All-Electric Thruster Control of a Geostationary Communications Satellite

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Investigation of an all-electric thruster combination for precise, long-life control of a nonspinning geostationary satellite without momentum exchange devices is presented. Electric thrusting devices offer certain unique advantages for this application and are assumed for both orbit and attitude control. Maneuvers associated with deployment, acquisition, and on-station operations are described in detail. A specific satellite configuration is assumed as a realistic model for use during the 1977–84 period, and thruster components and required performance parameters are selected based on estimated 1975 technology. Mercury electron bombardment engines were chosen for large-impulse maneuvers, and solid Teflon pulsed plasma thrusters were picked for small impulse-bit functions. Restrictions on thruster site selections resulted in a nonoptimum propulsion subsystem configuration. Nevertheless, adverse interaction with communications operations appear minimal, and full thruster redundancy is available. It is concluded that an all-electric thruster control system is feasible and can satisfy mission requirements for a large class of nonspinning satellites.

Introduction

POTENTIAL improvements in communications services provided by satellites of limited size and weight must have associated increases in position and attitude accuracy. Thus, development of new generation communications satellites will require a consideration of advanced propulsion technology with respect to precise, long-life control. Inherent reliability, repeatability, and redundancy in the control system are of primary importance in component selection. Electric thrusters offer certain unique advantages to this application, and this investigation considers the feasibility of using devices for primary satellite control. Performance capabilities of electric thrusters during attitude and orbit maneuvers are assessed, and recommendations based on several factors are offered. Schemes for controlling a "reference" satellite of specified configuration are formulated, based on partial simulation of stationkeeping dynamics and thrusting logic.

The use of electric thrusters for synchronous satellite control has been studied and proposed since the early 1960's.[1] In addition, the motion of such a satellite under the influence of a radial thruster which is mechanically vectored has been investigated and suggested as a technique for both attitude control and stationkeeping.[2] Experimental electric thrusters of varying types and sizes have been flown in space successfully,[3] e.g., SERT I and II and ATS-D. Furthermore, electric microthrusters have been in operation for east-west (E-W) stationkeeping of the synchronous, spin-stabilized Lincoln Experimental Satellite, LES-6, for more than three years.[4] The study reported here considers recent advances in thruster technology, and evaluations are based on over-all satellite performance objectives for a specific mission and spacecraft. Thus, interactions of forces and hardware components significantly influence the selection of control elements and operations.

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The specific satellite configuration under consideration is assumed to be launched with an Atlas/Centaur vehicle and injected into geostationary orbit with an apogee kick motor. On-orbit mass is limited by the booster selection and assumed to be initially 716 kg (1574 lbm). This spacecraft would be actively controlled and have an Earth-oriented primary module with sun-oriented solar cell panels, as illustrated in Fig. 1. Specific properties of interest are listed in Table 1. Variations of mass and inertia over the satellite lifetime can be ignored for this study. Propellant calculations and thruster selections are based on an assumed in-orbit life of seven years, in the period from 1977 to 1984. The sequence of interest in the mission profile commences at termination of "rough" despin after apogee kick execution. Of course, certain systems aspects require consideration of events occurring before apogee firing. Nevertheless, primary concern is limited to on-station operation, attitude acquisition and reacquisition, and station establishment and relocation.

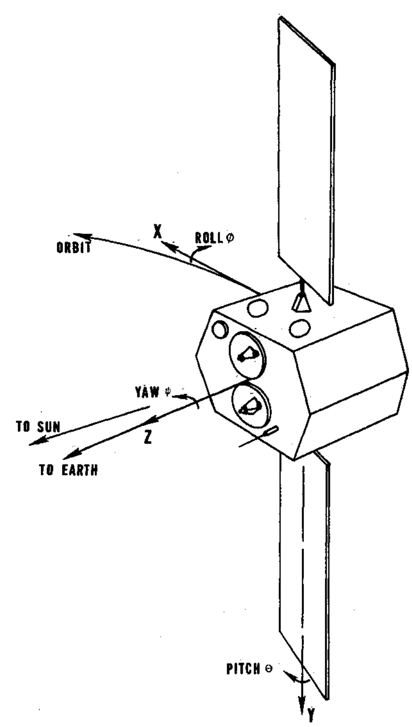

Fig. 1 Reference satellite configuration.

Table 1 Reference satellite properties

Initial on-orbit mass	716 kg (1,574 lbm)
Moments of inertia	
a) Panels deployed	
$I_X = I_Z$	2,000 kg-m^2 (1,475 slug-ft^2)
I_Y (panels locked)	500 kg-m^2 (369 slug-ft^2)
I_Y (panels free)	440 kg-m^2 (325 slug-ft^2)
b) Panels stowed	
$I_X = I_Y$	510 kg-m^2 (376 slug-ft^2)
I_Z	540 kg-m^2 (398 slug-ft^2)
Solar array area	20 m^2 (215 ft^2)
Initial power available	1.72 kw (Max.)
Useful life in orbit	7 years (1977–1984)

Mission and Control Requirements

Active control of a geostationary satellite entails several functional aspects. These include longitude and lattitude maintenance within limits corresponding to contractual ground station tracking and coverage agreements, and pointing requirements for multiple narrow-beam antennas. A high degree of attitude-pointing accuracy is most critical for ground stations far away from the boresight location. Allowable orbit and attitude limits are listed in Table 2 for this study. Ranges of accuracy limits will affect thrust impulse sizes and duty cycles in those cases where a perturbation results in periodic motion of amplitude falling within the corresponding accuracy range.

