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FINAL REPORT

Performance of Low-Power Pulsed Arcjets

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submitted to:

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March 7, 1995

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Re: NAG3-1360: **FINAL REPORT**

Dear John:

Gary Willmes and I previously promised you a pulsed arcjet final report to present the results for the experiments with a truncated nozzle. This letter describes where we stand today and our near term plans. I am requesting that, based on our significant performance increases, you continue funding for another year, which will enable Gary to finish his Ph. D. thesis work.

As you recall, we had concluded that viscous losses in the nozzle were one of the two dominant loss mechanisms in the pulsed arcjet, the other one being heat transfer losses in the capillary. We have been investigating the effect of area ratio on pulsed arcjet performance based on the theory and experimental observations that viscous losses become more severe as the Reynolds number based on d^* decreases below approximately 300. At these Reynolds numbers, the directed kinetic energy in the propellant is converted to thermal energy by viscous dissipation, and a long nozzle may decelerate rather than accelerate the flow. We wrote a quick-and-dirty quasi-1D steady FORTRAN code to estimate how much to shorten the nozzle. Based on these results, we machined the original 20-degree half-angle conical nozzle from an area ratio of 200 down to 35, followed by an additional decrease to 15. The results from the 35:1 case are attached and show a performance improvement of approximately 50 seconds I_{sp} with the lower area ratio, up to the 240 second range. The second figure shows large efficiency increases at lower specific energy, obtained by increasing the mass flow rate at roughly the same input power, to a maximum of 37%. These results appear to confirm our expectations of higher performance at lower area ratios and higher Reynolds numbers. We expect the 15:1 nozzle to raise the I_{sp} further to the 275 second range, when the tests are completed.

We have encountered a boron nitride material problem involving the dielectric strength of the boron nitride at high temperature. Due to this problem, we have been unable with the present design to obtain good performance measurements for the 15:1 nozzle. Since the shorter truncated nozzle has less radiating surface area, the capillary runs at a higher temperature. We are also running at a slightly higher breakdown voltage, and the combination of higher temperature and voltage has exceeded the capability of the boron nitride to hold off the 2 - 3 kV breakdown voltage. We have tested a number of different combinations of alumina insulator sleeves as a short-term fix but have been unable to obtain long-enough steady state operation for performance measurements due to a limited selection of in-stock material. There are two possible solutions. One is to radius a sharp corner on the nozzle to reduce the breakdown E-field, and this will take 3 - 4 weeks before we can resume testing. Second, we are investigating switching to Si_3N_4. This will take longer.

In parallel with the area ratio investigation, we have implemented a number of system improvements. We have installed a water-cooled enclosure around the thrust stand based on the NASA design in order to eliminate thermal drift. We modified the propellant injection so that the flow enters the capillary axially through the cathode rather than radially through the capillary wall. This reduces blowback and makes the capillary fabrication simpler, and we expect it to improve the heat transfer characteristics in the capillary by eliminating recirculation. Finally, we have designed and installed a simple control circuit for the power supply that allows us to control the pulse rate externally at a desired level. Since the pulse rate is roughly proportional to input power level, this gives us much better control over the input power as well.

We feel that a 2-D, second-order time and space-accurate numerical solution is necessary to describe the post-pulse unsteady, viscous capillary and nozzle flow with heat loss, using the MacCormack finite difference algorithm. Although this work has not been started, the algorithm has been proven elsewhere, and requires only the application of time-dependent B.C.'s.

The higher temperature capillary material will allow us to complete the area ratio investigation, followed by additional experiments to improve the thermal characteristics. We believe that these results will assist the development of other ultra-low-power electrothermal thrusters, which are likely to encounter some of the same loss mechanisms that we are encountering.

In summary, the pulsed arcjet program has demonstrated solid, real achievement on a tight budget. I hope that you will see fit to continue support for another year.

Sincerely,

Rodney L. Burton

Rodney L. Burton
Associate Professor

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Contents

Performance of Low Power Pulsed Arcjets

1.0 Review of Experimental Results

Phase I (September 93 - January 94) The first build of the radiation-cooled pulsed arcjet, including a low-inductance current path, was completed in October 1993. The first phase of testing was successfully conducted through mid-December at UIUC, and the thruster demonstrated stable operation at power levels from 88 to 319 watts. All planned measurements were obtained including breakdown voltage, input power, and discharge pulse shape for two different propellants, $N_2 + 2H_2$ and helium. A number of different capillary lengths, cathode geometry's, and capacitor combinations were tested to determine their effects on thruster electrical characteristics. The ringing effect was reduced from 4-5 rings down to 2-3 rings, indicating a close impedance match between the external circuit and the arc resistance. The thruster met the goal of 30 minutes continuous run time, which was selected as the requirement for thrust stand testing. Performance testing was performed on February 1, 1994 at NASA LeRC with a measured Isp of 176 seconds and 3 percent efficiency.

Phase II (February 94 - August 94) Post-test analysis and inspection of the pulsed arcjet following the NASA LeRC test indicated that damage sustained in shipment had caused leakage around the cathode-capillary seal and ultimately caused loose BN particles to become lodged in the nozzle throat. A design change was made to the seal which increased the cathode diameter from 3.18 mm to 4.76 mm. The experimental performance measurements and calculated numerical results indicated that the thruster thermal efficiency was a significant loss mechanism and needed improvement. To reduce radiation and conduction heat transfer losses, a new thruster body was fabricated with a wall thickness reduced from 6.35 mm to 0.81 mm. A new nozzle was fabricated with a 40 percent reduction in thermal mass and a smaller area ratio, decreased from 250:1 to 150:1. Initial tests at UIUC showed that the thermal characteristics of the thruster had improved because the nozzle temperature reached 1220 K for 125 watts input power. Subsequent performance tests at NASA LeRC gave a measured Isp of 177 to 194 seconds and 3.5 to 4.5 percent efficiency over a range of specific power from 40 to 80 MJ/kg.

Phase III (September 94 - Present) Post-test inspections of the BN capillary revealed that leakage around the cathode-capillary seal was still occurring, most likely due to thermal expansion of the thruster body. A new cathode design was implemented which put the cathode into compression with the capillary to eliminate this leakage path. Visual observations in previous tests had noted significant sparks in the thruster plume, leading to the conclusion that the thruster specific power was too high. Results from the numerical model indicated that heat transfer efficiency would improve at shorter capillary lengths. The UIUC thrust stand was completed and checked out, and the initial performance tests with the modified cathode design showed that cathode-capillary leakage was eliminated. The sparks were eliminated at lower specific powers from 20 to 40 MJ/kg. The measured specific impulse with the short 2.5 mm capillary was from 170 to 180 seconds, and total efficiency increased to a range from 5.3 and 8.2 percent. Performance testing for selected configurations is still in progress.

