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Paper Session III-A - On-Orbit Characterization of Electric Propulsion in Leo Satellites

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ABSTRACT

Because of the current high cost for space experiments on large and small space qualified platforms, alternate methods of space characterization must be explored. Utilizing commercial or military satellites as testbeds for subsystems is a potential platform for small devices. Electric propulsion is a viable and upcoming subsystem that is of high interest to planetary mission engineers as well as commercial satellite developers. The cost of space demonstration, and the risk associated with non-space tested components, is a major driver in the reluctant admittance into the satellite and space experiment world for electric propulsion. It is proposed that by incorporating small lightweight electric propulsion devices onto small satellites as external or "bolt-on" experiments, an increase in the number of flight opportunities can occur. Specific problems that will be addressed are spacecraft body interaction, contamination effects, thermal interface problems, power conditioning control electronics, and propulsion feed system interfaces.

Introduction

Traditional approaches to space qualified hardware encompass rigorous ground test methods and operations. Risk reduction is achieved through extensive environmental and qualification tests to a component or subsystem. Although the majority of component, subsystem, and even system problems can be worked out this way, a particular area of difficulty is characterizing the system interaction of a subsystems operation in space, a specific example being electric propulsion devices.

Electric propulsion devices produce various forms of ionized and uncharged contaminant exhaust during normal operation. In addition, they generally require larger amounts of power over time than other subsystems. Thus, effects of the electric propulsion device interaction with the overall spacecraft are not easy to discover on the ground, given the hard vacuum requirement for actual operations. The most robust approach to characterization of an electric propulsion device with its host system is, therefore, in space. But this presents the disadvantages of high risk to an overall mission, because limited testing can be accomplished prior to its operation.

This paper will address the concept of packaging an electric propulsion device into "payload" constraints. Currently there are several programs underway within the US and Europe that are developing both small and large electric propulsion demonstration experiments.[1][2][3] Although this approach is not new, the idea of reducing the electric propulsion system power and propellant requirements, can open up more potential flight opportunities via small, lightweight military and civilian satellites. The types of electric propulsion devices proposed to test include de-rated ion engines, stationary plasma thrusters, and hydrogen and ammonia arcjets. Magnetoplasmadynamic(MPD) thruster testing is not proposed on small satellites due to its large size and power requirements.

Electric Propulsion Devices

Electric propulsion devices, whether classified as arcjet or ion engine technology, typically are designed for specific operating points. These points are set based on both physical characteristics of the thruster, for instance grid size and diameter on an ion engine, and nozzle characteristics on an arcjet, as well as the input power applied, varying both current and voltage. Applied power is most often the primary characteristic that couples an electric propulsion device with a spacecraft.

Limiting these characteristics to what is readily and realistically available on board a small lightweight spacecraft(here defined as Pegasus constrained volume and less than 250 kg to LEO), defines operating constraints, and therefore, specific electric propulsion devices that can be flown. Although certainly not true in all cases, a basic assumption shall be made on the linearity of electric propulsion devices to decreases and increases in operating parameters around their nominal operating points. Figures 1 and 2 show respectively graphs of off-nominal ground test operations for an arcjet and plasma engine. As shown, a trend can be seen on the thrust and Isp for both, that increases with power. Thus, by flying and characterizing a lower power electric propulsion system, extrapolations to higher power systems can be made on both performance and operating characteristics directly.

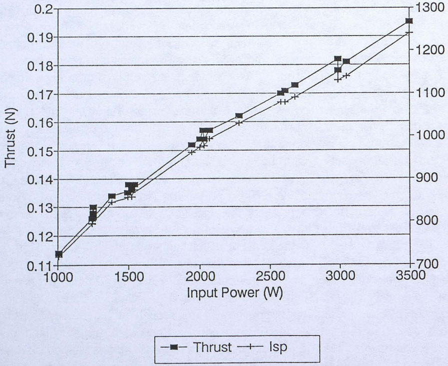

Figure 1. Arcjet Engine Operating Characteristics[4]

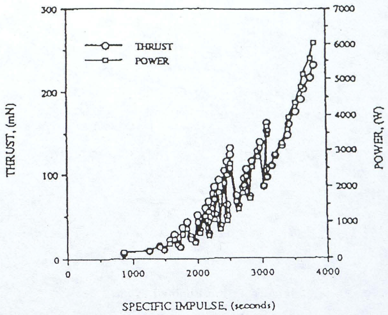

Figure 2. Ion Engine Operating Characteristics[5]

Figure 3 shows a representative arcjet system layout. Figure 4 shows a representative ion engine layout.

