HALL THRUSTER DIRECT DRIVE DEMONSTRATION

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Abstract

The use of electric propulsion systems to improve spacecraft mission performance is becoming commonplace. Conventional electric propulsion systems require power processing units to convert electrical energy from the spacecraft power source to the levels required for thruster operation. One possible option to simplify electric propulsion system is the elimination of a portion or all of the power processing unit with a direct connection of the thruster to a solar array. A joint NASA/NRL effort has been initiated to experimentally determine the efficacy of this approach. To facilitate this direct drive experimentation, a concentrator solar array testbed was installed at the NASA Lewis Research Center. In the initial experiments discussed in this paper, a Stationary Plasma Thruster (SPT), was operated directly from the solar array with a peak power of 1 kW at approximately 250 VDC output. Auxiliary power to energize the thruster magnets and ignite the thruster discharge was provided by laboratory-type power supplies. Data were taken with both laboratory power supply and solar array drive to compare electrical characteristics. The thruster was observed to operate with reduced current oscillations with solar array drive when compared to data taken with laboratory power supplies. Some differences in system volt-ampere characteristics were also noted. While operating on the solar array, the thruster discharge current was increased to the point of solar array voltage collapse without extinction of the discharge. Recovery from this condition was possible by the reduction of anode flow rate. In general the thruster operated nominally in the direct drive configuration which indicated that this is a viable option for thruster operation if mission and cost analyses indicate an overall benefit from this configuration.

Introduction

Electric propulsion (EP) systems have entered the age of application. Several commercial geosynchronous communications spacecraft are now equipped with arcjet thrusters for the north-south stationkeeping (NSSK) maneuvers[1] and there are plans for the installation of small gridded, ion thrusters for this function.[2] Hall thrusters, widely applied in Russia for the NSSK role,[3] are now being considered as a viable candidate on commercial communications spacecraft. In addition to these auxiliary propulsion roles, a 30 cm ion thruster is baselined as the primary propulsion system for the first flight of the New Millennium planetary spacecraft.[4]

Whichever thruster is selected, a typical EP system can be generically illustrated as in Figure 1. There are four major components, the electrical power source, the thruster, propellant feed system, and the power processor.

In conventional electric propulsion systems, a power processing unit (PPU) is required to condition the electrical power from the spacecraft to meet the requirements for thruster operation. The mass of the PPU and the required thermal management must be charged to the EP system. Thus a portion of the propellant mass saved by the application of EP is taken up by the PPU. In the state-of-the-art NSTAR system, for example, the PPU specific mass is 5.2 kg/kW.[5]

Elimination of some or all of the PPU power supplies can result in a significant cost and mass savings to the EP system. In a direct drive system, for example, one or more of the PPU power supplies can be eliminated by connecting the electric thruster directly to the electric power source, typically a solar array. A simplified block diagram of this type of system is shown in Figure 2.

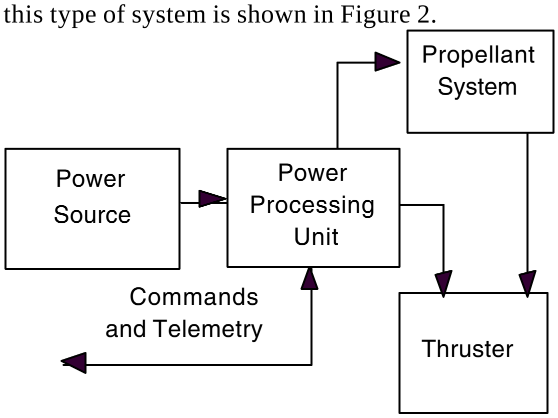

Figure 1. Generic Electric Propulsion System

Direct drive has been successfully demonstrated in the past by Gooder with an ion thruster.[6] In that work, the beam power supply was replaced with a high voltage (> 1kV) solar array.

Recent flight data indicate that concentrator type solar cells can be successfully operated with biases as high as 500 V in low Earth orbit (LEO).[7] Stationary Plasma Thruster (SPTs) and Thrusters with Anode Layer (TALs) operate at or below this voltage and are currently of significant interest to the EP community. For the demonstration described in this paper, an SPT was selected for this initial direct drive demonstration.

