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Astronavigation System as an Autonomous Enhancement Suite for a Strapdown Inertial Navigation System: An Evaluation

Jamshaid Ali

Center for Control and Instrumentation, NESCOM, Islamabad, Pakistan, E-mail: ali.jamshaid@hotmail.com

The determination of spacecraft coordinates is embodied by a navigation solution with high degree of autonomy, high precision and reliability. The foremost requirement for the solution of this problem is a strapdown inertial navigation system (SINS), which is autonomous, reliable and jamming resistant, but at the same time, errors in SINS increase with time. Therefore, for the SINS error correction, astronavigation system (ANS) is used as an additional navaid. This paper considers the ANS as an enhancement source for the SINS and presents its theoretical and practical aspects. The typical ANS makes use of star-trackers which are expensive, complicated in their structure and demand a-priori definition and vigilant preparation of each onboard attitude fix. To make this system cost effective and simple, an advanced configuration employs a charge coupled device (CCD) based star sensor rigidly mounted on a strapdown inertial measurement unit (SIMU). Consequently, in this evaluation, the ANS makes use of CCD based star sensors. Simulation results are presented to demonstrate the validity of the method for improving the navigation reliability and performance.

Keywords: Astronavigation, inertial navigation, Kalman filter, spacecraft, star sensor

1. INTRODUCTION

INERTIAL NAVIGATION, the ability to determine the location, orientation and motion of a vehicle, is an essential capability of any autonomous vehicle. The navigation accuracy and reliability requirements for a guided weapon and that of an autonomous space vehicle are not different. Both require high precision navigation solutions, in some cases that of the autonomous vehicle is down to centimeters, and both require the system to provide this data reliably. The major difference is the duration requirement to which the inertial navigation system (INS) is allowed to function without any external aiding, a function of the accuracy required. For civilian applications, this is quite short, in the order of seconds, because some sort of external aiding can be used.

The aspiration, however, is to endow with the navigation solution from the strapdown inertial navigation system (SINS) for as long as possible with no external aiding. This is due to the fact that aiding information cannot be pledged to come in at fixed intervals, and in any autonomous vehicle navigation, fault detection is vital and this requires accurate navigation solutions from individual systems. The cost of an inertial measurement unit (IMU) directs its accuracy and, in general, civilian applications necessitate low cost IMUs. These units, however, cause significant errors, which in turn instigate navigation solutions to drift significantly with time.

The focus of this research is to provide an assessment of the astronavigation system (ANS) which aids strapdown inertial measurement unit (SIMU) to make accurate and reliable navigation solutions available for low cost implementations. These navigation systems pose both challenging research and engineering problems. However, success in space vehicle implementations for major research has illustrated the capability and significance of such a navigation system. In a strapdown arrangement, the sensors experience the full effects of vehicle motion, and thus higher bandwidth and dynamic range are required. The higher dynamic range in turn affects the stability of scale factor terms, and may also introduce larger non-linearity errors. Higher bandwidth implies noisier data provided by the sensors. The ANS augmentation technique serves to compensate and calibrate errors in the attitude measuring sensors.

The SIMU data is based on principles of dead-reckoning and used in ships, spacecrafts, land vehicles and planes. The accuracy of these devices is not adequate for many of today's high precision, long duration sea, space, aircraft, and long range flight missions because of the inertial sensor errors which increase with time and are caused by initial orientation error, accelerometer bias and gyro drift. The SINS accuracy can be improved by augmentation with position or velocity updates from other navigation sensors. Although, the global positioning system (GPS) has received much attention for providing augmentation for SINS, an autonomous sensor is always preferable because active sensors such as the GPS may be unavailable as a result of hostile attacks on, or reliability failure of radio based navigation aids during the critical phase of the mission. Moreover, the current Missile Technology Control Regime (MTCR) limits the sale of GPS receivers that can function at speeds higher than 515 m/s and altitude greater than 18 km. Therefore, for third world countries, it is impracticable to purchase and use these controlled GPS receivers for their space program.

The ANS, as has been used in the past , is nonetheless popular and relates to its application in the modern space missions. This autonomous suite of ANS is free from radio aids, which may be unreliable, jammable, or unavailable during a wartime encounter. However, until now, the aided astro-inertial system has been hardly studied in domestic field because of its limited use in military missions. Therefore, in this study, we suggest SINS error compensation schemes using attitude information calculated by the ANS computer.

