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Power Electronics Development for the SPT-100 Thruster

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POWER ELECTRONICS DEVELOPMENT FOR THE SPT-100 THRUSTER

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ABSTRACT

Russian electric propulsion technologies have recently become available in the world market. Of significant interest is the Stationary Plasma Thruster, (SPT) which has a significant flight heritage in the former Soviet space program. The SPT has performance levels of up to 1600 seconds of specific impulse at a thrust efficiency of 0.50. Studies have shown that this level of performance is well suited for stationkeeping applications, and the SPT-100, with a 1.35 kW input power level is presently being evaluated for use on Western commercial satellites. Under a program sponsored by the Innovative Science and Technology Division of the Ballistic Missile Defense Organization, a team of U.S. electric propulsion specialists observed the operation of the SPT-100 in Russia. Under this same program, power electronics were developed to operate the SPT-100 to characterize thruster performance and operation in the US. The power electronics consisted of a discharge, cathode heater and pulse ignitor power supplies to operate the thruster with manual flow control. A Russian designed matching network was incorporated in the discharge supply to ensure proper operation with the thruster. The cathode heater power supply and ignitor were derived from ongoing development projects. No attempts were made to augment thruster electromagnet current in this effort. The power electronics successfully started and operated the SPT-100 thruster in performance tests at NASA Lewis, with minimal oscillations in the discharge current. The efficiency of the main discharge supply was measured at 0.92, and straightforward modifications were identified which could increase the efficiency to 0.94.

INTRODUCTION

In the recent past, Russian electric propulsion technologies have been made available in the world market. A class of thrusters known as Stationary Plasma Thrusters (SPTs) have emerged with performances of 1600 seconds specific impulse at a thrust efficiency of 0.50 with a power inputs of 0.4 to 1.35 kW using xenon propellant.[1] This level of specific impulse has been shown to be well suited for north-south stationkeeping and orbit raising applications.[2] Work on the SPT began in the Soviet Union in 1964, with the first space flight occurring in 1972. During this flight, the SPT operated for a total of 150 hours and demonstrated compatibility with the host spacecraft.[3] Two low power versions of the SPT, designated the SPT-50 and the SPT-70, have flown. The "-50" and "-70" designation indicate the diameter of the discharge chamber in millimeters. These lower power versions have a significant flight history, having flown on "Meteor," "Ekran" and "Gorizont" spacecraft,[4] but the SPT-100 has never flown.

The Innovative Science and Technology Division of the Ballistic Missile Defense Organization (BMDO/IST) recently sponsored a trip to Russia during which a team of U.S. electric propulsion experts evaluated the performance of the 1.35 kW SPT or SPT-100.[5]

A power processing unit (PPU) development program was initiated at NASA Lewis Research Center (NASA LeRC) in a program sponsored by BMDO/IST to determine the interface characteristics of the thruster and for performance and integration testing of an SPT-100, obtained from the FAKEL Enterprise, in the US. Since little detailed information was known about the load characteristics of the SPT-100, the PPU was designed according to information in the public domain on SPT operation, and NASA-Lewis experience with similar devices. Additionally, no information about the propellant flow controller was available, so the closed-loop regulation of thruster power via flow control was excluded from this effort.

This paper describes the development of the SPT-100 PPU, and includes the results of thruster integration tests.

THRUSTER OPERATION

A cross section of the SPT-100 thruster is shown in Figure 1. Xenon gas is injected into the annular discharge chamber through the anode electrode, which is situated at the rear of the chamber. Xenon is also provided to the hollow cathode, which is situated external to the thruster. A ceramic insulator lines the

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entire length of the discharge chamber. The discharge is initiated by first heating the hollow cathode to a suitable temperature with a resistance heater. When the cathode is at a sufficiently high temperature to facilitate electron emission, a high voltage pulse train is applied to the ignitor electrode to initiate the discharge.[6]

The discharge electrons flow from the cathode to the anode, passing through the electromagnets as shown in figure 1, and this magnetic field causes the electrons to drift to the anode in a spiral path. The xenon gas is ionized by collisions with these electrons, and the combination of the magnetic fields and the electron current causes an electric field to be set up with equipotential lines that are parallel to the anode and exit plane of the thruster. This electric field then accelerates the xenon ions electrostatically. The physical processes within the thruster are presented in detail by Bugrova et al..[3]

POWER SUPPLY CONFIGURATION

A block diagram of the power supplies required to operate the SPT and the thruster / power interfaces are shown in Figure 1. A discharge, cathode heater and a pulse ignitor power supply are required to operate the thruster with manual flow control. The discharge supply is a voltage regulated power supply which is connected to the thruster anode and the electromagnet. In this configuration, the discharge current excites the electromagnet, setting up a magnetic field in the discharge chamber that is proportional to the discharge current. No attempts were made in this work to augment the magnet current with an additional power source.