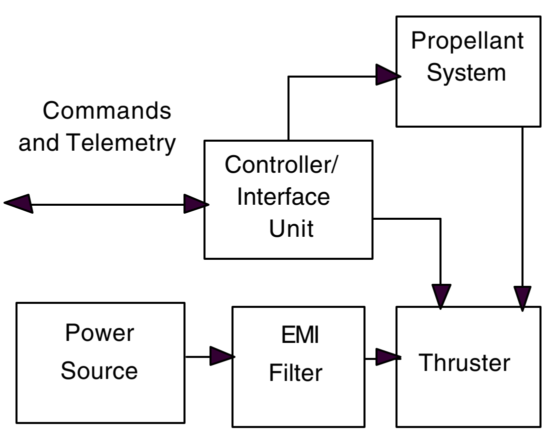

Figure 2. Direct Drive Electric Propulsion System

Conventional SPT PPUs developed in the US have been built with low impedance, or constant voltage outputs.[8,9] A thruster with anode layer (TAL) has also been operated on a large capacitor, with a very low output impedance, in long pulse mode.[10] Each system has demonstrated successful thruster operation and development efforts have documented thruster characteristics such as high inrush currents at startup. In addition, some empirical studies have determined that simpler PPU output filter topologies than those originally thought necessary can be successfully applied to the thruster discharge power supply.[11]

This paper describes a joint NASA Lewis/Naval Research Laboratory effort to demonstrate the feasibility of eliminating the discharge power supply from a PPU by using a direct connection to a solar array. Thruster interface considerations and overall spacecraft power management and distribution (PMAD) are also discussed. Preliminary thruster integration data are presented with a comparison of some thruster interface characteristics with both conventional and direct drive configurations.

Background

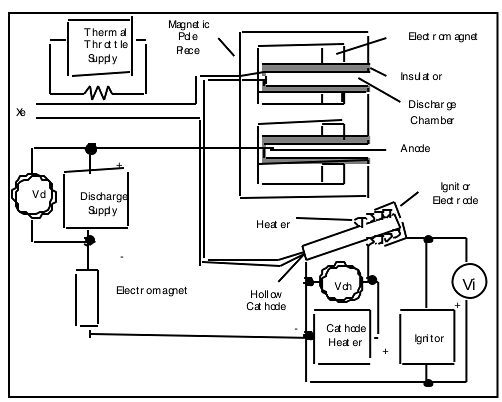

Figure 3. Hall Thruster PPU Architecture

A typical Hall thruster PPU architecture is shown in Figure 3. In steady state operation, only the discharge and thermal throttle power supplies are required. The thermal throttle is a flow control device which uses a heating element in conjunction with a capillary tube to control the flow of xenon into the thruster.[3] In the absence of a thermal throttle, an alternate flow system with its own power supply is required to regulate thruster power. The other power supplies for the cathode heater and ignitor electrode are required only for thruster ignition. Finally, in some applications it may be advantageous to provide a separate power supply for the electromagnet. In Figure 3, the electromagnet is connected in series with the thruster cathode, to permit magnet excitation without an external power supply.

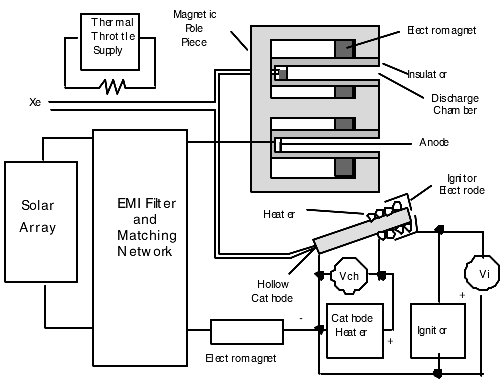

Figure 4. Direct Drive Configuration

In the direct drive demonstration described herein, the discharge power supply, which processes over 95% of the thruster power, was replaced with a direct connection to a solar array of 250 V output with an EMI filter, in an effort to simplify the electric propulsion system. (See Figure 4.)

PMAD Considerations

From an architectural standpoint, the power management and distribution (PMAD) configuration must be carefully considered when designing for the non-thrusting periods of the mission. Hall thrusters typically require a minimum of 200 VDC for proper operation, which is well in excess of the common 28 - 120 VDC range bus voltages which encompass most state-of-the-art spacecraft.

