Engineering Notes

Optimization of Ion Engine Control Systems for Synchronous Satellites

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Nomenclature

a_E = acceleration of earth with respect to sun and moon
a_{sat} = acceleration of satellite with respect to sun and moon
K_m^2 = moon's gravitational constant
K_s^2 = sun's gravitational constant
$\bar{r}$ = vector distance between satellite and earth
$\ddot{\bar{r}}$ = acceleration of satellite with respect to earth
$\bar{R}$ = vector distance between moon and earth
$\bar{R}_s$ = vector distance between sun and earth
Z = perpendicular distance from equatorial plane
α = right ascension of moon
δ = declination of moon
ζ = declination of sun
η = right ascension of sun
ν = right ascension of satellite

Introduction

An early possible application for ion propulsion systems is the orbit control of synchronous satellites. The design and operating mode of such a control system depends to a great extent on the requirements of the mission. In order to optimize the system, therefore, it is necessary to know in detail the magnitude and nature of the major forces that tend to perturb the satellite orbit. The major contribution to the total impulse required to maintain the orbit of a stationary satellite is the effect of the solar-lunar attraction. Initially, stationary satellite control systems (both chemical and ion) were designed to correct this solar-lunar perturbation by periodic thrusting. Although this pulsed correction mode may be efficient from the standpoint of total propellant utilized (and, therefore, most efficient for chemical engines), it adds an energy storage subsystem requirement to the ion propulsion control system. For a given size satellite, the weight of the energy storage system required to operate an ion propulsion system in a pulsed mode is determined primarily by the length of thrust interval per orbit correction. If it were possible, therefore, to fire more often, resulting in a shorter thrust period, a considerable reduction in system weight could be realized. Furthermore, if it were possible to correct continuously at sufficiently low thrust levels to run the ion engines directly off the primary power source, the energy storage system could be eliminated. For this reason, a study was performed to determine the exact nature of the solar-lunar perturbations and to determine the optimum thrust level and sequence for maintaining the orbit of a stationary satellite.

Solar-Lunar Perturbations

The force of attraction from the sun and moon can be divided into three components; two, the radial and tangential, lie in the equatorial plane, and the third lies normal to this plane. The radial and tangential components produce cyclic oscillations in satellite radius and longitude. It has been shown[1] that these oscillations have a small maximum amplitude throughout each satellite orbital period, and that at the end of that time both the change and rate of change of satellite radius and longitude will have assumed their initial zero values. Hence, no corrective thrust is required to counteract the radial or tangential force components.

The component of force normal to the equatorial plane will, on the other hand, cause an oscillation that results in an increase in satellite orbit inclination at the initial rate of almost one degree per year. The amplitude of this oscillation will grow to a maximum of about 20° in about 40 years.[2] If the satellite is to remain stationary in the equatorial plane, it is necessary to correct this perturbation. The optimum mode of correction and the magnitude of thrust required can be determined by solving the appropriate differential equations, taking corrective thrust into consideration.

The equation of motion of a satellite relative to the earth can be found by considering the attractive force equations. Referring to Fig. 1,

$$\bar{a}_E = -K_s^2 \bar{R}_s/R_s^3 - K_m^2\bar{R}/R^3 \quad (1)$$

and similarly,

$$\bar{a}_{sat} = -K_s^2(\bar{R}_s + \bar{r})/|\bar{R}_s + \bar{r}|^3 - K_m^2(\bar{R} + \bar{r})/|\bar{R} + \bar{r}|^3 \quad (2)$$

The total acceleration of the satellite with respect to the earth due to the sun, moon, and earth can be written

$$\ddot{\bar{r}} = \bar{a}_{sat} - \bar{a}_E - K^2\bar{r}/r^3 \quad (3)$$

Finally, by combining Eqs. (1–3), it can be seen that

$$\ddot{\bar{r}} + K^2\bar{r}/r^3 = K_m^2[(\bar{R} + \bar{r})/|\bar{R} + \bar{r}|^3 - \bar{R}/R^3] - K_s^2[(\bar{R}_s + \bar{r})/|\bar{R}_s + \bar{r}|^3 - \bar{R}_s/R_s^3] \quad (4)$$

Equation (4) is the general differential equation, in vector form, which describes the satellite's orbital motion in the presence of a spherical earth, the sun, and the moon. By dividing the vector equation into its components, the equation