The cathode heater requires a constant current DC power supply to provide power for preheating the cathode in the ignition process. This power supply operates at a single setpoint for a fixed time interval prior to the application of the ignitor pulses. Power must be removed from the heater at discharge ignition to prevent an over-temperature condition on the cathode.

The ignitor supply provides a high voltage pulse train between the cathode and ignitor electrode to provide a localized discharge between these electrodes. When this discharge is established, the main discharge also ignites and the ignitor supply is turned off. All voltages measured during thruster integration tests were as shown in the figure 1.

LOAD CHARACTERISTICS

Critical to this work were the volt-ampere characteristics of the discharge, the impedance of the cathode heater, and the ignition sequence. Volt-ampere characteristics have been published,[3] and show that the discharge current is not a strong function of discharge voltage, but rather of propellant flow rate and magnetic field produced by the electromagnets. A simple voltage regulated power supply was envisioned for the main discharge with a nominal 300 VDC output at 1.35 kW.

The cathode heater impedance was assumed to be in a range similar to those utilized on U.S. hollow cathodes, such as the Solar Electric Propulsion System (SEPS) 30 cm ion engine,[7] and the Space Station Freedom (SSF) Plasma Contactor. Power electronics to operate cathode heaters were already under development for the Space Station Freedom (SSF) plasma contactor system,[8] and were slightly modified for use in this effort. Some questions also remained concerning the operation of the starting electrode. The configuration of the hollow cathode and starter electrode was similar to a neutralizer / keeper assembly on an ion engine.[6,7] The amplitude and duration of the pulses were also unknown. Additionally, little was known about the actual starting sequence, which involved the cathode heater power levels and duration of the pre-heat phase.

When SPT operation was monitored in Russia measurements of the discharge voltage and current while the thruster was running indicated variations in discharge current at a 20 - 30 kHz fundamental frequency.[5] These oscillations are shown in Figure 2. A "matching network" was installed between the Russian power electronics and the thruster. The purpose of this matching network was to dampen these oscillations, and a schematic of the matching network was provided. From this information it was decided to incorporate the matching network with the output filter of the discharge power supply and monitor the discharge voltage ahead of the matching network for regulation purposes.

The heater impedance and the details of the ignition sequence were also obtained in Russia. After the initial breakdown between the ignitor and cathode electrodes, a discharge is maintained only for several milliseconds until the main discharge is established. Operation of the closed-loop flow control system was considered proprietary to FAKEL which, resulted in the omission of this function from the NASA Lewis PPU. Manual flow control was accomplished using commercially available flow controllers.

DESIGN DETAILS

An input voltage range for the PPU of 115-130 VDC was selected to alleviate the transformer design difficulties associated with large step-up ratios required with low input voltages and to facilitate the use of existing hardware. These decisions were also made to expedite the development process, as time was limited. A switching frequency of 20 kHz for all power supplies was selected as a compromise between low mass and high efficiency, and also took advantage of the experience base at LeRC with PPUs operating at this frequency.[9] Specifications for each power supply were compiled and are summarized in Tables I through III.

Discharge Supply

The discharge supply accounts for the largest amount of power in the PPU, and a full bridge topology was chosen for the power stage. A schematic of the discharge supply appears in Figure 3. This topology has the advantage of a single primary winding, compared to the bifilar windings of the parallel or push-pull topology. A low inductance power stage layout designed in previous works was used which eliminated the need for passive snubbers on the power switches.[10] No input filters for electromagnetic compatibility were included in the design. A simple LC output filter was originally envisioned for the power supply, however, the matching network required for the thruster would also serve as a suitable output filter. A small inductor was used in series with the matching network to form a fast response LC network with the first capacitor of the matching network to provide a filtered feedback signal for the voltage regulation control loop. The matching network contained a 15 mH inductor. The design was not optimized for low mass, but had a mass of less than 0.75 kg, indicating that a lower mass is achievable with a more detailed design. The discharge current was used to excite the thruster electromagnets, and no attempts were made to augment the electromagnet current with a separate power supply.

To maintain input / output isolation and isolation from chassis ground, a Hall effect current sensor was used to sense output voltage. The output voltage of the power supply was dropped across a 3 kΩ sense resistor, and the current through the resistor was measured as a feedback parameter. Regulation of the resistor current maintained a constant output voltage. Unfortunately, this voltage sensing technique dissipated 30 W of power at the nominal output voltage. The power level could have been significantly reduced by the incorporation of a more sensitive sensor and the addition of more turns of wire through the sensor window, which would combine to allow a larger sense resistor to be used. Time constraints prohibited this modification. The efficiency data published here include this loss. This voltage sensing technique maintained an impedance of greater than 1 MΩ between the discharge supply and system ground. Discharge current was also monitored for the current limit feedback loop and to remove power from the pulser and cathode heater power supplies when the discharge current exceeded 1.5 A.

