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NASA TECHNICAL MEMORANDUM

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(NASA-TM-X-71648) DEVELOPMENT OF A UNIFIED GUIDANCE SYSTEM FOR GEOCENTRIC TRANSFER (NASA) 13 p HC $3.25 CSCL 22A

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DEVELOPMENT OF A UNIFIED GUIDANCE SYSTEM FOR GEOCENTRIC TRANSFER

James E. Cake and John D. Regetz, Jr.
Lewis Research Center
Cleveland, Ohio 44135

TECHNICAL PAPER to be presented at
Eleventh Electric Propulsion Conference sponsored
by American Institute of Aeronautics and Astronautics
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E-8224

Abstract

A method is presented for open loop guidance of a solar electric propulsion spacecraft to geosynchronous orbit. The method consists of determining the thrust vector profiles on the ground with an optimization computer program, and performing updates based on the difference between the actual trajectory and that predicted with a precision simulation computer program. The motivation for performing the guidance analysis during the mission planning phase is discussed, and a spacecraft design option that employs attitude orientation constraints is presented. The improvements required in both the optimization program and simulation program are set forth, together with the efforts to integrate the programs into the ground support software for the guidance system.

I. Introduction

The recent Solar Electric Propulsion mission feasibility and design studies within NASA and industry have stimulated the development of trajectory optimization programs and other mission analysis tools required for SEP mission design. The efforts of the Lewis Research Center have been directed primarily toward the development of the mission analysis tools for geocentric missions, with particular emphasis on bringing existing mission analysis tools to a state of development such that the impact of the Guidance, Navigation, and Control subsystem design upon SEP spacecraft and thruster subsystems may be properly assessed. This impact may be defined in terms of subsystem hardware and operational requirements, relative cost, and reliability.

This paper describes the development of the ground software to effect the open loop cruise guidance of SEP geocentric transfer missions. The software comprises the SECKSPOT computer program, a trajectory optimization program developed by the Charles Stark Draper Laboratory, and the SEOR program, a detailed simulation program developed by the Analytical Mechanics Associates, Incorporated. When integrated, the programs will permit the determination of the impact of the Guidance, Navigation, and Control subsystem upon the SEP spacecraft systems as part of the preflight mission design studies. Several SEP design options are available to provide the attitude sensing and attitude maneuver authority necessary to achieve the required thrust vector directions. An option that imposes attitude and thrust vector orientation constraints as opposed to no constraints is presented together with the advantages and disadvantages from a guidance viewpoint already identified for each option. An overview is presented of the propulsion system model improvements being made to each program, the modifications to study the attitude orientation constraint option in both programs and the method of integrating the two programs.

II. Cruise Guidance Design Motivation

The motivation for the design of the cruise guidance during the pre-project or preliminary mission planning phase is that for geocentric transfer missions this state of development is required to define adequately the requirements of the SEP spacecraft and thruster systems. Some of the requirements to be identified during the guidance system development are those on: thruster throttling range and rate; thruster gimballing; attitude control pointing; attitude sensing; solar array pointing; thermal subsystem; and telemetry and command antennas. This identification is of particular importance when the requirements for a group of geocentric missions are to be integrated into a common set of ion thruster and power processor requirements. An analysis of the cost, reliability, and operational simplicity of the spacecraft systems which meet the requirements, establishes the impact of the guidance system on the spacecraft design. It also provides insight for project management into the tradeoffs of the guidance system performance versus the spacecraft design complexity and program costs, and permits an early establishment of the technology requirements for the ion thruster and attitude control systems.

This approach of developing a guidance and navigation subsystem definition during the preliminary SEP spacecraft design has also been advocated in reference 1. One of the most obvious questions is why must the design and evaluation of the guidance system be undertaken before the subsystems hardware requirements are finalized. The preliminary hardware requirements are typically derived by generating the reference trajectory with a trajectory optimization computer program. The error analyses conducted as part of the guidance system development and evaluation may impose a set of hardware requirements different from those requirements identified by the reference trajectory. In some instances, the changes in the hardware requirements may be small enough to still be accomodated by the baseline design. However, for other cases the final design requirements may be different enough to suggest a different hardware selection. Should the required system prove too operationally complex and expensive, it may be advantageous to compromise or relax the guidance success requirements (transfer time, delivered mass, etc.) and select a simpler and possibly less expensive spacecraft design. Another reason that the reference trajectory may lead to erroneous requirements is that the present optimization program may prove inadequate to generate realistic and accurate reference trajectories from which the hardware requirements may be identified. This is because the optimization programs are rather restrictive in their environmental and propulsion system models, and therefore the optimum solutions and required thrust directions determined from these programs may be different from those obtained if a detailed simulation and accurate representation of the spacecraft and environment were employed. The restrictive nature of these models arises because of the use of these programs for mission analysis and the attendant requirement to generate optimum solutions within a reasonable expenditure of computer time. Consequently, the thruster throttling model, the solar array degradation model, and the solar array power model are usually simplified in the optimization program to provide a tractable two point boundary value problem. Additionally, most optimization programs employ averaging techniques in formulating the state and costate equations. There is therefore a need to verify the performance of the steering law or thrust directions provided by the optimization program with another program that couples a detailed environmental and spacecraft simulation with a precision trajectory generation.

