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Performance Evaluation of a 50 kW Hall Thruster

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December 1999

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PERFORMANCE EVALUATION OF A 50 KW HALL THRUSTER

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ABSTRACT

An experimental investigation was conducted on a laboratory model Hall thruster designed to operate at power levels up to 50 kW. During this investigation the engine's performance was characterized over a range of discharge currents from 10 to 36 A and a range of discharge voltages from 200 to 800 V. Operating on the Russian cathode a maximum thrust of 966 mN was measured at 35.6 A and 713.0 V. This corresponded to a specific impulse of 3325 s and an efficiency of 62%. The maximum power the engine was operated at was 25kW. Additional testing was conducted using a NASA cathode designed for higher current operation. During this testing, thrust over 1 N was measured at 40.2 A and 548.9 V. Several issues related to operation of Hall thrusters at these high powers were encountered.

Introduction

Beginning in 1990, NASA initiated an effort to evaluate Hall thrusters due to their unique combination of relatively high specific impulse and favorable thrust density. These efforts primarily focused on the Russian Stationary Plasma Thruster (SPT) and Thruster with Anode Layer (TAL). These engines represent technology developed to a relative state of maturity over several decades in the former Soviet Union. As an example, over 70 SPT's have been used operationally on Soviet and Russian spacecraft since 1971[1]. The anode layer thruster, which potentially offers higher performance than SPT type thrusters, was never flown on a Soviet or Russian spacecraft, but has now been demonstrated on-board a Western experimental spacecraft with its first successful firing on October 23, 1998[2].

In recent years with advances in space power systems[3] and the consideration of new orbital trajectories[4], government mission designers have begun reevaluating the potential of leveraging these technologies to reduce launch mass for various potential missions. These missions include a manned mission to Mars[4] and a space power beaming satellite[5]. In order to validate the engine technology needed for such missions at the high total powers being considered, NASA's Advanced Space Transportation Program has undertaken a demonstration of a high power Hall thruster capable of operation at power levels approaching 50 kW.

The objective of this demonstration was to assess the suitability of this approach and to gain insight into the relevant issues with operating high power Hall thrusters of this type. The performance of one high power engine was experimentally evaluated during this investigation. These results are included in this paper for a wide range of operating conditions. The problems encountered during these tests are also discussed.

Apparatus and Procedure

The Hall thruster used in this investigation, referred to as the TM-50, was a laboratory model anode layer thruster obtained from the Central Research Institute of Machine Building (TsNIIMASH). This engine was developed several years ago to evaluate the concept of applying Hall thruster technology at the tens of kilowatt power level, but was never tested to its full designed capabilities due to facility constraints[6]. The anticipated maximum input power for this engine was 50 kW. A photograph of the hardware is shown in Figure 1.

The device had an average discharge chamber diameter of 200mm and overall dimensions of 310mm x 310mm x 160 mm. The thruster mass was approximately 30.5 kg. The thruster employed a conductive discharge chamber as do all anode layer thrusters. In this case, however, the discharge chamber was segmented into two stages allowing for operation in various electrical configurations. The first configuration, termed single stage was achieved by floating the second stage anode. An alternative single stage configuration may be achieved by electrically shorting the two anodes. Referring to Figure 2a, these different electrical configurations are achieved by placing the switch in position A or B respectively. Another configuration, shown in Figure 2b that was briefly investigated was two stage operation. In this configuration, each segment is powered with a separate power supply as discussed in more detail later.

The cathode is mounted in the center of thruster along the thruster axis. Initial testing was completed using a Russian hollow cathode that was delivered with the thruster. This cathode was designed to operate at current levels up to 20 A with a xenon propellant flow rate from 1-3 mg/s. Subsequent testing at higher power levels was conducted with a NASA hollow cathode that has higher emission current capabilities. This cathode is based on the Plasma Contactor design originally developed for the International Space Station.[7]

The thrust stand used in these tests was originally designed for a 0.5 MW magnetoplasmadynamic (MPD) thruster.[8] A photograph of the thrust stand is provided as Figure 3. The thrust stand held the thruster with its horizontal axis approximately coincident with the centerline of the vacuum facility. The stand was an inverted pendulum type design which allowed up to 25 mm of travel along the sensitive axis. An elastic restoring force was provided by the deformation of a load spring as well as the support flexures. This gave the system a natural frequency of approximately 0.25 Hz. Thrust induced deflections were measured using a linear variable differential transformer (LVDT) and recorded on a strip chart recorder. Transient motion of the thrust stand was critically damped with an electronic feedback circuit acting through an electromagnetic actuator. The damper only applied force when motion was detected, and did not affect steady state thrust measurements.

Xenon gas for the cathode and main discharge was transferred onto the thrust stand using two separate propellant flexures. These consisted of 3mm diameter stainless steel tubes with one end anchored to the stationary facility, and the other end attached to the moving structure of the thrust stand. Electrical current for the cathode and anode was transferred through the stand to the thruster using water cooled electrical flexures. The conductive spans were isolated at each end with plastic tubing. Electrical currents for the cathode keeper, heater, and electromagnets were transferred through the thrust stand, to the thruster with multi-conductor ribbon cables.

