# Ion Thruster Systems with Thrust Vector Deflection

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Ion thruster systems may be used either for primary propulsion or stationkeeping of stationary satellites. In either case the most efficient system designs require continuous thruster operation to utilize efficiently the source of electrical power. If the thrust deflection can be employed, there will be no need for additional attitude-control thrusters. For instance, three thrusters with vectorable thrust directions pointing in the north, south, and east (or west) directions can control the stationkeeping functions as well as the pitch, roll, and yaw of a synchronous satellite.

The means for controlling the direction of thrust may be either purely electrical (electrostatic) or electromechanical, whereby some part of the thruster is physically moved by an electrically generated force. The electrostatic systems have rapid response, but are typically more difficult to construct than the slower mechanical systems. While a detailed study of each mission is required to define the system requirements accurately, the criteria listed in Table 1 generally prevail.

## Design Criteria

For synchronous satellites, the stationkeeping thrust required is determined primarily by the vehicle mass, while the attitude-control torque requirements are influenced mainly by the vehicle geometry. When deflectable ion thrusters are employed to satisfy both requirements, the maximum beam deflection angle becomes a major system parameter, since this angle and the allowable control moment arm directly determine the control torque possible from a given thruster size. Therefore, thrusters with large deflection capabilities will allow greater variances in total system design and spacecraft configurations.

Although control systems can be designed using thrusters having either 1-axis or 2-axis deflection capability, the latter permits the minimum number of thrusters. Figure 1 represents a conceptual design using three thrusters, each with two axis deflection capability, thus achieving both 3-axis attitude control and station-keeping.

Many studies of ion propulsion systems for interplanetary missions[4] have shown that, in general, they should be made up of arrays of thruster modules (typically 30 cm in diameter). Some of these studies have investigated methods of using this prime propulsion thruster array to provide attitude control too. Both 3-axis attitude control and the nulling of center-of-thrust to center-of-mass off-sets can be accomplished by control system concepts which provide 2-axis bidirectional mechanical translation of the thruster array and mechanical gimbaling of selected symmetrical pairs of outboard engines. If a 2-axis beam deflection capability is developed, complete 3-axis attitude control can be achieved without gimbals or translators. In relatively large arrays, 3-axis control could, in fact, be provided by engines with 1-axis beam deflection systems. In this case, groups of two thrusters with single-axis deflection can be mounted such that their deflection axes are orthogonal and can substitute for the single 2-axis deflectable thrusters. It appears that for most practical spacecraft, maximum deflection angles of ±10° should be adequate.

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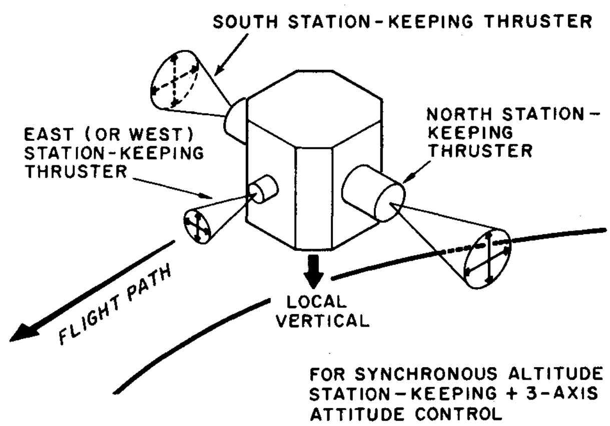


**Fig. 1 Ion engine satellite control system.**

In general, a high-frequency, open-loop response is not required, nor is it particularly desirable for high-powered solar-electric missions utilizing small main bodies with very large solar panel appendages because of the possibility of setting up mechanical resonances in the large, relatively weak structures.

## Theory, Design, and Test of Electrostatic Systems

The theory for the electrostatic deflection of electron beams and ion beams in a space-charge-free environment is well documented.[1] This simple theory breaks down in the typical case encountered in an ion thruster—namely a space-charge-limited beam which may nearly fill the accelerator aperture. This situation can only be treated using iterative techniques on an analog or digital computer. Even with such computers, it is difficult to calculate exactly the trajectories for a pencil type beam whose trajectories are deflected so as to be asymmetric with respect to the geometric axis of the system. This problem is currently under investigation, but has not yet been solved. Two general guidelines that arise from the computations are 1), the beam should be focused to as small a diameter as possible at the point where it traverses the accelerator, so that maximum transverse motion is possible before it intercepts the accelerator electrode, and 2) the thickness of the accelerator should be minimized to again prevent direct interception on the downstream edge. It is not practical to use a single large set of deflecting plates to deflect the entire beam after it has been electrically neutralized, because the deflecting fields will spatially separate the electrons and positive ions.

