NAVIGATION AND ATTITUDE REFERENCE FOR AUTONOMOUS SATELLITE LAUNCH AND ORBITAL OPERATIONS

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ABSTRACT

This paper investigates the navigation and attitude reference performance of a strapdown system for applications to autonomous satellite launch and orbital operations. It is assumed that satellite payloads are integrated into existing missile systems and that the boost, orbit insertion and in-orbit operations of the satellite are performed autonomously without relying on external support facilities. Autonomous and long term accurate navigation and attitude reference are provided by a strapdown inertial navigation system aided by a star sensor and earth landmark sensor. Sensor measurement geometry and navigation and attitude update mechanizations are discussed. Performance analysis data are presented for following functional elements:

a. prelaunch alignment,
b. boost navigation and attitude reference,
c. post boost stellar attitude and navigation updates,
d. orbital navigation update using sensor landmark measurements,
e. in-orbit stellar attitude update and gyro calibration.

The system performance are shown to satisfy the requirements of a large class of satellite payload applications.

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1. INTRODUCTION

This paper explores the feasibility of a guidance system using currently available sensor technology for autonomous satellite launch and orbital applications. One possibility of such applications may be stemmed from considerations on the survivability of satellite functions vital to national defense. Reconnaissance and communication payloads may be integrated into existing silo or submarine based missile systems to enhance their initial survivability. These payloads will be boosted and deployed without relying on external launch support and mission control facilities that may not be available during a national emergency. To support such mission applications a system consisting of onboard sensing and data processing elements is required to provide the navigation and attitude reference functions with necessary accuracy over the duration of the mission.

This paper considers an Autonomous Navigation System (ANS) consisting of an IMU, a star sensor, a down sensor for landmark sightings, a radar altimeter, a long term stable clock, and an on-board computer and resident softwares. To facilitate the discussions here, two autonomous satellite launch missions are hypothesized. The first involves the boost and deployment of a payload satellite into a 185 Km circular orbit, the second, a 370 Km x 40000 Km 12-hour Molnya orbit after temporary dwellings in parking orbit for appropriate orbit phasing. These two missions are believed suitable for a wide class of satellite payloads and within the capabilities of available booster candidates. To provide navigation and attitude reference functions supporting these missions, the ANS system elements are mechanized into the following operation modes:

(1) IMU self alignment mode prior to boost,

(2) Pure inertial mode during boost and ΔV thrustings for orbit insertion and in-orbit maneuvers,

(3) Stellar inertial mode for post boost and in-orbit attitude update and gyro calibration,

(4) Orbital navigation mode for in-orbit navigation and update.

An artist's conception of ANS in an autonomous satellite mission is presented in Figure 1.

In the following discussions the ANS performances for each of the functional elements are assessed. The error budget assumed in the performance analysis is summarized in Table 1. The sensor characteristics are based on existing system capabilities (Reference 1.) The down sensor landmark sighting accuracy is projected from the engineering test model developed on the Autonomous Navigation Technology (ANT) program (Reference 2.) It should be pointed out here that results reported here do not represent a particular mission operation nor a specific system design. Rather, they should be interpreted as what can be achieved with currently available technology for functional elements likely to be included in any autonomous satellite applications.

2. IMU SELF ALIGNMENT

IMU self alignment is performed prior to boost. It involves the determination of the ANS attitude defined in terms of a coordinate transformation matrix between the inertially fixed navigation frame and an ANS fixed frame to which sensor measurements are referenced. It is accomplished through velocity matching processing supplemented by an optical azimuth reference. For a land based system the reference velocity is calculated from earth rate and known location of launch site. For a submarine based system the velocity and location of the vehicle are provided by a shipborne IMU. The level components of the self alignment errors are limited by the bias errors of the horizontal accelerometers and the reference velocity errors with the latter more significant for a mobile system than a land fixed system. Optical azimuth reference is required to supplement that obtained from a pure velocity matching. Without such an optical reference the azimuth error would become excessive because of the inability to reorient a strapdown IMU for calibrating out the gyro bias errors.

Assuming the ANS IMU and shipborne IMU characteristics, the alignment accuracy is found to be adequate to support a boost and orbit insertion within the landmark acquisition bounds for later navigation update.

Since satellite payloads are integrated into existing missile systems, it is possible that the booster IMU will be retained. A boost phase velocity matching may be performed to align the ANS IMU with respect to the booster IMU. Being a platform system with reorientation capability the booster IMU can be aligned with respect to the navigation frame more accurately than a strapdown system with equivalent quality sensors. In this case the azimuth alignment for ANS becomes non-critical. Discussions on a technique of in-flight alignment transfer is contained in Reference 3.