The thrust stand was calibrated in place and under vacuum. A mono-filament nylon line was attached to the rear of the thruster and aligned with the thrust vector. The line passed over a pulley and was drawn by weights which could be cycled on and off. The thrust stand response to the calibration weights was recorded on the strip chart. This calibration could then be used to interpret the displacement measured during thruster operation. Cycling the calibration weights revealed hysterisis on the order of 1% of the full scale value. Zero drift was also observed over periods in which the thruster was fired. The amount of drift was dependent on the power level and duration of the test. Since the drift rate was fairly consistent, thrust data were corrected for this error.

Commercially available DC power supplies were used to operate the engine. Separate supplies were used to power the inner and outer magnets, cathode heater, keeper and ignitor. Two 60 A, 500 V discharge supplies configured in series were used for single stage operation. A low voltage, high current keeper supply protected by a blocking diode was used in parallel with a high voltage, low current ignitor supply for cathode ignition and operation. An output filter consisting of a 3.8 Ω ballast resistor in series with the anode and 100 μF capacitor between cathode and anode was used. Back to back 75 volt Zener diodes were used to clamp the floating potential of the cathode with respect to ground.

All performance data reported herein were taken with the thruster operating in the single stage configuration. A schematic diagram of the electrical configuration used for the investigation is provided as Figure 2a. The thruster was operated in the two stage arrangement to verify the integrity of the power system. Electrically this configuration differed from single stage operation by placing each of the two separate 60 A, 500 V discharge power supplies on a different segment of the anode. This configuration is shown schematically in Figure 2b.

Data were collected using a commercially available data acquisition system. Discharge current and voltage, magnet current and voltage, floating potential, keeper current, anode and cathode flow rates, and engine temperatures were monitored. Data were sampled at a rate of 1Hz. Simple internal mathematical functions allowed preliminary calculation of power, thrust, specific impulse, and efficiency.

A laboratory feed system was used to provide propellant to the anode and cathode at the desired mass flow rates via commercially available flow controllers. The accuracy of these devices quoted by the manufacturer was ± 1.0% full scale. Flow calibrations were performed before and after testing using a constant volume method. Commercially available research grade xenon with a purity of 99.999% was used. The feed system was helium leak checked prior to operation of the propulsion system.

All testing was conducted in the main volume of Vacuum Facility 5 at NASA GRC. A detailed description of the facility can be found in ref. 9. This chamber is 4.6 m in diameter and 19 m in length. It is equipped with 20 diffusion pumps which are 0.8 m in diameter. This facility pumps xenon at approximately 100,000 l/s with only the diffusion pumps operating. The facility is also equipped with a gaseous helium cryo system which pumps xenon at an additional 1.5 million l/s. A photograph of the facility is provided as Figure 4. The facility was initially operated only on diffusion pumps to facilitate rapid facility turnaround if problems were detected with the hardware or test setup. Data were collected for low propellant flow operating points. As engine flow rates were increased, the cryo system was started. The highest tank pressure experienced during testing was 5.3×10^{-5} torr measured using ionization gauges located on the facility wall, calibrated on air and uncorrected for xenon.

Following the facility pumpdown sequence, the engine was permitted to remain at vacuum for 24 hours prior to operation to allow for outgassing of the hydroscopic components of the engine. This procedure was recommended by the manufacturer TsNIIMASH. The current limits of the discharge power supplies were set to 25 A prior to operation. The Russian cold start cathode provided with the engine was ignited by applying high voltage between the cathode and keeper. The xenon mass flow rate was set to 3 mg/s for ignition and 1.5 mg/s for steady state operation. The inner and outer magnet supplies were switched on with current values of 1.0 A and 0.5 A respectively. With the cathode ignited, magnets energized, and an anode flow rate of 10 mg/s, the main discharge was started by ramping up the discharge voltage until breakdown occurred. The inner magnet current typically was reduced from the suggested value of 1.0 A to 0.5 A to facilitate breakdown at a lower discharge voltage value. With the reduced inner magnet current, breakdown consistently occurred at a discharge voltage less than 100 V. The discharge voltage was increased to 300 V over a period of 0.5 hours in 50 V increments to allow for hardware outgassing. The power level was increased to 10 kW by increasing the discharge current and voltage values in a stepwise fashion. This procedure was followed to allow the thruster to reach thermal equilibrium. Prior to making performance measurements, the engine was shut down and a thrust calibration was performed as described previously.

During performance evaluations the engine was operated by setting the mass flow rate to the desired value and varying the discharge voltage. When oscillations in the discharge current created thruster instabilities, the inner and outer magnets were optimized by attempting to minimize the magnitude of the discharge current.

Results and Discussion

During this investigation, performance data were taken with the engine operating in the single stage electrical configuration at power levels ranging from 2.0 to 25.4 kW. At the highest power level of 25.4 kW, the engine produced 966 mN of thrust with specific impulse of 3325 s and efficiency 62%. At an alternate operating point, thrust over 1 N was achieved with specific impulse of 2936 s and 66% efficiency. Several issues regarding testing this high power engine were encountered.

Performance

The majority of data discussed herein were taken with the engine operating on the Russian cathode provided. Because the device was not optimized for operation with this Hall thruster, specific impulse and efficiency values were calculated without the mass flow of the cathode. Also, the efficiencies reported were calculated using the discharge power only and did not include the power dissipated in the magnets (typically less than 10 W).

