RHETTWEPDM DEVELOPMENT TESTING

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ABSTRACT

As part of the development of the Thruster with Anode Layer for the RHETTWEPDM flight, several issues were addressed including, plume impacts, performance, and thermal characterization. The plume profile was found to be similar to other 1.5 kW-class Hall thrusters, which provided a spacecraft impacts database. The performance was characterized for firings down to 30 s duration. At some conditions the thruster exhibited a phenomenon in which the current was initially elevated and decreased in a stepwise fashion by up to 26% several seconds after ignition with negligible effect on thrust. In general the thruster exhibited steady, repeatable operation with performance ranging from 1270 s specific impulse at 600 W to 1440 s at 900 W. Thermal profiles of the thruster were obtained to aid in model thermal model for spacecraft integration development. Finally, the thruster was subjected to operation at -90°C to verify operation in the on-orbit environment.

INTRODUCTION

The major objective of the electric propulsion program sponsored by the Science and Technology Directorate of the BMDO is the development of innovative, high-performance propulsion technology for applications such as orbit insertion, repositioning, and stationkeeping for a broad range of government and commercial space missions. In 1990, the BMDO took the lead in identifying advanced spacecraft propulsion technology developed in the Former Soviet Union with potential applications for US government and commercial missions. Russian Hall thruster technology was identified as a technology with potentially high impact. A government team evaluated the technology in Russia and recommended further life evaluation and integration testing in the US. Three versions of the 1.5 kW Hall thruster were obtained for evaluation: (1) SPT-100, (2) T-100, and (3) TAL D-55. Program focus over the past three years has been on propulsion system development and transfer of the technology to the US user community under the NASA Lewis managed Russian Hall Effect Thruster Technology (RHETT) program. The program has a broad base of contributors with efforts supported by US industry and academia, several Russian institutes, NASA Lewis, JPL, and NRL. RHETT1 completed in early 1996 was a Hall thruster propulsion pallet ground demonstration. RHETT2 was the technology program providing Hall thruster propulsion system hardware to the NRL Electric Propulsion Demonstration Module (EPDM) flight. EPDM will be the first Western flight of a Hall thruster system with a mix of Russian and US technologies.

The RHETT2/EPDM programs selected the Thruster w/Anode Layer model D-55 (Figure 1) version of the Hall thruster for development and qualification. During the RHETT2 flight hardware development several open technical issues were addressed to enable successful integration with the spacecraft. Foremost among these were issues associated with interactions of the thruster plume with spacecraft surfaces. Further, thruster performance, including stability at derated power conditions and the repeatability of the impulse bit during short firing durations needed to be quantified across a range of powers. Finally, thruster thermal characteristics needed to be defined. Thermal issues included obtaining detailed information on the thruster thermal response while in a flight representative mounting configuration and understanding the ability of the thruster to operate at low temperatures expected on the mission. The
results of those development tests are summarized in this paper.

PLUME INTERACTIONS

Interactions of the Hall thruster plume with spacecraft surfaces has been a concern since the inception of the RHET2 program. High velocity ions exiting the thrusters at nominally 16,000 m/s can have a significant effect on solar arrays and other spacecraft surfaces. Early technology evaluation work investigated the effects of the SPT-100 plume with solar array materials including cover slides and interconnect materials. The results showed an area of high erosion ~65° from the thruster centerline, a contamination zone between 70 and 80° where material from the thruster was deposited, and a zone of minimal impact (>80°). Measurements of the ion plume density were obtained for the TAL D-55 and showed that the far-field plume was very similar in structure to other Hall thrusters, as shown in Figure 2. From those data it was determined that the earlier testing on SPT-100 was likely directly transportable and formed a database for model development.