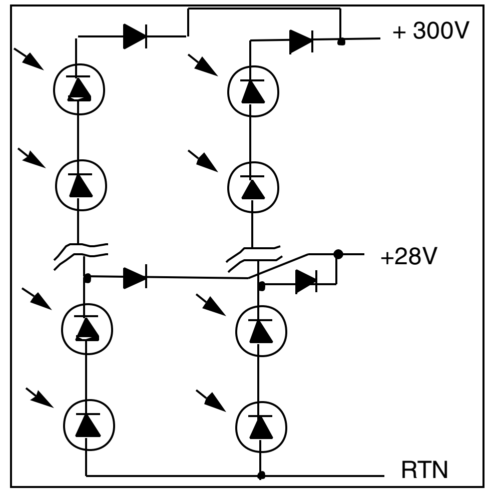

Figure 5. Tapped solar array system

Several power management and distribution (PMAD) options are available to maintain a more conventional spacecraft bus voltage. For example, a switching network to configure the array outputs in a series/parallel configuration can be employed. A system of this type, however, also adds weight and complexity and so mitigates somewhat the simplification provided by the direct drive option.

There are other possibilities for PMAD that do not involve array switching. If strings of series connected solar cells connected in parallel are used to produce a nominal 300 V for propulsion, these strings could be tapped at the 28 or 100 V levels for payload power, as shown in Figure 5. This would result in 9 - 33 % of the EOL power available for the spacecraft payload. Another option is the use of standard, military qualified 270 VDC input power converters in the individual avionics boxes, or as a central bus down converter to produce a nominal 28-100 V bus as shown in Figure 6. These converters are available in power densities of 120 Watts per cubic inch with efficiencies of 0.9. Using converters of this type, 90 % of the EOL power of the array would be available for the spacecraft payload in non-thrusting periods.

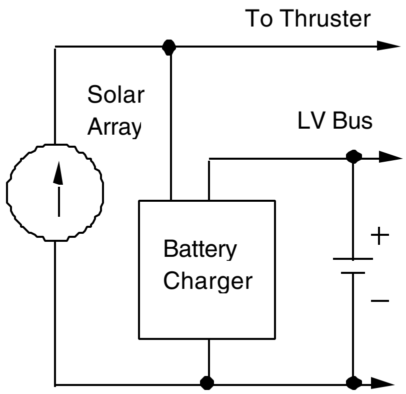

Figure 6. PMAD with low voltage battery bus

Apparatus

Advanced Concentrator Array

The concentrator testbed uses three photovoltaic concentrator arrays to provide the electrical power from available sunlight. These three arrays are special versions of a terrestrial concentrator array employing linear fresnel lenses similar to the type used with the recently space-qualified linear concentrator technology.[12]

Each array incorporates two concentrator modules with 6 square meters of combined sun-capturing lens aperture area. Full two-axis sun tracking is provided by a microprocessor-based, open-loop (no sun sensor), solar-powered, sun-pointing system. Each concentrator module uses an acrylic plastic linear Fresnel lens to focus sunlight into a bright line on a photovoltaic receiver as shown in Figure 7. The geometric concentration ratio of the module, corresponding to the lens aperture width divided by the cell active width, is 21X. The volume between the lens and receiver is enclosed with an aluminum housing structure which supports the lens and receiver while also minimizing dirt and moisture infiltration into the module. The modules are also equipped with a wind sensor which automatically stows the arrays in a safe position in the event of sustained high winds.

The photovoltaic receiver is comprised of series connected, single crystal silicon solar cells mounted on an extruded aluminum heat sink as shown in Figure 7. The heat sink provides natural convection air cooling of the cells, maintaining the cell temperature at about 30º C above the ambient air temperature in peak sunlight conditions.

The three arrays were specially designed and constructed to provide very high voltage levels. With all six receivers wired in series, voltages as high as 300 V can be obtained. This was accomplished by dicing 10 cm long cells, which are normally used in the terrestrial application of this array type, into three 3 cm long cells. 115 of these shorter cells were then wired into a series circuit on each receiver.

