
Thomas Harle

Submitted for the degree of PhD from the University of Surrey

Surrey Space Centre
Department of Electronic Engineering
Faculty of Engineering and Physical Sciences
University of Surrey
Guildford, Surrey, GU2 7XH, U.K.

December 2013

© Thomas Harle 2013
Abstract

This thesis describes an experimental investigation into the performance of a radio frequency plasma thruster (RFPT). This type of thruster does not rely on the use of high voltage ion acceleration grids or beam neutralisers which are typically life limiting elements of an electric thruster. The RFPT excites an atomic or molecular gas into a plasma using an external RF antenna. This plasma is contained by a cylindrical dielectric source tube which is open at one end. An axial magnetic field is applied to the plasma using one or more solenoids or a permanent magnet source. The magnetic field is usually applied such that it is constant throughout the length of the source tube, after which the field diverges. The plasma couples with the imposed magnetic field, generating internal field structures, which can be used to regulate both the power coupling and the rate at which the plasma diffuses out of the source tube. In this study, the thruster performance is measured directly, thus providing missions designers with accurate performance data which can be used to assess applicability of the technology to future missions.

A re-configurable lab prototype thruster was constructed and a pendulum type thrust balance was developed in order to make direct measurements of the thrust produced by the Surrey Space Centre (SSC) RFPT. The balance has been developed to allow mounting of the thruster together with the necessary RF and DC electrical feeds as well as the propellant feeds and incorporates sensors which allow measurements to be made in an RF plasma environment. A low magnetic field mode of RFPT operation was investigated in order to assess whether this mode of operation may be able to provide performance increases at reduced imposed magnetic fields ($\leq 20$ mT), which could enable the serious consideration of this technology for future flight opportunities. The lowered magnetic fields used here may reduce the risk of interference with spacecraft subsystems and perturbation to the spacecraft orbit, when compared to many of the proposed plasma thrusters which use fields in excess of 40 mT.

Direct thrust measurements of an RFPT operating in a low field mode are presented as a function of the propellant flow, RF power and for two source tube lengths. The thrust is shown to peak at a field strength, $B_0$, which is demonstrated to vary with RF power and propellant flow. The peaks are also shown to correspond generally to peaks in the source plasma density, plasma potential and in some cases to exhaust ion beam current. Ion energy distribution function measurements show that strong beams are not present in thruster configurations which use a 170 mm length source tube but are shown to increase in strength when using a shorter 85 mm length tube.

The low field mode is shown to generally provide performance increases over the non-magnetised case by a factor of 4. Low field mode thrust measurements using the shorter 85 mm source tube and matching double saddle antenna revealed enhanced peak performance gains over the 170 mm length case, resulting in an increase of the thrust efficiency, $\eta$, by up to a factor of 15. The performance of the RFPT operating in this configuration is shown to produce enhanced or equivalent performance, when compared to similar state of the art configurations but at generally lower imposed magnetic fields. This particular optimisation may make possible a first flight opportunity for the RFPT as weak magnetic fields may be supplied easily by electromagnets. Electromagnets can be easily deactivated or field reversed to avoid interference with spacecraft subsystems and orbital parameters unlike permanent magnets which may be required to generate higher magnetic fields. Operation of the RFPT in the low field mode with an 85 mm source has also revealed new low field mode behaviour compared to the 170 mm case, producing sudden, large reductions in the performance of the thruster for imposed fields beyond $B_0$ which are shown to correspond to drops in source plasma density, plasma resistance and exhaust beam current.
Email: t.harle@surrey.ac.uk
Acknowledgements

I feel privileged and deeply humbled to have worked with such knowledgeable, interesting, talented and admirable people over the last few years. Dave and Andy, Bill, Raphael, Emily, Richard thank you for keeping the labs running in the face of so many difficulties. Bob, Richard, Phil and Gerald the parts you produced were beautiful. Karen, your unfaltering sunny disposition, kind words and legendary abilities to keep the Space Centre running are truly incredible. We would all be lost in space and time without you, thanks! Vaios, thank you for first giving me the opportunity to work at the Surrey Space Centre and for your supervision, we have come a long way since then! Sabrina, it was an absolute pleasure working with and learning from you, thank you for all your help. Thanks also to Phil Palmer, who’s advice is so potent and keen, it need only be administered in a sentence. Rod Boswell and Christine Charles for their invaluable expertise and advice, their incredible hospitality, the crash course in surfing, snorkeling and Jazz improve. Thanks also to the rest of the SP3 team who made me feel very welcome during my brief stay. Thanks also to Nick Braithwaite, Mark Bowden and Ziad Otel of the Open university for the many interesting discussions and tutelage in the science of hairpin probes and for loaning us our first RF power generator! Paolo Bianco and Howard Gray for the many interesting conversations and insights into the practicalities of spacecraft engineering and the space industries approach to EP. Matthew Perren and Robert Lainé of Astrium, for introducing us to the ANU and providing the funding for this project and for the many interesting discussions. Thanks to the SSC PhD and RA team for generally being awesome and for your support, especially Aaron who willingly read through these pages.

I would like to dedicate this thesis to my fiancée Nikki, my Father and my Mother, who all supported me so much through this project.
Publications

The studies conducted for this thesis have lead to the following publications:


Contents

Abstract i
Acknowledgements i
Publications i
Table of contents vii
List of figures x

1 Introduction 1
1.1 Electric Propulsion ................................. 1
1.2 Objectives ........................................ 6

2 Background 9
2.1 The Power Supply Penalty ............................ 9
2.2 Electric Propulsion Missions .......................... 12
    2.2.1 Station Keeping and Attitude Control Applications .......................... 12
    2.2.2 Graveyard Orbit Insertion From Geostationary Orbit ......................... 12
    2.2.3 Interplanetary Missions .................................. 15
2.3 Current Flight Technologies .......................... 16
    2.3.1 Resistojets ...................................... 16
    2.3.2 Arcjets .......................................... 18
    2.3.3 Microwave Electro-thermal Thrusters ..................................... 19
    2.3.4 Pulsed Plasma Thrusters ................................ 21
    2.3.5 Ion Engines ....................................... 22
    2.3.6 Hall Thrusters ..................................... 25
2.4 RF Plasma Thrusters .................................... 26
    2.4.1 Whistler Waves ................................... 26
    2.4.2 Helicon Waves .................................... 29
    2.4.3 Variable Specific Impulse Magneto-plasma Rocket (VASIMR) ................. 33
    2.4.4 Mini Helicon Thruster ................................ 34
    2.4.5 Helicon Plasma Hydrazine Combined Micro Thruster ......................... 35
    2.4.6 High Power Helicon Thruster ................................ 36
    2.4.7 Low/Medium Power Helicon Thruster (LPHT) ................................. 37
    2.4.8 Helicon Double Layer Thruster (HDLT)/HPT ................................. 38
        2.4.8.1 Current-free Double-layer Formation ................................. 38
        2.4.8.2 Low Magnetic Field Helicon Waves ................................. 40
## Contents

2.5 Summary .................................................. 41

3 Equipment and Procedures ............................... 43
  3.1 Thruster Design ............................................. 44
    3.1.1 Source Tube ........................................... 44
    3.1.2 Antennas ............................................. 44
    3.1.3 Solenoids ........................................... 47
    3.1.4 Propellant ........................................... 48
  3.2 Laboratory Facilities ..................................... 49
    3.2.1 The *Pegasus* Vacuum Chamber ...................... 49
    3.2.2 Volumetric Flow Controller ......................... 54
    3.2.3 Radio Frequency Power Generator .................... 54
    3.2.4 Matching Network .................................... 56
    3.2.5 Standing Wave Ratio Meter ......................... 57
  3.3 Probes and Diagnostics .................................. 58
    3.3.1 Retarding Potential Analyser ....................... 58
    3.3.2 Microwave Resonator Probe ......................... 62
    3.3.3 Langmuir Probe ..................................... 64
    3.3.4 RF Current Probe .................................... 66

4 Thrust Balance Development ............................. 69
  4.1 Experimental Arrangement ............................... 69
  4.2 Interferences and Mitigations .......................... 70
  4.3 Calibration and Data Analysis ........................... 74
  4.4 Cold Gas Measurements .................................. 77
  4.5 RFPT Operation ........................................... 77

5 Experimental Results ..................................... 81
  5.1 Thrust Measurements ..................................... 81
    5.1.1 Variation with Imposed Magnetic Field and RF Power .. 81
    5.1.2 Variation with Imposed Magnetic Field and Propellant Flow 83
  5.2 Plasma Density Measurements ............................ 86
    5.2.1 Variation with Magnetic Field and Axial Location: 170 mm Source Tube 86
    5.2.2 Variation with Magnetic Field, Axial Location and RF Power: 85 mm Source Tube 86
  5.3 IVDF Measurements ...................................... 89
    5.3.1 Variation with Magnetic Field and RF power: 170 mm Source Tube 89
    5.3.2 Variation with magnetic field: 85 mm source tube .......... 91
    5.3.3 Variation with RF power at $B_0$: 85 mm source tube .......... 92
  5.4 Summary .................................................. 94

6 Discussions ............................................... 97
  6.1 The SSC RFPT Low Field Mode ............................ 98
    6.1.1 Current Free Double Layer Formation ................. 98
      6.1.1.1 Thermal Ion Magnetisation .......................... 98
      6.1.1.2 Ion Beam Formation ............................... 99
      6.1.1.3 Plasma Density Characteristics .................... 100
    6.1.2 Low Field Helicon Waves .............................. 101
6.1.2.1 Plasma Density Peaks and Ion beam Formation ............. 101
6.1.2.2 The Behaviour of $B_0$ ............................................. 103
6.2 Performance Characteristics of the SSC RFPT ......................... 107
  6.2.1 Thrust Production in RFPTs ........................................ 107
  6.2.2 RFPT Performance Comparisons .................................... 112
6.3 Summary ................................................................. 117

7 Conclusions ........................................................................ 119
  7.1 Summary of Contributions ................................................ 120
  7.2 Recommendations for Future work ..................................... 121
List of Figures

2.1 Total available $\Delta V$ as a function of exhaust velocity $v_e$ for a spacecraft of 2000 kg dry mass (not including the power supply) and 200 kg of available propellant .................................................. 11
2.2 Diagram of a low thrust trajectory between two circular, coplanar ($\Delta i = 0$) orbits of radius $r_1$ and $r_2$, which could be the operational and graveyard orbit of a mission, respectively ................................................................. 14
2.3 The MR-501b Electrothermal Hydrazine Thruster (EHT) developed by Aerojet, picture courtesy of Aerojet Redmond ................................................................................................................................. 17
2.4 A schematic diagram of the ARTUR-2 arcjet thruster ................................................................................................................................. 18
2.5 A schematic diagram of the Ablative Pulsed Plasma Thruster ......................................................................................................................... 22
2.6 A comparison between typical gridded ion thruster acceleration grid geometry (above) and the modified dual stage, 4 grid arrangement used in the DS4G thruster (below) ........................................................................................................... 23
2.7 A cutaway view of the T5 hollow cathode thruster ............................................................................................................................................. 24
2.8 A cross-section of an SPT-series Hall thruster ............................................................................................................................................. 25
2.9 Sketch of the dispersion relation for electromagnetic waves travelling in an infinite magnetised plasma .................................................................................................................................................................. 29
2.10 A cutaway diagram of the VASIMR thruster showing the two plasma heating stages which may be throttled to vary the specific impulse ..................................................................................................................... 33
2.11 A photo of the mHTX thruster firing .................................................................................................................................................................. 35
2.12 Schematic of the HPH.COM project thruster concept. Original image from [1] ........................................................................................................... 36
2.13 A photo of the HPH thruster before during firing ............................................................................................................................................. 37
2.14 A photo of the LPHT showing the thruster mounted on the inverted pendulum type thrust balance .................................................................................................................................................................. 38
2.15 A schematic diagram of the Chi-Kung helicon thruster ............................................................................................................................................. 39
3.1 Photos of the 130 mm (a) and 85 mm (b) antennas coated in boron nitride .................................................................................................................. 45
3.2 Plot of electron density and imposed field required to satisfy the helicon dispersion relation. Plotted here for $\omega=13.56$ MHz, $r=22.5$ mm and antenna lengths 85 mm (squares) and 130 mm (circles) ...................................................................................................................... 46
3.3 Graphs showing the axial magnetic field as a function of the axial position (Z) for ‘both solenoids’ on (a) and ‘exhaust only’ (b) cases for a 1, 2 and 1.5 A current flow. The calculated fields are plotted as solid lines and the measured values are plotted as squares, diamonds and triangles. The vertical red line indicates the position of the source tube exit aperture and the grey boxes are representations of the antenna and its positioning. Profiles were generated from simulations created using the FeMM finite element analysis software package ........................................................................................................... 48
3.4 Photos of the SSC RFPT. The source tube supports were 3D printed using a first generation MakerBot ® ................................................................................................................................................................. 49
3.5 The *Pegasus* vacuum testing facility, used for all experiments presented in this thesis. ................................................. 50
3.6 Graph showing the calculated neutral gas pressure in the source tube as a function of the propellant flow for a downstream vacuum and when corrected for finite facility pressures. ......................................................... 52
3.7 Pressure of the *Pegasus* vacuum chamber as measured using two gauges ................................................................. 53
3.8 The Bronkhorst EL-FLOW® F-201CV-200-AAD-22-V volumetric flow controller used to supply propellant to the thruster throughout these investigations. ....................... 55
3.9 The front panel of the MKS *SurePower* RF power supply .................... 55
3.10 Photo of the matching network (a) and simplified circuit diagram (b). Note the blocking capacitors (brown in colour) and RF current probe (yellow). .......... 56
3.11 The standing wave ratio (SWR) meter used to monitor the forward and reflected power between the generator and the load. ............................................. 58
3.12 Schematic and biasing scheme of the retarding field energy analyser. The ground, $G_g$, repeller $G_r$, discriminator $G_d$ and suppressor grids are shown as broken lines. The final electrode is the collection plate. These grids and the collector are biased according to the sketched voltage plot. The blue arrow indicates that the voltage of the discriminator is increased with time. ............................... 58
3.13 Example RPA current voltage profiles. ........................................ 61
3.14 Examples of ion velocity distribution functions. ........................................ 62
3.15 Schematic of the hairpin resonator probe. The hairpin itself is mounted on a ceramic support which slides onto a glass outer casing which contained the coaxial cable leading out of the chamber. .................................................. 63
3.16 Sketch of the inverted reflection peaks shifting from $f_0$ to $f_r$ in the presence of a plasma. ......................................................... 64
3.17 Schematic of the Langmuir probe and circuit diagram. .......................... 65
3.18 The ‘Ion Physics’ RF current probe used to measure the current passing through the RF connections to the antenna. The output of the probe was then used to calculate the change in the plasma resistance. These trends were later used to identify the location of low field mode peak in the plasma density and thrust. . . 66
4.1 Schematic of the SSC pendulum type thrust balance, showing the installation position of the plasma thruster and the z-axis scale referred to throughout this document. The ‘fixed plate’ is fixed relative to the chamber and the moving plate swings freely thus allowing it to move in response to an applied force. In this schematic only one of the two antenna power feed rods can be seen. The second is parallel to the one shown and separated from the first by approximately 20 mm. 71
4.2 Example plot showing the effect on the thrust balance displacement of the solenoid feed lines when current was supplied to the solenoids. The solenoids are activated at $t=20$ s. This effect was mitigated entirely through careful rearrangement of the current feed lines and replacement of all magnetic components with non magnetic counterparts. This strategy entirely mitigated the effects shown here. .......... 72
4.3 Free body force diagram of the calibration system used to apply a known forces to the moving plate of the thrust balance. ................................. 74
4.4 Response of the thrust balance to applied calibration forces and the corresponding calibration curve. ......................................................... 75
4.5 Theoretical and measured cold gas thrust made using Krypton exhausted from a 6.35 mm diameter tube. The standard deviation of these measurements were calculated as 0.1 mN. This represents the accuracy of the balance used to determine the accuracy of the balance. The statistical deviation of the of the measurements were calculated at 0.1 mN ................................. 78

4.6 Thrust versus forward RF power for an argon propellant flow of 0.5 mg/s and a zero applied magnetic field. The theoretical cold gas thrust generated from 0.5 mg/s of argon is shown for comparison. ................................. 79

5.1 Thrust as a function of the imposed magnetic field for a range of forward RF powers and for source tube lengths of 170 and 85 mm. ................................. 82

5.2 Normalised plasma resistance as a function of the imposed magnetic field and RF power for a propellant flow of 0.5 mg/s and while using the 85 mm source tube. The voltage across the output of the RF current probe was measured while operating the SSC RFPT Eq. 3.22 was then used to calculate the plasma resistance given an estimate of the antenna resistance. This was then normalised to the peak resistance of the 500 W condition to allow comparison. The voltage measured across the output is linearly proportional to the current flowing to the antenna, thus dips in the voltage indicate a decrease in current to the antenna and an increase in the plasma resistance. In this way, $B_0$ could be quickly identified, allowing the highest performance points to be measured without the need to record the thrust across the entire low field mode. ................................. 84

5.3 Thrust as a function of the imposed magnetic field strength for a propellant flow rate of 0.6 mg/s and for 200, 300 and 500 W while using the 170 mm source tube length (a). Thrust as a function of the imposed magnetic field strength for a range of propellant flow rates and a constant forward RF power of 200 W, also for the 170 mm source tube (b). Thrust as a function of RF power while operating at $B_0$ while using the 85 mm source tube and propellant flow rates of 0.3 and 0.5 mg/s (c). ................................. 85

5.4 Hairpin resonator probe electron density measurements as a function of the imposed magnetic field strength for the 170 mm source tube, 200 W RF power and 0.5 mg/s and propellant flow. ................................. 87

5.5 Ion density measurements made using a Langmuir probe; positioned 20 mm, in the plasma plume as a function of the imposed magnetic field for a propellant flow of 0.5 mg/s and an input power of 200 W (a), as a function of the axial distance along the length of the source tube for an input power of 200 W and an applied $B_0$ field of 6.8 mT (b) and positioned -45 mm into the source tube as a function of the input power for a propellant flow of 0.5 mg/s (c) ................................. 88

5.6 Normalised IEDFs measured as a function of the imposed magnetic field strength while using the 170 mm source tube, a propellant flow of 0.5 mg/s and for RF power levels of (a) 200 W and (b) 400 W. The beam and local plasma potential are plotted superimposed upon the IVDFs with triangles and circles respectively. The total and beam current are shown below each plot for (c) 200 W and (d) 400 W. (graphs colour on-line) ................................. 90

5.7 Normalised IEDFs measured as a function of imposed magnetic field strength while operating the SSC RFPT with the 85 mm source tube, an RF power of 200 W and a propellant flow of 0.5 mg/s (a). The corresponding normalised ion beam and total collected current is shown in (b). (color online) ................................. 92
5.8 Evolution of the IVDF’s as a function of forward RF power while operating the RFPT at $B_0$ and using a propellant flow of (a) 0.3 mg/s and (b) 0.5 mg/s. The total current collected by the RPA and the ion beam current corresponding to each flow condition are plotted beneath. (color online)  

6.1 Magnetic field strength $B_0$ as a function of the RF power used to sustain the plasma while using the 85 mm and 170 mm source tubes. The propellant flow rate used in both cases was 0.5 mg/s.
Chapter 1

Introduction

1.1 Electric Propulsion

Spacecraft propulsion systems can be broadly divided into two categories; high thrust systems, used to lift payloads from the surface of massive bodies, and low thrust systems, used to reposition a spacecraft while in an orbital trajectory or in deep space. In this thesis we investigate one of the latter, low thrust, space borne systems. A space mission may require on orbit propulsion to complete a variety of manoeuvres from orbit maintenance to plane changes and orbital transfers. The propulsion system is used to apply forces to the spacecraft, thereby changing its velocity. This force is generated though the expulsion of material and can be expressed as:

\[ F = v_e \frac{dm_s}{dt} \]  \hspace{1cm} (1.1)

where \( m_s \), is the wet mass of the spacecraft (payload plus propellant) and \( v_e \) is the velocity of the ejected material or simply, the exhaust velocity. This force acts on the spacecraft,

\[ m_s \frac{dv_s}{dt} = v_e \frac{dm_s}{dt} \]  \hspace{1cm} (1.2)

where \( v_s \) is the speed of the spacecraft. This differential equation can be solved by separating the variables and integrating. The left hand side can be integrated over a small change in the spacecraft’s velocity from some initial velocity, \( v_i \), to a final velocity, \( v_f \). The right hand side can be integrated over a change in the spacecraft’s mass (as propellant is expelled) from an initial mass, \( m_i \) to a final mass \( m_f \).
\[ \int_{v_i}^{v_f} dv_s = v_e \int_{m_i}^{m_f} \frac{1}{m_s} dm_s \] (1.3)

Integration of eq. 1.3 yields the renowned Tsiolkovsky [2] or ideal rocket equation,

\[ v_f - v_i = \Delta V = v_e \ln \left( \frac{m_i}{m_f} \right) \] (1.4)

which describes how a spacecraft’s velocity changes as a result of the expulsion of material through its propulsion system. From Eq. 1.4 we see that in order to produce large instantaneous or a succession of small changes in a spacecraft’s velocity (\(\Delta V\)), which over the course of a mission is often sizeable, a large change in mass is required or a correspondingly high exhaust velocity. The former requires that a large reservoir of mass i.e. propellant/fuel, be stored on board the spacecraft, whereas the latter depends on the amount of kinetic energy that can be imparted to the exhaust gas. These considerations are summarised in the metric given in Eq. 1.5, the specific impulse, \(I_{sp}\),

\[ I_{sp} = \frac{F}{mg_0} = \frac{v_e}{g_0} \] (1.5)

where \(\dot{m}\) is the mass flow of the exhaust gases and \(g_0\) is the acceleration due to gravity at sea level. The specific impulse is used to compare the mass efficiency of rockets i.e. how efficiently the thruster produces thrust for a given mass flow rate. High thrust impulsive manoeuvres, such as the Hohmann transfer, typically employ chemical based propulsion systems. The performance of such systems is dependent on the extent to which chemical reactions in the fuel can transfer kinetic energy to the exhaust gases. This coupling of the chemical energy source to the thrusting mechanism limits the exhaust velocity of the gases and leads to a characteristically low specific impulse. Transporting large quantities of fuel into space burdens the launch vehicle and consumes large portions of the mass budget, thus displacing useful payload. This situation can be avoided if the exhaust velocity of the gases can be increased. This then reduces the required mass of fuel/propellant allowing increased scientific/commercial payloads to be launched. Correspondingly, for a fixed payload mass, the reduction in the wet mass can be used to simply reduce launch costs through reduction of the total launch mass. Furthermore, the increased \(I_{sp}\) and total \(\Delta V\) offered by such rockets can allow greater mission lifetimes and enable new classes of missions requiring low thrust high \(I_{sp}\) propulsion.

Electric propulsion (EP) systems essentially decouple the thruster’s energy source from the
1.1. Electric Propulsion

propellant supply, which is instead energised using an electric power supply, with the aim of creating a high energy gas or plasma. These high energy particles are accelerated and expelled through the use of electromagnetic body forces \((\mathbf{J} \times \mathbf{B})\), high potential ion extraction grids or simply by expansion through a nozzle. Since its first application in the 1960s, EP has been used on a range of missions. It has been an enabling technology for ambitious high \(\Delta V\) deep-space missions, such as the National Aeronautics and Space Administration (NASA) Deep Space One comet fly-by and the Japan Aerospace and Exploration Agency, Hyabusa asteroid sample return mission. Electric propulsion has also been used to enable highly innovative scientific missions such as the ESA GOCE gravity mapper where it was used to offset atmospheric drag in a super low (~250 km) altitude orbit. Within the commercial sector, longer mission lifetimes or reduced launch costs allow a greater economic return on the initial satellite investment. Consequently, the trend is now towards all electric, high \(\Delta V\) propulsion systems to provide earth orbit maintenance, such as North-South and East-West station keeping. This trend is exemplified by the introduction of the 702 line of satellites by Boeing and the current European Space Agency (ESA) Next Generation Platform (NGP) projects which aim to develop a competitive all electric geostationary (GEO) platform.

There are a number interesting trends currently emerging across the space industry, which will drive the space propulsion requirements of future missions. New rules are being introduced across Europe and the United States, requiring de-orbit capabilities to be built in to new spacecraft in order to manage the accumulation of man made space debris. A spacecraft must then retain a propellant reserve, or some other de-orbit mechanism to be used at the end of the satellite’s life. This new constraint modifies the mass, volume and economic trade-off which will be considered when designing a compliant satellite.

Low cost off-the-shelf components have driven the growth of the small low cost satellites market, pioneered by Surrey Space Technology Limited (SSTL). Commercial launches are now regularly provided by SpaceX which support the International Space Station as well deliver a range satellites into space. SpaceX and others have also proposed that the space economy could be expanded into the solar system through the harvesting of space resources, such as water from the moon and asteroids, as a step towards the colonisation of Mars by humankind. Here EP systems would be crucial to provide the high \(\Delta V\) necessary to efficiently move between bodies in deep space. Such ambitious missions require new ideas in propulsion if they are to be realised.

An example of such an idea might be the use of in-situ resources to provide fuel and propellent
for the chemical and EP systems. Such missions would not be limited by the amount of propellant or fuel initially launched with the spacecraft but would also need to be capable of also using the abundance of hydrogen, oxygen, carbon dioxide and other molecular propellants found across the range of extra terrestrial bodies.

Currently, a large range of EP systems have been used in orbit, with many thousands of hours of flight heritage. However, most current EP technologies possess non-ideal elements, as a result of their design and governing physics which make them poorly suited to these future space missions. Ion extraction grids, used by the Kaufmann type thrusters, are held at high voltages and used to extract and accelerate ions from a plasma reservoir. Such grids are prone to plasma erosion processes, reducing the efficiency over time and limiting the lifetime of the system. The ubiquitous hollow cathode is used to provide neutralising electrons to an ion beam on the vast majority of EP systems. These cathodes require a lengthy warm-up period and contain chemically sensitive elements which may be poisoned if either the cathode or the main plasma discharge is not formed of an inert gas. They also often contain heating elements which undergo many heating cycles and fatigue with age.

A more ideal electric thruster would comprise of a minimum of components and not require sensitive critical components which are exposed to plasmas. In order to be compatible with the exotic missions of the future, it would also be capable of using a broad range of atomic and molecular propellants, while providing the performance required by the specific mission at a high propellant efficiency. Many of these criteria are met by a class of thrusters known broadly as radio frequency (RF) plasma thrusters. This type of thruster consists, in its simplest form, of a dielectric source tube and an external antenna. The tube is flooded with propellant gas and the external antenna is used to radiate radio frequency power which ionises the propellant, thus forming a plasma. A magnetic field may also be imposed in order to manipulate the plasma as it exits the source tube. Ions are not extracted by high voltage grids in this scenario, but rather flow down a density gradient along with their counter part electrons. This produces an exhaust which self neutralises thus removing the need for a separate hollow cathode electron source.

The isolation of the antenna from the plasma, coupled with a mechanically robust and chemically inert source tube allows for the possibility of using a broad range of atomic or molecular propellants. Such a technology could enable the realisation of missions which would previously have been inconceivable, for example the use of hydrazine contaminated helium, used to pressurise chemical propulsion systems, as a propellant in a ‘last gasp’ de-orbit or graveyard orbit
1.1. Electric Propulsion

insertion system. Similarly, in-situ propellant usage during deep space missions could be seriously considered for the first time. As well as these exotic missions, RF plasma thrusters may also be used to provide a low complexity, robust and consequently long-life solution to more common place propulsion requirements, such as spacecraft orbit maintenance.

As a consequence of these possible benefits, RF plasma thruster research has become popular and a number of projects now exist to investigate their operation. Devices currently under investigation range from low power devices ($\leq 500$ W) to multi kilowatt behemoths and vary in their geometry, approach to plasma excitation and magnetic field configuration. However, in order to serve the ultimate end of application to in-space propulsion, experiments verifying the performance of thrusters under a range of operating conditions are required. Theoretical investigations into the performance of both low and high power RF plasma thrusters have provided interesting insights into the physics of these thrusters as well as some predictions of their performance.

Experimental investigations into high power RF plasma thrusters, such as the Vasimr described in chapter 2, have provided convincing performance measurements using indirect methods. These studies make a compelling argument for the use of this technology in deep space missions such as asteroid tours or missions to the outer planets. Unfortunately, the sheer size and complexity of such a high powered thruster, together with the supporting infrastructure, make experimental investigation practically inaccessible to all but the most well resourced institutions. Consequently, the majority of past and current projects, including the work presented in this thesis, are restricted to low power devices. The current interest and success in bringing the benefits of electric propulsion to small satellites has further fuelled these investigations and emphasized the need for rigorous performance characterisation.

