Navigation and Pointing Timeline Development for an Autonomous Free-Flying Shuttle Payload

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This paper describes the techniques and analyses used to construct the Spartan 101 mission pointing timeline for the Orbiter detached portion of its flight. It covers the design process that had to transform requirements and constraints originating from the experimenter, the Space Transportation System, and the payload hardware design into a workable set of pointing instructions sequenced to mission events. The paper discusses the technical approach used for initial attitude acquisition, determining target visibility, observation scheduling, and calculation of payload maneuvers.

Introduction

SPARTAN 101 was flown in June 1985 aboard Shuttle flight 51-G. On the fourth mission day, the Spartan payload was deployed by the crew, using the remote manipulator system (RMS), into free flight. After release and verification of payload operation, the Orbiter performed a separation maneuver, and the payload subsequently executed a pointing program to observe several celestial targets for both science and attitude update purposes. Approximately 48 h later, the payload was retrieved by the Orbiter, restowed in the Orbiter bay and returned to Earth.

This paper is a description of the techniques and analyses that were used to construct the programmed timeline. The description covers a planning process that integrated requirements from the experimenter, the space transportation system (STS), and the payload hardware into a timeline. The ultimate form of this timeline is a distinct set of payload commands stored onboard. Such commands include, for example, attitude maneuvers, sensor operation, experiment turn-on, turn-off, etc. These actions are similar to those executed by a sounding rocket sequencer for good reason; the attitude control system for Spartan is a derivative of the STRAP V system developed by the Goddard Space Flight Center for sounding rocket attitude control. Typical rocket payloads execute a preprogrammed timeline lasting several minutes. Spartan 101, conceived as an extension of sounding rocket-type payloads had to operate nearly 48 h in low Earth orbit. Obviously, the sequence of events, i.e., timeline, is much more extensive.

Aside from the length of the mission and the fact that the payload is in orbit, certain other constraints made this process complex. These constraints include:

1) No capability to communicate with the spacecraft after it has been unberthed from the Orbiter, so that the timeline could not be altered during detached operation.

2) All flight software was stored in read-only memory chips, which were installed during payload integration months before launch. From this point on until payload unberthing, a total of only 8 bits of data could be changed. Therefore, significant changes of any pointing parameter required the physical replacement of memory chips embedded in the payload, and this could not be done during on-line integration at the Kennedy Space Center. Consequently, the program had to be structured to last through a launch window lasting at least four weeks.

3) The initial inertial attitude of the Spartan payload was transferred from the attitude of the Shuttle RMS at deployment. This attitude could not be guaranteed by the STS to be sufficiently accurate to assure a safe star acquisition by the payload star tracker after release. Since accurate pointing requires periodic stellar updates, it was necessary to incorporate a capability for an initial acquisition using sun sensors to acquire the sun, followed by a payload rotation to allow the star tracker to scan for a selected guide star. The program parameters due to the sun/star geometry were sensitive to the launch date and yet had to be determined and programmed prior to payload integration.

4) As mentioned earlier, the attitude control system design was based on a sounding rocket attitude control system which was required to operate for only a few minutes. To handle the greatly extended time of operation (48 h), the Spartan design had the addition of a microcomputer that loaded the sequencer with consecutive programs stored in the computer memory. Through these programs, the timeline was executed. However, to meet launch readiness dates, it was not possible to incorporate software that could perform any form of numerical computation on the timeline parameters or adjust the program for uncertainties in orbit parameters.

5) The cold gas thruster system, also derived from sounding rockets, had limited propellant capability requiring economy in the number of maneuvers.

These limitations were the result of a need to keep the payload hardware and STS interfaces as simple as practical for the first Spartan payload. As the Spartan program gains experience and the payload design is enhanced, these restrictions will no doubt be relaxed, thus extending capability on future missions. For autonomous payloads in general, the Spartan 101 mission illustrates the worst case in terms of timeline planning for observations of fixed celestial targets.

Payload Description

The Spartan 101 Payload is shown in Fig. 1. In the bay, the payload is secured by a release and engage mechanism (REM), which is attached to the flight support structure. During free flight, the detached payload operates as a three-axis stabilized observatory capable of maneuvering to any attitude at preprogrammed times ultimately fixed to the time of deploy. The maneuvers are made with small cold gas thrusters mounted to the rear of the payload.

The payload consists of a large sunshade attached to a box-like structure in which the experiment and support subsystems

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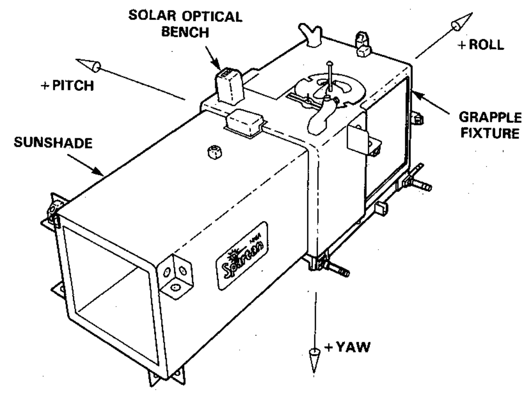


Fig. 1 Spartan 101 Payload.

are housed. The experiment, a large-aperture x-ray telescope, is coaligned with a star tracker to view out the sunshade. Except during observations, the apparatus is covered by a large door to keep out light and contamination. The experiment and star tracker are comounted on an optical bench. The bench is kinematically attached to the payload primary structure and, hence, remains free of distortion due to structural stresses. Also mounted on the optical bench are two tuned rotor inertial gyros, providing a full three-axis inertial reference. Fine sun sensors used during the initial attitude acquisition are mounted to a solar optical bench at the top of the payload, which in turn is rigidly attached to the optical bench in the interior. Coarse solar sensors are placed on mounts on the top and front to augment the field of view of the fine sun sensors.

Support systems for the experiment are physically placed behind the experiment. These subsystems include the batteries, the payload functional control subsystem, the attitude control electronics, and cold gas thruster system used for attitude control.

Attitude control is maintained by a three-axis stabilized system, which uses the gyros as the primary reference. Optical sensors are switched into the control loop to provide position signals when necessary to update the gyro reference.

Commands to the pitch, roll, and yaw control loops originate from one of two devices. One device, a sounding roket-derived sequencer, clocks most of the attitude control system functions by changing the state of 48 hard-wired switches at the completion of software encoded time intervals. Each sequencer program can execute a maximum of 57 timed intervals from 0.001 ms – 102.3 s long. After 57 intervals, the sequencer must be reloaded with the next sequencer program.