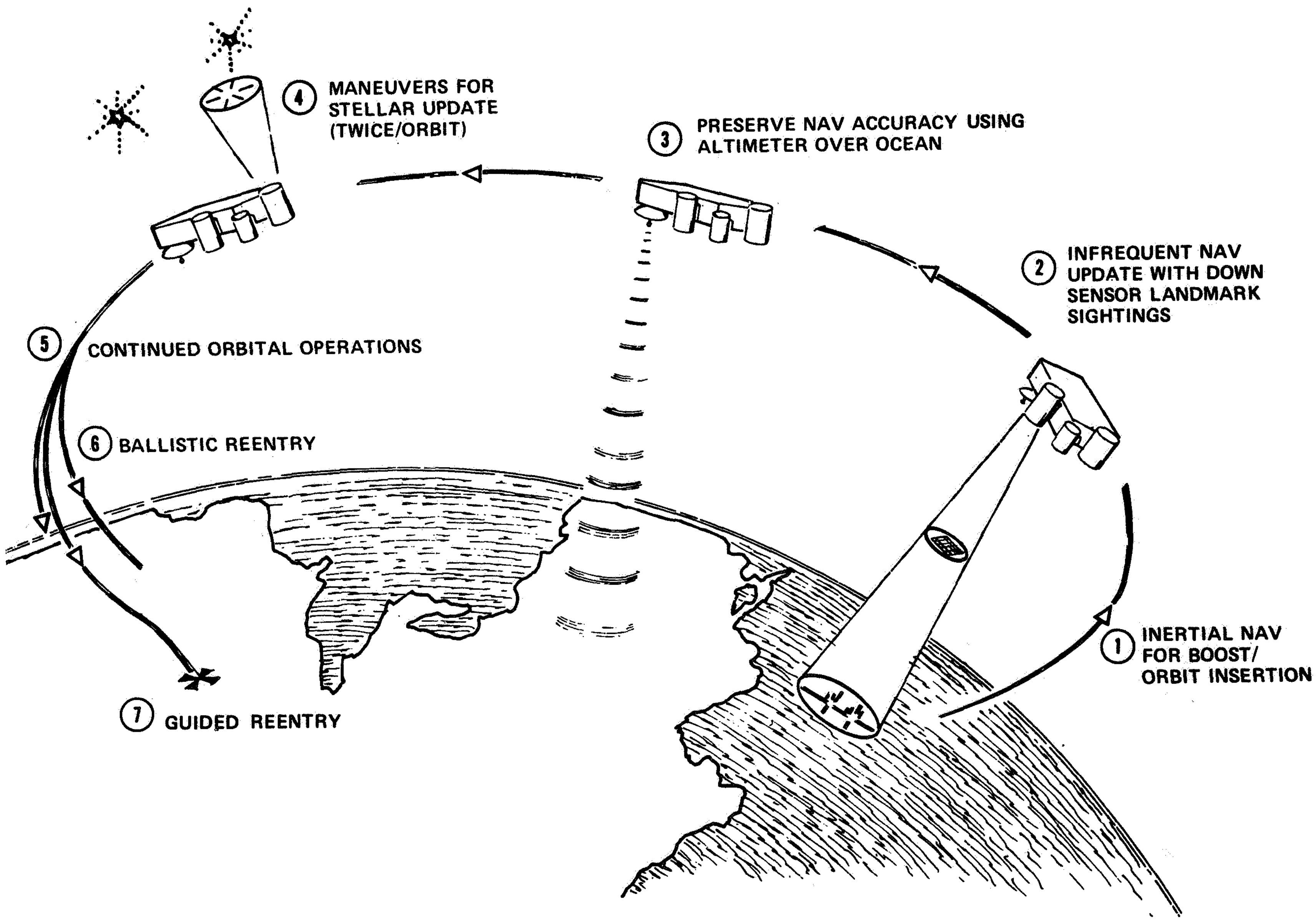


FIGURE 1. AUTONOMOUS NAVIGATION SYSTEM OPERATIONS BY MISSION PHASES

TABLE 1. ANS ERROR BUDGET (1 SIGMA)

GYRO:

● BIAS	0.04 DEG/HR
● SCALE FACTOR	75 PPM
● MUSA	0.02 DEG/HR/G
● MUIA	0.01 DEG/HR/G
● ANISO	0.01 DEG/HR/G^2
● MISALIGNMENT	13 ARC-SEC

ACCELEROMETER:

● BIAS	30 μG
● SCALE FACTOR	35 PPM
● SF ASYMMETRY	10 PPM
● MISALIGNMENT	7 ARC-SEC

STAR SENSOR:

● BIAS	6 ARC-SEC
● RANDOM	6 ARC-SEC

DOWN SENSOR:

● BIAS	30 ARC-SEC
● RANDOM	30 ARC-SEC

ALTIMETER:

● RANDOM	50 FEET

MODELING RESIDUE:

● GEOID ALTITUDE	45 FEET
● GRAVITY	1.25 μG
● DRAG: EXOSPHERIC TEMP	100^o /K
C_D	10 PERCENT

The alignment errors will propagate into position and velocity errors during boost. These errors can be significantly reduced via a post boost stellar update.

3. INERTIAL NAVIGATION

Inertial navigation is performed for all powered portions of an autonomous satellite mission including boost and ΔV thrustings for orbit insertion and in-orbit maneuvers. The gyro measured angular rate is integrated using the attitude initialized during self alignment. This attitude reference solution is used to transform the applied accelerations measured by the vehicle fixed accelerometers into the inertially fixed navigation frame. Vehicle position and velocity are obtained through integrating the measured thrust acceleration and the gravitational acceleration evaluated from a gravity model. These navigations and attitude reference solutions are used for generating booster steering and thrust termination commands during boost as well as the vehicle attitude control and ΔV thrustings for post boost maneuvers performed as part of orbit insertion and orbit transfer.

Boost navigation and attitude reference error sensitivities to IMU and initial alignment errors have been evaluated considering realistic missile boost trajectories. It is found that the initial alignment error dominates inflight sensor errors in their contributions to boost position and velocity errors. However, it is found that the boost errors are well within the acquisition bounds assumed for down sensor landmark sightings for in-orbit navigation updates.

Since the boost navigation states are computed using the attitude reference solution for coordinate transformation, the sensed acceleration provides the linkage between the navigation errors and the attitude errors. Through integration of the error propagation equation using the sensed acceleration as the driving function, it is possible to compute the correlation between the navigation error and the attitude error which is dominated by initial alignment error. This correlation allows the derivation of navigation error estimates from a post boost stellar attitude update.

The inertial navigation performance during thrustings for orbital maneuvers is evaluated assuming a ΔV level of approximately 2500 m/sec. This corresponds to the perigee burn required for the 12 hour Molnya orbit or a 18.5° plane change for the low altitude circular orbit. Assuming ANS IMU characteristics the velocity errors accumulated during the ΔV thrusting along the vehicle roll (nominal thrust direction), pitch and yaw axes are only 0.09, 0.26 and 0.22 m/sec (1σ) respectively. With these disturbances the total navigation errors are still within the down sensor acquisition bounds.

4. STELLAR INERTIAL ATTITUDE REFERENCE

The attitude solution obtained from integration of gyro angular rate measurements provide the basic attitude reference throughout the mission duration. Star sensor measurements are acquired for attitude and navigation update immediately following boost and for periodic in-orbit attitude update and gyro calibration.