The feasibility of deflecting a single pencil-type beam or a strip beam have been demonstrated.[2,3] The initial beam deflection system fabricated and tested was based on the two-dimensional array of pencil-type beamlets commonly extracted from a conventional electron bombardment ion

**Table 1 System requirements**

| Application | System characteristic: Deflection angle | Deflection direction | Response time |
|---|---|---|---|
| Synchronous satellite control | As large as possible | Two-axis highly desirable | Fast ($10^{-4}$ sec) for high pointing accuracy<br>Slow (sec); adequate for most applications |
| Prime propulsion systems | 10° probably adequate | Single-axis sufficient<br>Two-axis provides more versatility | Slow (sec); adequate |

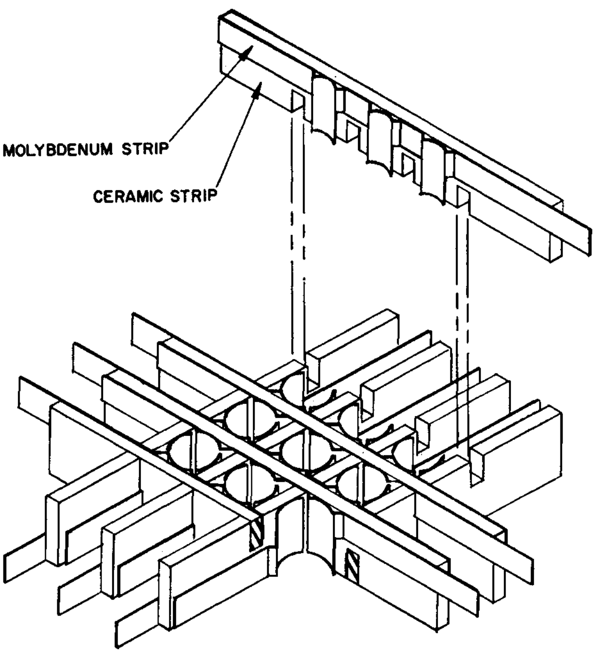


**Fig. 2 Interlocking 9-hole grid array.**

thruster. Fabrication of an accelerator electrode required for this type of system is particularly difficult, because each beam aperture must contain four deflection plates that are operated at different potentials. The deflection plates must also be electrically insulated from their nearest neighbors in adjacent apertures, but electrically connected to the plate in a corresponding position throughout the array of beam apertures. Conceptually this may be accomplished by making the accelerator from a ceramic sheet, brazing four small plates into each aperture, and making the necessary interconnections. A mechanical design concept which simplifies the fabrication procedure consists of thin, flat ceramic strips which are slotted to form an interlocking structure when assembled. The electrodes are accurately formed from a continuous molybdenum strip and brazed to each side of the ceramic, resulting in a symmetrical unit from which the final

