

INVITED PAPER

## Advanced Navigation System for Aircraft Applications

G. Satheesh Reddy\* and V.K. Saraswat

*Research Centre Imarat, Hyderabad-500 069, India*

*Defence Research and Development Organisation, DRDO Bhawan, Delhi-110 011, India*

*\*E-mail: sat.gundra@gmail.com*

### ABSTRACT

Various forms of navigation are present in today's world, leading from satellite based navigation to several archaic forms of navigation like star gazing. Now, lots of technologies are available to achieve this but with certain limitations. For example, FOG based navigation provides accuracy with in  $0.1^0-10^0$  range which is not sufficient for various military applications. Therefore, there is a need to design a system which will have better accuracy and thus requires development of ring laser gyro-based inertial systems. This paper concentrates on the aided navigation system based on ring laser gyro of 0.01 deg/hr class and GPS - GLONASS to further enhance the capability of system in terms of accuracy. The usage of such systems not only provides accurate results momentarily but it also persists for longer duration with the aid of GPS - GLONASS for applications like aircraft, ship and long range missiles. The system provides accuracy of the level of 1 Nm/hr in pure navigation and 30 m with the aid of GPS - GLONASS. Apart from this, the availability of gyro-compass and baro-inertial algorithms further enhances the system capabilities and made them self dependent to the major extent.

**Keywords:** Navigation system, ring laser gyro, fibre optic gyro, inertial navigation system

### 1. INTRODUCTION

The inertial navigation is based on the Newtonian laws of kinematics and motion. As the Newton's laws are essentially regressive in nature, they depend on their past values and summate over time for all future position and velocity. As the sensors employed in inertial navigation are polluted with various forms of errors, and due to the Newton's laws these errors get accumulated over time. This results in unavoidable exponential increase in position error over time.

The GPS-GLONASS (GPSGLS) itself is insufficient to provide a reliable navigation solution as the position random noise is large and not suitable for aircraft control loops. The GPSGLS and INS systems complement each other symbiotically to provide high frequency, low long term error navigation. A suitable Kalman filter of 17-error states, including gyro and accelerometer biases is implemented for the purpose. The estimated errors are feedback in closed loop to improve the dynamic performance of the system.

Also, uncertainty in the initial state of the navigation, viz., the initial attitude, position and velocity lead to incorrect state propagation for all time. It is hence essential to correctly define the initial state of navigation. Attitude initialization is obtained using self and aided alignment schemes. The idea of alignment is elaborated and suitable algorithms for various forms of alignment, viz., gyrocompass alignment, stored heading alignment, inserted heading alignment, and in-flight alignment are presented. The position and velocity differential

equations are solved taking into account the coning and sculling compensation arising due to oscillatory and rotatory coning motion for the local level navigation solution.

For aircraft a stand-alone height information sensor, like the Baro-altimeter commonly available with aircraft is used to limit the instability in the vertical channel. A four state Kalman filter is designed to estimate the vertical velocity and position error including the bias states of the altimeter.

The other aircraft functions viz., steering and magnetic heading computation are provided for aiding the Pilot and are part of the standard requirements of aircraft navigation.

Results of various test cases are presented at the end followed with conclusion. They include simulation test results, van trials and aircraft sorties.

### 2. PRINCIPLE OF INERTIAL NAVIGATION

Given an accelerometer that measures specific force, it is possible to provide estimates of velocity and position through successive integrations. If the velocity and position are used in navigation it is required that it can be related to a specific reference frame. Furthermore a continuous orientation of the accelerometers is needed with respect to the reference frame. This is done through gyroscopes that provide a measure of the attitude with respect to the same reference frame. A typical inertial navigation system (INS) is therefore made up of three accelerometers and three gyroscopes mounted in an orthogonal triad in order to determine the position and orientation of the

INS. Principle of inertial navigation is illustrated in Fig. 1.

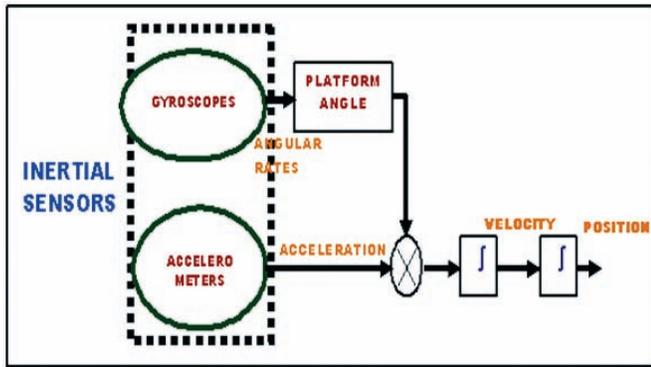


Figure 1. Principle of inertial navigation.