The ANS star sensor consists of a telescope with a set of six detector slits placed on its image plane as depicted in Figure 2. A transit pulse is produced when the image of a star moves across a detecing slit. The basic star sensor measurement is the time when the star transit occurs. Each transit time measurement contains only one component of vehicle attitude information. Multiple transits from a single pass over the reference star can be combined to provide the equivalent measurement of the LOS-vector to the star. Vehicle maneuvers that provides repeated multiple scans between two reference stars (Figure 2) produce star transit measurements containing information for 3 axes attitude update as well as gyro bias and scale factor calibrations. A discussion of the star sighting strategy and Kalman filter mechanization for stellar attitude update is contained in Reference 4.

For post boost stellar update a single scan over a star located in the down range direction provides transits with maximum observability on the initial azimuth error, the dominant boost navigation error source. The navigation error can be obtained from stellar attitude measurements using the correlations between these two types of errors described earlier. Cross-track velocity error due to initial azimuth uncertainty can be reduced to 0.23 m/sec assuming ANS star sensor accuracy and 7800 m/sec vehicle velocity for 185 Km circular orbit.

In-orbit stellar update will be accomplished by performing the Figure 2 star acquisition maneuvers once per half orbit while the vehicle is over polar regions. Assuming ANS gyro and star sensor characteristics and 45 minute between update maneuvers the peak vehicle attitude errors can be bounded to approximately .15 m rad (Fig. 3) with gyro bias trimmed to 0.005 deg/hr (1σ) and scale factor calibrated to 25 PPM (1σ).

5. AUTONOMOUS ORBITAL NAVIGATION

After orbit insertion, the accelerometer output will be bypassed in the orbital navigation computation to avoid integration of the sensor bias error which is more significant than the input drag acceleration. An atmospheric density model will be carried onboard for drag evaluation. Due to errors in the initial conditions handed over from boost and uncertainties in the acceleration models, the error buildup of the orbital navigation solution requires periodic updates using down sensor landmark measurements. Additional navigation updates are provided by radar altimeter or horizon sensor measurements that are more frequently available.

