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by Bruce E. LeRoy and John D. Regetz
Lewis Research Center
Cleveland, Ohio 44135

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Bruce E. LeRoy and John D. Regetz
Lewis Reserach Center
National Aeronautics and Space Administration
Cleveland, Ohio

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Abstract

A study of the Attitude Control System (ACS) is made for a solar electric propulsion geosynchronous transfer mission. The basic mission considered is spacecraft injection into a low altitude, inclined orbit followed by low thrust orbit changing to achieve geosynchronous orbit. Because of the extended thrusting time, the mission performance is a strong function of the attitude control system. Two attitude control system design options for an example mission evolve from consideration of the spacecraft configuration, the environmental disturbances, and the probable ACS modes of operation. The impact of these design options on other spacecraft subsystems is discussed. The paper presents a discussion of the factors which must be considered in determining the ACS actuation and sensing subsystems. The effects of the actuation and sensing subsystems on the mission performance are also considered.

Introduction

Recently, much interest has been evinced in the use of Solar Electric Propulsion (SEP) to transfer a payload from a low-energy geocentric orbit to one of higher energy, generally geosynchronous. The high specific impulse of electric thrusters permits transfer of a significantly larger payload than does chemical propulsion, for a given launch vehicle. Similarly, a given final payload can be delivered by a smaller, less expensive launch vehicle using electric propulsion. Hand-in-hand with the high specific impulse, however, goes low thrust. Thus, instead of a matter of minutes and hours, the transfer times using SEP become tens and even hundreds of days. In terms of percentage of overall mission time, therefore, the orbit transfer phase becomes a much more significant part of the mission than is true for chemical transfer. The geosynchronous transfer mission presents an extremely wide diversity in spacecraft environment characteristics and operational requirements.

During orbit transfer, the spacecraft ACS is required to orient the SEP thrust vector so as to accomplish the desired orbit changes, while at the same time keeping solar arrays oriented toward the sun and permitting unbroken communication ability with the earth. It must do this for an extended period of time, in the face of a severely changing spacecraft environment. Because of the extended thrusting time, mission performance is a much stronger function of ACS design and performance than with chemical transfer. Similarly, the ACS design impacts other spacecraft systems, notably the thermal control, communications, and power systems. Because of these factors, it is important to consider the feasibility and desireability of various attitude control system concepts early in the mission planning effort.

This paper presents a discussion of the basic factors which must be considered in the design of an attitude control system for a SEP orbit transfer mission. The basis of the discussion is work done recently at the Lewis Research Center in support of the proposed SERT C spacecraft.(1) For this reason, the sample mission and spacecraft described herein have similar, although not identical, characteristics to SERT C. These are described more fully in the Mission Description and Spacecraft Description Sections. A section is also included which describes the range of environmental conditions encountered by the spacecraft during orbit transfer.

Using the sample mission and spacecraft, this paper does the following: First, it identifies the basic modes of operation required of the ACS; Second, it examines qualitatively the tradeoffs between various ACS components and design options; and third, it identifies the impact of various ACS design options on other spacecraft systems and on mission performance. It is not the purpose of this paper to develop detailed requirements on the ACS, but to describe qualitatively the unique problems engendered by the use of low thrust SEP for orbit transfer. It is felt that the information and discussions presented herein will provide the system engineers with a basis from which they can more readily evolve detailed spacecraft subsystem designs.

Mission Description

General Mission. The basic solar electric geosynchronous transfer mission can be summarized as simply the application of electric propulsion to transfer a spacecraft from a low altitude orbit to a geosynchronous orbit. While the final orbit is well specified; the initial orbit depends on many factors. Spacecraft weight determines the initial orbit semimajor axis, and the launch vehicle determines the initial orbit eccentricity. Generally we may expect a launch from ETR, and thus an initial orbit inclination between 25^{o} and 30^{o}.

For any given initial and final orbits, the transfer may be optimized to minimize the transfer time subject to the spacecraft acceleration capability. The result of the optimizing process is a thrust vector orientation history for the mission. This orientation history then serves as the required guidance reference to which the ACS controls the thrust vector. Generally, a component of acceleration is required in the direction of orbit velocity to increase the semimajor axis. An out-of-plane acceleration component is required to decrease the inclination. During any one orbit this component is a maximum in one sense at the ascending node and maximum in the opposite sense at the descending node, changes in eccentricity are then controlled by the radial component of acceleration. Although radial component phasing is variable during the mission, both radially inward and outward components are required during any one orbit.

Example Mission. The example mission considered is similar to that in Ref. 1. The initial orbit is circular with an altitude of about 0.5 earth radius. The initial inclination is 28.5^{o}.

For this initial orbit, the optimum thrust vectoring requires an approximately sinusoidal out-of-plane acceleration. The radial acceleration is ignored since it is small. The assumptions leading to this optimum thrust vectoring for this mission are that the maximum array power is continuously available, there is no shadowing, and the array power output gradually degrades due to high energy particle flux. As in the reference the magnitude of the out-of-plane acceleration increases during the mission.

Spacecraft Description

General. The physical configuration of a solar electric propulsion spacecraft is largely dictated by the thermal and power subsystems. Generally, the configuration is that of a rigid center body and two flexible solar arrays attached to opposite center body faces. The solar arrays have a single degree of rotational freedom with respect to the center body, and power from the arrays is transmitted to the center body through slipring assemblies. For geosynchronous orbit operations (earth pointing) the axis of array rotation would be north-south, and the north and south faces of the center body would be used for thermal control. The 30 cm thrusters used for prime propulsion must be mounted to the center body so that thruster plume impingement on the solar arrays is avoided.

There are several configurations of thruster mountings which may be considered. The basic difference in configurations is whether each thruster is mounted and gimballed separately or whether the thrusters are grouped and mounted on a gimballed platform. From the attitude control point of view the platform mounting is less attractive for the following reasons: (1) if the prime propulsion system is to be used for attitude control, at least two platforms are required for production of control torques about all three spacecraft axis, and (2) for any two thruster fixed to a platform the thrust vectors may be slightly skewed thus creating a disturbance torque on the spacecraft which cannot be eliminated by platform gimballing. If each of the thrusters is individually gimballed, then the thrusters may be arranged in a line ("1 by" configuration), or in pairs ("2 by" configuration), etc. Either the line or pair configuration is favored since thermal control of the thrusters and power processors for these configurations is more easily accomplished than for other configurations. In addition, the line or pair configuration lends itself to modular design. Conceivably each module contains thruster, gimbals, power processor, thermal control and common propellant plumbing.(2) The modules would form the basic structure of the spacecraft.

