Electrical Propulsion for Control of Stationary Satellites

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The application of electric propulsion engines to attitude control and station keeping of 24-hr stationary satellites is analyzed and compared with the performance of contemporary cold-gas, monopropellant, and bipropellant propulsion systems. Both a 500-lb spin-stabilized and a 1500-lb 3-axes-controlled satellite compatible with current NASA boost vehicles are examined, and propulsion system effects on mission duration and maneuver requirements are compared. Solar electric propulsion is shown to be superior to chemical propulsion for long term (>1 yr) station keeping and 3-axes-attitude control of the larger satellite. Cold-gas and chemical propulsion are superior for attitude control and provide strong competition for electric propulsion in the station keeping of the smaller spin-stabilized satellite.

Nomenclature

a_p = semimajor axis of perturbing body orbit, ft
a_s = radius of satellite orbit, ft
A = angle between earth polar axis and line connecting center of earth with center of curvature of orbit axis path, deg
B = angle between perturbing body orbit axis and line connecting center of earth with center of curvature of orbit axis path, deg
d = distance of cp from c.g., ft
D_c = thrust duty cycle, %
E_f = feed system warmup energy, w-hr
E_t = thrustor warmup energy, w-hr
Ft = impulse, lb-sec
g = gravitational constant, ft/sec²
h = cylinder height, ft
i_1 = inclination of satellite orbit plane with respect to equator, deg
i_2 = inclination of satellite orbit plane with respect to orbit plane of perturbing body, deg
I_0 = vehicle inertia
J_2 = first-order earth oblateness coefficient
K_1 = precession rate coefficient due to earth oblateness
K_2 = precession rate coefficient due to lunisolar gravitation
l = thrustor moment arm, ft
M = mass of perturbing body, slug
m = satellite mass
n = satellite orbit angular velocity, rad/sec
p = solar radiation pressure, lb/ft²
P_a = average power required, w
P_c = continuous power required, w
P_f = feed system operating power, w
P_t = thrustor operating power, w
r = cylinder radius, ft
R = radius of curvature of path of orbit plane vector, rad
R_0 = radius of earth, ft
S = vehicle area, ft²
T_d = disturbance torque, ft-lb
V_1 = motion of satellite orbit axis due to earth oblateness, rad/sec
V_2 = motion of satellite orbit axis due to perturbing body, rad/sec
α = half-angle of thrust pulse, deg
γ = cylinder inclination, deg
θ = maneuver angle about any principal vehicle axis, deg
θ_d = attitude angular accuracy, deg
τ = time interval between thrustings, hr; period of orbit inclination cycle
ω_0 = vehicle spin rate, rad/sec

Introduction

To maintain proper orientation of a 24-hr satellite for a number of years against external disturbance torques due to solar radiation pressure, gravity gradient, magnetic fields, and micrometeorite impact and against such internal effects as gas leakage and moving parts is a formidable problem for the spacecraft designer. Two methods for maintaining such orientation will be examined: spin stabilization of the entire spacecraft and 3-axes stabilization by means of pulsed thrustors arranged to produce torque about the principal vehicle axes.

This report considers the application of electric propulsion to attitude control and station keeping for possible stationary satellites of each type (Table 1) and compares it with the performance of contemporary cold-gas, hot-gas, and bipropellant systems (Table 2).

The spin-stabilized vehicle resembles the NASA-Hughes Advanced Syncom communication satellite. The 3-axes-controlled vehicle represents current concepts for an advanced weather satellite. The chemical and cold-gas thrustors and power systems considered are compatible with present Hughes space vehicle design practice (Tables 2 and 3)[1]; the electric engines considered are direct descendants of those developed by NASA for Solar Electric Rocket Test (SERT). The thrustors used do not represent an ultimate in engine development but rather estimates of existing engine capability in order that the status as it exists today might

Table 1 Vehicles considered

	Launched by	
	Atlas Agena	Atlas Centaur
Stabilization type	Spin	3-axes controlled
Weight, lb	500	1500
Diameter, ft	5	9
Inertia, slug-ft²	50	470
Thrustor moment arm, ft	2	4
Spin speed, rpm	100	0
Alignment accuracy, deg	0.5	0.5
Turn-around time, 90°, hr	24	$\frac{1}{4}$