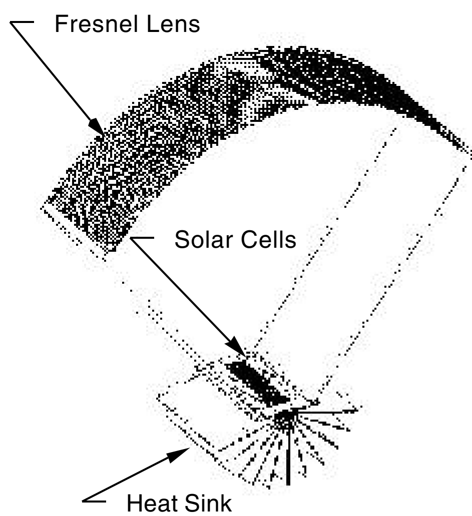

Figure 7. Concentrator Array Cross Section

Each receiver of the individual arrays is wired in series, configuring each array to a 100 V nominal output at the peak power point of approximately 350 W. Power is brought to a central switching panel via high flexibility welding cable which is tightly twisted to minimize both the DC resistance and inductance of the distribution system. The switching panel allows the remote switching of the array outputs into either series or parallel configurations and allows the user to direct array power to one of three high vacuum facilities in the building. The switching panel also contains safety interlocks to short the array outputs in the event of a breach of the closed conduit system or the activation of an emergency stop sequence at the remote power output sites. The normal off-line configuration of the arrays also requires a shorting of the outputs for safety. Typical maximum array output power is on the order of 1 kW.

Thruster

An SPT thruster, specifically a T-160 model obtained from the Keldish Research Center (KeRC) in Russia, was installed in a large space simulation testbed and used for this demonstration[13]. The T-160 is a Stationary Plasma Thruster (SPT) with a discharge chamber diameter of 160 mm and a nominal operating power of 4.5 kW.[14] The cathodes on this particular thruster were of the cold start variety, which means that a cathode heater was not present, and a power supply to drive it was not required. Xenon flow was provided to the thruster via commercial mass flow controllers.

Conventional Power System

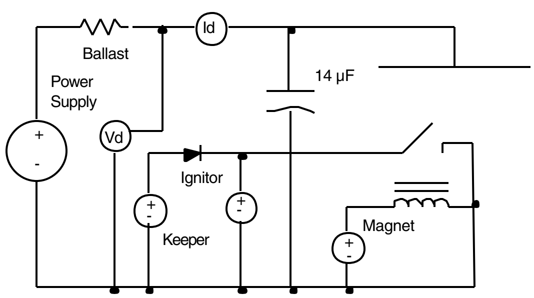

Figure 8. Conventional Drive Experiment Configuration

Electrical power was provided to the thruster via an assembly of laboratory power supplies to operate the thruster cathode and electromagnet. The conventional drive experimental configuration was as shown in Figure 8. The thruster was equipped with cold start cathode which negated the need for a cathode heater power supply. The EMI filter consisted of a single 14 µF capacitor connected across the anode and cathode of the thruster in close proximity to the thruster in the vacuum facility.Discharge power was supplied by a standard laboratory power supply with a series ballast resistor as shown. The nominal value used was 5?.

Concentrator Array Based Power System

The electrical configuration shown in Figure 9 was used when powering the thruster from the solar array. The laboratory power supply and ballast resistor were replaced with the solar array output. The thruster cathode and electromagnet were powered as in the conventional power system

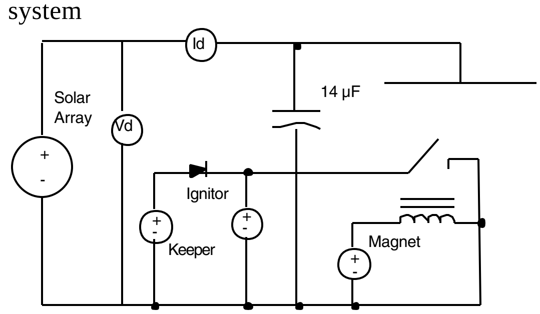

Figure 9. Direct Drive Experimental Setup

.Electrical Measurements

In all experiments, the electrical measurements were taken at the locations shown in Figures 8 and 9, For the conventional power system, the discharge voltage, Vd, was measured from the thruster cathode to the output side of the ballast resistor. The discharge or anode current was measured, as shown, on the anode line to the thruster. For the direct drive configuration, discharge voltage was measured at the output of the solar array at the vacuum facility feed through. Discharge or anode current was measured as in the conventional system. For the conventional The capacitor current was not measured for this experiment due to the location of the capacitor at the thruster in the vacuum facility.

