LOCAL VERTICAL/LOCAL HORIZONTAL ATTITUDE CONTROL FOR SPARTAN SPACECRAFT

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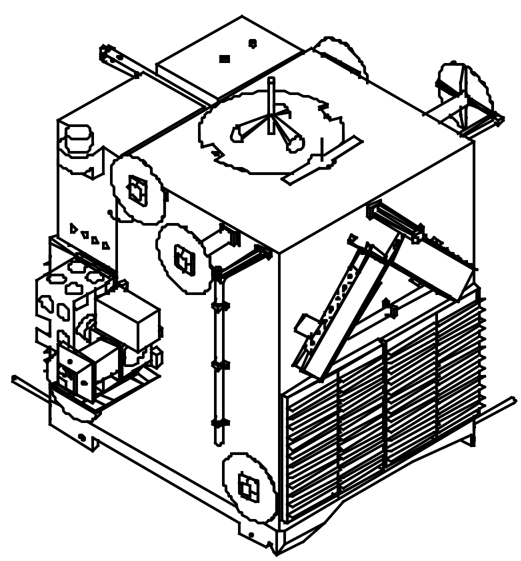
Abstract

A Spartan spacecraft attitude control system was reconfigured to provide attitude pointing with respect to a Local Vertical/Local Horizontal reference frame even though the baseline system uses only an initial start attitude, sun sensors, and star tracker/gyros for defining the spacecraft attitude. No earth sensors of any kind are used. Deployed from the orbiter for two days, usually for solar and stellar inertial pointing, Spartan missions use pointing programs that must be written months prior to launch with bit-level changes possible only in the on-orbit period prior to deployment. With these limitations, Spartan missions 206 and 207, flown in 1996, had requirements to point with respect to the orbital velocity vector, the direction of which in an inertial frame, varies greatly due to launch time, launch date and deploy time uncertainties. The description concentrates on the LVLH to inertial geometry, attitude maneuvering scheme, and timing considerations.

Introduction

The Spartan spacecraft has no numerical computation capability and no RF command link. As a result, all attitude maneuvers must be calculated on the ground and programmed into the attitude control software before launch. In the past, this has not been a problem because Spartan missions have either been solar or inertial stellar pointing and have not required particularly complex sequences of maneuvers. Pointing to the earth or any other direction in a Local Vertical/Local Horizontal (LVLH) frame of reference, however, is not so straightforward because

Figure 1. The Spartan 206 Spacecraft.

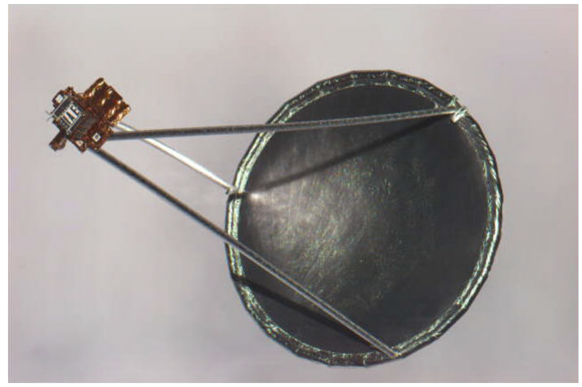


in low earth orbit this frame rotates nominally at 4 degrees/minute with respect to an inertial reference. Furthermore, the orbit plane which always contains the nadir vector continuously changes orientation due to nodal regression. For this reason, on most spacecraft, the simplest LVLH attitude pointing (e.g. nadir pointing) is usually done with the aid of earth horizon sensors. Earth sensing using gyros updated by star trackers must wrestle with time-variable geometry between LVLH and celestial references. The procedure described in this paper was implemented on two separate missions of the Spartan spacecraft (Spartan 206 and 207) in 1996. The procedure was designed to meet LVLH pointing requirements in lieu of redesign of the spacecraft's attitude

Figure 2. Spartan 206 in the Orbiter Payload Bay.

control hardware to accommodate costly earth horizon sensors. The LVLH pointing starts subsequent to deployment at a specified attitude after release from the space shuttle's Remote Manipulator System (RMS). Spartan 206 used this procedure to maintain an LVLH attitude for 30 orbits before going into an inertial orientation for retrieval. A drawing of the Spartan 206 spacecraft is shown in Figure 1. Figure 2

Figure 3. Spartan 207 Orbiting with Antenna Inflated.



shows a photograph of the Spartan 206 spacecraft in the orbiter payload bay. The procedure for Spartan 207/Inflatable Antenna Experiment did not include stellar updates because the LVLH orientation was maintained for only one orbit which was not long enough to incur unacceptable gyro errors. It did, however, perform an initial sun acquisition and then maneuvered to an inertial orientation parallel to the orbit plane before starting an LVLH rotation as explained below. The Spartan 207 is shown with the antenna inflated in Figure 3.

