AIAA 94-3012

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30th AIAA/ASME/SAE/ASEE Joint Propulsion Conference
June 27-29, 1994 / Indianapolis, IN

Preliminary Characterization of a Low Power End-Hall Thruster

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June 28, 1994

Abstract

Electron temperature and number density were measured near the exit of a 500 Watt end-Hall thruster using argon as propellant. Because of the low magnetic field strengths measured in the discharge chamber, a number of modifications were made to the original design, including fabricating soft-iron pole pieces to channel the magnetic field from the solenoid. These changes resulted in a three-fold increase in magnetic field strength. Centerline values of n_e and T_e taken ~1 inch from the exit plane of the modified engine were approximately 4×10^{10} cm^{-3} and 2 eV, respectively, at a discharge current of 5 A, a voltage of ~90 V, and total mass flow rate of 1.7 mg/s. Because the radial magnetic field strength of the modified device never exceeded 80 G, the discharge voltage of the engine was much too low to yield significant thruster performance. Emission spectra of the plume showed no evidence of doubly-ionized argon. Most of the prominent peaks in the spectra are ArII lines. All measurements were performed in a 30-ft-long by 20-ft-diam. vacuum chamber that is pumped by six 32-in-diam. diffusion pumps. Chamber pressure was maintained to 3.7×10^{-5} Torr during thruster operation.

Nomenclature

B, B_o	= magnetic field strength, T
e	= elementary charge, 1.6×10^{-19} C
k	= Boltzmann's constant, 1.38×10^{-23} J/K
n_e	= electron number density, cm^{-3}
T_e	= electron temperature, eV
ΔV_p	= variation in plasma potential, V
$\sigma_{\parallel}$	= parallel electrical conductivity, mho/m
$\sigma_{\perp}$	= perpendicular electrical conductivity, mho/m

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Introduction

Because of its high thrust efficiency at exhaust velocities that are optimum for North-South station-keeping, a great deal of interest has emerged in using Hall thrusters onboard American spacecraft. With performance potentially superior to that of arcjets, the closed-drift Hall thruster would not only serve as an excellent device for satellite station-keeping, but potentially could be used for orbit raising and satellite repositioning.

Although capable of excellent performance, the closed-drift Hall thruster is plagued by excessive insulator wall erosion and plume divergence. American spacecraft designers are reluctant to use these devices for fear that ablated insulator material may coat sensitive spacecraft components and that high energy ions from the highly divergent exhaust plume may damage the spacecraft.

To address these issues, a program was recently initiated at the University of Michigan to study Hall thruster plumes, the goal of which is to develop means of integrating these devices with communications satellites. As a start, electron temperatures and number densities were obtained near the exit plane of a 500 W end-Hall thruster via single Langmuir probe and emission spectroscopy. This effort is the precursor to a more extensive program to study spacecraft integration issues of Hall accelerators.

Background

The Hall thruster is an electrostatic engine that was developed in the 1960's to alleviate the thrust density limitation of ion engines that result from space-charge effects within the acceleration volume. Hall thrusters were also attractive from the standpoint that since grids are not required to accelerate ions, they do not suffer from the large grid erosion rates which were characteristic of the ion engines at the time. Interest in the Hall thruster waned in the early 1970's, however, because of budgetary cuts and because American researchers were never able to demonstrate that Hall thrusters could operate at thrust efficiencies near those achieved with ion engines[1, 2, 3].

Throughout this period, research on Hall thrusters in the Soviet Union flourished. Hall thrusters were first tested in space in 1971 with immediate success[4, 5]. Since then, over sixty Hall thrusters have been used on spacecraft, primarily for East-West station-keeping or for auxiliary propulsion. In January of 1994, the first SPT-100 was flown in space, on the Russian GALS spacecraft as the primary source of North-South station-keeping propulsion. This event marks the first time that a Hall thruster was used as a major source of spacecraft propulsion.

Recently, because of tests conducted at the NASA Lewis Research Center (LeRC) and the Jet Propulsion Laboratory (JPL) which showed that the SPT-100 is capable of operating at specific impulses in excess of 1500 sec at thrust efficiencies of approximately 50%, there has been a great resurgence of interest in using Hall thrusters on Western spacecraft[6, 7]. Clearly such a device, with performance superior to that of arcjets, would not only serve as an excellent thruster for North-South station-keeping, but potentially could be scaled in power for orbit raising and for interplanetary missions. However, before these devices gain widespread acceptance outside of Russia, a number of spacecraft integration issues have to be resolved.

Experimental Apparatus

The end-Hall Thruster

There are two types of Hall thrusters that have been studied at great lengths; the end-Hall thruster and the closed-drift thruster. Variants of the latter include the stationary plasma thruster (SPT) and the anode layer thruster (a good review of closed-drift Hall thruster technology is given in Ref. [1]).

