Vacuum Arc Thruster for CubeSat Propulsion

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Abstract: The vacuum arc thruster system, which includes an inductive energy storage power processing unit, has been tested in the past and has shown to have a variety of characteristics making it a suitable thruster for small spacecraft. Amongst those characteristics are low system dry mass of less than 300g, the ability to adjust the individual impulse bit from 50nN-s to 30 μN-s by using simple TTL level signals, a thrust-to-power ratio of the order 10μN/W and the ability to run without the need for a DC-DC converter from bus voltages as low as 5VDC. These characteristics have made the vacuum arc thruster the propulsion system of choice for the University of Illinois 2-cube CubeSat (10 x 10 x 20 cm) designed for April 2004 launch. The Illinois Observing NanoSatellite (ION) includes a scientific mission to view the airglow layer of the atmosphere and a CMOS camera for space and Earth photography.

Introduction

Development of nanosatellites is presently a strong interest of the USAF as well as the NASA, DARPA and MDA[1-3]. Spacecraft designs tend towards smaller, less expensive vehicles & distributed functionality. NASA's future vision is one of re-programmable/re-configurable, autonomous systems; small, overlapping instruments; small, inexpensive nano-satellites. Examples include the Nanosatellite Program and the Orion Formation Experiment. This new trend evokes the same advantages that drive the trend in computing towards distributed, parallel computing and the internet. There are already examples of distributed satellite networks, such as TDRSS, Intelsat, GPS, Iridium, Globalstar and SBIRS (high and low). However, while these are groups of satellites designed to accomplish a common goal, they are nevertheless 'non-cooperating'. The new wave of USAF constellations will be groups of vehicles that interact and cooperate to achieve mission goals. In such groups, vehicle pointing and positioning will be managed collectively; fleets will evolve over time, extending and enhancing the overall capabilities; self-controlling vehicles will eliminate the need for extensive ground support. From a programmatic perspective, the concept is to replace multi-instrument observatories with low-cost, short lead-time spacecraft that would allow adaptation to changing conditions. This in turn mitigates the risk that not all formation flying applications provide full programmatic benefits.

As an enabling technique for these missions, new types of micro- and nano-thrusters are required that offer a wide range of impulse bits from nN-s to μN-s at overall thrust efficiencies ~10%, with very low (<1 kg) total thruster and PPU mass. Scaling existing electric propulsion engines such as Xe ion engines and Hall thrusters down to ~1-10W power levels is not practical. The unavoidable overhead mass of propellant tankage, flow controls and plumbing in Xe ion engines and the increasing magnetic field with decreasing size in Hall thrusters makes their overall efficiency unacceptably low at these power levels.

Building upon an earlier development of a vacuum arc thruster [4] we have added a throttle to overcome the limitations of other propulsion systems. By adapting the simple technology of an inductive energy storage circuit PPU and combining it with powerful electronics control we have developed a throttleable vacuum arc thruster that can be remotely adjusted to deliver individual impulse bits from 0.25μNs to 50μNs with repetition rates varying from 1–1000 Hz, while all the time providing an I_{sp} of >1000s. This thruster will be well suited for precision control of spacecraft constellations including those used for optical interferometry, while offering low mass (<200g including the PPU), simplicity (no moving parts), high efficiency (≈10%) and low manufacturing cost. A first in-space test will be performed on board the UIUC Cubesat.

The propulsion system aboard ION will consist of one power-processing unit (PPU) that supplies power to four VAT thruster heads. The VAT thrusters work in a manner similar to a spark plug, with an electric arc being created from anode to cathode. The interaction between electric arc and cathode results in cathode material being ejected from its surface at high velocity. This produces a highly efficient, low-thrust method of propulsion. In the case of ION, the aluminum frame of the satellite will serve as the cathode material (with the electric arc localized to specific areas), effectively utilizing the frame as propellant in a self-consuming fashion.

Space qualification of the system will be accomplished by verification of successful firing using various on-board diagnostics. Validation of this system will enable its future use for attitude control, orbit maintenance, and perhaps even orbital maneuvers such as orbit raising/lowering and de-orbit. Such capability opens the door to missions that were previously impossible for satellites of ION's size, and provides the promise of lower launch costs and increased launch opportunities by allowing CubeSat-class satellites to piggy-back on vehicles with non-ideal orbits, followed by transfer to final desired orbit.