At the time this project was started, little evidence existed to support the performance claims which had been made in the literature [3, 4, 5] which were based largely on plasma parameter measurements. It is only in the last few years that experimental investigations have produced reliable direct measurements of the performance of low power RF plasma thrusters. Many of these advances have been made as a result of work started in this project as part of a fruitful collaboration between the Surrey Space Centre (SSC) and the Australian National University (ANU), supported by EADS Astrium. Investigations carried out at the SSC and described in this thesis, in parallel with work carried out by our collaborators at the ANU, have now provided the first direct performance measurements of this type of thruster. These measurements have
opened the door to mission designers who may now consider this technology for future missions.

In this thesis, the performance of a radio frequency plasma thruster (RFPT) is measured directly while operating in a low magnetic field (6-12 mT) mode. The application of weak magnetic fields to RF plasma sources has been shown to induce mode changes which may increase the performance of an RFPT. Low magnetic fields $\leq 40$ mT can be easily generated by electromagnets which can be deactivated and field reversed if required, whereas strong fields $\geq 40$ mT are more easily provided by permanent magnets. From a satellite systems engineering perspective, the use of strong magnetic fields is undesirable, especially if generated by permanent magnets, as the fields may cause disturbances to the satellite orbit through interactions with the earth’s magnetic field and may also interfere with satellite subsystems. Experiments on the Chi-Kung device indicated that the formation of an ion beam was linked to the weak radial confinement of source plasma ions by a low axial magnetic field, depending on the radius of the dielectric plasma confinement tube [6]. Weak magnetic field mode changes in similar devices have been observed in helicon discharges, whereby the source plasma density has been observed to increase over a small range of magnetic field strengths [7, 8, 9]. The following objectives were set in order to assess directly the effect of low field $\leq 400$ G modes of operation on the performance of a low power ($\leq 500$ W) RFPT.

1.2 Objectives

1. Development of a pendulum thrust balance to directly measure the thrust of an RFPT.

Previous performance estimates of RF plasma thrusters had previously been limited to indirect methods and plasma probe measurements. In order to directly measure the thrust of the SSC RFPT, a thrust balance and measurement system was developed which is capable of rejecting both RF input power noise as well as plasma noise while providing a measurement precision of 0.1 mN.

2. Development of an RFPT designed to take advantage of a low magnetic field modes of operation.

Radio frequency plasma thrusters may be operated in a number of modes which can be accessed depending on the conditions of the RF plasma. Low magnetic field modes have been shown to efficiently produce high density plasmas at low powers and low magnetic
1.2. Objectives

fields. An RFPT was designed to operate within a low field mode so that the possible performance benefits could be assessed.

3. Direct thrust measurements of an RFPT operating in a low magnetic field mode.

The extent to which low field modes may be used to enhance the performance of an RF plasma thruster had been investigated through indirect methods, or through plasma probe measurements. The thrust balance and methods developed during the first part of this project are used to make direct thrust measurements of the SSC RFPT operating in a low field mode.

4. Direct thrust measurements of the SSC RFPT operating in a low magnetic field mode with a compact source tube.

The impact of a propulsion system on a satellites mass and volume budget is an important consideration for satellite systems engineers. Investigations into the effect of RFPT source tube length on the low field mode behaviour and the effect on performance had not been investigated previously. Direct thrust measurements of the SSC RFPT using an 85 mm source tube (compared to the 170 mm of objective 3) while operating in the low magnetic field mode were made in order to explore the possibility of an increased performance in a low magnetic field, low volume system.
Chapter 2

Background

In this chapter, a summary is given of previous studies related to the work conducted for this thesis. Firstly, some general comments are made on how the mass of an EP power supply scales with the thruster’s performance and how this affects the mission trade-offs mentioned in the previous chapter. A basic mission analysis is then presented for each of the three main missions for which EP is currently used. This is intended to serve as a basis for comparison between the various thrusters presented in this thesis, thus allowing the reader to put performance numbers into perspective through simple calculations. This is followed by an overview of the most common electric propulsion technologies, their performances and applications. Finally, a summary of previous and current RFPT projects and their main results are presented, together with a summary of the relevant mechanisms of power deposition and ion acceleration which are often associated with their operation.

2.1 The Power Supply Penalty

The fundamental driving force behind EP development is the increasing requirement for higher exhaust velocities at high thrust efficiency in order to reduce the launch mass of a spacecraft or enable high ΔV missions. In this case, the power being supplied to an EP system comes not from the propellant, but from the power supply unit (PSU) and solar panels. The thrust efficiency, \( \eta \), of an EP system is defined as the ratio of the power lost through the exhaust gases to the input electrical power. The power lost through the exhaust, \( P_e \) is given by,

\[
P_e = \frac{\dot{m}v_e^2}{2}
\]  

(2.1)
where \( \dot{m} \) is the mass flow of the propellant and \( v_e \) is the exhaust velocity. We can then define the thrust efficiency \( \eta \) as the ratio of the input power, \( P \), to the power lost through the kinetic energy of the exhaust particles,

\[
\eta = \frac{P_e}{P} = \frac{\dot{m} v_e^2}{2P} = \frac{T^2}{2\dot{m}P} \tag{2.2}
\]

The definition of the thrust, \( T \), given in Eq. 1.1, is commonly substituted into Eq. 2.2, which gives the commonly quoted form of the thrust efficiency. The power supply will have some characteristic specific power, \( \xi \), defined as the power which can be supplied per unit mass,

\[
\xi = \frac{P}{m_b} \tag{2.3}
\]

where \( P \) is the output power and \( m_b \) is the total mass of the power supply unit. This includes the mass of the solar panels, which are assumed here to grow linearly in area and mass as the power requirements increase. Incorporating this parameter into Eq. 2.2 and rearranging, we gain an expression for how the mass of the power system grows as the exhaust velocity is increased.

\[
m_b = \frac{\dot{m} v_e^2}{2\eta \xi} \tag{2.4}
\]

In this example we are expelling a fixed mass of propellant over a given burn time, \( t_b \), thus we can replace \( \dot{m} \) in Eq 2.4 with,

\[
\dot{m} = \frac{M_p}{t_b} \tag{2.5}
\]

At first, it might seem sensible to increase the exhaust velocity as much as possible in order to maximise the total \( \Delta V \) provided by a given volume of propellant, i.e. maximise the \( I_{sp} \). However, Eq. 2.4 shows that the mass of the power supply, \( m_b \), increases linearly with the exhaust velocity. To see how this affects the total \( \Delta V \) which can be provide by a rocket, we can incorporate an extra term within an expression for the dry mass of the spacecraft and insert this into Eq. 1.4, as defined in chapter 1. Let the initial mass, \( m_i \) and the final dry mass, \( m_f \), represent the mass of the spacecraft before and after using all the propellant in its tanks,

\[
m_i = M_s + M_p
\]

\[
m_f = M_s
\]
and

\[ M_s = m_s + m_b. \]  \hspace{1cm} (2.6)

\( M_p \), is the mass of the propellant and \( M_s \), is the dry mass spacecraft structure minus propellant and includes our expression for the mass of the power supply, \( m_b \). The rocket equation is now,

\[ \Delta V = v_e \ln \left( \frac{M_s + M_p}{M_s} \right) \]  \hspace{1cm} (2.7)

and \( m_s \), is the mass of the spacecraft minus the power supply unit. Using this modified rocket equation, the effect of the finite power supply mass on the total \( \Delta V \) supplied by a given propellant mass can be examined as a function of the exhaust velocity.

\begin{figure}[h]
\centering
\includegraphics[width=0.5\textwidth]{fig2.1.png}
\caption{Total available \( \Delta V \) as a function of exhaust velocity \( v_e \) for a spacecraft of 2000 kg dry mass (not including the power supply) and 200 kg of available propellant. The total burn time is fixed at 16 weeks. This example follows the geostationary (GEO) station keeping requirements presented in section 2.2.1 and summarised in table 2.1.}
\end{figure}

For this simple model, let’s fix the thrust efficiency, \( \eta \), and the specific power of the power supply, \( \xi \). Let us also fix the burn time, \( t_b \), taken to complete a general manoeuvre or set of manoeuvres, as well as the propellant supply mass. Figure 2.1 shows how the available \( \Delta V \) changes with increasing exhaust velocity. For increasing exhaust velocity, the available \( \Delta V \) increases to a maximum before decreasing. This can be interpreted as increasing the acceleration voltage of an idealised ion thruster. The performance gains begin to be offset by the increase in
the mass of the dry structure, lowering the mass fraction $m_i/m_f$ of Eq.1.4 and thereby lowering the available $\Delta V$.

2.2 Electric Propulsion Missions

In this section a basic mission analysis is presented for each of the main missions with which EP technologies are typically associated. The analysis is intended to provide reasonable estimates for the $\Delta V$ costs of these missions, and may be used to provide context to the performance parameters quoted in later section for the current state of the art thrusters. Equally, these estimates will contextualise the measurements of the SSC RFPT performance which are presented in Chapter 5.

2.2.1 Station Keeping and Attitude Control Applications

Station keeping and orbit maintenance are the most common functions for on-orbit propulsive manoeuvres. After the insertion of a spacecraft into the desired operational orbit, the mission lifetime depends on the ability to maintain the orbit. For GEO telecommunications satellites and sun-synchronous orbits, North-South and East-West orbit corrections are necessary for the duration of the mission to counteract orbital drift. A 2000 kg GEO satellite may need a $\Delta V$ of $\sim 50 \text{ m s}^{-1}$ per year [10] for North-South station keeping over a 15 year mission lifetime. A traditional chemical propulsion system with a specific impulse of 230 s would require 788 kg of propellant whereas, an RFPT with a specific impulse of 800 s would require only 200 kg of propellant to provide the equivalent $\Delta V$.

2.2.2 Graveyard Orbit Insertion From Geostationary Orbit

Guidelines are now in place to protect regions of space with commercial value from space debris. Satellites approaching the end of their operational lifetime should be capable of deorbiting or modifying their orbit to a stable ‘end of life’ orbit. For a GEO satellite, orbits must be raised to an altitude, at perigee, of 36,500 km. For low earth satellites the orbit must decay within 25 years. The Hohmann manoeuvre is typically used to transfer between orbits when using a chemical propulsion system. The simplest manoeuvre uses at least two high thrust impulsive burns to first move into an elliptical orbit and then to circularise the trajectory. Electric propulsion systems are typified by low thrust levels at high specific impulse and must use alternative
manoeuvres. The low thrust orbital transfer problem was first described by Edelbaum, who considered a transfer between two circular trajectories at a relative angle (inclination) $\Delta i$ with respect to each other through the application of one or more constant thrusting forces. Figure 2.2 below illustrates such a trajectory for the case of $\Delta i = 0$. The expression for the $\Delta V$ required for this manoeuvre was derived by Edelbaum [11] as,

$$\Delta V = \left( v_0^2 - 2v_1v_0 \cos \frac{\pi}{2} \Delta i + v_1^2 \right)^{\frac{1}{2}}. \quad (2.8)$$

where, $V_0$ and $V_1$ are the initial and final velocities of the satellite in the initial and final orbits. This expression simplifies in the case where the relative angle between the initial and final orbits is zero, i.e. $\Delta i = 0$, giving,

$$\Delta V = \left( v_0^2 - v_1^2 \right)^{\frac{1}{2}}. \quad (2.9)$$

This is simply the magnitude of the difference between the satellite velocities at each altitude ($r_1$ and $r_2$ in Fig. 2.2). The initial and final orbital velocities can be calculated from,

$$v = \sqrt{\frac{GM}{r}} \quad (2.10)$$

where $M$ is the planetary mass (Earth in this example) and $G$ is the gravitational constant. To complete such a manoeuvre, a constant thrusting force tangent to the orbit is applied, resulting in a constant acceleration. This allows the transfer time, $t$, to be calculated from,

$$t = \frac{\Delta V}{a} \quad (2.11)$$

where $a$ is the constant acceleration. This acceleration can be calculated from $a = T/M_s$, where $M_s$ is the mass of the spacecraft and $T$ is the thrust. We assume in this example, that the spacecraft acceleration is constant. In practice during the manoeuvre, the mass of the spacecraft is changing with time as propellant is expelled, and therefore the thrust applied would need to change over the course of the manoeuvre. The thrust and spacecraft mass used to calculate the acceleration can then be thought of as being averaged over the course of the manoeuvre.

Using Eq.2.9, the $\Delta V$ required to move a spacecraft from an initial altitude of 35,786 km to a graveyard orbit of 35,500 km is calculated as 257 m s$^{-1}$. A spacecraft weighing 2000 kg and using an RFPT with an $I_{sp}$ of 800 s would require 6.6 kg of propellant and take 21 weeks to complete. For comparison, a spacecraft completing a similar co-planar circularised Hohmann
type manoeuvre using a chemical engine\textsuperscript{1} with a $I_{sp}$ of 230 s would require 230 kg of fuel. This extra fuel would add to that required for the operational phase of the mission.

The extra fuel needed by a chemical thruster to complete a graveyard insertion manoeuvre, suggests an economic case for using an RFPT on a GEO satellite. The Helicon Double Layer Thruster (HDLT) \cite{4}, a type of RFPT discussed later in section 2.4.8, has been shown to operate on a wide range of propellants. Satellite providers who do not want to switch to an all electric system could add an RFPT into their system that uses residual fuel and pressurising gas (usually helium) as a propellant, providing a post mission disposal option. The cost of launching 1 kg of mass into a geostationary transfer orbit (GTO) is approximately €20,000 \cite{12}, power is available post mission and an RFPT system could weight less than 50 kg. While these numbers are only estimates, they are useful to illustrate how the RFPT in particular could help to reduce the financial impact of regulations imposed on satellite operators.

\textsuperscript{1}This calculation is based on the performance of EADS Astrium’s CHT 10 mono-propellant hydrazine thruster.
2.2.3 Interplanetary Missions

Once launched into an earth escape orbit, an interplanetary flight trajectory may be achieved through the use of gravity assisted manoeuvres alone. Long transfer times can place a significant strain on the satellite hardware, which must be capable of operation following extended exposure to the deep space environment. Chemical propulsion may be used with gravity assisted manoeuvres to reduce the transfer times by allowing high $\Delta V$ manoeuvres at any stage of the mission. Conversely, the significantly higher specific impulses allowed through the use of EP can allow higher payload mass fractions to be launched, while providing a capable propulsion system. The $\Delta V$ required to transfer from a low earth orbit (LEO) of altitude 2000 km to a circular, low Mars orbit of altitude 250 km may be calculated by again solving the Edelbaum equation. The orbit of the satellite is assumed to be circular within the LEO and the speed at apogee is assumed to be the sum of the velocity of the satellite about the Earth and the speed of the Earth about the sun. Similarly, it is assumed that the velocity of the satellite at the destination is the sum of the velocity of Mars about the Sun and the velocity of a body orbiting about Mars at an altitude of 250 km. It is also assumed that the orbit of Mars about the sun is circular and that the Mars and Earth orbits are coplanar.

<table>
<thead>
<tr>
<th>Mission</th>
<th>$\Delta V$ (m s$^{-1}$)</th>
<th>Thrust (mN)</th>
<th>Propellant Mass (kg)</th>
<th>Maneouvre Time</th>
</tr>
</thead>
<tbody>
<tr>
<td>Graveyard orbit insertion</td>
<td>257</td>
<td>10</td>
<td>6.6</td>
<td>21 weeks</td>
</tr>
<tr>
<td>GEO station-keeping/Yr</td>
<td>50</td>
<td>10</td>
<td>200</td>
<td>16 weeks</td>
</tr>
<tr>
<td>LEO to low Mars orbit</td>
<td>10000</td>
<td>100</td>
<td>405</td>
<td>3.1 years</td>
</tr>
</tbody>
</table>

Tab. 2.1: Summary of mission requirement estimates for the three most common missions for which EP is typically used.

Proceeding in this manner and using an array of 10 RFPTs with a combined specific impulse of 2000 s and thrust of 100 mN yields a $\Delta V$ value of 10 km s$^{-1}$, an orbital transfer time of 3.1 years and a propellant usage of 405 kg for a space craft mass of 1000 kg. This mission scenario would not be suitable for a manned flight owing to the long transfer times, however scientific missions and payload freight journeys would be ideally suited to this scenario. Table 2.1 compares the thruster performance requirements and maneouvre durations for the three scenarios discussed above. In each case the thruster is fired continuously. For on orbit propulsion tasks and the
interplanetary primary propulsion application, a spacecraft mass of 2000 kg and 1000 kg is assumed, respectively.

2.3 Current Flight Technologies

There are a large number of EP technologies currently in use. While the physical principles behind the operation of each technology varies, the primary motivation remains the same. That is, to provide thrust at a high specific impulse in order to make mass savings through efficient propellant usage. In the following sections, details of the performances of various flight qualified technologies are given, as well as a historical perspective.

2.3.1 Resistojets

Resistojets are electro-thermal thrusters which generate thrust through the resistive heating and expulsion of propellant through an exhaust nozzle. The relatively simple operating principle of the technology has allowed easy miniaturisation (~25 cm³) and application to small spacecraft. This has been of special interest to small satellite companies such as SSTL who have used resistojets extensively [13, 14]. The resistojet is commonly used together with catalytically decomposed hydrazine, due to the widespread use of hydrazine in chemical propulsion systems. Many other propellants have been explored, including H₂O, high test peroxide (HTP), N₂O [14] and NH₃ [15]; their selection being based on factors such as ease of catalytic decomposition, cost per kg and environmental concerns. The resistojet has a vast heritage of use starting in 1965 with the first successful space firing on board the ‘Vela’ nuclear detonation detection satellite, which had a 90 W nitrogen-fed ‘helical coil heater’ thruster with had an $I_{sp}$ of 123 s and thrust of 186 mN [2].

The American made MR-501b hydrazine resistojet thruster, developed by Aerojet Redmond and shown in Figure 2.3, utilised a peak input of 493 W, produced a thrust of 369 mN, an $I_{sp}$ of 303 s [16] and weighed 0.89 kg. A total of 124 MR-501b units were delivered into space and by 1995 there were 24 satellites using this model. The SSTL Mark-I N₂O resistojet has a 30 by 120 mm thrust chamber which was operated at 28 V and was designed to produce 24 W cm⁻³ around the heating element. After much development, an iteration of the Mark series, the Mark-IV, was used aboard the UoSAT-12 satellite, which was launched in April 1999. The thruster produced 125 mN of thrust, had an $I_{sp}$ of 127 s, consumed 100 W of power and
Fig. 2.3: The MR-501b Electrothermal Hydrazine Thruster (EHT) developed by Aerojet, picture courtesy of Aerojet Redmond

weighed 1.24 kg. Investigations into the feasibility and mission compatibility of H$_2$O as a Mark-IV propellant were conducted [14, 17] and it was flight qualified for usage aboard the first of the MightySat II family of satellites$^2$. Using de-ionised H$_2$O heated to 1000 K, the thruster generates 45 mN of thrust and an $I_{sp}$ of 152 s while using 100 W of power, demonstrating the appeal of using water for missions requiring higher specific impulse. In recent years, research efforts have begun on the feasibility of ‘micro resistojets’, a class of resistojet using the same basic principles as outlined above. Micro fabrication techniques are employed to create heating chambers and nozzles capable of delivering micro-newton thrust levels at higher $I_{sp}$ than can be achieved by traditional scaling of resistojet technology. The Free-Molecule Micro-Resistojet (FMMR) developed by the Air Force Research Laboratory, California has been characterised and simulated [18, 19] showing thrust levels of 129 $\mu$N and $I_{sp}$ of 79.2 s. While these levels of thrust would be unsuitable for most traditional small satellites <500 kg, the group assert the relevance of the FMMR to nano-satellites (1-10 kg) claiming that a 45 degree slew in 60 s at a thrust level of 0.3 mN could be attained$^3$.

The performance of the resistojet is limited by the extent to which the heating element can transfer thermal energy to the working fluid, which occurs through conduction from the thruster walls. Increasing the temperature further still reduces the number of materials which can reliably

$^2$These plans were subsequently abandoned in favour of using an advanced Pulse Plasma Thruster

$^3$Correct for a 10 kg cylindrical satellite of dimensions 14.50 cm diameter and 24.92 cm height giving a moment of inertia about the spin axis of 0.0263 kg m$^2$. 

be used to construct the thruster. Work has been done to address this specific problem, through
the development of high temperature alloys such as a Platinum-Thorium Oxide \([20]\). There is
however, a limit to the achievable \(I_{sp}\) of resistojets, at which point more effective energy transfer
mechanisms become favourable. The resistojet therefore marks the boundary between ionised
and non-ionised gas EP technologies.

### 2.3.2 Arcjets

![Arcjet diagram](image)

**Fig. 2.4:** A schematic diagram of the ARTUR-2 arcjet thruster \([21]\) developed by The Institute
of Space Systems (IRS), University Stuttgart as part of the ‘Small Satellite program’. Image
courtesy of The University of Stuttgart.

Arcjet thrusters, like resistojets, are an electro thermal technology which, instead of trans-
ferring thermal energy to the gas through surface heating, create an arc discharge through the
propellant allowing a DC current to be run through the ionised gas, which heats it further.
Thermal energy from the arc is transferred to the propellant through joule heating, raising the
temperature of the inner core of the gas in the arc chamber. The gas then reaches thermody-
namic equilibrium through convection, diffusion and conduction, before expanding through the
exhaust nozzle and creating thrust. The swift heating of the inner core of the propellant in the
discharge chamber allows the gas flow to achieve a higher temperature than the chamber walls,
thus allowing higher exhaust velocities than is possible by surface heating in the resistojet.

Initial iterations of the arcjet did not aim to control the arc discharge, which was allowed to
start and terminate at any point on the cathode and anode. During operation of the thruster,
the DC current generates a magnetic field which constricts the partially ionised gas column.
This self constriction was at first uncontrolled and strongly modified by current density, leading to a magnetic ‘pinch’ which in extreme cases led to the arc becoming severed. Several methods have been employed to prevent the magnetic pinch effect. By introducing the propellant at an oblique angle to the column, so as to introduce a ‘swirling’ motion, a vortex can form around the arc, which stabilises the arc column. Physical constriction of the arc using a long narrow arc chamber has also been used to stabilise the discharge and this method is implemented in the majority of modern arc jet thrusters. The MR-507 hydrazine arcjet developed by Primex Redmond generates 150 mN of thrust at an $I_{sp}$ of 465 s, uses 1.4 kW of power and gives a thrust efficiency of $\sim 30\%$ [22]. A range of thrusters based on the MR-507 have now been developed for a variety of operating ranges i.e. low voltage and power, high $I_{sp}$. These are summarised in table 2.2. A system using the MR-510 for example, consists of four thrusters, two active and two of which are redundant. Each thruster draws 2.2 kW of power and operates on average at $\sim 92\%$ efficiency. The thrusters generate 222-258 mN of thrust with an $I_{sp}$ of 570-600 s, thus fulfilling the need for a high power high performance thruster required by Lockheed Martin Corp’s A2100 series communication bus.

The ARTUR-2 arc jet developed by The Institute of Space Systems (IRS), University of Stuttgart, shown in Figure 2.4, is an ammonia arcjet designed to work in conjunction with a cluster of In-stationary Pulsed Magneto-plasmadynamic Thrusters (I-MPDS) to propel an all electric scientific spacecraft to the moon [21]. The arcjet has been shown to generate a thrust\footnote{Values recorded at a mass flow rate of 10 mg/s, performance figures quoted were found experimentally for hydrogen} of 82.25 mN and have an $I_{sp}$ of 816 s with an input power of 1 kW, which represents an exceptionally high specific impulse for an arcjet system.

2.3.3 Microwave Electro-thermal Thrusters

Microwave electrothermal thrusters (METs) use a microwave source to heat a high pressure gas volume which is expelled through a physical nozzle to create thrust. The technology is interesting to operators as it is possible to achieve very high power transfer efficiencies to the working fluid through careful design of the resonant cavity. The mechanism of power deposition in the gas volume has been described as a three stage process [23]. Electrons are heated by incoming microwaves, which then collide with neutral particles, leading to rotational and vibrational excitation. The excited neutral particles then scatter off each other and thermalise in the bulk.
### Tab. 2.2: MR-series Thrusters

<table>
<thead>
<tr>
<th>Thruster</th>
<th>$I_{sp}$ (s)</th>
<th>Thrust (mN)</th>
<th>Input Power (kW)/ Voltage (V))</th>
<th>Flight Status</th>
</tr>
</thead>
<tbody>
<tr>
<td>MR-507</td>
<td>465</td>
<td>150</td>
<td>1.4</td>
<td></td>
</tr>
<tr>
<td>MR-508</td>
<td>502</td>
<td>230</td>
<td>1.8 / 65-96</td>
<td>Flight qualified (1991) for use on the Martin Marietta Astro Space communication satellite, 28 produced, 6 flown</td>
</tr>
<tr>
<td>MR-509</td>
<td>502</td>
<td>250</td>
<td>1.63/ 65</td>
<td>Flight qualified (1997) for use on the Lockheed Martin Corp’s series 7000 communications bus, designed as a low power unit</td>
</tr>
<tr>
<td>MR-510</td>
<td>600</td>
<td>250</td>
<td>2.2 (per thruster)</td>
<td>Flight qualified (1997) for use on the Lockheed Martin Corp’s A2100 communication bus</td>
</tr>
<tr>
<td>MR-512</td>
<td>497-522</td>
<td>225-246</td>
<td>1.78/ 31-50</td>
<td>Flight qualified (1998) for use on the NASA Data Relay Test Satellite (DRTS) designed for flexible $I_{sp}$ and thrust while requiring low input voltages</td>
</tr>
<tr>
<td>A/B</td>
<td>/462-492</td>
<td>/262-286</td>
<td></td>
<td></td>
</tr>
</tbody>
</table>

In this case, the heated neutral particles provide the largest contribution to the thrust by design. This allows for the use of a range of propellants including Ar, N$_2$ and H$_2$O, as the exhaust gases are neutral/quasi-neutral and therefore do not require a hollow cathode neutraliser, which would be damaged by most molecular propellants.

The MET is similar to the SSC RFPT while operating without an imposed magnetic field in a few ways. High frequency power is used to ionise the propellant, they can both operate using a range of propellants and do not require hollow cathode neutralisers. Unlike the SSC RFPT, however, METs are typically operated at high pressures, on order 10 kPa in the plasma cavity,
in order to achieve stable and efficient operation of the microwave heating stage. This required high pressure leads to a high throughput of propellant and a low specific impulse compared to mainstream EP technologies, such as the Hall effect thruster. Few direct measurements of the performance of such thrusters have been published, however a thrust efficiency of 35% at an $I_{sp}$ of 250 s was measured at NASA Glenn \[23\] while running a 1 kW MET using helium and nitrogen.

### 2.3.4 Pulsed Plasma Thrusters

Pulsed plasma thrusters (PPT) create thrust by discharging an arc across two electrodes in close proximity to a propellant material, often Teflon, which is ablated and ionised. The ionised propellant is then accelerated along the electrodes by a $J \times B$ force produced by the arc current and the self-generated magnet field. Pulsed plasma thrusters are designed to operate in a pulsed mode with each pulse being approximately 10 $\mu$s duration generating very small thrust levels ($< 2$ mN). The pulsed operating mode of the PPT requires large amounts of energy to be discharged in a short time, usually using a bank of capacitors. The PPT has been used on a number of spacecraft to date (see ref. \[24\] and references therein), with the first use being aboard the Russian built ZONDA-1 satellite, for use in orbit raising and life extending missions as well as primary propulsion uses and fine tuned attitude control. The MightySat II mission was initially to use a resistojet technology, however an advanced PPT demonstration found favour instead. The thruster, developed by Olin Aerospace Company under contract with NASA Lewis Research Centre, operated on a nominal 100 W of power and could produce an $I_{sp}$ of up to 1200 s. The pulsed mode of operation allows a range of impulses to be generated depending on the requirements of the mission over time, simply by increasing or decreasing the frequency of impulses.

The Earth Observing 1 (EO-1)\[25\] mission used PPT technology capable of generating an $I_{sp}$ of 650 s at 12 W of input power, variable up to 1400 s at an input power of 70 W. The low power inputs versus specific impulse make the thruster an attractive option for small low power satellites requiring fine tuned control. The ability to vary thrust levels allows for the PPT to take a more dominant role in the control of spacecraft, rather than needing a plethora of thrusters all rated at separate thrust levels.
Chapter 2. Background

2.3.5 Ion Engines

Ion engines or Kaufman type thrusters create a reservoir of ions which are focused and accelerated using magnetic fields and an array of grids positioned at the thruster’s exhaust. Modern ion engines also include a hollow cathode electron source to neutralise the outgoing ion beam. Ion thrusters can achieve higher $I_{sp}$ than electrothermal devices owing to the use of electrostatic body forces to directly accelerate the ions. The top half of Figure 2.6 illustrates a typical grid arrangement.