### 3. RING LASER GYRO BASED INS+GPS+GLONASS (RINS) ARCHITECTURE

The ring laser gyro based inertial navigation system plus the global positioning system (RINS) consists of three ring laser gyros (RLGs) and three accelerometers and other processing electronics to give the total navigation information.

Three RLGs and three accelerometers are strapped on to a structure called the strap down block. The inertial sensors are mounted in the strap down block in an orthogonal configuration with the sensitive axis of the gyros and those of accelerometers in parallel configuration. This block is mounted on the chassis through vibration isolators. Different processing electronics (gyro, accelerometer, navigation processor electronics, and GPSGLS receiver electronics) will process the information from the RLGs, accelerometers and RF signal from GPSGLS antenna to generate the navigation information. They are housed inside the chassis. GPGLS antenna with its preamplifier is mounted outside on the aircraft's body. The architecture diagram of RINS is shown in Fig. 2.

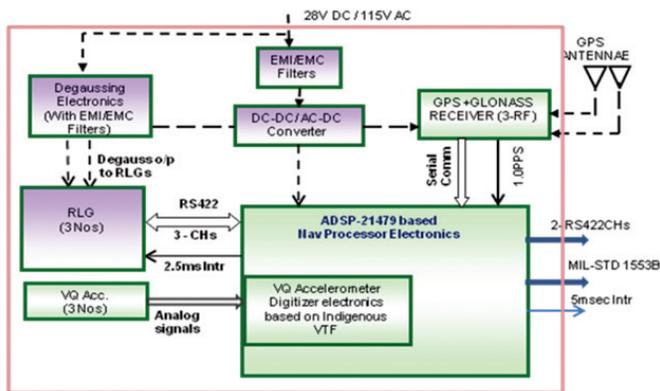


Figure 2. Architecture diagram of RINS.

## 4. SYSTEM DESIGN DESCRIPTION

### 4.1 System External and Internal Interfaces

Various interfaces of the RINS system are as follows:

#### 4.1.1 External Interfaces

- Power supply (28V DC/115VAC,400 Hz)
- Dual RF input

- Mil1553B and ARINC-429, and RS-422

#### 4.1.2 Internal Interfaces

- Ring laser gyros and accelerometers
- Navigation processing electronics
- Accelerometer electronics
- GPSGLS receiver electronics
- De-gaussing electronics
- DC-DC/AC-DC power module

## 4.2 Sensors

### 4.2.1 Accelerometers

High accuracy Force balanced pendulous accelerometers are used in the RINS system. These accelerometers are mounted along three mutually perpendicular axes of the strap down element. Each accelerometer measures the acceleration of the vehicle along its sensitive axis. The force-balanced accelerometer consists of a proof mass, which moves along acceleration sensitive axis. The pick-off device can detect the deviation of the position of the mass from the null position. The output of the pick-off device resulting from the motion of the mass is fed to a force balance coil, which produces the force required to return the proof mass to its null position. The flow of current in the force balance coil is a measure of acceleration of the body along the sensitive axis of the accelerometer. A typical VQ accelerometer is shown in Fig. 3.

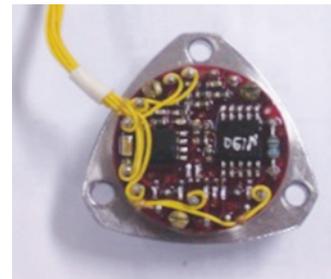


Figure 3. VQ accelerometer.

### Accelerometer specifications

- Dynamic range :  $\pm 50$  g
- Bias stability : 30  $\mu$ g
- Scale factor stability : 30 ppm

### 4.2.2 Ring Laser Gyro

The ring laser gyro is an inertial sensor that provides an output frequency proportional to the inertial angular rate about its sensitive axis. It is an optical gyro whose working principle is based on Sagnac effect.

This instrument is an essential requirement for navigation and control of a moving vehicle. It is required to determine the direction along which the vehicle is moving with respect to a reference set of coordinate frame. The property of gyroscope is basically defined by three parameters - scale factor stability, the zero bias stability, and the sensitivity to the environment. The above factors will determine the accuracy of the orientation of the vehicle, which can be measured and subsequently used for controlling and guiding the moving vehicle. A ring laser gyro is shown in Fig. 4.

