Hollow Cathode Thruster Design and Development for Small Satellites

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Direct thrust measurements and plasma plume diagnostics using a retarding field energy analyser have been performed for a developmental Hollow Cathode Thruster for discharge power of up to 53W for krypton and xenon. A maximum thrust of 1.6±0.1mN and specific impulse of 85±5s at 20sccm xenon and 53W discharge power are measured. The modular construction of the device allows altering the distance between keeper electrode and orifice. Propulsive performance is decreased considerably by increasing the distance between keeper and orifice from 1mm to 3mm. RFEA measurements 25cm downstream the HCT keeper operating with Krypton indicate the presence of a higher energy secondary ion population. The architecture and development status of a flight experiment consisting of a flight worthy HCT, a Flow Control Unit and a Power Processing Unit are also presented. The flight experiment is designed to operate from a small spacecraft bus. A PPU prototype has been manufactured and successfully tested with the developmental HCT.

I. Introduction

HOLLOW Cathode Thrusters (HCTs) is an active field of research aiming to develop secondary propulsion for large spacecraft (S/C) platforms, such as telecom S/C with installed electric propulsion (EP) systems, or primary propulsion for small S/C, with enhanced performance compared to resistojets. Hollow Cathodes (HCs) have demonstrated reliable performance1 in EP thrusters as main electron sources/beam neutralisers and as plasma contactors on the International Space Station (ISS)2. HCs were investigated in the 1970s for applicability as standalone thrusters and part of an all-electric propulsion suite for GEO platforms3,4 with early thrust measurements performed with mercury HCs5. Recent experimental investigations with inert gases have demonstrated thrust and specific impulse levels useful for S/C propulsive applications for small platforms and all-electric GEO S/C6. Thrust creation mechanism is postulated to be mainly due to gas dynamics at low power/discharge current regimes and magno-hydro-dynamic at high current and low flow rate7. High temperature neutrals present in the HCs8 and high energy ions9 observed may contribute to the thrust generated by the HCTs.

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HCT research and development in the Surrey Space Centre (SSC) are focused on propulsion for small, limited on-board resources S/C in low Earth orbit (LEO). In that scope, a laboratory prototype HCT is under testing using direct thrust measurements and retarding field analyser (RFEA) measurements, in an effort to characterise its performance and select a baseline configuration for an HCT flight experiment. The flight HCT will be an experimental enhancement to the platform of TechDemoSat-1. The experiment comprises of a flight-worthy HCT, Power Processing Unit (PPU) and Flow Control Unit (FCU), all under development in the SSC, using mainly commercial off the self components (COTS). Initial test results of the prototype thruster, the architecture of the flight implementation and early test results of the PPU prototype are presented.

II. Experimental Set up

A. Hollow Cathode Thruster (HCT)

A laboratory prototype HCT has been manufactured in the SSC and is under characterisation (Fig.1a). It utilises an enclosed keeper with no secondary anode or external ion acceleration/extraction device. The overall envelope is Ø40mm x 50mm length. The cathode tube and keeper are made of molybdenum. The cathode tube houses a type S insert swaged into position by its electrical leads. The insert is procured by Semicon Associates and measures 3mm O.D. x 1mm ID x 11mm able to provide a nominal discharge current of 3.4A. The laboratory HCT has been designed in a modular way in order to be able to test different geometrical arrangements for the orifice, keeper bore and keeper to orifice distance. Two cathode tubes are available with 1mm and 0.5mm orifice diameters, for the same orifice plate thickness. Keeper bore can be 4mm or 1mm, and the distance between orifice plate and keeper can be varied by use of a ceramic spacer situated between the Al₂O₃ insulation and the cathode tube. Thrust measurements of the HCT with 1mm orifice and 4mm keeper bore for 1mm and 3mm distance between orifice and keeper are shown in section III.

Insulator and interface are made from machinable grade alumina (Al₂O₃). The heater is made by 0.25mm tantalum wire coiled in grooves of a boron nitride (BN AX05 grade) cylinder encapsulated in a cylindrical BN AX05 cylindrical sheath. Heater leads out of the HCT are made from thicker 0.5mm molybdenum wire sheathed in 0.5mm OD Al₂O₃ tubes. No radiation shield is installed in the laboratory HCT thus the heater power required is 100-120W at a current of approximately 6A. While the laboratory HCT reliability is satisfactory, several changes are made in the flight unit under preparation as described in section IV to increase robustness, reliability and survivability of the launch environment.