Ion thrusters have traditionally formed the basis of EP development in the UK with the T5 and T6 being at the forefront of research. Development of the T-series thrusters began in 1962 when the Royal Academy of Engineering (RAE) started an EP research program with the aim to develop thrusters to provide attitude control and station keeping [27]. The initial design, which became the T-1 thruster, was a simple gridded ion thruster using mercury as a propellant. Mercury was vaporised using a porous tungsten filament which could control the flow into an acceleration chamber. The ions were accelerated using a single grid biased at 1.5 kV. It was at this time that a hollow cathode electron source was added at the exhaust aperture in order to neutralise and prevent ions interacting with and charging the spacecraft. This arrangement gave a specific impulse of $\sim 3000$ s, nearly 6 times the specific impulse generated by contemporary arcjet systems.

The design of the thruster has been developed heavily over the last 40 years culminating in the modern T5 and T6 thrusters. These thrusters use different variants of the hollow cathode known as the T5 and T6. The T5 cathode, pictured in figure 2.7, and the T6 operate in the same way, with variations only in performance. The propellant is passed through a heated
2.3. Current Flight Technologies

Fig. 2.6: A comparison between typical gridded ion thruster acceleration grid geometry (above) and the modified dual stage, 4 grid arrangement used in the DS4G thruster (below). Original images from [26]

cylinder of thermionically emissive material, termed the ‘insert’. A voltage is then applied to an external electrode initialising a discharge, at which point the heater may be deactivated as ion bombardment of the dispenser insert is sufficient to maintain the thermionic emission temperature. Once ions have been created, an external magnetic field enhances ionisation and directs the ions, through cyclotron motion, towards an array of grids at the thruster aperture. The grids extract and accelerate the ions from the plasma and a second external hollow cathode (usually a second T5/T6 cathode), which creates an electron beam to neutralise the exhaust. The deceleration grid shown in the top half of figure 2.6 is held at a potential sufficient to stop the neutralising electrons from migrating past the acceleration grid and interacting the ion source.

The thruster grids and cathode inserts are continually eroded by the ions that they come in contact with. This processes is known as sputtering. While this process limits the life time of an ion thruster, system [29], nominal operation of the T5 and T6 thrusters has been demonstrated when cathode inserts are severely (25-50%) depleted. Such is the confidence in the T5 thruster that they have been used on high profile missions such as ESA’s Gravity field
Chapter 2. Background

Fig. 2.7: A cutaway view of the T5 hollow cathode thruster [28].

and steady-state Ocean Circulation Explorer (GOCE). The T5 thruster, used in this mission for drag compensation [30], generated between 0.6-20.6 mN of thrust, had an $I_{sp}$ of 500-3500 s and drew 55-585 W across the thrust range. The ability to throttle the thrust with a resolution of 12 $\mu$N allowed fine tuning of the satellite speed to account for atmospheric drag experienced within a 250 km orbit.

The Dual-Stage 4-Grid Ion Thruster (DS4G) was proposed by Fearn [31] and was built and characterised by the Australian National University, Space Plasma and Propulsion group. The thruster concept was inspired by neutral particle beam injectors, used in controlled fusion systems, which typically generate particle energies of 80-100 keV. The DS4G uses a modified grid arrangement which effectively decouples ion extraction from the bulk plasma from the acceleration phase, allowing the extracted ions to be focussed before they are accelerated (shown in comparison with standard geometries in Figure 2.6). This results in higher acceleration efficiencies and a decrease in the sputtering rate of the acceleration grids, thus extending the life of the thruster. The thruster was demonstrated to produce 2.7 mN of thrust [26], have an $I_{sp}$ of 14000 s while operating at 300 W of input power and have a 63% thrust efficiency. The impressive specific impulse of the engine has led to suggestions that the thruster may be suitable for propulsion technologies involved with manned/robotic missions to the moon or Mars, as well as scientific missions to the outer solar system [31].
2.3.6 Hall Thrusters

Hall thrusters have become one of the most successfully implemented EP technologies to date, due in part to their extensive development in Russia over the last 30 years. These thrusters come in two types, the Stationary Plasma Thruster (SPT) and the Thruster with Anode Layer (TAL). In general, all variants of the Hall thruster share a common operating principle, with most modern advances being made in life time extension (sputter reduction) and thrust/power optimisation.

With reference to Figure 2.8, electrons produced at an external hollow cathode are drawn into the main chamber by an axial electric field, $E$. Once in the cylindrical channel, the electrons orbit the imposed, radial, magnetic field lines $B$. The electrons are also subjected to an $E \times B$ force which drives an azimuthal Hall drift i.e. around the thruster’s axis of symmetry. The combination of these motions drive the ionisation of neutral particles, which are fed from the anode. The imposed magnetic field is strong enough to weakly confine the electrons within the ionisation chamber, the ions, however, are accelerated out of the ionisation chamber by the axial electric field, forming the exhaust plume.

![Figure 2.8: A cut away diagram of an SPT-series Hall thruster. Original image from [32]](image)

The most extensively used SPT series thrusters are the SPT-100, with 56 units being used for spacecraft positioning and North-South and East-West station keeping between 1995-2001. The SPT-100 generates 80 mN of thrust at a specific impulse of 1600 s when consuming a power of 1.35 kW. The design included modifications arising from experience of previous SPT units, whereby the magnetic field geometry is modified to peak outside the aperture of the
thruster using a so called ‘magnetic shunt’. This modification has been shown [33] to reduce ion erosion of the chamber, thus extending the life of the thruster from 3000 to 9000 hours. High power Hall thrusters for use in orbit transfers are also under development. The SPT-290 has been demonstrated to produce an impressive 1.1 N of thrust at 20 kW of input power [34] and the 50 kW TAL thruster, developed at the Central Scientific Institute for Machine Building in Korolev Russia, produces a thrust of up to 2.5 N [35].

2.4 RF Plasma Thrusters

Radio frequency plasma thrusters (RFPT) are a relatively new class of thruster which have received increasing attention over the last 10 years. The thrusters generally operate by exciting an atomic or molecular gas into a plasma using an RF antenna. This plasma is generally bounded by a cylindrical dielectric source tube which is open at one end. An axial magnetic field is applied to the plasma using one or more solenoids or a permanent magnet source. The magnetic field is often applied such that it is constant throughout the length of the source tube, after which the field strength weakens. The plasma interacts with the imposed magnetic field, generating internal field structures, which can be used to regulate both the power input and the rate at which the plasma diffuses out of the source tube. The helicon wave sustained plasma is a popular and efficient method of creating a plasma and is introduced in the sections following. The various RF plasma thrusters which are or have been under investigation are then presented, with special attention being given to the Helicon Double Layer Thruster (HDLT) as it is both the most active and most relevant to the study presented in this thesis.

2.4.1 Whistler Waves

The helicon wave mode is a common feature among RF plasma thrusters as it has been shown to very efficiently generate high density plasmas [36]. Helicon waves are part of a class of plasma waves known as ‘whistles’ which were first encountered in World War I as an audible artefact picked up through military amplifiers, used in attempts to listen in on the opposition’s radio communications. Theories of the origins of these signals were developed most coherently by Storey [37] and form the basis for the modern understanding of the phenomenon. Lightening strikes create a localised perturbation in the ionosphere, which excites a broad spectrum of waves that propagate out through the plasma. The waves propagate at different velocities depending
on their frequency as a result of the frequency dependence of the plasma’s refractive index. These electromagnetic waves would be picked up by the communications equipment and sound like ‘whistles’ which descended in pitch as the various frequency components arrived.

The dispersion of such waves through the plasma is described by a dispersion relation, which relates the wave’s frequency to the plasma’s refractive index and thus to the various characteristic frequencies and properties of the plasma. For low frequency waves travelling through the Earth’s weakly magnetised ionosphere, the dispersion relation is given by,

\[ N^2 \approx \frac{\omega^2}{\omega_c \cos \theta - \omega} \]  

(2.12)

where \( N \) is the refractive index, \( \omega \) is the frequency of the propagating wave and \( \theta \) is the angle between the wave propagation vector and the magnetic field vector. The plasma frequency, \( \omega_{pe} \) is given by,

\[ \omega_{pe} = \sqrt{n_e q^2 / m_e \epsilon_0} \]  

(2.13)

where \( n_e \) is the plasma electron density, \( q \) is the electron charge, \( m_e \) is the mass of the electron and \( \epsilon_0 \) is the permittivity of free space. The electron cyclotron frequency, \( \omega_{ce} \), is the rate at which electrons orbit imposed magnetic field lines and is given by

\[ \omega_{ce} = \frac{|q| B}{m_e} \]  

(2.14)

where \( B \) is the imposed magnetic field. When the denominator of Eq. 2.12 is zero, \( N^2 \to \infty \), indicating that a resonance is occurring. This condition is met when,

\[ \theta_{res} = \arccos \left( \frac{\omega}{\omega_{ce}} \right) \]  

(2.15)

where \( \theta_{res} \) defines the angle of a cone around the imposed magnetic field within which the wave phase velocity is contained at resonance. For the plasma source used for this thesis, \( \omega_{ce} \approx 1.75 \text{ GHz} \), which gives \( \theta_{res} \sim 90^\circ \). In fact, the energy of the whistler waves has been shown to propagate only within a much narrower cone of \( \sim 19^\circ \) about the magnetic field vector and therefore essentially follow the earth’s magnetic field lines [38].

Early studies of electromagnetic wave propagation through magnetised plasmas in free space [39] paved the way to understanding the propagation of whistler waves in plasmas constrained by a physical boundary in laboratory plasmas. These waves were dubbed ‘helicons’, owing to the
helical pattern traced by the rotating $B$ field vector in the propagating wave. Research into the structure and propagation of helicon waves led to the discovery that through exploiting resonant helicon modes, plasmas can be generated much more efficiently than is the case for capacitive and inductive plasmas. Such studies were pioneered by Boswell [36], who produced a near 100% ionised plasma with an electron density above $10^{12}$ cm$^{-3}$ using an input power flux of less than 5 W cm$^{-2}$ and a static magnetic field of 100-758 G by launching helicon waves using a double saddle field antenna. Since that time, helicon waves have received much attention in both the development of the wave theory and the design and characterisation of a diverse range of helicon sources.

The general dispersion relation for waves travelling in a cold magnetised infinite plasma, parallel to the imposed magnetic field, is given by [38],

$$N_{L/R}^2 = 1 \pm \frac{\omega_{pe}^2}{\omega_{ce}^2 \omega \left(1 - \frac{\omega^2}{c^2}\right) \left(1 + \frac{\omega_{ci}}{\omega} - \frac{\omega}{\omega_{ce}}\right)}$$  \hspace{1cm} (2.16)

where $N_{L/R}$ is the left hand (LH) and right hand (RH) polarised wave refractive index, $v$ is the wave velocity and $c$ is the speed of light. The splitting of the dispersion relation into LH and RH branches is a result of the cyclotronic motion of charged particles about the imposed magnetic field, which causes a circular polarisation of the waves. A sketch of these functions [40] is shown in Figure 2.9.

The relative magnitudes of $\omega_{ce}$, $\omega_{ci}$ and the driving RF frequency $\omega$ governs the generation of the various wave modes which a magnetised plasma may support. With reference to Fig. 2.9, the helicon wave is a RH polarised wave which exists in the frequency range $\omega_{ci} \ll \omega \ll \omega_{ce}$. In this limit, and for $\omega_{ce} \ll \omega_{pe}$ (low imposed magnetic fields) and $\omega \ll \omega_{pe}$ (low frequency waves) the general dispersion relation given by eq. 2.16 becomes,

$$N^2 = \frac{k^2 c^2}{\omega^2} = \frac{\omega_{pe}^2}{\omega \omega_{ce}}$$  \hspace{1cm} (2.17)

The application of fixed boundary conditions, such as through confinement of the plasma within a dielectric source tube, modifies Whistler wave propagation. Where, in free space, Whistler waves only occur as right-hand (RH) circularly polarised waves, helicons may propagate as both RH and left-hand (LH) polarised waves.

Both RH and LH circularly polarised electromagnetic waves travelling in a vacuum produce a displacement current $J_d$ which acts in the direction of the electric field vector, thus re-generating
2.4. RF Plasma Thrusters

Fig. 2.9: Sketch of the dispersion relation eq. 2.16 showing the limits of the Alfven (longitudinal) wave mode and the helicon mode. The solid line represents the left hand polarised wave and the dotted line the right hand polarised wave. Branches of the dispersion relation which lie below the ω axis have a negative refractive index and can not propagate. Thus the left hand polarised waves in the range ω<sub>ci</sub> ≪ ω ≪ ω<sub>ce</sub> do not propagate.

the B field. By introducing a constant magnetic field, B<sub>0</sub>, in the direction of the wave propagation, as is the case for whistler waves, the displacement current becomes negligible compared to the J<sub>E×B</sub><sub>0</sub> current [41]. In the case of the left hand polarised whistler, this current is not oriented such that the B field of the wave is re-generated and consequently the wave cannot propagate. For cylindrically confined whistler wave; the current, J<sub>c</sub>, must go to zero as one approaches the boundary. This results in a slowing of the current transport at the plasma boundary and a subsequent build up of charge. This leads to an electrostatic field which modifies the whistler wave sufficiently to allow the left hand polarised wave to propagate [41].

2.4.2 Helicon Waves

In order to design a thruster which allows the propagation of the helicon within a source tube driven by an antenna, the boundary conditions must be taken into account which will ultimately
modify the dispersion relation. Starting from the linearised form of Maxwell’s equations, [42],

\[ \nabla \times E = -\frac{\partial B}{\partial t} \]  
\[ (2.18) \]

\[ \nabla \times B = \mu_0 j \]  
\[ (2.19) \]

\[ E = j \times B_0 / qn_0 \]  
\[ (2.20) \]

where \( n_0 \) and \( B_0 \) are the density and magnetic field at equilibrium respectively and \( B, E \) and \( j \) are the wave magnetic field, electric field and current which are sinusoidally varying quantities.

In this approach we neglect the displacement current from eq. 2.19. As we have seen earlier, the helicon propagates in a region where \( \omega \ll \omega_{ce} \), so we can assume that the plasma current is carried by the prevailing \( E \times B \) drift. Assuming that perturbations to the wave magnetic field take the general form of \( \exp(i(m\theta + kz - \omega t)) \) and substituting into eq. 2.18 and 2.20,

\[-\frac{\partial B}{\partial t} = i\omega B \]  
\[ (2.21) \]

\[ = \nabla \times E = j \times B_0 / qn_0 \]  
\[ (2.22) \]

\[ = (B_0, \nabla) j / qn_0 = (ikB_0 / qn_0 j) \]  
\[ (2.23) \]

Substituting eq. 2.19 for \( j \) into 2.23 gives,

\[ B = \left( \frac{\omega \mu_0 qn_0}{k \omega_{ce}} \right)^{-1} \nabla \times B \]  
\[ (2.24) \]

and a convenient quantity \( \alpha \) is then defined which contains the electron cyclotron and plasma frequencies,

\[ \alpha = \left( \frac{\omega qn_0}{k \omega_{ce}} \right) = \frac{\omega \omega_{ce}^2}{k \omega_{ce}^2} \]  
\[ (2.25) \]

Substituting this into equation 2.24 gives,

\[ \nabla \times B = \alpha B \]  
\[ (2.26) \]

the curl of which gives,

\[ \nabla^2 B + \alpha^2 B = 0. \]  
\[ (2.27) \]

Equation 2.27 is a Helmholtz equation, the solutions to which are Bessel’s functions, \( J_m \) and their derivatives \( J'_m \). These functions describe cylindrical harmonics and can be used to describe
the allowable modes of helicon wave propagation within the cylindrical plasma. The axial, $B_z$, component of Eq. 2.27 in cylindrical coordinates is given by [42],

$$B''_z + \frac{1}{r} B'_z + \left( T^2 - \frac{m^2}{r^2} \right) B_z = 0$$  \hspace{1cm} (2.28)$$

where $r$, is the cylinder radius (dielectric source tube) and $T$ is the transverse wave number which is related to $\alpha$ and the absolute wave number, $k$, by,

$$T^2 = \alpha^2 - k^2$$  \hspace{1cm} (2.29)$$

The radial, $B_r$, axial, $B_z$, and azimuthal, $B_\theta$, solutions to eq. 2.27 in terms of their Bessel’s functions are,

$$B_z = CJ_m(Tr)$$  \hspace{1cm} (2.30)$$

$$B_r = \frac{iC}{T^2} \left( \frac{m}{r} \alpha J_m(Tr) - k J'_m(Tr) \right)$$  \hspace{1cm} (2.31)$$

$$B_\theta = \frac{C}{T^2} \left( \frac{m}{r} k J_m(Tr) - \alpha J'_m(Tr) \right)$$  \hspace{1cm} (2.32)$$

where $C$ is a constant. At the boundary of the cylinder, the radial component, $j_r$, of the current, $j$ must go to zero. The current, $j$, can be found by substituting Eq. 2.26 into Eq 2.19, giving,

$$j = \frac{\alpha}{\mu_0} B$$  \hspace{1cm} (2.34)$$

The radial component is then,

$$j_r = \left( \frac{\alpha}{\mu_0} \right) B_r = 0$$  \hspace{1cm} (2.35)$$

which sets $B_r = 0$ at the boundary, $r$, thus from Eq. 2.32 the radial boundary condition is given by [42]

$$krJ'_m(Tr) + m\alpha J_m(Tr) = 0$$  \hspace{1cm} (2.36)$$

This boundary condition can be approximated for long and thin tubes, provided $T \simeq \alpha$ and $kr \ll 1$, thus modifying dispersion relation Eq. 2.25 giving
where the approximate boundary condition \( J_1(Tr) \approx 0 \) for the lowest Bessel’s root \( m = 1 \), sets \( Tr = 3.83 \) \cite{42}. Thus for a given source tube radius, antenna length and driving frequency, the plasma density and imposed magnetic field strength required to allow the wave to propagate are prescribed and may be calculated. This relation will be used later in section 3.1 where the design of the SSC RFPT is discussed.

If we are successful in launching propagating helicon waves, though correct matching of the parameters in Eq. 2.37, then these waves will carry energy into the plasma. Power absorption by plasmas in general proceeds through both ‘collisional’ and ‘collisionless’ processes. Collisional or ohmic heating of the plasma results from collisions between plasma electrons and the background neutral particles. This dependence on the electron-neutral collision frequency means that this process tends to occur at higher neutral pressures. At lower neutral pressures, collisionless or stochastic heating of electrons plays a more dominant role and occurs in the sheath regions as the electrons encounter the oscillating RF electric field. Electrons nearing the sheath will encounter an electric field which moves back and forth at RF frequencies, will be ‘pushed’ back and reflected by the electric field, gaining energy in the processes. Helicon wave heating of the electrons is another form of collisionless heating and results from electrons being acceleration by the electric field of the passing wave \cite{38}. As the wave field perturbs the electrons, it looses energy and is eventually damped.

Heating of the electrons increases the likelihood of ionisation resulting from electron-neutral collisions and will therefore tend to increase the number density of the plasma. Recent experiments and modelling of RFPTs, aimed at quantifying the mechanism of thrust generation, have shown separate contributions to the total thrust produced from an axial electron pressure force on the source tube back wall and Lorenz forces arising from diamagnetic currents acting on the diverging magnetic field \cite{43}. Thus through increasing the electron temperature and density, a larger contribution to the thrust from the electron pressure can be expected although at present this process is still poorly understood. This will be discussed in more detail in chapter 6.

The use of helicon waves to efficiently increase the plasma density is the main route through which the performance of RF plasma thrusters can be increased. This is in contrast to many of the thrusters which have been discussed in previous sections where, for example in the case of the ion thruster, a high density ion beam is formed through the use of an external circuit
and grids in order to produce thrust. While ion beams can be formed within RFPTs, it is not thought to be the main mechanism though which thrust is generated within these thrusters. Indeed, later in section 2.4.8.1 the formation of strong ion beams is revisited in the context of current free double layer formation at low imposed magnetic fields and it is found that, in fact, ion beam forming double layers do not contribute directly to the thrust of RF plasma thrusters, but instead increase the source electron density through the formation of source directed electron beams.

2.4.3 Variable Specific Impulse Magneto-plasma Rocket (VASIMR)

The Variable Specific Impulse Magnetoplasma Rocket (VASIMR) is an electrode-less propulsion system which has been in development at NASA since 1977. Recent developments are being made by the Ad Astra company, which aims to develop a commercially available solution for space propulsion for ambitious high ∆V manoeuvres beyond low earth orbit such as manned and robotic missions to the outer planets. The engine operation and construction can be divided into three sections (shown in Fig. 2.10); plasma source, ion heating and the plasma expansion area.

![Fig. 2.10: A cut away diagram of the VASIMR thruster showing the two plasma heating stages which may be throttled to vary the specific impulse. Original image from [44]](image)

The first stage consists of a helicon wave sustained plasma, typically fed with hydrogen, helium or argon gas. This helicon stage operates as described in section 2.4.1. The plasma is then injected into an ion cyclotron resonance heating stage. In this region, a strong magnetic field is applied to the plasma such that the ion gyro frequency, $\omega_{ci}$, is at resonance with a driving RF electric field, supplied from a separate antenna, allowing efficient heating of the plasma ions.
Using Eq. 6.2 and replacing the electron mass, $m_e$, with the ion mass, $m_i$, the applied field necessary for a resonance condition can be found. When using argon as the propellant, it is found that in order to avoid applied magnetic fields in excess of 1 T, the ion resonance heating antenna should be driven at a frequency of 3–4 MHz. In the final ‘magnetic nozzle’ stage, the heated ions flow downstream along the magnetic field which weakens as the field diverges. As the field weakens, the gyroradius of the plasma ions and electrons gradually increases. During this adiabatic expansion of the plasma, the energy of the ions which was largely directed around the imposed field lines as gyro-motion, is converted to axial motion. The magnetic field throughout the thruster is strong enough to radially confine the plasma ions, dramatically reducing the ion flux to the walls and thus reducing ion bombardment erosion of the thruster materials.

Initial experimental studies have shown the VASIMR thruster to have a specific impulse of up to 10000 s at a thrust level of 0.1 N while drawing 10 kW of power [45]. Future performance estimates of up to 5 N of thrust at an $I_{sp}$ 4000–5000 s for an input power of 200 kW have been predicted [46]. The two stage construction of the VASIMR thruster allows each stage to be throttled or ramped up to vary the specific impulse of the engine, allowing greater compatibility with missions. The VASIMR thruster is designed specifically to accommodate extremely high power and large scale missions, making the technology, in its current form, incompatible with small satellite missions. This is not to say, however, that principles from the project could not be applied to small scale thrusters, for example implementing the use of an ion cyclotron resonance second stage to throttle the specific impulse.

2.4.4 Mini Helicon Thruster

The ‘MHTX@MIT’ project at the Massachusetts Institute of Technology (MIT) developed an RF plasma thruster which was designed to use a ‘Shoji’ type antenna to generate a 3 cm diameter helicon wave sustained plasma which expands through a magnetic nozzle [5]. The thruster is shown firing in Fig. 2.11. The group aimed to radially confine both ions and electrons within the source tube. Thus, for a hydrogen plasma with an assumed average ion energy of ~1 eV, an applied magnetic field of $B \geq 0.1$ T reduced the ion gyro-radius to below the source tube radius thus radially confining the ions [5, 47].

The thruster has been characterised through the use of spectroscopy and electric probe measurements [49] and models have also been developed to predict the theoretical maximum performance of the system [47, 50]. Ion energy distribution function measurements and spectroscopy
2.4. RF Plasma Thrusters

Fig. 2.11: A photo showing the mHTX thruster firing, while operating on 1.9 mg/s Argon propellant, an imposed magnetic field of 0.18 T and an input power of 1000 W. Original image from [48]

of the plasma was used to estimate the specific impulse and thrust levels [49] which varied from 6–20 mN for input powers up to 1000 W and argon propellant flows of 0.35 mg/s. A method of ‘direct weighing’ was also reported to give a direct measurement of the thrust produced by the system, measured as 10 mN for an input power of 700 W and an argon propellant flow rate of 10 sccm which corresponds to a thrust efficiency of $\sim 20\%$ [47].

2.4.5 Helicon Plasma Hydrazine Combined Micro Thruster

The European Union Seventh Framework (FP7) funded project ‘HPH.COM’ aimed to develop a helicon wave sustained plasma thruster which could work in conjunction with an integrated chemical hydrazine propulsion system [51]. A schematic of this concept is shown in figure 2.12.

The thruster was conceived to be used on board small satellites to provide both high thrust, low specific impulse and low thrust high specific impulse capabilities for attitude and position control. The high thrust mode would operate as a typical chemical thruster system, whereas for low thrust tasks, the decomposed hydrazine products would be ionised using a helicon excitation antenna in an attempt to increase the specific impulse. A feasibility study was conducted to assess the dual mode functionality, however, it is not clear from the available literature whether this was shown to be technically feasible.

The thruster was designed to consume 50 W of RF power and to generate 1.5 mN of thrust at a specific impulse of 1200 s [51]. An imposed magnetic field is supplied by an array of
permanent magnets which creates a field profile that varies substantially along the axis of the thruster. Beneath the antenna, the axial component of the magnetic field was measured as 20 mT, which increases to a maximum of 0.14 T before diverging and weakening to 80 mT at the thruster orifice and then to zero in the far field[52].

Thrust measurements were made at the National Aerospace University, Ukraine by mounting the thruster upon a pendulum thrust balance while operating using 8 W of RF power and an argon mass flow of 0.12 mg/s, which was reported to produce a thrust of 0.5 mN at a specific impulse of 422 s, giving a thrust efficiency of 13%. Thrust measurements at higher powers were reported to be impossible due to ‘unacceptable errors’ and are instead estimated using ‘current measurements’ [1]. The thruster is reported to have been tested at an Onera testing site, however the performances could not be reproduced, which has been attributed to a possible interaction between the thrusters’ magnetic nozzle and the vacuum chamber. The thruster is detailed to use a new type of compact, high impedance antenna, called the ‘S-helicon’, which is reported to be very efficient at creating a plasma. No details of the antenna could be found at the time of writing as the antenna is under patent protection.

2.4.6 High Power Helicon Thruster

The High Power Helicon thruster HPH was designed not as a thruster, but as a solar wind simulator. The device was used to provide simulated solar wind for the Mini-Magnetospheric Plasma Propulsion thruster. It was later realised that this device itself could be used as a propulsion
technology. The thruster works in a pulsed regime, effectively creating high power discharge within a resonant inductor capacitor resistor (LCR) circuit which resonates at a predetermined frequency [53]. When the plasma discharge has initiated, it appears as a parallel inductance which modifies the resonant circuit to oscillate at a new frequency. A photo of the HPH thruster firing is shown in figure 2.13.

![Photo of the HPH thruster before and during firing](image)

**Fig. 2.13:** A photo of the HPH thruster before and during firing. Original image from [53]

The thruster operates at a RMS power of ~250 kW in the vacuum chamber and each ‘shot’ lasts roughly 1 ms. Plasma densities of $5 \times 10^{17} \text{m}^{-3}$ have been measured downstream of the thruster, suggesting a very high degree of ionization of the propellant. Few publications are available related to this technology as a thruster and in fact, no measurements of the thrusters performance are available to the author’s knowledge.

### 2.4.7 Low/Medium Power Helicon Thruster (LPHT)

The Georgia Institute of Technology have developed an RF plasma thruster designed to operate over an RF power range of 1–1.5 kW and a power frequency range of 7–13.56 MHz [54]. A 70 cm length solenoid was used to provide an axial applied magnetic field of up to 0.12 T and the thruster has been operated in argon up to propellant flows of ~15 mg/s. While the literature documents the thruster as being mounted upon a thrust stand (see Fig. 2.14) no performance measurements have been published to the best of the author’s knowledge.
2.4.8 Helicon Double Layer Thruster (HDLT)/HPT

The helicon double layer Thruster (HDLT) was designed by the Space Plasma, Power and Propulsion group at the ANU. The HDLT was conceived to operate based on the generation of an inductive or helicon wave sustained plasma using an external double saddle antenna, usually operating at 13.56 MHz. One prototype device named *Chi-Kung* was used to investigate the formation of a current free double layer, near the interface between the plasma source tube aperture and a down-stream diffusion chamber at $z = 25$ cm. [55], shown in Fig. 2.15. The HDLT and the related diffusion chamber experiments have revealed many modes of operation, some of which may be advantageous for use within a thruster. Two particular and promising modes are the ‘low field helicon mode’ and the ‘current free double layer mode’, which are described in more detail in the sections following.

