly longer than any other catalyst bed.

Inspection of the catalyst material under a microscope revealed that the pitting of the particle surface had further increased from that observed in Figure 5-35. This is illustrated in the right-hand image in Figure 5-36, which also shows a duller surface than observed previously. It is possible that deposits of silver oxides are beginning to occur, however no reduction in performance was observed.

**5.9 Discussion of Compressed Powder Catalyst Results**

The testing data for the compressed silver powder catalyst beds has indicated that it is possible to decompose hydrogen peroxide reliably in a small diameter chamber. The performance of the catalyst bed varied with the mass flow rate of propellant, length of the bed and packing density of the catalyst material. In order to characterise the performance of the catalyst bed and draw data trends each of these variables is considered separately.
5.9.1 Propellant Mass Flow Rate

The initial data recorded during testing implied that the propellant mass flow rate was too high and caused unstable decomposition. As a result of this, following the installation of a needle valve, testing proceeded at low propellant mass flow rates. Analysis of the data generated suggested that a higher decomposition temperature would result from an increase in the propellant mass flow rate, which resulted in additional tests taking place. The results from these tests were presented in section 5.8.2.4. Combining the all the propellant mass flow rate data with the decomposition temperature achieved in each case allows Figure 5-37 to be generated.

\begin{figure}[h]
\centering
\includegraphics[width=\textwidth]{decomposition_temperature_vs_mass_flow_rate.png}
\caption{Effect of Propellant Mass Flow Rate on Decomposition Temperature}
\end{figure}

The approximately logarithmic trend observed previously is evident once again. The data collected at low propellant mass flow rates shows great variation in the decomposition temperature achieved indicating the need for operation at higher flow rates. The data presented in Figure 5-37 contains results from tests of catalyst beds ranging in both diameter and length. No significant differences are present between the different sizes of catalyst bed although the mass of silver contained in the beds ranges considerably.

To determine whether the dimensions of a catalyst bed has any significant implications on the resultant efficiency of the decomposition reaction, data collected in a previous research effort was consulted [Coxhill'02]. The previous research utilised a catalyst bed constructed from compressed silver gauzes, packed inside a chamber 12.7 mm in diameter. The data selected for comparison here was recorded while testing a catalyst bed 10 mm in length, with a packing density of approximately 0.5. This agrees well with the fine silver powder used and the ratio of
dimensions is similar to that used for the high mass flow rate tests. Figure 5-38 shows the data generated by this pack together with data from a compressed silver powder catalyst 5 mm in length in the 6.7 mm diameter chamber.

Figure 5-38: Comparison of Propellant Mass Flow Rate Effect on Decomposition Temperature

From these data it can be seen that the trend observed previously remains in evidence for larger catalyst beds. In addition it is clear that the smaller diameter bed generates a superior performance at lower propellant mass flow rates. This indicates that further miniaturisation of the catalyst bed is required to optimise performance at low propellant mass flow rates.

5.9.2 Decomposition Chamber Internal Diameter

The performance of a compressed powder catalyst bed was evaluated for three different decomposition chamber diameters. A comparison of the data generated for a nominal propellant mass flow rate of 0.5 gs\(^{-1}\) is given in Figure 5-39. The spread of results in each case indicates variations in both the catalyst bed length and exact mass flow rate.
Overall the data exhibits a general parabolic trend, showing that a catalyst bed diameter of approximately 7.7 mm is optimal for this propellant mass flow rate. The implications of this data are difficult to evaluate directly as the thermal characteristics of the decomposition chamber were neglected at the design stage. The models discussed in chapter 4 predicted the thermal losses from a decomposition chamber that varied in total diameter, but maintained a constant wall thickness. To predict the thermal losses from the decomposition chambers tested, the data generated by the models is converted be a function of the wall thickness to chamber external diameter ratio. The data trend that results is shown in Figure 5-40 together with a linear trendline, which approximates the data. From this trendline it is possible to estimate the thermal losses from the three decomposition chambers tested.

The wall thickness of the steel surrounding the 9.7 mm diameter catalyst bed was 1.5 mm. This increased to 2.5 mm for the 7.7 mm diameter bed and 3 mm for the 6.7 mm diameter bed. The red lines superimposed onto Figure 5-40 indicate the wall thickness to total chamber diameter of each of these catalyst bed arrangements together with the thermal losses predicted. Table 5-9 then summarises the data extracted from these predictions.
Chapter 5: The Decomposition Chamber

Predicted Thermal Losses from the Decomposition Chamber

\[ y = 1.23 \times 10^7 x + 7.55 \times 10^4 \]

Figure 5-40: Predicted Thermal Losses from the Decomposition Chamber

<table>
<thead>
<tr>
<th>Catalyst Bed Diameter (mm)</th>
<th>Wall Thickness to External Diameter Ratio</th>
<th>Predicted Heat Flux per Unit Surface Area (Wm(^{-2}))</th>
</tr>
</thead>
<tbody>
<tr>
<td>9.7</td>
<td>0.118</td>
<td>(1.527 \times 10^6)</td>
</tr>
<tr>
<td>7.7</td>
<td>0.197</td>
<td>(2.498 \times 10^6)</td>
</tr>
<tr>
<td>6.7</td>
<td>0.236</td>
<td>(2.978 \times 10^6)</td>
</tr>
</tbody>
</table>

Table 5-9: Summary of Thermal Loss Predictions

The magnitude of the losses predicted indicates the need for assessment of the thermal characteristics of the decomposition chamber. In summary, it was found that both the 7.7 mm and 6.7 mm diameter catalyst beds performed better than the 9.7 mm diameter catalyst bed, although higher thermal losses were predicted. This indicates that the decomposition chamber can be miniaturised and the efficiency of the reaction maintained.

5.9.3 Catalyst Bed Length

Three different catalyst bed lengths were used to assess its influence on the efficiency of the decomposition reaction. Historical data suggests that a long thin catalyst bed is preferred as this ensures that the propellant has sufficient time to decompose. As the dimensions of the catalyst bed reduce the thermal losses incurred through the use of a long thin catalyst bed increase. The effect of this is reflected in the data generated through testing. Figure 5-41 shows data relating to the decomposition temperature achieved for a given catalyst bed length. The data used to create this chart is selected at a propellant mass flow rate of approximately 0.5 gs\(^{-1}\). Overall it can be seen that shorter catalyst beds generate higher decomposition temperatures, which confirms that
the catalyst bed geometry should be short relative to the diameter. It is thought that optimisation of the thermal characteristics of the decomposition chamber would further enhance these results.

**Figure 5-41: Effect of Catalyst Bed Length on Decomposition Temperature**

5.10 Conclusions

The monolithic catalyst bed produced good decomposition characteristics in terms of the temperature achieved, but it suffered from severe pressure instabilities. In terms of chemistry the use of manganese oxide as a catalyst for the decomposition hydrogen peroxide is preferred to silver due to the lower activation energy it requires. However, the severe pressure instabilities observed when using this catalyst bed in combination with the associated degradation of the catalyst material lead to the conclusion that extensive additional characterisation of this catalyst bed is required before any miniaturisation could take place.

There is evidence in the literature that the method of production of a manganese oxide catalyst has a significant impact on the activity observed [Ivanova'02]. The impact of this is clear from the changes in the catalyst loading observed using different methods. It was unfortunate that a change in the chilled environment led to the initial deposition method no longer being successful. It was at this point that the flash vaporisation technique was developed, however the uneven deposition led to significant reductions in the performance observed. While the manganese oxide catalyst exhibits a high activity with hydrogen peroxide, it is clear that a more reliable deposition method is required to ensure a minimum level of catalyst loading. Once this is achieved, more testing may be considered in order to optimise the performance of the decomposition reaction.
Chapter 5: The Decomposition Chamber

The silver powder catalyst beds exhibited repeatable stable decomposition. The primary conclusion from this work is that the silver powder is a suitable alternative to silver gauzes for use in a hydrogen peroxide catalyst bed. It is possible to stably and repeatedly decompose hydrogen peroxide in a chamber that contains a catalyst bed 6.7 mm in diameter and 5 mm in length at a propellant mass flow rate in excess of 4 g s\(^{-1}\).