Example Sapcecraft. The example spacecraft configuration is shown in Fig. 1 and is basically similar to that used in other studies (1,3,4,5,6). The center body dimensions are 1.22- by 1.22- by 3.25 m, and it houses most spacecraft subsystems. A payload section is provided on the spacecraft + x axis, and the four 30 cm prime propulsion thrusters are mounted on the spacecraft - x face. Each thruster is two-axis gimballed, and the "2 by" configuration was chosen to minimize the required gimbal angles. For the "2 by" configuration total gimbal motion to point each thruster from parallel with the x axis through the center of mass (located about 1.8 m from the - x face) is about 13°. If an in-line configuration is chosen, the required gimbal motion for out-board thrusters is about 27°.

Power is provided by the two solar arrays mounted through the + y and - y faces of the spacecraft. The arrays are mounted as near the center of mass as possible to minimize solar pressure disturbances. Each array is 3.05 m by 16.75 m and provides about 5.5 kW of power. The solar arrays are the dominating contributors to the spacecraft moments of inertia. The inertias are about 22 175, 875 and 22 850 kg m^2 in the x, y and z axes for a total spacecraft mass of 1000 kg. The mass of the deployed arrays is 200 kg.

The nominal acceleration is along the spacecraft + x axis for all four thrusters or for diagnally opposite (symmetric) thrusters operating. To produce the out-of-plane accelerations required for our example mission, the entire spacecraft is rotated as opposed to gimballing the thrusters. To define the spacecraft rotational motions we first establish a reference axis system shown in Fig. 2(a). The X_I, Y_I, and Z_I system is the earth centered inertial system with X_I along the vernal equinox. The X_R, Y_R, and Z_R reference system is a satellite centered system with Z_R toward earth center, Y_R perpendicular to the orbit plane, and X_R perpendicular to the radius vector and nominally along the orbit velocity vector. When the yaw, pitch and roll angles are all zero the spacecraft axes (fig. 1) coincide with the reference axes. Any spacecraft orientation may be achieved as shown in Fig. 2(b) by first yawing the spacecraft, then pitching the spacecraft, then rolling the spacecraft. For symmetric operating thrusters, the out-of-plane acceleration is achieved by simply yawing the spacecraft in approximately sinusoidal fashion over one orbit. If non-symmetric or an odd number of thrusters is operating, the spacecraft yaw and pitch angles must be biased to accelerate the spacecraft in the proper direction with respect to the reference axes. The maximum acceleration is achieved when all thrust vectors are parallel and the net thrust is through the center of mass.

Disturbances Torques

The disturbance torques acting on the spacecraft during orbit transfer may be separated into two categories. The first category includes the environmental torques; the second category includes disturbances produced by the spacecraft subsystems.

Environmental. The major environmental disturbances for the transfer mission are gravity-gradient torque, magnetic torques and solar pressure torques. Table I shows the maximum values of these disturbances for low altitude and synchronous altitude. The low altitude gravity-gradient torque is the maximum environmental torque and will be evident primarily about the roll axis for our example mission and spacecraft. The magnetic torque was estimated for a net spacecraft dipole of 10 A m^2. The estimated disturbance due to solar pressure is based on a 0.1 m separation of the spacecraft center of pressure and center of mass. Aerodynamic drag was found to be negligable at the initial altitude. The high altitude values of gravity-gradient and magnetic torques are two orders of magnitude less than the initial values. The solar pressure disturbance is essentially independant of altitude.

Spacecraft Induced Torques. Disturbances induced by the spacecraft subsystems are highly complex and deserving of detailed examination that is beyond the scope of this paper. The effects of solar array flexibility have been studied for linearized cases, and the reader is referred to the Refs. 3,4,5,7,8,8, and 10. The essential problem concerning flexibility is that the solar array flexible motions induce small motions of the center body. These motions, while not unstable in themselves, may be sensed and magnified by the ACS, resulting in a deterioration in performance or instability. Another potential source of disturbance is the mercury propellant management system. As the propellent is used we can expect a very slow shift in spacecraft center of mass. In addition, higher frequency center of mass shifts and inertial torques can be present due to propellent slosh. Thruster gimballing may also produce center of mass shifts and inertial torque. The thruster motion inertial torque can lead to the familiar "tail wags dog" effect. The critical frequency (that frequency at which the inertial torque equals the torque produced by thrust misalignment from the center of mass) has been estimated for our example spacecraft at 0.155 rad/sec for a single operating thruster. Any static thruster misalignment from the center of mass will also produce a disturbance torque. A single thrust vector misaligned by 0.1^o can produce a disturbance torque of 4.3×10^{-4} Nm in the yaw-pitch plane. This is equivalent to an error of about 0.3 cm in the knowledge of center of mass location. A disturbance torque about the roll axis of about 1.4×10^{-4} Nm is possible for a differential misalignment of 0.1^o for two thrust vectors. Torques produced by thrust vector misalignments can be the dominating disturbance at high altitudes.

Modes of Operation

The purpose of this section is to identify the probable modes of operation for our example mission. Table II presents a summary of this section.

Aquisition. Since we have not specified the booster, the rate and orientation initial conditions from which the spacecraft must acquire are considered random. The requirement on the ACS during this mode of operation can then be listed as: (1) nulling spacecraft rates, (2) acquiring the sun, (3) insuring orderly deployment of the solar arrays, and (4) acquiring the second position reference (earth or star). Functions 1, 2, and 3 must be accomplished within a fixed time limit, since batteries will probably be used for power until the arrays are deployed. For this mode the ACS requires rate sensing and sun sensing. Specific requirements on the rate sensing components cannot be specified until a booster is chosen. Given the random initial orientation, 4π steradian sun sensor coverage will probably be required. The method of control actuation most suitable for this mode is a low impulse auxilary propulsion system. Some argument for using wheels for actuation can be made; however, using wheels alone severely limits the acceptable acquisition initial conditions.

Checkout. A probable mode of operation is a checkout period during which the spacecraft subsystems, excluding thrusters, are exercised. This mode may last from a few hours to a few days and allows establishment of the spacecraft status prior to initiation of prime propulsion. At the simplest level, the ACS requirements during this mode are to maintain yaw, pitch, and roll angles approximately near zero and assure adequate housekeeping power from the solar arrays. Some maneuvers may be required to checkout communication antenna patterns and/or the thermal control system in the presence of high earth albedo. A choice in sensors is not implied by this mode; however, we note that control actuation is not necessarily limited to an auxilary low impulse propulsion system, and wheels may be considered.

