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An Examination of the Technology in Non-Chemical Propulsion Systems

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ABSTRACT

The foundation of space exploration is in-space propulsion systems, which are becomingprogressively more crucial as businesses and governments put more low-earth orbit(LEO) satellite constellations into operation. Space sustainability may be greatly aided byincreasing a satellite's propulsion power and maneuverability.

For electrodynamic propulsion, In order to choose the best technology for systemrestoration, preliminary studies of the tether performance can be done using thecalculated results along the anticipated MXER tether route.For a given lB, a varied number of tethers is necessary depending on the ratio betweenthe lengths of the insulated and naked segments that is chosen. The use of electrodynamictethers with a circular crosssection demands either numerous tether designs or totaltether lengths that are far longer than what is currently envisaged for the MXER facilityin order to achieve the needed payload acceleration.For Space Thermal Stream Propulsion, one of the solutions to enabling solar streampropulsion to move in any direction is to use a variety of add-on reflector reflectorswhich enables spacecraft powered by solar thermal stream propulsion to go in anydirection, including away from the sun.This review paper analyses three unknown forms of advanced propulsion, namely,Electrodynamic Tether Propulsion, Solar Sail Propulsion and the Pulse Plasma Thruster.

KEYWORDS: Low-EarthOrbit, Tether, Propulsion, Thermal Stream Propulsion,Electrodynamic Tether Propulsion, Solar Sail Propulsion, Pulse Plasma Thruster.

1. INTRODUCTION

Since the commencement of the space programme, people have been fascinated withlarge, powerful rockets, such as NASA's Saturn V rocket that sent Apollo to the surfaceof the moon or the Space Launch System, which will launch astronauts on Artemis backto the Moon with millions of pounds of thrust.But what if the most potent propulsion device NASA has at its disposal only generatesless than one pound of force at 200,000 mph? What if it is less expensive, more versatile,and fuel-efficient?Electric propulsion in space is this revolutionary concept. Compared to chemicapropulsion systems, it can lower the quantity of fuel, or propellant, needed by up to 90%,saving millions in launch costs and enhancing mission flexibility.Chemical propulsion creates a brief, intense thrust, or what we see as fire, by combining afuel and an oxidizer to transform energy held inthe chemical bonds of the propellants.Although it is loud and entertaining, it is not very effective.

A lot of the requirements and restrictions associated with keeping propellants aboard areremoved by an electric propulsion system, which uses electricity

produced by either solarpanels (solar electric propulsion) or a nuclear reactor (nuclear electric propulsion) tocreate thrust. Ionising, or positively charging, inert gas propellants like Xenon andKrypton (no, it's not from Superman's home planet) is then done using that power after ithas been converted. The ions are then accelerated and forced out of the thruster by anelectrostatic (gridded ion) field or a mix of electric and magnetic fields (Hall effectthruster), which causes the spacecraft to accelerate to extremely high speeds over time.And instead of burning, its exhaust leaves a dazzling, bluish-green trail that resemblessomething from science fiction.

In-space propulsion systems are the cornerstone of space exploration and are becomingincreasingly important as corporations and governments launch more constellations oflow-earth orbit (LEO) satellites. Increasing a satellite's propulsion and maneuverabilitymay substantially help space sustainability. Better smallsats will eventually be able toattain their intended orbit, avoid collisions with space debris, and safely deorbit after theyare no longer in use thanks to the development and testing of new technologies likeelectric propulsion systems (EPS). Without affecting their capacity to carry out activitieslike maintaining camera imagery, communications, and other mission-critical projects,EPS and other innovative technologies may be deployed in smaller spacecraft.

2. ELECTRODYNAMIC PLASMA THRUSTER

Such thrusters are known for their durability. A variety of system configurations may beused to produce electrodynamic propulsion based on the interaction of a conductingtether with the ambient magnetic field. For various uses, bare tethers, bare and insulatedtethers with a balloon termination, and insulated tethers with a grid-sphere terminationhave all been proposed. Momentum-Exchange/Electrodynamic Reboost (MXER) tetherfacility's proposed electrodynamic tether thruster has the ability to offer a totally reusablein-space propulsion infrastructure and significantly lower propulsion costs for severalspace missions. A revolving, about 100 km long tether with parts positioned all along itslength will make up the MXER installation. For the required payload acceleration, theusage of electrodynamic tethers with a circular cross section necessitates either a multipletether design or a total tether length that is far longer than what is currently anticipatedfor the MXER facility. In contrast, a tape cross section may do the necessary reboost with the projected tether length in less time than necessary. The effect of magnetic shieldinghas been shown to be minimal for the projected equatorial orbit, but it should be takeninto account in studies for orbits with high inclination.