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Table 2 Thrustors available

Parameters	Cold-gas jet		Resisto-jet		Chemical	Ion
Propellant	N_2	NH_3	NH_3	H_2O_2	N_2O_4-UDMH	Cesium
Thrust, lb	0.002–1.0	0.002–1.0	0.001–0.1	5	5	0.00001–0.01
Specific impulse, sec	60	100	425	160	240	4500
Tankage factor, %	200	10	10	30	20	50
Thrustor power, kw/lb	...	...	20	...	...	200
Feed system power, each, w	...	...	...	...	...	5
Control power, w	...	...	...	...	...	10
Warmup energy, w-hr	...	...	...	...	...	0.75
Warmup time, sec	...	...	...	...	...	30
Thrustor and misc. weight, each, lb	0.5	0.5	1.0	1.0	5.0	2.0

be examined. Dual redundancy is assumed for all engines and power conditioning equipment. Although this practice penalizes the electric system more severely than the chemical system, the very long mission times involved make some degree of redundancy advisable. Further discussion of the application of ion engines is given by Molitor.[2]

Attitude Control

Spin Stabilization

The use of vehicle spin to maintain orientation greatly simplifies the attitude-control system at the expense of making any maneuvering costly in terms of impulse required. Precession of a spinning satellite at the 24-hr-orbit altitude is caused primarily by solar pressure. For a cylindrical satellite with perfectly reflecting ends and with completely absorbing sides, this precession rate is given by[3]

$$d\theta/dt = \pi P r^2 h \sin 2\gamma / 8 I_0 \omega_0$$

For the vehicle considered (Table 1), this precession rate has a maximum value of 3.4°/yr. The total impulse required to precess the vehicle by means of thrustors located at the periphery (Fig. 1) is given by the equation

$$Ft = I_0 \omega_0 \alpha \theta / l \sin\alpha$$

The impulse required to precess through 3.4° is 16.3 lb-sec. A 90°-maneuver would then require a 430-lb-sec impulse. To make such a maneuver in a 24-hr period, a minimum thrust of 0.03 lb would be required.

Three-Axes Control

The use of pulsed jet thrustors acting about 3 vehicle axes allows complete freedom of orientation but requires a rather large impulse to maintain vehicle orientation against disturbance torques. At the 24-hr altitude, the normal component of solar radiation pressure produces the principal disturbance torque, which is $T_d = 2pSd$.

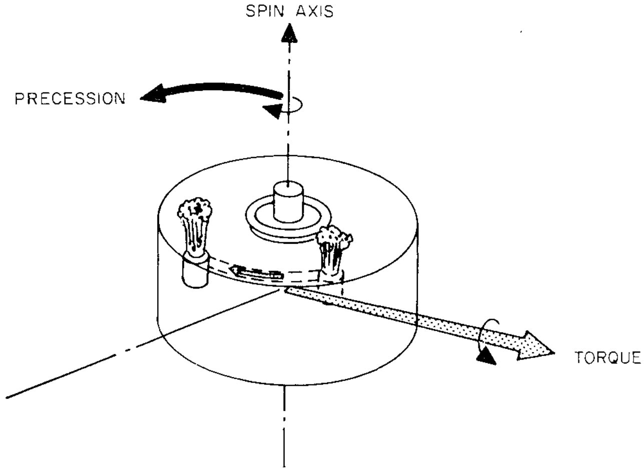

Fig. 1 Orientation control.

Table 3 Power system

Parameter	Power conditioning, ion engines	Battery, eclipse	Solar panel
Efficiency, %	80	65	10
Specific weight, lb/kw	30	125/hr	133 directed 400 nondirected
Life at 75-% depth discharge, cycles	...	300	...
Life at 50-% depth discharge, cycles	...	1200	...