A 180 Ω resistor was used to pre-load the discharge power supply to ensure proper voltage regulation when the discharge current was zero. This resistor was also switched out when the discharge current exceeded 1.5 A.

Cathode Heater

The cathode heater power supply used was derived from previous work,[8] and a schematic of the power stage of the cathode heater appears in Figure 4. Due to the low power required by the heater, and the relatively short operating time, a simple push-pull converter was used. The output filter simply consisted of an inductor, which limited the output ripple to 10 %. This power supply was commanded on by remote control, but was automatically shut off by a logic level output of the discharge supply when the discharge current exceeded 1.5 A.

Pulse Ignitor

A simple pulse transformer was used to generate the starting pulse train applied to the ignitor electrode. The operation of this circuit has been described in detail elsewhere.[10] Sufficient energy is stored in the transformer primary to sustain the low voltage discharge for several milliseconds after the initial breakdown.

FABRICATION AND RESISTIVE LOAD TESTING

The PPU was fabricated on a 45 cm X 60 cm X 1 cm fiberglass sheet. 110 VAC input power converters were used to generate the control and gate drive power. The entire assembly was enclosed in a perforated plastic cover with two fans for safety and cooling purposes, resulting in final assembled dimensions of 45 cm X 60 cm X 15 cm. No attempt was made to minimize the size of the PPU, and a flight type PPU would have smaller dimensions. A photograph of the completed PPU appears in Figure 5. A remote control box was constructed to house the on / off switches for the power supplies and potentiometers to control the cathode heater current and the discharge voltage.

Discharge

The discharge supply was connected to a resistive load to evaluate the efficiency of the power supply and to evaluate its line and load regulation characteristics. Input and output voltages were measured with digital multimeters, and the input and output currents were measured using high frequency current probes with a digital oscilloscope. The discharge supply efficiency was measured at 0.92, including the 30 W dissipation of the voltage sensor. If this loss was removed, the efficiency increased to 0.94. The regulation and ripple measurements corresponded to those listed in Table I.

Cathode Heater

The cathode heater was connected to a resistive load, and the efficiency measured as described above for the discharge supply. The efficiency was 0.77 at full power, and the regulation and ripple were within the specifications outlined in Table II.

Pulse Ignitor

An oscillograph of the pulse ignitor output appears in Figure 6. The peak pulse amplitude was 375 V, and the duration was 30 µs. Based on in-house experience with hollow cathodes, these parameters were judged to be appropriate to ensure reliable ignition. The normal breakdown voltage for ion thruster neutralizers with similar geometries and well conditioned cathodes is generally less than 100 V.[11] If a breakdown occurs in the early stages of the pulse, the remaining energy will sustain the discharge until the inductor discharges, as demonstrated with arcjet ignitors.[10]

THRUSTER INTEGRATION TESTS

The thruster was installed in a 4.6 m diameter, 18.3 m long vacuum facility. The thruster was mounted on a thrust stand which was described in detail elsewhere.[12] Xenon flow control was provided using a regulated pressure gas supply in conjunction with a metering valve. Flow rate was measured using calibrated thermal conductivity type flow meters.

The thruster data presented herein were taken during the first twenty hours of SPT-100 operation at NASA Lewis. Subsequent data have shown oscillatory behavior not described here. Full details of SPT-100 operation are presented in a companion paper.[13] Xenon flow was established at the specified flow rate for 1.35 kW operation and the cathode heater and discharge supplies were turned on. After the cathode pre-heat period, the pulse ignitor was turned on, and ignition of the thruster was instantaneous. An oscillograph of a typical thruster start appears in Figure 7. The discharge voltage sags to 90 V at ignition, and recovers to the nominal 300 V value within 2 ms. The discharge current overshoots the nominal value of 4.5 A to a peak of over 10 A. After the initial start transient, the thruster settled into quasi-stable operation, characterized by current oscillations of 6 A_{p-p} centered around the nominal value of 4.5 A. The discharge voltage also oscillated 75 V_{p-p} about the nominal 300 V setpoint. The discharge voltage measured included the voltage drop across the electromagnet, which was measured in these tests to include power loss in the magnets for thruster efficiency calculations. Direct anode to cathode voltage measurements were not made, and the large voltage oscillations may be a result of di/dt voltage drops across the electromagnets. These oscillations damped out after 15-30 minutes of thruster operation. The amount of time needed for the oscillations to damp out depended on the amount of time the thruster was in vacuum prior to ignition, and the number of cycles in the test session. The settling time decreased to as little as three minutes for a thruster which had been operated several hours and left in the vacuum facility at low pressure.