The SECKSPOT computer program(2),(3),(4), developed by the Charles Stark Draper Laboratory and the SEOR computer program(5) developed by the Analytical Mechanics Associates, Incorporated are examples of optimization and simulation programs, respectively, which have been selected to integrate into the ground software for the proposed cruise guidance method. The development of the original versions of both programs was supported by the Goddard Space Flight Center. As the optimization program, SECKSPOT determines the thrust directions required to provide a minimum time trajectory between the initial and final orbits specified as input data. SEOR does not solve a boundary value problem but rather generates a precision trajectory using the thrust direction information provided by SECKSPOT. Therefore, SEOR provides not only a check on the SECKSPOT solution but, as explained below, a means to suggest and develop any required improvements in SECKSPOT's environmental and spacecraft models.

III. Cruise Guidance Approach

The basic approach to the proposed open loop cruise guidance system is shown in Figure 1. Upon injection into the parking orbit by the launch vehicle, the STDN network will track the spacecraft and an orbit determination will be performed. The SECKSPOT program will then be used to target to the final conditions and determine the required thrust vector directions. The thrust directions are first input to the SEOR program in the form of state and costate information. The trajectory predicted by SEOR and in particular, the final orbit conditions will be compared to that obtained from the SECKSPOT solution. If the difference in the final orbit conditions calculated by the programs is unacceptable, some iteration may be required between SECKSPOT and SEOR to arrive at the target conditions with SEOR.

The nature of the interface between the SECKSPOT and SEOR programs, and the information transmitted to the spacecraft is best explained via an example. Consider an orbit raising mission from a low altitude, inclined circular parking orbit to geosynchronous orbit. The initial and final conditions and environmental effects included in the SECKSPOT program run are listed in Table I. The run assumes that the solar array surfaces are maintained normal to the sunline and that the thruster beam power is proportional to the power available from the thruster section of the solar array. The propulsion system parameters and spacecraft operational characteristics employed do not represent hardware requirements or preferred operational characteristics but rather were selected to illustrate the guidance method for a typical mission.

Figure 2 shows the time histories of the semimajor axis, inclination, and eccentricity. The ion thruster shutdown time during periods of earth shadowing causes the eccentricity buildup. For this case, the shutdown period did not include any time delay for restarting the thrusters after the spacecraft emerges from the shadow.

In SECKSPOT, the state vector comprises the five equinoctial orbit elements, mass, and 1 MeV fluence, and the costate comprises the adjoints to this seven element state. Figure 3 shows the time histories of the equinoctial orbit elements and their adjoint variables or costate as computed by SECKSPOT. The significance of this state and costate information is not necessarily in the values themselves, but rather in the slowly varying nature of the data with mission time. This suggests that the state and costate information may be transmitted to the spacecraft computer as coefficients of a function fitted to the data. The state and costate information can be transformed into the control or thrust vector in the equinoctial coordinate system, which in turn can be transformed into in-orbit plane and out-of-orbit plane thrust directions.(2) Figures 4 and 5 show the out-of-orbit plane and in-orbit plane thrust directions, respectively for several orbits during the example mission. The small in-plane thrust component is required to null the eccentricity.

Because the SEOR program requires a priori knowledge of the thrust vector directions, the equinoctial state and costate vectors output from SECKSPOT will be input to SEOR which will transform this information into the required thrust directions as the integration proceeds in SEOR. Several options as to the form of the thrust direction information actually transmitted to the spacecraft will be investigated as the guidance system development proceeds. The options to be studied include state and costate information, in-plane and out-of-plane thrust directions, and precomputed attitude angles as a function of argument of latitude or position in orbit. Transforming the state and costate information into precomputed attitude angles on the ground and transmitting this data to the spacecraft reduces the computational requirements of the onboard computer. An algorithm to compute the argument of latitude is stored in the onboard computer and updated with navigation information from ground tracking, or a limited onboard navigation system might be employed to feed the position in the orbit to the guidance algorithms.

During the orbit raising, navigation information is provided by STDN tracking. When the difference between the actual spacecraft position and that predicted by SEOR exceeds some predetermined level, the mission is reoptimized from the current state to the target. SECKSPOT computes a new set of thrust directions which update the guidance algorithms in the onboard computer. This retargeting may be frequent in the early part of the mission when knowledge of the biases in the thrust subsystem and power subsystems is small. Examples of early propulsion system uncertainties are solar array degradation, individual thruster and power processor performance, and thrust vector misalignment.

The current version of the SECKSPOT program is not capable of targeting on a final position or longitude in orbit. To achieve geosynchronous orbit at a particular longitude, a terminal guidance system must be activated shortly before attainment of geosynchronous altitude.

IV. Attitude Orientation Constraints

The requirement to orient the ion thrust vector in the proper out-of-orbit plane and in-orbit plane directions as illustrated in the example mission has a major impact on the design of the SEP spacecraft attitude control system. The ideal control system, capable of providing these thrust directions and maintaining the solar array surfaces normal to the sunline, would indeed effect the minimum time or optimum trajectory. Those features of the attitude control system which are impacted by this requirement include the attitude sensing and attitude maneuvering capability.