Experimental Procedure

For comparison, the thruster was first operated on laboratory power supplies and then switched over to operation on the solar array. The parameters of interest for these tests were the discharge voltage and current characteristics during startup and steady state operation at various power levels up to the peak power point of the solar array. The thruster operating condition would also be adjusted so that the thruster demanded more power than the solar array could deliver to investigate the operating characteristics in this regime.

Thruster Ignition

Thruster ignition was accomplished by first establishing an anode and cathode flow rate that corresponded to normal operating flow with an anode current of approximately 4 ADC. The ignitor power supply was then energized to 600 V with a current limit set to 0.5 ADC to initiate a local discharge between the cathode and the ignitor electrode. When the ignitor current was stable at 0.5 A, the keeper power supply was energized with a current setpoint of 4.0 ADC. Keeper current was maintained throughout thruster operations to ensure that the minimum cathode emission current was greater than 4.0 ADC. This was required to maintain the proper cathode temperature for emission since the T-160 was operated at only 1 kW or less during the direct drive portion of the experiment. At the throttled back condition, anode current alone may have been insufficient to maintain stable operation. Nominal T-160 power is 4.5 kW, (15 A @ 300 VDC). After the ignitor discharge was established, the thruster magnet power supply was energized and the magnet current was set to 6.5 ADC. Following the application of magnet current, the discharge power source was turned on to complete the ignition sequence. An oscilloscope was used to capture the discharge ignition transient. The thruster was first operated on laboratory power supplies to validate the system setup and determine if the operating point of 1 kW was stable. Following the first thruster ignition, the thruster was allowed to operate at 1 kW for 30 min to drive off any absorbed water vapor.

Direct Drive Operation

Following the 30 minute system checkout test, the laboratory discharge power supply was shut down and disconnected, and the solar array power source connected to the thruster. The thruster was re-ignited and startup data was taken at the 1 kW operating point. The thruster power was then throttled by varying the mass flow rate to the thruster and oscillographs of steady state anode voltage and current were taken at all throttled points. The thruster was throttled down to 2.8 A of anode current in the array constant voltage regime, and throttled up until the array collapsed into the constant current region of operation. The thruster power was then reduced to document the recovery of the system to nominal operation.

Conventional Power System Operation

The above procedure was repeated with laboratory power supplies for comparison of the volt-ampere characteristics of the thruster. Steady state oscillographs of the anode voltage and current were not taken with the thruster operating on laboratory power supplies at all throttled points. No attempts were made to acquire thruster performance data during these tests.

Results and Discussion

Ignition of the thruster on laboratory power supplies was straightforward and similar to ignition characteristics previously recorded.[8,9,11] Oscilloscope traces of discharge voltage and current during a typical start were as shown in Figure 10. The ramp up of the discharge voltage was due to the characteristics of the laboratory power supply. Steady-state operation at

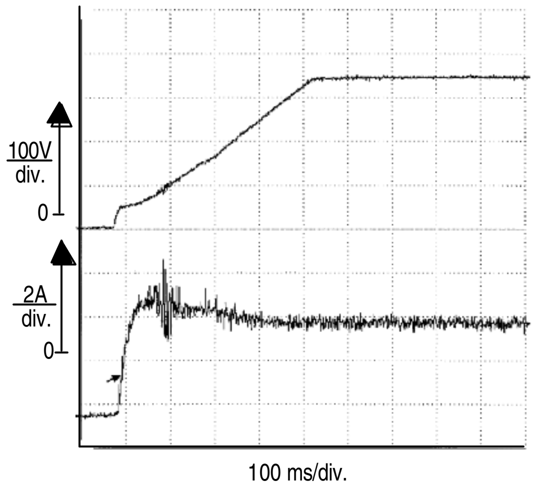

Figure 10. Thruster start on laboratory power supply

approximately 1 kW was as shown in Figure 11. Current oscillations of 1-2 Ap-p were noted.