![Laboratory HCT model overview](image1)

![HCT on the thrust stand](image2)

Figure 1. HCT and Thrust Stand

B. Thrust Stand and Thrust Measurement Procedure

An inverted pendulum thrust stand has been developed in-house to perform direct thrust measurements of the HCT and other thrusters under development in the SSC (Fig.1b). Thrust is measured by comparing the deflection caused by thrust with the deflection caused by the in-situ application of a known calibration mass. In this way the ‘null’ position of the thrust stand does not need to be known accurately as thrust is determined by change in deflection between ‘thruster on’ and ‘thruster off’ and the application of a calibration factor that is measured before every set of thrust measurements. The mechanical design of the thrust stand consists of a moving table that supports
the thruster and flexures connected to supporting struts. The rigidity of the flexures dictates the amount of deflection of the thrust stand and can be varied depending on the mass of the thruster undergoing characterization. Electrical harnessing and propellant feed-line to the thruster are formed into flexible coils to minimize damping and ‘stick’ effects on the deflection measurement. Deflection is measured by a laser optical displacement sensor reflected on a rigid strut attached to the moving table of the thrust stand. The laser sensor has an internal ADC (analogue to digital converter) and RS422 interface. Calibration is done using a 9.5g mass suspended from the thrust stand and displaced by a remotely driven stepper motor. The stepper motor has a resolution of 200 steps per revolution, and operates in 1/8 step mode at a frequency of 100 Hz. Deflection of the calibration mass from zero-applied position (vertical) is determined by the number of steps the stepper motor is driven, and the applied force to the thrust stand is determined via triangulation. By altering the number of steps, the calibration force can be adjusted. The calibration force used for the HCT is set to 1.66mN.

The procedure for measuring thrust starts with the collection of, typically, 10 calibration datasets for the setup under test. Each calibration dataset is 200s long with 2, equally spaced in time, application-retractions of the calibration mass by use of the stepper motor. The calibration factor is determined for each set and the average of all the sets is calculated. The standard deviation has been found to be <2% of the average calibration factor value for the thrust measurements of this investigation.

Thrust from the HCT is measured in two ways:
- By cycling the discharge off-on in 10s intervals while keeping propellant flow on, and the heater on at high power (70W), to allow the discharge to turn back on. Hot gas thrust is measured separately and added to the discharge thrust contribution.
- By turning off the gas from the mass flow controller total thrust from the HCT is measured. The voltage limiter of the discharge power supply is set to value close to the discharge voltage to prevent a high power discharge while the gas flow decays.

C. Retarding Field Analyser (RFEA)

A RFEA has been used to measure ion energy distribution functions and determine plasma potential and secondary ion population in the plume of the HCT. The RFEA is placed at a fixed position 25cm downstream the HCT on the central axis. The RFEA used consists of 4 grids (ground, repeller, discriminator, suppressor and collector plate) separated by insulation. The discriminator voltage sweeps 0-90V, repeller is held at -90V, suppressor at -9V and collector at -18V with respect to ground. The technique is described extensively in the literature\cite{10,12}.

III. Experimental Results

Figure 2 (a-c) shows representative ion energy distribution functions (IEDFs) for krypton discharge. Discharge current is 3.2A between 44W and 54W depending on the flow rate. Keeper bore is 4mm, orifice 1mm and keeper-orifice distance 1mm for these tests. Data in Fig.2 are for flow rates 25sccm to 15sccm in 5sccm increments. The magnitude of the secondary ion population observed in all cases, increases with the reduction of flow rate and increase of discharge voltage. The difference between plasma potential and the secondary ion population also increases, with the trend being more noticeable between 20sccm and 15sccm (Fig. 2 (d)). The high standard uncertainty of the data obtained for 25sccm is attributed to a low frequency voltage oscillation often observed in the discharge at high flow rates, which does not occur at lower flow rates.