Vacuum Arc Thrusters

The VAT was designed and built based on an inductive energy storage (IES) circuit PPU and simple thruster head geometry. In the PPU, an inductor is charged through a semiconductor switch. When the switch is opened, a voltage peak Ldi/dt is produced, which breaks down the thin metal film coated anode-cathode insulator surface at relatively low voltage levels (≈200V). The typical resistance of this metal film coated insulator surface is ~100 Ω. Porosity of this surface and/or small gaps in the metal film generate micro-plasmas by high electric field breakdown. These micro-plasmas expand into the vacuum and allow current to flow directly from the cathode to the anode along a lower resistance plasma discharge path (~10's of mΩ) than the initial, thin film, surface discharge path. The current that was flowing in the solid-state switch (for ≤ 1 μs) is fully switched to the vacuum arc load. Typical currents of ~100 A (for ~100-500 μs) are conducted with voltages of ~25-30 V. Consequently, most of the magnetic energy stored in the inductor is deposited into the plasma pulse. The efficiency of the PPU may thus be >90%.

Based on this inductive energy storage approach, a PPU was designed to accept external TTL level signals to adjust the energy and the repetition rate of individual plasma pulses. This was accomplished by adjusting the trigger signal to the semiconductor switch. The design of the PPU for the throttleable VAT is based on the basic design of the inductive energy storage circuit as shown in Figure 1.

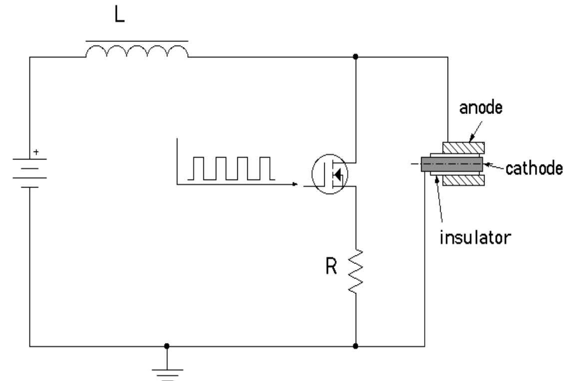

Fig. 1: Equivalent circuit of controllable, Inductive Energy Store (IES) PPU for the VAT.

A semiconductor switch is triggered by an incoming signal represented in the figure as a rectangular signal. By varying the length of the signal, the level of the current in the switch and thereby the energy stored in the inductor can be adjusted. This in turn changes the amount of energy transferred to the arc and the impulse bit of the individual arc pulse.

The repetition rate of the individual pulse can be easily changed by varying the frequency of the input signal.. The arc current waveforms for different inductor charge times are shown in figure 2. The input signals were generated by a simple, 5 V signal generator.

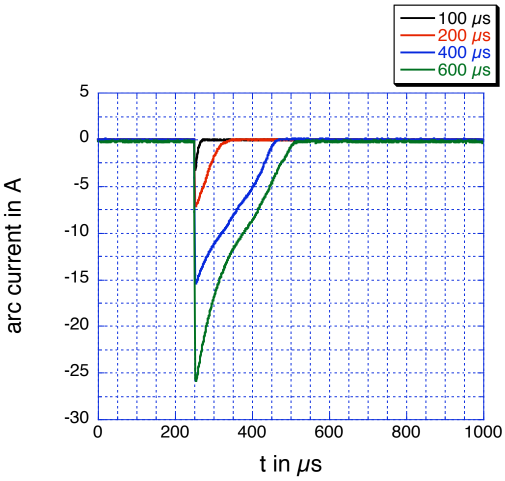

Figure 2: Arc current as a function of inductor charging time. As expected, the arc current and the pulse duration increase with increasing inductor charging time.