The relationship between thrust and power is shown for eight mass flow rates in Figure 5. The measured thrust values reported ranged from 147 mN to 966 mN. Thrust of 1011 mN was achieved at the power level of 22 kW with specific impulse of 2936 s and 66 % efficiency (this point was taken using the NASA cathode). While a comprehensive tabulation of those results taken with the NASA cathode is not presented here, a table of the performance test results taken with the Russian cathode is provided in Table 1.

Discharge current is plotted as a function of anode mass flow rate in Figure 6. Discharge current is typically directly proportional to anode mass flow rate. In the case of the TM-50 operating at high mass flow rates, the relationship deviates from linear. From this plot, it can be seen that the discharge current is significantly higher for a given mass flow rate; for example, at the highest mass flow rate of 29.6 mg/s, a discharge current value of 35.6 A was measured. Also, from the plot it can be seen that there is a vertical spread of the discharge current values for a given mass flow rate. This behavior was attributed to operation with a non-optimized magnetic field. As noted previously, magnets were adjusted such that the discharge current was minimized. This procedure was not implemented at each data point because identification of the optimum performance points was not the objective of this test sequence given the time constraints of this investigation.

Specific impulse and efficiency versus thruster power are plotted for the eight mass flow rates tested in Figure 7 and 8 respectively. Specific impulse increased with voltage in nearly all cases. Cases do exist were a increase in voltage did not yield an increase in specific impulse, but in those instances, there was a correspondingly sharp drop in thruster efficiency. This behavior was attributed to non-optimized magnets which caused a modal change in thruster operation. The trend of increasing specific impulse with voltage did come at the expense of efficiency as discussed below.

Efficiency versus discharge voltage is plotted for eight anode mass flow rates in Figure 9. Efficiency increased with discharge voltage up to approximately 500 V at which point the efficiency began to decrease as the voltage was increased further. This behavior exists for all mass flow rates at which a discharge voltage of 500 V was exceeded. As a result, operation of large Hall thrusters of this type over 500 V does not seem advantageous from a performance perspective. However, it may be possible to obtain high efficiency at discharge voltages >500 V by operating in a two stage configuration.

When operating in the two stage configuration, the discharge is initiated with the accelerating supply between the cathode and anode 2, which is the segment closest to the exit plane. The discharge supply between anode 1 and anode 2 is then energized. Application of accelerating voltages between 500 V and 2000 V allows for operation in a high specific impulse and efficiency mode with relatively low thrust. An efficiency of over 70% was calculated for data taken in Russia while operating with an anode mass flow rate of 10.16 mg/s, discharge voltage of 330 V, and an accelerating voltage 900 V[6]. Relatively high thrust with correspondingly lower specific impulse and efficiency can be achieved by operating in the single stage mode. Utilizing the favorable characteristics of each mode of operation, a two stage thruster could be used to perform multiple mission requirements. Only a short duration test sequence was conducted in the two stage configuration at NASA GRC to verify proper operation of the engine and laboratory power system.

Testing

In general, the thruster operated without incident at power levels below 10 kW. Current oscillations were easily minimized by adjusting the magnetic field. Engine breakdown occurred readily as described above. However, at elevated power levels (>10 kW) anomalous arcing between anode and thruster body periodically occurred. Specifically, arcing occurred between anode 2 and the graphite guard rings which protect the magnetic system. There were a very small number of arc occurrences at power levels below 10 kW but the behavior was more prevalent at voltages above 400 V and anode mass flow rates above 25 mg/s. As a result no performance data were obtained for powers above 25 kW .

Several factors were identified as possible influences on the arc occurrences: the temperature of the engine, facility pressure, internal engine materials, magnetic field effects, cathode flow and position, capacitor size, temporal response of the discharge power supplies, or an anomalous connection in the electrical scheme. Several of these items have been investigated without conclusive evidence of a significant impact on the engine's operation. It has been demonstrated that sustained versus unstained arc occurrences have a dependence on the value of the discharge current limit; for example, sustained arcs do not occur readily with the discharge current limit set below 20 A. A comprehensive evaluation is currently underway in an attempt to more completely understand this behavior.

Several issues pertinent to the test apparatus became apparent while testing at high power levels. A high temperature elastomer tubing was used to make the final anode connections between the thruster and thrust stand. This tubing was used to provide electric isolation of the anode. While testing at high powers, one of the three tubes was damaged as a result of an over-temperature condition. This in turn caused a propellant leak in the anode line. For subsequent tests, the distance between the thruster and the elastomer to metal tubing connection was increased to reduce the temperature at the isolator interface.

Additional electrical isolation issues were also manifest during high power testing. For this test series, electrical connections inside the vacuum chamber were made using mechanical fasteners. Silicone tape was used at the connection for electrical isolation. This method has been used in previous Hall thruster investigations without consequence. Problems were experienced in maintaining the integrity of the electrical isolation on the high current connections. There were several instances of electrical arcs to facility ground at these locations. Additional plastic insulation, fiberglass sleeving, and silicone tubing were added in subsequent tests to mitigate these problems.