The decomposition temperature achieved with the compressed silver pack was influenced by various factors, including the propellant mass flow rate as well as the catalyst bed diameter and length. In many cases the temperature failed to reach a steady state value, indicating that the performance values calculated will change with optimisation of the thermal characteristics. The characteristic at start-up was found to improve at higher propellant mass flow rates with a decomposition temperature in excess of 500 °C reached within 15 s of operation. It is thought that optimisation of the thermal characteristics of the chamber will further increase the temperature start-up characteristic as well as the maximum temperature achieved.

Miniaturisation of the catalyst bed was found to improve the decomposition characteristics recorded. At a given mass flow rate, the shortest catalyst beds were found to produce the highest temperatures for each diameter of catalyst bed tested. This indicates that a smaller diameter catalyst bed is required to accommodate low propellant mass flow rates. The compressed silver powder catalyst bed also demonstrated cold-start capability, with some of the tests being conducted with an ambient temperature of 0 °C and a propellant temperature of 4 °C.

Two different grades of silver powder were tested and the performance observed varied dramatically. The coarse powder performed well in the largest catalyst beds tested, but generated severe instabilities in a smaller chamber. The finer powder performed acceptably in all cases tested and is therefore recommended for further testing work.

It is thought that the test set up itself contributed to some of the oscillations observed in the chamber pressure readings due to the length of the feed lines used. In addition it is believed that the change in the method of mass flow rate measurement to the scales approach caused further instabilities, which were not present when the coriolis mass flow meter was used. Additional testing with an optimised test set up would confirm these assessments.

There was insufficient time available to develop a test chamber using silicon as the structural material. The thermal losses encountered by the steel chamber used indicate that silicon would increase these losses significantly. The geometry of the decomposition chamber studied is suitable for further miniaturisation, however it is unlikely that MEMS manufacturing methods would be suitable for its manufacture. It is therefore concluded that a different design perspective will be required to efficiently and effectively miniaturise a hydrogen peroxide thruster for manufacture using MEMS technologies.
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In summary this work has shown that a compressed silver powder is suitable for use as a catalyst material to decompose hydrogen peroxide in a decomposition chamber ranging in diameter from 9.7 mm to 3.3 mm, with a propellant mass flow rate of up to 5 gs⁻¹.
6 The Nozzle

This chapter will summarise the work completed that relates to the miniaturisation of an exhaust nozzle. Theory relating to the performance evaluation of a complete thruster was introduced in chapter 4. This will be expanded upon with respect to the performance of a nozzle. The sources of losses that may severely degrade the performance of a thruster were also discussed in chapter 4. The implication of these losses on the performance of a nozzle will be considered in more detail here. The development of the flow within a nozzle is then investigated through use of numerical models. These enable the evaluation of the performance of a nozzle as it reduces in dimension. The results from the models will be discussed and evaluated with respect to their impact on the thruster performance and the literature previously published.

6.1 Isentropic Nozzle Theory

The purpose of the exhaust nozzle component of a rocket engine is to expand and accelerate the hot gases produced upstream, thereby generating thrust with minimal loss. Use of isentropic theory enables the design of an ideal nozzle, which expands the exhaust gas perfectly. The condition of perfect expansion may be shown to occur when the pressure of the exhaust gas at the exit plane of the nozzle is equal to that of the ambient environment. For a rocket nozzle operating in the vacuum of space this is possible in theory, however in reality a compromise is made between the expansion achieved and the mass of the nozzle required. To address this a pressure ratio, \( p_e / p_c \), is used, which relates the pressure in the combustion chamber, \( p_c \), to the pressure at the exit plane from the nozzle, \( p_e \). The magnitude of this ratio is different for different types of rocket engine. Typically for a monopropellant engine it should be approximately 65. The theory in this section is derived from reference [Hill’92] unless otherwise stated.

6.1.1 Nozzle Performance Evaluation

The performance of the decomposition chamber was characterised previously in chapter 5 using the characteristic exhaust velocity, \( c^* \). This allowed evaluation of the efficiency of the chamber without consideration of the nozzle. The performance of a nozzle is usually evaluated through use of the thrust coefficient, \( c_t \), which is a measure of nozzle effectiveness. The value provides an indication of how the presence of the divergent section of a nozzle magnifies the thrust that would otherwise have been generated by the unexpanded gases. It is defined by Equation 6-1 with respect to thrust, \( T \), chamber pressure, \( p_c \) and throat area, \( A^* \). An alternative definition is also
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possible for the ideal rocket case where it may be derived as a function of the pressure ratio, \( \frac{p_c}{p_e} \) and the ratio of specific heats of the fluid, \( \gamma \), given in Equation 6-2.

\[
c_f = \frac{\tau}{p_c A_e}
\]

Equation 6-1: Definition of the Thrust Coefficient

\[
c_f = \left( \frac{2\gamma^2}{\gamma - 1} \right) \left( \frac{2}{\gamma + 1} \right) \left[ 1 - \left( \frac{p_e}{p_c} \right)^{\frac{\gamma - 1}{\gamma}} \right] + \left( \frac{p_e}{p_c} \right) A_e
\]

Equation 6-2: Thrust Coefficient for the Ideal Rocket

This second equation demonstrates that for the ideal case the thrust coefficient is a function of the nozzle geometry only, as \( \gamma \) remains constant for a given propellant. The pressure term can be regarded as a measure of the nozzle efficiency with respect to the actual pressure ratio. A theoretical value of \( c_f \) may be calculated from Equation 6-2 for a given pressure ratio. To calculate a value for \( c_f \) that is valid for a real nozzle an alternative approach is required. The thrust generated by a rocket engine may be defined as a function of the mass flow rate of propellant, \( \dot{m} \) and the effective exhaust velocity, \( u_{\text{ef}} \) as shown in Equation 6-3. The effective exhaust velocity was defined in chapter 4 in equation 4-11 and is the weighted average flow velocity across the exit plane of the nozzle.

\[
T = \dot{m} u_{\text{ef}}
\]

Equation 6-3: Thrust as a Function of Mass Flow Rate and Effective Exhaust Velocity

This may be combined with the definition of \( c^* \) in Equation 5-2 and Equation 6-1 to give an empirical definition of the thrust coefficient, given in Equation 6-4.

\[
c_f = \frac{u_{\text{ef}}}{c^*}
\]

Equation 6-4: Empirical Definition of the Thrust Coefficient

A value of \( c_f \) may be calculated using Equation 6-4 in combination with velocity data from numerical models. The ideal value of \( c_f \) can then be compared with this prediction to provide an indication of the efficiency of the nozzle.

6.1.2 Geometric Parameters

The design of a rocket nozzle is based upon thermodynamic relations, which describe the expansion of a perfect gas. These are used to develop the geometric parameters required to size
the nozzle. The approach used to generate the parameters required for the models used here is now summarised.

The first parameter calculated was the mass flow rate. The definition of the thrust coefficient, given in Equation 6-1, may be combined with that of $c^*$, given in Equation 5-2, to generate an equation for thrust as shown in Equation 6-5. Using empirical values for both $c^*$ and $c_f$, the required mass flow rate for a given magnitude of thrust can be calculated. Knowledge of the mass flow rate together with the conditions within the decomposition chamber enables calculation of the throat area required to generate supersonic flow using Equation 5-2, the definition for $c^*$.

$$
\dot{m} = \rho c^* c_f
$$

Equation 6-5: Thrust as a Function of $c^*$ and $c_f$

The expansion of the flow is the next consideration. The pressure ratio, $p_e/p_r$, defines the degree to which the flow is expanded. The size of nozzle required to achieve this is defined by the area ratio, $A_*/A_e$. This ratio is defined in Equation 6-6, where $\gamma$, is the ratio of specific heats of the exhaust products [Sutton'01]. The derivation of this equation is based upon the pressure ratio, $p_e/p_r$, and the assumption that the flow will become supersonic at the throat of the nozzle. Knowledge of the area ratio in addition to the throat area allows calculation of the required exit area of the nozzle.

$$
\frac{A_*}{A_e} = \left(\frac{\gamma+1}{\gamma+1}\right)^{\frac{1}{2}} \left(\frac{p_e}{p_r}\right)^{\frac{1}{\gamma-1}} \left[1-\left(\frac{p_e}{p_r}\right)^{\frac{\gamma-1}{\gamma+1}}\right]
$$

Equation 6-6: Area Ratio Definition [Sutton'01]

The area ratio defined the size of the exit of the nozzle. The rate at which the nozzle expands to that size is defined by the expansion half-angle, $\alpha$. The length of nozzle, $L_n$, required to expand the flow appropriately is calculated using Equation 6-7, where $d^*$ is the throat diameter.

$$
L_n = \frac{1}{2} d^* \sqrt{\left(\frac{A_*}{A_e} - 1\right) \cot \alpha}
$$

Equation 6-7: Nozzle Length

6.2 Boundary Layers

The theory outlined above assumes an ideal flow, which includes the assumption of an inviscid fluid. In practice the viscosity of a fluid cannot be ignored, as it is the source of various inefficiencies may degrade the performance of a nozzle. In particular the viscosity of the fluid is linked to the development of a boundary layer, which will modify the behaviour of a flow. The
concept of a boundary layer was introduced in chapter 4 with reference to flow through a channel. Within a nozzle a boundary layer can significantly reduce the volume available for the expansion of the flow. The classification of a boundary layer is considered here together with the numerical methods used to describe the velocity distribution within it.