Table 2 On-station mission requirements

Beam pointing accuracy (maximum deviation from local vertical)	0.05°–0.25°
Longitude maintenance limits (E-W deviation)	0.05°–0.1°
Latitude maintenance limits (N-S deviation)	0.1°

In addition to nominal on-station functions, the control system must provide attitude and station acquisition and reacquisition impulses. Attitude reorientation performance is judged on the basis of time required to complete maneuvers, assuming sufficient thrust levels are available. Initial station establishment and station change maneuvers are also judged primarily on the basis of time required in comparison to other thruster types, e.g., monopropellant. Table 3 summarizes reasonable, but somewhat arbitrary, off-station time requirements for both attitude and orbit acquisition and change. This indicates that initial orientation and station establishment may be permitted up to 12 days, during 10 of which the communications subsystems could be tested and prepared for operation upon reaching the nominal longitude position. Station changes of 120° completed in less than 20 days are very expensive in terms of power for electric thrusters and propellant for hydrazine rockets.[5] Figure 2 gives power required vs maneuver time for a 120° longitude change using a bombardment thruster with estimated 1975 performance and a power conditioner efficiency of 85%. In addition, the associated propellant consumption for continuous thrusting ($I_{sp} = 2200$ sec) is shown. Loss of north-south (N-S) stationkeeping capacity is noted, assuming common fuel tanks and a specific impulse of 3500 sec for the N-S engines. If a station change is anticipated, extra propellant may be added before launch to avoid this loss of latitude control capability.

Table 3 Off-station mission requirements

Attitude acquisition time	18–48 hr
Attitude reacquisition time (no despin required)	18–24 hr
Station establishment [maximum injection error of 30.5 m/sec (100 fps)]	10 days
Station change (120° longitude shift)	15–50 days

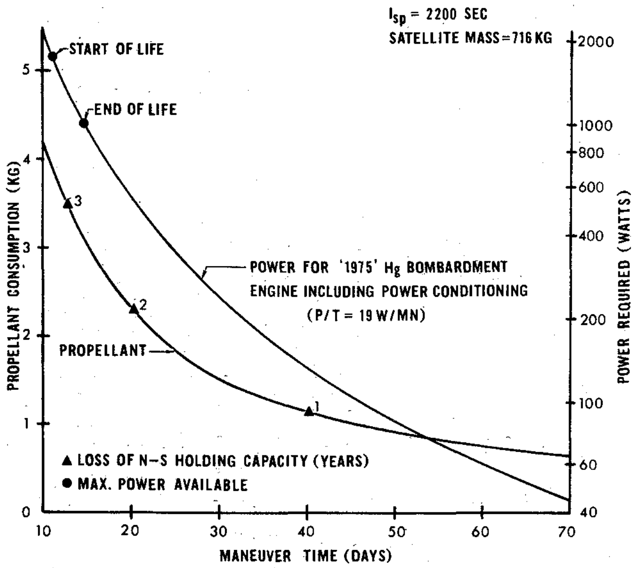

Fig. 2 Electric power and propellant consumption for 120° longitude change.

No requirements need be specified for radial position since small errors in altitude do not affect satellite performance. Furthermore, active longitude control also corrects radial errors because of the coupling between orbital velocity and altitude for circular orbits.

Control requirements are the necessary responses to both perturbations and commanded deviations from an existing situation. This definition applies to both orbit and attitude maneuvers, each of which has varying requirements depending upon the situation, e.g., acquisition, reacquisition, station and attitude maintenance, etc. Nominal orientation torque requirements include responding to perturbing moments, from outside as well as internal sources, after initial acquisition has been completed. Significant torques result from solar pressure and station change thruster misalignment, as summarized in Table 4. A simplified form of the estimated solar torque components is used in which the initial time corresponds to the 6:00 a.m. or 6:00 p.m. orbital position. Variations with time are due to the rotation of both the Earth-oriented central-body and the coordinate system. Torque magnitudes used here were derived by extrapolation of the analysis presented in Ref. 6 and are dependent upon reflectivity and geometry. Misalignment of longitude change thrusters results from the contributions of two uncertainties: center of mass location and net thrust direction relative to mounting alignment. A total alignment error of 0.1° is assumed for a worst case. Other sources of attitude perturbation can be ignored during nominal operation at synchronous altitude.