A second device, a microcomputer, has stored in its memory the entire set of sequencer programs. When a sequencer program is completed, the sequencer can signal the microcomputer to load a "next sequencer program," and so forth. The sequencer can also request the microcomputer to quickly load certain tables of information into the control loop that configures the gains and selects which sensors are in control. For certain situations, the microcomputer can regulate the control loops directly. This is typical during the time the sequencer is being loaded with a "next sequencer program."

The Spartan 101 payload used a coordinate system shown in Fig. 1. The roll, pitch, and yaw axes are oriented according to sounding rocket convention rather than satellite convention. As shown, the experiment and star tracker line of sight were parallel to the negative roll axis, while the grapple fixture center post and the fine sun sensor line of sight are parallel to the negative yaw axes. Positive or clockwise maneuvers are defined as right-handed rotations about the positive direction of any control axis.

Outline of the Procedure

The construction of the timeline required solutions to two fundamental problems:

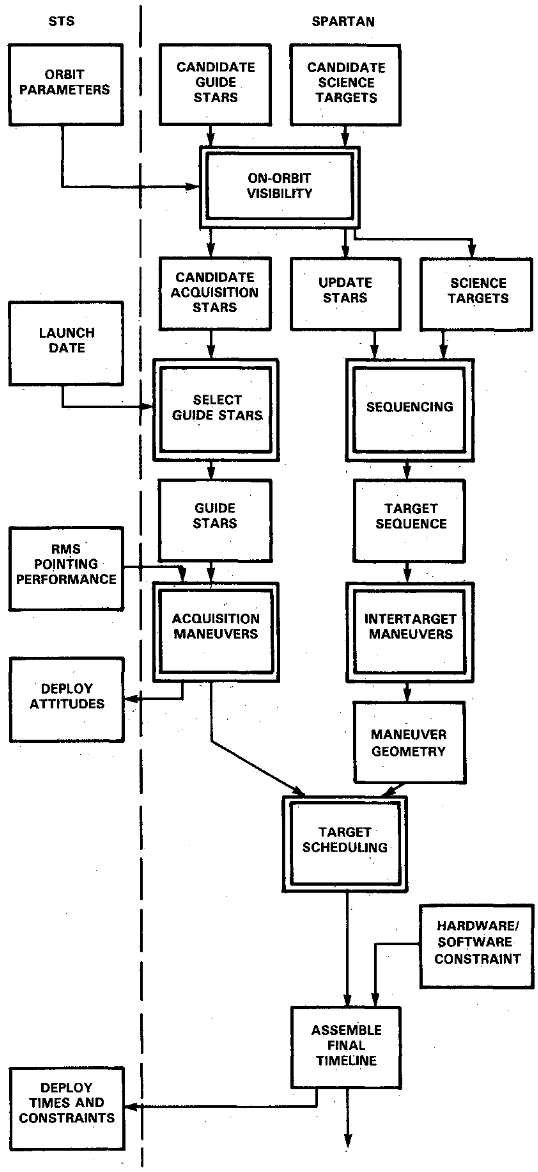


Fig. 2 Information flow for timeline construction.

1) Given the potential deploy errors, how can the payload obtain a three-axis inertial reference with errors sufficiently low to ensure the accurate attitude pointing required ($\pm$ 3 arcmin)?

2) Once the reference is obtained, the payload can perform its science pointing following the preflight stored timeline. How should this timeline be constructed to deal with the constraints indicated earlier?

Figure 2 shows the flow of information starting with experiment and STS inputs and ending with the timeline ready for encoding into flight software.

The first task was to determine the science and attitude control targets. The science instrument, an x-ray telescope, had a field of view of 3 deg $\times$ 5 arcmin. The science observations required small slow scans (1.5–2 deg at 16–24 deg/h) of x-ray sources. For the attitude control system, star pairs had to be selected for the star tracker to perform periodic gyro updates (once per orbit during orbit night) and initial acquisition. Owing to limitations of the star tracker and the attitude update algorithm, no star could be dimmer that +3 magnitude, and the two stars had to be separated by angles no greater than 90 deg or less than 25 deg. Additionally, it was desired that the first star used in acquisition be as near as possible to 90 deg from the sun in order to avoid attitude error cross-coupling.

The list of candidate targets was evaluated using programs to ascertain orbital visibility, described later. To perform this task required knowledge of the orbit from the STS. The position of a circular orbit in celestial coordinates is given by the orbit inclination and the orbit right ascension of ascending node (RAAN). With this information and the orbit altitude, it is possible to determine the "visibility window" of each target; i.e., when does it rise and when does it set on the Earth limb as seen from an orbiting spacecraft?

From this point on, the acquisition and timelining tasks are performed somewhat independently.

Inertial Attitude Reference

A method of obtaining and periodically updating the attitude reference had to be implemented. This method was required to operate autonomously and make use of the attitude sensors that had been incorporated on the Spartan payload, i.e., the coarse solar sensors, the fine sun sensor, and the star tracker.

The attitude reference was updated on each orbit by viewing two preselected stars in succession. This technique has been used for many years on sounding rocket attitude control systems. First, the star tracker locks on to a sufficiently bright star, called the guide star. The payload is maneuvered so that the guide star is centered in the 8 × 8 deg square field of view (FOV) of the tracker, thus defining two axes. The only significant error is around the roll axis. To eliminate this error, a single fixed-length maneuver is performed to a second star, called an update star. Because of the presence of the roll error, the update star will not be centered in the FOV of the star tracker at the completion of the maneuver. This offset is measured by the attitude control system (ACS) as the update star is captured and subsequently used to perform a roll correction maneuver, which cancels the remaining error (the roll error is determined by multiplying the measured offset angle by the cotangent of the maneuver angle between the two stars).

Limitations in this technique are that too large a pointing error will cause the update star to be missed completely. Also, if the update star is too close to the guide star, the error update will be inadequate due to poor resolution. A good distance is 30 deg. Other constraints on the selection of stars are that they be visible during the orbit night long enough to allow the acquisition (so that illuminated particles of debris are not mistaken for stars during the daytime) and that there be no comparatively bright stars nearby since the star tracker is designed to lock on to the brightest star in the FOV.

Obtaining the Initial Attitude Reference

When the Spartan attitude control system was to assume control after deployment by the Shuttle RMS, the maximum projected error in any axis was close to 10 deg, due to the effects of various STS and payload error sources. The science objectives required the ACS to reduce this error to 3 arcmin.