2.4.8.1 Current-free Double-layer Formation

When a plasma comes in to contact with a dielectric boundary, electrons, being more mobile than ions, escape quickly to the walls, which develops a net negative charge. The resulting electric field eventually reflects the majority of the incoming electrons and accelerates ions towards the walls. This plasma structure is known as the plasma sheath and this particular example is of the formation of an ion sheath. Ion acceleration by the sheath electric field into a surface is often exploited for ion implantation processes in order to modify surface properties of a substrate [57].
The ion sheath can be assumed to have a negligible electron density. Thus, unlike the bulk plasma, the assumption of quasi neutrality does not hold in the sheath region. A double layer can be thought of as a ‘double sheath’ consisting of an ion sheath attached to an electron sheath, similarly breaking quasi-neutrality and capable of sustaining an electric field and localized potential difference [55]. This effect was first observed early in the century by Langmuir [58], who documented a cathode double layer consisting of a negatively charged inner layer followed by a positively charged outer layer. More recently the formation, structure and characteristics of double layers have been modelled and characterized extensively in laboratory plasmas [59, 60, 61, 62] and has also been found to explain astrophysical phenomena such as atmospheric aurora and solar coronal discharges [63, 64].

The structure of double layers, such as those observed in the Chi-Kung experiment [61] and others [65, 66] are similar to the ion sheaths found in the vicinity of plasma boundaries. These types of double layers often spontaneously form as a mostly collisionless (27–267 mPa) plasma expands out of a bounding volume such as a cylinder open at one end. This has also been observed in the presence of a diverging magnetic field and are typically of the order of tens to hundreds of Debye lengths in thickness and sustain a potential difference of around 25 V. These are termed ‘current free’ double layers, as the potential drop results from internally generated currents and does not require an external DC circuit to occur.

Successful models of current free double layers have been constructed [67] which assume four particle species; thermal ions and electrons (following a Maxwellian distribution) and accelerated
ions and electrons all of which follow a Maxwellian distribution. A supersonic $\sim 2.1c_s$ ion beam of argon was indeed found to be generated from a double layer which formed from a high density low pressure helicon sustained RF plasma source [56]. Evidence has also been found that electrons within in the source possess sufficient energy to overcome the potential barrier of the double layer [56, 68]. Measurements of the electron energy distribution (EED) upstream of the double layer yielded essentially Maxwellian energies with a high energy depletion. Measurements of the EED downstream of the double layer reflected the depletion of the upstream distribution through an increase the number of high energy electrons [68]. This supports the theory that the plasma exhaust remains quasi-neutral, allowing recombination and neutralisation in the plasma exhaust.

Within the HDLT, the accelerated ions were thought to contribute towards the production of thrust. However, it is now understood that the current free double layer can not ‘produce’ thrust [69], but rather accelerates downstream electrons to form an electron beam. This beam travels upstream into the plasma source and is responsible for increased ionisation just upstream of the double layer, which in turn can increases the thrust. The thrust force which is generated is also now understood to be the sum of two main contributions [43, 70], an electron pressure on the back wall of the source tube and an electron pressure on the diverging magnetic field, which in some cases are equal in contribution.

The role of the magnetic field in double layer formation has been suggested to be related to ion gyro-motion and the critical point at which the ion gyro-radius is equal to the source tube inner diameter, occurring at 5 mT in the Chi-Kung experiment. At this point, a sharp increase in upstream plasma density and plasma potential occurs [71]. The double layer was found to be insensitive to increases in field strengths above the level required for the formation of the potential structure. Further evidence of the dependence of double layer formation on ion gyro radius was presented in an investigation relating various source tube inner radii to the appearance of an ion beam for a similar low pressure (53–80 mPa) and low power (200 W) system [6].

### 2.4.8.2 Low Magnetic Field Helicon Waves

Low field helicon propagation and absorption in low power cylindrical RF discharges has been observed in simple laboratory discharges [7, 9] and investigated in the context of power deposition in a HDLT [8] with a diverging magnetic field. In both studies the role of the imposed magnetic
field in power deposition was investigated and was demonstrated to control the plasma density, resulting in a peak in source density occurring over a 4 mT magnetic field range. Operation of the HDLT with low applied magnetic field strengths is attractive in terms of the thruster design as low magnetic fields may be supplied easily using low mass and low power electromagnets which is advantageous from a satellite engineering perspective.

The HDLT has been operated in a ‘low magnetic field mode’ and characterised in terms of thrust, ion density, plasma potential and beam current downstream of the source tube exit as function of applied magnetic field strength. Peaks in these parameters were found to exist for imposed magnetic field strengths of 2–3 mT. The thrust estimates in this cases however were made through indirect measurements using a momentum flux measuring instrument (MFMI) [72].

2.5 Summary

A review of modern EP space missions illustrates the increasingly high ∆Vs requirements, which have driven the development of a range of EP technologies. The high exhaust velocities produced by EP increases the ∆V per kg of propellant. This in turn can allow reductions in the wet mass of the satellites, cost savings through reduced launch mass or increased payload masses for the same cost.

The demands of novel future missions will need to be met by the development of novel EP technologies which address the weaknesses inherent in the current range of propulsion systems, such as the use of hollow cathode neutralisers, high voltage acceleration grids system complexity and limited propellant compatibility. A review of the current state of RFPT development showed that this technology could offer a range of novel capabilities, such as broad propellant compatibility and long thruster operational lifetimes while maintaining a low system complexity. The technology has been widely investigated through experiments and theoretical studies to assess its performance and to understand the fundamentals of its operation.

The review revealed that helicon waves have been used extensively in high (∼100 mT) imposed magnetic fields regimes, which have been shown to efficiently produce high density plasmas. This property is at the heart of the interest in helicon waves for propulsion purposes as it can increase the efficiency with which ions can be produced.

While performance increases could be provided through use of high field helicon waves, they are unsuitable for use on board earth orbiting satellites as strong magnetic fields may interfere
with satellite subsystems, and may induce torque forces on the satellite as the field interacts with the earth’s magnetic dipole. Furthermore, the requirement of strong magnetic fields has lead to a number of literature studies opting to use permanent magnets to produce the required magnetic fields. This is not favourable from a systems design perspective as the magnets could not be deactivated or field reversed to reduce the impact on the satellite and orbit.

Low magnetic field modes requiring relatively low imposed magnetic fields (2–30 mT) have been shown to increase the plasma density either through the development of plasma double layers or through the launching and absorption of low field helicon waves. Experiments and theoretical advances have explored the plasma dynamics within such sources, including the dispersion of low field helicons in low diverging magnetic fields and the formation and characteristics of plasma double layers in expanding cylindrical plasmas.

It has been proposed in the literature that such modes could improve the performance of an un-magnetised RFPT and initial indirect thrust estimates have been made which suggest that this is true. From a systems engineering perspective, an RFPT optimised to use magnetic fields below 20 mT would minimise the potential impact of the thruster on the spacecraft and its orbit during operation. Such a field strength could also be easily generated by solenoid magnets which can be deactiveated or field reversed if required.

While literature studies have made use of indirect methods, the performance of an RFPT operating in a low magnetic field mode has not been directly measured. For the RFPT to be considered as a flight technology its performance should be directly measured to establish the extent of its compatibility with the mission examples outlined earlier. The objectives set at the end of chapter 1 have now been motivated. Chapters 3 and 4 describe the equipment, facilities and methods used to address these objectives.
Chapter 3

Equipment and Procedures

In this chapter, descriptions are given of the diagnostic equipment and experimental methodologies that were used to address the objectives set in chapter 1. This includes descriptions of the relevant physical principles behind the operation of the instruments as well as data analysis and interpretation techniques.

Much of the equipment used during this project was available commercially, such as the RF power generator, standing wave ratio (SWR) meter, RF current probe and volumetric flow meter, described in sections 3.2.3, 3.2.5, 3.3.4 and 3.2.2 respectively. The retarding potential analyser (RPA), matching network and Langmuir probe, described in sections 3.3.1, 3.2.4 and 3.3.3, respectively, were built and provided by our collaborators at the ANU. The microwave resonance plasma probe is described in section 3.3.2 and was built and provided by the Open University (OU), who also provided training and their substantial expertise. The Pegasus vacuum chamber is an in-house facility and was the setting for all experiments carried out during this project. The chamber and its capabilities are described in section 3.2.1. The SSC did not possess an RFPT at the start of this project, therefore the commissioning of a prototype was an early requirement. The thruster was designed and constructed in house, although the manufacture of a number of the components was outsourced. The details and motivations behind the thruster’s design are given in section 3.1.

In chapter 1, the objective was set to measure the thrust produced by the RFPT, while operating in the low field mode. In chapter 2, it was highlighted that all electric propulsion thrusters which have been successfully flown have had their performance experimentally verified through direct thrust measurements while operating fully immersed in a space simulation cham-
Chapter 3. Equipment and Procedures

ber. Direct thrust balance measurements have been demonstrated to be a more accurate way of assessing the momentum which could be imparted to a spacecraft by a thruster compared to devices like the MFMI. Thrust balances have been used to measure the performance of a range of EP technologies, however, at the commencement of this project no studies had been completed to directly verify the performance of an RFPT. Thus, the design construction and validation of such a balance was also a requirement of this project. The detailed description of the design and validation of the thrust balance is presented in chapter 4.

3.1 Thruster Design

The SSC RFPT was designed to allow experimental flexibility within the objectives set in chapter 1 and compatibility with the thrust balance designed along side it. As a result, the RFPT does not resemble a self contained ‘engineering model’ thruster, but rather an experimental or proof of concept model. In this section, each component is outlined and described.

3.1.1 Source Tube

An off-the-shelf borosilicate glass tube with a 45 mm inner diameter, 2.5 mm wall thickness and 170 mm length was chosen as the dielectric plasma source tube. The sizing of the source tube radius was chosen based on geometries described in the literature that have been found to produce low field modes. Low field modes have been observed in several studies while using 10–100 mm radius source tubes with lengths from 95–300 mm [7, 8, 77, 78]. The tube chosen has a moveable back plate to allow the length to be varied, thus allowing investigation into the performance produced by low field modes while using a reduced length antenna and source tube aspect ratio (objective 4).

3.1.2 Antennas

In chapter 2, the dispersion relation for the helicon wave in a cylinder was derived, which can be used to design a plasma source that allows propagation of a helicon wave. The length of the antenna is incorporated into this relation and can be set initially. Two lengths of double saddle antenna have been used here, one of 130 mm and the other of 85 mm, shown in Fig 3.1. Since the source tube radius has been set to reflect a common literature geometry, and we are fixing the driving frequency, \( \omega \) at 13.56 MHz (see section 3.2.3), requirements are set on the plasma
3.1. Thruster Design

![Fig. 3.1: Photos of the 130 mm (a) and 85 mm (b) antennas coated in boron nitride.](image)

Density and imposed magnetic field in order to allow helicon wave propagation. For an imposed magnetic field of 10 mT, and a source radius of 22.5 mm, the plasma density calculated from Eq. 2.37 is \( \sim 2 \times 10^{-19} \text{ m}^{-3} \). One can also see from Eq. 2.37 that, as the impose magnetic field is increased, the density must also increase in order to maintain the resonant condition, shown in Fig. 3.2 for both antenna lengths. These density values will be referred to in later chapters where we will find that peaks in the plasma density as a function of the imposed field can be found at much lower plasma densities and that this is characteristic of low field modes in general.

Both the source tube and antenna lengths are to be varied between two values. Irrespective of helicon wave aspects of the power deposition, it is expected that plasma densities should be higher in the case of the shorter antenna and source tube combination, since for a fixed power between the two geometries, the power density will have increased. This is advantageous in the context of a thruster design, as volume and mass budgets are both important when considering application to a satellite. The matching of the antenna and source tube length is also prudent as plasmas tend to form beneath the antenna. Any length of the antenna in excess of the source tube size will simply radiate into free space, reducing the power transfer efficiency. For this reason, both antennas were placed centrally along the axis of the source tube volume.

In the case of the ion magnetisation mode, the main variable responsible for the effect is the imposed magnetic field, as this is what controls the extent of the ion magnetisation. Studies reporting on ion gyro radius related ion beam formation have used both the double saddle [71] and the simple double loop antenna [6]. Given that this particular effect is independent of the antennae geometry, the double saddle type is used for both the 85 mm and 130 mm antennas in this study.
Fig. 3.2: Plot of electron density and imposed field required to satisfy the helicon dispersion relation. Plotted here for \(\omega = 13.56\) MHz, \(r = 22.5\) mm and antenna lengths 85 mm (squares) and 130 mm (circles)

An issue observed during the initial phase of testing of the SSC RFPT was the sputtering and subsequent deposition of the material of the antenna by ambient plasma resulting from ‘Micro arcing’. This is a phenomenon commonly encountered by the plasma reactor community and many investigations have been conducted into the various causes and dependencies [79]. The net result of the arcing in the SSC RFPT is the vaporisation of the copper antenna and deposition onto the source tube and surrounding supports. It was often the case during early operation of the RFPT that copper plated images of the antenna were found embedded on the source tube outer surface. For the purposes of these investigations, two mitigation strategies were employed. In the first instance, boron nitride aerosol spray was used to create a high temperature resistant insulating layer around the antenna to avoid sputtering. This is the reason behind the white coating seen in the photos the antennae in Fig. 3.1. The second strategy was the insertion of a series of blocking capacitors between the RF generator and the RF antenna, which is discussed in more detail in section 3.2.4.
3.1.3 Solenoids

The imposed magnetic field was generated by two solenoids, which were designed and wound in house. The solenoids and their formers were designed to produce peak magnetic fields of up to 40 mT and to act as mounting points for the thruster. Throughout this thesis, the two solenoids are refereed to as the ‘source’ and ‘exhaust’ solenoid, as one is positioned closer to the plasma source and the other closer to the plasma exhaust.

The dispersion relation derived in section 2.4.2 for the helicon wave assumes a constant and purely axial imposed magnetic field. Since the waves are launched from underneath the antenna, the solenoids were positioned so as to align the peak magnetic field with the antenna. This provided a magnetic field that varied by approximately 1.5 mT, and thus does not significantly change the plasma density required for helicon wave propagation. However, since two antennas and source tube lengths were used in this study, two solenoid configurations were employed to ensure that the peak field region was axially aligned with the antenna. When using the 130 mm antenna, this was achieved by separating the solenoids by 10 mm and aligning the end of the exhaust solenoid former to the front of the source tube aperture. When using the 85 mm length antenna, the peak field region of the exhaust solenoid was axially aligned with the antenna by aligning the front of the solenoid former with the source tube aperture. The solenoids and their positions relative to the antenna and source tube are also shown schematically for the 130 mm antenna length case in chapter 4, Fig. 4.1.

A finite element analysis of the axial magnetic field produced by the two solenoid configurations is shown in Fig 3.3 for a current of 1 A supplied to the coils. In the case where both solenoids are used together, some asymmetry exists in the magnetic field at the peak field point, which is the result of source and exhaust solenoids having a different number of windings; 335 and 340 turns respectively. This was taken into account in the finite elements software by entering the actual number of windings in each solenoid. Measurements of the axial component of the magnetic field were made using a magnetic field probe at the peak field point and were found to be in good agreement to within ±5 mT of the simulated values. Due to the geometry of the probe and the solenoid formers it was not possible to measure the peak field strength for a single solenoid, however, measurements of the magnetic field downstream and upstream of the solenoids, as well as at the central region for the dual solenoid case (measured in the 10 mm spacing between the solenoids), have been plotted with the calculated values in Fig. 3.3. For
Chapter 3. Equipment and Procedures

Fig. 3.3: Graphs showing the axial magnetic field as a function of the axial position (Z) for ‘both solenoids’ on (a) and ‘exhaust only’ (b) cases for a 1, 2 and 1.5 A current flow. The calculated fields are plotted as solid lines and the measured values are plotted as squares, diamonds and triangles. The vertical red line indicates the position of the source tube exit aperture and the grey boxes are representations of the antenna and its positioning. Profiles were generated from simulations created using the FeMM finite element analysis software package.

the dual solenoid arrangement, it is reasonable to assume that the agreement between the measured peak field strength and the calculated value should continue to higher supplied currents. Similarly, in the single solenoid case, it is reasonable to assume that the agreement between the calculated and measured downstream fields should continue into the central region of the solenoid and up to higher currents. Thus, for the remainder of the thesis, the peak field strength quoted for the single solenoid is the calculated value of the field strength will be quoted. For the dual solenoid case, direct measurements were possible and are the values quoted.

3.1.4 Propellant

Argon is a staple gas for laboratory plasma discharges as it is inexpensive, easily obtainable and relatively easy to ionise, with a first ionisation potential of 15.7 eV. As a propulsion technology, xenon is a more desirable choice, with a lower ionisation potential of 12.13 eV and a mass of 131.29 g/mol compared to 39.95 g/mol for argon. Thus, for a given input power and mass flow, the efficiency of the thruster utilising Xe as propellant would increase along with the thrust, owing to the larger atomic mass. However, low field mode studies have all used argon as the
3.2 Laboratory Facilities

3.2.1 The Pegasus Vacuum Chamber

Flight qualified electric propulsion systems operate in the vacuum of space. In order to test such systems on the ground, a space-like environment must be created, which is generally achieved through the use of a vacuum chamber and pump arrangement. As discussed in chapter 2, precursor RFPT experiments have been performed in which a vacuum is generated downstream of a source tube which is connected to a diffusion chamber and pump [62, 68, 56]. These systems typically create a base pressure of 0.1 mPa and operate at a pressure of 10-400 mPa, depending on the gas flow rate into the system. This arrangement is an attractive option as it allows easy access to the exterior of the device during operation and does not require a large vacuum vessel. The source and exhaust plasmas are also accessible to plasma probe measurements, spectroscopy and MFMI measurements, which can be used to estimate the performance of the device. However, measurements of the total momentum transferred to the device (were it attached to a satellite for example) are not possible due to the mechanical dependencies. To measure the total momentum transfer, the thruster must be fully immersed within a vacuum and mounted on a device capable of quantifying the momentum imparted to it by the ejected plasma.

For the thesis experiments, the vacuum environment was created using the Pegasus space
simulation facility, shown in Fig. 3.5. The system consists of a 1.2 m long, 0.5 m radius cylindrical steel chamber, connected to a rotary roughing pump and a turbo molecular pump. The system generates a base pressure of about 0.1 mPa and an operating pressure of 10-100 mPa depending upon the rate of inflowing gas, which is close to the ‘externally attached’ experiments. The chamber has a large number of ports which may be used to provide optical access, electrical feeds and gas flow. The gas propellants were supplied to the vacuum side of the chamber through a Bronkhurst flow controller\(^1\), which is calibrated to supply up to 200 sccm of air, Ar, Kr, Xe, N\(_2\), CO\(_2\) and H\(_2\).

![Figure 3.5](image)

**Fig. 3.5:** The *Pegasus* vacuum testing facility, used for all experiments presented in this thesis.

It is practically impossible to create a pure vacuum within a ground facility, such as would be encountered in deep space or even at low earth orbit. As a result, the pressure in the source tube may be increased above what would be expected while operating in space. In this section we will explore first the ideal scenario of an RFPT operating in the vacuum of space, where a pure vacuum is found down stream of the source tube, before considering the effect on the source tube pressure of a finite pressure within the vacuum facility.

In the ideal scenario of a pure downstream vacuum, neutrals injected into the source tube at some mass flow, \(\dot{m}_p\), travel downstream and are lost from the system via the source tube exit at some exhaust mass flow rate, \(\dot{m}_e\). The one way particle flux, \(\Gamma\), across a unit area is given by,
3.2. Laboratory Facilities

\[ \Gamma = \frac{n_s \bar{C}}{4} \]  

(3.1)

where \( n_s \) is the gas density in the source tube and \( \bar{C} \) is the average velocity of the gas particles, given by

\[ \bar{C} = \sqrt{\frac{8k_B T}{\pi m}} \]  

(3.2)

where, \( T \) is the gas temperature, \( k_B \) is Boltzmann’s constant and \( m \) is the mass of the gas atom.

Since there are no other inflows or outflows of gas in this case, the propellant flow into the source tube is equal to the exhaust mass flow from the source tube of exit area \( A_s \),

\[ \dot{m}_p = A_s n_s m \bar{C} / 4. \]  

(3.3)

The gas pressure in the source tube is given by,

\[ P_s = n_s k_B T \]  

(3.4)

thus, rearranging 3.3 and substituting into 3.4,

\[ P_s = \frac{4 k_B \dot{m}_p}{A_s m \bar{C}} \]  

(3.5)

The source tube neutral pressure is shown in Fig. 3.6 as a function of the propellant flow. In this case, the source tube pressures vary from about 0.1–0.25 Pa over the Ar flow range used in this thesis of 0.3–0.6 mg/s.

The experimental arrangement used in this thesis deviates from this ideal case, as the vacuum facility is not capable of sustaining a pure downstream vacuum as a result of the finite pumping speed and instead has a measured facility pressure, \( P_f \). The source tube pressure is now determined by a balance of escaping gases into the chamber and ingested gas from the chamber into the source tube. A contribution to this balance comes from the source tube exit and also from the fact that the propellant feed tube is a 3.175 mm diameter steel pipe which is coaxially aligned with a 5 mm hole in the rear off the source tube. While this allows the thruster and thrust balance to be mechanically decoupled, thus removing the propellant feeds’ mechanical resistance (see section 4.2), this also creates a clearance gap of area \( A_c \) through which gas can escape and re-enter the source tube. The resulting overall mass flow balance is then,
Fig. 3.6: Graph showing the calculated neutral Ar gas pressure in the source tube as a function of the propellant flow for a downstream vacuum and when corrected for finite facility pressures. The corrected source tube pressures are calculated from Eq. 3.9. The measured chamber pressure, $P_c$, is used in this calculation and is shown as a function of the propellant flow into the chamber in Fig. 3.7.

\[ \dot{m}_p + \dot{m}_{is} + \dot{m}_{ic} = \dot{m}_{ec} + \dot{m}_{es} \]  

(3.6)

where $\dot{m}_{is}$ and $\dot{m}_{ic}$ are the mass flows ingested into the source tube through the source exit and the propellant feed clearance respectively and $\dot{m}_{ec}$ and $\dot{m}_{es}$ are the mass flows escaping from the source tube through the propellant feed clearance and the source tube exit. The mass flows into the source tube through the clearance area, $A_c$ and source tube exit area $A_s$, are determined by the pressure of the facility and are given by,

\[ \dot{m}_{is/c} = \frac{A_{s/c} m P_f C}{4 k_B T} \]  

(3.7)

and the mass flows out of the source tube area are determined by the pressure in the source tube and area given by,

\[ \dot{m}_{es/c} = \frac{A_{s/c} m P_s C}{4 k_B T} \]  

(3.8)

Equating all the mass flow terms according to Eq. 3.6 and simplifying we find that,
3.2. Laboratory Facilities

Fig. 3.7: Pressure of the Pegasus vacuum chamber as measured using two gauges a Leybold PTR90 dual cold cathode and pirani gauge, which claims a measurement range 1 µPa–100 kPa and a Pfeiffer CMR 365 single gauge with a claimed measurement range of 1 mPa–10 Pa.

\[ P_s = \frac{4k_B T \dot{m}_p}{mC(A_c + A_s)} + P_f \]  

which can be used to determine the effect of the facility pressure on the source tube pressure.

The pressure of the Pegasus facility was measured as a function of \( \dot{m}_p \) and is shown in Fig 3.7. The pressure in the vacuum chamber can be approximated to an altitude above the average sea level using the international standard atmosphere model [80]. The maximum background pressure at which the thruster was operated within this chamber was approximately 100 mPa which corresponds to an altitude of approximately 80-100 km. The exact altitude is difficult to compute as it fluctuates in response to day/night cycles as well as with solar activity. It is clear however that the simulated altitude is far from the desired altitude of about 400–700 km which is generally accepted to be a low earth orbit and is the likely first destination for a flight qualified RFPT. When the background facility pressure is accounted for, the resulting source tube pressure roughly doubles to 0.2–0.4 Pa over the range of propellant flow rates used in this study. This is also shown added to Fig 3.6.

Throughout this analysis it has been assumed collisions do not significantly contribute towards the dynamics of the source. If neutral particle collisions are sufficiently frequent, a normal shock
may form at the exit of the source tube, the effects of which have been neglected here. The
spacial extent of a normal shock wave is typically several times the mean free path of the
particles. In this case, the maximum corrected pressure in the source tube is $\sim 0.35$ Pa while
operating at propellant flow rate of $0.7$ mg/s of argon. This corresponds to a mean free path of
about 10 mm. Under these circumstances a weak normal shock may develop within the source
tube with a thickness smaller than the tube length. However since the source tube does not
have a diverging nozzle, the Mach number will not increase significantly when moving from the
source tube region to the vacuum chamber. Detailed modelling should be done however to more
fully explore neutral particle dynamics of the source tube. The influence of facility effects on the
operation of helicon courses in the context of plasma thrusters is an area of ongoing research.
Initial experimental [81] and Monte Carlo simulation investigations have can be found in the
literature [82] as well as more general investigations into neutral dynamics driven helicon source
phenomena such as of neutral heating and depletion [83, 84]. Given the breadth of dynamics that
can be observed in this source, a separate rigours exposition of the neutral dynamics should be
under taken which will allow the effect to be properly quantified. Until this is done, the analysis
presented serves to illustrate why a propellant flow limit of $0.7$ mg/s has been imposed, as
higher flows may cause the testing environment to significantly deviate from an approximation
of a space environment through the development of normal shocks at the source tube orifice.

3.2.2 Volumetric Flow Controller

Propellant flow into the vacuum chamber was metered using the Bronkhorst volumetric flow
controller, also referred to as a mass flow controller or MFC, pictured in Fig. 3.8. The MFC
is controlled digitally through an RS232 cable and a software interface. The manufacturer
specifications state that up to 200 sccm of propellant may be supplied with an accuracy of
$\pm 0.5\%$ of the commanded flow set point.

3.2.3 Radio Frequency Power Generator

The RF power supply to the antenna used throughout these investigations was the MKS
SurePower® RF plasma generator model QL3513. The generator is capable of supplying a
maximum of 3500 W of 13.56 MHz power into a $50$ Ω load. The manufacturer’s specifications
state that the generator can supply constant power to within $\pm 1\%$ for forward powers between
350 and 3500 W and $\pm 3.5\%$ for forward powers below 350 W. Digital circuits within the power
3.2. Laboratory Facilities

Fig. 3.8: The Bronkhorst EL-FLOW® F-201CV-200-AAD-22-V volumetric flow controller used to supply propellant to the thruster throughout these investigations.

Fig. 3.9: The front panel of the MKS SurePower RF power supply supply measure both the forward and reflected power and are able to adjust the frequency of the power to allow some degree of load matching. For the duration of these experiments, the automatic load matching features were disabled so that a constant frequency of 13.56 MHz was used. The forward power sent to the plasma is partitioned into power deposited into the plasma and power lost in heating the antenna and matchbox component. Ideally, these losses would be quantified to allow the actual power delivered to the plasma, however at the time these experiments were completed the necessary equipment was not available to allow this. As a result, the RF power levels quoted in this thesis are the forward RF powers as measured using the standing wave ratio meter (see section 3.2.5). Using the forward power in calculations of the
thrust efficiency will produce an underestimate, as not all the power will be delivered to the plasma.

### 3.2.4 Matching Network

The antenna and plasma both have inductive components. It is necessary, therefore, to match the impedance of these loads to the impedance of the power supply. This is achieved by adding a network of adjustable capacitors between the power supply and the antenna, commonly referred to as a matching network. The matching network used in these experiments is pictured in Fig. 3.10a alongside its equivalent circuit diagram.

In the absence of a plasma, the power supply impedance is first matched to the antenna impedance alone, through manual adjustment of the vacuum capacitors, $C_{\text{tune}}$ and $C_{\text{match}}$. At this stage the forward power is dissipated through a combination of emitted radiation into free space and heating of the antenna and of the components within the tuning network. Once the plasma has initiated, the load impedance changes considerably and the capacitors are re-adjusted to match the power supply impedance to the new load impedance to maximise the forward power to the plasma.

![Matching Network](image)

**Fig. 3.10:** Photo of the matching network (a) and simplified circuit diagram (b). Note the blocking capacitors (brown in colour) and RF current probe (yellow).

