An Investigation of a Low Cost Bi-Propellant Rocket Engine for Small Satellites

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"Those who cannot remember the past are condemned to repeat it"
Abstract

The goal of the research was to investigate a low cost bi-propellant engine of 40N thrust, suitable for integration into a small spacecraft. Small spacecraft are often launched as secondary payloads, where the primary spacecraft dictates the orbit. Thus propulsion is fast becoming a necessity to change the initial orbit into the mission-required orbit. The low-cost driver forced the use of non-toxic or ‘green’ propellants and the imposed volume constraints drove the requirement for a high density Isp.

The propellant combination chosen was kerosene and hydrogen peroxide (HTP), both readily available and cheap liquids. Unfortunately HTP suffers from the intrinsic decomposition into oxygen and water during storage, which in a sealed tank causes the pressure to rise until rupture occurs. The compatibility of fluoropolymer and aluminium tank liner materials was measured through decomposition rate determination, chemical analysis and mechanical pull tests. The results demonstrated that while pure aluminium is superior to the fluoropolymer FEP, the latter has the ability to passively vent oxygen. This enabled a passively vented tank to be produced, which demonstrated sealed storage over 18 days.

In order to maintain the low cost approach, high temperature alloys for the thruster were avoided. A range of cooling options were considered, however for the purpose of maximising the density Isp and minimising the complexity of the engineering, the novel decomposed HTP cooling system was chosen. The theory was evaluated with five practical engines built and tested. Combustion experiments determined the ignition conditions, combustion efficiency and stability. Cooling experiments were executed to determine the effectiveness of nitrogen and decomposed hydrogen peroxide coolant for the chamber and nozzle of the engine. The test data showed that this form of cooling is effective and a density Isp performance comparable to a commercial hypergolic bi-propellant engine is possible.
Summary

The capabilities of small satellites are constantly increasing and propulsion is fast becoming a requirement for a great number of missions. The majority of small satellite launches are as secondary payloads where the primary payload will dictate the orbit, however this orbit may not be ideal for the secondary satellite. A satellite will have much greater mission flexibility if it possesses the ability to change its own orbit – this in turn also maximises the available launch opportunities. On board propulsion also opens up the possibility of interplanetary missions for small satellites – offering the possibility of big science on a small budget.

The key to success for small satellites is achieving good performance but at a low cost and on rapid timescales and this philosophy also extends to propulsion. Traditional hypergolic bi-propellant rocket engines, while providing high performance, are correspondingly expensive. Hydrazine and nitrogen tetroxide are both extremely hazardous propellants due to their toxicity and carcinogenic nature. These propellants therefore require expensive self-contained breathing apparatus for personnel and a plethora of other ground support equipment. Alternatives to these engines do exist – such as electrothermal resistojets with propellants as friendly as water - however these provide relatively low thrusts and require large electrical input powers which typically is in scant supply on small spacecraft. Non-toxic, commonly referred to as ‘green’, monopropellant systems using for example Nitrous Oxide may provide a solution, however the propellant density is low, which for volume limited small spacecraft, is unattractive. Hybrid engines, using inert solid fuels and non-toxic liquid oxidisers, have been developed for spacecraft use and provide good Isp, however it has become apparent that hybrid engines do not scale well in terms of delta-V.

The goal of this research was to develop an engine, suitable for integration into a small satellite, providing 40N thrust with the flexibility of a bi-propellant engine but incurring low material cost and using green propellants. The propellant combination chosen was kerosene and hydrogen peroxide, both readily available and cheap liquids. Neither propellant poses a significant risk to the environment and hence requires only minimal safety equipment. In addition a high Isp is achievable for a flame temperature cooler than a hypergolic bi-propellant engine.

The specific problems with the use of hydrogen peroxide and kerosene in a bi-propellant rocket engine centre on the issues of engine cooling and oxidiser storage. While a high Isp is achievable in an un-cooled engine, with conventional (and hence low cost) materials, the engine would quickly burn out due to the high temperatures. Thus a cooling system is required that
allows maximum performance to be achieved whilst using materials that are easily procured and manufactured. The requirement to limit cost excludes the use of refractory metals and directed the emphasis onto stainless steel and high temperature super alloys such as Haynes and Inconels.

Hydrogen peroxide suffers from the intrinsic decomposition into oxygen and water during storage, which in a sealed tank causes the pressure to increase until rupture occurs. Thus the long-term storage of hydrogen peroxide was investigated with the aim of reducing the overall decomposition rate and developing a method of pressure relief in a zero-g vacuum environment without contaminating the spacecraft. Research into the oxidiser tank construction demonstrated that, while pure aluminium is superior to PFA and FEP in terms of compatibility, the use of fluoropolymers with their ability to permeate oxygen enables a passively vented tank system where peroxide could be sealed and stored without undue pressure build-up. Both PFA and FEP demonstrated remarkable resistance to hydrogen peroxide and, even after 4 months of exposure, there was no discernable surface chemistry difference. Similarly, hydrogen peroxide stored with the polymers showed no signs of reduced performance when catalytically decomposed. A passively vented tank was constructed which demonstrated that safe storage of hydrogen peroxide for extended periods without degradation in performance is possible.

In order to use low-cost materials for the engine, the nozzle temperature needs to be minimised during operation and a novel method of film-cooling was proposed. The theoretical analysis of engine cooling options showed that the decomposed peroxide gas coolant system has significant advantages, particularly in the context of simplified engineering. Regenerative cooling was determined not to be feasible on the entire engine due the low oxidiser flow rate. Fuel film cooling was demonstrated to be a viable option, however as it uses the low-density propellant for cooling, the overall density Isp suffered. The theory was evaluated with five practical engines built and tested. Initially a combustor using low cost components to limit production costs was developed and, once an efficient combustor was successfully operating, the cooling issue was investigated using a number of designs. With the inclusion of a flow separator, the decomposed hydrogen peroxide cooling method was demonstrated to cool the entire engine adequately to allow prolonged operation. Pivotal to the success was the Haynes 230 nozzle, which demonstrated remarkable oxidation resistance and high temperature survivability. The impulse per unit volume of the developed engine was similar to an equivalent commercial hypergolic thrusters. This demonstrated that for a volume-limited small spacecraft, comparable performance to a commercial hypergolic engine is possible but at approximately 25% of the manufacturing cost. When taking account of the simplified propellant handling, satellite integration and safety requirements, the final cost of the hydrogen peroxide/kerosene engine is expected to be around 10% of the comparable hypergolic engine.
The research has concluded that a decomposed film-cooled hydrogen peroxide/kerosene bi-propellant engine provides a low-cost, safe and efficient solution for small satellites. This paves the way for high volumetric efficiency propulsion, rivalling commercial hypergolic propulsion systems. The long-term safe storage of hydrogen peroxide, has also been demonstrated.

In summary, the research has both theoretically and practically demonstrated the design for a low cost thruster, suitable for small spacecraft, and a storage system for hydrogen peroxide. The work has contributed the following to the current state-of-the-art in satellite propulsion:

- First demonstration of decomposed hydrogen peroxide film cooling of a bi-propellant engine
- Demonstration of a low-cost hydrogen peroxide/kerosene combustor
- Identification of suitable nozzle material for gaseous oxidiser-cooled engines
- Demonstration of a permeable tank for long-term, safe storage of hydrogen peroxide
- New data, in terms of chemical compatibility and mechanical strength, of PFA and FEP when in contact with concentrated hydrogen peroxide.

Five international conference papers have been presented and one paper has been submitted to refereed journals for publication. A second journal paper is currently in progress. The BNSC funded a research contract to investigate green bi-propellant propulsion and two progress reports have been submitted on this work.
Acknowledgments

The author would like to warmly thank all those who contributed to this research programme, for without their help this would not have been possible. In particular the author would like to thank Malcolm Paul for his unerring support and guidance on what proved to be an interesting and rewarding subject. His experience and knowledge of propulsion systems in general and in particular Hydrogen Peroxide and Kerosene systems proved vital to the success of this project. Without his help, the research simply would not have accomplished what it did.

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## Nomenclature

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<th>Specific Subscript</th>
<th>Definition</th>
</tr>
</thead>
<tbody>
<tr>
<td>( R )</td>
<td></td>
<td>Universal gas constant = 8.3143 J mol(^{-1}) K(^{-1})</td>
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<tr>
<td>( \alpha )</td>
<td></td>
<td>Thermal conductivity divided by volumetric heat capacity (m(^2)s(^{-1}))</td>
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<td></td>
<td>Nozzle</td>
<td>Divergence half angle of the nozzles exit cone (°)</td>
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<tr>
<td>( w )</td>
<td></td>
<td>Absorptivity of the chamber wall, always taken as unity.</td>
</tr>
<tr>
<td>( \beta )</td>
<td></td>
<td>Volumetric thermal expansion coefficient (K(^{-1}))</td>
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<tr>
<td>( \Delta V )</td>
<td></td>
<td>Velocity change (ms(^{-1}))</td>
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<tr>
<td>( \gamma )</td>
<td></td>
<td>Ratio of the specific heats ( c_p / c_v )</td>
</tr>
<tr>
<td>( \epsilon )</td>
<td></td>
<td>Emissivity</td>
</tr>
<tr>
<td>( \eta )</td>
<td>( T )</td>
<td>Thermal effectiveness parameter, defined in equation 2.44</td>
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<tr>
<td>( \theta )</td>
<td>( \text{inc} )</td>
<td>Inclination change, angle (°)</td>
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<td>( \kappa )</td>
<td>1</td>
<td>Liquid phase homogeneous decomposition rate, s(^{-1})</td>
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<tr>
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<td>2</td>
<td>Liquid phase heterogeneous decomposition rate, s(^{-1})</td>
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<td></td>
<td>3</td>
<td>Vapour phase heterogeneous decomposition rate, s(^{-1})</td>
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<td>( \lambda )</td>
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<td>Nozzle efficiency due to thrust divergence</td>
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<tr>
<td>( \mu )</td>
<td></td>
<td>Absolute viscosity (kg m(^{-1}) s(^{-1}))</td>
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<tr>
<td>( k )</td>
<td></td>
<td>Kinematic viscosity (m(^2)s(^{-1}))</td>
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<tr>
<td>( g_c )</td>
<td></td>
<td>Product of Universal Gravitational Constant (G) and mass of the Earth = 3.986 \times 10^{14} m^3 s^{-2}</td>
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<td>( \Pi )</td>
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<td>Permeability of polymer (mm cm(^{-3}) m(^{-2}) day(^{-1}) bar(^{-1}))</td>
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<tr>
<td>( \rho )</td>
<td></td>
<td>Density (kg m(^{-3}))</td>
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<td></td>
<td>( a )</td>
<td>Density of coolant (air) (kg m(^{-3}))</td>
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<td>( g )</td>
<td>Density of hot gas (kg m(^{-3}))</td>
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<td></td>
<td>( s_g )</td>
<td>Average specific gravity of all the propellants in a complete system.</td>
</tr>
<tr>
<td>( \rho_{\text{Isp}} )</td>
<td></td>
<td>Product of the average specific gravity of the propellants and the Isp, termed density Isp (s)</td>
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<tr>
<td>( \sigma )</td>
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<td>Stefan-Boltzmanns Constant = 5.6696 \times 10^{-8} W m^{-2} K^{-4}</td>
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<td>Symbol</td>
<td>Definition</td>
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<tr>
<td>$\tau$</td>
<td>Polymer thickness (mm)</td>
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<td>$\psi$</td>
<td>Convective cooling loss coefficient (WK$^{-1}$)</td>
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<tr>
<td>$a$</td>
<td>Semi Major Axis (m)</td>
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<td>$h$</td>
<td>Hole diameter (m)</td>
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<tr>
<td>$A$</td>
<td>Cross sectional area ($m^2$)</td>
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</tr>
<tr>
<td>$\text{sur}$</td>
<td>Surface area ($m^2$)</td>
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<tr>
<td>$c$</td>
<td>Specific heat capacity at constant pressure (Jkg$^{-1}$K$^{-1}$)</td>
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</tr>
<tr>
<td>$v$</td>
<td>Specific heat capacity at constant pressure (Jkg$^{-1}$K$^{-1}$)</td>
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<tr>
<td>$C$</td>
<td>HTP Concentration of hydrogen peroxide by mass (%) 100% = Pure $H_2O_2$</td>
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<td></td>
<td>2 or 5 Elastic deformation coefficients</td>
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<td>$C^*$</td>
<td>Characteristic velocity ($ms^{-1}$) defined in equation 2.30</td>
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<tr>
<td>$\text{CF}$</td>
<td>Leak compensation factor</td>
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<tr>
<td>$d$</td>
<td>Diameter (m)</td>
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<tr>
<td>$4\text{hyd}$</td>
<td>$4 \times$ Hydraulic Depth. Hydraulic Depth is defined as the cross sectional flow area divided by the wetted perimeter (m)</td>
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<td>$D$</td>
<td>Plate Constant</td>
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<td>$E$</td>
<td>Orbital (Kinetic + Potential) Energy (J)</td>
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<tr>
<td>$m$</td>
<td>Elastic Modulus (Nm$^{-2}$)</td>
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<td>$f$</td>
<td>1 Pressure rise rate (Bar day$^{-1}$)</td>
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<td></td>
<td>2 Leak rate (day$^{-1}$)</td>
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<td>$g$</td>
<td>0 Acceleration due to gravity at sea level = 9.80665 $ms^2$</td>
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<td>Convective heat transfer coefficient (Wm$^{-1}$K$^{-1}$)</td>
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<tr>
<td>$H$</td>
<td>Enthalpy (Jkg$^{-1}$)</td>
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<td>$c$</td>
<td>Enthalpy of gas in the chamber (Jkg$^{-1}$)</td>
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<tr>
<td>ablative</td>
<td>Enthalpy of material ablation (Jkg$^{-1}$)</td>
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<tr>
<td>$fg$</td>
<td>Enthalpy of vaporisation (Jkg$^{-1}$)</td>
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<tr>
<td>$i$</td>
<td>index</td>
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<tr>
<td>$\text{Isp}$</td>
<td>Specific Impulse (s)</td>
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<tr>
<td></td>
<td>Overall Specific Impulse once cooling requirements have been considered (s)</td>
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<td>main Isp of the propellant flow used for combustion only (s)</td>
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<tr>
<td>$k$</td>
<td>Thermal conductivity (Wm$^{-1}$K$^{-1}$)</td>
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<tr>
<td>$K$</td>
<td>orif Head loss coefficient for an orifice</td>
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<tr>
<td>$L$</td>
<td>Length (m)</td>
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<td>11,14 Elastic deformation coefficients</td>
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<td>Symbol</td>
<td>Definition</td>
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<td>--------</td>
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<tr>
<td>( L^* )</td>
<td>'star' of the chamber, defined by equation 3.7. Units are (m) in SI, however rocketry commonly uses inches</td>
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<tr>
<td>( m )</td>
<td>Instantaneous mass of spacecraft (kg)</td>
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</tr>
<tr>
<td>( f )</td>
<td>Final mass of spacecraft (kg)</td>
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</tr>
<tr>
<td>( i )</td>
<td>Initial mass of spacecraft (kg)</td>
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<tr>
<td>( \text{prop} )</td>
<td>Mass of propellants (kg)</td>
<td></td>
</tr>
<tr>
<td>( r )</td>
<td>Molecular mass (kgmol(^{-1}))</td>
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<tr>
<td>( \dot{m} )</td>
<td>Mass flow (kgs(^{-1})). Calculations use the SI unit, however for convenience all flow rates are given in gs(^{-1})</td>
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<td>( M )</td>
<td>Mach number = ( \frac{V}{V_s} )</td>
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<td>( n )</td>
<td>Number of subscript</td>
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<td>( p )</td>
<td>Poisson ratio</td>
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</tr>
<tr>
<td>( P )</td>
<td>Pressure (Nm(^{-2}))</td>
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</tr>
<tr>
<td>( e )</td>
<td>Exhaust plane pressure (Nm(^{-2}))</td>
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<tr>
<td>( \dot{Q} )</td>
<td>Heat Power (W)</td>
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<tr>
<td>( \dot{Q}_{\text{total}} )</td>
<td>Sum of all the heat fluxes associated with the engine (W)</td>
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</tr>
<tr>
<td>( r )</td>
<td>Radius (m)</td>
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<tr>
<td>( R )</td>
<td>Instantaneous orbital radius (m)</td>
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<tr>
<td>( g )</td>
<td>Gas constant for gas in question = ( \mathcal{R} / m_r ) (m(^2)s(^{-2})K(^{-1}))</td>
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<td>( T )</td>
<td>Absolute temperature (K)</td>
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<td>Coolant (air) temperature</td>
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<td>( \text{vap} )</td>
<td>Temperature of vaporised film coolant (K)</td>
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<tr>
<td>( t )</td>
<td>Time (s)</td>
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<td>( \text{burn} )</td>
<td>Burn duration (s)</td>
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<td>( V )</td>
<td>Speed of fluid flow (ms(^{-1}))</td>
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<td>( e )</td>
<td>Effective exhaust speed (ms(^{-1}))</td>
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<td>( s )</td>
<td>Sonic speed (ms(^{-1}))</td>
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<td>Velocity of satellite (ms(^{-1}))</td>
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<td>Coolant (air) speed (ms(^{-1}))</td>
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<td>( g )</td>
<td>Hot gas speed (ms(^{-1}))</td>
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<td>( v )</td>
<td>Volume (m(^3))</td>
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<td>( W )</td>
<td>Width (m)</td>
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<td>c</td>
<td>Property of gas in chamber region</td>
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<tr>
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<td>Parallel chamber section</td>
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<td>Convergent section of the nozzle</td>
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<td>Property of combustion gases</td>
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<td>Hydrogen peroxide</td>
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<td>Transfer of parameter into engine</td>
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<td>Injector</td>
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xviii
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<td>Cool side of chamber wall</td>
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<td>wh</td>
<td>Hot side of chamber wall</td>
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## Abbreviations

<table>
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<tr>
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<td>Attitude Control System</td>
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<td>Analogue to Digital</td>
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<td>ADC</td>
<td>Analogue to Digital Converter</td>
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<td>AFRL</td>
<td>Air Force Rocket Laboratory</td>
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<td>AKM</td>
<td>Apogee Kick Motor</td>
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<td>AOL</td>
<td>Active Oxygen Loss</td>
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<td>Ariane Structure For Auxiliary Payloads</td>
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<td>Atmosphere</td>
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<td>Beginning of Life</td>
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<td>Commercial Off The Shelf</td>
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<td>GSE</td>
<td>Ground Support Equipment</td>
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<td>Geostationary Transfer Orbit</td>
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<td>H2O2</td>
<td>Hydrogen Peroxide</td>
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<td>HAN</td>
<td>Hydroxyl Ammonium Nitrate</td>
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<tr>
<td>Abbreviations</td>
<td>Definition</td>
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<tr>
<td>HDPE</td>
<td>High Density Polyethylene</td>
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<td>HTP</td>
<td>High Test Peroxide (Specifically 90%)</td>
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<td>HTPB</td>
<td>Hydroxyl Terminated Polybutyl Rubber</td>
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<td>ICBM</td>
<td>Inter-Continental Ballistic Missile</td>
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<td>ICP-MS</td>
<td>Inductively Coupled Plasma Mass Spectrometer</td>
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<td>Liquid Engine Royal Ordnance Space</td>
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<td>LTT</td>
<td>Low Thrust Thruster</td>
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<td>MEMS</td>
<td>Micro Electro Mechanical Systems</td>
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<td>MMH</td>
<td>Monomethyl Hydrazine</td>
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<tr>
<td>MON</td>
<td>Mixed Oxides Of Nitrogen, NTO + NO</td>
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<tr>
<td>MSL</td>
<td>Mean Slant Length</td>
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<td>N₂O</td>
<td>Nitrous Oxide</td>
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<td>NASA</td>
<td>National Aeronautics And Space Administration</td>
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<td>NHMF</td>
<td>Non Toxic Hypergolic Miscible Fuels</td>
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<td>Nitrogen Tetroxide</td>
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<td>Polyvinyl Chloride</td>
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</tr>
<tr>
<td>RAF</td>
<td>Royal Air Force</td>
</tr>
<tr>
<td>RATO</td>
<td>Rocket Assisted Take Off</td>
</tr>
<tr>
<td>RCS</td>
<td>Reaction Control System</td>
</tr>
<tr>
<td>RFC</td>
<td>Radiation/Film Cooled</td>
</tr>
<tr>
<td>RPM</td>
<td>Revolutions per minute</td>
</tr>
<tr>
<td>RF</td>
<td>Radio Frequency</td>
</tr>
<tr>
<td>RS</td>
<td>Radio Spares, Engineering Parts Supplier</td>
</tr>
<tr>
<td>sat</td>
<td>Saturated</td>
</tr>
<tr>
<td>S/V</td>
<td>Surface To Volume Ratio</td>
</tr>
<tr>
<td>SCAPE</td>
<td>Self Contained Atmospheric Ensemble</td>
</tr>
<tr>
<td>SH</td>
<td>Super Heated</td>
</tr>
<tr>
<td>SI</td>
<td>International System Of Units</td>
</tr>
<tr>
<td>SSME</td>
<td>Space Shuttle Main Engine</td>
</tr>
<tr>
<td>SSTL</td>
<td>Surrey Satellite Technology Ltd</td>
</tr>
<tr>
<td>STK</td>
<td>Satellite Tool Kit</td>
</tr>
<tr>
<td>STP</td>
<td>Solar Thermal Propulsion</td>
</tr>
<tr>
<td>ST&amp;P</td>
<td>Standard Temperature And Pressure</td>
</tr>
<tr>
<td>TC</td>
<td>Thermocouple</td>
</tr>
<tr>
<td>TCM</td>
<td>Thermocouple Multiplexer</td>
</tr>
<tr>
<td>TEAN</td>
<td>Triethanol Ammonium Nitrate</td>
</tr>
<tr>
<td>UDMH</td>
<td>Unsymmetrical Dimethyl Hydrazine</td>
</tr>
<tr>
<td>UoSat</td>
<td>University Of Surrey Satellite</td>
</tr>
<tr>
<td>USAF</td>
<td>United States Air Force</td>
</tr>
<tr>
<td>VFP</td>
<td>Vortex Flow Pancake</td>
</tr>
<tr>
<td>VL</td>
<td>Volume Limited</td>
</tr>
<tr>
<td>WWII</td>
<td>World War II</td>
</tr>
<tr>
<td>XIPS</td>
<td>Xenon Ion Propulsion System</td>
</tr>
<tr>
<td>XPS</td>
<td>X-Ray Photoelectron Spectroscopy</td>
</tr>
</tbody>
</table>
Chapter 1

1 Introduction

1.1 Introduction
1.2 The Small Spacecraft Revolution
1.3 Missions Flown as Secondary Payloads
1.4 The Need for Propulsion
1.5 Delta V Requirements for Mission Scenarios
1.6 Scope of the Research
1.7 Novel Work Undertaken
1 Introduction

1.1 Introduction

This chapter details the envelope of the research along with the specific reasons for selecting the research topic. It specifically discusses how small satellites have been made effective with the use of the secondary launch capacity of modern launch vehicles, the advantages of placing small satellites into various orbits and the magnitude of the propulsive capability required to achieve this. Finally the chapter details the novel work that has been accomplished during this research.

1.2 The Small Spacecraft Revolution

Since the launch of the first artificial satellite with Sputnik 1 in 1957, most satellites have become increasingly larger and heavier than their predecessors. It has been the demand from the commercial and scientific world for increasingly more complex and power hungry equipment to be installed on board. These factors have driven the mass from the original Sputnik with 87 kg [SMITHSONIAN 99] to a typical modern geostationary communication satellite with a mass of over 1,700 kg [SAT ENCYLO 02]. However these large satellites are expensive. The launch vehicle operators charge a rate per kilogram of payload with the cost dependant upon the actual launch vehicle in question. For big commercial payloads such as telecommunication satellites that require large antennas, miniaturisation is not an option. However for a great number of earth observation and scientific satellites, shrinking the dimensions is a very real possibility. During the early 1980's serious consideration was given to scaling down the mass and volume of satellites to provide a more cost effective system [MICCI 00]. The goal was to produce a satellite capable of performing a useful role with a mass barely more than the original Sputnik. Realisation of this idea was brought about through progress in microelectronic architecture. It was primarily the advancement of microprocessors [UNDERWOOD 98] that allowed satellites to reduce their mass, volume and most significantly their power requirement. Reducing the dimensions of complex electronic circuitry by replacing large transistor arrays with integrated circuits helped a great deal towards this goal. This enabled smaller solar arrays to be used and so small satellites became feasible. Thus for the first time, small and lightweight satellites could compete in terms of functionality with larger, more expensive satellites. With a 100 kg satellite being able to be launched as a secondary payload for a small fraction of the cost of a large satellite, Universities could suddenly afford to launch their own spacecraft. Arguably the greatest exponent of the small satellite revolution has been Surrey Satellite Technology Ltd, who have so far built and operated a total of 20 satellites, starting with UoSat-1 back in 1981. This functioned
until it burnt up upon re-entry in 1989. NASA has adopted a similar policy for a range of interplanetary missions under the mantra of 'smaller, faster, cheaper'. While NASA has achieved a degree of success with this policy, the interplanetary projects have not enjoyed the same level of cost reduction as one might have expected [McCURDY 01]. The size of small satellites currently under consideration varies dramatically, however the key feature is that almost without exception they are secondary payloads. The mass-classification of small satellites spans two orders of magnitude and the generally accepted names for mass sizes are shown below in table 1.1. These classes of spacecraft are all designed to be either secondary payloads or clustered into a multiple satellite launch configuration. Most small spacecraft to date have required insertion into LEO,

<table>
<thead>
<tr>
<th>Class</th>
<th>Mass</th>
<th>Example</th>
</tr>
</thead>
<tbody>
<tr>
<td>Pico Satellite</td>
<td>&lt;1 kg</td>
<td>Palm Sat</td>
</tr>
<tr>
<td>Nano Satellite</td>
<td>1 kg to 10 kg</td>
<td>Snap-1 Technology Demo</td>
</tr>
<tr>
<td>Micro Satellite</td>
<td>10 kg to 100 kg</td>
<td>UoSat-3 RF Communications</td>
</tr>
<tr>
<td>Mini Satellite</td>
<td>100 kg to 500 kg</td>
<td>UoSat-12 Earth Observation</td>
</tr>
</tbody>
</table>

Table 1.1: Satellite mass classifications

Figure 1.1: Ariane ASAP configuration
however there has been considerable interest in placing small satellites into HEO, GEO and interplanetary trajectories [SWEETING 00].

The emergence of a large number of small but highly useful satellites demanded some form of launch vehicle strategy. Dedicated launchers such as the Scout and Pegasus vehicles could meet this demand. However the secondary payload arena was the most attractive from a cost aspect. In such a system, a large ‘primary’ traditional satellite is launched with small ‘secondary’ satellites riding piggyback. Currently there are a number of launch vehicles capable of lifting small payloads with three distinct methods of payload insertion. The most common method, which has zero impact on the primary payload, is to use the adaptor ring concept. In the case of Ariane’s ASAP\(^1\), the ring sits below the main satellite and has space to accommodate several small satellites. Shown in figure 1.1 is ASAP-1 of Ariane mission V35 with six spacecraft, two of which are UoSat-3 and UoSat-4 on board. The engineers give scale to the primary and secondary payloads. Small satellites are located on the ring during ascent and are released upon telecommand after the main satellite has separated. Adding an ASAP ring costs approximately $1.2M, which is shared between all the small satellite payloads operators on board [SSTL 02].

\[1\] ASAP is an acronym for Ariane Structure for Auxiliary Payloads and is essentially a ring that sits on top of the final stage of the launch vehicle.
An alternative method is to attach the secondary payload onto the primary. This has the advantage that the launch vehicle is not modified in any way, it is the satellite manufacturer who has to make provisions for the secondary payload. This method was used to good effect with the Russian Cosmos launch vehicle used to launch of SSTL’s SNAP-1 and Tsinghua-1 satellites. The Tsinghua-1 satellite is shown in figure 1.2, attached to the Russian military satellite. This is generally the least favourable option as satellite companies regard their payload as their only concern. Any extra complication that could threaten the mission, such as a secondary payload, would be viewed with suspicion and in the final analysis would not be worth the risk. Launch vehicle operators, however, want to maximise their profits and a minimal risk addition is usually quite welcome. A third option is to use the adaptor cone at the base of the primary to house a small satellite. In this configuration it is neither the primary payload nor the launch vehicle that is modified but instead the third-stage-to-primary interface. The use of the secondary payload adapter ring, modified adaptor cone or a hard point on the primary spacecraft presents a great opportunity for small payloads launches at reduced cost, however the actual orbit is very much at the mercy of the primaries requirements. Thus, finding the actual launch you wish to be aboard can require a considerable wait if the orbital requirements are highly specific.

1.3 Missions Flown as Secondary Payloads

To date a great number of missions have utilised the secondary launch system, the largest number with 80 such missions [NASA 02] being flown on board the shuttle. This is not a true representation of small satellites, as often the payload on board the shuttle never becomes a free-flyer. However, all the expendable launch vehicles release their secondary payload and thus represent true small satellite launch vehicles. Of the SSTL missions flown, table 1.2 summarises the launch vehicles that were used. SSTL initially used US and European launchers but in recent times, the emergence of a large number of old Russian ICBM’s, converted to satellite launchers, has come to the forefront. These launchers are cheaper than their Western counterparts and so there has been a trend towards the use of Russian boosters, as can be seen from the chronologically ordered table.
<table>
<thead>
<tr>
<th>Spacecraft</th>
<th>Launch Date</th>
<th>Launch Vehicle</th>
</tr>
</thead>
<tbody>
<tr>
<td>UoSAT-1</td>
<td>October 1981</td>
<td>Delta</td>
</tr>
<tr>
<td>UoSAT-2</td>
<td>March 1984</td>
<td>Delta</td>
</tr>
<tr>
<td>UoSAT-3</td>
<td>January 1990</td>
<td>Ariane V35 ASAP</td>
</tr>
<tr>
<td>UoSAT-4</td>
<td></td>
<td></td>
</tr>
<tr>
<td>UoSAT-5</td>
<td>July 1991</td>
<td>Ariane ASAP</td>
</tr>
<tr>
<td>KitSAT-1</td>
<td>August 1992</td>
<td>Ariane V52 ASAP</td>
</tr>
<tr>
<td>S80/T</td>
<td></td>
<td></td>
</tr>
<tr>
<td>PoSAT-1</td>
<td>September 1993</td>
<td>Ariane</td>
</tr>
<tr>
<td>HealthSAT-II</td>
<td></td>
<td></td>
</tr>
<tr>
<td>Cerise</td>
<td>July 1995</td>
<td>Ariane</td>
</tr>
<tr>
<td>FASat-Alpha</td>
<td>August 1995</td>
<td>Tsyklon</td>
</tr>
<tr>
<td>FASat-Bravo</td>
<td>July 1998</td>
<td>Zenit</td>
</tr>
<tr>
<td>Thai-Phut</td>
<td></td>
<td></td>
</tr>
<tr>
<td>Clementine</td>
<td>December 1999</td>
<td>Ariane</td>
</tr>
<tr>
<td>UoSAT-12</td>
<td>April 1999</td>
<td>Dnepr (Primary payload)</td>
</tr>
<tr>
<td>Tiung-Sat-1</td>
<td>August 2000</td>
<td>Dnepr</td>
</tr>
<tr>
<td>SNAP-1</td>
<td>June 2000</td>
<td>Cosmos (Released from Primary satellite)</td>
</tr>
<tr>
<td>Tsingha-1</td>
<td></td>
<td></td>
</tr>
<tr>
<td>Picosat</td>
<td>September 2001</td>
<td>Athena</td>
</tr>
</tbody>
</table>

Table 1.2: Launch vehicles used by SSTL

1.4 The Need for Propulsion

The secondary launch system has been shown to be reliable, cheap and abundant. However the particular orbit they are placed into is very much dependent upon the primary payload's orbit. As an example, large telecommunication satellites are first injected into a Geostationary Transfer Orbit (GTO) by the launch vehicle. This orbit will have an apogee of 36,000 km and a perigee of a few hundred km. The primary spacecraft can circularise this orbit using its own Apogee Kick Motor (AKM), however any secondary payloads will be stuck in this highly eccentric orbit. Thus
for an earth observation mission such an orbit would not be acceptable. Without propulsion on board, the spacecraft would have to choose another mission to ride piggyback on, with perhaps months or years of waiting. This potentially increases the cost dramatically through extended programmes. However with propulsion, the secondary payload could change to a low earth orbit if required, thus rendering the original opportunity perfectly feasible. The pertinent benefit with propulsion is the wide degree of flexibility it gives the small satellite supplier. This has the potential to maximise the use of secondary launches and speed up lead times, thus reducing costs. However this flexibility is not without its flaws. While launch agencies are happy to place dumb payloads on board that present little or no risk to the primary, they might not be so enthusiastic about the primary sitting on top of tanks of hypergolic fluids. Thus safety, both to personnel and the primary payload is extremely important when considering propulsion.

As well as providing enhanced flexibility on the choice of launch vehicle, the mission can also be far more dynamic. Recently a lot of interest has focused on the concept of formation flying where several vehicles, which are spatially separated, fly in unison [INALHAN 00]. While it is possible to maintain nominal phasing of the constellation without propulsion (variable drag is an option) for accurate constellation control propulsion must be used. Similarly an Earth Observation satellite may require repositioning in terms of inclination or altitude. Or if the orbit is sun-synchronous, the local time of day at which the passage occurs may need to be altered or maintained. Finally there may come a point when satellites will be required to have propulsion on board because of orbital debris issues. This growing concern could force international law to impose strict requirements on all satellites. Currently once a GEO satellite expires due to reaching the end of its life, the vehicle should be removed from this slot. If a satellite company fails to remove the expired satellite they may not be granted a new licence for another GEO mission. In addition there are moves are afoot to impose a mandatory de-orbit or graveyard burn before the satellite fails [GIBBONS 90][LARSON 97]. This is a clear requirement for small satellites to incorporate propulsion.

1.4.1 Maximising Launch Opportunities and Functionality

Placing propulsion on board is not the answer to every launch issue however. As an example of exaggerated proportions, consider a GTO situation with the primary payload finally residing in GEO orbit. This would be useless for a low earth polar orbit secondary satellite mission. While it is possible to place enough propellants on board to perform a plane change of 90 degrees and bring the secondary payload back to LEO, it would essentially be a 'flying gas can'. So from a useful payload point of view, the mission would be pointless. Instead, each launch opportunity must be carefully analysed and the exact delta V calculated. However GTO would make a good starting point for a low cost interplanetary or lunar mission.
The mass fraction of propellants that are required on board is a function of the initial and final orbits of the secondary spacecraft and the efficiency of the propulsion system. The actual launch scenario chosen will be based upon costs and schedules, thus the small satellite has to be dynamic in its propulsion requirements in order to gain maximum benefit from secondary launches. Thus the emphasis is the propulsion system must be capable of being optimised for different delta V missions relatively easily. In the optimal design, only the alteration of the propellant volume would be required. The mass fraction is not the only concern, volume is a key issue too. Small satellites are very volume limited and this is usually the prime driver not the mass. Thus the propulsion system must be compact. The measure of performance used for rocket engines is the Isp term, described in §2.2. This metric describes the available impulse from a unit mass of the propellant. A better metric for small satellites is the density Isp parameter. This gives the impulse per unit volume and hence is ideal for assessing propulsion systems for small spacecraft.

1.5 Delta V Requirements for Mission Scenarios

To provide a ‘feel’ for the magnitude of propulsion required to perform useful functions, a series of scenarios were envisaged, and the delta V calculated. When the orbit is within the ‘potential well’ of the Earth, standard Hohmann transfer calculations were used. However when considering a lunar trajectory, Satellite Tool Kit’s (STK) numerical integrator was to be used to calculate the delta V requirement.

1.5.1 GTO to GEO

This is the most common form of orbit transfer that is routinely executed by telecommunications satellites. The initial orbit is highly elliptical with an apogee altitude of 35,786 km and a perigee of 250 km. Typically there will also be an inclination of the launch site to cancel out. A launch from KSC requires 28.5 degrees to be eliminated from the inclination. With the satellite in GTO, it will be moving at apogee with a speed of 1.58 kms⁻¹, and as the inclination burn delta V is proportional to the velocity, it is most efficient to perform the burn at this point. This requires a velocity change in the North-South direction of 0.78 kms⁻¹. There must also be another velocity to circularise the orbit into full GEO. This requires an East-West velocity change of 1.48 kms⁻¹. Through vector addition, this can be achieved in a single burn of 1.67 kms⁻¹. For a launch closer to the equator such as Korou, there is considerably less inclination to ‘burn out’ and this reduces the total delta V requirement down to 1.49 kms⁻¹.
1.5.2 GEO to Super-Sync Orbit

Currently the Geostationary belt is rapidly becoming saturated with spacecraft. Hence it is important to keep this particular orbit free of dead or outdated spacecraft to prevent paying for a 'dead' orbital slot. Thus a spacecraft launched into GEO must have the capability to get out again. The most common way is to boost the spacecraft into a higher orbit, called super-sync where the period of revolution is greater than 24 hours. It is important to do two burns as a single impulsive burn would create an elliptical orbit with a high apogee, but the perigee would slice straight through the geostationary orbit. While there is no set altitude where super sync orbit begins, a working figure of 42,000km was arbitrarily chosen by the author. The initial burn will place the spacecraft into super sync transfer orbit, which requires a delta V of 95 ms\(^{-1}\). Following that, at apogee another burn of 96 ms\(^{-1}\) is required to circularise it, giving a total delta V requirement of 191 ms\(^{-1}\). Thus it can be seen that the delta V requirement to remove the satellite to a comfortable height from GEO is relatively small in comparison to initially getting the spacecraft there. However this low delta V requirement is not without risk. If a propulsion failure occurred at the end of life, then a firing command could well explode the spacecraft creating a very bad situation for this orbital slot, now contaminated with debris.

1.5.3 GTO/LEO to Lunar Orbit

The final orbit to be considered was the transfer from a Low Earth Orbit or Geostationary Transfer Orbit to a Lunar Transfer Orbit, then finally into Lunar Orbit. SSTL has considered a lunar mission for capability demonstration as well as possible scientific purposes. Satellite Tool Kit (STK) was used for calculations and starting from a LEO orbit the total delta V required to insert into a Low Lunar Orbit is approximately 4.0 kms\(^{-1}\). However when starting from a GTO orbit, this figure reduces to 1.5 kms\(^{-1}\), comparable to GEO insertion from GTO.

1.5.4 Thrust Requirement

Generally impulsive type manoeuvres are the most efficient orbital transfers and this lends itself to a high thrust system. Additionally it is desirable to minimise the time spent in transfer orbits to reduce the radiation exposure and reduce the time required to commission the spacecraft. Both requirements lead to a high thrust system. There are many missions envisioned in the near future by SSTL that require high thrust and high delta V propulsion [JASON 01]. This is a clear manifesto for high performance propulsion. The issue is how large a thrust the spacecraft can tolerate. Using UoSAT-12 as a typical Mini Satellite, operating in 3-axis control mode, the maximum thrust was calculated. The cold gas thrusters could generate a maximum torque of 0.5Nm. Hence the thrust range, with a maximum displacement from the centre of gravity of
10mm, is of the order of 50N. A small spacecraft can tolerate a higher thrust through the use of a spinning platform due to the cancellation of the thrust misalignment. However communications become more difficult with a spinning platform and a de-spun platform is not applicable for small low cost satellites. Hence a peak thrust of 50N was chosen as this covers the worst possible case of a 3-axis stabilised spacecraft.

1.6 Scope of the Research

The scope of the research was to demonstrate that an HTP/Kerosene bi-propellant rocket engine, constructed from non-exotic materials, could offer advantages over commercially available engines. To perform this analysis both the storage of HTP and the engine design was investigated with the latter built around the concept of maximising density Isp and minimising costs. It is known that small satellites being extremely power and volume limited are therefore not suitable for high thrust electric propulsion. This leaves the chemical propulsion as an option if the high delta V’s and thrusts, described in §1.5 are required. While monopropellant technologies are simple, the performance is inferior to hybrid and bi-propellant systems. Hybrids however lack the flexibility that bi-propellants offer in terms of delta V scaling. For different delta V missions the quantity of propellant needs to be altered and with the hybrid this results in a redesigned chamber. It was determined that the system that gave the greatest flexibility and performance was the bi-propellant engine and this coupled with the use of non toxic ‘green’ propellants, offers the potential for low-cost and high-performance propulsion. HTP, being a high-density oxidiser, can provide high density Isp’s and when coupled with a cooling system that uses the oxidiser promises to reduce the engineering costs. The novel oxidiser film-cooling concept does not require the same high precision engineering required by commercial engines and it can operate without the use of high temperature refractory metals. Thus the complete package offers an overall reduction of cost, both in terms of development and actually using the engine on a flight mission. While HTP is a good oxidiser in terms of performance, it also has inherent problems with decomposition in a sealed tank causing a pressure rise. Thus a range of materials was tested as possible tank liners and the concept of the passively vented tank was examined. Through the use of a tank that allows Oxygen to permeate through the walls, an equilibrium pressure can be maintained. This was investigated and culminated in a demonstration tank.
1.7 Novel Work Undertaken

The research undertaken was on two specific areas; use of HTP and Kerosene in an Oxidiser cooled engine and the storage of HTP for extended periods. The engine research resulted in the following significant achievements;

- Demonstration of a gas cooled combustion chamber
- Demonstration of oxidiser film cooled nozzle
- Determination of minimum flow rates for adequate cooling of the nozzle and chamber
- A low cost design for injector and combustor system has been developed
- A low cost film cooled nozzle has been developed
- A design has been made which offers similar density Isp to commercial systems, yet is considerably less expensive to manufacture, test and fly

The HTP tank research has achieved the following significant achievements;

- Determination that PFA and FEP polymers are good Class I materials with HTP
- Pure Aluminium of 99.999% purity is superior to PFA or FEP
- Very little effect of the HTP upon the PFA or FEP in terms of mechanical strength
- No oxidation of the polymers was recorded
- Storing HTP in contact with PFA/FEP does not adversely effect the decomposition in a catalyst pack
- Design and implementation of a gas permeable tank
- Demonstration that a permeable tank is feasible and offers a way of on orbit extended storage
Chapter 2

2 Literature Survey

2.1 Introduction
2.2 Propulsion Basics
2.3 Propulsion Options
2.4 Thermodynamics of Choked Flow
2.5 Review of Hydrogen Peroxide
2.6 Hydrogen Peroxide as a Monopropellant
2.7 Hydrogen Peroxide and Kerosene Rocket Engines
2.8 Propellant Tanks for Hydrogen Peroxide
2.9 Conclusions of the Literature Survey
Chapter 2: Literature Survey

2.1 Introduction

This chapter discusses the propulsion options that are available for small satellites and reviews the pertinent equations of rocket propulsion. The green bi-propellant engine satisfies the demands of high thrust and high ΔV requirements. Hydrogen peroxide and its decomposition are therefore reviewed in detail. The use of HTP and kerosene is discussed in past applications and the cooling methods applicable to this type of engine are reviewed. Finally the storage of HTP is examined with reference to flown tank systems.

2.2 Propulsion Basics

There are several types of rocket engines currently available for providing satellite propulsion. In all of these engines the working fluid is expelled from the rear of the engine and the resulting momentum change produces an equal and opposite thrust, thus propelling the spacecraft. The method of acceleration of the working fluid is specific to the type of engine being used and involves the energy being utilised from electrical, chemical or nuclear sources [HUMBLE 95]. Primarily for small satellites the following rocket propulsion systems are of interest [SUTTON 92][MUELLER 00].

1. Cold Gas Thrusters
2. Solid Propellant Motors
3. Monopropellant Engines
4. Hybrid Engines
5. Liquid Bi-Propellant Engines
6. Augmented Cold/Hot Gas Thrusters
7. Thermal Electric Propulsion
8. Electromagnetic Propulsion
9. Electrostatic Propulsion

In all the above systems, if the effective exhaust velocity is defined as $V_e$ and the mass of the spacecraft as $m$, then the thrust that is produced is given by equation 2.1. The effective exhaust velocity is a measure of the efficiency of the engine. Specific impulse (Isp) is the commonly used term for describing the efficiency of an engine and, as described below, the two are related.
Chapter 2: Literature Survey

\[ \text{Thrust} = \frac{dm}{dt} V_e \]  \hspace{1cm} (2.1)

If \( V_{sat} \) is defined as the velocity of the spacecraft, then the thrust will produce a rate of change of \( V_{sat} \) given in equation 2.2.

\[ m \frac{dV_{sat}}{dt} = -\frac{dm}{dt} V_e \]  \hspace{1cm} (2.2)

Hence the change in velocity can be rewritten as equations 2.3 and 2.4, where \( m_i \) and \( m_f \) are the initial and final masses of the spacecraft respectively.

\[ dV_{sat} = -V_e \frac{dm}{m} \]

\[ \int_{0}^{\Delta v} dV_{sat} = -V_e \int_{m_i}^{m_f} \frac{dm}{m} \]  \hspace{1cm} (2.3)(2.4)

Thus by integration of the equation 2.4 the classical Tsiolkovsky equation [HUMBLE 95] is derived, which is more commonly known as the Rocket Equation, given in equation 2.5

\[ \Delta V = V_e \ln \left( \frac{m_i}{m_f} \right) \]  \hspace{1cm} (2.5)

If the effective exhaust velocity in a vacuum is known, this equation gives the change in velocity regardless of the propulsion mechanism used. It does not say anything about the efficiency of the system, this is in fact tied up in the effective exhaust velocity term. A common measure for describing the efficiency of a propulsion system is the Specific Impulse term. Specific Impulse, or Isp, is defined as the quantity of impulse achieved per unit weight of propellant used.

\[ Isp = \frac{\text{Total Impulse}}{\text{Mass of Propellants} \times g_0} = \frac{\text{Thrust} \times t_{\text{burn}}}{m_{\text{prop}} \times g_0} = \frac{\text{kgms}^{-1}}{\text{kg} \times \text{ms}^{-2}} = s \]  \hspace{1cm} (2.6)

Thus from equation 2.6 it can be seen that the unit of Isp is seconds. Equation 2.6 can be rearranged to arrive at the thrust definition.

\[ \text{Thrust} = Isp \times g_0 \times \frac{m_{\text{prop}}}{t_{\text{burn}}} = Isp \times g_0 \times \frac{dm}{dt} \]  \hspace{1cm} (2.7)

Hence by comparing equations 2.7 and 2.1, it can be seen that the effective exhaust velocity term used in the rocket equation 2.5, is related to the Isp through the equation 2.8.
The final parameter that is useful when comparing propulsion systems is the density Isp term. This term gives a measure for the compactness of the propellants and is defined by equation 2.9, where \( \rho_{sg} \) is the specific gravity of the propellants.

\[
\rho_{Isp} = \rho_{sg} \times Isp
\]

This gives units of seconds again as the specific gravity by definition is dimensionless. For a given \( \Delta V \) requirement a system that has a high density Isp will have a lower propellant volume than a lower density Isp system. For small satellites, where mass is not usually a problem, and volume is, the density Isp term is arguably more important than the true Isp [ZAKIROV 01].

2.3 Propulsion Options

When selecting the type of propulsion to use on a satellite there are a number of factors that need to be studied [SELLERS 96]. These nine factors are listed below, with the particular mission dictating which of the nine have more precedence than the others.

1. Propellant mass
2. Propellant volume
3. Time to Complete \( \Delta V \)
4. Power
5. System Price
6. Technical Risk
7. Safety Cost
8. Integration Cost

So far SSTL has flown satellites with propulsion on board capable of drag compensation and limited orbit raising capacity. In the mid to long term there will come a need for significantly higher performance propulsion, whether this be an insertion apogee burn for geo-synchronous circularisation or a de-orbit burn to return satellites to earth, freeing up orbital slots [PHIPPS 00][ECKART 00]. The problems specific to small satellites reside with the propellant volume issue and power available. UoSat-12, which was SSTL’s first mini-satellite with a mass of 300kg, had only 140W of on orbit average power. The volume available for propellant storage was only 30 litres, split between the cold gas and resistojet propellants. Taking this spacecraft as an example of a typical volume and power available for a small satellite, demonstrates the severe requirements imposed on the propulsion system. The propulsion system review showed that electrical propulsion requires too much power for a small spacecraft to provide. Typical power
requirements for even mN of thrust are in the kW range for gridded ion thrusters. While electric propulsion cannot be used for the main propulsion system, for changing the orbit where high thrust is demanded, the ACS requirements may well be satisfied by low power electrical systems. This leaves the chemical propulsion as the low power choice for a main propulsion system. Monopropellants and bi-propellants have the great advantage that different missions only need the propellant tank volume to change. This is an advantage over the hybrid system, where the combustion chamber has to be altered in dimensions to accommodate more, or less fuel. However, for high performance propulsion, monopropellant systems have too low an Isp, especially when green\(^1\) propellants are considered. Nitrous Oxide has been considered as a possible monopropellant, but with only a maximum Isp of 206 seconds and poor density Isp of 153 seconds [ZAKIROV 00] this is not in the regime of high performance propulsion. This leaves the bi-propellant engine as the propulsion system of choice for a wide range of small satellite missions. This conclusion from the analysis was not surprising, considering the large number of commercial satellites that use bi-propellant systems as their prime means of propulsion. However what is specific to small satellites is the use of green propellants as opposed to the toxic hypergolics. By removing the stringent ground support equipment (GSE) requirement of the hypergolics, the overall system cost can be dramatically reduced, thus making green bi-propellant engines truly competitive. While there are a myriad of fuels that can be burnt with an oxidiser, the number of green oxidisers is essentially limited to Nitrous Oxide and HTP. Nitrous Oxide and HTP have densities of 745 and 1393 kg/m\(^3\) respectively and it is clear that to satisfy the high-density criteria, then HTP is the suggested oxidiser. Research has reported that Nitrous Oxide, in addition to being self-pressurising, could be used as a pressurant for the fuel as well [ZAKIROV 00]. This would eliminate the requirement for an extra pressurising tank, and hence increase the effective density of the propellants. In comparison, HTP requires a pressurant and hence the overall system density would reduce.