Since the attitude error could exceed the star tracker's FOV, the major problem was to find the guide star. The scheme described here (called an "acquisition scheme" for "acquiring" the guide star) starts by acquiring and pointing the negative yaw axis at the sun (this axis is hereafter called the grapple fixture axis because on Spartan 101 the grapple fixture is on this side). This is accomplished by using the solar sensors onboard the payload. The solar sensors have a FOV of 270 deg, so the sun can easily be found, even in the presence of large attitude errors. Also, the requested deployment attitude of the payload from the RMS is with the grapple fixture pointed toward the sun, so that the initial attitude error should be within 10 deg. Solar acquisition occurs approximately one orbit after deployment in order to give the shuttle a chance to move far enough away so that it will not bias the solar sensors nor occult the sun. During the interval from deploy to the sun acquisition, the payload was stabilized at its deploy attitude by cold gas thrusters operating on gyro control.

To understand how the guide star is acquired, first, consider a situation in which the guide star is exactly 90 deg from the sun (Fig. 3). When the grapple fixture is pointed at the sun, the star tracker (which is pointed orthogonal to the grapple fixture axis) can be rotated about the grapple fixture axis until the guide star is acquired. As can be seen in the figure, after solar acquisition, all pointing errors are contained in the star tracker axis. Therefore, the requested position of the star tracker at deployment must be far enough to one side so that when the slew to the guide star is made, the star tracker will not be on the side of the star opposite the direction of the slew. If the star were not located in the same direction as that of the slew, the vehicle would be required to rotate almost 360 deg before the guide star is found. A slew that large causes other problems, not the least of which is insufficient time to do the acquisition. So in determining the initial star tracker pointing position, the assumed initial pointing error becomes very important.

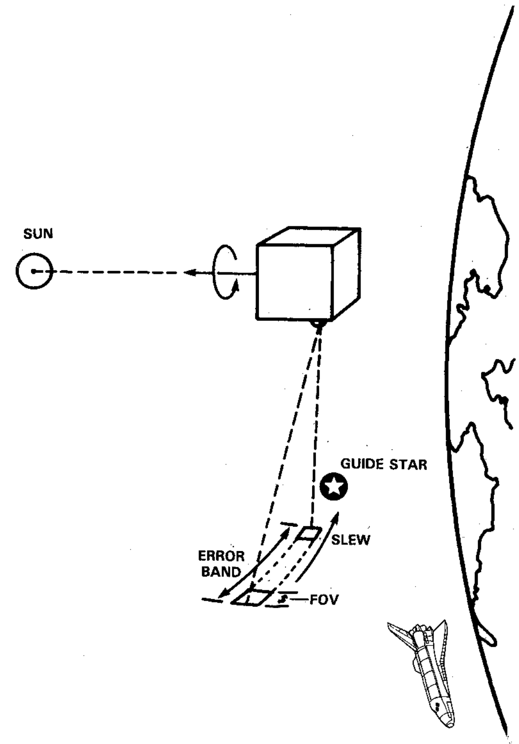


Fig. 3 Geometry when the Sun-Spartan-Guide Star Angle = 90 deg.

Also of concern here is the following: as the tracker slews to the guide star, it will lock on other stars along the scan path, checking to see if they are the correct magnitude, i.e., if they are the guide star. These are stars bright enough to draw the attention of the star tracker but not of the same brightness as the guide star. If the tracker magnitude discrimination logic determines that a star is not the guide star, the algorithm will reject it, place the spacecraft back on track, and continue on the way to the guide star (when the spacecraft gets "back on track," it automatically moves ahead 5 deg along the slew path so that it will not continually be acquiring the same star over and over again). These rejected stars are called "intercept stars." For the design of Spartan 101, it was decided there should be no more than 5 intercept stars, to reduce the possibility of accumulating error and missing the guide star (i.e., if more than 5 intercept stars are found when doing the analysis, another guide star must be chosen). Actually, with the guide star used in the flight, it turned out that there were no intercept stars.

Now, consider the case in which the guide star is not quite 90 deg from the sun. Figure 4 illustrates what was done to compensate for a different sun-guide star angle. Before slewing to the guide star, a pitch maneuver off the sun is performed to compensate for the difference in this angle. Given

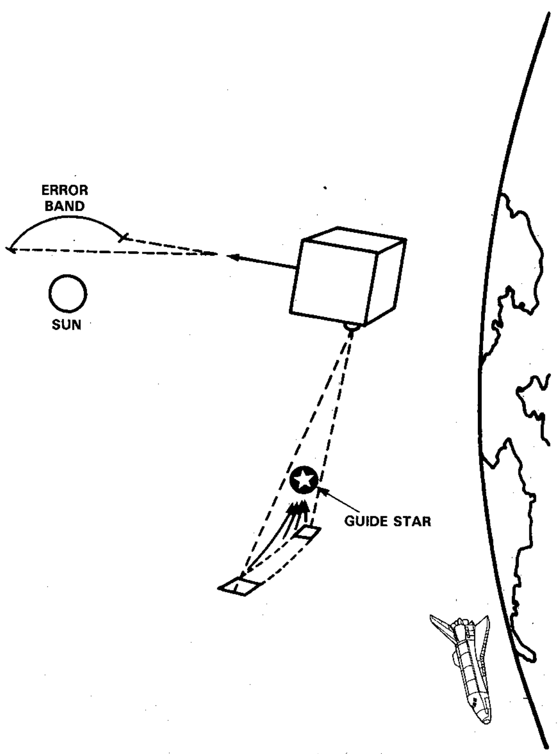


Fig. 4 Star Tracker Field of View has several possible paths when the sun-guide star angle is not 90 deg.

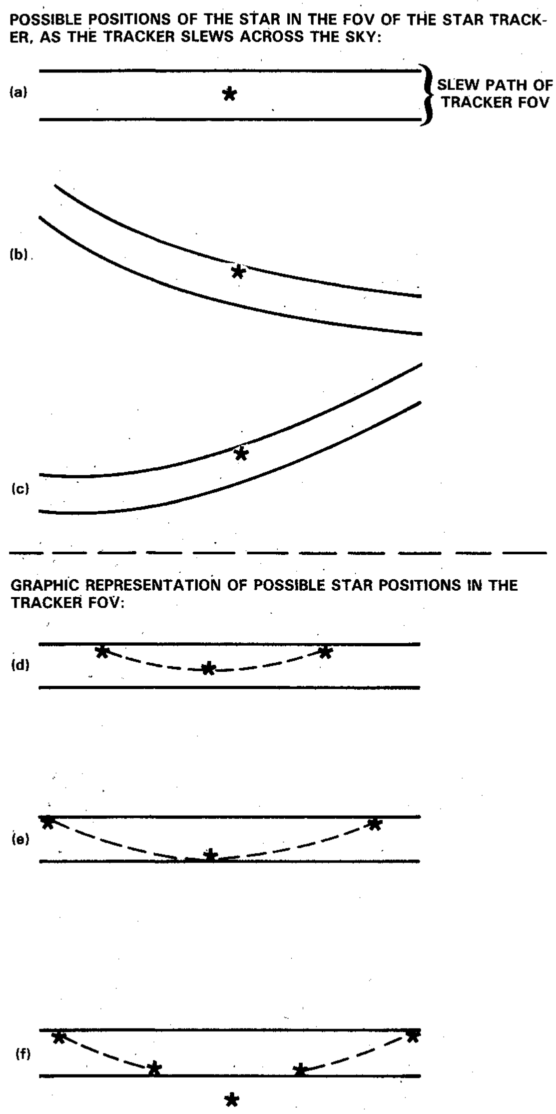


Fig. 5 Relation of slew path to guide star.