The tether system must be able to quickly recover its orbital energy and momentum after each payload transfer operation if it is to boost several payloads. Anode contactors are necessary for positively biased tether systems in order for them to be able to gather electrons from ionospheric plasma. The bare wire tether technology is also being taken into consideration for the MXER tether since active plasma contactors need the usage of propellant and may require significantly more mass. However, due to the creation of a magnetic field surrounding the wire, there have been concerns raised regarding the bare tether self-shielding at high current levels. The functioning of tethers that flow huge currents, such those necessary for the MXER idea, may be impacted by such a field. Based on a review of tether system performance for the mission parameters, the decision of tether design for a particular mission is made. If the current distribution along the tether for the satellite trajectory is known, it is possible to compute several metrics defining tether performance, such as system acceleration and efficiency. The efficiency estimates for the anticipated trajectory of the MXER tether employing circular and tape cross sections and taking into account potential magnetic shielding effects based on Khazanov's findings are shown below.

The MXER tether is anticipated to fly in an equatorial elliptical orbit with perigee between 300 and 500 km and apogee between 5000 and 10,000 km. The selected orbit would depend on the tether's tip velocity, which in turn depends on the required orbital transfer and the tensile strength limits of the material. We'll look at a trajectory with an apogee of 8500 km and a perigee of 300 km. The induced electric field in the 300–900 km altitude range for such a trajectory is shown. It has been assumed that there is a 90° angle between the satellite's velocity and the magnetic field of the planet. The final column displays the flight times beginning at perigee. The facility spins at a period of around 400 s and has an

orbital period of 3.06 hours. The availability of source electrical potentials is one of the constraints on tether performance. 5 kV seems to be a suitable maximum potential. For the insulated tether section, the potential drop along the tether may be computed using equations with constant current. The potential drop is greatest (along with the current) at 300 km altitude in the ionosphere and for ϑm = 0°.

The radius of the circular tether, as well as the maximum collected current, the thrust energy, and the amount of electrical energy used in one orbital period, are all constrained by the available source voltage, the ratio between the lengths of the bare and insulated segments, and a particular lB. This selection solely affects the cross section to perimeter ratio of the tape geometry; the tape width and thickness are still flexible. Notably, the minimum tape cross-section area, which is the cross section of a square, is defined by the recommended cross section to perimeter ratio. The current and tether radius for a circular tether are presented as functions of lB and. The permissible tether radius is constrained by the requirement that it should be less than the plasma Debye length in the densest ionosphere layer (0.2 cm), which is the condition of validity of the OML technique. The maximum considered current is assumed to be 50A. Additional requirements for the OML theory validity on the tape's thickness, h, and width, d, are considered to be h + d 4D 1 cm. Note that the source power need is determined by the choice of current since the source voltage is specified. Under the aforementioned parameters, the available thrust work per facility revolution, or Kt/N, has been determined. According to Hoyt [2000], the payload capacity is assumed to be 2500 kg, with 54 GJ of energy needed to restore the facility's orbit following the launch of the payload. To regain the position of the MXER facility within 100 days after launch (68.9 MJ), the thrust work, Kt/N, normalized to the work per facility revolution, is provided. The available lB and, on the other hand, rely on the selected current and the tether cross-section geometry, whereas the mandated lB and are dependent solely on the tether volume SL in this study. The amount of mass needed for tethers with circular and rectangular cross sections is the same because Kt/N for the selected lB and depends solely on the tether volume to get the required work.

3. SOLAR THERMAL STREAM PROPULSION

Since the commencement of the space programme, people have been fascinated with large, powerful rockets, such as NASA's Saturn V rocket that sent Apollo to the surface of the moon or the Space Launch System, which will launch astronauts on Artemis back to the Moon with millions of pounds of thrust. But what if the most potent propulsion device NASA has at its disposal only generates less than one pound of force at 200,000 mph? What if it is less expensive, more versatile, and fuel-efficient?