\(^1\) There is no clear definition of green propellants, however it is generally accepted that they are propellants whose toxicity values are classified as low and only minimal protective clothing is required for handling. In addition, upon combustion the propellants must produce benign products that can safely be vented to atmosphere.
2.4 Thermodynamics of Choked Flow

Considering a convergent/divergent nozzle, shown in figure 2.1, with the exit plane at 1 Bar and chamber pressure initially at 1 Bar also, the mass flow rate of a nozzle will correspondingly be zero. Increasing the mass flow from zero to a small but finite value is achieved by increasing the chamber pressure. The velocity in the chamber will still be close to zero, however in the narrow throat region there will be a significant velocity. This high velocity gas will then slow down as it expands in the exit cone. The mass flow can be increased and this will result in the velocity through the throat increasing still further and hence a lower pressure will be generated in accordance with Bernoulli’s equation. If the mass flow rate is increased so that sonic velocity of the gas is achieved at the throat then choked conditions will have been achieved. From the conservation of mass flow, conservation of energy and theory of supersonic flows, equation 2.10 can be derived.

\[
\frac{dA}{A} = \frac{dV}{V} \left( \frac{V^2}{V_s^2} - 1 \right)
\]

Where ‘A’ is the cross sectional area as a function of distance along the nozzle and ‘V’ is the velocity at that point. It can be seen that when \( V = V_s \) then the rate of change of area must be zero. That is Mach 1 can only occur at the parallel section at the point of the throat. This does not mean however that the throat will always be at Mach 1. For low flow rates the Mach number will be less than unity. Increasing the mass flow rate further will raise the pressure of the throat, its temperature and hence velocity, however it will still have a Mach number of 1. The expansion after the throat can either be subsonic or supersonic. The parameter that dictates the type of flow is the throat pressure. If the throat pressure is greater than the ambient pressure then expansion will cause acceleration. If the reverse is true and the throat pressure is below ambient pressure, then the pressure must increase to meet the exit conditions and thus there will be reduction in velocity, and hence it will be a subsonic flow. In order to determine if a system will have

![Figure 2.1: The DeLaval nozzle](image-url)
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supersonic efflux, the throat pressure must be determined. In order to make the analysis possible the following assumptions have to be made:

1. Adiabatic
2. Isentropic
3. One dimensional flow
4. Steady flow
5. Ideal gas
6. Choked flow

From the conservation of energy, composed of internal and kinetic energy, equation 2.11 describes the flow.

\[ H_c + \frac{1}{2}V_c^2 = H_i + \frac{1}{2}V_i^2 \]  \hspace{1cm} (2.11)

Where;

- \( H \): Specific enthalpy of the combustion gases (Jkg\(^{-1}\))
- \( V \): Velocity of the combustion gases (ms\(^{-1}\))

Subscript 'c' refers to the chamber and 'i' refers to an arbitrary part of the nozzle.

The velocity in the chamber section will be approximately zero and recasting the enthalpy in terms of the heat capacity, \( c_p \) and the thermodynamic temperature, \( T \).

\[ T_c c_p = T_i c_p + \frac{1}{2}V_i^2 \]  \hspace{1cm} (2.12)

The velocity can be expressed as Mach number so that;

\[ V_i = M \sqrt{\gamma R_g T_i} \]  \hspace{1cm} (2.13)

Hence

\[ T_c = T_i \left[ 1 + \frac{\gamma R_g}{2c_p} M^2 \right] \]  \hspace{1cm} (2.14)

However for an ideal gas \( R_g = c_p - c_v \) and \( \gamma = c_p / c_v \). Thus this equation can be rewritten as

\[ T_c = T_i \left[ 1 + (\gamma - 1) \frac{M^2}{2} \right] \]  \hspace{1cm} (2.15)
As the flow is assumed to have constant entropy, the isentropic relations can be used to relate pressure to temperature, given in equation 2.16.

\[
\frac{T_c}{T_i} = \left(\frac{P_c}{P_i}\right)^{\frac{y-1}{y}}
\]  

Thus

\[
\frac{P_c}{P_i} = \left[1 + \frac{(y-1)M^2}{2}\right]^{\frac{1}{y-1}}
\]  

When the mass flow rate is increased to the point where sonic flow occurs at the throat, the Mach number is by definition 1 and hence this now gives the condition for sonic flow.

\[
\frac{P_c}{P_i} = \left[1 + \frac{(y-1)}{2}\right]^{\frac{1}{y-1}}
\]  

This defines the condition for sonic velocity at the throat. However this does not necessarily mean that the flow past the nozzle will be sonic. For this to be the case as well, the throat pressure must be above the ambient pressure, and hence

\[
P_c \left[1 + \frac{y-1}{2}\right]^{\frac{1}{1-y}} > P_{amb}
\]  

Thus by knowing \(\gamma\), the ambient pressure and the chamber pressure, the exit flow can be determined to be either supersonic or subsonic. The chamber pressure \(P_c\) is determined from the thermo-chemical data of the propellants and leads to the concept of the \(C^*\) parameter. Starting with the definition of the mass flow rate.

\[
\dot{m}_{prop} = \rho A V
\]  

It is known, under choked conditions, at the throat the velocity is the sonic and hence

\[
\dot{m}_{prop} = \rho_c A_t \sqrt{\gamma R_e T_i}
\]  

The conditions in the chamber itself are known from the thermo-chemistry, thus the equation 2.21 needs to be re-written in terms of the chamber parameters. By applying the isentropic relation, given in equation 2.22 and equation 2.15, equation 2.23 is derived.

\[
\frac{P_c}{P_i} = \left(\frac{T_c}{T_i}\right)^{\frac{1}{y-1}} \Rightarrow \frac{P_c}{P_i} = \left(1 + \frac{y-1}{2} M^2\right)^{\frac{1}{y-1}}
\]  

2-8
Hence at the throat as $M=1$;

$$\frac{\rho_c}{\rho_t} = \left(1 + \frac{\gamma - 1}{2}\right)^{\gamma - 1}$$

(2.24)

Substituting equation 2.24 into 2.21 and using the isentropic equation for temperature in terms of $\gamma$, equation 2.25 can be derived.

$$m_{prop} = \rho_c \left(1 + \frac{\gamma - 1}{2}\right)^{1-\gamma} A_t \sqrt{\frac{R_g T_c}{1 + \frac{\gamma - 1}{2}}}$$

(2.25)

The ideal gas density term can be re-written as $\rho_e = P_e / R_g T_e$

$$m_{prop} = \frac{P_e A_t}{\sqrt{\gamma R_g T_c}} \gamma \left(1 + \frac{\gamma - 1}{2}\right)^{1+\gamma}$$

(2.26)

Hence the parameters $\gamma$ and $T_c$, which are obtained from the thermo-chemistry of the reaction, can then be used to calculate the pressure for a given throat area and mass flow rate. This equation can be re-written as equation 2.27.

$$C^* = \frac{P_e A_t}{m_{prop}}$$

(2.27)

Where

$$C^* = \frac{\sqrt{\gamma R_g T_c}}{\gamma \left(1 + \frac{\gamma - 1}{2}\right)^{1+\gamma}}$$

(2.28)

Hence it can be seen that $C^*$ is a function of just the thermodynamic parameters and thus a theoretical $C^*$ can be calculated. As $P_e$, $A_t$ and $m_{prop}$ can be measured, an empirical $C^*$, using equation 2.27, can be calculated. The empirical $C^*$ will always be lower than theoretical due to incomplete combustion, so the ratio of $C^*_{\text{empirical}}$ to $C^*_{\text{practical}}$ can be used as a measure of combustion efficiency.

Returning to the original problem, by substituting equation 2.26 into 2.19, the condition for supersonic flow is defined.
2.5 Review of Hydrogen Peroxide

It is well known that there are issues regarding the storability of the oxidiser and thus the oxidiser was reviewed in terms of its physical properties, handling requirements and storage issues.

2.5.1 Physical Properties

Hydrogen peroxide is a clear liquid at room temperature with the appearance of water. The formula for hydrogen peroxide is H$_2$O$_2$ and it is readily decomposed in the presence of a catalyst according to equation 2.30. If the reaction takes place isothermally, then liquid water will be produced. However in propulsive decomposition where approximately adiabatic conditions are observed, the water will be in the gas phase.

\[
2H_2O_2 \rightarrow 2H_2O + O_2
\]  

The method by which this decomposition occurs is not a straightforward process but can involve between 10 and 20 intermediate steps [PEROX HANDBOOK 67]. The rate determining step has yet to be isolated, but it is believed to involve electron transfer and the decomposition is shown empirically to be first order i.e. proportional to concentration [PEROXIDE 55]. Hydrogen peroxide is rarely in its pure anhydrous state, mainly due to the hygroscopic nature, thus there is always a percentage of water in the liquid. For propulsive applications it is vital that all the hydrogen peroxide decomposes into gas and no liquid remains. This puts a limit on the minimum concentration that can be used for propulsion, and has been determined to be 64.7% [BECCO 54]. The higher the concentration, the higher the performance of the bi-propellant propulsion system, and it is usually desired to use at least 90%. As the density of the hydrogen peroxide-water solutions varies with concentration, shown graphically in figure 2.2 [BECCO 54], the easiest way to empirically determine the concentration of HTP is to measure the density. For concentrations in the range of 80 to 100% and temperatures in the range of 4 to 27 °C, a linear formula that best fits the data was applied and is shown in equation 2.31, the accuracy is ± 0.001.
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1.46
1.44
1.42
1.38

Figure 2.2: Density of hydrogen peroxide/water solutions

\[
\rho_{\text{kg}} = \left( -6.420 \times 10^{-6} \right) T C_{\text{HTP}} + \left( 5.565 \times 10^{-3} \right) C_{\text{HTP}} - \left( 4.603 \times 10^{-4} \right) T + 9.128 \times 10^{-1} \tag{2.31}\]

The physical parameters for 90% hydrogen peroxide (HTP) are shown in table 2.1 along with water for comparison [BECCO 54][PEROXIDE 55][CRC 96].

2.5.2 Handling Requirements

The handling requirements are very different to water due to its strongly oxidising properties. When HTP is handled it is vital to wear protective clothing to prevent spills causing burns. Standard PVC chemical suits are sufficient to provide adequate protection, and no breathing equipment is required. This is due to the very low vapour pressure of the HTP even at very high concentrations. The specific requirements are well documented [HAZARDS 69] and a summary is given in table 2.2
### Table 2.1: Physical properties of 90% hydrogen peroxide

<table>
<thead>
<tr>
<th>Physical Parameter at ST&amp;P</th>
<th>90% Hydrogen Peroxide</th>
<th>Pure Water</th>
</tr>
</thead>
<tbody>
<tr>
<td>Freezing Point, °C</td>
<td>-11.25</td>
<td>0</td>
</tr>
<tr>
<td>Boiling Point, °C</td>
<td>148</td>
<td>100</td>
</tr>
<tr>
<td>Dynamic Viscosity, centipoises</td>
<td>1.262</td>
<td>1.002</td>
</tr>
<tr>
<td>Vapour Pressure, mBar</td>
<td>3.45</td>
<td>23.4</td>
</tr>
<tr>
<td>Heat Capacity, Jkg⁻¹K⁻¹</td>
<td>2,763.3</td>
<td>4,181.8</td>
</tr>
<tr>
<td>Surface Tension, Nm⁻¹</td>
<td>79.1</td>
<td>72.8</td>
</tr>
<tr>
<td>Density, kgm⁻³</td>
<td>1,393</td>
<td>1,000</td>
</tr>
<tr>
<td>Thermal Conductivity</td>
<td>0.4438</td>
<td>0.5984</td>
</tr>
</tbody>
</table>

### Table 2.2: Protective clothing for the handling of hydrogen peroxide

<table>
<thead>
<tr>
<th>Area of Protection</th>
<th>Protection Methods</th>
</tr>
</thead>
<tbody>
<tr>
<td>Hand and Foot Protection</td>
<td>Vinyl coated gloves MIL-G-43196</td>
</tr>
<tr>
<td></td>
<td>Surgical gloves applicable</td>
</tr>
<tr>
<td></td>
<td>Natural rubber boots</td>
</tr>
<tr>
<td>Head, Face and Body Protection</td>
<td>Dacron clothing, or Dynel fabric for routine operations. Emergency repairs, require vinyl coveralls. Clothing wetted with HTP should be flushed with water</td>
</tr>
<tr>
<td>Respiratory Protection</td>
<td>Ordinarily not required unless a mist is present</td>
</tr>
<tr>
<td>Other Safety Equipment</td>
<td>Safety showers and eye wash facilities should be present. An adequate water supply should be available for dilution, flushing and fire fighting.</td>
</tr>
</tbody>
</table>

Besides the oxidation properties and hence risk of fire, the biggest cause for concern is the detonation of the HTP vapour. It has been proven that when HTP is heated, the vapour that is produced can detonate with velocity of $2,042 \text{ m s}^{-1}$ [PEROXIDE 55]. In the ambient atmosphere it has also been proven that the detonable region for 90% HTP is above $120^\circ \text{C}$ [PEROXIDE 55].
Thus providing the HTP is kept below this temperature, detonations involving the vapour are avoided. Despite seemingly contradictory evidence, current thinking is that hydrogen peroxide of 95% concentration or lower is not shock detonable, and under normal conditions, it is impossible to obtain a propagating shock wave [PEROXIDE 55]. However, it should be noted that these tests used pure HTP, and it can be expected that an organically contaminated sample is likely to be shock sensitive. This may provide an explanation as to why the Germans in WWII found 88% to be shock sensitive. All this evidence points to the safe handling of HTP is achievable providing that strict cleanliness is observed, any spills are immediately washed down with water and protective clothing is always worn.

2.5.3 Storage Issues

The decomposition of hydrogen peroxide in any concentration is catalysed by a host of elements, and it is virtually impossible to eliminate all the catalysts present in a storage vessel. Hence some degree of decomposition will always occur over time, lowering the concentration of the oxidiser and generating oxygen. The elements that are known to be catalytic with hydrogen peroxide are shown in figure 2.3. Due to this decomposition, there will always be a pressure rise in a sealed tank due to the evolution of oxygen, which is the primary source of storage problems. The loss of concentration is a minor problem in comparison to the pressure rise. Hydrogen peroxide naturally decomposes through two mechanisms, homogenous catalysis and heterogeneous catalysis. As the name suggests, homogenous catalysis is the catalytic effect of impurities in solution, while heterogeneous refers to the catalytic effect of the container walls on the hydrogen peroxide [PEROXIDE 55]. The factors affecting the storability of hydrogen peroxide centre on the intrinsic compatibility of the tank material and the capacity of the tank to withstand high pressures. The rate at which the build pressure rise occurs is governed by the following factors [PEROX HANDBOOK 67].
Concentration

The higher the concentration of the hydrogen peroxide, the lower the decomposition rate at high temperatures relative to lower concentration hydrogen peroxide. This is primarily because the ionisation effect from the water molecules is suppressed [McCORMICK 65]. This ionisation leads to auto decomposition, as no contaminants are causing the decomposition, it is simply a product of its own physical nature. From the propulsion point of view the higher the concentration the greater the performance and the longer the hydrogen peroxide can be stored for.

Purity

If the organic and inorganic contaminants are removed from the hydrogen peroxide solution then the stability of the liquid will increase. With extremely pure hydrogen peroxide, it has been demonstrated that the Active Oxygen Loss due to auto decomposition is less than 0.03 % / year [PEROX HANDBOOK 67]. The AOL is defined in equation 2.32.

Temperature

It is generally accepted that the decomposition rate increases 2.3 times for each 10°C temperature increase [PEROX HANDBOOK 67]. However between 65 and 90°C it has been reported that this rule no longer applies, and another decomposition mechanism takes over [BOXWELL 00]. However this is not important as spacecraft extremely rarely exceeds 60°C.
Container Surfaces

The material from which the tank is constructed is of prime importance in dictating the decomposition rate. The tank material must be made from a Class 1 material. Materials designated Class 2 and Class 3 are for limited contact time only. The surface finish is also important [PEROX HANDBOOK 67]. The surface to volume ratio indicates the ratio of heterogeneous to homogeneous decomposition rate, and as homogeneous rate is very low for pure hydrogen peroxide, it is advantageous to have a small surface to volume ratio. This is achieved by engineering a very smooth tank wall with a spherical geometry. Passivation of the container surfaces is of concern as well. The surface must be contaminant free and this is achieved by a procedure of degreasing, acid pickling and finally propellant conditioning [McCORMICK 65]. Table 2.3 shows materials that are compatible with HTP and are classified as Class 1. The AOL is in percent per week at 66° C (151°F) [PEROX HANDBOOK 67] and is defined by equation 2.32.

\[
AOL = \left( \frac{m_{HTP,1}C_{HTP,1} - m_{HTP,2}C_{HTP,2}}{m_{HTP,1}C_{HTP,1}} \right) \times 100
\]  

(2.32)

Where \(m_{HTP,1}\) and \(m_{HTP,2}\) are the masses of the HTP samples before and after the test. \(C_{HTP,1}\) and \(C_{HTP,2}\) are the concentrations of the HTP by mass (percent/100) at the start and end of the test. Hence AOL is the percentage change in mass of just the H\(_2\)O\(_2\) due to oxygen evolution. The aluminium alloys that have been proven compatible are all the low copper alloys, which is in agreement with the fact that copper is a catalyst. Aluminium however is defined as not having catalytic activity [PEROXIDE 55]. It is interesting to note that Kel-F coated 1060 Al has a much lower activity rate than either of the individual materials. This is probably more of an anomaly than a real effect. Unfortunately the fluorinated polymers such as Kel-F, are all referred to by their trade names and hence the exact compositions of these materials are usually propriety. Other literature [McCORMICK 65] agrees that 1060 and 5652 aluminium alloys are compatible materials along with Fluorel 1421 by 3M, Viton A-77-545 by DuPont and 9711 Silicone. Interestingly it has been quoted that the lowest decomposition rates are achieved using Pyrex glass [PEROX HANDBOOK 67], while another source [McCORMICK 65] lists it as unknown.
<table>
<thead>
<tr>
<th>Material Type</th>
<th>Material Grade</th>
<th>Comment</th>
<th>AOL in test</th>
</tr>
</thead>
<tbody>
<tr>
<td>Al Alloy Cast</td>
<td>B356</td>
<td></td>
<td>2.8</td>
</tr>
<tr>
<td>Al Alloy Wrought</td>
<td>1060</td>
<td></td>
<td>1.5</td>
</tr>
<tr>
<td>Al Alloy Wrought</td>
<td>1060 (99.6)</td>
<td>99.6% pure Al</td>
<td>1.5</td>
</tr>
<tr>
<td>Al Alloy Wrought</td>
<td>1160</td>
<td></td>
<td>1.5</td>
</tr>
<tr>
<td>Al Alloy Wrought</td>
<td>1260</td>
<td></td>
<td>1.5</td>
</tr>
<tr>
<td>Al Alloy Wrought</td>
<td>1260</td>
<td>Pre-treated with H₂O₂ for 24 hours</td>
<td>0.7 to 0.8</td>
</tr>
<tr>
<td>Al Alloy Wrought</td>
<td>5052 (52S)</td>
<td></td>
<td>2.6</td>
</tr>
<tr>
<td>Al Alloy Wrought</td>
<td>5254 (XF54S)</td>
<td></td>
<td>0.3</td>
</tr>
<tr>
<td>Al Alloy Wrought</td>
<td>5254</td>
<td></td>
<td>1.7</td>
</tr>
<tr>
<td>Al Alloy Wrought</td>
<td>5254-0</td>
<td></td>
<td>0.3</td>
</tr>
<tr>
<td>Al Alloy Wrought</td>
<td>5254-H34</td>
<td></td>
<td>0.3</td>
</tr>
<tr>
<td>Al Alloy Wrought</td>
<td>5652-0</td>
<td>Requires anodising</td>
<td>5.0</td>
</tr>
<tr>
<td>Al Alloy Wrought</td>
<td>7072 (72S)</td>
<td></td>
<td>2.1</td>
</tr>
<tr>
<td>Tantalum</td>
<td>Pure material</td>
<td>No effect on material</td>
<td>Unknown</td>
</tr>
<tr>
<td>Polymer</td>
<td>Aclar</td>
<td>Allied Chemical</td>
<td>1.2</td>
</tr>
<tr>
<td>Polymer</td>
<td>Fluorel 2140</td>
<td>3M</td>
<td>0.3</td>
</tr>
<tr>
<td>Polymer</td>
<td>Fluorel 4121</td>
<td>Seals Eastern Inc</td>
<td>2.4</td>
</tr>
<tr>
<td>Polymer</td>
<td>Kel F</td>
<td>3M</td>
<td>1.3</td>
</tr>
<tr>
<td>Polymer</td>
<td>Kel F 800</td>
<td>3M</td>
<td>0.33</td>
</tr>
<tr>
<td>Polymer</td>
<td>Viton A</td>
<td>David Clark Co.</td>
<td>Unknown</td>
</tr>
<tr>
<td>Polymer</td>
<td>High Density Polythene</td>
<td>General Electric</td>
<td>2.9</td>
</tr>
<tr>
<td>Polymer</td>
<td>Vinylidon-Fluoride</td>
<td>Pall Filtration Corp.</td>
<td>Unknown</td>
</tr>
<tr>
<td>Polymer</td>
<td>Viton B (80S)</td>
<td>DuPont</td>
<td>0.6</td>
</tr>
<tr>
<td>Polymer</td>
<td>Teflon (white)</td>
<td>DuPont</td>
<td>2.8</td>
</tr>
<tr>
<td>Polymer</td>
<td>Teflon 100X FEP</td>
<td>DuPont</td>
<td>2.6</td>
</tr>
<tr>
<td>Polymer</td>
<td>Teflon Tape</td>
<td>DuPont</td>
<td>2.8</td>
</tr>
<tr>
<td>Polymer</td>
<td>Teflon Welded</td>
<td>DuPont</td>
<td>2.7</td>
</tr>
<tr>
<td>Polymer</td>
<td>Teflon TFE</td>
<td>DuPont</td>
<td>1.5</td>
</tr>
<tr>
<td>Polymer</td>
<td>Mylar 'A'</td>
<td>DuPont</td>
<td>5.0</td>
</tr>
<tr>
<td>Polymer</td>
<td>Mylar 'B'</td>
<td>DuPont</td>
<td>1.7</td>
</tr>
<tr>
<td>Glass</td>
<td>Pyrex</td>
<td></td>
<td>Unknown</td>
</tr>
<tr>
<td>Ceramic</td>
<td>Pyroceram</td>
<td>Corning Glass.</td>
<td>Unknown</td>
</tr>
<tr>
<td>Coating</td>
<td>Kel-F on 1060Al</td>
<td>Zero decomposition!</td>
<td>0.0</td>
</tr>
</tbody>
</table>

Table 2.3: Compatibility of materials with HTP from 1967 data [PEROX HANDBOOK 67]
Passivation

Passivation is the process of removing contaminants from the tank to increase its compatibility. The process is designed to remove the organic and catalytic elements present on the tank wall. In order for a tank to store hydrogen peroxide for extended periods it is vital that it goes through a passivation procedure. Studies have shown that without the surface treatment, the tank will never perform as well as a passivated tank [PEROX HANDBOOK 67]. From the literature there appears to be no common consensus on how to passivate parts. This may be down to different practices at different institutions, however a compromise can be reached by selecting the pertinent and common sections from the literature surveyed [PEROX HANDBOOK 67] [McCORMICK 65]. This is procedure is presented in table 2.4.

Stabilisation

The purpose of stabilisation is not to prevent the pure hydrogen peroxide from decomposing, but rather to prevent contamination from causing the catalytic decomposition [McCORMICK 65]. If the hydrogen peroxide can be kept perfectly clean and free of contaminants, then there would be no need for any stabilisers. However hydrogen peroxide tanks always suffers from incomplete removal of organic solvents, acids, detergents, chromic acid cleaning solutions, rinse waters or inadequate cleaning. The stabilisers that are added are used to inactivate catalytically-active substances, either dissolved or suspended in solution. Stabilisers do not stop the auto decomposition processes [PEROXIDE 55]. Of prime concern in the storage of hydrogen peroxide are the catalytic ions Cr^{3+}, Fe^{3+}, Cu^{2+}, Ni^{2+}, PO_4^{3-}, SO_4^{2-}, NO_3^{-}, Cl^{-} [PEROXIDE 55] and it is the purpose of the stabiliser to inactivate these species. Sodium stannate (Na_2SnO_3.3H_2O) forms by its hydrolysis colloidal hydrous stannic oxide, SnO_2.xH_2O. This absorbs catalytic ions such as ferric ions [PEROXIDE 55]. If the stannate is added in a too higher concentration, the pH rises and the stability of the hydrogen peroxide is reduced due to polarisation from the OH^- molecule. There is an optimum pH for the stannate to work effectively and this is between 3.5 and 6 [PEROXIDE 55]. The control of the amount of stannate in the hydrogen peroxide is left up to the manufacturer of their particular ‘flavour’ of hydrogen peroxide. Sodium pyrophosphate (Na_4P_2O_7.10H_2O) is a good stabiliser if Fe^{3+} contamination proves a problem. It is not however very effective against even a few ppm of copper compounds [PEROXIDE 55].
Table 2.4: Passivation procedure for HTP compatible materials

<table>
<thead>
<tr>
<th></th>
<th>Metals</th>
<th>Non Metals</th>
</tr>
</thead>
<tbody>
<tr>
<td></td>
<td>Stainless Steel</td>
<td>Aluminium Alloy</td>
</tr>
<tr>
<td><strong>Degreasing and Cleaning</strong></td>
<td>Cleaned with hot Decon 90 at 60°C. Scrubbed clean with distilled water to remove all traces of detergent. Handling after this stage should only occur with tongs/clean gloves.</td>
<td><strong>Degreasing and Cleaning</strong></td>
</tr>
<tr>
<td><strong>Descaling (Pickling)</strong></td>
<td>Immersed in room temperature 2M HF acid and 5M Nitric acid for no longer than 60 mins. After descaling rinse with distilled water at room temp.</td>
<td>Immersed in 0.5M HF acid 5 M Nitric acid for between 30s and 5 min at 45 °C max. After descaling, rinse with distilled water at room temp.</td>
</tr>
<tr>
<td><strong>Basic Passivation</strong></td>
<td>Immersed in 10M Nitric acid at 20°C for a min of 30 min duration. This should then be rinsed with distilled water and progress immediately onto the next stage.</td>
<td>Immersed in 10M Nitric acid at 20°C for a period of 1 hour. This should then be rinsed with distilled water and progress immediately onto the next stage.</td>
</tr>
<tr>
<td><strong>Propellant Conditioning</strong></td>
<td>Fill with 35% hydrogen peroxide and leave to stand for 24 hours. Following this fill with 90% HTP for 12 hours.</td>
<td></td>
</tr>
<tr>
<td><strong>System Assembly</strong></td>
<td>After system integration, the propellant conditioning with 35% hydrogen peroxide is required for 24 hours and then refilled with 90% HTP and repeated with 35% then 90%</td>
<td></td>
</tr>
</tbody>
</table>

The literature points to the fact that using stabilisers is a necessity in order to keep the decomposition rate down to a minimum, as absolute purity can neither be expected nor achieved in a working environment. However there is a problem with the stabilisers when used with catalyst decomposers. The phosphates in the hydrogen peroxide can cause a rapid increase of the pressure drop across the catalyst pack. Tin and aluminium in the hydrogen peroxide are deposited on the catalyst surface and this reduces the overall catalyst surface area while nitrates, phosphates and chlorides cause excessive erosion of the catalyst material. The stannates and phosphates that are used to ‘tie up’ the catalytic ions in the liquid will also try and do the same in the catalyst decomposer. This results in steady poisoning of the catalyst [PEROXIDE 55].

The process of decomposition occurs, as previously described, through heterogeneous and homogeneous channels, however there is also the vapour to consider. Hence there are five possible ways that the hydrogen peroxide can decompose.

1. Homogeneous reaction in the liquid phase between hydrogen peroxide and dissolved catalytic elements.
2. Heterogeneous reaction of the liquid phase, both bulk liquid and condensed vapour, with surfaces of the container and suspended particles.

3. Heterogeneous reaction of the film of condensed hydrogen peroxide on surfaces in contact with vapour phase.

4. Heterogeneous reaction of the vapour phase on dry surfaces.

5. Homogeneous decomposition of the vapour phase.

The reaction of the vapour phase on dry surfaces probably does not occur at room temperatures and reaction of the condensed film is of such a low significance it can be ignored [PEROX HANDBOOK 67]. The overall decomposition of the hydrogen peroxide can be summarised by equation 2.33.

\[
\frac{d(C_{HTP})}{dt} = \kappa_1(C_{HTP}) + \kappa_2(C_{HTP})(S/V)_{\text{liquid}} + \kappa_3(C_{HTP})(s/V)_{\text{vapour}} \tag{2.33}
\]

Where \( C_{HTP} \) is the concentration of the hydrogen peroxide and water solution, \( \kappa_1 \) is the liquid phase homogeneous decomposition rate, \( \kappa_2 \) is the liquid phase heterogeneous rate, \( \kappa_3 \) is the vapour phase heterogeneous decomposition rate. \((S/V)_{\text{liquid}}\) is the liquid surface to volume ratio and \((s/V)_{\text{vapour}}\) is the vapour phase surface to volume ratio. It has been found that the homogeneous decomposition rate for both stabilised and un-stabilised hydrogen peroxide is less than 0.03% AOL per year at 100°C, meaning that the vast majority of the decomposition stems from the heterogeneous decomposition.

2.5.4 Radiation Effect Upon Hydrogen Peroxide and Tank Polymers

As spacecraft are often required to operate in high radiation environments, such as the Van Allen radiation belts, radiolysis of the HTP and the effect upon the storage tank has to be taken into account. Metal storage tanks will be largely unaffected, but polymers can suffer serious degradation. The effect of radiation upon hydrogen peroxide has been investigated in the past, however the research performed was fragmented and inconsistent. The research into the effect of X-rays upon low concentrations of hydrogen peroxide shows a relationship where the number of moles of HTP decomposed per unit dosage increases as the square root of the concentration and decreases as the square root of the X-ray intensity [FRICKE 35][JOHNSON 51]. Actual rate constants are given for these experiments, but their applicability to HTP is limited. It has been
found that when the concentration is increased to high levels (>30%) the radiolysis mechanism now changes, as long chain reactions are occurring. This results in a rate of decomposition that is linearly proportional to concentration and increases with the square root of the radiation intensity [PEROXIDE 55]. Unfortunately no rate constants are given, thus calculations cannot be performed. In addition there appears to have been no research performed into forms of radiation other than X-rays. Electrons and protons would be a significant radiation risk, especially on GTO type orbits and their effect upon the hydrogen peroxide has yet to be determined. The general conclusion is that work, specific to rocket grade hydrogen peroxide decomposition, needs to be performed with simulated space radiation fluxes in order to build up an accurate picture of the effect on orbit. Arguably the most appropriate way to accomplish this would be to actually fly a sample accompanied by radiation detectors and monitor the decomposition in situ.

The use of polymers in space tanks also forces an understanding of the radiation effects to be appreciated. It is well known that the combination of X-rays (through bremsstralung), electrons, trapped protons and solar flare protons all contribute to the degradation of PTFE type materials. While FEP is superior to PTFE, the total maximum dose that it can receive is between $10^6$ and $10^7$ rads before the structure begins to break down [HANSEN 01]. A rad is defined as 0.01J of received energy per kg of material. After $10^6$ rads the long chains are broken down into shorter lengths, which results in embrittlement. What was not determined however is the degradation of chemical inertness prior to the $10^6$ rads limit. It would be expected that with increasing total dose the level the compatibility would decrease. Numerical values of this parameter are absent from the literature and thus it would have to be determined experimentally. However if the $10^6$ rad limit is taken as the maximum, a Molniya type orbit with 1mm thickness of aluminium shielding the FEP would be able to survive for 2 years before serious degradation would occur. Similarly a LEO, 12:00 LT, sun-sync orbit would be able to survive for 40 years before the limit is exceeded [HANSEN 01].

2.6 Hydrogen Peroxide as a Monopropellant

The simplest use of hydrogen peroxide is in a catalytic monopropellant system and examples of these include the Reaction Control Systems (RCS) for the Bell X-1 supersonic aircraft, North American X-15 rocket plane, USAF NF-104A, US Mercury spacecraft, Centaur Upper stage RCS, Centaur Propellant Settling, Scout rocket roll control, Little Joe II Apollo capsule test vehicle, Burner II, SATAR, SYNCOM satellite, COMSAT satellite and HS303A bus [McCORMICK 65]. In all the applications, HTP was chosen where reliability was paramount. In addition to actual propulsive decomposition of the hydrogen peroxide, it can also be used as a gas generator. This has been used to power the turbo-pumps of the German V-2 WWII missile and
the Redstone US missile [McCORMICK 65] [STOKES 98], which was used to launch the manned mercury spacecraft into a sub-orbital trajectory. A typical configuration for a monopropellant thruster is shown in figure 2.4. This thruster is equipped with a low temperature start capability through the implementation of the radial injection and scroll catalyst arrangement. If low temperature starts are not an issue then direct injection into the screen catalyst zone is permitted. It is believed that under low temperature conditions and certainly before the HTP changes phase, the catalytic decomposition is initiated by silver dissolving into solution and then precipitating out [ANDREWS 90][BAUMGARTNER 63]. Thus by adding the scroll section, the HTP can pick up a large quantity of silver before it moves into the main catalyst zone, thus initiating the decomposition process. At higher temperatures the dissolving and re-precipitation effect is replaced by another mechanism which is rate determined by the heat transfer in the pack [BAUMGARTNER 63b]. Thus there is no need for the scroll section if the pack is hot when the HTP is allowed to flow. Even when using cold HTP, some form of electrical heating can be implemented to allow the pack to be used without a scroll section [McCORMICK 65]. Recently General Kinetics LLC of Aliso Viejo CA, have qualified a series of monopropellant thrusters and gas generators for hydrogen peroxide use. These thrusters are between 3 and 25 lbf (13N and 111N) and are designed for satellite and launch vehicle attitude control [GEN KIN 99].

Considerable work into the hydrogen peroxide catalysts has been performed in the past and the general conclusion is that silver screens perform the best. Porcelain stones impregnated with metal salts such as Potassium Permanganate or Calcium Permanganate are an alternative to silver screens, which were investigated in the post WWII era. The stone type of catalyst was used in the gas generators of the V2 turbo-pumps but post WWII the catalyst stones were replaced in favour of silver screens for non-naval applications. The principal advantage of the silver screens is it can tolerate a higher loading factor and has potentially longer life. However stones impregnated with

Figure 2.4: Typical ACS monopropellant engine
permanganates, are tolerable to chloride ions, whereas silver is not, and hence were preferred for submarine torpedoes [LIQUID PROP 57].

To date, work still continues into producing a more effective catalyst with hydrogen peroxide and recently the work has centred around the use of alumina (Al₂O₃) as the substrate and silver nitrate deposited on the surface [ELOIRDI 01]. The principal advantage of this is that it has a large surface area in comparison to the silver gauze and hence should be more effective. However tests showed that while the alumina/silver system was more active at lower temperatures than silver gauze but at higher temperatures (>373 K) the silver gauze was just as effective [ELOIRDI 01]. Other modern investigations into catalyst packs have centred purely on silver gauze or coated silver gauze screens [MORLAN 99].

2.7 Hydrogen Peroxide and Kerosene Rocket Engines

Hydrogen peroxide as the oxidiser and kerosene as the fuel for a bi-propellant rocket engine dates back to the 1940's where considerable amount of work was conducted on the subject particularly in the UK. Amongst the engines that were developed, the following were notable

1. DeHavilland Super Sprite, rocket assisted take off pack
2. Saunders Roe Spectre, rocket power system for the SR-53 interceptor aircraft
3. Napiers Scorpion, rocket propulsion of the RAF Lightning interceptor aircraft
4. Bristol Siddeley Gamma 2 / Gamma 201, launch vehicle engine for Black Knight and Black Arrow
5. Rolls Royce Gamma 301 / Stentor, Blue Steel nuclear stand off bomb propulsion unit
6. KP series of engines, for tactical missiles
7. Beta Mk I and Mk II development engines at R.O. Westcott

In addition to the UK, the US was also interested in hydrogen peroxide/kerosene engines in the late 1950's and this reached fruition with the AR series of engines, produced by Rocketdyne for the USAF. The culmination of the effort became the AR2-3 engine of 6,000lbf (26,700N) thrust used to good effect in the NF-104, the rocket powered Lockheed F-104 Starfighter. The AR2-3 was later designated the LR-40 when it was developed for the US Navy by Reaction Motors Inc [WERNIMONT 99].
The vast majority of the hydrogen peroxide/kerosene rocket engines used a turbo pump to feed the propellants into the chamber. The single exception to the rule appears to be the KP series of rocket engines. Even the relatively simple Super Sprite, which was designed as a RATO unit, utilised a pump fed system. The KP-1 engine directly injected the HTP and kerosene and used a hypergolic slug to ignite the propellants. The KP-3 3,000 lbf (13,345 N) engine was the first in the series that employed thermal ignition and used silver coated screens, however due to long ignition delays, 83% HTP was selected over the 80% HTP which the design was initially baselined for. This engine was later used as the sustainer propulsion system on the Red Shoes ground-to-air missile [HARLOW 99]. All of these engines used the relatively high specific heat capacity of the hydrogen peroxide for regenerative cooling. This was further aided by the fact that the O/F ratio was exceptionally high. In the Super Sprite, which was deliberately run oxidiser rich, with a ratio of 22:1, thus giving large quantities of oxidiser for cooling of the combustion chamber, throat and nozzle [SNELL 63]. Other engines such as the Spectre ran at a mixture ratio of 9:1 which being close to stochiometric produced higher chamber temperatures and consequently high specific impulses. Figure 2.5 shows the chamber region of the Spectre 4 engine including the fuel injection system and catalyst pack. In the Spectre engine, part of the HTP flow was decomposed on a catalyst pack, which powered the main turbo-pump. The rest of the HTP flowed into the main catalyst pack and then into the combustion chamber where it was burnt with the fuel sprayed in through injectors. The turbo-pump exhaust was also fed into the
Chapter 2: Literature Survey

chamber, thus making an efficient system, with this cycle known as ‘topping’. Cooling was achieved by passing the total flow of the HTP around the chamber and then into the respective catalyst packs, thus achieving a zero loss cooling system. This is the same system as used in the US LR-40 engine. The Gamma series of engines used a spray injector to atomise the fuel entering the chamber whereupon it was burnt with the decomposed HTP thus creating an auto-ignition system. However an entirely different form of fuel atomisation using decomposed HTP has been demonstrated. It has been known that a high velocity gas has the capacity to produce effective atomisation of a jet of liquid, in this case fuel. Thus by decomposing the HTP and injecting the fuel at a high shear velocity, effective atomisation can occur [ANDREWS 90]. This technique was implemented recently with a 1,000lbf (4,450 N) thrust test engine developed by Kaiser Marquardt and Orbital Sciences Corporation. The test engine used ring type injectors where fuel is sprayed in radially into the steam port through which the high velocity steam and oxygen caused atomisation [WU 99][FULLER 00].

2.7.1 Performance of Flown Engines

The vast majority of bi-propellant hydrogen peroxide/kerosene engines have been designed to operate at sea level or high altitude, few if any are designed to operate in space. Arguably the highest performance engine was the Bristol Siddeley Gamma 301, which was the successor to the Gamma 201 as flown on the Black Knight and Black Arrow launch vehicles. The Gamma 301 engine and the AR2-3 engine had impressive performance characteristics shown in table 2.5 [SUNLEY 90][VENTURE 00]. These figures show that the engines are capable of producing around 250 seconds of Isp in a vacuum, even with the relatively short expansion nozzles.

<table>
<thead>
<tr>
<th>Parameter</th>
<th>Gamma 301</th>
<th>AR2-3</th>
</tr>
</thead>
<tbody>
<tr>
<td>Thrust</td>
<td>Variable 17,000lbf to 21,600lbf (75.6 to 96.1 kN)</td>
<td>3,300-6,000lbf (14.7 to 26.7 kN)</td>
</tr>
<tr>
<td>Area Expansion ratio</td>
<td>60:1</td>
<td>12:1</td>
</tr>
<tr>
<td>Chamber pressure at maximum thrust/ Bar</td>
<td>41.4</td>
<td>19.3</td>
</tr>
<tr>
<td>Isp at sea level / s</td>
<td>251</td>
<td>232</td>
</tr>
<tr>
<td>Isp in vacuum / s</td>
<td>265</td>
<td>245</td>
</tr>
</tbody>
</table>

Table 2.5: Gamma 301 and AR2-3 engine performance parameters
By extending the nozzles to the usual 150:1 or 200:1 vacuum engine expansion ratio, the Isp will correspondingly increase and an Isp of 300 seconds should be theoretically possible.

### 2.7.2 Cooling Methods

Engineering a bi-propellant rocket engine that produces good combustion is vital to achieve an efficient system, but the cooling of the engine is equally critical. At the thrust levels and durations applicable for a small satellites, the engine has to be actively cooled to prevent the engine from melting in all but the most exotic of chamber materials. A 2D carbon-carbon radiatively cooled engine was produced, however this is extremely expensive due to manufacturing costs [SUTTON 92]. The two dominant mechanisms by which heat is transferred in a rocket engine is forced convection and radiation. It can be shown that in the typical rocket chamber/nozzle geometry, radiation is dominant in the chamber region where the velocities are low. However in the throat region, where there are sonic speeds, the convective heating component becomes dominant. Figure 2.6 shows the typical profile of the heat transfer. Whenever there is a temperature difference, as occurs between the hot combustion gases and the cooler wall, heat will flow. The convective and radiative heat transfer components are shown in equations 2.34 and 2.35 respectively [LIQUID PROP 57b].

\[
\dot{Q}_{\text{con}} = A_{\text{sur}} h_{\text{gas}} (T_1 - T_2) \quad (2.34)
\]

\[
\dot{Q}_{\text{rad}} = \sigma A_{\text{sur}} \varepsilon_{\text{gas}} \alpha_w (T_{\text{gas}}^4 - T_w^4) \quad (2.35)
\]

Where

- \(T_1 - T_2\) = Temperature difference over which the transfer occurs (K)
- \(h_{\text{gas}}\) = Convective heat transfer coefficient of the hot gases (Wm\(^{-2}\)K\(^{-1}\))
- \(A_{\text{sur}}\) = Area over which heat transfer occurs (m\(^2\))
- \(\sigma\) = Stefan-Boltzmann constant, \(5.6696 \times 10^{-8} \text{Wm}^{-2} \text{K}^{-4}\)
- \(\alpha_w\) = Absorptivity of the chamber wall, taken as unity due to the chamber acting as a cavity.
- \(\varepsilon_{\text{gas}}\) = Emissivity of the combustion gases
- \(T_{\text{gas}}\) = Gas temperature (K)
- \(\dot{Q}\) = Power (W)

The temperature gradient across a cross section through an engine can be approximated to a series of steps as shown in figure 2.7, where it can be seen that the temperature of the gases is the highest in the centre (\(T_c\)) but as moving radially outwards, the temperature drops off down to the hot side of the chamber wall (\(T_{wh}\)). This drop off is due to the boundary layer that forms against
the wall and acts as an insulator. As the wall also provides a degree of resistance to the flow of heat, there will be a temperature drop across the wall to the cooler side \(T_{wc}\). Finally there will be a drop down to the ambient air temperature \(T_{air}\). When the engine is operating in a vacuum, radiation is the only form of natural cooling that can occur. Thus without any form of forced cooling, the engine will reach equilibrium according to equation 2.36.

\[
\sigma e_w T_{we}^4 = \sigma e_{gas} \alpha_{wh} (T_{gas}^4 - T_{wh}^4) + h_{gas} (T_{gas} - T_{wh})
\]  

(2.36)

The factor that prevents an immediate equilibrium temperature calculation through equation 2.36 is the convective heat transfer coefficient \(h_{gas}\). This parameter has been calculated empirically and is a function of the Reynolds, Prandtl and Nusselt numbers. The form of the equation depends...
upon the nature of the fluid as gases have a fixed Prandtl number irrespective of the gas in question. The convective heat transfer coefficients for the gases and liquids, \( h_{\text{gas}} \) and \( h_{\text{liq}} \), are shown in equations 2.37 and 2.38 respectively [LIQUID PROP 57b].

\[
\begin{align*}
    h_{\text{gas}} &= 0.021 \frac{k_{\text{gas}}}{d_{\text{hyd}}} \left( \frac{m_{\text{gas}} d_{\text{hyd}}}{A \mu_{\text{gas}}} \right)^{0.8} \\
    h_{\text{liq}} &= 0.023 \frac{k_{\text{liq}}}{d_{\text{hyd}}} \left( \frac{m_{\text{liq}} d_{\text{hyd}}}{A \mu_{\text{liq}}} \right)^{0.8} \left( \frac{C_p \mu_{\text{liq}}}{k_{\text{liq}}} \right)^{0.333}
\end{align*}
\] (2.37, 2.38)

Where,

- \( k \) = Thermal conductivity, Wm\(^{-1}\)K\(^{-1}\)
- \( d_{\text{hyd}} = 4 \times \) Hydraulic Depth\(^1\), m
- \( m \) = Mass flow, liq=liquid, g=gas
- \( A \) = Cross sectional area, m\(^2\)
- \( \mu \) = Absolute viscosity, kgm\(^{-1}\)s\(^{-1}\)
- \( C_p \) = Specific heat capacity at constant pressure, Jkg\(^{-1}\)K\(^{-1}\)

Thus by knowing the physical parameters of the gas and equations 2.35, 2.36, 2.37 and 2.38 it is possible to determine the equilibrium temperature of the rocket engine. The use of these equations has been shown to agree well with actual tested engines [LIQUID PROP 57b].

Ideally an engine would constructed from low cost materials such as stainless steel. However stainless steel cannot tolerate a temperature above 1,000°C without serious loss of strength and subsequent melting [SUTTON 92]. Nickel and Cobalt alloys are tolerable of higher temperatures, up to 1,150°C and offer a higher degree of oxidation resistance [SUTTON 92] [LAWRENCE 98]. Alternatively a high temperature alloy can be used instead, albeit with an order of magnitude price increase. Table 2.6 shows the refractory materials that can be used as combustion chambers [ROYAL ORD] bearing in mind that typical combustion temperatures are around 2,500K or greater.

---

\(^1\) Hydraulic Depth is defined as the cross sectional flow area divided by the wetted perimeter
<table>
<thead>
<tr>
<th>Material</th>
<th>History</th>
<th>Temperature Limit /K</th>
<th>400N Apogee Chamber Cost</th>
</tr>
</thead>
<tbody>
<tr>
<td>Columbium</td>
<td>Use on Space Shuttle and LEROS with Hitempco coating</td>
<td>1,673</td>
<td>£6,000</td>
</tr>
<tr>
<td>Iridium lined Rhenium</td>
<td>Kaiser Marquardt and Aerojet developed chambers</td>
<td>2,423</td>
<td>£50,000</td>
</tr>
<tr>
<td>Iridium/32% Rhenium lined Tantalum-Tungsten</td>
<td>5 lbf (22N) chamber tested successfully</td>
<td>2,173</td>
<td>£12,000</td>
</tr>
<tr>
<td>Platinum/Rhenium</td>
<td>Tested on ARC 5lbf (22N) engine</td>
<td>2,550</td>
<td>£28,000</td>
</tr>
</tbody>
</table>

Table 2.6: High temperature refractory alloys

Even when using the high temperature alloys, radiation cooling alone is not enough to prevent a bi-propellant engine from melting and hence an active cooling system would have to be employed. The cooling methods used on commercial engines are usually liquid film cooling, ablative cooling, regenerative cooling or a combination of these systems.