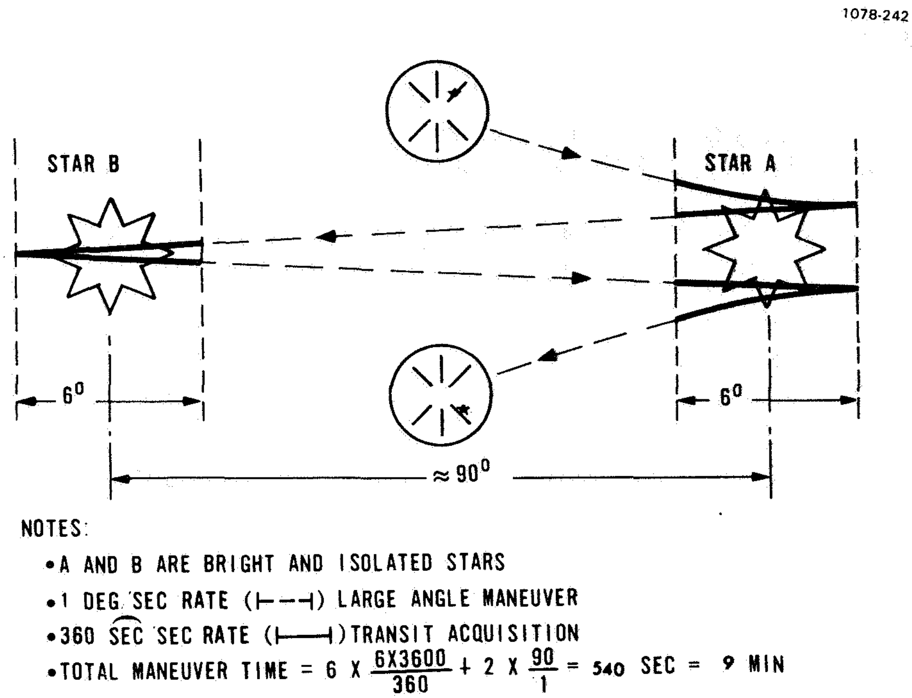


FIGURE 2. STAR ACQUISITION ATTITUDE MANEUVER PROFILE

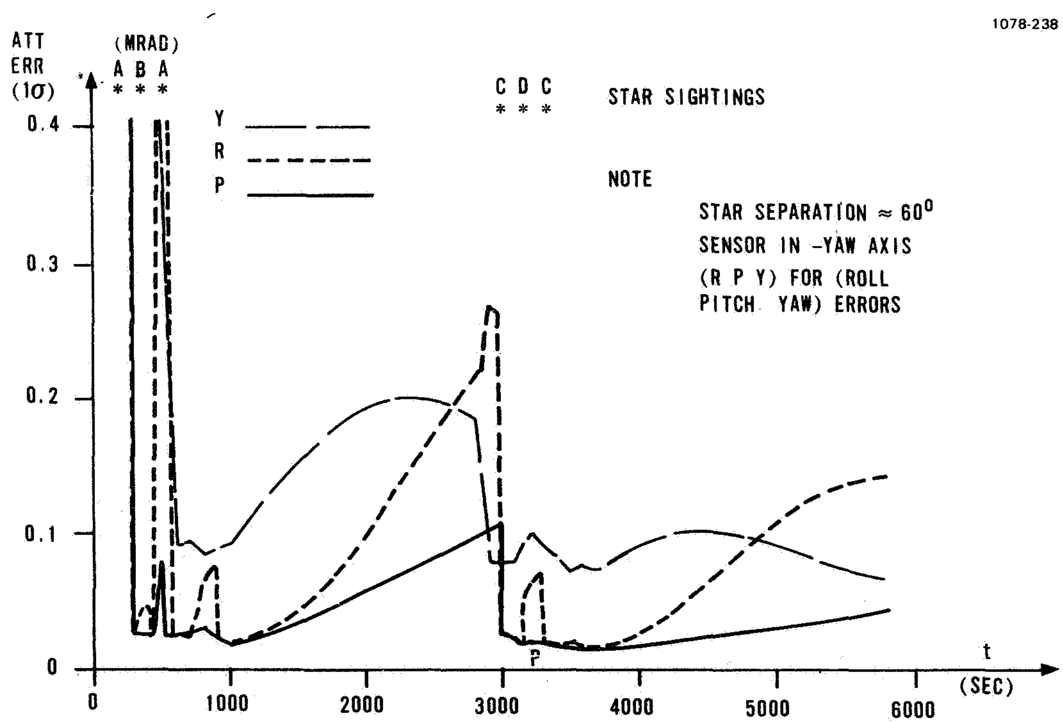


FIGURE 3. STELLAR INERTIAL ATTITUDE REFERENCE PERFORMANCE PLOTS

The ANS down sensor mechanization and navigation update concepts were originally developed on the Autonomous Navigation Technology (ANT) program performed for the Air Force (Reference 2). The ANT down sensor is an electro-optical device consisting of a telescope with two linear silicon detector arrays placed in its image plane as shown in Figure 4. Due to the vehicle motion, successive samplings of a detector array provide a digital image of the original terrain scene. Edge enhancement and feature extraction operations are performed to detect the presence of linear earth features (e.g. highways, rivers and coastlines), and to determine the orientation and the centroid of the segment of the feature appearing in the sensor Field Of View (FOV) as depicted in Figure 5. Upon down sensor linear feature detection candidate landmarks from an onboard catalog will be examined to identify the features sighted. The procedure involves a comparison of measured feature orientation against the cataloged landmark azimuth. The normal miss from the sensor look point to the candidate landmark that survives the orientation screening are then computed. The sensor look point is defined as the ground intercept of the LOS to the down sensor measured feature centroid. The normal miss is the minimal distance from look point to the linear landmark. The correct candidate is chosen as the one with the normal miss within an a priori tolerance. The landmark sighting is implemented for navigation update only if a unique candidate is identified. The normal miss will be the sensor measurement parameter used in the formulation of the Kalman filter for navigation update. Since the normal miss is a scalar each down sensor landmark sighting provides only one component of navigation state. As an example a landmark oriented normal to the orbit flight path will provide position update in the in-track component. Multiple sightings of landmarks with varied orientations provide complete observability in vehicle position and velocity. Details of navigation concepts are contained in Ref. 5.

In-orbit autonomous navigation performances are evaluated for the 185 Km circular orbit with the sunlit portion of the ground track shown in Figure 6. A down sensor landmark sighting schedule is hypothesized by first masking out the sunlit ground track using the average seasonal cloud cover probability. A 750 Km mean distance between linear landmark sightings is then applied to obtain the schedule shown in Figure 7. A covariance analysis program is utilized for navigation performance evaluation. Navigation errors for a three orbit period are plotted in Figure 7. These results are obtained assuming the down sensor, radar altimter and acceleration modeling errors defined in Table 1. Initial position and velocity errors of 2 Km/axis (1σ) and 2 m/sec/axis (1σ) are assumed. These errors are extremely conservative when compared with the boost inertial navigation performance. The stellar inertial attitude reference is not simulated in detail in the navigation analysis. It is characterized by a simple model consisting of 0.5mrad/axis (1σ) bias and 0.5 m rad/axis (1σ) random errors. Again these errors are conservative in view of the steady state attitude reference errors reported earlier. Navigation error plots shown in Figure 7 indicate that the high observability of down sensor landmark sightings enables rapid convergence from large initial

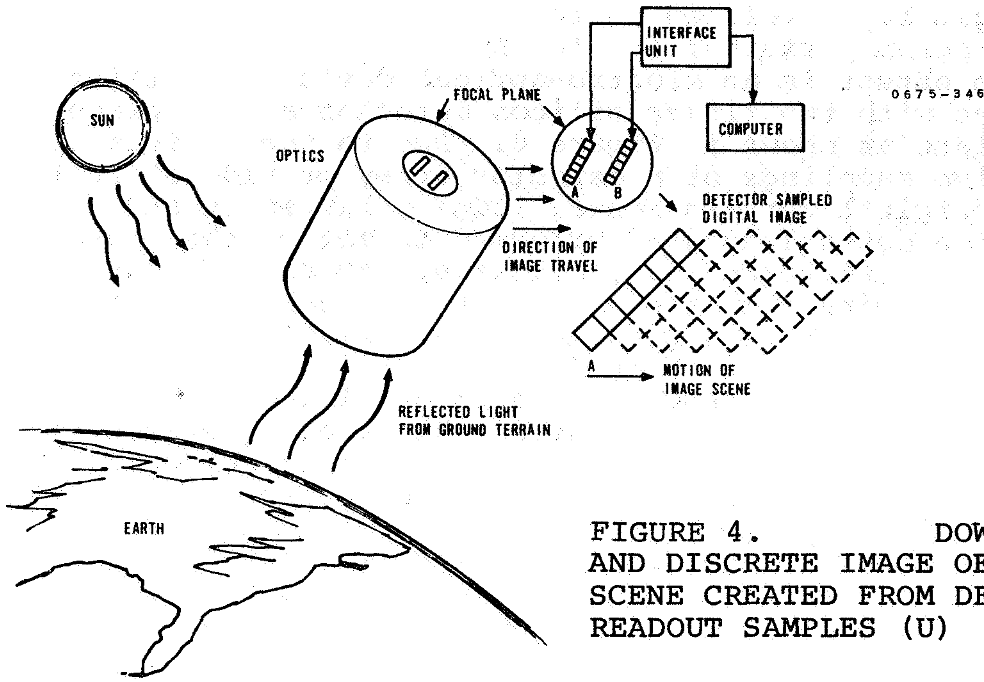
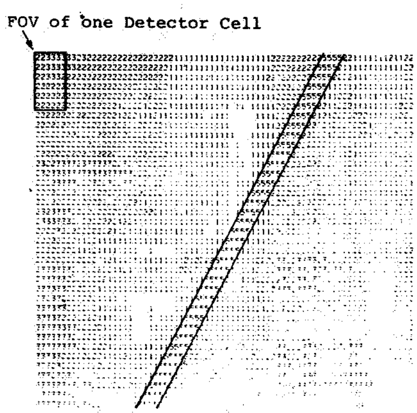
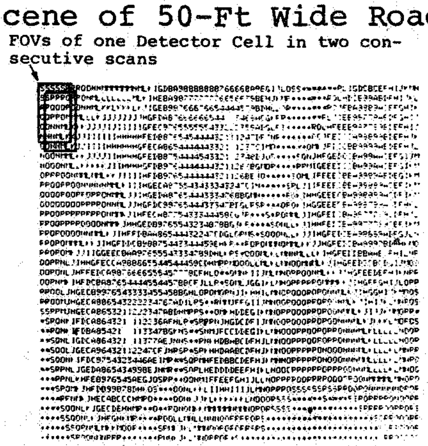


