TORQUE CONTROL OF HALL PROPELLED SMALL SPACECRAFT

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Abstract

The attitude control problem of small spacecraft propelled by a single Hall thruster is addressed in relation with a new research satellite concept being investigated at the Asher Space Research Institute. During the mission transfer phase, thrust generated torque perturbations are dominant. The control approach is to use the thruster to discharge the momentum accumulated by the reaction wheels due to its own operation and due to external sources. In particular, the momentum in the thrust direction is discharged by flipping the direction of the “swirl” torque, generated as a result of the interaction of the accelerated ions with the thruster magnetic field. The ability to flip the “swirl” torque direction during thruster operation was demonstrated experimentally with a Soreq built Hall thruster.

1. Introduction

As is well known, the main advantage of using electric propulsion in space missions is the characteristic large specific impulse, which enables significant savings in propellant mass. This advantage becomes most beneficial when large Δv missions are considered. In the case of small spacecraft, the use of electric propulsion is practical provided that the dry mass of the propulsion system can be minimized. In this regard, the Hall thruster represents a combination of properties which makes it attractive relative to other types of electric propulsion for use onboard small spacecraft: high specific impulse, as compared to electrothermal thrusters, and compactness, as compared to the ion engine.

In the last few years, a Hall thruster technology development program has been conducted at the Propulsion Physics laboratory in Soreq. This effort included an experimental parametric study of various magnetic and geometric configurations of the Hall thruster in a broad range of operating conditions [1-7]. Emphasis was put on performance improvements in the sub-kilowatt power range where efficiencies of 40-55% and specific impulses of up to 2000s were obtained. The experience gained serves as a basis for the development of flight model thrusters. Meanwhile, Soreq is involved with the Technion, Rafael and other parties in various aspects of the implementation of Hall based propulsion systems.

Research activities at the Asher Space Institute in the Technion focus on the design and development of small research satellites. So far, these efforts have materialized in the design and construction of a 48 kg, three-axis stabilized micro-satellite, Gurvin TechSat II [8], carrying a miniature visual CCD camera for earth imaging, and six more experiments, including Ozone monitoring, high Tc superconductivity and more. Gurvin TechSat II was launched on July 1998 by a Zenith launcher as a piggyback on the Russian satellite “Resurs” to a 650-800 km orbit and have operated successfully since then.

At present, Asher space institute researchers are involved in the study and design of electrically propelled small spacecraft for both near earth and deep space missions. While the later include longer term Asteroid belt and Mercury visits, the near earth study considers a very low perigee, high eccentricity, eclipse free orbit for regional remote sensing. The low perigee helps in minimizing the satellite mass and the high eccentricity reduces the drag problem. To avoid a perigee drift in latitude, an inclination of 116.5°, the so-called critical inclination, is required also. The satellite is intended to be launched as a piggyback on ARIANE to a GTO orbit. The large Δv transfer from this orbit to the final one necessitates the use of electric propulsion.

The integration of an electric propulsion system in a very small satellite poses many new challenges. The tight mass, space and power constraints limit the number of thrusters. Moreover, the trend of decrease in the Isp and efficiency as the operating power of a single thruster is reduced in the sub-kilowatt range [1-3,5,6,], favors the use of a single thruster operating
at the maximum available power (and an additional thruster for redundancy). One of the most important issues that have to be addressed is the attitude control problem. As the mass of the spacecraft is reduced, its moment of inertia scales as \( l^3 \) (\( l \) - spacecraft characteristic length), while the main external source of attitude errors, the drag generated torque, scales only as \( l \), and thus its effect becomes more severe for small satellites. Moreover, during the extended thruster operating time in the transfer phase, the thrust generated torque becomes dominant. This torque can be divided into the torque perpendicular to the thrust, which is mainly due to thrust vector errors and mass center uncertainties, and the “swirl” torque [9] along the thrust vector which results from the interaction of the accelerated ions and the thruster magnetic field.

In contrast to impulsive \( \Delta V \) space flight, electrically propelled flight is thus characterized by a tight coupling between the orbital and attitude dynamics. Therefore, it is just reasonable to build a unified attitude and orbital control system with the Hall thruster(s) creating simultaneously the required thrust and torque vectors. The approach chosen is as follows. The attitude control is accomplished by a set of reaction wheels. The two components of the accumulated momentum perpendicular to the thrust vector are unloaded by slightly canting the thruster, while the momentum in the thrust direction is discharged by changing repeatedly the direction of the current in the thruster coils. The ability to perform repeatedly such a change without turning off the thruster, as may be required during the transfer phase, was successfully demonstrated using a Soreq built Hall thruster.