Thruster Calibration. Whether or not the 30 cm thrusters are used for attitude control actuation, the thruster start-up mode is envisioned as a calibration of the thruster and gimbal systems. This is necessary due to the potentially large disturbance torques from thruster misalignment and the inability to measure precisely the location of the deployed spacecraft center of mass. Before operation, each thruster would be aligned through the prelaunch calculated location of the center of mass to minimize initial disturbances. Once in operation, each thrust vector would be aligned through the current center of mass. This process would provide information on thruster-thrust vector misalignment, center mass location, gimbal system backlash, etc. before proceeding to vector the spacecraft acceleration. The ACS would be used to aid the calibration, since position, rate and control torque information can be used to determine thrust vector misalignment. Obviously if a short time limit is imposed on the calibration, then sensor accuracy affects the ability to align the thrusters. For our example spacecraft and the disturbances quoted for a 0.1^o thruster misalignment, the pitch motion is 2^o in slightly over 6 minutes. The equivalent torque in yaw produces a 0.1^o motion in slightly over 7 minutes. Thus if position information is used in the calibration, the yaw accuracy should be about 0.1^o. Since the 30 cm thrusters would not be used for control actuation during this mode, the auxilary propulsion system or momentum storage would be used.

Thrust Steering. The thrust steering mode is the major operating mode during orbit transfer. In this mode the thrust vector is pointed according to the guidance equations to raise altitude, and eliminate the eccentricity and inclination. For maximum acceleration the operating thrusters are gimballed so that the thrust vectors are parallel and the net thrust is through the center of mass. For our example mission and spacecraft, the optimum guidance may be approximated by simply yawing the spacecraft in sinusoidal fashion. This, of course, assumes that all four or symmetric pairs of thrusters are operating. The magnitude of the pitch and yaw biases when operating three thrusters is 3.2^o for our example spacecraft.

If continuous maximum power is required, then the ACS must roll the spacecraft so that the pitch axis is perpendicular to the spacecraft-sun line. Figure 3 shows an example of the yaw, roll and

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array rotational motions required as a function of satellite argument in the orbit (see fig. 1). The orbit parameters for Fig. 3 are an ascending mode at 0° and an inclination of 20°. The choice of date locates the sun at the vernal equinox. Figure 4 expands the analysis of the required roll motion for maximum power to include additional sun-orbit relative locations. For all relative positions of the sun and orbit, the roll motion required for maximum power is cyclic and continuous. For large roll motion, gimballed or multiple sensing is required, and conceivably the gimballed prime propulsion thrusters could be used for control actuation. The sensing, actuation and the necessity for maximum power for this mode of operation are subjects for tradeoff studies appearing later in this paper.

Orbit Trim. Another distinct mode of operation may be required when the spacecraft is near synchronous orbit. During this mode, final adjustments to orbit parameters are made and some coasting may be required to achieve the proper longitude in geosynchronous orbit. A reduction in eccentricity without changing the orbit semi-major axis is accomplished by accelerating the spacecraft in the - X_R direction at orbit perigee and accelerating in the $+X_R$ direction at orbit apogee. A reduction in orbit inclination is accomplished by accelerating the spacecraft in the $+Y_R$ direction at the descending mode. No new implications on the ACS configuration are found by considering this mode.

Eclipse. During orbit transfer and geosynchronous orbit the spacecraft will be eclipsed by the earth. Obviously, during this mode the sun is not available for position reference, and the prime propulsion thrusters cannot be used for control actuation. Just prior to intering the eclipse the 30 cm thrusters would be turned off. The ACS, then, is required to maintain guidance pointing when the thrust is terminated and when thrust is initiated after eclipse. Unless all thrusters are turned off and on simultaneously, or unless the thrust is pointed through the center of mass, the spacecraft will be subjected to rather large short term disturbances. For our example spacecraft, the distrubance torque produced by a single thruster, whose thrust vector is parallel to the X axis, is 0.058 N.m. in the Y-Z spacecraft plane.

Momentum Dumping. If momentum storage devices are used for control during any of the above modes, then periodic momentum dumping is required. The ACS must use the auxilary propulsion or prime propulsion systems for this purpose.

Design Option Impact

As noted in the previous sections the geosynchronous transfer mission can be optimized to minimize the transfer time. The optimizing process presently used generally assume that the solar arrays are always pointed directly at the sun so that the maximum power is available. If the solar arrays are misaligned from the sun the thrusters must be throttled thus reducing the spacecraft acceleration and requiring more time to perform the orbit transfer. Yet, orienting the spacecraft to achieve maximum power produces penalties in terms of net payload mass and ACS complexity. Given the larger roll angles for our example mission shown in figs. 3 and 4, one can safely argue that the sensing system is more complex and the control energy expended is greater than if roll motion is limited to small angles or even zero. Thus, in subsequent discussions on the ACS sensing and actuation subsystems we will consider both unlimited and limited roll motion as viable design options for the thrust steering mode.

Obviously, the guidance and navigation analysis is more complex if the roll motion is restricted. In addition to shadowing and solar degradation, the effects of throttling the thrusters to match the large power variations over one orbit must now be included in the analysis. The effects of both design options on other spacecraft subsystems is discussed below for out example mission.

Thermal. If roll motion is unrestricted during the thrust steering mode, then the +Y and -Y faces used for thermal control will never see direct sunlight. However, the control faces will see large earth albedo effects at the lower altitudes. If roll motion is restricted to small angles, then the problem of earth albedo is eliminated, but some direct sunlight will be seen by the control faces. Note that some direct sunlight will be seen by the control faces during earth pointy geosynchronous orbit operations, and some direct sunlight may be seen during thrust calibration and orbit train modes. Thus, the thermal control faces will probably be designed to accommodate some direct sunlight, and it remains to be determined if a mass or performance penalty must be assessed against the thermal system for restricted roll motion during the thrust steering mode.

Communications. If roll motion is unrestricted, the communications antenna system must have acceptable gain pattern nulls in the pitch-yaw spacecraft plane. If roll motion is restricted, directional, higher gain antennas can be used since the spacecraft yaw axis is earth oriented.

Power and Thruster Systems. If roll motion is unrestricted, the thrusters would be throttled slowly to match the degradation in array power due to high energy particle flux. Under this condition, the power system could probably monitor or calculate the array peak power easily. When roll motion is restricted, rather fast variations occur in available array power. Care must be taken to avoid electrical collapsing of the array by demanding too much power for thrusters. The power system must then be able to calculate, measure, or predict power during any one orbit, and the thrusters must be throttled accordingly. For the mission analyzed in Ref. 1, the power variation over the orbit which required the widest range of thruster throttling is shown in Fig. 5, curve I. We can expect similar variations in our example mission. The power ratio is just the cosine of the array misalignment from the sun, and it varies from about .71 to 1.0 with four distinct throttling areas over this orbit. The average power is about 90%, and thus the average acceleration is 90% of the maximum available. Causes II and III show that some improvement in throttling range and average power can be achieved if small roll motions are allowed. Curve II is for roll angles less than 10° and shows the throttling range to be about .77 to 1.0 with a definite "flattening"of two of the throttling areas. Curve III is for roll angles less than 20° and shows the throttling range to be about .83 to 1.0 with a complete elimination of two throttling areas.

