THE ADVANCED SPACE PROPULSION PROGRAMMES
AT AEA TECHNOLOGY, CULHAM LABORATORY

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ABSTRACT

AEA Technology's Culham Laboratory has been involved in advanced propulsion projects for many years. A large part of the work of the Laboratory in this area has been directed at the development of ion thrusters and to this end two thruster systems have been developed. These are aimed at a variety of missions from north-south station keeping to primary propulsion. In addition, in recent years a considerable effort has been devoted to other advanced propulsion concepts ranging from arcjets and plasma thrusters to fusion powered systems. The objective of this paper is to give an overview of these projects and to provide a status report on the more important ones. The future direction of the advanced propulsion programmes at AEA Technology is also discussed.

1. INTRODUCTION

The development of advanced, non-chemical, propulsion systems has been an area of great activity from the beginning of the space age. The incentive for this development is due to the fact that advanced propulsion systems offer the capability of substantially higher specific impulses than those obtainable with conventional chemical rockets. This is a desirable attribute since increased specific impulse leads to a reduction in the amount of propellant required to perform a given manoeuvre or mission. Consequently such propulsion systems will allow space vehicles to carry larger payloads, to have longer operational lifetimes or to have lower launch masses than those propelled by chemical rockets. Indeed, in some cases, the propellant savings offered by advanced propulsion systems are so large that the mission would be inconceivable without them.

Over the last three decades AEA Technology, through its Culham Laboratory, has been involved in a large variety of advanced propulsion projects. Initially this work was directed towards the development of ion thrusters which, subsequently, has culminated in the design of two thruster systems for satellite station-keeping and primary propulsion tasks. This technology is now mature and a large part of the current programme is involved in life-testing, electromagnetic compatibility studies and operational and performance demonstration. In addition, considerable effort has been directed at examining the possible applications of ion thrusters. This has included small spacecraft as well as deep space and interplanetary probes.

Complementary to the above work the Laboratory has also been involved in the study and development of a number of electrothermal propulsion concepts. This programme has examined arcjet thrusters, where computational studies have been used to determine factors that affect performance, and inductively coupled plasma thrusters, where an experimental study has examined the preliminary performance potential of the concept. This latter work is particularly interesting, in part because the discharge in an inductively coupled plasma thruster is electrodeless. Consequently such devices will avoid electrode erosion problems which can have a serious affect on the lifetime of arcjet thrusters.

Both electrostatic and electrothermal propulsion systems of the kind mentioned above are all forms of electric propulsion and all require the use of the spacecraft's power system for operation. However, there are other forms of propulsion system which are not so restricted. Probably the best known of these alternative systems is the nuclear thermal system whereby a nuclear reactor is used directly to heat the propellant which is then exhausted through a conventional nozzle. Most of the effort that has been expended in studying and developing these systems has been directed at nuclear
fission systems. However, in a recent study AEA Technology has examined the feasibility of using a fusion reactor to heat the propellant. This study was based on using the most credible current fusion confinement concept, the tokamak, and restricting extrapolations in confinement physics to those that are reliably expected. The results of this study are encouraging in that the predicted performance is well in excess of that of an equivalent chemical system for missions involving several hundred tonne payloads in Earth-Lunar space.

Instead of using a nuclear power source to heat the propellant another attractive propulsion concept relies on using a laser beam transmitted from a remote location to heat the propellant to high temperature. Since the power source is not carried by the vehicle impressive performance characteristics might be expected from such a concept. In fact in a study of a vehicle propelled by laser heating, undertaken by the Laboratory, such performance was indeed demonstrated although the study emphasised the importance of the complex physics of the laser-propellant heating process.

The above discussion is a summary of some of the main advanced propulsion concepts that have been studied by AEA Technology. In the following sections these propulsion concepts are discussed in more detail.

2. ION THRUSTERS

2.1 Programme History

The research and development programme that has culminated in the design of the UK-10 and UK-25 ion thrusters has its origins in the late 1960's. During this time the then Royal Aerospace Establishment (RAE) and AEA Technology, successfully developed a 10 cm diameter electron-bombardment type ion thruster. The primary application of the thruster was for north-south station-keeping (NSSK) on large communications satellites in geostationary Earth orbit. At the start of the programme the propellant selected was mercury giving a nominal thrust of 10 mN and an ion beam velocity of 30 kms⁻¹. In operation the thruster demonstrated very high mass utilisation, electrical efficiency and stability over very long periods of operation. Indeed life-test data from both the thruster and its critical components confirmed its suitability for all missions then under consideration. With the completion of life-testing and the design of the propellant supply and monitoring equipment (PSME) and the power conditioning and control system (PCCS) the thruster was close to flight qualification. At this stage the original programme was terminated and not restarted until the mid 1980's.

When the current programme was initiated in 1985 the original thruster design was resurrected, although for spacecraft compatibility reasons a change in propellant from mercury to xenon was made. Also changes in the mass of communications satellites meant that there was potential demand for thrusters with thrust up to 25 mN. Consequently most of the development and qualification activity is now concentrated at the 25 mN level for which a new PCCE is being provided. This thruster system is known as the UK-10 ion thruster.