the range of points at which the star tracker may end up pointing due to the initial pointing error, after the pitch maneuver is performed, the grapple fixture axis will not end up pointing at one point in space but rather will have an error band of points at which it may end up pointing (as shown in Fig. 4). This error band will be an arc, or section of a cone around the sun, the half-angle of which is equal to the amount of the pitch maneuver. Now, given that these are different inertial points at which the grapple fixture may end up pointing, the slew to the guide star (a rotation about the grapple fixture axis) will cause the star tracker to travel different paths to the star (indicated in Fig. 4 by the arrows pointing to the guide star). This means that different intercept stars will be seen. What is needed now is a mathematical model of this situation, so it can be determined, given a certain pitch maneuver, an initial deployment attitude, an assumed amount of initial pointing error, and sun-guide star geometry, whether the guide star will be acquired or not. Or better yet, given the sun-guide star geometry and assumed initial pointing error, what should the pitch maneuver be and where should the star tracker be initially pointed to assure guide star acquisition?

Geometric Model

To understand the geometry involved, first, consider how the star appears in the star tracker FOV, as the tracker takes different paths to the guide star. Figure 5 illustrates some different cases:

a) The case in which the star will end up right in the center of the FOV as the tracker slews to the star. This corresponds to the grapple fixture pointing to the center of its error band with a specific angle off the sun.

b) The case in which the grapple fixture axis is pointed at the edge of its error band; then as the Spartan slews about the grapple fixture axis towards the guide star, the star will appear in the tracker FOV at the edge (if the grapple fixture was past its allowable error band, the guide star would end up outside the slew path of the tracker FOV and be missed).

c) The same case as b) but with the grapple fixture starting on the other side of its error band and the tracker slew path in the opposite direction.

d) Cases a–c using a frame of reference fixed to the star tracker FOV slew path, illustrating the range of possible positions of the guide star when it is acquired (this is a graphical representation only — it does not reflect the real situation).

e) The same case as d) with this exception: when the star is in the center position, it is at the "bottom" of the star tracker — this results from choosing a slightly different pitch angle off the sun. Doing this increases the span in which the star is visible. This, in turn, relates to a broader error band allowed in the grapple fixture axis.

f) Another possibility corresponds to the following situation: when the grapple fixture is pointed at the center of its error band, the guide star will not be acquired by the star tracker. The grapple fixture must rotate around its error band a certain amount before the star will start being visible. Then the star will be visible as the grapple fixture rotates a certain amount more. In other words, instead of the grapple fixture allowable error band looking as shown in Fig. 6a, it will look as shown in Fig. 6b.

Being able to calculate where these error bands are will allow the initial deployment attitude (i.e., star tracker pointing) to be calculated.

A broader illustration of the whole situation can be seen in Fig. 7. This shows the grapple fixture error band around the sun at the top of the illustration, with the guide star at the bottom. Three possible paths of the star tracker are illustrated —

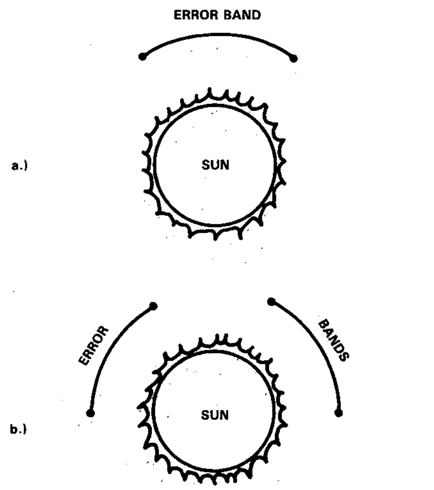


Fig. 6 Possible error bands for grapple fixture axis.

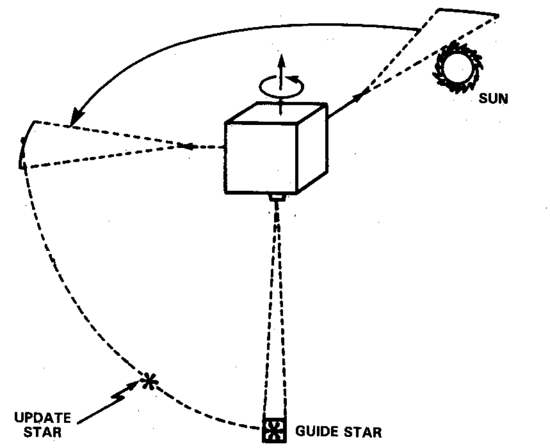


Fig. 8 Roll maneuver to set up capture of update star.

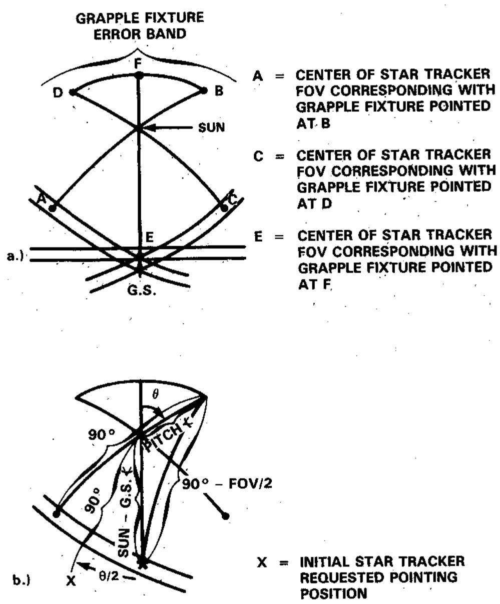
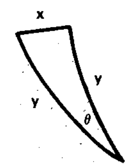


Fig. 7 Geometric model for calculating acquisition.



By spherical trigonometry:

$$\cos x = \cos y \cos y + \sin y \sin y \cos \theta$$

which reduces to:

$$y = \cos^{-1}\left[\sqrt{\frac{\cos x - \cos \theta}{1 - \cos \theta}}\right]$$

x = FOV of star tracker

θ = inertial error of third axis pointing

y = maximum degrees update star can be from guide star and still be found with the star tracker, given x and θ.