After the low perigee mission profile is described in section 2, section 3 addresses the attitude control problem. Rough estimates for the different sources of torque perturbations on the small spacecraft are given, and the approach to manage the thrust generated torque is described. Then, section 4 concentrates on the “swirl” torque and the ability to change its direction without turning off the thruster. Experimental results, demonstrating the feasibility of operating in this mode, are described.

### 2. A very low perigee satellite

A leading idea of the mission design was to build an operational orbit with a perigee \( \leq 200 \text{ km} \). 2-3 times below the heights of typical remote sensing satellites. For a given resolution, the shorter observation distances allow dramatic reductions in size, mass, and cost of the remote sensing instrumentation and hence of the spacecraft on the whole. Due to the aerodynamic drag, a satellite orbiting at such a low height has a lifetime of a few days or, at best, weeks.

A way to extend the lifetime up to a few years is to raise the apogee. Drag is known to reduce slowly the apogee while leaving the perigee almost untouched till nearly zeroing the eccentricity. Some electric propulsion thrusting is assumed to maintain the apogee at the required height.

Drag is only one of the factors perturbing the satellite orbit. They are dominated by the oblateness of the Earth gravity field. It leaves the size and the shape of the orbit untouched, but brings it to a complex and fairly fast rotation with typical angular rates of a few degrees per day. The perigee participating in this rotation drifts rapidly in latitude and local time of perigee passage. If a specific region is the subject of remote sensing, this drift is unacceptable. One can eliminate it by a careful choice of the equatorial inclination and the apogee height. The chosen elements of the operational orbit are shown in the table below. This particular orbit is especially favorable: a satellite placed in it never gets to the shadow behind the Earth.

<table>
<thead>
<tr>
<th>Perigee height</th>
<th>( \leq 200 \text{ km} )</th>
</tr>
</thead>
<tbody>
<tr>
<td>Apogee height</td>
<td>8300 km</td>
</tr>
<tr>
<td>Inclination</td>
<td>116.5°</td>
</tr>
<tr>
<td>Latitude of Perigee</td>
<td>35° or 145°</td>
</tr>
<tr>
<td>Local time at ascending node</td>
<td>7(^h)±1(^h) or 17(^h)±1(^h)</td>
</tr>
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The satellite is intended to be injected into a Geosynchronous Transfer Orbit (GTO) as a piggyback by the ARIANE launcher. Orbital parameters of the GTO are:

<table>
<thead>
<tr>
<th>Perigee height</th>
<th>200 km</th>
</tr>
</thead>
<tbody>
<tr>
<td>Apogee height</td>
<td>36000 km</td>
</tr>
<tr>
<td>Inclination</td>
<td>5°</td>
</tr>
</tbody>
</table>

After separation from the launcher, the satellite activates its electric propulsion with the thrust directed perpendicular to the orbit plane to move to the operational orbit. The transfer is carried out in two phases: (1) orbital plane rotation and (2) apogee descent. Electric propulsion drives the first phase, while the second phase uses aerodynamic braking, with minor assistance from EP for maintaining the desired perigee altitude and hence the apogee descent rate. In one of the scenarios being investigated, the inclination transfer lasts about 4 months while the aerobraking phase lasts two months or more depending on the actual strategy to be implemented.

The schematic drawing of the satellite in Fig. 1 shows two unfolded solar panels and one of the thrusters. The thrust vector of each thruster is parallel to the solar panel plane and passes through the spacecraft.
center of gravity. The wet spacecraft mass is expected to be in the range of 100-150 kg including 20-30 kg of xenon propellant and about 20 kg reserved for payload. The electric power supplied by the panels is in the range of 500-800 Watts.

3. Torque perturbation estimates and the attitude control approach

In both the transfer and the operational orbits, the satellite will experience torque perturbations due to external sources: aerodynamic drag, gravity, magnetic and solar pressure. The drag and solar torques are proportional to the frontal area times the distance between the mass and pressure centers. The magnetic torque depends on the mass of ferromagnetic materials while the gravity torque is known to be a linear function of the inertia tensor. The following table gives rough estimates for the absolute values, peak and average, of the external perturbing torques that are experienced by a satellite, similar to the one under study, on a similar operational orbit.