Concluding Remarks

An investigation was conducted to assess the performance of a high power Hall thruster. Performance measurements were made at power levels ranging from 2.0 to 25.4 kW. At the highest power level, the engine produced 966 mN of thrust with specific impulse 3325 s and efficiency of 62%. Higher power levels were not achieved due to arc occurrences between the anode of the second stage and the thruster guard rings. An investigation is currently being conducted to determine the cause of the arcing phenomena. Additional performance measurements will be taken at power levels in excess of 25 kW for the single stage configuration after the arcing issues have been resolved.

All data reported were taken with the thruster operating in the single stage configuration. An extensive investigation of the thruster's two-stage performance will be conducted in the future.

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Table 1: TM-50 Hall thruster performance data.

Discharge Voltage	Discharge Current	Power	Anode Flow	Cathode Flow	Thrust	Efficiency	Specific Impulse
V	A	W	mg/s	mg/s	mN		s
197.8	10.3	2037	10.3	1.5	147	0.52	1454
200.5	10.2	2045	10.3	1.5	147	0.51	1456
250.6	10.4	2606	10.3	1.5	169	0.53	1673
299.6	10.2	3056	10.3	1.5	176	0.49	1741
351.9	10.0	3519	10.3	1.5	189	0.49	1866
401.4	9.9	3974	10.3	1.5	202	0.50	1993
448.9	9.9	4444	10.3	1.5	213	0.49	2102
298.2	15.0	4473	14.9	1.5	255	0.49	1748
348.7	15.0	5231	14.9	1.5	276	0.49	1890
401.3	15.0	6020	14.9	1.5	305	0.52	2094
448.5	15.0	6728	14.9	1.5	324	0.53	2222
149.8	17.2	2577	16.3	1.5	191	0.44	1198
199.3	17.6	3508	16.3	1.5	239	0.50	1494
249.9	17.0	4248	16.3	1.5	281	0.57	1759
300.0	16.8	5040	16.3	1.5	313	0.60	1958
350.0	16.7	5845	16.3	1.5	336	0.59	2103
401.2	16.6	6660	16.3	1.5	358	0.59	2241
451.3	16.7	7537	16.3	1.5	376	0.58	2354
501.9	16.7	8382	16.3	1.5	395	0.57	2475
501.5	16.6	8325	16.3	1.5	390	0.56	2445
552.0	16.6	9163	16.3	1.5	407	0.56	2551
549.9	16.5	9073	16.3	1.5	411	0.57	2574
602.0	16.6	9993	16.3	1.5	429	0.57	2686
650.3	16.6	10795	16.3	1.5	441	0.55	2763
704.5	16.6	11695	16.3	1.5	456	0.55	2854
758.7	16.8	12746	16.3	1.5	470	0.53	2942
806.7	16.7	13472	16.3	1.5	484	0.53	3030
268.9	20.6	5539	18.4	1.5	336	0.55	1862
309.6	20.5	6347	18.4	1.5	361	0.56	2000
349.3	20.4	7126	18.4	1.5	388	0.57	2150
149.7	22.2	3323	20.2	1.5	255	0.49	1289
200.8	22.0	4418	20.2	1.5	314	0.55	1585
251.0	21.9	5497	20.2	1.5	360	0.58	1819
302.9	22.4	6785	20.2	1.5	409	0.61	2067
350.9	22.8	8001	20.2	1.5	443	0.61	2238
400.6	22.7	9094	20.2	1.5	474	0.61	2392
449.6	22.7	10206	20.2	1.5	500	0.61	2524
501.1	22.6	11325	20.2	1.5	528	0.61	2669
502.6	21.5	10806	20.2	1.5	516	0.61	2605

Table 1: continued

Discharge Voltage	Discharge Current	Power	Anode Flow	Cathode Flow	Thrust	Efficiency	Specific Impulse
V	A	W	mg/s	mg/s	mN		s
276.6	24.1	6666	21.5	1.5	403	0.57	1912
300.5	23.9	7182	21.5	1.5	424	0.58	2014
353.0	23.7	8366	21.5	1.5	462	0.59	2193
402.4	23.5	9456	21.5	1.5	495	0.60	2349
452.0	23.4	10577	21.5	1.5	528	0.61	2505
450.9	23.2	10461	21.5	1.5	528	0.62	2505
500.8	23.1	11568	21.5	1.5	557	0.62	2641
549.2	23.2	12741	21.5	1.5	583	0.62	2769
598.0	23.1	13814	21.5	1.5	603	0.61	2862
650.5	23.2	15092	21.5	1.5	627	0.61	2975
700.0	23.0	16100	21.5	1.5	623	0.56	2955
745.5	23.0	17147	21.5	1.5	662	0.59	3139
351.6	27.5	9669	24.7	1.5	550	0.63	2273
402.9	27.0	10878	24.7	1.5	594	0.66	2455
455.6	26.8	12210	24.7	1.5	636	0.67	2628
500.0	26.6	13300	24.7	1.5	661	0.66	2729
548.2	26.6	14582	24.7	1.5	695	0.67	2871
600.0	26.8	16080	24.7	1.5	726	0.66	2998
650.1	26.9	17488	24.7	1.5	753	0.66	3109
697.6	26.9	18765	24.7	1.5	775	0.65	3200
748.2	27.3	20426	24.7	1.5	802	0.64	3311
449.2	27.4	12308	24.8	1.5	624	0.64	2567
501.9	27.4	13752	24.8	1.5	655	0.63	2695
350.0	28.3	9905	24.7	1.5	557	0.63	2295
301.4	29.8	8982	24.7	1.5	509	0.58	2099
357.8	34.8	12451	29.6	1.5	668	0.60	2299
399.2	33.9	13533	29.6	1.5	711	0.63	2446
453.7	33.7	15290	29.6	1.5	764	0.64	2630
509.5	33.4	17017	29.6	1.5	817	0.66	2814
551.3	33.3	18358	29.6	1.5	855	0.67	2942
600.3	33.3	19990	29.6	1.5	891	0.67	3066
649.5	33.3	21628	29.6	1.5	929	0.67	3199
713.0	35.6	25383	29.6	1.5	966	0.62	3325