Fig. 9 Calculation to verify update star acquisition.

one at the two extremes and one in the center (the three cases corresponding to the three star positions illustrated in Fig. 5e). Figure 7b shows a picture similar to 7a, but the lengths of the sides of the spherical triangles have been added. These spherical triangle relations can be used to solve for θ in Fig. 7b. θ is the maximum allowable error in the grapple fixture pointing for a given pitch angle, which will still assure acquisition of the guide star. Notice that θ is only on one side of the sun-guide star line. This is to keep the corresponding star tracker position pointed to one side of the guide star to assure acquisition. It is important to notice here that θ is also the maximum allowable star tracker pointing error (because as the grapple fixture moves around the Spartan sun line, so also does the star tracker, by the same angle).

Hence, when θ is determined for a given pitch angle, it must be greater than or equal to the maximum assumed star tracker initial pointing error. If not, the grapple fixture may end up pointed out of its allowable error band, causing the guide star to be missed.

The pitch angle for Spartan 101 was determined by a computer program that starts from pitch = 0 and successively increments by 0.5 deg — each time calculating θ for that pitch angle, until a large enough θ is found. That corresponding pitch angle is the one used for that day (i.e., that sun-guide star geometry). Of course, this is all very dependent on what is used for the FOV of the star tracker.

All that remains now is to find the initial star tracker axis requested pointing position. This is determined by finding the inertial point that is 90 deg from the sun and an angular distance of $\theta/2$ from the sun-guide star line, as shown in Fig. 7b.

Note that these figures are based on the case in which the sun-guide star angle is less than 90 deg. A similar picture can be developed for the case in which the sun-guide star angle is greater than 90 deg. In this case, the original pitch maneuver would not be "up" off the sun but "down" off the sun.

Acquiring the Update Star

Consider that the guide star is now found and centered in the star tracker FOV. The next step is to acquire the update star. As illustrated in Fig. 8, the next maneuver performed is a roll about the star tracker axis. This is to put the grapple fixture axis, the star tracker axis (and the guide star), and the update star in the same plane. Actually, the exact position at which the grapple fixture axis is pointing is not known (because now that the star tracker is locked on to the guide star, all pointing errors are contained in the grapple fixture axis), so the center of its error band is used.

Now, the two-axis pitch maneuver to the update star can be made, and the attitude correction can be calculated, as described earlier.

To make sure the update star will be within the FOV of the star tracker, a spherical triangle can be constructed as in Fig. 9. Given the FOV of the star tracker and the assumed error in the grapple fixture axis, it can be determined whether a given update star (with a given distance from the guide star) will be acquired, with any of the possible pointing positions in the error band.

For initial acquisition, the guide star chosen for Spartan 101 was Vega and the update star Deneb. They are separated by

23.8 deg. The assumed maximum pointing error after deployment is ±12 deg, which included a margin exceeding that guaranteed by the STS. However, the grapple fixture axis pointing error will be much less, depending on the magnitude of the pitch maneuver off the sun. For example, a pitch of 2.9 deg gives a grapple fixture pointing error of 1.2 deg. A pitch of 23.5 deg gives a grapple fixture pointing error of 9.5 deg. In either case, there is plenty of margin when slewing to the guide star.

Following is a review of the Spartan acquisition scheme:

1) deploy Spartan with a three-axis attitude reference to within desired tolerance via the RMS.
2) Acquire sun.
3) Perform a pitch maneuver of the amount predetermined for that day.
4) Yaw to find guide star.
5) Perform a roll maneuver of the amount predetermined for that day.
6) Perform a pitch maneuver and acquire the update star.
7) Execute the science program.

Remote Adjust

The acquisition scheme described thus far will work on a given day for a given sun-star geometry. Now, there needs to be a way of having it work over the entire launch window, or about 28 days (this is the amount of time the Spartan project agreed it could handle launch slips, before the Spartan needs to be reprogrammed).

To do this, a scheme was developed called "remote adjust." By this method, one of 28 different selectable programs can be chosen before deployment. Each of these programs contain different pitch (off the sun) and roll (to get in the plane of the update star) maneuvers. That is all that changes — the rest of the science program remains the same. So then, the maneuvers chosen need to work for only one day (actually, maneuvers can be chosen to work for sometimes two or three days before a new set of maneuvers needs to be selected). Through the use of computer programs that make use of the geometric model described earlier, a chart can be generated listing the remote adjust maneuvers for each selectable acquisition program.

Developing the Pointing Timeline

Target Scheduling

Target scheduling is the arrangement of ACS navigation star observations, science target observations, orbital parameters, and hardware/software subsystem constraints into an orderly and robust sequence of events that can be transformed into the ACS flight software for execution during the mission. This sequence of events must be developed nearly four months prior to flight and must be applicable to all expected and contingency orbits and launch dates. These requirements are a mix of individually simple requirements that, when blended together, form a set of rather difficult and often contradictory constraints. These constraints dictated that an iterative approach to target scheduling be used. Numerous trade-offs were made between optimal science viewing, program robustness, and ACS/vehicle limitations throughout the development of the Spartan timeline.

The most critical parameters that drive the development of the pointing timeline will be described in the following paragraphs. The constraints each parameter imposes, along with the methodology chosen to deal with these constraints, will be discussed.

Launch Window

Unlike many other spacecraft, the Spartan program philosophy is to not impact the STS launch window if at all possible within the scope of the mission scientific objectives. Consequently, the Spartan 101 launch window was defined by the other payloads manifested on flight 51-G and by STS operational constraints. The Spartan 101 flight pointing timeline must be built to accommodate this window. Unfortunately, by not imposing constraints on the STS, the Spartan 101 timeline needed to be redone whenever the STS manifest or other payload constraints were changed. As a practical matter, however, Spartan 101 was forced to impose a requirement on the STS that they would not change the agreed-upon launch window after the launch minus three months date had passed. This approach, though sometimes inconvenient, seems to give the Spartan spacecraft many more STS manifest opportunties than would otherwise be possible.

Launch Date

Launch date serves primarily to bound the possible celestial locations of the sun, planets, and moon and defines the variations in orbital day/night that can be expected. Obviously, solar acquisition and sun pointing must occur during orbit day, preferably near orbit noon in order to avoid Earth albedo effects. Stellar acquisition and stellar updates were done only in orbit night in order to minimize the possibilities of false lock-ons or of tracking illuminated debris. Furthermore, it was decided that all stellar updates be performed near orbit midnight in order to make ACS navigation as immune as possible to off-nominal orbital parameters. Because of the desire to make ACS navigation activities as robust as possible, it was decided to design the Spartan 101 program for only a 28-day launch period, even though the composite 51-G window was much longer in duration.