During the 1980's, as today, there was considerable interest in the application of ion thrusters to primary propulsion so extending the range of possible missions. As a consequence in 1986 RAE and AEA Technology initiated the design of a larger ion thruster of 25 cm diameter and 200 mN nominal thrust. Again xenon was selected as a propellant and within a year the design performance was achieved. This larger thruster system is designated the UK-25.

The current status of the two thruster systems and their basic principles of operation is described in the following sections.

2.2 Thruster Operation

Both the UK-10 and the UK-25 ion thrusters are of the electron bombardment type. As such, a discharge is established between the cathode and anode, and the electrons in this discharge provide the means by which neutral gas is ionised. Figure 1 shows a schematic diagram applicable to both thrusters.
Earth Screen

Magnetic Solenoid

Magnetic Field lines

Flow Distributor

Anode

Cathode

Baffle

Cathode Heater

Cathode Keeper

Discharge Chamber

Neutraliser

Neutraliser Heater

Figure 1 Electron bombardment ion thruster schematic

With reference to the figure, gas is fed into both the hollow cathode and the main cylindrical discharge chamber. A coupling plasma inside the pole piece/baffle assembly provides the source of ionising electrons which produce the plasma in the main discharge chamber. The magnetic field configuration is shaped to illuminate the extraction system with this plasma in as uniform a manner as possible, while at the same time confining the ionising electrons for a sufficient time that a good source efficiency is achieved. Typical values of efficiency of the order of 250 W/A are found in such sources, at a mass utilisation efficiency of 90%. Ions from the main discharge plasma that drift into the extraction grid region are rapidly accelerated by the strong electric fields and form the emitted ion beam. This ion beam is neutralised by electrons emitted from a secondary hollow cathode or neutraliser. The grid assembly forms the downstream boundary of the discharge chamber.

Several features of the design are worth noting:

- Three independently controllable propellant feeds are used to supply the cathode, the distributor and the neutraliser respectively. These allow active throttling over a wide range.
- Solenoids are used to generate the magnetic field. This feature assists in providing flexibility, in particular enabling wide thrust and throttling ranges to be attained.
- The grids are dished inwards. This results in a low beam divergence (about 10°) and good resistance to inter-grid electrical breakdown as the thruster heats up.

2.3 UK-10 Thruster Status

As described above the UK-10 Ion Thruster Programme restarted in 1985 when the original mercury propellant T5 ion thruster was converted to run on xenon⁴. The decision for this change was made primarily due to concerns over the handling and contamination effects of mercury. The added benefit accrued was a simplification in the power supplies, since heaters to prevent mercury condensing in the thruster and for vapour production, could be omitted. The PSME requirements were complicated, however, by the need to store and flow regulate xenon.

Although testing resumed at 10 mN of thrust it was soon decided to increase the nominal thrust level to 25 mN. This was the level determined to be most desirable by potential customers, then perceived to be large communications satellite operators. Performance at this level, and at an exhaust velocity of 40 km/s, was proven satisfactorily. Indeed the UK-10 thruster has been operated up to thrust levels of 70 mN without major problems.

In 1988 the first flight application for the UK-10 ion thruster was offered. An European Space Agency technology satellite, ARTEMIS, (Advanced Relay TEdnology MIssion) became the first spacecraft destined to fly not only the UK-10 ion propulsion system, but also the German RITA (Radio Frequency Ion Thruster Assembly)⁵. For this application the thrust level of the UK-10 thruster has to be downrated to 18 mN due to spacecraft power limitations.

The UK-10 programme at the Culham Laboratory is directed at proving the operation of the thruster at 25 mN. Thruster characterisation and long term operation at this nominal thrust level are considered acceptable for the ARTEMIS spacecraft despite its reduced thrust requirements.
The main objectives of the current UK-10 programme, are given below.

- Define the nominal operating parameters for the UK-10 ion thruster. This has been done successfully and the details are given in Table 1.

- Characterise the ion beam and any contamination effects such as extraction grid sputtering. The ion beam will be monitored by complex diagnostics, to determine current profiles, divergence and xenon ion species ratios.*

- Investigation of dual and triple extraction grid systems to determine if a third earthed grid reduces grid sputter. 500 hour cycle tests on both the dual and triple grid sets will be performed. Diagnostics as above will be used to see if variations occur over this period.

- Perform detailed electromagnetic compatibility testing to characterise emissions from the thrusters.

- Finally a long duration cyclic lifetest will be performed to prove extended thruster operation. Initially 2000 hours of testing will be performed although increasing this to 6000 hours is under discussion.

The work currently being performed at AEA Technology is an integrated programme supported by the British National Space Centre (BNSC) and INTELSAT. Both contracts are monitored by the Defence Research Agency, Space Division, RAE Farnborough. The BNSC work is geared toward the conventional two grid UK-10 thruster and INTELSAT are supporting the triple grid work, although both organisations and ESA have full view of the work.

Figure 2 shows a photograph of the UK-10 ion thruster used in current testing. This thruster is already at engineering model (EM) status. In all of the testing described above an EM version of the PSME is used. A breadboard PCCE will be used for the 500 hour cyclic testing and EMC tests. It is planned that an EM PCCE will be used for the long duration cyclic tests. This is being developed by Matra Marconi Space UK Ltd*.