Stationkeeping requirements include only responses to outside forces. However, the over-all orbit control mission includes initial station acquisition and 120° longitude changes. Table 5 lists ranges of velocity increments associated with orbit control

Table 4 Nominal orientation torque requirements

Solar pressure	
$T_X = 2(1 - 2\sin\omega_0 t) \times 10^{-5}$ N · m	
$[1.48(1 - 2\sin\omega_0 t) \times 10^{-5}$ lb · ft$]$	
$T_Y = 10^{-4}\cos\omega_0 t$ N · m	
$(0.738 \times 10^{-4}\cos\omega_0 t$ lb · ft$)$	
$T_Z = -5 \times 10^{-5}\cos\omega_0 t$ N · m	
$(-3.7 \times 10^{-5}\cos\omega_0 t$ lb · ft$)$	
Station change thruster misalignment torque (due to 0.1° error)	8.5×10^{-5} N · m $(6.26 \times 10^{-5}$ lb · ft$)$

Table 5 Orbit correction requirements

On-station:	
Longitude (E-W)	0–1.83 m/s/yr (0–6.0 fps/yr)
Latitude (N-S)	40.2–51.3 m/s/yr (132–168 fps/yr)
Station acquisition:	
Injection error	0–30.5 m/sec (0–100 ft/sec)
120° Longitude change:	
Total velocity increment[5]	27.1–91.5 m/sec (89–300 fps)
(Drift time 50 to 15 days)	

requirements. The E-W holding requirement is a function of longitude only and is the result of equatorial asphericity (triaxiality), which is the only in-plane secular perturbation of significance to geostationary orbits.[1] Other in-plane disturbing forces are periodic and of low amplitude. Latitude corrections require the major portion of propellant and result from lunar-solar attraction of the satellite.[7] The initial station acquisition maneuver requires elimination of injection velocity error, assumed to be as much as 30.5 m/sec (100 fps). Actual performance[8] has been significantly better than the range indicated in Table 5. However, when no attitude control is provided during apogee transfer (as with an all-electric control system) further errors may be introduced. The upper limit on expected velocity error is considered conservative for this scheme. Finally, a station reacquisition capability will provide a possible in-orbit backup satellite. Station changes of up to 120° are anticipated with an assumed transfer time range of 15–50 days.

Sensor Requirements

In order to uniquely determine vehicle orientation at any time, at least two types of outside sensors are needed, one to detect deviation from the local vertical (Earth sensor) and the other to determine relative position of the sun (sun sensor). However, a third type of sensor is required here to determine yaw errors, e.g., star tracker. The combination of selected attitude control components, specified beam-pointing accuracy, and ground station locations indicate the need for a yaw sensor. Sensing of the local vertical may also be accomplished by using special beam patterns and modulation techniques. Sun sensors are necessary only for solar panel steering and maintenance of power. Complete spherical (4π sterad) aspect coverage by such sensors is suggested, because attitude motion during despin is somewhat uncertain. Output of sun sensors and rate gyros would then be used to determine angular rates. Of primary concern in the selection of an Earth sensor is acquisition performance. A range of $\pm 25°$, typical of such devices, is used here.

Orbit control can be achieved only with an accuracy less than the combined accuracy of orbit determination techniques. It is of critical importance to know the node position with at most a 5° uncertainty, in order to execute N-S thrusting efficiently and effectively. Furthermore, these corrections cannot be made without a precise measurement of latitude motion. This is also true for longitude drift. Fortunately, current techniques permit position and attitude measurement to within 0.01°.[8] Thus, the technology associated with attitude and orbit determination is capable of providing measurement and prediction accuracies required for this mission.

Electric Thruster Technology

The primary factor which will eventually lead to the operational use of such thrusters for N-S control and station changing seems to be specific impulse, which is easily ten times that of hydrazine. Attitude control applications await satisfactory reliability, life, power requirements, and repeatability. Thruster weight is not an important factor for high impulse applications. The elimination of mechanical valves, complicated feed systems, and other moving parts make some electric engines competitive with chemical monopropellant devices for low impulse functions, even at a slight weight penalty. Several electric devices have already flown in space, but selection for commercial venture is the final test of acceptance.

Table 6 Total impulse requirements

Orbit maneuvers:	
Station acquisition	65,400 N · s (14,700 lb · s)
N-S stationkeeping (7 yr)	219,500 N · s (49,400 lb · s)
E-W (worst case)	9,150 N · s (2,060 lb · s)
Attitude:	
Initial acquisition (worst case)	23.1 N · s (5.2 lb · s)
On-station (assumed)	26,700 N · s (6,000 lb · s)

A division of electric devices can be made on the basis of application. Attitude and longitude control of nonspinning bodies is ideally accomplished with microthrust engines, while N-S stationkeeping and longitude changes require millipound size thrusters. Total impulse requirements for the mission are listed in Table 6. The largest single impulse appearing is for N-S stationkeeping, which is an accurate estimate of the actual requirement.[5] Other impulse figures depend on station selection, final control system design, duty cycles, and uncertainties associated with injection, perturbations, and thrust alignment.

Holcomb[3] has made a comprehensive survey of all potential electric devices and their current status. It is not the intent to repeat this survey or make final thruster selections here. However, for the purposes of preliminary control system design, specific thruster types were chosen on the basis of past developments, test data, and flight experience. Orbit maneuvers can be performed quite satisfactorily with mercury electron bombardment ion thrusters. These devices offer millipound thrust at reasonable power levels and can use a common propellant supply system. Characteristics of two engine sizes are illustrated in Fig. 3. These performance curves are based on conservative extensions of current technology to 1975.[5] The 6.7 mn (1.5 mlb) engine is assumed for N-S stationkeeping with operation at a specific impulse of 3500 sec. This operating figure was obtained through a consideration of power conditioning and propellant weights. The 53.4 mn (12 mlb) thruster is to be used for initial station acquisition and longitude change maneuvers. Since high thrust is desirable at limited power, a specific impulse of 2200 sec was selected. Thrust level determinations are based on mission requirements. A power conditioning efficiency of 85% is assumed; thus, the over-all