Launch Time-of-Day

Launch time of day determines what orbit RAAN will be seen during the mission. The expected orbit RAAN changes approximately 15 deg for every hour change in launch time of day. Launch date defines expected orbit RAAN, but to a much lesser extent, i.e., 1-deg RAAN change for every day change in launch date. As we will see later, orbit RAAN is the primary orbit element that defines target occultation. Unfortunately, in the real world, launch time of day is the most "variable" variable in development of the STS mission timeline. This dictates that the payload timeline planners do some educated "second guessing" of the STS mission planners in order to develop a robust observation program.

Orbit Altitude

Orbit altitude defines what the orbit period will be during the mission. This is important because the spacecraft relies totally on elapsed time from deployment in order to infer orbit position. Orbit position can be implied by elapsed time only if one has an accurate and stable vehicle clock and accurate preflight knowledge of orbit period. Since the detached operation is only 48 h in duration, this approach is both technically feasible and very cost-effective. For the orbits considered on this mission, the orbit periods changed approximately 2.2 s for every 1-n.m. change in altitude.

Orbit Eccentricity

Orbit eccentricity directly implies, through Kepler's second law, the amount that the orbital angular velocity will vary throughout the orbit. Variation in orbital velocity is very detrimental to the target observation sequence, which relies completely on elapsed time for inferred orbit position. Consequently, it was decided to levy a requirement on the STS to provide a nearly circular orbit in order that an essentially constant angular velocity could be obtained. This requirement was easily met by the STS.

Orbit Inclination

Orbit inclination is the most stable of all the orbit variables, both in flight and in the planning phases of an STS mission. The inclination defines the orbit precession rate (i.e., change in RAAN) that will occur during the mission. It also has an effect on the target occultations.

Right Ascension of Assending Node (RAAN)

Right ascension of ascending node is defined as the celestial right ascension of the vehicle as it overflies the Earth's equator on the ascending side of the orbit. The expected RAAN range during the mission is the most critical orbital parameter affecting target visibility. As mentioned previously, the orbit RAAN range is determined primarily by the launch time and the STS ascent trajectory. Once in 28.5 deg-inclination low Earth orbit, the RAAN precesses at a rate of approximately 7.5 deg/day. Target visibility is predominantly constrained by occultation of the target by the Earth and, in some cases, the Earth's atmosphere. In low Earth orbit, the Earth forms a disk approximately 146 deg in diameter. The Spartan 101 science instruments could not view through the Earth's atmosphere below 17 deg above the horizon. Therefore, the effective occultation disk of the Earth increased to 180 deg in diameter (i.e., half the sky at any given time). The orgital motion of the spacecraft around the Earth would have the effect of rotating this occultation disk around the spacecraft once per orbit. Because orbit RAAN is constantly changing, the rotation of this occultation disk about the vehicle is seen to precess slowly with time, thus obscuring different portions of the sky as the mission progresses. The mathematical modeling of these motions is trivial if the following assumptions are made:

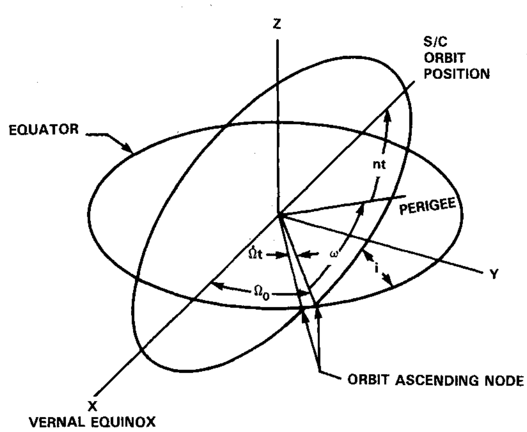
1) Circular orbit, which implies a constant orbit angular velocity.

2) Negligible aerodynamic drag, which implies a constant orbit throughout the mission.

3) Perfectly spherical Earth, which implies a constant zenith/Earth horizon angle.

These simplifications allow the vehicle orbital motion to be modeled with a zenith pointing unit vector that rotates about the Earth's center. If this unit vector is described in a right-handed XYZ coordinate frame, where the X axis is the equatorial plane and pointed toward the vernal equinox (zero node), the Y axis is the equatorial plane and normal to X, and the Z axis is normal to the equatorial plane and pointed to the northerly direction, then this vehicle unit vector can be related to the celestial location of any desired target. Recalling that the dot product of any two unit vectors is positive if they are pointed within 90 deg of each other and negative if they are not, it becomes apparent that if the payload zenith unit vector is rotated about the expected orbit path and the dot product between this unit vector and the desired target coordinates is constantly taken, then the polarity history of this dot product will reveal when the target is "visible" as a function of orbit position. If the orbit RAAN is incremented and the unit vector rotation and dot product sequence is repeated, it can then be seen how target visibility within the orbit path varies as a function of RAAN. Since orbit position is inferred only by elapsed time during the mission and since orbit position must be relative to the celestial coordinate frame of the targets, it seems logical that the time frame chosen must be related to these same celestial coordinates. The time it takes to orbit the Earth between successive ascending node crossings of the equator was chosen as the appropriate time reference frame. This orbital period is called the nodal period and is defined as two-pi radians divided by the sum of the rate of change of the mean anomaly and the rate of change of the argument of perigee (see Fig. 10). If the zero time reference point is set to be the time at which the vehicle crossed the equatorial plane on the ascending node of the orbit, then this provides a rigidly defined, well-behaved orbital clock. Vehicle deployment time is then specified relative to the orbital ascending node, thus linking deploy time to nodal time. Orbit position angle can now be accurately inferred from the time elapsed since deployment. Figure 11 illustrates the visibility of the primary Spartan 101 science target as a function of time from ascending node (orbit position) and ascending node location (RAAN). Similar visibility curves are generated for the other science targets, navigation stars, and the sun on the dates defined by the launch window. If a fixed range of RAAN's is chosen and a common subset of times the target is visible for all those RAAN's is extracted from the visibility curves, then a digital chart of guaranteed visibility windows can be constructed. Figure 12 is an example of this type

ORBIT NODAL PERIOD



i = ORBIT INCLINATION
ω = ARGUMENT OF PERIGEE
n = MEAN ANGULAR MOTION (RATE OF CHANGE OF MEAN ANOMALY)
Ω = RIGHT ASCENSION OF THE ASCENDING NODE
τ_N = NODAL PERIOD
t = ELAPSED TIME

NOTE: THE ARGUMENT OF PERIGEE IS REFERENCED TO THE EQUATOR, THEREFORE THE ORBIT NODAL PERIOD, τ_N, CAN BE DEFINED AS

$$\tau_N = 2\pi/(\dot{\omega} + n)$$

Fig. 10 Orbit nodal period.