Consequently, study of the ANS as an augmentation means for SINS is the subject of this paper. This paper is organized in 5 sections. In section 2, the theme of the astro-inertial navigation is presented. In section 3, details about the attitude measuring systems are presented. Section 4 presents an evaluation conducted on the SINS/ANS integrated navigation system. Some useful conclusions are drawn in section 5.

2. ASTRO-INERTIAL NAVIGATION SYSTEM

With the accessibility to precise charge coupled device CCD star sensors, fast computers, high density memory chips, and star pattern recognition and identification algorithms, it is now doable to find low priced solution for the ANS. Star observations are made by video cameras using CCD electro-optical star sensor, mounted on SIMU, generating a two dimensional image of a small section of the sky equal to field-of-view (FOV) of the star sensor. The star sensor will sense a star over a sample period. During this period, the photo-sensitive CCD collects a charge from each incident photon. The total charge on each pixel is read out at the end of a sample period to determine how many photons were incident upon each pixel. Centroiding is performed on the pixel readouts, and this centroid is taken as the star measurement. This image output from the camera goes to the onboard computer. Computer programs, through some video thresholding schemes, limit the number of stars in a field that will be processed. Stored in the computer memory is a star catalog with the celestial coordinates of each star, in some reference system, which will be used for star field recognition. Star field recognition is accomplished through some star identification techniques. Subsequent to successful identification, a star's right ascension and declination with respect to body frame are computed. Then, from the known star position in the real and reference image, axes misalignment angles are estimated to construct an attitude error matrix [1]-[3]. Fig.1 shows conceptual arrangement of the ANS.

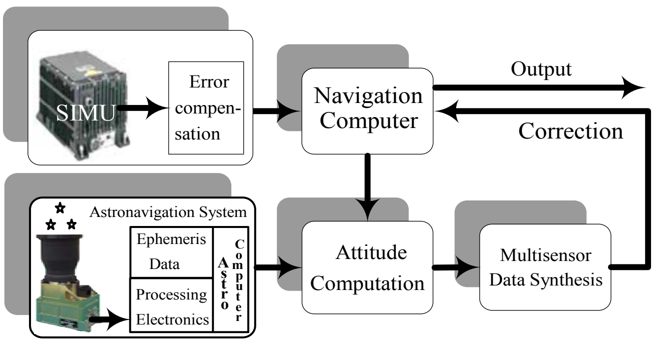

Fig.1 Astro-Inertial navigation system schema

It is vital for most space vehicles to know celestial referenced attitude from an onboard sensor. Usually, a quaternion or a direction cosine matrix is used to represent the attitude of the vehicle. These describe a rotation from an inertial space coordinate system to a coordinate system referenced to the attitude sensor. Successive coordinate rotations relate the attitude sensor coordinate system to the spacecraft body in pitch, yaw and roll. A combination of magnetometers, star trackers, sun sensors, horizon sensors, or star scanners is used on both spin stabilized and three-axis stabilized spacecraft for attitude determination. Star trackers are best suited for three-axis stabilized applications. In most applications, the output of the star tracker is used to update and correct drift in an inertial based reference system which provides high bandwidth attitude information. However, a gyroless spacecraft can use a mathematical model for attitude information. The star tracker then updates the state vector in this model [4].

3. ATTITUDE DETERMINATION SYSTEMS

A. Traditional Attitude Determination Systems

In the past, a variety of different attitude determination systems has been used. A traditional attitude determination system consisted of an absolute attitude reference sensor and an inertial sensor. The absolute attitude sensor determined the absolute pointing direction of the spacecraft at regular intervals. This measurement was used to calibrate the inertial sensor, which measured the changes in the attitude between the absolute calibrations [2].

The following technologies have been utilized for absolute attitude sensors [2]:

Magnetometers measure the size and orientation of a magnetic field. This strategy requires detailed knowledge of the magnetic field. Accuracies of 1 arc minute are obtainable. The precision depends on the geomagnetic model.

Radio frequency beacons can be used as a reference of pointing. However, if high accuracies are desired, a directional antenna is required. Accuracies of 1 arc minute are obtainable. Because only one reference is utilized, the method is only able to determine a pointing direction.

Horizon sensors detect the limb of the Earth, typically detecting infrared radiation. Accuracies of approximately 5 arc minutes are obtainable depending on the orbit.