The thruster was throttled over a discharge current range of 3 to 6 A to determine the stability of the thruster at these power levels. The thruster was stable and the current oscillations remained constant in amplitude and frequency regardless of the operating power of the thruster on the laboratory power supplies.

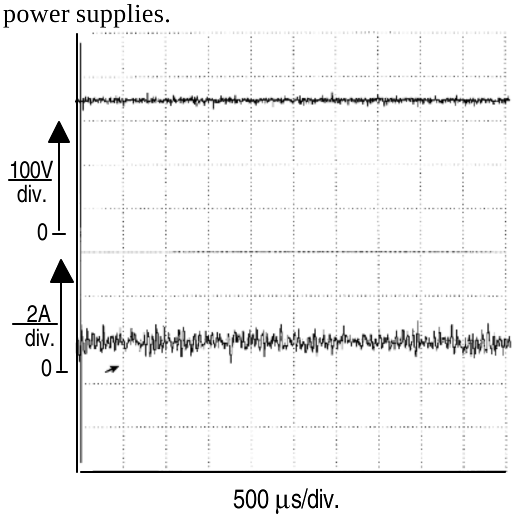

Figure 11. Steady-state operation on conventional power supplies

Solar Array Operation

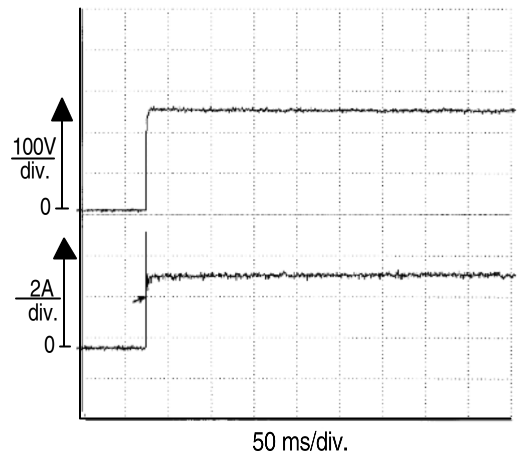

Figure 12. Discharge voltage and current during thruster ignition on solar array

Following the tests with the conventional power system, the T-160 was connected to the solar array. Thruster ignition was immediate and the voltage and current waveforms are shown in Figure 12. From the traces it can be seen that the array voltage does not exhibit any appreciable droop, and the thruster current does overshoot 2 A over the steady state value of 3.8 ADC. This level of overshoot is consistent with the short circuit array current of 6.3 ADC. Note the absence of discharge current oscillations immediately after the start of the thruster. This behavior was typical of thruster operation throughout the test until the operating point approached the peak power point of the solar array.

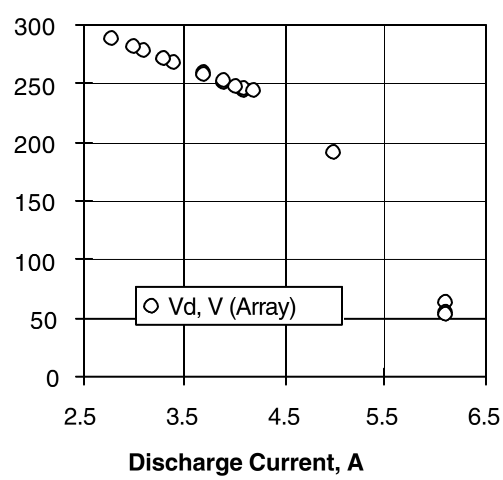

Figure 13. Solar array volt-ampere characteristic with T-160 thruster.

The solar array volt-ampere characteristic obtained while operating the T-160 was as plotted in Figure 13. A typical steady state oscillograph of discharge voltage and current was as shown in Figure 14. Note the absence of any current oscillations for this operating point of 267 V and 3.4 ADC. These data were typical of all operating points on the constant voltage side of the peak power point of the array.

