Overview of NASA's Advanced Electric Propulsion Concepts Activities

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ABSTRACT

Advanced electric propulsion research activities are currently underway that seek to addresses feasibility issues of a wide range of advanced concepts, and may result in the development of technologies that will enable exciting new missions within our solar system and beyond. Each research activity is described in terms of the present focus and potential future applications. Topics include micro-electric thrusters, electrodynamic tethers, high power plasma thrusters and related applications in materials processing, variable specific impulse plasma thrusters, pulsed inductive thrusters, computational techniques for thruster modeling, and advanced electric propulsion missions and systems studies.

INTRODUCTION

The most important accomplishment in electric propulsion during the past year has been the launch and successful operation of the New Millennium Deep Space 1 (DS-1) mission.[1] The is the first time that NASA has flown a planetary spacecraft employing Solar Electric Propulsion (SEP) with a xenon-propellant ion engine as the primary (i.e., main ΔV) propulsion system. (Earth-orbit flight tests in the 1960s were with mercury-propellant ion engines.) Thus, ion propulsion has made the transition from an advanced concept to one whose feasibility has been fully demonstrated in a flight test. Further technology development will address issues such as thruster lifetime, efficiency, mass (and specific mass, kg/kW_e), and power-per-thruster.

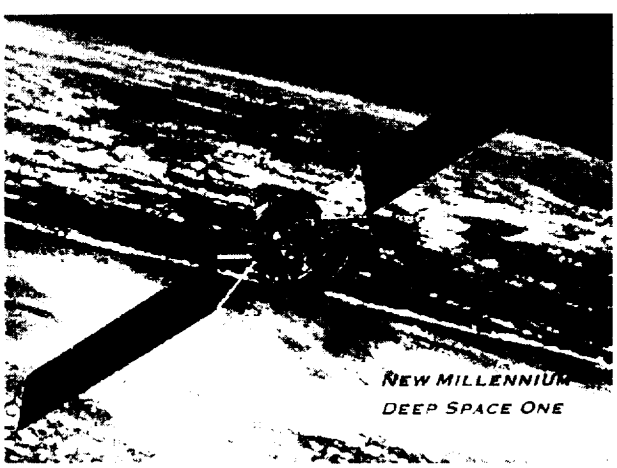

Figure 1. New Millennium Deep Space 1 Spacecraft

However, there are already missions appearing on NASA's science roadmap that cannot be done with existing propulsion technology, including near-term ion propulsion. Some of these missions will require impulse bits that are extraordinarily tiny and precise, while others will require high-power spacecraft that will pass by Voyager on their way out of our solar system.[2] To address the needs of ambitious missions of the 21st century, NASA has in place a research program that is focused on evaluating the feasibility of a variety of advanced propulsion concepts that offer a significant departure from existing capabilities. The focus of this paper is on advanced electric propulsion research being performed at several NASA centers, including the Marshall Space Flight Center (MSFC), the Glenn Research Center at Lewis Field (GRC), the Johnson Space Flight Center (JSC), and the Jet Propulsion Laboratory (JPL), and in conjunction with a number of Academic institutions and commercial organizations. Also, new, novel concepts and technologies are being identified and pursued in the industrial sector by means of the Small Business Innovative Research (SBIR) program.

The various advanced propulsion research activities are, in some cases, supported by more than one organization within NASA. There is a core budget allocated for the Propulsion Research Program within the Advanced Space Transportation Program (ASTP) by the Office of Aeronautics and Space Transportation Technology (OASTT) at NASA Headquarters. The ASTP is managed at the Marshall Space Flight Center. Since the end of 1997, the Advanced Propulsion Concepts activity at JPL has become a part of the ASTP Propulsion

Research Program, resulting in one integrated activity NASA-wide. The ASTP activity includes a broad range of advanced propulsion technologies such as advanced chemical propulsion, nuclear fission, fusion, and antimatter propulsion concepts, beamed-energy propulsion, tethers, advanced launch concepts, and breakthrough propulsion physics, in addition to advanced electric propulsion. Additional support for some of the advanced electric propulsion research described below has come from other sources within the ASTP, NASA Headquarters, discretionary grants at individual NASA centers, the NASA Institute for Advanced Concepts (NIAC), and SBIR grants.

NASA has undertaken some fascinating research of advanced electric propulsion technologies that may ultimately lead to accomplishing the most challenging, exciting deep space missions ever attempted. In the following sections, these propulsion technologies, which range from the very small and precise to the very large and powerful, are described and some of their potential applications are explored.

MICRO-ION ENGINES

Introduction

There currently exists a strong interest within the aerospace community for micropropulsion devices capable of delivering very small impulse bits over a wide range of thrust values, and for engine sizes and masses orders of magnitude smaller than are available with current technologies. This interest results from the drive to explore the feasibility of microspacecraft, typically viewed as spacecraft having wet masses on the order of 10-20 kg and below, as well as the need for fine attitude control of larger spacecraft like those envisioned for space interferometry missions.

While some micropropulsion devices will be used predominantly in pulsed operational modes for attitude control purposes, others may require a more continuous mode of operation. Two examples for such an application are for high-ΔV, primary propulsion on missions to small bodies (comets and asteroids), and for continuous disturbance torque compensation or drag make-up on larger spacecraft.[3]

In cases where spacecraft wet mass has to be kept low, the use of high specific impulse (I_{sp}) propulsion devices may be a necessity to keep required propellant masses small. One approach to high-I_{sp} micro-propulsion is miniaturization of ion propulsion technology.[4]

The feasibility of reducing ion engine sizes dramatically below state-of-the-art technology to engine diameters in the 2-3 cm range and thrust levels in the sub-mN range requires an investigation of several key issues. These include miniature accelerator grid system fabrication and operation, the replacement of hollow-cathode technologies with lower power consumption miniaturized cathode systems, the sustainability and efficient operation of high surface-to-volume ratio plasma discharges, and the feasibility of fabrication and operation of miniaturized power conditioning units and feed system components.

