Design, Development and Operation of a Laboratory Pulsed Plasma Thruster for the First Time in West Asia

By Abdolrahim REZAEIHA,1,2) Mehdi ANBARLOUI2) and Mohammad FARSHCHI1)

1) Sharif University of Technology, Tehran, Iran
2) Iran Space Research Institute (ISRI), Tehran, Iran

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Although the pulsed plasma thruster (PPT) was first utilized on a space mission in 1964, after more than four decades, it is still a space-rated technology which has performed various propulsion tasks, from station-keeping to three-axis attitude control for a variety of former missions. With respect to the rapid growth in the small satellite community and the growing interest for smaller satellites in recent years, the PPT is one of the promising electric propulsion devices for small satellites (e.g., CubeSats) due to the following advantages: simplicity, lightweight, robustness, low power consumption, low production cost and small dimensions. Therefore, a laboratory benchmark rectangular breech-fed pulsed plasma thruster using a self-inductor as a coupling element was designed, developed and successfully tested in a bell-type vacuum chamber at 10^{-4} Pa for the first time in west Asia (Iran). The PPT has been tested using a 35 μF, 2.5 kV oil-filled capacitor, producing an impulse bit varying from 300 μN·s to 1.3 mN·s at a maximum specific impulse of 1100 s. As a result a research program in Iran was initiated for working on PPTs and the miniaturization of PPTs while increasing the performance parameters. The present paper briefly reviews the PPT design and development.

Key Words: Pulsed Plasma Thruster, Design and Development, Operation, West Asia

Nomenclature

- \( \mu_0 \): magnetic permeability
- \( h \): distance between electrodes
- \( w \): electrode width
- \( V_0 \): capacitor voltage
- \( i \): discharge current
- \( E \): discharge energy
- \( L_t \): nozzle inductance gradient
- \( t \): time
- \( I_{\text{bit}} \): impulse bit
- \( I_{\text{sp}} \): specific impulse
- \( h \): thruster efficiency
- \( M_{\text{bit}} \): mass per shot

1. Introduction

There has been a growing interest within the space sector to develop smaller satellites, which reduces cost and development time. This trend has been followed by many universities worldwide actively participating in the development of small satellites. CubeSats are the focus of many studies as they offer the most demanding constraints for different subsystems in terms of power and mass. At the same time, their complex mission tasks make active attitude and orbit control a necessity. Therefore, they require propulsion systems which meet the performance requirements of these missions, whilst conforming to the stringent mass and power constraints imposed by satellites with a mass of less than 100 kg is crucial. This class of satellites may perform propulsive maneuvers including formation flying, satellite inspection, drag compensation, station-keeping and attitude control in future missions. The maximum velocity change requirement for these missions assuming a duration of 6 to 12 months is 300 m/s, which is within the expected performance range of PPTs. Mission analysis studies show that the use of onboard propulsion compared to reaction wheels or passive magnetic attitude control can dramatically increase mission capabilities for microsatellites.1)

Development of missions for small satellites has reinitiated interest in ablation-fed pulsed plasma microthrusters (μPPT). This interest stems from the ability of the PPT to operate at very low power levels, even at input powers of less than 10 W, while having low mass and size compared to other propulsion systems.2) The many other benefits of PPTs listed below are also very persuasive for designers:1)

1) Zero warm-up time, zero standby power.
2) Inert and fail-safe—no unpowered torques or forces.
3) Scaleable to performance requirements.
4) Usable on spinning or three-axis stabilized satellites.
5) Solid propellant advantages: no tankage, feedlines, seals, mechanical valves, easily measured propellant consumption, zero-g, cryogenic, vacuum compatible, noncorrosive, nontoxic, long shelf life,
not affected by rapid temperature changes, not
affected by variable high “g” loads.
6) Discreet impulse bits compatible with digital logic.
7) Variable thrust level.
8) Performance compatible with attitude control and
station-keeping requirements.
9) Operation at large variation in environmental
temperature.
10) Thrust vector control capability.
Although no propulsion system has yet been able to
completely meet the requirements of microsatellites; the
PPT is one of the promising propulsion systems which has
the potential to do so. The technical development areas for
small PPTs include reduction in mass and power, and
optimization of performance.
PPTs have been utilized in space missions since 1964,
and after more than four decades, they are still a
space-rated technology which has performed various
propulsion tasks, from station-keeping tasks to three-axis
attitude control for a variety of former missions. PPTs are
categorized according to their geometry and feeding
method as shown in Table 1.

Table 1  PPT types.