6.2.1 Types of Boundary Layer

A fluid flow is classified according to its Reynolds number as laminar, turbulent or transitional and the same classification extends to boundary layers. A laminar boundary layer is a smooth layer, which contains all the effects of fluid viscosity. A severe velocity gradient exists within a laminar boundary layer and little mixing occurs within the fluid, leading to high levels of shear stress within the fluid. In comparison within a turbulent boundary layer the fluid is no longer held into structured layers. Instead a series of small vortices is formed allowing the fluid to rotate. This leads to convective mixing between the fluid adjacent to the wall and the freestream flow, reducing the shear stresses observed. A transitional boundary layer encompasses elements of both laminar and turbulent boundary layers.

Upon exit from the decomposition chamber, the flow of exhaust gases will be slow and turbulent. As the flow progresses through the nozzle it is accelerated and if the nozzle is designed correctly it will reach supersonic velocity at the throat. Downstream of the throat the convergent section of the nozzle further accelerates the exhaust stream to increase the thrust produced. The low velocity of the flow leads to a low Reynolds number, which implies that it will quickly become laminar following exit from the decomposition chamber. The acceleration of the flow through the nozzle may cause the flow to become turbulent again therefore to ensure correct representation of the flow, a turbulent flow and boundary layer is considered. The turbulent boundary layer will now be considered in more detail, with all theory referenced from [Schlichting'00].

6.2.2 Structure of a Turbulent Boundary Layer

A turbulent boundary layer is thought to consist of three regions: one close to the wall, one close to the freestream fluid and one between the two. The viscosity effects are restricted to the layer close to the wall, the viscous sublayer, which is significantly thinner than the thickness of the complete boundary layer, denoted by $\delta$. The viscous sublayer exhibits behaviour that is similar to that of a laminar boundary layer with a laminar velocity profile and viscosity dominated flow. The exterior layer is called the overlap layer, as this accounts for the change between flow within and exterior to the boundary layer, the layer between the two is then called the buffer layer. The location of the different layers relative to each other and a wall is illustrated in Figure 6-1.
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Freestream Flow

Overlap Layer
Buffer Layer
Viscous Sublayer
Wall

Figure 6-1: Turbulent Boundary Layer

The order of magnitude of the thickness of each of these layers varies considerably. If a linear scale were used to measure the thickness of each layer, the thickness of the viscous sublayer would appear negligible in comparison to that of the overlap layer. It is impossible to ignore the presence of the viscous sublayer however, therefore a characteristic wall coordinate is introduced, $y^*$. The definition of $y^*$ is given in Equation 6-8, where $y$ is the linear distance from the wall and $v$ is the dynamic viscosity. The friction velocity, $u_*$, is defined in Equation 6-9, where $\bar{\tau}_w$ is the time averaged wall shear stress.

$$ y^* = \frac{y u_*}{v} $$

Equation 6-8: Definition of $y^*$

$$ u_* = \sqrt{\frac{\bar{\tau}_w}{\rho}} $$

Equation 6-9: Definition of Friction Velocity

Characterisation of a turbulent boundary layer is then possible through use of the $y^*$ coordinate as shown in Table 6-1.

<table>
<thead>
<tr>
<th>Layer</th>
<th>$0 \leq y^* &lt; 5$</th>
</tr>
</thead>
<tbody>
<tr>
<td>Viscous Sublayer</td>
<td>Buffer Layer</td>
</tr>
<tr>
<td>$5 \leq y^* &lt; 70$</td>
<td>Overlap Layer</td>
</tr>
<tr>
<td>$70 \leq y^*$</td>
<td></td>
</tr>
</tbody>
</table>

Table 6-1: Turbulent Boundary Layer Characterisation

6.2.3 Boundary Layer Development

There are various conditions that will promote or inhibit the growth of a boundary layer, in particular the pressure gradient. A favourable pressure gradient, where the static pressure increases in the direction of the flow will inhibit the growth of a boundary layer. An adverse pressure gradient, where the static pressure decreases in the direction of the flow, will in contrast encourage the growth of a boundary layer, eventually leading to flow separation. In the nozzle an adverse pressure gradient exists, as the flow is expanding, therefore a boundary layer develops causing a modification in the flow profile. The presence of the boundary layer will reduce the...
effectiveness of the nozzle, hence evaluation of the actual performance of the nozzle itself is possible though identification of the position and thickness of the boundary layer, \( \delta \).

The edge of the boundary layer may be determined in numerous ways although two are of particular interest and both relate to the flow velocity. The most common definition relates to the magnitude of the flow velocity. The edge of the boundary layer, where \( y = \delta \), is defined as the point where the velocity within the boundary layer is 99% of the freestream velocity, \( u_\infty \), giving the first condition shown in Equation 6-10.

\[
y = \delta \quad \text{when} \quad u = 0.99u_\infty \quad \text{and} \quad \frac{du}{dy} = 0
\]

Equation 6-10: Definition of the Edge of the Boundary Layer

6.2.4 Limitations of Classic Boundary Layer Theory

The theory outlined above is based upon empirical data that has been collected and analysed over the course of many years. The velocity of the flows involved is usually high and the characteristic dimensions large, leading to large Reynolds numbers and thin boundary layers. The limitations of this theory in relation to flows with low velocity and small characteristic dimensions are unknown. Flows with low Reynolds numbers have been investigated but uncertainty in the results has led to different conclusions [Hsieh'04, Mall'99]. In the absence of any other literature, throughout this analysis it was assumed that classic boundary layer theory was valid.

6.3 Development of Numerical Models

Following assessment of the relevant theory relating to a flow within an exhaust nozzle a series of numerical models was created. These were designed to investigate how the flow developing within a nozzle is modified due to the presence of a boundary layer, as the dimensions of the nozzle reduce.

The parameter selected to determine the size of the nozzle was the magnitude of thrust produced. Nine thrust levels were initially identified ranging from 500 mN to 1 mN. Four different types of geometry were investigated and the dimensions for each model were calculated based upon the theory summarised previously. A standard geometry was used as a baseline and the other models
were created as modifications of this to allow continuity between the different aspects investigated. Each of the three additional models was designed to assess the impact of changing one particular aspect of the nozzle. The wide expansion angle models considered increasing the expansion half-angle to allow the nozzle to self compensate for the presence of the boundary layer. The sharp throat models contained a sharp throat instead of a smooth contour to investigate if this affected the rate of boundary layer growth. Finally the convergent section only models were used to evaluate the thrust produced if no divergent nozzle section was present.

The literature revealed that there was little performance advantage from use of a contoured nozzle; hence a simple conical nozzle geometry was selected. The section of the model geometry corresponding to the exit of the chamber and convergent section of the nozzle remained identical in each case. The literature relating to the optimisation of a catalyst bed indicated that the convergence angle at the end of the chamber was immaterial [Willis’60]. As a result the diameter of the chamber was arbitrarily assumed to be 5 mm and the convergence angle of 60°.

The chamber pressure was assumed to be 5 bar in each case, although in practice optimisation of this would be required through experiment. The pressure ratio used was typical of monopropellant engines at 65 and the value of the ratio of specific heats, \( \gamma \), was taken to be 1.27 from characteristic data for the exhaust products of 85% concentration hydrogen peroxide [BECCO’54]. The particular aspects of each of the four different types of model are now discussed, with the initial geometry calculations outlined in the standard geometry section.