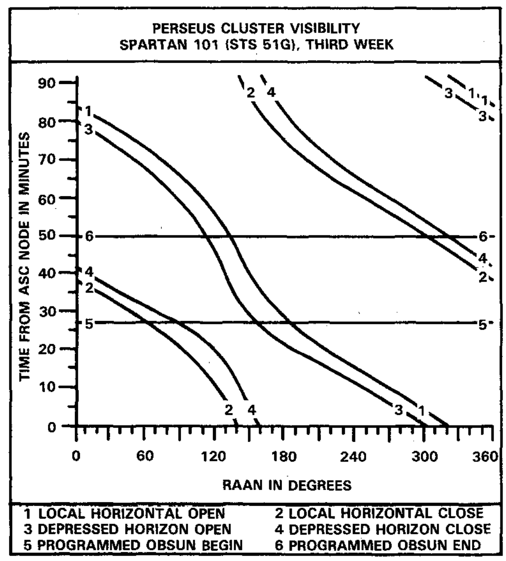


Fig. 11 Typical on-orbit visibility for a Spartan target.

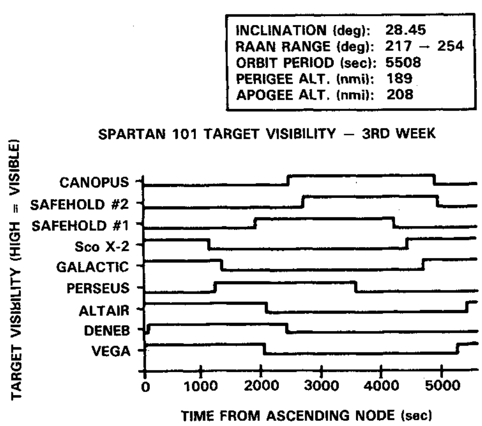


Fig. 12 Chart of guaranteed visibility times for Spartan targets.

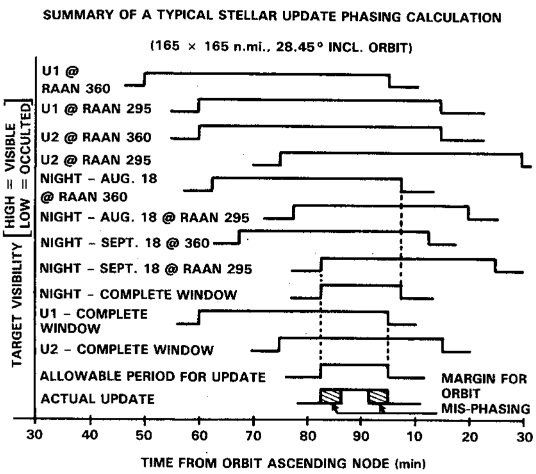


Fig. 13 Calculation of time from ascending node to do stellar update.

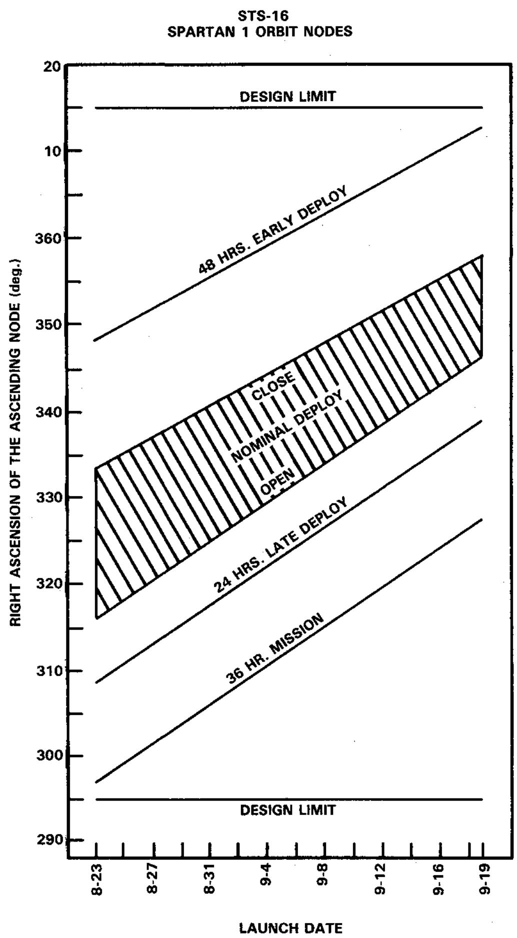


Fig. 14 Effect of early or late deploy on RAAN range.

of chart. Figure 13 illustrates the techniques used to find the common subset of visibility constraints. This type of chart provides the information needed to begin the assembly of a pointing timeline sequence.

Additional Constraints

In addition to the major parameters already discussed, numerous second-order variables, contingency plan effects, and design margins must be taken into account. These constraints fall under the following major headings:

1) RAAN range

a) The Spartan timeline was designed such that it was fully accurate if the vehicle was deployed either two days early or one day late. This had effect of adding 21.5 deg to the nominal RAAN range (see Fig. 14).

b) The Spartan timeline would also be accurate in the event that the STS launch window was extended 15 min any given day. This had the effect of adding 4 deg to the nominal RAAN range.

2) Timing Errors

a) Bias errors required a specified margin. The Spartan 101 deployment time was specified in nodal time with an error tolerance of ± 30 s. This left a 30-s time uncertainty in orbit position that had to be covered. Uncertainty in the true target locations also manifests itself as a bias error. A margin of ± 30 s of time was allocated for the effects of bias errors.

b) Cyclic errors can be caused by orbit eccentricity. Even though the instantaneous orbit velocity changes with time in an eccentric orbit, the orbit period remains constant. This introduces a cyclic timing (orbit position) error that averages to zero on an orbital basis. Cyclic timing errors are also caused by orbital oscillations produced by higher-order gravitational harmonics. A nonspherical Earth produces cyclic errors between the true occultation time and the calculated occultation time. A margin of ± 15 s of time was allocated for total effect of cyclic errors.

c) Secular timing errors are the most dangerous timing errors because a very small orbital error, when integrated over a 48-h mission, can become very large in the latter portion of the flight. Secular timing errors are those errors that cause the actual orbit nodal period. Potential causes of this type of error are insertion into the wrong orbit, orbit decay due to aerodynamic drag and solar pressure, and thruster misalignment that results in vehicle translation capability. The only protection against this type of error that was deemed reasonable was to schedule the ACS stellar and solar navigation sightings in the center of their respective viewing windows so that they had maximum tolerance to timing errors. This was possible because Spartan navigation sightings tended to be of quite short duration compared to the visibility windows. This strategy is not possible with science targets because the experimenter desired to maximize target viewing time.