An oscillograph of the thruster oscillations is shown in Figure 8. The oscillations were periodic, with a fundamental frequency of 12.5 kHz. The amplitude was somewhat variable, but averaged 3 A_{p-p}. The corresponding discharge voltage oscillations were approximately 50 V_{p-p}. These oscillations eventually damped out, and an oscillograph of stable thruster operation is shown in Figure 9. The current oscillations averaged 0.5 A_{p-p} with some excursions to 1 A_{p-p}. The voltage oscillations were less than 10 V_{p-p}. Occasionally during stable operation, the thruster would burst into oscillation for periods of less than 1 second. These oscillations had a fundamental frequency of 6 kHz and were variable in amplitude. The oscillations measured were of improper frequency to be associated with resonances or controls oscillations in the PPU.

CONCLUSIONS

Stationary plasma thrusters, which have a significant flight history in the former Soviet Union have recently become available to the world aerospace community. These thrusters have performance levels of 1600 s specific impulse at a thrust efficiency of 0.50. Under a program sponsored by BMDO/IST, a PPU was developed by NASA Lewis to enable performance and thruster integration testing in the US.

The PPU performed all functions associated with thruster operation with the exception of closed loop regulation of discharge current via flow control. A Russian-designed matching network was incorporated into the discharge power supply to ensure a proper match between the PPU and thruster. Discharge voltage was measured on the PPU side of the matching network to provide a stable feedback signal. This matching network included a 15 mH inductor which had a mass of less than 0.75 kg. The discharge supply efficiency was 0.92. Straightforward modifications were identified which would raise this efficiency to 0.94.

Thruster integration tests were successful, and the PPU started and operated the thruster without difficulty. The discharge current exhibited oscillatory behavior for the first several minutes of operation,with amplitudes of $6A_{p\text{-}p}$, and a fundamental frequency of 12.5 kHz. These oscillations eventually damped out to a 0.5 $A_{p\text{-}p}$ average amplitude. Voltage oscillations of approximately 75 $V_{p\text{-}p}$ were also associated with the large current oscillations. These voltage swings were thought to result from the interaction of the time varying discharge current with the inductance of the electromagnet. The current oscillations were of improper frequency to be associated with PPU resonances or the result of control instabilities.

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REFERENCES

1. Bober, A.S., et al., "State of Work on Electrical Thrusters in the USSR," IEPC-91-003, October, 1991.

2. Patterson, M.J., et. al.. "Experimental Investigation of a Closed-Drift Thruster," AIAA Paper 85-2060, 1985.

3. Bugrova, A.I. et al., "Physical Processes and Characteristics of Stationary Plasma Thrusters With Closed Electrons Drift," IEPC-91-079, October, 1991.

4. Zhurin, V.V., et al., "Some Problems of EPT Research in the USSR," AIAA Paper 90-2678, July, 1990

5. Brophy, J.R., "Stationary Plasma Thruster Evaluation in Russia, Summary Report," JPL Publication 92-4, 15 March, 1992.

6. Arkhipov, B.A., et al., "The Development of Cathodes-Compensators for Stationary Plasma Thrusters in the USSR," IEPC-91-023, October, 1991.

7. Anonymous, "30-Centimeter Ion Thrust Subsystem Design Manual," NASA TM-79191, June, 1979.

8. Hamley, J.A., et al., "Development of Power Electronics for the Space Station Freedom Plasma Contactor," IEPC-93-052, September, 1993.

9. Hamley, J.A., et al., "10 kW Power Electronics for Hydrogen Arcjets," Proceedings of the 1992 JANNAF Propulsion Meeting, pp. 539-553, (see also NASA TM-105614), February 1992.

10. Hamley, J.A. and Hill, G.M., "Power Electronics for Low Power Arcjets," AIAA Paper 91-1991, (see also NASA TM-104459), June, 1991.

11. Sarver-Verhey, T.R., "Extended Testing of Xenon Ion Thruster Hollow Cathodes," AIAA Paper 92-3204, July, 1992.

12. Haag, T.M. and Curran, F.M., "Arcjet Starting Reliability: A Multistart Test on Hydrogen / Nitrogen Mixtures," AIAA 87-1061, May, 1987.

13. Sankovic, J.M. et al., "Performance Evaluation of the Russian SPT-100 Thruster at NASA Lewis," IEPC-93-094, September, 1993.