Sun sensors can be constructed in numerous ways. They all utilize that the sun is the brightest object on the celestial sphere. Constructions vary from detecting whether the sun is present in a cone to high precision instruments that determine the orientation better than 1 arc minute. Because only one reference object is utilized, the sun sensor only determines the pointing direction toward the sun.

Solar panels can also be used as sun sensors. The currents from the different solar panels are monitored. Accuracies of 1 degree are obtainable. As the sun is the only reference used, only the pointing direction toward the sun can be determined.

Star trackers are, beyond dispute, the most accurate reference for pointing, because they utilize the fixed stars. The disadvantage of traditional star trackers was that they had to be externally locked onto known stars. Accuracies better than 1 arc second are obtainable. If more than one star is tracked, all three angles in the attitude are determined.

The following technologies have been utilized as inertial attitude sensors [2]:

Gyroscopes are spinning flywheels. If the orientation of the gyroscope is changed, the flywheel will apply a torque on the axis, which is proportional to the angular velocity of the gyroscope. High precision gyroscopes are very expensive. They also suffer from wear out and high drift rates.

Optical gyroscopes have the advantage of no moving parts. They consist of a coil of double optical fibers. The difference in light propagation in each direction of the optical fiber is proportional to the angular velocity of the optical gyroscope.

Orbit models are mathematical models of the spacecraft motion. It simulates the motion of the spacecraft which allows extrapolating the attitude to a given time.

B. *Star Sensors*

The hardware of the modern star sensors based on CCD technology is very compact and easy to implement. This star sensor has two major elements: a collecting lens gathering light from the sky and bringing it to a focus and a CCD. A CCD is a semiconductor chip which records the image into a matrix of small picture elements, pixels. The CCD is connected to the optical readout electronics, a high impedance buffer amplifier and a fast analog to digital converter, feeding the stellar computer which processes the optical image. The computer stores the ephemeris data necessary to identify and aid the detection of stellar signals in its memory [5]. The IMU detects angular and velocity increments which are processed in the inertial computer to give outputs of vehicle position and attitude. Attitude data is passed to the stellar computer to aid in locating and identifying stars and to stabilize the picture as the image moves over the CCD array because of vehicle rotations. The stellar computer passes angular data back to the inertial computer's Kalman filter to calibrate the gyro drifts.

The CCD detects photons falling on a pixel and integrates this as an electrical charge collected during the exposure time. The exposure time is a function of the blurring of the star image which can be tolerated. The performance of the star sensor is determined by the number of stars it can detect against the background of atmospheric radiation and internally generated CCD electronic noise. Typically, the attitude of a 3-axis stabilized spacecraft is determined by a star sensor. Orientation of the spacecraft can be determined based on the star observations. A modern star sensor is autonomous i.e. it automatically performs pattern recognition of the star constellations in the FOV and calculates the attitude quaternion with respect to the celestial sphere [6].

C. *Star Identification Techniques*

Algorithms that autonomously identify a star field with no a-priori information regarding orientation are ideally suited for use in attitude initialization. Current CCD star sensors provide a relatively inexpensive way to image the sky and extract information about the stellar locations and apparent brightness. A number of algorithms for star identification exist that can determine the correspondence between the viewed star field and a set of cataloged stars in a known reference frame [7].

It is well understood that taking image of the star in the sensor axis frame has no meaning unless the star is identified in the star catalog, so that its coordinates in the inertial reference frame are also known; the attitude of the spacecraft can also be calculated based on this knowledge. Here, we summarize some techniques developed for star identification using a star catalog onboard the spacecraft. Here, we will assume CCD electro-optical star sensor in conjunction with the star identification techniques. In a broad sense, star identification techniques are divided in three categories i.e. direct match, angular separation match and phase match.

In direct match technique, the estimated celestial coordinate frame must be sufficiently close to the star's celestial coordinate frame in the star catalog. An observation of the star in the star sensor is matched with a catalog star if $d(O,S) < \varepsilon$, where, O is the observed star unit vector in the estimated celestial coordinate and S is the catalog star unit vector in the true celestial coordinate; $d(O,S)$ is the angular distance between both vectors, and ε is the error window radius. The star observation is checked against all possible cataloged stars until an unambiguous and unique identification is found [8].