<table>
<thead>
<tr>
<th>Geometry</th>
<th>Feeding method</th>
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<tbody>
<tr>
<td>Rectangular</td>
<td>Side-fed</td>
</tr>
<tr>
<td>Coaxial</td>
<td>Breech-fed</td>
</tr>
<tr>
<td>Z-pinch</td>
<td></td>
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</table>

Figure 1 shows the schematic of a typical rectangular
breech-fed PPT using Teflon® (PTFE) as solid propellant.
Although gas-fed PPTs and liquid-fed the PPTs have been
tested successfully in laboratory environments, Teflon is
the propellant of choice for space missions. The use of a
solid propellant avoids using a complex feeding system as
the system has only one moving part; thus, the system
becomes simple and robust.

Fig. 1.  PPT schematic.

Altogether, with respect to the many advantages of
PPTs for microsatellites, a laboratory benchmark
rectangular breech-fed PPT using a self-inductor as a
coupling element has been designed and developed for the
first time in west Asia, and its performance was
investigated while varying the PPT discharge energy over
a wide range. The PPT main capacitor, which is a 35 μF,
2.5 kV oil-filled capacitor, was charged with a wide range
of voltages, ranging from 250 to 1750 V, making the
system stored energy range from less than 1 to 60 J, and
producing an impulse bit varying from 30 μN-s to 1.3
mN-s.

2.  Experimental Facilities

2.1 Vacuum chamber
PPT experiments were performed in a mid-sized
high-vacuum facility capable of achieving a chamber
pressure of $10^{-4}$ Pa while the thruster is working. The
bell-type vacuum chamber has dimensions of 0.4 m in
diameter and 0.4 m in length. It is evacuated by an oil
diffusion pump in conjunction with a rotary centrifugal
pump, while the pressure is monitored on different gauges.
The chamber is equipped with a number of feed-through
flanges and a Plexiglas window for visual inspection of
the PPT.

2.2 High-voltage probes
Two high-voltage probes capable of transmitting high
voltage of up to 15 kV to the oscilloscope with a
reduction ratio of 100:1 were used to record the PPT
capacitor discharge voltage and the PPT igniter plug arc
voltage.

2.3 Rogowski coil
A Rogowski coil was needed to record the PPT
discharge current pulse and to calculate the impulse bit of
the thruster. Therefore, a Rogowski coil with a peak
current measurement of 60 kA is used in the tests.

2.4 Power supply and digital oscilloscope
A 750 W power supply was used to power the dc-dc
boost converters used to convert the 24 V input power
from the power supply to the desired voltage to charge the
main capacitor and the PPT discharge initiating circuit. A
four-channel digital oscilloscope was used to record three
signals coming from the thruster.

3.  Laboratory Benchmark PPT

3.1 Electrodes
At the beginning, copper, brass and molybdenum were
considered as the options for the electrode material; but in
the end, a copper anode and cathode set was made. The
anode is 31 mm in width and the cathode is also 31 mm in
width, and they make the PPT nozzle 50 mm in length.
The distance between the electrodes is 31 mm. The anode
electrode has a 1.5 mm-deep shoulder to retain the Teflon
bar, and the cathode has a 12.7 mm hole for the igniter
plug location. Figure 2 shows a picture of the anode and
cathode.

3.2 Propellant
The propellant bar is 31 mm in width and 31 mm in
height, made of “Polytetrafluoroethylene,” or PTFE, and the propellant face is 9.61 cm$^2$. The propellant feed assembly is a spring which pushes the fuel bar against the shoulder to keep the distance between the propellant face and thrust chamber constant. Figure 3 shows a picture of the propellant bar.

3.3 Energy storage device
An oil-filled capacitor with a capacitance value of 40 μF (actual capacitance measured is 35 μF) and a 2.5 kV maximum voltage rating was used in the PPT system. The cylindrical capacitor has a diameter of 10 cm and length of 16 cm, and weighs about 1.75 kg. Figure 4 shows a picture of the capacitor.

3.4 Igniter plug
An annular semiconductor igniter plug with a 2 mm-diameter center electrode is used to produce a plasma puff to initiate the capacitor discharge current between the electrodes in the vacuum (Fig. 5). The plug is located inside the PPT cathode, while its cathode is electrically isolated from the thruster cathode. The igniter plug cathode is connected to the thruster cathode via a 270 μH inductor. The inductor is used to decrease the coupling current flowing from the thruster cathode to the plug cathode as a result of discharge chamber arc attachment to the plug face, which has a strong bearing on the accumulated plug deposit. The value of inductance was chosen according to the results of studies made by Graeme Aston and Lewis Pless shown in Fig. 6.

3.5 Discharge initiating circuit
The discharge initiating circuit has been designed and developed as a self-contained module as it receives 24 V DC input power from the Channel 1 power supply. Then, using a boost converter, it increases the voltage to 500 V DC, which directly transmitted to charge a 1 μF, 600 V capacitor. The capacitor is then discharged to the primary circuit of a step-up impulse transformer with a 1:3 ratio via an isolated gate bipolar transistor (IGBT) switch. The 1500 V current pulse coming from the secondary circuit of the impulse transformer fires the igniter plug. The voltage pulse of the individual igniter plug test (i.e., conducted when not installed in the PPT) under standard atmosphere conditions and at a vacuum pressure of $10^{-4}$ Pa are shown in Fig. 7, and a picture of igniter plug spark inside vacuum chamber can be seen in Fig. 8. Figure 9 shows the ignition circuit design.