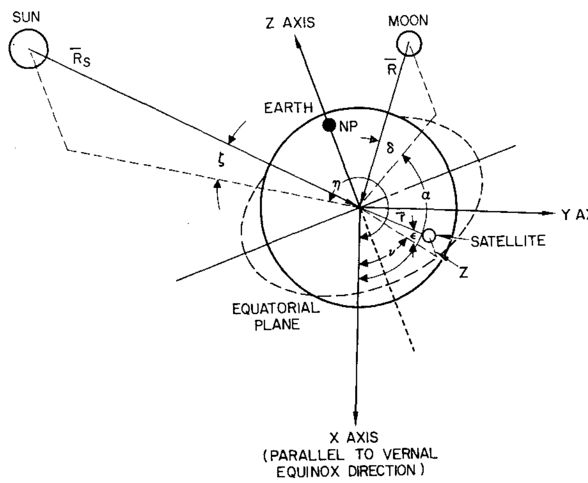

Fig. 1 Geocentric coordinate system.

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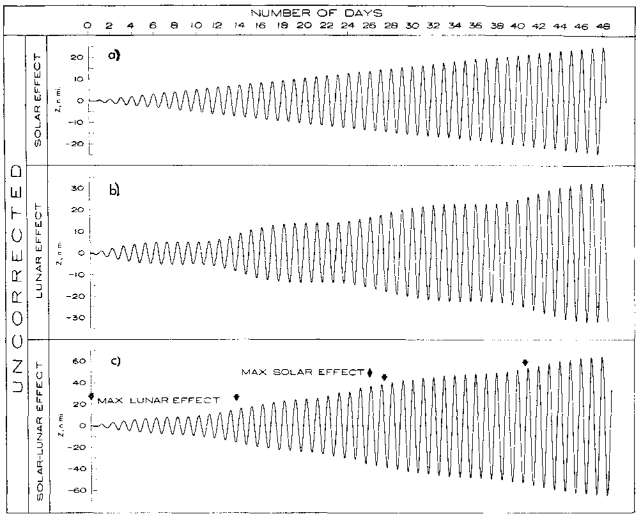


Fig. 2 Uncorrected latitudinal satellite motion.

which defines the latitudinal motion of the satellite is found to be

$$\ddot{Z} + K^2Z/r^3 = K_m{}^2r[3 \sin\delta \cos\delta \cos(\nu - \alpha)]/R^3 + K_s{}^2r[3 \sin\zeta \cos\zeta \cos(\nu - \eta)]/R_s{}^3 \quad (5)$$

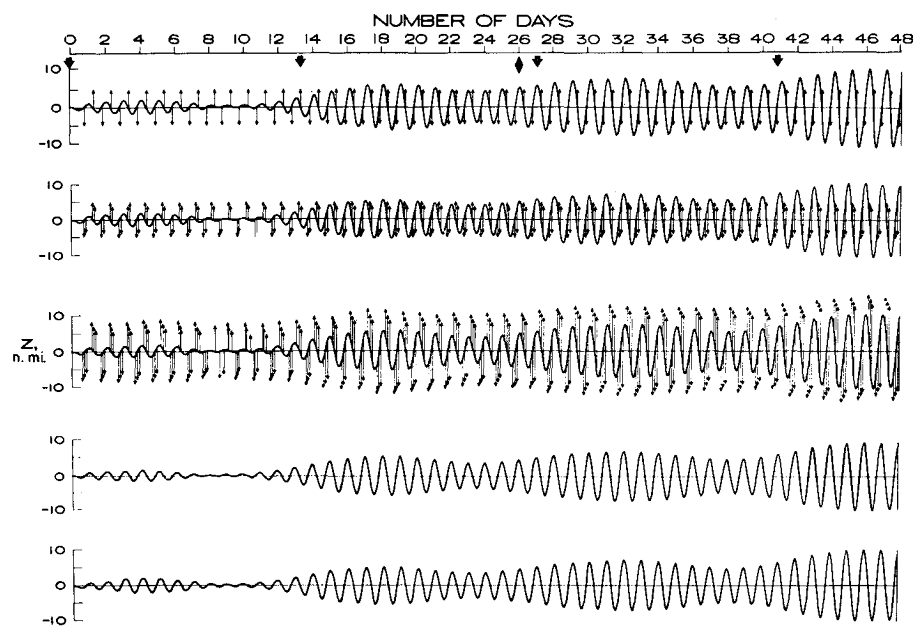
Equation (5) was programed on an analog computer; the uncorrected Z oscillation was plotted to simulate a 48-day interval during which the sun passed through a point of maximum perturbation. The maximum moon effect occurs every 13.6 days. Figure 2c illustrates the total solar-lunar effect on the orbital inclination of a satellite for the 48-day period starting November 26, 1965. The inclination at the end of this time is 0.16°. The effects of the solar and lunar perturbations are shown separately for the same period in Figs. 2a and 2b, respectively. From these curves the total change per year in orbital inclination was determined. The sun's contribution was found to be 0.30°, whereas that of the moon was 0.64° for a total of 0.94° per year. From this result the minimum amount of North-South station-keeping propulsion required daily can be calculated.