The feasibility of ion engine grids based on MEMS (Microelectromechanical Systems) fabrication techniques is being explored. Silicon-based MEMS techniques were investigated first due to the considerable heritage and experience available with these methods. Furthermore, MEMS fabrication can produce extremely small feature sizes within very tight tolerances of 1 μm or less. However, MEMS-fabrication of accelerator grids opens up a host of manufacturing and operations-related issues. Foremost among them is the selection of appropriate grid materials. Specifically, the grid insulator material isolating the screen and accelerator voltages from each other will have to stand off voltages on the order of 1.3 kV or more over distances of a few microns if current grid voltages and engine specific impulses are to be maintained.

While the grid technology represents one key component in a micro-ion engine, another area presenting unique feasibility issues is the cathode. With robust field emitter materials to withstand hostile thruster environments and low operating voltages, a Field Emitter Array (FEA) cathode is a plausible candidate as a low power and efficient electron source for micropropulsion systems.[5] Testing of single tips and arrays of FEAs has been conducted in higher pressure regimes than these devices typically see in flat panel display applications. Successful operation of these devices at higher pressures is one of the critical technologies for micro-ion engines as well as a promising technology for electric propulsion systems using in-situ-produced propellants (e.g., oxygen)[6] and plasma contactors for electrodynamic tethers.

Finally, discharge chamber modeling is a critical step in the development of an ion engine. The use of a gaseous discharge poses a key feasibility issue to this technology since in small plasma volumes a considerable degree of electron wall losses may occur.

Status

Micro-Accelerator Grids

As part of the on-going feasibility assessment of miniature ion engine accelerator grid systems, experiments aimed at determining the suitability of microfabricated thin oxide films as insulators in micro-ion engine grids have continued. An investigation of new grid/insulator materials and geometries is underway as it was evident from previous experiments[7,8] that the field concentration at specific locations play a major role in oxide insulator breakdown.

Silicon dioxide was investigated for use as a grid insulator material because it exhibits good electric insulating characteristics when compared to other materials used in silicon-based MEMS fabrication, and is already widely used in the microfabrication field. In order to study the suitability of silicon oxide for this application, both substrate or bulk electric breakdown characteristics as well as electric breakdown along the surface needed to be studied. Two sets of experiments were conducted using specially designed silicon oxide breakdown test chips to systematically study both modes of electric breakdown. Data obtained in these experiments may also be of relevance to other MEMS-based electric propulsion applications.

Figure 2. View of Substrate Breakdown Test-Chip

Substrate breakdown field strengths ranged between 600 - 700 V/μm. Tests performed at elevated temperatures revealed only a limited drop in breakdown field strength of about 15% as the temperature was increased from ambient to 400°C. Thus, as far as bulk breakdown field strength is concerned, silicon oxide films would appear suitable for use in ion engine grids. However, surface breakdown field strengths of only 200 V/μm were measured over a gap distance between contact pads of 5 μm. Thus, even a 5 μm film thickness (representing the approximate maximum silicon oxide thickness that can be deposited) would not prevent breakdown along the grid aperture walls at the required 1.3 kV. It therefore appears that surface breakdown field strengths of silicon oxides are not sufficient for these oxides to be used in ion engine grids. This experiment, however, does not precisely simulate the geometry found in microfabricated grids along aperture walls and thus any conclusions drawn remain preliminary.

Future work will include the investigation of very thick Plasma Enhanced Chemical Vapor Deposited (PECVD) oxides at thicknesses up to 15 μm, provided by Alberta Microelectronic Corporation, non-silicon based micromachining methods, or the development of multiple grid structures. Also, it may be possible to split the required voltage drop of about 1.3 kV between several grid pairs, the voltage drop between each pair being lower and inversely proportional to the total number of grids. Limits may arise as to how many grids may be stacked on top of each other due to intrinsic thin film stresses which may result in poor adhesion to the substrate. In the case of multiple stacked grids, thermally grown oxides may become of interest again, as the required oxide thickness decreases in such a stack and thermally grown oxides provide higher breakdown field strengths for comparable thicknesses.

Field Emitter Array Cathodes

Field emitter cathode arrays are micron-sized, batch-fabricated structures with sharp conical electron emitting electrodes and gate electrodes with small apertures that extract electrons. A single element of an array is shown in Figure 3. An array of these structures is shown in Figures 4-5. Typical FEAs employ molybdenum or silicon tips fabricated by etching and deposition techniques. The emitter tips are typically ~1 μm tall, have a radius of curvature down to tens of angstroms, and can be fabricated in arrays with packing densities greater than 10^7 tips/cm^2.

Work at the University of Michigan and JPL had been focused on the key feasibility issue of FEA cathode operation in higher pressure environments. This work includes modeling of field emission array cathode lifetime and emission limitations in electric propulsion systems.[5]