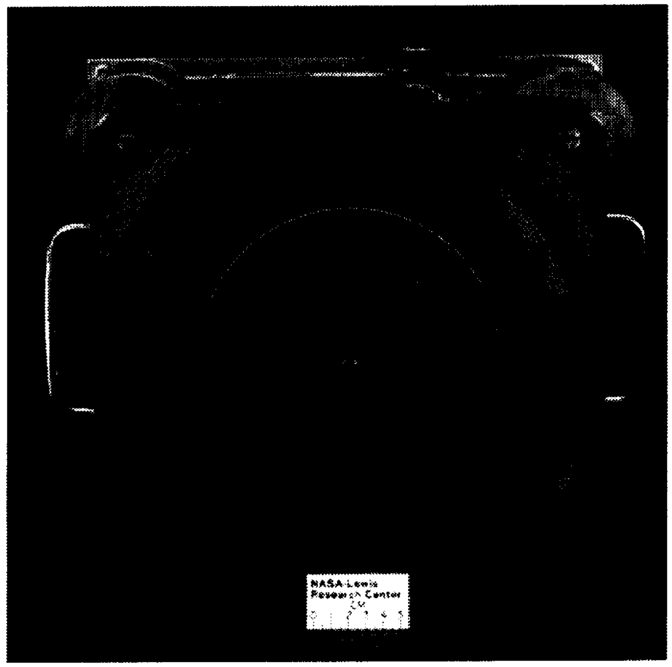

Figure 1: TM-50 Hall thruster with centrally mounted Russian cathode.

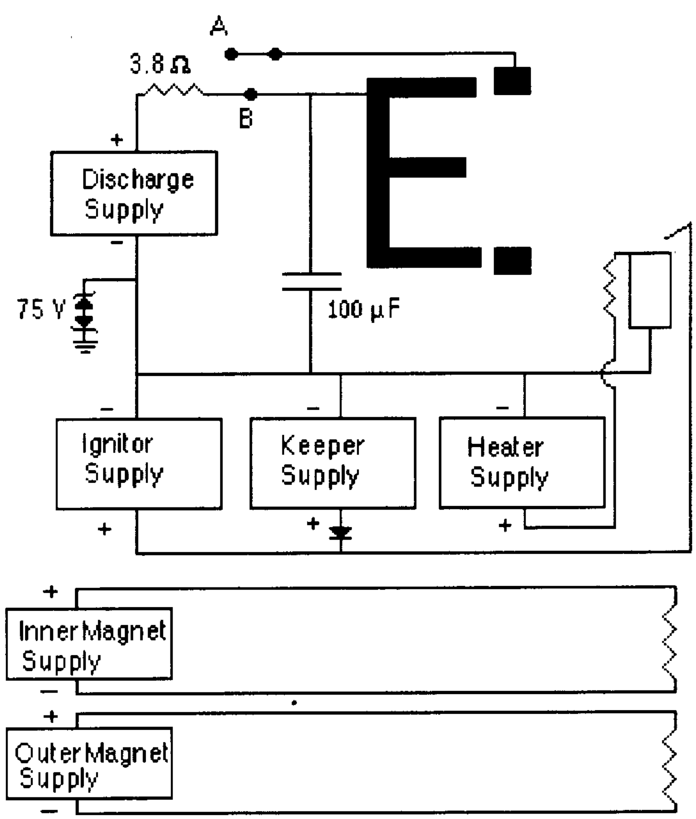

Figure 2a: Electrical schematic for the single stage configuration.