Electric propulsion in space is this revolutionary concept. Compared to chemical propulsion systems, it can lower the quantity of fuel, or propellant, needed by up to 90%, saving millions in launch costs and enhancing mission flexibility. Low-thrust approaches necessitate protracted wait times and cautious maneuvering, which is a serious drawback for missions involving tiny spacecraft. A better answer is required. For tiny spacecraft and CubeSats, we provide solar thermal propulsion assisted by carbon nanoparticles in this research.

Many of the proposed thermal propulsion methods call upon the use of a nuclear reactor to convert water into steam for propulsion. In our strategy, we want to maximize solar to thermal energy conversion and heat water by using solar concentrators that make use of carbon nanoparticles. It is possible to obtain 80% conversion efficiency using regular carbon nanoparticles. When Vanta-blackTM is used, these efficiencies increase to 99%. It would make use of a deployable parabolic solar concentrator. Concentrated solar energy is transformed into heat, which is then utilized to turn liquid water into steam at a high temperature. With the right concentrators and heat transmission systems, temperatures between 1000 oK and 3,000 oK may be reached. As a result, the Isp ranges from 190 to 320 s.

Four crucial aspects of our solar thermal steam propulsion technology stand out. First off, the propellant minimizes launch dangers from unintentional combustion and is safe, inexpensive, and simple to store. Second, the propellant may theoretically be produced on asteroids or comets utilizing in-situ resource utilization (ISRU). As opposed to electrolysis, there is no need to distill or purify the water. Thirdly, the method uses

energy from the Sun with significantly better conversion efficiency than electric propulsion does. The suggested system is not just capable of using photovoltaics, which only have conversion efficiency of between 29 and 35%. Fourth, the suggested system has a higher thrust capability than ion- or electrospray-based electric propulsion systems, which is essential for entering capture orbit or performing escape maneuvers. The suggested technology is elegant in that it approaches the performance of bi-propellant propulsion while avoiding the difficulties associated with propellant storage and safety. This suggested strategy's system-wide simplicity is what makes it tractable for ISRU. Other better performance approaches, such as water electrolysis, might eventually be bootstrapped to this technology if water collecting and processing on tiny bodies continue to be improved.

Using a parabolic solar thermal concentrator, solar energy is focused onto a surface covered in carbon nanoparticles. The sunlight is converted into heat by the carbon nanoparticles, which have a high light absorptivity of 80% to 99%. The water is subsequently exposed to the heat. With the help of a bell nozzle that has been particularly created, the water is heated into steam and fired outward into a jet. Moving reflecting mirrors will be added to the parabolic concentrator's rim. As the spaceship moves away from the sun, the reflecting mirrors will be employed to reflect sunlight into the concentrator. The spaceship has moveable reflecting mirrors that allow it to move in any direction. We contemplate a voyage from Low Earth Orbit (LEO) to the Martian moon Phobos and return for our spacecraft design. Phobos has an escape velocity of 12 m/s, and on the return journey, a delta-v of 3.5 km/s would be adequate to reach Earth LEO. The regolith from Phobos would be gathered into a storage tank and heated to release water before being used to extract water. The inner solar system can use the suggested solar thermal propulsion.

Beyond Mars, the approach is impossible due to the drop-off in solar insolation. The solar irradiation for a voyage to Phobos is 1365 W/m2 on Earth and 590 W/m2 close to Mars. The recommended parabolic dish size and the accuracy needed for the focus point both increase due to the decrease in insolation. By ensuring that the heat is directed directly onto the working fluid rather than a heat exchanger, direct solar collecting minimizes heat losses. Heat exchangers may make system design more difficult while also adding bulk to the whole spaceship. For high concentration ratios, a parabolic dish (point heating) rather than a parabolic trough (linear heating) is preferable.

4. PULSE PLASMA THRUSTER

In order to push a solid insulator (such as Teflon) over an exposed surface, pulsed plasma thrusters (PPTs) generate a pulsed, high-current discharge. The propellant material is abated (sublimated/vaporized) from its surface by the arc discharge, which ionizes and accelerates it to high speeds. Typically, a capacitor that is charged and discharged around once per second drives a short-duration current pulse.