FIGURE 4. DOWN SENSOR AND DISCRETE IMAGE OF TERRAIN SCENE CREATED FROM DETECTOR CELL READOUT SAMPLES (U)

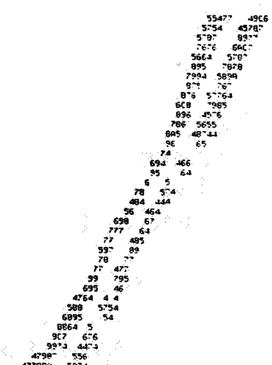


(a) Original Terrain Scene of 50-Ft Wide Road

(c) Thresholded Image of (b)



(b) Superpositioned Image From Consecutive Detector Readout Samples



(d) Gradient of Image of (c)

FIGURE 5. DOWN SENSOR LINEAR LANDMARK PROCESSING

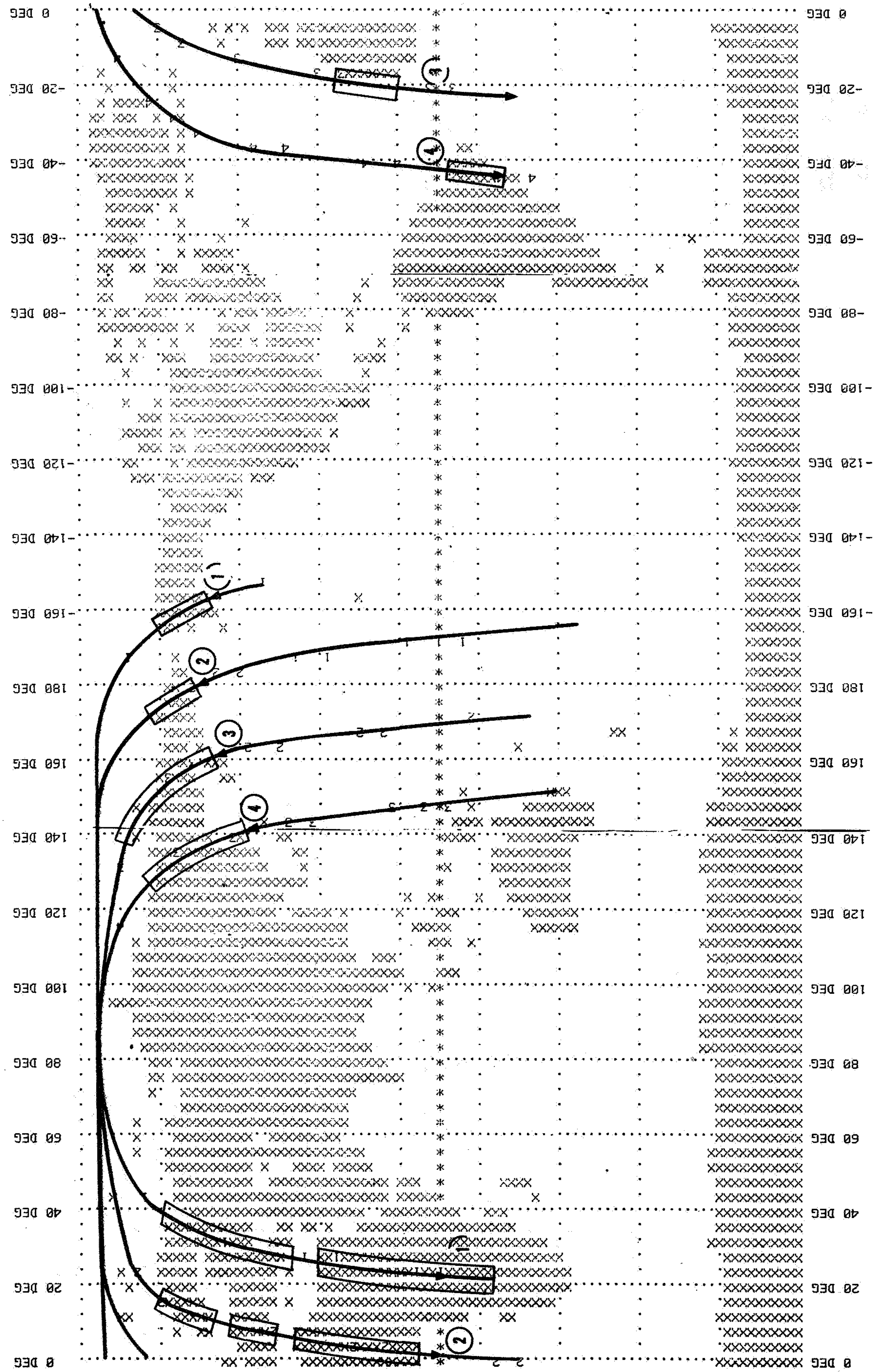


FIGURE 6. SUNLIT PORTION OF REFERENCE ORBIT CHOSEN FOR ANS IN-ORBIT NAVIGATION ANALYSIS

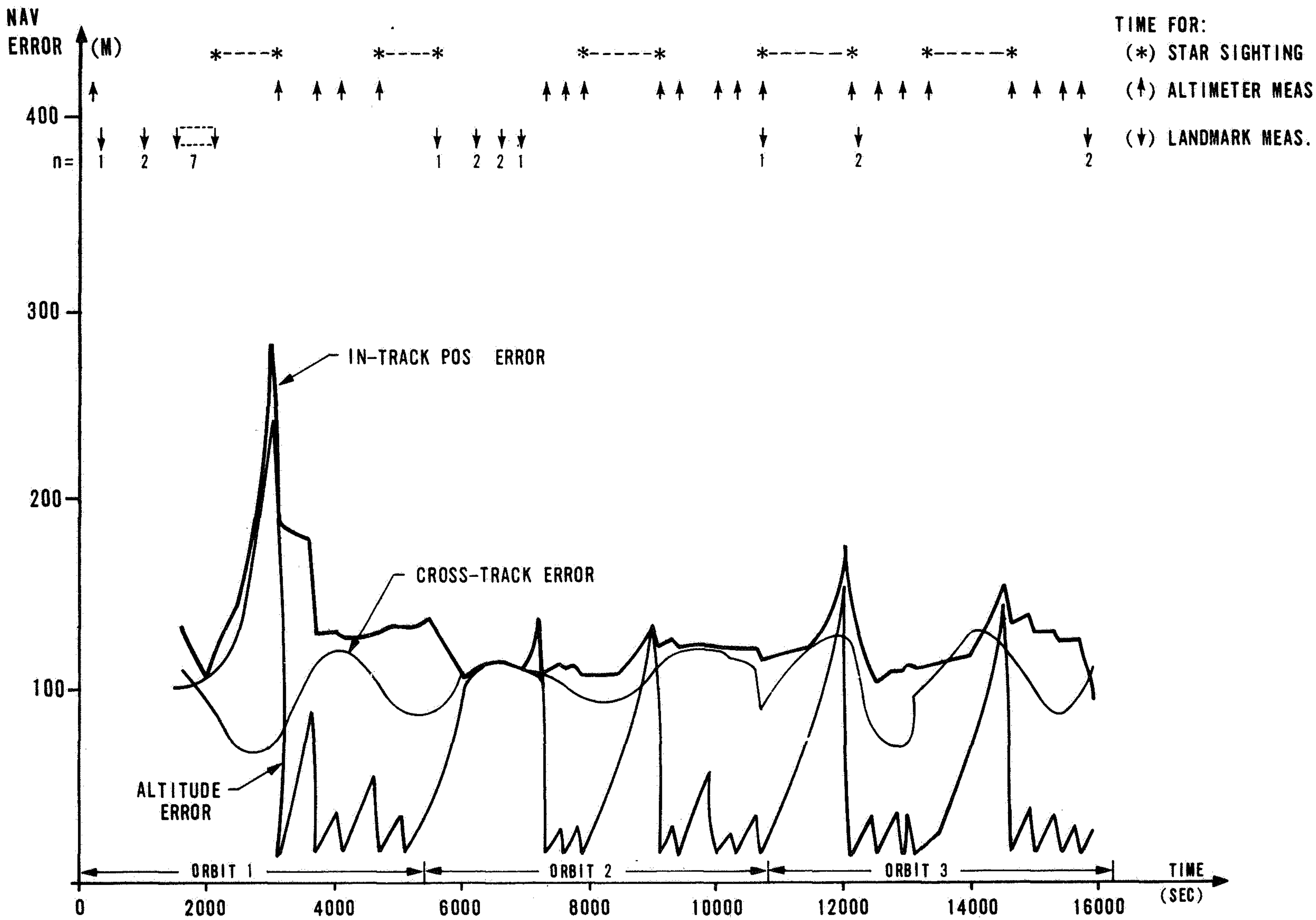


FIGURE 7. POSITION ERROR WITH LANDMARK UPDATES

errors. Also, navigation accuracy acquired from down sensor measurements can be preserved over intervals where landmark sightings are not available. This is accomplished by making radar altimeter measurements over the ocean where the geoid height can be accurately modeled onboard in $5^o \times 5^o$ grids. The cross coupling between the vertical and in-track components of navigation errors allows the bounding of in-track velocity error through vertical measurements. The ability to preserve navigation accuracy relieves the requirement on the landmark sighting frequency. A second case is run assuming identical conditions except that the landmark sightings after the first orbit are not implemented for navigation update. Navigation errors for this case are plotted in Figure 8 showing the ability of radar altimeter measurements in preserving the navigation accuracy over an extended period without landmark sightings. It should be pointed out here that once the attitude reference error is converged to steady state value the corresponding navigation error will be reduced to approximately 50 m (1σ) for in-track and cross track components.