Table I. Discharge Power Supply Specifications

Input Voltage:	115 - 130 VDC
Output Voltage:	300 VDC, regulated
Output Current:	4.5 ADC nominal 5.0 ADC current limit
Ripple:	< 3 % into resistive load at rated power
Regulation:	
Line:	< 1 % of setpoint
Load:	< 1 % of setpoint

Table II. Cathode Heater Power Supply Specifications

Input Voltage:	115 - 130 VDC
Output Voltage:	8 VDC maximum
Output Current:	12 ADC, regulated 15 ADC current limit
Ripple:	10 % peak to peak of DC value
Regulation:	
Line:	< 1 % of setpoint
Load:	< 1 % of setpoint

Table III. Pulse Ignitor Specifications

Input Voltage:	115 - 130 VDC
Pulse Amplitude:	400 V
Pulse Duration:	30 μs

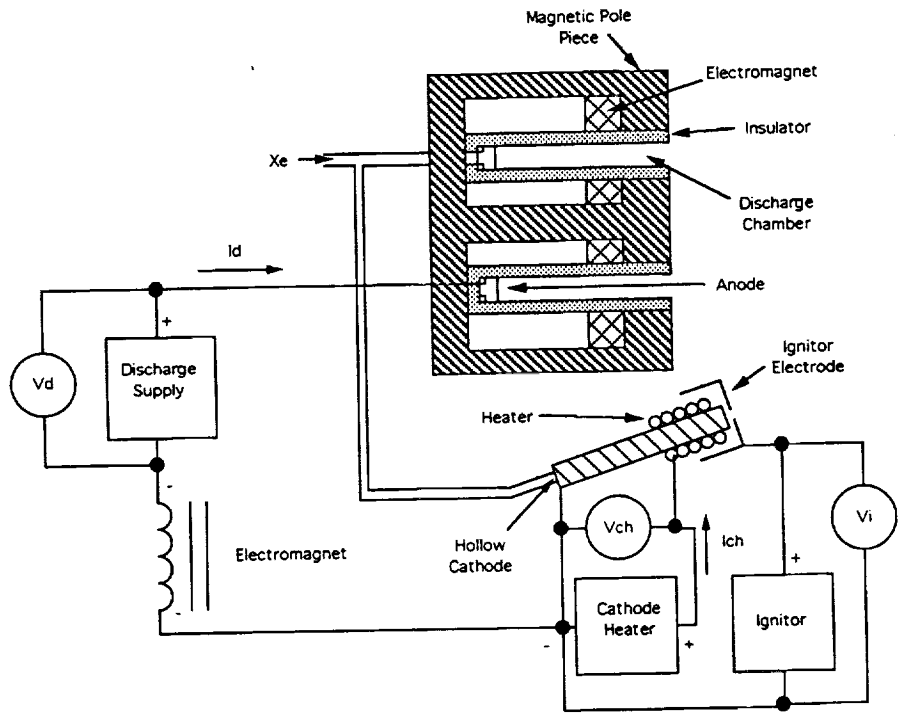

Figure 1. Simplified cross section and electrical schematic of the Stationary Plasma Thruster (SPT)

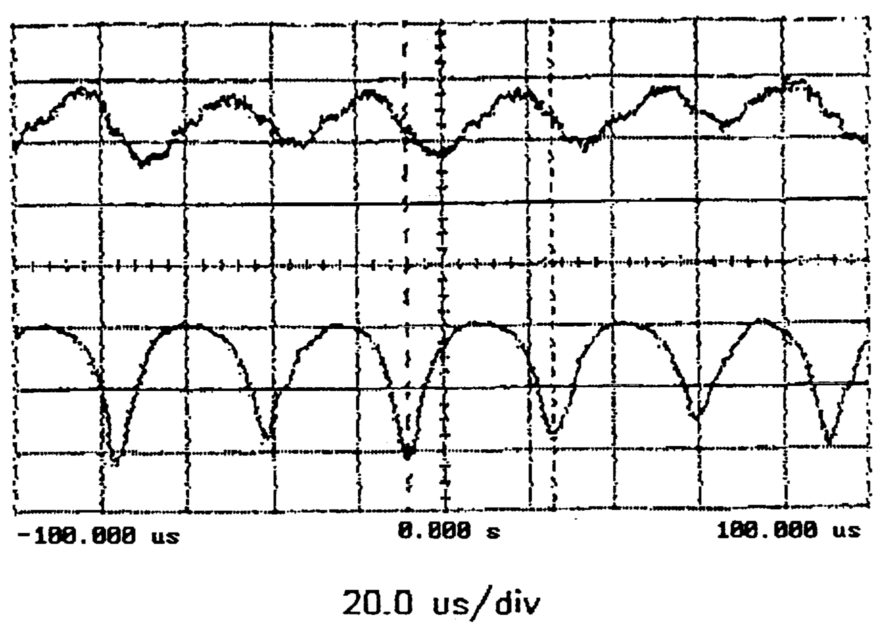

Figure 2. Discharge voltage and current waveforms showing current oscillations (Reference 5).
Upper trace: Discharge Voltage, 20 V / div. Lower trace: Discharge Current, 1.25 A / div.