Figure 6 shows a typical SEP spacecraft configuration and coordinate system definition. For this configuration the x axis lies in the orbit plane and has the same sense as the orbit velocity vector; the y axis is perpendicular to the orbit plane and directed south, and the z axis is parallel to the earth radius vector and directed toward the earth. For zero attitude errors, the spacecraft roll, pitch, and yaw axes are aligned with the x, y, and z axes respectively. The ion thrusters are mounted on the negative roll face of the spacecraft and the solar arrays may be rotated about their longitudinal axis which is aligned with the spacecraft pitch axis. The out-of-plane thrust component is provided by rotating the spacecraft in yaw and the in-plane component by rotating in pitch. The solar panels are maintained perpendicular to the sun by rolling the spacecraft until the panel longitudinal axis is perpendicular to the sunline and then rotating the panel normal to the sun. This method requires the use of a star tracker or gimballed earth horizon sensor for attitude sensing and sufficient control torque to provide the required pitch, roll, and yaw motions.

An attitude control design option proposed during the SERT C design study(6) reduces the attitude maneuvers to just yaw motion but provides less than optimum orbit raising performance. As indicated by the example mission, geosynchronous transfers via circular orbits require a relatively large out-of-orbit plane thrust component or yaw motion to effect a reduction in inclination angle and a small in-plane component or pitch motion to reduce the small eccentricity buildup (see Figure 2) caused by earth shadowing. The proposed system employs a non-gimballed, two axis earth horizon sensor for pitch and roll sensing, and a sun sensor-gyro combination to provide yaw sensing. Null operation of the horizon sensor requires that the spacecraft roll-pitch plane be maintained perpendicular to the earth radius vector, and therefore only yaw motion is permitted. Moreover, this yaw motion is unconstrained. One disadvantage of this system lies in its lack of capability to null the residual eccentricity due to shadowing during the orbit raising. Nulling the eccentricity at the end of the transfer increases the transfer time over that obtained when the eccentricity is nulled during the orbit raising. With the thrust acceleration available at the end of the example mission defined in Table I, approximately 6 days would be required to null an eccentricity of 0.05.

The amount of eccentricity buildup due to shadowing may be controlled and the attendant transfer time may be reduced by proper selection of the launch date and time. This is true for both the nonconstrained case and the constrained case defined above. Figure 7 shows the eccentricity buildup for the nonconstrained case at a different launch time. By selecting an initial longitude of the ascending node of -90°, the maximum eccentricity is 0.02 as compared to the eccentricity of 0.05 shown in Figure 2 for an initial node of 0°. As shown in Figure 8, the in-plane steering angle requirements are reduced to less than 3 degrees, compared to the 6 degree requirement shown in Figure 5 for the initial node of 0°. The mission time for this case is reduced to 179 days and the final mass is 720 kg.

The second disadvantage of the attitude constrained system is that because roll motion is not permitted, the solar arrays cannot be maintained normal to the sun line throughout the orbit revolution. The peak value of the solar array normal sun offset is determined not only by the magnitude of the yaw steering angle, but by the orientation of the orbit plane relative to the sunline. Consider the situation where the sun lies in the orbit plane and is perpendicular to the line of nodes as shown in Figure 9. A yaw steering program is employed to simultaneously change the semi-major axis and inclination. For no oblateness and no shadowing, the steering program is approximately a sine wave, with zero yaw angle at the antinode and maximum yaw angle at the node. By constraining the attitude of the center body, the arrays may be directed normal to the sunline at the antinodes, but at the nodes the angle between the array normal and the sun is equal to the magnitude of the yaw steering angle. Figure 10 shows the solar array power variation for that orbit which has the lowest value of power during the example orbit raising mission for some launch date and time. The use of a steering law based on constant power and no shadowing was assumed. During the orbit raising, the ion thrusters would be throttled equally to take advantage of the total array power available. The throttling requirement of 1.4:1 is well within the 2:1 throttling range required for stable operation of the 30 cm thrusters being developed. The preferred method of thruster throttling would be to throttle back on beam current and increase the beam voltage somewhat to maintain the thruster specific impulse near its full power value of 2900 seconds.

The effect of this power variation on the SEP mission performance has been evaluated for geosynchronous missions requiring up to a year of orbit raising time. The steering law employed in the simulation of the constrained case is the optimum law for circle-to-circle transfers between inclined orbits and assumes that the power over the orbit revolution is constant, and that there is no oblateness or shadow effect. It was found that for the constrained case, the total average power was approximately 90 percent of full power. The probable effect on the transfer time for the constrained case would be to increase it by a factor equal to the inverse of the average power ratio over the time for the nonconstrained case. For an average power of 90 percent, the transfer time is increased by 11 percent.

A summary of the preliminary SEP propulsion system and attitude control system requirements is presented in Table II, for the case of a SEP spacecraft having no attitude orientation constraints and for the case with the attitude orientation constraint of maintaining the spacecraft roll-pitch plane perpendicular to the Earth radius vector. The salient advantages of the constrained system over the nonconstrained system are the simplification of the attitude control system and a possible reduction in the thruster gimbal requirement if the thrusters are used to provide the attitude maneuvering. The disadvantage of the constrained case is the requirement of the propulsion system to track the varying solar array power and the attendant requirement on thruster throttling range and rate. Based upon the preliminary evaluation of the SEP performance and operational simplicity of the required attitude control system for the design option employing attitude orientation constraints, a study has been undertaken to determine the effect of attitude constraints on optional geocentric transfers. Modifications are being made to the SECKSPOT computer program so that the computation of the thrust directions is based on an optimization formulation which accounts for the power variations over the orbit revolution caused by constraining the roll-pitch plane to be perpendicular to the Earth radius vector. The transfer times for orbit raising trajectories using the thrust directions computed from this new formulation are expected to be smaller than those obtained with the steering law employed in the simulation discussed above. Therefore the orbit raising performance of the constrained case will be more competitive with that for the nonconstrained case.