In accordance with the angular separation match technique, the angular distances between pairs of sensed stars are compared to the angular distances between pairs of stars in the star catalog. Two stars are selected arbitrarily from a set of measured stars, and the corresponding angular separation is calculated using $d_m^{12} = \cos^{-1}(S_1 \cdot S_2)$, where, S_1 and S_2 are the direction of two stars as measured by the star sensor [8], [9]. Now, search is made in the finite region of the catalog around the approximated boresight of the sensor, for a pair of stars (i,j) that fulfils the condition of $\left| d(i,j) - d_m^{12} \right| \le \varepsilon$, where $d(i,j)$ is the calculated angular distance.

The phase match scheme is used for calculating the direction of spin axis in spinning satellites [9]. For all the star matching techniques, the reliability of the identification process can be increased by using additional star characteristics, such as magnitude and spectra. This additional data must also be loaded into the onboard computer.

D. *Attitude Determination*

Traditionally, to exploit stars as assistance in automatic navigation, we have to consider two diverse points of view.

The first of these involves mechanization of the conventional measures of celestial navigation, which has been applied for many years by human navigators aboard ships and in aircrafts. The other considers astronomical source as the nearly ideal inertial base and uses it to enhance the performance of the inertial sensors of an INS. The astronomical reference is used to purge the effects of the inertial sensor errors from the performance of the navigation system [3]. Since gyro drift is one of the major sources of navigation error, celestial aiding leads to remarkably increased navigation accuracy and greatly extended range and reliability of long range assignments.

Once a set of measured stars is identified, the observation vectors associated with them must be compared to the reference vectors in the star catalog to compute attitude information. The attitude of the spacecraft can be determined by either deterministic methods such as TRIAD, QUEST, etc. [9], [10] or by utilizing algorithms that combine dynamic and/kinematic models with sensor data.

The TRIAD method [9], [10] generates an attitude matrix deterministically when two sets of measured vectors towards known stars are available. Because of its simplicity, the algorithm was implemented in many missions as the most popular method for determining the three axis attitude for spacecraft that provides complete vector information. The major drawback of this method is that it can accommodate only two observations. When more than two observations are available, these can be utilized in this algorithm only by a cumbersome weighted combination of the attitude solutions for the various observation vector pairs.

The QUEST algorithm is an optimal least square, minimal variance algorithm which determines the attitude that achieves the best weighted least square solution for an arbitrary number of reference and observation vectors. This approach, applied in the MAGSAT mission in 1979, has become the most widely used algorithm for attitude estimation based upon star sensor measurements [10].

Thereafter, many different approaches have been suggested in the literature for attitude determination with a given set of measurements. According to the recent comparative study for the existing algorithms, the algorithms presented in [11]-[13] are revealed to be faster or more robust than pre-existing approaches to estimating the attitude matrix.

For attitude determination, a method is described in [14] for estimating the incremental angle and angular velocity of a spacecraft using integrated rate parameters with the help of a star sensor alone. The main advantage of this method is that the measured stars need not to be identified, whereas the identification of the star is necessary in earlier methods. Here, estimation can be carried out with all the available measurements by a simple linear estimator, even though with a time varying sensitivity matrix. The residuals of the estimated angular velocity by this method have a competent accuracy level.

4. ANS – AN AUTONOMOUS ENHANCEMENT SUITE FOR SINS

Astronavigation system, in current and prior sense, is based on star observations. Star observation in powered flight phase of a ballistic vehicle will carry out in-flight alignment of the SINS as well as position and velocity errors that occur due to attitude errors [15]. Periodic star sensing during the free-fall periods can be used to obtain gyro g-insensitive calibration coefficients [16].

Since above 13.7 km altitude, the daytime sky background, not in the direct vicinity of the sun, grows progressively darker with an increase in altitude until it essentially turns black giving an unobstructed view of the stars. The ANS augmentation comes into effect after lift-off of the spacecraft when it attains altitude above 22 km; from onwards stars are visible unhindered [17].

A. Coordinate Frames

Inertial navigation system theory necessitates precise description of the coordinate frames which are defined as follows [15], [16]:

Inertial frame (*i*-frame, $x_i y_i z_i$): It has origin in the Earth's center; z_i is normal to the equatorial plane; x_i lies in equatorial plane, its direction can be specified arbitrarily; y_i complements the right handed system.

Body frame (b-frame, $x_b y_b z_b$): It has origin in the center of the mass of the rocket; x_b is along the rocket longitudinal axis; z_b axis is perpendicular to the longitudinal plan of symmetry and y_b axis complements the right handed system.