The initial application of the RHET2 hardware was proposed on the Wake Shield Facility (WSF) 3 flight which is released from the Shuttle by the Remote Manipulator System. For that flight opportunity the thruster was to have been oriented as shown in Figure 3. The thruster would have been operated at 4 A and 300 V (near the design current level of 4.5 A) for a total of three hours while attached to the robotic arm. The effect of the plume on the arm material was of paramount concern. Some xenon ion flux was predicted to teach the arm and cause minimal erosion. The material eroded would consist of fluorinated ethylene propylene (FEP) of the beta cloth covering and aluminum from the WSF structure. The amount of material sputtered was predicted by using experiment correlations provided in Ref. 4. The projected depth of material etched is presented in Figure 4. The etch rate drops off rapidly with distance and angle (e.g. it decreases from 2160 microns/h at 0.4 m to 9 microns/h at 2.7 m). The three hours of thruster operation would nominally etch 0.8 microns of FEP on the robotic arm. Using data obtained on the atomic oxygen fluence from the STS-46 flight and the atomic oxygen erosion rates of FEP in low Earth orbit, it was determined that 3h of thruster operation would be equivalent to 60 h atomic oxygen exposure. The WSF did not materialize, but the effort was rolled into the EPDM flight program. The former analysis helped to demonstrate that although the impact of the plume on spacecraft surfaces can be severe, proper orientation of the thruster on the spacecraft can result in minimal impacts.

PERFORMANCE

Performance characterization of early versions of the TAL under high fidelity vacuum conditions was one of the first tests conducted during the evaluation stage and showed a significant range of power throttling capability and a predicted life of over 5000 h. The EPDM mission planned to operate the thruster in a derated power condition from 600 to 900 W. Simply reducing the flow rate at 300 V to reduce the current caused discharge current instabilities. Tests were conducted to determine a magnetic field level to allow lower power operation. It was determined that a reduction in the field strength of 37.5%, (reducing the magnet current from 4 A to 2.5 A), allowed quiet operation with significant margin between 600-900 W. A oscilloscope trace of the discharge voltage and current at the 600 W (2 A/300V) operating point is presented in Figure 5.

Because the EPDM mission required precise, short-duration firings, impulse-bit measurements as short as 30 s were made. These were done in addition to defining the steady-state performance at the derated EPDM power range.

All testing was performed on the Engineering Model engine which was identical in form and function to the two flight units. The performance of thruster at the EPDM operating range was measured using the same apparatus and techniques used previously. Thrust measurements were performed in the large space simulation chamber at NASA LeRC at background pressures below 3 x 10^-4 Torr (measured using ionization gauges calibrated on air and corrected for xenon) during thruster operation. The power system to operate the thruster was similar to that used previously. The cathode was an engineering model of the flight unit described in a companion paper.

The results of the testing are presented in Table I. At the nominal operating point (1386 W) the thruster provided 88.6 mN of thrust with a corresponding specific impulse of 1640 s. Performance at derated power conditions was lower, but sufficient for the EPDM, ranging from 1270 s specific impulse at 600 W to 1440 s at 900 W for those cases the cathode flow was kept at the same value as the nominal
condition, 0.55 mg/s, and the thruster was operating at steady-state.

The effects of short duration operation were also investigated. Similar anode flow rates of 2.66, 3.17, and 3.49 mg/s, which provided steady-state power levels of 600 W, 750 W, and 900 W, respectively, were used for firings of 30 s. Plots of thrust and discharge current versus time for a 30 s burn obtained with a strip chart recorder at the 2.66 mg/s anode flow setting is provided in Figure 6. The figure shows that the thrust has a top-hat appearance with a rise time <1 s. Similar results were obtained at the other flow settings. From Table I it can be seen that the current for the 30 s firings is higher than the steady-state values, and leads to slightly elevated thrust levels. The slightly elevated current at start-up is a phenomena noted during performance testing and is thought to be related to the thermal characteristics of the system. The current was found to steadily decrease by a few tenths of an ampere over tens of minutes until the thruster reached steady-state temperature. Another phenomena was also noted at the high end of the power range when the thruster was at ambient temperatures or below and consisted of a large shift in current level several seconds into the run. With the thruster operating at room temperature and at 3.49 mg/s of anode flow the initial current was 4.45 A and dropped to 3.28 A within 21 s in a stepwise fashion as shown in Figure 7. Repeated starts showed the same phenomenon until the thruster neared thermal equilibrium. The increased current did not contribute to increased thrust as shown in Figure 7(a). (note: the decreased voltage at the 4.45 A current level was due to the ballast resistance in the discharge power line.) The physics behind the phenomenon are not clear; however, the result is of concern for spacecraft integration, due to increased current draw from the power bus.