Also inside the matching network are a series of fixed capacitors, referred to as blocking capacitors, labelled in Fig. 3.10b as $C_b$. These were installed in order to reduce the micro arcing discussed in section 3.1.2. The reasoning behind this is not immediately apparent, however...
experiments aimed at understanding the cause of micro arcing describe the scenario simply [79]. Assuming that the applied RF cycle is symmetric; at each positive half of the cycle, electrons will be drawn to the antenna from the ambient plasma and with each negative half, ions will be drawn. Due to the differences in mobility between the ions and the electrons, a larger net negative current is drawn to the antenna than positive. The imbalance in charge drawn from the plasma raises the plasma potential higher and higher until a breakdown occurs across the plasma sheath, allowing the charge difference to re-balance. During this process, a large amount of energy is dissipated near the antenna which causes local heating and small ejections of antenna material. In order to remedy this situation, capacitors are connected to the antenna allowing the formation of a negative DC self bias. This reduces the difference between the antenna voltage and the plasma potential sufficiently to prevent arcs forming across the sheath. This was found to be a very effective strategy in these experiments and the micro arcing phenomenon was reduced considerably.

3.2.5 Standing Wave Ratio Meter

In the previous section the necessity of impedance matching within RF systems was described. In order to match the system manually through variable capacitors, a standing wave ratio meter is required which displays the forward and reflected power to the operator. The Daiwa® SWR meter has a frequency range of 1.8–200 MHz and a power range of up to 2 kW of forward power, which is ideal for the purposes of this study. The meter was placed between the RF generator and the matching network. Two needles display the forward and reflected power and their point of crossing represents the standing wave ratio. For the purpose of the experiments presented in this study, the system was tuned until the needles indicated that the forward power was maximised and the reflected power minimised. Typically, the reflected power was minimised to under 0.2 W, at which point the system is described as ‘well matched’. The meter data sheet quotes a tolerance of ±10%, which for the purposes of this study corresponds to a maximum of 5 W at the maximum power used in this study of 500 W. The SWR indicates the total forward power sent to the matchbox but it does not account for losses within the match box or for power which goes into joule heating the antenna. In order to gauge the distribution of power deposition, a current probe was used to gauge the current flowing to the antenna. This is discussed further in section 3.3.4.
Fig. 3.11: The standing wave ratio (SWR) meter used to monitor the forward and reflected power between the generator and the load.

3.3 Probes and Diagnostics

The following sections describe the plasma probes and diagnostics used in these experiments.

3.3.1 Retarding Potential Analyser

Fig. 3.12: Schematic and biasing scheme of the retarding field energy analyser. The ground, \( G_g \), repeller \( G_r \), discriminator \( G_d \) and suppressor grids are shown as broken lines. The final electrode is the collection plate. These grids and the collector are biased according to the sketched voltage plot. The blue arrow indicates that the voltage of the discriminator is increased with time.

The retarding potential analyser (RPA) can be used to measure the ion velocity distribution function (IVDF) of a plasma [85]. The RPA, shown schematically in Fig. 3.12, consists of a
small grounded metal box with a 4 mm diameter orifice and a series of which house three fine nickel grids and a collecting plate. These are stacked next each other but are electrically isolated from one by sheets of mica. Each grid and the collector is biased relative to the outer casing, which is grounded. The biasing scheme is shown graphically in Fig. 3.12. The first grid, known as the ‘repeller’, ‘G_r’, is held at a constant -90 V bias and is used to repel electrons and prevent them from entering the device. Ions which have entered the device meet the second grid, known as the ‘discriminator’ G_d, which ramps from 0–90 V and repels ions of increasingly high energy. Ions with sufficient energy to overcome the potential barrier of the discriminator are collected at the collector plate. A ‘suppressor’ grid G_s, located between the discriminator and the collector is held at -18 V to repel any electrons born through secondary emission within the probe. The current collected at the collecting plate is recorded as a function of the discriminator voltage, which produces the red curve shown plotted in Fig. 3.13.

This data acquisition procedure was automated using a LabView program which commands the PC data acquisition card to generate a small 0–10 V, which is then amplified and applied to the discriminator grid. The current collected at the collector plate is recorded by the software along with the discriminator voltage and stored for later analysis. The ion current collected by the analyser is given by [38],

\[ I = \beta \theta \int_{v_{\text{min}}}^{\infty} f(v) \, dv. \]  

(3.10)

where \( f(v) \) is the velocity distribution of the ions entering the aperture, \( v_{\text{min}} = \sqrt{2eV_d/M} \), \( \theta \) is the product of the transparencies of each grid through which the ions pass before reaching the collector, \( \beta \) is a constant which can be determined when \( v_{\text{min}} = 0 \), at which point the integral is given by the ion saturation current, \( qAne_s u_B \) where \( A \) is the collector area, \( n_s \) is the sheath ion density and \( u_B \) is the Bohm velocity [38]. Differentiating Eq. 3.10 with respect to the discriminator voltage gives,

\[ \frac{dI}{dV_d} = -\beta \theta \frac{q}{M} f(\sqrt{2eV_d/M}) \]  

(3.11)

The first derivative is then proportional to the ion velocity distribution function (IVDF) and is a measure of the distribution of the speeds of the ions directed towards the RPA aperture. After smoothing the raw current-voltage data using a Savitzky-Golay filter, a Matlab script was used to fit a function to the data, the derivative of which was then calculated. The first
derivatives of the sample current-voltage trace are plotted in Fig. 3.13.

In the absence of a beam, a single Gaussian distribution of ion velocities is seen. In this case the highest energy ions will be those which have rolled from the plasma potential, $V_p$, which can be identified from the peak position of the Gaussian curve. In practice this identification is made by fitting a single Gaussian curve to the derivative data $\frac{dI}{dV}$, from which the Gaussian parameters can be extracted easily. In some cases, a single Gaussian curve is not sufficient to provide a good fit and a second is required. An example of such a case is shown in Fig. 3.13a. In this case a second population of ions will have fallen though the plasma potential, plus some other potential structure which increased their energy over that of the first ‘background’ population. This secondary peak is referred to as the ‘beam population’ throughout this thesis and the position of the peak is defined as the ‘beam potential’. The assumptions of Gaussian populations of ions and the definitions used here follows the standard procedure as found in the literature [56]. Further to the plasma and beam potential, the area under the Gaussian curve can be calculated to gain the collected current.

When representing data obtained from the thruster as a function of an operating parameter, it is common to normalise each IVDF to the maximum collected current in that set in order to allow comparisons to be made easily as shown in Fig. 3.14a. To allow clearer visual inspection of such collections of IVDFs, it is common to represent them as a heat map allowing ion currents, plasma and beam potentials to evolve as a function of an independent variable.

The RPA measurements collected in this thesis are used to gain insight into the behaviour of the plasma, however there are limitations to the probe measurements. The probe can collect only a sample of the downstream plasma and consequently cannot give an absolute measurement of the exhaust current. As a result, the collected IVDFs may only indicate relative changes in the downstream plasma currents. Similarly, the potentials of the two ion populations (in cases where a beam is present) used in this thesis are only as a means to understand the trends in the directly measured thrust. The performance of the thruster may be estimated by computing the difference between the potential of the bulk plasma and the potential of the ion beam, which may then be equated with the kinetic energy of the ions and the ion velocity, using Eq. 3.12:

$$u_{ion} = \sqrt{\frac{2e(V_b - V_p)}{m_{ion}}}$$

(3.12)

where $u_{ion}$ is the velocity of the ions, $e$ is the fundamental charge, $V_b$ and $V_p$ are the potentials of the ion beam and the bulk plasma, respectively, and $m_{ion}$ is the mass of the ion. This in
3.3. Probes and Diagnostics

Fig. 3.13: Example RPA current voltage profiles illustrating a case where a beam is present (a) and the case where no beam is present (b). The fitted Gaussian curves are shown in both cases. In the beam carrying case, two fitted Gaussian curves correspond to the background plasma population (dotted curve) and beam population of ions (dot-dashed curve) and in the zero beam case, only a single Gaussian is needed (solid black curve). The local plasma potential, $V_p$, and beam potentials, $V_b$, are also marked with a solid vertical line at the peak of each Gaussian curve.

Turn may be used to calculate the specific impulse of the thruster if we make the simplifying assumption that all of the ions are travelling at the same velocity, $u_{ion}$, parallel to the thrust vector,

$$I_{sp} = \frac{u_{ion}}{g_0}$$  \hspace{1cm} (3.13)

where $g_0$ is the gravitational acceleration at sea level. However, in this thesis, such estimates will not be computed as the specific impulses which can be calculated may be misleading, due to the assumption that the IVDF of the plasma sampled by the RPA represents the IVDF of the entire exhaust plume. In this study, the performances of the RFPT will only be calculated from directly measured forces generated by the RFPT in order to provide an accurate measure of the performance. The RPA probe was positioned 10 cm downstream of the source tube orifice for all measurements made for this thesis.
Fig. 3.14: An example of a set of IVDF’s as a function of an arbitrary independent variable (a) and the equivalent contour map representation of the IVDF’s which is used throughout this thesis (b). The independent variable in this example was the applied magnetic field, however this could also be the RF power input or any other independent variable.

3.3.2 Microwave Resonator Probe

A microwave resonator probe, often referred to as a ‘hairpin’ probe, was supplied by the OU and used to measure the electron density of the RFPT plasma. A schematic of the probe head is shown in Fig. 3.15. The probe consists of an exposed 20 mm probe tip, with the end wires separate by 8 mm. The probe tip couples inductively with a small loop of wire which is connected to the inner conductor of a coaxial cable. The probe tip is also connected directly to the outer shielding of the coaxial cable. The tip is driven by a small current from a microwave generator, which sweeps over a frequency range of 2.4 to 4 GHz.

A directional coupler and oscilloscope are used between the microwave generator and the probe to monitor the forward and reflected signals between the generator and the probe (much like the standing wave ration meter described in section 3.2.5). At certain frequencies a resonance occurs and a standing wave produces a voltage maximum at the probe tip [86]. At this point, the reflected signal is at a minimum. This resonant frequency is given by [86],

\[ f_r = \frac{c}{4L\sqrt{\epsilon}} \] (3.14)
3.3. Probes and Diagnostics

Fig. 3.15: Schematic of the hairpin resonator probe. The hairpin itself is mounted on a ceramic support which slides onto a glass outer casing which contained the coaxial cable leading out of the chamber.

where, \(L\), is the length of the ‘U’ shaped tip, \(c\) is the speed of light in a vacuum and \(\epsilon\) is the relative permittivity in the vicinity of the probe. In the first instance, a frequency sweep is performed, with the probe positioned at the point where an electron density measurement is to be taken, and the vacuum frequency is determined empirically. The relative permittivity of a non-magnetised plasma is given by [86]

\[
\epsilon = 1 - \frac{f_p^2}{f^2} \tag{3.15}
\]

where the plasma frequency is given by,

\[
f_p = \sqrt{\frac{q^2 n_e}{\epsilon_0 m}} \tag{3.16}
\]

and \(q\) is the charge of the electron, \(m\) is the mass of the electron and \(n_e\) is the electron density. When the hairpin is surrounded by plasma, the presence of the electrons changes the relative permittivity between the probe tips and the resonance frequency increases, which is given by [87]

\[
f_r^2 = f_0^2 + f_p^2 \tag{3.17}
\]

The vacuum resonance frequency, \(f_0\), of the probe and the resulting change in resonance frequency may then be related to the electron density according to Eq. 3.18 [86],

\[
n_e = \frac{f_r^2 - f_0^2}{0.81} \tag{3.18}
\]

where \(f_r\) is the resonance frequency of the probe in GHz in the plasma and the electron number density \(n_e\) is in units of \(10^{16} \text{ m}^{-3}\). Figure 3.16 shows a sketch of a typical trace which would be
recorded on the oscilloscope, where the initial vacuum resonance frequency, $f_0$, is recorded and displayed next to the with plasma case, $f_r$, in order to measure the frequency shift.

![Graph showing inverted reflection peaks]

**Fig. 3.16:** Sketch of the inverted reflection peaks shifting from $f_0$ to $f_r$ in the presence of a plasma.

Equation 3.18 may be applied in the case of a weakly magnetised plasma i.e. when the electron cyclotron frequency, $f_{ce}$, is less than the plasma resonance frequency ($f_r > f_{ce}$), as is the case for the presented data [88]. Sheath effects around the probe tip have been investigated to determine the effect on the density measurements relative to other techniques, such as laser interferometry and Langmuir probe measurements [86]. In the present system, the sheath thickness is of order $10^{-1} \text{m}$, which is much less than the inter-wire separation. For this reason, sheath effects have been neglected for the experimental conditions under investigation in this thesis [89, 90].

### 3.3.3 Langmuir Probe

A Langmuir probe, supplied by the ANU, was used to measure the plasma density in source tube regions beyond $z=-40 \text{ mm}$. A schematic of the Langmuir probe used in these experiments is shown in Fig. 3.17.

The probe consists of a 2 mm diameter tungsten plate tip connected to a tungsten filament insulated by ceramic and formed into a long cylinder. The tungsten filament was connected in series with a 10 kΩ resistor and a constant voltage source biased at -85 V. By biasing the probe to a large negative potential, the probe could be operated in ion saturation mode. When a negative bias is applied to a conductor immersed in a plasma, electrons are repelled and an
3.3. Probes and Diagnostics

ion current is drawn to the probe. When biased sufficiently negatively, the current drawn will not increase with increased negative bias. This current is known as the ion saturation current, which can be expressed as [91],

$$ j_{\text{sat}} = 0.6eAnv_b $$  \hspace{1cm} (3.19)

where $A$ is the probe area which is exposed to the plasma, $n$ is the plasma density and $e$ is the charge on the electron. The Bohm condition velocity $v_b$ is, which is given by

$$ v_b = \sqrt{\frac{k_B T_e}{m_i}} $$  \hspace{1cm} (3.20)

where $T_e$ is the electron temperature, $m_i$ is the ion mass and $k_B$ is the Boltzmann constant. In these experiments, the probe was biased to -55 V while immersed in the plasma and the voltage drop across the 10 kΩ resistor was measured using a multimeter. The saturation current was calculated using Ohm’s law and related to the plasma density through Eq. 3.19.

The electron temperature was not measured in these experiments and instead an estimate was used of 5.5 eV. This value is based on measurements of the electron temperature made in two studies in which the source tube geometry, RF power levels and propellant flow were very similar to that in the present study [77, 75]. This introduces an uncertainty in the measurement of the ion densities that were made in this thesis however the present experiments are mostly concerned with the relative increase in the plasma parameters while operating the RFPT in the low field mode. While the electron temperature is not likely to differ much from this assumed
value, the plasma density measured in this thesis could be higher or lower by up to 10–20% if
the electron temperature were to deviate by 0.5–1 eV.

When the probe has a bias applied to it the sheath, which surrounds the tip, has a finite
thickness which results in an increase in the effective current collecting area of the probe. This
increase in $A$ was corrected for using Sheridan’s method [93] and used to correct the calculated
densities.

### 3.3.4 RF Current Probe

![RF Current Probe](Image)

**Fig. 3.18:** The ‘Ion Physics’ RF current probe used to measure the current passing through the
RF connections to the antenna. The output of the probe was then used to calculate the change
in the plasma resistance. These trends were later used to identify the location of low field mode
peak in the plasma density and thrust.

The ‘Ion Physics’ RF current probe used in these experiments and pictured in Fig. 3.18 allows
the RF current flowing through a conductor to be measured. The probe essentially consists of a
series of wire windings, rather like a solenoid, connected to a BNC cable. The current carrying
conductor is passed through the centre of the current probe and a voltage is generated in the
windings that is proportional to the current flowing as a result of the time varying magnetic field
generated by the current carrying conductor. This voltage can be measured using an oscilloscope
and used to infer the current flow. For the present experiments, the current probe was positioned
as indicated in Fig 3.10b and used to gauge the relative plasma resistance. The forward power
dissipated in the antenna and the plasma can be expressed as,

$$P_f = I_{RMS}^2(R_p + R_{ant})$$  \hspace{1cm} (3.21)

where $P_f$ is the forward power, $R_p$ is the resistance of the plasma and $R_{ant}$ is the resistance
of the antenna. The voltage output of the current probe is directly proportional to the current.
flowing through the feed through to the antenna. By substituting for the peak to peak current probe voltage $V_{pp}$, and using an estimate of the vacuum antenna resistance of 0.7 $\Omega$ [8], and rearranging Eq. 3.22 for the plasma resistance, we gain,

$$R_p = \frac{P_f - R_{ant}V_{pp}}{V_{pp}}$$  \hspace{1cm} (3.22)

The value of the antenna resistance used here is arbitrary as we are using the current probe here to identify changes in the plasma resistance, not to calculate the absolute resistance. A change in the plasma resistance indicates that more power is being deposited into the plasma.
Chapter 4

Thrust Balance Development

Introduction

In this chapter, the design and verification of the thrust balance instrument and experimental arrangement is presented. During the initial stages of testing with the balance, a number of issues were encountered relating to plasma interactions with instrument electronics as well as thruster magnetic field interactions. The sources of interference that were encountered during the design phase of the balance are outlined in this chapter as well as the strategies that were used to mitigate them. Simple tests were carried out to facilitate the extensive thrust balance debugging operation which resulted in the successful and reliable use of the thrust balance to measure thrust for a range of different thruster conditions.

4.1 Experimental Arrangement

A pendulum type thrust balance is the preferred method of measuring the thrust produced by an EP system as it most closely simulated the momentum exchange between a thruster and a satellite. When this project was started, pendulum thrust balances had not been used to characterise RFPTs as it is difficult in many cases to mount the thruster, which requires RF and DC power lines as well as and propellant feed lines.

The SSC pendulum type thrust balance is shown schematically in Fig. 4.1. The thruster to be characterised is fixed to the ‘moving plate’ of the balance, which at all times is allowed to swing freely. The moving plate is hung from the ‘fixed plate’ via 0.1 mm thick steel flexures which interface to the fixed and moving plates through interference fits. A ‘Micro-Epsilon’ IDL1700-
Chapter 4. Thrust Balance Development

2 optical triangulation laser is mounted to the fixed plate of the balance and aligned with a ceramic target which is mounted to the moving plate. The laser sensor is used to measure the distance between its sensor and the ceramic target. The laser operates over a 2 mm range which the target must remain between. If the laser is perfectly centred in that range, then the forces applied to the balance must not produce displacements in excess of 1 mm, which would push the sensor out of range. This corresponds to approximately 130 mN and represents the approximate upper limit of thrust which could be measured by this thrust balance.

The thruster was mounted on the thrust balance for all measurements made in this thesis. Figure 4.1 shows a z-axis scale with a number of scale tick marks. This axis will be referred to throughout this thesis to describe the positioning of probes used for these experiments. A convention is set here of the zero point being coincident with the exit of the thruster. The movable backplate, which allows the length of the source tube to be varied (see section 3.1.1), was placed at two positions also marked at 85 mm and 170 mm.

When using the 170 mm source tube length and both the source and exhaust solenoids, the peak magnetic field strength is located at the -85 mm mark. For the experiments in which the 85 mm source tube is used in conjunction with a single solenoid, the peak field strength occurs at the -45 mm mark. The position of the RPA for the experiments in this thesis is also marked at the 100 mm position.

4.2 Interferences and Mitigations

During the commissioning of the thrust balance a number of issues were encountered which created interferences both mechanical and electrical in nature. These interference had to be mitigated in order for thrust measurements to be made.

The RFPT needed to be supplied with current for the solenoids, RF power and propellant gasses. However the moving plate of the balance must be allowed to move freely at all times in order to be able to move in response to an applied force. The RF power delivery lines must attach to the thruster antenna, however the antenna could not be attached to the source tube ss this would cause mechanical interference. This was mitigated by attaching the antenna to 5 mm diameter copper rods which, when connected to the antenna, were able support their own weight and that of the antenna. This allows the antenna to ‘float’ in place and remain fixed relative to the rest of the thruster. The use of thick copper rods to supply the RF power served
Fig. 4.1: Schematic of the SSC pendulum type thrust balance, showing the installation position of the plasma thruster and the z-axis scale referred to throughout this document. The ‘fixed plate’ is fixed relative to the chamber and the moving plate swings freely thus allowing it to move in response to an applied force. In this schematic only one of the two antenna power feed rods can be seen. The second is parallel to the one shown and separated from the first by approximately 20 mm.

also to minimise the inductance of the system connected to the match box. This mitigated the mechanical interferences caused by the antenna and its associated feed lines.

The solenoid current supply lines, when attached to the mounted solenoids, introduce a resistive force to the moving plate. This issue was mitigated through the use of a remote controlled stepper motor which allowed the automated and in-situ calibration of the balance.
This allowed calibrations to be performed while the system was arranged as it would be during firing. Regular calibrations were performed to ensure that the calibration constant did not change over time due to any resistance due to the current feeds. Furthermore, the solenoid supply lines were constructed from thin sheets of laminated copper foil in order to reduce their mechanical resistance to the movement of the thrust balance.

The solenoids were also found to introduce a second source of mechanical interference. When current was supplied to the solenoids, a displacement was found to be produced in the thrust balance. This effect is shown in Fig. 4.2. This was due to a combination of the magnetic field acting on the fixed parts of the thrust balance as well as forces acting between the current carrying supply lines. The mitigation strategy for this issue was twofold. First, all magnetic parts the thrust balance were replaced by non magnetic counter parts. The current supply lines were then rearranged until the effect shown in Fig 4.2 could not be detected by the laser sensor. Thus the issue was entirely mitigated.

![Fig. 4.2: Example plot showing the effect on the thrust balance displacement of the solenoid feed lines when current was supplied to the solenoids. The solenoids are activated at t=20 s. This effect was mitigated entirely through careful rearrangement of the current feed lines and replacement of all magnetic components with non magnetic counterparts. This strategy entirely mitigated the effects shown here.](image)

The propellant feed line would typically be fixed and sealed to the rear end of the plasma source tube to supply the propellant as part of a flight model thruster. While this is possible when using this thrust balance, it was found that this introduced a large resistances to the motion of the moving plate. To mitigate this issue, a gas feed line was constructed from 3.175 mm steel
4.2. Interferences and Mitigations

tubing which was aligned coaxially with a small 5 mm diameter propellant feed hole in the rear of the source tube, which was allowed to electrically float. This required careful positioning each time the thrust balance was used, since the clearance around the gas line was only 0.6 mm. In this way propellant could be fed into the source tube while remaining mechanically isolated from the system and thus the issue was mitigated.

Initially, the displacement of the thrust balance was measured using a DC powered linear voltage displacement transducer (LVDT) which was later replaced by an AC powered LVDT. In both cases, operation of the RFPT revealed susceptibility to electromagnetic interference despite mitigation strategies in the form of mounting them within a grounded steel container. The interference manifested as a step change feature in the output trace of the devices when the RF power generator was activated with zero gas flow supplied to the RFPT. This would appear as an ‘apparent displacement’ but was in fact the result of currents induced in the sensor by the ambient RF power.

Initially, it was thought that once the RF power was being absorbed by the plasma that this would cease to be an issue. However after locking the moving plate of the balance in place with bolts and initiating a plasma, the step change feature was still observed. This was the primary motivation behind using the optical laser displacement sensor.

The laser sensor measures displacement through optical triangulation which is then digitally encoded and transmitted through a shielded coaxial cable. In this way the signal is shielded from analogue electromagnetic interference. This was confirmed by locking the thrust balance in place and initiating the plasma discharge while acquiring data. The plasma was supplied with the maximum power surveyed in this thesis of 500 W and no apparent displacements were found in the output of the laser sensor. This confirmed that this mitigation strategy was successful and that the laser displacement sensor would only record actual displacements of the balance.

While these sources of uncertainty could be entirely mitigated, there are other sources which are harder to quantify and were not accounted for in this thesis. As the thruster fires, heat may accumulate within the thrust balance structure. This may have the effect of changing the mechanical characteristics of the flexures and thus the characteristics of the thrust balance. While calibrations were performed regularly and in-situ, this effect should be explored further in future designs. The impact of plasma electrostatic noise was mitigated in the case of the laser sensor, however interference may be able to enter the laptop that was running the acquisition software. While the presence of such an effect was not observed, coaxial shielded cabling was
Chapter 4. Thrust Balance Development

used throughout in an attempt to mitigate if present.

4.3 Calibration and Data Analysis

The thrust balance was calibrated by finding the relationship between a series of forces applied to the moving plate of the balance and the resulting displacement of that moving plate. A mass was tethered to the moving plate of the balance and allowed to hang vertically. A remotely controlled stepper motor was used to displace the mass horizontally from its resting position using a second tether. Figure 4.3 shows a free body force diagram of the system. This results in a force being applied to the moving plate of the balance, which was calculated from Eq. 4.1. Increasingly large forces were applied to the moving plate by increasing the horizontal displacement of the mass.

Plots of the displacement of the moving plate in response to the applied forces and the corresponding calibration curve of balance displacement as a function of the calculated applied forces are shown in Fig. 4.4. The balance typically exhibits ~1 Hz simple harmonic motion while in use which can be seen in the blue line in Fig. 4.4a. This was filtered out using a Butterworth low pass filter, shown as the black line in the same figure.

![Free body force diagram of the calibration system used to apply a known forces to the moving plate of the thrust balance.](image)

\[ f = \frac{d}{l}mg \]  \hspace{1cm} (4.1)

Calibrating the balance through the application of increasingly large forces allows the linearity
4.3. Calibration and Data Analysis

of the balance to be checked across a range of applied forces. Figure 4.4b shows that the response of the balance is highly linear up to an applied force of $\sim 18$ mN which encompasses the thrust range that is expected to be generated by the experiments in this thesis. The gradient of the linear trend in Figure 4.4b gives the calibration constant of the system in mm/mN and gives us the correspondence between the displacement of the moving plate in response to an applied force.

![Figure 4.4a](image-a.png)

![Figure 4.4b](image-b.png)

**Fig. 4.4:** (a) Thrust balance displacement measured using the optical laser triangulation sensor as a function of time. Each ‘step’ represents an increasingly large calibration force being applied to the balance through successive displacements of the hanging calibration mass. Both the raw unfiltered data is shown (blue) and the result of applying a Butterworth low pass filter to the measurements (black). (b) The calibration curve of balance displacement as a function of the applied calibration force corresponding to the graph (a). The equation of the linear fit is shown overlain, where the gradient represents the calibration constant of the system in mm/mN. Note that the non-zero y-intercept at zero applied force represents the equilibrium point about which the balance oscillates. This is arbitrary as the laser displacement sensor operates over a 2 mm range that is calibrated by the manufacturer.

Thrust measurements were made by activating the thruster and recording the resulting displacement of the balance using the laser displacement sensor. This is repeated a number of times for a given operational set point. The data is filtered using the same Butterworth filter and the difference between the displacement of the balance at the ‘thruster on’ and ‘thruster off’ points
are extracted from the data using a MatLab script. The thrust is then calculated by dividing the collected displacements by the calibration constant, the results of which are averaged to gain the mean measured thrust.

The use of a stepper motor to displace the mass allows for highly repeatable and semi-automated calibration process, which may be completed while the vacuum chamber of closed and under vacuum and therefore under identical conditions to those under which thrust measurements will be taken. Since the characteristics of the balance could change over time if, for example, an electrical feed attached to the moving plate shifts position, it was important to be able to make in-situ calibrations of the balance.

The calibration mass was measured using a digital scale, the lengths of the tethers were measured using digital callipers and the balance displacement using the laser displacement sensor. The relative uncertainty in the calibration constant can be calculated by combining the uncertainties in each of these measurement instruments which was found through the manufacture specifications. The relative uncertainly in the calibration constant was calculated at 2%. This is small since the measurements were made using high precision digital instruments. For example in the case of the optical laser triangulation sensor, the manufacturer quotes a resolution of 0.1 µm which, using the calibration constant shown in Fig. 4.4b corresponds to a thrust measurement resolution of 0.01 mN.

When making thrust measurements through repeated firing of the thruster at a given operational set point, the standard error in the measured thrust is calculated and is typically about 0.1 mN. This is the precision of the balance and is calculated for every set of repeated measurements and is plotted centred upon the mean value of the measured thrust.

The preceding description of the calibration and thrust measurements procedure as well as the use of the standard error when quoting the uncertainly in the measured thrust has become standard practice within the literature however this is not the only way in which the thrust balance can be calibrated. Independent studies have now used this same thrust balance [74] and balances based on this design [75, 106, 70, 43] all of which find a similar standard error in their measurements. While this provides a degree of confidence in the measurements which are made in this thesis, the thrust balance should in future be independently verified using a known, well characterised thruster or load cell. An extensive characterisation of the thrust balance with respect to the plasma noise for two separate thrusters may be found in Ref. [94] and further SSC studies which uses the same thrust balance may be found in Refs. [73, 90, 95, 96, 76, 97].
4.4 Cold Gas Measurements

Since the standard error of the thrust balance is often quoted the thrust of a cold gas source was measured as a check, to establish how well the thrust balance measured the thrust of an unknown source attached to the balance.