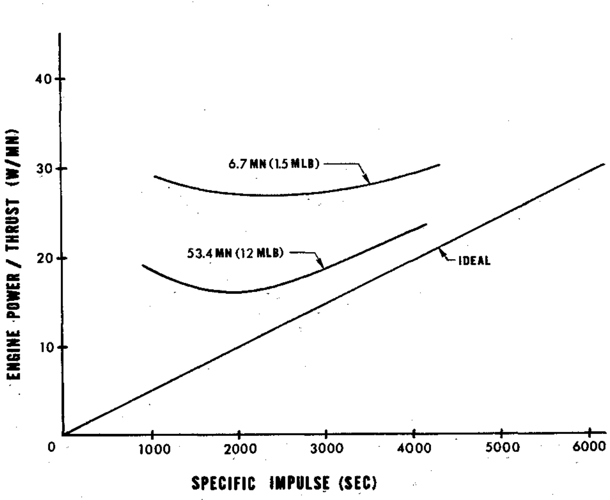

Fig. 3 Performance of mercury electron bombardment thrusters.

power/thrust ratios for the 6.7 and 53.4 mn engines are 33 and 19 w/mn, respectively.

Attitude control and E-W stationkeeping can be performed with Teflon pulsed plasma thrusters of the type used on LES-6 and developed for LES-7. These offer a unique combination of variable effective thrust, solid state design, simplicity, and self-containment. Although a common feed system is not possible, their inherent reliability and demonstrated flight performance overshadow any slight weight and power penalties. Characteristics reflecting current technology and mission requirements are listed in Table 7. These estimates are based on ground-demonstrations and flight performance.[3] Furthermore, all attitude and E-W stationkeeping units are identical, except possibly for propellant mass and mounting provisions. The configuration is similar in shape to that of LES-6 units, but the size is comparable to the LES-7 thrusters. Use of identical units will permit a minimum of development and testing. Discussions of thruster selection criteria follow.

Attitude Acquisition and Torquing Thruster Selection

Use of electric thrusters for attitude holding is considered feasible and desirable in many instances. However, a principal factor in selection for attitude control is their ability to perform acquisition in a reasonable amount of time with the power available. As a basis for comparison of acquisition schemes, the monopropellant attitude jet system with a momentum wheel is considered as the reference case. A general sequence has been formulated for such systems[6] and will be discussed before developing possible techniques using other equipment. Attitude acquisition begins with the satellite spinning about its Z (yaw) axis at about 1° per sec, possibly with some wobble (precession). The plane-change associated with the apogee burn requires that the momentum vector be 25.2° out of the equatorial plane. If apogee occurs at an ascending node, then the direction of the Z axis is 25.2° below this plane, as illustrated in Fig. 4. Initially, the spacecraft is in a stowed configuration. Solar panels are assumed to deploy so that the normal to these surfaces is parallel to Z. Thus, the sun angle σ, defined in the figure, is an indication of available solar power if the panels were deployed. The most favorable apogee firing time appears to be 6:00 a.m. or 6:00 p.m. during the winter or summer months, because the Z axis is almost parallel to the sun line, if no wobble occurs. Sun sensors and rate gyros verify initial acquisition spin rate and orientation so that attitude jets can reduce the spin from 1 to 0.1° per sec. The Z axis is then reoriented to bring σ to zero. Solar panels will then be deployed, and since the sun location is anticipated, solar power becomes available. The central satallite body can be rotated 90° about the Y (pitch) axis so that at 6:00 a.m. or 6:00 p.m. the Earth can be acquired by the sensor. Rate gyros and monopropellant jets maintain a spin rate about the X-axis (roll) to insure Earth sensing during ± 100 min of six o'clock. A minimum threshold spin rate for the monopropellant jets can be maintained. For example, a 0.1° per sec rate will allow three full revolutions of the satellite for scanning. Once Earth is acquired the X-axis can be rotated so that β is brought to zero, and panel despin motors are then engaged. Finally, the momentum wheel is spun up and the on-station mode may commence.

Table 7 Teflon pulsed plasma thruster characteristics

Power/thrust	49 w/mn (218 w/mlb)
Thrust	445 μn (100 μlb)
Specific impulse	1028 sec
Engine efficiency	10%
Power conditioner efficiency	85%
Weight per unit	
E-W thruster	5.2 kg (11.5 lb)
Attitude thruster	5.0 kg (11 lb)

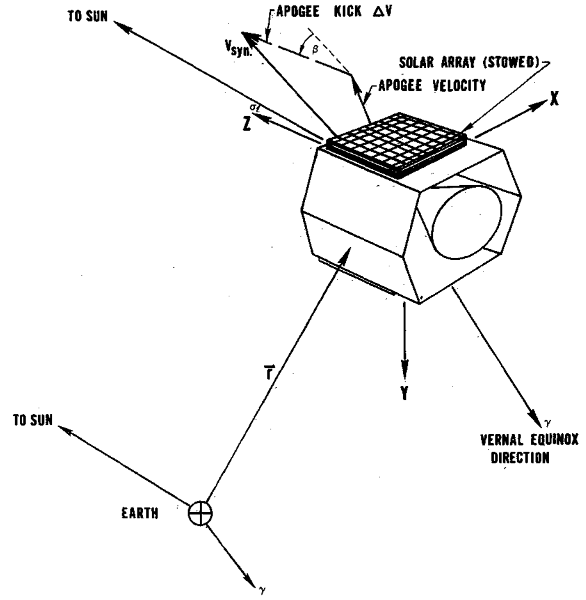

Fig. 4 Initial orbit and attitude situation for acquisition.