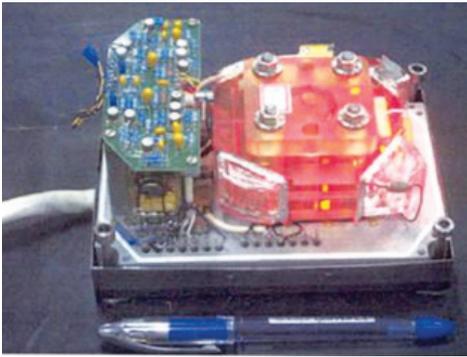


Figure 4. Ring laser gyro.

**RLG specifications**

- Input rate range :  $\pm 300$  deg/s
- Bias stability : 0.01 deg/hr
- Angular random walk: 0.003 deg/ $\sqrt{\text{hr}}$
- Scale factor stability : 10 ppm

**4.3 Processing Electronics**

**4.3.1 NAV Processor Module**

The navigation processor module has got numeric intensive navigation algorithms, which requires wide dynamic range, 64-bit floating point operations; on chip peripherals required are SRAM, DMA, high speed serial link, MAC for mathematical modeling, timer for real time clock generation, pipe lining, etc. ADSP-21479 processor is selected for the system.

The navigation processor performs the built in self tests for sensors health, peripherals checks, and interfaces and so on. The navigation processor has to acquire the three orthogonally mounted RLGs and accelerometer data and compensate the sensor data for errors like bias, misalignment and scale factor deviations and execute the coning and sculling algorithm to extract the true rotational rate in real time. Quaternions are computed based on four sample Miller’s algorithm. Pure navigation algorithm is to be executed to compute the incremental position and velocity. Navigation processor has to acquire GPSGLS receiver data at 1.0 s rate and check the validity of data. The data fusion of pure navigation and GPSGLS is done using a 17- state Kalman filter. The pure and hybrid navigation data is to be posted to flight computer. The navigation processor module is also responsible for all flight management functions like wise NRAM data storage in power failure condition, health checks of various peripheral interfaces.

**4.3.2 Accelerometers Data Processing Electronics**

The VQ accelerometer output is in the form of current signal. A high accuracy indigenously developed voltage to frequency converter (VTF) is used to process the accelerometer output. The output of VTF is the frequency pulses proportional to the acceleration. Each pulse corresponds to velocity increments.

**4.3.3 GPS - GLONASS Receiver**

GPS-GLONASS (GPSGLS) a 24-channel receiver developed by INSD, Research Centre Imarat in association

with private industry is used in RINS. The receiver takes the measurements of range to four satellites, four ranges being required to determine the four un-knowns; three spatial coordinates and time. Block diagram of receiver hardware is shown in Fig. 5.

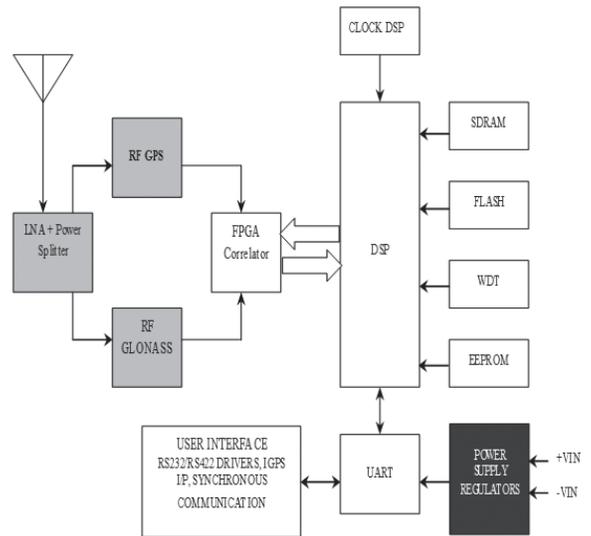


Figure 5. Block diagram of receiver hardware.

The signal from combined RF antenna is given to 2 RF down converters through LNA followed by power splitter. The RF section consists of RF down converter to frequency translate the signal to IF, ADC to Digitize the IF, DAC to control the gain of RF chain and RF clock (Reference clock for both the RF sections). The digitized IF samples are given to FPGA for further processing.