An additional improvement to the mission performance of the constrained case would be to allow a limited amount of pitch motion and still retain the non-gimballed horizon sensor. The magnitude of the pitch offset is limited to the field of view of the sensor and the accuracy available with the sensor over the field of view. This pitch freedom would permit small in-plane thrust offsets, such as those exhibited in Figures 5 and 8, to control the eccentricity during the orbit raising rather than waiting to null the eccentricity near geosynchronous orbit.

V. SECKSPOT Program Modifications

Several modifications are currently being made to the SECKSPOT computer program to improve its capability to simulate some aspects of the SEP systems and to add the capability to study the option of attitude orientation constraints. The objective is to add as much detail to SECKSPOT as possible without greatly increasing the program execution time. The ion thruster restart time after shadow is being added to the Earth shadow time to obtain a more realistic shutdown period caused by shadowing. The restart time is modeled as the sum of the time for the solar array to achieve operating temperature and the time for the thrusters to achieve full thrust after the solar array power has been applied to the power processor. Results from array thermal analyses and ion thruster hardware tests have been used to develop the model. A new Earth magnetic field model is being used to generate the solar array degradation model which is being generalized to consider various solar cell shielding thicknesses.(7) Subroutines are being added to calculate parameters that are useful in the spacecraft design. These parameters include in-plane and out-of-plane thrust directions, spacecraft attitude angles to achieve these thrust directions, and the solar array incidence angles on the spacecraft body. The major effort underway is the inclusion in the optimization problem of the attitude orientation constraint whereby the spacecraft roll-pitch plane is maintained perpendicular to the Earth radius vector.(8) The effect of the orientation constraint is being formulated in the equations of state and costate. As indicated previously, the attitude constraint causes the solar array power to vary over the orbit revolution. In SECKSPOT, the thrust will be assumed to be directly proportional to the array power and the specific impulse will be assumed to be constant.

VI. SEOR Program Modifications

The modifications being made to the SEOR program(9) include a detailed thruster configuration and throttling model, provisions for generating thrust vector orientations based on input from SECKSPOT, and addition of the thruster restart time to the shadow model. The thruster system configuration model will permit specifying the number and location of the individual thrusters and the direction of the thrust vector relative to the spacecraft coordinates. As the solar array power degrades due to particulate radiation, the thruster throttling model, based upon 30 cm thruster test data, will compare the thrust of the number of operating thrusters to that for operating one less thruster. If the total thrust can be increased, one of the thrusters will be shut down.

Steering data will be input to SEOR from SECKSPOT in the form of a table or curve of time histories of the state and costate vectors required to provide the minimum time trajectory. SEOR will transform these vectors into a thrust vector in the coordinate frame required by SEOR, and then as an option apply the attitude orientation constraint of maintaining the roll-pitch plane perpendicular to the Earth radius vector. The nature of the interface between the SECKSPOT and the SEOR program is sufficiently general to permit the generation of either circular or elliptic orbit raising trajectory simulations with SEOR.

One of the problems in synthesizing two programs such as these with their different techniques or level of detail in modeling the spacecraft and environmental simulation is that the thrust profiles generated by SECKSPOT when input to SEOR may not produce the same final conditions in SEOR. If the modeling in SEOR is correct, then the integration of these two programs will assist in improving the SECKSPOT models to that state required for the mission operations. One of the more probable areas where some disagreement may arise will be in the solar array degradation model and the solar array power model. Both programs will use the same electron and proton environment models as developed by the National Space Science Data Center. The damage coefficients relating the electron and proton flux to 1 MeV equivalent electron flux are input data to SEOR and to the degradation model computer code which interfaces with SECKSPOT, and can therefore be made the same for both programs. We will decouple the degradation models from the trajectory integration and compare for each program the results of the process of transforming from electron and proton flux to 1 MeV equivalent flux, and from 1 MeV flux to solar array power.

VII. Concluding Remarks

An approach to effect the open loop cruise guidance of a SEP spacecraft in geocentric transfer has been discussed. The guidance system consists of determining the ion thrust vector directions on the ground with an optimization computer program, and transferring this information to the spacecraft computer via ground command. This procedure is repeated when the difference between the trajectory predicted with a precision simulation computer program and the actual trajectory determined from ground tracking exceeds some predetermined level.

A spacecraft design option that employs attitude orientation constraints has been presented. Modifications to the optimization and simulation computer programs are currently underway and include: improvements in the propulsion system simulation; a new formulation in the optimization program to study the attitude orientation constraints; and the integration of the programs to transfer thrust direction information from the optimization program, SECKSPOT, to the simulation program, SEOR.

Upon completion of the integration or unification of these programs, the basic ground software to perform the cruise guidance will be in hand. Simulation of the cruise guidance will permit the refinement of the preliminary propulsion system and spacecraft hardware requirements defined by the optimization program for the design options under consideration. Upon the conclusion of the tradeoff studies the question of how worthwhile and practical is a truly optimal geocentric trajectory as compared to one which is less than optimal may finally be answered from the viewpoint of the spacecraft design.