With a 100-ft² area and 2.5-ft c.g. displacement, the 1500-lb vehicle can experience a torque as large as 5×10^{-5} ft-lb. Allowing for a safety factor of 2, the impulse required to maintain vehicle orientation on 1 axis is then 790 lb-sec/yr. This is the maximum impulse required to maintain vehicle orientation, provided the minimum impulse bit used is less than a critical value given by the equation

$$(Ft)_{\min} = 2(\theta_d I_0 T_d)^{1/2}/l$$

At maximum disturbance torque, this value of minimum impulse (10^{-2} lb-sec) will result in a coast time of 404 sec and in 79,000 thrustor operations per year on each vehicle axis. For smaller values of the minimum impulse bit, the total impulse required at zero disturbance torque will be re-

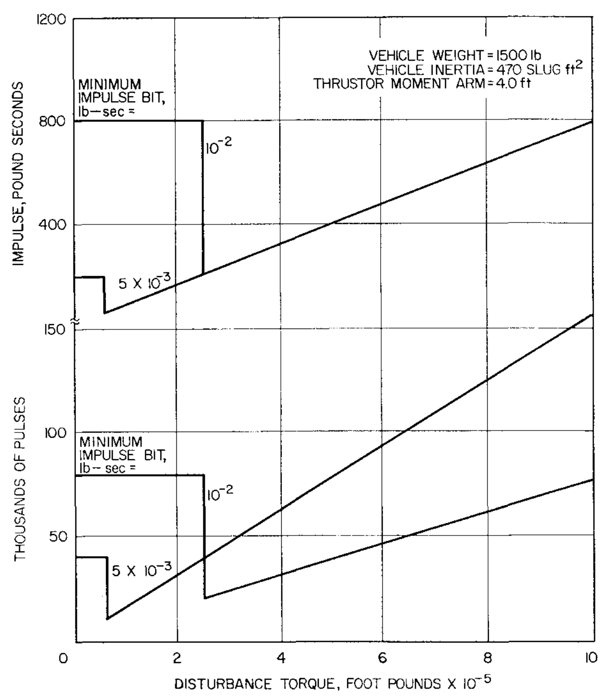

Fig. 2 Single-axis impulse and number of thrust pulses per year vs disturbance torque.

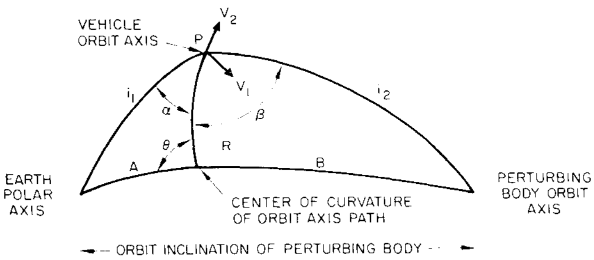

Fig. 3 Polar diagram of lunisolar perturbation geometry.

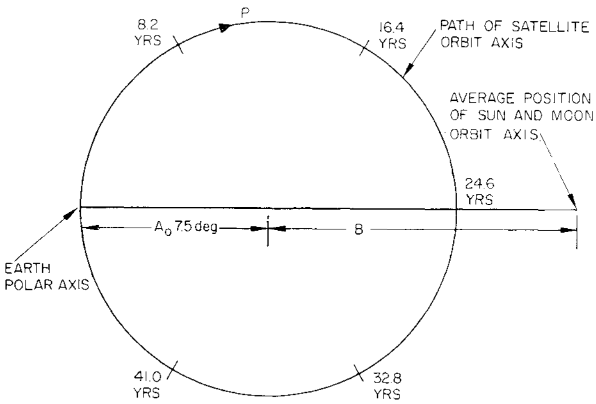

Fig. 4 Approximate orbit axis path of 24-hr satellite.

duced, but the number of thrust cycles required at maximum disturbance torque will increase. Figure 2 shows the total impulse and the number of thrustor operations per year required on each vehicle axis as a function of disturbance torque and minimum impulse bit. A minimum impulse bit of 10^{-2} lb-sec will be used for all control system designs.

Power Required

The average power required to operate the thrustors can be found by adding the average power required during thrusting; the average power required to ready the thrustor for operation, and any continuous power required:

$$P_a = P_c + (P_t + P_f)D_c + (E_t + E_f)/\tau$$

This is the power which must be provided by batteries while the vehicle is in the earth's shadow. Solar panels are sized to provide for peak power demands during normal operation.

Station Keeping

Perturbing Forces

To maintain a satellite stationary in a 24-hr orbit, small thrust impulses must be frequently applied, primarily because of two sources of perturbing forces, the triaxility effect and the sun-moon perturbation.