3) Lighting constraints

Solar acquisition and tracking was constrained to occur within ± 5 min of orbit noon in order to minimize Earth albedo effects and maximize tolerance to slips in launch date. For similar reasons, stellar updates were chosen to occur near orbit midnight. The star tracker LOS and instrument LOS had to be kept at least 35 deg from the sun line in order to prevent damage to the star tracker or scientific instruments. In addition, the star tracker was to be kept from viewing the illuminated Earth.

Priorities and Timeline Design Strategy

Once all constraints were understood it became apparent that many of the restrictions could not be simultaneously satisfied. Therefore, design guidelines had to be established before a detailed timeline could be developed. The following strategy was chosen:

1) Occultation phasing will be adequate for all expected and contingency RAAN.

2) Stellar updates will be given priority in the timeline development and will be positioned in the center of their guaranteed visibility windows. This guideline was established in order to ensure that the ACS maintained navigation capability for as long as possible in the event of an anomalistic orbit.

3) Science target observations will be optimized for the normal orbit.

4) The pointing program will be repetitive on an orbital basis in order to minimize onboard computer memory requirements.

Establishing the Observation Pointing Timeline Sequence

Once the target visibility windows were defined and the required timing margins were identified, the assembly of the observation pointing time could begin. Studying the digital guaranteed visibility window graphs (Fig. 12) for all desired science targets and navigation stars permits determination of an appropriate sequence for the observations. After the desired observation sequence is defined the maneuvers required to get from one target to the next need to be calculated. These maneuvers must be selected such that they satisfy all vehicle lighting constraints and minimize ACS gas consumption.

Maneuver Angle Determination

Computation of intertarget maneuvers was straightforward since the targets were inertially fixed and because only one of the three control axes is slewed at a time. The first step was to convert all targets to the same celestial coordinate systems. Targets were expressed in right ascension and declination in 1950 Epoch. No consideration was given to parallax errors or velocity aberration effects because they were too small to be of concern. The target coordinates are the directions in which the negative roll axis (experiment and star tracker look vectors) are to be pointed. Once the sequence of targets was given, the next step was to establish the payload azimuth angle on each target, i.e., the angle between the payload yaw axis and the meridian passing through the target coordinates. The selection of azimuths started with the requirement that only one maneuver could be made from the first to the second update star. Other target azimuths were then made to minimize maneuvers to the update stars.

Because the payload slews are only about the control axes, three ordered angles are generally required to maneuver from an arbitrary three-axis orientation to a second orientation. These are Euler angle sequences calculated using spherical trigonometry for great circles. Computer programs that perform this calculation for rocket payloads were used with no modification other than to provide five places of accuracy.

The set of Euler angle sequences from one orientation to the next, however, is not unique. For Spartan 101, it was decided to employ sequences of ROLL-PITCH-ROLL or ROLL-YAW-ROLL. Since ROLL maneuvers are rotations about the experiment look axis, either maneuver sequence will scan the experiment look through the shortest angular distance between the two targets.

It was obvious that storage requirements for the pointing program could be minimized by reusing repetitive maneuver sequences when possible. Although a given science target was scanned at different azimuths on different orbits, the payload was always returned to a fixed orientation before proceeding to another. Thus, the intertarget maneuvers did not vary from orbit to orbit. For Spartan 101, the set of maneuvers over one orbit carried the star tracker/experiment axis through angular path including two science targets and two update stars. Such a path is called a "science circuit."

After all the maneuvers in a circuit were calculated, their accuracy was verified by taking the total maneuver sequence and calculating a corresponding sequence of coordinate transformations starting with the orientation at the completion of acquisition. Checks were made through the sequence to ensure that critical body-fixed vectors were pointed to the desired targets.

The vehicle maneuver rates were previously selected by analysis of a host of ACS parameter optimization and sensitivity tests.

Assembly of the Observation Pointing Timeline

Once the observation and maneuver sequence was established, the final assembly of the pointing timeline could begin.

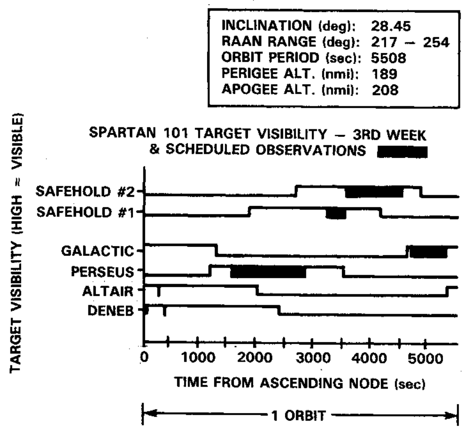


Fig. 15 Illustration of a scheduled observation sequence for one science circuit.

Since the maneuver rates were determined, the maneuver angles and rates could be combined with attitude fine mode capture times and sequencer load/verify timing overhead in order to determine the amount of time that must be allocated between the various target observations. In some cases, it was found that there was not sufficient time between the closing of an observation window and the opening of the next window to execute all the maneuvers and hardware/software overhead functions. When this occurred, the experimenter was forced to prioritize his targets so that the observations could be tailored accordingly. In other cases, there was excess time available between sequential target observations. In this case, timed waits at safehold attitudes were executed. These safehold parking locations were necessary in order to satisfy the vehicle lighting constraints while waiting for the next bona fide target to rise out of occultation into its guaranteed visibility window.

The two ACS navigation stars were scheduled to be tracked while in the center of their guaranteed visibility windows. The placement of these two navigation star observations establishes a starting point for assembling the remainder of the observation timeline. Science target observations were made as long as possible within the scope of the vehicle viewing constraints and design strategy guidelines. The total observation sequence is shown in Fig. 15. The two science targets (the Perseus Cluster and the Galactic Center) were observed once per orbit, every orbit, for the duration of their guaranteed visibility. The sequence repeats on an orbital basis and uses the orbit nodal period as the time reference. The repetitive nature of the timeline was chosen by design in order to minimize software memory requirements. Since any given target is visible on one particular orbit for a time period that is much longer than its guaranteed visibility window, it is apparent that this approach to target observation scheduling is suboptimum. However, it is a very practical approach, which balances the simplicity (and hence low-cost) of the spacecraft against the desire to do significant science in a timely manner.

Conclusions

The basic mission planning techniques described in this paper were used to construct the observation timeline that was executed during the flight of Spartan 101. The success of the flight provided the soundness of these concepts. The vehicle successfully acquired its initial desired attitude and updated its attitude via the navigation stars exactly as planned. Most of all, the mission experimenter obtained large amounts of scientific data that offer the promise of revealing new and exciting information about the universe.