The end-Hall thruster is a gridless electrostatic device in which the propellant is first ionized in a discharge chamber and then accelerated by an axial electric field produced by the interaction of the discharge current with an applied diverging axial/radial magnetic field. This field is supplied either through permanent magnets or by a solenoid. These thrusters are typically between ten and twenty centimeters in diameter and operate at power levels of a few kilowatts or less. End-Hall thrusters are also used as an ion source for plasma processing[8]. Their primary applications in this field include substrate cleaning and conditioning, reactive etching, and substrate stress control [8].

The applied magnetic field strength of these devices is usually set so that the electron conductivity parallel to the magnetic field lines is much greater than the transverse electron conductivity[9];

$$\sigma_{\parallel} \gg \sigma_{\perp}. \tag{1}$$

This typically corresponds to an applied field strength of a few hundred Gauss.

In accordance to Eqn. 1, one would expect equipotential lines to be parallel with magnetic field lines. However, numerous experiments have found that substantial voltage gradients exist parallel with the magnetic field in approximate agreement with the following expression:

$$\Delta V_p = \frac{kT_e}{e} \ln(\frac{B}{B_o}) \tag{2}$$

where B/B_o reflects the variation in magnetic field strength over the extent at which the change in plasma potential is ΔV_p[9]. It is this voltage drop, calculated over the entire volume of the discharge chamber, that is responsible for the acceleration of the ionized propellant. Typical discharge voltages are on the order of a few hundred volts.

Both the end-Hall and closed-drift thruster, in principle, are capable of producing specific impulses of over 1500 seconds with xenon at thrust efficiencies in excess of 50%. However, results of recent experiments suggest that the ultimate performance of the end-Hall thruster may be inferior to that of the closed-drift thruster[9].

Fig. 1 shows a cross sectional view of the 0.5 kW-class end-Hall that was supplied by the NASA LeRC for this study. The engine is composed primarily of four components: 1) the cathode (not shown on figure); 2) the anode; 3) the discharge chamber; and 4) the magnet (solenoid).

The cathode used for this study is the standard hollow cathode used in xenon ion thrusters at NASA LeRC[10]. The cathode consists of a 0.25-inch-diam. by 4-inch-long molybdenum-rhenium tube. A thoriated-tungsten endplate with a 60-mil-diam. orifice is electron-beam welded to one end. The other end of the cathode contains fittings for propellant delivery. The cathode insert is a sintered tungsten cylinder impregnated with 4BaO-CaO-Al_2O_3. The cathode is situated on the "12-o'clock" position above the discharge chamber, pointing approximately 45 degrees toward the center with its orifice some 0.75 inches downstream of the discharge chamber exit (2.75 in downstream of the anode).

To allow electron emission at low cathode surface temperatures and low erosion rates, the cathode insert must be activated. The heater used for activation is the standard helical-wound sheathed tantalum heater that is used to activate ion engine cathodes and plasma contactors. The coil is enclosed in multiple layers of tantalum foil

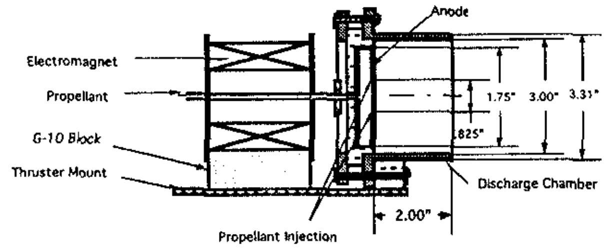

Figure 1: Schematic diagram of the 500 W end-Hall thruster.

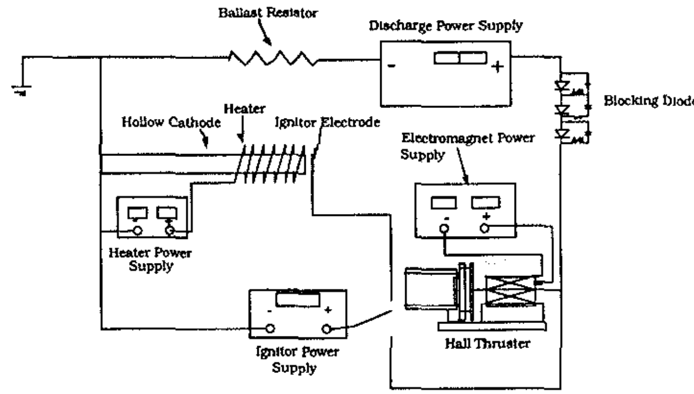

Figure 2: Power supply circuit of the end-Hall thruster.

to help prevent the cathode from radiating heat into space. A thermocouple is attached to the cathode to allow the temperature to be monitored during activation. Because this cathode had not been used in over a year, and was stored in a cabinet under atmospheric pressure for much of that time, it was first activated and started in an axisymmetric arc configuration in a rough-pumped 3-ft-diam. by 6-ft-long chamber. A planar anode was placed 0.75 inches from the cathode orifice. After ignition, the cathode maintained a current of 7.5 Amps with a discharge voltage of ~20 volts while flowing 35 sccm of argon through it at a tank pressure of 40 mTorr.

There was concern that back-streaming mechanical pump oil might contaminate the cathode, however, spectroscopic analysis of the anode surface with a scanning electron microscope (SEM) found no traces of the hydrocarbon pump oil anywhere on the anode. The cathode was installed on the end-Hall thruster once it could be reliably started in the configuration above.