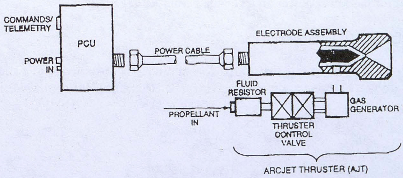

Figure 3. Arcjet System Layout(1.8kW Hydrazine RRC)

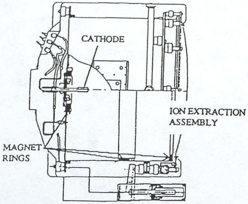

Figure 4. Ion Engine Layout

All electric propulsion systems include some type of power-related hardware which are used to implement electrical operation of the thrusters. This hardware includes the power processing units(PPU's), cabling, and depending upon the type of spacecraft operation, batteries, solar cells, or other power sources. Currently, PPU components are still large with respect to a small class spacecraft. Weight and volume constraints on a small spacecraft are such that packaging is a large consideration. Considering an electric propulsion device then, largely depends upon the volume available to insert the related hardware. Based on Figures 1 and 2, it is postulated that decreasing the input power to the electric propulsion unit, smaller components can be flown, including PPU's and their related cabling. This can be seen in the PPU's generated for the lower power arcjet engines from RRC/PED.[6]

<u>Electric Propulsion Integration on S/C</u>

To assess the premise that an EP system can be incorporated onto a small satellite, we will examine a sample spacecraft and electric propulsion device.

* EP Device Characteristics

Input Power	500W
Efficiency	30%
Isp	1000 sec
Total Propellant	4 kg

* Spacecraft Characteristics

Mass	225 kg
Total Power	600 W
Allowable Payload Weight	30 kg

For demonstration, it is proposed that a very short orbit transfer be initiated by the electric propulsion device on this satellite, after the primary mission has occurred. The mission parameters for this are listed below[7]:

Min. Altitude at startup	484 km
Final Altitude	800 km
Inclination	28.5 deg.
Total Delta V change	170 m/sec

The launch vehicle to be used would be less than 1000 Kg payload capacity. A Pegasus is used for example. The payload limitations are shown below to indicate the constraints on small spacecraft mass characteristics for various insertion altitudes:

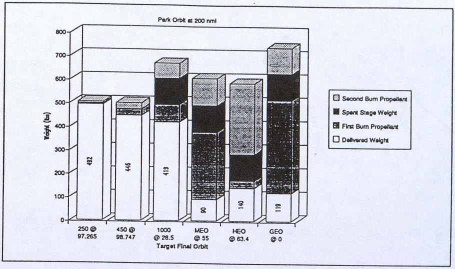

Figure 5. Pegasus Payload Capabilities Versus Altitude

Integration Methodology

There are two basic methods to integrate an EP system onto the spacecraft. The first consists of creating a structural module from the EP system components that attaches independently to the spacecraft. This module would "bolt-on" separately from other components and have minimal interface requirements. The second method incorporates the EP system internally designed with the spacecraft.

Bolt-on Method

The first method of attaching a module completely separate from other components of the spacecraft presents unique advantages and disadvantages. By creating an efficient package out of the EP system, this "payload" can be transported on a variety of different spacecraft within its diameter constraints. The interfaces between the spacecraft and the module are minimal, consisting of the mechanical fixture points or fixture plane, and the electrical umbilicals or wiring. All operation of the EP system would be carried out through a pre-defined set of logic parameters that are monitored and acted upon by the EP module. The module would execute its own EMI and thermal dissipation based on the spacecraft orientation and requirements. Figure 6 shows a concept designed by the Phillips Lab and the Jet Propulsion Laboratory.

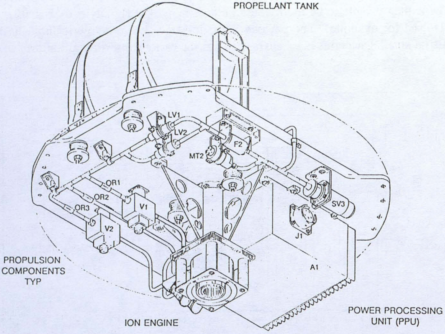

Figure 6. Electric Propulsion Module Concept.

This approach has been successfully achieved by the Russians in the development and flight operation of their Stationary Plasma Thrusters (SPT's) from the early 1970's[8].

Some disadvantages of this approach are apparent when trying to minimize overall payload weight. Because the EP module is independent, it must have a structure to hold all the components together. Based on previous payload development efforts, this structure would account for up to 15% of the EP module weight. Also, if the EP system is kept simple as outlined above(ie. no gimbaling mechanism allowed), there is only one specific attachment orientation that is allowed to the spacecraft to make the EP system effective in operation. Thrust vector offsets must be minimized by mechanical alignment prior to operation.