**Liquid Film Cooling**

In liquid film cooling, either the liquid fuel or liquid oxidiser is sprayed onto the inside walls of the combustion chamber. This produces a thin film of fluid against the wall that cools and insulates. As the fluid moves down the wall it reaches the boiling point and then changes phase into gas, but the fluid still stays as a film against the wall, thus protecting it. Equation 2.39 describes the thermal flux balance of liquid film cooling [HUMBLE 95], which was derived by assuming the cooling occurs a localised zone against the wall.

\[
(\dot{Q}_{\text{rad}} + \dot{Q}_{\text{con}})_{\text{in}} = \dot{Q}_{\text{wall}} + \dot{m}_{\text{film}} \int_{T_{\text{vap}}}^{T_{\text{in}}} \left( c_{p,\text{gas}} dT + H_{\text{fg}} + c_{p,\text{liq}} \left[ T_{\text{vap}} - T_{\text{in}} \right] \right)
\]  

(2.39)
Where

\[ \dot{Q}_{\text{rad}} = \text{Radiative heat flow into film cooling zone, (W)} \]
\[ \dot{Q}_{\text{con}} = \text{Convective heat flow into film cooling zone (W)} \]
\[ \dot{Q}_{\text{wall}} = \text{Heat flow into wall (W)} \]
\[ \dot{m}_{\text{film}} = \text{mass flow rate of the film coolant (kgs}^{-1}) \]
\[ c_{p,\text{gas}} = \text{Mean heat capacity of the vaporised liquid coolant (Jkg}^{-1}\text{K}^{-1}) \]
\[ H_{fg} = \text{Heat of vaporisation of the liquid (Jkg}^{-1}) \]
\[ c_{p,\text{liq}} = \text{Mean heat capacity of the liquid (Jkg}^{-1}\text{K}^{-1}) \]

The fuel or the oxidiser can be used as the coolant but more typically it is the fuel that is used. This is essentially because very hot oxidiser can present problems with the high temperature alloys oxidising. The Russian LTRE400N engine is exceptional in that it uses nitrogen tetroxide (the oxidiser) as the coolant [NIIMASH 00]. The primary driver for using the oxidiser is believed to lie in the density Isp advantage. Oxidisers generally have a higher density than the fuel, and hence by utilising the higher density propellant for cooling the overall density Isp can increase over a fuel film cooled engine. Fuel film cooling is used almost exclusively on all small bi- propellant rocket engines in the sub 1,000N thrust class, and this fact alone demonstrates the effectiveness. The cooling effect can be increased by increasing \( \dot{m}_{\text{film}} \) as can be seen from equation 2.39, but this fuel is not burnt and hence contributes little thrust to the overall system. Thus the overall Isp will decrease as a consequence. A variation on the standard film cooling option is the novel vortex oxidiser film coolant system. Orbitec have demonstrated a LOX/RP-1 engine, which uses a LOX vortex against the chamber walls as a film coolant. In this configuration, the combustion products do not contact the wall, thus allowing standard materials to be used [ORBITEC 01]. The engine is hypothesised to work well due to the stability of the vortex reducing film break-up and thus superior cooling efficiency.

**Ablative Cooling**

Ablative cooling is a process by which the nozzle and chamber erode thus cooling the engine. The nozzle is made from a material that has a large enthalpy of vaporisation, thus the change of state that occurs through erosion carries away large quantities of heat. In addition the charred products that remain have a low thermal conductivity, thus providing an insulative layer. Typically this would be used on solid rocket motors as opposed to liquid engines, but the system would still work on a liquid engine. However the constantly enlarging throat makes satellite
operations difficult because the chamber pressure is falling thus so will the thrust. From a satellite operator’s point of view, this adds an extra dimension of complexity that should be avoided [HAAG 01b]. The basic equation that governs ablative cooling is equation 2.40 with the parameters of ablative material mass flow rate and the enthalpy of material ablation usually found empirically. Another problem with ablative cooling is it usually lasts for only short periods. This is ideal for short one-shot solid systems, but not so appropriate for long burn times used on liquids.

\[ \left( \dot{Q}_{\text{rad}} + \dot{Q}_{\text{con}} \right)_{\text{in}} = \dot{m}_{\text{ablative}} H_{\text{ablative}} \] (2.40)

Where

\[ \dot{m}_{\text{ablative}} = \text{The mass flow of the ablative material (kgs}^{-1}) \]
\[ H_{\text{ablative}} = \text{Enthalpy of material ablation (Jkg}^{-1}) \]
\[ \dot{Q}_{\text{rad}} = \text{Radiative heat flow into ablative cooling zone (W)} \]
\[ \dot{Q}_{\text{con}} = \text{Convective heat flow into ablative cooling zone (W)} \]

**Regenerative Cooling**

The most efficient form of cooling available is the regenerative system where either the fuel or the oxidiser is fed around the outside of the combustion chamber and back into the engine. Using such a system, none of the propellants are lost in the process and most of the heat is recovered. Figure 2.8 shows a typical schematic of the cycle as used in the Spectre and AR2-3 engines. The effectiveness of regenerative cooling centres on the mass flow rate of the coolant and temperature rise over which the coolant remains stable. The higher the specific heat capacity, the lower the mass flow of the coolant needs to be to produce the same temperature rise. Equation 2.41 describes the cooling heat balance formally.

\[ \left( \dot{Q}_{\text{rad}} + \dot{Q}_{\text{con}} \right)_{\text{in}} = (\dot{Q}_{\text{rad}})_{\text{out}} + \dot{m}_{\text{cool}} c_p \left( T_{\text{cool, out}} - T_{\text{cool, in}} \right) \] (2.41)

Where

\[ \dot{Q}_{\text{rad}} = \text{Radiative heat flow into or out of regenerative cooling zone (W)} \]
\[ \dot{Q}_{\text{con}} = \text{Convective heat flow into regenerative cooling zone (W)} \]
Regenerative cooling is used almost exclusively for larger engines, of the several hundred Newton thrust level and above. The reason for the effectiveness at high thrusts is due to the high mass flow rate of the oxidiser or the fuel. As all the oxidiser or fuel first goes through the cooling jacket, very effective cooling at high mass flow rates is possible. A good example of this is the Gamma 201 engine as used on Black Knight where all the HTP is fed around the chamber and back into the catalyst pack [SUNLEY 90].

\[ m_{\text{cool}} = \text{Mass flow of coolant (kgs}^{-1}) \]

\[ c_p = \text{Mean heat capacity of fluid over temperature range of interest (Jkg}^{-1}\text{K}^{-1}) \]

\[ T_{\text{cool, in}} = \text{Temperature of the fluid entering the cooling region (K)} \]

\[ T_{\text{cool, out}} = \text{Temperature of the fluid leaving the cooling region (K)} \]
Alternative Cooling Schemes

Besides regenerative, ablative and film cooling there are other systems that are less well known. These systems make use of a cooler gas than the combustion products, injected into the chamber wall region, which insulates and cools the wall. Systems using the exhaust from turbo-pumps at 700 to 1,000°C, have been used in a similar way to cool the exit cones of large engines, the Rocketdyne F-1 engine being one example [SUTTON 92]. Figure 2.9 shows possible gas/liquid film cooling mechanisms [SUTTON 92].

The concept of using gas as the coolant and insulator has been researched before but in the application of aircraft cooling using atmospheric air. In addition the high performance jet engines have used the compressor air bleed forced out of holes in the turbine blades, forming a protective cooling sheath. This technology has however not been implemented to any great extent for cooling of rocket engines. The concept of gas cooling can be broken into three distinct mechanisms, Convection, Transpiration and Film Cooling. Figure 2.10 shows graphically the modes of operation of the three different cooling mechanisms. Convective cooling uses a coolant flowing on the outside of the chamber, while transpiration cooling forces coolant through a porous wall to form a protective layer on the inside of the wall. Finally film cooling forms a cool boundary layer on the inside of the chamber. From the work achieved, which considers a flat plate being cooled by air, transpiration cooling is the most effective system [ECKERT 53]. This was closely followed by the forced convection cooling mechanism. When a single injection port was used for film cooling the degree of cooling was low in comparison to the other mechanisms. However over short distances away from the injection port, film cooling was reported to be effective. If the number of injection ports was increased, so that the film boundary layer can be

Figure 2.9: Gas/liquid film cooling systems [SUTTON 92]
regenerated downstream, then film cooling becomes effective, but at the expense of extra complexity. It should be noted that as the number of injection ports increases, film cooling tends to transpiration cooling. Figure 2.11 shows the effect of the three cooling mechanisms in terms of the temperature difference ratio and coolant flow ratio for a flat plate. This data is presented under zero radiation conditions and Reynolds number of 10⁷. The temperature difference ratio is defined in equation 2.42 and the coolant flow ratio in equation 2.43

\[ TemperatureDifferenceRatio = \frac{T_w - T_a}{T_g - T_a} \]  \hspace{1cm} (2.42)

\[ CoolantFlowRatio = \frac{\rho_a V_a}{\rho_g V_g} \]  \hspace{1cm} (2.43)

Where:
- \( T_w \) = Temperature of the wall (K)
- \( T_a \) = Air temperature (coolant) (K)
- \( T_g \) = Hot gas temperature (K)
- \( \rho_a \) = Air (coolant) density (kg m⁻³)
- \( V_a \) = Air (coolant) velocity (m s⁻¹)
- \( \rho_g \) = Hot gas density (kg m⁻³)
- \( V_g \) = Hot gas velocity (m s⁻¹)
Chapter 2: Literature Survey

Number of slots, Cooling — Convection — Transpiration

Thermal

0.6

0.004 J0 0 8

Coolant-flow ratio, \( \frac{w*}{\rho g V_g} \)

0.002

0.006

0.008

0.010

0.012

Turbulent flow without radiation. Reynolds number, \( Re_a 10^7 \).

Figure 2.11: Comparison of gas cooling systems [ECKERT 53]

The thermal effectiveness parameter \( \eta_T \) is a measure of the temperature rise of the coolant air, and a value of 1 results in the coolant rising up to the wall temperature when it reaches the end of the plate. The definition of this parameter is given in equation 2.44

\[
\eta_T = \frac{T_{a,exitplate} - T_a}{T_w - T_a}
\]

The conclusions from the work [ECKERT 53] are that transpiration cooling is the most effective option, but arguably the most complicated to implement. Convective cooling offers the advantage of simpler implementation, but still modest performance. Film cooling can only be used to cool small areas, a short distance away from the injection point, but has the advantage over convective cooling that the hot gases are not directly in contact with the wall material.
2.8 Propellant Tanks for Hydrogen Peroxide

In order for hydrogen peroxide to be flown in space, there needs to be a way of storing the propellant on board for extended periods. As §2.5.3 describes, the HTP will naturally decompose on the tank walls, and it is the goal of a tank design to minimise the decomposition rate. Historically the most compatible tank materials have been the aluminium alloys 5254-O and 5254-H34 [McCORMICK 65]. But even with these materials, HTP will suffer a significant decomposition over a long period of time. Another problem with these materials is their strengths are very low. This makes the high-pressure storage vessels thick walled and consequently heavy. Table 2.7 compares the strength of 1060 aluminium, 5254 aluminium and 316 Stainless Steel [MATWEB 02].

<table>
<thead>
<tr>
<th>Material</th>
<th>Maximum Tensile Strength/MPa</th>
<th>Yield Tensile Stress/MPa</th>
<th>Tensile Modulus /GPa</th>
</tr>
</thead>
<tbody>
<tr>
<td>1060 Aluminium</td>
<td>70</td>
<td>30</td>
<td>69</td>
</tr>
<tr>
<td>5254 Aluminium</td>
<td>240</td>
<td>115</td>
<td>70</td>
</tr>
<tr>
<td>316 Stainless Steel</td>
<td>580</td>
<td>290</td>
<td>193</td>
</tr>
</tbody>
</table>

Table 2.7: Mechanical properties of tank materials

The alternative to a tank made entirely from a compatible material is to use an outer shell of some high strength material such as Titanium or Stainless Steel and use a compatible liner in contact with the HTP. A manufacturer of high strength hydrogen peroxide, Degussa, ship their propellant in polythene lined stainless steel [PAULS 99] drums, while FMC ship in 1060 grade aluminium alloy drums [YUAN 01]. The liner could either be a polymer, as in the Degussa drum case, or it could be pure aluminium deposited through electroplating, physical vapour or chemical vapour deposition. Using such a system could potentially reduce the mass of the tanks. It is known that the Syncom II satellite manufactured by Hughes (now owned by Boeing) used hydrogen peroxide for its reaction control system. This satellite operated on orbit for two years and used an all welded 1060 aluminium propellant tank and feed system [McCORMICK 65]. The Syncom II satellite, being a spinning system, had no need for a propellant management device (PMD) and could easily be vented, but no information as to whether this took place could be found. In non-spinning spacecraft the reported expulsion system of choice is to use a bladder type tank, using 3M Fluorel 4121 elastomer [McCORMICK 65]. It is unclear how long a bladder type system would be viable, due to oxygen evolution on the liquid side of the bladder. Gas free propellant
feed can be achieved by using the bladder to separate the liquid from the gas [BALLINGER 99], but in a HTP system this is only the case just after the tank has been filled. Thus after a period of time, there would be significant quantities of oxygen on the liquid side of the bladder which would introduce gas into the feed lines. An alternative system is to use a PMD constructed from 5254 aluminium to provide gas free flow to the feed system. The PMD relies upon surface tension of the propellant to fill a sponge at the base of the tank. Vanes provide a path for the liquid to flow and under low accelerations thus gas free feed is achieved. Such a system would be similar to the type of PMD’s that are found in propellant tanks of communication satellites. Typically in communication spacecraft there needs to be a feed for the thrusters regardless of the direction of firing, albeit with small accelerations [DEBRECENI 96] [TAM 96]. However if the requirement is imposed that only high accelerations will be required in one axis, then the PMD can be reduced in complexity. Figure 2.12 shows the schematic of a possible simplified PMD system for hydrogen peroxide.

As far as launch vehicles that use HTP are concerned, the exact propellant tank construction remains elusive, however several references refer to an aluminium alloy hydrogen peroxide tank for the Black Knight launch vehicle [SUNLEY 90][ANDREWS 90]. Thus it can be expected that either 1060 or 5254 grade alloys were used. For launch vehicles there is only limited concern with gas in the propellant lines as gravity and the acceleration of the vehicle provides the necessary separation. In addition the compatibility of the propellant tanks are not as important as in a satellite due to the limited time the HTP is stored for. It is known that the Soyuz-TM spacecraft uses hydrogen peroxide for its landing engine [FROLOV 98] and with an on orbit
lifetime of 6 months, it is clear that the HTP must be kept in very compatible tanks. It has been suggested that the tank is a bag arrangement that is impregnated with PTFE [WHITEHEAD 00]. This again raises the issue of gas being generated on the wrong side of the bladder. However as the engine is designed to work upon landing, the system is then operating in a 1 'g' environment and hence this ceases to be a problem.

2.9 Conclusions from the Literature Survey

Hydrogen peroxide makes an excellent oxidiser due to its high density and ease of ignition with fuels such as kerosene, however arguably the biggest concern is the storage of the oxidiser on orbit for extended periods. In any propellant tank it is vital to maintain absolute cleanliness and hydrogen peroxide is no exception. Using compatible materials, and the recommended passivation procedure, low decomposition rates are possible. The fact that the Syncom II satellite used HTP on orbit for two years demonstrates that on orbit storage for extended periods is indeed feasible. The standard method of using a bladder style tank for ensuring gas free liquid propellant flow is almost certainly unacceptable and thus the PMD design is the superior choice. Failing that, the spacecraft would have to be spin stabilised to provide the required gas/liquid separation. This puts excessive requirements on the mission and should be avoided.

Few if any HTP/kerosene engines have flown in space, with the exception of launch vehicles, however there is no reason to suspect that such engines would have problems in space. The literature survey has also identified the regular cooling systems that can be used on rocket engines and also unusual cooling systems that rely on gas as opposed to liquid cooling. Such a coolant option could be appropriate for a HTP system when used in conjunction with a gas generator catalyst pack. No literature was found on cooling systems for HTP/kerosene engines that use decomposed HTP as a coolant, but similar systems using turbo-pump exhaust products as a coolant were found.

A HTP/kerosene system should be efficient both from a mass and, crucially, volume point of view, with a performance comparable to a traditional hypergolic engine. The high density of the hydrogen peroxide, provides a high density Isp ideal for small spacecraft. The single biggest challenge facing the use of these propellants is the storage of the oxidiser for long periods on orbit. Closely following this challenge is the cooling of the combustion chamber/nozzle of the engine. The extreme heat fluxes intrinsic to rocket combustion must be dealt with in an efficient manner and the possibility of decomposed HTP film cooling is a potential option. Crucially the cost of exotic refractory metal chambers are excessive for low cost propulsion, thus alternative metals have to be considered such as Nickel and Cobalt alloys.
Chapter 3

3 Theoretical Thruster Design

3.1 Introduction
3.2 Propellant Choice
3.3 Cooling Mechanism Choice
3.4 Injector Design
3.5 Chamber Geometry Optimisation
3.6 Propellant Concentration Optimisation
3.7 Nozzle Geometry Selection
3.8 Conclusions
3 Theoretical Thruster Design

3.1 Introduction

This chapter discusses the theoretical design aspects of the bi-propellant engine. Initially a propellant choice is made on the basis of providing maximum density Isp while maintaining a 'green' propellant combination. The method of cooling the engine is analysed with reference to the standard cooling methods and additionally the novel gas film cooling system. Performance estimates of the cooling systems are made and the most appropriate method selected. The injector design is discussed with reference to previous research and the geometry of the chamber and propellant concentrations are theoretically optimised to maximise the density Isp. Finally the nozzle geometry is selected from both a cost of manufacture and performance basis.

3.2 Propellant Choice

The bi-propellant rocket engine can be fed by a multitude of possible oxidisers and fuels; imagination and chemical engineering expertise are the only limit on possible choices [CLARK 72]. However with the requirement of 'green' applied to the discussion, the number of potential candidates decreases dramatically. Using green propellants has the potential for reducing the cost of the system considerably due to reduced Ground Support Equipment (GSE) and safety infrastructure. As an example, the oxidiser used in the traditional hypergolic satellite engine is MON\(^1\), the acronym for Mixed Oxides of Nitrogen. This makes a very good oxidiser due to its high density and good performance with a wide range of fuels. It is particularly effective with hydrazine due to its hypergolic nature and an Isp in excess of 300 seconds [HUMBLE 95]. Unfortunately MON has very serious health implications, which prohibits its classification as a green propellant. The high vapour pressure results in highly toxic fumes from any spillage [SUTTON 92]. The requirement for safe handling of MON includes hand, head, face and body protection in the form of vinyl coated fibreglass protective suit and breathing apparatus [HAZARDS 69]. The strenuous safety procedures that have to be adhered to prohibits a low cost approach thus a green oxidiser, with few safety requirements, is attractive for a low cost system.

\(^1\)Nitrogen tetroxide is a liquid over a very narrow temperature and pressure range. It readily freezes or boils and as such it is usually mixed with the small amount Nitric Oxide, NO. This has the effect of suppressing the melting point but increases the vapour pressure. NO is added to \(\text{N}_2\text{O}_4\) in proportions between 2 and 30% total content by mass [SUTTON 92], and the mixture produced is termed MON, for Mixed Oxides of Nitrogen.
3.2.1 Potential Oxidisers

The two green oxidisers have been identified as potentially useful are hydrogen peroxide, H$_2$O$_2$ and nitrous oxide, N$_2$O. Their physical properties are listed below in table 3.1 [PEROX HANDBOOK 67] [MATHESON 66] [ANDREWS 90]. Neither propellant requires the extreme safety precautions associated with MON.

<table>
<thead>
<tr>
<th>Parameter</th>
<th>Nitrous Oxide</th>
<th>Hydrogen Peroxide, 90%</th>
</tr>
</thead>
<tbody>
<tr>
<td>State at ST&amp;P</td>
<td>Gas</td>
<td>Liquid</td>
</tr>
<tr>
<td>Vapour Pressure at 25°C</td>
<td>59.29 Bar</td>
<td>0.035 Bar (vapour over 60°C liquid mixture)</td>
</tr>
<tr>
<td>Boiling Point at 1 atm</td>
<td>-89.5°C</td>
<td>140.0°C</td>
</tr>
<tr>
<td>Freezing Point at 1 atm</td>
<td>-102.4°C</td>
<td>-10.0°C</td>
</tr>
<tr>
<td>Density (25°C liq)</td>
<td>641 kgm$^{-3}$</td>
<td>1,413 kgm$^{-3}$</td>
</tr>
<tr>
<td>Thermal Conductivity</td>
<td>0.0494 Wm$^{-1}$K$^{-1}$ (0°C and Gas Phase)</td>
<td>563.8 Wm$^{-1}$K$^{-1}$</td>
</tr>
<tr>
<td>Heat Capacity at 15°C and 1 atm</td>
<td>$c_p$ 839 Jkg$^{-1}$K$^{-1}$</td>
<td>$c_p$ 2,763.3 Jkg$^{-1}$K$^{-1}$</td>
</tr>
<tr>
<td>Oxygen content available after complete</td>
<td>36 g/100g</td>
<td>42.5g/100g</td>
</tr>
<tr>
<td>decomposition</td>
<td></td>
<td></td>
</tr>
</tbody>
</table>

Table 3.1: Physical properties of hydrogen peroxide and nitrous oxide

Nitrous oxide is supplied as a liquefied gas and as such there are standard safety precautions that have to be taken with a high-pressure gas - eye protection and gloves to handle the cylinder are all that are required. Nitrous oxide is non-toxic, but it is also a mild anaesthetic and an asphyxiant [MATHESON 66]. A well ventilated room is required, but crucially breathing apparatus are superfluous. Hydrogen peroxide on the other hand requires significantly higher levels of protection. The actual requirements are detailed in §2.5.2, but to summarise PVC clothing is required, but again no breathing apparatus. Nitrous oxide does have the added advantage of being self-pressurising due to its high vapour pressure. This raises the possibility of using the oxidiser to pressurise the fuel tank via a piston or diaphragm. A metal collapsible positive expulsion diaphragm tank such as the device produced by Atlantic Research Corporation [ARC 01b] would be suitable for this type of system. Using the oxidiser to pressurise the whole system would remove the need for an external pressurant tank and has the potential to reduce the overall volume of the system.
3.2.2 Potential Fuels

Limiting the selection of fuels to commonly available non-toxic liquids results in the possible candidates belonging to the hydrocarbon family. Table 3.2 shows the fuels that were considered. It should be noted that these fuels still require careful handling and are not without their faults. However in comparison to the hypergolic fuels such has hydrazine and its derivatives, the handling requirements are minimal [HAZARDS 69]. The safety precautions that need to be taken when working with these fuels are no more than the every day event of refilling a car with petrol or using pressurised gas on a camping stove.

<table>
<thead>
<tr>
<th></th>
<th>Kerosene (RP-1)</th>
<th>Ethane</th>
<th>Ethylene</th>
<th>Propane</th>
<th>Propylene</th>
</tr>
</thead>
<tbody>
<tr>
<td>Density</td>
<td>800-810</td>
<td>90.8</td>
<td>232.2 (Superheated Phase)</td>
<td>21.5</td>
<td>25.7</td>
</tr>
<tr>
<td>@ 25°C/kgm³</td>
<td>Liquid 346.3</td>
<td>Liquid 492</td>
<td>Liquid 505</td>
<td></td>
<td></td>
</tr>
<tr>
<td>BP 1 atm</td>
<td>177-274 °C</td>
<td>-88.6°C</td>
<td>-103.7°C</td>
<td>-42.1°C</td>
<td>-47.7°C</td>
</tr>
<tr>
<td>FP 1 atm</td>
<td>-40°C</td>
<td>-183.3°C</td>
<td>-169.5°C</td>
<td>-187.7°C</td>
<td>-185.3°C</td>
</tr>
<tr>
<td>Vapour Pressure</td>
<td>0.055Bar 10°C</td>
<td>43.5 Bar</td>
<td>80 Bar</td>
<td>9.9 Bar</td>
<td>11.9 Bar</td>
</tr>
<tr>
<td></td>
<td></td>
<td>27°C</td>
<td>38°C (SH)</td>
<td>27°C</td>
<td>27 °C</td>
</tr>
<tr>
<td></td>
<td></td>
<td></td>
<td>23.2 Bar</td>
<td>4.5 Bar</td>
<td>5.6 Bar</td>
</tr>
<tr>
<td></td>
<td></td>
<td></td>
<td>-1°C</td>
<td>-1°C</td>
<td>-1°C</td>
</tr>
</tbody>
</table>

Table 3.2: Physical properties of hydrocarbon fuels

3.2.3 Performance Discussion

A series of oxidisers and fuels were identified for their intrinsic green virtues and the propellant combinations were assessed for highest performance. As was discussed in the introduction, the propulsion volume envelope of small spacecraft is typically very limited, hence density Isp is of more importance to small spacecraft than true Isp. The Isp program by Curt Selph of AFRL [SELPH 94] was extensively used to calculate the true and density Isp of the various fuel and oxidiser combinations. The Isp code requires the propellant constituents, mixture ratio and expansion ratio inputs to produce a theoretical value. For the purpose of these calculations, the chamber pressure was set to 10 Bar with a 1000:1 pressure expansion ratio, thus producing a
pressure at the exit plane of 0.01 Bar. To validate the Isp code, a known engine was analysed. The space shuttle main engine (SSME) has a vacuum Isp of 453 sec with 200 Bar chamber pressure and 69:1 area ratio expansion nozzle [BOEING 02c]. The Isp software produced a result within 2% of this value, lending credibility to accurate theoretical results. Comparing the Isp of a smaller engine such as a 10N thruster may have been appropriate if data on the fuel film cooling flow could be found. However this was not possible and hence any comparison would have been corrupted leading to an erroneous result. The Isp code was then employed to calculate the performance of the combinations listed in table 3.3. The optimum O/F ratio was determined by an iterative process, which was then used to calculate the density Isp through multiplication of the average propellant specific gravity and the Isp. Furfuryl alcohol is also included in the table as an example of a high-density fuel other than kerosene. Furfuryl alcohol is not specifically a green propellant as it has some health and safety issues but not as severe as the traditional hypergolics.

<table>
<thead>
<tr>
<th>Fuel</th>
<th>Oxidiser</th>
<th>Isp /s in Vacuum Expanding to 0.01 Bar</th>
<th>Optimum O/F Ratio</th>
<th>Density Isp /s</th>
</tr>
</thead>
<tbody>
<tr>
<td>Kerosene (RP-1)</td>
<td>Nitrous Oxide</td>
<td>312</td>
<td>9:1</td>
<td>204</td>
</tr>
<tr>
<td>Ethane</td>
<td>Nitrous Oxide</td>
<td>314</td>
<td>10:1</td>
<td>187</td>
</tr>
<tr>
<td>Ethylene</td>
<td>Nitrous Oxide</td>
<td>319</td>
<td>9:1</td>
<td>173</td>
</tr>
<tr>
<td>Propane</td>
<td>Nitrous Oxide</td>
<td>302</td>
<td>10:1</td>
<td>188</td>
</tr>
<tr>
<td>Propylene</td>
<td>Nitrous Oxide</td>
<td>308</td>
<td>9:1</td>
<td>192</td>
</tr>
<tr>
<td>Kerosene (RP-1)</td>
<td>90% HTP</td>
<td>322</td>
<td>8:1</td>
<td>420</td>
</tr>
<tr>
<td>Furfuryl Alcohol</td>
<td>90% HTP</td>
<td>315</td>
<td>4:1</td>
<td>410</td>
</tr>
<tr>
<td>Ethane</td>
<td>90% HTP</td>
<td>323</td>
<td>10:1</td>
<td>357</td>
</tr>
<tr>
<td>Ethylene</td>
<td>90% HTP</td>
<td>330</td>
<td>8:1</td>
<td>298</td>
</tr>
<tr>
<td>Propane</td>
<td>90% HTP</td>
<td>312</td>
<td>8:1</td>
<td>365</td>
</tr>
<tr>
<td>Propylene</td>
<td>90% HTP</td>
<td>319</td>
<td>8:1</td>
<td>376</td>
</tr>
</tbody>
</table>

Table 3.3: Calculated performance of green Fuel/Oxidiser combinations
The table shows that both nitrous oxide and hydrogen peroxide produce similar Isp’s for all the fuels, with HTP marginally superior. However where hydrogen peroxide excels is in the density Isp performance, almost double the performance of nitrous oxide is achieveable.

The highest density Isp system from this analysis was achieved with kerosene and HTP (of 90% concentration) but this analysis ignores the potential benefit of using nitrous oxide as a pressurant. The hydrogen peroxide/kerosene combination would need and external pressurant source, which would lower the overall density Isp, this was investigated in §3.2.4 below.

### 3.2.4 Self Pressurising vs External Pressurant

In the case of nitrous oxide, the vapour pressure can pressurise both the fuel and itself, thus the density Isp calculated in table 3.3 is an accurate representation. With the hydrogen peroxide case, this requires an external pressurant, regulated down to tank pressure. It is common for the pressurant tanks to be charged to 200 Bar at the start of life [PAUL 02] and once the propellant tanks are empty, this reduces to just above chamber pressure. For the purpose of this discussion it is assumed that at end of life the system is at 10 Bar and at start of life 200 Bar. $v_{press}$ is the volume of the pressurant tanks and $v_{prop}$ is the volume of the propellant tanks.

At end of life: 

$$v_{prop} + v_{press} = \frac{n_{mol} gT}{10^6}$$  \hspace{1cm} (3.1)

At start of life 

$$v_{press} = \frac{n_{mol} gT}{2 \times 10^7}$$  \hspace{1cm} (3.2)

Thus it can be shown that 

$$v_{prop} = 19v_{press}$$  \hspace{1cm} (3.3)

<table>
<thead>
<tr>
<th>Fuel</th>
<th>Oxidiser</th>
<th>Isp /s in Vacuum Expanding to 0.01 Bar</th>
<th>Optimum O/F Ratio</th>
<th>Density Isp /s, corrected for pressurant source</th>
</tr>
</thead>
<tbody>
<tr>
<td>Kerosene (RP-1)</td>
<td>Nitrous Oxide</td>
<td>312</td>
<td>9:1</td>
<td>204</td>
</tr>
<tr>
<td>Propane</td>
<td>Nitrous Oxide</td>
<td>302</td>
<td>10:1</td>
<td>188</td>
</tr>
<tr>
<td>Propylene</td>
<td>Nitrous Oxide</td>
<td>308</td>
<td>9:1</td>
<td>192</td>
</tr>
<tr>
<td>Kerosene (RP-1)</td>
<td>90% HTP</td>
<td>322</td>
<td>8:1</td>
<td>399</td>
</tr>
</tbody>
</table>

Table 3.4: Performance of Fuel/Oxidisers after consideration of pressurant source
Hence the volume that the pressurant and the two propellant tanks will occupy is 1.052 times the volume of just the propellant tanks. This effectively reduces the overall density of the propellants to 95% of the previously calculated value. Hence the true density Isp can be compared with the self-pressurising nitrous oxide system, as shown above in Table 3.4.

This table demonstrates that from a performance basis the system using hydrogen peroxide/kerosene with an external pressurant is still far and away superior to any of the self-pressurising nitrous oxide systems. There are further advantages to be gained from this combination and this centres on ignition. While the two are not hypergolic, they can be thermally ignited [HARLOW 99]. Hydrogen peroxide can be passed through a silver catalyst producing 600°C steam and oxygen that can initiate auto-ignition with kerosene, alleviating the need for an ignition source. The same cannot be said for nitrous oxide systems that currently require a pyrotechnic or spark igniter [ZAKIROV 01]. A final comment on the nitrous oxide system focuses upon the ability to maintain vapour pressure. As the propellant tanks empty, gas has to fill the space above the liquid propellant. This involves a phase change through the liquefied gas boiling into vapour, thus requiring a significant quantity of energy. Specifically the latent heat of vaporisation is 200 kJkg⁻¹ [MATHESON 66]. Thus during the burn the nitrous oxide will cool and correspondingly its vapour pressure will fall. To compensate for this some form of heat recycling from the engine to warm the tanks would be required, raising the complexity considerably. Also the vapour pressure varies considerably with temperature. Solar heating will correspondingly shift the O/F ratio hence precise temperature control may be required. In the case of HTP, which uses a gaseous external pressurant, there is not a phase change so the issue would not be as severe. Mitigating this is the use of a pressure regulator, thus a constant flight pressure could be held for the entire mission up to the very last burn of the system.

3.2.5 Propellant Conclusions

The conclusion from the theoretical analysis of the propellants was that hydrogen peroxide and kerosene provides a green, high performance, easily ignitable oxidiser and fuel combination for a bi-propellant rocket engine. The use of nitrous oxide raises concerns of ignitability, as a spark or pyrotechnic igniter would have to be employed which could be avoided through the use of hydrogen peroxide. However the biggest criticism comes from the low density Isp. Thus HTP was chosen in preference to nitrous oxide, and the kerosene fuel was chosen to maximise this parameter.
3.3 Cooling Mechanism Choice

In order to select the cooling system, the heat fluxes in the engine need to be known. The convective heat and radiative heat transfer equations are shown in equations 3.4 and 3.5, where \( T_c \) and \( T_{wh} \) represents the two temperatures over which forced convection is taking place. Equation 3.6 defines the equilibrium temperature which occurs naturally, due to radiative and convective effects [SUTTON 92].

\[
\dot{Q}_{\text{conv}} = A_{sur}h_c(T_c - T_{wh}) \quad (3.4)
\]

\[
\dot{Q}_{\text{rad}} = \sigma A_{sur} \varepsilon_c \alpha_{wh} (T_c^4 - T_{wh}^4) \quad (3.5)
\]

\[
\alpha_{v}T_{wc}^4 = \alpha_{v} \varepsilon_c \alpha_{wh} (T_c^4 - T_{wh}^4) + h_c(T_c - T_{wh}) \quad (3.6)
\]

The convective heat transfer coefficient \( h_c \) can be expressed for the combustion gas as in equation 2.37. The two important parameters for the convective heat flux of the combustion gases are shown in table 3.5 [LIQUID PROP 57]. The actual calculation was performed by a computer model, which is detailed in §7.2.

<table>
<thead>
<tr>
<th>Parameter</th>
<th>Combustion Products</th>
</tr>
</thead>
<tbody>
<tr>
<td>Thermal conductivity, k</td>
<td>0.172 Jm⁻¹K⁻¹s⁻¹</td>
</tr>
<tr>
<td>Viscosity, ( \mu )</td>
<td>5.70 \times 10⁻⁵ kgm⁻¹s⁻¹</td>
</tr>
</tbody>
</table>

Table 3.5: Properties of HTP/kerosene combustion products

3.3.1 Cooling Techniques

Comparison of the cooling systems was achieved through a model of a standard combustion chamber with a diameter of 18mm. The \( L^* \) of the chamber, defined in equation 3.7, was set to 1.27m, as recommended for HTP/kerosene systems [LIQUID PROP 57].

![Test chamber dimensions](image)
Chapter 3: Theoretical Thruster Design

$$L^* = \frac{\text{ChamberVolume}}{\text{ThroatArea}} \quad (3.7)$$

The $L^*$ and diameter gave a mid aspect ratio chamber, common to many past designs. The convergent section was set at 30 degrees with a 15-degree divergent section, both half angles, with a 6.25:1 expansion cone, shown in figure 3.1. The pertinent chamber parameters are given in table 3.6. Using this configuration the heat flux into the chamber and nozzle sections were calculated. This was based upon a propellant flow rate of 13.0 gs$^{-1}$, which corresponds to a vacuum thrust of approximately 40N. The model was split into 1mm long sections and a computer program calculated the flame temperature at each location and corresponding heat flux with the cooling radiation heat flux subtracted, a full description of the model is given in §7.2. The wall temperature of the engine was taken to be 1,250K, the maximum that standard metals can be taken to. With the heat flux data available, it was possible to examine the various forms of cooling that must extract the considerable quantity of energy from the chamber and nozzle sections.

<table>
<thead>
<tr>
<th>Symbol</th>
<th>Parameter</th>
<th>Value</th>
</tr>
</thead>
<tbody>
<tr>
<td>L$_{ch}$</td>
<td>Chamber Length</td>
<td>5$\times$10$^{-2}$ m</td>
</tr>
<tr>
<td>d$_{ch}$</td>
<td>Chamber Diameter</td>
<td>1.8$\times$10$^{-2}$ m</td>
</tr>
<tr>
<td>d$_{th}$</td>
<td>Throat Diameter</td>
<td>4$ \times$10$^{-3}$ m</td>
</tr>
<tr>
<td>$\dot{Q}_{ch}$</td>
<td>Power into the Chamber</td>
<td>4,770 W</td>
</tr>
<tr>
<td>$\dot{Q}_{noz}$</td>
<td>Power into the Nozzle</td>
<td>1,978 W</td>
</tr>
</tbody>
</table>

Table 3.6: Test chamber parameters

3.3.1.1 Regenerative Cooling

The preferred choice for cooling a bi-propellant rocket engine is to use a regenerative cycle, thus ultimate performance is achieved. Concentrating on the parallel chamber initially, the total power into the walls from the heat of combustion was calculated to be 4,770W. The specific heat capacity of 90% hydrogen peroxide is 2.776 Jg$^{-1}$K$^{-1}$ [BECCO 54] and the HTP cannot be allowed to experience a temperature rise of more than 70°C, starting from 20°C, otherwise rapid decomposition is likely to occur [BECCO 54]. Hence with the maximum permissible temperature rise, 194.3 Jg$^{-1}$ of heat can be absorbed. The flow rate of the hydrogen peroxide in the 40N engine will be 10.85 gs$^{-1}$ at stochiometric and thus the available power that can be removed is 2,108 W. Hence by raising all the flow of HTP by 70°C, the power that this requires is insufficient to cool the chamber alone. This is before the nozzle has even been considered, thus regenerative cooling
on such a small engine and this chamber design is not possible using HTP. Kerosene cannot be used to assist the HTP for cooling due to coking of the coolant passages [CLARK 72]. This does not mean, however, that certain areas of the engine cannot be cooled in the regenerative fashion. The nozzle is an obvious area where regenerative cooling can greatly assist another cooling system and therefore be more efficient overall.

3.3.1.2 Fuel Film Cooling

A system employed in many hypergolic engines is the fuel film cooling option. Here fuel is sprayed onto the chamber wall, which initially forms a film of liquid that is rapidly heated by the hot combustion gases. Once the fuel has been vaporised it forms a boundary layer that rises in temperature as it is convectively and radiatively heated. The coolant fluid absorbs most of the heat consequentially reducing the transfer of heat to the wall. The total power that needs to be extracted from the engine is $P_{ch} + P_{loss} = 6,748 \text{W}$ which must be removed by the vaporization and heating of the excess fuel. The composition of kerosene is a mixture of hydrocarbons ranging from C9 to C16, and hence for the purpose of the analysis kerosene is assumed to be composed purely of decane (C10) [GALLANT 70]. The estimated properties of kerosene are shown in table 3.7. The issue of hydrocarbon cracking has been ignored in this first order estimate.

<table>
<thead>
<tr>
<th>Heat of Vaporisation (293K liquid to 550K gas) / Jg$^{-1}$</th>
<th>Boiling Point at 15 Bar /K</th>
<th>Gaseous Heat Capacity / Jg$^{-1}$K$^{-1}$</th>
</tr>
</thead>
<tbody>
<tr>
<td>377</td>
<td>550</td>
<td>2.4 at 700K</td>
</tr>
</tbody>
</table>

Table 3.7: Estimated properties of kerosene

The heat from the engine will take the fuel from its initial temperature of 293K up to its vaporisation point and then up to the preset maximum permissible temperature of 1,250K, limited by softening and melting of the chamber walls. Thus the total energy that this would require is 1,680 Jg$^{-1}$. Hence from the power calculated earlier, the mass flow rate of the fuel would have to be 3.1 gs$^{-1}$. In theory fuel film cooling is thermodynamically possible but the coolant flow rate is twice the main fuel flow rate with a flow of 1.45 gs$^{-1}$. Here the low-density propellant is being excessively used thus lowering the overall density Isp.
3.3.1.3 Ablative Cooling

Ablative cooling is a possibility on short duration firings, and situations where the level of thrust is allowed to vary. However, for satellite operations, a repeatable thrust characteristic and long burn times are a necessity for accurate operations. Thus the concept of an ablative nozzle was discounted in preference over a cooling system that has repeatability and longevity.

3.3.1.4 Liquid Oxidiser Film Cooling

An alternative to fuel film cooling is to use the oxidiser instead of the fuel. In such a system the liquid oxidiser is sprayed onto the wall, which performs the same function as the fuel did in fuel film cooling. This method is unusual because of the corrosion issue with the hot oxidiser on the chamber wall, however at least one engine utilises this technique, the Russian LTRE400N engine [NIIMASH 00]. The primary advantage of this method is the use of the high-density propellant for cooling. This increases the overall density Isp thus creating a more compact system. However when injecting liquid oxidiser into the chamber along with the liquid fuel, there is a substantial chance that an explosion may occur if the two propellants pool inside the chamber. During the British work on hydrogen peroxide in the 50’s and 60’s, liquid injection was abandoned in favour of decomposed HTP because of the safety implications [HARLOW 99]. Hence it was decided that liquid oxidiser injection would not be pursued.

3.3.1.5 Oxidiser Gas Film Cooling

The natural extension to the potentially dangerous liquid oxidiser film cooling is to inject decomposed hydrogen peroxide as the coolant. A catalyst pack would be used as a gas generator to convert the high-density liquid into steam and oxygen. Typically, the exhaust temperature of a catalyst pack will be 800K, with an efficiency factor $\eta$ of 90% and using 90% concentration HTP. Thus the available temperature rise for removing heat, before the wall temperature limit is reached, is 450K. This time however there will be no phase change, so all the energy absorption comes from raising the oxygen and steam mixture up to 1,250K, which again is the maximum permissible temperature, 100K less than the melting point of copper. The heat capacity of the steam and oxygen mixture at 800K is 1.90 Jg$^{-1}$ and does not vary appreciably between 500 and 1,250K. Hence the available heat that can be removed is 855 Jg$^{-1}$ and consequentially the required flow rate of HTP is 7.6 gs$^{-1}$. It may be possible to drop the flow rate further due to the insulation effect of the boundary layer. While the layer of coolant next to the wall is at 1,250K, further away the coolant can safely exceed this limit, thus improving the Isp.
3.3.2 Performance Estimation

It has been established, theoretically, that fuel film cooling and oxidiser gas film cooling are two ways in which a small bi-propellant engine can be cooled. In both systems the main flow of the combustion products constitutes the highest Isp and calculations from the Isp software show that 322 seconds is possible with a stochiometric mixture and a 300:1 expansion cone operating in a vacuum. The fuel or oxidiser film flow will leave the nozzle with a lower velocity due to the lower gas temperature of only 1,250K and the higher molecular weights. To find the equivalent Isp’s of the film cooling gases, the Isp program was run with only the coolant in the chamber and the temperature forced to 1,250K. The Isp generated from the program was taken to be the Isp that would be produced by the film coolant leaving the combustion chamber in a propulsive manner. The results are summarised in table 3.8.

<table>
<thead>
<tr>
<th>Isp through a 300:1 expansion ratio nozzle /s</th>
<th>Stochiometric mix of HTP and Kerosene</th>
<th>Kerosene at 1,250K</th>
<th>Decomposed HTP at 1,250K</th>
</tr>
</thead>
<tbody>
<tr>
<td></td>
<td>322</td>
<td>180</td>
<td>177</td>
</tr>
</tbody>
</table>

Table 3.8: Isp of components of the exhaust stream

The data can then be used to estimate the total Isp for a film cooled system and also its density Isp. This was achieved by assuming the main flow has the full stochiometric Isp, while the coolant flow will have a respectively lower Isp. The overall estimated Isp is then defined in equation 3.8, the propellant specific gravity is defined by equation 3.9 and the density Isp is defined in equation 3.10. The density of HTP is taken as 1.40 g/cm$^3$ and the density of kerosene is 0.80 g/cm$^3$. The results of this analysis are shown in table 3.9 for three configurations; purely fuel film cooling, purely oxidiser gas film cooling and the combination of a regeneratively cooled nozzle and oxidiser gas film cooled chamber.

\[
Isp_{\text{Overall}} = \frac{\dot{m}_{\text{main}} Isp_{\text{main}} + \dot{m}_{\text{cool}} Isp_{\text{cool}}}{\dot{m}_{\text{main}} + \dot{m}_{\text{cool}}} \quad (3.8)
\]

\[
\rho_{sp} = \frac{\rho_{sp} \dot{m}_{sp} + \rho_{fu} \dot{m}_{fu} + \rho_{cool} \dot{m}_{cool}}{\dot{m}_{sp} + \dot{m}_{fu} + \dot{m}_{cool}} \quad (3.9)
\]

\[
\rho Isp = \rho_{sp} \times Isp_{\text{Overall}} \quad (3.10)
\]
Chapter 3: Theoretical Thruster Design

<table>
<thead>
<tr>
<th>Engine Parameters</th>
<th>Fuel film cooled nozzle &amp; chamber</th>
<th>Ox Gas film cooled nozzle &amp; chamber</th>
<th>Ox gas film cooled chamber &amp; regen cooled nozzle</th>
</tr>
</thead>
<tbody>
<tr>
<td>Main Oxidiser Flow Rate / gs⁻¹</td>
<td>10.85</td>
<td>10.85</td>
<td>10.85</td>
</tr>
<tr>
<td>Main Fuel Flow Rate / gs⁻¹</td>
<td>1.45</td>
<td>1.45</td>
<td>1.45</td>
</tr>
<tr>
<td>Coolant Flow Rate / gs⁻¹</td>
<td>3.1</td>
<td>7.6</td>
<td>5.6</td>
</tr>
<tr>
<td>Overall Isp / s</td>
<td>293</td>
<td>267</td>
<td>277</td>
</tr>
<tr>
<td>Overall Propellant specific gravity</td>
<td>1.22</td>
<td>1.36</td>
<td>1.35</td>
</tr>
<tr>
<td>Density Isp / s</td>
<td>358</td>
<td>362</td>
<td>374</td>
</tr>
</tbody>
</table>

Table 3.9: Overall Isp of fuel film, oxidiser film and oxidiser film/regen cooling

3.3.3 Cooling Mechanism Conclusions

In both the oxidiser gas film cooled cases, the density Isp’s are marginally greater than the fuel film cooled case. These calculated figures are theoretical and in excess of what you would expect from a real system. Typically a 92% efficiency of the overall Isp can be expected [WU 99] from an actual system. This reduces the Isp and density Isp of the oxidiser gas cooled/regeneratively cooled case to 255 and 344 seconds respectively. The gas film cooled system does suffer from a reduced overall Isp, but volume is of more importance in small spacecraft than mass. However the main advantages to be gained centre on the engineering and testing of the design. In order to get an even distribution of coolant around the engine, as in a fuel film cooled system, many small holes are required to spray fuel onto the walls. Thus the coolant flow through each one will be small and this dictates a small orifice size, typically less than 0.1mm diameter, to maintain a decoupling pressure differential with the feed system. This small orifice size will be extremely prone to blockages which may go unnoticed resulting in an erroneous result and burn out of the chamber. When using gas film cooling, the dimensions of the orifices can be much larger and hence blockages are not an issue as hole diameters of 0.8mm or greater are required. This produces a simpler engineering solution and hence reduces the overall cost of the system, thus oxidiser gas film cooling was selected for the practical designs.
3.4 Injector Design

The design of the injector is the single biggest factor affecting the burn efficiency and smoothness of combustion. In the hydrogen peroxide/kerosene system, the oxidiser can be introduced into the chamber through the catalyst pack, which is effectively the oxidiser injector. The fuel system however requires a dedicated injection system and several methods are available. Like-like impingement of two kerosene jets produces a fine mist of fuel, which will then burn with decomposition products of the HTP. It has been reported [WU 99] that the C* efficiency of a like-like injection HTP/kerosene system can be as high as 95% of theoretical. This high efficiency does come at a price in the form of high precision and therefore costly engineering required to produce the design. Any deviation in the paths of the jets will cause a reduced atomisation process and result in a reduced overall efficiency. An alternative method is to use a swirl type injector, where the fuel exits as a conical sheet that rapidly disintegrates into droplets due to aerodynamic and viscous forces. This was the system employed on the hydrogen peroxide/kerosene research engines that led to the Gamma series of highly efficient launch vehicle propulsion units [LIQUID PROP 57b]. A final design whose chief advantage is simplicity is the direct jet injection. Rather than produce a spray pattern from the kerosene before it comes into contact with the oxidiser, this system uses the oxidiser to break up the kerosene flow. Whenever there is a large shear velocity between the flow of the fuel as a liquid jet and the fast moving gaseous oxidiser there will be a significant force introduced through the viscous aerodynamics. This results in atomisation of the liquid jet into droplets, which are then burnt in the oxidiser flow [FULLER 00]. It has been reported that a C* efficiency of around 92% is achievable using this injection mechanism [WU 99], while being lower than like-like system, efficiency is traded off with simplicity of construction. All that is required is the kerosene be injected through small orifice holes at an angle into the stream which maximises the shear velocity.

The diameter of the injector holes must be such that a pressure differential is maintained between the feed system and the chamber pressure. Typically for liquid systems the pressure drop should be around 10% of the chamber pressure [HUMBLE 95]. This is required to decouple the chamber oscillations from feed system, and thus prevent instability in the engine. The situation can be imagined where a pressure spike in the chamber raises the pressure above the tank pressure thus preventing propellant to flow into the chamber. The chamber pressure will therefore fall, allowing propellants to flow again, and an oscillatory process is initiated. The equation governing the required orifice area of each injector is given in equation 3.11. It was decided that a 4-injector configuration would be the maximum number of injectors that could be accommodated in the engine. The greater the number of injectors, the better the mixing will be due to a more even
propellant distribution, however there is a practical limit on the number that can be physically placed inside the combustion chamber.

\[
A_{\text{inj}} = \frac{n_{\text{inj}}}{\left(\frac{K_{\text{orif}}}{2n_{\text{inj}}A_{\text{inj}}}\right)} \sqrt{\frac{K_{\text{orif}}}{2\rho_{fu} \Delta P}}
\]  

(3.11)

Where

\(n_{\text{inj}}\) = Number of injectors

\(\dot{m}_{fu}\) = Total mass flow (kgs\(^{-1}\))

\(A_{\text{inj}}\) = Injector cross section (m\(^2\))

\(K_{\text{orif}}\) = Head loss coefficient (1.2 - 1.7 for radius to non radius)

\(\rho_{fu}\) = Density of kerosene (kgm\(^3\))

\(\Delta P\) = Pressure drop across the injector (Nm\(^2\))

The chamber pressure was estimated to be 15 Bar, thus the injectors required a 1.5 Bar \(\Delta P\) in order to decouple the pressure oscillations. The actual O/F sensitivity to the propellant tank pressures is discussed in §7.5.