Launch-inertial frame (il -frame, $x_{il} y_{il} z_{il}$): It has origin at the launch point; x_{il} axis aims towards the expected target location; y_{il} axis points upward vertically and z_{il} axis complements the right handed system. It is the reference frame in which navigation calculations are performed.

B. Attitude Error Model

Angular rate error equation is given as [15], [16]

$$\dot{\bar{\phi}}^n = \delta\bar{\omega}_{ib}^n + \delta\bar{\omega}_{ie}^n + \delta\bar{\omega}_{en}^n - (\bar{\omega}_{ie}^n + \bar{\omega}_{en}^n) \times \bar{\phi}^n \tag{1}$$

where n refers to the reference frame in which navigation calculations are performed; where $\delta\bar{\omega}_{ib}^n = \bar{\varepsilon}^n$ is gyro drift; $\bar{\omega}_{ie}^n$ is Earth's rate vector; $\bar{\omega}_{en}^n$ is the transport rate vector and $\bar{\phi}^n$ indicates SINS axes misalignment angles.

For spacecraft application, usually space-stabilized SINS mechanization is used, therefore in Eq. (1),

$$\delta\bar{\omega}_{ie}^n = 0; \quad \delta\bar{\omega}_{en}^n = 0; \quad (\bar{\omega}_{ie}^n + \bar{\omega}_{en}^n) \times \bar{\phi}^n = 0$$

thus

$$\dot{\bar{\phi}}^n = \delta\bar{\omega}_{ib}^n = C_b^n \bar{\varepsilon}^b \tag{2}$$

where C_b^n is the *b-n* frame transformation matrix

C. Velocity and Position Error Model

Velocity error equation is expressed as follows [15], [16]

$$\begin{aligned} \delta\dot{\bar{v}}^n &= \bar{f}^n \times \bar{\phi}^n + C_b^n \bar{\nabla}^b - (2\delta\bar{\omega}_{ie}^n + \bar{\omega}_{en}^n) \times \bar{v}^n \\ &- (2\delta\bar{\omega}_{ie}^n + \bar{\omega}_{en}^n) \times \delta\bar{v}^n + \delta\bar{g}^n \end{aligned} \tag{3}$$

where $\overline{f}^n$ is the specific force; $\overline{\nabla}^b$ represents accelerometer bias; $\overline{v}^n$ is velocity of the vehicle and $\delta\overline{g}^n$ is the acceleration error due to gravity that is

$$\delta\overline{g}^n = \begin{bmatrix} \partial g_x^n / \partial r_x^n & \partial g_x^n / \partial r_y^n & \partial g_x^n / \partial r_z^n \\ \partial g_y^n / \partial r_x^n & \partial g_y^n / \partial r_y^n & \partial g_y^n / \partial r_z^n \\ \partial g_z^n / \partial r_x^n & \partial g_z^n / \partial r_y^n & \partial g_z^n / \partial r_z^n \end{bmatrix} \begin{bmatrix} \delta r_x^n \\ \delta r_y^n \\ \delta r_z^n \end{bmatrix} \tag{4}$$

where $\delta\overline{r}$ denotes position error vector.

For a high accuracy demand of the navigation system, Earth is taken as an ellipsoid having axes of symmetry coinciding with the Earth's axis of rotation. In this case, the gravity vector takes into account the ellipsoid shape of the Earth. When using the space-stabilized mechanization for SINS, in Eq. (3),

$$(2\delta\overline{\omega}_{ie}^n + \overline{\omega}_{en}^n) \times \overline{v}^n = 0; \quad (2\delta\overline{\omega}_{ie}^n + \overline{\omega}_{en}^n) \times \delta\overline{v}^n = 0$$

thus $$\delta\dot{\overline{v}}^n = \overline{f}^n \times \overline{\phi}^n + C_b^n \overline{\nabla}^b + \delta\overline{g}^n \tag{5}$$

The ANS can be used to estimate or correct those velocity errors that occur due to misalignments. Dynamic accelerometer errors or errors caused by gravity uncertainties cannot be estimated using ANS aiding.

