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DESIGN OF THE TOPSAT MISSION*

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The currently preferred design for a global topography satellite mission poses a challenging task in orbit design and navigation since it requires close formation flying of two radar antenna equipped satellites. This paper describes the origin of the orbital requirements and some methods to achieve success. Preliminary analysis indicates that the mission is technically sound but would require precise navigation and intensive mission operations.

INTRODUCTION

Although the topographies of certain small areas of the Earth's land masses are well known, there is a need for a high resolution global map. Vertical accuracy of 5 meter or better, on a horizontal grid of 30 m would be a useful product to a number of customers. This paper presents the interesting astrodynamics involved with a dual satellite mission concept which was the leading candidate chosen out of an intensive study of so-called TOPSAT missions carried out at JPL. The basic mission would provide the required data for all the land masses between ± 70° latitude. The recently proven InSAR (Interferometric Synthetic Aperture Radar) method would be used to obtain the accuracy needed.

MISSION CONCEPTS

Three different methods of obtaining the InSAR images were considered. The simplest method which has been used in the past on such missions as Magellan, uses a single spacecraft with one antenna which remaps a certain area from a slightly different angle the next time the orbit flies over the area in question. Because the data are taken at different times there is some difficulty doing the orbit determination to reconstruct the geometric relationship of the two measurements. However, the biggest drawback of this

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method is the temporal correlation of the data which is lost due to the extended time period between the two measurements. This is a critical problem on the Earth where meteorological and biological events can change the surface properties over a short time scale, Thus this 'Repeat Pass Method' was not chosen as a viable option though it would be used as a back-up if one satellite in the dual satellite configuration described below happened to fail.

The primary choice is a dual satellite configuration with a L-band radar antenna on each satellite. One antenna would be used to transmit a signal which would be received at both satellites. The two satellites must remain in close proximity to achieve the proper measurement accuracy. Achieving and maintaining these two orbits is the subject of this paper.

The best alternate design is a single satellite with a pair of Ka-band antennae separated by an extended 12 meter boom. While the dual satellite requires precise navigation and reconstruction of the satellite to satellite distance, the single satellite requires highly accurate knowledge and control of the linear and angular relationship of the two antennae.

Other radar options looked at in lesser detail were C and X bands on a dual satellite system and X and Ku bands on a single satellite mission. All mission options now include a laser altimeter on each satellite. Altimeter measurements can be used to determine ice sheet topography or as a calibration check on the lnSAR measurements.

THE DUAL SATELLITE MISSION

Two preliminary designs for the dual satellite option were created for two of the different launch vehicles that could be used, The preferred configuration is two identical triangle trusses attached together inside a 40 foot shroud to be launched by a Titan 11S. Both the radar antennae and the solar panels would be rigidly attached to the truss structure. The alternate launch aboard a Delta 117920 would necessitate stacking the two satellites vertically and having deployable radar antennae and solar panels.

The satellites are injected together into the 565 km orbit where they are separated from the final stage still attached to each other. Below is a description of two methods to perform the subsequent separation of the two satellites into their initial configuration. After obtaining a good understanding of the flight properties of the satellites, in particular the differential drag experienced by the two satellites when passing through the same medium, the satellites can be brought together to their nominal separation for measurements. After calibrating the instruments, the mapping begins. In less than 84 days, a full global map is obtained. By launching at the proper time of the year into a "6AM-6PM" sun synchronous orbit, two of these global maps can be obtained well before entering the first solar occultation.

ORBIT PARAMETERS AND SWATH PATTERN

The radar look direction is perpendicular to the velocity direction and 30° off nadir (nominally chosen to the right). The dimension of the footprint in this direction is 35 km. However, to have a 10% overlap, the orbit was picked to have adjacent swaths 31.5 km apart. The following equatorial elements represent a sun synchronous, frozen orbit which creates these swaths by having a period which is slightly greater than one which repeats every 15 revs in one day.

a = 6942.811 km	h = 564.677 km
T = 5764.568 sec	i = 97.905°
e=- 0.001063	ω= 90.0°

Thus, as shown in Figure 1, the 16th rev is placed to the west of the first one. The groundtracks continue to drift westward until after 83.1 days (1246 revs) a global map is completed. The drift of the swaths due to drag should be small compared to their overlap if the drag make-up maneuvers described below are done properly. Note the small triangle depicted in Figure 1 which indicates that the rotation of the Earth causes the angle with which the swaths cross the equator (783°) to be more acute than the supplementary angle to the inclination (82.0950).