Background

Spartan is a NASA program originally intended to extend the observation time of sounding rocket astronomy experiments from five minutes-per-flight to two days while free-flying from the orbiter. In the nominal shuttle mission at the completion of its science observations, the Spartan payload goes to a low power mode with a stabilized attitude until RMS retrieval by the orbiter, reattachment to the flight support structure, and return to earth.

Low-cost and reuse have been major goals in the Spartan program and this has been achieved for the most part by a conscious effort to keep orbiter interfaces to a minimum, standardization of the spacecraft bus, and keeping procedures and operations as simple as practical. The original bus design, done between 1980 and 1982, was based on proven sounding rocket subsystems. At the current time an extensive effort is underway to enhance the Spartan design with new hardware based in large part on the Small Explorer Program. The Spartan bus design reflected in this paper, however, is based on technology that was state-of-the-art at the time the first Spartan payload was developed in the early 1980's. This includes an attitude control timeline entirely programmed before launch months prior to launch and consisting mostly of software that cannot be modified after loading into EPROM memories. A Z80 processor provides limited processing at various points in the timeline. To cope with pre-launch uncertainties such as the actual time of launch, several bit-level commands can be loaded into the Attitude Control System (ACS) by a member of the astronaut crew prior to deployment using a hardware bus. There is no RF link available after deployment so that immediately upon detachment from the RMS Spartan operates autonomously.

Not surprisingly, since its original flight in 1985, Spartan experimenters have desired more than the original celestial pointing capability. In some cases this has been accommodated. For example, sun-pointing was provided on three missions using sun sensor error signals that inherently correct for slow sun motion against the background stars.

Flight Software

The flight software consists of a main routine that calls up subroutines, referred to as sequencer programs, that send commands to the ACS. A series of attitude maneuvers can be set up within a sequencer program and wait functions can be used to control the length of time required for the sequencer program to run from beginning to end. This is important because the sequencer programs that perform the stellar updates are repeated every other orbit so that they must have the same duration. The timing in the main routine is controlled using the known duration of the sequencer programs and two onboard timers. On Spartan 206, a repeated loop is set up in the main routine with sequencer programs and timer settings whose sum is equal to twice the orbit period. Also, because sequencer programs are numbered, the loop can increment through a list of sequencer programs. This allows different maneuvers to be performed every other orbit as required for stellar updates.

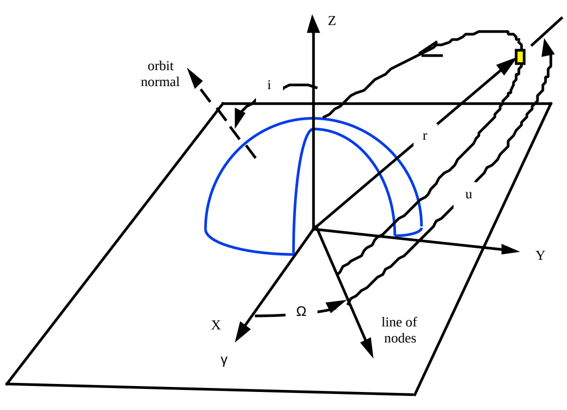
LVLH Orientation

Figure 4 shows the Earth Centered Inertial (ECI) frame, given by the X, Y and Z axes, and the spacecraft body frame, given by the roll, pitch and yaw axes. The ECI frame has the X and Y axes in the earth's equatorial plane with the X-axis pointing in the direction of the first nodes of Aries, shown by γ. In the LVLH orientation the roll axis points along the velocity vector, the pitch axis points in the direction of the negative orbit normal and the yaw axis points in the nadir direction.

To maintain the LVLH orientation shown in Figure 4, the spacecraft performs a constant right-handed rotation about its negative pitch axis at a rate equal to the orbit rate of the spacecraft. This rate is referred to as the LVLH rate. Therefore, the desired inertial attitude is time varying and depends on the orientation of the orbit plane and the spacecraft's attitude at any given time.

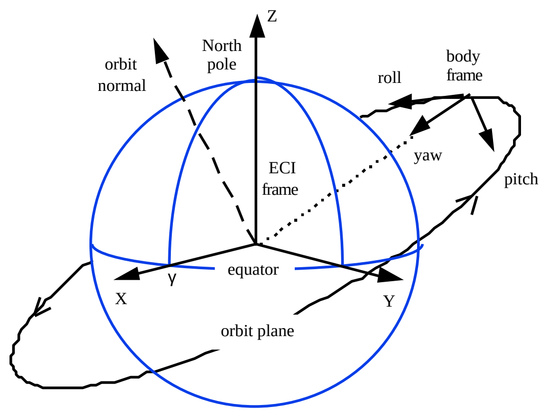
Orbital Elements

Figure 5. Orbital Element Definitions.