A 6.35 mm plastic tube was affixed to the moving plate of the thrust balance and a mass flow controller was used to meter the gas flow. The propellant flow was set and a short period of about 10 seconds was allowed to elapse in order for the chamber pressure to reach the new equilibrium. This was done before the data acquisition was started to avoid pressure differentials in the chamber which could influence the measurements. The data acquisition was then started, the propellant flow was stopped after 40 s and the data acquisition was stopped at \( t = 80 \) s. This procedure was repeated 5 times for each flow set point. The data was then filtered as described previously and the differences between the mean equilibrium positions of the balance were calculated. Since the gasses are exhausting into a vacuum of \( \sim 1 \) Pa, the gas is in the choked flow regime. The cold gas thrust of a choked propellant mass flow \( \dot{m} \) is given by Eq. 4.2,

\[
F = v_e \dot{m} = \sqrt{\gamma RT \dot{m}}
\]

where, \( v_e \) is the exhaust velocity of the gas, \( \gamma = \frac{c_p}{c_v} = 1.667 \) is the ratio of the specific heats for noble gases, \( R \) is the molar gas constant which is 99.2 J Kg\(^{-1}\)K\(^{-1}\) for krypton, and \( T \) is the absolute gas temperature and is assumed to be \( T = 298 \) K. Figure 4.5 shows the theoretically calculated and the mean measured cold gas thrust which are in good agreement to within 0.1 mN.

4.5 RFPT Operation

The capability of the thruster to measure the thrust of the RFPT was first assessed by running the thruster at zero applied field and as a function of the input RF power between 200 and 500 W and for a nominal argon flow rate of 0.5 mg/s. This was the minimum and maximum RF power investigated for this thesis. The measured thrust of the SSC RFPT as a function of the input power and zero applied magnetic field is shown in Fig. 4.6.

The thrust was found to increase from 0.1 mN at 200 W to 0.3 mN at 500 W. The error bars shown in Fig. 4.6 are the standard deviations calculated from 6–10 measurements and centred upon the mean thrust value. The standard deviation of the measurements was found to be
**Fig. 4.5:** Theoretical and measured cold gas thrust made using Krypton exhausted from a 6.35 mm diameter tube. The standard deviation of these measurements were calculated as 0.1 mN. This represents the accuracy of the balance used to determine the accuracy of the balance. The statistical deviation of the measurements were calculated at 0.1 mN about 0.1 mN at each power level. We shall not discuss in this chapter the origin of the thrust measured in this experiment as this is topic is introduced in chapter 6 and discussed in detail in relation to measured performance of the SSC RFPT and RFPTs found in the literature. This experiment did however demonstrate that measurements could be made of the SSC RFPT using the thrust balance and thus that the mitigation strategies have succeeded. In the next chapter, the results of experiments conducted for this thesis are presented.
**Fig. 4.6:** Thrust versus forward RF power for an argon propellant flow of 0.5 mg/s and a zero applied magnetic field. The theoretical cold gas thrust generated from 0.5 mg/s of argon is shown for comparison.
Chapter 5

Experimental Results

In the previous chapter the design of the thrust balance and the methodology behind its use was described. In this chapter, the results of experiments to determine the effect of the low magnetic field mode on the performance of the SSC RFPT are presented. The thrust, plasma density and the IVDFs have been measured and a section has been devoted to each of these parameters. These are presented as a function of the imposed magnetic field, propellant flow, forward RF power and in some cases the location along the thruster axis.

A secondary aim of the experiments was to ascertain the possibility of producing further performance enhancements by operating in the low field mode while using a more compact source tube length. This was motivated by the requirements of the project sponsors for a compact, high efficiency thruster. To this end, the peak thrust point of the low magnetic field mode was identified for this source tube length and subsequent measurements of the thrust, plasma density and IVDFs were made.

The results of the experiments for the two source tube lengths are presented side by side for ease of comparison and general descriptions of the trends are given. The features most relevant to the aims of this study are also identified and are discussed in more depth in chapter 6.

5.1 Thrust Measurements

5.1.1 Variation with Imposed Magnetic Field and RF Power

The thrust produced by the SSC RFPT as a function of the imposed magnetic field and forward RF power is shown in Fig. 5.1 for source tube lengths of 170 mm and 85 mm. At the 200 W
power level, the thrust starts low and steadily increases to a peak value, before decreasing at higher imposed magnetic fields for both the 170 and 85 mm source tubes. This peaked feature is the low magnetic field mode which is investigated in this thesis. The magnetic field strength for which the measured parameter (in this case thrust) peaks, is referred to hereafter as $B_0$.

Comparing the two source tube lengths, the $B_0$ thrust was found to peak at 1.2 mN at a $B_0$ field strength of 6.6 mT for the 85 mm tube, compared to 0.38 mN at a $B_0$ field strength of 9.5 mT for the 170 mm tube. This is an increase over the zero field thrust level by a factor of 3 for the 170 mm tube compared to a factor of 4 for the 85 mm tube. Interestingly, the thrust drops much more rapidly for fields above $B_0$ when using the 85 mm tube, occurring over a magnetic field increment of 0.4 mT. This field increment is generated by a solenoid current of 0.1 A, which was the lowest increment which could be supplied by the power supply used. It is for this reason that the region between the $B_0$ peak and higher magnetic fields was not explored experimentally any further for the 85 mm tube. The sudden drop in the thrust will however be discussed further in the next chapter, section 6.1.2, when the evidence for the presence of low field helicon waves is discussed in more detail.

![Figure 5.1: Thrust measurements as a function of imposed magnetic field for source tube lengths of 170 mm (a) and 85 mm (b). Measurements are shown for a constant propellant flow rate of 0.5 mg/s and for forward RF powers of 200–500 W.](image)
300 and 500 W, shown in Fig. 5.1a. In this case, thrust levels were generally higher for increased RF powers, which remains true across the range of applied magnetic fields. The value of $B_0$ was also found to increase with RF power, which can be seen in Fig. 5.1a as a shift in the peak positions to higher magnetic fields.

The 85 mm source tube was used in this study to explore the performance of the SSC RFPT while using a more compact source tube, in line with objective 4. Since the thrust was found to be maximised at $B_0$ for each RF power level while using the 170 mm source tube, the $B_0$ performance at increased power levels for the 85 mm source tube was of particular interest, as the thrust was found to be generally higher in this case compared to the longer source tube. As we are particularly interested in the high thrust $B_0$ point when considering the 85 mm source tube, it was not therefore necessary to measure the thrust as a detailed function of the imposed magnetic field. Instead, the $B_0$ point was identified by using the RF current probe, a technique which has been used extensively in the literature to identify low magnetic field modes [8, 98, 77].

Figure 5.2 shows the normalised plasma resistance measured using the RF current probe as a function of the imposed magnetic field. The peaks of these traces correspond to the low magnetic field mode peaks. The $B_0$ field strength was identified for power levels of up to 500 W while using the 85 mm source tube and the thrust levels were then measured at that point. The $B_0$ thrust measurements as a function of the RF power are shown for the 85 mm source tube in Fig. 5.1b. The thrust was found to increase with RF power and to be higher than the $B_0$ thrust measured for the same forward RF power level when using the 170 mm source tube. For example, at the highest power level of 500 W, the $B_0$ thrust using the 85 mm source tube is a factor of 4 higher than that of the 170 mm source tube case. This increase in the performance of the thruster will be discussed further in the next chapter.

### 5.1.2 Variation with Imposed Magnetic Field and Propellant Flow

The thrust produced while operating in the low field mode was also investigated as a function of the propellant flow. The operation of the SSC RFPT as a function of the propellant flow is of interest as there is typically a trade-off between the thrust of an electric thruster and the specific impulse which, for a constant power input, are related by the propellant mass flow, according to Eq. 1.5. As in the experiments of the previous section, the 170 mm source tube was used to investigate the detailed trend of the thrust as a function of the imposed magnetic field for a range of propellant flows, and the 85 mm source tube was used to investigate operation at the
Fig. 5.2: Normalised plasma resistance as a function of the imposed magnetic field and RF power for a propellant flow of 0.5 mg/s and while using the 85 mm source tube. The voltage across the output of the RF current probe was measured while operating the SSC RFPT Eq. 3.22 was then used to calculate the plasma resistance given an estimate of the antenna resistance. This was then normalised to the peak resistance of the 500 W condition to allow comparison. The voltage measured across the output is linearly proportional to the current flowing to the antenna, thus dips in the voltage indicate a decrease in current to the antenna and an increase in the plasma resistance. In this way, $B_0$ could be quickly identified, allowing the highest performance points to be measured without the need to record the thrust across the entire low field mode.

$B_0$ point.

Figure 5.3a shows thrust as a function of the imposed magnetic field for a range of power inputs and a propellant mass flow of 0.6 mg/s. The low field mode $B_0$ peaks are again identified and are found to move to higher field strengths with increasing RF power. With the RF power fixed at 200 W, the thrust was also measured for a small range of propellant flows as a function of the imposed magnetic field the results of which are shown in Fig. 5.3b. Again, low field mode peaks are found but do not shift noticeably with changes in the propellant flow. Interestingly, $B_0$ is $\sim 7$ mT for the 200 W power level in Fig. 5.3a which is noticeably lower than the $\sim 9$ mT found across the propellant flow range in Fig. 5.3b. This will be discussed further in the next
5.1. Thrust Measurements

The 85 mm source tube was again used to explore the maximum thrust $B_0$ points, this time while operating at a reduced propellant flow rate of 0.3 mg/s. Figure 5.3c shows the $B_0$ thrust
as a function of power for the 85 mm source tube for propellant flows of 0.3 and 0.5 mg/s, the latter being replotted from Fig. 5.1b for comparison. The thrust at 0.3 mg/s is found to be lower across the power range surveyed, compared to the 0.5 mg/s case, but is still higher by a factor of 1.6 compared to the 170 mm source tube case.

5.2 Plasma Density Measurements

Measurements of the plasma density were made while operating the SSC RFPT using both the 170 mm and 85 mm source tube lengths. These measurements were taken as a function of the axial distance along the thruster and as a function of the imposed magnetic field strength. Measurements of the plasma density when using the 85 mm source tube have also been made while operating at $B_0$ and as a function of power to correspond to the $B_0$ thrust measurements which are described in the previous section. The propellant flow rate was fixed at 0.5 mg/s for all measurements.

5.2.1 Variation with Magnetic Field and Axial Location: 170 mm Source Tube

Figure 5.4 summarises measurements made of the ion density while using the 170 mm source tube and an RF power of 200 W. Measurements were made at 20 mm intervals between $z=40$ mm and $z=-40$ mm and at each point measurements were taken as a function of the applied magnetic field. The ion density was found to increase with magnetic field strength to a peak value, before reducing for higher imposed fields. This trend was found to exist at each axial location surveyed. Interestingly, the $B_0$ field strength was found to increase from 8–10 mT as the probe was moved from the downstream position at 40 mm to -40 mm within the source tube. Comparing the ion density peak positions, $B_0$, to those identified from the thrust measurements shown in Fig. 5.1a, the peaks are found to be within 2 mT of each other for the 200 W RF power case and are therefore likely to be related to the same low field mode.

5.2.2 Variation with Magnetic Field, Axial Location and RF Power: 85 mm Source Tube

With the 85 mm source tube installed, the ion density was measured using a Langmuir probe as a function of the imposed magnetic field, the axial distance along the source tube and also as a
5.2. Plasma Density Measurements

Fig. 5.4: Electron density measurements made using a hairpin resonator probe as a function of the imposed magnetic field strength. Measurements were made using the 170 mm length source tube for an input power of 200 W and propellant flow of 0.5 mg/s. Measurements are shown for a range of axial positions relative to the source tube exit. The occurrence of the peak density is coincident with the peak in the thrust measurements shown in Fig. 5.1a.

Figure 5.5a shows the ion density as a function of the imposed magnetic field while the probe is positioned 20 mm downstream of the thruster orifice. The density increases with magnetic field and peaks at a field strength of 6.4 mT before decreasing at higher field strengths. While operating at this \( B_0 \) point and at the same 200 W power level, the ion density was measured along the axis of the thruster and is shown in Fig. 5.5b. The probe was recessed far into the rear of the source tube in this case and measurements were made beneath the antenna region. Here the plasma density is generally highest, as this is where the majority of the RF power is deposited. Thus, in moving from the furthest extent into the source tube at -55 mm, to where the antenna was centred at -40 mm, the plasma density increased slightly to a peak value. Moving further down the length of the source tube towards the orifice at 0 mm and to the furthest downstream location at 100 mm, the density smoothly decreases by about an order of magnitude.
Chapter 5. Experimental Results

The observation that the plasma density decreases smoothly along axis of the thruster while operating at the $B_0$ peak of the low field mode is important, as current free double layer containing plasmas have well documented axial ion density profiles. This will be revisited in the next chapter, where the possibility of current free double layer formation in the SSC RFPT is discussed.

**Fig. 5.5:** Ion density measurements made using a Langmuir probe; positioned 20 mm, in the plasma plume as a function of the imposed magnetic field for a propellant flow of 0.5 mg/s and an input power of 200 W (a), as a function of the axial distance along the length of the source tube for an input power of 200 W and an applied $B_0$ field of 6.8 mT (b) and positioned -45 mm into the source tube as a function of the input power for a propellant flow of 0.5 mg/s (c)
5.3 IVDF Measurements

Measurements of the IVDFs were made while operating the SSC RFPT with both the 170 and 85 mm source tubes. While using the 170 mm source tube, measurements were made as a function of the imposed magnetic field strength for two power levels (200 and 400 W) and for a fixed propellant flow of 0.5 mg/s. While using the 85 mm source tube, measurements were made as a function of the imposed magnetic field strength for an RF power of 200 W and a propellant flow of 0.5 mg/s, as well as a function of the RF power while operating at the $B_0$ field condition for propellant flows of 0.3 and 0.5 mg/s. All IVDFs were measured with the RPA located 10 cm downstream of the source tube orifice.

5.3.1 Variation with Magnetic Field and RF power: 170 mm Source Tube

Figure 5.6 shows IVDFs, measured as a function of the imposed magnetic field strength while using the 170 mm source tube, a propellant flow rate of 0.5 mg/s and for input powers of 200 and 400 W. The total current collected and the beam current for each power level are plotted below each of the IVDFs.

The total collected currents at both power levels were found to start low and to increase with the applied magnetic field up to the maximum collected current, which then decreases at higher field strengths. For both power levels, the magnetic field strength at which the total current is maximal is approximately 8 mT. Comparing this with the value of $B_0$ identified through the thrust measurements shown in Fig. 5.1a, a good agreement is found only at the 200 W power level. Figures 5.1a, 5.2 and 5.3 show that the value of $B_0$ increases with increasing RF power. We would expect therefore, that the $B_0$ field strength corresponding to the peak total current of the 400 W discharge, shown in Fig. 5.6d, would be higher than that of the 200 W discharge. This is found not to be the case and a possible explanation for this is discussed in the following chapter. Despite the disagreement in $B_0$ at the 400 W power level, the total current displays the general characteristic of the low field mode that are seen in both the thrust and the ion density measurements presented in the earlier sections. Indeed, it would be expected that an increase in the ion density within the source tube region should lead to a greater ion flow from the source tube orifice and downstream into the RPA to be collected.

The local plasma and beam potential for both power levels share some common features but differ sightly in their details. With reference to Figs. 5.6a and 5.6b, the beam potential tracks
Fig. 5.6: Normalised IEDFs measured as a function of the imposed magnetic field strength while using the 170 mm source tube, a propellant flow of 0.5 mg/s and for RF power levels of (a) 200 W and (b) 400 W. The beam and local plasma potential are plotted superimposed upon the IVDFs with triangles and circles respectively. The total and beam current are shown below each plot for (c) 200 W and (d) 400 W. (graphs colour on-line)

the trend of the plasma potential for both power levels. This should be expected, as the beam ions have rolled down the plasma potential and whatever potential drop has formed between the plasma source and the RPA orifice. Thus, changes in the plasma potential will modify the beam potential trend. The plasma potential is also found to increase by approximately 10 V
through the application of an 0.8 mT magnetic field in both cases. At higher imposed fields, the trends begin to differ. At 200 W, the beam and plasma potential stay relatively flat, increasing by around 10 V to a local maximum at about 5 mT. At this point both the beam and plasma potential begin to decrease to a local minimum at 8–9 mT, in correspondence with the increase in the total collected current to its maximum at 8–9 mT, shown in Fig. 5.6c. At 400 W, the plasma potential slowly decreases by 20 V across the applied magnetic field range, while the beam potential essentially remains flat at 80 V. For both power levels, the beam current was found to make only a minor contribution to the total current collected. At zero applied field, the beam contributions are around 20–25% which reduce to below 1% as the total currents reach their global maximum. The important observation here is that for both power levels, the total collected current increases to a peak in the same manner as the upstream and downstream ion density, shown in Fig. 5.4 and the thrust, shown in Fig. 5.1a. This fact will be referred to later in chapter 6, where we shall discuss the low field mode operation of the SSC RFPT in the context of low field helicon wave modes and current free double layer formation.

5.3.2 Variation with magnetic field: 85 mm source tube

Figure 5.6 shows the evolution of the IVDF measured as a function of the imposed magnetic field while using the 85 mm source tube and an RF power of 200 W. Here a more complex structure was found compared to the 170 mm source tube case.

Between the zero field condition and 6.6 mT, the plasma potential remains essentially constant, while the beam potential gradually increases by about 10 V. Over the same magnetic field range the beam current, shown in Fig. 5.7b, remains essentially constant and the total current increases slowly to its maximum at 6.6 mT. This results in the ion beam comprising about 28% of the total current at the zero field condition, which reduces to 16% of the total current at 6.6 mT. For magnetic fields above 6.6 mT the beam and plasma potential increase suddenly by around 15 V, the total current decrease to about 30% of its maximum and the beam current essentially reduces to below 1% of the total current. These changes in the potential and collected current occur over a magnetic field range of 6.6–7.3 mT, and correspond to the steep reduction in the thrust levels, shown in Fig. 5.1b, the drop in plasma resistance shown in Fig. 5.2 and the peak ion density shown in Fig. 5.5a. The correspondence between the turning points of each of these parameters to a field strength of about 6 mT will be discussed further in section 6.1.2 where it shall be compared to literature studies in which low field helicon waves have been observed at
Fig. 5.7: Normalised IEDFs measured as a function of imposed magnetic field strength while operating the SSC RFPT with the 85 mm source tube, an RF power of 200 W and a propellant flow of 0.5 mg/s (a). The corresponding normalised ion beam and total collected current is shown in (b). (color online)

similar field strengths and operating conditions.

5.3.3 Variation with RF power at $B_0$: 85 mm source tube

Measurements of the IVDF as a function of the RF power have been made while operating using the 85 mm length source tube, at the $B_0$ magnetic field condition, for an RF power of 200 W and propellant flows 0.3 and 0.5 mg/s. This complements the thrust measurements made while operating the source at $B_0$, presented in section 5.1, Fig. 5.3c. Figure 5.8 shows the evolution of the IVDF’s as a function of the RF power for 0.3 mg/s and 0.5 mg/s. For both flow conditions, the plasma potential remains constant; 10 V for the 0.3 mg/s case and 15 V for the 0.5 mg/s case across the RF power range. The beam potential however increases linearly with the power by about 10 V for both flow conditions. The beam and total current for the 0.3 the 0.5 mg/s flow conditions, shown in Figs. 5.8c and 5.8d, increase linearly with the RF power. This is to be expected, as we have already seen that the ion density within the source tube increases monotonically with the RF power, as for example in Fig. 5.5c. In both cases the contribution to the total current given by the ion beam remains essentially constant across the power range, however the contributions differ between the two flow conditions; 23% for the 0.3 mg/s case and
Fig. 5.8: Evolution of the IVDF’s as a function of forward RF power while operating the RFPT at $B_0$ and using a propellant flow of (a) 0.3 mg/s and (b) 0.5 mg/s. The total current collected by the RPA and the ion beam current corresponding to each flow condition are plotted beneath. (color online)

14% in the 0.5 mg/s case. The latter value agrees to within a couple of percent of the value calculated from Fig 5.7b at the $B_0$ field condition of 6.6 mT. It is also interesting to compare the beam contributions between the two source tube lengths as a function of the imposed magnetic field, now that values have been obtained for both.

We see that at the zero magnetic field condition the ion beam contributes between 20–30% of the total current collected when using both the 170 and 85 mm source tubes. However when
the total current reaches its global maximum at $B_0$, when the thrust is a maximum, the ion beam contributes below 1% of the total current for the 170 mm case and around 14–16% when using the 85 mm source tube case. This final observation will be revisited in the next chapter in section 6.1.1 where the evidence for current free double layer formation in the SSC RFPT is collected and explored.

5.4 Summary

The SSC RFPT was operated using two source tube lengths and over a range of propellant flows, RF powers and imposed magnetic fields. Measurements of the thrust, IVDF and ion density were made and these experimental results have been presented and described. The important features relevant to the investigation have been identified and instances of unexpected behaviour of the plasma source have been highlighted.

Peaks in the thrust, total current collected by an RPA and ion density were found for imposed magnetic fields of 6–12 mT when using both the 170 and 85 mm source tubes. This was identified as the low field mode and the field strength that produces a peak for a given set of parameters was defined at $B_0$. The field strength $B_0$ was found to generally increase with input RF power but also varied with axial location along the source tube, where weaker fields were found to produce ion density peaks within the source tube and stronger fields producing peaks in the downstream regions.

Use of the 85 mm source tube was found to increase the $B_0$ thrust levels by up to a factor of 4, compared to when using the 170 mm source tube. Use of the 85 mm source tube also resulted in an unexpected modification to the low field mode behaviour. This manifested as sudden drops in the thrust level, total current collected by the RPA and decreases in the plasma resistance for imposed fields above $B_0$. This is in contrast with the 170 mm source tube which produced smooth changes in all the measured parameters for fields above $B_0$.

Ion beams were found to be produced when using both the 170 mm and 85 mm source tube lengths. It was found that the beam comprises a larger fraction of the total current at the zero field condition, when using the 170 mm source tube, whereas for the 85 mm source tube the beam was found to be strongest at the maximum total current point i.e. at the $B_0$ field strength when the thrust was maximised. This ion beam produced at the $B_0$ point was also found to be enhanced from 14–16% of the total current to 23%.
In the next chapter we explore the nature of the low field mode observed here, in the context of low field modes investigated in the literature. The performance of the SSC RFPT while operating in the low field mode is also explored further and compared to RFPTs found in the literature.
Chapter 6

Discussions

In chapter 2 it was suggested that RFPTs could have advantages over current EP technologies, such as reduced complexity, increased lifetime and novel flexibility in terms of the range of propellants with which it may be operated. These new capabilities could allow novel mission scenarios to be realised, such as the use of in-situ propellants to allow long duration deep space missions and the economical use of hydrazine-pressurising gases as a propellant source to safely dispose of retired satellites.

Despite the advantages of RFPTs, their efficient operation generally requires the application of strong magnetic fields in order to leverage more efficient power deposition modes, commonly via helicon waves. Since strong magnetic fields are more easily supplied using permanent magnets, this has been investigated extensively, revealing promising efficiency levels. However, attaching strong magnets to a satellite produces orbit-perturbing torque forces through interactions with the Earth’s magnetosphere. The applied fields may also interact with payloads such as magnetometers that are used for navigation. These issues are compounded by the inability to deactivate or field reverse the magnets if required.

Helicon wave sustained plasmas have also been produced for weak applied magnetic fields, typically below 10 mT and for RF powers of below 500 W. An RFPT designed around the use of a low field helicon mode could benefit from increases in thrust efficiency while minimising the required magnetic field, which could be supplied easily with electromagnets. In this thesis, an empirical approach was adopted to explore this possibility. An RFPT was constructed and operated in a low magnetic field mode. A thrust balance instrument was developed to measure directly the thrust that it produced and plasma probes were used to characterise the plasma.
Chapter 6. Discussions

In the present chapter these results are discussed across two main sections. The first examines the low field mode operating in the SSC RFPT and the second compares the performance of the thruster to RFPTs which can be found in the literature. It is demonstrated that current free double layer formation is unlikely in the SSC RFPT according to the current understanding of their formation. We compare the behaviour of the plasma source with low field helicon sources and show that this is the most likely mechanism behind the behaviour of the SSC RFPT. Finally we show that the performance of the SSC RFPT is comparable to those which can be found in the literature in terms of thrust efficiency and specific impulse, but that it achieves this using generally lower applied magnetic fields.

6.1 The SSC RFPT Low Field Mode

6.1.1 Current Free Double Layer Formation

Magnetic field induced current free double layer formation has been shown to produce characteristic changes in the IVDF and the ion density of an expanding plasma. Measurements of the IVDF and ion density were made for this thesis and presented in sections 5.2 and 5.3. These measurements are used as our basis for comparison. The discussion follows three themes, thermal ion magnetisation, ion beam formation, and ion density characteristics, each of which pertain to a characteristic of current free double layer formation and is examined in turn.

6.1.1.1 Thermal Ion Magnetisation

Current free double layers have been observed to be ‘triggered’ by ion magnetisation [71, 6], that is that the magnetic field appeared to be the driving force behind the formation of the double layer structure. In these studies, ion energy distribution function measurements were made as a function of the imposed magnetic fields between 5 and 20 mT and for source tube radii ranging from 23 to 75 mm. Ion beam formation was found to correspond to the point of weak radial confinement of the source plasma ions, where the threshold of radial confinement was defined as the point at which the ion cyclotron radius equals the radius of the source tube. This was referred to as the ‘transition point’. The peaks in the measured thrust, plasma density and total collected current which were observed in the SSC RFPT at \( B_0 \) are also a kind of transition and we can test whether this is a transition to a double layer containing plasma.
The theory of formation assumes that it is the magnetisation of thermal ions within the source tube that drives the formation of the structure. An ion thermal energy of 0.2 eV was assumed in [6] based on measurements of a similar source made using laser induced florescence [99]. This value of the thermal ion energy was then used to calculate the ion gyroradius for a source tube radius of 23 mm, an RF power of 200 W and a chamber pressure of 80 mPa. These conditions have been investigated in this thesis for both the 170 mm and 85 mm source tube lengths and, in particular IVDF measurements have been made as a function of the imposed magnetic field [6]. Measurements of the thermal energy of ions in the SSC RFPT were not made in this thesis, however, we may calculate the gyroradius of the thermal ions within the SSC RFPT using a thermal energy estimate of 0.2 eV as doing so does not introduce more assumptions than are already present in the original study. The gyroradius of the source plasma ions can be calculated from Eq. 6.1,

\[ r = \frac{v_i}{\omega_{ic}} \]  

(6.1)

where the ion cyclotron frequency is given by,

\[ \omega_{ic} = \frac{|q|B}{m_i} \]  

(6.2)

and the thermal ion velocity is given by,

\[ v_i = \sqrt{\frac{8k_BT_i}{\pi m_i}} \]  

(6.3)

We wish to know whether the presence of a current free double layer is responsible for the peaks in the measured thrust at the field strength \( B_0 \), as presented in section 5.1. From Fig. 5.1, \( B_0 \) was found to be 9.5 mT while using the 170 mm source tube and 6.6 mT while using the 85 mm source tube while operating the SSC RFPT at 200 W. This gives a Larmour radius of 48 mm for the 170 mm source tube case and 69 mm for the 85 mm source tube, which are both over a factor of 2 greater than the SSC RFPT source tube radius of 22.5 mm. According to the theory of double layer formation given in [6], a double layer was not present in the SCC RFPT when the performance was found to peak.

### 6.1.1.2 Ion Beam Formation

The original studies did not observe ion beams for magnetic fields below 20 mT while using a 23 mm radius source tube. At the ‘transition point’, ion beams were formed and the strength
of the beam did not diminish with increasing imposed magnetic fields. When operating the SSC RFPT using the 170 mm source tube, very weak ion beams were formed at $B_0$, shown in Fig 5.6d, which comprised about 1% of the total collected current. This remained the case for higher applied magnetic fields above $B_0$.

When using the 85 mm source tube, the ion beam current at $B_0$ was a higher fraction of the total current, at about 16%. However, applied fields above $B_0$ drastically reduced the ion beam fraction to below 1% of the total collected current. In summary, we see that for both source tube lengths, the pattern of ion beam formation with respect to the imposed magnetic field does not support the existence of a current free double layer in the SSC RFPT.

### 6.1.1.3 Plasma Density Characteristics

Current free double layer formation produces characteristic patterns of ion density as a function of the axial location along the thruster and as a function of the imposed magnetic field. The transition to a current free double layer containing plasma, when triggered by the application of a magnetic field was found to produce a large increase in the plasma density which remained relatively constant for further increases in the applied field. Furthermore, once the transition had occurred, the ion density along the source tube axis and into the diffusion chamber did not vary in a smooth fashion, but instead contained a discontinuity located at the boundary between the source tube exit and the diffusion chamber [71, 62].