RELATIVE SATELLITE POSITIONING

The baseline distance is defined as the component of the satellite separation perpendicular to the velocity, as shown in Figure 2. It must be in the range of 800 m to 2 km for proper interferometric results. As shown in Figure 3, having two orbits which are identical except for a 2.02 km difference in the node crossings gives a baseline separation of 2 km at the equator and 8(KI m at 65° latitude. Because of the denser groundtracks at the higher latitudes (see Figure 4), good results can actually be obtained up to about 70° latitude. Coverage between ± 70° includes almost all the land areas of topographic interest. By increasing the equatorial baseline a little bit beyond 2 km, sornc more of the unmapped area could be covered in an extended mission if this is desired. The laser altimeterwill be taking high resolution measurements of the ice surface at latitudes beyond *70° during the nominal mission, Of course, due to the inclination of the orbit, the maximum latitude of the groundtrack is 82°. The resolution of the nadir pointing laser is depicted in the cartoon of the groundtracks shown in Figure 4.

The lag distance is the component of the satellite separation measured in the velocity direction. It is of utmost interest for the navigation of the two satellites. If the second satellite is at Position 1 on Figure 2, that is with no lag, the satellites will theoretically collide at the point where the two orbit planes intersect. However, at Position 2 where the satellites are at the equator at the same time, the lag is a 278 m. Since the lag distance in the Keplerian case remains constant, these two satellites will always be at the same latitude as each other while the baseline separation varies between ± 2 km throughout an orbit. This is the idealized design for the orbits; however in order to have a margin to accommodate the effect of small differential perturbations on the true orbits an extra lag of 400 m, shown as Position 3, is included in the final choice of orbit parameters. Note that there is a science requirement to keep the lag less than 1300 m to insure that there is adequate overlap of the footprints in the direction of the velocity (see Figure 1).

SEPARATION ANALYSIS

The two simple methods of putting the two satellites in separate orbits can be thought of as creating a lag distance and then establishing different orbital planes or vice versa putting them in different planes and then creating the lag. 'l'he following describes this in more detail.

In the first scenario, assume the two satellite payload is in a circular orbit at the final desired inclination, If a set of springs is used to separate the satellites by giving one a Δv in the initial velocity direction while the other gets a Av in the opposite direction, the former will get a slight increase in period (and apogee height) while the latter will get a decrease in period (and perigee). These would not have to be large Δv's, for example a 11.6 mm/s Av would create an along-track difference of 200 m every half rev. When the desired initial separation of a couple of kilometers was reached, the satellites could be re-circularized. The orbit plane separation could then be done by using the propulsion system to burn near the maximum and minimum latitudes in a direction perpendicular to the radius and velocity. ‘J’he burns would be in opposite directions for the two satellites, away from the equator to regress the node of the leading satellite, but towards the equator to progress the node of the lagging satellite. These burns would produce the desired 2.02 km separation in nodes.

In the second strategy, the combined satellites are launched into a slightly less (5.7 mdeg) than nominal inclined orbit but still assumed circular. At the highest (or lowest) latitude point, a spring system would be used to impart a Av of 2,36 m/s to each satellite in opposite directions perpendicular to the velocity and radius vectors. This results in one satellite on the ascending leg just before the nominal high latitude, while the other is on its descending leg as if it has just passed the high point. Thus, the node separation is achieved, ‘J’he two satellites would theoretically collide half an orbit later, so the satellite on the ascending leg has to be given an increase in velocity and/or the other satellite must get a decrease in velocity. Again, to achieve the difference in period these burns do not need to be large, a total Av of 5.8 mm/s gives the 100 m per half rev separation. Again, the satellites are re-circularized after the desired separation is reached.