<table>
<thead>
<tr>
<th>Table 1</th>
<th>The performance characteristics of the UK-10 and UK-25 ion thrusters</th>
</tr>
</thead>
<tbody>
<tr>
<td>Parameter</td>
<td>UK-10</td>
</tr>
<tr>
<td>Nominal Thrust (mN)</td>
<td>25</td>
</tr>
<tr>
<td>Beam Voltage (V)</td>
<td>1100</td>
</tr>
<tr>
<td>Specific Impulse (s)</td>
<td>3200</td>
</tr>
<tr>
<td>Thruster Power (w)</td>
<td>650</td>
</tr>
<tr>
<td>Propellant Utilisation Efficiency (%)</td>
<td>85</td>
</tr>
<tr>
<td>Total Efficiency (%)</td>
<td>76</td>
</tr>
<tr>
<td>Power to Thrust Ratio (W/mN)</td>
<td>26</td>
</tr>
</tbody>
</table>

Figure 2 The UK-10 ion thruster

The characterisation of the UK-10 thruster parameters has so far been performed using laboratory power supplies. A performance map is shown in Figure 3, where the propellant utilisation efficiency, at a fixed flow rate into the thruster, is plotted against the discharge plasma ion production costs (discharge power per amp of ion beam current). This is measured over varying discharge voltage and current levels. The 25 mN
nominal thrust case is at 85% propellant utilisation efficiency.

Ion probe measurements of the beam have shown a profile as shown in Figure 4. The 3 dimensional probe positions and computerised data acquisition system (DAS), allow rapid repeatable beam measurements to be made.

In summary the UK-10 ion thruster is advancing toward its first flight application on the ESA ARTEMIS spacecraft. The current work is geared towards proving a thruster that is safe and suitable for propulsion tasks on future spacecraft.

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Figure 3 UK-10 ion thruster performance map

2.4 UK-25 Thruster Status

The development of the UK-25 ion thruster has lagged behind the UK-10 by several years. The applications for primary propulsion are more limited, and at present no specific missions have been defined. The work has been funded and monitored by the Defence Research Agency, Space Division, RAE Farnborough. The UK-25 ion thruster is shown in Figure 5.

The design of the UK-25 thruster was derived by scaling up the UK-10 design to achieve a thrust level of nominally 200 mN. This necessitated an increase in accelerating potential and beam current. The resultant thruster had a grid diameter of 25 cm. Discharge chamber dimensions were also increased and a hollow cathode with a significantly higher current capability had to be developed. The nominal UK-25 ion thruster parameters are shown alongside those of the UK-10 thruster in Table 1. A performance map of the UK-25 thruster is shown in Figure 6 and the throttling range of the thruster in Figure 7.

Figure 4 UK-10 ion thruster beam profile

Figure 5 UK-25 ion thruster
The design of the ion extraction grids was done using in-house software codes. The higher accelerating potential of 2.2 kV, and second grid potential of -350 V maintain optimum ion optics over a grid gap of only 0.75 mm.

The main early problem encountered during testing was significant grid distortion which led to the extinguishing of the beam. This was found to be due to higher than expected power input onto the thruster outer magnetic field pole piece, which also supported the extraction grids. This thermal loading resulted in the distortion of the two grids and an enhancement of the electric fields between them. The combination of the two produced forces which eventually led to the two grids shorting. Subsequent analysis using a computer model showed that of the order of 50% of discharge power was being deposited in the pole piece area. Redesign initially concentrated on verification of the model by increasing the pole piece radiating area in order to decrease the rate of thermal loading. Further testing demonstrated successful and stable operation, although at higher grid and pole piece temperatures.

Another major area of investigation with the UK-25 thruster has been with regard to concerns over possible cathode erosion phenomena. The high current cathode necessary for the UK-25 thruster is similar to designs elsewhere which have been associated with large discharge chamber erosion problems. A study and experimental programme was conducted to determine the relevance of the phenomena to the UK-25 thruster and is summarised below. The UK-25 cathode assembly is shown in Figure 8.

Initially a survey of existing data on both mercury and xenon primary ion thrusters was conducted.
to look at cathode and cathode erosion information. In particular the phenomena of high energy jet ions emanating from the cathode was identified as a potential source of erosion. An experiment to look at cathode emissions in the UK-25 thruster was performed using a retarding potential analyser. This detected high energy ions, as predicted, which are thought to be produced by a magnetohydrodynamic mechanism. Solutions to this have been suggested, such as increasing the cathode orifice diameter. However further work is required and has been proposed.

In summary the current situation is that the UK-25 ion thruster will soon undergo characterisation tests in an attempt to investigate lifetime factors, thrust range limits and ion beam characteristics. The next phase of the programme will look at the requirements to operate the thruster with a less expensive propellant than xenon, such as krypton or argon. This is seen as critical for future flight opportunities. Other investigations into the cathode and system aspects are also proposed.

2.5 ESA Primary Propulsion

In parallel to the activities on the indigenous UK ion thruster programmes, AEA Technology is now also working with MBB of Germany and the University of Giessen, Germany on an ESA sponsored primary ion propulsion programme.