### 6.3.1 Standard Geometry

For each thrust level used, the mass flow rate, \( \dot{m} \), required was calculated using Equation 6-5. Equation 6-2 was used to calculate \( \gamma' \) using the parameters specified previously and assuming perfect expansion, giving \( \gamma' = 1.68 \). The value for \( c^* \) was calculated from Equation 5-3 assuming a ratio of specific heats of the decomposition products, \( \gamma \), of 1.27, giving \( c^* = 701 \text{ ms}^{-1} \).

Knowledge of the mass flow rate, \( \dot{m} \), allowed the calculation of the throat area of the nozzle through use of Equation 5-2. From the throat area the throat diameter was calculated using geometric relations. The area ratio was calculated using Equation 6-6 in combination with the same value of \( \gamma \) as before, giving \( A_T/A_e = 0.128 \). The exit area of the nozzle was then calculated using this in combination with the throat area, followed by the exit diameter.

Finally the length of the nozzle, \( L_m \), was calculated using Equation 6-7, where \( \alpha \), was assumed to be 15°. A summary of the relevant dimensions for the nine models created is provided in Table 6-2 and a schematic of the geometry is shown in Figure 6-2.
6.3.2 Wide Expansion Angle

The presence of a thick boundary layer is the key concern as the dimensions of a nozzle reduce. An alternative to using the standard geometry with an expansion half-angle of 15° is to increase the expansion angle. The effect of this modification on the performance observed was investigated using this model, where \( \alpha \) was increased to 20°. The area ratio used remained the same, hence the nozzle length reduced slightly to accommodate the increased expansion angle. Four of the nine basic thrust levels were selected for this analysis. The nozzle length dimensions used are summarised in Table 6-3 and the geometry is illustrated in Figure 6-3.

<table>
<thead>
<tr>
<th>Model</th>
<th>A</th>
<th>B</th>
<th>G</th>
<th>I</th>
</tr>
</thead>
<tbody>
<tr>
<td>( L_n ) (mm)</td>
<td>2.13</td>
<td>0.476</td>
<td>0.213</td>
<td>0.095</td>
</tr>
</tbody>
</table>

Table 6-3: Length Parameters for Wide Expansion Angle Model
6.3.3 Sharp Throat Profile

These models were based upon the same dimensions developed initially, however the throat contour was removed and replaced with a sharp throat. The creation of a perfectly smooth throat contour when the throat diameter is of the order of 50 μm would be difficult to achieve with current manufacturing techniques. A sharp throat will cause a modification to the flow profile observed, therefore these models were designed to allow the impact of this on the performance to be evaluated. Three of the four thrust levels selected previously were again used for these models and Figure 6-4 illustrates this geometry.

![Figure 6-4: Schematic of the Sharp Throat Geometry](image)

6.3.4 Convergent Section Only

The predicted presence of a thick boundary layer within the nozzle may result in the presence of the nozzle being of little advantage. To test this four models were created based upon four thrust levels, without the divergent section to evaluate the modification in performance observed. The initial throat curvature was retained to allow some straightening of the flow prior to exit, as illustrated by Figure 6-5.

![Figure 6-5: Schematic of the Convergent Section Only Geometry](image)
6.4 Solution Methodology

The geometry for each of the models described was created and meshed using the commercial pre-processor Gambit. The creation of a mesh causes the model to be split into many smaller elements. In the case of a two-dimensional model, the elements may be either a three sided or a four sided shape. The quality of a mesh may be evaluated in a number of different ways, including aspect ratio and skewness.

In many cases different sizes of element will be required in different regions of the model. In order to generate a stable model, the mesh should be created such that the size of the elements changes slowly to prevent the generation of errors in the governing equations during solution. It is recommended that if the size of adjacent elements is changing, this should be by less than 20% each time. The aspect ratio and skewness of an element determine how close it is to a regular shape. These parameters are summarised in Table 6-4 together with their recommended range and their values in the models created.

<table>
<thead>
<tr>
<th>Parameter</th>
<th>Description</th>
<th>Recommended Range</th>
<th>Maximum Value in Models</th>
</tr>
</thead>
<tbody>
<tr>
<td>Skewness</td>
<td>Ratio of the maximum and minimum internal angles of an element relative to an equilateral element.</td>
<td>Triangular Element: 30° - 90° Quadrilateral Element: 60° - 120° Skewness &lt; 0.9</td>
<td>&lt; 0.56</td>
</tr>
<tr>
<td>Aspect Ratio</td>
<td>Ratio of the longest edge length to the shortest edge length of an element.</td>
<td>Less than 5:1</td>
<td>&lt; 1.8</td>
</tr>
</tbody>
</table>

Table 6-4: Mesh Quality Parameters

The models were developed to investigate the presence of a boundary layer within the expanding section of the nozzle. To allow complete resolution of the boundary layer, the geometry for each model used was split into a number of sections. This allowed the mesh to be graded such that a fine mesh was present in the nozzle throat region. When each geometry was imported into Fluent® the y+ value at the near-wall mesh node was calculated to ensure a value of less than 5. This ensured that several elements would be present within the boundary layer enabling resolution of the different regions within it.

An example of a geometry without the mesh is shown in Figure 6-6. Each of the lines represents a boundary and each area represents a zone. The primary attributes of these are set in Gambit and refined within Fluent®. A boundary can be set to represent a variety of features, including a wall, an interior, an inlet or an exit. The interior setting is of particular importance in these models as this means that it is in effect invisible to the flow. In Figure 6-6 the black line indicates a wall, the blue dashed line an inlet/exit, the green dashed line a symmetry plane and the red dotted lines interior boundaries.
The wall boundary condition requires the velocity of the flow immediately adjacent to it to be zero. The inlet boundary condition allows the properties of the flow to be set in terms of pressure and velocity at the inlet plane, while the exit boundary condition allows the conditions exterior to the model to be set.

An example of a mesh generated for these models is shown in Figure 6-7. The mesh was created using both square and triangular elements, with the lines meshed first to ensure that a smooth mesh resulted. The model shown is nozzle A and the number of mesh points used is 61069. As the size of the nozzle component reduced, more and more nodes were required to generate a mesh of the required density. For each of the models used the number of mesh points was in the range 61069 – 116410.
Following generation of the mesh, the geometry was then imported into Fluent® and additional parameters, including boundary conditions were defined. For each model the inlet boundary conditions were set to include a chamber pressure of 5 bar and freestream temperature of 873 K. In addition a difference was set between the static and the gauge pressures at the inlet plane of the model, this to generate a flow velocity in the x-direction. The calculations used to generate the dimensions of the nozzle provided the mass flow rate for each case. From this the velocity required was calculated using Equation 6-11.

\[ m = \rho \nu \]

**Equation 6-11: The Continuity Equation**

The flow velocity was transformed into a pressure drop input through use of Equation 6-12. This is based upon Bernoulli’s equation for incompressible fluid flow, assuming that the upstream velocity is approximately zero. Here \( \Delta p \) indicates the difference between the gauge total and initial gauge pressures, where the gauge total pressure was always set to 5 bar (506625 Pa).

\[ \Delta p = \frac{1}{2} \rho \nu^2 \]

**Equation 6-12: Pressure Drop Inlet Condition Calculation**

The region of the model exterior to the nozzle was set to represent a space environment. To achieve this the pressure was set to zero bar, the temperature to 4 K and the exit boundaries were set to be pressure outlets. The material selected to flow through the nozzle in the simulation was based upon the hot exhaust products of the decomposition reaction. It was a mixture of water and oxygen, the properties of which were calculated using a mass balance approach assuming complete decomposition of 85% concentration hydrogen peroxide. The material accounts for the high viscosity of the mixture, which is of particular interest when considering the boundary layer. The attributes of the material that were included are summarised in Table 6-5. The density of the fluid was assumed to be an ideal gas as the flow was expected to be supersonic.