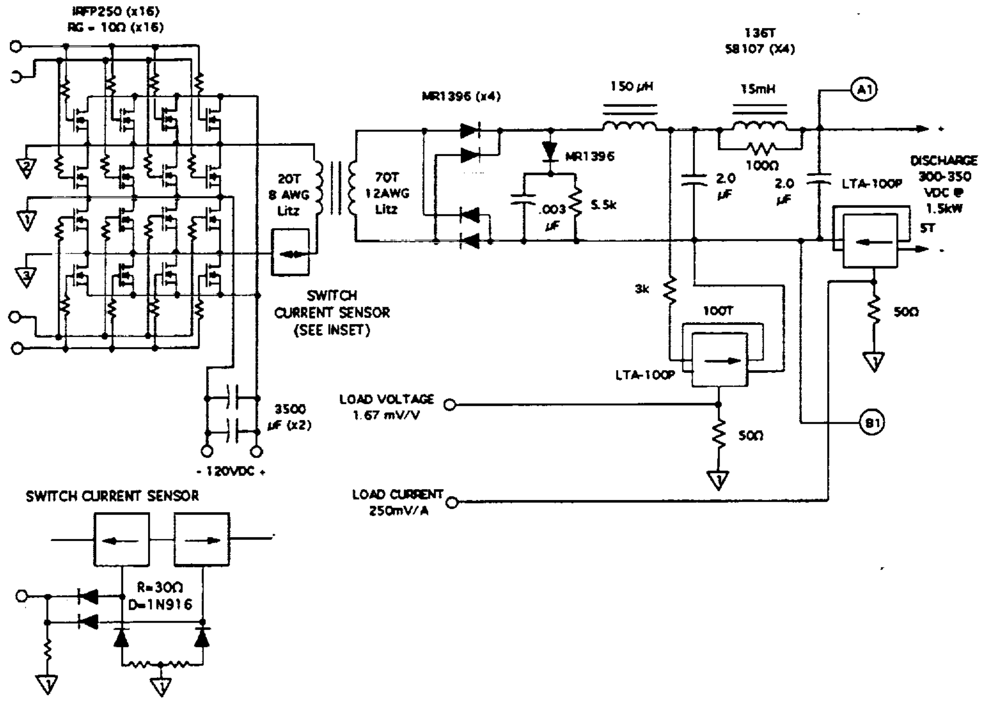

Figure 3. Schematic diagram of the discharge power supply.