The triaxial distortion of the earth's gravitational field gives rise to radial and tangential accelerations that are longitude-dependent. The radial component will only modify the required altitude of the satellite. The tangential component, however, causes an East-West drift. The magnitude of this component depends on the longitude of the station, but the location of the maxima are not precisely known.[4] Tracking data from the NASA-Hughes Syncom 2 communication satellite show this acceleration to be about 1.6×10^{-7} ft/sec^2 at 55° West longitude. Syncom 2 (at 55° West longitude) experienced an acceleration eastward, causing it to gain energy (altitude) and develop an apparent westward drift, until checked by onboard propulsion. This drift can be stopped by imparting to the vehicle a number of small velocity increments totaling about 7 fps/yr. If uncorrected, this acceleration would cause an East-West oscillation with an amplitude of up to 90° and a period between 1 and 5 yr.[4]

The primary effect of nearby celestial bodies on earth satellites is to precess the orbit plane about the orbit axis of the disturbing body. The inclination change is due to the component of the attraction of the celestial body normal to the satellite orbit. This component tends to pull the satellite orbit plane toward the "orbit plane" of the perturbing body. (In the case of the sun, this "orbit plane" would be the plane of the ecliptic.) The velocity of the orbit axis due to precession is given by[5]

$$V_2 = \tfrac{3}{4}(gM/a_p{}^3\omega_0)\sin i_2 \cos i_2$$

The maximum rate of change of inclination angle due to the sun and the moon are 0.269° and 0.679°/yr, respectively. This results in a maximum orbit inclination change of 0.948°/yr. In order to negate this and maintain the satellite orbit in the equatorial plane, velocity must be added to the satellite in a direction normal to the orbit plane. Because of its gyroscopic nature, the satellite orbit will precess on an axis through the point of applied torque or normal thrust. Therefore, if the satellite is to be precessed into the equatorial plane, thrust must be applied as the satellite crosses the equator, i.e., twice each day. The normal velocity increment imparted by thrusting adds vectorially to the orbital velocity of 10,087 fps, thus a net velocity increment of 176 fps is required to rotate the satellite orbit through 1°. In order to negate the solar and lunar perturbations of 0.948°/yr, a net correcting velocity of 0.466 fps/day is required.

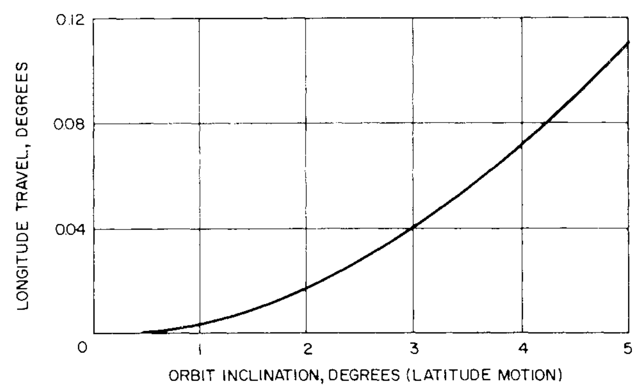

Fig. 5 Longitude motion due to orbit inclination for near stationary orbit.

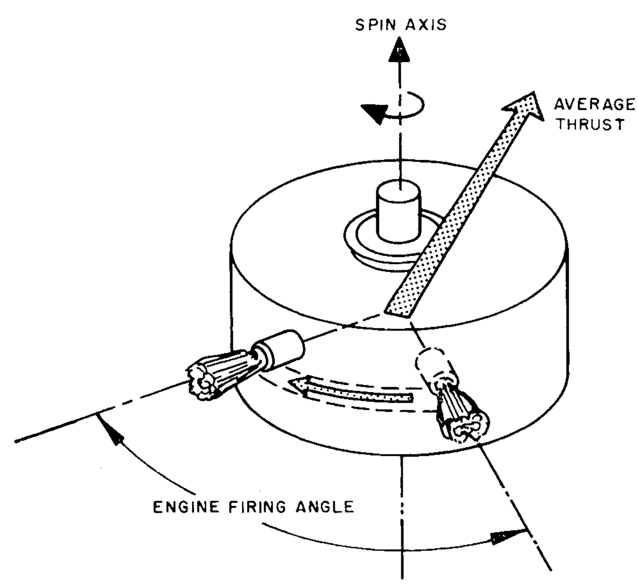

Fig. 6 Velocity control.