The selection of a highly reliable, low-mass, high-energy switching device for triggering the discharge initiation circuits was a significant design challenge. Several different types of devices were considered, including silicon-controlled rectifiers (SCRs), power transistors, power metal-oxide semiconductor field-effect transistors (MOSFETs), and IGBTs. The original PPT design for Lincoln Experimental Satellite (LES) 8 and 9 used SCRs. The power transistors were ruled out because of excessive base drive requirements. The MOSFETs were ruled out because of power and peak current limitations. The SCRs have the advantages of flight heritage and a higher resistance to radiation because of metal packaging. However, they are prone to latch up failure, they have an electrically hot case in a configuration that is difficult to

Fig. 2. Photograph of anode (top) and cathode (bottom) made of copper.

Fig. 3. Teflon fuel bar.

Fig. 4. The 35 μF, 2.5 kV capacitor.

Fig. 5. Igniter plug cross-section.
integrate on a low profile board, and have significantly higher mass than IGBTs. IGBTs were selected because they offer the following advantages over other devices:

- Higher peak current capacity, which maximizes spark plug peak voltage.
- Readily available in 1200 V configuration, which was almost twice the rating of other devices.
- Smallest size and mass.
- Latch proof design, yielding higher system reliability.

The energy stored in the discharge initiating circuit is only 0.125 J, while the circuit 1 μF capacitor is charged with 500 V to fire the plug. The voltage across the igniter plug terminals was measured when the plug working pressure varied from 10⁻⁵ Pa (atmospheric pressure) to 10⁻⁴ Pa, and it was observed that the breakdown voltage decreased from 1500 V at atmospheric pressure to 1200 V at 10⁻⁴ Pa.

4. Experimental Results

A schematic of the system used to monitor the PPT current and voltage is shown in Fig. 10. It shows that a resistor is put in series with the main capacitor, which helps to control the capacitor charging time. Apart from the discharge initiating circuit, another boost converter is used to increase the 24 V input power from the power supply to the capacitor desired charging voltage. Its output is adjustable between 500-1750 V.

The PPT discharge current curves are analyzed to provide an estimate of impulse bit, I_bit. The I_bit is related to the discharge current via Eq. 1 and is determined by integrating the discharge current curve using a numerical formula.

\[
I_{bit} = \frac{L'}{2} \int_{0}^{t} i^2 dt
\]

Here, the inductance gradient (L') is approximated by Eq. 2 and expressed in terms of permeability of free space, also known as the magnetic permeability constant (Eq. 3), the electrode separation (h) and electrode width (w).

\[
L' = \mu_0 \frac{h}{w}
\]

\[
\mu_0 = 4\pi \times 10^{-7}
\]

A PPT with an aspect ratio of 1 was investigated. The electrode configuration of h=31 mm and w=31 mm was selected in order to conform to previously tested geometries, and thus provide a basis for comparison with earlier models (Pottinger and Scharlemann, 2007; Benson and Arrington, 1999). The PPT was tested at discharge energies of 54, 39.3, 27.3, 17.5, 9.8, 4.3, 1.09, and 0.7 J. The tests with discharge energies of 4.3, 1.09, 0.7 and even 0.175 J were done only to prove the operation of PPT at these low voltages. The PPT successfully work at
capacitor charge voltages as low as 100 V, but stopped working when the voltage dropped below this and could not perform at 50 V.

$I_{\text{bit}}$ measurements for each discharge energy between 9.8 to 54 J were taken in 10 different tests. The results are shown in Fig. 11. There is almost a linear relationship between impulse bit and discharge energy in this range, as shown in Fig. 11. $I_{\text{bit}}$ measurements for 4.3, 1.09 and 0.7 J are also seen in Fig. 11, but they are not integrated in the curve fitting process. Table 2 shows the average $I_{\text{bit}}$ and related discharge energy for the PPT tested. Each data is the average of 10 data measurements in the tests.