2.0 Present Experimental Capabilities

The Electric Propulsion Laboratory at UIUC has in place all the capability and diagnostics required for performance testing of low power pulsed and DC arcjets. The UIUC thrust stand is operating with excellent accuracy and sensitivity at very low thrust levels. An important aspect of the experimental setup is the use of a PID controller to maintain a constant thruster position, which reduces hysterisis effects. Electrical noise from the arcjet induces some noise into the thrust signal, but this does not affect the measurement (see Fig. 1). The noise is expected to be eliminated with a low-pass filter or improved shielding. We have also implemented two additional signal conditioning circuits in the data acquisition process - one for breakdown voltage and one for pulse rate - to better track the thruster operating characteristics over a longer time scale than previously done (see Fig. 2 and 3). All appropriate calibrations are either complete or are in the final stages of checkout, including a mass flow controller calibration system which will be completed shortly.

3.0 Analysis of Performance Results

Although we have improved the sealing and thermal characteristics of the arcjet, additional work is still needed to improve the performance, particularly the specific impulse. We have been evaluating our experimental results compared to our numerical heat transfer model. The nozzle temperatures we determine experimentally appear to correspond with the heat transfer trends we calculate numerically. However, when we include estimates of nozzle, frozen flow and external circuit efficiencies, our specific impulse predictions are much higher than we measure experimentally. Accordingly, we have been readdressing some initial assumptions to determine which loss mechanism(s) we are underestimating.

We have tentatively concluded that the nozzle losses may be much greater than we have previously assumed. The two nozzle designs that have been tested are 20 degree cones with area ratios of 250:1 and 150:1. These parameters were selected based on the results of Curran, et al,[1] which showed the 20 degree cone with large area ratios to have slight performance benefits over other combinations of half angle and area ratio. However, a significant difference between the <u>very low</u> power pulsed arcjet at 4 mg/sec flowrate and the <u>low</u> power DC arcjet at 50 mg/sec flowrate is the Reynolds number based on throat diameter. For a typical 1 kW DC arcjet, Re is around 500, but for our 100 watt pulsed arcjet, Re is less than 200. An experiment by Rothe[2] showed that when the Reynolds number drops below about 300, the thrust coefficient becomes < 1, as viscous dissipation converts the directed kinetic energy into random thermal energy, increasing the static temperature and decelerating the flow. A DSMC analysis by Zelesnik[3] gives some numerical results for very low Re nozzle performance with nitrogen propellant. In particular for a case where the gas is at 1000 K and the nozzle wall is at 298 K, the nozzle efficiency defined as Isp(actual)/Isp(ideal) drops below 50 percent. We have recently applied a Navier-Stokes solver to our nozzle design to further substantiate our conclusion. The Navier-Stokes code is currently being developed at UIUC for DC arcjet flows under AFOSR sponsorship. The initial and boundary conditions used are equivalent to our nozzle immediately following a pulse discharge. The results for axial velocity and Mach number (Fig. 4 and 5) show qualitatively that the nozzle is quite possibly *decelerating* the flow and that our thrust coefficient may be approximately 1 or less. This accounts for the difference between our performance predictions and experiment.

A fundamental question regarding the low values of the thrust measurements involves whether the high losses are attributed to the low power, low Re regime in which we are attempting to operate, or whether the problem lies with the pulsed concept itself, or a combination of both. The nozzle loss mechanism would imply that this is a scaling problem likely to be present in all electrothermal devices at this operating condition, and to which our pulsed device is not immune. The thermal efficiency is related to both the small dimensions of the capillary as well as to the high peak temperature following a pulse discharge; however, compared to a DC device, a somewhat lower thermal efficiency could be offset by a higher frozen flow efficiency. To separate the scaling effects from the pulsed effects, we need to operate at a higher power level (and thus higher mass flow rate and Re) in order to determine whether the pulsed concept is feasible at any power and whether the present performance limits are a function of the low power scale.

4.0 Technical Approach for Next Research Phase

We feel that we are at a point in the pulsed arcjet program where we can make rapid progress and produce important results. The radiation-cooled device has matured to the extent that it can be reconfigured rapidly and reliably for different tests. All diagnostics are in order, and a range of numerical tools are in place for design and analysis. Based on the results from Phase III and our recent nozzle analyses, we see several highly promising approaches, and we propose to tackle all of them.

4.1. The scaling effects for very low Reynolds number nozzles must be identified. A very significant result would be experimental data showing a dramatic improvement in performance above a certain critical Reynolds number. Nozzle performance trends can be found in the literature[4] for cold flow nitrogen and hydrogen, but this data cannot be readily extrapolated to flows where the gas total temperature is much higher than the wall temperature, since the wall temperature can substantially affect the final result. Once we determine the scaling effects, we will be able to implement design rules and determine operating points with potential for much higher performance.

4.2. As previously discussed, viscous effects may affect the nozzle performance to the extent that large area ratios and small cone angles are inappropriate in the low Re regime. We feel that smaller area ratios and larger cone angles should be investigated, and we propose an experimental approach combined with a smaller numerical effort to identify these effects. For example, an area ratio of only 10 would give an ideal thrust coefficient of $C_F = 1.5$, without incurring significant viscous losses. We propose to run the pulsed arcjet with several reduced area ratio nozzle configurations at identical operating conditions, power levels, pulse rates, etc. The nozzle radiating surface area will not be significantly reduced so the nozzle temperature will remain reasonably constant.

4.3. The third approach is to increase the thermal efficiency by decreasing the heat transfer loss in the capillary. We feel that the way to address heat loss is by increasing the mean Mach number of the flow where heat addition occurs, thereby decreasing the residence time. We will design the capillary-nozzle interface such that the pulse discharge passes through the throat and attaches in the supersonic region of the nozzle. This is analogous to operating in a constricted mode similar to a conventional DC arcjet. The bulk of the ohmic heating will occur at Mach numbers between 0.5 and 1.0, as opposed to ohmic heating in the $M = 0.0$ to 0.5 range as in the present design.

5.0 Proposed Statement of Work for Very Low Power Pulsed Arcjets

5.1. Run pulsed arcjet with simulated hydrazine at 500 W, 16-32 mg/sec to identify performance trends, specific impulse and efficiency, versus Reynolds number and input power. Run cold flow performance tests in the low Re range to obtain a baseline thrust coefficient versus Re characteristic for simulated hydrazine. Previous testing has been performed in the 80-200 watt, 2-8 mg/sec range.

5.2. Modify the available pulsed arcjet nozzles with smaller area ratio and/or larger cone angle. Run hot and cold performance tests of pulsed arcjet with the modified nozzles. Apply an available Navier-Stokes numerical code to truncated nozzles, including 40:1 and 10:1 area ratios and 20 degree and 40 degree cone angles.

5.3. Fabricate a capillary geometry such that the arc passes through the constrictor and attaches in the supersonic region of the nozzle. Thermal efficiency trends will be identified by measuring nozzle surface temperature in addition to thrust.