The anode used for the thruster served as the primary propellant delivery system for the discharge. The anode is a 1.75-inch-diam molybdenum disc recessed within the 2-inch-long by 3-inch-diam. (i.d.) molybdenum discharge chamber (cf. Fig. 1). The anode contains twelve propellant injection holes, each located 0.825 inches radially from the anode centerline. The discharge chamber rests on a Macor block that is bolted to an aluminum thruster mount. The discharge chamber is insulated from the anode with boron-nitride.

The magnetic field is supplied by a solenoid located 0.25 inches behind the discharge chamber. The solenoid rests on a G-10 block that is bolted to the aluminum mount and consists of approximately 60 turns of #6 gage solid copper wire wrapped around a soft-iron spool. At a solenoid current of 10 A, the maximum radial magnetic field strength measured in the discharge chamber was approximately 12 G. The the maximum magnitude of the magnetic field within the chamber is estimated to be 25 G. Thus, modifications were made to the original design to increase field strength. These included surrounding the discharge chamber with a soft-iron ring, and using iron bars to more efficiently conduct the magnetic field from the solenoid to the ring. This modification increased the maximum field strength within the chamber to 75 G (at 10 A). Although this value is still too low to provide an effective means of ion acceleration, it is of sufficient strength to magnetize the electrons. Details of the design modifications are given in the next section.

Fig. 2 shows a schematic of the power supply circuit used to operate the thruster. Four power supplies were used for the thruster; one for the discharge, the cathode heater, the magnet, and the ignitor/keeper electrode. Hall thruster main discharge power is provided by a 10 kW Sorenson DCR 600-16T power supply that is capable of providing 16 Amps of current at 600 V. This power supply is connected in series with the thruster through a variable ballast resistor. Magnet and cathode heater power are provided by Kikisui voltage and current regulated DC power supplies. The ignitor supply was custom built for this application and is based on a design used at NASA LeRC to ignite ion thrusters. The ignitor can provide 2000 V DC at a maximum current of 200 mA. Although the negative lead of the ignitor is always connected to the cathode, the positive lead can be connected either to the anode or the ignitor electrode (cf. Fig. 2). A series of high current (12 A) 1000 V blocking diodes is placed along the discharge power supply line to protect the Sorenson from high voltage when the ignitor is used at the anode. However, since the thruster was found to ignite more readily when the ignitor electrode is used, only the ignitor electrode was used in starting the Hall thruster. Usually an ignitor voltage of 300-500 V was required to ignite the engine.

Hall thruster voltage is measured with Tektronix P6007 100:1 voltage probes clamped to the electrode leads. The voltage probe signals are collected and registered by a Tektronix AM501 operation amplifier. Digital multimeters are used to measure the discharge voltage as well. Hall thruster current is monitored with a Tektronix A6303 current sensor powered by a Tektronix AM503 current probe amplifier. Hall thruster voltage and current data are monitored and stored by the computerized data acquisition system described below.

For this study, the Hall thruster was operated on argon at a total mass flow rate of 5 mg/s or less. Propellant is supplied to the Hall thruster from compressed gas bottles through stainless-steel feed lines. Propellant line pressures are maintained to above atmospheric pressure throughout the delivery system to minimize the likelihood of cathode contamination through propellant line leaks. Cathode and discharge propellant flow are controlled and monitored separately with MKS 1159B mass flow controllers. The flow controllers are periodically calibrated with a specially constructed calibration rig that measures gas pressure and temperature changes with time in an evacuated chamber of known volume. The ideal gas equation of state is then employed to estimate the change in mass within the volume as a function of time. The propellant split is typically four-to-one, anode-to-cathode.

Thruster operation is visually monitored and stored with a Sony camcorder connected to a JVC Super VHS cassette recorder. A filter wheel is placed in front of the video camera to allow band-pass filtered images of the plume to be stored on film. Video frames are downloaded to and stored on a Macintosh computer for image processing and analysis through the use of a Radius frame grabbing system.

Test Facilities

All experiments reported were performed in a 30-ft-long by 20-ft-diam. stainless-steel vacuum chamber (cf. Fig. 3). The facility is supported by six 32-inch-diam. diffusion pumps rated at 32,000 l/s each (with water-cooled coldtraps), backed by two 2,000 cfm blowers, and four 400 cfm mechanical pumps. These pumps give the facility an overall pumping speed for argon of over 100,000 l/s at 10^{-5} Torr. In addition, a Polycold PFC-1100 closed-loop water cryopump/cold-trap has been installed above two of the diffusion pumps. This unit doubles the water pumping speed of the facility, thus, greatly reducing the pump-down time needed for the chamber. The chamber can be evacuated from atmospheric pressure to 3×10^{-5} Torr in less than four hours.