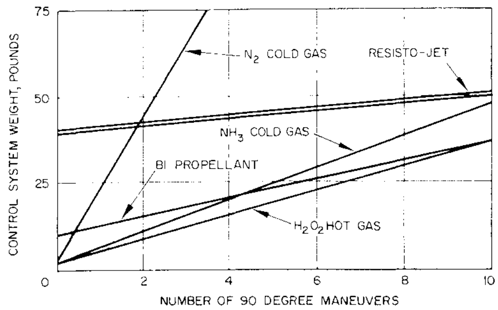

Fig. 7 Attitude control system weight vs maneuver angle for 500-lb vehicle.

If unchecked, these perturbations will not cause unlimited inclination. The path the orbit axis will follow on the geocentric sphere can be obtained as follows: The perturbing body will cause angular motion V_2 of the orbit axis normal to the direction of applied torque as shown in Fig. 3. In a similar manner, the oblateness of the earth will cause angular motion of the orbit axis V_1. These velocity components add vectorially to produce motion about some center of curvature. The radial and angular motion of the orbit axis about this point is[5]:

$$dR/dt = -V_1 \sin\alpha + V_2 \sin\beta$$

$$d\theta/dt = (V_1 \cos\alpha + V_2 \cos\beta)/\sin R$$

where

$$V_1 = \tfrac{3}{2} J_2 n_0 (R_0/a_s)^2 \sin i_1 \cos i_1 = K_1 \sin i_1 \cos i_1$$

$$V_2 = \tfrac{3}{4} (gM/a_p{}^3 n_0) \sin i_2 \cos i_2 = K_2 \sin i_2 \cos i_2$$

Setting R constant to find the center of curvature and eliminating α and β through spherical trigonometric identities yields

$$dR/dt = 0 = [-K_1 \sin A \cos i_1 + K_2 \sin B \cos i_2] \sin\theta$$

$$d\theta/dt = [K_1 \cos^2 i_1 + K_2 \cos^2 i_2]/\cos R$$

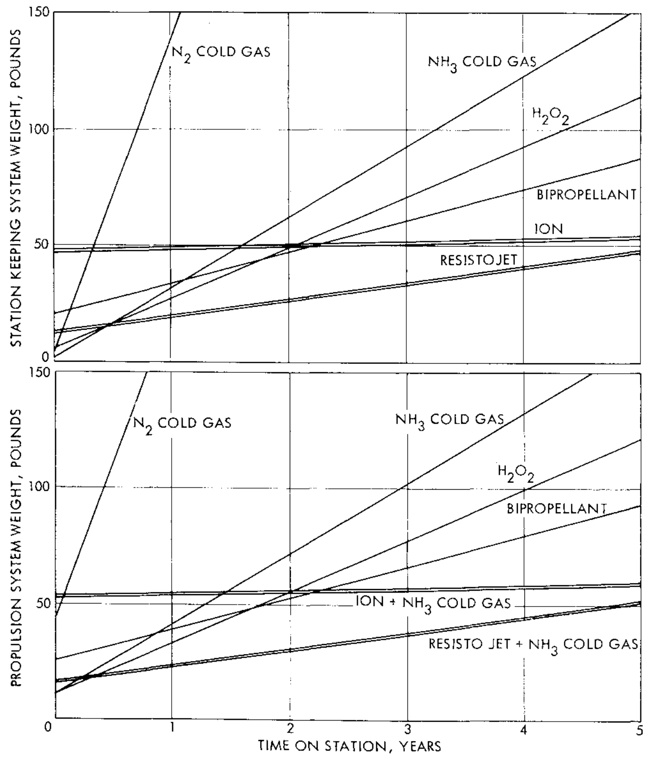

Fig. 8 Station-keeping and propulsion-system weights vs mission time for 500-lb vehicle.