Summary. In summary, we may say that for our example mission restricting roll motion simplifies the communications and attitude control systems, while unlimited roll motion simplifies mission analysis and the power system. It has yet to be determined how the thermal system is affected.

Actuation Tradeoffs

The purpose of this section is to examine in a qualitative manner the tradeoffs which may be made in the selection of the prime actuators for our example mission and in particular the thrust steering modes. The discussion is limited to the following items. (1) gimbaled 30 cm thrusters, (2) hot gas auxiliary propulsion system, (3) momentum wheels, and (4) small ion thrusters (8 cm, 5 cm). Items 1, 2, and 3 have already been mentioned in the discussions on probable modes of operation. The small ion thrusters are considered only because they may be required for station keeping in geosynchronous orbit.

During the thrust steering mode, the two basic ACS design options are either unlimited or limited roll motion. To assess the required control capability, a simple, single-axis rigid body analysis is made for a 1.5 R_E orbit radius. We assume that the commanded roll angle is a sinusoid at orbit frequency, and that gravity-gradient is the only external disturbance. A block diagram of the system is shown in fig. 6. The controller is a simple lead-lag system with an overall gain of 1.76 Nm/rad. This gain was chosen from a linear analysis of the closed-loop system for a 2% error in step response.

The results of this study are summarized in fig. 7. The figure shows the maximum torque, average torque, and maximum momentum storage required for control as a function of roll angle.

Unlimited Roll Motion. For unlimited roll motion the actuators must be capable of producing the maximum required torque of .027 Nm. This should be well within the range of momentum wheels and the auxiliary propulsion system. Small ion thrusters mounted on the ends of the arrays could also produce the maximum torque if they are gimballed. A gimbal angle of 20° is required for one thruster or about 10° using two thrusters. Using two 30 cm thrusters the differential gimballing required is about 20° and for four thrusters about 10°. Thus all considered actuators can produce the maximum required torque.

The average torque becomes important when considering the use of the gimballed 30 cm thrusters. To produce the maximum average of $1.9\text{x}10^{-2}$ Nm, requires 13.5° differential gimballing of two thrusters or 6.75° gimballing of all four thrusters. Thus for two thrusters there is a decrease in the net acceleration, due to gimballing of 3%, but for four thrusters the decrease is about 1%.

The maximum momentum storage requirement is important if wheels are considered for control. To store 70 Nm sec could conceivably require a wheel mass of 47 Kg. The estimate is based on extropolation of current hardware weights or the use of multiple wheels. We may also use the maximum momentum to estimate the auxiliary propulsion system propelent usage. Since the momentum build up occurs over half an orbit, the total momentum required per orbit is 140 Nm sec. Assuming a system specific impulse of 200 sec, the propellent used for our example spacecraft is .114 kg per orbit.

Limited Roll Motion. As can be seen from fig. 7 the control requirements are drastically reduced as the commanded roll angle magnitude decreases. If we assume that the roll motion is limited to about .3 radians ($\sim 18^{\circ}$), then the maximum torque is about $1.2\text{x}10^{-2}$ Nm, the maximum average torque is about $9.0\text{x}10^{-3}$ Nm and the maximum momentum is 30 Nm sec. In the same manner as above, we find that the gimballing required to produce the maximum torque is 8.5° for a single small thruster or two 30 cm thrusters and 4.25° for two small thrusters or four 30 cm thrusters. The 30 cm thruster gimballing to produce the average torque is about 6.5° for two thrusters and 3.25° for four thrusters. The attendant loss in average net acceleration is about 0.6% for two thrusters and about 0.2% for four thrusters. Considering the momentum, the required wheel mass is about 20 kg and the propellant usage is about .05 kg per orbit. Table III summarizes the above.

Tradeoffs. If the tradeoff criterion is the usable payload delivered to synchronous orbit in a fixed time, then based on the above discussions a "worst case" tradeoff can be made for the unlimited roll ACS design option. As a basis, consider the optimum mission with full power, maximum acceleration and no required control energy. Thus the mass used for control will be considered a penalty on the payload delivered in optimum fashion. The use of the 30 cm thrusters for control reduces the maximum available thrust by 1% to 3%. Thus to deliver the payload in the same fixed time required a comparable reduction in total spacecraft mass (to maintain maximum acceleration). For our example spacecraft with a base mass of 1000 kg, the corresponding penalty would be 10 to 30 kg. If wheels, small ion thrusters on the auxiliary propulsion system were used for control, then there would be no reduction in total spacecraft mass. However, a net reduction in usable payload (payload penalty) would occur due to the actuation mass. The penalty for the use of wheels is about 47 kg. The penalty for using the auxiliary propulsion system can be estimated by multiplying the propellant usage per orbit by the number of orbits during the thrust steering mode. Our "worst case" estimate for 400 orbits is 46 kg. Because the small ion thrusters were presumed available for synchronous orbit operations, the penalty for the use of these as actuators is quite small. The penalty is the sum of the extra propellant and additional array required for operation of the thrusters during the thrust steering mode. The extra propellant mass is 6.5 kg for 200 days operation, and the extra array mass is 6.5 kg for 300 watts power for two small thrusters. Thus the total payload penalty could be about 13 kg. If the small ion thrusters are not already part of the spacecraft, then this penalty increases to 30 kg with the inclusion of 17 kg for hardware (thrusters, tankage, gimbals, etc).

The payload penalties given above show that the 30 cm thrusters or the small ion thrusters are the best solutions for unlimited roll control. However, we must note that the 30 cm thruster and auxiliary propulsion system penalties are "worst case", since they were estimated by using a low altitude orbit to determine control requirements. The true penalty for using these systems would be less, since the gravity gradient disturbance decreases as the cube of the orbit radius ($1/R^3$). Thus, for a true comparison the integrated control requirements must be found over the duration of the thrust steering mode.

A complete comparison would also consider the complexity of the actuation system. For example, one would attempt to weigh the complexity of gimballing small ion thrusters mounted at the ends of rotating flexible arrays as compared to the straightforward application of wheels of the auxiliary propulsion system. In addition, one would consider the limited and unlimited roll control methods. For the restricted roll design the payload penalty due to actuation drops significantly. The "worst case" penalty for use of the 30 cm thrusters is 2-6 kg. The estimated wheel mass is 20 kg, and the worst case auxiliary propulsion system propellant is 20 kg. The penalty for using the small ion thrusters does not change, since control torque are generated by gimballing the continuously operating thrusters. Thus, the propellent, array and hardware mass penalties for using the small ion thrusters are the same for limited and unlimited roll control.