The FPGA contains 24 correlator channels. Among 24-channels 12-channels are dedicated for GPS and 12-channels for GLONASS. The correlation values from FPGA are read by DSP at a particular rate. The FPGA correlator also generates the sampling clock for IF digitization. The DSP section contains clock source for its operation. This hardware contains flash to store FPGA and DSP code. To store run-time code, which can be accessed at core clock speed 128Mb, SDRAM is provided. There is provision for EEPROM, in which we can store almanac information on-line. To monitor the system health, hardware watch dog timer is provided on board.

*Receiver specifications*

- Accuracy
- Position : 30 m
- Velocity : 0.15 m/s

**5. ALIGNMENT SCHEME OF RINS**

The need for accurate initial alignment is at best understated. Initial misalignment is one of the major error sources of strap down inertial navigation systems (SDINS). The error in initial misalignment translates into incorrect attitude, velocity and position information for all time. Inserted heading (IH), stored heading (SH), Gyrocompass (GC), and in-flight (IF) alignment scheme is developed to meet the alignment accuracies within limited specified time.

### 5.1 Gyrocompass Alignment

Gyrocompass alignment for strap-down systems is a process of virtual alignment since no actual rotation takes place to align the body frame with the navigation frame. The DCM gets estimated while the SDINS is essentially stationary. Gyrocompass alignment is comprised of an initial coarse estimation and later fine alignment. Fine alignment employs a Kalman filter to estimate the misalignment angle and corrects them to give true DCM.

#### 5.1.1 Coarse Alignment

The DCM constructs from 3 euler angles and it is a 3x3 matrix with 9 unknowns. To solve for the 9 unknowns, 9 equations should be available. The actual and measured earth gravity vector (including centripetal acceleration), earth rate vector and a combination of cross product of two vector in body and NED frame are related as follows.

$$\hat{C}_b^{ned} = \begin{pmatrix} g^{nedT} \\ (g^{ned} \times \omega_{ie}^{ned})^T \\ ((g^{ned} \times \omega_{ie}^{ned}) \times g^{ned})^T \end{pmatrix}^{-1} \begin{pmatrix} g^bT \\ (g^b \times \omega_{ie}^b)^T \\ ((g^b \times \omega_{ie}^b) \times g^b)^T \end{pmatrix}$$

The coarse estimate  $\hat{C}_b^{ned}$  is away from true  $C_b^{ned}$  due to sensor errors. Coarse alignment estimates roll and pitch angle very great degree of accuracy of the order of 0.01°, however yaw angle accuracy up to 1°. High accuracy in roll and pitch is due to greater observability of north and east accelerometers, while the yaw accuracy is limited due to poor observability of the east gyro.

#### 5.1.2 Fine Alignment

The coarse DCM is used to propagate in fine alignment and refines it with a Kalman filter. Kalman filter is used to predict the misalignment angles from noisy North and East accelerometers and East gyro measurements. The local level accelerometers and the east gyro measure null magnitude irrespective of the ambiguity of location and hence they are chosen as observables. The coarse DCM estimate is used to transform the sensor measurements in the body frame to the NED frame. The North and East velocity measurements are used for leveling and the angular velocity in the east is used for heading refinement. Figure 6 depicts the fine alignment scheme.

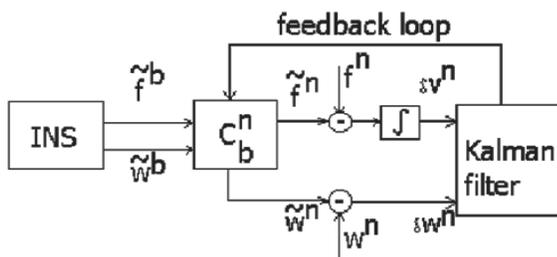


Figure 6. Block diagram of fine alignment scheme.

#### State model

$$\begin{bmatrix} \delta v^N \\ \delta v^E \\ \psi^N \\ \psi^E \\ \psi^D \end{bmatrix} = \begin{bmatrix} 0 & 2\omega^D & 0 & g & 0 \\ -2\omega^D & 0 & -g & 0 & 0 \\ 0 & 0 & 0 & \omega^D & 0 \\ 0 & 0 & -\omega^D & 0 & \omega^N \\ 0 & 0 & 0 & -\omega^N & 0 \end{bmatrix} \times \begin{bmatrix} \delta v^N \\ \delta v^E \\ \psi^N \\ \psi^E \\ \psi^D \end{bmatrix} + \begin{bmatrix} \delta f^N \\ \delta f^E \\ \delta \omega^N \\ \delta \omega^E \\ \delta \omega^D \end{bmatrix}$$

