Pulsed Plasma Thrusters for Orbit Transfer

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Introduction

THE use of electric propulsion for near earth missions has recently received increased attention with the recognition of possible long-duration missions involving payloads that require substantial levels of electrical power. If most of the mass of the power supply (including heat rejection) can be ascribed to the mission payload, then the optimum exhaust velocity can increase substantially above levels accessible with chemical thrusters. From classic mission analyses for electric propulsion,[1] the optimum exhaust speed is given by $u=(k\epsilon\alpha\tau)^{1/2}$ where $k=1$-2 depending on the ratio of u to the necessary velocity increment Δv; ϵ is the efficiency of thrust power from total power, α is the specific power associated *only* with the electric propulsion system, and τ is the mission time. If the total power system has a specific power of 30 W/kg for example, but 80% of this system is needed by the mission, then $\alpha=150$ W/kg. A thirty-day mission (with $k=2$ and $\epsilon=0.5$) would then have an optimum specific impulse of 2000 s.

There are several electric propulsion concepts that could satisfy this specific impulse requirement.[2] This Note discusses some aspects of a particular approach, based on a system that has already performed successfully (albeit modestly) on long term space missions,[3] the pulsed plasma microthruster (PPT). The principal interest is the introduction of electrical propulsion in a manner that can evolve as electrical power levels increase, drawing on actual flight experience while maintaining the physical processes in the thruster. Such introduction would benefit from experience with the PPT design.[4] That is, optimization of thruster performance can be sacrificed to system simplicity in order to match available power supplies and minimize risk. For example, the PPT uses a Teflon fuel block fed into the electrode region by a negator spring, and thus avoids repetitive valve operation and cryogenic fuel storage. Furthermore, arc discharges tend to operate at voltages that are independent of the charging voltage of the current source. The energy from a pulsed plasma thruster might vary slightly from shot to shot, but the requirement for voltage regulation of the power supply is much less severe than for other types of electric thruster.

This Note reviews some aspects of mission performance using pulsed plasma thrusters. Recent experimental work[5] suggests the possibility of a single thruster that could match increasing space-electric power levels.

Pulsed Specific Power

As noted in the Introduction, the optimum exhaust speed depends on the product of specific power α and mission time. For pulsed plasma thrusters, it is reasonable to expect that the specific mass α^{-1} will be dominated by the energy storage system. Such approximation is based on the recognition that most of the steady power components of the total electrical system (e.g., prime power source, heat rejection elements) may be used in performance of the spacecraft mission. The equivalent specific power of an energy storage component may be defined as the product of its specific energy E and the repetition rate f: $\alpha=Ef$. The product $\alpha\tau$ is then $\alpha\tau=EN$ where N is the number of shots in the course of the mission. The limiting value of N is the lifetime of the component.

The principal concern for pulsed plasma systems will probably be the capacitors used to accummulate energy from the power system between thruster firings. For the pulsed plasma microthruster,[3] a capacitor was developed with a product of specific energy density and lifetime that extrapolated to $EN=1.1\times10^9$ J/kg. This would be reasonably in excess of the $\alpha\tau$ used in the introduction ($\alpha\tau=3.9\times10^8$ J/kg). Since the pulsed plasma microthruster has operated on actual missions, firing once a second during the course of a year, it appears that a critical component for pulsed plasma thrusters applied to orbit raising has already been developed. (More recently, capacitors for quasi-steady MPD arcjet thrusters have been reported[6] with $E\approx80$ J/kg (kg and $N>10^7$).

Orbit Transfer LEO-to-GEO

It is useful to make some simple estimates of the mission capability that could be achieved merely by extrapolating PPT technology. The characteristic velocity increment needed for a LEO-to-GEO transfer is quoted as $\Delta v=6000$ m/s.[7] At a specific impulse of 2000 s, the fraction of initial mass delivered to GEO is then:

$$M_f/M_0=e^{-\Delta v/I_{sp}g_0}=0.74$$

The fractional mass of the propellant utilized is therefore:

$$M_p/M_0=1-(M_f/M_0)=0.26$$

To achieve this level of propellant utilization by ablation in the pulsed plasma thruster requires accummulation of ablation per capacitor discharge over the N shots during the mission. Experimentally, the rate of ablation is approximately 2.5 μg/J per shot. The total capacitor mass is then:

$$M_{\mathrm{cap}}=\frac{M_p}{(2.5\mu\mathrm{g/J})EN}$$
$$=0.1M_0$$

The mass of the payload (including the power source needed for both mission and propulsion tasks) is obtained by subtracting this capacitor mass from the delivered mass:

$$M_\ell=M_f-M_{\mathrm{cap}}=0.64M_0$$

This result overestimates M_ℓ by neglecting the mass of the thruster itself and other power conditioning required only by the propulsion system. However, a more exact calculation requires a more detailed specification of the mission; in particular, the power conditioning for propulsion could benefit from voltages higher than the 28 V generally available to meet mission needs, Therefore, it may be possible to access the spacecraft power utilizing mission-related power conditioning, rather than merely accepting bus voltage levels. The main point of the present calculation is that substantial fractions of

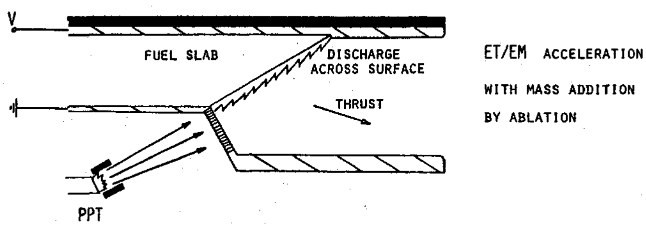

Fig. 1 Conceptual schematic of the use of a PPT with a second-stage thruster to obtain higher thrust by ablation of a fuel slab.