Using a head loss coefficient of 1.2 for injector, assuming a non-radiused design, and a kerosene flow rate of 1.45 gs\(^{-1}\), the area of the injector’s orifice was calculated to be \(2.865 \times 10^{-8} \text{ m}^2\), which results in a diameter of 0.18 mm. This was extremely convenient as 0.2mm diameter brass injectors are commercially available for blowtorches, thus reducing the manufacturing cost. The injectors must be angled into the flow of the decomposed HTP to maximise the atomisation of the kerosene. The optimum angle for this is clearly at 90° to the decomposed HTP flow. However there is a danger that the kerosene will not be fully atomised by the time it reaches the far wall of the chamber. Angling the spray down stream provides greater time for atomisation to occur thus

![Diagram of injector spray trajectories as a function of injection angle](image-url)

Figure 3.2 a,b,c : Injector spray trajectories as a function of injection angle
potentially producing a superior combustor. However, if the jets are angled too much downstream there will be a reduced shear velocity and the onset of spray atomisation will be inhibited. This reduces the effective length of the chamber and the design becomes unnecessarily long to compensate. A long chamber is more difficult to cool and hence produces a less efficient design. Figures 3.2 a, b and c show the three discussed cases.

3.4.1 Cold Flow Analysis Apparatus

The effect of the injection angle was examined empirically using a cold flow model to simulate the flows inside the chamber. The cold flow model utilised a Perspex chamber that allowed direct observation of the flows, which could easily be photographed. Simulating a circular cross section chamber is possible in Perspex but the optical effects make the image distorted and data analysis hard. Hence a 2D chamber was manufactured. The model used water as a surrogate for the kerosene and compressed nitrogen used in place of the decomposed HTP. The feed systems for the nitrogen and water used a regulated supply system with Coriolis flow meters to record the flow rate. The set up was similar to the actual engine feed systems, described in chapter 4. The experimental chamber is shown below in figure 3.3. The initial apparatus used an injection angle of 55° to the oxidiser flow for both injectors, the maximum angle a practical combustor could be made and this is detailed in §4.2.1. One injector was then modified to 25° for simulation of a much shallower injection angle.

To simulate the mass flow of HTP of 10.85gs\textsuperscript{-1} proved unfeasible, as excessively high nitrogen pressures would be required. The maximum flow rate with 20 Bar nitrogen pressure was only 8gs\textsuperscript{-1}. The water flow rate was also difficult to set due to the low flow rates involved, however a total water flow of 1gs\textsuperscript{-1} was possible. As there were only two injectors in the cold flow model and the actual engine had four injectors, this gave a model “O/F” ratio of 4:1. While this was not
ideall, it was assumed that the “O/F” was close enough to provide useful data. The density of decomposed HTP at 600°C and 15 Bar (a typical chamber pressure) is $4.92 \times 10^{-3} \text{ kgm}^{-3}$. With a feed supply of 20 Bar the chamber pressure in the Perspex chamber was 4 Bar absolute. At this pressure and an ambient temperature, the density of nitrogen is $4.48 \times 10^{-3} \text{ kgm}^{-3}$. Thus the momentum flux of the decomposed HTP in the real engine and the nitrogen in the cold flow model are nearly identical. The apparatus was set up with a camera facing the flat front of the chamber and the nitrogen was turned on. The water was then injected and a series of photographs taken of the chamber and spray patterns. It was important to take the photographs quickly as the chamber rapidly became obscured with fine water droplets forming on the walls. Once the cold flow model had been dried out, the test was repeated, this time without any nitrogen flow. This was done to witness the comparison between the two spray patterns. The apparatus was then modified with one of the injectors angle altered from the original angle of 55° to 25°. The same procedure was then applied to photograph the new configuration.

3.4.2 Cold Flow Analysis Results

With the injection angle set at 55° the following spray pattern was observed, shown in figure 3.4. The evidence indicates that atomisation is occurring as can be seen by the feathery trails of the water jets. However the atomised water is not entrained in the flow of nitrogen as was intuitively expected. Instead the spray impinges upon the walls of the chamber forming a build up of water.

![Figure 3.4: Cold flow test for 55° injector angle](image)
Chapter 3: Theoretical Thruster Design

Atomisation Point

Figure 3.5: Cold flow tests for 25° injector angle

This does have implications regarding the combustion efficiency as fuel collecting on the walls would not burn efficiently, leading to a reduced Isp. The 25° injection angle was therefore chosen to reduce impingement upon the chamber walls. It was anticipated that less atomisation might occur due to the lower shear velocity differential in the shallow angle configuration. Figure 3.5 shows the results of the shallow angle test. The results of this test are harder to observe due to water droplets obscuring the view. However impingement of the water on the chamber wall still occurred, this time further down the chamber. The highlighted trajectory in figure 3.5 also shows the atomisation point where the jet decayed into a spray. In comparison to figure 3.4, this point is considerably further from the injector head. The implication of this is that the chamber would have to be made longer in order to accommodate the delayed atomisation distance. In addition the degree of atomisation in this configuration appeared less pronounced than the 55° case.

Thus from these tests the 55° case was adopted for the actual engine configuration due to apparently superior atomisation and reduced chamber length. It was felt that as long as adequate performance was produced in hot firings the injector design was acceptable. Optimisation of the injector was not the primary goal of this research.

3.5 Chamber Geometry Optimisation

The chamber is defined as the region of the rocket engine, downstream of the injectors and up to and including the start of the expansion cone. In this volume, rapid mixing and combustion takes
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place continuously with the geometry affecting the efficiency. The first concern of the chamber is to provide enough volume for the initial ignition to establish combustion conditions. Secondly there needs to be adequate ‘stay time’ to produce nearly complete combustion. The metric that pertains to this characteristic is the $L^*$ (pronounced L star) of the engine. $L^*$ is defined in equation 3.7. An HTP/kerosene system should have an $L^*$ of between 40 to 51 inches (1.01 to 1.30m) to achieve good combustion [LIQUID PROP 57b]. This parameter says nothing about the actual geometry of the chamber so it is perfectly feasible to have a short squat chamber and a long thin chamber, yet still have similar combustion efficiencies providing the $L^*$ is the same. Where the two chambers will vary however is in the total heat flux into the engine’s walls. With a large surface area, the heat flux will be appreciably higher than a small surface area combustion chamber due to the effect of the boundary layer heating. The ideal chamber geometry will be sphere due to the surface to volume ratio being minimised. However such a design is difficult to manufacture and a cylindrical chamber section would have to be used instead. There exist two scenarios where the chamber geometry can be optimised.

a) Oxidiser film cooled chamber section and regeneratively cooled nozzle

b) Oxidiser film cooled chamber and film/radiatively cooled nozzle

In case ‘b’ minimisation of chamber surface area solves the posed problem. This results in a cylindrical chamber that approximates in its aspect ratio to a spherical design. In case ‘a’ the analysis is more complicated as the two methods of removing heat are coupled together. In order to perform this analysis, a theoretical chamber geometry was defined that allowed the diameter of the cylinder to be varied. The convergent section was set at 30° half angle in accordance with many rocket engine designs [HUMBLE 95]. This configuration is shown in figure 3.6. The throat diameter is set to 4mm, in order to give the correct chamber pressure at the required thrust. Thus the length of the convergent section is given by equation 3.12, the total chamber volume in 3.13 and the $L^*$ in equation 3.14.

\[ L_{Con} = \frac{d_{Ch} - 4 \times 10^{-3}}{2 \tan 30} \]  

\[ V_{Ch+Con} = \frac{1}{3} \pi L_{Con} \left( 2 \times 10^{-6} + 2 \times 10^{-3} \times d_{Ch} + d_{Ch}^2 \right) + \pi \left( \frac{d_{Ch}}{2} \right)^2 L_{Con} \]  

\[ L^* = 1.3m = \frac{V_{Ch+Con}}{\pi \left( 2 \times 10^{-3} \right)^2} \]
Hence $L_{Ch}$ and $d_{Ch}$ can vary over a range of values and produce in theory the same $C^*$. However the power that needs to be extracted from the chamber and nozzle will vary quite dramatically with these parameters. The cooling mechanism for the nozzle, which consists of the convergent and divergent section, is achieved through regenerative cooling. The parallel section of the chamber however is cooled using the decomposed HTP gas. The flow paths of the regenerative and oxidiser gas cooled systems are shown in figure 3.7. The total flow of HTP from the propellant tank is designated $\dot{m}_{cool}$ which is fed through the regeneratively cooled nozzle which then splits into two flow paths. Part of the flow is fed into the chamber where it combusts with kerosene after passing through the main catalyst pack. The remainder of the flow is passed

Figure 3.6: Optimised chamber

Figure 3.7: Flow path of the oxidiser in the regenerative / gas cooled system
through a second catalyst pack and the products are used to cool the main chamber region. Minimisation of the heat into the chamber walls can be achieved by constructing a wide and short chamber. However if the chamber is made too wide, the heat flow into the nozzle section will increase too much due to the larger surface area. From §3.3.1.1 the maximum temperature rise of the HTP is 70°C, when starting from a typical 20°C ambient temperature. The heat capacity of 2.763 Jg⁻¹K⁻¹ which gives a maximum power extraction in the nozzle of 193.41 Jg⁻¹. The heat capacity of the decomposed HTP in the chamber section is approximately 1.9 Jg⁻¹K⁻¹. In this section there is no concern about the decomposed HTP getting too hot, instead it is the temperature limit of the chamber material that is the governing factor, and a nominal limit of 1,250K is chosen, this being 100K less than the melting point of copper. From the previous analysis, the available heat that can be removed is 855.0 Jg⁻¹. Thus the minimum flow rates in the nozzle and the chamber coolant are defined by equations 3.15 and 3.16.

\[ m_{cool} = \frac{\dot{Q}_{Net}}{193.4} \quad (3.15) \]
\[ m_{film} = \frac{\dot{Q}_{Ch}}{855.0} \quad (3.16) \]

Hence if \( m_{film} \) is greater than \( m_{cool} - \dot{m}_{ax} \) then \( m_{cool} \) will have to be increased. However if the reverse is true, then the calculated \( m_{cool} \) is acceptable. The performance of the engine is calculated by assuming the \( m_{main} \) component (\( \dot{m}_{fu} + \dot{m}_{ax} \)) produces an Isp of 296 seconds and the \( m_{film} \) produces 177 seconds, due to it effectively being a monopropellant system. These Isp’s are calculated from the theoretical values and an efficiency factor of 0.92 applied, the reasoning behind this is described in §3.3.3. The weighted average of these two Isp’s produces the overall effective Isp.

\[ Isp = \frac{\dot{m}_{main} \times 296 + \dot{m}_{film} \times 177}{\dot{m}_{main} + \dot{m}_{film}} \quad (3.17) \]

The computer simulation described in §7.2 was employed to calculate the heat fluxes for the different geometry of chambers considered. The results of this analysis are shown in figure 3.8 with the highest Isp performance of the engine of approximately 270 seconds with a chamber diameter of 26mm. It has a clearly defined maximum demonstrating the importance of selecting the correct chamber geometry.
The actual design of the regeneratively cooled nozzle is also of interest, as typically such a system would involve complex engineering of coiled pipes braised together to form a nozzle. On large engines such as missile or launch vehicle this is a valid design. However on a nozzle with only a 4mm diameter throat the process becomes expensive through the use of advanced fabrication techniques. As the project strives for simplification of the engineering, the concept of a regeneratively-cooled nozzle is not necessarily in agreement with this philosophy. However it has been established through a numerical model detailed in §7.3 that a copper nozzle for an 18mm diameter chamber can be cooled using two parallel cooling channels drilled either side of the throat. This only forms a single cooling loop but due to the high thermal conductivity of copper, enables the whole of the nozzle to be cooled. However there is an upper limit to the diameter of the regenerative nozzle before inadequately cooled regions develop. The optimum chamber design is one with a diameter of 26mm and a length of 25mm, which gives a conservative Isp estimate of 269 seconds. However this does not guarantee that such a system is actually physically possible to run. The problem can be visualised by considering two nozzles for two different chamber diameters as shown in figure 3.9. In nozzle ‘a’ the convergent region is relatively close to the coolant channels, thus effectively cooling it. However in nozzle ‘b’ there is a considerable area of the convergent section that is not near a coolant channel, thus limited cooling will occur. Copper was deliberately chosen for its high thermal conductivity to limit this potential problem. However to give an indication as to whether this would be a problem the

Figure 3.8: Isp vs chamber diameter
regenerative nozzle simulation was adapted to run with a chamber diameter of 26mm. Figure 3.10 shows the steady state results after 30 seconds of burn time. The main combustion flow exceeds the temperature scale and hence is false colour. The cooling channels are visible on the plot as the dark circles located either side of the throat. The convergent section is hottest part of the nozzle, as was expected, with a peak temperature of 950K. This is within the tolerable limit of the copper and so should not pose a problem. Thus according to the simulation data a chamber of 26mm diameter and a regeneratively cooled nozzle is perfectly possible. Unfortunately using such a large mass of copper as the nozzle is mass wasteful, and hence the simulation was adapted to simulate a thinner walled nozzle section, this time maintaining a 5 mm wall thickness, which produced the thermal map shown in figure 3.11. The map demonstrates that the convergent section of the nozzle would reach an upper temperature of 1,200K, thus very close to the melting point.
Extending the wall thickness to 8mm brings the maximum temperature within the acceptable range and thus specifies the minimum wall thickness. Hence the conclusion of the chamber optimisation is that a chamber of 26mm diameter and length 25 mm, with a copper regeneratively cooled nozzle, of wall thickness 8mm, would be adequately cooled and should produce a realistic Isp of 269 seconds with a 300:1 expansion ratio nozzle.

### 3.6 Propellant Concentration Optimisation

When hydrogen peroxide is used for propulsive purposes the concentration is desired to be between 70% and 100%. The feedstock used in this research was of 90% concentration and was selected initially to be of the highest concentration available. The maximum concentration is usually chosen as this maximises the Isp available from the chemical reaction. Also with increasing concentration the density of the HTP increases and hence the overall density Isp will increase substantially. As an illustration of this, the following data in table 3.10 demonstrates performance against concentration for a hydrogen peroxide/kerosene system. However in the combined regenerative and film cooling system, raising the concentration of the HTP has a detrimental effect. Firstly the combustion flame temperature increases with concentration thus making the system harder to cool.
Chapter 3: Theoretical Thruster Design

<table>
<thead>
<tr>
<th>HTP Concentration</th>
<th>Optimum O/F</th>
<th>Theoretical Maximum Isp /s</th>
<th>Overall Propellant Specific gravity</th>
<th>Theoretical Maximum density Isp /s</th>
<th>Stochiometric flame temperature /K</th>
</tr>
</thead>
<tbody>
<tr>
<td>70</td>
<td>10</td>
<td>279</td>
<td>1.24</td>
<td>345</td>
<td>2,226</td>
</tr>
<tr>
<td>75</td>
<td>9</td>
<td>291</td>
<td>1.27</td>
<td>370</td>
<td>2,350</td>
</tr>
<tr>
<td>80</td>
<td>9</td>
<td>302</td>
<td>1.30</td>
<td>393</td>
<td>2,500</td>
</tr>
<tr>
<td>85</td>
<td>8</td>
<td>311</td>
<td>1.32</td>
<td>411</td>
<td>2,625</td>
</tr>
<tr>
<td>90</td>
<td>8</td>
<td>321</td>
<td>1.34</td>
<td>430</td>
<td>2,713</td>
</tr>
<tr>
<td>95</td>
<td>7</td>
<td>327</td>
<td>1.36</td>
<td>445</td>
<td>2,820</td>
</tr>
<tr>
<td>100</td>
<td>7</td>
<td>329</td>
<td>1.38</td>
<td>454</td>
<td>2,894</td>
</tr>
</tbody>
</table>

Table 3.10: Performance of hydrogen peroxide/kerosene as a function of concentration

Secondly the exhaust temperature of gas generator used to cool the parallel chamber section will increase accordingly with concentration, producing a lower ΔT over which heat can be absorbed. Thus at higher concentrations the coolant flow rate would have to be increased to compensate for a lower heat capacity. Additionally the specific heat capacity of the hydrogen peroxide decreases with increasing concentration, thus compounding the issue of regeneratively cooling the nozzle.

As an example 70% HTP has a specific heat capacity of 3.019 Jg⁻¹K⁻¹ while 100% has 2.633 Jg⁻¹K⁻¹. Hence at the higher concentrations, the flow rate would have to be increased through the nozzle in order to prevent the maximum temperature of the HTP from being exceeded. Table 3.11, below, summarises the parameters of the HTP between 70 and 100% concentration.

<table>
<thead>
<tr>
<th>HTP concentration</th>
<th>Liquid $c_p$ /Jg⁻¹K⁻¹</th>
<th>Monopropellant adiabatic decomposition temperature /K</th>
<th>Isp of monoprop gas at 1,250K /s</th>
<th>Average $c_p$ of monoprop gas, 800-1250K /Jg⁻¹K⁻¹</th>
</tr>
</thead>
<tbody>
<tr>
<td>70</td>
<td>3.019</td>
<td>401</td>
<td>182</td>
<td>2.03</td>
</tr>
<tr>
<td>75</td>
<td>2.954</td>
<td>499</td>
<td>181</td>
<td>2.00</td>
</tr>
<tr>
<td>80</td>
<td>-2.885</td>
<td>600</td>
<td>179</td>
<td>1.96</td>
</tr>
<tr>
<td>85</td>
<td>2.822</td>
<td>701</td>
<td>178</td>
<td>1.93</td>
</tr>
<tr>
<td>90</td>
<td>2.759</td>
<td>800</td>
<td>177</td>
<td>1.90</td>
</tr>
<tr>
<td>95</td>
<td>2.696</td>
<td>998</td>
<td>175</td>
<td>1.88</td>
</tr>
<tr>
<td>100</td>
<td>2.633</td>
<td>1,003</td>
<td>174</td>
<td>1.83</td>
</tr>
</tbody>
</table>

Table 3.11: Properties of HTP and its decomposition products as a function of concentration

3-25
The design of the chamber described earlier in §3.5 was specific only to the 90% concentration propellant, and hence using the same chamber geometry for the other concentrations will not be optimum. Thus the chamber optimisation method with each of the concentrations listed above was performed, with the following parameters altered.

1. HTP liquid specific heat capacity
2. HTP gas specific heat capacity
3. Monopropellant Isp (at 1,250K)
4. Bi-Propellant Isp at stochiometric O/F
5. Flame temperature of the main flow
6. HTP density

To make the simulation realistic, 92% efficiencies of the theoretical Isp’s have been chosen. Table 3.12 shows the results of the analysis for the optimal chamber with each concentration of HTP. The parallel chamber-section in these cases is film cooled and regenerative cooling is relied upon for the nozzle cooling.

<table>
<thead>
<tr>
<th>Concentration of HTP %</th>
<th>Coolant flow rate gs⁻¹</th>
<th>Chamber Diameter / mm</th>
<th>Chamber Length / mm</th>
<th>Isp /s</th>
<th>Density Isp /s</th>
</tr>
</thead>
<tbody>
<tr>
<td>70</td>
<td>0.4</td>
<td>34.0</td>
<td>13.4</td>
<td>270.9</td>
<td>332.1</td>
</tr>
<tr>
<td>75</td>
<td>1.2</td>
<td>29.0</td>
<td>18.5</td>
<td>276.4</td>
<td>349.6</td>
</tr>
<tr>
<td>80</td>
<td>2.6</td>
<td>26.0</td>
<td>23.0</td>
<td>277.0</td>
<td>360.2</td>
</tr>
<tr>
<td>85</td>
<td>4.4</td>
<td>26.0</td>
<td>23.0</td>
<td>272.5</td>
<td>362.8</td>
</tr>
<tr>
<td>90</td>
<td>6.4</td>
<td>26.0</td>
<td>23.0</td>
<td>269.0</td>
<td>366.7</td>
</tr>
<tr>
<td>95</td>
<td>10.3</td>
<td>28.0</td>
<td>19.8</td>
<td>256.5</td>
<td>327.2</td>
</tr>
<tr>
<td>100</td>
<td>19.4</td>
<td>35.0</td>
<td>12.7</td>
<td>233.4</td>
<td>335.0</td>
</tr>
</tbody>
</table>

Table 3.12: Cooled engine performance versus HTP concentration

The simulation therefore shows that the optimum propellant is not 100% HTP as one might have assumed. Additionally, the maximum Isp and maximum density Isp are not coincident on the same propellant concentration. Instead, the highest Isp is achieved with 80% HTP, while the maximum density Isp is reached with 90% propellant. Thus two versions of the engine can be envisaged, one designed for volume-limited envelopes (suffix VL) and mass-limited envelopes (suffix ML). The volume-limited system is of most interest to small satellites and this happens to
be a very convenient result as the standard feedstock is 90%. Thus the propellant requires no modification in order to reach top density Isp performance.

3.6.1 Comparison of Performance with Commercial Systems

The hypothesised engine is rated at 40N while the closest commercial engines to this thrust are the 10N family of engines. ARC UK manufacture two 10N bi-propellant engines in its LEROS family, the LEROS 10 and the LEROS LTT. Both use the propellants monomethyl hydrazine and nitrogen tetroxide in the form of MON. The performances of these engines are shown below in table 3.13 with comparison to the HTP/kerosene engine, both ML and VL versions.

<table>
<thead>
<tr>
<th>Parameter</th>
<th>LEROS 10</th>
<th>LEROS LTT</th>
<th>ML version</th>
<th>VL Version</th>
</tr>
</thead>
<tbody>
<tr>
<td>Isp/s</td>
<td>286</td>
<td>274</td>
<td>277</td>
<td>269</td>
</tr>
<tr>
<td>O/F</td>
<td>1.65</td>
<td>1.65</td>
<td>11.06</td>
<td>12.49</td>
</tr>
<tr>
<td>DensIsp /s</td>
<td>351</td>
<td>336</td>
<td>360</td>
<td>367</td>
</tr>
<tr>
<td>Expansion Ratio</td>
<td>336:1</td>
<td>336:1</td>
<td>300:1</td>
<td>300:1</td>
</tr>
<tr>
<td>Thrust / N</td>
<td>10</td>
<td>10</td>
<td>40</td>
<td>40</td>
</tr>
</tbody>
</table>

Table 3.13: Comparison of performance with commercial systems

Thus the density Isp performance of the both versions are superior to that of the LEROS 10 and LEROS LTT. The true Isp is worse than that of the LEROS 10. However the predicted performance is impressive, making it a very attractive engine from an end performance point of view.

3.7 Nozzle Geometry Selection

The choice of the nozzle geometry is the least important part of the whole engine. A simple conical nozzle will suffice without a large loss in performance. However the nozzle must be smooth and free of sharp edges. Any step changes will suffer highly localised heating and as a result will burn out. In addition the throat section of the engine must be smoothly contoured to reduce shock disturbances as the gas passes through the trans-sonic region. The standard throat contour that is commonly used is a radiused convergent section of 1.5 times the radius of the throat, and the divergent section 0.382 r_th [SUTTON 92], shown in figure 3.12. While this is easy to manufacture on engines with throat radii of several centimetres, when the throat dimension is merely 4mm in diameter, manufacturing becomes difficult. The convergent radius of 3mm should be possible, but the divergent radius of 0.764mm is only possible if spark erosion techniques are used.
The nozzle expansion cone itself can either be an optimised contoured bell shape, or a simple conical expansion cone. The big advantage with a contoured nozzle, is the nozzle can be made shorter for the same expansion ratio and the divergence angle at the exit will typically be less. The ideal situation is to have a nozzle, which produces a parallel jet and hence directs all of the thrust in the required direction. In a practical nozzle where the gas flow upon exiting the nozzle is not parallel, there is a component of the thrust that is orthogonal to the main thrust direction. Although this extra component is symmetrical and therefore cancels out, it does produce a reduction in the overall efficiency. It can be shown that the efficiency ($\lambda$) of the conical nozzle (excluding friction) is given by equation 3.18, where $\alpha$ is the half angle of the diverging cone.

$$\lambda = \frac{1}{2} (1 + \cos \alpha_{\text{Nozzle}})$$  \hspace{1cm} (3.18)

Typically, when a simple cone is used as the expander, a half angle of 15° is used. This produces an efficiency of 98.3%. Thus on a real system with theoretically 270 seconds of specific impulse, the actual specific impulse will drop to 265 seconds. If, however, the cone was reduced to 10° then the length will increase considerably, but the loss will be 99.2%, giving an Isp of 268 seconds. This is eliminated if long bell shape is used. This loss is completely eliminated if a contoured nozzle is used, whose divergence angle decreases to zero at the exit. Unfortunately such a nozzle is as long as a 10° cone and is usually considered too long and hence too massive [SUTTON 92]. The third type of nozzle is the Rao or shortened contour nozzle, which achieves close to parallel flow of the gases upon exit [RAO 58]. Such a nozzle is around 80% the length of 15° cone, but has only around 8° of divergence at the exit, thus giving 99.5% efficiency. The actual contour of a shortened contour nozzle is based upon the cancellation of expansion shock waves in the initial expansion section, with the compressive shocks in the turning contour. The method of computing the exact contour is based on supersonic aerodynamic flow, the method of characteristics and the properties of expanding gas flow [RAO 58]. An approximate nozzle contour can be found by taking the dimensions off a sectioned LEROS 22N thruster, which has a throat diameter of 4mm, the same as the hydrogen peroxide/kerosene engine. Shown in figure 3-28.
Chapter 3: Theoretical Thruster Design

Nozzle Geometries

![Graph of Nozzle Geometries](image)

Figure 3.13: LEROS, 10° and 15° cone nozzle geometries

3.13 is a graph of the contours of the LEROS engine along with the 10° and 15° conical nozzles. Hence it can be seen that in order to get close to the efficiency of the LEROS nozzle, the conical nozzle will have to be twice as long. Table 3.14 summarises the properties of these three types of nozzle all with an expansion ratio of 200:1, with a nozzle made from copper. It was determined in §3.5 that a regeneratively cooled nozzle needs to be at least 8mm in thickness. However this is only in the throat region as in the expansion cone the gases fall below the melting point of copper. The expansion cone can be any thickness and a value of 2mm wall thickness was chosen to make the manufacturing less expensive.

<table>
<thead>
<tr>
<th>Nozzle</th>
<th>Length /mm</th>
<th>Mass /g</th>
<th>λ</th>
</tr>
</thead>
<tbody>
<tr>
<td>LEROS Contour</td>
<td>75</td>
<td>167</td>
<td>0.995</td>
</tr>
<tr>
<td>15° Cone</td>
<td>100</td>
<td>193</td>
<td>0.983</td>
</tr>
<tr>
<td>10° Cone</td>
<td>150</td>
<td>281</td>
<td>0.992</td>
</tr>
</tbody>
</table>

Table 3.14: Nozzle geometry comparison

The 15° expansion cone provides a length that is similar to the LEROS contour, and is not significantly heavier (only 15%). Choosing a 10° cone, reduces the Isp loss, however the mass is
almost 70% more than the LEROS nozzle. There is an additional factor that so far has not been discussed and this is effect of skin friction. The longer the nozzle, the greater the surface area, and hence the greater the drag. This will reduce the efficiency of the 10° cone, making it less attractive. The other issue is manufacturing. The process by which a contoured nozzle is made would require considerable amount of tooling, while a simple conical nozzle could be produced in any well-equipped workshop. If performance was the key objective then the LEROS type contour would be chosen, however the design philosophy calls for cost reduction but still keeping reasonable performance. The 15° conical nozzle satisfies this criterion, as performance reduction is acceptable in return for a vastly simplified nozzle design. It is interesting to note that the Black Arrow launch vehicle used conical nozzles on the second stage, despite it operating at close to vacuum conditions. Clearly the extra complication of a contoured nozzle was not worth the slight performance increase and was omitted. Hence the 15° nozzle was chosen for the practical designs.

3.8 Conclusions

The theoretical analysis has shown that for a volume limited spacecraft the green propellant combination of hydrogen peroxide and kerosene provides excellent performance, particularly in terms of density Isp. When combined with oxidiser film cooling of the chamber, the performance is preserved and the use of precision-drilled orifices is avoided. In addition the density Isp performance is possibly superior to the more traditional fuel film cooling option. The performance is further increased through the use of a simple regeneratively cooled nozzle. Using this configuration an optimal hydrogen peroxide concentration of 90% was established with chamber dimensions of 26mm diameter and 23 mm in length. Finally the possibility of using a Rao contoured nozzle was examined, however the small performance increase for considerably greater engineering complexity was deemed an unacceptable trade and a standard conical nozzle was chosen.
Chapter 4

4 Thruster Research—Engines and Apparatus

4.1 Introduction
4.2 Engine and Nozzle Designs
4.3 Experimental Test Apparatus
Chapter 4: Thruster Research-Engines and Apparatus

4 Thruster Research-Engines and Apparatus

4.1 Introduction

This chapter presents a detailed description of each of the four engines used in the research and their associated nozzles. The experimental setup required to perform the actual firings is also presented. The data acquisition system used to analyse the performance of each engine is discussed along with a description of the sensors used.

4.2 Engine and Nozzle Designs

The goal of the thruster research was to demonstrate that through the use of decomposed HTP as a gas film coolant, a low cost engine of comparable density Isp to a commercial hypergolic system could be produced. Attempting to design a complete engine from the start was unlikely to be successful and so the problem was tackled in a series of steps. The first engine was designed to test an un-cooled system using low cost components, which should operate with a moderate to high efficiency factor. The very first test engine built, designated the MkI engine, was little more than a kerosene injector added to the end of a catalyst pack. Spontaneous ignition was not achieved and thus it will not be discussed further. The MkII engine was the first design of a test engine engineered from scratch. This engine performed well and returned good C* performance, demonstrating the effectiveness of the combustor design. The MkIII engine implemented the decomposed HTP cooling system with the same combustor design from the MkII engine. The lessons learnt from the MkIII engine then led to the successful MkIV engine.

4.2.1 MkII Engine Description

The MkII engine was designed to solely test the combustion and ignition performance of the chosen combustor design. The chamber was left un-cooled and hence used a large mass of copper as the combustion chamber to act as a heat sink. Calculations showed that 10 seconds of burn time was easily tolerable with the expected heat fluxes and the mass of copper used. Accurate performance data can be obtained in very short burn times of less than 10 seconds, thus the heat sink is adequate for the purpose intended. The nozzle, however, was water-cooled and utilised the same nozzles that were used for the VFP Hybrid Rocket project [HAAG 99] and the use of this nozzle forced the chamber diameter to 18mm. The combustor unit itself consisted of a central port through which decomposed hydrogen peroxide was fed and surrounding it are four kerosene injectors. The injectors are commercially available parts, primarily designed for blowtorches and
an injector is shown in figure 4.1. Due to the orifice diameter of 0.2 mm, they are close to ideal for this engine. The angle at which the injectors are incident to the flow is the maximum that can be achieved while still allowing the injectors to be unscrewed out of the manifold. The injector configuration is also shown in figure 4.1. The catalyst pack was interfaced onto the combustor assembly through 6xM3 bolts and this forms the fuel/oxidiser delivery unit. The chamber and nozzle mates onto the remaining open face of the injector and are clamped into position using external tie bolts. The MkII engine is shown in the CAD model in figure 4.2 and the engineering
drawings are shown in Appendix A. The catalyst pack selected for the bi-propellant research was leftover hardware from a previous research program and consequently the dimensions were not optimised. The physical length of the catalyst pack is 40mm, however it was found the optimum length of the compressed silver gauze was 20mm. Thus throughout the experiments half the length of the catalyst pack was left as a void. All the catalyst packs used in this research consisted of one hundred Ø12.7mm pure silver discs. Each disc was fabricated from 40×36 mesh, Ø0.2mm wire.

The engine design incorporated a number of metal-to-metal face seals, which were made gas tight through the use of a high temperature silicone sealant. The design of the engine had safety in mind with the use of external tie bolts, which in the event of an over pressurisation would stretch and allow the gas to escape through several seals.

Figure 4.3: CAD model of MkIII engine
4.2.2 MkIII Engine Description

The purpose of the MkIII engine was to test the concept of oxidiser gas film cooling. The MkIII engine used the successful combustor design from the MkII engine to retain continuity and avoid combustion issues. The gas film cooling was implemented using a series of 20×0.8mm injection ports positioned immediately downstream of the injector and situated parallel and adjacent to the chamber wall. This orifice diameter was chosen to produce a 1.5 Bar ΔP across each injector when fed with decomposed HTP at a flow rate of 6 g s\(^{-1}\). This was chosen to supply an adequate decoupling pressure differential between the feed supply and the chamber pressure. Due to the construction of the engine, the MkIII version had a widened chamber diameter of 30mm to accommodate the gas film cooling manifold and injector ring. The optimum diameter is 26mm according to the analysis in §3.5 however the practicality of using the MkII combustor forced the slight diameter increase over optimum. The complete CAD model of the MkIII engine is shown in figure 4.3.

4.2.3 MkIV Engine Description

The MkIV engine was designed to improve the cooling effect of decomposed HTP through the use of the coolant in a more effective manner. A copper flow separator for the chamber region was incorporated into the MkIV design whose design theory is outlined in §4.2.4. The flow separator performs the function of preventing the coolant gases meeting the combustion gases in the parallel chamber region. The device fits snugly inside the outer stainless steel chamber and allows the decomposed HTP coolant to spiral around the helix, between the outer stainless steel and the inner copper. The flow of coolant therefore cools the flow separator via forced convection in an analogous way to the cooling jacket of a regeneratively cooled engine. Crucially, once the gas has reached the aft end of the spiral it is not piped into the top of the engine but instead is fed into the convergent part of the nozzle. Thus the coolant then operates in true film cooling mode. The bottom section of flow separator was designed as a flange with a series of 30 holes of Ø0.8mm drilled in it. This was designed to distribute the decomposed HTP evenly around the periphery of the entrance to the convergent nozzle.

The MkII combustor design was again used, as there was little reason for change, thus all the engines MkII to MkIV used the same combustor design. In addition to the redesigned coolant system, the MkIV engine also incorporated an additional oxidiser manifold immediately downstream of the injectors. This was designed to aid atomisation and improve the C* of the engine. The manifold could be fed by a separate catalyst pack and eight radially aligned Ø1mm holes injected the additional decomposed HTP into the centre of the chamber. Figure 4.4 shows the design of the MkIV engine. The complete sectioned drawing of the MkIV engine is given in Appendix A. From the theory in §3.5, the optimum diameter is 26mm when using a
regeneratively cooled nozzle, however the MkIV engine used a chamber diameter of 18mm. This was simply a practical requirement of mating the MkII combustor unit to the MkIV chamber. While being non-optimal, it still provided the same cooling mechanism and hence valid data.

### 4.2.4 Design Theory of the Flow Separator for the MkIV Engine

The flow separator was carefully designed to facilitate adequate cooling of the engine yet be easy to manufacture. A simple tube flow separator was initially proposed, however it became apparent that such a design would not have a high enough convective heat transfer coefficient ($h_g$) to adequately cool the engine. Thus the coolant flow path had to be altered to increase the $h_g$ value and this was achieved by introducing a spiral flow path. The increased velocity and larger surface area over which heat could be removed allowed greater heat flux into the coolant for a given flow rate and $\Delta T$. The actual dimensions of the groove cut into the copper tube to form the helix will therefore dictate the value of $h_g$ and the total surface area.

It was known from calculations, performed previously in §3.3.1, the heat flux into the parallel chamber region is of the order of 4,770W. This power must be removed by the coolant flow around the outside of the spiral. Equation 2.37 specifies the convective cooling coefficient for the
gas coolant. The hydraulic depth and the cross sectional area of the channel are specified by the geometry of the groove profile and both factors feed into equation 2.37. A cooling spiral of generic geometry was considered, shown in figure 4.5. The width and height of each groove and the pitch of the spiral are all parameters that can be varied. For ease of manufacture the depth of each groove was set to 1mm. Thus the pitch and the groove width are the two unconstrained parameters. In order to calculate the power into the coolant, once $h_g$ is calculated, the temperature difference between the hot copper wall and the coolant needs to be known. Clearly the coolant temperature will rise as it travels down the spiral. Similarly the copper wall temperature will rise with distance down the chamber. The average $\Delta T$ between the coolant and the wall temperature was desired to be no greater than 100K otherwise the available coolant temperature rise would be severely reduced. With 100K $\Delta T$ the coolant can only rise to 1,150K, as at the end of the flow separator the wall temperature will be up to 1,250K, the maximum permissible temperature. The analysis proceeded through the selection of a geometry for the groove profile and computing the value of $h_g$ using equation 2.37. The total area from the groove profile multiplied by $h_g$ and the average $\Delta T$ of 100K gave the total power than can be removed from the engine using this groove profile. If this power was greater than 4,770W then this was an acceptable geometry. A series of profiles were analysed and the profile with a pitch (P) of 4.5mm and a groove width (W) of 3mm was chosen as this more than satisfied the power requirements and imposed no excessive machining demands. At a HTP flow rate of 6.5 gs$^{-1}$, the chosen profile has an estimated $h_g$ of 7,432 WK$^{-1}$m$^{-2}$, which gave a $\Delta T$ of 103K.
4.2.5 Nozzle Designs

Three types of nozzle were used in the research, high flow liquid cooled, low flow liquid cooled and radiation/film cooled. The high flow liquid cooled nozzle was designed to use water as a total loss coolant at a flow rate greatly in excess of the propellant flow rates. The purpose of such a nozzle was to allow an engine to be fired for extended periods to determine combustion performance. The high flow liquid cooled nozzle, which was specifically designed for a previous research project, was named the Vla copper nozzle. The engineering drawings for this nozzle are included in Appendix A. Two versions of this nozzle were made, one with a throat diameter of 4mm, designated Vla and a 5mm version, designated Vlb. Both nozzles had an expansion ratio of 4.1:1. The design incorporated wide bores for the flow path around the throat region to facilitate high flow rates. The water channels were separated from the combustion flow by approximately 1mm of copper at the thinnest point. The water was introduced through a single inlet, which splits internally into two paths that travel either side of the throat. After the throat they rejoin into a single path and exit the nozzle via a single pipe. The nozzle only interfaced onto the MkII chamber, thus another high flow liquid cooled nozzle was designed for the MkIII engine. This nozzle with a 4mm throat diameter was designated the Vlc copper nozzle and is shown in figure 4.6. The purpose of this nozzle was to again allow long burn times but this time for the purpose of determining effectiveness of the chamber coolant. No significant thermal changes between the Vla/Vlb and Vlc copper nozzles were made; the latter merely had a larger convergent section to accommodate a larger chamber diameter.

Figure 4.6: Section views of the Vlc copper nozzle
The V2a copper nozzle was designed to simulate a flight type regenerative nozzle. Thus it was engineered to operate at a low flow rate of approximately the same rate as the hydrogen peroxide used for combustion. As the V1a/V1b nozzles were overly cooled, the nozzle remained cool and thus conduction would occur from the chamber into the nozzle. To prevent this unwanted conduction it is desired that the nozzle operate at a similar average temperature to the chamber, thus the nozzle was designed to operate at an average temperature of approximately 1,100K. The liquid HTP entering the coolant inlet will be heated and hence an average coolant temperature of 323K was chosen. Using these approximations, heat transfer will occur over a 777K ΔT. The power into the nozzle region, as calculated in §3.3.1 is approximately 1,978W, thus the product of cooling channel area and the convective heat transfer coefficient needs to be large enough to extract this quantity of power. The product of $h_g$ and the area of the coolant channels area needs to be 2.5 W K\(^{-1}\) or higher in order to extract 1,978W for a 777K ΔT. From the flow properties of hydrogen peroxide, it was calculated that a coolant channel of $\varnothing 1 \times 10$mm provided a large enough heat transfer to keep the nozzle adequately cooled. This nozzle is shown in figure 4.7.
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The final sets of nozzles are the radiation/film-cooled versions designed for the purpose of testing the effectiveness of nitrogen and decomposed hydrogen peroxide as a film coolant. These nozzles are designated RFC-SS and RFC-230 which refers to the radiation film cooled nozzles fabricated from Stainless Steel and the Haynes 230 Nickel/Chromium/Tungsten alloy. These nozzles were employed for use on the MkIV engine only and all had 4mm diameter throats and an area expansion ratio of 6.25:1. The stainless steel version was designed to test the effectiveness of nitrogen as the coolant, while RFC-230, was specifically designed for use with decomposed hydrogen peroxide. Haynes-230 was chosen as the nozzle material in the latter case to tolerate a high temperature oxidising environment. This alloy is not significantly more expensive than regular stainless steel and hence can still be termed low cost. Figure 4.8 shows the CAD model of the radiation film cooled nozzles. The associated engineering drawings can be found in Appendix A.

4.3 Experimental Test Apparatus

In the following sections the experimental test set up used for each of the engines are discussed along with details of the sensors used. All tests were carried out at the SSTL rocket test facility based at E-Site on the Westcott Venture Park, Aylesbury, Buckinghamshire. The basic test configuration of the MkII engine was also used for the MkIII and MkIV engines with changes made for the coolant feed systems and measured parameters.

4.3.1 MkII Engine Test Set Up

The three most important parameters to be measured in a liquid rocket engine are the mass flow rates of both propellants and the chamber pressure. From these the performance in terms of C* can be calculated and hence an overall Isp predicted. A direct calculation of Isp through thrust measurements was not attempted due to the C* value being just as important as Isp when testing at sea level. A vacuum facility was not available, nor would the budget allow such testing. Thus sea level thrust measurements were not pursued.

Both the hydrogen peroxide and kerosene tank pressures were instrumented to ensure that the tank pressures would be logged for future reference and to determine that a failure mid-burn was not due to a tank de-pressurising. The manifold pressures of both the HTP and kerosene lines, directly upstream of the catalyst pack and the injectors, were also instrumented. These transducers were not always present due to the limited number of data channels available and were very much a secondary level of instrumentation.

The performance of the catalyst pack was critical to the success or otherwise of the engine. As such a thermocouple was situated in the exhaust stream of the pack to return data on the
decomposition temperature. This temperature measurement was extremely important for controlling the temperature for the ignition determination tests.

Control of the engine itself was accomplished remotely using electronically actuated solenoid valves. The propellant tanks themselves could be pressurised and vented on command, as could the propellant control valves leading to the engine. As the propellants are pressure fed an inert gas was required to force the liquids from their tanks. Nitrogen was selected as the low-cost pressurant and separate bottles were used for the fuel and oxidiser. This was chosen to avoid vapour from the oxidiser tank passing through the pressurant lines to the fuel tank or vice versa and causing an explosion. Control of the propellant tank pressures was performed through the use of manual pressure regulators. Control of the solenoid valves was achieved using manual switches as opposed to computer control to maximise safety. In the event of power failure, the propellant control valves fall naturally closed and the tank vents open while cutting off the nitrogen pressurisation and thus the system was made inherently safe. This experimental set up is shown in figure 4.9.
4.3.2 MkIII Engine Test Set Up

The MkIII engine used precisely the same feed system for the main propellants as the MkII engine. However, the MkIII engine required a second hydrogen peroxide tank and feed system for the oxidiser gas-cooling operation. This second tank used the same pressurant reservoir as the primary hydrogen peroxide tank, but used its own regulator, thus the pressure could be set independently. The oxidiser gas cooling tank was fed to the second catalyst pack and then into the cooling manifold ring as described in section 4.2.2. The schematic for the MkIII test configuration is shown in the figure 4.10 below. As well as using decomposed hydrogen peroxide...
as a coolant, nitrogen was also tested. This was accomplished with minimal changes through emptying the second tank of hydrogen peroxide and after a small amount of purging, allowing a flow of nitrogen to blow through the whole system.

The instrumentation was also altered over the basic MkII set up. It was vital to know the temperature of the stainless steel chamber and thus nine bead style K type thermocouples were welded to the outside of the chamber. Three columns of three thermocouples were positioned at 120° locations. The middle thermocouple was welded half way down the chamber and the other two 10mm from either end as shown in figure 4.11. Thermocouples 6,7 and 8 coincided with the angular location of an injector, while the other two columns are located between injectors. The thermocouples were interfaced to the data acquisition system through a multiplexer. In addition to this a third flow meter was used to measure the flow of coolant HTP to the second catalyst pack.

4.3.3 MkIV Engine Test Set Up

The MkIV engine used a similar configuration to the MkIII engine except a third HTP tank and feed system was used for some of the experiments. There were four configurations for the MkIV engine designated phases A through D.

- Phase A: Test of chamber cooling, using N₂ or decomposed hydrogen peroxide
- Phase B: Test of regenerative type nozzle using liquid HTP and water. Chamber cooled with N₂ or decomposed hydrogen peroxide
- Phase C: Test of film cooled nozzles RFC-SS and RFC-230. Chamber cooled with N₂ or decomposed hydrogen peroxide
- Phase D: Test of the C* performance with extra oxidiser injection. Un-cooled chamber.
In phase A the V1c copper water-cooled nozzle was used with the same feed system as the MkIII engine. Both nitrogen and decomposed HTP could be fed into the engine as a coolant through the flow separator to test chamber cooling. Thermocouples located at the head and aft end of the flow separator provided information on the gas temperature of the coolant.

In the phase B configuration the V1c and V2a copper nozzles were used. Liquid HTP or water was used as the nozzle coolant to simulate the effect of a regeneratively cooled system. In the HTP cooled nozzle cases, the exhausted HTP was not fed back into the engine as in a regenerative system, but merely collected in a drum. A thermocouple in the coolant exhaust line recorded the
coolant temperature rise. As all the flows needed to be varied independently, a third tank feeding the nozzle was implemented. This operated in an identical fashion to the previously described feed systems and accordingly a flow meter measured the mass flow. The schematic for the phase B configuration is shown in figure 4.12.

In phase C the copper nozzles were replaced with the RFC-SS and RFC-230 nozzles and the same feed system as the phase A configuration was used. The purpose of this set up was to test the possibility of using a radiation/film cooled nozzle. In this configuration the chamber and nozzle was cooled using either decomposed HTP or nitrogen. Four thermocouples were welded to the throat of the nozzle at 90° locations to record the effectiveness of the film coolant. Figure 4.13 shows the location of the thermocouples on the RFC-230 nozzle.

The final configuration, phase D, was designed to measure the C* performance improvements through the addition of extra oxidiser. The chamber coolant used in phases A through C artificially raised the chamber pressure which would give an erroneously high C* and hence was removed for this exercise. A consequence of this was the burn times had to remain short to avoid burnouts. The V1c copper water-cooled nozzle was however retained to prevent the throat melting. The feed system configuration was similar to the MkIII engine, however the secondary catalyst pack was fed into the additional oxidiser manifold at the head of the chamber. This allowed extra oxidiser to be injected to aid the combustion process through better atomisation and mixing. The schematic for this set up is shown in figure 4.14.
Figure 4.14: MkIV engine phase D apparatus schematic
4.3.4 Instrumentation

The instrumentation system consisted of the sensors attached to the rocket engine and the data acquisition system to log the values. The sensors outputed a milliamp signal, which was then converted into a voltage for acquisition by the analogue to digital converter (ADC) card residing inside the PC. The data acquisition card (DAC), manufactured by National Instruments, is capable of 12 bit A-D conversion over 16 channels at a frequency of 16kHz per channel. The milliamp signal passed through a 220Ω resistor connected between the signal input on the card and ground. This produced a voltage between 0.88 and 4.4V and thus fits in the 0-5V range of the card. The software used to drive the card was the National Instruments LabVIEW system. This is an object orientated graphical programming language using building blocks and connecting wires to build the application. Two important pieces of software were written for the project; the acquisition and the calibration software. The calibration software was written to reduce the time spent calibrating the sensors and combined the best-fit line routine and data sampling in one convenient package.

The propellant flow rates were measured using Coriolis Flow Meters. These devices are mass flow instruments and require no correction for density; they also have no moving parts and hence are extremely accurate. Typical accuracies are of the order of 1% of the scale reading and are thus significantly superior to mechanical impeller type devices. The wetted parts of the devices used are all stainless steel and are therefore compatible with hydrogen peroxide. Initial experiments with a turbine flow meter encountered severe problems with HTP on the bearings in the instrument, thus the Coriolis meter provided a solution to compatibility issues. The output of these devices is a 4-20mA signal that is proportional to the flow. This signal output was chosen so that long leads from the sensor to the instrumentation system could be used without loss of signal gain as would be experienced with a voltage output system. Four flow meters were used in the project, two being of the 0-20 gs\(^{-1}\) variety and two 0-100 gs\(^{-1}\). A useful property of the more advanced high mass flow meters was the ability to measure density very accurately, typically to +/- 0.001 gcm\(^{-3}\). This provided the opportunity to record the concentration of the hydrogen peroxide to within 1% at time of firing. Calibration of the actual sensors was not required as they are essentially calibrated once at the factory and the settings remain constant over the lifetime of the instrument. The milliamp output did however need to be calibrated, which was achieved through a water catch-and-weigh test.

Static pressure was measured using standard low cost pressure transmitters, manufactured by Gems Sensors. The quoted accuracy of these devices is typically 0.25% with a maximum error of
±1%. The sensors work on a diaphragm and strain gauge principal that is connected to an amplified bridge. The output of these sensors, when fed with a 24V excitation voltage produces a 4-20mA linear response to pressure. Two ranges of sensors were chosen these being 0-25 Bar and 0-100 Bar gauge pressure. The response time of these sensors was never accurately determined, however from pressure spikes experience during live recordings it was apparent that a sub ms was obtainable, producing an upper Nyquist frequency response of approximately 500Hz. In an attempt to protect the diaphragm from the hot combustion gases, water was injected into the device. However this had a side effect of water hammer, which upon ignition regularly destroyed the sensors. Once this practice was abandoned the survivability of these sensors increased markedly. Calibration of the pressure transmitters was accomplished through a qualified test gauge calibrated to 1%.