Making the small angle approximations $\sin A = A$, $\sin B = B$, $\cos i_1 = \cos i_2 = \cos R = 1$ yields a circular orbit path whose axis is inclined at an angle A from the polar axis, and with a cycle period of $\tau = 2\pi/(K_1 + K_2)$. Putting in numbers and using the mean value for the lunar inclination:

$$K_1(\text{earth}) = 4.88 \sin i_1 \cos i_1, \text{ deg/yr}$$

$$K_2(\text{sun and moon}) = 2.30 \sin i_2 \cos i_2, \text{ deg/yr}$$

$$A + B = 23.5°$$

$$A = (A + B)/(1 + K_1/K_2) = 7.5°$$

Therefore a maximum inclination of 15° is reached, and the approximate orbit axis path is circular with a period of about 50 yr. Figure 4 shows the approximate orbit axis path resulting from the small angle approximations. The maximum error in the small angle approximations occurs at $i_1 = 0$ and are about 8% in radius of curvature and 5% in angular velocity. This orbit inclination causes the satellite to oscillate in longitude as well as latitude. However, as shown in Fig. 5, the longitudinal motion due to orbit inclination is negligible compared with that in latitude.

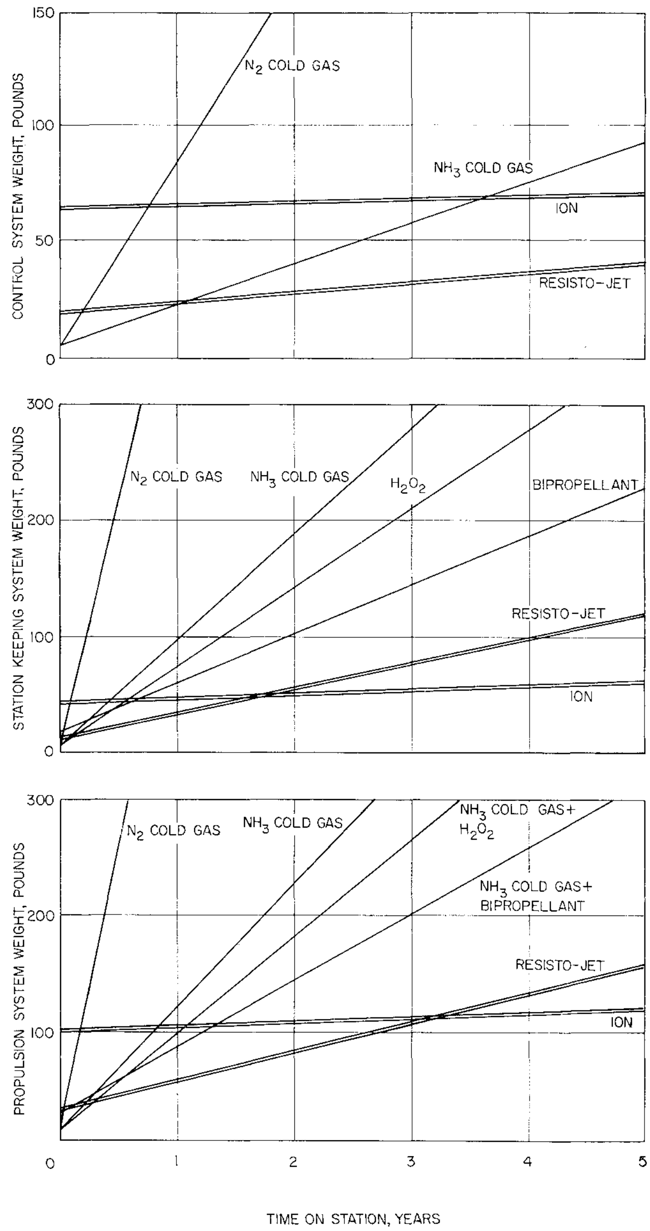

Fig. 9 Propulsion system weights vs mission time for 1500-lb vehicle.

Table 4 Propulsion system for 90°-maneuver for 500-lb spin-stabilized vehicle (430-lb-sec impulse)

Parameter	Cold-gas jet		Chemical		Resisto-jet
Propellant	N_2	NH_3	H_2O_2	N_2O_4-UDMH	NH_3
Specific impulse	60	100	160	240	425
Maneuver time, hr	0.75	0.75	0.15	0.15	24
Operating thrust, lb	1.0	1.0	5.0	5.0	0.03
Average thrust	0.167	0.167	0.835	0.835	0.005
Powers, w:					
Operating engine	...	...	...	...	100
Average engine	...	...	...	...	100
Weights, lb:					
Solar panel	...	...	...	...	40.0
Two thrustors	1.0	1.0	2.0	10.0	2.0
Propellant	7.2	4.3	2.7	1.8	1.0
Tankage	14.4	0.4	0.8	0.9	0.1
Propulsion system	22.6	5.7	5.5	12.7	43.1