Presented as Paper 84-1931 at the AIAA/SAE/ASME 20th Joint Propulsion Conference, Cincinnati, Ohio, June 11-13, 1984; received Aug. 16, 1984; revision received March 8, 1985.

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the mass at LEO can be delivered usefully to GEO with technology that is based on present capabilities.

Variation of Mission Requirements

If the capacitor lifetime depends only on the number of shots, then higher power missions (shorter transfer times) can be accomplished simply with higher repetition rates and/or longer current pulsewidths. That is, to the extent that the capacitors dominate the specific mass, $\alpha\tau(=EN)=$ constant. The optimum specific impulse is also constant, so the delivered mass fraction depends only on the necessary Δv for the mission.

The size of the capacitor supply will increase with the size of the mission payload. For fixed transfer time, the average thrust required will also increase with payload mass. Since the number of shots N is set by the capacitor lifetime, the repetition rate is fixed in this case and the current pulsewidth increases with the mass of capacitors.

Recent experiments[5] indicate that plasma flow conditions similar to the PPT flow can be maintained for the length of the current pulse in a two-stage thruster system. This system uses an actual microthruster (for the LES 8/9 mission) to initiate current flow in an ablation thruster driven by a pulse forming network. The arrangement is shown conceptually in Fig. 1. Data suggest that after the initial microthruster plasma clears the second-stage, quasi-steady exhaust speeds of $1.7\text{-}2.5\times10^4$ m/s are achieved for the remainder of the flat-topped current pulse ($\sim160\ \mu s$). The impulse per discharge appears to scale with system energy over an order of magnitude variation in PFN energy. Preliminary estimates of kinetic efficiency divided by input electrical energy indicate an efficiency of 23-72%. It would thus appear possible to vary current pulsewidth and maintain useful efficiency at a specific impulse of about 2000 s.

Concluding Remarks

From earlier work on pulsed plasma microthrusters and recent efforts involving extended, constant current waveforms, the technology for a family of pulsed plasma thrusters could reasonably be applied to near-Earth missions, including orbit transfer. Such a family would allow electric propulsion to be introduced into the U.S. inventory in an evolutionary manner as space-electrical power becomes available.[8] In particular, it appears that an ablation arc on a Teflon slab, with combined electrothermal and electromagnetic contributions to thrust, provides an adequate propulsion mechanism in the I_{sp} range of 2000 s. Such an arc can be operated in both pulsed and quasi-steady conditions indicating that thruster physics can be maintained as prime-power levels extend from kilowatts to megawatts. Operational experience could thus be shared within this family of thrusters, providing confidence in performance as higher power missions are defined. Such confidence is probably critical to the practical introduction of electric propulsion in U.S. space missions over the next few decades.

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Explosion Phenomenon from Contact of Hypergolic Liquids

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Introduction

THE phenomenon of explosions induced by the contract of hypergolic liquid propellants has potential importance for the combustion process in rocket engines and for safety procedures. Friedman et al.[1] conducted experiments on the mechanisms of the explosions, using a falling droplet apparatus and specially designed devices to observe the process with a high-speed camera (up to ~7000 frames/s).

Based on sequences of ultrahigh-speed motion pictures having a sufficient power of resolution ($\sim10^5$ frames/s) and experimental observations of the characteristics of the explosion, such as the strength of the shock waves produced and the time lags of their occurrence, we suggested in Ref. 2 that the explosion in the case of a N_2H_4 (droplet)/N_2O_4(pool) system is caused by a sudden gasification of the thin surface layer of N_2O_4.

In the present investigation, we took ultrahigh-speed motion pictures of the explosion in the reversed system [N_2O_4(droplet)/N_2H_4(pool)]. The fluid dynamical process from the contact of the two liquids to the occurrence of an explosion was investigated in detail and compared with records of the N_2H_4(droplet)/N_2O_4(pool) system. There are marked differences between the properties (i.e., density, surface tension, boiling point, etc.) of N_2H_4 and N_2O_4.

Experimental Procedure

The experimental apparatus was similar to that described in Ref. 2. A Beckman and Whitley ultrahigh-speed camera was equipped with a xenon flash lamp having a delay circuit that could be operated for the required lighting period. Because the explosion induced by the contact of the hypergolic propellants could not be reproduced consistently, the time lag was varied in each test run. Thus, it was difficult to obtain the ultrahigh-speed motion pictures of the explosion phenomenon (the operating time of the high-speed camera was only 0.5 ms at 10^5 frames/s) and we had to accept the low possibility for obtaining successful records.

Results and Discussion

A selected run of high-speed motion picture records for the N_2H_4(droplet)/N_2O_4(pool) system is shown in Fig. 1. It was

Received Dec. 1, 1984; revision received April 10, 1985.

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