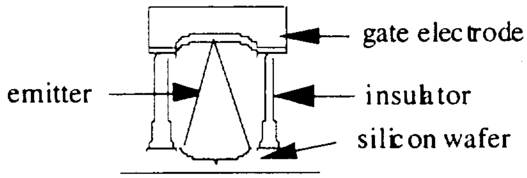

Figure 3: Schematic of a Single Element of a Field Emitter Array

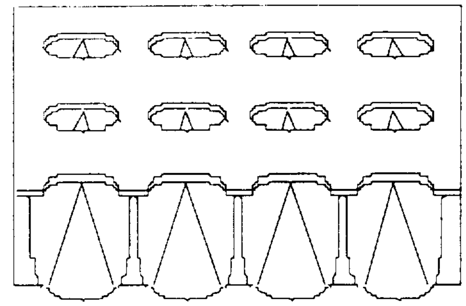

Figure 4: Schematic of a Field Emitter Array

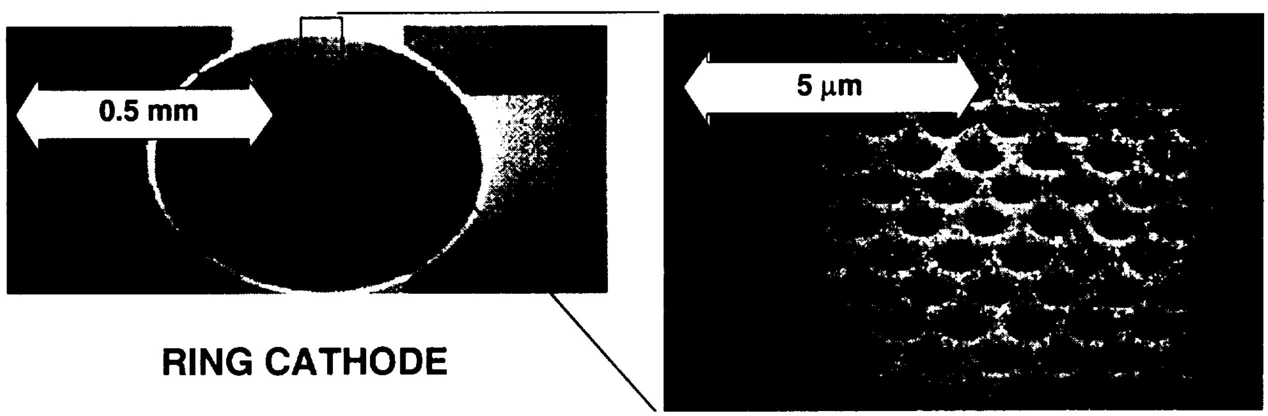

Figure 5. Ion-Beam Lithography Molybdenum Field Emitter Arrays (Photo courtesy Stanford Research Institute)

For example, individual field emission array (FEA) cathode tip failures were examined in a 10x10 tip array. Failures were generated at elevated pressures, then examined using a scanning electron microscope and compared to failures generated at lower pressures. Also, work on FEA cathode fabrication continued with deposition of the molybdenum gate electrode on silicon wafers with an oxide insulator at the California Institute of Technology (Caltech). Three photolithography masks were designed for patterning the molybdenum gate electrode.

Field emission array cathode lifetime and emission limitations in electric propulsion systems were modeled. The focus of lifetime limitations has been on tip sputtering in hostile thruster environments which degrades tip performance in time. Different models have been coupled to predict tip lifetimes as a function of cathode materials and configuration, operating voltages, and local neutral particle pressures. Predicted lifetimes are much lower than those typically observed. It is believed that the weakness with the model is in the sputter yield data. More accurate data are needed, and should be available soon.

A micro-einzel lens for focusing and decelerating electron beams, and repelling ions was also designed. Production of a multi-electrode micro-sandwich has been the recent focus. Work on the fabrication of the einzel lens structure continued with the deposition of sandwich structures at Caltech. Photolithography masks have been designed and fabricated for pattern arrays of 10,000 apertures per structure. Reactive ion etching is now being employed to etch the apertures in the sandwich structure.

Finally, a sheath model was developed to predict field emitter cathode space charge limitations when emitting electrons into a plasma where a virtual anode will collect the electron current (instead of a physical anode commonly used with these cathodes). This model, still under development, is being used to predict the lifetime of the cathodes in the hostile environment of a Hall thruster. This model was used to estimate cathode lifetimes and optimize cathode material. It was determined that the lifetime of silicon cathodes will exceed that of molybdenum cathodes. Therefore, cathode testing began with silicon cathodes.

Discharge Chamber Modeling

Initial work in this area was performed by Yashko and Hastings at MIT, studying MEMS-based discharge volumes.[9] Results obtained within that study indicated that targeted discharge chamber volumes were too small and led to a re-configuration of the micro-ion engine project, now focusing on a MEMS-hybrid design approach, i.e., a miniature, conventionally fabricated discharge chamber interfaced with various MEMS-based components, such as the field emitter arrays and, potentially, microfabricated grids. Additional modeling on these somewhat larger discharge volumes is about to be initiated in a collaboration between the Air Force Research Laboratory (AFRL) and JPL. A Direct Simulation Monte Carlo (DSMC) code available at AFRL will be used to model neutral flows. Results obtained with that code will be used as input values for a Particle-in-Cell (PIC) code available at JPL, simulating the charged particle behavior. This iterative approach (running a DSMC code first, followed by a PIC code) is feasible because ionization fraction remain small; for example, an ionization fraction of less than 10 % is expected in the cases of micro-ion engines. Results will be used to determine how small the discharge chamber volume can be designed without having to machine actual chambers and fabricate thrusters to measure performances, an approach which would be time-consuming and incompatible with current research budgets. Data of interest to be used as design criteria for ion thrusters will be discharge efficiency or electron production cost. Other data of interest include discharge fractions. Different chamber geometries, neutral flow patterns, grid transparencies, as well as discharge concepts will be explored for various discharge pressures and power levels. Following a narrowing-down of design parameters, thruster units will be built and performances verified.

ELECTRODYNAMIC TETHERS

Introduction

Electrodynamic (ED) tethers offer a method of propellant-free propulsion for orbit transfer and power generation in the Earth's magnetosphere. By using the voltage induced in a long conducting tether moving through the Earth's magnetic field and closing the circuit through the surrounding plasma, a drag force is created that can be used to lower a spacecraft orbit and simultaneously generate power. Alternatively, using power supplied by the spacecraft, driving a current down a tether produces a force in the reboost direction along the orbit. Actual current in a tether is determined by the ability of a tether's plasma contactor to eject electrons, and for a bare conducting tether and endmass to collect electrons. Feasibility studies have also been undertaken to examine the use of electrodynamic tethers in the Jovian magnetosphere.[10] (Jupiter's rapid rotation produces a condition where a tether can produce power and raise orbit passively and simultaneously at orbits above 2.2 Jupiter radii.)