6.0 References

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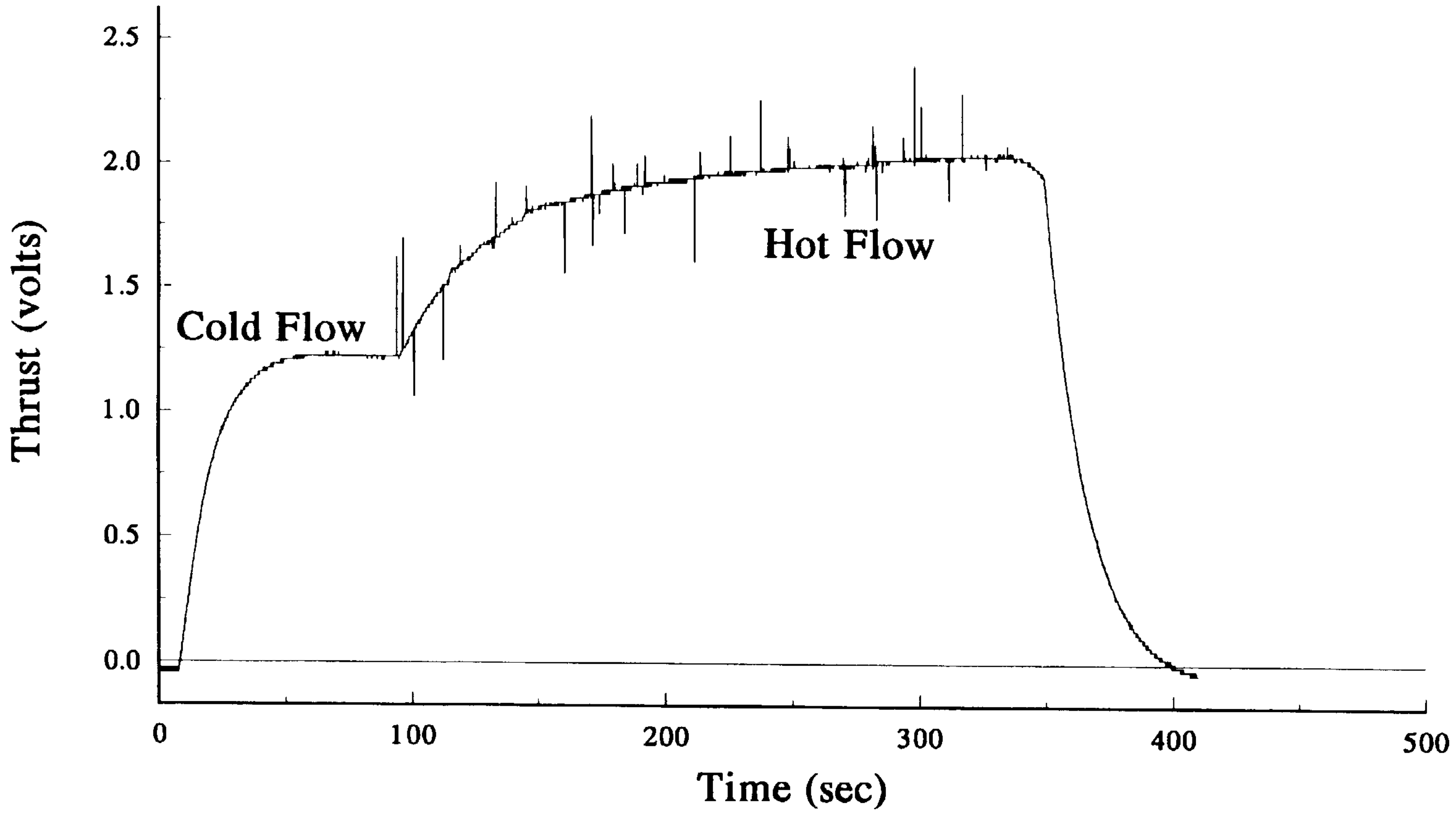