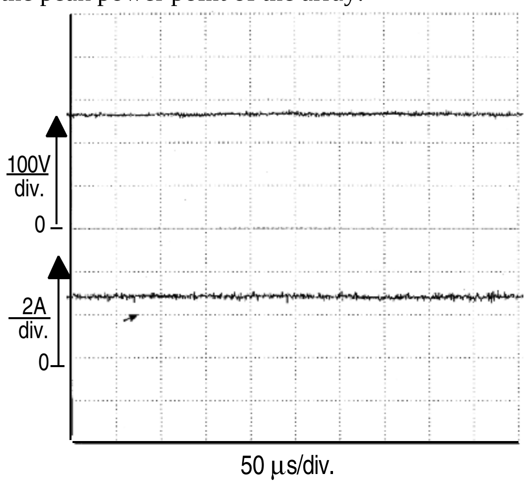

Figure 14. Steady state discharge voltage and current waveform

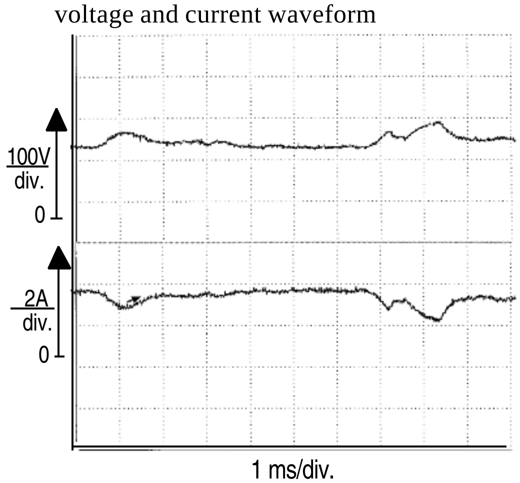

Figure 15. Discharge voltage and current waveforms at onset of oscillatory behavior near array peak power point

As thruster power was increased (by increasing the mass flow rate) to the point that the peak power operating point of the array was approached and surpassed, the discharge voltage and current were as shown in Figure 15. The thruster operated nominally, however an occasional excursion or perturbations in thruster voltage and current were noted as the operating point began to move on the array volt-ampere characteristic. These excursions, as shown in Figure 15, were characterized by a decrease in discharge current, which caused an increase in discharge voltage. The increase in discharge voltage was expected due to the volt-ampere characteristic of the system (Figure 13). The cause of this oscillatory behavior was not clear and warrants further exploration. It was surmised that these oscillations were due to the increase in array impedance near the peak power point. As mass flow rate was increased further, the oscillatory behavior increased, as shown in Figure 16.

At still higher mass flow rates, the solar array collapsed as expected into a constant current mode. The current in this case was 6.1 ADC with a discharge voltage of 54 V. Figure 17 shows the discharge voltage and current while the thruster operates in this regime. In this mode, the thruster plume appeared dimmer on visual inspection, but operation appeared otherwise normal. The thruster returned to a nominal operating point as the mass flow rate was reduced.

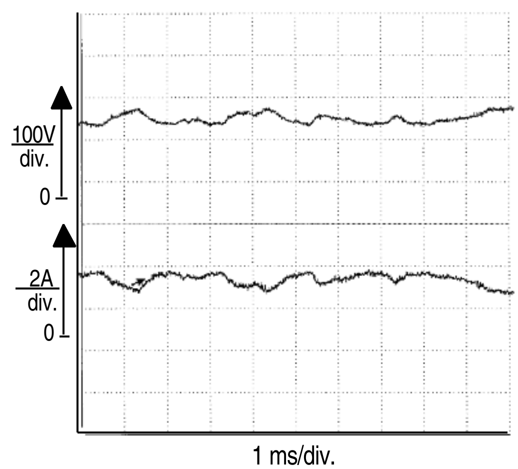

Figure 16. Discharge voltage and current oscillations at increased flow rate prior to array collapse

The SPT nominally operates in constant current mode for discharge voltages in excess of 100 V at a fixed mass flow rate.[15] Further, the discharge current is reported to be proportional to anode mass flow rate. These characteristics were not observed for all operating conditions tested. It must be noted here that the T-160 has a nominal operating power of 4.5 kW. During these experiments, the maximum power reached was only 1.8 kW on the conventional power system and 1 kW on the solar array. To determine if this was a real characteristic of solar array operation, or an artifact of operating the T-160 at a highly throttled condition, data from conventional and solar array drive were compared. The results of these measurement are presented in figure 18.