Status

A near-term demonstration of ED tether technology is the ProSEDS (Propulsive Small Expendable Deployer System) mission, under development at NASA MSFC, which has been selected as a Future X mission. ProSEDS is an effort to validate bare-wire tether technology in space and demonstrate the use of electrodynamic tether propulsion for the accelerated deorbit of a Delta II second stage. The planned 2000 launch of the ProSEDS flight experiment is a precursor to several applications, including electrodynamic tether upper stages, and International Space Station reboosting.

ProSEDS is a 20-km long tether, 5 km of which is bare aluminum wire. The tether will be released in an upward deployment from a 400-km orbit. Approximately 100 Watts is anticipated to recharges batteries for downlink telemetry, along with a 5 to 10 km per day altitude drop of the Delta II upper stage from its initial altitude.

The use of electrodynamic tethers for deorbit propulsion would eliminate the need to preserve onboard fuel for deorbit maneuvers at the end of life. As such, there is significant commercial interest in the technology for use on planned telecommunications satellite systems; this application is being explored by Tethers Unlimited under the Small Business Innovative Research Program.

HIGH-POWER ELECTRIC PROPULSION THRUSTERS

Introduction

Many planetary spacecraft missions using SEP can use ion engines operating at modest power levels, such as the NSTAR (NASA SEP Technology Application Readiness) ion engine on the New Millennium DS-1 spacecraft (2.5 kW_e). However, high-power SEP or nuclear electric propulsion (NEP) power systems and thrusters with a high power-per-thruster will ultimately be required to efficiently process the hundreds of kilowatts to megawatts of power required to enable ambitious missions of the 21st Century. Thus, part of the effort in the advanced electric propulsion area is aimed at development of high-power thruster concepts.

For example, Hall thrusters are gridless ion engines that produce thrust by electrostatically accelerating plasma ions out of an annular discharge chamber. The concept was originally conceived in the USA, but it was only in the former Soviet Union where it was successfully developed into an efficient propulsion device. These thrusters have an I_{sp} in the range of 1500 to 2500 lb_f-s/lb_m that makes them ideal for near-Earth-space missions. Hall thrusters have been used on a number of Earth orbit missions at thruster power levels up to about 5 kWe. Advanced Hall thrusters with an order-of-magnitude greater power per thruster will be needed for future cis-lunar orbit raising missions (e.g., in support of lunar base cargo missions) for SEP vehicles operating in the few hundred kW_e range.

Similarly, MW-class SEP or NEP vehicles will be needed to support human planetary exploration missions. The Lorentz Force Accelerator (LFA) is a high-power, steady-state magneto-plasma-dynamic (MPD) thruster that uses a multi-channel cathode and lithium (Li) propellant to provide high efficiency and long lifetime previously unattainable with gas-fed MPDs. A 500-hour lifetime test of a 500-kW_e Li-LFA at Energia in Russia has shown negligible erosion and a thrust efficiency of 55% at a specific impulse (I_{sp}) of 4500 lb_f-s/lb_m and a thrust of 12.5 N. The demonstrated capability of a single thruster to process MW-level power with a 25 N/MW thrust-to-power ratio at high specific impulse renders the Li-LFA an ideally suited option for heavy cargo and piloted missions for planetary exploration. For example, the figure below illustrates an NEP system that was studied by JPL and the DoE Energy Technology Engineering Center (ETEC) for a Mars Cargo Mission.[11] The vehicle was designed to deliver a 90-MT payload to Mars in support of a Piloted Mars Mission where the crew travel to Mars in a separate vehicle. The vehicle employed three SP-100 derivative nuclear-electric power systems (with dynamic thermal-to-electric power conversion) that provided a total of 1.5 MW_e, and Li-propellant LFA thrusters.

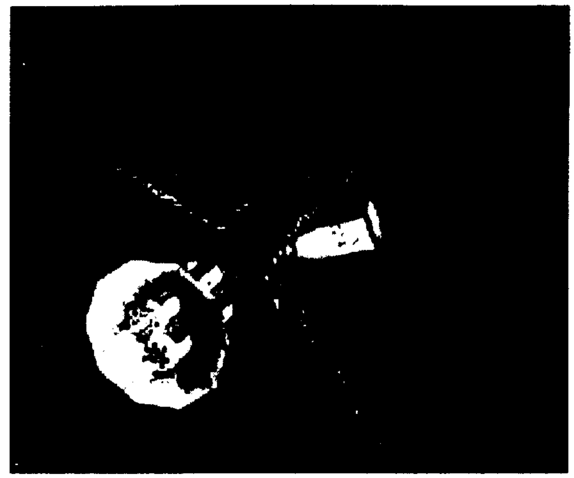

Figure 6. 1.5-MW_e NEP with Li-LFA Thrusters for the Mars Cargo Mission

Status

The NASA GRC is currently testing a 50-kW_e Hall thruster for high-power orbit raising mission applications. This work is described in detail by Dunning and Sankovic.[12]

Li-propellant LFA work is a cooperative effort between the Moscow Aviation Institute (30-years experience in designing and testing Li-LFAs, high-power engine testing, modeling of acceleration processes), Princeton University (thruster performance and plasma diagnostics, thruster physics modeling), Thermacore, Inc. (Li handling experience, refractory metal and heat pipe technology, flight hardware capability), MSFC (anode thermal modeling), and JPL (high-current cathode physics modeling, endurance testing, mission/systems analyses) as lead.