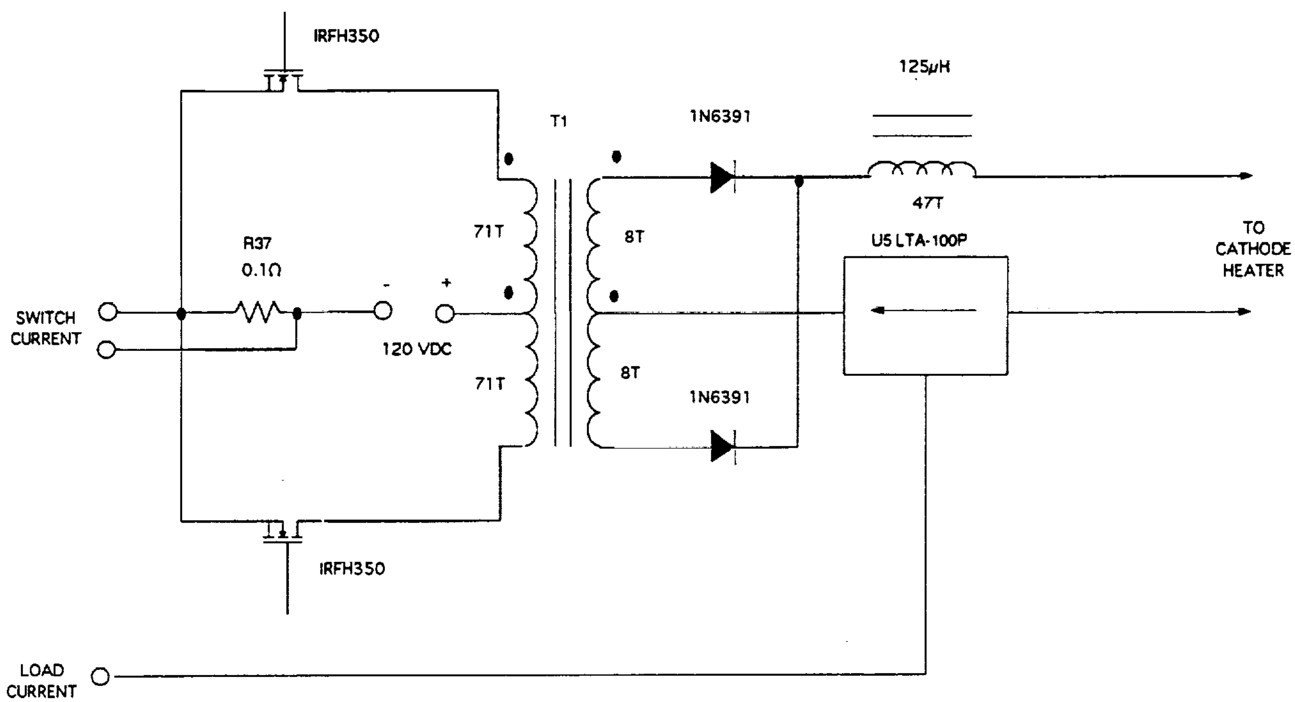

Figure 4. Schematic diagram of the cathode heater power supply.

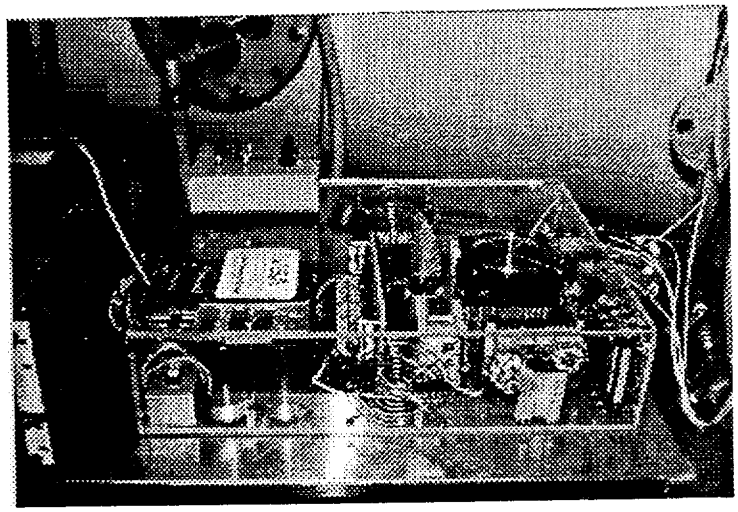

Figure 5. Photograph of completed SPT-100 PPU

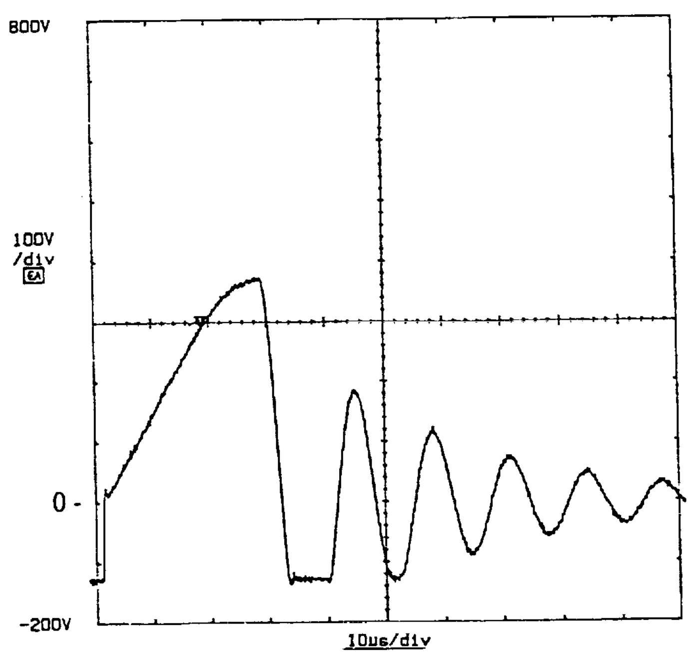

Figure 6. Pulse ignitor output.