Temperature measurements were exclusively carried out using K type thermocouples. Two versions were used, these being the electrically insulated 1 or 1.5 mm diameter rod style devices or the un-insulated bead style thermocouples. These devices work up to 1,000°C and are thus ideal for the range of temperatures experienced. The response time of these devices was low, primarily due to their thermal inertia particularly for the electrically insulated rod style devices. The bead style thermocouples had a better response due to their low thermal mass. The output voltage signal of the thermocouples is approximately linear with temperature over the required range, however some form of amplification is required. Standard thermocouple transmitter amplifiers were extensively used for the insulated rod thermocouples. These amplifiers are cold junction compensated and when excited by 24V produce a 4-20mA signal. They are also programmable to cover a user selectable temperature range. It was discovered however that when used with the un-insulated bead style devices, two or more thermocouples attached to the same conducting metal part caused the output to always jump to 20mA regardless of temperature. This problem was solved by using a custom built differential amplifier to drive these thermocouples. Calibration of the thermocouples was very easy due to the acquisition of a thermocouple simulator. This device, supplied by RS, allows the user to select a temperature and the unit outputs a voltage that a K type thermocouple would generate at the selected temperature. Calibration was a simple matter of recording the amplified voltage and fitting the best line. As the data acquisition card was only capable of recording 16 channels simultaneously, this proved an insufficient number for recording the numerous thermocouples that were used in the MkIII and MkIV engines. Thus a thermocouple multiplexer (TCM) was designed and built to mix 15 channels of thermocouples down to 2 channels. A consequence of the multiplexing process was that each channel is monitored for 20ms periods at the base sample speed. Thus in 1 second each channel is revisited three times. This did not present a problem due to the intrinsically low response time of the thermocouples.
Chapter 5

5 Thruster Research—Experimental Results

5.1 Introduction
5.2 Hot Firing Experimental Procedure
5.3 MkII Engine Combustion Experiments
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5.7 MkIII Engine Cooling Experiments
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5 Thruster Research-Combustion Experiments

5.1 Introduction

This chapter presents the experimental research that was carried out on the bi-propellant thruster. The experiments are split into the combustion and cooling analyses. The combustion experiments established the ignition conditions and combustion efficiency. The cooling experiments determined the effectiveness of the oxidiser gas cooling and the minimum flow rates to cool the nozzle.

5.2 Hot Firing Experimental Procedure

The following section describes the procedure for igniting the engines for the purpose of determining the combustion efficiency, stability and the effectiveness of the various cooling systems. Preparing the engine for ignition always proceeded in the same manner and this was common to all the experiments.

Initially with the tanks empty and the system depressurised, the filling procedure could begin. The fuel tank was filled first and once this had been completed and sealed, the oxidiser filling would proceed. Correct protective clothing, described in §2.5.2, was always worn when handling hydrogen peroxide. A transfer jug was filled directly from the storage drum, which upon carriage into the firing bay, was poured into the HTP run tank. Any spills were washed down with water, and then both tanks were sealed. At this point the water supply was connected to the water-cooled nozzle inlet. The primary valves on the separate nitrogen pressurisation bottles were opened, allowing the respective regulators to be pressurised. The required pressure for each tank was set using the hand-operated regulators. The propellant tanks were then pressurised using the electrically operated solenoid valves, commanded from the control room. Opening these valves connected the downstream side of the regulators to its respective propellant tank. If the HTP control valve was opened with the catalyst cold, there would be a “wash-out” and liquid peroxide would coarse through with very little decomposition. Thus it was important to pre-warm the catalyst. This was achieved by pulsing the HTP control valve to allow a small quantity of HTP into the pack. The heat of decomposition increased the pack temperature, which could be verified on the thermocouple read out. Once a temperature in excess of 100°C was recorded, the firing could commence. Each firing had a selected burn duration, which was entered into the data recording system plus a margin of 10 seconds. The firing then proceeded with a countdown from
3. At the count of 1 the data acquisition system was started and at zero the HTP propellant control valve opened. The engine was then run in monopropellant mode for approximately 2 to 3 seconds before the kerosene propellant control valve was opened and full ignition achieved. The oxidiser lead was deliberately selected to ensure high enough temperatures to ignite the kerosene cleanly. After the required burn duration, or if an anomaly occurred, the engine was shut down by simultaneous closure of both propellant control valves. On early tests it became apparent that the HTP feed required a choke placed upstream of the catalyst pack to decouple the pressure oscillations. The choke was achieved through a needle valve placed inline, which could be wound in and out at will. The procedure for solving the instability was to run the engine in monopropellant mode and wind in the needle valve until the oscillations ceased to be ‘felt’ in the pipe work. Typically, with the engine off, the valve could be backed off slightly to reduce the pressure drop, yet retain oscillation free monopropellant performance.

5.3 MkII Engine Combustion Experiments

The MkII engine was designed to assess the performance of the hydrogen peroxide and kerosene combustor and as such no concessions were made to cooling the engine in a flight type configuration. The combustion stability, C* engine performance and ignition tests were carried out upon this engine. The experimental procedure for each test and the results are described below.

5.3.1 Ignition Tests

The method of thermal ignition relies upon the high temperature oxidiser reacting with the fuel resulting in combustion. In operation, initially just the oxidiser flows until high temperature oxidiser gas is generated, whereupon the kerosene can be injected. In a high performance system it is vital to keep the monopropellant period short to maximise the overall time averaged Isp. Thus it had to be established over what range of parameters ignition actually occurs. The three primary factors effecting the ignition characteristics are; chamber pressure prior to ignition, gaseous oxidiser temperature and the L* of the engine [LIQUID PROP 57b]. The degree of atomisation will also be a factor, but by keeping the propellant flow rates approximately constant, this parameter will similar. By changing the nozzle’s throat diameter, the chamber pressure can be altered over significant ranges. The temperature of the gaseous oxidiser at the point of attempted ignition was selected by varying the time between the oxidiser lead and the injection of the kerosene. The rise of the exhaust temperature of the catalyst pack was fairly gradual and the maximum temperature of approximately 650°C was not achieved until several seconds after the
propellant valve was opened making it relatively easy to control the temperature. The experimental set up for these tests was the standard MkII configuration, detailed in §4.3.1

5.3.1.1 Procedure

The experiment proceeded with the selection of the 4mm throat diameter nozzle and assembly of the engine. Propellants were filled as described in §5.2 and the tank pressures selected. Initially the actual tank pressures were estimated from the pressure drop seen through water tests of the kerosene injectors and monopropellant runs of the catalyst pack.

A series of monopropellant tests were carried out to determine the flow rate over which the chamber pressure corresponded to between 7 and 14 Bar. Once this was completed, hot firings were attempted. The HTP tank pressure was selected to give a pre-ignition chamber pressure of approximately 7 Bar. A kerosene tank pressure was selected, which upon ignition should give an O/F of 7.5 and the tests were commenced. In the first burn, the engine was run in monopropellant mode for 5 seconds in order to drive the oxidiser temperature to approximately 650°C. At this point the kerosene was injected and ignition or otherwise was observed. If ignition occurred, the plume attained its characteristic orange signature with a series of shock diamonds. After completion of the burn, the recorded data was then reviewed with specific reference to the flow rates of the kerosene and hydrogen peroxide. If the O/F value was out of the range of 6:1 to 9:1, the test was repeated with a modified kerosene pressure to bring the O/F within this range. It was primarily desired to get an O/F ratio during the burn of approximately 7.5 (stochiometric) but a lateral tolerance of between 6 and 9 was chosen due to the flame temperature and Isp performance curves being relatively flat over this range. If ignition did not occur, thick white smoke issued from the exhaust and the test was immediately aborted with a full shut down sequence. Without ignition the desired chamber pressure was not achieved. As a greater pressure drop occurred in the catalyst pack as compared to the injectors, the kerosene flow increased more than the oxidiser flow and so a very fuel rich mixture was produced. This could generate erroneous results indicating that for a particular pressure, temperature and L* ignition was not achievable when in reality it was because prior to ignition the mixture was too fuel rich. To alleviate this on these tests, the kerosene pressure was reduced and the test repeated to determine if ignition could be achieved. The experiment was then repeated with a lower ignition temperature, accomplished through the same procedure as described above, but using a shorter monopropellant period. Choosing an exact temperature was inaccurate using this process, however repeatedly performing this exercise generated enough data to cover the required range between 350 and 650°C. The exercise was then repeated with an increased hydrogen peroxide tank pressure, producing a higher chamber pressure prior to ignition in approximately 1 Bar steps through to 14 Bar. The full range of ignition temperatures was again repeated. Once the tests with the V1a Ø4mm nozzle were
complete, the engine was disassembled, the nozzle removed and replaced with the V1b Ø5mm nozzle. The full parameter space was again tested for ignitability in the same manner as described above. In total forty runs produced uniquely useful data, split evenly between the 4 and 5mm throat diameter nozzles. The results of which are shown below in §5.3.1.2. While the oxidiser temperature could have been reduced still further to observe the cut off from reliable ignition to unreliable ignition it was decided not to pursue this on grounds of safety. With a pre-warmed catalyst pack, 350°C oxidiser was readily achieved within 0.10-0.25 seconds of opening the control valve. The only way to achieve lower temperatures was to start from a cold catalyst pack as opposed to a warm one. This poses the potential problem of flooding the chamber with liquid HTP. With the addition of kerosene, the chamber would therefore become explosive which clearly one would wish to avoid.

5.3.1.2 Results

With the Ø4mm nozzle the engine ignited over the full range of conditions, all the way down to 7 Bar chamber pressure and 350°C oxidiser temperature. Figure 5.1 shows the data of a typical firing run with the 4mm nozzle where the oxidiser is shown leading the fuel injection by 4
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Figure 5.2: Mk II ignition parameters (Ø4mm Nozzle)

Figure 5.3: Mk II ignition parameters (Ø5mm Nozzle)
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Figure 5.4: Typical ignition failure of MkII engine with Ø5mm nozzle

Figure 5.5: Typical ignition success of MkII engine with Ø5mm nozzle
seconds. The complete range of test points are shown in figure 5.2 where each data point represents an ignition result. Once the Ø4mm nozzle had been replaced with a Ø5mm nozzle, ignition problems were recorded for pressures below 13 Bar. Figure 5.4 shows the typical data trace recorded of an attempted run where ignition did not occur and for comparison figure 5.5 shows the data where ignition was successful. Below a chamber pressure of 13 Bar ignition became unreliable and below 12 Bar no ignition was observed. At this pressure the temperature of the oxidiser had no measurable effect upon the ignitability of the kerosene. For chamber pressures above 13 Bar, the system also exhibited a lack of sensitivity to the oxidiser temperature between 450 and 650°C where ignition did occur. The full plot of test points is given in figure 5.3 where successful ignition and ignition failures are indicated.

5.3.1.3 Discussion

It is apparent that changing the L* from 75 inches with the Ø4mm nozzle case to the L* of 48 inches in the Ø5mm nozzle case had greatly effected the ignitability of the system. It also demonstrates the requirement to select the correct nozzle throat size for the desired thrust. For a given C* value the mass flow rate, which is dictated by the required thrust, governs the chamber pressure for a given throat area. Having a large throat would result in a low chamber pressure, however as the data has shown, this can make the engine un-ignitable. As the Ø4mm nozzle gave good results over a wide range of temperatures and pressures, it was decided to retain this diameter in the MkIII and MkIV engine designs. With a 200°C preheated catalyst pack, ignition temperatures can be reached in less than 0.25 seconds when used with a Ø4mm nozzle. This is an acceptable ignition delay and on long burns would be insignificant in terms of overall Isp.

5.3.2 C* Performance

The C* of the engine dictates the overall performance and it is vital to achieve a high C* in order to achieve a high Isp. The MkII engine configuration allowed the use of either two or four kerosene injectors. Two injectors have a higher pressure drop across the orifices due to the higher flow rate and therefore the sensitivity of the O/F shift to chamber pressure is reduced, this is discussed in detail in §7.5. However it was also hypothesised that using four injectors as opposed to two would give a higher C* through better propellant distribution and hence better mixing. Thus a series of experiments were conducted to assess the C* performance over an O/F range of 6 to 9 for the two injector and four injector configurations. The experimental set up used was the standard MkII system described in §4.3.1 with the V1a Ø4mm throat water-cooled nozzle.
5.3.2.1 Procedure

With the engine disassembled, two diametrically opposite injectors were unscrewed and replaced with injectors whose orifices had been Silver braised shut. The engine was reassembled and prepared for firing. As pressure and flow measurements are critical in this experiment for producing accurate C* values, a calibration procedure was carried out. An O/F was selected as the target O/F, the first being approximately 6. The hydrogen peroxide tank was set to 30 Bar and the kerosene tank to 25 Bar. A short burn was accomplished using these settings to observe the actual O/F produced. Analysis of the data in this ‘trim burn’ allowed any corrections required to the kerosene pressure. Once the required O/F had been met, the engine was run for approximately 5 seconds. This data was then used to get an accurate C* value by averaging the centre 2 seconds of the burn. The test was repeated at the same pressure settings for two more runs. Slight differences in flow rates for these subsequent burns were recorded due to thermal effects. An increased O/F was then chosen and the test repeated up to an O/F of 9. This O/F range encompasses the optimum operating conditions on either side of stochiometric. Once the tests with the two-injector configuration had been completed, the engine was disassembled and the two blank injectors replaced with unmodified components. The same test series, as described above, was carried out with the 4-injector configuration, resulting in 36 useful data firings.

5.3.2.2 Results

Tables 5.1 and 5.2 show the series of runs for the two-injector and four-injector configurations with the later system achieving a C* of 92±2.1% of theoretical. The theoretical C* values were calculated from the Isp code developed by Curt Selph of AFRL [SELPH 94]. The two injector configuration, however, only achieved around 80±2% C* efficiency. The experimental error in the empirical C* stems from the measurement inaccuracy of the chamber pressure, throat area and the mass flow rate. The diameter of the nozzle was specified to be 4.00±0.05 mm which then provides the error on the area. Thermal expansion of the nozzle during the burn will occur and consequentially produce an erroneous C*, however since the water-cooled nozzle which is combined with short burn times, the effect was minimal.

The results clearly show that the four-injector configuration is preferential over the two-injector design. It would seem logical that increasing the number of injectors further would produce superior results. However with this engine design, fitting five or six screw-in injectors proved impossible and thus this avenue of research was not tested. In addition increasing the number of injectors reduces the pressure drop across each one to a point where instability would be expected thus this configuration was unlikely to produce smooth combustion even if it was physical possible to fit more injectors into the system.
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<th>C*% HTP monoprop</th>
<th>O/F</th>
<th>Fuel Flow Rate/ gs(^{-1})</th>
<th>Empirical C* / ms(^{-1})</th>
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<td>12.2±0.3</td>
<td>96.4</td>
<td>6.2±0.3</td>
<td>2.0±0.1</td>
<td>1277.0±28.8</td>
<td>1600.7</td>
<td>79.8±1.8</td>
</tr>
<tr>
<td>11.4±0.2</td>
<td>92.6</td>
<td>7.6±0.5</td>
<td>1.5±0.1</td>
<td>1269.9±28.6</td>
<td>1610.5</td>
<td>78.9±1.8</td>
</tr>
<tr>
<td>12.9±0.4</td>
<td>95.7</td>
<td>8.6±0.4</td>
<td>1.5±0.1</td>
<td>1212.1±27.3</td>
<td>1580.4</td>
<td>76.7±1.8</td>
</tr>
<tr>
<td>12.7±0.2</td>
<td>93.5</td>
<td>9.8±0.2</td>
<td>1.3±0.1</td>
<td>1268.3±28.6</td>
<td>1545.6</td>
<td>82.1±1.9</td>
</tr>
<tr>
<td>12.9±0.3</td>
<td>95.6</td>
<td>9.9±0.3</td>
<td>1.3±0.1</td>
<td>1230.1±27.7</td>
<td>1545.5</td>
<td>79.6±1.8</td>
</tr>
</tbody>
</table>

Table 5.1: C* performance of the two injector configuration

<table>
<thead>
<tr>
<th>Ox Flow Rate/ gs(^{-1})</th>
<th>C*% HTP monoprop</th>
<th>O/F</th>
<th>Fuel Flow Rate/ gs(^{-1})</th>
<th>Empirical C* / ms(^{-1})</th>
<th>Theoretical C* / ms(^{-1})</th>
<th>C* %</th>
</tr>
</thead>
<tbody>
<tr>
<td>9.3±0.2</td>
<td>92.6</td>
<td>6.2±0.2</td>
<td>1.5±0.1</td>
<td>1432.6±32.3</td>
<td>1600.7</td>
<td>89.5±2.0</td>
</tr>
<tr>
<td>9.2±0.3</td>
<td>90.4</td>
<td>6.6±0.3</td>
<td>1.4±0.1</td>
<td>1458.2±32.8</td>
<td>1607.4</td>
<td>90.7±2.1</td>
</tr>
<tr>
<td>9.3±0.2</td>
<td>91.3</td>
<td>6.8±0.5</td>
<td>1.4±0.1</td>
<td>1438.2±32.4</td>
<td>1609.3</td>
<td>89.4±2.0</td>
</tr>
<tr>
<td>9.4±0.4</td>
<td>91.6</td>
<td>7.4±0.2</td>
<td>1.3±0.1</td>
<td>1486.3±33.5</td>
<td>1606.9</td>
<td>92.5±2.1</td>
</tr>
<tr>
<td>9.6±0.1</td>
<td>92.5</td>
<td>8.0±0.3</td>
<td>1.2±0.1</td>
<td>1421.9±32.0</td>
<td>1596.2</td>
<td>89.1±2.0</td>
</tr>
<tr>
<td>9.7±0.2</td>
<td>93.2</td>
<td>8.2±0.3</td>
<td>1.2±0.1</td>
<td>1443.7±32.2</td>
<td>1590.9</td>
<td>90.8±2.1</td>
</tr>
</tbody>
</table>

Table 5.2: C* Performance of the four injector configuration

5.3.2.3 Discussion

The peak C* value was recorded with four injectors and an O/F ratio of 7.4:1 where a value of 92.5±2.1 percent of the theoretical value, was achieved. It is apparent from the data that the performance of the combustor is close to commercial hypergolic systems within experimental error. It has also been reported that similar combustion efficiencies of 92% were obtained with a larger engine that used the same injection system as the MkII engine [WU 99]. In addition with the two and four injector configurations there appears to be no trend of combustion efficiency with the O/F ratio.

The use of a two-injector configuration produced considerably poorer combustion efficiencies than the four-injector configuration. It was expected that the former should produce lower efficiency and this was vindicated with the results.
5.3.3 Combustion Stability

The combustion stability over a range of O/F values and thrusts was assessed to determine the operational limits of the combustor design. It was known from theory that the decoupling pressure drop across the injectors was low due to the slightly oversized orifice diameters. Thus when operating at low thrusts and in an oxidiser rich regime, the kerosene flow becomes so low that the pressure-drop across the injector is insignificant. Estimated vacuum thrusts of between 25 and 60N were chosen to test the combustor, corresponding to a total propellant flow rate of between 8 and 20 gs\(^{-1}\). The O/F values chosen were again between 6 and 9:1 to span either side of the stochiometric value of 7.5:1. Two nozzle sizes were chosen, the Ø4mm V1a copper nozzle and the Ø5mm V1b copper nozzle. These were chosen to observe any increase in stability of the engine with an enlarged L* provided by the smaller nozzle. The standard MkII engine set up was used for this experiment, details of which are given in §4.3.1

5.3.3.1 Procedure

With the engine set up in a four-injector configuration and the Ø4mm nozzle, an estimate of the HTP tank pressure required to produce a flow of 7gs\(^{-1}\) was dialled into the regulator. The same process was applied to the kerosene tank, this time for a target flow of 1gs\(^{-1}\). At these flow rates and with an estimated vacuum Isp of 300s, the thrust would be approximately 25N. A trim burn was then performed to adjust the O/F selection and the make changes if necessary. Once the O/F was chosen a long burn of 10 seconds was attempted. During the burn it was apparent if significant oscillations were present due to the noise and variations in the plume. After the firing the actual combustion chamber pressure trace was analysed and the amplitudes relative to the average chamber pressure, which when expressed as a percentage was the metric used for combustion stability. The experiment was repeated for a thrust of approximately 40N. The engine was then dismantled and the V1a Ø4mm nozzle replaced with the V1b Ø5mm nozzle. This allowed higher thrusts to be achieved for the same chamber pressure. Combustion stability with this new configuration for a calculated thrust of approximately 60N was then tested. Fifteen firings were completed over the three thrust ranges.

5.3.3.2 Results

Shown in table 5.3 are the results from the combustion stability tests. The calculated approximate vacuum thrust was derived from the total mass flow rate of the propellants and the actual empirical C* value. A thrust coefficient of 1.97 was assumed throughout. The percentage oscillation of the chamber pressure was calculated for just the centre portion of the main burn using the following method; The average chamber pressure of the main burn section was
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<table>
<thead>
<tr>
<th>Approximate Calculated Vacuum Thrust / N</th>
<th>O/F</th>
<th>Nozzle Diameter /mm</th>
<th>Chamber Pressure/ Bar</th>
<th>% Oscillation Amplitude</th>
</tr>
</thead>
<tbody>
<tr>
<td>24.8</td>
<td>4.1</td>
<td>4.0</td>
<td>8.9±0.1</td>
<td>5.6</td>
</tr>
<tr>
<td>24.0</td>
<td>5.1</td>
<td>4.0</td>
<td>8.7±0.1</td>
<td>6.9</td>
</tr>
<tr>
<td>25.1</td>
<td>7.2</td>
<td>4.0</td>
<td>9.2±0.1</td>
<td>6.5</td>
</tr>
<tr>
<td>24.9</td>
<td>9.4</td>
<td>4.0</td>
<td>9.1±0.1</td>
<td>5.5</td>
</tr>
<tr>
<td>37.4</td>
<td>4.6</td>
<td>4.0</td>
<td>14.7±0.1</td>
<td>6.8</td>
</tr>
<tr>
<td>38.7</td>
<td>5.7</td>
<td>4.0</td>
<td>14.4±0.1</td>
<td>7.6</td>
</tr>
<tr>
<td>37.4</td>
<td>6.4</td>
<td>4.0</td>
<td>13.8±0.1</td>
<td>7.2</td>
</tr>
<tr>
<td>37.4</td>
<td>7.1</td>
<td>4.0</td>
<td>15.9±0.2</td>
<td>8.8</td>
</tr>
<tr>
<td>36.9</td>
<td>8.1</td>
<td>4.0</td>
<td>14.3±0.1</td>
<td>7.0</td>
</tr>
<tr>
<td>38.6</td>
<td>9.7</td>
<td>4.0</td>
<td>16.4±0.2</td>
<td>6.1</td>
</tr>
<tr>
<td>65.2</td>
<td>4.6</td>
<td>5.0</td>
<td>11.0±0.1</td>
<td>3.6</td>
</tr>
<tr>
<td>64.1</td>
<td>5.6</td>
<td>5.0</td>
<td>10.6±0.1</td>
<td>5.7</td>
</tr>
<tr>
<td>61.0</td>
<td>6.8</td>
<td>5.0</td>
<td>10.0±0.1</td>
<td>5.0</td>
</tr>
<tr>
<td>60.6</td>
<td>7.6</td>
<td>5.0</td>
<td>11.2±0.1</td>
<td>5.4</td>
</tr>
<tr>
<td>65.2</td>
<td>8.4</td>
<td>5.0</td>
<td>10.9±0.1</td>
<td>4.6</td>
</tr>
</tbody>
</table>

Table 5.3: Combustion stability results

calculated and an approximate value of the oscillation amplitude estimated. The number of points that fell outside of this amplitude band either side of the mean were then counted using EXCEL. The amplitude was then iteratively adjusted until 10% of the points fell outside of the selected oscillation amplitude. Hence 90% of the time the oscillations are lower than the selected oscillation amplitude.

5.3.3.3 Discussion

The results show that regardless of the thrust, the combustion chamber oscillations are between 3.6 and 8.8% of the average combustion chamber pressure. There is however no correlation with O/F. It was expected that as the mixture was pushed towards the fuel rich side of stochiometric, the fuel flow increases and thus produces a greater pressure drop across the injectors, decoupling the oscillations in the chamber from the feed system. Hence one would expect to see reduced oscillation amplitude. However as this was not seen this points to the fact that the combustion oscillation is intrinsic to the propellant combustion and is not a function of the feed system. It has been stated that the smooth combustion is defined as chamber pressure oscillations of less than
±5% of the average chamber pressure [SUTTON 92]. While the engine was not operating within this region for most of its envelope, it does not exceed this by a significant margin and hence can still be termed moderately smooth combustion. The implications of this are important from the engine cooling aspect. When oscillatory behaviour is induced, turbulence is generated which can increase the convective heat transfer to the chamber walls. In addition, turbulence causes break up of film coolant flows, both of which lead to an engine that is more difficult to cool.

5.3.4 MkII Engine Combustion Conclusions

The results from the experiments performed on the MkII engine show that the 4-injector configuration coupled with a 4mm nozzle provides a simple and efficient design for a hydrogen peroxide and kerosene combustor. Ignition was readily achieved with chamber pressures above 7 Bar prior to kerosene injection and temperatures down to as low as 350°C. This provides a system, which could achieve full chamber pressure within 250ms of opening the propellant valves. While the performance of this combustor is not as high as a good hypergolic liquid bi-propellant system [SUTTON 92] with a peak C* performance of 92.5±2.1% of theoretical, it is comparable to the performance of a similar injection scheme reported previously [WU 99]. The combustion smoothness was rougher than commercial engines with recorded values as high as 8.8% variation of the $P_c$. This oscillation amplitude is not excessive as it only just falls outside of what is nominally termed smooth.

5.4 MkIV Engine Combustion Experiments

The experiments described below assessed the performance of two methods of combustion enhancement implemented on the MkIV engine. Both methods relied upon increased atomisation and turbulence in the injection region. This would lead to better mixing and hence greater combustion efficiency. It was the goal of this research to raise the C* performance above the MkII level of 92.5±2.1% to a value closer to commercial hypergolic engines.

5.4.1 C* Performance Improvements

As the fuel atomisation is produced through aerodynamic shear forces of the fast moving oxidiser, the faster the velocity the greater the degree of atomisation and hence mixing. The MkIV engine provided two mechanisms to produce the necessary velocity increase

1. Choking of the central oxidiser port
2. Additional oxidiser injection

By constricting the oxidiser inlet port into the chamber from its original diameter of 9mm down to 5.3mm, a three-fold increase in velocity and hence increased atomisation would occur. This was
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expected to lead to an enhanced C* value. In addition, the MkIV engine also had the provision for introducing extra decomposed HTP radially into the chamber through eight Ø1mm injection ports. This was again designed to assist the combustion process through the high velocity jets of oxidiser providing large shear velocities.

The first set of tests did not include the additional oxidiser and just employed choking of the central oxidiser port. The experimental set up used was the Phase D configuration of the MkIV system, described in §4.3.3. The second set of tests used the full Ø9mm oxidiser inlet port but with additional oxidiser injection and again used the Phase D configuration.

5.4.1.1 Procedure

The MkIV engine was disassembled and the injector block removed. A stainless steel plug was then inserted into the injector port and hammered into location to produce a snug fit. This plug was then drilled out to Ø5.3mm and the complete engine reassembled. No flight propellant cooling was attempted, instead the chamber remained un-cooled and the V2a copper water-cooled nozzle was used. If a chamber coolant was used, it would raise the combustion chamber pressure thus producing erroneous C* measurements. The pressures required to produce an O/F of 7.5:1 were known from previous experimental data acquired at the start of the MkIV tests and these were set on the regulators. For the operation of the engine same procedure as described in §5.2 was followed. The engine was then disassembled and the central bore drilled out to Ø6.3mm.

The test was repeated again and finally the central bore was enlarged to the full Ø9.0mm and the same test performed.

For second set of tests with the radially injected additional oxidiser the chamber region again remained un-cooled and the V2a copper water-cooled nozzle was used to prevent throat melting. The first run in this configuration was to produce a C* value with zero additional oxidiser. After this run a series of 5 second burns were executed with flow rates of additional oxidiser varied between 3 and 8 gs\(^{-1}\). At each flow rate four runs were conducted, producing twenty runs in total. As the flow rate of the additional oxidiser increased, the main flow of HTP had to be correspondingly decreased to maintain a constant chamber pressure. Thus with every new oxidiser flow rate, a series of trim burns were performed to attain an approximately stochiometric O/F and a chamber pressure of approximately 20 Bar. The sequence for firing the engine required warming of both catalyst packs. The main oxidiser valve was then opened followed by the additional oxidiser valve, which was then allowed to burn in monopropellant mode for 2 seconds. Following this the kerosene was injected and ignition occurred.
5.4.1.2 Results

Choking the oxidiser inlet port to produce a higher velocity proved detrimental to the performance of the engine as ignition failed to occur. With either the Ø5.3mm or the Ø6.0mm choke, the engine did not ignite and consequentially no C* values could be obtained. Once the choke was removed and the full Ø9.0mm inlet port was used, reliable ignition occurred.

The introduction of extra oxidiser, injected radially into the chamber produced superior results as reliable ignition occurred and the C* performance was shown to increase. Table 5.4 gives the details of these results.

<table>
<thead>
<tr>
<th>Oxidiser Flow Rate/ gs⁻¹</th>
<th>Extra Oxidiser Flow Rate/ gs⁻¹</th>
<th>O/F</th>
<th>Theoretical C* / ms⁻¹</th>
<th>Empirical C* / ms⁻¹</th>
<th>C* %</th>
</tr>
</thead>
<tbody>
<tr>
<td>12.7±0.3</td>
<td>0±0.0</td>
<td>6.70±0.4</td>
<td>1608.6</td>
<td>1417.2±31.9</td>
<td>88.1±2.0%</td>
</tr>
<tr>
<td>11.7±0.2</td>
<td>3.2±0.2</td>
<td>6.89±0.5</td>
<td>1606.7</td>
<td>1434.8±32.3</td>
<td>89.3±2.0%</td>
</tr>
<tr>
<td>10.9±0.2</td>
<td>4.5±0.3</td>
<td>6.80±0.5</td>
<td>1609.3</td>
<td>1475.7±33.2</td>
<td>91.7±2.1%</td>
</tr>
<tr>
<td>11.2±0.4</td>
<td>4.8±0.3</td>
<td>8.40±0.6</td>
<td>1585.3</td>
<td>1455.3±32.8</td>
<td>91.8±2.1%</td>
</tr>
<tr>
<td>8.7±0.3</td>
<td>8.1±0.3</td>
<td>7.44±0.5</td>
<td>1606.9</td>
<td>1471.9±33.1</td>
<td>91.6±2.1%</td>
</tr>
</tbody>
</table>

Table 5.4: C* results of MkIV engine with additional oxidiser injection

5.4.1.3 Discussion

The failure of the choked MkIV engine can be attributed to the ‘stay-time’ of the fuel in the oxidiser core. With the choke in place the cross sectional area of the central oxidiser core was reduced, resulting in the injected kerosene being subjected to higher shear forces but for a shorter period of time. In addition as the kerosene jet passes through the hot oxidiser, the kerosene is heated and upon reaching its ignition point, will burn. Reducing this heating time will have a detrimental effect upon the ignitability. This was compounded by the temperature reduction of the oxidiser associated with the velocity increase. However this fact alone cannot explain the ignition failure. Ignition studies, discussed in §5.3.1, demonstrated that the temperature of the oxidiser has little dependency upon the ignition conditions, providing it is above 350°C. Thus there is a clearly a trade between the velocity of the oxidiser and the ‘stay-time’ of the fuel in the oxidiser core. The fact that reducing the bore to Ø6.3mm caused ignition failure indicates that the Ø9.0mm inlet port is close to the minimum diameter.
The data from the additional oxidiser tests show that there is a possible increase in $C^*$ of the order of 2% over the standard MkIV injector without additional oxidiser injection. However with the error bars being ±2%, it makes this a tenuous judgement. This evidence coupled with the fact that a $C^*$ value of 92% of theoretical was recorded in the MkII engine (without the extra oxidiser) implies that the effect of the additional oxidiser is limited. The discrepancy between the MkII engine and the MkIV engine of 2%, is due to the different chamber configurations. Both engines used an identical combustor and Ø18mm combustion chambers, but the MkII has an $L^*$ of 75 inches (1.91m) while the MkIV has an $L^*$ of only 55 inches (1.40m). Thus the MkII engine would be expected to return higher performance due to the extended period over which stream tube combustion can occur.

5.5 Engine Combustion Conclusions

From the one hundred and fourteen useful firings the simple combustor design proved to be an effective system over vacuum thrust ranges of between 25 to 60N.
The ignition tests demonstrated that with the MkII engine with the 4mm throat diameter nozzle and an $L^*$ of 75 inches (1.91m), reliable ignition was readily achieved over a range of pre-ignition pressures from 7 through to 14 Bar. Ignition occurred over the full range of oxidiser temperatures tested and reliable ignition above 350°C was achieved. Using a 5mm throat diameter nozzle and an $L^*$ of 48 inches (1.22m) resulted in a minimum pre-ignition chamber pressure of approximately 13 Bar. This demonstrates the importance of maintaining a high pre-ignition chamber pressure in a hydrogen peroxide/kerosene system. As reliable ignition was always required, the 4mm throat diameter nozzle was used in all further engine designs.
The $C^*$ performance of the MkII engine in the four-injector kerosene configuration was comparable to designs of injectors tested elsewhere, returning approximately 92% of expected $C^*$ value. Choosing a two-injector configuration produced a dramatically lower $C^*$ value of only 80% if theoretical. Also the use of the MkII injector in the MkIV engine produced a slightly suppressed $C^*$ and this was concluded to be due to the shorter chamber, as all other engine dimensions parameters remained constant.
The success of the combustor design provided a useful research tool for the investigation of the engine cooling, discussed in the following sections. As such no further development was carried out on the combustor design and the $C^*$ efficiency of 90-92% was judged to be adequate for the research purposes.
5.6 Engine Cooling Experiments

This section presents the experimental research that was carried out into cooling of the bi-propellant thruster. The chamber and nozzle cooling aspects are divided into two sections with the relevant procedures, results and discussions presented for each one in turn. In the first section the cooling experiments of the MkIII and MkIV chambers are presented. In the final section the various flight-type nozzles are examined.

5.7 MkIII Engine Cooling Experiments

The MkIII engine was designed to incorporate gas film cooling in the parallel chamber section but use the same combustor as used in the MkII design. The tests on the MkIII engine were designed to assess the degree of cooling experienced on the parallel chamber section. Two experiments were performed; the first used nitrogen as the gas film coolant, while the second used decomposed hydrogen peroxide. Thermocouples, described in §4.3.2, were welded onto the chamber to provide information on the effectiveness of the cooling.

5.7.1 Nitrogen Cooled Chamber

The nitrogen injection on the inside periphery of the MkIII chamber was intended to produce a cool boundary layer, thus protecting the stainless steel chamber from the high temperature combustion products. The choice of nitrogen as a coolant was not intended to simulate flight propellants but to test the effectiveness of gas cooling. Using nitrogen avoids the issues of combustibility of the coolant with the fuel and hence provides a clear picture of the coolants effectiveness. The test set up for this experiment is described in §4.3.2.

5.7.1.1 Procedure

With the engine set up according to §4.3.2 with the V1c copper water-cooled nozzle, a series of 5-second un-cooled test firings at an O/F of 7.5 were attempted to confirm operation of the engine. This obtained data on the un-cooled chamber temperature rise. As nitrogen was not introduced the same procedure as detailed in §5.2 was followed. Following this test, a series of 'trim burns' were performed to set the flow rates of the hydrogen peroxide, kerosene and nitrogen to 13.5, 1.8 and 4.0 gs\(^{-1}\) respectively, adjusting propellant pressures as required. A deliberately low nitrogen coolant flow rate of 4.0 gs\(^{-1}\) was selected. This was done so the nitrogen flow could be switched on and off during a run to show the effect of the coolant without seriously affecting the O/F ratio. A large mass flow of nitrogen would increase the chamber pressure, greatly affecting the kerosene flow rate. This would lead to erroneous results due to variation in flame temperature. The target
O/F was 7.5:1 and even with nitrogen at just 4.0 g s⁻¹ the O/F changed from 5.2:1 to 9.1:1 upon N₂ injection. The runs required ignition without the coolant and after 2 seconds of ignition the coolant was introduced for 3 seconds. It was not expected that this flow rate of coolant would adequately cool the engine; the theory suggested a higher rate would be required. However the thermocouple data would show the degree of cooling that was achieved.

5.7.1.2 Results

Without any form of cooling the MkIII engine ran in a similar fashion to the MkII engine and after 5 seconds of burn time, the chamber began to glow as expected. Figure 5.6 shows the data recorded from a typical un-cooled run and figure 5.7 shows the chamber thermocouple temperatures. From ignition the temperature rise was near linear to the point where the engine was shut down. The peak temperature recorded was 850°C however at this point the lowest temperature recorded was only 560°C, giving a 290°C difference between sections of the chamber. This demonstrates that ‘even’ heating in the chamber was not taking place but rather certain areas were becoming localised hot spots. Referring to figure 4.11, the temperatures of the chamber can be shown in the form of a map representing the cylindrical projection of the chamber onto a 2D plane. This is shown below in table 5.5 and constitutes averaged data over four runs at an O/F of 7.5±0.5:1 after 5 seconds of ignition.

![Figure 5.6: Typical firing data for the un-cooled MkIII engine](image-url)
Table 5.5: Average temperature map of un-cooled engine after 5 seconds of ignition

When nitrogen was introduced as the coolant, the thermocouple results demonstrated that cooling near the coolant injection region was taking place, however down stream of this the coolant had little effect. Thermocouples 0, 3 and 6 all register significant effects upon injection of the nitrogen and with reference to figure 4.11, these are located 10mm downstream of the coolant injection manifold. Figure 5.8 shows the typical firing data of a run where the nitrogen coolant was introduced for 3 seconds and then shut off. The data indicates that upon coolant injection the instability of the engine increases as can be seen from the chamber pressure trace. Figure 5.9 shows the data recorded from the thermocouples for one run. In this run it is quite obvious that

![Figure 5.7: Typical chamber temperature rise of un-cooled MkIII engine](image)
Figure 5.8: Typical firing data for pulsed $\text{N}_2$ cooling of the MkIII engine

Figure 5.9: Typical chamber temperature rise of $\text{N}_2$ cooled MkIII engine
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upon injection of the nitrogen, the rate of increase of temperature of channels 0, 3 and 6 drops dramatically. The temperature increase rate of the remaining channels does reduce but not as effectively as those closest to the injector. In addition, the channels mid way down the chamber record the same temperature as the channels at the bottom of the chamber. From the four runs comprising of two seconds of un-cooled combustion and then 3 seconds of combustion with nitrogen cooling, an average chamber temperature map was generated, shown in table 5.6.

| Injector End | TC 0 439±50°C | TC 3 480±50°C | TC 6 432±50°C |
| TC 1 636±50°C | TC 4 656±50°C | TC 7 622±50°C |
| TC 2 595±50°C | TC 5 605±50°C | TC 8 611±50°C |

Nozzle End

Table 5.6: Temperature map of N₂-cooled engine after 5 seconds of burn time

5.7.1.3 Discussion

In the un-cooled case, the data shows the injector end of the chamber was relatively cool which was most likely due to partial combustion of the propellants and hence a suppressed flame temperature. There is also the possibility that fuel was being sprayed onto the walls cooling the very top of the chamber. The mid section of the chamber was significantly hotter than the injector end and this was due to almost complete combustion occurring by the time the propellants passes this point, causing a high degree of heating. Further down, the chamber temperatures cool again due to the chamber being oversized. Once all the propellants have been burnt, the extra chamber length is simply radiating energy and hence cooling the combustion products. This was expected due to the $L^*$ of the chamber being 133 inches (3.38m) while the recommended $L^*$ for an engine with these propellants is only 50 inches (1.27m). There was also a variation in temperature around the chamber at a fixed distance axially down the chamber. Observing the values for thermocouples 0, 3 and 6 shows that adjacent to the injector, the temperature was the lowest while between the injectors the temperature was higher at the two measured locations. This is consistent with the hypothesis that fuel was impinging upon the walls of the chamber, initially causing local cooling. Further down the chamber, at the mid way section, there is still a disparity in temperatures however this time the thermocouple in between an injector was reading the lowest temperature, while the other thermocouple, also in between an injector, was reading the highest value. This implies asymmetric mixing and hence preferential hot spots on the chamber.

The nitrogen cooled MkIII engine demonstrated pronounced oscillations immediately after the nitrogen flow began. This was believed to be due to the higher chamber pressure. Raising the

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chamber pressure reduces the pressure differential between the coolant tank and the engine, thus making the system more susceptible to feed system instability. In the nitrogen cooled case, there was again asymmetric heating of the chamber, but significant cooling was experienced at the injector end of the chamber. In addition, the mid and aft thermocouples 1, 2, 4, 5, 7 and 8 all recorded similar temperatures. In the un-cooled case, the mid section was significantly hotter than the lower section. This implies the nitrogen coolant was still partially effective by the time it reached this region. However at the aft end of the chamber the temperatures in the cooled and un-cooled cases were similar, suggesting the film coolant had mixed with the main flow and the cooling effect had been lost.

It is believed that the nitrogen coolant, while having an effect close to the injection port, rapidly loses its effect due to turbulent mixing with the main combustion flow. As mixing occurs, a greater percentage of the gas against the wall is composed of combustion products and hence a greater heat flux into the chamber wall will occur.

5.7.2 Hydrogen Peroxide Cooled Chamber

The injected decomposed hydrogen peroxide coolant was intended to form a warm boundary layer against the wall of the chamber, thus cooling and insulating the wall from the hot combustion products. Unlike the nitrogen coolant, the hydrogen peroxide is reactive with the fuel and as such it was vital that the main propellant flow was run either at exactly stochiometric or oxidiser rich to ensure that the any un-burnt fuel does not burn with the oxidiser boundary layer. While the available temperature rise of the decomposed HTP is lower than the nitrogen coolant, the lower temperature differential can provide a more stable film coolant. Kelvin Helmholtz instability is excited by a velocity and temperature differential, thus the higher temperature HTP coolant would inherently be more stable than the cool nitrogen.

5.7.2.1 Procedure

The MkIII engine was assembled and configured in accordance with the set up described in §4.3.2. A main flow O/F of approximately 8.5 was aimed for with an absolute minimum of 8.0 in order to keep the main flow oxidiser rich. From the purely theoretical calculations in §3.3.1.5, the required mass flow rate of the coolant flow was approximately 7.6 g/s. Through a series of short 'trim burns' the pressures in the 3 propellant tanks were set such that the main flow had an O/F of 8.5, the main oxidiser was set to a flow rate of 12.0 g/s and the coolant oxidiser at 7.6 g/s. The engine was then run for an intended 10 seconds or until an intense orange glow from the chamber was seen. The test was then repeated three more times with these settings. The procedure for operating the engine was identical to the scheme performed in §5.7.1.1 with the exception that the
coolant was introduced before ignition as opposed to during it. This was a safety requirement as the catalyst pack used for decomposing the coolant HTP must be operating when ignition occurs. By running both packs before injecting the kerosene, good operation of both could be ascertained and thus the possibility of injecting liquid HTP was avoided.

5.7.2.2 Results

Figure 5.10 shows the firing data of a typical run with the decomposed HTP film coolant set to 7.6gs⁻¹ and an O/F of 8.5. Figure 5.11 shows the temperature rise of the chamber for the same run. It was noticed on the four runs operated with these flow parameters that the firing duration before the chamber started to glow was similar to the un-cooled case. The average temperature map of the four runs with 5 seconds of ignition, is shown in table 5.7. While all the thermocouples exhibited a temperature reduction as compared to the un-cooled case, shown in table 5.6, thermocouples 7 and 8 are substantially hotter than the other thermocouples. Thermocouple 7 in particular was less than 100°C below the un-cooled engine temperature. In addition, it is clear from the trend in figure 5.11 that the equilibrium temperature for thermocouples 7 and 8 are in excess of 1,000°C.
5.7.2.3 Discussion

The effect of the decomposed HTP coolant was limited when the entire chamber is considered and in particular thermocouples 7 and 8 recorded temperatures that were similar to the un-cooled case. The suppressed temperatures for the remaining thermocouples indicate that the coolant is indeed having an effect. The data recorded also suggests that at these locations a sub 1,000°C equilibrium temperature may be achievable. In order to determine the equilibrium temperature the engine must be run for longer periods. However the excessive temperatures in the zones where thermocouples 7 and 8 were located prohibited this. The cause of this excessive heating in these areas is believed to be due to the location of the kerosene injectors. Thermocouples 6,7 and 8 are opposite an injector and hence any un-burnt fuel will be sprayed on to the wall. The initially
un-burnt fuel will then combust with the decomposed 'coolant' HTP causing full combustion temperatures to be generated and hence ineffective cooling. In between the injectors, the mixture will be heavily oxidiser rich, as is desired, and hence cooling was achieved.

5.7.3 MkIII Engine Conclusions

It was concluded that the method of injecting either an oxidiser or an inert gas into the chamber of the MkIII engine produced inadequate cooling of the whole engine. The data recorded from the nitrogen injection test suggested that turbulence in the chamber resulted in failure to cool the entire chamber. While significant cooling was seen close to the injection manifold, the effectiveness rapidly diminished with distance away from this point. In the case of hydrogen peroxide coolant, no significant cooling was observed inline with the kerosene injectors. It is believed that fuel impingement onto the oxidiser rich layer near the wall generates full combustion temperatures thus negating any benefit from the introduction of coolant oxidiser. The concept of fuel impingement was confirmed with the cold flow model described in §3.4.1, which showed that not all the atomised fuel is carried away downstream, but instead a significant proportion does indeed impinge onto the walls. The experiments conducted with the MkIII engine do not show that oxidiser film cooling is impractical; merely the implementation in this configuration does not work. If the injector could be changed to limit or eliminate the quantity of fuel impinging upon the wall and a more laminar flow created, again through an alternative injector, the coolant performance should be dramatically increased. However the most robust solution to the problem was through the use of a flow separator to keep the coolant and main flows physically apart while in the chamber section. It is apparent from the C* measurements that good combustion was being obtained and a corollary of this is the fuel in the combustion gases, upon reaching convergent section of the nozzle, must be almost completely burnt. Thus injecting the oxidiser as a film coolant into the convergent region of the nozzle would not result in combustion.

5.8 MkIV Engine Experiments

The MkIV engine was designed with the lessons learnt from the MkIII engine and consequentially incorporated a copper flow separator, discussed in §4.2.3 and §4.2.4. With the new configuration, the engine was again tested using nitrogen and hydrogen peroxide as the coolant gases in order to make a direct comparison to the MkIII engine. Thermocouples were again welded to the chamber wall to provide information on the outer wall temperature. However the pertinent values for the MkIV engine are the coolant temperatures at the start and end of the flow separator.
Measurements of the actual flow separator temperature proved impractical due to the geometry of the engine.

5.8.1 Nitrogen Cooled Chamber

The initial experiments were only concerned with cooling of the chamber in a flight type manner and hence the V1c copper water-cooled nozzle was used. This removed the complication of possible nozzle burnouts. The same feed system as used in the MkIII case was used and the engine was set up in the MkIV Phase A configuration, discussed in §4.3.3. This time however thermocouples in the coolant flow, at the top and bottom of the flow separator, measured the coolant gas temperature and provide information on the heat flux into the coolant. With reference to figure 4.11, only thermocouples 0,1 and 2 were welded to the MkIV chamber. This was done to reduce the quantity of data recorded and remove superfluous channels. There was concern that the water-cooled nozzle could act as a heat sink, conducting heat into the coolant water. This would artificially reduce the recorded heat flux into the chamber. Thus two more runs were conducted with the water-cooled nozzle replaced with the RFC-SS nozzle, described in §4.2.5.

5.8.1.1 Procedure

The procedure for operating the engine was exactly the same as for the nitrogen cooled MkIII engine which is discussed in §5.7.1. With the V1c copper water cooled nozzle in place, a series of 'trim burns' were first executed to set the flow of the main HTP to 12 gs⁻¹, the fuel flow to 1.8 gs⁻¹ and the nitrogen coolant flow to 6.5 gs⁻¹. Once this was complete a burn of 10 seconds duration was performed. After the burn the data was analysed and the temperatures noted, the test was repeated with series of burns of increasing duration, finishing with a 60 second burn duration. The nitrogen flow rate was then systematically reduced to 5.5gs⁻¹ over a series of three further burns, again starting with a 10 second burn and working up to a 60 second burn. The V1c copper nozzle was then replace with the RFC-SS nozzle and the tests repeated. Using the same nitrogen flow rates of between 5.5 and 6.5 gs⁻¹ was not possible due to lack of cooling of the RFC-SS nozzle. Hence a higher flow rate had to be selected and a series of burns with the nitrogen flowing at 12.4 gs⁻¹ was performed.