Other variations of these methods exist, such as using the springs to impart most of the node separation and the along track component at the same time. This would involve further burns to refine the asymmetries in inclination. A detailed analysis which includes the attitude control and propulsion factors is needed to determine the optimum strategy. 1 lowever, the preliminary conclusions that can be drawn from the above two methods is that the first method does not take advantage of the Av available from standard springs and requires large propulsion bums but is relatively safe. The second method is efficient in its manner of obtaining its velocity changes but relies on a crucial burn right after the spring separation. Both of the descriptions above have ignored the differential drag effects due to different orientations of the two satellites which is discussed below.

NAVIGATION STRATEGY

The navigation of TOPSAT can be divided into the absolute control of the pair of the satellites and the more complex relative positioning with respect to each other. The Global Positioning System (GPS) will be used for both purposes. A relative knowledge of 3 mm in position is needed to properly reconstruct the interferometry measurements. This high accuracy also permits the precise navigation that is required to maintain the satellites at the proper separation.

In a comparative sense, the absolute navigation requirements are less severe, however the atmospheric drag at this altitude will require both satellites to be raised on the order of 100 m every 10 days. The actual frequency of these drag make-up maneuvers, will be dependent upon the highly variable atmospheric density. It is important to note that a 1998 or 1999 launch would imply lower densities to be encountered while a later launch would be in the solar maximum period, thereby resulting in higher densities and more frequent maneuvers. The absolute error requirement is mitigated by the fact that the antennae can be tilted slightly to maintain the proper swath pattern on the ground. Nevertheless, the drag make-up maneuvers still have to be done accurately to avoid differential errors,

In between these drag make-up maneuvers there would have to be Relative Station-Keeping (RSK) maneuvers about every 3 days to maintain the proper separation distance. Another possibility is to very accurately do one third of the drag make-up every time a RSK was performed. Preliminary analysis has indicated that the two major effects which contribute to separation changes are differential drag and the radial orbit difference induced by maneuver execution errors, The mission design requires that the two satellites bc of identical construction, particularly in cross-sectional area and mass, However, a conservative *a priori* value of 2% was assumed for the possible deviation of the drag effect of either satellite from nominal, The relative drag effect should be estimated to a high precision from early tracking results.

An earlier analysis made some incautious assumptions but provided some precursory results which produced three conclusions: a) If the IZ zone is defined to be between 280 m and 1100 m lag distance, the first time that the IZ is entered, the nominal value of 680 m should not be used as the target but a more conservative value (890 m was chosen) should be used b) With maneuver errors of 0.1 mm/s, there can be a resultant oscillatory behavior in the lag distance of a magnitude that is a substantial potion of the IZ. 'l'his resulted in a tighter requirement of 0.05 mm/s being recommended and accepted by the propulsion engineer c) The differential drag uncertainty should be reduced to less than its *a priori* value of 2% before entering the IZ. These ideas were incorporated in the following analysis. Figure 5 should be used for reference.

Initially, the lag distance is assumed to be about 2 km after the initial separation and check out of the satellites. The differential drag uncertainty is assumed to be its *a priori* value of 2%. The orbit determination done by the GPS is very accurate, so for this analysis the differential position uncertainty is considered negligible for the epoch when the orbit solution is determined. In this case a solution is obtained one day before the sequence starts (Day - 1), which predicts that the separation will beat a specific value (1985 m was chosen arbitrarily) one day later, With a 2% drag uncertainty and the Standard 1977 atmosphere, this implies a ± 15 m uncertainty at Day O. A maneuver design occurs between Days -1 and O and is executed at day O with a target of 890 m (halfway between 680 and 1100), Both satellites are assumed to have an execution error of 0.05 mm/s in opposite directions. In the case where the lead satellite gets an excessive boost and thus slows down and the lagging satellite gets the opposite, there will be an excess decrease in the lag distance. In this case the unknown drag of the lagging satellite is assumed to be larger than expected so that the lag distance again will be decreased. This combination of events gives the lower bound of the lag distance error bars, as shown in Figure 5. The similar but opposite set of events gives the upper bound.