Fig 1. Pulsed Arcjet Thrust

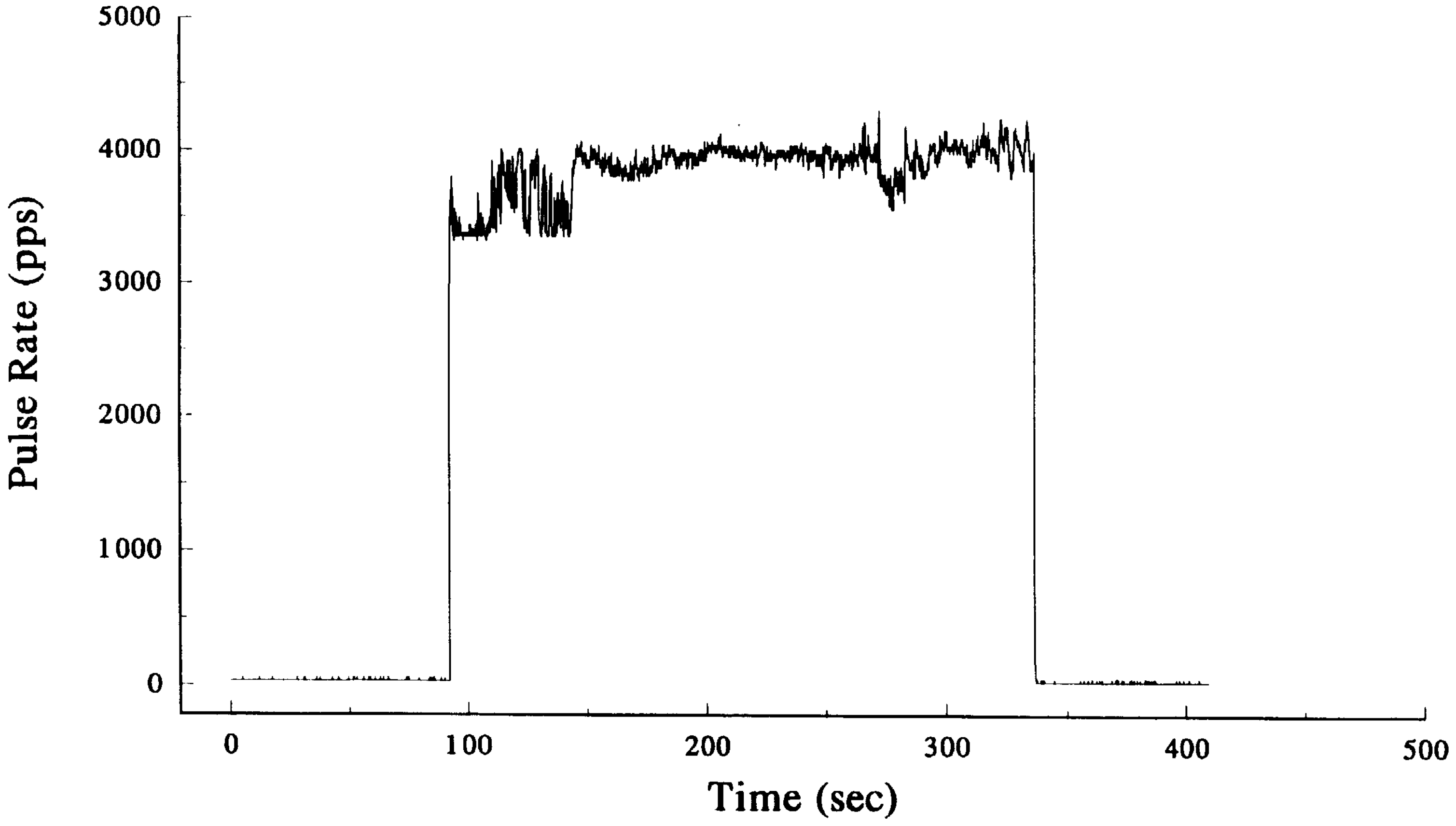

Fig 2. Pulse Rate

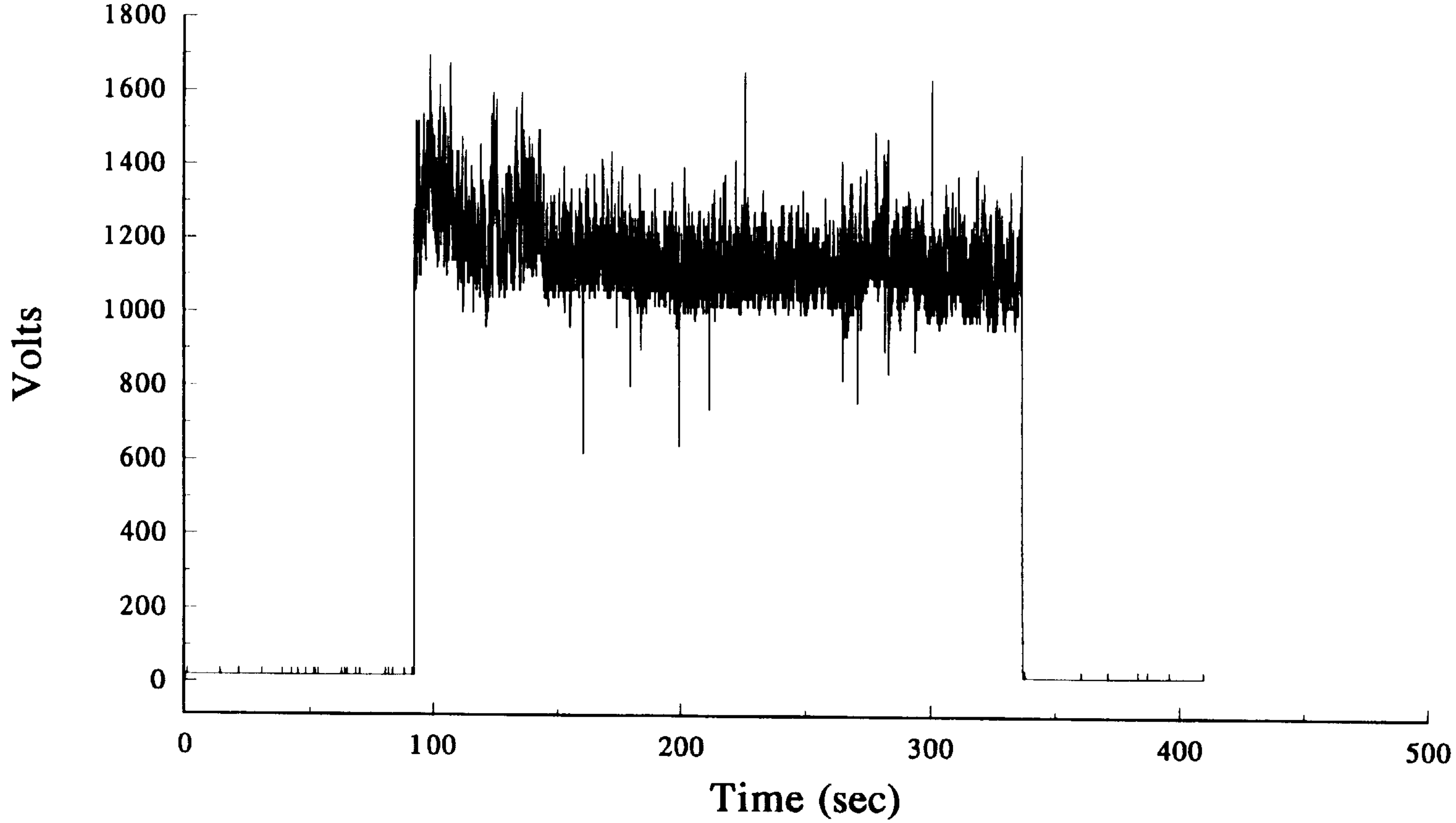

Fig 3. Breakdown Voltage

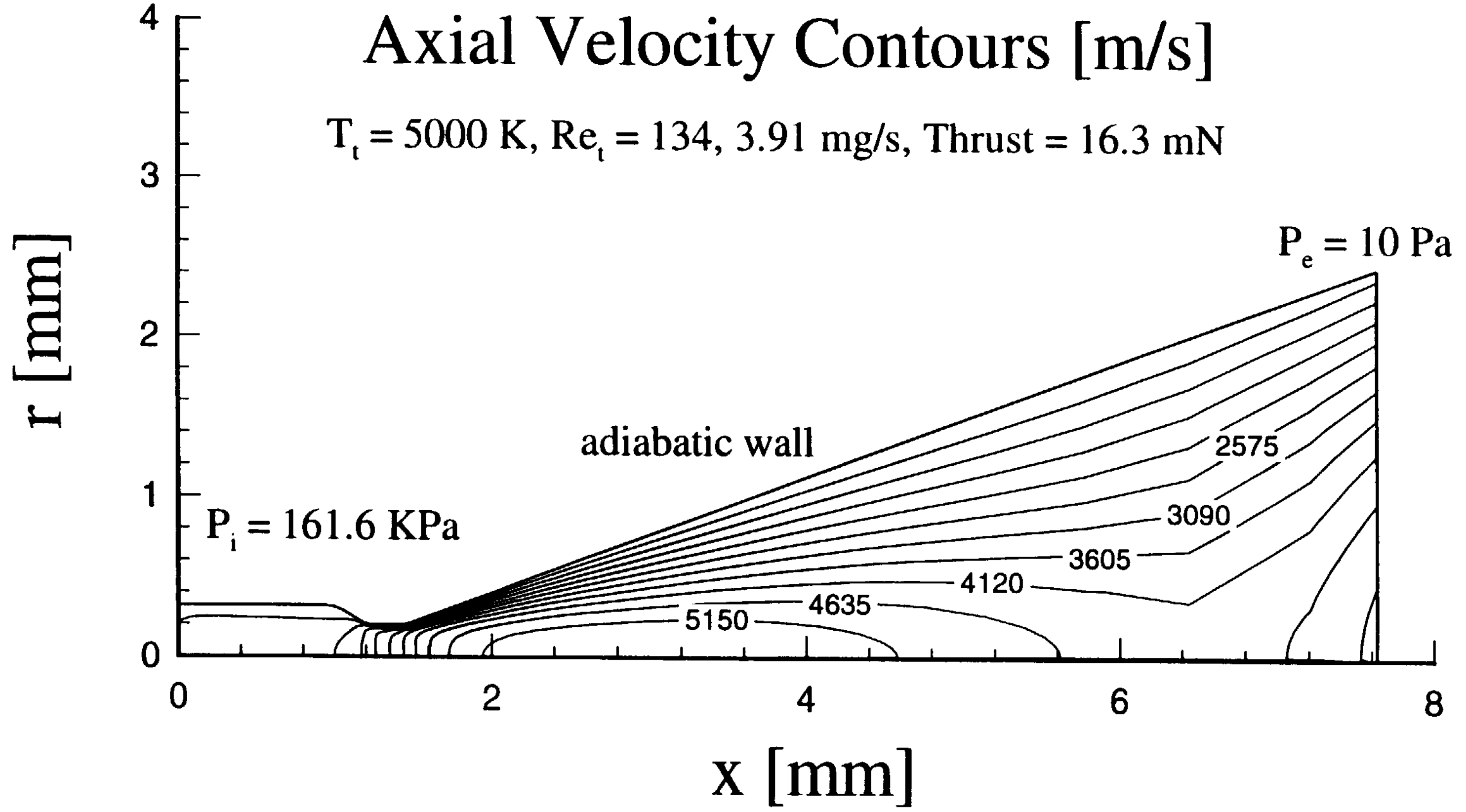
Axial Velocity Contours [m/s]
T_t = 5000 K, Re_t = 134, 3.91 mg/s, Thrust = 16.3 mN
P_e = 10 Pa
adiabatic wall
P_i = 161.6 KPa
2575
3090
3605
4120
4635
5150
r [mm]
x [mm]
0
1
2
3
4
0
2
4
6
8

FIG 4

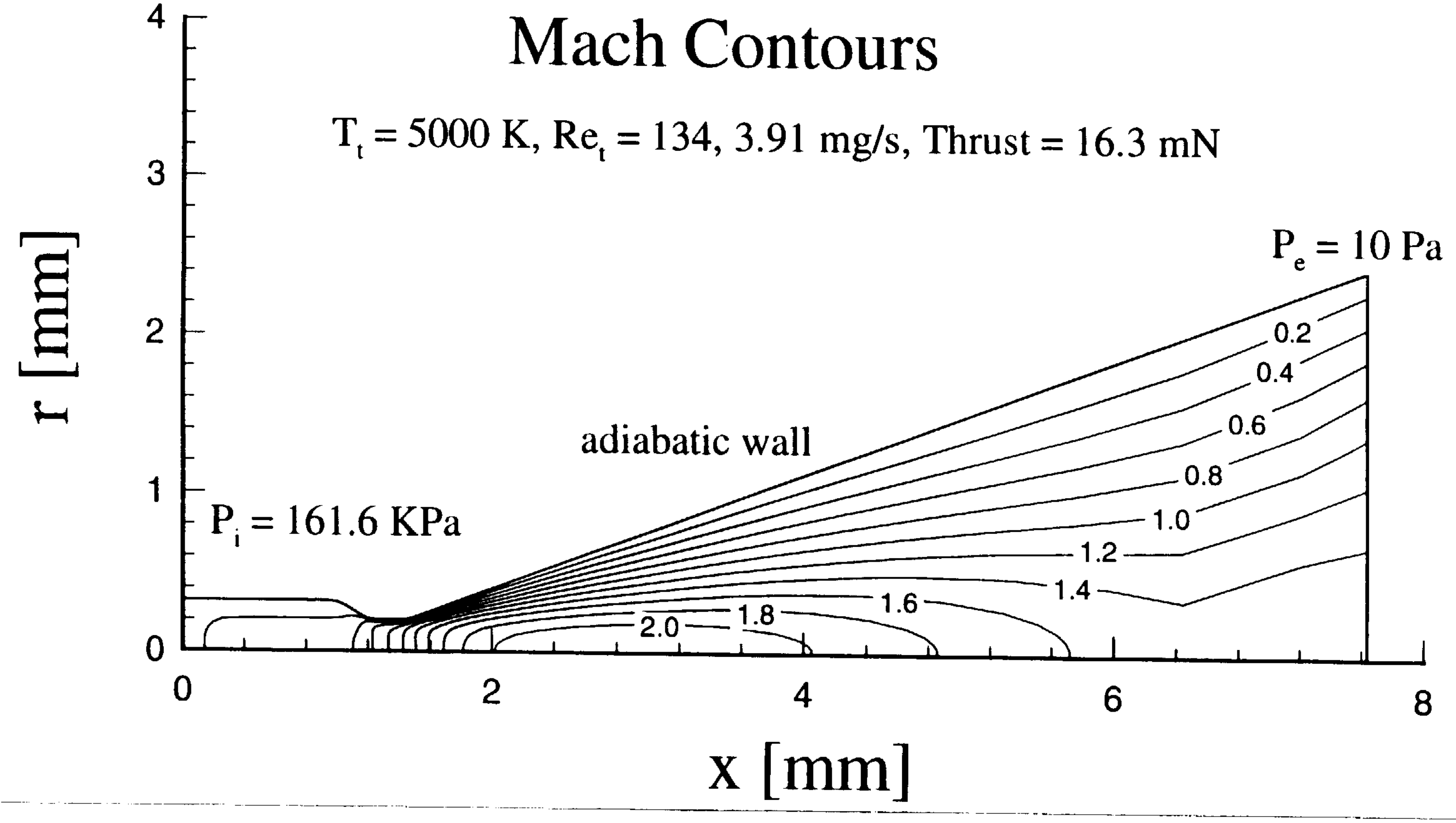
Mach Contours
T_t = 5000 K, Re_t = 134, 3.91 mg/s, Thrust = 16.3 mN
P_e = 10 Pa
P_i = 161.6 KPa
adiabatic wall
0.2
0.4
0.6
0.8
1.0
1.2
1.4
1.6
1.8
2.0
r [mm]
x [mm]
0
1
2
3
4
0
2
4
6
8

FIG 5

AIAA 94-3125

Thrust Performance of a Very Low Power Pulsed Arcjet

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THRUST PERFORMANCE OF A VERY LOW POWER PULSED ARCJET

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Abstract

This paper discusses testing and modeling of a very low power (150 W) pulsed arcjet. Electrical energy is transferred to the propellant from a capacitive energy store, which delivers ~1 μs current pulses at several kHz to the electrodes. All the energy addition to the propellant occurs upstream of the nozzle in a cylindrical 2.5 mm diameter, 5 mm length capillary. The extremely short duration current pulse raises the propellant pressure and temperature in the capillary to approximately 5 atm and 10,000 deg K. The high enthalpy gas is expanded through a tungsten nozzle with a 150:1 area ratio and a 0.4 mm diameter throat. Thrust measurements are obtained for simulated hydrazine ($N_2 + 2\ H_2$) propellant at flow rates from 2.5 to 4.0×10^{-6} kg/sec and 100-200 watts input power. Specific impulses are between 175 and 195 seconds with efficiencies from 2.0 to 4.6 percent. Estimates of thermal efficiency using a 1-D time-dependent heat conduction model indicate that over 80 percent of the energy deposited in the propellant is transferred to the capillary walls by conduction before the propellant passes the nozzle throat. Design changes are proposed that will result in a significant increase in thermal efficiency and thruster performance.