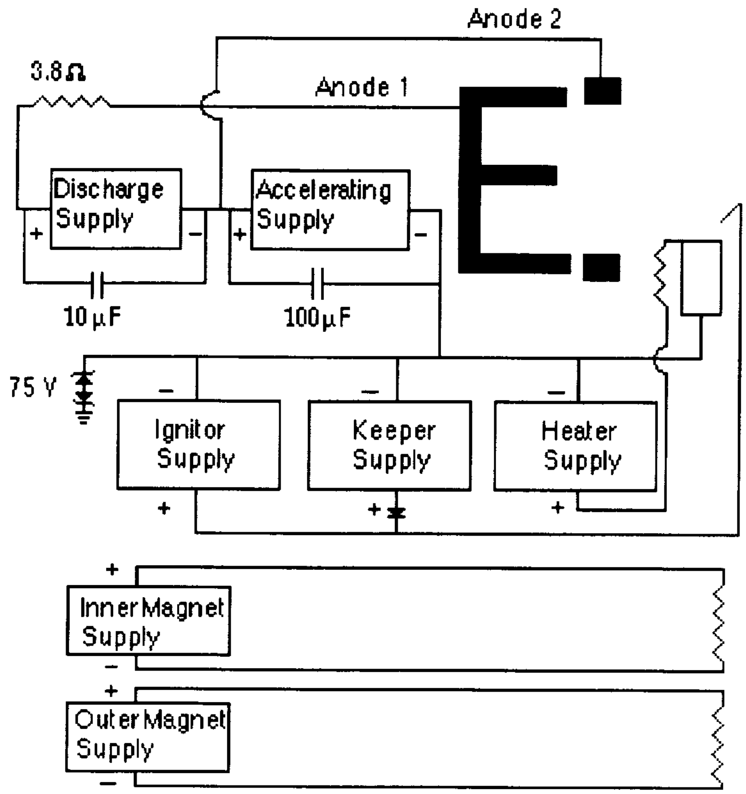

Figure 2b: Electrical schematic for two stage configuration.

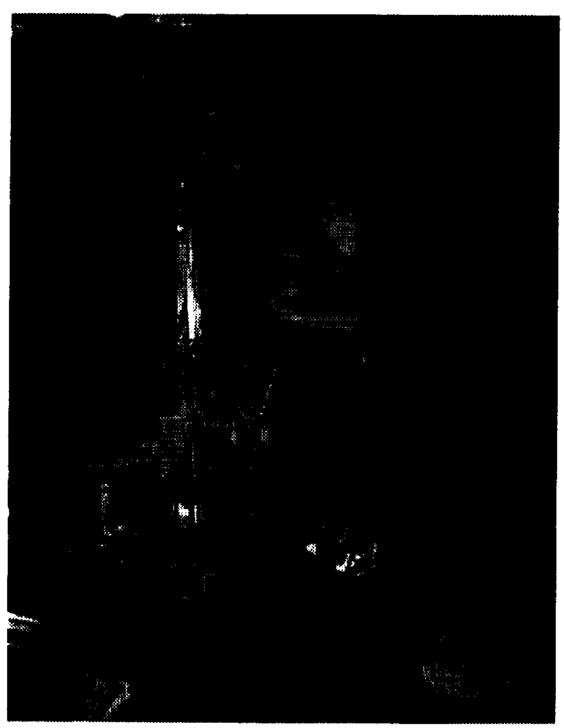

Figure 3: TM-50 mounted on thrust stand in Vacuum Facility 5 at NASA GRC.

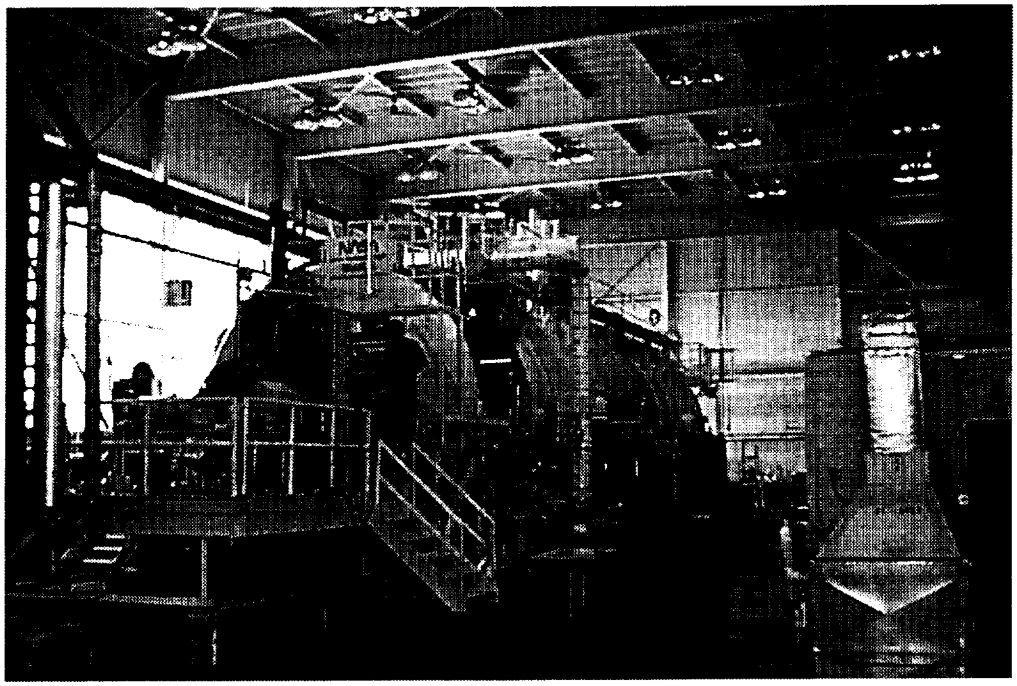

Figure 4: Vacuum Facility 5 at NASA GRC.