With the additional tracking, the drag uncertainty is assumed to be 1% between Days O and 3. This is combined with the above described execution errors to give an uncertainty in the control of the satellite lag at Day 3 to bc ± 161 m. The lower drag uncertainty does give a resultant improvement in the lag knowledge to ±8 m on Day 3, assuming, as before, a perfect orbit determination solution on Day 2, The maneuver designed between Days 2 and 3 and executed on Day 3, now chooses 680m as the target.

The next series of results follows the same pattern with the drag uncertainty decreasing by a half while the maneuver execution errors remain the same. The minimum value of drag uncertainty assumed was 0.125% with a resulting control band off 89 m and a knowledge of ± 1 m. Originally it was assumed that the differential drag could be estimated to a high precision (like 0.1 %) and compensated by a small difference in altitude. However, since a constant change in altitude gives a linear change in lag while the differential drag gives a quadratic effect and is dependent upon the absolute drag which always has some prediction uncertainty, this compensation will always have some residual error. Also, recent discussions suggest that here may be uncorrelated errors in the attitude of the two satellites which would result in a random unestimatable drag difference. Investigations of methods to minimize the drag differences such as curved shields on the leading faces of both satellites have been suggested,

SUMMARY

Preliminary results have shown that the dual satellite TOPSAT mission is a viable option from a navigation standpoint, A separate covariance analysis by the GPS measurement group at JPL has confirmed the results. Further mission design work along with the other subsystems analysis is planned for the near future.

ACKNOWLEDGMENT

Several subsystems such as radar science and propulsion aided in the understanding of their specialties to help do this study. The navigation and tracking sections at JPL both performed complimentary analysis which supported this effort. Roy Kakuda produced some of the original concepts for the orbit design of the mission as well as handling the mission engineering which combines the science and spacecraft requirements with the astrod ynamics involved,