<table>
<thead>
<tr>
<th>Torque</th>
<th>Peak value Nm</th>
<th>Average value Nm</th>
</tr>
</thead>
<tbody>
<tr>
<td>Drag</td>
<td>$10^{-3}$</td>
<td>&lt; $3 \cdot 10^{-5}$</td>
</tr>
<tr>
<td>Gravity</td>
<td>$10^{-4}$</td>
<td>&lt; $10^{-5}$</td>
</tr>
<tr>
<td>Magnetic</td>
<td>$10^{-4}$</td>
<td>&lt; $10^{-5}$</td>
</tr>
<tr>
<td>Solar</td>
<td>10^{-5}</td>
<td>&lt; 10^{-5}</td>
</tr>
</tbody>
</table>

The drag torque was estimated taking a surface area including the solar panels of about 4 m², a maximum allowed attack angle of 0.5°, and a typical distance between the surface area and the mass center of 1 m. Every orbit, the satellite is only a few minutes at a height below 200 km. At a higher height the drag is exponentially smaller and hence neglected in the estimate of the average value. The gravity, magnetic and solar torques represent worst attitude cases.

As for the thrust generated torque, we consider here the situation in which the spacecraft is propelled by a single operating Hall thruster. As mentioned above, the thrust generated torque can be divided into two kinds. The first results from the thrust direction not passing through the satellite mass center. This torque is perpendicular to the direction of the thrust and is given by $N_t = TA/T$. One source for a perturbation of this kind is thrust vector misalignment that can change during the operating lifetime of the thruster and is estimated to be of the order of 1° [9]. Taking a distance of ≈ 30 cm between the thruster and the satellite mass center, we get $ΔT ≈ 5$ mm. Another source for such torque perturbations is the uncertainty in the location of the mass center, which can be a result, for example, in the case of two symmetrically placed xenon tanks, of uneven propellant expenditure. If we take a distance of ≈ 25 cm between tank center and satellite mass center and an uneven propellant expenditure from the tanks of less than 1 kg [9], we get for a spacecraft of 150 kg, $ΔT = 1.6$ mm. Other causes of mass center uncertainties, including design accuracy, are expected also to be in the mm range. We assume then that the overall $ΔT$ will be about 1 cm. Then, for thrust levels of 30-50 mN, we get a torque perturbation of $≈ 3 \cdot 10^{-4}$ Nm.

The other kind of thrust generated torque, which is a unique property of the Hall thruster, is the “swirl” torque, resulting from the Lorentz force in the azimuthal direction around the thruster axis that the accelerated ions experience due to the thrust radial magnetic field. The azimuthal ion velocity at the thruster exit gives rise to a reactive azimuthal force and, as a result, a torque along the thrust direction. Integrating along the thruster axis to get the azimuthal ion velocity, and on the cross section and ion mass flow, one obtains for the “swirl” torque, $N_{sw} = I_t φ / 2π$, where $I_t$ is the ion current and $φ$ is the radial magnetic flux of the thruster. Some contribution to the “swirl” torque comes also from the electrons [9], which under the influence of the magnetic field rotate in the discharge channel in the same direction as the ions (Hall current). Some of the electrons transfer the momentum back to the thruster by collisions with the channel walls, however some of the electrons collide with the ions and increase their azimuthal momentum. Hence, we can say that $N_{sw} ≤ I_e φ / 2π$, where $I_e$ is the total discharge current of the thruster (ions and electrons). To calculate $φ$ we have used the results of a magnetostatic simulation of the thruster magnetic circuit, which showed a very good agreement with the measured magnetic field profiles [1]. Using typical discharge current and magnetic coil current values for operating points in the 500-800
Watts range, we obtain for the “swirl torque” values of 2-4 \times 10^{-5} \text{ Nm}.

The implementation of the mission profile requires a variety of attitude modes to point the thrust vector, the antennas etc, with a point accuracy within tens of arc minutes. More precise attitude maintenance is most probably required for the remote-sensing mode. The attitude control is intended to be accomplished by a three dimensional set of reaction wheels using the feedback from a star field optical sensor and rate gyro. When the thruster is off, the accumulated momentum will be discharged by magnetotorquers. When the thruster is on, the approach is to use it to discharge its own generated momentum. The discharge of the momentum perpendicular to the thrust vector will be done by gimbaling the thruster. A limited gimbal angle of less than 4° is sufficient. Since rough torque control is sufficient for momentum discharge, the gimbal may cope with only a finite number of discrete positions. This can be accomplished by positioning the thruster on a plate, which is supported by three independently fed solenoid actuators. Fig. 2 is a schematic representation of such an arrangement, showing two of the actuators. A current impulse tumbles the plunger of the solenoid drive from one extreme position to the other, so the plate can be placed in one of eight angular positions, seven of them different.