<u>VAT performance</u>

The vacuum arc thruster efficiency is defined as the product of the power processor unit (PPU) efficiency and the vacuum arc efficiency:

$$\eta = \eta_{PPU}\eta_{arc}$$

where η_{PPU} is the PPU efficiency and η_{arc} represents the vacuum arc efficiency as defined below.

$$\eta_{PPU} = \frac{V_{arc} \cdot I_{arc}}{V_{PS} \cdot I_{PS}}$$

$$\eta_{arc} = \frac{\dot{m}_{arc} v^2}{2 I_{arc} V_{arc}} \alpha$$

where the "arc" and "PS" subscripts distinguish between the plasma arc and power supply voltage and current parameters respectively. α represents a loss mechanism due to arc divergence and is assumed to be 0.8, *v* represents the plasma streaming velocity and the ion mass flow rate $\dot{m}_{arc}$ [kg/s] can be represented as,

$$\dot{m}_{arc} = r I_{arc} m / Z,$$

where *r* is the ion to arc current ratio, m is the cathode material ion mass, and Z is the effective ion charge state [Z*e] of the discharge plasma.

Based on these calculations we can determine the performance of the VAT by measuring the charge state distribution, the streaming velocity and the arc-to-ion-current ratio. The measurement methods have been described elsewhere [4]. The results for Cromium are shown in table 1.

Table 1: performance

Material	*v* [m/s]	*r*	*Z/e*	η_{arc}
Cr	≈17000	7%	1.5	≈5%

Thrust Measurements

In order to verify predictions made from plasma parameter measurements, thrust and power measurements were conducted using facilities at JPL. The parameters measured were:

- Impulse
- Plasma current – I_{arc}
- Plasma voltage - V_{arc}
- Power supply current - I_{PS}
- Power supply voltage - V_{PS}

Because the individual impulse bit is very small, the TVAT was operated repetitively over 0.5 second periods and the total impulse was integrated over the number of pulses produced during this pulse burst. The TVAT was operated at a variety of conditions:

- The inductor charging time, and therefore the energy stored in the inductor, was varied from 250 μs to 500 μs.
- The repetition rate, and therefore the number of pulses produced during the 0.5 s interval, was varied from 50 Hz to 1kHz
- The bus voltage, and therefore the maximum current that can be conducted through the inductor during a given charging time, was varied from 14 V to 30 V

It was demonstrated impressively [5] how well the VAT can be controlled by either adjusting the charge time, the repetition rate or the supply voltage. The efficiency of the VAT remains constant over the operating range as shown in figure 3.

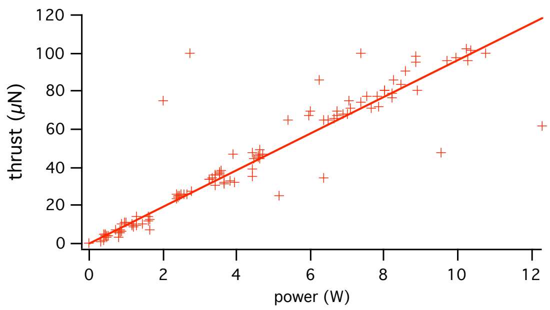

Figure 3: thrust vs. power for Cr TVAT

The data points are on a well defined straight line which indicates that it is not important by which means the power is increased. The slope of the fit and therefore the achievable thrust to power ratio amounts to 9.6 μN/W, which translates into an efficiency of 8.1%. However, during the experiments it was observed that operation simultaneously at high frequency *and* long charging time could not be supported by the power supply. But this is merely a limitation of our available power supply.

To determine the overall system efficiency the energy lost in the PPU has to be taken into account. There are three primary loss mechanisms in the PPU.

- Resistive loss in the coil and in the semiconductor
- Energy left in the inductor at the end of the pulse and its subsequent dissipation
- Energy lost in the trigger electronics

All these loss mechanisms are included when the thrust to power, with respect to the power drawn form the power supply, is computed as shown in figure 4.

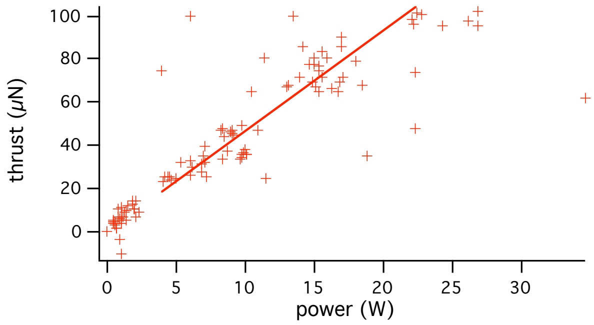

Figure 4: thrust-to-system power

Thrust-to-power in this case amounts to 5μN/W or an efficiency of 4.2%, which implies that the PPU is only ≈ 52 % efficient. This contradicts earlier predictions of ≈90% efficiency. Those earlier measurements and predictions were based on single shot data. Experiments performed under a NASA SBIR (NAS3-02064) confirmed the tendency of decreasing efficiency at high frequency operation. However, very recently this problem has been solved by improving the gate driver for the semiconductor so that now a PPU efficiency of 80% can be achieved.