The position error equation for the SINS is given as

$$\delta\dot{\overline{r}}^n = \delta\overline{v}^n \tag{6}$$

D Evaluation via Simulation

An in-flight alignment technique for the SINS employing a star pattern recognition procedure for identifying stars sensed by a CCD electro optical star sensor is presented here. Collinearity equations are used [15] to estimate the sensor frame star coordinates and the conventional least square differential correction method is used to estimate the unknown orientation angles. A comparison of this attitude with the attitude estimated by the SINS provides axes misalignment angles, an observation to the Kalman filter [18]. Simulations using Kalman filter are carried out for SINS employing space-stabilized navigation frame. Analysis shows that celestial updates are effective in estimating and compensating for gyro errors as well as the position and velocity errors that occur due to the SINS misalignments.

The discrete Kalman filter realization used in this paper is the direct feedback where the estimated errors are fed back to the SINS, thus minimizing the evolution of the observed errors; those are to be delivered as an observation to the Kalman filter [1]. In this simulation, quaternion is obtained from the corrected attitude matrix and is fed back for attitude error compensation.

E. Experimentation Setup

Simulation is an important step towards solution of an engineering problem. An engineering simulation entails mathematical modeling and computer assisted investigation. To validate and corroborate the designed multi-sensor navigation data synthesis technique, simulation data is generated through a half-physical simulation setup established in the laboratory [19]. Simulation data consists of specific force, angular rate, position, velocity, attitude, ANS estimated attitude, SIMU

In this setup, spacecraft trajectory and SIMU simulator are used to simulate the flight path and the SIMU outputs. The SIMU outputs comprise angular rate and specific force. The ANS augmentation comes into effect about 40 seconds after lift-off of the vehicle. At this time, guide stars are extracted from the star catalogue stored in the computer. These guide stars are simulated using a physical star sensor simulator. Light of stars is simulated by a liquid crystal light valve and detected by the CCD electro-optical star sensor. Star pattern recognition is carried out to identify the stars in the reference star catalogue. Once stars have been identified using a star identification technique, the attitude is estimated through an attitude determination procedure. Experimental view of the half-physical ANS simulation setup is shown in the Fig. 2.

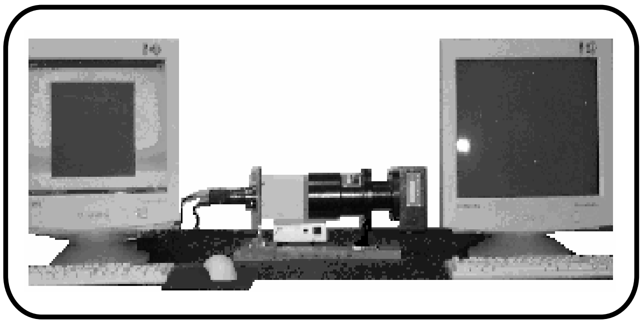

Fig. 2. Experimental view of the ANS simulation

F. Velocity and Position Error Compensation

In the SINS/ANS integrated navigation system, observability of the velocity and position errors is poor. From the simulation, we estimate axes misalignment angles and the constant gyro drifts. This drift represents gyro turn-on to turn-on constant drift. We can only compensate fixed gyro drift that is estimated during laboratory calibration of the SIMU. Turn-on to turn-on constant drift is estimated using ANS aiding. Thus, from the known gyro errors, we can estimate and compensate for the position errors that occur due to the attitude errors. This position error compensation takes place when the vehicle is out of atmosphere and the astronavigation comes into effect.

The simulation results for the SINS/ANS integrated system are depicted in the following figures where Fig. 3 depicts velocity error compensation whereas Fig. 4 shows position error compensation.