Cathode modeling by JPL has indicated that the addition of a small fraction of barium to the lithium propellant stream has the potential to substantially decrease the cathode operating temperature (yielding large gains in lifetime). The Moscow Aviation Institute (MAI) began preliminary tests of a 30 kWe Li-fed LFA thruster this year with barium addition. These experiments showed a 350-400°C drop in cathode temperature with the proper flow of barium. Too much barium addition to the propellant flow resulted in high discharge voltages and low currents. Follow-on experiments will focus on more controlled barium addition. A 30-kWe engine was also used in wear tests. Scaling relations were developed to properly simulate conditions in a high-power engine. Results indicated that a calculated wear rate of 13.5 ng/Coulomb appeared to be dominated by ignition erosion; steady-state erosion is expected to be 0.1-1.0 ng/Coulomb, implying engine lifetimes of hundreds to a few thousand hours. Finally, MAI has begun further characterization of the Li-LFA thruster performance at 200 kW_e and a study of the effect of barium addition on cathode temperature.[13] Preliminary measurements of performance yielded 49% efficiency at 185 kW_e and an I_{sp} of 4100 lb_f-s/lb_m.

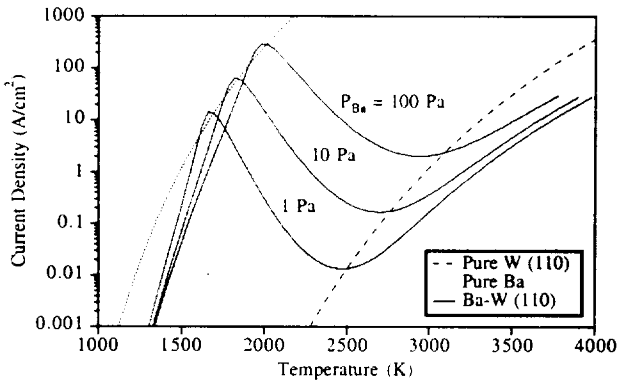

Figure 7. Li-LFA Cathode Thermal Model Showing Effect of Barium Addition

Figure 8. Moscow Aviation Institute (MAI) 200 kW_e Li-LFA Thruster

Princeton University and Thermacore, Inc. have developed the open-ended heat pipe Li-LFA shown in Figure 9. Princeton is continuing to perform tests of a high power thrust stand with an argon-fed LFA prior to testing an MAI 30 kWe engine. Work on LFA discharge modeling using a novel solution technique has focused on the development of a 1-D model which has been benchmarked against analytical solutions for certain cases. This model is being extended to 2-D and applied to Li-fed thrusters. A 3-D thermal model has also been developed.

Figure 9. Open-Ended Heat Pipe Li-LFA

Finally, JPL has begun facilities modifications in preparation for endurance testing of LFA thrusters.

DIAMOND FILM DEPOSITION TECHNOLOGY

Introduction

Diamond possesses a combination of properties which makes it uniquely suited to a number of scientific, military, and industrial applications. Work performed by Caltech and JPL over the last several years has identified potential benefits of diamond-like materials in advanced thrusters. One such application is the use of chemically vapor deposited (CVD) polycrystalline diamond as a coating for ion thruster grids subject to the life-limiting effects of sputter erosion from charge exchange ions.[14,15] The first phase of this activity was the measurement of the sputter yield of polycrystalline diamond relative to molybdenum and carbon-carbon composite (candidate ion engine grid materials). Other potential propulsion related applications include coatings for Hall thruster insulators, nozzle throats in chemical rockets, and fabrication of cold cathode field emitters.

One approach to large scale production of chemically vapor deposited diamond is the use of plasma discharges to create the required chemical environment for growth. DC-arcjet sources have been successfully used in the synthesis of diamond and have demonstrated higher rates of deposition than other sources using different gas activation processes. Materials processing with DC-arcjets represents an effective dual use of this technology which has benefited from extensive studies characterizing arcjet thrusters performed over the years. Other electric propulsion technologies may also possess unique advantages for materials processing. Although the Lorentz Force Accelerator (LFA) may be well suited to diamond synthesis, its potential has not been explored.

Status

Beginning this fiscal year, several tests were performed by Caltech researchers using a hydrogen/argon LFA plasma source located in the JPL Cathode Test Facility. Initial testing was aimed at determining the optimal location for a water-cooled molybdenum substrate to be used as a surface for diamond film deposition. Substrate location is of critical importance to the overall process because of its effect on temperature of the growth surface, flux of important species to the growing film, and also because of the desire to have optical access for spectroscopic analysis of the boundary layer directly above the growing film. For a typical operating current of approximately 900 A, a substrate location was selected which resulted in surface temperatures in the 800 – 1100°C range.

Testing then began using methane gas addition. In the Plasma Assisted Chemical Vapor Deposition Process (PACVD), methane serves as the source of carbon for diamond film growth. The methane is injected downstream of the anode directly into the high velocity hydrogen/argon plume from the Lorentz Force Accelerator.

When methane is injected into the high-enthalpy hydrogen/argon plume, it dissociates into a variety of species including acetylene, methyl, and atomic and molecular carbon. Chemical kinetic calculations performed using the Sandia CHEMKIN software package suggested that shorter residence times of the methane in the high-enthalpy plume would increase methyl concentrations (which is favorable for diamond growth) at the growth surface. These calculations suggested that moving the injector closer to the substrate would result in a shorter residence time for the methane in the plume and a correspondingly higher concentration of methyl at the surface of the substrate. This was confirmed in a subsequent 70 min. run which produced the first evidence of diamond crystallites. A second test lasting 75 min. confirmed the initial results and produced a continuous film 4.5 microns thick. Scanning Electron Microscope (SEM) images of this film are shown below.

