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DEVELOPMENT OF A UNIFIED GUIDANCE SYSTEM FOR GEOCENTRIC TRANSFER

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TECHNICAL PAPER to be presented at
Eleventh Electric Propulsion Conference sponsored by American Institute of Aeronautics and Astronautics
New Orleans, Louisiana, March 19-21, 1975
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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 precise 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 upon the 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 SDO 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 progeflight 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.
The run assumes that the solar array surface are maintained normal to the uunline and that the represent hardware requirements or preferred oper-
array. The propulsion system parameters and space-
available from the thruster section of the solar
quired between SECKSPOT and SEOR to arrive at the
of state and custard information. 'Che trajectory
vehicle, the SIGN network will track the spacecraft
injection d,nto the parking orbit by the launch
thruster boom power is proportional to the power
from a low altitude, inclined circular parking
transmitted to the spacecraft is beet explained via
time delay for restarting the thrusters after the space-
craft emerges frum 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 nown 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 vector
as the integration proceeds in SEOR. Several op-
tions 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 and position
in orbit. Transforming the state and costate
information into precomputed attitude angles on
the ground and transmitting this data to the space-
craft reduces the computational requirements of the
onboard computer. An algorithm to compute the
argument of latitude is stored in the onboard com-
puter 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 differ-
ence between the actual spacecraft position and that
predicted by SEOR exceeds some predetermined level,
the mission is reoptimized from the current state
state to the target. SECKSPOT computes a new set of
thrust directions which update the guidance algo-
rithms in the onboard computer. This retargeting
may be frequent in the early part of the mission
when knowledge of the biases in the thrust sub-
system and power subsystem is small. Examples of
early propulsion system uncertainties are solar
array degradation, individual thruster and power
processor performance, and thrust vector misalign-
ment.
The current version of the SEEKSPOT 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 attitude.

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 (5) 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 shown in 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. As such only yaw motion is permitted. Moreover, 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 and reduces the total transfer time. 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 0°, 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 5 degrees, compared to the 6 degree requirement shown in Figure 9 for the initial node of 0°. The mission time for this case is reduced to 170 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 antinodes and maximum yaw angle at the nodes. 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 1.41 is well within the 2:1 throttling range required for stable operation of the 30 cm thrusters being developed. The preferred method of constraining yaw might 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 thrust. 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 1, 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 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 within 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 being forced 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 system 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 its computation 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 analysis 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 takes into account 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. 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 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 mission time trajectory. SEOR will transform those vectors into a thrust vector in the coordinate frame required by SEOR, and the 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 most 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. This 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 affect 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 direction on the ground with an optimization computer program, and transmitting 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 programs 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.

References


TABLE 1. EXAMPLE MISSION CHARACTERISTICS

A. Input Data

<table>
<thead>
<tr>
<th>Parameter</th>
<th>Value</th>
</tr>
</thead>
<tbody>
<tr>
<td>Initial semi-major axis, $a_0$, km</td>
<td>9528</td>
</tr>
<tr>
<td>Initial eccentricity, $e_0$</td>
<td>0</td>
</tr>
<tr>
<td>Initial inclination, $i_0$, deg</td>
<td>28.3</td>
</tr>
<tr>
<td>Initial argument of periapsis, $\omega_0$, deg</td>
<td>0</td>
</tr>
<tr>
<td>Initial longitude of ascending node, $\Omega_0$, deg</td>
<td>0</td>
</tr>
<tr>
<td>Final semi-major axis, $a_F$, km</td>
<td>42164</td>
</tr>
<tr>
<td>Final eccentricity, $e_F$</td>
<td>0</td>
</tr>
<tr>
<td>Final inclination, $i_F$, deg</td>
<td>28.3</td>
</tr>
<tr>
<td>Final argument of periapsis, $\omega_F$</td>
<td>Free</td>
</tr>
<tr>
<td>Final longitude of ascending node, $\Omega_F$</td>
<td>Free</td>
</tr>
<tr>
<td>Launch date</td>
<td>January 1, 1980</td>
</tr>
<tr>
<td>Mass, kg</td>
<td>850</td>
</tr>
<tr>
<td>Thruster beam power, kw</td>
<td>4.87*</td>
</tr>
<tr>
<td>Specific impulse, sec</td>
<td>2900</td>
</tr>
</tbody>
</table>

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

B. Results

<table>
<thead>
<tr>
<th>Parameter</th>
<th>Value</th>
</tr>
</thead>
<tbody>
<tr>
<td>Transfer time, days</td>
<td>187</td>
</tr>
<tr>
<td>Final mass, kg</td>
<td>721</td>
</tr>
<tr>
<td>Power ratio</td>
<td>0.70</td>
</tr>
</tbody>
</table>

*Equivalent to 2.6, 30 cm thrusters

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

<table>
<thead>
<tr>
<th></th>
<th>W/Constraints</th>
<th>W/Constraints</th>
</tr>
</thead>
<tbody>
<tr>
<td>Attitude Control</td>
<td></td>
<td></td>
</tr>
<tr>
<td>Attitude sensor</td>
<td>Star tracker or gimbaled horizon sensor</td>
<td>Non-gimbaled horizon sensor</td>
</tr>
<tr>
<td>Control torque</td>
<td>Roll, yaw, and pitch</td>
<td>Yaw only</td>
</tr>
<tr>
<td>Propulsion System</td>
<td></td>
<td></td>
</tr>
<tr>
<td>Solar array power</td>
<td>Constant over revolution</td>
<td>Varies over orbit rev.: power processors must track available power</td>
</tr>
<tr>
<td>Thruster throttle effect, range/rate</td>
<td>1. Degradation effect (2:1)/(slow)</td>
<td>1. Degradation effect (2:1)/(slow)</td>
</tr>
<tr>
<td></td>
<td>2. Power variation due to attitude constraints (1:4:1)/(15/min.)</td>
<td></td>
</tr>
<tr>
<td>Thruster gimbal</td>
<td>Center of mass alignment and possible control torques about all axes</td>
<td>Center of mass alignment and possible control torques about all axes</td>
</tr>
<tr>
<td>Hipion Performance</td>
<td>Optimal</td>
<td>Sub-optimal</td>
</tr>
</tbody>
</table>
Figure 1. - Cruise Guidance System Block Diagram.
Figure 2. - Time histories of classical orbit elements.
Figure 3. Time histories of equinoctial elements and their adjoints.
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.
Figure 7. - Eccentricity history when initial node is \(-90^\circ\).

Figure 8. - In orbit plane thrust directions for an initial node of \(-90^\circ\).
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.