#### Measurement model

$$H = \begin{pmatrix} 0 & g & 0 & 0 & 0 & 0 \\ -g & 0 & 0 & 0 & 0 & 0 \\ -\omega \sin L & 0 & -\omega \cos L & 0 & 0 & 0 \end{pmatrix}$$

#### State vector

$$\Psi^N \Psi^E \Psi^D \delta \omega^N \delta \omega^E \delta \omega^D$$

The scheme is tested on 28 cm path length ring laser gyro (0.01°/hr bias accuracy) based INS (RINS) mounted on a two axis accurate rate table. The error in Azimuth (Yaw) is found to be within 0.2° after 6 min GC (2 min course and 4 min fine alignment). The roll and pitch angle are within 0.01°.

### 5.2 Stored Heading Alignment

The purpose of stored heading is to obtain attitude of an aircraft as was stored when the aircraft has come to halt from a previous sortie. The attitude at the end of the earlier sortie is stored in the NVRAM and is re-loaded for the present sortie without altering the position of the aircraft at the hangar. It is obvious that the stored heading alignment is faster than the other alignment procedures.

### 5.3 Inserted Heading Alignment

The heading ( $\psi$ ) is provided by measuring externally and the other two angles  $\phi, \theta$  are measured using the accelerometer output resolved along the reference frame. The reference frame for leveling is local north, east and down in the geodetic frame. In the process of alignment, the accelerometers output along the X and Y of the reference frame is compared with the surveyed gravitational force vector and the error is determined. The attitude (Quaternion) is corrected with this error with an appropriate gain. The leveling process continues in the closed loop, and it takes 15 s to correct the initial attitude error of 1° to 10 arc s.

Leveling is carried out for 30 s and at the end of this cycle the attitude re-initialization is carried out. The angles  $\theta$  and  $\phi$  are extracted from the last quaternion and  $\psi_{INITIAL}$  are used to form a fresh quaternion.

### 5.4 In-flight Alignment

The INS-GPGLS features an in-flight alignment capability. This allows achieving alignment procedure using the GPGLS computed solution. In-flight alignment may be initiated either in-flight while aircraft is in motion, or as a continuation of an interrupted ground alignment procedure.

The aircraft is required to fulfill either of the below

mentioned two criteria. First is to move the aircraft with maneuvers and second is sudden changes in velocity i.e. acceleration or deceleration. The coarse alignment duration is 3 min, which provides coarse attitude. This coarse attitude is fed to the INS-GPGLS six state ( $\Delta\psi$ ,  $\Delta\Phi$ ,  $\Delta\theta$ ,  $\Delta V_n$ ,  $\Delta V_e$ , and  $\Delta V_d$ ) alignment Kalman filter as initial attitude estimate. The INS-GPGLS filter would correct the coarse estimate of attitude and provides fine attitude and velocity thus achieve alignment in flight. After seven minutes of fine alignment the accuracy achieved in  $\psi$ ,  $\Phi$ , and  $\theta$  are  $0.2^\circ$ ,  $0.01^\circ$ , and  $0.01^\circ$  respectively.

## 6. AIDED NAVIGATION TECHNIQUES

Although very high accurate inertial sensors are available but still the navigation accuracies cannot be met for long duration without using the satellite (GPS, GLONASS) navigation. By integrating the unique and complementary characteristics of each system into one integrated INS-GPSGLS system, accuracies as well as additional benefits can be achieved even though unattainable by either system independently. A good integration with GNSS ensures system availability over an extended period of time.

Altitude damping techniques (based on barometer data) for aircrafts have evolved to ensure precise INS solution availability in GPS denied environment.

### 6.1 INS-GPGLS Integration

INS provides autonomous and smooth navigation outputs, but these trends to drift with time in unbounded manner. Satnav position information has bounded errors.

Taking the advantage of combined GPS-GLONASS systems, the new hybrid navigation scheme has been developed to fuse the data from INS, GPS, and GLONASS systems by using the raw measurements (pseudo-ranges and delta pseudo-ranges) of GG Receiver and which are synchronized to INS clock (real-time algorithms are adopted). The INS system is implemented in the strap down configuration and the extended Kalman filter is used to fuse the Inertial and GPS-GLONASS raw measurements data. However, the GPS and GLONASS clocks are not exactly synchronized. So, for computing the combined position solution, it requires a minimum of 5 satellites from GPS-GLONASS (GG) to get a position fix.