Chamber pressure is measured with MKS model 919 hot-cathode ionization gauges located on vacuum ports on either side of the chamber, and by a MKS Type 317HA baratron capacitance manometer located in the center of the chamber (cf. Fig. 3). Background chamber pressure, as determined by the ionization gauges, is maintained at 0.005 Pa (4×10^{-5} Torr) or less during Hall thruster operation. Although this pressure is higher than what has been shown to be desirable for making Hall thruster plume measurements[11], the ultimate test

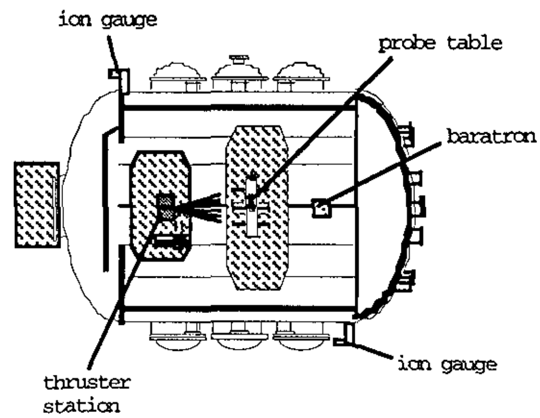

Figure 3: Vacuum facility.

pressure for these conditions is expected to drop to below 3×10^{-5} Torr in the near future once minor modifications are made to facility.

Data acquisition is accomplished through a Macintosh based system developed by National Instruments (LabVIEW). The system includes four SCXI-1120 8-channel isolation amplifiers that are used in conjunction with a NB-MIO-16XH-18 data acquisition board. This configuration provides thirty-two isolated differential input channels. The data acquisition system monitors thruster, diagnostics, and tank operations and is controlled by LABVIEW II software.

The thruster is mounted to a NASA LeRC designed inverted pendulum thrust stand[12] that sits on a movable platform near one end of the chamber (cf. Fig. 3). The thruster is mounted to the thrust stand through a Macor slab. In-situ thrust stand calibration is performed by loading and off-loading small weights to simulate thrust via a remotely-controlled stepper motor driven pulley system. A linear curve-fit of thrust stand deflection vs. thrust is then obtained and used for performance measurements. Soon after the thruster is turned off, a post-test calibration is performed. The springs of the stand were made with extra stiffness to minimize thruster deflection, thus allowing plume measurements to be made in parallel with thrust measurements.

Thrust data based on the predetermined curve fit is displayed and simultaneously stored by the data acquisition system: The data acquisition system also reads cathode and anode mass flow rates, current, and voltage, enabling it to calculate thrust efficiency and specific impulse in real-time.

Because the facility does not yet have water handling capability, the entire thrust stand is enclosed in a stainless-steel shell which will soon be replaced with a water-cooled copper shell. Currently, pre and post-run calibration curve fits, which usually differ only in intercept, are used to account for thermal drifts.

Plume diagnostics are performed through the use of a state-of-the-art probe positioning system developed by New England Affiliated Technologies (NEAT). The table contains two rotary platforms on a linear stage with 5 ft of travel in the radial direction on a 3 ft travel axial stage. The rotary actuators allow rotation of the Langmuir probes to minimize measurement errors caused by probe misalignment with the flow[13], and for characterizing the local flow field. The system allows for sweeps of two probe rakes at a time at radial speeds in excess of 2 ft/s with an absolute position accuracy of 10 mils. The system is driven by a Macintosh computer running LABVIEW II software. Like the thruster station, the entire probe positioning system is mounted on a movable platform to allow for measurements to be made throughout the chamber (cf. Fig. 3).

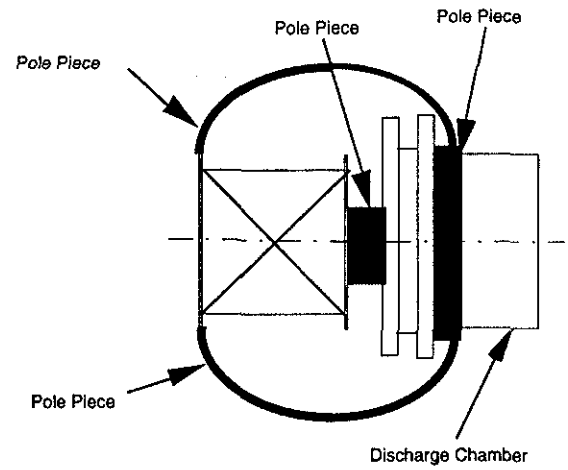

Figure 4: Schematic of the modified end-Hall thruster.

Thruster Modifications

Hall thrusters typically operate with magnetic field strengths between 200 and a 1000 Gauss and discharge voltages of a few hundred volts (for a few Amperes of current). Since the maximum field strength in the discharge chamber of the original thruster is predicted to be less than 100 G for the maximum current the solenoid can tolerate (30 A), the likelihood of achieving acceptable performance with this device is nil.

Initial end-Hall thruster runs verified that the magnetic field strength was far too low in the discharge chamber to provide effective thrust augmentation. Increasing the solenoid current from 1 to 10 A had little effect on the discharge voltage. At a discharge current of 5 A, and a total argon mass flow rate of 2.4 mg/s with a 25% cathode flow fraction (i.e., 0.6 mg through the cathode and 1.8 mg/s through the anode), the discharge voltage remained below 55 volts. Considering the losses incurred from ionization alone (i.e., tens of eV per ion), this voltage is far too low to yield acceptable thruster performance.