The prime payload penalty for the restricted roll design is due to the reduction in power for the misalignment of the array from the sun line. In the example given in the previous section (roll less than 20°) the reduction in average power over that orbit was 4%. As above, we can compare this to a 40 kg reduction in payload, if this is the average power for the mission. Obviously, the power reduction for the entire thrust steering mode must be determined for an accurate comparison of design options.

Examining actuation alone does not provide a complete basis for comparing the limited and unlimited design options. We will examine sensing requirements in the next section.

Sensing Tradeoffs

The purpose of this section is to examine in a qualitative manner the tradeoffs which may be made in selecting the prime sensing system for our example mission. In particular, we will deal with the thrust steering mode of operation.

For three axis stabilization of any spacecraft, at least two non-parallel reference vectors must be known by the control system. In discussions on modes of operation, we have indicated that one such reference would be provided by sun sensors. For acquisition the sun sensors would be mounted on the centerbody. Once the arrays are deployed, array mounted sun sensors can provide reference information. Assuming that the sun sensors provide one reference vector, then the second reference can be provided by earth sensing, star sensing, or inertial reference system. These three sensing methods or suitable combinations of them will be discussed for the limited and unlimited roll ACS design options. As in the previous section payload penalties for the methods of sensing will be determined from the sensing system masses.

Limited Roll Motion. If roll motion is limited, a natural mounting location for an earth sensor is on the +Z face of the spacecraft centerbody. In this location, the sensor provides roll and pitch information, and it is not affected by the spacecraft yaw motions required during the thrust steering mode. If the linear range of the sensor is small, or a function of the apparent earth size, some small angle gimballing of the sensor may be required to permit spacecraft roll angles up to 20°. The chief disadvantage in using the earth sensor in conjunction with sun sensors is that during equinox periods the earth vector and sun vector are nearly paralled twice each orbit. Attitude reference information during these periods is not complete, and therefore additional reference must be provided. This additional reference can be provided for the least mass by using a single axis inertial reference gyro. The mass of the total earth sensing system is estimated at about 20 kg. This total assumes that the sensor mass is about 14 kg, that the inertial component is about 3kg, and that the sensor gimballing requires 3 kg. The sensor mass is probably the largest of current hardware. Array mass for this sensor is negligible since the maximum expected power usage is 20 watts.

Unlike the earth sensor, the star sensor presents mounting problems to the system designer. Mounting locations on the +Y or -Y face of the spacecraft are occulted by the array as the array rotates with respect to the centerbody. Other centerbody locations such as the +Z and -Z faces would require more than one sensor for complete coverage. One apparent suitable mounting location is on the tip of a deployed solar array. The sensor must be two-axis gimballed to maintain star contact during the thrust steering maneuvers. A sketch of such a mounting is shown in fig. 8. An additional problem is the choice of suitable reference stars for low altitude orbits. Fig. 9 shows the required star declination to prevent earth occultation as a function of orbit radius and inclination. The problem of earth occultation can be circumvented in two ways. First, an additional inertial reference may be provided during occultation periods; or second, reference stars with large differences in right ascension can be used over different parts of the orbit. The mass weight for the star tracker and gimbal system is estimated at 14 kg. This is based on the ATM star tracker which has similar gimbal requirements.

An inertial reference unit (IRU) has no mounting problems. However, the IRU is generally less accurate than the optical sensors discussed above. This is because the IRU is subject to drift. The IRU must be "updated" or recalibrated periodically using external sensors (sun and earth or star), and thus the IRU at best is no more accurate than the sensors used for calibration. The estimated mass weight of the IRU is 16 kg. This is based on the minimum of representative hardware now in existance, and includes the array mass weight required to generate the IRU power of about 100 watts.

Unrestricted Roll Motion. In considering the large roll angles required for our example mission, it is apparent that mounting change or additional sensors are required for earth or star sensing. To maintain earth contact a single earth sensor would be deployed from the +X face (payload area) of the spacecraft. A 360° gimbal would then be necessary. For complete star tracker coverage, two sensors mounted on opposing arrays (fig. 8) are now necessary. The mass of the earth sensing system for this design option is estimated at 35 kg. The mass of the star sensing system is estimated at 28 kg. No increase in the IRU mass estimate is required.

Tradeoffs. Again we will use the criterion of usable payload delivered to synchronous orbit in a fixed time. The basis is again the optimum mission. Sun sensing is assumed to be part of the existing spacecraft, therefore the payload penalties do not include sun sensing. The payload penalties for using earth sensing are as given above at 20 kg for limited roll motion and 35 kg for unlimited roll motion. The penalty for using star sensing is 14 kg for the limited roll motion design and 28 kg for the unlimited roll motion design. The penalty for using the IRU is greater than the 16 kg mentioned above. We should also include 14 kg for either a star tracker or a body fixed mounted earth sensor to be used with the sun sensors for IRU calibration.

The tradeoff as presented here is not complete in that weighing of system complexity is not made. The earth sensor, for example, supplies roll and pitch information directly for the limited roll motion design. For this case, the star tracker does not supply roll and pitch information directly and in addition presents the problem of having the sensor and actuator separated by a flexible array. Due to its mounting location the star tracker will sense not only the rigid body motions but the flexible motions of the array as well. Thus, care must be taken in processing the star tracker data to avoid undesirable or unstable ACS response. A weighing of such factors as these may be largely judgemental in nature and is not made here.

Concluding Remarks

We have examined separately the payload penalties due to actuation and sensing for limited and unlimited roll motion ACS designs. The actuation penalties, sensing penalties and power penalties for our example mission can be combined so that the limited and unlimited roll motion designs can be compared. The comparison is summarized in Table IV. The first subtable is for the limited roll design option. The largest single penalty is due to the decrease in average power. A 4% reduction in power was assumed from fig. 5, and the resulting payload penalty is 40 kg for our example spacecraft. Actuator penalties are listed down the left side of the subtable, and sensor penalties are listed along the top. The subtable, then, lists the total penalty depending on the choice of sensor and actuator. A similar listing is presented for the unlimited roll design option. Remember that the total penalty is the reduction in useable payload delivered in a fixed time. For our example mission the penalty is caused by the inclusion of sensing other than sun sensing, any reduction in net thrust due to decreased power or 30 cm thruster gimballing, and the control energy which must be expended during the thrust steering mode.

The table shows that minimum penalties are associated with the use of the 30 cm thrusters or the small ion thrusters should they already exist for station keeping purposes. Note again, that the auxiliary propulsion penalty and the 30 cm thruster penalty are worst cases since they were based on low altitude control energy expenditures. The table also shows that minimum penalties are associated with star sensing. This may or may not be surprising since the mass estimate for the earth and star sensing system were the largest for the current hardware, and the mass estimate for the IMU was about the smallest for current hardware. With these estimates, it appears that there is little difference in sensing system choice for unlimited roll motion design, but for the limited roll motion design the earth and star sensing systems clearly cost less in terms of payload penalty. Any reduction in the star sensing or earth sensing systems mass estimates could make these systems more desirable for the unlimited roll design option.