<table>
<thead>
<tr>
<th>Property</th>
<th>Value</th>
</tr>
</thead>
<tbody>
<tr>
<td>Thermal Conductivity, ( k )</td>
<td>0.0547 Wm(^{-1})K(^{-1})</td>
</tr>
<tr>
<td>Specific Heat Capacity, ( c_p )</td>
<td>1596 Jkg(^{-1})K(^{-1})</td>
</tr>
<tr>
<td>Bulk Viscosity, ( \mu )</td>
<td>3.06 x 10(^{-5}) kgm(^{-1})s(^{-1})</td>
</tr>
<tr>
<td>Average Molecular Mass, ( \bar{M} )</td>
<td>21.822</td>
</tr>
</tbody>
</table>

**Table 6-5: Estimated Properties of the Exhaust Gas**

Following definition of a model, the solution was initialised and solved using the 2-dimensional double precision, axi-symmetric, coupled, implicit solver combined with the \( k-\varepsilon \) turbulence model. The 2-dimensional solver was used to minimise computational time, while the selection of double precision allowed an increase in accuracy. The axi-symmetric solver ensured the calculation of out of plane effects without the need for a full 3-dimensional model. Finally the
coupled implicit solver was used as it solves the governing equations simultaneously using both known and unknown variables from neighbouring cells and is well suited to supersonic flows.

To improve the stability of the solution, the standard $k$-$\varepsilon$ model was used to generate the initial solution and this was changed to the realisable $k$-$\varepsilon$ model for the final iteration run. In addition an enhanced wall treatment was used to ensure the correct treatment of the near-wall region.

Once initialised the solution was iterated until the residuals indicated a change of less than $10^{-3}$ in all cases. The solution procedure consisted of four stages. The first stage used a first order solution scheme for the flow as well as the turbulence kinetic energy and dissipation. Following convergence, the second stage changed the solution scheme for the flow to second order, until convergence was again achieved. The third stage changed the solution scheme for both the turbulence kinetic energy and dissipation to second order. Finally the fourth stage required the turbulence model to be changed to the realisable model. This method allowed the model to remain stable throughout the iteration procedure.

### 6.4.1 Limitations of Numerical Modelling

When considering the results from these models the particular conditions applied in each region should be noted. This is of particular importance with reference to the region exterior to the nozzle, where the conditions were set to be a vacuum. The exhaust flow will quickly become rarefied under these conditions and the Navier-Stokes equations used by Fluent® are no longer valid. The exterior region was included to ensure that the flow profile inside the nozzle was representative, however the data in this region itself is of little interest.

### 6.5 Presentation of Results

The results from each of the models created may be presented in a number of different formats dependent upon the particular aspects that are of interest. The various different forms of data presentation relevant to this research will be demonstrated to justify the reasoning for the approach used to evaluate the data generated.

Contours of velocity magnitude within the nozzle may be displayed as illustrated in Figure 6-8, this is model C, where the diameter of the throat is 0.432 mm. From this it is possible to identify the presence of a boundary layer within the divergent section of the nozzle reducing the expansion half-angle. The precise location of the edge of the boundary layer is difficult to determine exactly due to the cluster of contours present.
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Figure 6-8: Contours of Velocity Magnitude

Alternatively the data may be presented in a chart format, which provides an evaluation of the velocity at predefined locations along the nozzle axis. The position of these locations is shown in Figure 6-9, which is again model C.

Figure 6-9: Location of Sample Planes

The first three sample planes (Upstream Throat, Throat and Downstream Throat) are clustered around the throat to allow a better resolution of the boundary layer development. The remaining planes (1, 2, 3 and Noz Exit) are then spaced equidistant throughout the length of the nozzle with the last located at the exit plane of the nozzle. The velocity data relating to the six sample planes from model C are illustrated in Figure 6-10. The yellow and blue traces represent the velocity profiles immediately adjacent to the nozzle throat and share the same maximum radial coordinate.
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Figure 6-10: Plot of Velocity Magnitude vs. Radial Coordinate

The three profiles relating to locations close to the throat exhibit characteristics that are indicative of a distinct boundary layer. The trace increases to a distinct peak velocity before dropping sharply to zero. The peak indicates the edge of the boundary layer, with the sharp decrease in velocity indicating the high velocity gradient within it. Each of the traces that correspond to a location further downstream demonstrates a similar general shape, but the peak disappears altogether. This change in the shape of the velocity trace prevents the identification of the edge of the boundary layer as a function of the freestream velocity.

The alternative is to consider the rate of change of axial velocity with respect to radial position. At the edge of the boundary layer velocity of the flow within the boundary layer is the same as the free stream velocity, therefore the rate of change of axial velocity with respect to radial position will be zero.

Figure 6-11: Contours of the Rate of Change of Axial Velocity
Figure 6-11 shows contours of the rate of change of axial velocity with respect to the change in radial coordinate again for model C. The scale was set to consider any change that was less than zero, indicating deceleration of the flow. The edge of the boundary layer is then clearly visible. The bulges at the exit of the nozzle indicate the rapid change in direction of the flow that occurs at this point. This is illustrated in more detail by Figure 6-12, which shows the velocity vectors at this point in model C.

![Figure 6-12: Velocity Vectors at the Nozzle Exit](image)

The same sample planes that were used previously were used to generate Figure 6-13. This shows the magnitude of the rate of change of axial velocity with respect to the change in radial coordinate as a function of the radial position. It can be seen that the traces all cross the y-axis at approximately zero thereby indicating that little or no variation is present at the centre line of the nozzle. Each of the traces exhibits similar characteristics, with a period at approximately zero followed by a sharp turn to become negative. The radial coordinate at which this change in
direction occurs provides the location of the edge of the boundary layer. The gradient of the trace as it becomes progressively negative is indicative of the intensity of the shear stresses within the boundary layer at that location. In addition in this plot the three traces corresponding to the planes in the vicinity of the nozzle throat indicate an acceleration of the flow, which is to be expected. The trace that relates to the sample plane at the exit from the nozzle, shown in pink exhibits a corner like feature. This corresponds to the bulges noted in Figure 6-11, which represent the rapid turning of the flow as it exits the nozzle.

Identification of the precise location of the edge of the boundary layer is still difficult from this chart, therefore the data was exported and processed using MS Excel®. The results of this will be discussed in more detail in the following section.

6.6 Model Results

Four different nozzle geometries were investigated using numerical modelling. The performance of each type of geometry will be summarised first, followed by a comparison between the different geometries.

6.6.1 Standard Geometry Model Results

Each of the nine thrust levels described was evaluated using the standard geometry model. To ensure that the flow predicted by the models remained within the capabilities of the software, the Reynolds and Knudsen numbers of the flow were calculated for the throat conditions of each model. The figures calculated are shown in Table 6-6.

<table>
<thead>
<tr>
<th>Model</th>
<th>A</th>
<th>B</th>
<th>C</th>
<th>D</th>
<th>E</th>
<th>F</th>
<th>G</th>
<th>H</th>
<th>I</th>
</tr>
</thead>
<tbody>
<tr>
<td>$d^*(\text{mm})$</td>
<td>0.836</td>
<td>0.591</td>
<td>0.418</td>
<td>0.264</td>
<td>0.187</td>
<td>0.118</td>
<td>0.084</td>
<td>0.059</td>
<td>0.037</td>
</tr>
<tr>
<td>$Re$</td>
<td>16187.98</td>
<td>11446.12</td>
<td>8093.27</td>
<td>5134.40</td>
<td>3656.92</td>
<td>2313.75</td>
<td>1632.85</td>
<td>1140.84</td>
<td>715.84</td>
</tr>
<tr>
<td>$Kn$</td>
<td>7.36E-05</td>
<td>0.000104</td>
<td>0.000147</td>
<td>0.000234</td>
<td>0.000317</td>
<td>0.000511</td>
<td>0.000722</td>
<td>0.001039</td>
<td>0.001653</td>
</tr>
</tbody>
</table>

Table 6-6: Summary of Throat Reynolds and Knudsen Numbers

Overall the results from these models show that as the dimensions of the nozzle reduce, the boundary layer present within the nozzle at a given location thickens relative to the diameter at that point. The rate at which this occurs also increases with reducing nozzle dimension. This is demonstrated by Figure 6-14, which shows the position of the edge of the boundary layer in terms of a non-dimensional radial coordinate with respect to the throat diameter.
Figure 6-14: Assessment of the Position of the Edge of the Boundary Layer

The curves shown indicate that the rate at which the boundary layer is developing within the nozzle increases as the nozzle size reduces. To assess this directly the position of the boundary layer relative to the initial geometry was evaluated for each case and the effective expansion half-angle of the nozzle was calculated. This is illustrated by Figure 6-15, where the predicted position of the boundary layer for each thrust magnitude is shown relative to a nominal geometry. The data relating to the effective expansion half-angle of the nozzle, $\theta_{eff}$, is presented in Table 6-7.