5.8.1.2 Results

The data recorded showed that the chamber, with the water-cooled nozzle present, could be adequately cooled for a range of nitrogen flow rates between 5.5 and 6.5 gs⁻¹. Figure 5.12 shows a typical 60-second firing with the N₂ coolant flowing at 5.5gs⁻¹. Figure 5.13 shows the thermocouple data from the sensors located in the coolant flow at the start and end of
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Figure 5.12: Typical firing data of MkIV engine with N₂ cooling

Figure 5.13: Forward and Aft thermocouple measurements of MkIV N₂ cooled chamber
the flow separator for two different mass flows of nitrogen. Table 5.8 provides a summary of the recorded data.

<table>
<thead>
<tr>
<th>Run Number</th>
<th>N₂ Tank Temp / °C</th>
<th>Nozzle Type</th>
<th>N₂ flow rate / g s⁻¹</th>
<th>Aft Temp / °C</th>
<th>Engine O/F ratio</th>
</tr>
</thead>
<tbody>
<tr>
<td>1</td>
<td>15±2</td>
<td>V1c</td>
<td>6.5±0.1</td>
<td>380±10</td>
<td>6.8±0.2</td>
</tr>
<tr>
<td>2</td>
<td>15±2</td>
<td>V1c</td>
<td>6.3±0.1</td>
<td>384±10</td>
<td>6.5±0.2</td>
</tr>
<tr>
<td>3</td>
<td>15±2</td>
<td>V1c</td>
<td>5.8±0.1</td>
<td>445±10</td>
<td>6.7±0.2</td>
</tr>
<tr>
<td>4</td>
<td>15±2</td>
<td>V1c</td>
<td>5.5±0.1</td>
<td>501±10</td>
<td>7.0±0.2</td>
</tr>
<tr>
<td>5</td>
<td>15±2</td>
<td>RFC-SS</td>
<td>12.4±0.1</td>
<td>262±10</td>
<td>7.6±0.2</td>
</tr>
<tr>
<td>6</td>
<td>15±2</td>
<td>RFC-SS</td>
<td>11.5±0.1</td>
<td>296±10</td>
<td>7.8±0.2</td>
</tr>
</tbody>
</table>

Table 5.8: Summary of nitrogen cooled chamber data of MkIV engine

5.8.1.3 Discussion

With the provision of the thermocouples at the start and end of the flow separator, the temperature rise of the coolant could be measured and hence the heat flux in the parallel chamber section calculated. According to the theory of convective and radiative heat transfer, the power into the chamber section of the 40N engine of 18mm diameter and 50mm long should be of the order of 4,770 W, see §3.3.1. This includes the heat that is lost due to radiation from the hot chamber, and it also assumes the temperature of the copper insert is 1,250K as estimated from the design of the coolant spiral in §4.2.4. Knowledge of the actual temperature of the coolant spiral is not critical in determining the power into the chamber section due to the dominance of the radiation term. As this term is a function of the temperatures to the fourth power, even if the wall temperature was completely ignored it would only produce 5% difference in the estimated radiation power. The convective term has a far greater dependence upon the wall temperature, however this term is small in the chamber region and hence the estimate of the flow separator temperature is valid.

By making the assumption that the flow separator is at approximately 1,250K the empirical powers recorded in the experiments could be compared to the theoretical values. The power into the chamber is simply calculated by the product of the temperature rise of the nitrogen, its average specific heat capacity and the flow rate. The results of this analysis are shown in table 5.9.
### Table 5.9: Chamber power data for MkIV N₂ cooled chamber

<table>
<thead>
<tr>
<th>Run</th>
<th>N₂ Inlet Temp/°C</th>
<th>Nozzle Type</th>
<th>N₂ Final Temp/°C</th>
<th>N₂ Flow Rate/ gs⁻¹</th>
<th>Average Chamber Temp /°C</th>
<th>Empirical Power into Chamber/W</th>
</tr>
</thead>
<tbody>
<tr>
<td>1</td>
<td>15±2</td>
<td>V1c</td>
<td>380±10</td>
<td>6.5±0.1</td>
<td>280±10</td>
<td>2,790±150</td>
</tr>
<tr>
<td>2</td>
<td>15±2</td>
<td>V1c</td>
<td>384±10</td>
<td>6.3±0.1</td>
<td>278±10</td>
<td>2,734±150</td>
</tr>
<tr>
<td>3</td>
<td>15±2</td>
<td>V1c</td>
<td>445±10</td>
<td>5.8 ±0.1</td>
<td>332±10</td>
<td>2,933±150</td>
</tr>
<tr>
<td>4</td>
<td>15±2</td>
<td>V1c</td>
<td>475±10</td>
<td>5.5±0.1</td>
<td>329±10</td>
<td>2,783±150</td>
</tr>
<tr>
<td>5</td>
<td>15±2</td>
<td>RFC-SS</td>
<td>262±10</td>
<td>12.4±0.1</td>
<td>184±10</td>
<td>3,601±200</td>
</tr>
<tr>
<td>6</td>
<td>15±2</td>
<td>RFC-SS</td>
<td>296±10</td>
<td>11.5±0.1</td>
<td>202±10</td>
<td>3,605±200</td>
</tr>
</tbody>
</table>

The expected power into the chamber was approximately 4,770W and in the water-cooled cases on runs 1 through 4, a value as low as 57% of this was recorded. In runs 5 and 6, close to theoretical powers of 75% were obtained. The disparity in the recorded heat fluxes can be explained through the convective cooling experienced when testing at sea level. In runs 5 and 6, the higher flow rate of nitrogen caused a lower average exterior chamber temperature and hence lower convective losses. Consequentially the recorded power from the N₂ temperature rise is closer to the theoretical power. An estimate of the convective cooling losses can be made, if the theoretical chamber heat flux is correct. The surface area over which the forced convective cooling takes place is constant and the cooling can be expressed as the power lost per Kelvin. Equation 5.1 therefore defines the convective heat loss and 5.2 the definition of the coefficient.

\[
\dot{Q}_{\text{conv,loss}} = \psi \left( T_w - T_{\text{air}} \right)
\]  

\[
\psi = \frac{\dot{Q}_{\text{Ch,ther}}} {\dot{Q}_{\text{Ch,emp}}}
\]  

From equation 5.2, the convective heat loss coefficient can be calculated from the data in table 5.9 which gave an average value of 6.8±0.7 WK⁻¹. This produces an interesting result, as from this analysis it is possible to run the engine with no coolant flowing and still achieve an equilibrium temperature below the melting point of copper. From the theoretical analysis 4,770W of heat must be removed and using the convective heat loss coefficient, an equilibrium temperature of 701±72K above ambient temperature would produce the required degree of cooling. This was never tested due to the construction of the flow separator. Engineering judgement indicated that the knife edge of the aft end of the spiral would not survive without any coolant flowing. The

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MkIII engine due to its vertical mounting had a lower convective heat loss coefficient and hence would have reached equilibrium at a much higher temperature as indicated by the data recorded. The results show that over the five unique flow rates of nitrogen, hence five unique average chamber temperatures, the power lost due to convection can be accurately described by the convective heat loss coefficient. If this were not the case then the determined error would have been larger than the 10% error recorded. Likewise the estimate of the power into the chamber was also accurate.

5.8.2 Hydrogen Peroxide Cooled Chamber

The aim of this experiment was to show the chamber could be partially cooled using decomposed hydrogen peroxide. With the higher temperatures of the coolant, the convective losses will be consequentially higher and hence the measured power from the coolant temperature rise was expected to be considerably lower. From the previous analysis the minimum flow rate of coolant hydrogen peroxide is clearly zero due to convective heat losses.

The nitrogen supply was switched for hydrogen peroxide and the same MkIV Phase A configuration used. The temperature of the flow separator was not measured so just the thermocouples attached to the outside of the chamber and in the coolant flow were used to record the performance.

5.8.2.1 Procedure

The procedure for operating the MkIV engine with decomposed HTP was exactly the same as for the MkIII engine, which is discussed in §5.7.2.1. With the V1c copper water cooled nozzle in place, a series of ‘trim burns’ were first executed to set the flow of the main HTP to 12 gs\(^{-1}\), the fuel flow to 1.5 gs\(^{-1}\) and the HTP coolant flow to 7.4 gs\(^{-1}\). The engine was then burnt for 60 seconds. The engine was then run at lower coolant flow rates of between 6.0 and 3.7 gs\(^{-1}\), each run repeated twice with 100 second burn durations at an O/F of 7.5±0.5.

5.8.2.2 Results

Figure 5.14, below, shows the results of the burn with the coolant HTP set 4.2 gs\(^{-1}\) and with an engine O/F of 8:1. Three thermocouples were attached to the chamber were located 10mm from the either end of the parallel chamber section, dubbed top, mid and aft. Visual observations confirmed that symmetrical heating occurred as expected. Figure 5.15 below shows the temperature rise for the forward and aft thermocouples in coolant flow for the case where the coolant oxidiser is flowing at 4.2gs\(^{\prime}\). Table 5.10 shows the average chamber temperature derived from the three chamber thermocouples, for the coolant flow rates of between 2.9 and 8.0 gs\(^{-1}\).
Figure 5.14: Chamber temperature rise when cooled with HTP

Figure 5.15: HTP chamber coolant temperature rise
Table 5.10: Average chamber and coolant temperatures as a function of the HTP coolant flow rate

5.8.2.3 Discussion

Figure 5.14 clearly demonstrates the low temperature rise between the forward and aft temperatures. This is in sharp contrast to figure 5.13 where large differences were recorded. This is attributable to the convective heat losses as most of the heat is lost to convection, not through the coolant temperature rise. Knowing the convective heat loss coefficient, the power into the chamber can be calculated if the estimated heat capacity of the decomposed hydrogen peroxide is correct. Thus table 5.11 below shows the measured power from the coolant temperature rise, the expected heat lost to convection and the total heat flux expressed as a percentage of the theoretical power.

Table 5.11: Estimated power fluxes in the HTP cooled chamber

The data shows that once the convective heat losses have been taken into account the total power is close to theoretical and always within experimental error.
5.8.3 MkIV Chamber Cooling Conclusions

The MkIV engine demonstrated that the chamber could be adequately cooled with gaseous nitrogen and from this a convective heat loss coefficient was calculated. The nitrogen cooling experiments demonstrated the effectiveness of the flow separator as a low equilibrium chamber temperature could be readily achieved, in stark contrast the MkIII engine. The decomposed hydrogen peroxide cooling experiments confirmed the previously calculated convective heat loss coefficient. This coupled with the estimated heat capacity of decomposed hydrogen peroxide demonstrated that an accurate estimate of the heat fluxes in the engine could be established. Confirmation of the convective heat transfer coefficient of the flow separator was not performed due to the lack of temperature measurements of the flow separator itself.

This work paved the way for the most relevant part of the experimental work, the cooling of the nozzle. The nozzle being small has a small surface area to convect and radiate heat away. Thus the cooling issues are extreme in this location and importantly the forced convective cooling due to sea level testing would not mask the results to the same degree as in the chamber's case.

5.9 Flight Nozzle Experiments

Two types of potential flight nozzles were investigated, these being simple regenerative cooled nozzles and a radiation/film cooled versions. Details of the construction of the two types are given in §4.2.5. The use of a regeneratively cooled nozzle is the optimum solution as it potentially reduces the total quantity of propellant used for cooling and hence maximises the specific impulse. The radiation/film-cooled nozzle sacrifices the performance of a regeneratively cooled nozzle for the simplicity of construction. Thus both systems have advantages and were investigated. In all cases the MkTV engine was used to test both regenerative and radiation/film cooled nozzles, with propellant flow rates set to give approximately 40N of vacuum thrust.

5.9.1 Regeneratively Cooled Nozzle

The MkIV engine was never actually run in a regenerative mode but instead a simulation was attempted. Rather than feeding the propellant around the nozzle and back into the engine the output was to a collection vessel. This facilitated isolation of any potential problems with cooling the nozzle from the combustion chamber.

It was known from calculations that the V1c water-cooled nozzle would fail under low flows of the coolant fluid. This nozzle was designed to run with water at a flow rate in excess of 30gs$^{-1}$. In such a configuration it performed well, however if the flow rate was reduced to 15gs$^{-1}$, which represents the total flow of HTP in the engine, the nozzle would melt due to the convective heat transfer coefficient being too low. Thus a new nozzle was designed to operate at realistic
propellant flow rates, designated the V2a copper nozzle. However, useful data could be obtained through operation of the V1c copper nozzle at a high flow rate of HTP as a direct comparison with water. Subsequently the V2a copper nozzle was also attempted with both water and hydrogen peroxide.

5.9.1.1 Procedure

The MkTV engine was set up in the Phase B configuration (see § 4.3.3) with nitrogen used as the chamber coolant. The usual water feed into the V1c copper nozzle was also used. A thermocouple on the exhaust monitored the coolant rise throughout the burn. The test proceeded with a ‘trim burn’ to ensure the O/F was approximately 7.5. The water coolant flow through the cooling loop was trimmed to 16g/s. The firing sequence was immediately commenced and a burn of 60 seconds duration was executed. The test was then repeated for another 60-second repeat burn. The set up was then converted to accept the liquid HTP as the nozzle’s coolant and the flow rate adjusted to 28g/s to compensate for HTP’s reduced heat capacity. A burn of 60 seconds duration was again attempted. Upon shutdown, the coolant was allowed to flow through the nozzle to cool the copper for an additional 30 seconds. If all flows, including the coolant, were turned off simultaneously the stagnant HTP inside the hot copper nozzle may well have rapidly decomposed causing problems.

The V1c copper nozzle was removed and replaced with the V2a copper nozzle, specifically designed for low flow rates. The new V2a copper nozzle was first cooled with water and later hydrogen peroxide. This was performed in an identical manner to the V1c nozzle tests described above. The water flow rate was set to 11g/s and a burn duration lasting 60 seconds was executed. This run was again repeated for another 60-second burn. The MkTV set up was then modified to feed HTP instead of water into the nozzle. With these modifications complete and the HTP feed trimmed to 20 g/s, another burn lasting for 60 seconds was attempted. Deliberately high flow rates of the hydrogen peroxide were chosen for the first series of runs to avoid overheating. It was anticipated that upon successful results, the flow rate would then be reduced.

5.9.1.2 Results

In the V1c copper water-cooled case, as shown in figure 5.15, the temperature of the coolant rose to an equilibrium value as was expected over the 60-second burn duration. The O/F was 7.0:1 for this run and the water coolant flow was set at 16g/s with a similar result obtained with the repeat burn. When the water was switched for HTP as shown in figure 5.16, the same pattern was not followed. The temperature of the exhausted HTP rose as in the case for water, but upon reaching a temperature of 40°C the flow rate began to reduce and the recorded temperature
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Figure 5.15: Water-coolant temperature rise in V1c nozzle at 16gs⁻¹

Figure 5.16: HTP-coolant temperature rise in V1c nozzle
developed an oscillation and spiked. Once the flow had passed through this region, equilibrium conditions were reached and the remaining part of the burn continued without any oscillations. On grounds of safety the test was not repeated due to this event.

Tests with the V2a nozzle showed the nozzle worked well with water and similar data to figure 5.15 was produced, albeit with a higher exhausted coolant equilibrium temperature of 67°C. When this nozzle was tested with hydrogen peroxide the rapid oscillations seen on the V1c nozzle returned, however the system did not pass through this region, unlike the V1c nozzle. Figure 5.17 shows the temperature trace recorded of the burn. Using the V2a nozzle the HTP developed problems after 70 seconds of ignition. At this point the flow rate tailed off to a low level despite the pressure in the feed tank remaining constant.

**5.9.1.3 Discussion**

From the data recorded with the HTP cooling of the V1c nozzle, the temperature spike was either due to the decomposition of the HTP or extreme localised heating attributable to the nozzle design. It can be seen from figure 4.6 of the V1c water-cooled nozzle, that the HTP splits and rejoins as it passes around the throat. This raises the possibility that a vapour lock could occur in one of the channels, thus preferentially forcing HTP through the other channel. This would result in increased cooling on one side, but reduced cooling on the other, possibly raising the HTP to a high temperature where rapid decomposition would take place. The solution to the problem was believed to lie in the V2a nozzle. Using this nozzle, the HTP has only a single path to follow and
therefore cannot form vapour locks. However the V2a nozzle performed more erratically than the V1c nozzle. The data shown in figure 5.17 demonstrates a reduction of flow rate with time. The tank pressures were recorded to be constant and this implies the frictional losses in the cooling channels are increasing, causing a greater pressure drop. The lower flow rate correspondingly caused a greater temperature rise in the coolant and hence a positive feedback effect was set up. The conclusion from the V2a nozzle data was that decomposition was again occurring within the cooling channels of the nozzle, most probably due to the warm HTP on the copper, a known mild catalyst [LIQUID PROP 57]. This effect was more pronounced in the V2a nozzle than the V1c nozzle because of the narrow diameter cooling channels. In order to get the correct convective heat transfer coefficient the V2a coolant holes had to be fabricated with a diameter of 1.0 mm. Thus any decomposition of the HTP evolves oxygen, which readily impedes the flow of HTP in the narrow channels. However in the wide bore V1c nozzle, the effect was reduced. Thus it can be concluded that the use of a pure copper regeneratively cooled nozzle is not a practical solution with this design. However if the coolant channels were coated on the inside with a thin layer of pure aluminium a fraction of a millimetre thick, it is likely the nozzle would work. By removing the catalytic nature of the nozzle, the HTP will behave like water to a far high temperature and therefore would function as expected.

5.9.2 Radiation/Film Cooled Nozzle

As well as investigating the regeneratively cooled nozzle option, the film-cooled nozzle was also researched. It was found that the Haynes-230 alloy provides exceptional oxidation and high temperature resistance, yet the cost is similar to stainless steel. This allows a simple nozzle to be manufactured that in principle is compatible with oxidiser film cooling. As the procurement and fabrication of such a nozzle was predicted to take some considerable time, an identical geometry nozzle was manufactured from stainless steel. This would not be used with decomposed HTP due to the oxidation problems, but it could be used with nitrogen instead. This would give a useful insight into the gas film cooling problems, free from fuel combustion issues. Thus three nozzles were manufactured, a stainless steel nozzle designated RFC-SS and two Haynes-230 nozzles, designated RFC-230. These nozzles are discussed in detail in §4.2.5.

5.9.2.1 Procedure

In the initial set of experiments the MkIV engine was configured with the RFC-SS nozzle and the chamber coolant manifold fed with nitrogen. The engine was set up in the Phase C configuration described in §4.3.3. Thermocouples were spot welded to the nozzle in the locations shown in figure 4.13. In this configuration it was critical that the throat of the nozzle was visible during
firings and as such a mirror was placed on the far side of the engine to facilitate direct viewing of the complete throat. While the thermocouples gave temperature output of the throat, during data capture the actual values remained hidden from the user due to the method of acquisition. Hence the engine was run on a judgement basis. If the throat colour exceeded what was determined to be its limit the engine was shut down. After several runs, the colour and the actual temperature could be correlated and through experience a better appreciation of the temperature during the burn could be established. It still remained an imprecise method however, and only after the burn, when the data was analysed, could the exact temperature be determined. The engine was initially tested with an O/F set at 7.5 with the nitrogen flowing at approximately 4.5 gs\(^{-1}\). After a series of 'trim burns' to set up these parameters, two 10-second burns were attempted. If excessive temperatures were estimated at the throat during this period the engine was shut down prematurely. The tests were repeated at an increasing nitrogen flow rate of 8 and 10.5 gs\(^{-1}\). At every increase in coolant flow rate, a series of 'trim burns' had to be repeated to bring the O/F back to the required value. Once the Haynes-230 nozzle had been manufactured, the experimental set-up was reconfigured to accept the new RFC-230 nozzle and fresh thermocouples welded to it. The nitrogen feed system, for the chamber section was replaced with the HTP feed system and a series of 'trim burns' were performed to ensure the O/F was set at 7.5 and the coolant flow rate was at 8gs\(^{-1}\). On completion of the trim burns, two 30-second firing were attempted. Following this a series of tests designed to reduce the coolant flow rate were attempted. The coolant flow was reduced from 6.5gs\(^{-1}\) down to 2.5 gs\(^{-1}\) in a sequence of steps. At each new selection of coolant flow rates, a series of 'trim burns' were carried out to maintain the O/F at 7.5-8.0. Three runs to equilibrium were carried out at each flow rate, which required a 30 second burn.

5.9.2.2 Results

With the engine set at an O/F of 7.5 and the nitrogen film coolant flowing at between 4.5 and 8.0 gs\(^{-1}\) it became apparent that the nozzle was not being adequately cooled. Temperatures in excess of 700°C at the throat were recorded in less than three seconds of burn time. The data trend also showed that an equilibrium temperature of less than 1,000°C was unlikely to occur. When the nitrogen flow rate was increased to 10.5 gs\(^{-1}\), adequate cooling was achieved with an equilibrium throat temperature of 800°C. Figure 6.18 shows a summary of three flow rates of nitrogen tested, with the maximum thermocouple temperature shows for each flow rate. In the 4.5 and 8.0 gs\(^{-1}\) cases, the test was aborted due to excessive temperatures at the throat. Figure 5.19 shows all the thermocouple readings from the nozzle for a typical run with a nitrogen coolant flow rate of 10.5 gs\(^{-1}\).
Figure 5.18: Peak throat temperatures recorded for three unique N₂ flow rates

Figure 5.19: RFC-SS nozzle operating with N₂ film coolant at 10.5g/s⁻¹
Figure 5.20: Asymmetric temperature of the first RFC-230 nozzle prior to burn out

Figure 5.21: Burn out of the RFC-230 nozzle
Figure 5.22: Throat temperatures of RFC-230 nozzle during extended burn

Figure 5.23: Peak throat temperature plotted as a function of the coolant HTP flow rate
The initial tests of RFC-230 nozzle with the coolant HTP flow rate set at $8\text{gs}^{-1}$, resulted in the nozzle burning out after 15 seconds. The data for this particular run is shown in figure 5.20 and the cause of failure, attributed to asymmetric cooling, is discussed in §5.9.2.3. Figure 5.21 shows an image of how the nozzle failed. The second RFC-230 nozzle with the new flow separator performed far better and adequate cooling with a HTP flow rate of just $2.5\text{gs}^{-1}$ was achieved. Figure 5.22 shows the thermocouple data for the throat during a 30 second burn with the coolant flowing at $6.5\text{ gs}^{-1}$. Figure 5.23 shows the peak throat temperature as a function of coolant flow rate. The error bars on this chart were chosen to encompass the spread of temperatures recorded on the three runs with the same coolant flow rate.

5.9.2.3 Discussion

The data from the nitrogen cooling experiments demonstrated that cooling using a gas is possible in this configuration, but a large mass flow was required to achieve this. There is also a lack of symmetry around the nozzle’s throat. There was approximately $200^\circ\text{C}$ difference between the highest and lowest values recorded which indicates asymmetric coolant distribution. While this was not a problem in terms of engine cooling, it could be an issue in terms of the alignment of the thrust vector. If more coolant is travelling down one region as compared to the diametrically opposite region, then the flow will be vectored off axis. It is known from the test data that the temperature of the injected N$_2$ coolant into the nozzle region is only at $250^\circ\text{C}$ at equilibrium. This introduces a large temperature difference between the coolant and the hot combustion gases. A velocity and temperature difference between two parallel flows can cause Kelvin Helmhotlz instability [BETCHOV 67], which leads to turbulent mixing. The density of the nitrogen at the chamber pressure and recorded temperature is $13.1\ \text{kgm}^3$, which at the flow rate of $12.5\text{gs}^{-1}$ leads to an injection velocity of $93\text{ms}^{-1}$. The main flow of combustion products will however have a velocity of $25\text{ms}^{-1}$, thus creating a significant velocity gradient. With HTP as the coolant, the gas temperature is much higher and hence there will be a lower temperature difference leading to less turbulence.

The failure of the first RFC-230 nozzle can be attributed to highly unsymmetrical cooling of the nozzle as can be seen from figure 5.20. There was over $300^\circ\text{C}$ difference between opposite locations on the throat and this is believed to be due to the damaged flow separator. From the image shown below in figure 2.24, the injection ports of flow separator are distorted. This is believed to be due to excessive temperatures and mechanical damage during handling. It can be seen in figure 5.21 that the throat region has become slightly eroded during the burn, which most likely caused massive turbulence down stream, destroying any residual film cooling. This allowed the hot combustion gases to directly impinge on the metal in the expansion cone and
hence the wall melted through. The burn through point occurred slightly down stream of the location where the failed thermocouple was located, thus giving no data on the actual temperature at which it failed. From the data that was recorded, there was a large spread of temperatures around the throat, but importantly a quasi-equilibrium state had been reached. The lowest temperature recorded was only 730°C while the highest, diametrically opposite, was 1,031°C. It can be deduced from the fact that the burn through did not occur at the 1,031°C location that the temperature at that burn point was in excess of this and most likely in excess of the nominal working limit of 1140°C [HAYNES 02]. The results suggested that if an even distribution of coolant could be maintained around the nozzle then cooling with HTP is a real possibility. This hypothesis was made on the assumption that the average temperature of the thermocouples would be the temperature recorded with an even coolant flow.

Once the flow separator had been replaced the new RFC-230 nozzle worked extremely well with the decomposed HTP coolant adequately cooling the engine down to 2.5 gs⁻¹. There was however, a degree of asymmetric cooling and this is believed to be due to the design of flow separator. As a single helix was used, it is possible for coolant to be preferentially introduced into one area of the nozzle, starving the opposite side. This could be rectified by manufacturing a 4 helix design which would produce a more even coolant distribution. Naturally in a vacuum the minimum flow rate would have to be in excess of the 2.5gs⁻¹ to prevent the throat from melting, however the area over which convection occurs is small, only 1.2×10⁻³ m² and from the
convective heat transfer equation calculated previously the power lost is merely 150±24W and correspondingly the flow increase will also be small. Thus it can be said that the radiation film cooled nozzle tests are similar to vacuum conditions.

5.9.3 Flight Nozzle Conclusions
The results from the V2a and RFC-230 nozzles show that for a simple bi-propellant system the film cooled option is superior in terms of simplicity and performance. While the V2a nozzle worked well with water, the use of HTP was prohibited by the catalytic nature of the copper upon the oxidiser. Alternatives to copper such as aluminium would not be able to retain their strength at the high temperatures required which leaves the RFC-230 nozzle as the ideal candidate. Providing symmetrical coolant distribution is achieved, the nozzle worked well to flow rates as low as 2.5 gs⁻¹ and this combined with its inherent simplicity provides an elegant solution. In vacuum operation, the chamber will reach a higher temperature due to the absence of the convective cooling. This results in an anticipated coolant temperature of 877°C entering the nozzle as opposed to 600°C as was experienced in the sea level tests. Thus it can be seen that the temperature rise of the nozzle was 375°C above the injection temperature in the case where the nozzle was cooled with a flow rate of 2.5gs⁻¹. Thus it can be expected that if this flow rate was used in a vacuum, the nozzle would reach approximately 1,250°C, thus resulting in the nozzle destruction. Hence the minimum vacuum flow rate, determined from figure 5.23, would be 3.5 gs⁻¹. Compensating for the small convective loss, results in an expected flow rate of 3.8gs⁻¹.

5.10 Engine Cooling Conclusions
The research has demonstrated that both the chamber and nozzle of a small bi-propellant rocket engine with a low cost injector can be cooled using decomposed hydrogen peroxide when tested at sea level. The highlight of the research was the demonstration that the RFC-230 nozzle could be kept well within its maximum service temperature with a hydrogen peroxide coolant flow rate of only 2.5gs⁻¹.
From the theoretical model, the empirical convective cooling heat transfer coefficient was determined for the MkIV engine using nitrogen as the coolant. When the nitrogen was replaced with decomposed HTP the convective cooling issue became more severe. However the empirical transfer coefficient still provided an accurate estimation of the heat lost due to convection and the calculated heat fluxes agreed well with the theoretical analysis. Applying the data to vacuum conditions, an estimate of the overall performance can be made. The theoretical flow rate of hydrogen peroxide required to cool the chamber section, when allowed to rise from 800K to 1,150K (100K temperature difference to the copper flow separator)
is 7.2 gs\(^{-1}\). This is higher than the value calculated in chapter 3 due to the introduction of the flow separator. There has to be a temperature difference between the flow separator and the coolant gas for convective cooling to take place. The flow separator has an upper limit of 1,250K and hence with the design of flow separator, the peak coolant temperature can only be 1,150K hence slightly reducing the performance. The performance can be increased through the use of a flow separator with a tighter helical spiral and hence higher convective heat transfer coefficient. In such design a lower temperature difference between the flow separator and the coolant would still maintain the correct heat flow. This allows a greater temperature rise of the coolant before the wall melts.

The results suggest that the RFC-230 nozzle would be adequately cooled in a vacuum with a flow rate above 3.8 gs\(^{-1}\). Thus the overall system, chamber and nozzle, could be cooled with minimum flow rate for the chamber of 7.2 gs\(^{-1}\). Assuming 92% efficiency of combustion, the estimated vacuum Isp for this engine is 252s. However the density Isp of 343s is comparable to the LEROS-LTT engine, demonstrating the impressive performance that can be achieved in a volume constrained spacecraft.

Confirmation of these performance figures requires vacuum firings, which would be particularly useful for determining the effectiveness of the chamber cooling aspect. While the theory indicates that in vacuum the temperatures would be kept within limits, the convective heat transfer of the flow separator needs to be established, not just theoretically. This requires embedded thermocouples in the flow separator itself and a vacuum firing capability, both of which were not available for this project.
Chapter 6

6 Hydrogen Peroxide Tank Research

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6 Hydrogen Peroxide Tank Research

6.1 Introduction

This chapter presents the investigation into possible tank materials applicable for the on orbit storage of hydrogen peroxide. The specific problems with long-term storage are discussed and possible materials identified. Two distinct philosophies for extended on orbit storage are considered; extreme inertness or ability to vent in a zero g environment. The experimental procedures to determine the most suitable candidates are described along with a description of the experimental apparatus. These tests involved decomposition rate determination, mechanical pull tests and chemical stability investigations using the X-Ray Photoelectron Spectroscopy technique. Finally a flight-type passively vented tank was tested and the results presented.

6.2 Tank Design and Philosophy

The vast majority of applications that have used hydrogen peroxide can be classified as launch or limited lifetime vehicles [McCORMICK 65]. When a time delay between filling the vehicle with propellant and actually using it is short, as in the launch vehicle case, the tank issue is trivial. It is well known that passivated low copper aluminium alloy such as the 1060 grade makes a good tank material in terms of compatibility [PEROX HANDBOOK 67]. However, over extended periods of the order of months, the pressure will rise significantly within the tank if no venting is allowed to occur. In the launch vehicle case, the HTP simply does not have enough time to cause problems through a pressure rise [ANDREWS 90]. In a tactical missile case, the vehicle needs to be stored for a long period and as such a vent valve can be installed to remove excess oxygen [PAULS 99]. The HTP concentration will decrease as a result of long-term storage but this consequence is not as important as the pressure rise [WHITEHEAD 99]. In the case of a satellite that is expected to stay on orbit for several years, the issue is more complex. There are two distinct methods that can be applied to reduce or stop the pressure rise rate of the tank. The first method is to use an exceptionally clean and well passivated propellant tank with a large ullage volume. Decomposition will occur, albeit at a low rate, and when coupled with a large volume into which the oxygen can accumulate the result is a low rate of pressure rise. This method is not particularly desirable due to the wasteful use of spacecraft volume. The principal advantage of hydrogen peroxide is the high density and hence high density Isp [WU 99] as demonstrated in chapter 3. Deliberately increasing the propellants tank's volume kills the effective density, resulting in a system that could be no better than a nitrous oxide system, see §3.2.3. The alternative philosophy is to use a vented tank. In such a system, there would be a high fill fraction
and the evolved oxygen would be vented. In a zero g environment it is imperative that the vent valve separates liquid from the oxygen, otherwise it would be possible to loose a considerable volume of propellant in an 'oxygen dump' exercise. Typically liquids are separated from gas in a zero ‘g’ environment through a sponge and vane Propellant Management Device [TAM 96]. This method is not desirable for a HTP system as the large surface area promotes decomposition. However, an alternative novel method of tank venting was proposed for this project. The principal decomposition of hydrogen peroxide occurs on the oxidiser/tank interface [PEROX HANDBOOK 67], generating oxygen bubbles against the tank liner. Through the use of a liner that is permeable to the oxygen but not the liquid, a passive relief valve can be created. The pressure in the tank will therefore increase to a pressure where the permeation rate is equal to the decomposition rate. This pressure can be selected by the thickness and composition of the liner. The tank venting method still requires the tank material to be extremely compatible with the hydrogen peroxide. The main advantage of such a system is it does not require valves to accomplish the venting as it constantly bleeds oxygen at a low rate, thus can be expected to be very reliable. The inevitable side effect is a wasteful use of the pressurant. When a propulsion burn is required, the tank would be pressurised and the engine fired. After the burn had been executed, the pressurant would be isolated from the propellant tank and the pressure in the tank would therefore decay. At the next firing the same process would have to be followed. Compared to a hydrazine system where the tank is pressurised once for a firing and then maintained at that pressure over its lifetime, the HTP system looks wasteful. However as HTP would be used in an orbital manoeuvring engine, the number of times the tank is pressurised is low, and thus will have little impact on the overall propulsion system volume.

6.3 Coupon Sample Testing

To assess the possibility of either pressure control method, a range of candidate materials were identified. Once the candidate materials were chosen, a decomposition test with the HTP was performed. A pressure vessel containing the coupon sample of the tank material and hydrogen peroxide was hermetically sealed. The resulting evolved oxygen from the decomposition pressurised the entire system. By monitoring the pressure via a sensor, the decomposition rate could be established. This direct method of decomposition measurement was chosen for simplicity and low cost. Ideally microcalorimetry would be the preferred option due to its safety and sensitivity [DAVIS 00] but this fell outside of the constraints of the research programme.
6.3.1 Choice of Candidate Materials

From the literature survey, low-copper aluminium alloy tanks were the preferred option for storing hydrogen peroxide [PEROX HANDBOOK 67]. However typically the alloy as in the case of 1060 Al is only 99.6% pure aluminium. It has been noted that pure aluminium is not a catalyst while copper is [PEROXIDE 55] and as such it was expected that pure aluminium would produce a lower decomposition rate than 1060 Al. Pure aluminium cannot be used as a tank material alone, principally because the material’s ultimate tensile strength is too low to be used as a pressure vessel. However pure aluminium can be coated onto the inside of a high strength Al alloy, or Titanium tank. This would create a compatible surface coating yet still retain the strength and hence lightness of the outer shell material. The literature failed to show any evidence of tests conducted with extremely pure aluminium. 99.999% pure Al can readily be procured from chemical suppliers [ALDRICH 00] and was chosen as a potential candidate material.

In addition to pure aluminium, the fluoropolymers PFA and FEP were also chosen. Most of the work accomplished in the 50’s and 60’s pointed to the fact that PTFE was a good material, but not as compatible as 1060 aluminium [ANDREWS 90]. FEP was also available at this time and this produced similar results to the PTFE [PEROX HANDBOOK 67]. PFA however was not commercially available, so this can be regarded as the next generation of fluoropolymers. While there are many more fluoropolymers that can be considered, there is considerable experience in moulding the FEP and PFA thermoplastics, which lends credibility to a practical design [HOLSCOT 00]. In addition both polymers have the physical property of being permeable to oxygen. However, easily condensable liquids such as Water and hydrogen peroxide tend not to pass through [RICHARDS 73][DuPONT 02]. This makes PFA and FEP ideal candidates for the HTP storage research. Thus the three materials considered for this research were PFA, FEP and pure aluminium. Kitchen foil grade aluminium was also investigated as a possible replacement to research grade pure aluminium.

6.3.2 Experimental Apparatus

The basic experimental set up consisted of a borosilicate glass pressure vessel, made by Fischer Porter, which will be referred to from now on as a Fischer Porter (FP) bottle and a test sample of tank material. The sample of polymer used in the experiments had a surface area of 59.6±0.5 cm². The volume of HTP used to submerge the samples was 126±10cm³ giving a surface to volume ratio of 0.47cm⁻¹, equivalent to a spherical 1.1 litre tank. The aluminium tests used samples of 50.0±0.2cm² surface area, giving a similar surface to volume ratio. Borosilicate glass has been shown to be very compatible with hydrogen peroxide [PEROXIDE 55] making it an ideal storage
vessel. The FP bottle was kept in a constant temperature water bath such that constant decomposition rates could be observed. Hydrogen peroxide and the coupon sample of choice could be introduced into the FP bottle and the system sealed. As decomposition occurred the subsequent pressure rise was measured through a transducer located a distance from the pressure vessel, linked via a PFA plastic tube. Shown above is a schematic of the system in figure 6.1. The apparatus included four such sets of pressure vessels to allow simultaneous measurement of four samples. To provide an accurate pressure rise rate, each system must be shown to be leak proof and this was accomplished through a test pressurisation using an external oxygen supply. In addition to a pressure transducer, a dial gauge was also included to allow manual reading of the pressure inside the vessels. It was determined that a transducer failure may instil a false sense of security, as the pressure may in reality be dangerously high. The length of PFA pipe leading from the clamped bung in the top the FP bottle was deliberately chosen to reduce the risk of decomposition in the pressure delivery line. As the bath remained warm (32°C) while the ambient air is typically around 10°C, there was a tendency for the HTP and water vapour to condense on the cold pipe. If a steel pipe was used then significant decomposition would occur, producing erroneous results. However using cleaned PFA pipes, this effect was minimised. As there was a significant safety risk with pressurised hydrogen peroxide in glass vessels, the bath containing the FP bottles was located inside the shipping container, at the Westcott test site, and the dial gauges and valves positioned outside. This allowed any vessels that exceeded the 6 Bar limit to be safely vented without exposing personnel to a potentially dangerous situation.
The biggest concern regarding the experimental set up was the number of joints presenting possible leak paths. While every effort to reduce the number was attempted, joints were unavoidable. With a system that has a leak rate on the same order of magnitude as the decomposition of hydrogen peroxide, the actual decomposition rate would be significantly masked. Thus it was important to reduce the leaks to an absolute minimum and also determine the magnitude of the residual leak rate. The method used to perform this is discussed in §6.3.5.1.

The pressure build up in the vessels was artificially reduced by the pressure loss due to the leaks. $f_1$ is defined as the pressure rise rate in Bar/day and $f_2$ is the leak rate, defined as the pressure lost per Bar of pressure per day. Hence the overall change in pressure is given by equation 6.1.

$$dP = f_1 dt + f_2 P dt$$  \hspace{1cm}  (6.1)

Solving this integral and rearranging the equation results in the following equation.

$$P = \frac{f_1}{f_2} \left(e^{f_2 t} - 1\right)$$  \hspace{1cm}  (6.3)

However, the actual decomposition rate would produce a pressure rise in a totally sealed system of $P = f_1 t$. Hence the ratio of the true pressure to the compensated pressure is;

$$C_{FP} = \frac{f_2 t}{(e^{f_2 t} - 1)}$$  \hspace{1cm}  (6.4)

Hence $C_{FP}$ is the pressure compensation factor. Thus by knowing the leak rate and applying this correction factor, it was possible to arrive at the true decomposition rate, as if the system was completely hermetically sealed.

6.3.3 Data Logging System

The pressures inside the four FP bottles were measured using 0-6 Bar pressure transmitters supplied by RS components. These devices produced a 4-20mA current signal, allowing the data to be transmitted over significant distances. The 486 PC equipped with the data acquisition card resided in the control room at the Westcott site and cables ran outside to the HTP test apparatus. Along with the pressure measurements, two thermocouples immersed in the water bath provided data on the bath temperature. This was nominally maintained at 32°C using a 100W thermostatically controlled fish tank heater. The PC was equipped with an uninterruptible power supply to prevent data from being lost in the event of power failure. In addition the data was automatically saved periodically on the PC every 10 minutes to prevent large data losses due to a
computer crash. The data acquisition card was driven by the LabVIEW software produced by National Instruments with custom made acquisition code. Calibration and data handling was performed from the PC and once configured, could be left to autonomously collect data indefinitely.

6.3.4 Passivation Procedure

Hydrogen peroxide will decompose readily on any organic contamination and a wide range of metals. As such it was vital that the surfaces of both the FP bottles and the coupon samples were cleaned and passivated to as high a standard as possible. From the literature survey it was found that a satisfactory passivation procedure had been established and it was decided to adopt this process [McCORMICK 65] [PEROX HANDBOOK 67], see §2.5.3. The coupon sample of the test material and also the FP bottle itself required passivation and the most effective way to accomplish this was to place the sample inside another FP bottle and passivate the two FP bottles plus contents together. Then when it came time to use the coupon sample, it was removed from the passivated environment and placed in the test FP bottle.

Initial cleaning was performed in the Chemistry department by filling each bottle with Decon 90 (10% solution) at 60°C and agitating periodically over an hour. The solution was then poured away and the bottle rinsed out with copious quantities of distilled water until no further soap bubbles were formed. The bottles were then filled with 5M Nitric acid at room temperature and allowed to stand for 1 hour. The acid was then poured away and the bottle again flushed with distilled water. The bottles were then bagged up inside sterile grade plastic bags and transported to the Westcott site for propellant conditioning. On arrival the bottles were filled with 90% hydrogen peroxide and allowed to stand for 1 week at room temperature. The bags were placed over the neck of the bottles to reduce contamination. The hydrogen peroxide was introduced into the FP bottles using an all PTFE/PFA feed system that was thoroughly flushed through with the propellant to avoid metallic ion contamination. After one week had expired the HTP in the bottles was then decanted off and the FP bottles dried on a radiator. At this point passivation was then complete.

6.3.5 Experimental Work

The experimental work was performed on seven samples, four fluoropolymer and three aluminium samples. Leak tests were performed upon the apparatus prior to actual hydrogen peroxide tests, which were conducted over a period of two days while the actual live tests each took place over 15 days. As the apparatus contributes to the overall decomposition rate, it was important to record the 'blank' rate (dP_{blank}/dt) prior to the immersion of the sample. Using such a
scheme, the increased rate due to the surface area of the sample could be calculated. The blank decomposition rate was known, as was the rate with the immersed sample \( \frac{dP_{\text{blank}}}{dt} \). The difference between these values was due to the surface area of the polymer samples causing decomposition. A useful way of expressing the decomposition rate was in terms of the volume of oxygen at ST&P evolved per year per square centimetre of the coupon sample. This figure does however assume that there is zero decomposition if the surface area decreases to zero and hence no auto-decomposition. This is not exactly the case, but as auto-decomposition is small, it can be ignored [PEROX HANDBOOK 67]. In order to calculate the value, knowledge of the ullage volume was required and this was determined by theoretically analysing the pipe work. The calculated volume was 104±8 cm³. This value had a large error mainly because of the inaccuracy of filling the bottles with ±10g of hydrogen peroxide and problems with measuring the volume inside bends of the pipe work. Using equation 6.5 it was therefore possible to calculate the decomposition rate due to the surface area of the samples

\[
\text{DecompRate} = \frac{dP_{\text{blank}} - dP_{\text{blank}}}{dt} \frac{V_{\text{ullage}}}{A_{\text{sur,samp}} P_{\text{am}}}
\]  

(6.5)

6.3.5.1 Procedure

The initial test performed was the oxygen leak check. With the FP bottles empty of HTP and dry, an oxygen supply through a regulator, was connected to the relief valve/pressurisation port and the regulator increased to pressurise the system to 6 Bar. The relief valve was then closed and the oxygen supply removed. The pressure in the system was then monitored over a period of 2 days to determine the leak rate. Through the use of PTFE paste on the ferules used in the fittings, leak rates were reduced to a satisfactory low value. Similarly the PFA components that join the pipe work to the Fischer Porter bottle had to be bonded using epoxy resin for the same reason.

Upon completion of the leak test, the next step was to record the background decomposition rate of just the apparatus without any samples present. The passivated sample bottles were filled with 126±10cm³ of 90% hydrogen peroxide and connected to the rest of the system. As the initial test was a control, no material samples were used. Once all the bottles were placed in the water bath, the immersion heater was turned on. The set up is shown in figure 6.2. Over a period of two hours the temperature stabilised and the relief valves were then closed, creating a sealed system. The logging software was started and the apparatus left for a period of 15 days. This value was
chosen as the empirical leaks are small over this period yet a trend line could easily be fitted with the quantity of data generated. The apparatus was observed periodically for any excessive pressure rises, or the water bath running dry. After 15 days, the test was terminated and the system vented. The passivated polymer coupon samples were then removed from their respective bottles and placed inside the FP bottles containing the HTP. The original bottles, now containing a coupon sample and the HTP, were resealed into the system again and the test was repeated. The vents were sealed again and the logging system started for a second period of 15 days. Careful monitoring of the apparatus was again essential to prevent unwanted failures. At the end of this second period the system was again vented and the data examined.

The whole experiment was repeated with the aluminium samples in place of the polymer samples. The experiment proceeded from the initial measurement of the leak rates of the apparatus and passivation through to tests with aluminium samples submerged in the HTP. In the latter case, three samples of aluminium were used. A high purity 99.999% pure aluminium sample and two lower purity commercial household foil grade samples were chosen. The temperature of the water bath, remained constant at 32.0±2.0°C throughout these experiments while the heater was on. Unfortunately during the ‘blank’ test the heater tripped out causing a period where the bath was at an ambient temperature of approximately 15°C.
6.3.5.2 Leak Test Results

The following data presents the results of the leak test carried out prior to the polymer and aluminium sample tests. Figure 6.3 shows the leak rates for the four separate FP bottle systems used for the polymer tests. Figure 6.4 shows similar leak data for the aluminium tests, however P3 was not used to acquire experimental data.

![Figure 6.3: Leak test results for polymer apparatus](image)

![Figure 6.4: Leak test results for Aluminium apparatus](image)
6.3.5.3 PFA/FEP Test Results

Shown below in figure 6.5 is the data recorded for the ‘blank’ runs of the apparatus subsequently used for the tests of the PFA and FEP samples. The data shown has been compensated for the small but significant leaks of the system by applying leak compensation factor through equation 6.4. The linear trends produced indicate that the decomposition rate is approximately constant with respect to time.

The figure 6.6 shown below, gives the pressure rise with time of the actual samples immersed in hydrogen peroxide. This test took place over a 15-day period and again the data is corrected for leaks through equation 6.4. While the data is not precisely linear it is clear that they are following a general trend, with a slight variation superimposed upon the pressure rise rates. Crucially all four curves deviate from the straight lines in a similar way which implies the temperature of system is varying slightly thus causing a change in decomposition rate and ullage pressure. With both sets of data recorded it was now possible to calculate the decomposition rate due to just the polymer sample, which is shown in table 6.1. The pressure rise due to just the apparatus is described as \( \frac{dP_{\text{blank}}}{dt} \), while the decomposition due the sample and also the apparatus is labelled \( \frac{dP_{\text{sample}}}{dt} \).

![Graph showing pressure rise over time](image-url)
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6.3.5.4 Aluminium Test Results

The ‘blank’ decomposition test data is presented below in figure 6.7, compensated for leaks. Unfortunately part way through the 15-day test, the immersion heater circuit breaker tripped out causing the temperature of the bath to fall. This lowered the decomposition rate and reduced the pressure in the ullage volume. Once this was realised, the heater was manually reset and the test continued as per normal. The erroneous part of the data is clearly visible in the graph, however
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Figure 6.7: Pressure rise for ‘blank’ HTP tests (II)

Figure 6.8: Pressure rise for peroxide in contact with aluminium
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the overall trend remains unaltered and an accurate background rate was still established. Figure 6.8 shows the data recorded of the pressure rise due to the decomposition upon the aluminium samples, again compensated for the leaks in the system.

Using equation 6.5, the rate in terms of volume of evolved oxygen at ST&P per square centimetre of sample per year was calculated and this data is shown below in table 6.2

<table>
<thead>
<tr>
<th>Sample</th>
<th>( \frac{dP_{\text{blank}}}{dt} )</th>
<th>( \frac{dP_{\text{sample}}}{dt} )</th>
<th>Decomposition Rate cm(^3)year(^{-1})cm(^{-2})</th>
</tr>
</thead>
<tbody>
<tr>
<td>P1 Household Foil</td>
<td>27.70</td>
<td>35.61</td>
<td>16.0±1.3</td>
</tr>
<tr>
<td>P2 Household Foil</td>
<td>24.93</td>
<td>32.12</td>
<td>15.9±1.3</td>
</tr>
<tr>
<td>P4 99.999% Pure</td>
<td>18.83</td>
<td>24.35</td>
<td>11.5±0.9</td>
</tr>
</tbody>
</table>

Table 6.2: HTP in contact with aluminium decomposition rates

6.3.5.5 Discussion

Leak Mitigation Tests

Sealing of the system to reduce the leak rates to an absolute minimum required considerable effort and the data presented in figures 6.3 and 6.4 was the result of several months of work into leak mitigation. It was noted on the leak test performed prior to testing the polymers that the system labelled P3 exhibited an increased leak rate as compared to the other FP bottles. Despite extensive endeavours, the source of this leak remained elusive. This was eventually deemed acceptable due to the relatively low leak rate and the ability to compensate for leaks according to equation 6.4.

When the leak tests were again performed on the system, this time prior to the aluminium samples the P3 apparatus suffered an excessive leak rate and as such was inadequate for the purpose of a long-term test. The most likely reason for this failure was the removal and subsequent reattachment of the pressure transmitters. These transmitters were required for another task and it transpires that upon reconnection, the P3 system did not seal properly. This was not an issue however as only three samples were on test, thus the P3 system was redundant. The remaining systems, P1, P2 and P3 all recorded similar leak rates to each other and the previous leak test.
**Polymer Sample Tests**

All the polymer samples exhibited a similar decomposition rate, with the two PFA polymer samples recording both the highest and lowest rates. The difference is most likely to be due to slight variations in the cleanliness of the samples and apparatus. It was however apparent that all the polymers gave similar decomposition rates. This places confidence in the experiment, as a large difference on one of the samples would indicate improper cleaning. An estimated AOL was determined from this data, corrected for temperature, and for the sample sizes used the AOL was 1.9%. This is similar to values recorded in table 2.3, however the surface to volume ratio was not specified in the source data and thus direct comparisons cannot be made.