**THERMAL CHARACTERIZATION**

In order to model the thermal effects of the thruster on the spacecraft, temperature profiles of the thruster were obtained. For this, the thruster was held on a flight representative mount constructed from aluminum. The thruster was instrumented along with the cathode with eight thermocouples. The assembly was attached to a water-cooled plate which was maintained at a constant 19°C. The thruster was fired at the 3.49 mg/s anode flow condition for test durations ranging from 30 s to 3 h. No noticeable thermal effect was observed in the 30 s run. The results of the long duration run are provided in Figure 8. For that run the discharge current was decreased to 2.68 A by the end of the run, significantly lower than the 3.1 A recorded at then beginning. The thermal data acquired was used to construct and validate a thermal model by NRL for the spaceflight environment conditions.

Another thermal concern was with the ability of the thruster to start at the cold temperatures anticipated for the EPDM flight. The low temperature operating specification was set at -60°C with external heaters and -90°C with no heating. Because the TAL body is at cathode potential (isolated from the spacecraft body), attaching heaters to the thruster is a reliability risk. If a heater should fail, the thruster body and hence the cathode could potentially be shorted to the spacecraft ground; therefore, the use of heaters on the thruster was to be avoided. The thruster and cathode were mounted in Vacuum Facility 5 in a liquid nitrogen cooled thermal shroud as shown in Figure 9. A laboratory feed system and power supplies were used to operate the thruster. During the testing several problems with freezing of the xenon in the propellant lines were encountered. This required addition of line heaters which are not a problem since the propellant system is at spacecraft potential.

The thruster was successfully cooled to -90°C and fired. Similar to the other development tests, the cold-soak test was also conducted at the 3.5 mg/s anode flow condition. The parameter of interest was the discharge current, and a plot of current versus time is shown in Figure 10. Similar to the 30 s burn conducted at ambient conditions presented in Fig. 7(b), the -90°C run showed a stepwise change in current from 4.3 A down to 3.1 A in approximately 1 min. The current then steadily decreased throughout the run until a current excursion was encountered. It is believed that that excursion was caused by external conditions, and was not related to normal thruster operation. Several minutes prior to the incident, the liquid nitrogen to the thermal shroud was shut off in order to allow the temperature of the thruster to rise more quickly. Throughout the test series problems were encountered with xenon propellant freezing in the lines. It is hypothesized that as the shroud warmed so did a section of propellant line containing frozen propellant. The additional propellant caused the temporary rise in discharge current. Also included in Figure 9 is a plot of a subsequent run. As expected with a the second run did not experience a stepwise current change after ignition because the thruster was warm; however, the current was initially...
CONCLUDING REMARKS

Low-power Hall thruster technology evaluation has been ongoing in the United States for several years. The technology has matured to the point of being selected for use on a US Government spacecraft through the RHETT/EPDM initiatives. The thruster chosen for flight qualification and demonstration was the Thruster with Anode Layer model D-55. In preparation for flight, further development testing beyond that completed under the RHETT technology evaluation phase, was required. Three major areas of concern were addressed, including plume interactions, performance, and thermal characterization.

Plume characterization was completed in an early program phase and showed similarity with the T-100 and the SPT-100. The SPT-100 data was assumed directly transportable and formed a database for simple model development. Analysis was completed to compare the effects of the thruster plume on LEO spacecraft surfaces. In one example investigated, the effects of erosion from the plume (with short exposure times) were comparable to atomic oxygen degradation (continuous). With careful positioning of the thruster on the spacecraft, effects from the plume can result in minimal impacts.