Integral Method

The second method using an EP system is to integrate all the components into the preliminary and final design of the spacecraft. To develop an efficient and economical union, this would mean a high degree of interaction by the payload designer between the EP system and the spacecraft early on. This allows utilization of unused physical volume of the spacecraft for additional instrumentation. It also allows a decrease in the EP system overall weight because extra structural elements do not need to be developed to house the entire structure. Existing or easily modifiable internal bulkheads and interface plates can be used on the spacecraft. Specific electric propulsion sizing can be accomplished more accurately, because of the flexibility in the various subsystem components(ie. different propulsion tanks, various thrust levels and thruster sizes, etc.) Balancing the spacecrafts axis of inertia to its required mission constraints(eg. viewing requirements), allows optimization of the consumables necessary.[9] Thus, flexibility in placement of the electric propulsion system components can decrease the required amount of propellant. (Although this is not necessarily a consideration in demonstration missions, it is a primary consideration in operational missions.)

This approach has been used more often than modular or wafer stage integration. The SERT experiments in the 70's and 80's specifically used this approach, as the spacecraft was designed for the electric propulsion device[10]. RRC has developed a low power hydrazine arcjet for specific use on the GE Series 7000 geosynchronous satellites for north/south stationkeeping.[11] These thrusters are incorporated into the design, and are specifically suited for this size and mass spacecraft.

In terms of demonstration missions, a small solar ion propulsion spacecraft was proposed for lunar science missions.[12] The concept was to develop a lightweight spacecraft to lower overall launch and mission costs to allow continuos space science exploration of the lunar surface. The spacecraft weighed approximately 150 kg, with approximately 1450 W of power at the beginning of the mission.

A disadvantage to this approach is the incumbent increase in complexity on the overall spacecraft design due to inclusion of another payload. Thermal and electrical interface concerns will increase, and a more detailed modeling effort may have to be done to keep thermal and electrical interference effects from becoming a problem.

Characterization

Tradeoffs between ground testing and space testing are normally made with regards to finances and mission opportunities for systems that utilize electric propulsion devices. Space qualification of any system is expensive and complicated. However, characterization of the electric propulsion systems overall performance interaction with the host spacecraft must be done on-orbit. Thus, ways to achieve this characterization must be done with low-cost and light weight components, or procedures dealing with the host spacecrafts capabilities.

Normally, spacecraft manufacturers do not ground test the EP system while it is attached to the S/C. Tank sputter and contamination are the biggest concerns with operational ground vacuum tests of integrated EP system on S/C bus. However, due to cost and risk, an arcjet was tested on a FLTSATCOM spacecraft at TRW to look at EMI and thermal effects.[13](It has been proposed that a "bell jar" can be placed over the spacecraft bus and critical components, isolating the electric propulsion device from the spacecraft, which would allow the electric thruster to be run inside a vacuum chamber as a ground systems test.[14])

Thus, characterization of the EP system interaction to the spacecraft is usually reserved for space. Characterization is extremely important to assess overall risk for future satellite designers. Space characterization is very important to verify the ground based data, thereby updating effective models to decrease future space testing. Several important parameters are specified here to characterize the EP system.

Characterization parameters for EP systems include;

* Life and degradation of performance
* Thrust performance
* Beam divergence angle and subsequent plume energies
* Electrical interaction of the plume with the spacecraft
* Thruster startup, and transition to steady state
* Thruster interaction with the spacecraft outgassing
* Thruster interaction with the overall environment
* Aspects of solar array degradation.

These and other parameters can be separated into areas of applications, namely

- Spacecraft Operation
- Array(Power source) Degradation
- Thruster/Spacecraft Interaction
- Thruster/Environment Interaction
- Thruster Performance

Spacecraft Operation

The parameters that characterize an electric propulsion device relative to the spacecraft can be described by the physical processes that drive the thruster. The particular voltage and current sent into the PPU, and into the thruster describes relative efficiency of the physical operation of the thruster. The specific hardware requirement can be easily met on a small spacecraft with voltage and current probes.

Propellant pressure gauges can measure pressure in the tanks, and be used to characterize the feed system performance. Varying the mass flow rate changes the performance parameters of the electric thruster. Thus, placing pressure gauges along the feed system path can reveal relative pressure drops versus time, and help understand transients in the thruster operation.