ION Design [6]

The University of Illinois power processing unit, designed and built by Alameda Applied Sciences Corp. and shown in figure 5 incorporates the same inductive energy storage design. The ION PPU was designed to control 4 thrusters individually. The PPU has a modular design (figure 6) with one main board housing the inductor and timing circuit. Each thruster is connected to a separate control board which houses the semiconductor switch used to control the current through the inductor. PPU size and mass have been driven by the CubeSat requirements and amount to 4 x 4 x 4 cm and 150 g, respectively. The PPU is powered from a 12-24 V power bus.

Four TTL level control signals determine which thrusters are operational. Multiple control signals can be sent in order to fire multiple thrusters simultaneously. In this case, the arc will randomly initiate on one of the two thruster heads. Another way to fire two thrusters simultaneously is to alternate the control signal. In this fashion, the relative power ratio going to each thruster can be controlled.

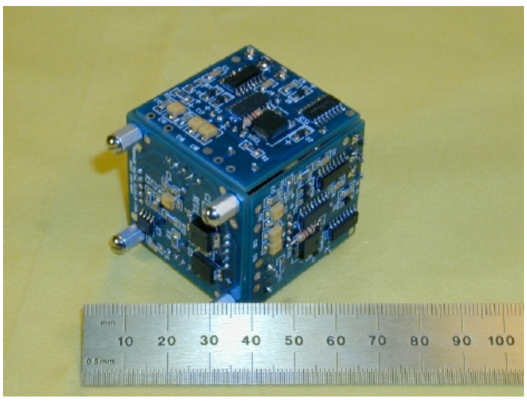

Figure 5 Vacuum Arc Thruster PPU designed for use aboard ION

Figure 6: Vacuum Arc Thruster PPU designed for use aboard ION, Open View

The ION PPU can be fired in two modes of operation. First, an onboard timing circuit can be used to fire the thrusters at a pre-set pulse frequency and power. This mode of operation requires only a single activation control signal. The second mode of operation for the PPU is one where the ION on-board computer sends the PPU a square wave signal generated by a pulse width modulator circuit (PWM) to control the switch for each thruster. In this manner, the computer can control the pulse frequency as well as the energy per pulse.

<u>Thrusters</u>

The original VAT design incorporated a cylindrical thruster. With this design, the thruster was to be placed in the feet of the satellite. It was determined that the cylindrical geometry would require significant insulation, as well as tight tolerance machining to operate reliably. It was also unclear if a cylindrical design would withstand launch vibration. A sandwich or "BLT" geometry was adopted instead. In testing, the BLT geometry proved to be more reliable, as well as easier to manufacture. The first BLT design is a sandwich of copper, ceramic, and titanium. The arc forms between the center (titanium) and outer electrodes. The original BLT design was scaled to a width of approximately 1 cm.

During the initial design, the high density and high I_{sp} of the tungsten electrode were favored. The tungsten could provide the highest DV given a limited mass and volume; however, in order to demonstrate a thruster system with a large volume of fuel for possible future missions, the BLT thrusters were incorporated into the satellite structure in such a way that the aluminum structure became the fuel. The final design of the BLT is shown in figure 7. The anode is separated from the structure (cathode) with two high-alumina ceramic plates. The arc will either attach to the satellite structure or to the aluminum bar used to clamp the BLT together.