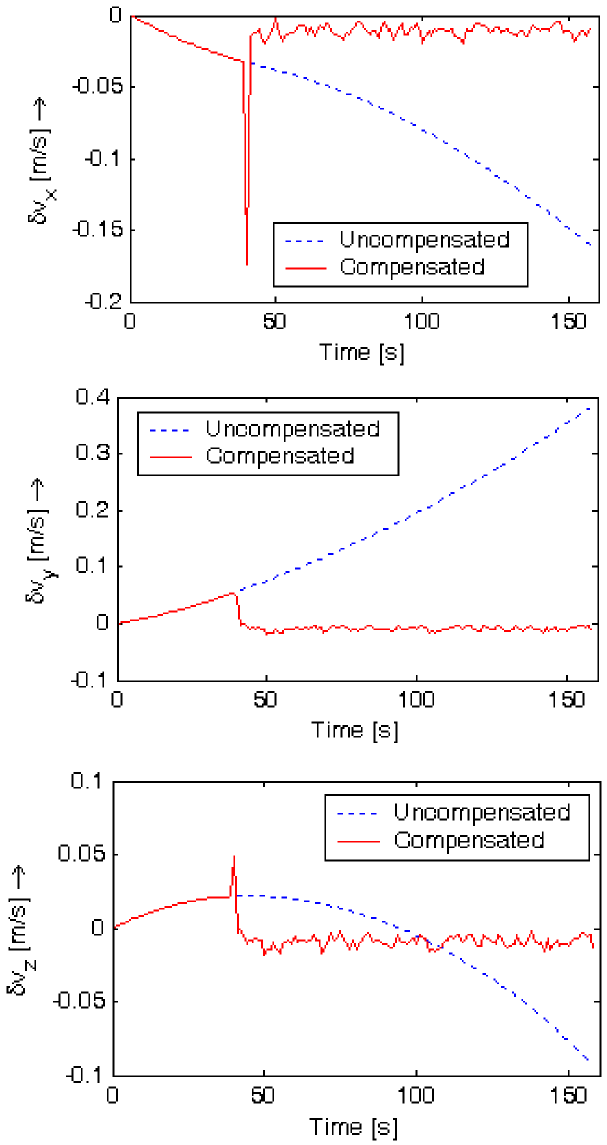

Fig.3 Compensated and uncompensated velocity error

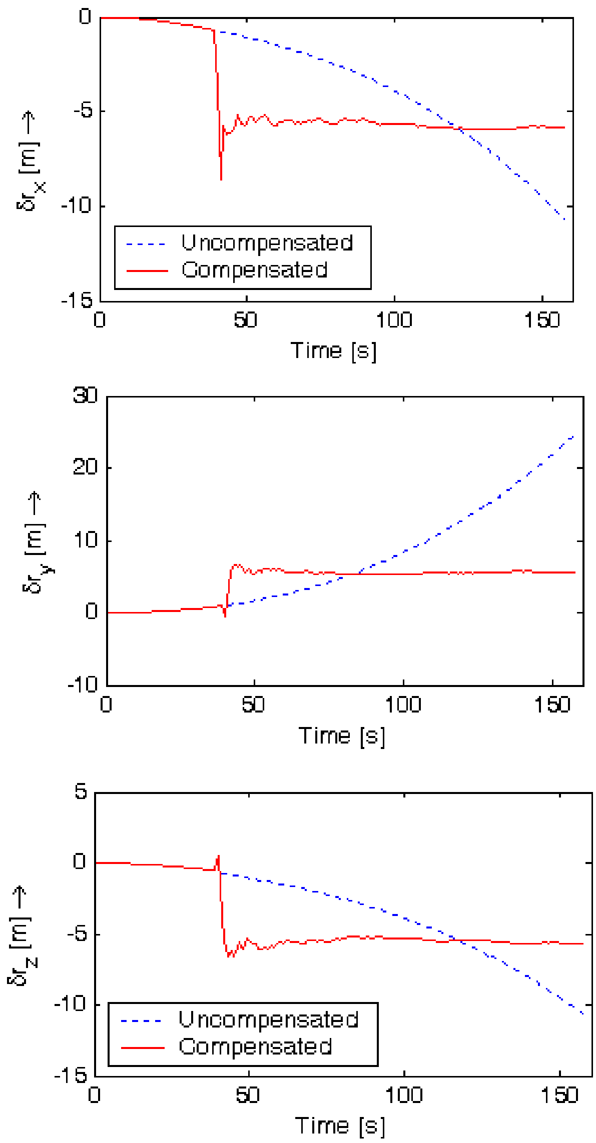

Fig.4 Compensated and uncompensated position error

5. Conclusions

The SINS integrated with ANS yields reliable mission capability and enhanced navigational accuracy for spacecrafts. This paper presents an evaluation of the astronavigation systems as an autonomous augmentation source for the SINS. Emphasis has been put on the state of the art technologies for the star sensing and the efficient techniques for processing the ANS data. The current analysis presents a brief description of the state of the art star sensor, attitude and angular velocity algorithms, and the integration of the ANS with the SINS. This assessment has been done with an aspiration to present a systematic theoretical background and overview of the inclusive research on the subject.

A linear error model for the space-stabilized SINS has been employed for a spacecraft application. This integration results in estimation of gyro drift and compensation of the velocity and position errors that occur due to the SINS axes misalignment angles.

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