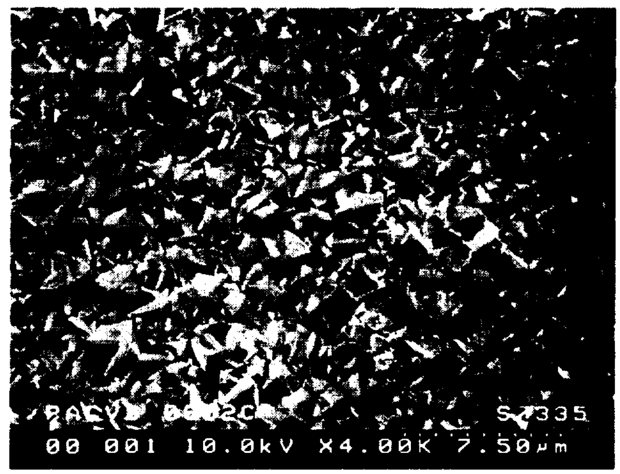

Figure 10. Scanning Electron Microscope (SEM) Image of Diamond Film (Magnified 4000 X) Deposited in Cathode Test Facility Using Lorentz Force Accelerator as Plasma Source
(Run time was 75 minutes with a discharge power level of 14 kW)

Figure 11. SEM Image of Diamond Film (Magnified 40,000 X) to Show Faceting of Single Crystal

Figure 12. SEM Image of Edge View of Diamond Film (Magnified 10,000 X) Used to Determine Thickness (Approximately 4.5 µm)

Having established the viability of the process, the facility was modified with a port installed to enable optical access of the substrate. This port will be used for spectroscopic observation. Additional tests were then performed to investigate the effects on growth rate and film quality of different methane/hydrogen gas ratios. These tests are continuing.

VARIABLE SPECIFIC IMPULSE HIGH POWER PLASMA THRUSTERS

Introduction

The VASIMR (Variable Specific Impulse Magneto-plasma Rocket) thruster represents an application to propulsion of RF and microwave heating methods and magnetic confinement technologies originally developed for fusion power research. The device is electrodeless and uses both electrothermal and electromagnetic processes to convert electrical power into directed kinetic energy.

The distinguishing characteristic of this high-power (potentially up to 10 MW_e per thruster) device is the ability to vary thrust and specific impulse during a mission over a wider range than a comparable-power ion or MPD/LFA system. This is accomplished by the injection of neutral (cold) hydrogen into the hot, high-energy exhaust stream of plasma exiting the engine. For example, the VASIMR has the potential to deliver relatively high thrust (1000 - 2000 N) and high specific impulse (3000 - 30,000 lb_f-s/lb_m) at multi-megawatt power levels, thus providing in a single thruster an engine that is competitive with high powered NEP systems (which operate at lower thrust), and nuclear thermal rocket systems (which have a much lower I_{sp}). This feature could benefit missions such as high-payload, low-speed, minimum-energy cargo transits as well as lower-payload, high-speed piloted trips. Additionally, because of its relatively high thrust at high I_{sp}, powered abort modes essential for human interplanetary missions may be enabled by this technology if it could be successfully developed.

Status

Work on the VASIMR has been underway since the early 1980s, first at the MIT Plasma Fusion Center, and currently at the Advanced Space Propulsion Laboratory at NASA JSC in conjunction with the University of Texas at Austin, the University of Maryland, Rice University, University of Michigan, and MIT. Most of the work to date has focused on understanding different aspects of the ion cyclotron heating. A large part of this was studying the different types of wave propagation through which energy is coupled to the plasma. In addition to experimental investigations, numerical models have been developed to study wave propagation and energy coupling efficiency as well as plasma and neutral jet interactions in the exhaust stream. Some mission analyses have been performed to explore the advantages of varying both thrust and specific impulse for a variety of human and robotic Mars missions.

The large, laboratory-scale thruster at JSC just recently began operation on hydrogen. In addition, JSC has received funding for a Phase A study of the use of a smaller, 10-kW_e VASIMR thruster on the Radiation Technology Demonstration (RAD) mission. This mission would use an SEP system to travel through the van Allen radiation belts; eight micro-satellites would be deployed at different altitudes to evaluate the effects of a slow SEP transfer through the radiation fields that would be encountered in a LEO-to-GEO SEP mission. Another important goal would be the flight validation of a 10-kW_e xenon-propellant Hall thruster, as well as the 10-kW_e hydrogen-propellant VASIMR thruster. An interesting synergism with the liquid hydrogen VASIMR propellant would be its use to cool the VASIMR's superconducting magnets developed by Oak Ridge National Laboratory.

PULSED INDUCTIVE THRUSTERS

Introduction

The pulsed inductive thruster (PIT) uses a flat induction coil (approximately 1-m diameter) and a fast gas valve to inject a few milligrams of propellant over the coil. Once the gas has been injected, a bank of high-voltage, high-energy storage capacitors is discharged providing a large azimuthal current pulse to the coil. The time-varying electromagnetic field caused by the current pulse ionizes the propellant gas and causes the ionized gas to accelerate away from the coil. Because the energy is inductively coupled into the plasma, the device can be designed so that the plasma has minimal contact with thruster surfaces, resulting in minimal erosion of thruster components.

Another advantage is that the PIT can be operated on a variety of propellants, such as hydrazine, ammonia, argon and carbon-dioxide, and at specific-impulses ranging from 1,000 to 6,000 lb_f-s/lb_m. Typically the efficiency ranges between 20 and 40% below 3,000 lb_f-s/lb_m and between 30 and 60% in the 3,000 to 6,000 lb_f-s/lb_m range. In order to make this system practical, development of a long-life, high-repetition-rate propellant valve is needed. Another feasibility issue is the development of high-repetition-rate capacitors and the associated solid state switches. Recent development of these devices for other applications suggest that they may achieve desired lifetimes; however, incorporation and testing of these components on a PIT is required to verify this.

Status

Previously, development work on the PIT had been done in the private sector at TRW. Recently, NASA MSFC has begun investigation of the PIT (supported in part by NASA GRC) in conjunction with Auburn University.

NASA INSTITUTE FOR ADVANCED CONCEPTS (NIAC) RESEARCH PROGRAMS

Introduction

There are a number of advanced propulsion concepts that are being supported by the NASA Institute for Advanced Concepts (NIAC) research program.[16] This activity seeks to identify and nurture new, innovative advanced concepts from industry and academia.