ESA has decided that, rather than develop either the German RIT-35 or UK-25 ion thrusters as the ESA standard large ion thruster, there is a significant synergy to be found in combining the best from the two systems. Accepting that RF sources do not have cathode erosion or single point failure problems it has been decided that such a source will form the basis for the ESA thruster.

The expertise of Culham in ion extraction systems is being utilised for the new thruster. The programme is in early days, and it is expected that a laboratory model of the new thruster should be available for test in 1992.

2.6 Large Advanced Ion Thrusters

The UK-10 and UK-25 ion thrusters, described above, are capable of providing the propulsion on all existing and planned missions that call for ion propulsion. However, it is likely that sometime in the next century there will be a requirement for very large vehicles with correspondingly large propulsion systems. It is of interest, therefore, to examine the ultimate performance limits of ion thrusters.

Such a study was performed some years ago by AEA Technology and made use of the extensive experience of the Culham Laboratory in the design of ion sources for neutral beam injectors in fusion experiments. Such sources, and particularly those developed for the Joint European Torus (JET) experiment, are typically of high power and incorporate much of the technology required for advanced ion thrusters. A typical 4.8 MW ion source developed for the JET experiment by the Culham Laboratory is shown in Figure 9. This is a magnetic multipole ion source giving large areas of uniform plasma whilst operating at a low source pressure and a high efficiency of positive ion production.

Table 2 gives the principle characteristics of an advanced ion thruster based on argon propellant and derived from the above ion source technology. Such a thruster might power a deep space probe to the edges of the Solar System.

Figure 9 A 4.8 MW ion source
Table 2 Characteristics of an advanced ion thruster

<table>
<thead>
<tr>
<th>Propellant</th>
<th>Argon</th>
</tr>
</thead>
<tbody>
<tr>
<td>Beam Voltage (kV)</td>
<td>9</td>
</tr>
<tr>
<td>Exhaust Velocity (km/sec)</td>
<td>188</td>
</tr>
<tr>
<td>Beam Current (A)</td>
<td>320</td>
</tr>
<tr>
<td>Thrust (N)</td>
<td>27.5</td>
</tr>
<tr>
<td>Discharge Losses (W/A)</td>
<td>100</td>
</tr>
<tr>
<td>Total Power (kW)</td>
<td>2,919</td>
</tr>
<tr>
<td>Power Efficiency</td>
<td>0.989</td>
</tr>
<tr>
<td>Propellant Efficiency</td>
<td>0.9</td>
</tr>
<tr>
<td>Thruster Diameter (m)</td>
<td>0.5</td>
</tr>
</tbody>
</table>

2.7 Applications of Ion Thrusters

There are many potential applications of ion thrusters to space missions, ranging from north-south station keeping and drag make-up to deep space missions such as comet or asteroid rendezvous. Most of these missions have been examined in considerable detail over the years and the advantages of using ion thrusters well expounded. In this section some less well studied applications of ion thrusters are discussed.

One that has been investigated at AEA Technology is the use of the UK-10 size ion thruster as a major propulsion system on small spacecraft. In such cases the low thrust can provide sufficient acceleration to perform useful orbital manoeuvring whilst staying within achievable power limits.

The main direction of this investigation has been in the design of an solar powered, ion thruster propelled, orbit raising spacecraft system known as ARGOS\textsuperscript{13}. This ARGOS concept (shown in Figure 10), is of a small (< 1m\textsuperscript{3}, < 100 kg) spacecraft that can transfer payloads from small launch vehicles to useful orbits in excess of 1000 km altitude. The ion propulsion system, due to the high specific impulse, offers mass enhancement capabilities. The versatility of a commandable on/off propulsion system and the significant end of transfer solar array power are also attractive. Figure 11 shows the transfer time from a 300 km initial circular orbit for ARGOS. The not insignificant durations are still not unreasonable.

There are a number of other applications for small ion thruster technology on small satellites, some of which are discussed below.

- Orbit maintenance in Low Earth Orbit (LEO) is enhanced by ion thrusters due to the reduced propellant mass and volume requirements, and the smooth manoeuvring from low thrust.
- Maintenance of orbit inclination otherwise affected by the Earth, Sun and Moon perturbative effects. Again the high specific impulse is propellant mass efficient.
- Concepts such as ARGOS can also act as a basic generic spacecraft bus for environmental and scientific payloads in LEO. Orbit raising and lowering can be performed at ease.
- Although the mass advantage for small
geostationary spacecraft is not significant for NSSK manoeuvres say, they may still be an advantage to their use. The transfer orbit may be partially performed by an ion thruster giving mass savings, as can orbit position acquisition.

Figure 11 ARGOS transfer duration in LEO

The application of small ion thrusters to small satellite markets is expected to produce an interesting expansion of advanced propulsion usage. It is hoped that the generally higher risk of the small satellite industry will allow quicker uptake of ion thruster systems than by the conservative conventional spacecraft operators.