4. “Swirl” torque flipping during thruster operation

The mission under consideration requires thruster operation for a large fraction of the inclination transfer phase. Then, the typical operating period is expected to be much longer than the time interval between “swirl” torque flipping, which is determined by the reaction wheel capacity. If, for example, the “swirl” torque together with the drag torque in the thrust direction amount to a value of about 5 \times 10^{-5} \text{ Nm} and we take a reaction wheel of 0.4 Nmsec., then, a flip is required approximately every two hours. It seems then that the “swirl” torque flipping have to be implemented many times during planned thruster operating periods. The straightforward way to perform this maneuver is to turn off the thruster, then switch the coil current direction and turn on the thruster again. However, operating in this mode would result in a significant increase in the number of thruster on/off cycles (by hundreds or maybe thousands), which would have a strong negative effect on thruster (in particular – cathode) operating lifetime. Moreover, since a minimal delay is required after each turn off, and the turn on procedure requires also a few minutes, it may also undesirably affect the thruster operating program.

The problems associated with frequent thruster turn on and off could be avoided if the coil current direction could be changed during thruster operation. However, the following difficulty was anticipated in this case. The speed by which the coil current is changed is limited by the large inductance of the thruster coils. During that time, the magnetic field in the thruster is weak or vanishes and, as a result, the thruster impedance may drop dramatically, followed by a sharp rise in thruster discharge current. This, it was suspected, could lead to unstable operation, damage due to spot mode, and/or thruster turn off.

The feasibility of coil current direction change during thruster operation was verified experimentally with a Soreq built Hall thruster [1]. Initially, electromechanical switches (relays) were used to change the current direction. Using this scheme, the coil current direction change was tried in a few tens of attempts and was successful in most of them. Nevertheless, in about 15% of the attempts the thruster was turned off. In addition, the coil current pulse during the direction switching was observed to be accompanied by sharp spikes which can be attributed to the phenomenon of bouncing of...
mechanical contacts. To avoid these problems, a second set of experiments was conducted using electronic switching. For that purpose, a logic controlled, dual polarity, Darlington type current source was built and used as the coil’s supply. This scheme was tried also in tens of attempts for different thruster operating conditions at thruster discharge voltage and power levels of 150-300 V and 200-500 Watts respectively, and was successful in all of them. Smooth coil and discharge current pulses during direction switching were observed. Fig. 3 shows four samples of the coil current pulse at switching for different conditions: when the thruster is off, for thruster operation at a discharge voltage of 200 V and coil current change in both directions, and for thruster operation at a discharge voltage of 250 V. The thruster mass flow rate was 1.8 mg/sec. As can be seen, the shape of the pulse, and hence the characteristic time of coil current direction change, is not affected by thruster operation, nor by the operating voltage or the direction of the change.

Fig. 3: Four cases of coil current direction changes: when the thruster is off, when the thruster operates at 200 V and the coil current is changed in both directions, and when the thruster operates at 250 V. In all operating cases the mass flow rate was 1.8 mg/sec.

The inductance of the series connected thruster coils is calculated from the expression \( L = 2E_m/I^3 \), where \( I \) is the coil current and \( E_m \) is the total magnetic energy, which was obtained by integration over the magnetic field solution of the above mentioned magnetostatic simulation to give \( L \approx 27 \text{ mH} \). The total resistance of the coils together with the shunt resistance (for current measurement), all connected in series, was \( \approx 6 \Omega \). The obtained time constant, \( L/R \approx 4.5 \text{ msec} \), is in accordance with the switching time demonstrated in Fig. 3.

Fig. 4 shows together the coil and the discharge current pulses for a discharge voltage of 200 V and a mass flow rate of 1.8 mg/sec. As can be seen, the discharge current increases from the steady state value of a little less than 2 A up to a peak value of 16 A which is delayed by approximately 4 msec after the switching. Fig. 5 shows discharge current pulses during switching for three discharge voltage values: 150, 200, 250 V. As can be seen, while during Hall thruster steady state operation the discharge current is mainly dependent on the mass flow and almost independent of the discharge voltage, the current pulse during switching is affected by the discharge voltage. When the discharge voltage is decreased, so does the peak discharge current. This behavior could potentially be used to minimize the current rise during coil current direction change by decreasing the operating discharge voltage just prior to the switching. Additional current limiting schemes are also being considered, even though there were no indications for instabilities, spot mode operation or any other damaging effect as a result of such a current pulse.

Fig. 4: Coil and discharge current pulses during coil current direction switching, for a discharge voltage of 200 V and mass flow rate of 1.8 mg/sec.
5. Conclusion

The attitude control problem of Hall propelled small spacecraft was addressed. During long thruster operating periods, expected in the transfer phase of a mission, thrust generated torque perturbations are dominant. Nevertheless, thruster operation itself could be incorporated as part of the attitude control to discharge the accumulated momentum. In particular, “swirl” torque direction flipping can help in discharging the momentum in the thrust direction. Due to the extended thruster operating periods, it may be required to perform this maneuver during thruster operation. The ability to operate in this mode was demonstrated experimentally.

References