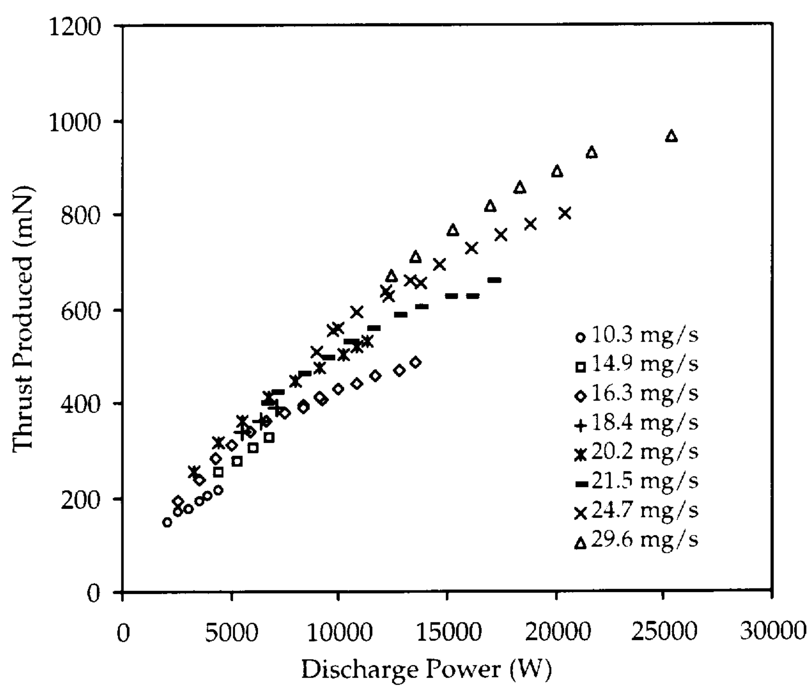

Figure 5: Thrust produced by the TM-50 versus discharge power for eight different anode flow rates.

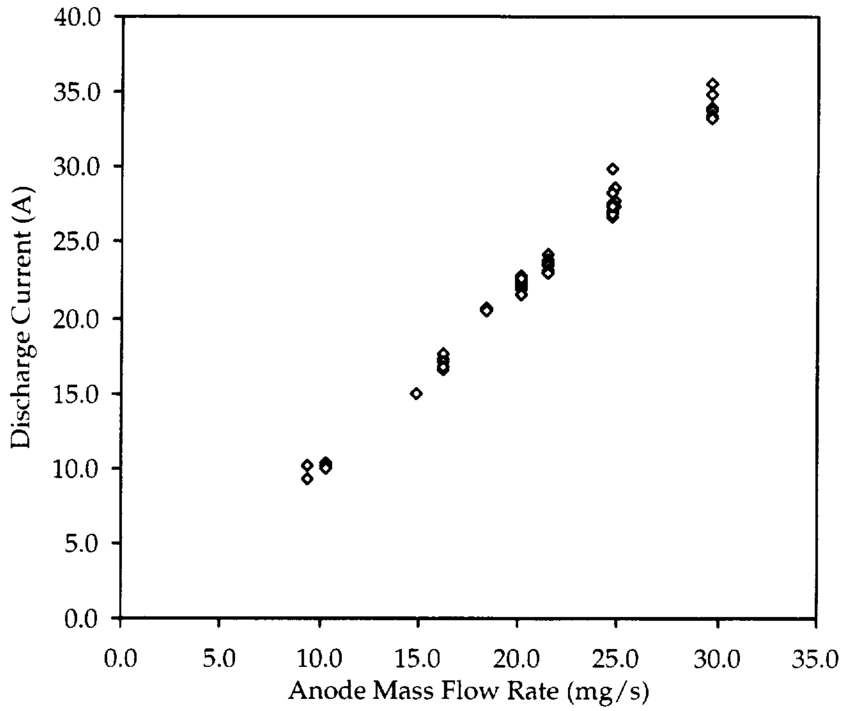

Figure 6: Discharge current versus anode mass flow rate.

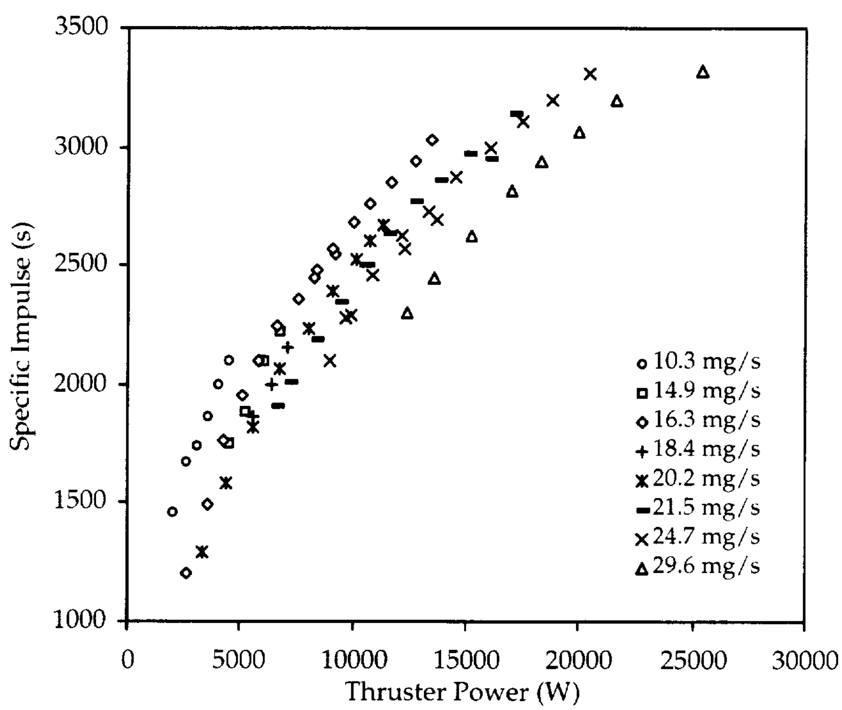

Figure 7: Specific Impulse versus thruster power for eight different anode flow rates.