It should be noted that a considerable number of thrust impulses are required over small time intervals in several instances. Elimination of wobble must be accomplished quickly with battery power in order to deploy the panels. It is not clear whether or not the 1° per sec rate is too high for panel deployment. If so, a rapid reduction to 0.1° per sec would then be desirable. Reorientation of the Z axis to bring σ to zero is not critical in terms of time, but for Earth acquisition rapid attitude change will significantly simplify the maneuver, although only one revolution is needed in a period of 3 hr and 20 min. Once the Earth reaches the sensor field of view, it is desirable to stop rotation before the image leaves this field again.

The case in which all torques are produced by electric thrusters presents special problems, particularly when a large impulse is required, e.g., for Earth acquisition. Consider the general situation in which there is wobble initially. Sun sensors and rate gyros will yield angular momentum components with time. Thrusters must be operated intermittently on batteries to stabilize about the Z axis. If possible, the solar panels should be deployed if σ is reasonably small. Then further despin with yaw thrusters will eliminate momentum. Rotation of Z to bring σ to zero is next. Then the central body rotates 90° for the Earth acquisition sequence. Finally, β is brought to zero and panel despin is carried out. Impulse requirements for acquisition maneuvers have been estimated and are listed in Table 8. For dewobble, a worst case of 1° per sec is assumed about the transverse axis. In stowed configuration, $I_X \simeq I_Y \simeq 510$ kg · m^2 (376 slug-ft^2). All cases assume a single thruster is used to generate torque about each axis at a 0.91 m (3 ft) moment arm. Two possible situations occur for despin about Z from 1 to 0.1° per sec. If panels are still stowed, then batteries are used, but the required impulse is more because deployment reduces spin to 0.27° per sec without applying torque. To rotate the Z axis into the sun direction a worst case is considered in which $\sigma = 90°$ initially. Two 3-hr continuous thrusts are used, reversing direction at $\sigma = 45°$ with a thrust level of 29.3 μn (6.6 μlb). The 90° central body rotation was assumed to be a 3-hr thrust maneuver, again reversing direction at 45°. Spin-up for Earth acquisition is designed to allow despin within 40° once Earth enters the field of view, because the Earth sensor acquisition field is $\pm 25°$. At that point, $\beta \leq 23.5°$ and can be brought to zero within a few hours.

Attitude acquisition impulses appearing in Table 8 were obtained in a somewhat arbitrary manner. However, this does provide a basis for a "first-order" evaluation of potential thrusters to perform acquisition without monopropellant

Table 8 Impulse requirements for attitude acquisition maneuvers

Maneuver	Power source	Estimated impulse[a] ($N \cdot s$)	Remarks
Dewobble	Batteries	9.8	$I_X = I_Y = 510$ kg · m^2
Despin, panels stowed	Batteries	9.3	$I_Z = 540$ kg · m^2
Despin, panels deployed	Solar cells	6.7	$I_Z = 2{,}000$ kg · m^2
Z-axis into Sun ($\sigma \to 0$)	Solar cells (batteries if σ is large)	0.64	$I_X = 2{,}000$ kg · m^2; $f = 29.3$ μn for 6 hr continuous thrusting if $\sigma = 90°$ initially
90° central body rotation	Solar cells	0.14	$I_Y = 440$ kg · m^2 (without panels); $f = 25.8$ μn for 3 continuous thrusting
Spin-up about X axis	Solar cells	1.15	$I_X = 2{,}000$ kg · m^2; $f = 428$ μn; $\omega = 0.52 \times 10^{-3}$ rad/sec
Despin about X axis	Solar cells	1.15	
X-axis into plane ($\beta \to 0$)	Solar cells	0.16	0.74 hr up and 0.74 down[b] if $\beta = 23.5°$, $I_Z = 2{,}000$ kg · m^2; $f = 29.3$ μn
Panel despin	Solar cells	34.7	$I_Y = 440$ kg · m^2 assume spin-up of body to 15°/hr

[a] Assuming single thruster at 3-ft moment arm.
[b] Assuming despin within 40° rotation in roll.

systems. Primary properties of interest here are thrust and power consumption. Other properties, such as weight, specific impulse, and reliability, must be included in considerations after the initial thruster selection is made on the basis of acquisition capabilities. Holcomb[3] considers six thrusters for attitude control duties. However, the first three of these were eliminated from consideration because of their high weights or low total impulse capabilities. Table 9 lists the other candidate thrusters, acquisition maneuvers, and time and energy associated with each combination thereof. Effective thrust level and power consumption are given in each case. Also, the LES-7 device is shown with two thrust levels; the lower one corresponds to 1 pulse/sec (pps) and the higher one to 8 pps, which is available only for short intervals. The critical factors of concern are the total time of acquisition and the time to despin about the X-axis for Earth acquisition (item 6). Since at least an angular rate of 0.52×10^{-3} rad/sec is necessary to sweep one revolution during Earth passage through the orbital arc corresponding to the Earth sensor field of view, the time to despin must not exceed 0.74 hr. The only device listed which satisfies this requirement is the LES-7 Teflon thruster. Total time of the acquisition maneuvers is somewhat greater than the total thrust time shown in Table 2, because a certain amount of "drift" time is required between maneuvers. The total drift time will be at least 18 hr. Of course, it is desirable to complete acquisition in a reasonably short time, e.g., within two days.