A circular orbit of radius r is assumed. The orientation of the orbit plane relative to the ECI frame is given by two angles, the inclination, i, and the Right Ascension of the Ascending Node, Ω, as shown in Figure 5. The inclination is measured from the Z-axis to the orbit normal and Ω is measured in the equatorial plane from the X-axis to the point on the line nodes where the spacecraft passes from the southern to the northern hemisphere. The position of the spacecraft in the orbit plane is given by the argument of latitude, u, which is measured in the orbit plane from the line of nodes to the position of the spacecraft[1].

Figure 4. The LVLH Orientation and Spacecraft Roll, Pitch and Yaw Axes.



Orbit Rate and LVLH Rotation Rate

As stated earlier, in order to maintain the LVLH orientation the spacecraft must rotate at a rate equal to the orbit rate, n. The orbit rate for circular orbits is given by equation (1).

$$n = \sqrt{\frac{\mu_E}{r^3}} \qquad (1)$$

μ_E is the gravitational parameter of the earth. The orbit rate must be known in advance so that the pitch gyro torquing rate can be calibrated to the LVLH rotation rate. This means that the radius of the orbit must be known in advance. A difference in the orbit rate and the LVLH rotation rate will produce an offset from the LVLH orientation about the pitch axis. This offset can be removed periodically by performing stellar updates.

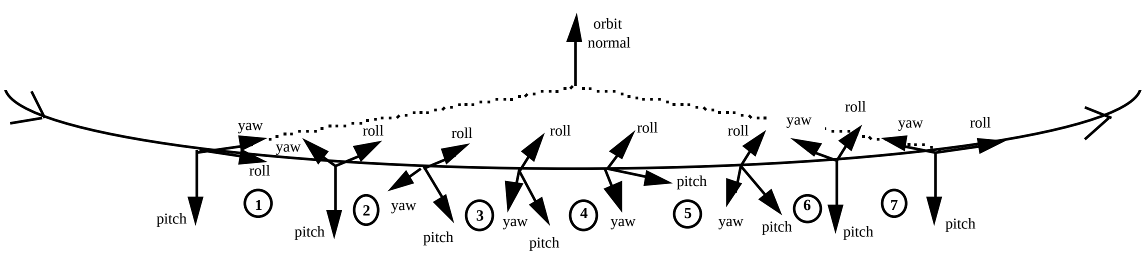
Motion of the Line of Nodes

The earth is not a perfect sphere but bulges slightly around the equator. The resulting gravitational perturbation causes the line of nodes to move longitudinally. For orbits where the inclination is less than 90°, the line of nodes will move westward, corresponding to a decrease in the right ascension of the ascending node. The average change in the right ascension of the ascending node per orbit is given by equation (2)[2].

$$\frac{\Delta\Omega}{\text{orbit}} = -3\ \pi\ J_2\ \frac{R_E^2}{r^2}\ \cos(i) \qquad (2)$$

J_2 is the second order spherical harmonic correction to the earth's gravitational field and R_E is the radius of the earth. This gravitational perturbation does not affect the orientation of the spacecraft so stellar updates are also used to reorient the spacecraft with the precessing orbit plane.

Figure 6. Stellar Update Example Showing Maneuvers (Star Tracker Boresight Pointing Along the Pitch Axis).



Initial Acquisition

The spacecraft is released from the RMS at a specified orientation. A standard stellar acquisition[3] is then made consisting of a sun acquisition followed by a stellar update (described below). The entire initial acquisition is completed in slightly under two orbits. The spacecraft then maneuvers to an inertial orientation with its roll and yaw axes in the orbit plane and performs a pitch "catch-up" maneuver to obtain the initial LVLH orientation. The LVLH orientation is maintained by starting an orbit rate pitch rotation when the spacecraft reaches the correct argument of latitude.