In section 5.2, the plasma density was measured as a function of the imposed magnetic field and axial location for the two source tube lengths. Measurements of the plasma density as a function of the imposed magnetic field (shown in Figs. 5.4 and 5.5a) show that rather than increasing to a constant maximum with the applied magnetic field, the density increases to a peak value and then decreases for fields above the $B_0$ field strength. This was observed while using both the 170 and 85 mm source tubes.

Measurements of the axial ion density characteristics of the SSC RFPT while using the 85 mm source, presented in section 5.2, Fig. 5.5b, show a smooth decrease in the ion density as a function of the axial distance. This does not exhibit the kind of discontinuity at the source tube exit (located at the 0 mm position) that was found in the literature. Thus, again, the experimental measurements made of the SSC RFPT plasma properties do not support the possibility of current free double layer formation.

In each case outlined in sections, 6.1.1.1, 6.1.1.2 and 6.1.1.3 the behaviour of the SSC RFPT
does not conform to trends found in the literature that are associated with current free double
layer formation. Thus, we conclude that a current free double layer was not responsible for the
performance peaks found for applied magnetic fields $B_0$. This ends our discussion of current
free double layer formation and we now turn our focus to the low field helicon wave mode.

6.1.2 Low Field Helicon Waves

Helicon wave sustained plasmas exhibit bulk properties that vary characteristically with the
imposed magnetic field and input RF power. In the case of low field helicon waves, the most
obvious characteristic is the presence of peaks in the plasma density, plasma resistance and total
current collected by an RPA, as a function of the imposed magnetic field.

These properties have been measured within the SSC RFPT and in this section we examine
their behaviour compared to low field helicon sustained plasmas which can be found in the
literature. We show that although direct evidence can not be provided in the form of wave field
measurements, strong evidence exists to support the existence of a low field helicon mode within
the SSC RFPT.

6.1.2.1 Plasma Density Peaks and Ion beam Formation

Peaks in the plasma density have been observed experientially \cite{78, 7, 98} in helicon wave produced
plasmas, which have also been the subject of modelling efforts \cite{78, 9}. These peaks have been
shown to accompany the production of ion beams \cite{77, 98}, peaks in the plasma resistance \cite{9, 77}
and, more recently, peaks in thrust levels measured using an MFMI \cite{77}. Since the geometry
and power levels used in these studies vary considerably, we shall make comparisons between the
SSC RFPT and these studies on the basis of the relative changes in experimentally measured
parameters.

In section 5.2, the plasma density of the SSC RFPT was shown to increase to a peak value as
a function of the applied magnetic field for a given RF power. The plasma density was found to
increase by a factor of 6–20 for the 170 mm and 85 mm source tubes respectively, when measured
20 mm downstream of the source tube exit and for an RF power of 200 W. Low field helicon
mode literature studies also show large relative density increases as a function of the applied
field of a factors of 3–10 \cite{77, 98, 7, 8}. The minimum and maximum densities varied between
these studies and were different to those measured for this thesis, owing to differences in RF
power levels (200–400 W) and the location at which the measurements were taken (within the
source [98, 7] and downstream of the SSC RFPT and in [77]).

The total collected current from a downstream RPA can be compared with plasma density measurements made within the source tube, since ions born in the source flow downstream and may be collected. Increases in the total current that mimic the plasma density trends [8, 100] as a function of the imposed magnetic field have been found through RPA measurements. In section 5.3, Fig. 5.6, the total current collected downstream of the SSC RFPT was found to also follow the trend of the plasma density measured within and downstream of the source tube, shown in Fig. 5.4, while using the 170 mm source tube.

Ion beams were also found to form at the peak of the low field mode [77, 98]. In section 5.3, we saw that the low field mode peak did not lead to the formation of strong ion beams, which formed about 1% of the total collected current. In contrast, when using the 85 mm source tube, ion beams were found to form at the peak of the low field mode and made up about 16% of the total collected current. Since we have already established that there is not a double layer forming as a result of ion magnetisation, another process must be forming the beams at $B_0$. It has been suggested that ion beams can form whenever there is a plasma density gradient between the source plasma region and some downstream region, into which the plasma flows [71, 101, 102]. Such ion beam formation through geometric expansion from a tube has also been seen without the application of a magnetic field and in the absence of a double layer at the source tube exit [103].

The increased strength of the ion beam when using the 85 mm source, compared to the 170 mm source, may result from the factor of 30 difference in the plasma density at the source tube exit. The formation of ion beams in low field helicon modes is not yet fully understood and current investigations [100] aim to develop the subject further. Similarly, the formation of ion beams due to geometric expansion of a plasma and the interplay with the magnetic field is a subject of current research [104], particularly in cases where no magnetic field is required [103], as this has obvious advantages in the context of a plasma thruster. Thus, future investigations of low field operation of the SSC RFPT should map the spacial potential structures to allow further understanding of the ion acceleration and beam formation mechanisms.

Through comparisons between experimentally observed behaviour of low field plasma sources found in the literature and the SSC RFPT, we have found some consistencies between them in support of the existence of a low field mode within the SSC RFPT. Trends of the plasma density as a function of the applied magnetic field were found to be qualitatively similar between
6.1. The SSC RFPT Low Field Mode

the sources, and increases in the plasma density of a factor of 6–22 over the zero applied field
density were found between all the sources reviewed here, including the SSC RFPT. Although
the relationship between ion beam formation and low field helicon modes is still the subject of
investigation, they are found to form with some consistency among low field sources and within
the SSC RFPT. However it is acknowledged here and in the literature that additional processes
may contribute towards their formation.

6.1.2.2 The Behaviour of $B_0$

The plasma density, total collected by the RPA parameters discussed in section 6.1.2.1 peaked
at certain applied magnetic field strengths, referred to in this thesis as $B_0$, which may also be
compared between sources. The low field mode literature generally refers explicitly to helicon
wave modes to explain the observed plasma behaviour [77, 8]. Since we know from the simple
bounded helicon dispersion relation Eq. 2.37, page 32 that the condition for propagation of the
wave depends on the density of the plasma and the applied magnetic field strength, we expect
that the measured values in a laboratory plasma should increase linearly with each other.

Literature investigations of low field modes do indeed find that the peak magnetic field and
the plasma density vary linearly with each other. For a vacuum immersed thruster system [77],
the ion densities produced at the peak of the low field mode ranged from $0.5–3.5 \times 10^{16}$ m$^{-3}$ and
the corresponding $B_0$ ranged from 2.4–4 mT for power inputs of 130–470 W. In the case of a
diffusion chamber type experiment [8], density peaks of $0.2–2 \times 10^{17}$ m$^{-3}$ were produced from a
$B_0$ range of 1.6–4.8 mT for RF powers of 50–400 W. In both cases, the increasing power input
was described to produce a linear increase in the plasma density and $B_0$.

In chapter 5, it was noted that the value of $B_0$ shifted to higher magnetic fields when the RF
power was increased. This trend was observed in the measurements of thrust while using both
the 170 mm source tube, shown in Figs. 5.1a and 5.3a, and the 85 mm source tube, shown in
Fig. 5.1b. The trend was also reflected in the measurements of the plasma resistance shown in
Fig. 5.2. Figure 6.1 shows the observed values of $B_0$ plotted as a function of the input power
for both source tubes used in the SSC RFPT for a power range of 200–500 W and a propellant
flow rate of 0.5 mg/s.

The values of $B_0$ were found to increase linearly with the applied field for both source tube
lengths. The plasma density at $B_0$ as a function of the RF power was also found to increase
linearly between $0.9–2 \times 10^{18}$ m$^{-3}$ for applied magnetic fields of 6–9 mT and RF powers of 200–
Fig. 6.1: Magnetic field strength $B_0$ as a function of the RF power used to sustain the plasma while using the 85 mm and 170 mm source tubes. The propellant flow rate used in both cases was 0.5 mg/s.

500 W. This behaviour matches with that observed in the literature and thus provides support for the existence of a low field helicon mode in the SSC RFPT.

While the linear trends in the values of $B_0$ and plasma density of the SSC RFPT support the existence of a low field helicon mode, it was found that both are higher than those found in the literature values [8, 77]. We can perhaps find an explanation for this by looking at another trend found within the SSC RFPT data across the two source tube sizes.

When using the 85 mm source tube, the $B_0$ field strengths were generally higher by about 4 mT across the range of RF power, when compared to the 170 mm source tube case. Comparing the densities between the two source tubes at the same point, in this case we choose the source tube exit ($z=0$ mm), we find that the density is higher by a factor of 30 in the 85 mm source tube. If the plasma is being sustained by helicon wave phenomena we would expect that higher plasma densities would require higher magnetic fields in order to satisfy the helicon relation, which is found to be the case with the SSC RFPT.

When comparing the differences in the plasma densities between the SSC RFPT and the literature, we find that the volume of the 85 mm source tube is a factor of 16 smaller than that
of Refs. [8] and [77] and that the volume of the 170 mm tube is a factor of 8 smaller. Since roughly the same RF power range is used between these studies, this leads to a corresponding increase in the input RF power density (W/m$^3$). This has the effect of increasing the plasma density, in the same way that increasing the RF input power would for a fixed source tube volume. Therefore, we should expect that values of $B_0$ and plasma density should be higher in the SSC RFPT when using the 85 mm source tube, compared to the 170 mm source tube and that both should exhibit higher plasma densities and $B_0$ values compared to the literature studies.

The preceding description of the behaviour of $B_0$ is, however, based on an incomplete understanding of the exact mechanism of low field helicon plasma heating within the SSC RFPT. A range of phenomena have been suggested to contribute towards the heating of plasma within so-called low field helicon plasma sources. These include the possibility of reflections of the helicon waves from the source tube dielectric boundaries [78], resulting in constructive interference and subsequent damping of the waves, as well as the possibility of downstream electron cyclotron resonance regions in the downstream region of the source tube [105].

We are attempting to gather evidence to support the existence of a low field helicon mode in operation within the SSC RFPT. So far we have compared the behaviours of the plasma parameters which were experimentally accessible within the capabilities of the SSC laboratory. We can perform another check based again on the simple model of bounded helicon wave dispersion, reproduced on page 29 in Fig. 2.9. This plot shows that the frequency of a helicon wave must lie between the ion cyclotron frequency, $\omega_{ci}$, and the electron cyclotron frequency, $\omega_{ce}$, frequencies within the plasma. These frequencies can be calculated from,

$$ f = \frac{|q|B}{2\pi m} $$

where $B$ is the applied magnetic field, $q$ is the fundamental charge and $m$ is the mass of the ion/electron. For an applied field of 8 mT, which is of the same order as the applied fields that are used throughout this thesis, $\omega_{ci}$ is calculated as 3.2 kHz and $\omega_{ce}$ as 0.2 GHz. Since we are using a driving frequency of $\omega = 13.56$ MHz and as we saw from page 29 in Fig. 2.9 that $\omega_{ci} \ll \omega \ll \omega_{ce}$, it is at least possible that helicon waves are present within the SSC RFPT, since our driving frequency lies between these bounds.

The experimental evidence related to low field modes in operation in the SSC RFPT has now been presented. First, it was reasoned that current free double layer formation was unlikely, due
to the broad disagreement between the behaviour of the SSC RFPT and those sources found in the literature which have been demonstrated to contain current free double layers. We saw that the current theory of magnetic field driven double layer formation requires that the ion gyroradius be smaller than the source tube radius, which was not the case at any point during this investigation. We also found that the profile of plasma density as a function of the axial distance did not contain characteristic discontinuities located at the source tube orifice, as has been found within double layer containing plasmas.

We then explored the low field helicon mode and found a general agreement between the behaviour of the SSC RFPT plasma source and low field helicon sources found in the literature. We demonstrated that the propagation of helicon waves is at least possible within the SSC RFPT by calculating the ion and electron gyro frequencies and showing that the driving frequency lies between them, thus satisfying a wave propagation condition. We showed that peaks exist in the plasma density and total collected current within a narrow range of applied magnetic fields, which are a widely observed feature of low field modes. Furthermore, we showed that the magnitude of the plasma density increase towards the peak values was consistent with that found in a number of literature studies.

Finally, we found that a linear trend exists between the value of $B_0$, the input RF power and the plasma density within the SSC RFPT, which is consistent with experimental trends found in the low field helicon mode literature. More convincingly, this trend is also in agreement with the linear trend between the plasma density and the applied magnetic field that is prescribed by the simple dispersion relation for the cylindrically bound helicon wave.

In order to concretely confirm the existence of helicon waves within the SSC RFPT source, direct measurements of the helicon wave fields are required. However, this was beyond the capabilities of the SSC plasma laboratory. Despite this, a number of features have been found within the plasma that are clearly related to those found in the literature and we find that these features can be explained by the existence of low field helicon modes.

This concludes the first half of this chapter relating to the SSC RFPT low field mode. In the next section we discuss the measured performance of the thruster while operating in this mode and how this compares to the performance of RFPT’s in the literature.
6.2 Performance Characteristics of the SSC RFPT

In this section, we shall discuss the thrust measurements made of the RFPT while operating in the low field mode. While some early studies can be found [47], the systematic and direct assessment of RFPTs operating over a range of parameters is relatively recent and was catalysed by the development of the thrust balance at the SSC. Many varied geometries of RFPT exist in the literature and few comparisons are made between the various thrusters which have been tested. Typically, the source tube dimensions, antenna length and geometry and magnetic field geometry differ between studies initially, and the propellant flow, RF power and, in some instances, the magnetic field strength are varied.

In this section, we shall adopt the approach of comparing the thrust efficiency of systems, which was defined in Eq. 2.2 on page 10. This quantity captures the essential performance of a thruster as the ratio of the power delivered to the system to the kinetic power of the exhaust products. This quantity allows useful comparisons between thrusters to be drawn as it removes differences between the thrusters and gives an overall performance metric.

Before comparing the performance of the SSC RFPT to other thrusters, it is useful to review the current understanding of thrust production within RFPTs. We will see how the thrust produced by an RFPT can depend strongly on the geometry of the source tube and how measurement of the plasma density within the source tube can provide a good prediction to first order of the thrust produced by an RFPT. This will allow further insight into the performance differences between the SSC RFPT while operating in the low field mode and those found in the literature.

6.2.1 Thrust Production in RFPTs

The thrust produced by RFPTs has been the subject of modelling and experimental efforts for both magnetised [70, 106, 107, 108] and unmagnetised plasmas [109, 75]. The interaction between the plasma and the diverging magnetic field has been shown to be very complex [107] and the development of accurate models is an area of ongoing investigation. Despite the complexity of the models and early development stage, some important advances have been made that can help us to understand the differences between the performances of RFPTs, which are compared to the SSC RFPT in the next section.

Theoretical investigations into the dynamics of collisionless [83] and collisional [109] plasmas
predicted that the maximum electron pressure within the source tube is a key parameter that
determines the thrust produced by an RFPT. In this model, it is suggested that the electron
pressure within the source tube region can be ‘converted’ into ion momentum [75] through the
ambipolar field or the electric field of a current free double layer if present [106]. In this model,
the thrust, $T$, of an RFPT is determined by the electron pressure, $p_e$, on the back wall of the
source tube [83],

$$T = q2\pi \int_0^{r_s} r p_e(r, z_0) \, dr.$$  \hspace{1cm} (6.5)

where $q$ is the fundamental charge, $r_s$ is the radius of the source tube and $z_0$ is denoted here
as the position of the back wall of the source tube. Thus, for a cylindrical source tube of cross
section $A = 2\pi r_s^2$, a radially constant electron temperature, $T_e$, and a radially averaged electron
density measured at the maximum density point within the source tube, $\langle n \rangle_{\text{max}}$ [75],

$$T = q\langle n \rangle_{\text{max}} T_e A$$ \hspace{1cm} (6.6)

This model can be understood from a particle dynamics perspective. Ions entering the sheath
at the back wall of the source tube with some momentum flux are accelerated by the sheath
voltage and gain momentum, which is lost from the thruster. As the massive ions hit the wall, the
momentum lost to the ions is exactly balanced through the collisions. Thus, no net momentum
is gained by the thruster through the ions striking the back wall, but energy is simply lost from
the system as heat in the collision events [83]. This is a source of inefficiency that occurs at
every dielectric surface within the channel.

The electrons approaching the sheath are reflected from the negatively charged wall (reflected
by the sheath) and the source tube gains a momentum that is proportional to the electron
pressure. Thus, in this model of the RFPT, the ions lost to the open end of the source tube
carry a momentum flux that is equal to the electron pressure multiplied by the back wall area,
which, for a cylindrical source tube, is just the cross sectional area, $A$. This final interpretation
of the model suggests that by increasing the area of the back wall, i.e. making $r_s = r_s(z)$, the
thrust could be enhanced. In fact, this was the motivation behind the use of a conical shaped
source tube in recent RFPT studies [70].

Finally, since this model assumes that the total neutral, ion and electron momentum is
conserved, this model remains valid even when the plasma is collisional and a significant fraction
of the thrust is produced from charge exchange neutral particles [83, 106]. This means that a
measurement of the plasma density alone could allow for a prediction of the thrust produced by an RFPT even if a mixture of fast ions and charge exchange neutrals are generating the thrust.

Several experimental studies [75, 70, 74] have provided support for this model, which is being extended further to include the effects of the magnetic field. In one study [75], thrust predictions based on this model were compared to direct thrust measurements of an RFPT for two source tube geometries of 95 mm and 175 mm length and of 32 mm radius. These geometries are very close to the SSC RFPT source tubes of 170 mm and 85 mm length and 22.5 mm radius, which makes it an interesting case for comparison which will be explored shortly.

The model was found to be in good agreement with the measured thrust [75] when using measured values of the electron temperature and radially averaged plasma density. The study also provided a perspective on the main power losses within an RFPT by examining the power balance of the thruster. We recall from earlier in the discussion that ions impacting the boundaries of the source tube represent a net loss of energy from the system, which does not produce a useful net thrust [83, 75]. Thus, an RFPT with a high aspect ratio cylindrical source tube will incur high ion wall losses along the radial boundary, thus reducing the efficiency of the plasma source, as well as reducing the RF power density in the source. Conversely, a shorter source tube of the same radius will incur a lower ion wall loss penalty and the RF power density will be increased. This will have the effect of increasing the efficiency of the plasma source and increasing the density of the plasma. Indeed, it has been suggested that a reasonable strategy for optimisation of the source is to eliminate the wall losses completely by strongly confining the ions in the source by using strong magnetic fields [75]. This strategy has also been implemented to increase the efficiency of some RFPT plasma sources [107, 110, 47].

The main points to be understood are that the source tube cross sectional area and electron pressure determine the thrust levels and that the total internal source tube area determines the wasteful ion loss, and thus in part the efficiency of the thruster. Thus, the simple relation given in Eq. 6.6 tells us that the RFPT plasma source efficiency can be optimised through modification of the source tube geometry [70] and through control of the wall losses using a magnetic field [107, 110, 47].

We began the review of this particular model by pointing out that this is a simplified picture of the RFPT that does not include the effects of an applied magnetic field. As we saw in section 6.1.2, the behaviour of the SSC RFPT plasma source is not straight forward and it is not clear that even the more complex RFPT models are equipped to deal with the presence
of low field helicon waves. Despite this, the discussion does have an important bearing on the interpretation of the performance trends observed in the SSC RFPT, which uses an applied magnetic field and was investigated for two source tube lengths.

When no magnetic field was applied to the SSC RFPT, the plasma density (measured at \( z = 20 \text{ mm} \)) was found to increase from \( 1.3 \times 10^{15} \text{ m}^{-3} \) for the 170 mm source length, to \( 2.9 \times 10^{16} \text{ m}^{-3} \) when using the 85 mm tube, for an RF power of 200 W and a 0.5 mg/s propellant flow. This also corresponded to an increase in the thrust from 0.13 mN to 0.3 mN. When this model was compared to direct thrust measurements using source tubes of 95 mm and 175 length and 32mm radius [75] a similar increase in the thrust of about a factor of 2 was found when switching between the two source tube lengths.

In the context of the previous discussion, this can be readily understood to be the result of reduced ion losses to the wall, leading to an increase in the power efficiency of the source and an increase in plasma density. While these are the downstream density values, the maximum density clearly will have increased within the source tube, but will have decreased to the quoted downstream values, as seen in Fig. 5.5b.

Similarly, the plasma density at the peak of the low field mode \( B_0 \) when using the 170 mm source for the same RF power and propellant flow rate as above, increases from \( 7.8 \times 10^{15} \text{ m}^{-3} \) to \( 1.27 \times 10^{17} \text{ m}^{-3} \) when using the 85 mm source tube. This corresponds to a thrust increase from 0.3 mN to 1.22 mN.

As in the zero applied field case, the efficiency of the plasma source has increased through reduction of the ion loss surface, leading to an increase in the plasma density and hence the thrust produced though the electron pressure. Furthermore, the application of the magnetic field has provided access to the low field mode and enhanced the density further so we see that the thrust has now increased by a factor of 4 in this case. It is not clear at this stage what determines the enhanced increase in the thrust when the magnetic field is applied, in excess of the effect due to the increased plasma density. However, more advanced electron pressure based RFPT models suggest that the thrust force due to the electron pressure acting on the diverging field contributes towards the thrust levels [106].

The model described here requires that the radially averaged plasma density at the maximum density point be used for calculation of the thrust, along with a measurement of the electron temperature. Since these models were still in development at the time that experiments for this thesis were being completed, the appropriate measurements were not made and therefore more
6.2. Performance Characteristics of the SSC RFPT

definitive calculations using the model cannot be made. This will however form the basis of future studies with the SSC RFPT.

The axial plasma density of the SSC RFPT was however measured while using the 85 mm source tube, shown in Fig.5.5b. Since the RF power levels and source tube geometry are very similar between the present study and that of Lafleur[75], it is interesting to use the radial density average factor of \( \sim 0.58 \) that was experimentally determined for the 95 mm source tube and an estimate of the electron temperature of 5.5 eV that we have used previously for the ion density calculations. Using these values and the peak plasma density of \( 1 \times 10^{18} \text{m}^{-3} \), we calculate a thrust of 0.8 mN at the peak of the low field mode for the 85 mm source tube, compared to the directly measured value of 1.2 mN. This is an interesting calculation to make as it shows that, despite the use of an electron temperature estimate and the presence of a magnetic field, the model gives at least the correct order of magnitude of thrust. However, we cannot draw firm conclusions from this due to the fact that we have used estimates for the electron temperature, but instead we acknowledge that this picture of the RFPT operation captures at least part of the thrust production process and that the electron pressure plays a large role.

In summary, we have presented and discussed the final result of a current model of RFPT thrust production. We saw how the model gives insight into how the geometry of the source tube can influence both the thrust produced by the thruster, through the electron pressure acting on the back wall of the tube, and how the length, in the case of the commonly used cylindrical source tube, can have an impact on the thrust by increasing the efficiency of the plasma source through the control of ion losses to the walls. Finally, we showed how the SSC RFPT conforms to this model by highlighting that the plasma density and the thrust both increase as predicted when the source tube length is reduced for a given power and propellant flow. We then demonstrated the use of the model by using an estimate of the electron temperature for the SSC RFPT, the measured maximum plasma density and a radial averaging factor from a study that used a closely matched source tube. This was shown to under-predict the thrust by a factor of 1.5, illustrating both the limitations of this model, which does not take into account the effect of the magnetic field, and the need for careful measurements of the internal plasma density structure and electron temperature within RFPT source tubes if the model is to be used to make concrete predictions of RFPT performance.

This concludes our review of the RFPT thrust production model and in the section following we shall make some broad comparisons between the performance of the SSC RFPT measured
here and RFPT found in the literature.

6.2.2 RFPT Performance Comparisons

In this section, we establish how the low field mode of SSC RFPT operation compares to RFPTs which can be found in the literature. We show that in particular, the low field mode can only provide a performance advantage in terms of thrust efficiency and specific impulse if the geometry of the source tube has already been optimised to reduce power losses to its walls. We demonstrate that the low field mode can not provide a performance advantage over systems that have a well optimised source tube geometry and that make use of strong magnetic fields to access more efficient power deposition regimes than are possible through use of the low field mode. Despite these negative results, we also find that when the low field mode is used in conjunction with a source tube that reduces ion losses, the specific impulse and thrust efficiency are increased favourably over RFPTs that use a similar tube geometry but do not use a magnetic field. Thus, this provides a compromise solution between performance and the strength of the applied magnetic field.

The development of low power RFPTs has recently become popular given the range of possible advantages they could provide, as described at the beginning of this chapter. The introduction of the pendulum balance to measure the thrust of RFPTs directly was an important step towards raising its technology readiness level and thus towards a maiden flight opportunity. Since the present investigation makes use of direct thrust measurements using a pendulum thrust balance, the comparisons are restricted to those studies which have made direct thrust measurements.

The majority of the experiments which have been conducted in this thesis were carried out in order to ascertain the nature of the low field mode, which was argued in the previous section to be a low field helicon mode. In this section however, we wish to establish how well the low field mode can optimise an RFPT and thus we make use of the higher power thrust measurements made while operating at the peak of the low field mode, $B_0$. These measurements were made specifically to explore the upper limits of the low field mode performance in the SSC RFPT, in an attempt to reveal performance advantages.

Similar attempts to optimise the performance of RFPTs have led to a large number of thruster geometries and operating points being investigated. Table 6.1 contains information extracted from the literature pertaining to studies which have made direct thrust measurements of an RFPT. It contains measured system parameters; the thrust level, plasma density and electron
temperature where measured, as well as the value of free parameters, such as the applied magnetic field strength and propellant flow. It also contains a number of practical system parameters: the radius and length of the source tube and the type of antenna used.

Recall from chapter 1 page 2 and chapter 2 page 10, that the thrust efficiency, $\eta$, describes how efficiently an EP thruster converts the input power into kinetic power of the exhaust products and that the specific impulse, $I_{sp}$, gives a measure of how well the thruster is utilising its propellant. The use of $\eta$ and $I_{sp}$ allow EP systems to be compared to each other. These values were calculated for each system using the propellant flow rates, measured thrust and input powers quoted and were entered into the table along with the RF power density, as this can also give some insight into the performance differences found between the thrusters.

A nominal RF power level of 500 W was set to facilitate comparisons. Correspondingly, the SSC RFPT appears three times in the table representing the configurations for which thrust data has been obtained at a power of 500 W. This table will be referred to throughout the remainder of this chapter and comparisons will be made through references to a numerical label, shown preceding the name of the author in the first column of Tab 6.1. The label will be referred to in brackets, for example “...system (1)” refers to the RFPT system configuration found at entry (1) in the table 6.1. A discussion of all the entries is not required for us to reach our conclusions, however, they have been included in the table for completeness, as few tables making such broad comparisons can be found in the literature and this may serve as a useful reference for future studies.

Since we are interested in finding ways to increase the performance of RFPTs, we start by comparing the performance extremes. Entry (9) contains the highest performing system, investigated by Takahashi [112] where $\eta$ was 2.8% and the $I_{sp}$ was 640 s. The lowest performing system is entry (1), which is the SSC RFPT, where $\eta$ was 0.02 % and the $I_{sp}$ was 60 s.

The differences in performance between these two configurations can be attributed to both the differences in the source tube geometries and the use of an applied magnetic field. We saw in the previous section that the internal source tube wall area and the cross sectional area are both important parameters in determining the overall performance of an RFPT.

Thus, for a given internal source area, the aspect ratio will determine the efficiency of the source. These two systems have essentially the same ion loss surface area, however, the SSC RFPT has a high aspect ratio in this configuration of 3.8, compared to Takahashi’s of 1.9. This will increase the ratio of the ion loss surface to the surface available for electron pressure to
Tab. 6.1: Table summarising the direct measurements made of a range of RFPTs. Common performance metrics have been calculated for a nominal 500 W of RF power. The studies have been ordered according to the strength of the applied magnetic field. The use of permanent magnets is indicated next to the quoted field strengths as (PM) for permanent magnetic and (EM) for electromagnets.