Defined 19 states for navigation Kalman filter are as follows:

$$\delta x = (2\delta b, 2\delta d, \delta R^e, \delta v^e, \delta \Psi^e, \delta \omega_b, \delta a_b)$$

Where,  $\delta R^e$ ,  $\delta v^e$  and  $\delta \Psi^e$  are the INS error states for position, velocity and tilt,  $\delta \omega_b$  and  $\delta a_b$  are the gyro drift and accelerometer bias error states and  $2\delta b$ ,  $2\delta d$  are the GG receiver clock bias and clock drift errors w.r.t. to GPS and GLONASS measurements.

#### 6.1.1 Observation Process

The measurement matrix  $z_k$  is processed by the filter to yield the best estimates of the error state  $x_k$ .  $z_{(k|k-1)} = H_k x_{(k|k-1)}$  the observation matrix  $H_k$  relates the components of the true state vector to the measurements.

The observation matrix for pseudo-range is as follows:

$$H_p = \begin{pmatrix} 1 & 0 & 1 & 0 & 0 & 0 & 0 & 0 & 0 & 0 & 0 & 0 & 0 & 0 & 0 & 0 & 0 & 0 & 0 \end{pmatrix}_{1 \times 19}$$

The observation matrix for Delta-range is as follows:

$$H_{\delta p} = \begin{pmatrix} 0 & 1 & 0 & 1 & 0 & 0 & 0 & 0 & 0 & 0 & 0 & 0 & 0 & 0 & 0 & 0 & 0 & 0 & 0 \end{pmatrix}_{1 \times 19}$$

In the observation matrixes the first two elements represent the GPS clock bias and clock drift observations. The next two elements are the GLONASS clock bias and clock drift observations. The Measurement noise values for  $P_r$  and  $D_p r$  are 2 m and 0.02 m/s respectively. The navigation scheme of INS-GPS-GLO is shown in Fig. 7.

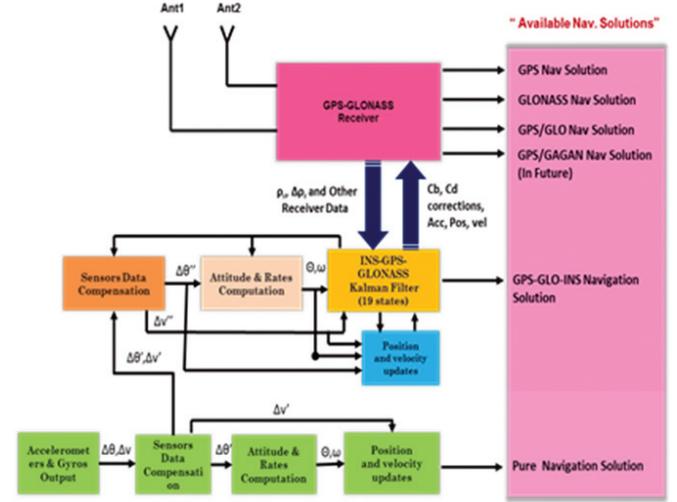


Figure 7. INS-GPS-GLONASS navigation scheme.

The fusion filter is driven by number of navigation solutions available with GPS-GLONASS receiver. Therefore, it becomes obvious to take best decision in real-time to get optimized hybrid navigation solution so as to guide the vehicle in the desired accuracy.

The performance of the GPS-GLONASS aided inertial navigation systems are evaluated for real time in the Lab as well as for the simulated Flight trajectories of various high dynamics. The GPS-GLONASS outage, degradation in satellites signal and error in the position solution were simulated to see the hybrid navigation performance

### 6.2 Baro Inertial Aiding

The error in the vertical channel directly contributes to error in altitude estimate. Due to schuler damping, the horizontal errors are bounded and oscillate with a time period of 84 min. Due to undamped vertical channel, the errors are not bounded and hence it needs to be controlled in absence of GPS. Altitude measurements from the baro-altimeter which is commonly available on all aircrafts are used in conjunction with the INS measurements, to keep the error growth in vertical channel within bounds. The baro-altimeter data and the INS data are fused with a Kalman filter to obtain an accurate estimate of the flight altitude.

A four state Kalman filter is designed for optimal mixing of baro-altitude data with inertial height data. The Markov model for the baro altitude would keep the baro-altimeter error bounded. The mixing block diagram is shown in Fig. 8.