Status

Many of these research tasks have just recently begun; the advanced electric propulsion concepts selected for study are listed below. Detailed descriptions can be obtained from the NIAC website (URL: http://www.niac.usra.edu).

- "Low-Cost Space Transportation Using Electron Spiral Toroid (EST) Propulsion"
  Clint Seward, Electron Power Systems, Inc.
- "Pulsed Plasma Power Generation"
  Clark W. Hawk, University of Alabama-Huntsville, Propulsion Research Center
- "The Mini-Magnetospheric Plasma Propulsion, M2P2"
  Robert M. Winglee, University of Washington
- "Primary Propulsion for Deep Space Exploration"
  Dr. Michael R. LaPointe, Horizon Tech Development Group
- "The Magnetic Sail"
  Robert Zubrin, Pioneer Astronautics

VIRTUAL EXPERIMENTS FOR ADVANCED ELECTRIC PROPULSION USING MASSIVELY PARALLEL SUPERCOMPUTERS

Introduction

Advances in computer hardware and software make it possible to perform detailed numerical simulations of electric propulsion plasma interactions that, until recently, would have been impractical without the use of numerous simplifying approximations. These computing advances make it possible to numerically model, with a high degree of fidelity, the various plasma processes occurring inside the electric thruster. The models can be used to optimize thruster performance,[17] as well as perform detailed simulations of interactions between the electric propulsion thruster (e.g., thruster electromagnetic field, exhaust plume, etc.) and the spacecraft's subsystems (e.g., telecommunications, science instruments, etc.).[18] Thus, we can now perform "virtual" experiments that would otherwise be difficult or impossible to do in an actual laboratory environment.

The DS-1 mission provides a space-based laboratory for investigating ion engine interactions with the spacecraft. This allows preliminary validation of computer models with actual measurements.[1] For example, preliminary results from the IDS and PEPE instruments of DS-1 for the ion engine-induced plasma environment have been analyzed. The measurements indicate a charge-exchange ion current of 10^{-7} A/cm^2 and a charge-exchange ion density of 10^6 cm^{-3} at the IDS location. The observations agree well with computer simulation results by Wang et al. published in 1996.[18]

Status

In a joint research effort with the Air Force Research Laboratory, JPL has developed a full 3-D particle-in-cell (PIC) simulation model to study plasma interactions associated with non-neutralized ion beam emissions in space. This model may also be applied to study the electron dynamics in the near field of an ion thruster. Additional research is aimed at developing a multi-purpose 3-dimensional hybrid electromagnetic PIC code for global-scale kinetic plasma simulations (on state-of-the-art parallel supercomputers) to model the effects of electric propulsion thrusters on science measurements.

Also, in a joint study with the University of Michigan, JPL has developed a 3-dimensional particle simulation model that can predict space charge limited emission from field-emitter cathodes.

ADVANCED ELECTRIC PROPULSION MISSION ANALYSES STUDIES

Mission analysis studies provide a mechanism for assessing the mission benefits (e.g., initial mass in low Earth orbit, IMLEO, trip time, net payload mass, etc.) of a given advanced propulsion concept. These analyses also allow us to assess what level of technology performance (e.g., thruster specific impulse, efficiency, lifetime, specific mass, etc.) is required to achieve a required degree of mission performance. Finally, these analyses can help focus our technology investment strategy by identifying the parameters that have the greatest impact on overall mission performance. For example, there may be situations where the demonstrated specific impulse (I_{sp}) of an electric thruster is adequate, because increases in I_{sp} (for a given power level) result in a significant increase in trip time with little reduction in IMLEO. By contrast, the mission analyses might show that the thruster efficiency has a significant impact on trip time, thus suggesting technology investments in improving efficiency.

For example, Figures 13-16 illustrate the results of a study of a nuclear electric propulsion (NEP) vehicle with advanced ion thrusters used to rendezvous with a Kuiper Belt Object (KBO) at 40 A.U. There, chemical landers are deployed to obtain surface and cores samples for analysis. This mission is of interest as a potential interstellar precursor because KBO surface samples could contain particles captured from the interstellar medium. (By contrast, core samples would contain undisturbed material condensed out of the primordial solar nebula.)

A low-thrust trajectory computer code is used to determine the optimum I_{sp}, the spacecraft initial (wet) mass divided by the total thruster "jet" or exhaust power (P_{jet}), the final (dry) mass divided by P_{jet}, and propellant mass divided by P_{jet}, all as a function of trip time for an injection of C_3=0 from a Delta IV (Heavy) launch. Given the spacecraft's initial mass from the launch vehicle injection capability, we then find P_{jet}, the spacecraft final mass and propellant mass, and the total system "bus" electric power (P_e) for a given overall propulsion system efficiency. With the final (dry) mass known, it is then possible to determine the net amount of payload that can be delivered based on the various values of propulsion system specific mass (kg/kW$_e$) and propellant tankage.

In general, there will be a complex interplay between I_{sp}, specific mass, and efficiency of the various system components; mission analyses allow us to numerically map out the mission performance space resulting from changes in the various system parameters. For example, specific impulses in excess of 9,000 lb$_f$-s/lb$_m$ are needed for trip times less than 13 years. Note that as trip time decreases, I_{sp} also must decrease in order to produce a higher thrust. This has the result of requiring more propellant, which ultimately reduces the net amount of useful payload, as shown in Figure 14.

Interestingly, the total bus electric power is relatively insensitive to trip time, remaining close to 100 kW$_e$ for trip times of 10-13 years. Thus, a 25-50 kW$_e$ power per thruster is needed to keep the number of thrusters to a reasonable number (i.e., 4 or 2 engines, respectively).