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TABLE I. EXAMPLE MISSION CHARACTERISTICS

A. Input Data

Initial semi-major axis, a_0, km	9528
Initial eccentricity, e_0	0
Initial inclination, i_0, deg	28.3
Initial argument of perigee, ω_0, deg	0
Initial longitude of ascending node, Ω_0, deg	0
Final semi-major axis, a_F, km	42164
Final eccentricity, e_F	0
Final inclination, i_F, deg	28.3
Final argument of perigee, ω_F	Free
Final longitude of ascending node, Ω_F	Free
Launch date	January 1, 1980
Mass, kg	850
Thruster beam power, kw	4.83*
Specific impulse, sec	2900

Oblateness and shadowing effects included.
Degradation included (6 mil coverglass, infinite backshielding)

B. Results

Transfer time, days	187
Final mass, kg	721
Power ratio	0.70

*Equivalent to 2.6, 30 cm thrusters

TABLE II. SUMMARY OF PRELIMINARY PROPULSION SYSTEM AND ATTITUDE CONTROL REQUIREMENTS FOR GEOSYNCHRONOUS MISSION WITH AND WITHOUT ATTITUDE ORIENTATION CONSTRAINTS

	W/O Constraints	W/Constraints
Attitude Control:		
Attitude sensor	Star tracker or gimballed horizon sensor	Non-gimballed Earth horizon sensor
Control torque	Roll, yaw, and pitch	Yaw only
Propulsion System:		
Solar array power	Constant over revolution	Varies over orbit rev.: power processors must track available power
Thruster throttle effect, range/rate	1. Degradation effect (2:1)/(slow)	1. Degradation effect (2:1)/(slow) 2. Power variation due to attitude constraints (1.4:1)/(1%/min.)
Thruster gimbal	Center of mass alignment and possible control torques about all axes	Center of mass alignment and possible control torques about all axes
Mission Performance:	Optimal	Sub-optimal

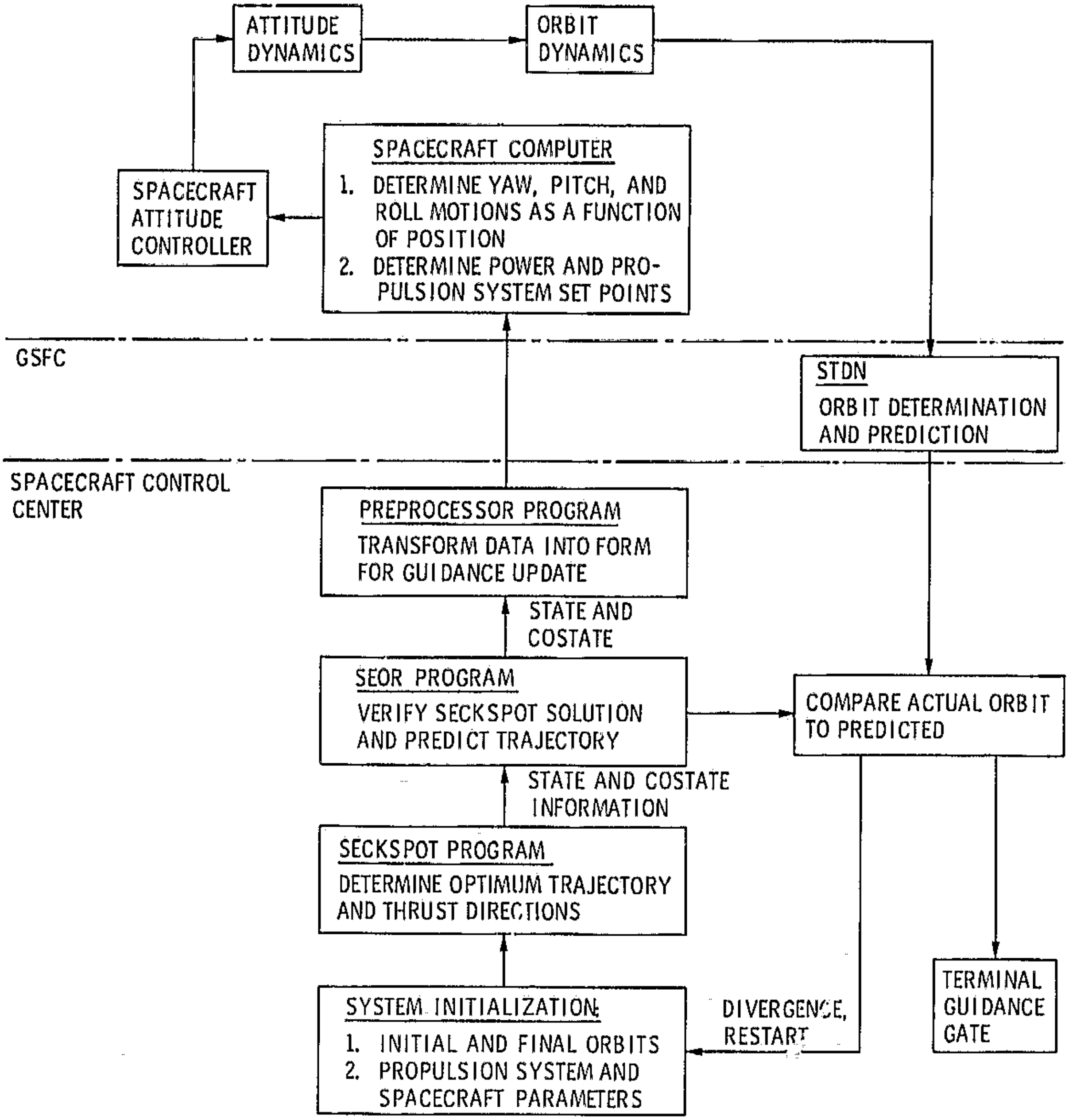

Figure 1. - Cruise Guidance System Block Diagram.