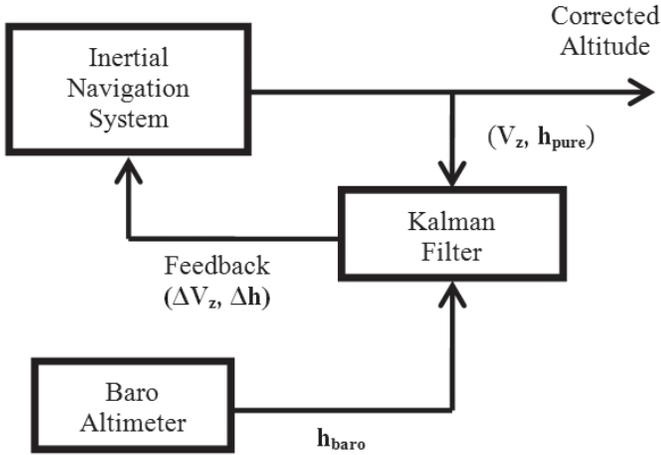


Figure 8. Baro inertial integration.

State Model

$$\begin{bmatrix} \Delta h_{ins} \\ \Delta v_{dins} \\ \Delta h_{baro} \end{bmatrix}_{K+} = \begin{bmatrix} 1 & \delta t & 0 & 0 \\ 0 & 1 & 0 & 0 \\ 0 & 0 & 1 & \delta t \end{bmatrix} \begin{bmatrix} \Delta h_{ins} \\ \Delta v_{dins} \\ \Delta h_{baro} \\ \Delta \hat{h}_{baro} \end{bmatrix}_K + \begin{bmatrix} 0 \\ W_{acc} \\ W_{baro} \\ W \end{bmatrix}$$

Where  $\alpha = (\beta^2 + w^2)^{1/2}$   $X_{k+1} = \phi_k X_k + W$

Process noise

$$Q = E [WW^T] \text{ and } W \triangleq [0, W_{acc}, W_{baro}, w_{baro}]$$

$$Q_k \approx \begin{pmatrix} A\delta t^3 / 3 & A\delta t^2 / 2 & 0 & 0 \\ A\delta t^2 / 2 & A\delta t & 0 & 0 \\ 0 & 0 & 2\beta\sigma^2\delta t & 0 \\ 0 & 0 & 0 & (\alpha - 2\beta)^2 2\beta\sigma^2\delta t \end{pmatrix}$$

Measurement model:

The measurement is the difference between INS altitude and baro altitude,  $Z = (h_{ins} - h_{baro})$

The third and fourth state is optional, generally it has been observed that the altitude information from the combat aircraft for fusion is best combination of baro-altimeter, radio altimeter, radar and pitot tube, where the error is maximum 50 ft.

The algorithm is simulated with various flight trajectories. Several van-trials and aircraft sorties were carried out. The result of one of the aircraft sortie conducted at HAL, Bangalore on avro aircraft is plotted and shown in Fig. 9.

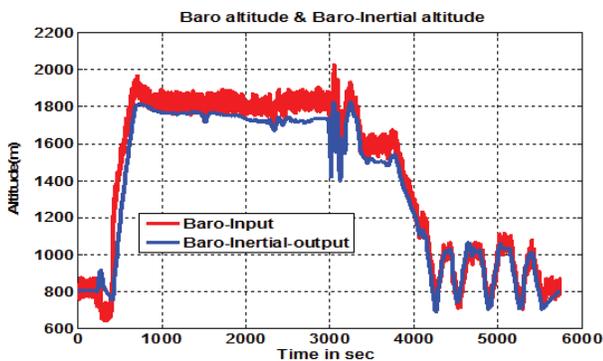


Figure 9. Baro Inertial performance.