Navigation performance for the 12-hour Molnya orbit is analyzed considering the ground track shown in Figure 9. To maximize down sensor landmark sighting opportunity it is mounted with the LOS pointed 15^o from the orbit plane and 15^o ahead of the nadir. It is shown that extending the limiting altitude of down sensor can significantly increase the landmark availability. The performance analysis results given in Figure 10 are obtained corresponding to h_1 = 1852 Km. Initial position and velocity errors assumed here are 500 m/axis (1σ) and 0.5 m/sec/axis (1σ). They are conservative compared with the navigation accuracy achievable in the low altitude parking orbit. Intrack and cross track position errors plotted in Figure 10 suggest that the LOS from vehicle to a point on Earth's surface can be computed to within 0.05 m rad at the apogee altitude using the onboard position information.

6. Summary and Conclusions

An ANS system concept has been described in this paper. It is shown how various system components can be mechanized to provide navigation and attitude reference supporting an autonomous satellite mission. The ANS performance has been evaluated for two hypothetical missions based upon existing sensor characteristics. The ANS navigation and attitude reference performances are evaluated for four major functional elements likely to be included in any autonomous satellite launch missions. These results show that it is feasible to develop an ANS system using existing sensor technology. In fact, most of the ANS functions (alignment, boost navigation, and stellar update) have already been demonstrated in various operational systems. The most critical ANS function is that of autonomous navigation update using down sensor landmark sightings. An engineering model down sensor has been built and tested in the ANT program. The remaining task is to prove the down sensor and navigation update concepts through a flight experiment.