In order to increase the electrical conductivity of the acceleration chamber, the spark plug is concurrently ignited (by a short discharge) by the spring that feeds the propellant, which is typically solid, between the two electrodes (anode and cathode). Now that a current loop has been completed, a magnetic field is being produced at the same time that electricity from the Power Processing Unit (PPU) travels to the electrodes through the capacitor and into the arc. The propellant is destroyed by the electric arc that forms, and ionized plasma is created. The Lorentz Force produced by the electric arc and the induced magnetic field then accelerates the plasma.

The benefits of a PPT include its capacity to deliver small impulse bits for precise maneuvering, robustness due to the ability to programme impulse bits to meet mission requirements, design simplicity because it can use a wide range of propellants (solid/liquid), and its capacity to maintain constant specific impulse and efficiency over a wide range of input power levels. These benefits, however, come at the expense of problems with electrode degradation, the presence of macroparticles in the plume as a result of uneven ablation, and extremely poor thruster performance. According to the rule of conservation of momentum, the thrust generated is computed. The link between the force (thrust) produced by charged particles traveling through a self-induced magnetic field is described by the Lorentz force, which applies to all electric thrusters. The amount of thrust generated also relies on the ion charge (q), the total of all impact forces per particle (propellant) over all particles ((Pi)k), and the velocity of the particles (ui). The thrust

produced, the mass flow rate of propellant (m), and efficiency () all affect the ionized propellant's effective exit velocity. The effective exit velocity may also be estimated using the anode and cathode radii (Ra, Rc).

According to analyses, 45–11% of the ablated mass per pulse in a PPT is ejected over a longer time period than the 0.5 s main discharge and at insignificant rates compared to the effective plume exhaust velocity (1500 m/s). PPTs have a very poor efficiency (10–20%) because of delayed ablation and particle emissions. During a pulse, particles are released that may interact with the surrounding plasma and are sometimes referred to as Macro-Particles (MP). Additionally, particulate emissions use up about 40% of the total mass of propellants while producing only 1% of the overall thrust. The specific impulse and efficiency of the thruster can be significantly increased if late ablation is reduced or completely avoided. Teflon is a common inert and non-toxic solid propellant used in PPTs, thus it is crucial to understand the phenomena that arise when it is utilized. Charring of electrodes is one example of this phenomenon. In an experiment, it was discovered that the electrodes' preferred ablation causes current density and Teflon surface temperature to peak there. Due to the carbon flux that was returned from the plasma, microscopic examination of the charred area revealed the presence of carbon. Under the carbon char, metal layers originating from electrode erosion have occasionally been discovered.

The difficulties with propellant conversion efficiency impeding the breakdown of propellant gas make it difficult to develop an effective pulsed plasma accelerator. The difficulty in transmitting all energy input to the ionized propellant (gas) and electrode degradation provide further difficulties. Early PPT technology was updated to address efficiency and electrode wear problems, leading to the development of the Planar Pulsed Inductive Thruster (PIT). PIT is an electrode-free system that can use a larger range of propellants, such as CO2, ammonia, hydrazine, and water. PITs have also shown that they can function with efficiency that is mostly constant throughout a broad range of Isp. They could be able to convert large amounts of electricity into reasonably high thrust with just one thruster.

Fully ionized plasma jets with a high velocity are produced by vacuum arc discharges that vaporize and consume cathode material (solid propellant) in vacuum. Since the 1960s, Vacuum Arc Thrusters (VATs), a type of such device, have been researched for use in propulsion. Electrostatic accelerators like ion or Hall thrusters may also employ vacuum arcs as plasma sources. Due to the lack of a gas feed system, VATs have the potential to be lighter and more efficient than conventional thrusters. They also allow for discrete pulse operation without sacrificing plasma production efficiency, which enables the fine-tuning of spacecraft maneuvers.