It was decided that a modification was needed to increase the field strength in the discharge chamber. Before such a modification could be made, however, the reasons why the initial design failed had to be determined.

The possible reasons why a solenoid can produce low magnetic field strengths in a discharge chamber are numerous. In general, low field strengths are caused by low solenoid current, poor solenoid positioning, and improper utilization of pole pieces or multiple solenoids for field enhancement and/or shaping.

The original thruster had one solenoid placed 0.25 inches behind the discharge chamber. In this configuration, much of the magnetic field from the solenoid simply emanates in the vacuum between the spool and the anode. Our approach in rectifying this situation was twofold: 1) move the solenoid as close to the discharge chamber as possible, and 2) add pole pieces to channel as much of solenoid's magnetic field across the remaining gap to the discharge chamber as possible.

A schematic of the modified device is shown in Fig. 4. The solenoid was moved slightly closer to the anode, to within ~0.20 inches of the discharge chamber (the closest possible with this design). Highly permeable pieces of soft-iron were machined into pole pieces to conduct the magnetic field to the discharge chamber. An iron tube was placed between the anode and the electromagnet, and an iron ring was slipped around the discharge chamber such that the rear portion of the ring is aligned with the anode surface. To complete the magnetic circuit, two iron loops were fastened to the ring pole piece and the rear pole of the electromagnet. This design places iron along most of the path that the most intense portion of the solenoid magnetic field would naturally propagate through. The portion of the circuit that is open corresponds to the anode region of the discharge.

Fig. 5 shows a comparison of the radial magnetic field profile of the original thruster compared to that after the modifications were made. Small radial and axial Hall probes were used to map the magnetic field throughout the discharge chamber. The position of the Hall element with respect to the sensor tip dictated how close to a boundary measurements could be made. In general, the modifications resulted in a three-fold increase in radial magnetic field strength for a given solenoid current. With an operating current of 10 Amps, the magnetic field of the original design was observed to fall off to negligible values within 0.25 inches of the anode, whereas reasonable values, however small, were measured after the modifications were made.

Fig. 6 shows axial magnetic field profiles for the modi-

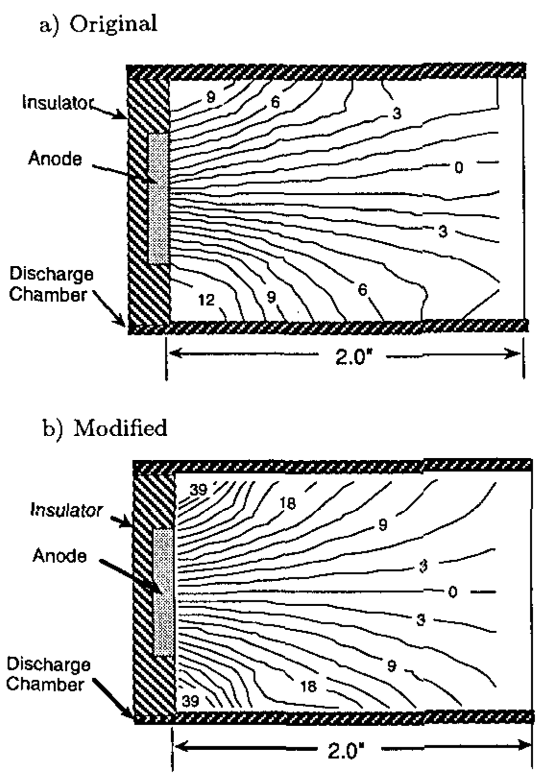

Figure 5: Solenoid radial magnetic field profile (all values are in Gauss).