From these considerations it is concluded that a LES-7 type Teflon engine designed to operate at 445 μn (100 μlb) thrust can handle acquisition within a two-day period. Furthermore, in selecting thruster units for attitude control, reliability and simplicity are of prime importance. Thus, Teflon pulsed plasma thrusters were selected for all attitude functions. Each attitude thruster is assumed to be independent and have a mass of 5 kg (11 lb), including 0.77 kg (1.7 lbm) of propellant. At one pulse per second (445 μn), 21.8 w are required to run each unit. This will incur a slight power penalty in terms of weight, possibly on the order of a few kilograms. However, attitude thrusters can be operated sequentially rather than simultaneously to minimize power requirements. Weight reductions may be possible by combining common elements while maintaining full redundancy. These thrusters must provide attitude holding torques after acquisition. Table 4 lists a maximum perturbing torque magnitude of 10^{-4} N · m (0.738×10^{-4} lb · ft). Thus, a thruster at a 0.91 m (3 ft) moment arm need only be

Table 9 Time and energy to perform attitude acquisition

	Thruster				
	LES-6 teflon	LES-7 teflon		EOS ion	TRW colloid
Maneuver	17.8 μn, 1.41 w	400 μn, 19.6 w	3,180 μn, 156.8 w	89 μn, 26 w	38.6 μn, 1.39 w
1) Dewobble (batteries)	153. hr 216. w-hr	6.8 hr 133. w-hr	0.85 hr 133. w-hr	30.6 hr 795. w-hr	70.2 hr 97.5 w-hr
2) Despin (stowed) (batteries)	146. hr 208. w-hr	6.5 hr 127. w-hr	0.81 hr 127. w-hr	29.2 hr 760. w-hr	67. hr 93. w-hr
3) Z-axis rotation ($\sigma \to 0$)	9.9 hr 14. w-hr	0.44 hr 8.6 w-hr	0.06 hr 8.6 w-hr	2. hr 52. w-hr	4.6 hr 6.4 w-hr
4) 90° central body rotation	2.2 hr 3.1 w-hr	0.1 hr 2. w-hr	0.012 hr 2. w-hr	0.45 hr 11.7 w-hr	1. hr 1.4 w-hr
5) Spin-up about X	17.8 hr 25.1 w-hr	0.8 hr 15.7 w-hr	0.1 hr 15.7 w-hr	3.6 hr 92.6 w-hr	8.2 hr 11.4 w-hr
6) Despin about X	17.8 hr 25.1 w-hr	0.8 hr 15.7 w-hr	0.1 hr 15.7 w-hr	3.6 hr 92.6 w-hr	8.2 hr 11.4 w-hr
7) Z-axis ($\beta \to 0$)	2.6 hr	1.2 hr	0.1 hr	0.5 hr	1.2 hr
8) Panel despin	32.4 min 0.77 w-hr	1.5 min 0.47 w-hr	0.18 min 0.47 w-hr	6.5 min 2.8 w-yr	14.9 min 0.35 w-hr
Total thrust time (hr)	349.8	16.6	2.0	70.1	160.6
Total energy on batteries (w-hr)	424.	260.	260.	1,555.	190.5

capable of 110 μn (24.6 μlb) thrust. This is easily accomplished with Teflon units at low pulse rates. A total impulse of 15,400 N · s (3,460 lb · s) is required to continuously cancel the cyclic torque of T_Y for seven years. This can be done with two Teflon engines, each using 0.77 kg (1.7 lbm) of propellant. Thus, an all-Teflon attitude control system can be recommended. The EOS ion and TRW colloid engines require more complicated propellant storage and feed systems, and both appear to have thrust levels too low for acquisition. Furthermore, Teflon devices provide variable thrust (pulse rate) over wide ranges.