Second, because plasma is generated from the cathode and transported out of the thruster channel by the plasma pressure gradient alone, the directional efficiency of the thrust generated is highly dependent on the geometry of the thruster electrodes. As a result, VATs also have two significant drawbacks. First, the force generated per pulse is non-adjustable for each specified cathode material, and the thrust level can be adjusted only by varying the pulse duty cycle. None of the PPTs on the list have a history with CubeSat, although two of them have flown on larger satellites for secondary propulsion, and one of the VATs has flown on a CubeSat mission.

5. ANALYSIS

An electrodynamic tether's rotation raises the mass to total impulse ratio, 1/t, while having no effect on the electrical power efficiency. Depending on the ratio between the lengths of the insulated and naked segments that is chosen, a different number of tethers are needed for a particular lB. For example, two tethers with = 3.2 or four tethers with = 4.4 can be chosen if lB = 1.0. For those two circumstances, a total tether length of around 30 km for = 3.2 and 80 km for = 4.4 is required, respectively. Based on the total number of tethers, the mass varies just slightly (0.62 MT and 0.60 MT, respectively).

The use of electrodynamic tethers with a circular cross section demands either numerous tether designs or total tether lengths that are far longer than what is currently envisaged for the MXER facility in order to achieve the needed payload acceleration. In contrast, a tape cross section may complete the required reboost in less time than necessary using the estimated tether length. For the predicted equatorial orbit, magnetic shielding has been found to have a negligible impact; however, studies for orbits with high inclination should take this into consideration.

For Space Thermal Stream Propulsion, The concentration ratio, or the ratio of the aperture (parabolic dish) area to receiver area, determines the heat flux incident on the receiver. The propellant with the metal covered in carbon nanotubes will be heated directly by the parabolic concentrator. The highest temperature reached increases with increasing concentration ratio. greater Isp results from greater temperatures. These findings imply that Isp and thrust are trade offs in design. Low thrust allows for the achievement of high Isp while maximizing the use of the incoming solar energy. The system's capacity to adjust the Isp and thrust performance brings up some intriguing options. The system may be run at high Isp during transit, high thrust during capture orbits, and high thrust during gravity escape maneuvers. The data show that the Isp might vary between 190 and 320 s. We contrast the feasible delta-v ratio with the dry mass ratio. For a dry mass ratio of 0.3, a spacecraft may reach a delta-v of 4 km/s at an Isp of 320 s. This comes close to our electrolysis propulsion system's delta-v. Additionally, according to a specific design strategy, solar thermal propulsion system spacecraft may move in any direction, including away from the sun. A variety of additional reflector mirrors are used to achieve this. There has only been a little amount of testing of solar thermal concentrators in space up to this point.

For Pulse Plasma Thruster, VATs have two significant drawbacks: first, the force generated per pulse is non-adjustable for each specified cathode material, and the thrust level can only be adjusted by changing the pulse duty cycle; and second, because the plasma is generated from the cathode and is only transported out of the thruster channel by the plasma pressure gradient, the directional efficiency of the thrust generated is highly reliant on the geometry of the thruster electrodes.

6. CONCLUSION

For Space Thermal Stream Propulsion, It is suggested converting water into high-temperature steam using metals coated with carbon nanoparticles. A tiny spaceship would use the steam to produce propulsion. The use of carbon nanoparticles significantly improves the efficiency of turning solar energy into thermal energy, even though solar thermal propulsion has been proposed earlier. It may be feasible to surpass the efficiency of photovoltaic systems, which only convert 29–35% of incoming solar energy into electrical energy for electric propulsion, by using promising nanoparticles like VantablackTM, which absorbs 99.9% of sunlight and converts it into heat. This will make it possible for tiny, lightweight spacecraft to use solar-thermal propulsion. Fully ionized plasma jets with a high velocity are produced by vacuum arc discharges that vaporize and consume cathode material (solid propellant) in vacuum. Since the 1960s, Vacuum Arc Thrusters (VATs), a type of such device, have been researched for use in propulsion. Electrostatic accelerators like ion or Hall thrusters may also employ vacuum arcs as plasma sources. Due to the lack of a gas feed system, VATs have the potential to be lighter and more efficient than conventional thrusters. They also allow for discrete pulse operation without sacrificing plasma production efficiency, which enables the fine-tuning of spacecraft maneuvers.

Conflict of interest statement

Authors declare that they do not have any conflict of interest.

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