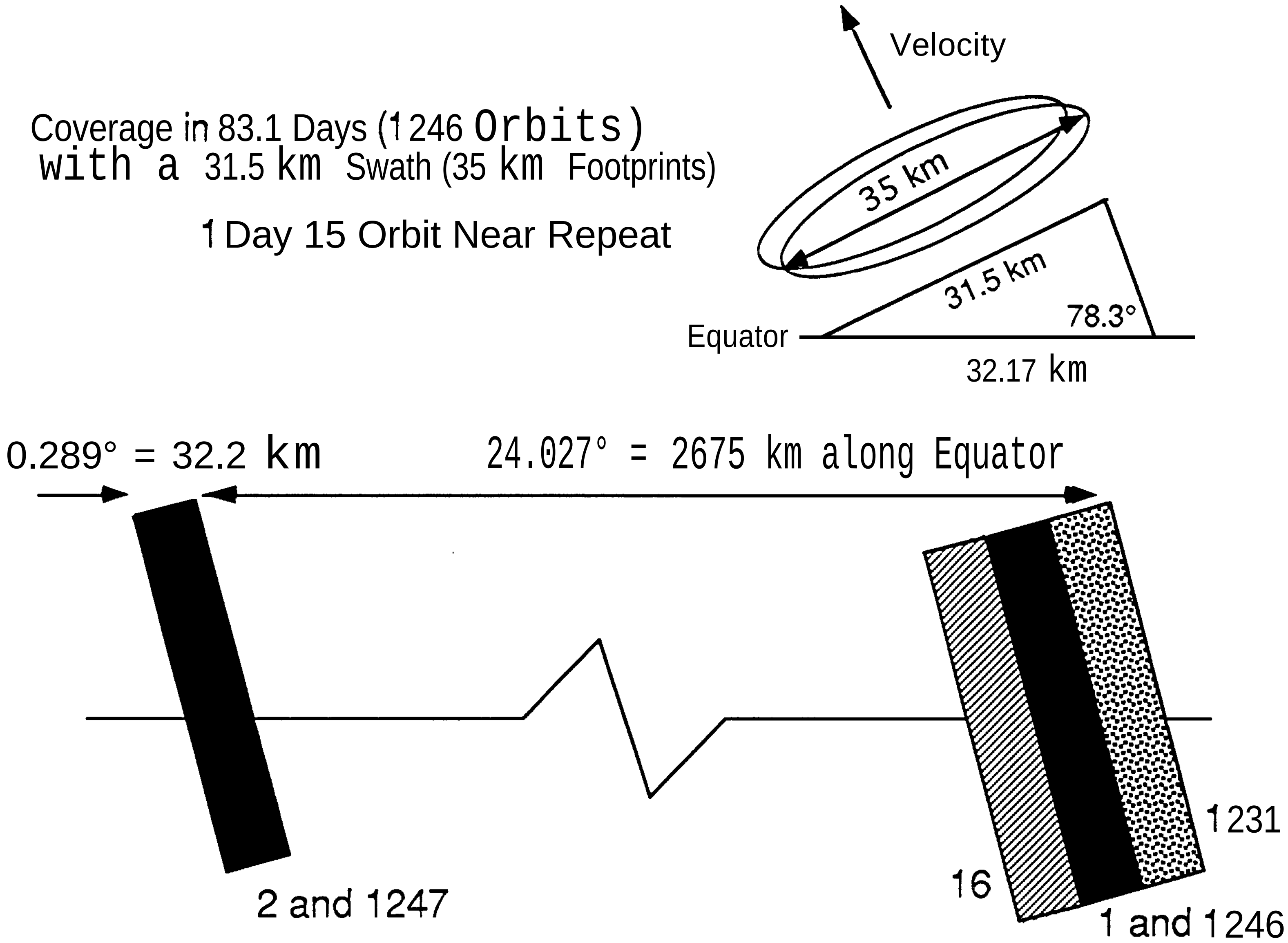


Figure 1 Swath Pattern

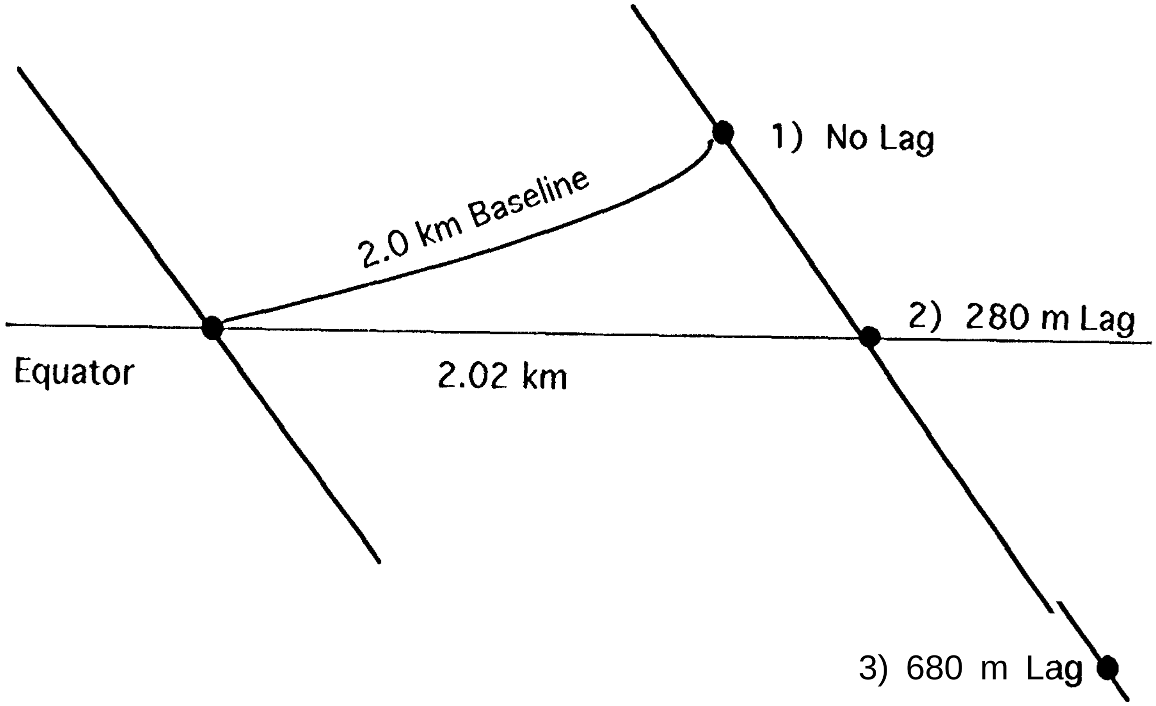