Shown in Fig.3 are thrust measurements of the discharge contribution to thrust at a krypton flow rate of 25sccm for increasing discharge current and same HCT geometry as with RFEA results of Fig.2. Testing is done by cycling the discharge off-on with the heater at 70W and propellant flow constantly on. Each data point is the average of 10 measurements and with the standard uncertainty of the measurements representing error bars. Thrust increase over hot gas levels is in the range of 0.05mN which is small and close to the sensitivity of the thrust stand and laser sensor used for this investigation. Even though the uncertainty is large compared to the thrust measured, there is thrust consistently detected albeit low.
Thrust measurements for xenon (Fig.4) under the same testing conditions (heater input, geometry and 25sccm flow) as with krypton (Fig.3), indicate an order of magnitude higher contribution to thrust from discharge with thrust increasing with increasing discharge power. The total thrust of the HCT for this case is represented by the circles in Fig. 5(a) and is determined by summing the thrust contribution of discharge alone (Fig. 4) and hot gas thrust, which is measured separately. For higher discharge power total thrust is measured by turning off the propellant from the mass flow controller and allowing thrust to decay. For the cases of 3.2A with low heater power (30W) and heater off (squares of Fig.5a), total thrust is considerably higher than when the heater is set at higher power levels (circles of Fig.5a), indicating that heater input doesn’t increase thrust produced by the HCT. At 20sccm xenon, with the same geometry configuration and heater input (30W and heater off), total thrust levels are similar to those measured at 25sccm. As a result specific impulse at 20sccm is higher (crosses, Fig.5).

**Figure 2. Retarding Field Energy Analyzer Experimental Results for Krypton Discharge and 20cm RFEA Distance from HCT Keeper.**
Increasing distance between keeper and orifice plate at 3mm (triangles Fig.5) at 3.2A, for low (30W) and no heater, input produce noticeably lower thrust and specific impulse than for the 1 mm keeper-orifice distance.

Propulsive performance, for the investigated conditions, is shown in Fig.5 with maximum thrust of 1.6±0.1mN and specific impulse of 85±5s at 20sccm xenon and 53W discharge power.

Figure 3. Thrust contribution of 25sccm krypton discharge for 2.0A, 2.4A, 2.8A, 3.2A discharge current. Heater input 70W.

Figure 4. Thrust contribution of 25sccm xenon discharge for 2.0A, 2.4A, 2.8A, 3.2A current drawn. Heater input 70W.

(a) Total thrust of the HCT.
(b) Specific impulse of the HCT.
Circles: Discharge thrust contribution of Fig.4 summed with hot gas contribution measured separately (1mm orifice, 4mm keeper bore, 1mm keeper-orifice distance).
Squares: Total thrust measured at once for 3.2A discharge current with heater at 30W and heater off (1mm orifice, 4mm keeper bore, 1mm keeper-orifice distance).
Crosses: Total thrust measured at once for 3.2A discharge current with heater at 30W and heater off (1mm orifice, 4mm keeper bore, 3mm keeper-orifice distance).

Figure 5. Propulsive performance of the HCT operating with xenon as a function of discharge power

IV. Flight Implementation of the HCT

A. Overview
An HCT module is under development by the SSC and SSTL for a flight opportunity on TechDemoSat-1 (TDS-1) as an experimental enhancement to the S/C platform. TDS-1 is UK Space Agency funded, and SSTL built S/C based on the SSTL-150 platform. The S/C has a propulsion module with a total of 1.5kg of xenon to be shared between the enhanced resistojet and the HCT. The HCT module comprises of the Hollow Cathode Thruster
(HCT), Flow Control Unit (FCU), and Power Processing Unit (PPU), developed to require minimal changes to the platform. The baseline configuration of the HCT is the 1mm orifice, 4mm keeper bore and 1mm keeper to orifice distance, without external anode, operating at 20sccm of xenon.

The thrust axis of the HCT does not have to pass through the centre of gravity (CoG) of the S/C and thus produces a small disturbance torque that will be offset by the reaction wheels of the S/C. This is because energy allocation for the HCT is maximum 25Wh per orbit, thus the HCT can be fired for no more than 20 minutes every two orbits. This firing duration may not be enough to determine thrust produced accurately solely based on the orbital altitude change. The rotation speed of the wheels during the operation of the HCT will aid in determining the thrust produced at increased accuracy.