Figure 6-15: Predicted Boundary Layer Position
Table 6-7: Summary of Data for Standard Geometry Models

The rate of boundary layer development is further demonstrated by Figure 6-16, which shows the effective expansion half-angle of the nozzle as a function of throat diameter. It illustrates that the growth rate observed exhibits a logarithmic behaviour similar to that shown in Figure 6-14. The reduction in the expansion half-angle indicates that progressively less expansion of the flow is occurring. In the case of the smallest model, Nozzle I, corresponding to a thrust level of 1 mN, barely any expansion of the flow is observed with an effective expansion angle of 4.23° resulting.

![Figure 6-16: Predicted Reduction in Nozzle Expansion Half-Angle](image-url)

The effective area of the nozzle at the throat and exit plane was calculated for each model to estimate the predicted reduction in area due to the presence of a boundary layer. To evaluate the predicted reduction in $c_f$ the effective exhaust velocity across the exit plane of each nozzle was also required. This was achieved by using a mass averaged approach to allow for the variation in velocity observed. This, in combination with the value of $c_f$ for each nozzle, allowed the calculation of $c_f$ in each case. The values calculated in each case are given in Table 6-7.
The data in Table 6-7 shows that overall the exit pressure in each case is comparable, indicating a similar level of expansion is achieved. Consideration of the \( c_f \) predicted in each case reveals that the performance of the nozzle reduces with decreasing dimension. This is a result of the relative boundary layer thickness increasing, reducing the throat and exit areas of the nozzle. The \( c_f \) efficiency estimate is based upon the ideal value of \( c_f \) for these nozzle geometries, which is the same in each case at 2.05. This value assumes perfect expansion and no losses. Inspection of the \( c_f \) efficiency reveals the extent of the reduction in performance in each case, with the smallest nozzle exhibiting the lowest efficiency as expected. Inspection of the magnitude of thrust predicted reveals that this is also affected by the presence of the thick boundary layer. The thrust efficiency data was calculated using an ideal thrust value in each case, based upon perfect expansion. This data again illustrates that the rate of performance loss increases.

### 6.6.2 Wide Expansion Angle Model Results

The results from the standard geometry models indicated that as the dimension of the nozzle reduced further, the effect of the presence of the boundary layer increased. The four models selected for investigation with an increased expansion half angle were A, E, G and I, corresponding to 500, 25, 5 and 1 mN of thrust respectively. It was decided that a cluster of models towards the smaller end of the thrust range investigated would provide the optimal data resolution possible in the time available.

A summary of the data generated by these models is presented in Table 6-8. Inspection of the exit pressure data reveals that a similar level of expansion is achieved in each case, although the \( c_f \) achieved varies significantly. In the case of the largest model, nozzle A, an improvement in both the \( c_f \) and \( T \) efficiency was achieved in comparison to the standard geometry. As the geometry reduces in size this improvement disappears, with a reduction in performance observed in the other three models. The overall implications of these results will be discussed further in section 6.7.2.

<table>
<thead>
<tr>
<th>Model</th>
<th>A</th>
<th>B</th>
<th>C</th>
<th>D</th>
</tr>
</thead>
<tbody>
<tr>
<td>( P_e ) (Pa)</td>
<td>12664.67</td>
<td>12665.63</td>
<td>12102.8</td>
<td>12064.19</td>
</tr>
<tr>
<td>( \sigma )</td>
<td>2.01</td>
<td>1.71</td>
<td>1.66</td>
<td>1.60</td>
</tr>
<tr>
<td>( c_f ) efficiency</td>
<td>97.68%</td>
<td>83.44%</td>
<td>80.63%</td>
<td>77.89%</td>
</tr>
<tr>
<td>( T ) (mN)</td>
<td>523.15</td>
<td>20.73</td>
<td>3.78</td>
<td>0.64</td>
</tr>
<tr>
<td>( T ) efficiency</td>
<td>91.56%</td>
<td>72.51%</td>
<td>65.56%</td>
<td>57.13%</td>
</tr>
<tr>
<td>Effective ( \alpha_{out} ) (°)</td>
<td>19.25</td>
<td>15.03</td>
<td>11.51</td>
<td>7.10</td>
</tr>
<tr>
<td>% Reduction in ( A^* )</td>
<td>6.26%</td>
<td>13.10%</td>
<td>18.69%</td>
<td>26.66%</td>
</tr>
<tr>
<td>% Reduction in ( A_e )</td>
<td>13.58%</td>
<td>39.57%</td>
<td>57.84%</td>
<td>77.88%</td>
</tr>
</tbody>
</table>

Table 6-8: Summary of Data for Wide Expansion Angle Models
6.6.3 Sharp Throat Profile Model Results

Figure 6-11 illustrated the position of the boundary layer in the standard geometry model with reference to contours of the change in axial velocity with respect to radial direction. From this image it is possible to see that the boundary layer begins to develop within the convergent section of the nozzle, approximately at the point where the curvature for the throat of the nozzle begins. To investigate the impact of this on the performance of the nozzle, three models were created with a sharp throat profile. A summary of the data generated by these models is given in Table 6-9.

<table>
<thead>
<tr>
<th>Model</th>
<th>A</th>
<th>E</th>
<th>I</th>
</tr>
</thead>
<tbody>
<tr>
<td>$P_e$ (Pa)</td>
<td>12004.86</td>
<td>11958.73</td>
<td>11671.27</td>
</tr>
<tr>
<td>$\gamma_f$ efficiency</td>
<td>93.22%</td>
<td>90.86%</td>
<td>87.65%</td>
</tr>
<tr>
<td>$T$ (mN)</td>
<td>512.16</td>
<td>24.38</td>
<td>0.87</td>
</tr>
<tr>
<td>$T_f$ efficiency</td>
<td>89.64%</td>
<td>85.27%</td>
<td>78.07%</td>
</tr>
<tr>
<td>Effective $\alpha_f$ ($^\circ$)</td>
<td>13.55</td>
<td>10.18</td>
<td>5.45</td>
</tr>
<tr>
<td>% Reduction in $A_f$</td>
<td>3.84%</td>
<td>6.15%</td>
<td>10.93%</td>
</tr>
<tr>
<td>% Reduction in $A_e$</td>
<td>14.52%</td>
<td>40.60%</td>
<td>71.56%</td>
</tr>
</tbody>
</table>

Table 6-9: Summary of Data for Sharp Throat Profile Models

The data from these models shows that there is little difference in the performance observed in comparison to the standard geometry. In general a smooth throat contour is used to provide a slow turning of the exhaust to prevent separation of the flow. If a flow separates within a nozzle a significant reduction in performance is observed. Consideration of the velocity contours from these models reveals that the flow remains attached at all times as shown in Figure 6-17.

Figure 6-17: Velocity Contours within a Sharp Throat Profile Model

The data from the sharp throat models reveals that overall the boundary layer grows at a rate similar to that observed for the standard geometry case and the $\gamma_f$ predicted is comparable. This confirms that there is no loss in performance as a result of the removal of the throat contour. In
terms of the $T$ predicted, the sharp throat profile produces improved results because the boundary layer at the throat is significantly reduced. In addition in the case of the smallest model, nozzle I, the thickness of the boundary layer is reduced throughout the nozzle, as illustrated by Figure 6-18. This is discussed further in section 6.7.2.

![Figure 6-18: Comparison of Boundary Layer Position](image)

### 6.6.4 Convergent Section Only Model Results

The final set of models considered the effect on the performance of the thruster if the expanding section of the nozzle was removed completely. Four thrust levels were considered and a summary of the data obtained is presented in Table 6-10. From the data presented it is clear that the lack of expansion of the flow would lead to a significant degradation in performance. The predicted $c_f$ and $T$ in each case are below 50% of the design values, with little variation between the models.