Figure 2 **Orbit Geometry at the Equator**

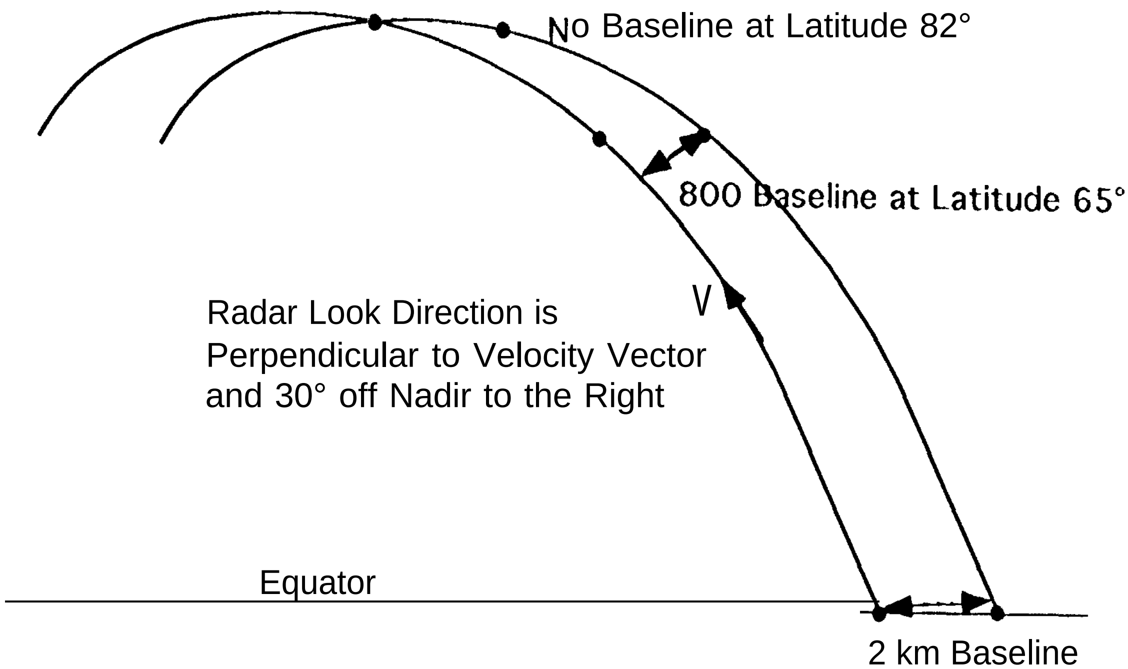