Propellant mass flow will be derived based on the average plenum pressure and the calibrated flow restrictor of the FCU thus allowing for calculation of specific impulse.

B. HCT Flight Unit

HCT flight unit is based on the laboratory HCT, with changes to increase robustness and survivability in the launch environment and increase heater lifetime and efficiency (Fig.6). The single spiral heater of the laboratory HCT has been replaced by two double spiral heaters (main and redundant). A thicker 0.5mm tantalum wire is used for increased heater robustness at the expense of increased current requirement to reach temperature. Thus, resistive loses in the PPU, switches and diodes are relatively increased. Hot to cold junctions are formed by swaging the wire in 0.5mm ID, 0.8mm OD tantalum tubes. The resulting conductor is 0.8mm OD reducing heat dissipation on the part of the heater wire outside the ceramic heater body. Heater leads for power return are terminated on the cathode tube, heater live are connected to insulated interfaces inside the HCT and bolts serve as leads to outside the HCT. Heater body is BN AX05 as in the laboratory HCT, keeper and cathode tube are tantalum, and the ceramic insulations on the cooler parts of the HCT are from BN Grade M26. A radiation shield made out of 0.025mm tantalum foil will encapsulate the heater reducing the power requirement to the heater to start the discharge.

C. Flow Control Unit

The FCU is located downstream the low pressure plenum, which operates nominally at 1bar. A 1/8 inch porous stainless steel flow restrictor calibrated for 20sccm is used (Fig.7b). The restrictor itself is press-fit in a custom manifold connected to the HCT with flexible 1/16 inch tubing. 1/8 inch COTS ceramic break downstream the restrictor manifold isolates the propellant feed-lines from the HCT. 1/8 to 1/16 inch adaptors are used at the HCT, ceramic break and restrictor outlet. Piping connections are weld or brazed except for the restrictor manifold outlet that is 37° flared. Solenoid valves (shown as 'V' in Fig.7a) are not controlled by the HCT experiment. An engineering model of the FCU is under preparation and will be used for testing with the qualification model (QM) of the HCT.
D. Power Processing Unit

A Power Processing Unit (PPU) for the HCT has been designed and is currently under development in the SSC. The PPU drives the discharge and the heaters of the HCT, and is capable of controlling output current of discharge and heater depending on the operating state of the HCT. The PPU is located adjacent to the propulsion module controller unit, and the module box size is 260x86x26.3mm (Fig. 8). The unit consists of two printed-circuit boards to accommodate communication/control (daughter board, top) and power converter electronics components (main board, bottom). PPU is connected to the 28VDC unregulated spacecraft power bus through a dedicated power switch in the power distribution module.

The PPU is designed to operate at 28±6VDC at a peak power output of 150W. Telemetry/telecommands between the unit and OBC are sent/received through CAN-BUS network. The thruster is connected by a D-SUB-15 pin connector that carries power to the HCT keeper (discharge), redundant heaters, common power return path, and a spacecraft chassis ground.

1. Description of the PPU and functionality

The PPU consists of two isolated switching converters for High Voltage (HV) and Low Voltage (LV), a non-isolated buck-converter which is used as the Heater Power Supply (HPS) converter, and a Siemens C515c micro-controller (Fig. 9). Power consumption of the PPU during standby is typically 1.7W. In this unit, each converter uses a different type of COTS PWM controller to evaluate their performance in-orbit. The architecture of the isolated switching converters is inherited from a Capacitor Charge Regulator that is also developed in the SSC. The LV converter supplies power not only to the discharge, but also to the HPS converter. The output direction of the LV converter is set by switching SW1 in Fig. 9, and enabling/disabling the HPS converter. The auxiliary power converter is a COTS isolated type switching converter and used for powering the micro-processor, sensors, and all PWM (Pulse Width Modulation) controllers of the converters thus the input and the output of the PPU are isolated. These isolations are required for the HCT to be compatible with the spacecraft grounding scheme and also allow the HCT to operate at floating potential with respect to spacecraft ground level (connected to the spacecraft ground through a resistor).