A comparison of the subtables shows that the unlimited roll design option payload penalties are generally less than the penalties for the limited roll design option. Thus, the unlimited roll design should deliver more useable payload to synchronous orbit. The minimum values for each subtable are for star sensing and 30 cm thruster actuation. A comparison of these minimum values shows about a 20 kg advantage for the unlimited roll design option. This is about 2% of the basic spacecraft mass. If we are now allowed to consider the variation in flight time, the 2% payload penalty represents only a 4 to 5 day increase in a 200 day mission for a fixed mass payload. Reductions in estimates of star and earth sensing systems masses would tend to make the unlimited roll design more attractive; however, the payload penalties are not strong functions of the sensing systems masses. If, for example, the star sensing systems mass were ignored, then we would be comparing payload penalties of 42-46 kg for the limited roll design with penalties of 10-30 kg for the unlimited roll design. This represents an ultimate difference of 32 kg or about 3% of the basic spacecraft mass.

We see, then, that a very real tradeoff exists between the ACS design options, the other spacecraft subsystems (thermal, power, etc.), and the mission performance represented by delivered payload. It is apparent that additional effort can be expended to refine the payload penalties for the ACS design options, and determine weighings for attitude control and guidance and navigation systems complexity. Such an effort would produce a realistic spacecraft design for a geosynchronous transfer mission.

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Table I Disturbance torques

Low altitude peak values

Gravity gradient	1.5×10^{-2} Nm
Magnetic	1.0×10^{-4} Nm
Solar pressure	1.5×10^{-4} Nm

High altitude and geosynchronous orbit peak values

Gravity gradient	1.7×10^{-4} Nm
Magnetic	1.0×10^{-6} Nm
Solar pressure	1.5×10^{-4} Nm

Table II

Modes of operation	Comments
Acquisition	Rate gyros, sun sensors, aux. propulsion, earth or star sensor
Checkout	Aux. propulsion, wheels
Thruster calibration	Thruster gimbals, aux. propulsion, wheels
Thrust steering	Yaw-roll motion, actuation and sensing tradeoffs
Orbit trim	180° single axis maneuvers
Eclipse	Aux. propulsion, wheels no sun sensing
Momentum dumping	Aux. propulsion or prime thrusters

Table III

	Unlimited roll	Limited roll
Small ion thrusters gimbal angles (.0044 N)	20°-1 thruster 10°-2 thruster	8.5°-1 thruster 4.25°-2 thruster
30 cm thrusters (.13 N)		
Gimbal angles	20°-2 thruster 10°-4 thruster	8.5°-2 thruster 4.25°-4 thruster
Over reduction in acceleration	3%-2 thruster 1%-4 thruster	0.6%-2 thruster .2%-4 thruster
Wheel (mass)	47 kg	20 kg
Auxiliary propulsion system (ISP = 200 sec) propellent mass	.114 kg/orbit	.05 kg/orbit

Table IV Payload penalties

Limited roll

Power penalty = 40 kg

Actuator	Sensor: Earth 20 kg	Star 14 kg	IMU 30 kg
30 cm thrusters 2-6 kg	62-66 kg	56-60 kg	72-76 kg
Small ion thrusters 13-30 kg	73-90 kg	67-84 kg	83-100 kg
Wheels 20 kg	80 kg	74 kg	90 kg
Aux. propulsion 20 kg	80 kg	74 kg	90 kg

Unlimited roll

Actuator	Sensor: Earth 35 kg	Star 28 kg	IMU 30 kg
30 cm thrusters 10-30 kg	45-65 kg	38-58 kg	40-60 kg
Small ion thrusters 13-30 kg	48-65 kg	41-58 kg	43-60 kg
Wheels 47 kg	82 kg	75 kg	77 kg
Aux. propulsion 46 kg	81 kg	74 kg	76 kg

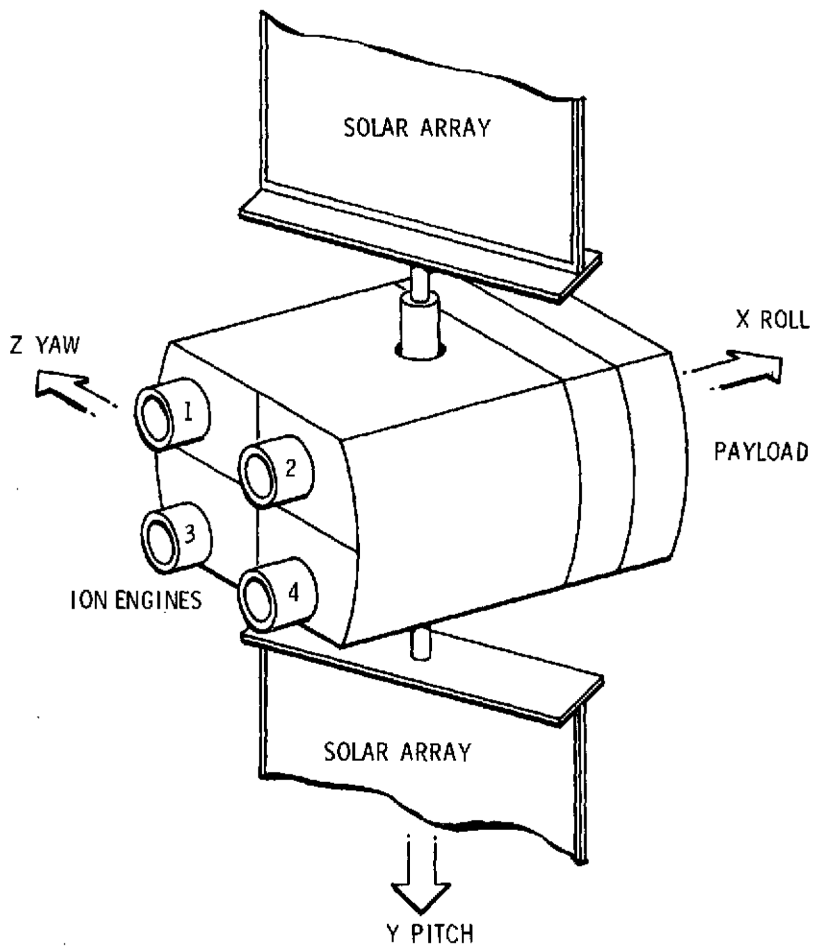

Figure 1. - Spacecraft configuration.