Temperature probes(eg. thermocouples or thermoresistors) show absolute temperatures on the PPU, the thruster body, thermal signatures near the thruster heat or isolation shield, and transients in the overall spacecraft temperature.

Array Degradation

Spacecraft interaction includes the articulated appendages that pose the largest area of exposure to an electric propulsion device plume, the solar arrays. Degradation to the arrays has been a subject of argument about possible effects. Some proponents believe that the plume effects will essentially sputter contaminants off of the array, rather than on. It may be that characterization of this phenomenon must be determined in the combined environment of space. Diagnostic equipment used to determine this effect can include carefully placed soar cells in varying positions near the plume; radiometers of several wavelengths to assess radiation emission effects; and possibly neutral mass spectrometers with different mass ranges to assess propellant products and sputter contaminants. Of these, only the solar cells would easily be integrated onto a small spacecraft.

Thruster/Spacecraft Interaction

The characteristics described under the spacecraft operation description all apply to describe the thruster to spacecraft interaction . Some additional diagnostic tools can be applied to further ascertain certain issues. Radio emission probes, or radio frequency antennas tuned to certain wavebands, can be used to determine interference to onboard telemetry. Also, by careful placement and use of the spacecraft telemetry equipment, the primary diagnostic RF equipment may be the spacecraft operational system.

Further interaction may be determined by electron temperature and density probes. Normally referred to as langmuir probes, these can determine relative dispersion of plume effects.(Specific electron densities must be considered. Depending upon their intensities, the langmuir probe may not be feasible due to a high accuracy requirement in positioning, which may not be possible on a small spacecraft.[15][16])

Again, a mass spectrometer can determine sputter or contamination products deposited by the plume itself, or the plumes interaction with the spacecraft body.

Additionally, electromagnetic noise and electromagnetic interference (EMI) affects, present or absent by the operation/non-operation of the thruster, are particularly important in assessing a thrusters interaction. Electrostatic field sensors and transformers placed at various points on the spacecraft surface can measure electric field intensity, frequency of noise pulses, and characterization of the EMI effects during operation.[17]

Thruster/Environment Interaction

Specific interaction between the thruster or its exhaust and the ambient environment is also of concern. Radio frequency antenna, radiometers, electron temperature and density probes, and mass spectrometers could all contribute to the diagnoses of how the thruster interacts with the ambient environment. Of particular interest is the relative voltage or charge buildup between the thruster exhaust and the spacecraft.

Thruster Performance

Almost all of the aforementioned diagnostic tools will help to determine the overall thruster performance. Internal operating characteristics can be determined from the basic health and status information presented in temperature, voltage/current, pressure and radiation data. In addition, high heat strain gauges may be used to characterize the physical processes that occur on and around the thruster itself.

Thrust performance(ie. force imposed by the thruster on the spacecraft) can be determined from actual operation of the spacecraft over a given amount of time. Ephemeris data can be used to calculate the thrust given an operating time and altitude. By demonstrating an orbit transfer, obvious performance calculations can be made with reference to the spacecrafts movement, easily tracked by a space surveillance system. Thrust performance can also be determined by the use of accelerometers along the longitudinal axis of the spacecraft. The use of onboard GPS data can also be used to determine exact inertial position over time.

Life

In any examination of electric propulsion, another characteristic must be discussed. Based upon the premise of limited propellant and power available for an EP system, life testing would not be a characteristic examined on a small satellite with electric propulsion. Life testing is normally done in ground facilities. This also allows testing of the entire power and propulsion subsystem and its characterization over different operating points.[13] Flow rate data could be determined by propellant pressurization drop over time versus the operating time. This could be characterized fairly accurately on the ground but can be varified in orbit.

It may be possible to achieve results on life test using accelerated methods. This would be done running a given electric propulsion device above its rated power level at a fraction of its stated life, and using a statistical averaging method to calculated lifetime performance.[18] It is questionable, without the actual retrieval of the device to measure any physical deformities, that a good life test characterization could be made with this method. Possible applications of remote cameras to examine the electric thruster nozzle for obvious signs of wear would provide degradation data.

Conclusion

By decreasing the input power and propellant required to operate a specific electric propulsion device, small experimental packages can be flown on numerous small satellites. Traceability to large powered systems can be done with small electric propulsion systems for future operations. Due to the mass and volume constraints that small spacecraft would impose, multiple missions with similar thrusters might have to be performed to evaluate different characteristic data, using distinct sets of instrumentation. Each mission could be tailored for a specific set of diagnostics equipment and characterization requirements, thus increasing the general operational knowledge over a series of experiments. This approach can be done today with existing technology and experimental apparatus, for use in tomorrows larger operational systems.

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