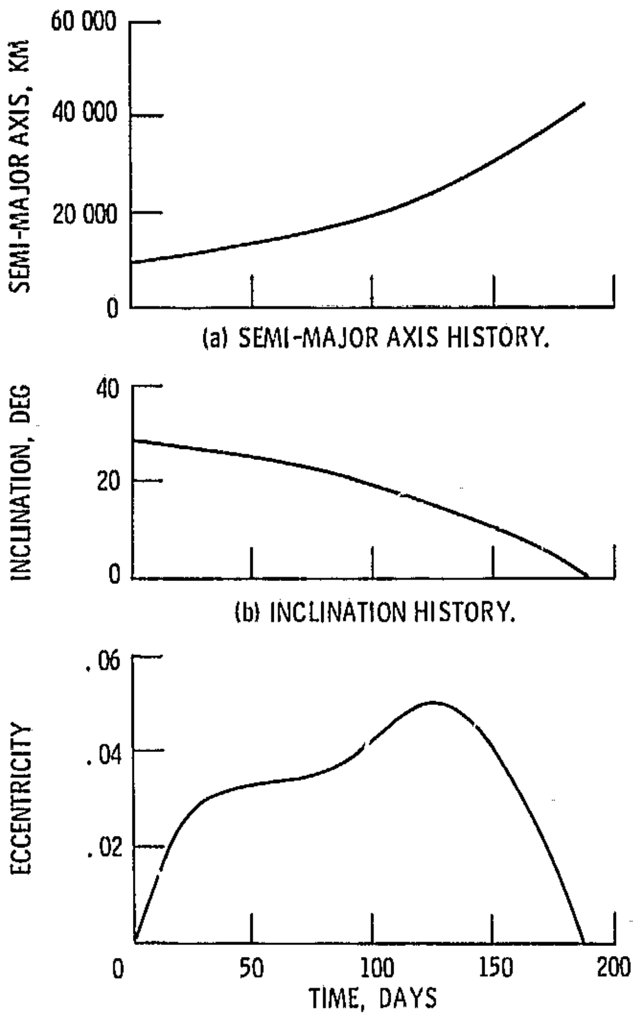

Figure 2. - Time histories of classical orbit elements.

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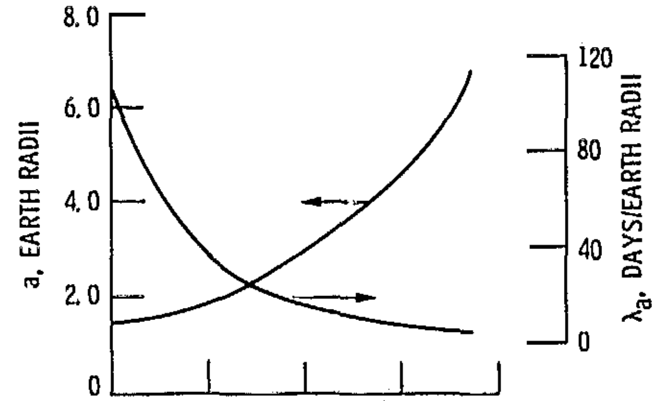
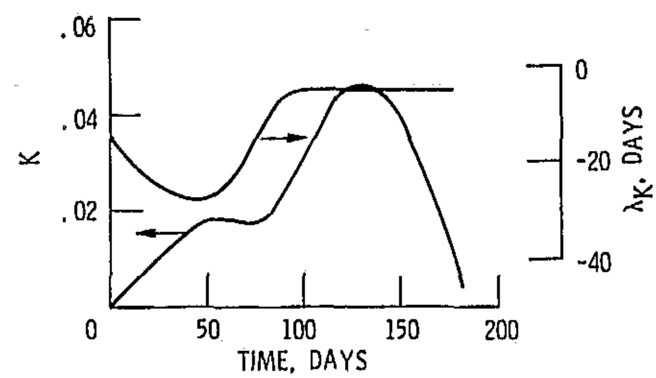


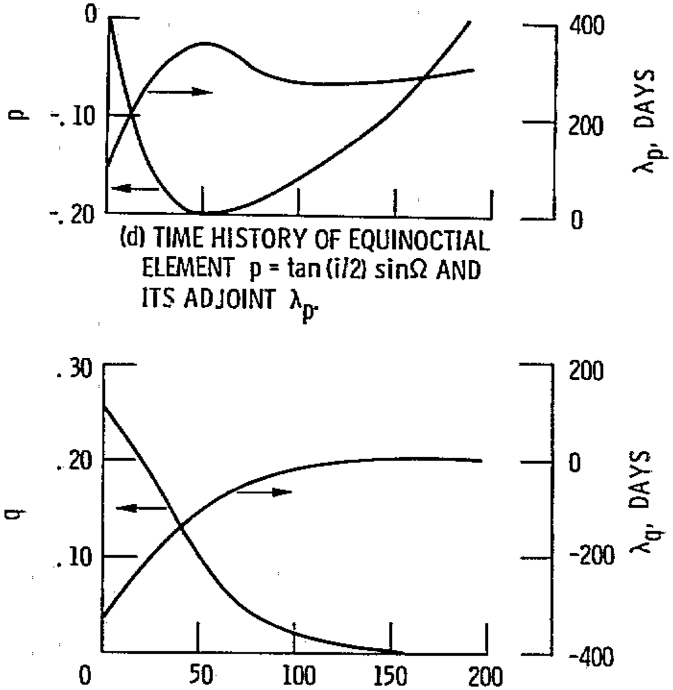
(a) TIME HISTORY OF SEMI-MAJOR AXIS, a, AND ITS ADJOINT, λ_a.