Nomenclature

C	PFN capacitance, F
d^*	nozzle throat diameter, m
D	capillary diameter, m
h	propellant enthalpy, J/kg
I_{peak}	peak discharge current, amps
L	capillary length, m
L_t	inductance, H
m_{cap}	propellant mass in capillary
$\dot{m}$	propellant mass flow rate, kg/s
n	particle density, m^{-3}
pps	pulse rate, s^{-1}
P	power, W
ρ	propellant density, kg/m^3
r	radius, m
R	total circuit resistance, ohms
R_{arc}	arc resistance, ohms
R_{ext}	external circuit resistance, ohms
t_p	arc discharge time, s
V_b	breakdown voltage
V_t	voltage at peak current
κ	thermal conductivity, W/m K
τ	pulse cycle time (= 1/pps), s
T	propellant temperature, K

I. Introduction

Electric propulsion for spacecraft primary propulsion, attitude control and station-keeping has been developed in various forms since the late 1950's. Of the class of electrothermal thrusters, the steady DC arcjet is the most developed technology for low-medium power requirements (several kW), but has the disadvantage of decreasing efficiency, small electrode gap and unstable operational modes at very low power levels (less than 500 W). The present trend to smaller satellites with little power available for propulsion results in a need for thrusters which can operate in the 50-200 watt range. In particular, constellations of small, low-orbit communications satellites have a requirement for efficient propulsion for orbit-raising and drag makeup.

A relatively new candidate for the low power regime is an arcjet which operates in a pulsed mode.[1-6] With this device, a DC power suppy charges a capacitor bank of a pulse forming network (PFN), which is electrically connected to the anode (nozzle) and the cathode. When the breakdown voltage of the propellant is reached, approximately 1000 volts, the capacitor bank discharges, and the energy is transferred by ohmic heating to the propellant in approximately 1 μsec. The resulting high temperature, high pressure plasma (10,000 K, 5 atm) expands through the nozzle while the capacitor bank recharges. This pulsed mode of operation allows much higher peak pressure to

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pulse time to obtain the energy dissipated in the capacitors. In the second method, a thermocouple mounted on a capacitor allowed an estimate of power loss by multiplying the estimated capacitor heat capacity by dT/dt. Both methods showed the power loss in the capacitors to be less than 5 watts.

The DC resistance in the transmission path is negligible; however as the inductance gets extremely small, the rise time of the current pulse becomes extremely short, raising the high frequency resistance of the transmission path, also referred to as a skin depth effect. An estimate of resistance at an inductance of 0.2 μH and current rise times of 200 nsec gives a calculated high frequency resistance of 0.006 ohms, which is negligible compared to the internal capacitor resistance of 0.2 ohms.

An objective of the PFN design is that the pulse energy be transferred in a single pulse without oscillating between the anode and the cathode. That is, the complete RLC circuit including the arc impedance should be near-critically damped. An underdamped circuit with an exponentially decaying sinusoid for the current should be avoided because it can cause accelerated electrode wear by high coulomb transfer for a given discharge energy. If the circuit is critically damped, all the energy will be transferred without current oscillation when $CR^2/4L_t = 1$. To maximize this parameter, the current path between the capacitors and the capillary is made coaxial to minimize the inductance. In the real thruster, the arc resistance is not constant, but estimating the arc resistance as V_t/I_{peak} gives R_{arc} approximately 5.1 to 6.4 ohms and the transfer efficiency $1-R_{ext}/R > 95$ percent. Approximately 70% of the energy delivered to the arc is transferred in the first half cycle.

Thermal Characteristics

With the thruster running, the tungsten nozzle and the front 1 cm of the thruster body glow orange. A type K thermocouple was mounted on the nozzle, giving the initial temperature ramp rate, heat-up time, and steady state temperature (Fig. 6). The nozzle achieves 95 percent of its 1123 K steady state temperature after 4 minutes. A rough estimate of the anode heat loss can be calculated from the slope of the temperature curve, dT/dt at t=0 and the known nozzle heat capacity. This calculation shows that approximately 40 watts goes into anode heating at 135 watts input power, or 30 percent of the total. Heat conduction through the cathode and the front insulator can be estimated as 30 watts or possibly higher at steady state. These results show that the thermal efficiency is extremely low.

Thruster Performance

Performance testing was conducted at NASA Lewis Research Center on the pulsed arcjet at power levels between 100 and 200 watts. Mass flow rates tested were from 2.5 to 4.0 x 10^{-6} kg/sec of simulated hydrazine. This range of flow rates was selected to keep the breakdown voltage and the energy per pulse at reasonable levels, giving specific powers between 38 and 60 MJ/kg. Two PFN configurations were used with $C = 0.05$ μf and $C = 0.10$ μf. Cold flow I_{sp}'s were 105 seconds and nozzle efficiencies were 83 percent. The steady cold flow results compare well with the experimental data of Whalen[8] for the same nozzle area ratio, cone half angle, and Reynolds number. Measured hot flow I_{sp}'s are shown in Figure 7 and are between 175 and 195 seconds for all tests with total efficiencies of 2.0 to 4.6 percent. The thrust response after power is turned on is shown in Figure 8.

Electrode/Capillary Wear

Total operating time on the thruster was 63 minutes at an average input power of 150 watts for a total energy input of 5.7 x 10^5 joules and 10^7 pulses. There was no appreciable change in the capillary diameter due to erosion, although a black coating formed on the inner surface. Very little of this black discoloration appears at the propellant feed hole inlet, indicating that the hot gas in the capillary does not reverse flow into the thruster after each pulse discharge. The anode shows uniform material removal/erosion where the arc attaches. There is no indication of arc attachment inside the convergent section, and the nozzle throat is in the as-machined condition. The material loss of the cathode was not measured but shows signs of smoothening of rough edges. Material loss appears to be minimal.

IV. Heat Transfer Model

A numerical model has been developed to describe the time-dependent energy transfer mechanisms in the pulsed arcjet. Because the arc discharge time is much shorter than the time between pulses, $t_p << \tau$, the arc discharge can be approximated as a discrete event in which the energy stored on the capacitor bank is transferred temperature and thus also decreases the rate at which propellant is ejected from the capillary.