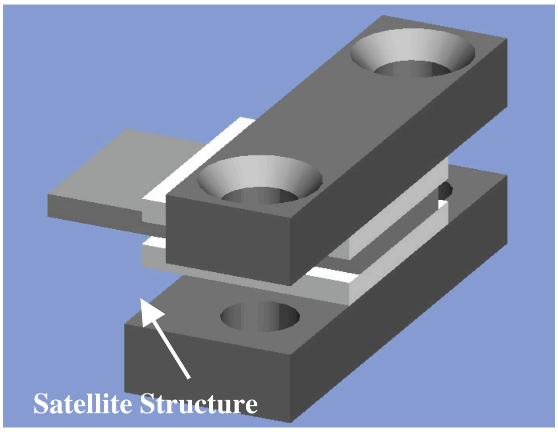

Figure 7: BLT design incorporated into satellite structure.

The four thrusters are located on the satellite in such a way as to allow both translation and 2 axis rotation. Two thrusters are placed on each of the 10 cm x 10 cm faces of the satellite. Each thruster is in the opposite corner of that face as shown in figure 8. With this layout, when thrusters 1 and 2 or thruster 3 and 4 are fired simultaneously, the satellite experiences translation. When thrusters 1 and 3 or thruster 2 and 4 are fired, the satellite experiences rotation about one axis. Firing thrusters 1 and 4 or thrusters 2 and 3 rotates the satellite about another axis.

Due to the location of the thrusters, torque will not be applied through the center of mass, thus likely resulting in something other than pure rotation. The dynamics of this motion are yet to be simulated.

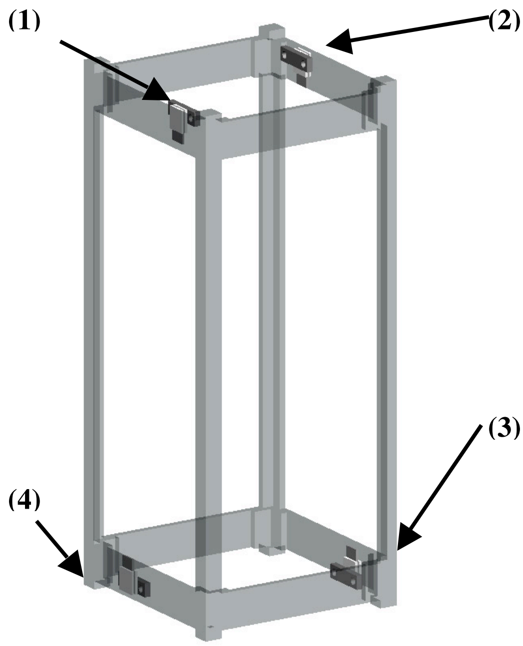

Figure 8 BLT Thruster locations on Satellite

Diagnostics

One issue that has often confronted mission designers is whether the exhaust from thrusters will redeposit on solar panels or optical experiments. Such deposition can reduce solar panel lifetime and efficiency. In order to estimate this effect, a conductive deposition monitor (CDM) experiment will be flown to help determine the amount of deposition accumulating on the satellite due to the thruster.

The CDM is a 1 cm square ceramic plate, with wire leads attached to opposite sides. As the fuel from the thrusters deposits on the ceramic plate, the resistance between the two wires will fall. A circuit measures the resistance between the leads. Since a very small amount of deposition is expected, the circuit is designed to be very sensitive between infinite resistance and 10 MW.

A number of methods have been devised to help verify the operation of the thrusters. At the very basic level, the attitude control system will be able to detect any change in attitude using the on-board magnetometer. Although this is not a direct determination that the thrusters are functioning properly, it is enough to verify that an end has been achieved.

The attitude control system will also be capable of compensating for the thrust with the on-board torque coils. The amount of current through the torque coils necessary to compensate for the thrusters can then be converted into a force or torque produced by the thrusters. This method can be used as an in-flight thrust-stand to measure the exact thrust produced by the thrusters.

The ION computer system is capable of measuring the temperature of the satellite in up to 63 locations. The temperature of the inductor, as well as one of the IGBT switches and it's corresponding thruster will be monitored. This will provide a means to protect the PPU and thrusters from overheating due to normal operation. In case of a failure resulting in a short circuit, the temperature measurement will help to protect and diagnose the system for future operation.

Conclusion

An in-space test of the vacuum arc thruster is planned on board the UIUC cubesat. Ground based measurements have favored the vacuum arc thruster as a propulsion system based on its low mass and simplicity and performance. In space tests will verify the concept of a self-consuming spacecraft while testing basic performance of the thruster and addressing contamination and interface issues.

Acknowledgement:

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