The use of larger ion thrusters, such as the UK-25, opens the door to a whole range of deep space missions many of which have received considerable attention over the years. An important part of the electric propulsion system, however, is the power supply, which must generate several tens of kilowatts over many days. Basically the choice of power system lies between solar arrays or a nuclear reactor and both systems have there advantages and disadvantages. Figure 12 shows an artist impression of a deep space probe propelled by an array of UK-25 type ion thrusters and powered by a nuclear reactor\(^4\).

Figure 12 A nuclear powered deep space probe

3. ELECTROTHERMAL PROPULSION

Electrothermal propulsion concepts are similar in operation to chemical rockets in the sense that thrust is obtained by expanding a heated gas through a converging-diverging nozzle. However, unlike chemical systems where performance is limited by the specific energy of the chemical process, electrothermal rockets can achieve high specific impulses. In fact, the specific impulse attainable by an electrothermal thruster is simply dependent on the amount of electrical energy that can be added to a given mass of propellant.

The main issues pertaining to such systems are ones of material suitability, thermal efficiency and the means by which the electrical energy is converted into directed kinetic energy.

In this section two electrothermal propulsion concepts are discussed and work undertaken by the Culham Laboratory described. The two thruster types are the so-called arcjet and a device called an Inductively Coupled Plasma (ICP) thruster. Work on these two systems has taken place over the last three years and includes both computational as well as experimental studies. The incentive for studying these devices is described below.
Arcjets and Inductively Coupled Plasma thrusters offer potential propulsive performance in a regime which is not accessible to either conventional chemical systems nor ion thrusters. This regime is one in which relatively high thrust is produced in combination with high specific impulse and would be advantageous to missions where rapid in-orbit manoeuvrability and optimum propellant mass are desirable. Typically both ICP and Arcjet thrusters would operate with characteristics intermediate between the high thrust, low specific impulse of chemical systems and the low thrust, high specific impulse of ion thrusters.

3.1 Arcjets

Historically arcjet thrusters were amongst the first electrothermal propulsion systems to be examined with early work being performed during the late 50's/early 60's. After an interval of some years when little work was done interest in arcjets was revived in the US in the mid 80's and today arcjets represent one of the most developed of the electric propulsion systems.

The basic operating principle behind an arcjet is illustrated in Figure 13. The objective is to heat the propellant gas by passing an electric arc between two electrodes past which the gas flows. In general the cathode is located axially in the device and the anode is formed into a combined plenum chamber/constrictor/nozzle configuration. The flow of gas in the constrictor and the constrictor walls confine the arc along the constrictor centre line.

Currently arcjet thrusters have achieved specific impulses of around 750 seconds and efficiencies of about 30%. However lifetime is a key concern and modern systems have only been able to achieve lifetimes of 1000 hours or so. Lifetime is determined largely by electrode erosion which is a key characteristic of these systems.

Despite the interest and effort expended in developing arcjet thrusters there has been relatively little detailed computational analysis of the features that affect arcjet performance. Whilst this is beginning to change with the completion of a number of computational studies, most improvements in performance have been the result of experimental observation rather than theoretical understanding.

![Schematic diagram of an arcjet](image)

Partly as a consequence of this AEA Technology have recently undertaken a computational study with the objective of developing a computer program to model the principal features of arcjet operation. Rather than constructing a complex multi-dimensional fluid dynamics code the study undertook to develop a simple one-dimensional model that nevertheless included most of the important physical processes. The advantage of this is that the resultant program is relatively fast, runs on a personal computer and can be used to generate a large amount of performance data rapidly.

The computer model works by combining equations describing the transonic fluid dynamics, the heat transport and the non-equilibrium chemistry. The model therefore includes radiative, conductive and convective heat flow, friction and chemical effects in the propellant.

For given input parameters the solution is obtained by integrating two equations. The first describes the acceleration of the propellant as it passes through the arcjet and the second describes the corresponding degree of dissociation of the propellant. Together with an approximate expression for the equation
of state for the dissociating propellant the solution allows the performance of the arcjet model to be determined. The current model is capable of modelling hydrogen, nitrogen, ammonia and hydrazine, the latter two by assuming an appropriate mixture of hydrogen and nitrogen. Since nitrogen does not dissociate appreciably until high temperatures (~ 6000 K) are reached only hydrogen dissociation need be modeled. This approximation nevertheless appears to yield very good results.

A typical solution from the model is illustrated in Figure 14 which shows the pressure, temperature and velocity at various locations along the length of a 30 kW arcjet. One can clearly see how the temperature and pressure fall very rapidly in the region of the throat while the velocity increases throughout the nozzle region. This behaviour is to be expected.

An important objective of the study was to verify the program by comparing its results to published experimental data. Figure 15 shows a comparison of experimental and model values of the thrust as a function of arcjet power for ammonia arcjets. One can see that the agreement is very good considering the simplified nature of the model and the range of arcjet powers over which the comparison is made. The discrepancy is largely due to the fact that the wall temperature and inlet temperature remain fixed which is not the case in reality because of heat conduction. This effect will be included in a future version of the model.

In general the calculations performed with the model agree well with experimental data not only for ammonia arcjet systems but also for hydrogen and hydrazine arcjets. The results suggest that the constrictor has an important role in allowing efficient heat input to the flow while the flow is accelerating. Dissociation and radial heat transport to the walls are also important in determining the overall efficiency of arcjets.