Further insight into the pulsed arcjet performance can be obtained by examining the impulse achieved during the first 50 μsec "pulse" compared with the impulse obtained during the 200 μsec "tail". The total impulse obtained in a single pulse is 2.2×10^{-6} N-s with 55 percent in the pulse and the remaining 45 percent in the tail. We define a propellant efficiency as the ratio of the propellant mass ejected during the pulse to the total mass passing through the capillary in one complete cycle. Since the total mass that must go through the capillary in a single cycle is mass flow rate/pulse rate, or 5.0×10^{-10} kg, then from Figure 12, 32 percent of the propellant is ejected in the "pulse" and the propellant efficiency is 32 percent. Roughly speaking, the mass ejected during the pulse is at a temperature characteristic of the arc while the mass ejected in the tail is at a temperature characteristic of the nozzle. The average performance of the pulsed arcjet is low because a large fraction the propellant mass in the tail exits the nozzle at a relatively low velocity, even though the I_{sp} for the mass ejected in the pulse is high.

The obvious question that arises is whether the performance of the low power pulsed arcjet can be increased significantly. To obtain a performance increase, the thermal efficiency must be raised, since conduction heat loss appears to be by far the highest loss mechanism. We make a simple scaling argument to indicate a direction for design improvement. We wish to increase the heat loss time, which scales as $m_{cap} h_0 / (\kappa \, dT/dr \, \pi \, D \, L)$. Assuming that $h_0 / (\kappa \, dT/dr \, D)$ is invariant, then the heat loss time scales as m_{cap}/L or as $n\pi D^2/4$. Since the Paschen breakdown voltage, V_b, is a function of nL, it may be possible to increase n and decrease L, raising thermal efficiency without changing the breakdown voltage. This suggests that the pulsed arcjet performance can be improved with shorter, higher pressure capillaries.

VI. Acknowledgments

This work was supported by Grant NAG 3-1360, awarded by the NASA Lewis Research Center. J. M. Sankovic is the grant monitor. We especially wish to thank R. M. Myers for many stimulating discussions, T. W. Haag for technical support in obtaining the performance data, and K. Elam for fabricating the thruster components. We also gratefully acknowledge the Aeronautical and Astronautical Engineering Department of the University of Illinois for additional funding support.

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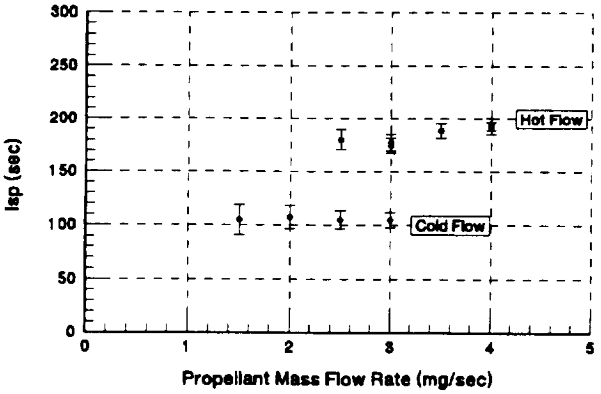

Fig 7. Specific impulse from experiment. Input powers between 125 and 190 watts. Efficiencies from 2.0 to 4.6 percent.

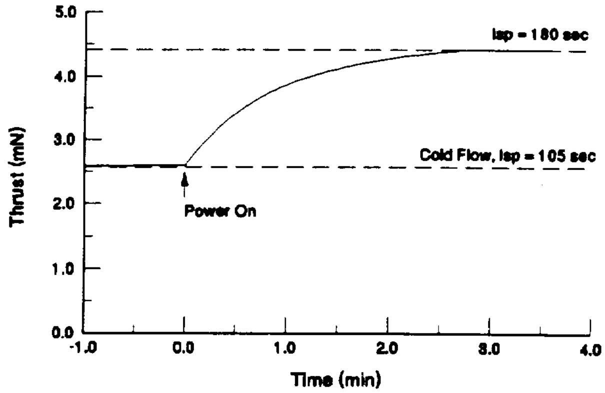

Fig 8. Initial thrust increase after power application. 155 watts input power, 2.5 mg/sec.

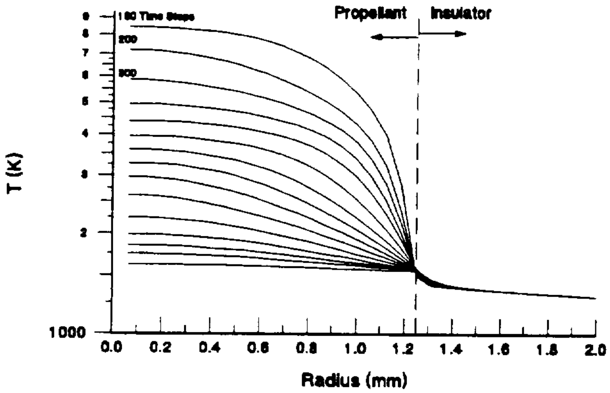

Fig 9. Calculated temperature profiles in propellant and insulator following a pulse at 100 time step intervals.

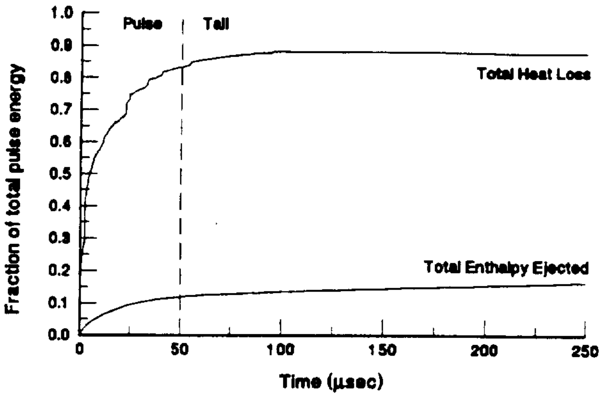

Fig 10. Comparison of conduction heat loss and total enthalpy ejected from the nozzle in a single pulse.

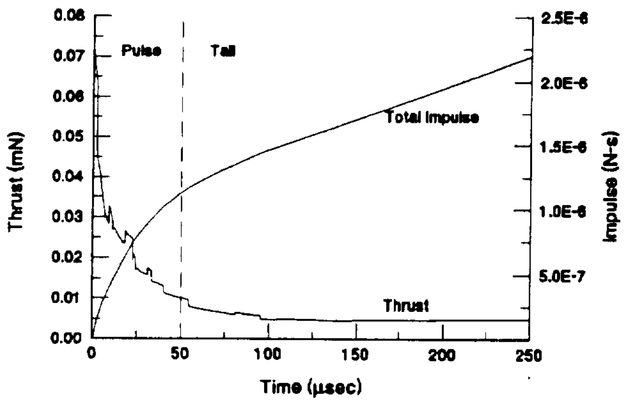

Fig 11. Total impulse and instantaneous thrust during one cycle. Discontinuities are from fluid properties interpolation scheme.