(b) TIME HISTORY OF EQUINOCTIAL ELEMENT, $h = e \sin(\omega + \Omega)$, AND ITS ADJOINT λ_h.



(c) TIME HISTORY OF EQUINOCTIAL ELEMENT $K = e \cos(\omega + \Omega)$ AND ITS ADJOINT λ_K.

Figure 3. - Time histories of equinoctial elements and their adjoints.



(d) TIME HISTORY OF EQUINOCTIAL ELEMENT $p = \tan(i/2) \sin\Omega$ AND ITS ADJOINT λ_p.

(e) TIME HISTORY OF EQUINOCTIAL ELEMENT $q = \tan(i/2) \cos\Omega$ AND ITS ADJOINT λ_q.

Figure 3. - Concluded.

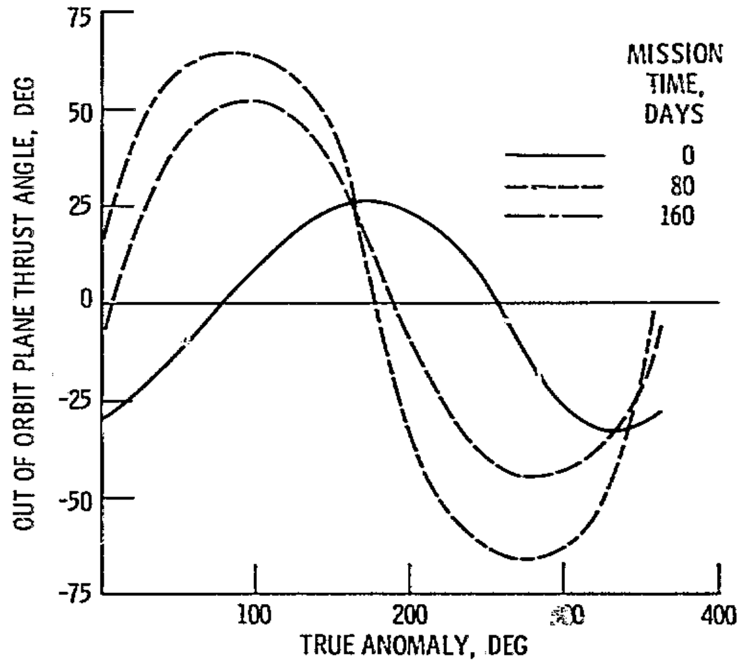


Figure 4. - Out of orbit plane thrust directions.

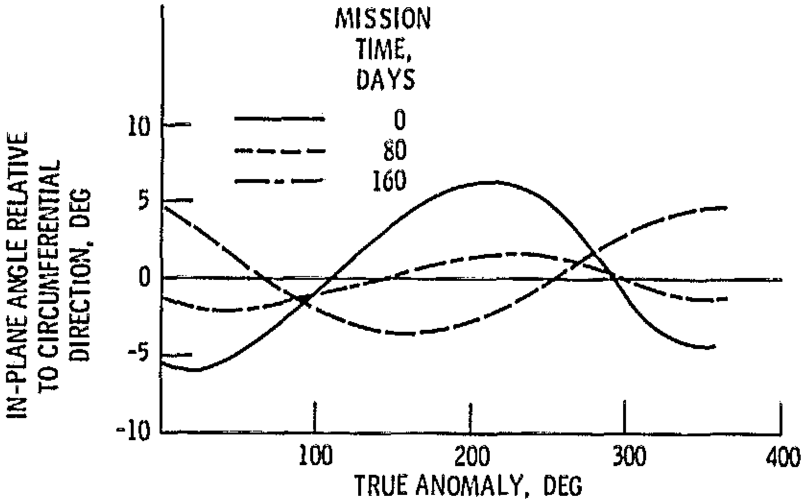


Figure 5. - In orbit plane thrust directions.

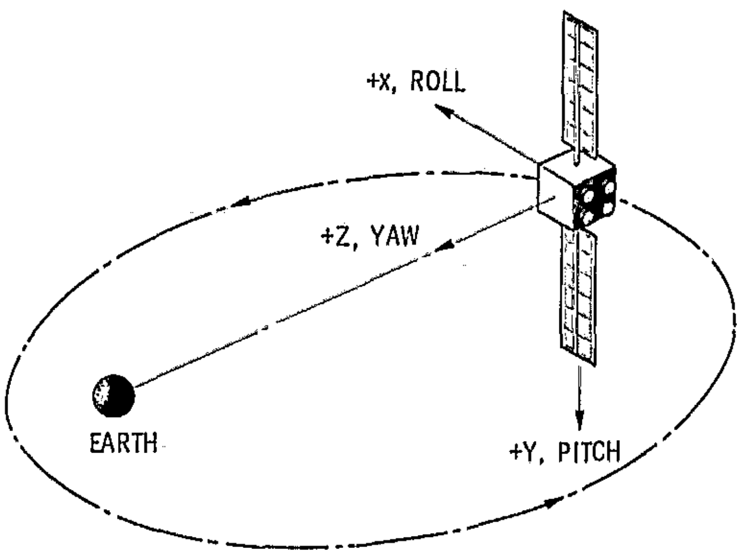


Figure 6. - Spacecraft configuration and coordinate definitions.