**Aluminium Sample Tests**

It can be seen from the graphs that the linear trend with time was not as apparent as the polymer case. All traces initially rise faster than the polymer samples for a short period, but then reduce. As all traces are equally effected this was again determined to be due to a temperature effect upon the system. While the water bath was designed to maintain a constant temperature, it transpired that the thermostat had temporarily failed, causing the bath temperature to rise to approximately 40°C. After a period of 6 days the heater returned to normal operation and the test continued. Despite these factors the data was good enough to fit a straight line to and obtain decomposition rates for the second part of the graph. The results show that the pure aluminium is superior to the polymer samples and the commercially available household foil. The increase in decomposition rate in the latter is most likely due to the higher copper content. In addition directly observing the pure aluminium gave an indication of its compatibility as the household aluminium, over a period of days, attained many small bubbles attached to its surface. In the pure aluminium case, this did not occur except for at the edges of the 50×50mm sample.

**Conclusions**

The tests performed on the PFA, FEP, pure aluminium and household foil grade aluminium showed that high purity aluminium is superior to the other materials. This was not an unexpected result considering the literature review had suggested this from the onset. However aluminium does not allow for the passive venting discussed earlier, so while the compatibility of the polymers are lower than pure aluminium, the former material may actually be of more use in a flight system.

The data shows that PFA and FEP polymers have a similar decomposition effect upon the HTP. Both polymers evolve approximately 17-20cm³ of oxygen at ST&P for every square centimetre of
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tank material and year of exposure. This compared to the pure aluminium of just 12.8 cm³ year⁻¹ cm², demonstrates that the polymers are approximately 1.5 times worse than the aluminium. The household aluminium is similar to the polymer cases, with both evolving around 17-20cm³ of oxygen per square centimetre per year. To put the data into perspective, a 10 litre spherical tank was considered with an ullage volume of 15%. The wetted surface area of such a tank is 2,245cm² and the tank is assumed not to vent during this period. The table 6.3 below demonstrates the pressure rise that would be expected in such a tank after 1 year on orbit at a constant temperature of 32°C. This is higher than a spacecraft would be designed to run, but it is not uncommon for these temperatures to be maintained for extended periods.

<table>
<thead>
<tr>
<th></th>
<th>PFE or FEP</th>
<th>99.999% Aluminium</th>
</tr>
</thead>
<tbody>
<tr>
<td>Pressure/Bar</td>
<td>25.7±2.2</td>
<td>19.2±1.5</td>
</tr>
</tbody>
</table>

Table 6.3: Pressure rise in a realistic spacecraft tank

A 19.2 Bar increase per year in the case of the pure aluminium tank is tolerable for a short term spacecraft mission of 3 months, especially when operated in a blow down mode. Indeed in such a case where the tank is pressurised initially, the decomposition of the HTP with time helps to maintain tank pressure as propellant is expelled. A 25.9 Bar increase over a period of year in the polymer case is higher than the aluminium, however with the ability to passively vent the oxygen through the tank walls an equilibrium pressure that is within bounds may be feasible. Thus the conclusion to be drawn from this section of the research is that a pure aluminium tank is acceptable for missions of up to 3-6 months, however after this, a polymer tank looks more attractive due to its intrinsic venting capability. In either the case a stress bearing shell is required which would be made from titanium or another high strength but inevitably incompatible material.

6.4 Polymer Permeation Tests

In the permeable tank concept, it is desired that the tank liner polymer is permeable to oxygen but not to the liquid HTP. The literature shows that liquids and easily condensable vapours when presented with a polymer of dissimilar functional groups to the liquid, will have greatly reduced permeability [RICHARDS 73]. Thus it can be expected that FEP and PFA will have limited permeability with hydrogen peroxide. As well as chemical permeation through a material there is also mechanical porosity, due to microscopic holes in the polymer. FEP and PFA being melt
processible thermoplastics are virtually devoid of micro-holes and permeation should only occur through the chemical mechanism. PTFE however, has a porous structure and as such is prone to mechanical porosity as well as chemical permeation [FLUOROPOLYMER 97]. From the literature there is plenty of data regarding the permeability of oxygen and other gases, however permeability data of concentrated hydrogen peroxide is non-existent. Thus the polymers were investigated in terms of both liquid hydrogen peroxide and oxygen permeability, the latter to test for similarity with the literature. If the liquid HTP permeation rate is significant, then the material in question cannot be used as a tank liner, however a small rate is tolerable. It was not expected that there would be significant issue however, this is based on the fact that FEP and PFA are readily used to line water tanks, without any concern. The International Space Station (ISS) uses PFA lined water tanks demonstrating this point [HOLSCOT 00]. Hydrogen peroxide, as far as the polymer is concerned, is similar to water and hence a similar permeation rate would be expected. To empirically demonstrate the permeation rates, a series of experiments were conducted using oxygen and hydrogen peroxide as the permeants.

The permeability of FEP and PFA to pure oxygen at room temperature is well documented and follows a relationship where the rate is proportional to the pressure differential across the membrane. The table 6.4 below shows the published permeation rates for various gases [WESTERN 01]. The data is given in terms of the polymer thickness in millimetres.

<table>
<thead>
<tr>
<th>Gas</th>
<th>Permeability @ 25°C</th>
</tr>
</thead>
<tbody>
<tr>
<td></td>
<td>FEP / mm cm⁻² m⁻² 24 h⁻¹ atm⁻¹</td>
</tr>
<tr>
<td>Carbon dioxide</td>
<td>657.86</td>
</tr>
<tr>
<td>Hydrogen</td>
<td>866.14</td>
</tr>
<tr>
<td>Nitrogen</td>
<td>127.00</td>
</tr>
<tr>
<td>Oxygen</td>
<td>294.64</td>
</tr>
</tbody>
</table>

Table 6.4: Polymer permeability to gases

There are several different grades of FEP and PFA with similar physical properties but not identical. Thus it was important to establish that permeability for the test fluoropolymer materials kindly donated by Holscot Industrial Linings. Typically permeability rates are measured using a membrane with pure oxygen on one side and a carrier gas such as Argon flowing on the far side, this transports permeated oxygen to a Mass Spectrometer where the oxygen quantity can easily be
calculated [PERMEABILITY 85]. However a Mass Spectrometer was out of the scope of this project and an alternative system had to be established. The method chosen was to use a column of water with the base in a reservoir. With the permeant gas feed at the top, a slight negative pressure is generated below the disc which is used to draw the permeated oxygen into the column. As the water level decreases the volume of oxygen in the column represents the volume of permeated oxygen at a given temperature and pressure.

6.4.1 Procedure

Oxygen permeation

The test apparatus for determination of the oxygen permeability is shown above in figure 6.9. The polymer membrane was trapped between the two halves of the high density polythene assembly. Oxygen pressurant was applied to the top side of the membrane and the permeated oxygen was collected in the water column. The area over which permeation occurred was calculated to be $3.14 \pm 0.05 \text{ cm}^2$ and the thickness of polymer used was $0.5 \pm 0.05 \text{ mm}$. The experiment proceeded with the assembly of the permeation apparatus with the chosen polymer disc sandwiched
between the two halves. The water column was filled and connected via a flexible polythene pipe to the permeation vessel. The regulated oxygen supply was then connected to the pressurant port and adjusted to 3.2 Bar gauge pressure (4.2 Bar absolute). As there was a 0.2 Bar partial pressure of oxygen on the far side, due to the atmosphere, this results in a 4.0 Bar differential across the polymer. The pressurisation caused the polymer to dish down slightly, but due to the bore being 20mm diameter, this was not excessive. The water column was then monitored daily over 12 days and the volume of permeated oxygen recorded along with ambient temperature. The volume in the column was a function of temperature which was necessarily also recorded. Oxygen does dissolve into the water over extended periods, but due to the lack of agitation and small surface area this was proven to be insignificant.

**Hydrogen Peroxide Permeation**

The liquid HTP permeation rate was measured in similar way to the oxygen test, however for this experiment a silver catalyst disc was placed on the far side of the membrane. This allowed the permeating hydrogen peroxide to decompose upon the catalyst, generating oxygen which could then be measured in the water column. This modified apparatus is shown in 6.10.

The test was simply a repeat of the oxygen test with a 4 Bar pressure differential across the membrane. The set up of the apparatus proceeded with assembly of the apparatus with the polymer sample, previously cleaned with IPA and distilled water. The reservoir above the polymer was then filled with hydrogen peroxide and the top cap was sealed onto the apparatus to
pressurise the system. The apparatus was then left and the water column monitored periodically for one month.

6.4.2 Results

Oxygen Permeation

Table 6.5 below, shows the data recorded for the FEP and PFA, along with the calculated permeability. The volume increase is corrected for temperature differences to ST&P conditions.

<table>
<thead>
<tr>
<th>Sample</th>
<th>Volume Increase/cm³</th>
<th>Duration of Test/days</th>
<th>Permeability/mmcm³m⁻²24h⁻¹atm⁻¹</th>
</tr>
</thead>
<tbody>
<tr>
<td>PFA</td>
<td>8.29±0.5</td>
<td>12.333</td>
<td>267.64±53.85</td>
</tr>
<tr>
<td>FEP</td>
<td>5.83±0.5</td>
<td>12.333</td>
<td>188.16±42.65</td>
</tr>
</tbody>
</table>

Table 6.5: Measured oxygen permeability data

Hydrogen Peroxide Permeation

Table 6.6 below, shows the data recorded for the FEP and PFA samples. The volume increase is corrected for temperature differences to ST&P conditions.

<table>
<thead>
<tr>
<th>Pressure Differential</th>
<th>Sample</th>
<th>Volume Increase/cm³</th>
<th>Test Duration/days</th>
<th>Rate/cm³year⁻¹</th>
</tr>
</thead>
<tbody>
<tr>
<td>4.0 Bar</td>
<td>PFA</td>
<td>16.02±0.5</td>
<td>30.0</td>
<td>194.9±6.1</td>
</tr>
<tr>
<td></td>
<td>FEP</td>
<td>12.12±0.5</td>
<td>30.0</td>
<td>147.5±6.1</td>
</tr>
</tbody>
</table>

Table 6.6: Liquid HTP permeation data

6.4.3 Discussion

Oxygen Permeation

The gaseous oxygen permeation experiment produced results similar to the manufacturers specification, however due to the nature of the experiment the error bars on the permeability values are significant. The average temperature of the apparatus was 12°C, considerably lower
than the manufacturers test temperature of 25°C.. Literature predicted at the 12°C, the permeability should fall to approximately 60% of the 25°C value [DuPONT 02]. Hence the results generated were both within 7% of the manufacturers value proving the validity of the experiment. Additionally the permeation rate for a 1mm thick liner with a pressure differential of 1 Bar is approximately one third of the measured decomposition rate. Thus a permeable tank with a 1mm thick liner would reach an equilibrium pressure at approximately 3 Bar. This shows that the permeation and decomposition rates are the correct order of magnitude to make the concept of a passively vented tank feasible.

Hydrogen Peroxide Permeation

It was discovered upon disassembly of the apparatus that significant oxygen bubble formation had occurred over the complete surface of the polymer, due to HTP decomposition. Thus a significant proportion of the permeant was oxygen not liquid hydrogen peroxide. It was also noted from these experiments that the HTP under 4 Bar of pressure produced a much higher rate than expected as virtually no permeation was predicted from theory. The bubbles had clearly produced an erroneous result as it was not known the proportion of liquid HTP to oxygen that permeates through. In the worst case, oxygen was covering the entire surface area of the sample and thus no liquid HTP was in contact with the polymer. Using the permeation rates found empirically, shown in table 6.5, and assuming complete oxygen bubble coverage, the following rates would theoretically be obtained shown in table 6.7, with the actual data also given for comparison.

<table>
<thead>
<tr>
<th>Sample</th>
<th>Empirical 4.0 Bar $\text{HTP}/ \text{cm}^3 \text{year}^{-1}$</th>
<th>Theoretical 4.0 Bar $\Delta P \text{O}_2/ \text{cm}^3 \text{year}^{-1}$</th>
</tr>
</thead>
<tbody>
<tr>
<td>PFA</td>
<td>194.9±6.1</td>
<td>263.5±52.7</td>
</tr>
<tr>
<td>FEP</td>
<td>147.5±6.1</td>
<td>172.6±39.1</td>
</tr>
</tbody>
</table>

Table 6.7: Comparison data of liquid permeation and theoretical oxygen permeation

These calculations show that if oxygen was indeed present over the entire surface, a higher permeation rate would be expected. It was logical to assume that only a partial coverage of oxygen bubbles was present on the surface, however the percentage remains unknown. The data points to the possibility that very limited quantities of liquid HTP permeates through and it is essentially just permeated oxygen that is causing the volume increase. Without performing a very long-term weight loss test, the permeation rate of liquid HTP remains inconclusive and this area is proposed for research in the continuing work.
6.5 Polymer Strength Post HTP Exposure

It is vital that the polymer retains its strength and plastic properties after prolonged exposure to hyrdogen peroxide in order to retain the tank integrity. While the metallic outer shell will be the load bearing support, any embrittlement of the plastic may cause cracking under thermal expansion. Similarly softening of the plastic can cause problems with enhanced cold flowing and degredation of seals on joints. The three parameters that were measured are maximum tensile strength, offset yield strength and tensile modulus. Maximum tensile strength measures the overall strength of the material and this parameter is used to observe any increase or decrease in strength after exposure to HTP. The offset yield strength is the stress required to produce a permanent offset in the length of the sample by 2%. This metric gives information about the onset of plastic behaviour. The final parameter, the tensile modulus, gives the strength over the non-plastic regime. Embrittlement of the polymer would become apparent with an increase in the tensile modulus and early destructuion at the onset of the plastic deformation regime [ARRIDGE 75]. The industry standard method of producing strength data on materials is to perform a pull test. In such a study of a sample, the material is clamped between two jaws, one static and the other dynamic. The dynamic jaw is then motor driven such that the material is stretched as the

![Image of 'Instron' pull test apparatus]
distance between the jaws increases. The displacement and the load on the sample are automatically measured throughout and from this data a stress/strain plot can be produced. The same methodology was applied to the PFA and FEP samples and was conducted on the Instron Pull Test Machine, figure 6.11, located in the Mechanical Engineering department of the University. The shape for the polymer test samples is a dog bone profile with the dimensions shown in the figure 6.12. The shape is important to prevent necking of the clamped section of the sample under tensile stress, producing erroneous results. It was critical that all the samples had the same dimensions in order to produce comparable results and as such a CNC milling machine was employed to perform the manufacturing. In total 18 samples were manufactured, 9 of the PFA and 9 of FEP, all 0.5mm thick.

6.5.1 Procedure

Twelve of the eighteen dog bone samples (six of PFA and six of FEP) were prepared by degreasing with IPA. The residue was rinsed off with distilled water and the samples were placed in two Pyrex beakers, one for each type of polymer. The beakers were then filled with 90% hydrogen peroxide so all the samples were fully submerged. The remaining six dog bone samples were simply cleaned and were not exposed to hydrogen peroxide. The strength of the polymers were then measured with following degrees of exposure.

1) No Exposure, virgin samples

2) 2 Month exposure in 90% HTP at room temperature

3) 4 Month exposure in 90% HTP at room temperature

At each interval, samples were removed from the HTP, 3 of PFA and 3 of FEP. The sample was rinsed with distilled water and allowed to naturally dry. Within 24 hours of removal from the HTP, the sample was clamped into the jaws of the pull test instrument and the test could begin. The jaws were clamped such that the end tags of the material were in direct contact with the face of the jaws. All the tests were performed with this same set up to retain near identical test
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conditions. The pull rate was always set to 10mm/min and the data was then recorded over the period of the test which typically lasted for about 10 minutes. The test was completed upon the breakage of the polymer and the maximum tensile strength recorded. The data recorded in the form of a stress/strain graph was then used to calculate the maximum tensile strength, offset yield strength and tensile modulus.

6.5.2 Results

For brevity the complete family of stress/strain curves of the samples are not shown, but instead two graphs showing FEP with zero exposure and with 4 month exposure are presented in figures 6.13 and 6.14. The data interpreted from the recorded data is summarised in table 6.8. As three samples were taken at every test point, an average of the three could be determined and the error value then encompasses the spread of the values for this data point.

<table>
<thead>
<tr>
<th>Exposure and Polymer type</th>
<th>Max Tensile Strength /MPa</th>
<th>Offset Yield Stress/MPa</th>
<th>Tensile Modulus/MPa</th>
</tr>
</thead>
<tbody>
<tr>
<td>0 Month FEP</td>
<td>128±38</td>
<td>8.5±0.3</td>
<td>414±21</td>
</tr>
<tr>
<td>2 Month FEP</td>
<td>77.4±22</td>
<td>7.6±0.5</td>
<td>371±35</td>
</tr>
<tr>
<td>4 Month FEP</td>
<td>78.8±28</td>
<td>8.2±0.4</td>
<td>412±12</td>
</tr>
<tr>
<td>0 Month PFA</td>
<td>75±28</td>
<td>9.1±0.4</td>
<td>428±23</td>
</tr>
<tr>
<td>2 Month PFA</td>
<td>86±21</td>
<td>7.9±0.3</td>
<td>375±25</td>
</tr>
<tr>
<td>4 Month PFA</td>
<td>145±35</td>
<td>7.0±0.8</td>
<td>423±18</td>
</tr>
</tbody>
</table>

Table 6.8: Polymer pull test results

6.5.3 Discussion

The data showed that there was a large scatter in terms of maximum tensile strength for both the FEP and PFA cases. This is most likely due to slightly incorrect mounting of the sample with non-parallel jaws, causing excessive side loading. In the FEP case there was not a significant trend of this parameter with HTP exposure. However the PFA case demonstrated a possible reduction with exposure to HTP. The offset yield stress of the FEP material also showed no trend with exposure to HTP, however the PFA case did exhibit a possible trend. In the
Figure 6.13: Stress/Strain data of virgin FEP sample

Figure 6.14: Stress/Strain data of FEP sample after 4 month exposure
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PFA case, there was some evidence to suggest that the onset of plastic deformation is occurring at a lower stress with increased exposure to HTP. However the experimental error is large enough to cast doubt on this possible trend. Finally the tensile modulus showed no trend with exposure for either the PFA or FEP cases.

The conclusion that can be drawn from the tests is there appears to be no significant effect, mechanically, on the performance of FEP polymer over a 4 month continuous exposure to hydrogen peroxide. The PFA plastic, while maintaining the same tensile modulus throughout the tests, exhibited a possible reduction in the stress required to cause plastic deformation. Thus out of the two polymers FEP is the preferential material. Neither polymer showed any discolouration during the test, nor was there a measurable weight gain of the samples. Hence the conclusion was that FEP is unaffected by exposure to HTP while the PFA suffers possible minor degradation.

6.6 Chemical Analysis

As well as performing mechanical tests upon the liner materials, chemical analysis was also performed to identify any signs of performance degradation that might occur. Oxidation of the polymer liner by the hydrogen peroxide can lead to a reduced compatibility through a break down of the polymer structure. Similarly any material that is shed by the polymer into the hydrogen peroxide will eventually be used in the catalyst pack, potentially causing catalyst failure. Surface analyses can be performed with a wide variety of instruments, however XPS was chosen for this application due to the ability to discern specific elements with a quantitative data output of concentrations [ALMQUIST 67]. XPS is a surface analysis technique that relies upon the detection of photoelectrons that are emitted from a sample when bombarded with X-Rays in a vacuum. When the photon strikes an atom in a molecule it is ionised and a core electron is emitted. The electron that is emitted has a specific binding energy. Hence by measuring the energy of the emitted photoelectron it is possible to determine which shell on what element the photon had struck. A spectrum of binding energy on the x-axis and counts of photoelectrons on the y-axis is produced from this analytical technique. Thus by generating the characteristic spectrum it is possible to determine the elements present in the sample.
6.6.1 XPS of Polymers

The experiment was conducted to observe any difference between FEP and PFA plastics that had been exposed to hydrogen peroxide when compared to virgin samples of the polymers. The two areas that differences were likely to occur was in the oxidation of the polymer itself and contamination of the surface from the stabilisers found in the hydrogen peroxide. The XPS technique is sensitive enough to detect shifts in electron energies of the carbon atom due to the element it is bonded to. As such it is possible using the XPS technique to discern C-C, C-O and C-F bonds [FLUOROPOLYMER 97]. With a virgin polymer sample it was expected that the vast majority of the carbon would be found bonded to the fluorine atom. However after oxidation the C-O bond will become more prevalent. The trace elements of sodium, lead and phosphorous, all present in hydrogen peroxide as stabilisers would also show on the polymers, due to slight adsorption onto the surface.

6.6.1.1 Procedure

The experiment proceeded by first preparing two sets of samples in FEP and PFA. Eight 1 cm² samples were cut from the polymer sheets (four of FEP and four of PFA) and degreased using IPA. They were then thoroughly rinsed in distilled water. One set of samples (two FEP and two PFA) were wrapped in aluminium foil and stored. The remainder were placed in a cleaned beaker and filled with hydrogen peroxide of 90% concentration and left for 2 months. At the end of this period the samples were removed from the beaker, rinsed in distilled water and dried over a radiator. The samples were only ever handled by the edges thus limiting contamination. The samples, once dry, were packaged up in aluminium foil and sent of for analysis to the XPS laboratory. aluminium foil was chosen as the packaging material as this was recommended to give minimum contamination [GREAVES 01].

6.6.1.2 Results

The global spectra of the FEP and PFA samples before and after the 2-month exposure are shown in figures 6.15 through to 6.18. The carbon peak is examined in detail in figures 6.19 through to 6.22. Table 6.9 shows the area ratios of the C-F to C-O to C-C peaks. The area ratios are used as these represent the ratio of species present per unit area. All the area ratios are normalised such that the C-C bond peak has an area of 1.00.
Figure 6.15: PFA Spectrum prior to peroxide immersion

Figure 6.16: PFA Spectrum post 2 month immersion
Figure 6.17: FEP Spectrum prior to immersion

Figure 6.18: FEP Spectrum post 2 month immersion
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Figure 6.19: PFA C1s Spectrum prior to immersion

Figure 6.20: PFA C1s Spectrum post 2 month immersion
Figure 6.21: FEP C1s Spectrum prior to immersion

Figure 6.22: FEP C1s Spectrum post 2 month immersion
### Table 6.9: Comparison of degree of fluorination before and after exposure to HTP

<table>
<thead>
<tr>
<th>Samples</th>
<th>C-F</th>
<th>C-O</th>
<th>C-C</th>
</tr>
</thead>
<tbody>
<tr>
<td>Virgin PFA</td>
<td>55.56</td>
<td>6.11</td>
<td>1.00</td>
</tr>
<tr>
<td>2 month exposure PFA</td>
<td>48.15</td>
<td>5.31</td>
<td>1.00</td>
</tr>
<tr>
<td>Virgin FEP</td>
<td>39.91</td>
<td>5.47</td>
<td>1.00</td>
</tr>
<tr>
<td>2 month exposure FEP</td>
<td>31.25</td>
<td>5.00</td>
<td>1.00</td>
</tr>
</tbody>
</table>

6.6.1.3 Discussion

The FEP spectra showed no differences on the full sweep chart between post exposure and virgin sample. The PFA charts do however show a difference with a slight peak at 486eV signifying the presence of tin. As tin is used as a stabiliser in the HTP it was not surprising that the PFA polymer absorbed some of this metal. However one would expect the FEP would do likewise, but the spectrum did not show this metal present. When the carbon peak was examined in detail, as shown in figures 6.19 through to 6.22, the binding energy shifts could be seen due to the nature of the bonding. Although C1s shell has a binding energy of 288eV, when this atom is bonded to oxygen, the binding energy is shifted to 281eV and a carbon-carbon bond to 278eV. The fluorine-carbon bond remains un-shifted at 288eV. In all the samples the C-F, C-O and C-C bonds are clearly visible with the C-F bond dominating the spectrum. It was apparent from the data that the size of the C-F peak to the C-O peak did not vary appreciable in either the PFA or FEP case after exposure to HTP when compared to the virgin samples. This demonstrates that oxidation of the sample is not occurring to a measurable degree over the two month sample period.

6.6.2 XPS of Silver Catalysts

As well as the HTP affecting the polymers it is conceivable that polymer can detrimentally affect the catalyst performance. An analogous event occurred with hydrazine monopropellant systems [BALLINGER 99]. It was common practice to use elastomer bladder tanks for hydrazine tanks, which were manufactured from a Silicon-based rubber. However it was found that the Silicon leached out and poisoned the catalyst. In a similar way, organic components leached out of the polymer can cause problems on silver catalyst packs [McCORMICK 65] thus it was important to
observe the silver catalyst pack before and after exposure to hydrogen peroxide that had been stored in contact with the polymers. The samples of polymers used had a surface area of approximately $59.6 \pm 0.5\text{cm}^2$ with a HTP volume of $126 \pm 10\text{cm}^3$. This gave the equivalent surface area to volume ratio of a 1.1 litre tank.

### 6.6.2.1 Procedure

The experiment used the same hydrogen peroxide that was employed in the decomposition compatibility tests as described in §6.3 and as such maximum cleanliness was maintained due to the strictness with which the original test was conducted. Samples of silver gauze (from which the catalyst packs were made) were degreased with IPA and rinsed with distilled water. Two samples were packaged up for analysis by XPS in the virgin state while the remaining six were used for the HTP decomposition. The hydrogen peroxide that had been in contact with the FEP or PFA polymer for two months was decanted into a small, cleaned reaction beaker. $20\text{cm}^3$ of HTP was used in each test and a single cleaned $12.7\text{ mm}$ diameter silver gauze disc was dropped into it. Violent decomposition then occurred which resulted in the liquid being totally consumed leaving a residue on the silver that was later analysed for contamination. The $20\text{cm}^3$ of hydrogen peroxide per silver disc is equivalent to the whole catalyst pack decomposing $2.8\text{ kg}$ of HTP, which was chosen as a flight realistic value. As well as performing a test with stored HTP, samples straight from the Degussa supplied storage drum were also decomposed to provide a control. All the samples, once dry, were then wrapped up in aluminium foil and sent off for XPS analysis.

### 6.6.2.2 Results

Figure 6.23 shows a typical wide sweep survey of a sample of silver gauze that had been cleaned, but not introduced to hydrogen peroxide. Figure 6.24 shows a typical wide sweep survey of the ‘control’ silver catalyst after it had been exposed to the HTP that was stored in the Degussa supplied drum. It was known through practical experiments that storage in the Degussa drum for extended periods was perfectly acceptable, and thus the spectrum produced is typical of an acceptable spectrum. Figures 6.25 and 6.26 show the results of the analysis performed on the silver discs after exposure to hydrogen peroxide stored in contact with PFA and FEP polymers respectively.
Figure 6.23: Ag catalyst prior to HTP exposure

Figure 6.24: Ag catalyst after exposure to storage drum HTP
Figure 6.25: Ag after exposure to HTP stored with PFA

Figure 6.26: Ag after exposure to HTP stored with FEP
6.6.2.3 Discussion

Prior to exposure of hydrogen peroxide, the silver disc had oxygen present on the surface as can be seen by the peak at 530 eV. This may however be due to water contamination. As hydrogen cannot be detected using XPS, the source of the oxygen remains inconclusive. The main peak however is at 368 eV corresponding to silver with a minor peak at 284 eV, which is the C1s binding energy. Clearly carbon should not be present and was probably introduced through the IPA cleaning process. While the alcohol itself is completely volatile, the small but significant heavy hydrocarbon residue may well have caused this carbon contamination.

After exposure to the control HTP, obtained directly from the Degussa drum, the spectra showed the presence of tin and sodium. The tin showed up with a large double peak at 486 and 494 eV and the sodium at 1071 eV. The carbon peak at 287 eV had actually reduced relative to the pure silver test previously discussed. This indicates that the sample HTP was not contaminating the silver with organics, but with tin and sodium compounds. As discussed previously tin is used as a stabiliser thus it was expected to be deposited onto the silver. In addition phosphates are also added to stabilise the HTP, usually in the form of sodium pyrophosphate, explaining the sodium contamination. The phosphorous also shows up as a very small peak at 133 eV.

Figure 6.25 shows the XPS data of the silver catalyst when used to decompose HTP stored with PFA for the period of 2 months. It can be seen from this figure that the same elements are present as the control test, Ag, O, C, Sn, Na and a small quantity of phosphorous. Also the proportions of these elements are extremely similar, with the exception of carbon, which is marginally greater. The carbon is most likely contamination from sample preparation as opposed to leaching from the polymer, mainly because a relatively high carbon content was witnessed on the pure Ag sample, demonstrating external contamination. Figure 6.26 shows the XPS data of the silver catalyst when used to decompose HTP stored with FEP for the period of 2 months. Again all the previously discussed elements are present and the spectrum is nearly identical demonstrating both repeatability in the results and lack of any significant effects of storage with the FEP polymer.

Thus it can be concluded that over a 2 month period the residue left on the catalyst used to decompose the HTP, which was stored with PFA or FEP, was virtually identical in composition to the residue left behind if a sample of the bulk stored HTP was decomposed. Thus there should be no problems with the catalytic decomposition of hydrogen peroxide after storage with either FEP or PFA polymers.

6.6.3 Chemical Analysis Conclusions

The analysis of the polymers consistently showed that over a 2-month exposure period, there was no degradation of the surface through oxidation. Visible inspection of the sample also showed no
discolouration throughout the extended test. The silver catalysts showed no sign of increased degradation when exposed to HTP stored in contact with the polymers. Virtually identical spectra were obtained with the bulk-stored HTP, demonstrating that storage in either PFA or FEP lined tank is perfectly acceptable.

6.7 Deformation of PFA/FEP Under Pressure and Temperature

When using a polymer liner as a permeable membrane, the stress bearing outer shell must be both strong enough to contain the pressure and allow the permeated oxygen to escape. The method proposed to achieve this was to cut a series of concentric circular grooves on the inside of the tanks stress-bearing shell. These grooves, when joined together via radial grooves to a vent port, provides a leak path through the liner to the vacuum of space. This type of tank design is shown in figure 6.27 Using the concept imposes constraints upon the tank design. The polymer that sits on top of the vent grooves must not extrude to fill this gap and seal off the channel for venting the tank. The degree of deformation can be calculated theoretically using standard stress calculations and the tensile modulus of the polymer, details of which are given in §7.4.1. However, both PFA and FEP are well above their glass transition temperatures [FLUOROPOLYMER 97] and hence will cold flow. Fortunately PFA and FEP cold flow considerably less than PTFE [FLUOROPOLYMER 97]. To observe the effects of extrusion a test was designed where the polymer could be exposed to high pressure and temperature of 40 Bar and 50°C. The end plate allowed the polymer to extrude through a 1.6mm diameter hole, chosen to simulate the grooves of similar dimensions. The degree of extrusion provides qualitative data on this characteristic.

![Permeable tank design](image)

Figure 6.27: Permeable tank design
6.7.1 Procedure

The schematic in figure 6.28 shows the apparatus used to conduct the test. The assembly was fastened together with the 0.5 mm thick sample of FEP polymer sandwiched between the end plate and the manifold. The system was then pressurised through a regulator set at 40 Bar and placed on a heater, which was adjusted to give a polymer temperature of 50°C. This temperature was chosen as an upper limit to the spacecraft’s expected operating regime. The test was then run for an hour and then disassembled for inspection. The test was then repeated with PFA in place of the FEP. Once the test had been completed the samples were examined for signs of serious extrusion. While this test does not prove over extended periods that cold flow may or may not cause a problem, it provides immediate data on the short-term effects. If the tests show extensive deformations it would demonstrate that cold flow is a large problem and the sample would most likely unsuitable for the required task. However, small deformations would demonstrate more chance of success and is merely a qualitative indication of suitability.

6.7.2 Results

No significant extrusion of either polymer occurred, however a visible spot indicating slight extrusion was noticed with both polymers. Figure 6.29 shows the FEP polymer immediately after the hour-long test. Very similar results were obtained with the PFA sample.
6.7.3 Discussion

While neither polymer demonstrated significant deformation, this does not guarantee that over very long durations the plastic would not significantly cold flow. The results merely show that despite the plastics being above their respective glass transition temperatures, cold flowing does not appear to be as important an issue as first assumed. Ideally an extremely long term test of a year or more would be required to prove or disprove this concern.

6.8 Flight-Type Passively Vented HTP Tank

The culmination of the propellant tank work was the production and a test of a tank to simulate long term sealed storage using the permeation system shown in figure 6.27. The aim of the experiment was to prove that hydrogen peroxide could be sealed and left unattended for extended periods with only a vent port open for the oxygen to escape. Crucially the tank should not allow liquid to leak and could therefore be used as a flight propellant tank.

The operation of a permeable liner tank in a 1 ‘g’ environment would be quite different to zero ‘g’, however the principle of permeation remains the same. In a 1 ‘g’ environment the liquid would naturally fill the bottom of the vessel. As such the bubbles of oxygen that are formed would detach from the tank wall and rise to the surface. This permeation of the oxygen will therefore primarily occur over the hemisphere at the top of the tank. In the zero ‘g’ case, the liquid will form a sphere inside of the tank. As the polymer is hydrophobic, the contact area with the tank wall will be minimised. Additionally the oxygen bubbles generated will stay in contact.
with the wall and thus the majority of the liner will be available for permeating oxygen. The tank was tested in a 1 ‘g’ environment, and thus it was designed accordingly.

As the permeable liner sat on the grooved stress bearing shell there was a range of possible permeation paths that could occur. The liner was not designed to be bonded to the outer shell and was merely held in place through the internal pressure. In the extreme case where the liner seals onto the tank wall in-between the groove channels, the effective liner thickness will increase due to the discreet leak channels. Conversely if no appreciable sealing occurs and the whole hemisphere is essentially a continuous leak path, the effective thickness of the liner will be the physical thickness. The former case is mathematically examined in §7.4.2.

The spherical tank was designed to have a total volumetric capacity of 1.5 litres (Ø142mm). With a 75% fill fraction, the quantity of hydrogen peroxide to be stored was 1.125 litres. As the test was designed to be relatively safe, low equilibrium pressures of below 10 Bar were desired. The exposed area at the top of the tank where permeation could occur was 205.4cm² and the liquid HTP was in contact with 428.1 cm² of FEP. From the coupon tests performed earlier at 32°C, the expected quantity of generated oxygen over 1 year was 7,491.8 cm³ at ST&P. At equilibrium pressure, this flow rate of gas must be permeated. Using a 1mm thick FEP liner, the following equilibrium pressures, in table 6.10, were calculated. Two cases were considered; where the liner doesn’t seal causing a low equilibrium pressure, or sealing does occur generating a higher equilibrium pressure. The maximum pressure case was taken using vent grooves spaced at 5° intervals. The engineering drawings in Appendix A show the details of the tank.

<table>
<thead>
<tr>
<th>Environment</th>
<th>Estimated tank pressure</th>
</tr>
</thead>
<tbody>
<tr>
<td></td>
<td>Maximum / Bar</td>
</tr>
<tr>
<td>1 ‘g’</td>
<td>7.5</td>
</tr>
<tr>
<td>0 ‘g’</td>
<td>2.4</td>
</tr>
</tbody>
</table>

Table 6.10: Predicted equilibrium pressures for demonstration tank
Figure 6.30: Tank half with FEP liner installed

Figure 6.31: Outside of aluminium stress bearing shell
The predicted pressures are relatively low, demonstrating that a tank using a 1mm thick FEP liner would be an acceptable test. In the worst case where the liner seals itself to the tank walls, the spacing of the vent grooves was sufficiently close to keep the pressure from exceeding the 10 Bar arbitrary limit. Figure 6.30 shows the inside of the hemispherical tank half with the FEP liner installed. Figure 6.31 shows the outside of the tank with the location of the oxygen vent port indicated.

6.8.1 Procedure

Permeation Test

The experiment proceeded with an initial leak proof test to 10 Bar with nitrogen after which the system was set up to test the oxygen permeation rate through the FEP liner. The purpose of this test was to show that the permeability was similar to the tests performed in §6.4. The tank was filled with 1.125 litres of distilled water, thus simulating an 80% full tank. The dome at the top of the tank allows oxygen permeation over an area of 205.4cm². The tank was then pressurised to 2.0 Bar and the oxygen permeated through the top hemisphere was collected in a water column.

Hydrogen Peroxide Storage Test

After the permeation was complete, the tank was set up to store hydrogen peroxide in a live test.
The tank was cleaned and passivated using the same procedure as was carried out on the coupon tests described in §6.3.4. The tank was then filled with 1.125 litres of hydrogen peroxide and the pressure transducer attached directly to the top of the vessel. There was a possibility that gas could leak out of the transducer joints making it seem as though the tank was venting through the liner where in fact it was just leaking through a hole in the system. Thus a configuration was devised where a HDPE secondary containment vessel (SCV) fitted over the transducer assembly, which was then connected to a water column. Any leaks will produce a volume increase and this registered on the water column. Photographs of the assembly are shown in figure 6.32 and a schematic of the assembly in figure 6.33. The system was then pressurised to 4.0 Bar with nitrogen and the pressurisation valve closed. The pressurant line was removed and a blanking cap was sealed onto the open end of the valve. The electrical connections from the transducer were joined to the ‘feed throughs’ in the SCV and the unit was slid into place over the boss. Sealant was applied to the boss/SCV interface, and around the exhaust port tube at the top of the SCV. Once the sealant had set, the water column was attached to the SCV and the temperature noted. Any gas that leaks through the joints in the system increased the volume, which caused the water level to drop. As the pressure inside the SCV was slightly below the ambient pressure due to the head of water, air will also be drawn into the system through any leak holes. Thus, only if the system is entirely leak tight will the water level remain constant (for a given temperature).
Chapter 6: Hydrogen Peroxide Tank Research

test was then conducted over a period of 18 days, chosen purely on the grounds of the limited time available.

6.8.2 Results

Permeability Test

The initial oxygen permeability test upon the tank agreed well with values recorded very close to what was expected. The results are summarised in the table 6.11 below, with the calculated volume derived from the empirical tests conducted earlier and the assumption that the liner did not seal onto the stress bearing shell. The volumes are corrected for ST&P conditions.

<table>
<thead>
<tr>
<th>Test Duration</th>
<th>Test Pressure</th>
<th>Volume Permeated</th>
<th>Volume Expected</th>
</tr>
</thead>
<tbody>
<tr>
<td>4.0 days</td>
<td>2.0 Bar</td>
<td>30.6 cm³</td>
<td>31.0 cm³</td>
</tr>
</tbody>
</table>

Table 6.11: Oxygen permeability of the test tank

Hydrogen Peroxide Storage Test

The results from the test with hydrogen peroxide stored for 18 days demonstrated a slight decrease from the initial pressure of 4.0 Bar down to approximately 3.9 Bar over this duration. The tank did exhibit a cycling effect with temperature, as this parameter was not controlled. The data for the test is presented in figure 6.34. The secondary containment vessel recorded a volume increase, when corrected for temperature variations, of 19.6 cm³ over this same period.

6.8.3 Discussion

The results from the oxygen permeation test show that the liner is not sealing onto the outer stress bearing shell. If sealing was occurring and hence a greater effective thickness, the permeation rate would have been lower than the value recorded. This indicates that the equilibrium pressure expected from the live hydrogen peroxide test would be 3.4 Bar. Hence a pressure decrease from the initial 4.0 Bar charging pressure was expected. The results of the live HTP test shown in figure 6.34 demonstrated that there was no appreciable decrease in pressure over the 18 day test. The measured volume increase from the secondary containment vessel was only 19.6 cm³. Leakage into the SCV was the only possible leak path other than through the liner due to the hydrogen peroxide level being higher than the equatorial bolt flange. This leak rate corresponded to an effective loss of 0.1 Bar over the 18-day
Figure 6.34: Pressure and temperature record of flight-type hydrogen peroxide tank

Figure 6.35: Pressure loss in flight type hydrogen peroxide tank, compensated for temperature
test, insignificant for the purposes of this test. If the tank was completely sealed, that is the vent port was closed, the expected pressure rise would have been 1.2 Bar. It was known that only 0.1 Bar was lost through leaks, thus the liner is clearly performing as desired. Figure 6.35 shows the pressure data corrected for temperature and a low pass filter applied. The best-fit line is also shown in the figure.

The equilibrium pressure from the tests was approximately 3.97±0.05 Bar, higher than 3.4 Bar calculated prior to the test. This can be explained through enhanced decomposition rate in the tank over the coupon samples. It was known that the cleaning procedure for the tank proved more difficult to perform than the coupon samples. It was noted that rinsing out of the Decon 90 was particularly difficult. This may have resulted in a reduced compatibility surface in contact with the hydrogen peroxide. However the disparity is not large enough to cause concern.

6.9 HTP Tank Research Conclusions

The research has demonstrated the PFA and FEP polymers are very compatible with concentrated hydrogen peroxide, recording only 17-20cm³ of evolved oxygen at ST&P per square centimeter per year. In terms of AOL this is approximately 1.9%, which when compared to table 2.3 demonstrates favourable results. However neither polymer is as compatible as pure Aluminum, recording just 12.8cm³ of evolved oxygen at ST&P per square centimeter per year. This apparently low decomposition rate would still produce a 19.2 Bar increase in an 85% full 10 litre spherical tank over the course of a year. This level may be acceptable for a mission of less than a year, however any longer and the pressure rise would almost certainly be unacceptable. Thus the concept of a passively vented tank would therefore become more attractive.

The oxygen permeation tests performed upon the polymers demonstrated that similar permeation rates to the manufacturers specifications were obtained. Tests designed to calculate liquid permeation rates proved unsuccessful due to decomposition upon the polymers surface. This made the oxygen and liquid HTP rates indistinguishable. Thus the best estimate that can be made is from the known water permeation rates through FEP. It was known that the water transmission rate through a membrane ≥ 0.25mm thick is approximately 0.025 g per 625cm² per day [DuPONT 02]. With a 10 litre spherical tank, with a 1mm thick FEP liner, the expected water permeation rate is just 32.7g per year, and a similar value can be expected for hydrogen peroxide. Thus 0.3% of the initial propellant volume per year will be lost due to permeation.

The tests performed upon the fluoropolymers to determine the strengths after exposure to hydrogen peroxide demonstrated a remarkable resistance to the oxidiser. Crucially the tensile modulus did not vary after 4 months of exposure as compared to the virgin sample. This
demonstrates that embrittlement is not taking place, a vital requirement for a space tank where thermal cycling will occur.

The resistance to the oxidiser was confirmed with the XPS examination of the polymers. No discernable surface chemistry difference between the sample before or after the exposure could be found. In particular the ratio of the C-F bonds to the C-O bonds remained constant and hence oxidation is not taking place to any appreciable degree. The analysis of silver catalysts used to decompose hydrogen peroxide stored with the polymers showed no adverse effects. Thus it can be concluded that the oxidiser can be stored for extended periods in a PFA or FEP tank with no adverse effects to either the tank liner or the hydrogen peroxide.

Finally the construction and evaluation of a flight type tank demonstrated that hydrogen peroxide can be stored for extended durations in a vessel which is leak tight to liquids. An 18 day test resulted in an equilibrium pressure of approximately 3.9 Bar without any significant loss of pressurant through unwanted leaks in the tank/transducer interface. In addition an elevated temperature test on the extrusive properties of the polymer, indicated that liner should maintain its function over extended periods.

This is the first time that a passively vented hydrogen peroxide tank has been demonstrated and the tests conducted indicate that this system represents a viable way of storing hydrogen peroxide on orbit for extended periods.
Chapter 7

7 Theoretical and Numerical Models

7.1 Introduction
7.2 Chamber Heat Flux Model
7.3 Regenerative Nozzle Simulation
7.4 Passively-Vented Tank Channel Analysis
7.5 Sensitivity of O/F to fuel pressure
Chapter 7: Theoretical and Numerical Models

7 Theoretical and Numerical Models

7.1 Introduction

This chapter is concerned with the theoretical and numerical models that have been used to calculate the design parameters of the thruster and the hydrogen peroxide tank system.

7.2 Chamber Heat Flux Model

It was important to accurately assess the heat flux into the engine’s walls to determine the most efficient cooling system, as was performed in §3.3. Thus a numerical model was developed to calculate the heat fluxes for a given geometry. Convective and radiative heating both occur in the parallel chamber region, with radiation being the dominant term, however as the gas enters the convergent section the velocity increases and the convective heat transfer becomes dominant over the radiation term. The engine will also radiate heat and, when tested at sea level, be subjected to convective cooling through the external air passing over it. Clearly in space convective cooling will not occur and thus the model ignores this cooling mechanism.

In order to calculate the heat fluxes a chamber wall temperature must be known and the simulation requires this to be provided by the user. Typically the wall temperatures of rocket engines are run as hot as is possible and hence the maximum service temperature was chosen. As the chamber diameter is a function of the distance down the chamber, the heat flux will vary accordingly. The model was required to provide heat flux data in three specific zones; the parallel chamber section, convergent section and divergent section. The parameter ‘x’ is defined as the distance down the chamber with its origin at the injector face. With reference to figure 3.6, the diameter ‘d’ of the engine can be described in the following equations.

If $x \leq L_{ch}$ then $d = d_{ch}$  \hspace{1cm} (7.1)

If $x > L_{ch}$ and $\frac{d_{ch} - d_{th}}{2Tan30^\circ} + L_{ch}$ then $d = d_{ch} - 2(x - L_{ch})Tan30^\circ$  \hspace{1cm} (7.2)

If $x > \frac{d_{ch} - d_{th}}{2Tan30^\circ} + L_{ch}$ then $d = 2Tan15^\circ\left(x - \frac{d_{ch} - d_{th}}{2Tan30^\circ} + L_{ch}\right) + d_{th}$  \hspace{1cm} (7.3)

The simulation splits the engine into elemental slices of width $\Delta x$ and the heat fluxes calculated for each element. No conduction was assumed between wall elements due to the isothermal
approximation. While this is appropriate in the parallel chamber region, in the throat region this is less precise. However this technique has been used for the analysis of practical engines and has been shown to be accurate [LIQUID PROP 57b].

The heat flux into the slice through radiation and convection was calculated using equations 3.4 and 3.5 with the convective heat transfer coefficient calculated using equation 2.39. These equations expressed in terms of the actual model are shown below with the nomenclature defined previously.

\[ Q_{\text{rad}} = \sigma \pi d \Delta x \varepsilon \alpha_w (T_e^4 - T_w^4) \]  
\[ (7.4) \]

\[ Q_{\text{conv}} = \pi d \Delta x h_c \left( T_e - T_w \right) \]  
\[ (7.5) \]

\[ h_c = 0.021 \frac{k_c}{d_{4xy}} \left( \frac{\dot{m}_{\text{main}} d_{4xy}}{\pi \left( \frac{d}{2} \right)^2 \mu_c} \right)^{0.8} \]  
\[ (7.6) \]

The engine will also radiate heat energy from each section. For simplicity the internal diameter was taken to be the same as the external diameter, thus the internal and external areas of the section are the same. The power lost due to radiation is shown in equation 7.7.

\[ Q_{\text{rad,out}} = \sigma \varepsilon \pi d \Delta x T_w^4 \]  
\[ (7.7) \]

The heat flux into a section of the chamber of length \( \Delta x \), whose diameter is \( d \), is therefore given in equation 7.8.

\[ Q_{\text{in}} = \pi d \Delta x [h_c (T_e - T_w) + \sigma \varepsilon \alpha_w (T_e^4 - T_w^4) - \sigma \varepsilon w T_w^4] \]  
\[ (7.8) \]

Thus the model proceeds by calculating the power input at every segment along the length of the chamber and nozzle. However the diameter, coefficient of convective heat transfer, and temperature of the gas are all functions of position along the chamber. Thus the total power into the engine is described by equation 7.9 where \( d \), \( h_c \) are \( T_e \) are functions of \( \Delta x_i \).

\[ Q_{\text{total}} = \sum_{i=0}^{L_{\text{ch}}-\Delta x} \pi d (\Delta x_i) \Delta x [h_c (\Delta x_i) \left( T_e (\Delta x_i) - T_w \right) + \sigma \varepsilon \alpha_w (T_e (\Delta x_i)^4 - T_w^4) - \sigma \varepsilon w T_w^4] \]  
\[ (7.9) \]
The gas temperature remains approximately constant throughout the parallel chamber region and can be calculated using the Isp code [SELPH 94]. Once the flow reaches the convergent section, with the accompanying velocity increase, the temperature falls, and as such there is a strong dependency of temperature upon position down the chamber.