The topology of the HV converter is a forward type switching converter –using single Nch MOS-FET, outputs a voltage of 150V and the maximum output current of 0.5A as current source. This converter is specifically used to initiate the discharge of the HCT. Use of this HV converter at steady state discharge (<25V) is not efficient. Therefore the other converter (LV) is used to supply high discharge current with high efficiency. A push-pull topology current mode switching converter is used for LV – using dual Nch MOS-FETs. The LV converter outputs 30V and a maximum output current of 5A. Both HV and LV converters have Constant Voltage (CV) feedback circuits and Constant Current (CC) feedback circuits. CC mode is used for steady state discharge with the current limit of LV converter adjusted from the micro-controller for throttling.

Main or redundant HCT heater is selected by solid state switches (SW2 and SW3 in Fig.9). The HPS converter outputs current from 2.5A (early heating stage) to 15A (final heating stage). In order to improve the efficiency at high heater current, synchronous rectification is used. This converter is also able to control output current from the micro-controller. The micro-controller also determines the approximate heater temperature by calculating its resistance from the output voltage and the output current of the HPS. This temperature information is used for automatic thruster control. For example, when the heater temperature reaches pre-set temperature, the micro-controller sends a command to enable HV converter.

Once the discharge becomes stable on the HV converter, the micro-controller turns off the HPS converter, sets a current limit for the LV converter (throttle level), and closes a switch (SW1 Fig.9) in order to power the discharge (keeper). The discharge now runs of both HV and LV converters in parallel. When the discharge is stable on the LV converter (i.e. CC mode at set throttle level), HV converter is switched off by the micro-controller.
2. Prototyping and Testing of the PPU with the laboratory HCT

A prototype of the HV (Fig. 10) and LV (Fig. 11) converters of the PPU has been manufactured early in the development process. In order to gain confidence in the PPU design, the prototype of HV converter has been tested with the laboratory HCT. In this test, Krypton was used as propellant. The output voltage and current signal from the HV converter were acquired by an ADC on a PIC micro-controller. Moreover, this test aimed to check any noise related problems (i.e. disruptions and reboots of the micro-controller) during HCT operation along with validating the grounding method of the PPU. Data were logged on a laptop computer through RS-232 serial connection.

The HV converter prototype was connected in parallel with a lab bench power supply (CV: 30V, CC: 3A) to allow the heater to switch off after start-up, as the current limit of 0.5A from the HV converter is not sufficient to sustain the discharge without heater operation. Only the HV converter was used for the start-up of the HCT, and once the HV converter stabilized at 0.5A (CC limit), the lab power supply was activated manually, followed by deactivation of the heater and the HV converter. This process was repeated several times at different gas flow rates (20±5 sccm). In all cases, the HCT started successfully. In addition, there were neither recorded disruptions in the data logging nor rebooting of the micro-controller. Fig.12 shows output voltage and current transients during a HCT start-up along with the stabilization of the discharge. Fig.13 is a picture of the laboratory HCT operating during this test.

The engineering model of HV, LV, HPS converters are in preparation with tests planned first with the lab HCT followed by testing with the QM of the HCT.
V. Conclusion

A prototype HCT has been through initial thrust characterization at 20 and 25sccm xenon and discharge power <55W, yielding a maximum thrust of 1.6mN and 85s specific impulse. Keeper to orifice plate distance seems to affect propulsive performance, with measured values of thrust being considerably lower when distance was increased from 1mm to 3mm. Plume RFEA measurements 25cm downstream the keeper while operating with krypton indicate the presence of a higher energy secondary ion population, but very small increase in thrust is measured under these conditions when the discharge is turned on (0.05mN). Operation with xenon yields higher thrust increase (0.2mN) with the discharge on, but no RFEA data are available yet to compare between the two gases.

A flight worthy unit with the baseline performance of the prototype HCT is developed as an experimental enhancement to the TDS-1 S/C platform with improvements in the robustness and heater reliability of the unit. Along with the thruster a PPU able to operate the HCT from the 28VDC unregulated bus is developed and a prototype of the discharge initiation HV converter has been tested successfully with the prototype HCT. The engineering models of the PPU converters are currently under preparation in the SSC and will be fully tested with the qualification unit of the HCT.

Testing, both thrust measurements and plasma plume diagnostics, will be extended at lower flow rates/higher power regime, with potential performance gains. Robustness of the PPU and S/C impact of the HCT operating at regimes that intense electromagnetic noise is created will also be investigated.

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References