Stellar Updates

Stellar updates are performed every other orbit and are used to periodically reorient the spacecraft with the precessing orbit plane and to remove errors due to gyro drift and due to differences in the LVLH rate and the calibrated pitch gyro torquing rate. Each stellar update consists of two parts. First the spacecraft performs three maneuvers about its control axes to point the star tracker at a relatively bright star, referred to as the guide star, to update the two gyro axes perpendicular to the star tracker boresight. Secondly, the errors about the star tracker boresight are removed by maneuvering to and capturing a second star called the update star. The spacecraft then maneuvers back to the precessed orbit plane which now has a slightly different Ω. The difference is equal to the amount that Ω has changed since the last update two orbits earlier according to equation (2). This means that i must also be known in order to calculate the maneuvers. Each stellar update sequencer program is associated with a slightly lower Ω and the loop in the main routine increments through them. An example of the sequence of maneuvers for a stellar update is depicted in Figure 6. This example has the star tracker boresight pointing along the pitch axis.

The sequencer maneuvers are :

1. A pitch maneuver to bring the yaw axis parallel to the negative projection of the guide star on the orbit plane.
2. A roll maneuver to point the star tracker at the guide star.
3. A pitch maneuver to align the roll axis perpendicular to the plane formed by the vectors to the update and guide stars. **After this maneuver the star tracker locks on the guide star and updates the roll and yaw axes.**
4. A roll maneuver to point the star tracker at the update star. **After this maneuver the star tracker locks on the update star and updates the pitch axis.**
5. A pitch maneuver that brings the roll axis into the orbit plane associated with a new Ω.
6. A roll maneuver that brings the pitch axis parallel to the orbit normal.
7. A pitch maneuver that brings the spacecraft back into the LVLH orientation associated with the spacecraft's argument of latitude after the elapsed time since the start of the stellar update sequencer program.

In maintaining the LVLH orientation, the angle about the spacecraft's pitch axis, relative to the ECI frame, depends on the spacecraft's position in the orbit plane, given by the argument of latitude. For this reason, the argument of latitude when the spacecraft maneuvers to and from the LVLH orientation must be known in order to calculate the maneuvers in advance and program them into the flight software. The argument of latitude is set by the position of orbit sunset because the stellar update must be performed during orbit night.

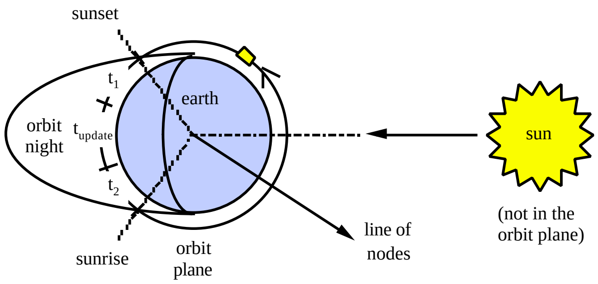
<u>Orbit Night</u>

The argument of latitude at sunset depends on the r, i and Ω of the orbit and the right ascension and declination of the sun, α_{sun} and δ_{sun}, and is given by equation (3)[3].

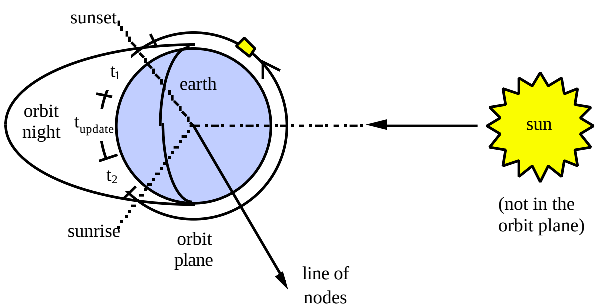
$$u_{sunset} = u_{sun} + \pi - \cos^{-1}\frac{\left(1 - \frac{R_E^{\ 2}}{r^2}\right)^{\frac{1}{2}}}{\cos(\delta_{sun})} \qquad (3)$$

u_{sun} is the argument of latitude to the projection of the solar vector on the orbit plane which depends on the position of the sun and the orientation of the orbit given by Ω and i. Figure 7 shows the position of orbit night in the orbit plane.

Figure 8. Correction Using t_1 and t_2 for the Motion of the Line of Nodes.



(a) Initial case with $t_1 = t_2$ and t_1 starting at orbit sunset.



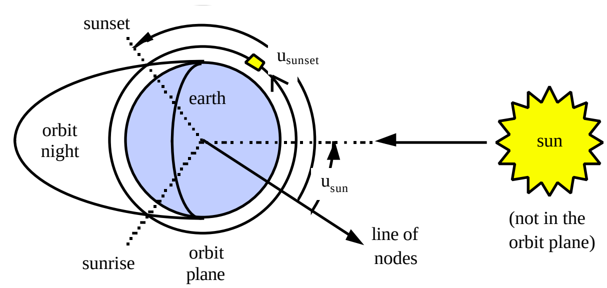
(b) Later case showing the motion of the line of nodes with $t_1 > t_2$ and t_1 starting before orbit sunset.