From the assumptions that the flow is:

1. Isentropic
2. One dimensional flow
3. Steady flow
4. Ideal gas
5. Choked flow

It can be shown that the area ratios of two parts of the engine are related to their respective Mach numbers according to equation 7.10.

\[
\frac{A_2}{A_1} = \frac{M_1}{M_2} \left[ \frac{1 + \frac{\gamma - 1}{2} M_2^2}{1 + \frac{\gamma - 1}{2} M_1^2} \right]^{\frac{\gamma + 1}{\gamma - 1}}
\]  

(7.10)

Realising that under choked flow conditions, the throat will be at sonic velocity, and using the isentropic relationship for temperature, it is possible to calculate the temperature as a function of position down the chamber. This was accomplished by using a numerical solver routine, to obtain the Mach number at the desired location. From equation 7.11, the temperature can then be found.

\[
\frac{T_c}{T} = 1 + \frac{\gamma - 1}{2} M^2
\]  

(7.11)

Using the solutions to equation 7.11 and equation 7.9, a program was written to calculate the power into the chamber and nozzle for a given chamber geometry. The heat fluxes calculated in the parallel chamber region were validated against real data generated from the MkIV engine. While the convective losses complicated the analysis, the practical data agreed well with theory. The code for simulation is included in the accompanying CD-ROM.
7.3 Regenerative Nozzle Simulation

The fabrication of a simple regeneratively cooled nozzle requires minimisation of the number of cooling passages in the nozzle. The simplest system envisaged, yet still retain a degree of symmetry, was two circular cross section passages either side of the throat. Liquid would therefore flow through one, loop back round at the end and return on the far side as shown in figure 4.7. The problem with this type of nozzle is the degree of conduction needs to be high enough to cool the entire nozzle using only the two heat sinks. The channels are located next to the throat to cool the area of maximum heat flux. However further upstream there will also be severe heating which will not be near a coolant channel. Hence a temperature differential between different areas of the nozzle will be generated. A computer simulation was designed to determine if the temperature differential would be so high that melting would occur in parts of the nozzle.

The simulation split the 25 mm long nozzle into cubes of material, side length 1mm. The matrix space was set at 25mm × 25mm × 25mm. Thus producing a model with 15,625 elements. The matrix was mapped out with the geometry of the nozzle system in the following manner. A cube can be copper, air, combustion products, HTP coolant or the feed pipe. Thus each cube had a number assigned to it to specify which one of the above materials it contains. Initially the copper was set to room temperature of 293K. The coolant temperature was fixed throughout the simulation at a predefined temperature of 350K, and the combustion gas temperature was set by its position down the nozzle. This temperature was calculated theoretically from fundamental rocket thermodynamics using equations 7.10 and 7.11.

The simulation proceeded with constant time step iterations where the new temperature matrix is calculated from the previous one, which was then copied back to the original matrix. Heat transfer was handled in four ways. If a copper cube borders on to a hot combustion cube then convective and radiative heat transfer into the cube will occur which is a function of the gases physical properties and the diameter of the nozzle at that point. If the copper cube borders another copper cube and providing there is a temperature difference, conduction will occur. If a copper cube borders onto a coolant cube then convective heat transfer will occur from the copper into the coolant through the surface area of the cubes' face. The convective heat transfer coefficient was calculated from the dimensions of the cooling channel. Finally if a copper cube borders onto an air cube, then radiation cooling will take place, which is a function of the copper cubes' temperature. If a copper cube borders onto the feed pipe then no heat transfer is allowed to take place. This is a vast simplification, however it produces an upper limit on the nozzle temperature, which is the important value. Clearly in reality heat will be transferred from the hot copper to the
coolant feed pipes. However there may well be a small but significant air gap between the pipe and the copper nozzle, which would significantly affect the conduction of heat. Thus the worst possible scenario was assumed with no heat transfer occurring. The program proceeded by interrogating every cube in the matrix and checking if it met any of the conditions set out above for the immediately adjacent cubes shown in figure 7.1. The power in and out due to the thermal heat transfer is then multiplied by the time step and the heat fluence during this time calculated. Thus by knowing the heat capacity, the temperature rise of the element can be calculated. A problem of stability was encountered during the execution of the program. When the time step was too large the program produced erroneous results that were not physically possible. However reducing the time step to a lower value solved this problem and acceptable results were produced. The time step was chosen to be 0.1 ms, which provided reasonable execution times. Shown below in figure 7.2 is the flow chart for the execution of the program. The program was evaluated to verify correct results were being obtained. With the simulation set to no coolant flow and zero external radiation the program was run for a simulation time of 10 seconds. The power into the nozzle was known from the chamber heat flux model and hence by knowing the mass of the copper used for the nozzle and the heat capacity a simple check was performed to calculate the average expected temperature. The results from the program agreed to within 30K of the expected temperature thus confirming the code. The simulation was then altered to remove the combustion gases but turn on the coolant flow. The simulation was run with an initial starting condition of 1,000K and the program was run for 10 seconds to determine the temperature reduction. This again agreed closely with the expected results from the theory. Finally the radiative losses to the atmosphere were checked.
Chapter 7: Theoretical and Numerical Models

Figure 7.2: Nozzle simulation program flow chart
Upon confirmation of a working model, the simulation was run with all the heat fluxes enabled on a Ø20mm nozzle. The temperature charts below show the cross sectional data for the nozzle at times of 5, 10 and 30 seconds after ignition. The location of two planes are shown in figure 7.3 as viewed from the top. Figures 7.4 through to 7.6 are of plane B, while figure 7.7 is in plane A

Figure 7.3: Locations of the cross sectional planes, viewed along the nozzles axis

Figure 7.4: Temperature map of plane B after 5 seconds of combustion
Figure 7.5: Temperature map of plane B after 10 seconds of combustion

Figure 7.6: Temperature map of plane B after 30 seconds of combustion
The results of the nozzle simulation showed that the nozzle would survive with just the simple, two channel cooling loop. The peak temperature would reach approximately 860K and hence be considerably below the melting temperature of copper at 1,357K. Also as expected the upstream convergent section did get the hottest as this was the furthest away from the cooling loops.

From the models generated, an actual copper nozzle was built and tested which performed adequately during a burn of over 60 seconds with no melting reported. A detailed analysis using embedded thermocouples to validate the model was planned. However the insurmountable problems experienced with the hydrogen peroxide decomposing in the cooling channels switched the attention to film cooling and hence this work was stopped.
7.4 Passively-Vented Tank Channel Analysis

In the passively vented tank system the gas permeable FEP polymer bleeds oxygen into venting channels, which prevents pressure build up within the tank itself. The pressure will rise in the tank until an equilibrium value is reached whereupon the leak rate of the oxygen will equal the oxygen evolution due to decomposition. Figure 7.8 shows a small section of the tank with the FEP liner situated on top of the tank wall with the vent channel shown. The channels can be placed at regular intervals as shown in figure 7.9 and the distance between the channels and the FEP thickness will govern the equilibrium pressure. In addition the FEP will deform over a period of time and hence it will extrude into the venting channels. Thus it is important to determine the depth and width of the channels required to prevent sealing of the vents.

7.4.1 Deformation of the Tank Liner

It is possible to calculate the mechanical deformation of the FEP into the grooves of specified dimensions through a stress analysis. This analysis does not take into account the cold flow that occurs with FEP, however this can be determined empirically. The deformation of sheet materials such as FEP, into a circular cavity is a standard problem [ROARK 75]. With reference to the figure 7.10, the deformation displacement \( y_b \) is given in equation 7.12.

![Figure 7.8: Vent channel geometry](image)
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Figure 7.9: Cross section schematic of tank venting channels

\[ y_b = \frac{P a_h^4}{D} \left( \frac{C_2}{C_5} - L_{11} \right) \] 
(7.12)

\[ D = \frac{E_m \tau^3}{12(1 - \nu^2)} \] 
(7.13)

Where:
- \( P \) = Applied pressure (Nm\(^2\))
- \( a_h \) = Hole radius (m)
- \( E_m \) = Modulus of Elasticity (Nm\(^2\))
- \( \tau \) = Thickness of material (m)
- \( \nu \) = Poisson Ratio

With the following constants specific to the problem in question.

\( C_2 = 0.25 \) \( L_{14} = 0.0625 \) \( C_5 = 0.5 \) \( L_{11} = 0.015625 \)

At 25°C the modulus of elasticity is 600MPa, and when a 0.5mm thick sheet is deformed by 40

\[ 2a_h \]

Figure 7.10: Deformation of FEP sheet under pressure
Bar pressure into a 10mm diameter hole, the maximum displacement is 4.1mm. Limiting the hole diameter to 1mm diameter and increasing the thickness of the FEP to 1mm, reduces the maximum displacement to 0.0004mm, a negligible value. Although the hole scenario is not truly representative of a linear channel, the degree of deformation should be similar and hence should not present a problem.

7.4.2 Effective Liner Thickness

If the liner seals against the tank wall, the greater the distance between the vent channels the greater the effective thickness of the FEP and thus the higher the equilibrium pressure. This effective liner thickness can be calculated if a small cross section of the channel/liner system is considered as in figure 7.11. If an oxygen bubble is situated over area ‘b’, the distance to reach the vent channel will simply be the thickness of the FEP. The oxygen bubble located over area ‘a’ however will have a distance \( \sqrt{\left(\frac{W}{2}\right)^2 + t_{FEP}^2} \) and thus will be longer. The average slant distance will give the effective depth of the FEP. Equation 7.14 describes the volume of oxygen at ST&P that is permeated through the FEP.

\[
v = \frac{\Pi \times A_{sur} \times t \times \Delta P}{\tau}
\]  
(7.14)

Where;

\( \Pi \) = Permeability (mm\( \cdot \)m\( ^3 \)/m\( ^2 \)/day/atm

\( A_{sur} \) = area over which permeation occurs (m\( ^2 \))

\( t \) = Time over which permeation occurs (days)

\( \Delta P \) = Pressure difference between tank and external pressure (vacuum) (Bar)

\( \tau \) = Thickness of polymer (mm)

Hence when all other parameters are kept constant, the volume of gas is proportional to the reciprocal of the thickness. In the above example the volume of gas through the two routes ‘a’ and ‘b’ can be calculated. However this is simply taking into account the two highlighted paths that could be taken. The sum of the paths, of width \( W_{ve} \), need to be taken into account giving the total volume permeated, shown in equation 7.15

\[
v_{total} = \sum \left( \frac{\Pi \times A_{sur,ve} \times \Delta P \times t \times 1}{SlantLength} \right)
\]  
(7.15)

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Inside the sigma term, there will be \( n \) discrete slant lengths, which when multiplied by the area will equal the total area of the analysis width, \( W \). Hence this can be rewritten as equation 7.16:

\[
\nu_{total} = \Pi \times A_{sur} \times \Delta \times t \times \left( \frac{1}{\frac{1}{n} \sum (Slant \_Length)} \right)
\]  \hspace{1cm} (7.16)

Thus the final term is the average of the reciprocal slant lengths. The mean slant length (MSL) can be calculated and hence recasting 7.16 in terms of this new parameter, equation 7.17 is derived.

\[
\nu_{total} = \Pi \times A_{sur} \times \Delta \times t \times \frac{1}{MSL}
\]  \hspace{1cm} (7.17)

Hence by comparison to the equation 7.14, the effective thickness can be defined.

\[
\tau_{eff} = MSL
\]  \hspace{1cm} (7.18)

The MSL value for the vent channel geometry can be calculated by considering figure 7.12. The slant distance as a function of position \( x \), is given in equation 7.19.

\[
Slant = \sqrt{\tau_{FEP}^2 + x^2}
\]  \hspace{1cm} (7.19)
Hence the mean reciprocal slant length is given in equation 7.20

\[
MSL = \frac{1}{W} \int_{-\frac{W}{2}}^{\frac{W}{2}} \frac{x}{\sqrt{\tau_{FEP}^2 + x^2}} dx
\]  

(7.20)

This can be evaluated and thus the MSL value and hence effective thickness is derived.

\[
MSL = \frac{W}{2} \sqrt{\frac{W^2}{4} + \tau_{FEP}^2}\left[\ln\left(\frac{\left(\frac{W}{2} + \sqrt{\frac{W^2}{4} + \tau_{FEP}^2}\right)}{\left(-\frac{W}{2} + \sqrt{\frac{W^2}{4} + \tau_{FEP}^2}\right)}\right)\right]
\]  

(7.21)

Figure 7.12: Derivation of the MSL value
7.5 Sensitivity of O/F to Fuel Pressure

It was known from empirical tests that the pressure drop across the injectors was nominally around 1.5 Bar while the pressure drop across the catalyst pack was approximately 10 Bar. Thus it was expected that the sensitivity of the O/F to the propellant pressures would be significantly greater for the fuel than the oxidiser. This results in an inherent problem of the system, forcing strict control on the pressure of the fuel tank. A theoretical analysis of this effect was required and in order to produce an accurate model, it was important to characterise the injectors. The critical factor when using the injector flow rate formula, equation 7.22, is an accurate value for the head loss coefficient $K_{onf}$.

$$ m_{fu} = A_{inf} \sqrt{\frac{2 \rho_{fu} \Delta P}{K_{onf}}} \quad (7.22) $$

The diameters of the injectors are 0.2mm, sized according to the flow requirements derived in §3.4. To calculate the value of $K_{onf}$, which should be between 1.2 and 1.7, flow tests were conducted. The pressure in a length of pipe filled with water was varied by applying helium pressurant to it. A valve between the helium reservoir and the pipe was used to start and stop the flow. The test then proceeded by timing the duration it took to empty the contents of the pipe into a beaker which was then subsequently weighed. From this the mass flow rate of the water could
be calculated. The results of this are plotted on the graph of mass flow on the y-axis and $\sqrt{\Delta P}$ on the x-axis, shown in figure 7.14. From the calibration plot it is evident that the injector follows the classical equation for liquid flows through a small orifice. The slope of the graph, which related to the value of $K_{\text{orif}}$ through equation 7.23, was found to be $1.289 \times 10^{-6} \text{kgs}^{-1} \text{Pa}^{0.5}$.

$$Slope = A_{\text{inf}} \sqrt{\frac{2 \rho_{fu}}{K_{\text{orif}}}}$$

(7.23)

Thus applying the known constants;

$A_{\text{inf}} = 3.14159 \times 10^{-8} \text{ m}^2$

$\rho_{fu} = 1,000 \text{ kgm}^{-3}$

Hence the K value arrived at is 1.18, very close to the expected value of 1.20 for an ideal radiused injector. As a check to see if the MkII engine was operating as the cold flow injector predicted, previously recorded data was examined. Amongst the data recorded was the kerosene tank pressure, the combustion chamber pressure and the fuel flow rate. While the chamber pressure transducer was accurately calibrated regularly, the fuel pressure sensor was not so rigorously checked. This is mainly because the fuel pressure was only used as an indication, the fuel flow and chamber pressure are the important factor for $C^*$ measurements. However it was felt the accuracy was sufficient to build up a statistical interpretation of the $K_{\text{orif}}$ value in the system.

![Figure 7.14: In system, injector characterisation](image-url)
Chapter 7: Theoretical and Numerical Models

Above in figure 7.14 is the graph of mass flow against the square root of pressure drop. The scatter on the best-fit linear line is attributable to the inaccuracy in the fuel pressure sensor, as well as the usual random scatter with the chamber pressure and flow rate. Obtaining the slope from the graph and compensating for four injectors, translates to a $K_{orif}$ value of 1.20, which agrees closely with the cold flow tests with water.

The fuel pressure set on the tank was assumed to be the same as that in the fuel manifold of the engine, and so the fuel flow rate will be a function of the pressure differential between the chamber and the tank according to equation 7.24

$$\dot{m}_{fu} = \frac{n_{inj} A_{inj}}{K_{orif}} \sqrt{\frac{2 \rho_{fu} (P_{fu} - P_C)}{}}$$  \hspace{1cm} (7.24)

Where;
$n_{inj} =$ Number of injectors (4)
$A_{inj} =$ Cross section area of injector m$^2$ ($3.14159 \times 10^{-8}$ m$^2$)
$K_{orif} =$ Head Loss (1.18)
$P_{fu} =$ Kerosene Tank Pressure (Nm$^{-2}$)
$P_C =$ Chamber Pressure, (Nm$^{-2}$)
$\rho_{fu} =$ Fuel Density kgm$^{-3}$ (800 kgm$^{-3}$)

If the mixture is at stochiometric O/F then for a given total propellant flow rate, the combustion chamber pressure will be a maximum. If the fuel pressure is increased, then the mixture will move into a fuel rich regime and the $C^*$ value will fall, thus for a given mass flow rate, the chamber pressure will also fall. However the lower chamber pressure allows more fuel into the chamber, consequentially raising the pressure due to the higher propellant mass flow rate. Thus an equilibrium pressure will be achieved that is on the fuel rich side of stochiometric. If however the fuel pressure is lowered from stochiometric, then again the chamber pressure will fall causing the fuel flow rate to increase but this time it will bring it back towards a stochiometric O/F ratio. Below in table 7.1 is a list of $C^*$ values against fuel mass flow rate for a constant oxidiser flow rate of 12.0 gs$^{-1}$. Although the oxidiser flow rate is affected by the chamber pressure, it is highly insensitive. From the empirical data, with a chamber pressure of 9.8 Bar and a HTP tank pressure of 30 Bar, the flow rate was 12.68 gs$^{-1}$. Increasing the chamber pressure to 17.96 Bar (due to ignition) reduced the flow rate to 11.30 gs$^{-1}$. Thus increasing the pressure drop by a factor of 1.7 merely increases the oxidiser flow rate by a factor of 1.1. Thus over small chamber pressure ranges the oxidiser flow rate can be assumed to be independent of $P_C$.  

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Chapter 7: Theoretical and Numerical Models

<table>
<thead>
<tr>
<th>Oxidiser Flow / gs⁻¹</th>
<th>Fuel Flow / gs⁻¹</th>
<th>O/F</th>
<th>C* / ms⁻¹</th>
<th>P₀ / Bar</th>
</tr>
</thead>
<tbody>
<tr>
<td>12.0</td>
<td>4.00</td>
<td>3.0</td>
<td>1403.7</td>
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</tr>
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<td>1512.1</td>
<td>18.05</td>
</tr>
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<td>2.40</td>
<td>5.0</td>
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</tr>
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<td>7.0</td>
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<td>1.00</td>
<td>12.0</td>
<td>1480.2</td>
<td>15.31</td>
</tr>
</tbody>
</table>

Table 7.1: Theoretical engine values as a function of fixed oxidiser flow and shifting O/F

<table>
<thead>
<tr>
<th>Oxidiser Flow / gs⁻¹</th>
<th>Fuel Flow / gs⁻¹</th>
<th>Fuel Pressure required/ Bar</th>
<th>O/F</th>
</tr>
</thead>
<tbody>
<tr>
<td>12.0</td>
<td>4.00</td>
<td>25.3</td>
<td>3.0</td>
</tr>
<tr>
<td>12.0</td>
<td>3.00</td>
<td>22.2</td>
<td>4.0</td>
</tr>
<tr>
<td>12.0</td>
<td>2.40</td>
<td>20.9</td>
<td>5.0</td>
</tr>
<tr>
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<td>2.00</td>
<td>19.6</td>
<td>6.0</td>
</tr>
<tr>
<td>12.0</td>
<td>1.71</td>
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</tr>
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<td>12.0</td>
<td>1.20</td>
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</tr>
<tr>
<td>12.0</td>
<td>1.09</td>
<td>16.2</td>
<td>11.0</td>
</tr>
<tr>
<td>12.0</td>
<td>1.00</td>
<td>15.7</td>
<td>12.0</td>
</tr>
</tbody>
</table>

Table 7.2: Fuel pressures required to maintain a range of O/F values

From the injector flow rate equation 7.22 and from the injector calibration run, it was possible to calculate the theoretical ΔP across the injectors. Thus this data coupled with the chamber pressure gives the theoretical tank pressure required to maintain the chosen O/F ratio, shown in table 7.2

A graph fuel pressure against O/F is shown in figure 7.15 along with the best-fit equation for the trace given in equation 7.25.
\[
O/F = -2.240 \times 10^{-3} P_{fa}^3 + 2.348 \times 10^{-1} P_{fa}^2 - 7.703 P_{fa} + 83.84
\] (7.25)

Differentiating equation 7.25 with respect to \( P_{fa} \) gives the rate of change of \( O/F \) with varying tank pressure.

\[
\frac{d(O/F)}{dP_{fa}} = -6.72 \times 10^{-3} P_{fa}^2 + 4.696 \times 10^{-1} P_{fa} - 7.703
\] (7.26)

Thus given the hypothetical condition that the engine is operating at an \( O/F \) ratio of 9:1, then the sensitivity of the \( O/F \) to the tank pressure is \(-1.56\) per Bar of kerosene tank pressure. Hence dropping the tank pressure by 1 Bar takes the mixture ratio from being oxidiser rich to stochiometric. This demonstrates the sensitivity of the fuel pressure is severe and essentially stems from the large \( O/F \) value required for the engine and the small pressure drop across the injectors. If the system was converted to a single 0.2mm injector, the pressure drop would increase. In this configuration the fuel pressure has to be much higher (40 Bar nominal) to compensate for the reduced number of injectors, however the differential of the best-fit equation shows that a sensitivity, at an \( O/F \) ratio of 9:1, of \(0.103\) \( O/F \) units per Bar of tank pressure. Hence the sensitivity over a small range has all but vanished.

\[
y = -2.240E-03x^3 + 2.348E-01x^2 - 7.703E+00x + 8.384E+01
\]

Figure 7.15: Shifting O/F as a function of kerosene tank pressure
This is a very important result when considering a pressure fed system as employed on a satellite. The regulator on board will be subject to an error around a mean value, plus the regulator cannot vent, so even if flight pressure is maintained accurately, solar heating of the propellant tank will raise the pressure. In a flight configuration with a nominal tank pressure of 20 Bar, and with a temperature range of between 0 to 30°C, a pressure variation of 2 Bar would occur, which translates into serious O/F shift for the four injector configuration. If the engine operated stochiometrically at 0°C, the O/F would shift to just 4.5:1 at 30°C. However empirical results have shown that having fewer injectors leads to significant reduction in C* performance. Thus the best method is to keep the 4-injector configuration and place a choke upstream of the fuel manifold. This raises the working pressure of the kerosene tank, but this is a minor hindrance in the pursuit of a static O/F, which is perfectly feasible.
Chapter 8

8 Flight Design and Implementation

8.1 Introduction
8.2 Flight Engine
8.3 Flight Oxidiser Tank
8.4 Propulsion System Configuration
8.5 Conclusions
8 Flight Designs and Implementation

8.1 Introduction

This chapter discusses the implementation of the hydrogen peroxide/kerosene bi-propellant rocket engine on a small satellite. A possible flight engine is discussed with reference to a flight-type engine, designated the MkV, designed for the purposes of demonstrating such an engine could be adequately packaged. The oxidiser tank is also analysed and the potential issues with thermal effects and the decomposition of the hydrogen peroxide identified. Finally the whole propulsion system is discussed with a mechanism to provide gas free propellants during the main burn and determination of the delta V achieved.

8.2 Flight Engine

The research engines MkII through to MkIV demonstrated that an efficient engine could be cooled using decomposed hydrogen peroxide. The test set up and configuration of these engines did not lend themselves to a flight design due to the instrumentation required. In addition the engines had to be dismantled regularly, which drove the design to large heavy flanges. This created an unacceptable geometry and mass for a flight system. Thus from the research, the MkV engine was designed to demonstrate that the low cost hydrogen peroxide/kerosene engine could be adequately packaged.

8.2.1 Design

The biggest issue regarding the design of a flight engine was the catalyst packs as these had to be incorporated into the injector assembly with minimal volume. Using a coaxially mounted catalyst packs drives the length of the whole engine to excessively high values, thus mounting perpendicular to the thrust axis provides a more compact solution. The MkV engine was designed to test the practicality of packaging the catalyst packs in such a configuration. This would determine if there are issues with the decomposed oxidiser being fed through a 90° bend. The chamber diameter was also widened to reduce the overall length while keeping the same L*. The nozzle was chosen to be a heavy copper nozzle operating in film-cooled mode. This was selected in preference to the Haynes 230 nozzle as the wider chamber diameter had an unknown effect on the degree of film cooling. As this engine was designed to test packaging as opposed to performance, the heavier but more robust copper nozzle was chosen. The MkV engine design is
Figure 8.1: MkV engine layout
Figure 8.2: MkV engine during sea level test firing

Figure 8.3: Conceptual design of the MkV space rated thruster
shown in figure 8.1. The total mass of the MkV engine was 930g and had a length of 100mm. The proposed extended space rated version with the 200:1 Haynes 230 expansion nozzle has a mass of 1,030g shown in figure 8.3. This makes the mass of a flight system far greater than an equivalent LEROS engine with a mass of only 224g. The MkV engine had screw joints and was thus not truly representative of a flight design, however in a flight system the joints would be welded, but the envelope of the engine would remain the same. The MkV engine is shown during a sea level burn in figure 8.2. The MkV engine performed as expected, with ignition and stable combustion very similar to the MkIV engine demonstrating the validity of the design. The engineering drawings for this engine are given in Appendix A.

8.2.2 Implementation

In a flight system, the method of preheating the catalyst packs with pulses of hydrogen peroxide is not desirable due to the lack of repeatability and indeterminate thrusts generated. Thus an electrical system that heats the catalyst packs to approximately 100°C would be the preferred option. As the catalyst packs are in thermal contact with the injector block, heating the packs also heats the assembly too. This is unfortunate as the quantity of heat required to raise the whole injector assembly by 100°C (assuming a 0°C start temperature) is 28kJ. A 100W heater with no losses would require 5 minutes of operation to achieve this, which is unacceptable. Thus the solution is to sleeve the catalyst packs with an electrical heater, over-wrapped with ceramic fibre insulation and then inserted into the manifold block. This allows just the packs themselves to be heated electrically. The total mass of silver used in the catalyst packs is approximately 40g, which with a 25W heater requires a warm up period of just 38 seconds. This does not impose an unacceptably high demand upon the electrical power system.

A thermal stand off between the engine itself and the spacecraft is required to reduce heat soak back into the structure. Due to the nature of the engine it is permissible to use large dribble volumes on the propellant feeds. Thus the final propellant control valves can be located inside the spacecraft, away from the hot engine. This relaxes the requirements upon the valves and hence drives the system to a lower overall cost. Traditional hypergolic engines require small dribble volumes to ensure both propellants reach the chamber at precisely the correct time. This is not the case in the hydrogen peroxide/kerosene engine due to the monopropellant period prior to ignition. From the experimental runs, the typical propellant pressures required for smooth operation of the engine was 40 Bar on the hydrogen peroxide tank and 20 Bar on the kerosene tank. This produced a chamber pressure of 18 Bar. The large pressure differences between the oxidiser tank and the chamber was partially due to the high catalyst pack ΔP and the deliberate choke placed upstream of the engine to decouple oscillations. A flight engine would require similar decoupling pressure difference and hence the HTP tank would nominally be pressurised to 40 Bar. As the

8-5
same regulator would be used for the oxidiser and the fuel, the fuel tank should also be pressurised to this high pressure. This would then be choked down through a visco-jet\textsuperscript{1} type device to obtain the correct flow rate. The research discovered that maintaining correct O/F during a burn was problematic due to the sensitivity to fuel pressure. The required choking consequentially reduces the sensitivity of the O/F to tank pressure. From §7.5 a variation in O/F of approximately 0.1 would occur per Bar of pressure change, which with a -30°C to 60°C range would result in an O/F shift of only ±0.6, well within limits. Setting up of the actual O/F of the engine requires calibration on the ground. With both tanks pressurised to the nominal flight pressure of 40 Bar, visco-jets can be placed in the fuel line to limit the flow and hence achieve stochiometric O/F performance.

8.3 Flight Oxidiser Tank

The research into the long-term storage of hydrogen peroxide demonstrated that the FEP polymer provided adequate compatibility and the capacity to passively vent the excess oxygen. The challenge would be to take the conceptual research phase and produce a flight tank.

8.3.1 Design

The flight oxidiser tank would be similar in construction to the tank tested in the research and would consist of a FEP liner and an outer stress bearing shell. The outer shell would be Titanium for reduced mass. The research tank used grooves cut into the tank for venting of the excess oxygen. A flight tank would use a porous liner in place of the grooves, such as open cell metal foam to avoid the stress issues and reduce the thickness of the tank wall. This area needs to be examined in detail, however it is primarily engineering development. When designing the tank it is important to select the correct thickness of liner to avoid excessive equilibrium pressure at high temperatures. The decomposition rate and permeability of the liner are both functions of temperature. The decomposition rate of the HTP can be described by equation 8.1 [PEROX HANDBOOK 67] and the permeability of the liner follows a temperature dependency as in equation 8.2 [DuPONT 02].

\textsuperscript{1} A visco-jet is a device manufactured by the Lee Company, USA, which is essentially a choke. Rather than having a single hole, multiple tortuous paths are made thus the same pressure drop occurs but with a lower risk of clogging.
Figure 8.4: Equilibrium pressure ratio as a function of temperature

\[ R = R_0 \cdot 2.3^{T/10} \]  \hspace{1cm} (8.1)

\[ \Pi = \Pi_0 e^{T/10.5} \]  \hspace{1cm} (8.2)

The ratio of the permeability to the rate of decomposition gives the ratio of the equilibrium pressure in the tank to the equilibrium pressure at \( T_0 \) conditions taken as 0°C. Figure 8.4 above shows the dependency of the equilibrium pressure upon the temperature. The upper temperature limit that the tank should be designed for is 60°C, where the equilibrium pressure will be approximately 21 times that of 0°C. As the tank is designed to operate at 40 Bar, this is the upper limit of the natural equilibrium pressure. If this value were higher then correct O/F of the engine could not be guaranteed. Thus the flight tank should be designed for an equilibrium pressure of 40 Bar at 60°C, which results in equilibrium pressure of 3 Bar at the nominal spacecraft temperature of approximately 20°C.

### 8.3.2 Implementation

While the tank design allows an equilibrium pressure to be generated, the concentration will decrease with time, which again is a function of the temperature of the propellant tank. With the concentration decrease, the density will also decrease. Table 8.1 shows the effect upon concentration and density after 1 year of on orbit storage in a 15-litre tank at various temperatures, assuming an initial concentration of 90%. Providing the temperature is kept below 40°C,
acceptable concentration reductions occur. However at 60°C a serious decrease in concentration would occur and if such a temperature was maintained, the propellant would be unsuitable for propulsion after 4 years of storage.

<table>
<thead>
<tr>
<th>Temperature /°C</th>
<th>Concentration loss per year</th>
<th>Density after 1 year /kgm⁻³</th>
<th>Density at BOL /kgm⁻³</th>
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</thead>
<tbody>
<tr>
<td>0</td>
<td>0.03%</td>
<td>1413.5</td>
<td>1413.7</td>
</tr>
<tr>
<td>16</td>
<td>0.13%</td>
<td>1396.1</td>
<td>1397.0</td>
</tr>
<tr>
<td>32</td>
<td>0.51%</td>
<td>1380.4</td>
<td>1377.7</td>
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<tr>
<td>40</td>
<td>1.01%</td>
<td>1366.5</td>
<td>1372.1</td>
</tr>
<tr>
<td>60</td>
<td>5.56%</td>
<td>1322.6</td>
<td>1351.4</td>
</tr>
</tbody>
</table>

Table 8.1: Concentration loss as a function of temperature

The launch vehicle operators demand that the propellant tanks on board the spacecraft must not vent while attached to the launch vehicle. Thus the passively vented hydrogen peroxide tank must be fully sealed during the launch phase of the mission. This would be achieved using a pyro-valve on the tank vent port. Prior to arriving on orbit, the valve would be closed and thus the pressure inside the tank would increase. Once safely separated from the launch vehicle, the pyro-valve would fire, allowing the tank to vent as desired. Thus the ullage volume in the tank must be large enough to accumulate the pressure rise when the tank is in the un-vented phase on the launch vehicle. If a 15-litre spherical tank is considered as the flight tank and the total time from filling to arrival on orbit is no more than 3 months at an average temperature no more than 32°C, the ullage volume required to prevent the nominal 40 Bar limit being exceeded is 312cm³ or just 2.1% of the tank volume. However if the tank was filled at 0°C and sealed and the temperature rose to 40°C, the density change would reduce the ullage volume. Thus the tank would have to be designed for a propellant density at the highest expected temperature. Hence a 15-litre tank should be filled with 20.153 kg of 90% hydrogen peroxide. This gives a nominal ullage volume at 16°C immediately after filling of 574cm³. When expressed as a percentage, the required ullage is only 3.8% of the total tank volume. It is advisable to make the volume larger to provide a safety margin and as such 5% tank ullage would more than satisfy the requirement.
8.4 Propulsion System Configuration

The research led to a design of a low cost hydrogen peroxide/kerosene rocket engine and a tank to store the oxidiser. The following section describes how this could be implemented on a 100 kg SSTL enhanced microsatellite platform to achieve a $\Delta V$ of $1.5 \text{ km/s}$. This is the approximate $\Delta V$ required for a lunar mission, starting in a GTO orbit and finally residing in low lunar orbit.

8.4.1 Flight Configuration

The packaging of propellant tanks in a micro satellite is always challenging and this is further compounded by the engine’s total O/F being 12.4:1 giving a volumetric O/F ratio of 7.1:1. With an overall Isp of 252 seconds and a desired delta V capability of $1.5 \text{ km/s}$ for a GTO to lunar orbit, the total mass of propellants required on a 100 kg spacecraft is 46.00 kg. Thus fuel mass is 3.43 kg while the oxidiser mass is 42.57 kg. Hence the total oxidiser volume would be 30.4 litres, while the fuel volume would be 4.3 litres. A standard SSTL enhanced micro satellite bus has an available footprint of 600mm square. Figure 8.5 demonstrates one possible arrangement of the

![Figure 8.5: Propellant and pressurant tank configuration](image_url)
tanks along with 16.4 litres of nitrogen pressurant. As was previously mentioned, the permeable tank design is wasteful on pressurant and the hence the tanks need to be larger than a conventional system. It is also envisaged that such a system would use cold gas for the attitude control system, which further increases the nitrogen requirement. Figure 8.6, above, shows the schematic for a flight system. Most of the system is similar to any hypergolic bi-propellant propulsion system,
however there are several crucial differences. As the tank is designed to permeate gas, the pressurant must be isolated from the tank in between burns. Thus there is a latch valve placed after the N$_2$ regulator and before the hydrogen peroxide tank. When a burn is required, the latch valve is opened and the tank pressurised to 40 Bar. Upon completion of the burn the valve is closed, thus preserving the pressurant. The remaining residual pressurant in the tank will bleed off over time to its nominal equilibrium value, dictated by the polymers' permeability and the decomposition rate of the hydrogen peroxide.

8.4.2 Gas Free Propellant Feed

In order to maintain a low decomposition rate in the propellant tank it is vital that the surface to volume ratio of the storage tank is minimised. Thus it is desirable to avoid the use of a PMD. Due to the nature of the engine this is possible and offers the ability to use simplified tanks. This does however come at the expense of more complicated operation of the hydrogen peroxide/kerosene thruster as compared to a hypergolic engine. The engine has to operate in monopropellant mode first to build up a high chamber pressure and temperature prior to fuel injection. This typically requires 0.25-0.50 seconds depending upon the degree of pre-heating performed. Without any form of PMD, the oxidiser lines will be filled with a mixture of propellant and pressurant. Upon opening the primary control valve, the mixture will be purged. This will provide an acceleration that will consequentially settle the oxidiser and fuel tanks. The transition from gas to liquid feed will be erratic, generating fluctuating decomposition temperatures. Smooth running of the catalyst pack can be determined from the thermocouple located in the engine. This measures the decomposition temperature and once this has reached above a threshold of 500°C and is no longer erratic, the Kerosene would be injected. The Kerosene tank requires no PMD or diaphragm due to the settling effect of the monopropellant burn phase. The injection of the fuel would occur automatically through the engine control unit (ECU) whose sole job is to monitor the performance of the engine, driven by the main on board computer (OBC). Hence three distinct phases of the engine would occur upon startup; Pressurant/monopropellant mode, pure monopropellant mode and finally bi-propellant mode. Shut down would be immediate and simply requires simultaneous closure of both primary propellant valves.

8.4.3 Delta V Determination

The fundamental problem with the staged approach to engine ignition is the variability of the length of the monopropellant period. Thus will vary depending upon the length of time to remove the gas bubbles from the oxidiser feed line. Thus the ΔV would not be known accurately until after the burn has been completed when orbital elements could be determined for the spacecraft.
An alternative system that is based upon an inertial navigation system is to utilise three orthogonal mounted accelerometers. Micro electro-mechanical systems (MEMS) technology permits such sensors capable of measuring down to micro ‘g’ accelerations to be packaged into volumes no bigger than a cube of side length 5cm [SICKER 02]. Thus during the entire burn period, starting with the monopropellant phase, the acceleration can be integrated numerically by the ECU to arrive at the current ΔV imparted. Once the pre-programmed total ΔV has been achieved the ECU shuts the engine down, thus achieving accurate burns.

8.5 Conclusions

The implementation of a hydrogen peroxide/kerosene rocket engine on board a small satellite is quite different to a regular hypergolic engine through several factors. Primarily the storage of the oxidiser such that the tank is allowed to permeate oxygen and the ignition sequence are the biggest differences. In addition to calculate the exact delta V imparted, simply timing the burn duration is not an accurate option, instead accelerometers would have to be used. This raises the complexity of the on board processing, but due to the modern sophisticated low power electronics, this should not present undue concerns. The system while being different to a traditional hypergolic propulsion system, is a practical solution to small satellite propulsion albeit one which trades ease of operation with cost of the overall system. The hydrogen peroxide/kerosene propulsion system is certainly more complex to operate than a commercial hypergolic system however the gain is a cheaper engine to manufacture, approximately 25% of a commercial engine, and much cheaper ground support equipment. As breathing apparatus and highly specialised transfer vessels are not required, a cost as low as 10% of the hypergolic GSE has been estimated.
Chapter 9

9 Conclusions and Further Research

9.1 Introduction
9.2 Thruster Research Conclusions
9.3 Further Thruster Research
9.4 Hydrogen Peroxide Tank Research Conclusions
9.5 Further Hydrogen Peroxide Tank Research
Chapter 9: Conclusions and Further Research

9 Conclusions and Further Research

9.1 Introduction

This final chapter describes the conclusions from the research and summarises the experimental results. As with every research project, there are several areas where further work is required to fully characterise the engine and tank designs. The further research topics are highlighted and the necessary work required to develop a flight engine and tank is discussed.

9.2 Thruster Research Conclusions

The research has demonstrated that the use of hydrogen peroxide and kerosene has the capacity to produce a density Isp performance comparable to small hypergolic thrusters even when using non-exotic constructional materials. Through the use of decomposed HTP gas film cooling the design of the engine remains relatively simple as precision-drilled components are avoided. This renders the cost of the engine considerably lower than a commercial engine. The MkV engine with the full expansion cone would cost less that £2,000 to fabricate. Specifically the research has demonstrated the following:

- A simple low cost injector was designed and this performed adequately well, returning a C* of between 90 and 92% of the theoretical value. The concept of shear flow atomisation is applicable to engines on this scale and offers a simple and low cost alternative to expensive precision-drilled injectors used in hypergolic engines.

- Oxidiser film cooling, with the gas injected at the head of the chamber, is not possible when combined with the low cost COTS injector. It was hypothesised that the un-burnt fuel impinges on the oxidiser rich layer near the chamber wall causing combustion and hence a failure of the coolant. An impinging jet injector more akin to a hypergolic engine may allow head injected oxidiser film cooling to function correctly.

- Through the use of a copper flow separator, oxidiser gas cooling was shown to be an effective way of cooling the parallel chamber region of the engine. Both nitrogen and decomposed hydrogen peroxide performed the coolant role. Due to the convective heat losses in the decomposed HTP cooled case, the power into the coolant was much lower than the calculated heat flux. However the nitrogen cooled cases allowed calculation of the convective heat loss coefficient, which produced consistent results with the HTP cooled case. The conclusion is the calculated heat flux is close to the actual heat flux and hence in vacuum conditions the thruster should operate as the theory predicts.
• The simple regeneratively cooled nozzle failed to work with hydrogen peroxide due to decomposition of the propellant on the warm copper. The nozzle performed well when fed with water, demonstrating that the thermal properties of the system are adequate and the concept of a simplified regeneratively cooled nozzle is valid. However the material incompatibility issue caused the failure. This can be resolved by coating the inside of the coolant channels with pure aluminium, thus removing the compatibility issues. The entire nozzle could not be made from aluminium due to the low melting point and inadequate thermal conductivity.

• The Haynes-230 nozzle operating in oxidiser film/radiation cooled mode performed well and demonstrated a remarkable oxidation resistance. When testing at sea level, the nozzle could be cooled with the oxidiser coolant flowing at just 2.5gs⁻¹. When corrected for the small convective losses and the higher coolant inlet temperature, a vacuum firing would require a coolant flow rate of 3.8gs⁻¹. Thus it can be concluded that oxidiser film cooling using decomposed hydrogen peroxide is perfectly feasible.

• The practical nozzle and chamber cooling experiments have demonstrated that an oxidiser film cooled engine is feasible for sea level tests. Additionally these results coupled with the theoretical analysis show that the MkIV engine operating with a main propellant flow rate of 12.3 gs⁻¹ and a coolant rate of 7.2gs⁻¹ would be adequately cooled in a vacuum. This would give an estimated vacuum Isp of 252s. The density Isp however would be 343s and hence fall between the LEROS LTT and LEROS 10 thruster performance. This demonstrates that for volume limited vehicles such as small satellites the research is of interest.

9.3 Further Thruster Research

The research into the use of hydrogen peroxide and kerosene in a low cost oxidiser gas film cooled engine demonstrated that such a design is practical and identified suitable nozzle and chamber materials. The research was by no means definitive and there are several areas where further research needs to be conducted. This work is both of academic and commercial interest if the engine was designed to fly.

• Vacuum firings of the MkIV engine need to be conducted to determine the effectiveness of the chamber cooled with decomposed hydrogen peroxide, with the convective air cooling mechanism removed.
• The flow separator needs to be instrumented with thermocouples embedded into the wall to determine an accurate $h_g$ value. The research performed merely used the theoretical value for $h_g$ and this needs to be confirmed through empirical tests.

• The actual $I_{sp}$ of the engine needs to be determined with vacuum firings on a thrust balance. This would determine the accuracy of the weighted $I_{sp}$ calculations, described in equation 3.8, and hence refine the theoretical model.

• The effect of using a wider chamber diameter with a film cooled Haynes 230 nozzle needs to be investigated. It is known that the RFC-230 nozzle used in the research works very well over a range of coolant flow rates. However in order to reduce the heat flux into the chamber, it needs to be widened. Hence a Haynes nozzle with a larger convergent section needs to be examined in a similar way to the RFC-230 experiments. As the diameter of the chamber increases, there will come a point where the flow rate required to cool the chamber equals the flow rate required to cool the nozzle and this will be the point where the engine is optimised.

• The Kelvin Helmholtz instability between parallel flows is believed to be the reason for the film coolant breakup in the nozzle. This effect can be modelled using CFD and could form an entire doctoral research programme.

• The asymmetrical coolant distribution of the throat temperatures needs to be investigated as this is a possible cause of off-axis vectored thrust. It is expected that cutting multiple helix grooves into the flow separator may solve this problem.

• A system to electrically heat the catalyst packs needs to be devised and while this is not a research topic in itself, it is an area that must be investigated before a true flight engine can be devised.

• Finally the flight engine that would emerge from this research must be tested over a full environmental test plan over a range of temperatures and propellant gas saturation levels. The heat soak back into the spacecraft would also need to be assessed before the engine could be expected to fly.

9.4 Hydrogen Peroxide Tank Research Conclusions

The research into the storage of hydrogen peroxide was particularly successful and demonstrated for the first time that liquid sealed storage of the oxidiser can be achieved without a continuously increasing ullage pressure. This presents for the first time a real practical solution for on orbit long term storage. Specifically the research has demonstrated the following;
Chapter 9: Conclusions and Further Research

- FEP and PFA polymers have similar compatibility with hydrogen peroxide and have been shown to be nearly as compatible as pure aluminium.

- Upon exposure of the PFA and FEP polymers to 90% hydrogen peroxide over a period of 4 months, their mechanical properties showed little change. In particular no discernable trend was identified with the FEP polymer. Thus it can be concluded that the FEP polymer makes an ideal liner for a hydrogen peroxide storage tank.

- The X-ray Photoelectron Spectrometry (XPS) surface analysis of the PFA and FEP polymer showed no sign of oxidation between samples immersed in 90% hydrogen peroxide and fresh samples. This again indicates the polymers are highly compatible with the oxidiser.

- Hydrogen peroxide that had been stored with either the PFA or FEP readily decomposed upon contact with the silver catalyst. The XPS spectra obtained were not significantly different to catalysts used to decompose the HTP stored in the bulk container. Thus it can be concluded that the catalytic decomposition of hydrogen peroxide is not degraded when the oxidiser has been stored in FEP or PFA lined tanks.

- It has been shown that the intrinsic oxygen permeability of the FEP and PFA polymer allows the concept of a passively vented hydrogen peroxide tank. A tank was designed to incorporate an FEP liner that demonstrated storage of hydrogen peroxide over an 18-day period. Crucially the pressure remained constant over the test and hence demonstrated the possibility of long-term liquid sealed storage.

9.5 Further Hydrogen Peroxide Tank Research

The culmination of the long-term storage research was the demonstration tank, which produced excellent results. However, this tank would require more research before it could be flown on board a satellite. The effects of radiation are essentially unknown on both the hydrogen peroxide and the compatibility of the polymer. Thus the following areas must be investigated;

- Effect of radiation upon the decomposition of high strength (90%) hydrogen peroxide. The increase in decomposition rate due to radiation must be quantified which can be simulated on the ground using a radiation source. This is particularly important when selecting the equilibrium pressure for the passively vented tank. Work has been performed in the past on the effect with low concentration hydrogen peroxide, but research on high concentration has so far not been conducted.
It is well known that the effect of radiation upon the fluoropolymers is an increase in cross-linking, which causes embrittlement. However it is not known if there is also a loss of compatibility of the polymer with hydrogen peroxide. Thus the compatibility with hydrogen peroxide of a series of samples exposed to varying degrees of absorbed dose should be assessed.

The effect of the vented oxygen from the tank needs to be determined. Ionising radiation will tend to convert molecular $O_2$ into atomic $O$, which can cause damage to the structure. The effect needs to be examined in detail. In addition there is the possibility that an oxygen scrubber can be placed on the vent of the tank. Thus all the evolved oxygen is then absorbed into the scrubber and none is expelled from the spacecraft.

The research failed to ascertain the liquid permeability rate of the hydrogen peroxide through the FEP and PFA. Thus an accurate test needs to be conducted to determine this factor. It is expected that the rate should be similar to water vapour.

The test tank used in the research provided useful information on the practicality of a permeable tank. The use of grooves cut into the tank walls for the purpose of venting makes the tank unacceptably heavy due to the thick walls. Thus an alternative leak path needs to be devised and the possibility of open cell metal foam is one possibility. Research needs to be conducted into this area to provide a satisfactory flight solution.

Finally a flight tank needs to be constructed and hydrogen peroxide sealed and stored for the duration of one year, with thermal cycling. After this period the HTP should be tested for concentration loss and then test fired in a live engine.
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"Investigation into a low cost Bi-Propellant Engine for Small Satellites"
Technical aspect accepted, necessary style changes currently in progress for publication

"A Novel Tank Design for the Long-Term Storage of Hydrogen Peroxide"
Currently in progress, due to be submitted to AIAA Journal of Propulsion and Power, Jan 03.
"BNSC Preliminary Bi-Propellant Research Report"

Phase I report submitted to the British National Space Centre as part of the contract to investigate green bi-propellant propulsion.

"BNSC Phase I and II Bi-Propellant Research Report"

Phase I and II report submitted to BNSC as part of the contract to investigate green bi-propellant propulsion.

Interview with Meridian Television describing the reasons for performing the research into Hydrogen Peroxide and Kerosene. The interview was shown during May 2001.
Appendix A

Engineering Drawings
SECTION A-A  SECTION B-B

DO NOT SCALE IF IN DOUBT ASK!

MATERIAL: STAINLESS STEEL
GRADE: 316
TEMPER:

UNLESS OTHERWISE SPECIFIED

GENERAL TOLERANCES
DECIMAL X /0.30 /0.50
XX /0.10
XXX /0.05

1. ALL DIMENSIONS IN MILLIMETERS
2. BEAR ALL EDGES & EXTERNAL CORNERS TO 0.2
3. ALL UNSPECIFIED RADII R2.5
4. REMOVE SHARP FROM ALL HOLES

ADDITIONAL OXIDISER MANIFOLD AND INJECTOR PARTS

SCALE: 1:1
STOCK NUMBER

SURREY SATELLITE TECHNOLOGY LTD

DRAWN IGC
APPROVED

DATE

REVISION

1

DATE

1 OF 1
20 x Ø0.8 EQU SPC HOLES THRU' ON 29.0 PCD

Ø3.2 HOLE THRU' C'BORED Ø6.4 x 3

SECTION A-A
ALL GROOVES ARE 60 DEGREE "V".
RADIUS OF BOTTOM OF "V" APPROX
73.3 FROM TANK GEOMETRIC
CENTRE. DETAIL B SHOWS GROVE
PROFILE. 14 CONCENTRIC CIRCULAR
GROOVES CUT NORMAL TO THE
WALL AT ANGULAR LOCATIONS
SHOWN OPPOSITE.

MAXIMUM RADIUS ON BOTTOM
OF "V" 0.2. POSITION OF
GROOVES NEEDS TO BE WITHIN
3.0 OF SPECIFIED LOCATION

SECTION A-A

DETAIL B

GEOMETRIC CENTRE OF TANK
LINER SHOWN IN FINAL CONFIGURATION WITH WELDED EXIT PORT IN PLACE

THICKNESS OF LINER IS EITHER 1.0 OR 1.5 NOMINALLY. DRAWING SHOWS 1.0 THICKNESS.