Orbit Control Evaluations

A complete orbit control system must contain a variety of thrusters and redundant capability. Performance is required of this system for initial station establishment, stationkeeping, and longitude changes. Initial station establishment usually requires the elimination of injection errors in velocity. This maneuver is performed in conjunction with a longitude drift until the desired station is reached. If the sequence is executed properly, velocity and altitude errors are simultaneously diminished as the final site is approached. A "worst-case" situation is one in which a maximum injection error occurs so that the satellite is drifting away from the desired station, assumed to be within 120° of injection longitude. This would either require elimination of velocity errors in addition to inducing a positive drift, or waiting a longer time to allow the initial drift to continue around most of the orbit. Time to establish a station is a function of initial conditions and station change thruster performance. The maximum injection error is taken from Table 3 and is 30.5 m/sec (100 fps). If this results in drift toward (less than 180°) the final station, then thrust is required only to stop the spacecraft at the terminal stage of reaching the selected site. Otherwise more complicated maneuvers are necessary. The increase in orbital period due to 1% excess velocity, is 3%. Therefore, the drift rate due to the maximum velocity error is $-10.8°$ per day. Assuming a state-of-the-art thruster and 1 kw is available for station acquisition, this permits a thrust of 53.4 mn (12.0 mlb) and requires 4.75 days to stop on station. During this time, the last 24.3° of drift will occur. However, the total time to station may be as much as 25 days, because position acquisition cannot commence until attitude acquisition is complete and the probability that initial drift will be of desirable direction and magnitude is low. Thus, it is doubtful that electric thrusters will bring this satellite to station within the 10 day period specified in Table 3. A reduction in maximum expected injection error will not necessarily improve this situation. Realistic alternatives are to relax the time-to-station requirement to about 25 days or use a monopropellant system for this maneuver.

Station acquisition is ideally performed with high thrust. However, power is limited, which leads to the use of a high thrust/power engine. There is also the redundancy requirement which is satisfied by opposite facing engines, because a single unit can perform in either direction through satellite reorientation. Several candidate thrusters were considered, but only one seems to fulfill all important requirements and remain compatible with N-S engine selections. This is the mercury electron bombardment ion engine sized for 53.4 mn (12.0 mlb) with characteristics shown in Fig. 3. Common propellant tankage with N-S thrusters minimizes weight addition. Propellant mass required for a worst-case station acquisition corresponding to a velocity increment of 91.5 m/sec (300 fps) with a specific impulse of 2200 sec is 3.0 kg (6.7 lbm). Each thruster unit is estimated to have a mass of 4.1 kg (9.0 lbm) including feed lines. The power conditioner, which is common to both large station engines and N-S thrusters, with redundant components, represents about 7.5 kg (16.5 lbm). The same consideration for hydrazine would result in an analogous mass of about 45 kg (99 lbm), although the time to reach station is considerably less. Since full power is available for longitude shifts, it would be highly advantageous to use those initial station acquisition engines for station changes. It is conceivable that one of these two engines may not even be required for initial orbit maneuvers. Furthermore, either engine could perform the entire station change giving redundant capability for both major orbit maneuvers. Additional redundancy may be available from E-W stationkeeping thrusters, although these are envisioned as microthrusters and would require much greater time to execute a large velocity increment. For equal longitude shift times, hydrazine would require about five times the propellant mass of mercury bombardment thrusters. In summary, initial station acquisition engines should have sufficient reliability, life, and restart capability to perform a station change maneuver at some point during the seven-year satellite life.

Once the station has been established, the orbit control system can be engaged for stationkeeping, i.e., elimination of outside orbit perturbations resulting in violation of the position holding requirements listed in Table 2. Secular effects of uncontrolled orbit perturbations are well known,[7] and required velocity increments for station maintenance are listed in Table 5. Latitude requirements vary with inclination of the lunar orbit as measured from the equatorial plane; the average increment for the 1977-84 period being 43.5 m/sec/yr (143 ft/sec/yr). Application of impulsive thrust corrections can be easily handled in a simulation of stationkeeping dynamics; whereas low thrust devices complicate the situation due to factors associated with thrusting over finite intervals of time. Some work has been done with uncoupled correction schemes[1] in which N-S thrusting had no radial effect, and the single radial thruster concept mentioned previously has been proposed.[2] However, this latter approach is not practical in view of lubrication and reliability problems, as well as inefficient use of propellant. Since the N-S thrusters cannot be aligned with the pitch axis, but will be in the pitch-yaw plane, a radial component of thrust results as a byproduct of N-S corrections. This will effect motion due to coupling between radial and longitudinal dynamics. Analytical treatment of continuous or impulsive corrections is possible with some difficulty. However, low thrust devices must be applied discretely over finite periods. The most practical approach to investigating effects of such thrusters and demonstrating effective duty cycles is through direct simulation over several orbits. This offers an excellent opportunity to investigate undesirable coupling phenomena and secondary short-term effects. Results of such simulations were combined with other information to formulate a desirable combination of propellant utilization, power, and hardware.

Mercury electron bombardment thrusters were selected for N-S control, station acquisition, and major longitude changes. Because of duty cycle and communications requirements, two distinct engine sizes are recommended. N-S control is assumed to use four 6.7 mn (1.5 mlb) thrusters operating at a specific impulse of 3500 sec with a unit power/thrust ratio of 33 kw/n including a power conditioner efficiency of 85%. Station acquisition and change engines are selected as 53.4 mn (12.0 mlb) thrusters with a power/thrust ratio of 19 kw/n including the same power conditioner efficiency. These Mercury thrusters must provide a total impulse of 283,000 N · s (63,600 lb · s) and will have an estimated overall mass, including propellant, power penalty, and power conditioning, of 29 kg (64 lbm). These performance figures represent estimated 1975 technology, since this size hardware has not been developed or flight tested. Teflon pulsed plasma thrusters were selected for E-W control. Properties of these engines are listed in Table 7. Each unit has an estimated mass of 5.2 kg (11.5 lbm) including 1 kg (2.2 lbm) of propellant. Further advances in thruster technology may improve performance, but a conservative approach seems essential when commerical applications are the objective.