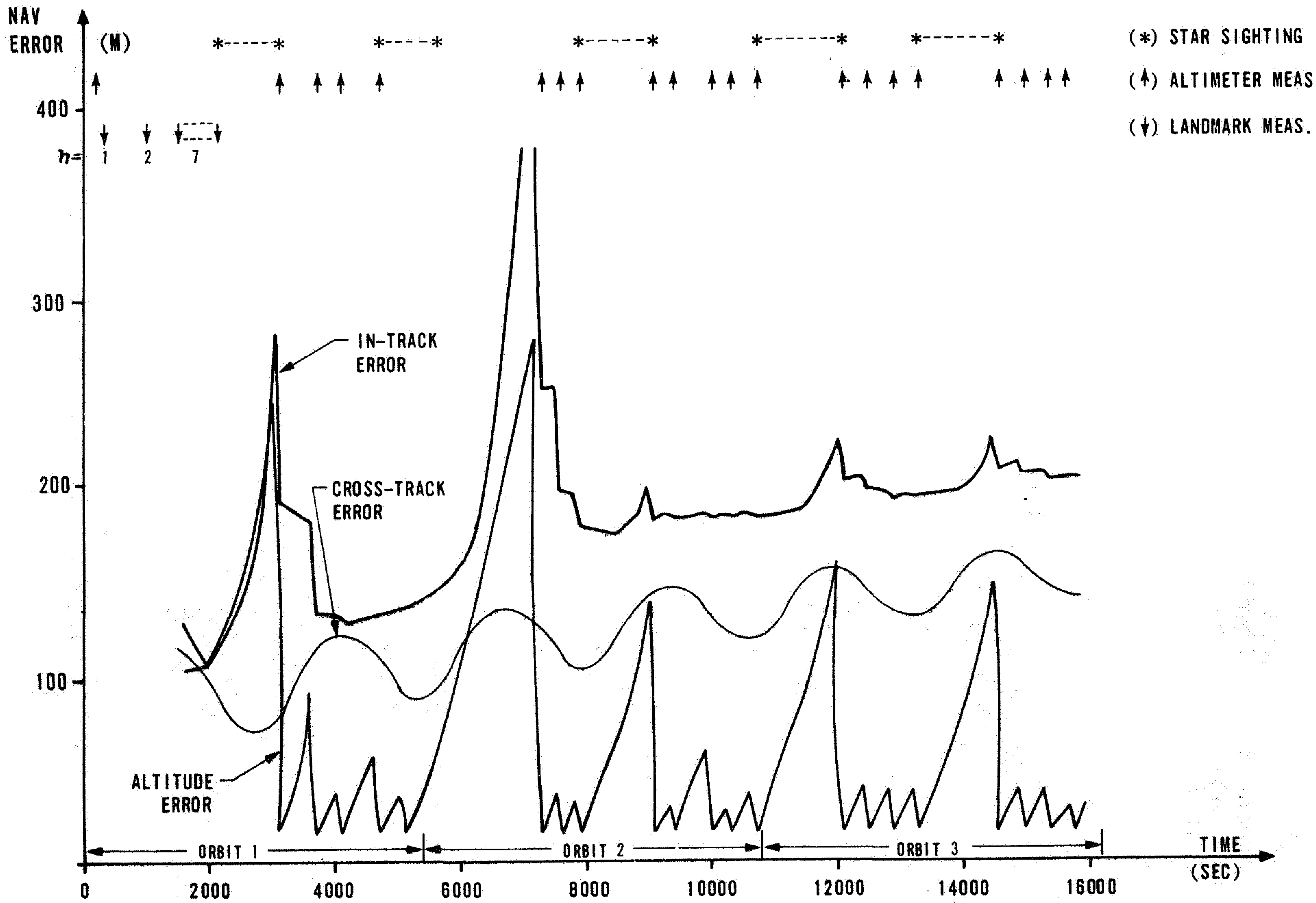


FIGURE 8. PLOTS OF POSITION ERRORS WITH 10 LANDMARK UPDATES

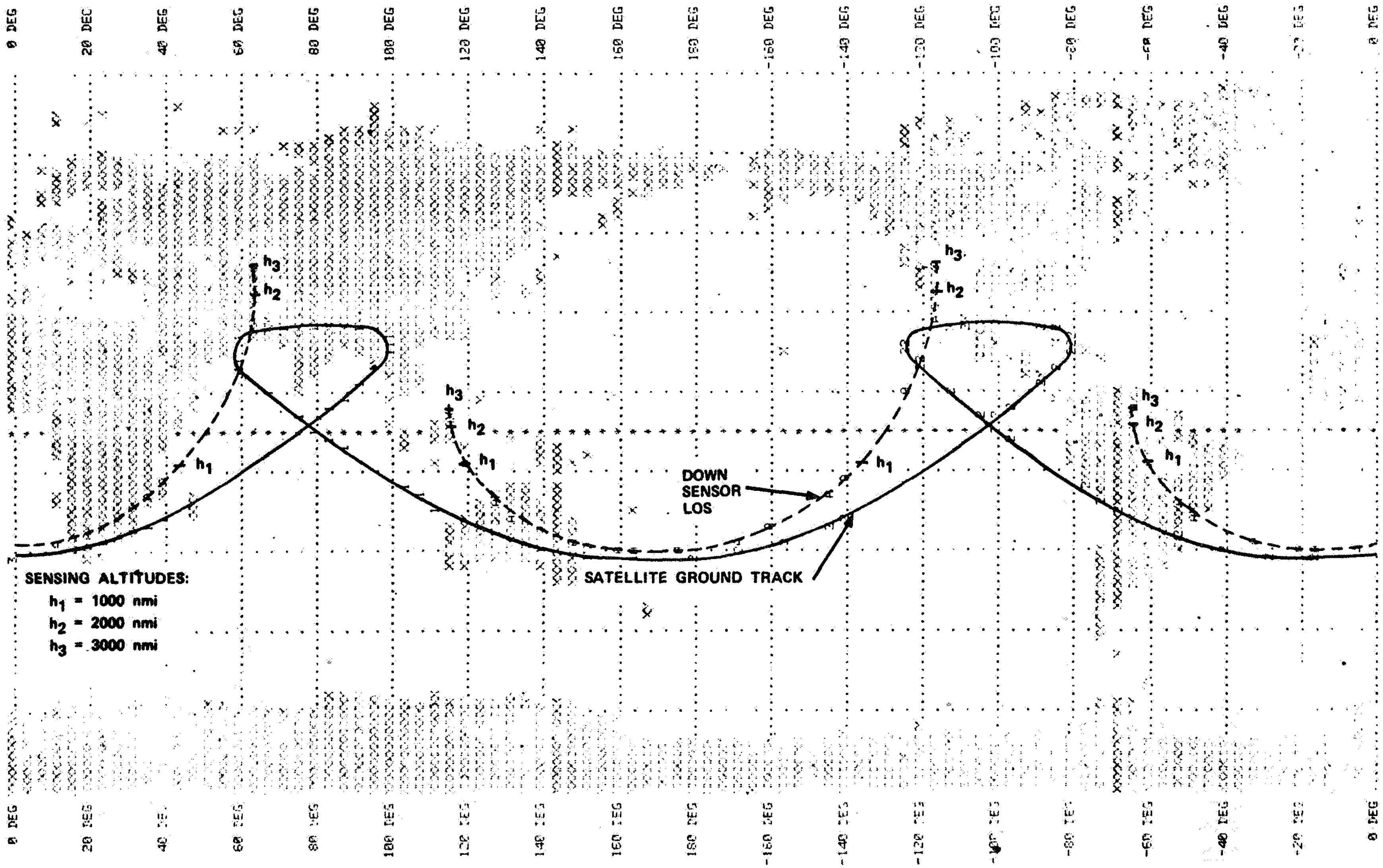


FIGURE 9. DOWN SENSOR LOS AND SATELLITE GROUND TRACK FOR MOLNYA ORBIT