<table>
<thead>
<tr>
<th>Model</th>
<th>A</th>
<th>E</th>
<th>G</th>
<th>I</th>
</tr>
</thead>
<tbody>
<tr>
<td>$P_e$ (Pa)</td>
<td>311660.2</td>
<td>312001.1</td>
<td>311988.1</td>
<td>311722.4</td>
</tr>
<tr>
<td>$\eta_f$</td>
<td>0.904</td>
<td>0.880</td>
<td>0.872</td>
<td>0.905</td>
</tr>
<tr>
<td>$\eta_f$ - efficiency</td>
<td>43.98%</td>
<td>42.81%</td>
<td>42.45%</td>
<td>44.06%</td>
</tr>
<tr>
<td>$T$ (mN)</td>
<td>241.79</td>
<td>11.21</td>
<td>2.16</td>
<td>0.41</td>
</tr>
<tr>
<td>$T$ - efficiency</td>
<td>43.98%</td>
<td>42.81%</td>
<td>42.45%</td>
<td>44.06%</td>
</tr>
<tr>
<td>% Reduction in $A^2$</td>
<td>3.78%</td>
<td>8.42%</td>
<td>11.92%</td>
<td>16.39%</td>
</tr>
</tbody>
</table>

Table 6-10: Summary of Data from Convergent Section Only Models

The reasons for this reduced performance are partially revealed through examination of the velocity vectors at the exit from the model, shown in Figure 6-19. The image shows how the convergent section of the nozzle accelerates and turns the flow.

As the flow exits the nozzle a *vena contracta* appears in the flow, causing the core flow to narrow further and accelerate. Following this the flow quickly diverges and appears to accelerate further due to the vacuum conditions exterior to the nozzle. The rapid divergence of the flow reduces the axial velocity of the flow hence reducing the momentum and thrust of the process. The effect of this may be reduced if an additional straight section were included prior to the flow exit.
6.7 Discussion of Results

The results from the various models considered were presented in the preceding sections. Overall they indicate that both the size and geometry of the nozzle have a direct effect on the performance that results. The implications of these results will now be discussed. A prediction of boundary layer thickness for the standard geometry model will be proposed and evaluation of nozzle performance in terms of $C_T$ and $\eta$ efficiency for each geometry considered will be examined.

6.7.1 Boundary Layer Thickness Prediction

Overall the results from the models generated have indicated that the rate at which the boundary layer develops increases as the nozzle dimension reduces. The data from the standard geometry models led to the development of Figure 6-16, which showed effective expansion half-angle of the nozzle as a function of the nozzle throat diameter. Using a heuristic approach the equation of the curve that approximates the data points generated was created. Figure 6-20 shows this approximation together with the equation of the line. The curve created fits the data points to within 1% with the exception of the two smallest throat diameters, which it fits to within 2%. From this equation it is possible to predict the point where the boundary layer will fill the nozzle entirely. This data indicates that under the flow conditions considered a nozzle with a throat diameter of 0.023 mm, corresponding to a thrust level of 0.36 mN will be completely filled with a boundary layer.
Chapter 6: The Nozzle

6.7.2 Comparison of Results

The four different geometries investigated have revealed that a significant reduction in performance will occur in all cases as the dimensions of the nozzle reduce. Through comparison of the data from the different models the performance of each geometry can be evaluated with respect to the standard geometry.

The \(c_f\) and \(\eta\) efficiencies are compared for each type of geometry in Figure 6-21 and Figure 6-22 respectively. It is expected that if the additional data points were calculated for the wide expansion and sharp throat models the shape of the trends observed would imitate that of the standard geometry.

From the trends presented it is clear that all of the modifications to the geometry considered have a direct effect on the performance that results. Increasing the expansion angle causes a similar effect in both the predicted \(c_f\) and \(\eta\) efficiency. In the case of the largest models, a significant improvement is observed, due to the additional expansion that occurs. This improvement swiftly disappears as the nozzle dimensions reduce and the boundary layer rapidly fills more of the nozzle. In comparison in the case of the sharp throat model there is no improvement in the predicted \(c_f\) efficiency in any of the cases considered. In addition there is no loss in performance predicted, which indicates that it is possible to utilise a sharp throat profile without performance deficit.

Figure 6-20: Approximating the Effective Nozzle Expansion Half-Angle
The $c_f$ efficiency of the sharp throat model shows a significant increase in the thrust level predicted. This is due to the predicted reduction in the boundary layer thickness at the throat of the nozzle. The convergent section only trace, on both the $c_f$ and $T$ comparison charts, indicates a further reduction in performance in comparison to all of the other geometries investigated. The differences in the data presented in these two charts relates to the losses considered in each case. The $c_f$ efficiency only considers the area loss, while the $T$ efficiency also includes drag losses. This shows that while the predicted performance of each of the geometries is significantly reduced, the presence of the divergent section still contributes appreciably to the resultant performance.
6.8 Conclusions

This chapter has considered the effect of miniaturisation on the performance of a nozzle. Through creating and solving a series of models using Fluent® the development of a boundary layer within the exhaust nozzle of a hydrogen peroxide thruster has been investigated. The results from the models have revealed that the size of the nozzle has a significant impact on the boundary layer that develops.

The rate at which the boundary layer develops with reducing nozzle dimension is highly non-linear and causes a significant reduction in the resultant performance at small scales. In the smallest case considered, corresponding to a thrust level of 1 mN, the nozzle divergence half-angle was reduced to 4° due to the presence of a thick boundary layer. Based upon the results from these models a relation between the throat diameter of the nozzle, \( d' \), and the effective nozzle expansion half-angle, \( \alpha_{\text{eff}} \), was proposed and is given in Equation 6-13. This relation is valid for throat diameters corresponding to the thrust range considered, 1 - 500 mN.

\[
\alpha_{\text{eff}} = 3 \ln(d' - 0.0188) + 0.0925d'^2 - 2.2978d' + 16.418
\]


From this equation it is predicted that the throat diameter corresponding to the point where the boundary layer will fill the nozzle entirely is 0.023 mm. This corresponds to a thrust level of 0.36 mN. In addition a critical throat diameter of 0.273 mm is proposed, which indicates the point where the presence of a boundary layer within the nozzle should be considered at the design stage.

The three additional geometries considered in this analysis show that small modifications in the geometry could result in significant changes in performance at small scales. The wide expansion angle model showed performance improvement at large scales, but no improvement at small scales. The sharp throat model demonstrated that the throat contour contributes little to the performance observed. In addition it showed that this geometry could result in an improvement in the thrust generated. Finally the convergent section only model showed that significant reductions in performance would result if the divergent section were removed altogether.
7 Conclusions & Future Work

The motivation for this research was to address the need for a propulsion system that was suitable for accommodation onboard a nanosatellite platform. This chapter will summarise the research completed towards this goal and detail how it fulfils the various aims set out in chapter 1. The contributions of this work to the present state of the art will then be summarised. Finally recommendations for future work to develop the research presented here into a flight-ready propulsion system will be outlined.

7.1 Summary of Conclusions

This research has demonstrated that it is possible to miniaturise a monopropellant thruster. In addition it has highlighted some of the complications associated with the process. The conclusions from this research are now summarised with reference to the different stages of research.

7.1.1 Derivation of Propulsion System Requirements

The selection of a nanosatellite platform, PalmSat, allowed the assessment of various mission scenarios and the derivation of a set of suitable mission requirements. Assessment of these requirements led to the following conclusions:

- At present no micropropulsion system development is capable of meeting the derived mission requirements within the constraints imposed by PalmSat.
- A requirement for a monopropellant system miniaturised such that it is capable of producing thrusts of 100 mN and 10 mN and operating for 5s durations was identified.
- Hydrogen peroxide is the most suitable monopropellant for use in this thruster due to its known characteristics in terms of catalytic behaviour and handling requirements.
- Miniaturisation of a decomposition chamber will cause an increase in thermal losses as the surface area to volume ratio increases.
- Miniaturisation of fluid channels will cause the relative effect of surface forces within a fluid to increase.
- Miniaturisation of the decomposition chamber and exhaust nozzle should be considered in detail to optimise performance.
7.1.2 The Decomposition Chamber

The decomposition characteristics of hydrogen peroxide are well understood due to its extensive use in the 1960's. Assessment of the recommendations for catalyst bed design from this time revealed that they were not appropriate for use at the thrust levels under consideration. An empirical approach was then used to determine the optimal geometry and morphology of a catalyst bed suitable for use in a thruster producing a thrust in the range 10 - 100 mN. This research generated the following conclusions:

- Use of a monolithic substrate supporting a manganese oxide catalyst generates good decomposition of hydrogen peroxide from a chemistry perspective.
- The energetic nature of the reaction associated with the decomposition of hydrogen peroxide when catalysed by manganese oxide causes severe structural degradation of a low density alumina foam substrate.
- A substrate of increased density together with even catalyst deposition is required to develop this catalyst bed into a suitable structure for use in a monopropellant thruster.
- A compressed silver powder catalyst bed, 6.7 mm in diameter and 5 mm in length is capable of operating at a propellant mass flow rate in excess of 4 gs⁻¹ and for a total duration in excess of 15 minutes.
- The propellant mass flow rate through a compressed silver powder catalyst bed influences the decomposition temperature achieved, with an increase in temperature observed at higher propellant mass flow rates.
- For a given propellant mass flow rate the shortest catalyst bed generated the highest decomposition temperatures for a given diameter of catalyst bed.
- Miniaturisation of a compressed silver powder catalyst bed improves performance at low propellant mass flow rates.