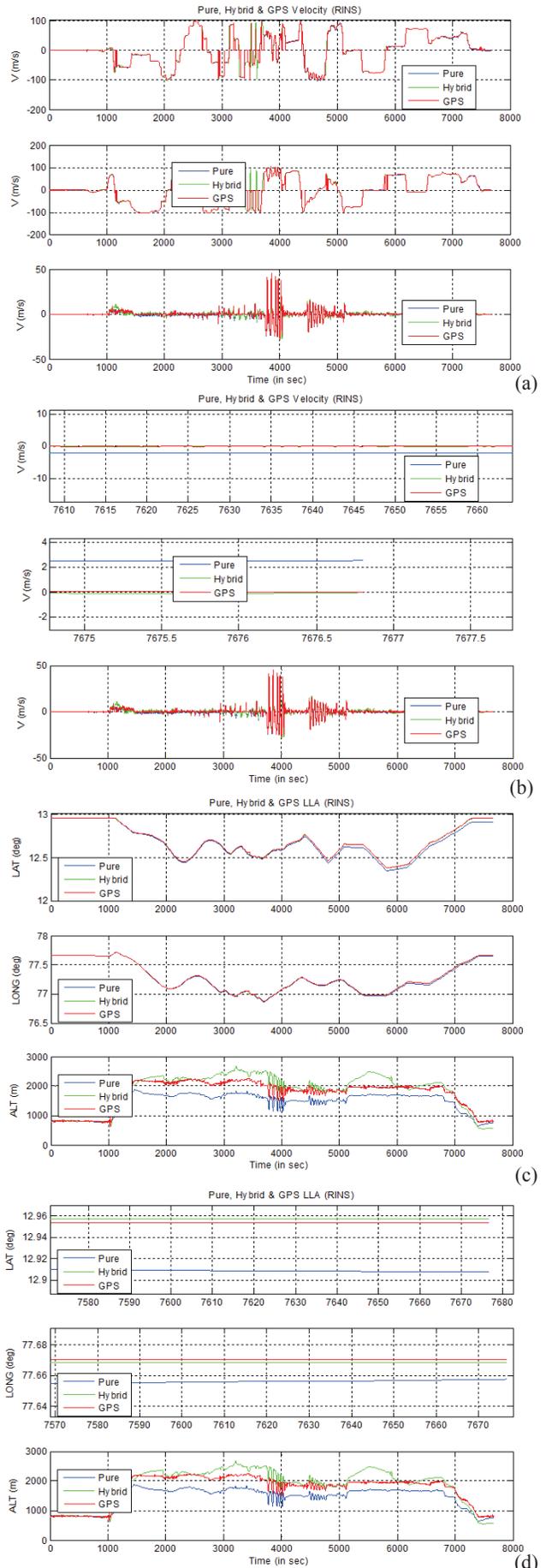


Figure 10. RINS performance in air-sortie.

## 7. AIRCRAFT SORTIES RESULTS

RINS system was tested number of times on HAL AVRO aircraft for the navigation performance. The GC, IF alignment and pure navigation performance along with Baro inertial Navigation and Hybrid KF navigation algorithm was tested by subjecting various conditions in flight and the results obtained was satisfactory. The results are plotted and shown in Fig.10.

## 8. CONCLUSIONS

The paper presented a high grade long duration aided inertial navigation system for long range missiles, combat aircrafts augmented with the measurements of the satellite navigation data, vehicle baro altimeter. The integration technique used was tightly coupled where the measurements was directly used to update the kalman filter. Simulation test carried out using a high dynamics RF simulator. System performance was verified for various aircraft flight trajectories with and without injection of sensor errors, alignment errors and GPS errors with position fix loss in the trajectory profile.

The performance of the proposed integrated navigation solution was demonstrated to be very competitive with the imported navigation system. The results have been examined to verify the suitability and satisfactory performance of the proposed solution even with degraded/multipath or totally denied GPS for long durations. Various van-trial and aircraft sorties were carried out to demonstrate the performance of RINS.

The efforts have resulted in building an indigenous state-of-the-art, ready to integrate aided inertial navigation system to the various needs of the Defence forces of India.

In the aftermath, the navigation systems of Indian origin will further have enhanced features like vision aided navigation, inclinometers, GAGAN, and IRNSS aided high precision.

## ACKNOWLEDGEMENTS

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## Contributors



**Mr Satheesh Reddy** obtained his BTech (Elect. Comm. Eng.) from JNTU Ananthapur and MS (Elect. Comm. Eng.) from JNTU, Hyderabad. Presently working as Associate Director of RCI, DRDO and spearheading the Technology Group of Navigation Systems. With more than two and half decades of experience, contributed towards design, development and productionisation of advanced Navigation technologies for the medium, long range and tactical missiles like Agni, Prithvi, Brahmos, Dhanush, Prahaar, AD, ANSP, Astra; Fighter Aircrafts like LCA, Jaguar, Sukhoi-30 and other strategic missions in India.



**Dr V.K. Saraswat** obtained his MTech from IISc, Bangalore and PhD (Propulsion Engineering) from Osmania University, Hyderabad. Presently working as a Scientific Advisor to Raksha Mantri (SA to RM), Director General, DRDO and Secretary, Department of Defence R&D, India. Under his leadership, India has embarked on a challenging, futuristic Air Defence Programme encompassing development of complex anti ballistic missile systems, Radars, C4I systems and integration of battle management resources into a national authority.