<table>
<thead>
<tr>
<th>Vo (V)</th>
<th>E (J)</th>
<th>$I_{\text{bit}}$ (μN-s)</th>
<th>$I_{\text{sp}}$ (s)</th>
<th>$M_{\text{bit}}$ (μg)</th>
<th>$\eta$</th>
</tr>
</thead>
<tbody>
<tr>
<td>750</td>
<td>9.84</td>
<td>476</td>
<td>200</td>
<td>242</td>
<td>5%</td>
</tr>
<tr>
<td>1000</td>
<td>17.5</td>
<td>663</td>
<td>366</td>
<td>184</td>
<td>7%</td>
</tr>
<tr>
<td>1250</td>
<td>27.3</td>
<td>943</td>
<td>525</td>
<td>183</td>
<td>9%</td>
</tr>
<tr>
<td>1500</td>
<td>39.3</td>
<td>1118</td>
<td>800</td>
<td>142</td>
<td>11%</td>
</tr>
<tr>
<td>1750</td>
<td>54</td>
<td>1323</td>
<td>1100</td>
<td>122</td>
<td>13%</td>
</tr>
</tbody>
</table>

Specific impulse ($I_{\text{sp}}$) is calculated according to Eq. 4, which is taken from Guman, 1976. This equation is valid only for breech-fed PPTs and gives an estimate of the system $I_{\text{sp}}$. $I_{\text{bit}}$ in Eq. 4 is in μlb-s.

\[
I_{\text{sp}} = \frac{560 \times E^{1.6}}{I_{\text{bit}}} \quad (4)
\]

A picture of the PPT in the vacuum chamber is shown in Fig. 12, and Fig. 13 shows a picture of the thruster main discharge that leads to producing thrust. Table 3 shows a comparison of our PPT performance with some other laboratory and flight-proven PPTs.

<table>
<thead>
<tr>
<th>Thruster</th>
<th>E (J)</th>
<th>$I_{\text{sp}}$ (s)</th>
<th>$I_{\text{bit}}$ (μN-s)</th>
<th>$I_{\text{sp}}$/E</th>
<th>$M_{\text{bit}}$/E</th>
<th>$\eta$</th>
</tr>
</thead>
<tbody>
<tr>
<td>LES-6</td>
<td>1.85</td>
<td>300</td>
<td>26</td>
<td>14</td>
<td>4.8</td>
<td>2%</td>
</tr>
<tr>
<td>SMS</td>
<td>8.4</td>
<td>450</td>
<td>133</td>
<td>15</td>
<td>3.4</td>
<td>3.7%</td>
</tr>
<tr>
<td>LES 8/9</td>
<td>20</td>
<td>1000</td>
<td>297</td>
<td>15</td>
<td>1.5</td>
<td>7.4%</td>
</tr>
<tr>
<td>NOVA</td>
<td>20</td>
<td>850</td>
<td>375</td>
<td>19</td>
<td>2.3</td>
<td>7.6%</td>
</tr>
<tr>
<td>Primex-NASA</td>
<td>43</td>
<td>1136</td>
<td>737</td>
<td>17</td>
<td>1.5</td>
<td>9.8%</td>
</tr>
<tr>
<td>Japan Lab</td>
<td>20.4</td>
<td>423</td>
<td>469</td>
<td>15</td>
<td>3.7</td>
<td>3.2%</td>
</tr>
<tr>
<td>China</td>
<td>23.9</td>
<td>990</td>
<td>448</td>
<td>19</td>
<td>1.9</td>
<td>9.3%</td>
</tr>
<tr>
<td>EO-1</td>
<td>24.4</td>
<td>1150</td>
<td>316</td>
<td>13</td>
<td>1.1</td>
<td>7.6%</td>
</tr>
<tr>
<td>Dawgstar</td>
<td>12.5</td>
<td>500</td>
<td>70</td>
<td>5.6</td>
<td>1</td>
<td>1.5%</td>
</tr>
<tr>
<td>Our PPT-1</td>
<td>27.3</td>
<td>525</td>
<td>943</td>
<td>34</td>
<td>6.7</td>
<td>9%</td>
</tr>
<tr>
<td>Our PPT-2</td>
<td>39.3</td>
<td>800</td>
<td>1118</td>
<td>28</td>
<td>3.6</td>
<td>11%</td>
</tr>
</tbody>
</table>

5. Conclusion

With respect to the movement towards smaller satellites, micro-propulsion systems need to be developed. The small size and mass, and low power of PPTs make them one of the best choices as a micro-thruster. In the first step, a laboratory benchmark PPT has been designed, developed and successfully tested at discharge energies from very low (1 J) up to 54 J. It uses a 270 μH self-inductor as a coupling-element to connect the igniter plug cathode to the thruster cathode. The PPT discharge current has been measured and analyzed, and the results show that the $I_{\text{bit}}$ varies from less than 300 μN-s up to more than 1 mN-s, and the $I_{\text{sp}}$ from 200 to 1100 s. This work has initiated a research program on PPTs and issues related to optimizing and miniaturizing them in West Asia and Iran.

References

Fig. 11. PPT impulse bit measured vs. discharge energy.

Fig. 12. PPT installed in the vacuum chamber.

Fig. 13. PPT discharge current while producing thrust shown in picture (right).