Future developments of the code will concentrate on improving aspects of the heat transfer to the walls and conduction through them, improving the modelling of the inlet region in order to better accommodate experimental data and incorporating some of the important physics of the arc. This latter feature may be achieved using correlations for voltage-current characteristics.

![Figure 14](image-url) Calculated pressure, temperature and velocity profiles along the length of a 30 kW arcjet

![Figure 15](image-url) Comparison of experimental with calculated thrust values

3.2 Inductively Coupled Plasma Thrusters

An inductively coupled plasma (ICP) thruster
operates by using a magnetic field alternating at high frequency to drive currents in a propellant and in so doing to heat the propellant to high temperature. Figure 16 shows a schematic representation of an ICP thruster. The magnetic field is generated by a coil surrounding the throat chamber and typically alternates at a frequency ranging from a few megahertz to a few tens of megahertz. Since the alternating magnetic field is axial within the coil region the currents induced in the propellant gas are azimuthal and strongest in the centre of the coil region. The propellant is heated to temperatures in excess of 5000 K by ohmic dissipation and the propellant in the hottest part of the discharge is partially ionised and strongly radiating. The high temperature gas is expanded through a conventional nozzle converting the thermal energy of the gas into directed kinetic energy and so producing thrust.

ICP thrusters are a derivative of the so-called inductively coupled plasma torch which has been used extensively for a number of industrial applications. However, despite this there has been little reported experimental work on ICP thrusters, although what has been reported indicates encouraging performance for such systems.

One of the main potential advantages of ICP thrusters over similar systems, such as arcjets, is the absence of electrodes. This is important since electrode erosion is one of the primary life limiting features of arcjets. Consequently the electrodeless nature of the discharge in an ICP thruster offers the prospect of greatly enhanced lifetime and therefore overall system performance.

Over the last few years AEA Technology has been studying the capability of ICP thrusters. Initially this has been through a series of preliminary experiments that have been performed in order to gauge the performance of such thrusters. These experiments are described in this section.

Figures 17 and 18 illustrate the experimental ICP thruster and its associated propellant and electrical supplies. The thruster consists of a fused silica tube with a copper nozzle arrangement at one end and a stainless steel closing flange with propellant supply inlet at the other. Surrounding the tube is a five-turn water-cooled induction coil supplied with radio-frequency power from a commercial RF generator at 2 MHz frequency and 6 kW maximum power output. The lower flange plate has a hole through which a graphite rod is inserted and is used for initiating the discharge. Instrumentation consists of a propellant flow meter, a number of pressure gauges and transducers and several thermocouples.

The thruster discharge is initiated by allowing the graphite rod to be heated to incandescence in the RF field and in so doing to produce a small pilot discharge around the rod. Once this has been achieved the application of further power leads to the establishment of the main discharge. This discharge or arc is very bright and is the result of circulating electric currents induced in the argon propellant by the time varying axial magnetic field. As noted in previous studies the gas flow configuration is important in stabilising the discharge and in the thruster the argon is introduced tangentially in order to achieve a vortex flow regime. This vortex flow prevents the discharge from reaching the tube walls which would otherwise cool the arc or lead to over-heating of the tube. In fact, at very low flow rates the discharge becomes unstable, eventually extinguishing itself.
The experimental programme was performed in two parts. The first part consisted of a series of cold flow tests and the second part comprised the main experimental programme in which the thruster was operated at up to full discharge power.

The cold flow tests were performed in order to obtain a discharge coefficient for the thruster nozzle. In the main part of the experimental programme the objective of the analysis was to derive the thermal power in the flow for a given electrical power to the coil of the thruster. In this case the thermal power was determined from an effective flow temperature derived from the perfect gas flow equations and using the experimentally determined nozzle discharge coefficient.

Experiments were performed at powers ranging from 4.2 kW to 4.8 kW. The results showed that the input power efficiency peaks at around 18%, corresponding to an effective flow temperature of 1450 K and a vacuum specific impulse of about 123 seconds (which could reach about 550 seconds in hydrogen). However, the efficiency, mean stagnation temperature and specific impulse all fall with increasing specific input.
power, a phenomena that is due in part to the way in which the flow is partitioned between the hot core and cooler boundary.  

The results obtained in this series of experiments are considered to be very favourable at this stage since no detailed optimisation of the flow characteristics or size of the thruster chamber have been performed. The efficiency obtained, for example, compares well with arcjets of similar power.

A more comprehensive experimental programme is planned for the future where more detail measurements will be made and where the importance of radiative heat transfer will be examined. This programme will include measurements of the thrust efficiency, specific impulse and energy transport mechanisms over a range of input powers up to 30 kW. Operation in nitrogen will also be attempted in order to examine the effect of dissociation.

4. ADVANCED PRIMARY PROPULSION SYSTEMS

In Section 2 the development of ion thrusters for primary propulsion was discussed. For a number of missions, particularly deep space missions, these propulsion systems offer attractive performance capabilities, largely because of their high specific impulse. However, ion thrusters, like all electric propulsion devices, suffer from the disadvantage of requiring an on-board power source that can generate the required electrical power. These power sources are heavy and for some missions would lead to unattractive weight penalties. Fortunately there are propulsion concepts that can overcome this difficulty and offer promise for a number of missions.