Orbit Maintenance Requirements

In order to counteract the change in inclination and maintain the satellite in the equatorial plane, velocity must be added to the satellite in a direction normal to the orbit plane. Because of the gyroscopic nature of a satellite in its orbit, the orbit will precess about an axis directed through the point of applied torque and normal to the thrust. If the satellite is to be precessed into the equatorial plane, corrective thrust can be applied most efficiently (in terms of total impulse) at the nodal points, i.e., satellite crossings of the equatorial plane. For a 24-hr satellite these crossings occur twice daily. As an example, assuming a thrust mode of two nodal firings per day, alternating in the North-South directions, depending on whether the satellite is crossing an ascending or descending node, the total impulse required each day to correct the solar-lunar perturbations (0.94° per year) on a 550-lb satellite is 7.8 lb-sec. If 1.5-mlb station-keeping engines are employed, thrust must be applied for 43.5 min per firing. (Because the required thrust time when correcting twice daily at the nodes can be determined analytically, a reference correcting mode can be established to which the effectiveness of other thrusting sequences may be compared.)

Special circuitry was added to the original computer program to simulate thrust interval and firing position. By means of this program it was possible to determine the variation in latitude, as defined by the parameter Z, of a synchronous satellite under the influence of the solar-lunar perturbations and any given thrust sequence. The study was performed for a 550-lb satellite using 1.5-mlb station-keeping engines (except in the cases where continuous thrust was employed); however, the results may be extrapolated to other size vehicles. In all cases thrust is applied in a direction to decrease the satellites velocity component normal to the equatorial plane. Figure 3a shows the effect of thrusting 43.5 min twice daily at the equatorial crossings (nodes). Since this thrust sequence has been shown to be effective, it will be used as the reference. In this case, the satellite is held to an inclination of 0.027°.

Various thrusting modes were investigated with as many as six firings per day. The length of the thrust interval was varied according to the scheme being tested. Figure 3b shows the result of thrusting each time the satellite crosses a node and exactly 3 hr afterward. It was found that each firing interval must be 25 min long to accomplish the same



a) Pulsed, 1.5 mlb, 2/day, 43.5 min each, at nodes

b) Pulsed, 1.5 mlb, 4/day, 25 min each, at nodes and 3 hr later

c) Pulsed, 1.5 mlb, 6/day, 18 min each, at nodes and 3 hr before and after

d) Continuous, 0.15 mlb

e) Continuous, less 2 hr at peak, 0.15 mlb

Fig. 3 Latitudinal control using various thrust sequences. Bold arrows at top indicate maximum lunar effect periods; diamond is maximum solar effect. Small arrows on parts a–c indicate application of thrust.

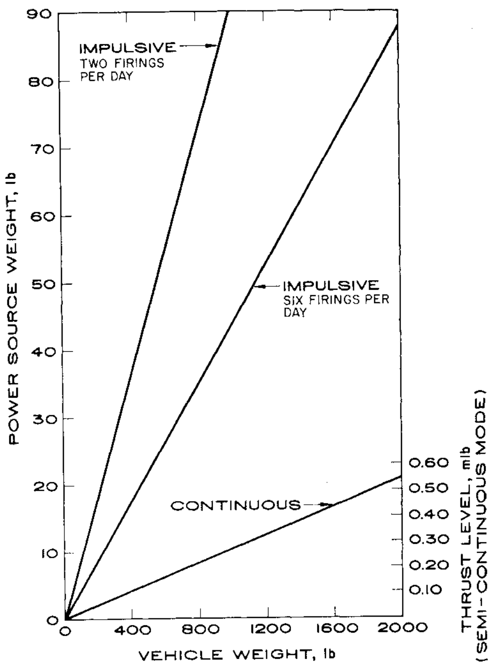


Fig. 4 Power supply weight and thrust level vs satellite weight.

result as firing twice daily at the nodal points. This represents a total daily thrust time of 104 min, corresponding to an impulse of 9.0 lb-sec.