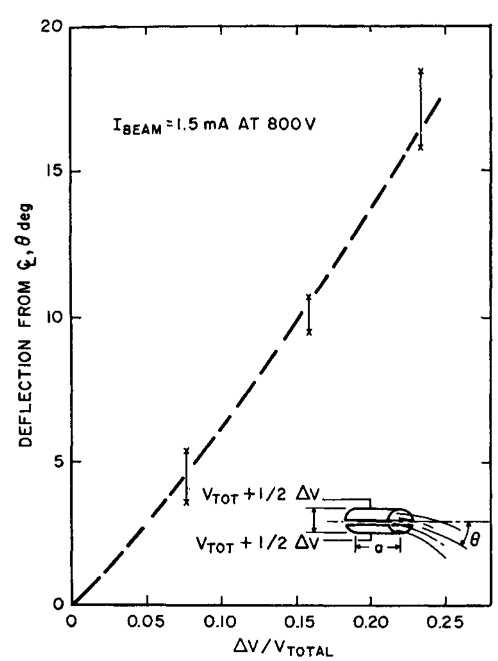


**Fig. 3 Experimental deflection of ion beam from nine-aperture array mercury bombardment thruster.**

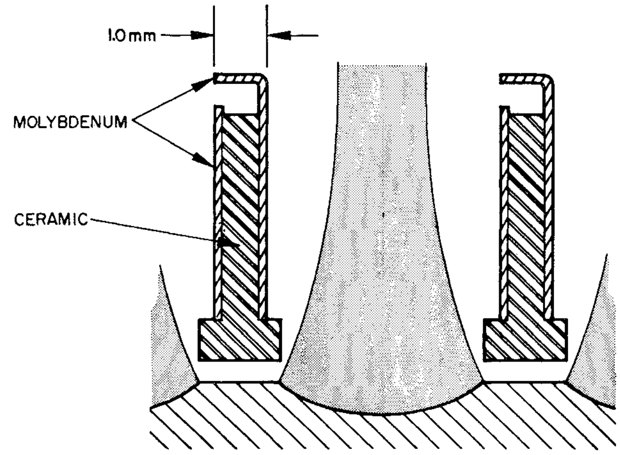


**Fig. 4 Deflecting strip cross section.**

assembly is constructed. The strips are all identical, symmetrical in form, and can be fabricated individually and inspected before integration into a complete electrode system (see Fig. 2). The electrode has high structural integrity and requires only a few electrical connections at the periphery of the array. A nine-aperture array was tested on the 5-cm thruster. This accel electrode is used in conjunction with a conventional screen electrode to form a two-grid optic system.

After operating the 5-cm thruster with conventional optics (97 holes) and extracting a beam current of ~32 ma with less than 1 ma accel drains, the 9-hole deflecting accel electrode was installed on this thruster with a screen electrode.

For this experiment pairs of deflecting elements were tied together to effectively form two half-cylindrical deflecting plates to permit the resulting beam deflection to be measured with the available probe mechanism, which moved in the horizontal plane. The thruster was tested in a 9-ft-diam × 18-ft oil-diffusion-pumped vacuum facility. Beam profiles were measured at the thruster and at a point 2.5 m downstream, where the shift in position was approximately 0.75 m. The test results of Fig. 3 illustrate a measured deflection angle of 18°.

A single-grid optic system was designed which incorporates the shadow shield feature and provides a dielectric surface at the plasma interface. The strips were fabricated as described previously, except that the ceramic strips were of a tee-shaped cross section and so formed an overlapping dielectric surface on the upstream side of the electrode. A schematic cross section of this design is shown in Fig. 4. Test results are not yet available. Recent experience at NASA-LeRC and Hughes Research Laboratories has indicated that thruster performance is improved with the single insulated grid electrode system.[5]

### Concluding Remarks

It has been demonstrated that the thrust vector from an electron bombardment ion thruster may be deflected by electrostatic techniques. Analytical studies and comparisons with experimental data by others (not presented herein) also have shown that electromechanical systems are feasible. While the electrostatic technique has the advantages of fast response, low power consumption and greater angular deflection, it remains to establish the best technique by experimentally comparing optimized systems of each design.‡ Better definition of the system requirements for a particular mission are required to define accurately the optimum system, of the type discussed in Ref. 4. It is certain that thrust de-

‡ This investigation is currently underway under Contract NAS 3-14058 recently awarded to the Hughes Aircraft Company by NASA Lewis Research Center.

flection can be accomplished and that its incorporation into the system will reduce the over-all weight and complexity of the system.

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# Approximate Analytic Modeling of a Ballistic Aerobraking Planetary Capture

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### Nomenclature

$B$ = vehicle ballistic coefficient, $B = C_DA/2M$
$R$ = flight-path radius of curvature
$r$ = distance from planet center
$s$ = distance along trajectory
$V$ = velocity of vehicle
$y$ = altitude, distance from planet surface
$\theta$ = true anomaly along trajectory
$\mu$ = gravitational constant
$\rho$ = density of atmosphere

### Introduction

SPACECRAFT approaching a planet may be captured by passing through the planet's atmosphere. Such maneuvers have been considered for Mars missions.[1,2] For the case of ballistic entry, simple analytic theories for the motion have been given.[3,4] This Note demonstrates the equivalence of these two theories, and gives a complete solution for the maximum deceleration due to drag in a form useful for preliminary mission planning.

The geometry of a ballistic aerobraking capture is given by Fig. 1. Maday[3] gives the vehicle equations of motion as

$$VdV/ds = -B\rho V^2 - (\mu/r^2)dr/ds,\ V^2/R = (\mu/R)d\theta/ds$$

and these equations may be combined by noting that $d\theta/ds = (d\theta/dr)(dr/ds)$. It is now assumed that 1) the trajectory may be modeled by means of a drag-free (Keplerian) trajectory, and 2) the vehicle is in the atmosphere for small values of $\theta$. The trajectory is then given by the expression

$$r = r_p\{1 + \tfrac{1}{2}[e/(1 + e)]\theta^2\} \quad (1)$$

where $r_p$ is radius at perigee, and $e$ is the eccentricity, given by $e = V_p{}^2r_0/\mu - 1$, and $r_0$ is the planet radius. Density varies exponentially with altitude, $\rho = \rho_p \exp[-k(r - r_p)]$.