Two such concepts have been studied by the Culham Laboratory and are described in this section. They are a propulsion system based on using a remotely generated laser beam to heat the propellant and a propulsion system based on a nuclear fusion reactor.

4.1 A Laser Powered Propulsion Concept

The laser propulsion concept is an attempt to decouple the energy source from the rocket vehicle and in so-doing to produce high thrust and high exhaust velocity without the need for a heavy on-board power source. Instead of using energy from chemical or nuclear reactions to heat a propellant, a powerful beam of laser energy is employed. This laser energy can, in principle, be used to heat the propellant to very high temperatures, resulting in high exhaust velocities.

The basic idea is to use a large, remote (eg. ground based) laser facility to produce a high power (multi-megawatt) laser beam which is then transmitted through space to a vehicle equipped to intercept it. The vehicle has a specially designed heating chamber into which the laser beam may be reflected using suitably shaped mirrors. Inside the heating chamber a so-called combustion wave is generated, close to the focus of the laser beam, which is characterised by a sharp rise in the temperature of the inflowing propellant. The energy is absorbed in the high temperature region by inverse bremsstrahlung, and this region then radiates and conducts energy forward, heating the oncoming cold gas up to a temperature at which the necessary electrons are produced. This mode must be started by some auxiliary technique, such as the application of a spark, in order to produce the electrons needed to initiate absorption.

Since there is no limit to the temperature that the propellant gas can be heated to, the chamber temperature can be raised to the metallurgical limits of the absorption chamber walls. This can allow very high propellant temperatures to be achieved, leading to exhaust velocities in excess of 20 kms.  

The laser powered vehicle concept studied by AEA Technology is an attempt at determining the kind of performance that might be expected using laser heating. The general arrangement of the vehicle is illustrated in Figure 20. The specific design point for the vehicle was a geostationary mission from LEO in which it was assumed that the laser retransmission mirror was in GEO and that 2700 seconds could be available to accomplish the first transfer orbit burn. A 500 MW continuous beam laser was assumed with 300 MW available at the vehicle after taking into account losses. About 6.8 tonnes of payload could be accommodated in
a vehicle of total initial mass of just under 18 tonnes. Table 3 gives a mass breakdown of the vehicle.

![Diagram](image)

**Figure 20** General arrangement of the laser-powered orbital transfer vehicle

**Table 3** Vehicle all-up masses (kg) (GEO round trip mission)

<table>
<thead>
<tr>
<th>Component</th>
<th>Mass (kg)</th>
</tr>
</thead>
<tbody>
<tr>
<td>Basic Mass</td>
<td>3901</td>
</tr>
<tr>
<td>Propellants LH₂</td>
<td>6358</td>
</tr>
<tr>
<td>LOX</td>
<td>702</td>
</tr>
<tr>
<td>Storables</td>
<td>200</td>
</tr>
<tr>
<td>Payload</td>
<td>6805</td>
</tr>
<tr>
<td><strong>Total</strong></td>
<td><strong>17,966</strong></td>
</tr>
</tbody>
</table>

An important aspect of the concept was the combustion or heating chamber and the nozzle. The heating chamber has to both accommodate the high temperatures and pressures of the propellant as well as to allow the laser beam to be brought to a focus inside. To do this a fused silica window was included in the upstream end of the chamber. Fused silica has a very low absorption coefficient in the wavelength range around 1.0 μm, which is probably the optimum wavelength for the laser beam on the basis of other criteria. Figure 21 shows the overall thruster arrangement and Table 4 a summary of the performance of each thruster.

![Diagram](image)

**Figure 21** The thruster configuration

**Table 4** Thruster parameters (each)

<table>
<thead>
<tr>
<th>Parameter</th>
<th>Value</th>
</tr>
</thead>
<tbody>
<tr>
<td>Exhaust velocity (kinematic equilibrium)</td>
<td>23,604 m/s</td>
</tr>
<tr>
<td>Actual exhaust velocity</td>
<td>22,895 m/s</td>
</tr>
<tr>
<td>Flow</td>
<td>0.3571 kg/s</td>
</tr>
<tr>
<td>Thrust</td>
<td>8177 N</td>
</tr>
<tr>
<td>Throat radius</td>
<td>0.02514 m</td>
</tr>
<tr>
<td>Exit radius</td>
<td>0.1509 m</td>
</tr>
<tr>
<td>Chamber (throat total pressure)</td>
<td>50 bar</td>
</tr>
</tbody>
</table>

The above study of a laser powered orbital transfer vehicle demonstrated that impressive performance could be achieved using a laser beam to heat the propellant. However, a very important part of
the process is the actual propellant-laser heating. Recent studies at AEA Technology and elsewhere\textsuperscript{34} indicate that very large amounts of the absorbed laser power are quickly re-radiated. Such large power flows would have serious consequences for the integrity of the combustion chamber. Further studies are clearly required to elucidate this problem.