Change in the Position of Sunset Due to Precession of the Line of Nodes

Two timers, set to values of t_1 and t_2, are used to position the update during the orbit night. The flight software's main routine is set up so that, initially, t_1 starts at orbit sunset and t_2 starts after the stellar update sequencer program has ended, and ends close to orbit sunrise. These times are constrained by equation (4).

$$t_1 + t_2 + t_{update} \leq t_{night} \qquad (4)$$

Figure 7. Position of Sunset Shown in the Orbit Plane.



t_{update} is the duration of the stellar update sequencer program and t_{night} is the approximate length of the orbit night. Initially, t_1 and t_2 are set equal to each other. Afterwards, however, an equal amount is added to t_1 and subtracted from t_2 each orbit. This amount is set approximately to the time required for the spacecraft to travel throughout the distance that u_{sunset} changes each orbit due to the motion of the line of nodes. This ensures that the update will always begin and end during orbit night. This is shown in Figures 8 (a) and (b).

Change in Position at Sunset Due to the Motion of the Sun

The calculated maneuvers had to be valid over a range of days to account for possible launch delays. A range of days was covered by varying t_1 and t_2. This was possible because the values for the timers were entered as bit level commands just before the spacecraft was deployed. The argument of latitude at sunset for a given orbit increases by about 1° /day for low inclination orbits. To avoid writing sequencer programs with maneuvers for each day, maneuvers were prepared for a day at the midpoint of a range of possible mission days and, instead of setting the initial values for t_1 and t_2 equal they were varied from day to day. Maneuvers are calculated with t_1 and t_2 set equal for a calendar day in the middle of the range. For flight days before the middle of the range the initial value for t_1 is set higher than t_2. The timer still starts at orbit sunset but the update occurs later in the orbit night. Conversely, for flight days after the middle of the range, t_1 is set lower than t_2. The sum of t_1 and t_2 always remains the same. These two cases are shown in Figure 9 (a) and (b). The case at the middle of the range of days is the same as the case shown in Figure 8 (a).

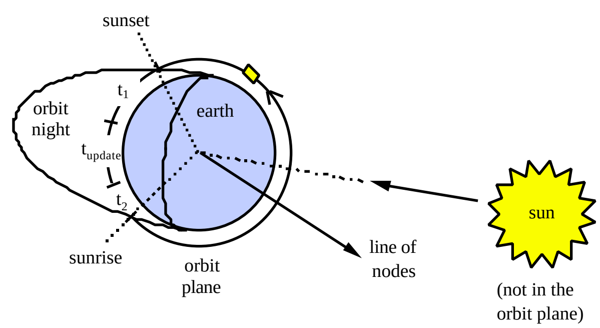
Correction for Differences in the Orbit Period

An error that cannot be corrected using stellar updates will occur if the actual orbit period is different than the orbit period assumed by the sequencer programs. This is handled by adding or subtracting the difference from the either the initial value for t_1 or t_2. This was done on orbit because the initial values for t_1 and t_2 are, as stated earlier, calculated during the mission and entered into the onboard computer by the astronaut crew just before deployment. The sequencer program with the maneuvers corresponding to the right ascension of the ascending node at deployment is also chosen at this time.

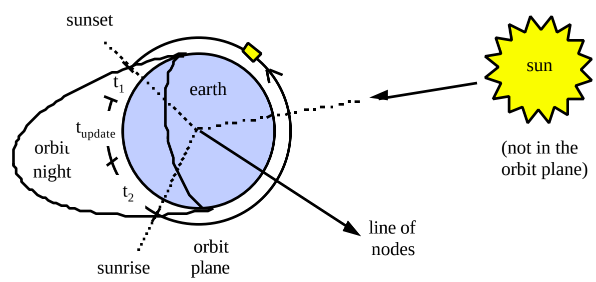
Conclusion

This paper describes a way to program an inertial pointing spacecraft to behave as an earth pointing spacecraft. The accuracy of this method depends on several factors. Because the radius of the orbit, the inclination and the orbit circularity (eccentricity) need to be specified in advance of launch, any differences during the actual mission will cause errors. Another source of error lies in the increment used for the right ascension of the ascending node but this can be reduced by performing more frequent stellar updates. The errors could also be eliminated by the addition of an RF command link for the Spartan carriers. This would allow maneuvers to be calculated on the ground and transmitted to the spacecraft.

Figure 9. Correction for the Motion of the Sun Using t_1 and t_2.



(a) Start of range of flight days with $t_1 > t_2$.



(b) End of range of flight days with $t_1 < t_2$.

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4. Agrawal, pp 81-82.

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