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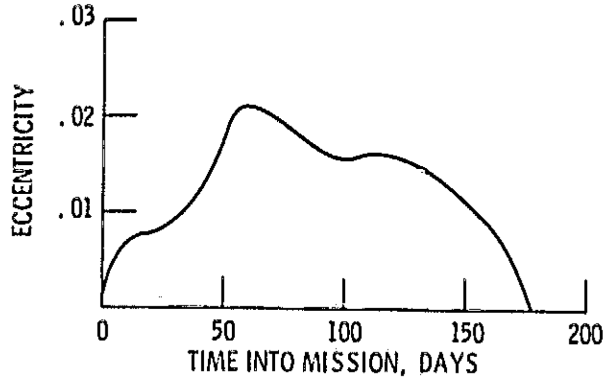

Figure 7. - Eccentricity history when initial node is -90°.

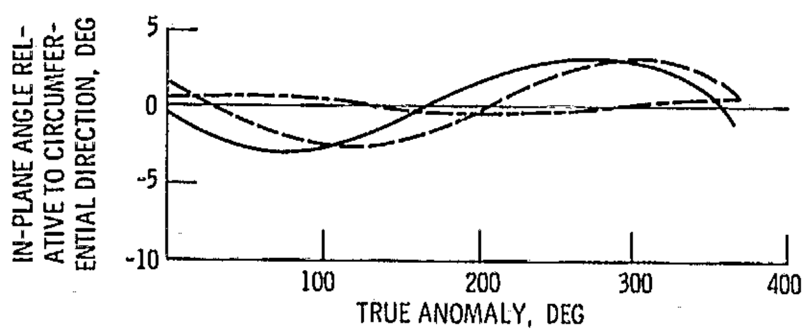

Figure 8. - In orbit plane thrust directions for an initial node of -90°.

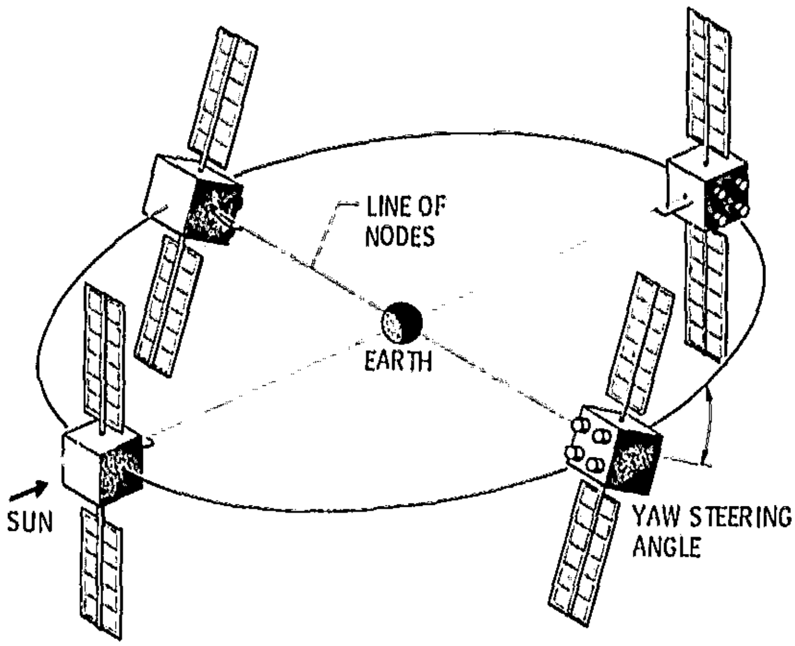

Figure 9. - Orbit-sun spacecraft geometry illustrating solar array power variations.

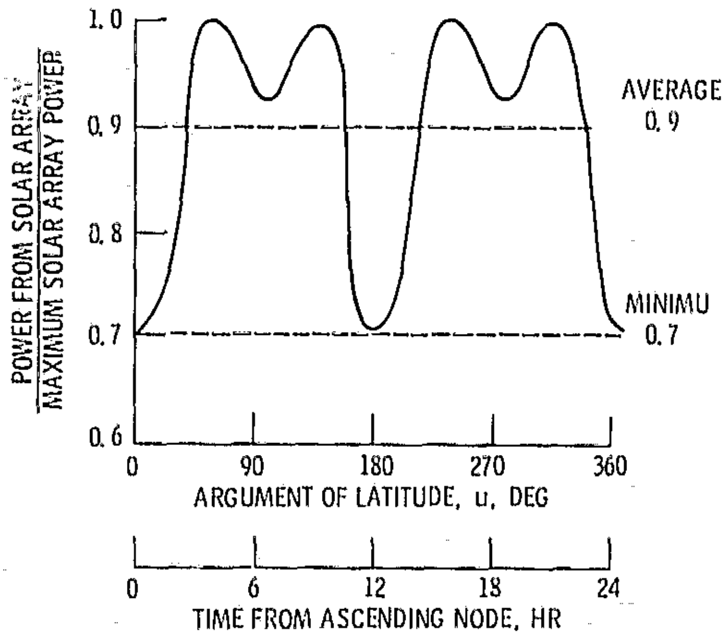

Figure 10. - Variation of solar array power over that orbit which has the lowest value of value of instantaneous power.