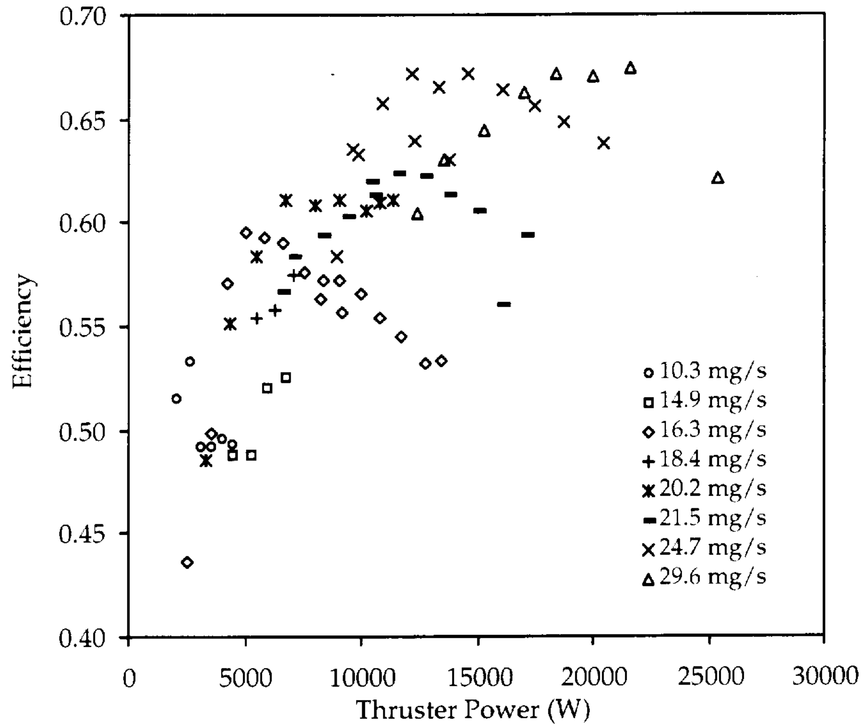

Figure 8: Thruster efficiency versus thruster power for eight different anode flow rates.

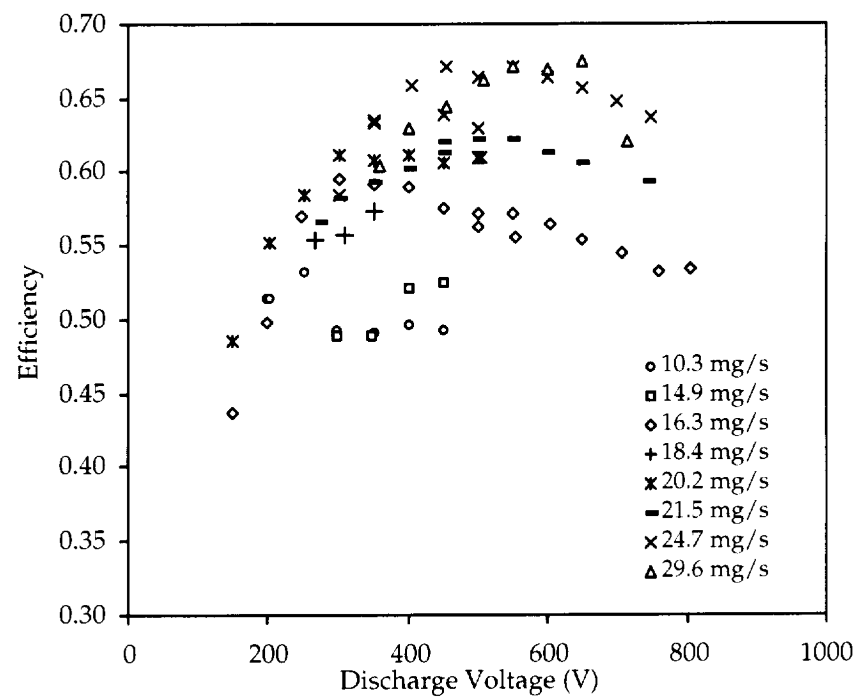

Figure 9: Thruster efficiency versus discharge voltage for eight different anode flow rates.

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13. ABSTRACT *(Maximum 200 words)*

An experimental investigation was conducted on a laboratory model Hall thruster designed to operate at power levels up to 50 kW. During this investigation the engine's performance was characterized over a range of discharge currents from 10 to 36 A and a range of discharge voltages from 200 to 800 V. Operating on the Russian cathode a maximum thrust of 966 mN was measured at 35.6 A and 713.0 V. This corresponded to a specific impulse of 3325 s and an efficiency of 62%. The maximum power the engine was operated at was 25kW. Additional testing was conducted using a NASA cathode designed for higher current operation. During this testing, thrust over 1 N was measured at 40.2 A and 548.9 V. Several issues related to operation of Hall thrusters at these high powers were encountered.

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