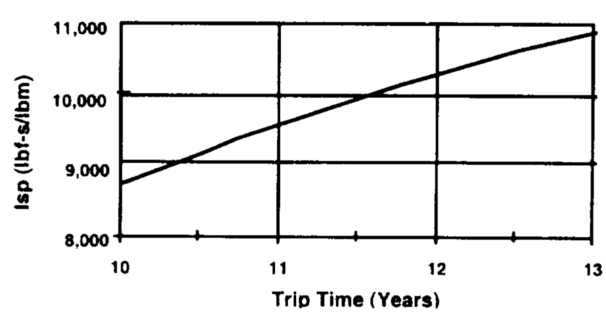

Figure 13. NEP Specific Impulse (I_{sp}) versus Trip Time for the KBO Rendezvous Mission

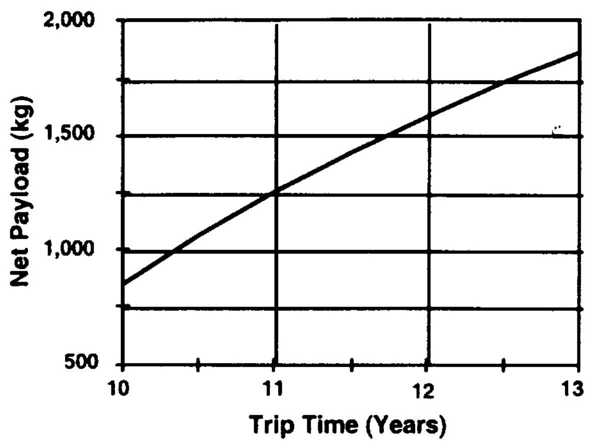

Figure 14. Net Payload Mass versus Trip Time for the KBO Rendezvous Mission

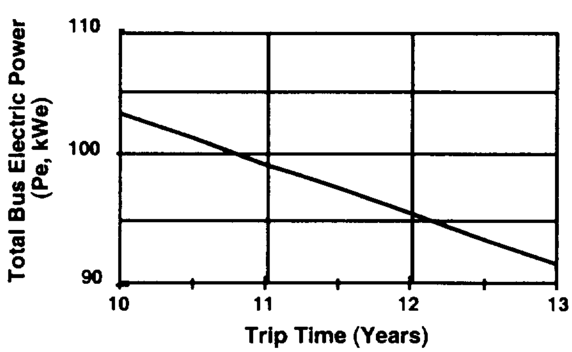

Figure 15. NEP Total Bus Electric Power (P_e) versus Trip Time for the KBO Rendezvous Mission

Finally, the cumulative thruster run-time is about 60% of the total flight time; this long run-time requirement can be satisfied by either pursuing improvements in thruster lifetime, or by adding additional thrusters (to be run in series), although the latter option has an adverse impact on the effective thruster system specific mass.

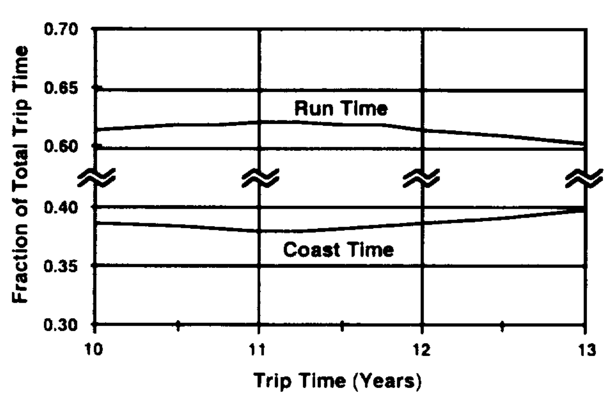

Figure 16. NEP Run Time versus Trip Time for the KBO Rendezvous Mission

Therefore, for the assumptions made in this NEP vehicle analysis, a future advanced ion engine development program aimed at meeting the technology needs of an interstellar precursor mission like a KBO rendezvous should seek to increase thruster I_{sp}s to the 9,000-11,000 lb_f-s/lb_m range, increase the power-per-thruster to 25-50 kWe, and increase thruster lifetimes to as much as 8 years per engine.

CONCLUSIONS

Advanced electric and plasma propulsion research activities are currently underway at NASA. These efforts are aimed at addressing the feasibility issues of a wide range of propulsion concepts (including micro-electric thrusters, electrodynamic tethers, high power plasma thrusters, variable specific impulse plasma thrusters, and pulsed inductive thrusters) through a combination of experiments and modeling/analyses. Such research will result in the development of technologies that will enable exciting new missions within our solar system and beyond.

For example, as primary propulsion, microthrusters may be enabling for fleets of microspacecraft designed for three-dimensional mapping of magnetic fields and particle distributions, or for performing global survey's of planets. Such fleets would be appropriate for "high risk" missions like a Saturn ring explorer, where loss of a single spacecraft does not impact the overall science return. For fine pointing and attitude control, micropropulsion will have to meet extremely demanding specifications to enable various interferometry missions.

Electrodynamic tethers have the potential of dramatically reducing the cost of orbit raising or lowering by eliminating the need for large amounts of propellant; this technology may be attractive for both commercial as well as NASA missions. For deep space missions, high-power electric propulsion thrusters will be necessary to conduct fast outer planet missions, human exploration missions, and interstellar precursor missions.

Thus, the present research program at NASA for investigating the feasibility of a variety of advanced electric propulsion concepts capable of meeting a wide range of mission applications (e.g., Watts to megawatts of power per thruster) emphasizes NASA's commitment to maintain U.S. preeminence in space in the 21st century.

ACKNOWLEDGMENTS

The research described in this paper was carried out by the Jet Propulsion Laboratory, California Institute of Technology, under a contract with the National Aeronautics and Space Administration.

The author wishes to express his thanks to the many people who contributed to the preparation of this paper, including John Blandino, Stephanie Leifer, Juergen Mueller, James Polk, and Joseph Wang of JPL, Colleen Marrese of the University of Michigan, John Cole and Les Johnson of NASA MSFC, John Sankovic of NASA GRC, and Franklin Chang-Diaz of NASA JSC.

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