The performance of the thruster was evaluated for short duration operation, typical of the EPDM mission. The thruster impulse-bit showed rapid rise and decay times (<1 s). At ambient and below operating temperatures and at the high end of the 600 W-900W power range, the thruster exhibited a repeatable mode change characterized by an initially elevated current level decreasing in a stepwise fashion by over 26% several seconds into the run. The phenomenon had negligible effect on thrust. The thruster also exhibited slow monotonic decrease in current (on the order of a few tenths of an ampere) during a long duration operation with a slight decrease in thrust.

The thermal characteristics of the thruster were defined for firing times ranging from 30 s to 3h in order to provide data for the development of a thermal model to aid in spacecraft integration. Initial results showed the thruster to be subjected to cold temperatures and required verifying thruster operation at -90 °C. The thruster successfully operated at that condition, but again exhibited a 26% step change in current during the first several minutes of operation along with an additional drop of 14% during the 90 minute run.

REFERENCES


ACKNOWLEDGMENTS

The help of Mr. Thomas Haag (NASA LeRC) in obtaining the thrust measurements and Messrs. Eli Green, Fred Jent, (NASA LeRC) and James Parkes (NYMA) for expert technical support is greatly appreciated.
### Table 1. Thruster with Anode Layer model D-55 performance

<table>
<thead>
<tr>
<th>Power (W)</th>
<th>Current (A)</th>
<th>Thrust (mN)</th>
<th>Anode Flow (m/s)</th>
<th>Cathode Flow (m/s)</th>
<th>Specific Imp. (s)</th>
<th>Efficiency</th>
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<tr>
<td>1388</td>
<td>4.0</td>
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<td>599</td>
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<td>1271</td>
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<tr>
<td>750</td>
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<td>50.2</td>
<td>3.17</td>
<td>0.55</td>
<td>1377</td>
<td>0.45</td>
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<tr>
<td>902</td>
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<td>1444</td>
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</tbody>
</table>

- 30 s Impulse-bit: warm thruster
- 50 s Impulse-bit: room temp. thruster
- Part 1
- Part 2

![Figure 1(a) Front view](image1)

Figure 1(a) Front view

![Figure 1(b) Side view](image2)

Figure 1(b) Side view

Figure 1. Thruster with Anode Layer model D-55 manufactured by TsNIIMash, Korolev, Russia

![Figure 2. Ion current density profiles for 1.5 kW-class Hall thrusters from Ref. 3](image3)

Figure 2. Ion current density profiles for 1.5 kW-class Hall thrusters from Ref. 3
Figure 3. Plume density contours for RHETT2 D-55 TAL mounted on Wake Shield Facility

Figure 4. Predicted depth of material sputtered by 3h of RHETT2 D-55 TAL operation at 1300 W
Figure 5. Oscillograph of discharge current and voltage of D-55 operating at 600 W with off-nominal magnet current level of 2.5 A. Time scale is 79.35 μs/div.

Figure 6(a). Thrust versus time

Figure 6(b). Discharge current versus time

Figure 6. TAL D-55 operating characteristics for a nominal 30 s firing duration at 669 W (anode flow rate 2.66 mg/s) and thruster warm prior to ignition (cont.).

Figure 7(a). Thrust versus time

Figure 7(b). Discharge current versus time

Figure 7(c). Discharge voltage versus time

Figure 7. TAL D-55 operating characteristics for a nominal 30 s firing duration at 1250/977 W (anode flow rate 3.49 mg/s) and thruster cold prior to ignition (complete).
Figure 8(a). Thermocouple location

Figure 8(b). Temperature profiles

Figure 9. Thermal characteristics of TAL D-55 at nominally 900 W

Figure 10. Thruster operation after cold soak to -90°C (300 V discharge voltage)

Figure 9. Test set-up for cold soak validation at NASA LeRC