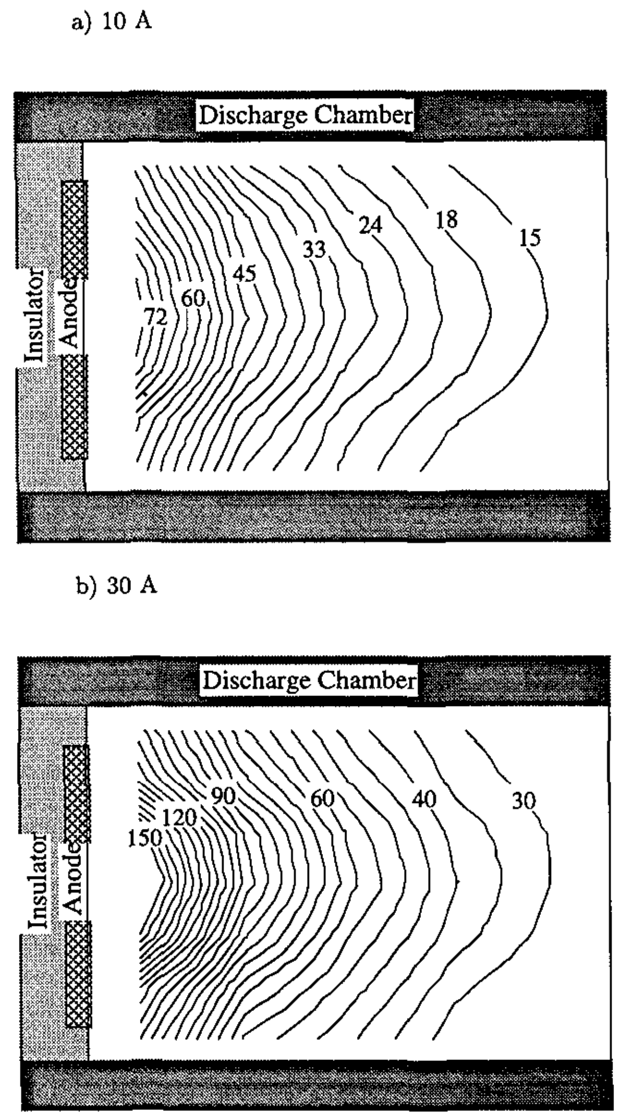

Figure 6: Axial magnetic field profile for the modified thruster (all values are in Gauss).

fied engine at 10 and 30 A solenoid currents. As expected, the axial component of the magnetic field is far greater than the radial strength. At 30 A, an axial field strength of 150 G was measured near the anode compared to a maximum radial strength of 80 G (cf. Fig. 7). Although an improvement over the original design, even at 30 A the modified magnetic circuit does not generate enough magnetic field. Unfortunately, time did not permit further modifications (e.g., use more turns of higher current capacity wire) to be made. These modifications will be made to the thruster over the next few months.

Results and Discussion

Fig. 8 shows a captured video image of the end-Hall thruster in operation both before and after the modification. Note that in part b), Langmuir probes can be seen probing the near-field plume of the modified thruster. As is evident from the images, the plume of the modified thruster appears brighter even though both thruster were operating at approximately the same mass flow rate, cathode flow fraction (25% of total), and discharge current. The difference is due to the fact that the solenoid current was set to 30 A in Fig. 8 b) and so the discharge

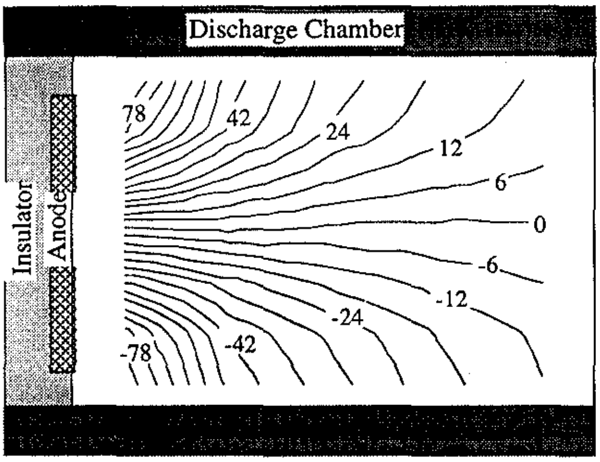

Figure 7: Radial magnetic field profile for the modified thruster with a solenoid current of 30 A (all values are in Gauss).

magnetic field strength was almost an order of magnitude higher than when Fig. 8 a) was taken. The discharge voltage for the thruster shown in part b) was almost twice as high as the original thruster at the same operating conditions (90 V vs. 55 V). As is expected, the discharge voltage was found to decrease with increasing mass flow rate.

There was concern of operating the solenoid at 30 Amps for extended durations of time for fear that the Kapton insulation would melt. A thermocouple was placed between two adjacent wires to ensure that the coil temperature remained below the 260°C limit of the tape. Unless otherwise noted, the discussion below pertains only to the modified thruster.

Performance

Attempts at making thrust measurements with this device were largely unsuccessful. A specific impulse of 190 sec was measure when the solenoid coil was set to ~2 A. Although "modest" in comparison to the SPT-100, the measured specific impulse is high enough to suggest that the argon heavy particle temperature is probably on the order of 1 eV, since most of this thrust is due to the random thermal expansion of the plasma from the chamber. Attempts at measuring thrust with higher solenoid currents failed (e.g., negative thrusts were measured), most likely because of insufficient shielding and/or thermal drifts. This thrust stand experienced similar problems while operating a 1 kW arcjet on pure hydrogen. The source of the problem was traced to a disconnected ground wire of the LVDT. Incorporating a water-cooled shell to the thrust stand should help determine if thruster noise is the cause for these problems.

a) Original

b) Modified

Figure 8: End-Hall thruster in operation (5 A and 1.7 mg/s of argon).

Langmuir Probe Measurements

A cylindrical single Langmuir probe, with a 9-mil-diam. by 0.75-inch-long tungsten wire collector electrode, was used to measure n_e and kT_e in the near-field thruster plume. The electrode of the probe is attached to the center conductor of a triaxial boom that is constructed of titanium with Teflon insulation. The boom is approximately 0.166 inches in diameter and 7 inches long.