7.1.3 The Nozzle

The performance of the exhaust nozzle is critical to the overall performance of the thruster. The development of a boundary layer within a nozzle will reduce the flow expansion causing a performance deficit. As the dimensions of the nozzle are reduced, the boundary layer begins to consume progressively more of the available volume. Numerical evaluation of a rocket nozzle led to the following conclusions:

- The rate of boundary layer development is highly non-linear, with significant performance reductions occurring within small dimension nozzles.
Chapter 7: Conclusions & Future Work

- A logarithmic relation exists between the throat diameter of a nozzle and the effective nozzle expansion half-angle as a result of the presence of a boundary layer. In addition a thrust magnitude of 50 mN corresponds to a critical point, below which the rate of boundary layer development increases.

- A throat diameter of 0.023 mm, corresponding to a thrust of 0.36 mN corresponds to the point where the boundary layer completely fills the nozzle throat.

7.2 Research Achievements

This research set out to fulfil the six aims set out in chapter 1. The PalmSat platform was selected to allow evaluation of the constraints placed upon a propulsion system. This addressed the first aim. The second two aims were concerned with the selection of a suitable candidate mission and determination of the most appropriate propulsion technology. The derivation of the mission and propulsion system requirements in chapter 3 met these aims.

In order to create a monopropellant thruster that operates efficiently at low thrust magnitudes, the physical implications of miniaturisation were considered. Assessment of manufacturing techniques together with analysis of modifications in heat transfer and fluid flow characteristics as a result of scaling enabled the consequences of miniaturisation to be highlighted. This was the subject of the fourth aim.

The fifth aim required the performance of a monopropellant decomposition chamber to be characterised as it was miniaturised. The research summarised in chapter 5 addressed this aim, through the development of two different types of catalyst bed. The effect of catalyst geometry and morphology was tested and the resultant performance evaluated.

The final aim was concerned with the evaluation of the performance of a rocket nozzle as it was miniaturised. This was addressed through the use of numerical modelling, to provide an indication as to the development of a boundary layer within the nozzle under steady state conditions.

7.3 Contribution to the State of the Art

The research presented has addressed the gap in published data relating to hydrogen peroxide thrusters and contributes to the state of the art in the following ways:

- Successful miniaturisation of a decomposition chamber capable of stably and repeatedly decomposing a flow of hydrogen peroxide at a rate of 0.5 g s\(^{-1}\) through an un-insulated catalyst bed 6.7 mm in diameter and 5 mm in length.
• Identification of a trend in the performance of a decomposition chamber according to the diameter of the chamber and mass flow rate of propellant.

• Identification of a trend in the development of a boundary layer within a thruster nozzle that suggests a logarithmic rate of growth.

• Identification of a critical point relating to nozzle size, which corresponds to the size of nozzle that is the smallest that may be considered without boundary layer analysis.

### 7.4 Future Work

Overall this research has addressed the need to understand the effect of miniaturisation on the performance of a monopropellant propulsion system. The conclusions from this research indicate that it is possible to miniaturise a monopropellant system for use onboard PalmSat, however additional development is required to extend the research completed into a flight-ready system. There are several areas that require attention, which are detailed in the following sections.

#### 7.4.1 Decomposition chamber

The research completed indicates that silver powder packed into a catalyst bed, 6.7 mm in diameter and 5 mm in length will operate efficiently at a propellant mass flow rate in excess of 1.25 $\text{g s}^{-1}$. The mission requirements developed for PalmSat indicate the need for a propellant mass flow rate ranging from 0.01 - 0.1 $\text{g s}^{-1}$. These values are up to two orders of magnitude lower than those currently in use. In order to complete this mission successfully additional miniaturisation of the decomposition chamber is therefore required. To determine the dimensions of the catalyst bed required an extension of the experimental work conducted for this research is recommended. To minimise instabilities in the results generated it is recommended that a new test set up is developed to reduce the length of feed lines between the decomposition chamber and the propellant tank. In addition the decomposition chamber itself should be miniaturised further to minimise the surrounding material and hence reduce thermal losses. The addition of an addition coriolis mass flow meter capable of operating at very low rates would further improve the accuracy of the results generated.

The conclusions from the testing completed indicate that to improve the overall efficiency of decomposition the thermal characteristics of the decomposition chamber should be addressed and a suitable heat shield should be installed. This will also allow the start-up characteristics of the system to be evaluated and optimised for short duration operation. In addition the decomposition characteristics of the compressed silver catalyst bed at low temperatures should be investigated further.
It is estimated that operating at a propellant mass flow rate of $1.25 \text{ g/s}^1$ the system would produce a thrust of approximately 1 N. Therefore if a short demonstration mission is of interest then the catalyst bed described may be appropriate.

The research conducted into the monolithic catalyst bed was suspended due to inconsistencies in the deposition of the catalyst material. The initial testing results indicated a good decomposition characteristic, therefore it is recommended that this avenue of research is pursued if a reliable deposition method is found. A key reason for researching the use of a manganese oxide catalyst was its low temperature capabilities.

### 7.4.2 Exhaust nozzle

The numerical modelling completed in chapter 6 investigated the effect of a boundary layer in a rocket nozzle under steady state conditions. The mission requirements indicate that the thruster will operate for durations of 5 - 12 s. The development of the boundary layer under transient conditions will therefore be important. The time taken for the boundary layer to develop and stabilise will influence the thrust generated in this time, therefore should be considered.

The material developed to represent the flow of exhaust gases in the nozzle was an estimate based upon the relative masses of the different components. This represented the viscosity of the fluid, but did not account for any fluctuations in the physical properties due to pressure and temperature. It is therefore recommended that additional modelling should be performed to investigate the influence of these properties on the results generated. In addition the results from the models created would be further enhanced through the addition of experimental data.

### 7.4.3 Propellant feed system

In order to supply propellant to the thruster at the required time and rate the propellant feed system should be capable of operating efficiently and quickly. To achieve this within the mass and power constraints imposed by PalmSat miniaturised valves will be required. Due to the other micropropulsion developments currently under way some miniaturised commercial valves are available, however the compatibility and performance of these will require detailed assessment.

### 7.4.4 Propellant storage

Storage of hydrogen peroxide on orbit is a complex issue due to the propellant instabilities. It will decompose at rate of approximately 1% per year, releasing gaseous oxygen in the process. A previous research study demonstrated that through the use of a gas permeable liner it was possible to passively store hydrogen peroxide in a sealed container for a period of 18 days [Coxhill'02]. The conclusion from this research was that the system would be appropriate for use for up to one
year. Further investigation of this propellant tank would be required to ensure the long-term capability required.

7.5 Publications

The work in this thesis has been presented to audiences via the following publications:


The first publication describes the development of and testing of a monolithic catalyst bed, presented in chapter 5 of this thesis. The presentation of this paper at the conference was supported by awards from the Royal Academy of Engineering and the School of Engineering and Physical Sciences at the University of Surrey.

The second paper details the numerical simulations conducted to evaluate the performance of a nozzle, as described in chapter 6.

The final publication summarises the results from the testing of the compressed silver powder catalyst bed, as described in chapter 5. The paper has been accepted for publication and will be presented in July 2006.
Bibliography


Bayt'97

Bayt'98

BECCO'54

Bermyn'00

Berner'02

Brutin'03

Cheung'01

Chojnacki'99

Choudhuri'00

Choudhuri'01a

Choudhuri'01b


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