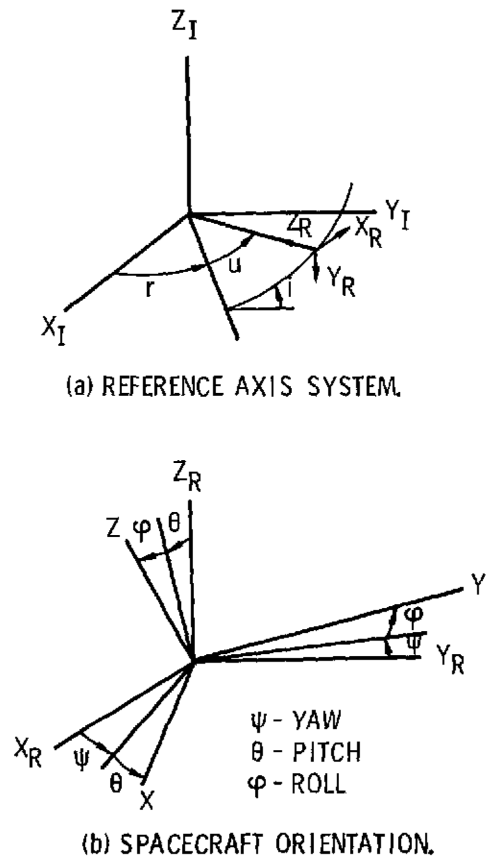

Figure 2.

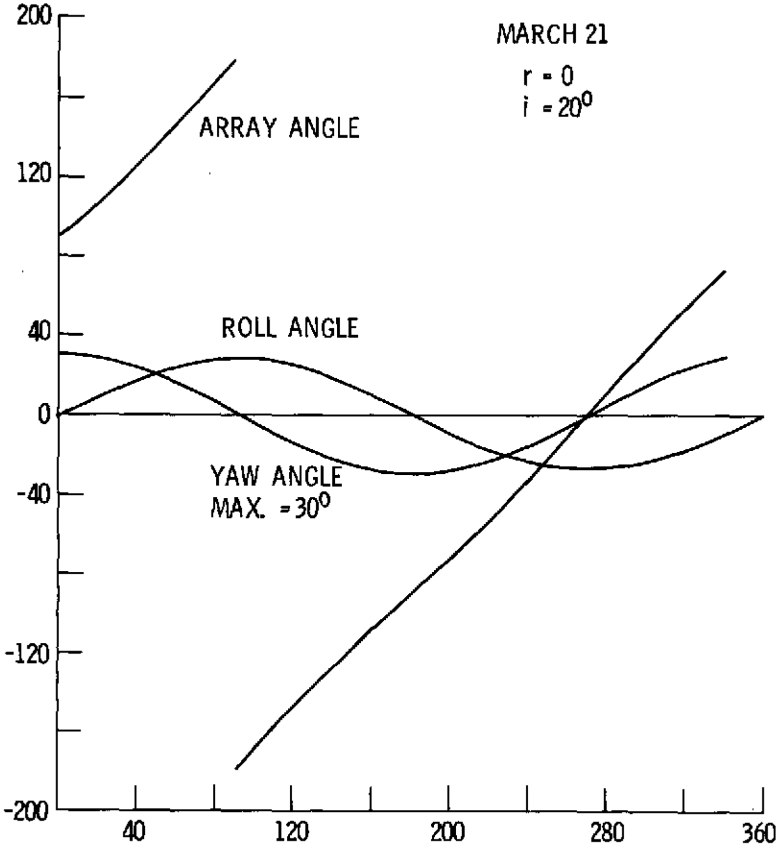

Figure 3.

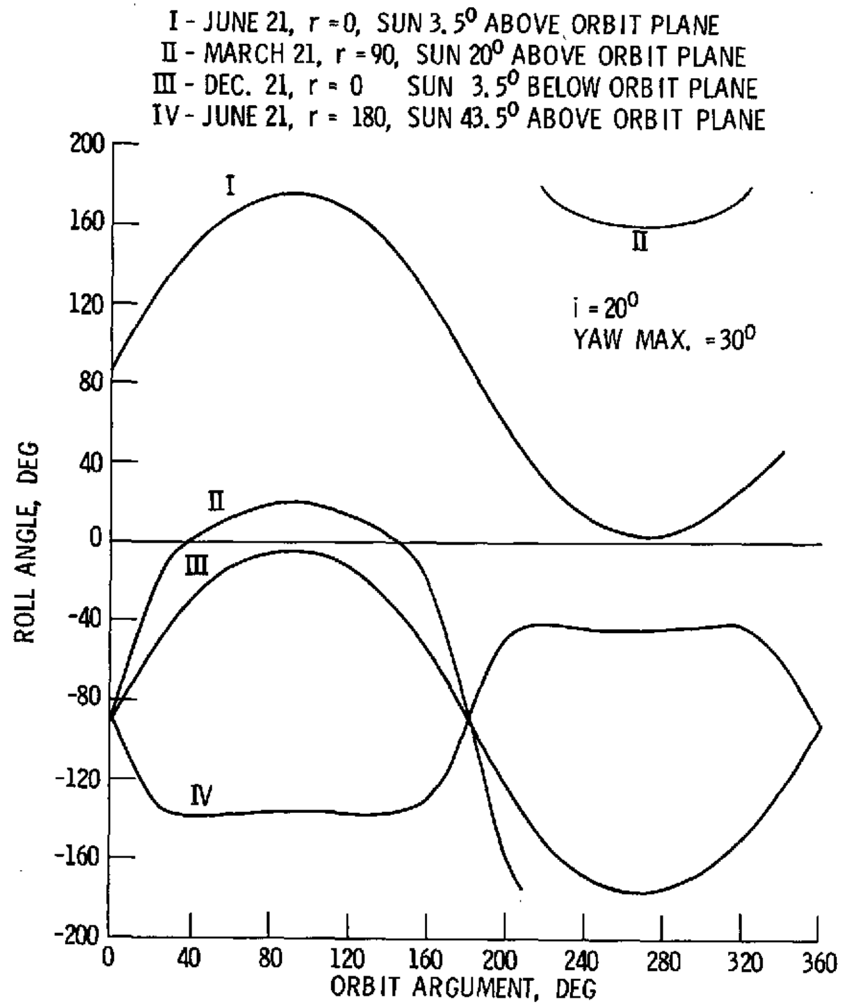

Figure 4. - Roll motion.