The collector electrode of the probe was biased with respect to the chamber wall by a Kepco model BOP 100-2M programmable bi-polar power supply. A function generator was used to provide the 12.7 Hz triangular source waveform that was amplified to ±50 V by the bi-polar supply.

Probe current through the small probe was measured with a 100 Ohm shunt. Probe voltages measured with re-

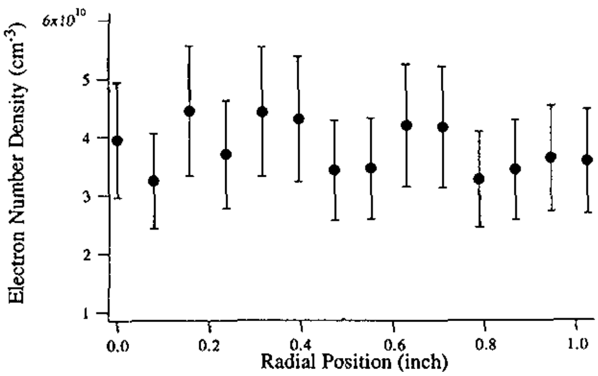

Figure 9: Electron number density profile 1.2 inches from thruster exit plane.

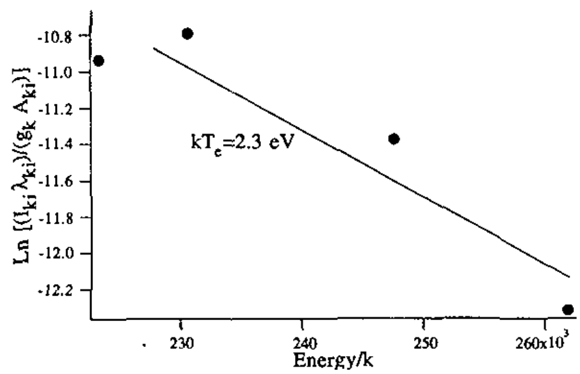

Figure 10: Boltzmann plot of excited states.

spect to tank ground were collected with voltage probes and operation amplifiers. Amplifier output signals were collected both by the data acquisition system for storage and later processing, and by a Tektronix 500 MHz digital oscilloscope for real-time processing. The data acquisition system stored 50 pairs of probe voltage-current data points per voltage ramp.

All near-field measurements were made with the tip of the Langmuir probe placed 1.2 inches downstream of the discharge chamber exit. For these measurements, the probe was quickly moved to the collection site, kept there long enough to collect ten ramps of data (~1 sec), and rapidly move out of the plume to allow for probe cooling. This approach also served as an effective means of cleaning the probe and could be used to probe within the discharge chamber.

Fig. 9 shows electron number density versus radius taken 1.2 inches from the thruster exit. The thruster was operating at a current of 5 A, a total mass flow rate of 1.7 mg/s (20% through the cathode), and a solenoid current of 30 A. Tank pressure was measured at 3.7×10^{-5} Torr during these measurements. Because of excessive signal noise on the Langmuir probe traces, realistic electron temperatures could not measured. Therefore, an electron temperature of 2 eV was used for determining the electron number density from the ion saturation current. The value of 2 eV is based spectroscopic data described below.

Over a 1 inch region, the electron number density remains relatively flat at a density of $\sim 4 \times 10^{10}$ cm^{-3} . The scatter in the data is a reflection of the periodic changes in thruster operating conditions that were observed.

Thruster current was maintained by operating the Sorenson power supply in constant current mode. Thus, discharge voltage is automatically adjusted to maintain the set current. Over the course of a few minutes, however, the discharge voltage would vary by as much as 20 volts (i.e., 85±20 V) before returning to its original value. The reason for this low frequency oscillation has not been determined, however, similar phenomena were seen in other Hall thruster experiments[6].

Emission Spectroscopy

Emission spectroscopy was used to estimate the electron temperature near the exit plane of the thruster. An achromatic lens was used to collect light 0.5 inches downstream of the thruster exit plane along the centerline. The light collected by the lens was focused onto a silica optical fiber which carried the light out of the vacuum chamber to the collection optics at the spectrometer. A Spex 500M spectrometer, fitted with an 1800 grooves/mm holographic grating blazed at 500 nm, was used to analyze the light collected at the entrance slit. Both the entrance slit and the exit slit were set at 100 microns to maximize the amount of light collected. A detailed description of the spectroscopic system can be found in Ref. [14].

Spectra were acquired in the 340-520 nm range in an attempt to obtain information from those excited states with energies near the ionization boundary. Such states are spaced closely together and are good candidates to be in equilibrium with the electrons. For these measurements, the thruster was operated at a current of 5 A, a total mass flow rate of 1.7 mg/s (20% through the cathode), and a solenoid current of 30 A.