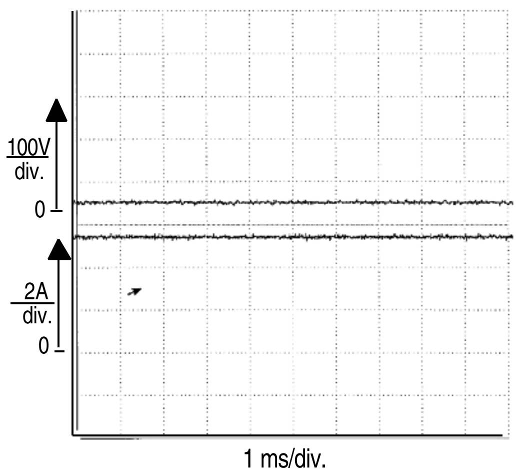

Figure 17. Discharge voltage and current waveforms while operating on collapsed array.

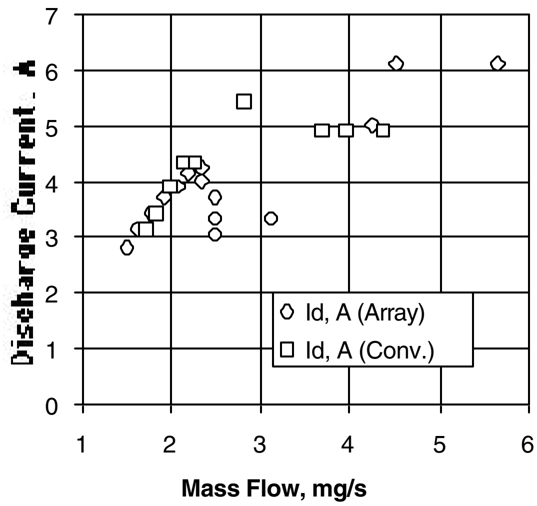

Figure 18. Anode (Id) current vs. Flow rate for solar array and conventional power supply drive

The array data show a clear drop in anode current as flow was increased above 2.3 mg/s. The conventional power supply data continue to increase, but also drops above 3.5 mg/s. The data begin to realign at flow rates in excess of 4 mg/s. These data may be the result of operating the T-160 off of its normal operating power of 4.5 kW. It is also hypothesized here that the discharge current vs. mass flow rate data while operating with solar array drive are the result of the combination of the thruster and solar array volt-ampere characteristics. Unfortunately, a variable electronic load of sufficient power was not available to sweep an array volt-ampere characteristic on resistive load.

Conclusions

Conventional electric propulsion systems utilize PPUs which represent a mass and cost penalty. Operating electric thrusters directly from solar arrays is one potential option for reducing system cost and complexity. This requires, however, the integration of high voltage solar arrays to the spacecraft. This represents a significant deviation from the current state of the art. To address this, several PMAD schemes adapting the required high voltage solar arrays to conventional spacecraft avionics were also discussed. These included solar array switching, array tapping, down converters, and application of 270 V input inverters to payloads.

In this paper, an SPT was operated with both convention power supplies and a 300 V solar array, which replaced the discharge power supply used in a PPU. The electrical characteristics of the thruster, specifically the discharge voltage and current waveforms and the discharge current vs. mass flow rate, were documented for both the conventional and solar array drive.

While operating the thruster discharge with solar array power, the cathode was ignited with auxiliary supplies and the array power was applied to the thruster anode and cathode in a stepwise fashion. Thruster ignition was immediate with little current overshoot or other deleterious transients.

Steady state operation below the peak power point of the solar array showed the absence of anode current oscillations observed while operating on laboratory power supplies. Current oscillations were noted as the peak power point was approached and surpassed. The cause of this behavior is unknown at this time, one potential explanation was that these oscillations were due to the proximity of the operating point to the constant current regime and were a transition between constant current and constant voltage operation on the array.

As anode flow rate was increased the array was collapsed into the constant current operating regime, which resulted in a low voltage discharge on the thruster. Of interest was the fact that the thruster did not extinguish during this transition and, in fact, operated normally. The thruster was returned uneventfully to normal operations at greater than 200 V by reducing the anode flow rate.

The oscillations near the peak power point resulted in a deviation in the proportionality of the anode current vs. flow rate that was expected as when the thruster was operated on laboratory power supplies.

In general, the SPT operated nominally from the solar array. Thus the drive scheme is a viable option if the system trade space warrants its application.

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