4.2 The Starlight Fusion Propulsion Concept

Nuclear energy, in the form of both fission and fusion, has a specific energy many orders of magnitude greater than that of chemical energy. Consequently, it offers the potential of making feasible a propulsion device of impressive capabilities and performance. While there have been many studies of fission powered propulsion systems the number of studies examining fusion based systems is much less. Moreover most of these studies have assumed extrapolations in fusion physics well beyond what can reliably be expected today.

In a recent study AEA Technology has revisited fusion propulsion by carrying out a preliminary study of a different type of fusion rocket\textsuperscript{35}. The objective of this study was to examine the most credible current fusion concept, the tokamak, and identify the possibilities it offered in terms of propulsion for near-Earth missions over the next 50 years. Some extrapolation from present experience is required since no controlled ignited fusion plasma has yet been obtained. However, the aim was to keep these extrapolations within the bounds of the current physics scaling laws and likely near-term technology. In particular the study assumed that the fusion reactor is simply an energy source which heats a propellant to a sufficiently high temperature to give a high exhaust velocity of around 10 \text{ km/s}.

A number of important features followed from these ground rules:

- Tokamak confinement concept
- Deuterium/tritium fuel
- Specific impulse – 1000s
- Magnetic field coils cooled by cryogenic propellant (normal conducting coils)
- Reactor indirectly heats the propellant
- Short operational lifetime (10's hours)
- No tritium breeding
- No specific vacuum pumping

A key feature of the design was the toroidal field coil system which was cooled to low temperature (65 K) and run at very high magnetic fields. This was crucial in producing a compact design. Table 5 shows the principal parameters of the concept which was named Starlight. Figure 22 is a schematic representation of the concept. The turbo machinery is partly to power the magnetic field coils and auxiliary machinery and partly to optimise the overall thermodynamic cycle.

The results from this study are encouraging as they yield a conceptual design and a highly compact fusion reactor which appears capable of achieving impressive performance without requiring extensive extrapolations in current technology. Clearly much of the design remains uncertain and further studies are required to elucidate it. Nevertheless the concept appears to be very competitive with chemical systems for large payloads (several hundred tonnes) in Earth-Lunar space.

![The Starlight fusion propulsion concept](image)

Figure 22 The Starlight fusion propulsion concept
Table 5 Starlight principal parameters

<table>
<thead>
<tr>
<th>FUSION REACTOR CORE</th>
<th></th>
</tr>
</thead>
<tbody>
<tr>
<td>Minor radius</td>
<td>1.08 m</td>
</tr>
<tr>
<td>Elongation</td>
<td>2.0</td>
</tr>
<tr>
<td>Major radius</td>
<td>2.38 m</td>
</tr>
<tr>
<td>Aspect ratio</td>
<td>2.2</td>
</tr>
<tr>
<td>Peak toroidal field</td>
<td>27.9 T</td>
</tr>
<tr>
<td>Toroidal field on axis</td>
<td>10.5 T</td>
</tr>
<tr>
<td>Safety factor</td>
<td>2.1</td>
</tr>
<tr>
<td>Plasma current</td>
<td>31 MA</td>
</tr>
<tr>
<td>Beta</td>
<td>9.5%</td>
</tr>
<tr>
<td>Total fusion power</td>
<td>10 GW</td>
</tr>
<tr>
<td>First wall neutron power</td>
<td>44 MW/m²</td>
</tr>
<tr>
<td>TF coil power dissipation</td>
<td>1 GW (0.5 GW)</td>
</tr>
</tbody>
</table>

<table>
<thead>
<tr>
<th>PROPULSION SYSTEM</th>
<th></th>
</tr>
</thead>
<tbody>
<tr>
<td>Exhaust velocity</td>
<td>9442 m/s</td>
</tr>
<tr>
<td>Propellant mass flow rate</td>
<td>199.3 kg/sec</td>
</tr>
<tr>
<td>Thrust</td>
<td>1.88 MN</td>
</tr>
<tr>
<td>Component Masses</td>
<td></td>
</tr>
<tr>
<td>(tonnes)</td>
<td></td>
</tr>
<tr>
<td>Toroidal Field Coils</td>
<td>75</td>
</tr>
<tr>
<td>Blanket</td>
<td>152</td>
</tr>
<tr>
<td>Shield (Shadow only)</td>
<td>57</td>
</tr>
<tr>
<td>Others</td>
<td>16</td>
</tr>
<tr>
<td>Total system mass</td>
<td>300</td>
</tr>
</tbody>
</table>

5. SUMMARY

The advanced propulsion programmes described in this paper cover a wide range of different propulsion concepts, several of which have been under development for many years. In the case of ion thruster development the UK-10 system is moving towards flight status and it is likely that in the next few years the system will achieve full space qualification. The larger UK-25 ion thruster, on the other hand, is currently undergoing characterisation and lifetime tests. In the next phase of the program operation in other propellants will be examined.

Concurrently with these programmes there are important studies being undertaken on the use of ion thrusters for a variety of missions. These studies will, hopefully, enhance the commercial prospects of ion thrusters and lead to their wider use on-board space vehicles.

In other areas AEA Technology has been active in developing novel concepts and in performing proof-of-principle and theoretical studies of new propulsion systems. These studies will continue with the emphasis being on pushing the concepts towards practical demonstration.

6. ACKNOWLEDGEMENTS

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