Table 5 Propulsion system for station keeping for 500-lb, spin-stabilized vehicle for 3 yr (8150-lb-sec effective impulse)

Parameter	Cold-gas jet		Chemical		Resisto-jet	Ion
Propellant	N_2	NH_3	H_2O_2	N_2O_4-UDMH	NH_3	Cesium
Specific impulse, sec	60	100	160	240	425	4500
Operating thrust, lb	1.0	1.0	5.0	5.0	0.001	0.00015
Duty cycle, %	0.0087	0.0087	0.00174	0.00174	8.7	87
Thrust period, min/day	...[a]	...[a]	...[a]	...[a]	2.1	20.9
Powers, w:						
Operating engine	...	...	...	...	20	50
Solar panel	...	...	...	...	20	63
Weights, lb:						
Power conversion	...	...	...	...	...	14
Solar panel	...	...	...	...	8	25
Four thrustors	2	2	4.0	20	4	8
Propellant	136	82	51	34	19	2.5
Tankage	272	8	15	7	2	1.5
Propulsion system	410	92	70	61	33	51

[a] Thrusting will occur on weekly or monthly intervals.

Orbit eccentricity can also cause considerable longitude motion (approximately 1° for each percent eccentricity). It can be removed by simply thrusting in the direction of satellite velocity at apogee or against satellite velocity at perigee. To remove it, an initial value of 1% in orbit eccentricity would require a total velocity increment of 100 fps. This correction would probably be made by the prime propulsion system used to place the satellite in a 24-hr orbit and will therefore not be considered.

Impulse Required for Station Keeping

The impulse required to maintain the 500-lb vehicle on station for 1 yr is then $\Delta VM = 2720$ lb-sec; 8150 lb-sec impulse is required to maintain the 1500-lb vehicle on station for 1 yr. This impulse must be applied both normal to the orbital plane (North-South) and in the direction of the orbital path (East-West). For the 3-axes-controlled vehicle this poses no problems. However, in order to thrust efficiently in both North-South and East-West directions, the spin-stabilized vehicle should have its spin axis normal to the orbit plane (North-South). In this orientation the major correction impulse (North-South) is made by thrusting continuously along the vehicle spin axis. The smaller correcting impulse (East-West) is imparted by a thrustor normal to the spin axis fired intermittently during vehicle rotation (Fig. 6).

Results

A number of contemporary propulsion devices were examined for station keeping and attitude control of the previously described vehicles. Specific weights used for thrustors, power conversion equipment, batteries, solar cells, tankage, and plumbing are compatible with present Hughes designs and are described in Tables 1, 2, and 3. The resisto-jet engine considered operates with continuous heating at a power level as low as 10 w. It seems possible to attain this low power level with proper scaling. The ion engine considered is more advanced than the SERT I engine, with appreciably lower power requirements and a specific impulse of 4500 sec. This type of engine is now under development at the Hughes Malibu Research Laboratories[2] under NASA contract.

Table 6 Propulsion system for 3-axes attitude-control 1500-lb vehicle for 3 yr (4740-lb-sec impulse)

Parameter	Cold-gas jet		Resisto-jet	Ion
Propellant	N_2	NH_3	NH_3	Cesium
Specific impulse	60	100	425	4500
Thrust level, mlb	0.5	0.5	1.0	0.25
Duty cycle per axis, %	5	5	2.5	10
Time for 90° maneuver, min	19.5	19.5	13.7	27.5
Powers, w:				
Operating engine	...	...	60	75
Continuous	...	...	60	25
Solar panel	...	...	60	95
Weights, lb:				
Eclipse battery	...	...	...	8
Power conversion	...	...	...	18
Solar panel	...	...	8.0	13
12 thrustors	6.0	6.0	12.0	24
Propellant	79	47.3	11	2[a]
Tankage	158	4.7	1	1
Propulsion system	243	58	32	66

[a] Three times single-axis propellant load.