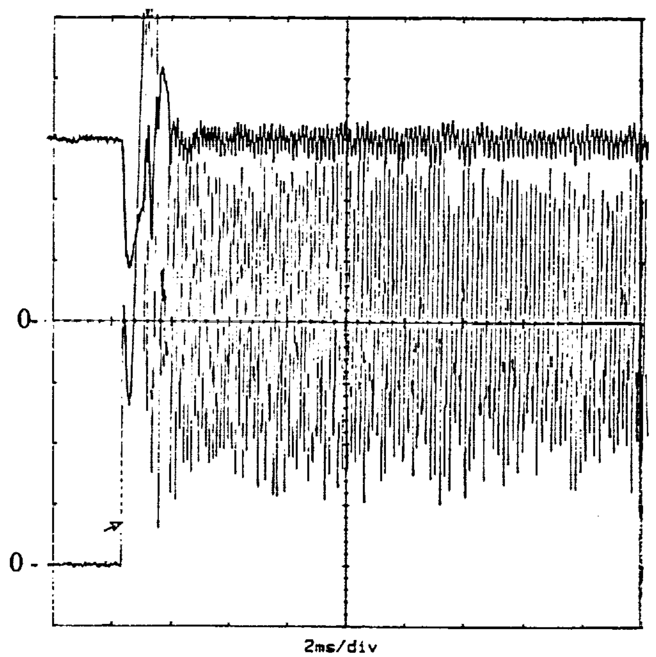

Figure 7. Typical thruster start. Upper trace: Discharge voltage, 100 V / div.
Lower trace: Discharge current, 1 A / Div.

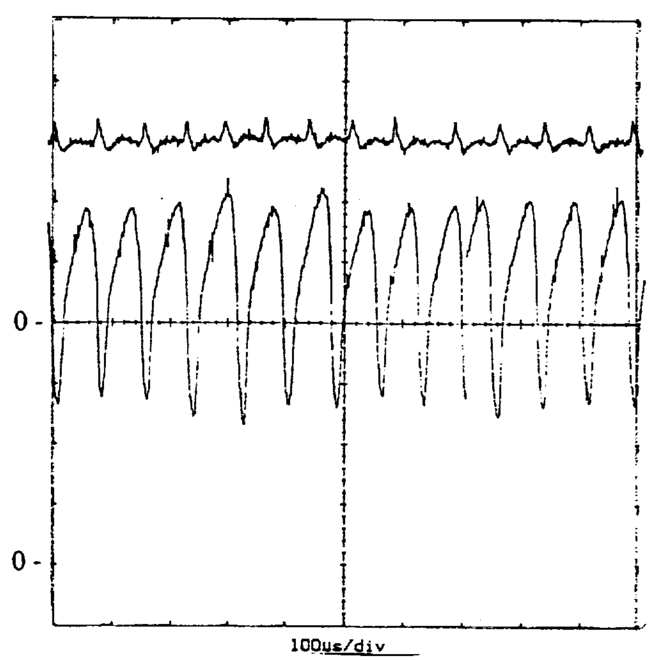

Figure 8. SPT operation with discharge current oscillations. Upper trace: Discharge voltage, 100 V / div.
Lower trace: Discharge current, 1 A / div.

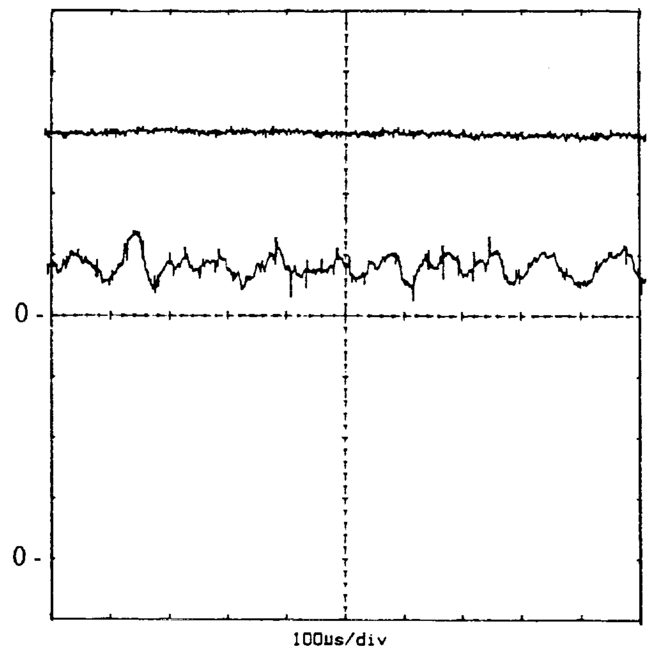

Figure 9. Stable SPT operation: Upper trace: Discharge Voltage, 100 V / div.
Lower trace: Discharge current, 1 A / div.

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13. ABSTRACT ***(Maximum 200 words)***

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14. SUBJECT TERMS

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