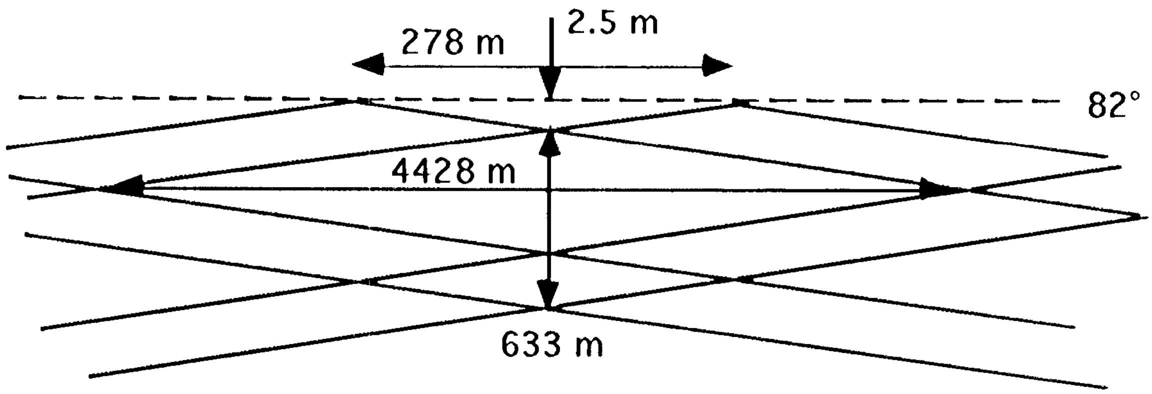
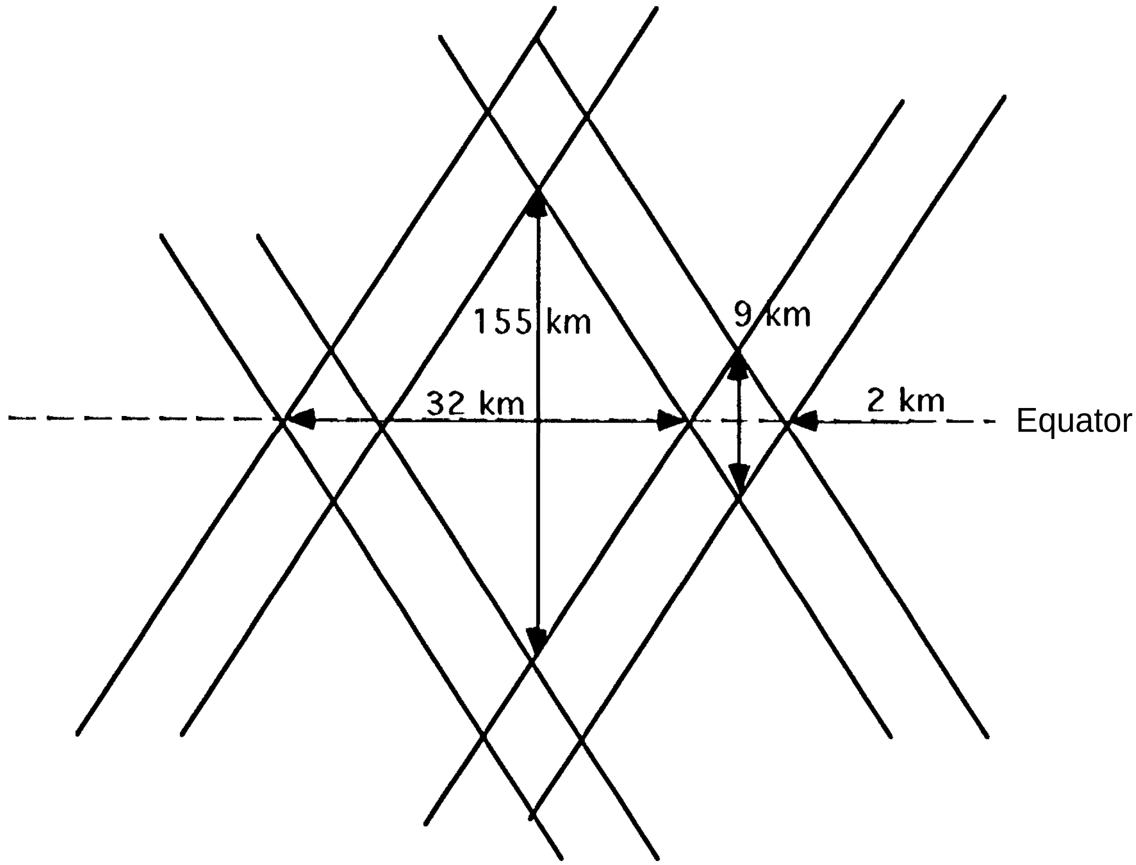