Table 7 Propulsion system for station keeping for 1500-lb vehicle for 3 yr (24,500-lb-sec effective impulse)

Parameter	Cold-gas jet		Chemical		Resisto-jet	Ion
Propellant	N_2	NH_3	H_2O_2	N_2O_4-UDMH	NH_3	Cesium
Specific impulse, sec	60	100	160	240	425	4500
Operating thrust, lb	1.0	1.0	5.0	5.0	0.001	0.0004
Duty cycle, %	0.026	0.026	0.0052	0.0052	26.5	87
Thrusting time/day, hr	...[a]	...[a]	...[a]	...[a]	6.4	20.9
Operating electrical power, w	...	...	...	...	20	110
Solar panel power, w	...	...	...	...	20	138
Weight of power conversions, lb	...	...	...	...	...	18
Weight of solar panel, lb	...	...	...	...	3	19
Weight of 8 thrustors, lb	4	4	8	20	8	16
Weight of propellant, lb	410	245	153	102	59	7.5
Weight of tankage, lb	820	25	46	20	6	3.5
Propulsion system weight, lb	1234	274	207	142	76	64

[a] Thrusting will occur on a weekly or monthly basis.

Spin-Stabilized Vehicle

Spin stabilization reduces the impulse required to maintain orientation to negligible values but makes maneuvering costly in terms of fuel consumption. Figure 7 shows the net weight of hot-gas, cold-gas, and resisto-jet propulsion systems as a function of total maneuver angle. Even though the maneuver rate assumed was very low (90°/day), the resisto-jet system is clearly inferior to the cold-gas systems for reasonable maneuver angles. Ion engine systems have been omitted from this study because no engine in development to date can produce the required thrust. For large maneuver angles all systems would be excessively heavy compared to a 3-axes control system. Weight breakdowns for these systems capable of maneuvering vehicle *A* through 90° are given in Table 4.

The upper part of Fig. 8 shows the net weights of various thrusting systems designed to negate (by frequent small impulses) both North-South and East-West perturbations as functions of time on station. A weight breakdown for the systems considered at a 3-yr station time is given in Table 5, which shows that power requirements impose a severe weight penalty for the ion engine systems in this time interval. The resisto-jet system shows a distinct advantage over other systems over most of the 5-yr period. The dual tasks of attitude control and station keeping can be performed by combinations of the previously described systems. The lower part of Fig. 8 shows the weights of hot-gas, cold-gas, and resisto-jet systems capable of turning the spin-stabilized vehicle through 180° and controlling it in attitude and position. A combined cold-gas and resisto-jet system is lightest in this application for most mission times considered.

Three-Axes Attitude-Controlled Vehicle

Cold-gas thrustors and electric engines were considered for 3-axes attitude control of the 1500-lb spacecraft. The upper part of Fig. 9 shows the control system weight (less sensing equipment) required as a function of time on station. The ammonia cold-gas system is the lightest for mission times under 1 yr. For longer mission times the resisto-jet system is considerably lighter than the best cold-gas system. A weight breakdown for the systems considered is shown in Table 6.

A velocity increment of 175 fps/yr is required for station keeping. The middle part of Fig. 9 shows the weights of cold-gas, monopropellant, bipropellant, and electrical propulsion systems required for a 1500-lb vehicle as a function of time on station; Table 7 gives the weight breakdown. With directed solar panels available for electrical energy, both electrical systems show clear weight advantages over other systems for times in excess of 1 yr.

The lower part of Fig. 9 shows the weights of cold-gas, combined cold-gas and bipropellant, resisto-jet, and ion engine systems capable of controlling the vehicle in position and attitude. Electric systems are lighter than the chemical system for mission times of over 1 yr. For the longer mission times, the ion engine system is lighter than the resisto-jet systems because of its lower propellant consumption.

Conclusion

Solar powered electric rockets were compared with conventional rockets for controlling both a spin-stabilized and an inertially oriented 24-hr-stationary satellite in position and attitude for periods of up to 5 yr. The conventional rocket systems were found to be lighter than electric propulsion systems for controlling the attitude of the spin-stabilized vehicle. The resisto-jet electric system was quite competitive with conventional rockets for station keeping the spin-stabilized vehicle. Both electric systems were found to offer considerable weight savings over conventional rocket systems for control of the inertially oriented vehicle.

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