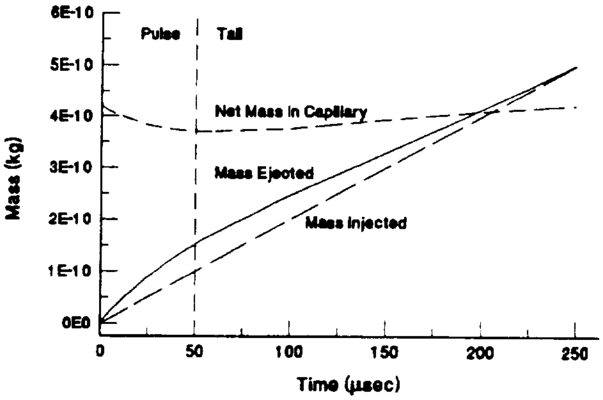

Fig 12. Propellant mass ejected from capillary in a single cycle. Mass is injected at a constant rate.

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Pulsed Arcjet Performance Measurements

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An electrothermal thruster which operates in a pulsed mode is being investigated at the University of Illinois. Electrothermal thrusters operating at high specific impulse with low power requirements offer significant benefits for the potentially large numbers of low orbiting "microsats" and for constellations of communications satellites. A low orbit microsat with body-mounted solar panels will have limited power available for electric propulsion. Consequently, pulsed arcjet propulsion systems for these satellites must be able to operate stably and efficiently at very low power levels.

In a pulsed electrothermal thruster, a DC power supply charges a capacitor bank which is electrically connected to the anode (nozzle) and the cathode. When the breakdown voltage of the propellant between the electrodes is reached, approximately 1000 volts, the capacitor bank discharges, and the energy on the PFN capacitors is transferred to the propellant by ohmic heating in an arc discharge. The heat addition occurs in the subsonic region upstream of the nozzle throat in approximately 1 µsec, and the high temperature plasma then exhausts through a nozzle on a comparatively longer time scale, approximately 1 msec. Because the arc discharge is essentially a discrete time event, stability of the arc is not a problem, and the pulsed arcjet can operate stably over a wide range of power levels and mass flowrates. A pulsed mode is expected to have lower frozen flow losses than a DC arcjet because the energy addition occurs in a relatively high pressure capillary where dissociation and ionization are reduced. Further, the pulsed arcjet is not as sensitive to cathode-to-anode gap length, which is much more critical for a DC arcjet at very low power.

The thruster assembly consists of the arcjet and capacitor energy store integrated within a single housing. The arcjet design is based on a typical NASA 1-kw laboratory arcjet with modifications for pulsed operation. The primary physical difference between the pulsed arcjet and the constricted DC arcjet is the large gap of several mm between the cathode and the anode. The arc discharge occurs entirely within a cylindrical cavity drilled in a boron nitride insulator. This cylindrical region is closed at one end with a 4.76 mm diameter, 2% thoriated tungsten cathode. At the other end is a conical tungsten nozzle with a 0.4 mm diameter throat, 150:1 area ratio and a 20 degree half angle. The thruster design allows capillaries of different lengths and diameters to be interchanged. Propellant is injected radially into the capillary through a 0.33 mm diameter feed hole in the boron nitride insulator. The front thruster body has a 0.80 mm wall thickness, which decreases heat conduction to the rear of the thruster and allowing the nozzle temperature to reach 1200 K or higher. Testing is performed at the University of Illinois in a 1.5 m^3 vacuum tank with a 50 mTorr background pressure when the thruster is running. The assembly is mounted as a single unit on a flexure-type thrust stand with a PID controller. The PID controller uses a counterforce damper coil to eliminate low frequency oscillations and to maintain a constant thruster position, allowing measurements at extremely low (0.1 mN), thrust levels and reducing hysterisis effects.

In a previous paper,[1] the results of pulsed arcjet testing at the University of Illinois were reported in which a pulsed electrothermal thruster using simulated hydrazine propellant was operated at 125 to 200 watts. Isp's measured were from 175 to 195 sec. The paper also presented a one-dimensional time-dependent numerical model which calculates propellant

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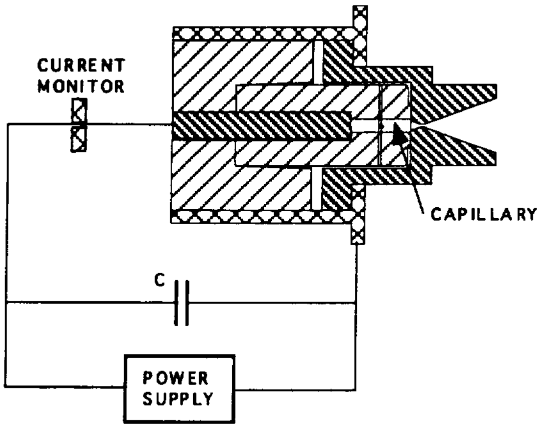
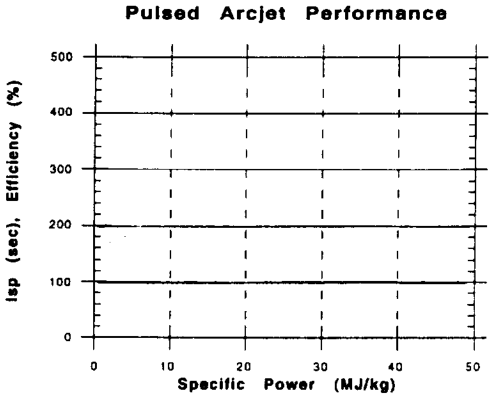
temperature profiles in the capillary versus time and provides an estimate of thruster thermal efficiency. This paper extends the previous research with experimental results for thrust, efficiency, and I_{sp} over a wide range of operating conditions, including different propellant flowrates, pulse rate, input power, input current, pulse energy, and capillary dimensions. In particular, the paper discusses whether a pulsed electrothermal device is practical at low power and the extent to which scaling effects degrade thruster performance. At very low power levels, and hence low flowrates and Reynolds numbers, viscous losses in the nozzle increase significantly and total efficiency and overall performance can decrease dramatically. An understanding of the energy loss mechanisms, including thermal efficiency and nozzle efficiency, are critical to the understanding of arcjet design criteria in this low power regime.



Schematic of electrodes, capillary and pulse forming network



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[1] Willmes, G.F., and Burton, R.L., "Thrust Performance of a Very Low Power Pulsed Arcjet," AIAA Paper 94-3125, June 1994.

[2] Willmes, G.F., and Burton, R.L., "Investigation of a Very Low Power Pulsed Arcjet," IEPC Paper 93-136, September,1993.

ACKNOWLEDGMENTS

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