Figure 3 **overall orbit Geomet ry**



High Latitude



Low Latitude

Figure 4 Ground Track Pattern

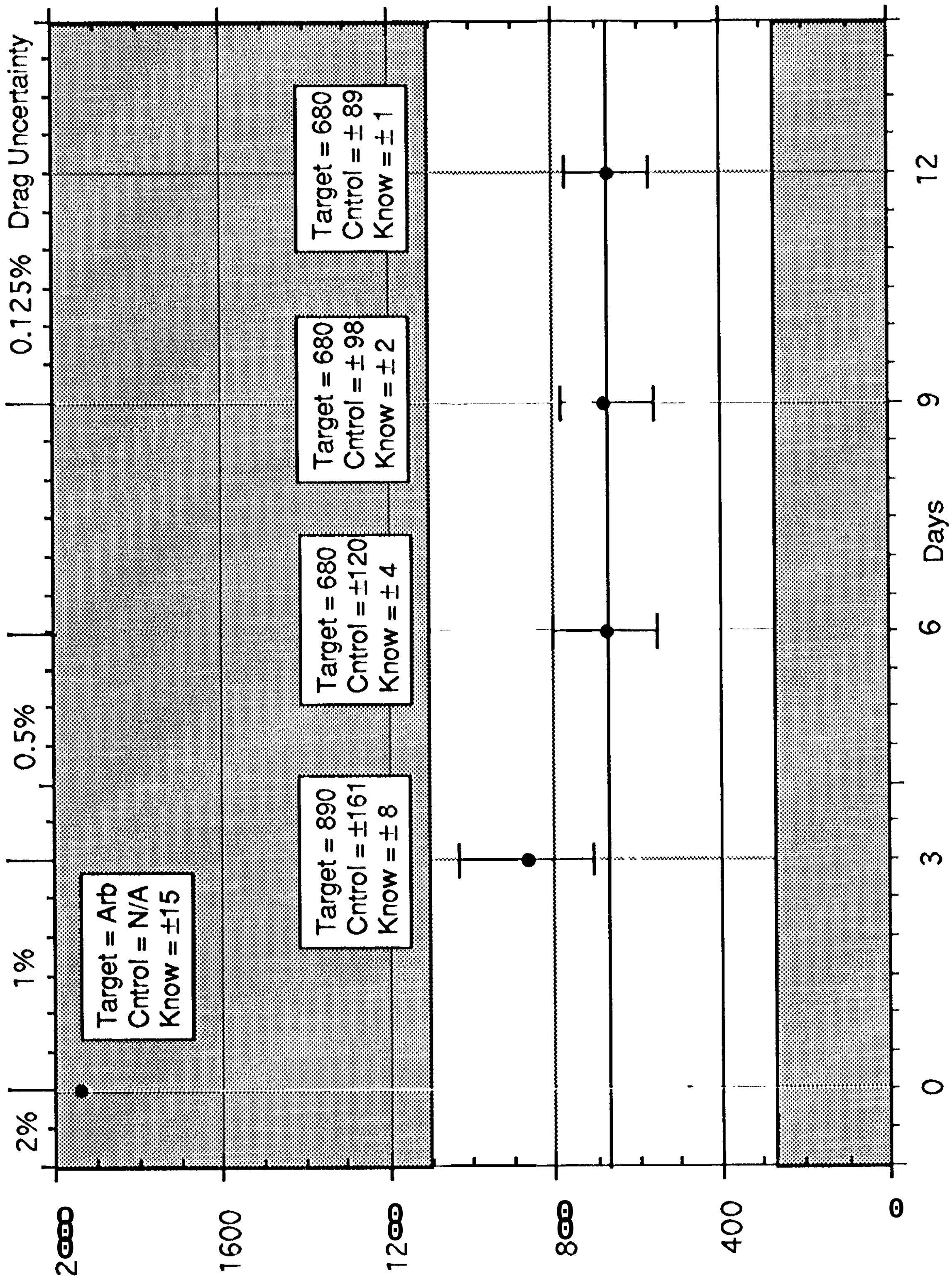


Figure S **Worst Case Initial Maneuver Sequence**