The equation of motion may be simplified for $|\theta| \ll 1$; the velocity change may then be found by quadratures, the integrations involving error integrals of large argument, which are

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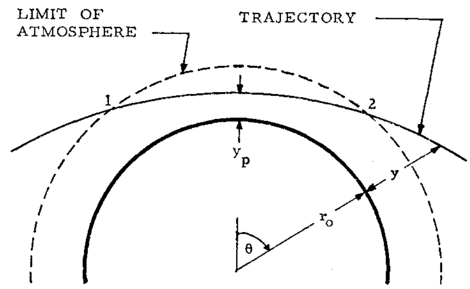


**Fig. 1 Geometry of an aerobraking capture maneuver.**

approximated by the complete error integral, with value $\pi^{1/2}/2$. In particular,[3]

$$(V_2/V_1) = \exp\{-B\rho_p[2\pi r_p(e + 1)/ke]^{1/2}\} \quad (2)$$

where $k$ is the atmosphere scale height, and, if $\rho_0$ is the density at ground level, then $\rho_p = \rho_0 e^{-ky_p}$.

In contrast, the second theory[4] explicitly neglects gravity in the equation of motion:

$$dV/dt = -\rho BV^2 \quad (3)$$

The quantity $m$, the mass of gas "encountered" by the spacecraft, or lying within the tube of space swept out by its area $A$, is introduced as a parameter coupling the velocity reduction to the trajectory. The rate of mass encounter is $dm/dt = \rho AV$, and $m$ is thus given by $m = (2M/C_D) \ln (V_1/V_2)$. Moreover, $m$ is given as a function of the orbit shape, $r = r(\theta)$:

$$m = A \int_{r(\theta)} \rho(r)ds \quad (4)$$

As in the first theory, the orbit shape is given by Eq. (1), and the function $\rho = \rho(r)$ is taken as $\rho = \rho_0 e^{-ky}$. Since $|\theta| \ll 1$, $y \ll r_0$, $s \simeq r_0 d\theta$. The quadrature in Eq. (4) thus also involves error integrals of large argument. The problem considered in Ref. (4) is the inverse of that of Maday: given $V_1$ and $V_2$, to find $y_p$. The solution to this trajectory-design problem is given:

$$y_p = -\frac{1}{k} \ln \left[\frac{1}{B\rho_0}\left(\frac{k}{2\pi r_p}\frac{e}{1 + e}\right)^{1/2} \ln \frac{V_1}{V_2}\right] \quad (5)$$

where $r_p = r_0 + y_p$; $r_p \simeq r_0$. Comparing Eqs. (5) and (2), it is seen that they are equivalent.

### Comparison of the Theories

The second method[4] explicitly neglects gravity; in this it resembles the Allen-Eggers[5] approximate theory for ballistic entry at high flight-path angles. Maday's theory explicitly takes account of gravity. However, Maday gives the influence of gravity as being zero; for the gravity term in his expression for $V_2/V_1$ involves the multiplier $(\theta_1 + \theta_2)$, and by the symmetry of Eq. (1), it is seen that this is zero, i.e. atmosphere entry and exit take place at the same value of $|\theta|$. Moreover, the second method allows the convenient study of trajectories whose shapes are given by expressions other than Eq. (1), which are to be used in Eq. (4). Orbits considered include a "double conic," with two branches each of which is given by an expression such as Eq. (1), but with different eccentricities for $\theta < 0$ and $\theta > 0$, and an expression similar to Eq. (1) but adding a term proportional to $\theta$. It is found that differences in $y_p$ from the value given by Eq. (5) are of the order of 1%.

Drag is proportional to $\rho V^2$ and we require, for the drag to be maximum, that $d(\rho V^2)/dt = 0$, or $V^2(d\rho/dt) + 2\rho V(dV/dt) = 0$. Using Eq. (3),

$$d\rho/ds - 2B\rho^2 = 0 \quad (6)$$

We have $s \simeq r_0\theta$; let $\rho(s) = \rho_p \exp[-k(y - y_p)]$. Using