Figure 3c illustrates the effectiveness of applying thrust 3 hr before and 3 hr after a node crossing as well as at the node, a total of six firings per day. The thrust time per firing required to control the vehicle was found to be 18 min for a total thrust time of 108 min per day and an impulse of 9.7 lb-sec.

It is apparent, therefore, that the number of firings executed each day can be increased and that the individual thrust periods will be shortened as a result. Another possible correction mode which was investigated was continuous thrusting at lower thrust levels. Figure 3d shows the result of a continuous 0.15-mlb thrust. This level is seen to be sufficient to control the satellite. However, thrust applied near and at the peaks of the oscillations is essentially useless due to the gyroscopic effect. Therefore, a correction curve again using 0.15 mlb continuous thrust, except for a 2-hr period when passing each peak, was plotted. Figure 3e illustrates the successful controlling of the satellite by this technique. The total impulse required, however, has increased to 10.8 lb-sec.

Conclusion

From the foregoing analyses, it is seen that the solar-lunar perturbations on a stationary satellite can be counteracted by thrusting alternately in the North and South (N-S) direction perpendicular to the satellite orbit. As shown, the most efficient N-S correction mode, in terms of total impulse, is thrust two times daily at the nodes. There are, however, other thrust sequences, such as multiple or continuous firing which can effectively control a stationary satellite.

The effect of thrust sequence on ion engine control system power supply weight is shown in Fig. 4. In both the pulsed and continuous correction modes, the weight of the satellite to be controlled determines the weight of the power supply (e.g., solar cells and batteries). In the latter case, the satellite weight also determines the required thrust level. Figure 4 shows, for the relatively high-thrust, impulsive-type control system the substantial weight savings ($\sim$50% for a 1000-lb satellite) afforded by correcting six rather than two times daily. The optimum control system operation, however, is seen to be the continuous (or semicontinuous) thrust mode. Since this mode of operation requires a continuous low thrust level (0.3 mlb for a 1000-lb satellite), the ion engines can be operated efficiently directly from the solar cells, thereby eliminating the need for batteries. In addition, the engine and power conditioning equipment will be somewhat reduced in weight because of the lower thrust level. For a 1000-lb vehicle, the power supply weight is reduced from 45 lb for the best pulsed mode operation to 10 lb for the continuous correction, representing an almost 100% reduction in total control system weight. As the satellite weight increases, the advantage of continuous correction becomes even more pronounced.

References

[1] Molitor, J. H. and Kaplan, M. H., "Optimization of ion engine control systems for synchronous satellites," AIAA Preprint 63-273 (June 1963).

[2] Frick, R. H., and Gordon, T. B., "Perturbations of synchronous satellites," Rand Rept. R-399-NASA, The Rand Corp., Santa Monica, Calif. (May 1962).

Fin Temperature Measurements: Nike Cajun Sounding Rocket

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Introduction

THIS report summarizes results of tests conducted on stabilizing fins developed for the Nike Cajun sounding rocket. Since these tests were completed, there have been approximately 85 flights using these fins in the NASA Sounding Rocket Program (50 Nike Cajuns and 35 Nike Apaches) with no evidence of structural failure. Payload weights (gross) have varied between 50 and 95 lb, and the rockets have been launched with effective launch angles varying between 2° and 17° from the vertical.

Tests

Structural load tests based upon anticipated flight loads and temperatures indicated that a panel and shroud fin design was capable of withstanding 3890 lb per fin at destruction at room temperature; at elevated temperature, 210°F, the fin withstood 1800 lb load per fin without reaching the yield point.[1]

A Nike Cajun two-stage solid propellant sounding rocket was flight tested to determine the temperature of the second-stage panel and shroud design fins under flight conditions. A schematic of the Nike Cajun sounding rocket (see Ref. 2) is shown in Fig. 1. The warpage problem previously associated with panel and shroud fins[3] is avoided by the use of an ex-

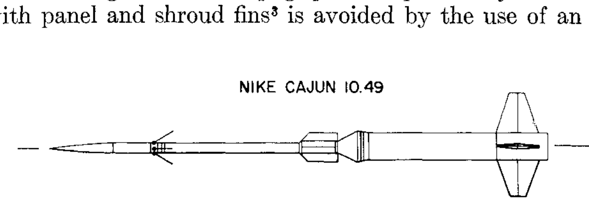

Fig. 1 Schematic of Nike Cajun sounding rocket.

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