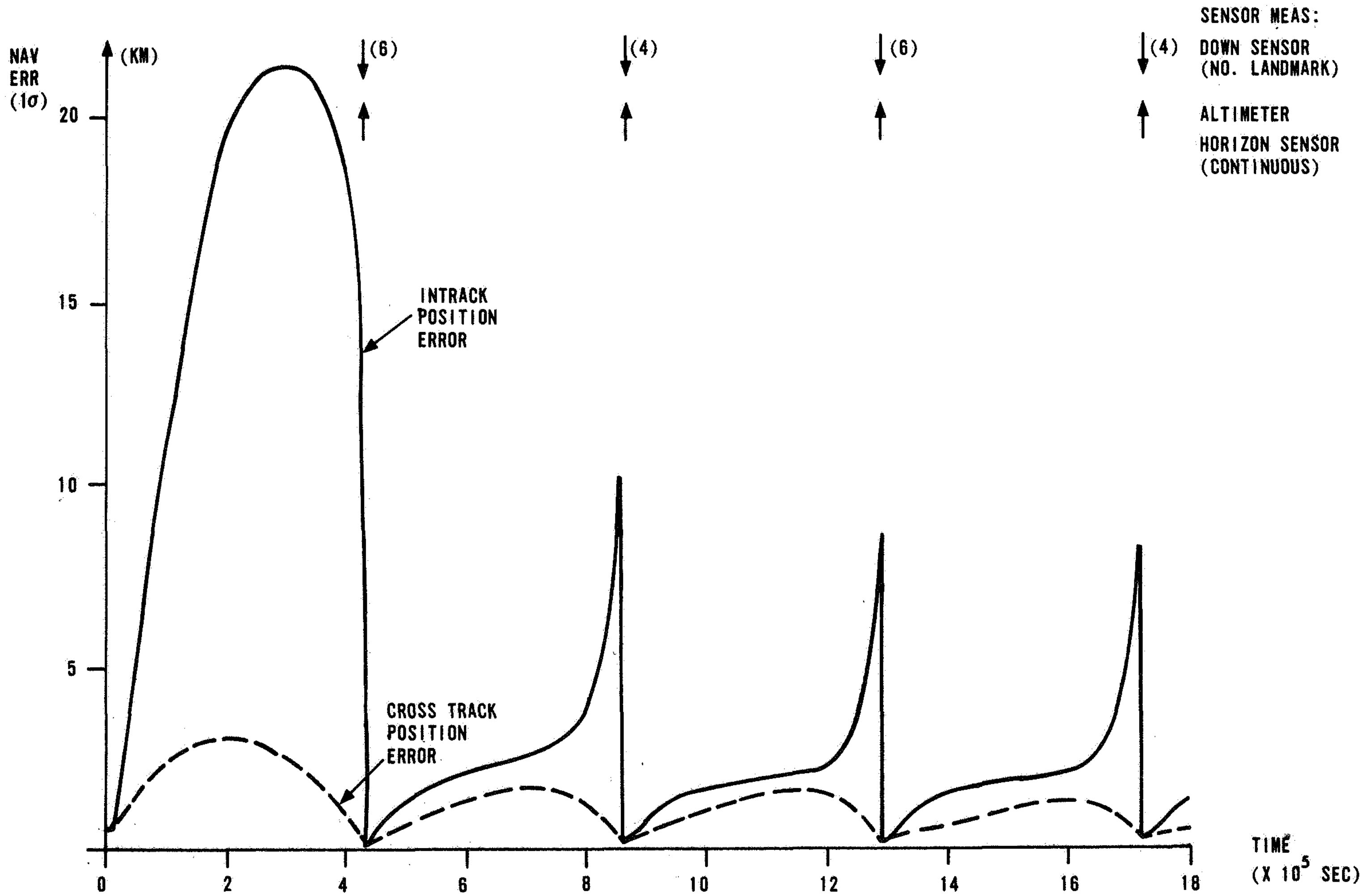


FIGURE 10. NAVIGATION ERROR FOR MOLNYA ORBIT WITH PERIGEE AT LOCAL 6 AM

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