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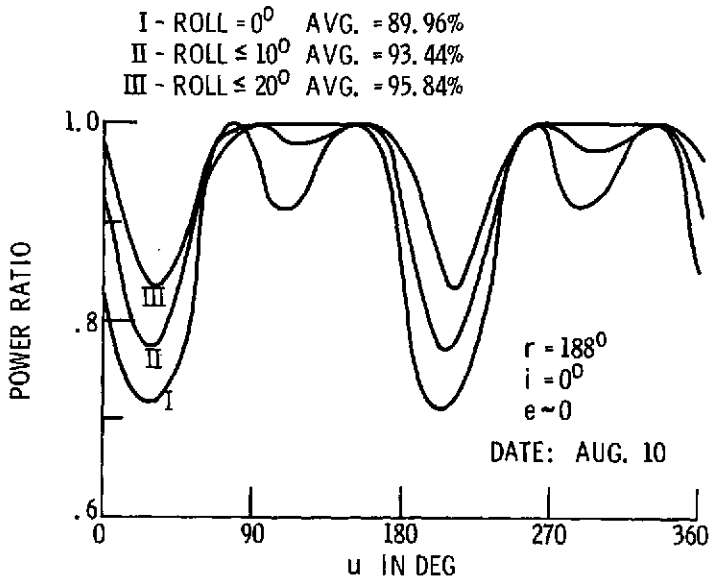

Figure 5. - Power ratio as a function of orbit position and allowed roll angle.

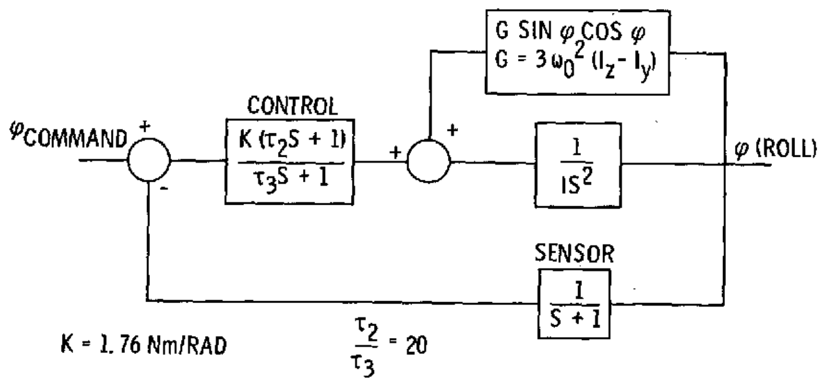

Figure 6. - Simplified roll control.

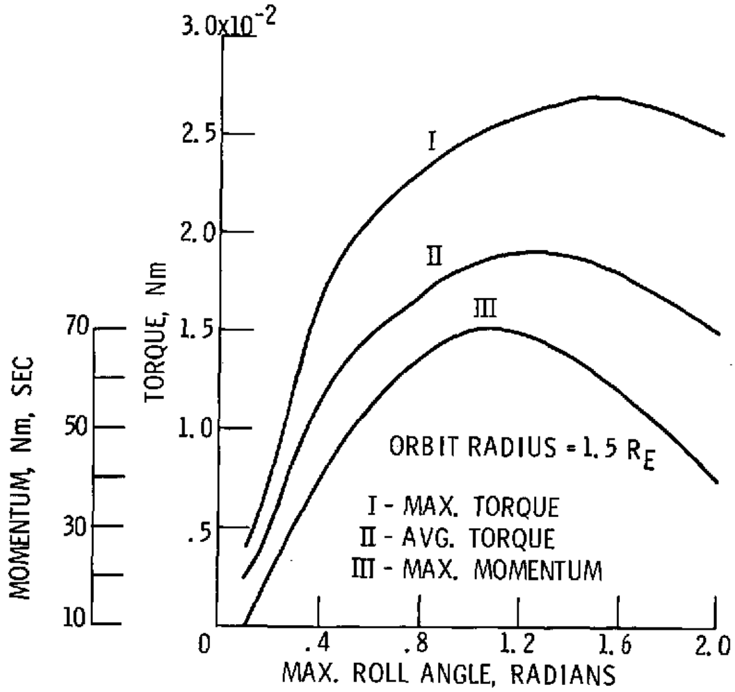

Figure 7. - Torque and momentum requirements for roll control.

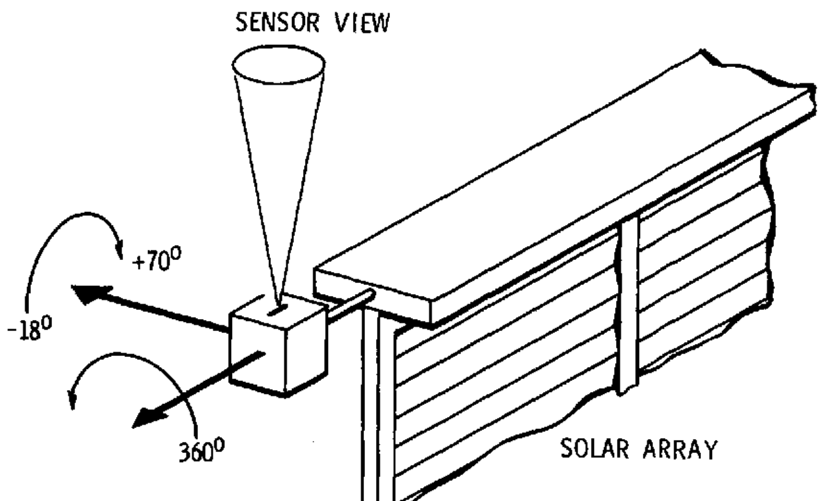

Figure 8. - Array mounted sensor.

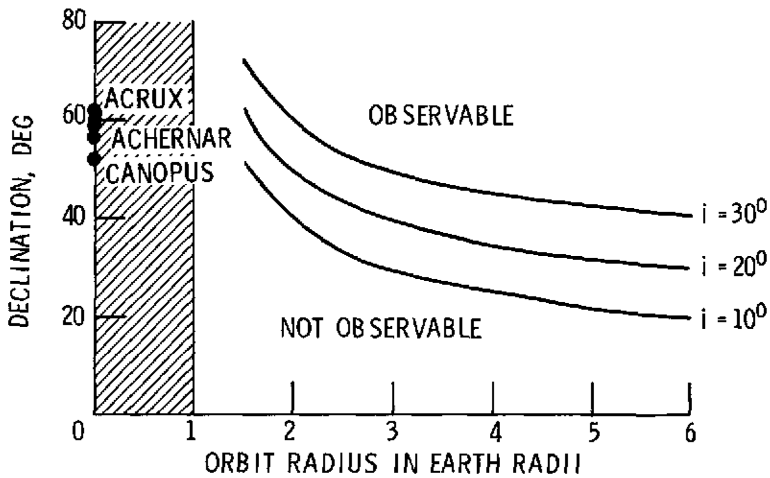

Figure 9. - Required declination for continuous observation of star.

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