<table>
<thead>
<tr>
<th>Antenna</th>
<th>Applied RF Density (W/m²)</th>
<th>Flow (mg/s)</th>
<th>Isp (s)</th>
<th>Thrust (mN)</th>
<th>R/L (mm)</th>
</tr>
</thead>
<tbody>
<tr>
<td>(1) SSC RFPT</td>
<td>0.7</td>
<td>0.1</td>
<td>182</td>
<td>0.5 (Ar)</td>
<td>551</td>
</tr>
<tr>
<td>(2) Lafleur [75]</td>
<td>1.5</td>
<td>0.1</td>
<td>255</td>
<td>0.9 (Ar)</td>
<td>87</td>
</tr>
<tr>
<td>(3) Lafleur [75]</td>
<td>2.6</td>
<td>1.4</td>
<td>555</td>
<td>0.5 (Ar)</td>
<td>3</td>
</tr>
<tr>
<td>(4) Pottinger [73]</td>
<td>4.5</td>
<td>2.8</td>
<td>640</td>
<td>0.72 (Ar)</td>
<td>7</td>
</tr>
<tr>
<td>(5) SSC RFPT</td>
<td>11</td>
<td>0.7</td>
<td>142</td>
<td>0.5 (Ar)</td>
<td>4</td>
</tr>
<tr>
<td>(6) SSC RFPT</td>
<td>12</td>
<td>0.1</td>
<td>110</td>
<td>1.5 (Ar)</td>
<td>0.9</td>
</tr>
<tr>
<td>(7) Takahashi [106]</td>
<td>18</td>
<td>2.2</td>
<td>551</td>
<td>0.72 (Ar)</td>
<td>0.3</td>
</tr>
<tr>
<td>(8) Takahashi [74]</td>
<td>20</td>
<td>0.4</td>
<td>255</td>
<td>0.6 (Ar)</td>
<td>0.9</td>
</tr>
<tr>
<td>(9) Takahashi [112]</td>
<td>30</td>
<td>1.3</td>
<td>150</td>
<td>1.5 (Ar)</td>
<td>0.1</td>
</tr>
<tr>
<td>(10) Williams [113]</td>
<td>45</td>
<td>0.4</td>
<td>150</td>
<td>1.5 (Ar)</td>
<td>0.1</td>
</tr>
</tbody>
</table>

Note: DS: double saddle, DL: double loop. The length of the antenna is given in millimetres for double saddle antennas, double loop antennas have a ~5 mm clearance about the source tube and can be approximated as the radius of the tube.
create a thrusting force, which is a factor in producing the comparably low source efficiency of the SSC RFPT. The importance of reducing this ratio can be appreciated more clearly through comparing entries (2) and (3), which show that for an unmagnetised RFPT, halving the ion loss surface while maintaining the same back wall area increases the plasma density by a factor of 2, the thrust produced by a factor of 3 and the thrust efficiency by a factor of 5.

In addition to having a lower aspect ratio than the SSC RFTP of entry (1), Takahashi’s system (9) uses a 30 mT magnetic field. To understand the effect of the applied field, we can compare the system with the SSC RFPT of entry (6), which has a similar source aspect ratio of 1.8 and is operating at the peak of the low field helicon mode. This allows us to understand better the effect of the performance enhancements gained using the applied field, in excess of the source tube geometry optimisation. In fact, the comparison demonstrates that the low field mode does not provide a performance advantage over systems which use a strong magnetic field and a source tube geometry which has been designed to reduce power losses at the wall.

The highest performing SSC RFPT arrangement is shown in entry (6), where the thrust was measured as 2.6 mN at a thrust efficiency of 1.4% and an $I_{sp}$ of 555 s. Comparing this to Takahashi’s system [112] at entry (9), we see that by reducing the applied magnetic field by 60%, the specific impulse reduces by 13%, the efficiency reduces by 50% and the thrust reduces by 40% . Since the power density in the SSC RFPT is actually higher at 3 W/cm$^3$, compared to Takahashi’s 1.3 W/cm$^3$, yet the density of the plasma in the SSC RFPT is 80% lower by comparison, it is clear that the use of the strong 30 mT magnetic field allows for more efficient power deposition into the plasma than is possible with the low field helicon mode.

The use of a low aspect ratio source tube and application of a strong magnetic field in system (9) resulted in a high thrust efficiency that can not be matched by a system operating in the low field mode using a similar source tube aspect ratio. However, the purpose of this discussion is to ascertain whether a compromise solution can be found between the strength of the applied magnetic field and the system performance through use of the low field helicon mode.

The use of strong magnetic fields is not a universally advantageous strategy to enhance RFPT performance if the source tube has not been optimised to reduce wall power losses. This assertion is illustrated well through comparison of the SSC RFPT of entry (6) with the system at entry (10) of Williams [113]. In system (10), the applied magnetic field of 45 mT was higher than in Takahashi’s (9), however, the thrust efficiency was 0.4% and the $I_{sp}$ was 150 s, both of which much reduced compared to the SSC RFPT of entry (6).
The thrust levels that Williams measured are comparable to the SSC RFPT (6) at \(\sim 2.5 \text{ mN}\), as are source tube aspect ratios at \(\sim 1.9\). However, the internal ion loss area is a factor of 10 larger than that of the SSC RFPT, thus, power lost to the walls through ion losses was a factor of 10 higher. This effect is compounded by the large volume of the source tube, which reduces the RF power density by a factor of 30 compared to the SSC RFPT. Since the magnetic field used in the SSC RFPT is much weaker at 10 mT, compared to 45 mT in Williams’ system, we conclude that the use of the low field mode, together with a source tube that increases the power density and reduces the wall losses, provides an advantage over a system with a higher applied magnetic field and an unhelpfully large ion loss surface. This is true both in terms of the performance increases it provides and the reduced magnetic fields required for operation.

The assertion that the application of a weak magnetic field to an RFPT is only advantageous if the source tube has already been optimised to reduce wall losses can be reinforced through comparing the SSC RFPT of entry (6) to the system at entry (4) of Pottinger \[73\], which also used a weak 10 mT applied field.

Firstly, however, we note that Pottinger used a source tube with the same aspect ratio as Williams, but that the thrust produced by Pottinger’s system was 30% lower and that the thrust efficiency was reduced by 50%. This is despite the fact that the power density is the same in both cases. Thus, the use of a weak 10 mT magnetic field by Pottinger did not provide performance advantages over the high field case of Williams. This was largely due to the fact that the applied field was insufficiently strong to allow access to more efficient power coupling regimes, such as a high field helicon wave mode, and therefore the inefficiency resulting from the large wall losses could not be balanced. Thus, the thrust efficiency was low.

Now comparing the SSC RFPT of entry (6) to Pottingers’ system, which both use a relatively weak applied magnetic field, we find that the SSC RFPT has an increased thrust of 45% and a higher efficiency of 1.4%, compared to 0.2%. This is because the source tube was optimised to reduce the wall losses in the first instance, allowing further performance enhancements to be made through use of the low field helicon mode. Since the applied magnetic field is the same, the conclusion again is that optimisation of the source tube geometry is an important initial design consideration towards enhancing the efficiency of the thruster before low field modes are employed to increase the efficiency further.

These last few comparisons have illustrated that the use of magnetic fields \(> 10 \text{ mT}\) can allow the efficiency of the source to be enhanced over that which can be provided by a low field mode,
as we saw in the comparison between the SSC RFPT and Takahashi’s system [112]. However, in order to reduce the strength of the applied magnetic field while maintaining a degree of efficient operation, the optimisation of the source tube geometry to reduce the ion wall losses is of critical importance to establish a baseline of efficient source operation. This was demonstrated through comparisons of the thrusters of Pottinger [73] and Williams [113] with the highest performing configuration of the SSC RFPT measured for this thesis.

The low field mode can have a favourable impact on the thrust efficiency and specific impulse of a system that has a power-efficient source tube design while using a weak magnetic field. We saw earlier in our discussion that Lafleur [75] found an increase in efficiency, specific impulse and thrust by simply halving the length of the source tube, shown by comparing table entries (2) and (3). The SSC RFPT of entry (6) uses a very similar source tube aspect ratio as in (3), and also produces a similar thrust level of 2.6–3 mN. This can be understood in the context of the model discussed earlier, since the plasma densities measured in the source of each RFPT are very similar. However, the efficiency and specific impulse of the SSC RFPT are both higher by a factor of 3. Since the plasma density is the same across both thrusters, yet the power density is increased by a factor of 2, it seems that the use of the low field helicon mode has facilitated an increase in the efficiency and the specific impulse of the thruster, over that gained by only adjusting the source tube geometry.

6.3 Summary

In this chapter, we explored the experimental measurements made in this thesis. We first pursued an understanding of the nature of the low field mode operating in the SSC RFPT by examining two magnetic field driven plasma phenomena; magnetic field driven current free double layer formation, and low field helicon wave modes. The prominent models proposed to describe each of these modes were presented and used to understand the observed behaviour of the SSC RFPT plasma source.

We then discussed the current model proposed to describe thrust production in RFPTs. This model was used to contextualise the thrust measurements made of the SSC RFPT and was further used to understand in some cases why the performance differs between the SSC RFPT and those RFPTs which can be found in the literature and have been characterised using direct thrust measurements. In the next chapter, we will draw together what has been learnt from this
discussion and state the contributions that have been made as a result of the work conducted for this thesis.
Chapter 7

Conclusions

An RFPT was constructed at the SSC to explore the possibility of using a low magnetic field mode to improve the performance while reducing the magnetic field required for efficient operation. A pendulum type thrust balance was constructed to allow measurement of the thrust produced and a range of plasma probes were used to gather measurements of the behaviour of the plasma source.

A low field mode was found to exists in the plasma source. This mode was found to produce peaks in the plasma density, total current collected by a downstream RPA, plasma resistance and the thrust of the RFPT as a function of the applied magnetic field. Two low field modes were considered as explanations of the observed behaviour; the formation of a current free double layer and a low field helicon mode. A current free double layer was determined not to be present in the SSC RFPT since the leading theory of magnetic field driven double layer formation says that the ion gyro radius must be smaller than the source tube radius, which was shown not to be the case in these experiments. Furthermore the pattern of ion beam formation as a function of the magnetic field was found to be inconsistent with experimental accounts of double layer formation found in the literature. Finally the profile of plasma density as a function of the axial thruster location did not contain characteristic discontinuities located at the source tube orifice, as has been found within double layer containing plasmas.

The low field mode in the SSC RFPT is proposed to be the result of the presence of low field helicon waves within the plasma source. This is possible since the driving RF frequency lies between the characteristic gyro frequencies of the ions and electrons in the SSC RFPT plasma and therefore a helicon wave could, in principle, propagate. This is likely, as the behaviour of the
low field helicon mode in the SSC RFPT was found to be consistent with low field helicon sources identified in the literature. This includes the observation of peaks in the plasma density and total collected current within a narrow range of applied magnetic fields, consistent in magnitude and character with widely reported peak formation in low field helicon modes. A linear trend also exists between the value of $B_0$, the input RF power and the plasma density within the SSC RFPT, consistent with experimental trends found in the low field helicon mode literature. This linear trend between the plasma density and the applied magnetic field was found to be consistent with the simple dispersion relation for the cylindrically bound helicon wave.

Through a review and comparison of studies in which RFPT thrust has been directly measured with the SSC RFPT it was found that the low field mode is unable to match the performance of RFPTs that leverage high magnetic field driven power deposition modes in conjunction with a well optimised source tube. The low field mode can however provided increases in the thrust efficiency and specific impulse over a zero magnetic field RFPT system if the source tube has already been optimised to reduce the power lost from the system by ion collisions with the source tube boundaries.

### 7.1 Summary of Contributions

New direct thrust measurements are provided of the performance of an RFPT while operating in a low magnetic field mode. New data is also provided of the performance of the thruster as a function of the length of the plasma source tube while operating in a low magnetic field mode, which demonstrates how the geometry may be modified while operating in this mode to increase the performance of an RFPT. In summary, the following contributions are offered:

- The development of a pendulum type thrust balance which is insensitive to plasma noise and capable of measuring the thrust produced by an RFPT with a precision of 0.1 mN.
- The first direct thrust measurements made of an RFPT while operating in a low magnetic field mode.
- The first observations and characterisation of new low magnetic field mode behaviour as a result of using a short 85 mm length source tube and 130 mm double saddle antenna that exhibits sudden drops in the plasma parameters.
7.2 Recommendations for Future work

- The first direct thrust measurements of low field mode operation of an RFPT using an 85 mm source tube that demonstrates improved thrust efficiency and specific impulse over non magnetised RFPTs which use a similar source tube geometry.

- The performance characteristics presented in this study bear many similarities to the low field helicon mode. Direct plasma wave measurements could be made using B-dot probes, for example in order to discern whether such waves are indeed present. This may also help to explain the abrupt changes in the behaviour of the plasma in the short source tube case.

- Experiments using the short source tube showed evidence of abrupt changes in the performance of the thruster as a function of the imposed field for RF powers up to 500 W, which corresponded to a $B_0$ field of 12.5 mT. The imposed field may be increased considerably before one might be motivated to switch to permanent magnets, which would be undesirable from a spacecraft engineering perspective. Therefore, an investigation into higher still forward powers, while tracking this particular low field mode, would be interesting as improved performances could be gained.

- In chapter 3, the effective back flow of gas into the source tube was calculated. To the best of the author’s knowledge, there are no comprehensive studies into the effects of the chamber back pressure on the performance of RFPTs. It is telling to note that in the case of technologies which have been used in space, such as the Hall effect thruster, the effect of gas ingestion is well characterised. The effect of gas ingestion on the thruster operating should be considered as part of the push towards a first flight opportunity.

- The source tube geometries used here were based on those found in the literature in order to facilitate comparison, although with an inclination towards a compact size to better suit the prospect of application on board a satellite. While the use of a 85 mm source in the low field mode has revealed good performance increases thus fulfilling one objective of this thesis, a greater range of source tube lengths should be investigated under identical magnetic field and propellant flow conditions as those found in the literature, to understand the limits at which further performance gains can be made while using the low field mode.

- The spacial potential structures of the SSC RFPT while operating in the Low field mode
operation should be mapped to allow further understanding of the ion acceleration and beam formation mechanisms as this is at present still poorly understood.

- Estimates were used here for the electron temperature and the thermal ion temperature when computing several parameters including the gyro radius of thermal ions and the plasma densities while using a Langmuir probe in ion saturation mode. Future work should include measurements of the electron temperature and thermal ion temperatures.

- The RPA was used in this investigation to measure the relative change in the ion current downstream of the thruster. In future studies, measurements of the absolute current taken downstream of the thruster could be used to corroborate the thrust measurements.

- The scope of this experimental investigation was limited due to the limitations in expertise and the lack of many pre-existing facilities. The importance of losses within the match box was not considered at the time the experiments were conducted due to lack of experience with RF systems. Consequently the forward RF power has been quoted throughout this thesis. In order to obtain increased confidence in the measured thruster efficiency, the matchbox losses should be quantified. At present the calculations under predict efficiency due to the fact that less power is being delivered to the thruster than the quoted forward power. Similarly, investigations aimed at diagnosing the nature of the low field mode, particularly for low volume source tubes will require measurements of the actual power delivered to the plasma.

- The SSC lab did not possess the hardware capability to measure directly the wave fields of the low field helicon mode. Measurements of these wave fields will allow an understanding of the processes driving the observed behaviour in the SSC RFPT plasma source which will be a theme of future work.

- The thrust balance was calibrated by using a mass tethered to the moving plate of the balance which was horizontally displaced to apply known calibration forces. While this was found to be adequate for the purposes of this thesis, future investigations should aim to independently verify the accuracy of the thrust balance through the use of a well characterised thruster or load cell. Further, experimental uncertainties that were not quantified in this study such as the thermal effects on the thrust balance should be quantitatively investigated.
Datasheet F-201CV
Mass Flow Controller for Gases

> Introduction
Bronkhorst High-Tech model F-201CV Mass Flow Controllers (MFCs) are suited for precise control of virtually all conventional process gases. The MFC consists of a thermal mass flow sensor, a precise control valve and a microprocessor based PID controller with signal and fieldbus conversion. As a function of a setpoint value, the flow controller swiftly adjusts the desired flow rate. The mass flow, expressed in normal litres or millilitres per minute or per hour, is provided as analog signal or digitally via RS232 or fieldbus. The flow range, wetted materials and orifice size for the control valve are determined depending of the type of gas and the process conditions of the application.

Although all specifications in this datasheet are believed to be accurate, the right is reserved to make changes without notice or obligation.

> Technical specifications

**Measurement / control system**
- **Accuracy (incl. linearity)**: ± 0,5% Rd plus ± 0,1% FS (Based on actual calibration)
- **Turndown**: 1 : 50 (in digital mode up to 1:187,5)
- **Multiple fluid capability**: • storage of max. 8 calibration curves
  • optional Multi Gas / Multi Range functionality up to 10 bar
- **Repeatability**: < ± 0,2% Rd
- **Settling time (controller)**: 1…2 seconds; option: down to 500 msec
- **Control stability**: ≤ ± 0,1% FS (typical for 1 l/min N₂)
- **Max. Kv-value**: 6,6 x 10⁻²
- **Temperature range**: -10…+70°C
- **Temperature sensitivity**: zero: < ± 0,05% FS/°C;
  span: < ± 0,05% Rd/°C
- **Leak integrity (outboard)**: < 2 x 10⁻⁹ mbar l/s He
- **Attitude sensitivity**: max. error at 90° off horizontal 0,2% FS
  at 1 bar, typical N₂
- **Warm-up time**: 30 min. for optimum accuracy
  2 min. for accuracy ± 2% FS

**Mechanical parts**
- **Material (wetted parts)**: stainless steel 316L or comparable
- **Pressure rating**: 64 bar abs
- **Process connections**: compression type or face seal male
- **Seals**: standard: Viton; options: EPDM, Kalrez
- **Ingress protection (housing)**: IP40

**Electrical properties**
- **Power supply**: +15…24 Vdc ±10%
- **Power consumption**: at voltage I/O at current I/O
  (based on N/C valve) 15 V 290 mA 320 mA
  24 V 200 mA 215 mA
- **Extra for fieldbus**: PROFIBUS DP: add 53 mA (15 V supply) or 30 mA (24 V supply)
  (if applicable) EtherCAT™: add 66 mA (15 V supply) or 41 mA (24 V supply)
- **Analog output**: 0…5 (10) Vdc, min. load impedance > 2 kΩ;
  0 (4)…20 mA (sourcing), max. load impedance < 250 Ω
- **Analog setpoint**: 0…5 (10) Vdc, min. load impedance > 100 kΩ;
  0 (4)…20 mA, load impedance ~250 Ω
- **Digital communication**: standard RS232; options: PROFIBUS DP, DeviceNet™,
  EtherCAT™, Modbus RTU/ASCII, FLOW-BUS

> Ranges (based on Air)

<table>
<thead>
<tr>
<th>Model</th>
<th>minimum</th>
<th>nominal</th>
<th>maximum</th>
</tr>
</thead>
<tbody>
<tr>
<td>F-201CV-020</td>
<td>0,16…8 ml/min</td>
<td>0,16…20 ml/min</td>
<td>0,16…30 ml/min</td>
</tr>
<tr>
<td>F-201CV-050</td>
<td>0,4…20 ml/min</td>
<td>0,4…50 ml/min</td>
<td>0,4…75 ml/min</td>
</tr>
<tr>
<td>F-201CV-100</td>
<td>0,8…40 ml/min</td>
<td>0,8…100 ml/min</td>
<td>0,8…150 ml/min</td>
</tr>
<tr>
<td>F-201CV-200</td>
<td>1,6…80 ml/min</td>
<td>1,6…200 ml/min</td>
<td>1,6…300 ml/min</td>
</tr>
<tr>
<td>F-201CV-500</td>
<td>4…200 ml/min</td>
<td>4…500 ml/min</td>
<td>4…750 ml/min</td>
</tr>
<tr>
<td>F-201CV-1K</td>
<td>8…400 ml/min</td>
<td>8…1000 ml/min</td>
<td>8…1500 ml/min</td>
</tr>
<tr>
<td>F-201CV-3K</td>
<td>16…800 ml/min</td>
<td>16…2000 ml/min</td>
<td>16…3000 ml/min</td>
</tr>
<tr>
<td>F-201CV-5K</td>
<td>32…1000 ml/min</td>
<td>32…3000 ml/min</td>
<td>32…4500 ml/min</td>
</tr>
<tr>
<td>F-201CV-10K</td>
<td>64…2000 ml/min</td>
<td>64…6000 ml/min</td>
<td>64…8000 ml/min</td>
</tr>
<tr>
<td>F-201CV-20K</td>
<td>128…2000 ml/min</td>
<td>128…4000 ml/min</td>
<td>128…6000 ml/min</td>
</tr>
</tbody>
</table>

Intermediate ranges are available.
## Model number identification

<table>
<thead>
<tr>
<th>F</th>
<th>N</th>
<th>N</th>
<th>A</th>
<th>A</th>
<th>A</th>
<th>N</th>
<th>A</th>
</tr>
</thead>
<tbody>
<tr>
<td>Base 2 Controller</td>
<td></td>
<td></td>
<td></td>
<td></td>
<td></td>
<td></td>
<td></td>
</tr>
<tr>
<td>Pressure rating 0 64 bar</td>
<td></td>
<td></td>
<td></td>
<td></td>
<td></td>
<td></td>
<td></td>
</tr>
<tr>
<td>Nominal range Factory selected</td>
<td></td>
<td></td>
<td></td>
<td></td>
<td></td>
<td></td>
<td></td>
</tr>
<tr>
<td>Communication (I/O)</td>
<td></td>
<td></td>
<td></td>
<td></td>
<td></td>
<td></td>
<td></td>
</tr>
<tr>
<td>A RS232 + analog (n/c control)</td>
<td></td>
<td></td>
<td></td>
<td></td>
<td></td>
<td></td>
<td></td>
</tr>
<tr>
<td>B RS232 + analog (n/o control)</td>
<td></td>
<td></td>
<td></td>
<td></td>
<td></td>
<td></td>
<td></td>
</tr>
<tr>
<td>D RS232 + DeviceNet (n/c)</td>
<td></td>
<td></td>
<td></td>
<td></td>
<td></td>
<td></td>
<td></td>
</tr>
<tr>
<td>E RS232 + DeviceNet (n/o)</td>
<td></td>
<td></td>
<td></td>
<td></td>
<td></td>
<td></td>
<td></td>
</tr>
<tr>
<td>M RS232 + Modbus (n/c)</td>
<td></td>
<td></td>
<td></td>
<td></td>
<td></td>
<td></td>
<td></td>
</tr>
<tr>
<td>N RS232 + Modbus (n/o)</td>
<td></td>
<td></td>
<td></td>
<td></td>
<td></td>
<td></td>
<td></td>
</tr>
<tr>
<td>P RS232 + PROFIBUS DP (n/c)</td>
<td></td>
<td></td>
<td></td>
<td></td>
<td></td>
<td></td>
<td></td>
</tr>
<tr>
<td>Q RS232 + PROFIBUS DP (n/o)</td>
<td></td>
<td></td>
<td></td>
<td></td>
<td></td>
<td></td>
<td></td>
</tr>
<tr>
<td>R RS232 + FLOW-BUS (n/c)</td>
<td></td>
<td></td>
<td></td>
<td></td>
<td></td>
<td></td>
<td></td>
</tr>
<tr>
<td>S RS232 + FLOW-BUS (n/o)</td>
<td></td>
<td></td>
<td></td>
<td></td>
<td></td>
<td></td>
<td></td>
</tr>
<tr>
<td>Internal seals</td>
<td></td>
<td></td>
<td></td>
<td></td>
<td></td>
<td></td>
<td></td>
</tr>
<tr>
<td>V Viton (factory standard)</td>
<td></td>
<td></td>
<td></td>
<td></td>
<td></td>
<td></td>
<td></td>
</tr>
<tr>
<td>E EPDM</td>
<td></td>
<td></td>
<td></td>
<td></td>
<td></td>
<td></td>
<td></td>
</tr>
<tr>
<td>K Kalrez (FKM)</td>
<td></td>
<td></td>
<td></td>
<td></td>
<td></td>
<td></td>
<td></td>
</tr>
<tr>
<td>Connections (in/out)</td>
<td></td>
<td></td>
<td></td>
<td></td>
<td></td>
<td></td>
<td></td>
</tr>
<tr>
<td>1 1/8” OD compression type</td>
<td></td>
<td></td>
<td></td>
<td></td>
<td></td>
<td></td>
<td></td>
</tr>
<tr>
<td>2 ¼” OD compression type</td>
<td></td>
<td></td>
<td></td>
<td></td>
<td></td>
<td></td>
<td></td>
</tr>
<tr>
<td>3 6 mm OD compression type</td>
<td></td>
<td></td>
<td></td>
<td></td>
<td></td>
<td></td>
<td></td>
</tr>
<tr>
<td>4 ½” Face seal male</td>
<td></td>
<td></td>
<td></td>
<td></td>
<td></td>
<td></td>
<td></td>
</tr>
<tr>
<td>8 other</td>
<td></td>
<td></td>
<td></td>
<td></td>
<td></td>
<td></td>
<td></td>
</tr>
<tr>
<td>9 none</td>
<td></td>
<td></td>
<td></td>
<td></td>
<td></td>
<td></td>
<td></td>
</tr>
</tbody>
</table>

## Thermal mass flow measuring principle

The heart of the thermal mass flow meter/controller is the sensor, that consists of a stainless steel capillary tube with resistance thermometer elements. A part of the gas flows through this bypass sensor, and is warmed up using resistance heating elements. Consequently the measured temperatures $T_1$ and $T_2$ drift apart. The temperature difference is directly proportional to mass flow through the sensor. In the main channel Bronkhorst High-Tech applies a patented laminar flow element consisting of a stack of stainless steel discs with precision-etched flow channels. Thanks to the perfect flow-split the sensor output is proportional to the total mass flow rate.

$$\Delta T = k \cdot C_p \cdot \Delta m$$

$\Delta T = T_2 - T_1$ in Kelvin

$C_p =$ specific heat

$m =$ mass flow

## State of the art digital design

Today's EL-FLOW® series are equipped with a digital pc-board, offering high accuracy, excellent temperature stability and fast response (settling times $t_{98}$ down to 500 msec). The basic digital pc-board contains all of the general functions needed for measurement and control. In addition to the standard RS232 output the instruments also offer analog I/O. Furthermore, an integrated interface board provides DeviceNet™, PROFIBUS DP, Modbus RTU/ASCII or FLOW-BUS protocols.

### Functional scheme of the thermal mass flow sensor

![Functional scheme of the thermal mass flow sensor](image)

### Functional scheme of the digital PC-board

![Functional scheme of the digital PC-board](image)
Hook-up diagram for analog or RS232 communication

Hook-up diagrams for fieldbus communication

For the available fieldbus options we refer to the various hook-up diagrams as indicated below. If you are viewing this datasheet in digital format, you may use the hyperlink to each of the drawings. Otherwise please visit the download section on www.bronkhorst.com or contact our local representatives.
> Dimensions (mm) and weight (kg)

- [Image of dimensions and weight]

> Options and accessories

- Multi-Gas / Multi-Range option, with free configuration software.
- Free software support for operation, monitoring, optimizing or to interface between digital instruments and windows software.
- IN-LINE filters for protection against particulates
- BRIGHT compact local Readout/Control modules
- E-8000 Power Supply/Readout systems
- Interconnecting cables for power and analog/digital communication
- PIPS Plug-in Power Supply

> Alternatives

- IQ+FLOW, world’s smallest Mass Flow Controller
- MFC with integrated 24V shut-off valve
- LOW-ΔP-FLOW series MFC for low pressure drop applications or corrosive gas service
- Metal sealed MFC for Semiconductor or other high purity applications
- Mass Flow Controller for standardised modular platform systems (top-mount version)
- Pre-assembled multi-channel solutions: series FLOW-SMS

---

**Dimension table adapters (RS-type)**

<table>
<thead>
<tr>
<th>Compression type</th>
<th>OD</th>
<th>Size A</th>
</tr>
</thead>
<tbody>
<tr>
<td>adapter 3 mm</td>
<td>26.1</td>
<td></td>
</tr>
<tr>
<td>adapter 6 mm</td>
<td>28.4</td>
<td></td>
</tr>
<tr>
<td>adapter 8 mm</td>
<td>29.4</td>
<td></td>
</tr>
<tr>
<td>adapter 10 mm</td>
<td>30.2</td>
<td></td>
</tr>
<tr>
<td>adapter 12 mm</td>
<td>32.5</td>
<td></td>
</tr>
<tr>
<td>adapter 1/8”</td>
<td>28.1</td>
<td></td>
</tr>
<tr>
<td>adapter 1/4”</td>
<td>28.4</td>
<td></td>
</tr>
<tr>
<td>adapter 3/8”</td>
<td>29.9</td>
<td></td>
</tr>
<tr>
<td>adapter 1/2”</td>
<td>32.7</td>
<td></td>
</tr>
</tbody>
</table>

*) Dimension A is typical finger-tight.
References


[34] S ZURBACK, P LASGORCEIX, and N CORNU. A 20kW High Power Hall Effect Thruster for Exploration. In 61st International Astronautical Congress, pages 1–9, Prauge, 2010. IAF.


[52] D Pavarin, F Ferri, M Manente, A Lucca Fabris, F Trezzolani, M Faenza, L Tasinato, O Tudisco, A Loyan, Y Protsan, A Tsaglov, A Selmo, K Katsonis, D Packan, J Jarrige, C Blanchard, P Q Elias, and J Bonnet. Thruster Development Set-up for the Helicon


References