Table 10 Operating modes for redundancy

Mode	Function							
	Pitch	Roll	Yaw	N	S	E	W	Station
Primary	P_1P_3	R_3R_4	Y_1Y_3	N_1	S_1	E	W	L_1 or L_2
Backup	P_4P_2	R_2R_1	Y_4Y_2	N_2	S_2	$Y_1 + Y_2$	$Y_3 + Y_4$	L_2 or L_1

Systems Considerations

The final decision on thruster selection will depend largely on interaction of the control system with mission objectives. Delivery of communications services is the primary concern of the satellite system. Any engine which cannot operate satisfactorily within this system will not be selected. Considerations must include performance, expected life, reliability, power requirements, and weight. Another important aspect which may determine the deciding vote is the physical interaction of thrusters with other satellite components and operations. Interaction effects include exhaust impingement and RF interference, power sharing, redundant modes, and dynamics coupling. Therefore, placement of electric thrusters on a communications satellite is critical to mission success. Site selection is based on minimizing exhaust and interference effects while applying proper forces and torques. Fully redundant modes for all control functions are assumed mandatory. Figure 5 illustrates one possible thruster site combination. Some units are "clustered" to take advantage of the few sites which offer a minimal number of adverse effects of the exhaust plume. Also, the Earth-face should be free of thrusters since RF interference might be excessive from engines mounted on or next to this face. The several operating modes available are listed in Table 10, with the primary scheme appearing first. A total of 14 Teflon units are assumed. Six mercury bombardment engines are sufficient for major orbit control requirements. Thus, the total propulsion subsystem mass is estimated as 99.4 kg (218 lbm).

The use of momentum devices for attitude control may significantly reduce the number of thrusters and total impulse. However, the added weight and other disadvantages of such devices must be compared to the advantages associated with fewer thrusters and less propellant. Of course, reliability and life estimates must also be acceptable, and a redundant mode must be available. E-W holding engines are pitched away from the E-W direction to permit placement of the station-change thrusters. The radial component due to this is of negligible consequence. Furthermore, pulsed devices, such as the attitude control engines, may be fired in sequence so that power drain is equivalent to only a portion of the units. Duty cycles for attitude control need not be constrained otherwise.

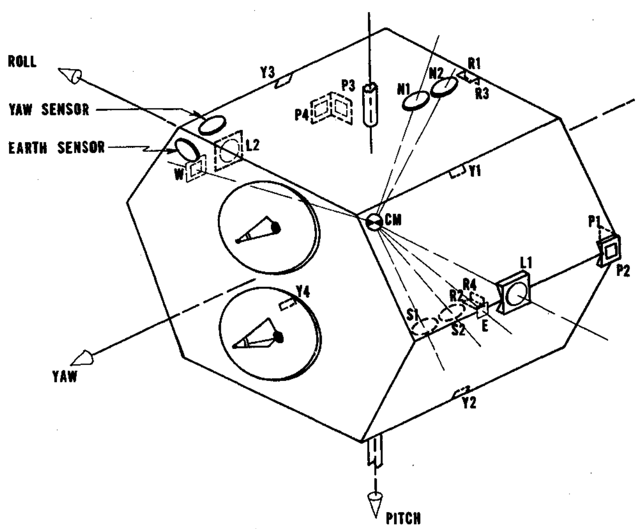

Fig. 5 Thruster arrangement.

The large N-S and station-change thrusters operate within some restrictions. Station change engines will require most of the power available in order to maneuver the satellite in reasonable time. Thus, no high-density communications functions can be performed while major longitude changes are occurring. This is consistent with operational philosophy. However, N-S thrusting must maintain a rather strict duty cycle because of restraints on firing positions around the orbit. Thus, there is a power penalty in weight to run these thrusters. The level of thrust is, however, high enough so that on a 50% daily duty cycle, yearly N-S corrections can be accomplished in only 155 days. Thus, no thrusting should be required during shadow periods. In any case, batteries will not be used to run large engines, because corresponding maneuvers are performed after panels are deployed and in sunlight.

Conclusions

The following conclusions can be made: 1) An all-electric thruster control system can provide complete control system impulses for the geostationary satellite studied. It is not as effective during initial injection and positioning as a hydrazine system, but is more efficient for on-station orbit control and more reliable for attitude holding. 2) Although electric thruster technology is rapidly advancing, a preliminary selection of thrusters has been made based on estimated 1975 state-of-the-art. Mercury electron bombardment engines are assumed for N-S corrections and large station-change maneuvers. Solid Teflon pulsed plasma thrusters were chosen for all attitude functions and E-W stationkeeping. 3) Restrictions on exhaust impingement and interference resulted in some nonoptimum thruster site selections. However, full redundancy is still possible, and adverse E-W coupling effects of N-S thrusters can be easily overcome by the E-W engines. 4) A specific satellite configuration was used here. However, limited variations in initial mass and shape should not effect the selection of thruster types or general conclusions. An increase in accuracy requirements would be limited by sensor performance and orbit determination methods, but the system selected here has considerable growth capability in terms of attitude and orbit-holding requirements.

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