The portion of the argon spectrum analyzed contained both singly ionized argon lines and neutral lines, however, the strongest signals obtained in this region of spectra came from ArII lines. To minimize errors associated with intensity measurements, only lines associated with singly ionized argon were used in the Boltzmann plot for calculating T_e. Although the lines acquired were on the order of 5 eV below the ionization potential of singly ionized argon, the lines were still somewhat closely spaced together

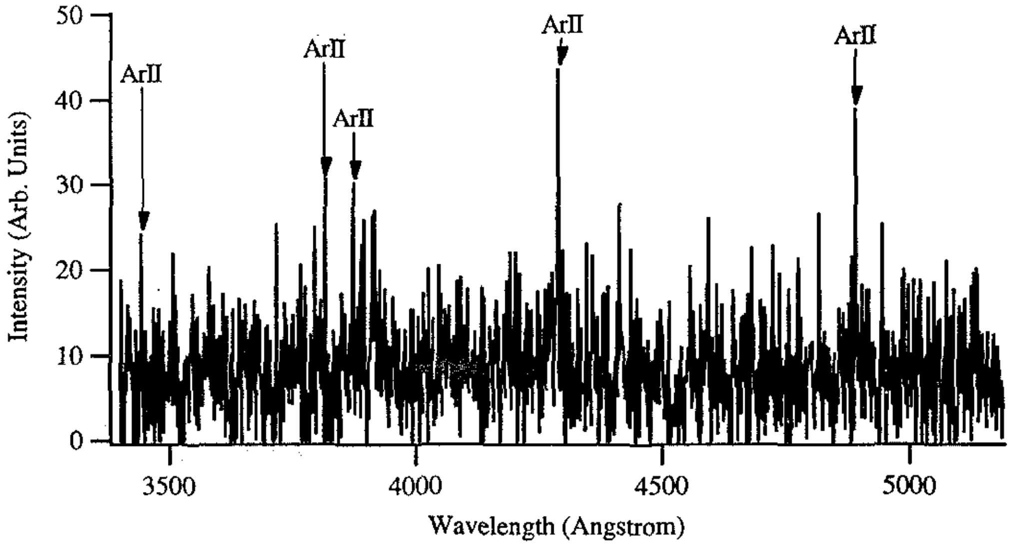

Figure 11: Measured emission spectra.

in energy.

Four lines - 381 nm, 428 nm, 440 nm, and 455 nm - were used for the Boltzmann plot of Fig. 10. The selected points from the spectra match the line-fit rather well. An electron temperature of is 2.3 eV is calculated from the slope. Because the intensity contributions come from a chord along the plume cross section, this temperature should be thought of as an average electron temperature across the centerline plume at this axial location.

The measured emission spectra from the engine are shown in Fig. 11. Labeled on the graph are prominent ArII lines. No strong lines for neutral or doubly ionized argon were observed.

Conclusions

Electron temperature and number density were measured near the exit of a low power end-Hall thruster. Because the magnetic field strength that the solenoid provided in the discharge chamber was low, a number of modifications were made to the original design. These included fabricating soft-iron pole pieces to channel the magnetic field from the solenoid, moving the solenoid closer to the discharge chamber, and increasing the operating solenoid current by monitoring its temperature and allowing it to cool off periodically. These changes resulted in a three-fold increase in magnetic field strength. Typical thruster operating conditions were 5 A discharge current, a voltage of $\sim$90 V, and total mass flow rate of 1.7 mg/s.

Centerline values of n_e and T_e were measured $\sim$1 inch from the exit plane of the modified engine by a Langmuir probe and emission spectroscopy, respectively. Measured values were 4×10^{10} cm^{-3} for n_e and 2 eV for T_e.

Because the radial magnetic field strength of the modified device never exceeded 80 G, the discharge voltage of the engine was much too low to yield significant thruster performance. Emission spectra of the plume showed no evidence of doubly-ionized argon: Most of the prominent peaks in the spectra are ArII lines.

Future work includes making more modifications to the thruster to improve its performance, taking additional Langmuir probe and emission spectroscopy measurements in the plume, and complete the construction of Hall probes to map the magnetic field within the discharge chamber. Future work will also include looking into facility effects, particularly the influence of tank pressure and background gas species on Hall thruster operation.

Contamination

The issue of testing Hall thrusters in a diffusion-pumped vacuum facility is always a source of concern. At issue is the effect deposition of diffusion pump oil on thruster components may have on thruster operation. Although most of the Hall thruster development in Russia was done using diffusion-pumped facilities, the availability of large oil-free vacuum environments at NASA LeRC, JPL, and FAKEL has cast a shadow of uncertainty onto the utility of performing Hall thruster tests with diffusion pumps.

An attempt will be made in the future to look into the effect that diffusion pump oil might have on Hall thruster performance and operation. SEM photographs will be taken of thruster components to determine if products from backstreaming diffusion pump oil are being deposited on thruster surfaces.

Acknowledgments

The authors would like to thank the technical support staff of the Aerospace Engineering Department for their help in maintaining the facilities, NASA LeRC for their technical assistance and instrumentation support, Mr. John Hamley (LeRC) for providing the schematics for the ignitor power supply, Mr. Arnaud Turlan for data processing, and Dr. Larry Brace of the University of Michigan's Space Physics Research Laboratory for use of his Langmuir probe. This work was supported through grants from NASA LeRC and BMDO (Dr. Frank Curran - contract monitor) and the University of Michigan Office of the Vice President for Research.

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