Scheduled Gain EKF for Data Fusion for re-entry Navigation

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Abstract

The Space Capsule Recovery Experiment (SRE) is the first Indian reentry mission in which a space vehicle is de-orbited from a low earth orbit and recovered in the Indian coastal waters on 22nd of January 2007. The main objective of SRE is to provide a microgravity platform for scientific experiments, demonstrate a host of new technologies for safe reentry of future manned space modules into the earth's atmosphere and safe landing. One of the major challenges in SRE is the Navigation and Guidance system. For the SRE, a precision hybrid navigation system based on an INS, and a GNSS receiver is used. A robust EKF for integration of the two systems is designed, developed and successfully used to achieve the mission. As per the mission plan, the capsule is to be de-orbited under the control of the closed loop NGC system to achieve the re-entry pill- box at 100 km altitude very precisely. Below 100 Km, the aerodynamically stable capsule follows ballistic flight, terminal velocity reduction is achieved by parachutes for safe splash down within the specified impact zone for recovery. The impact point accuracy is achieved by controlling the accuracy of the reentry pill box. This paper presents the major design aspects of the EKF and the flight performance of the Hybrid Navigation System.

Keywords

SRE : Space Capsule Recovery Experiment
GNSS : Global Navigation Satellite Receiver
INS : Inertial Navigation System
GAINS : GNSS Aided INS
CLG : Closed Loop Guidance
CEP : Circular Error Probable
NGC :Navigation Guidance Control

1. Introduction

The SRE, a 550Kg satellite (fig.1) is launched into a 625Km circular polar orbit on 10th January 2007, and maintained in the orbit for 12 days for carrying out microgravity experiments. The overall mission profile is shown in fig2 and 3. On the 10th day, an orbit transfer is carried out, by burning the thrusters for 140sec, under the control of the closed loop NGC system, to achieve a precise elliptical orbit which give daily repeating ground trace over SHAR ranges. On the 12th day, the final deboost burn is done to achieve atmospheric reentry under the NGC system. The target reentry pill box is specified at 100 Km which is achieved with great precision, so that ballistic descend trajectory provide the ground impact point CEP within specifications.

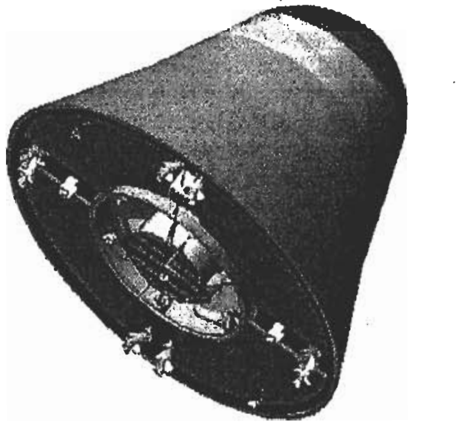


Fig 1. SRE Capsule

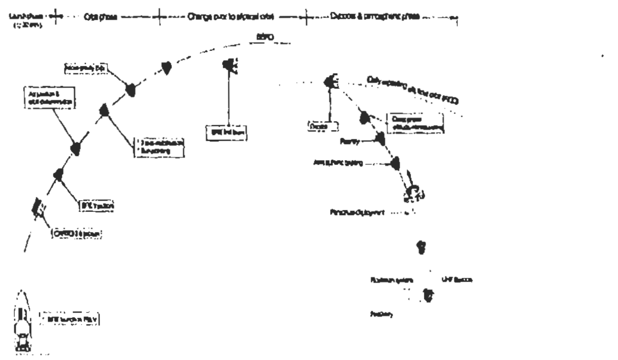


Fig 2. Mission Profile

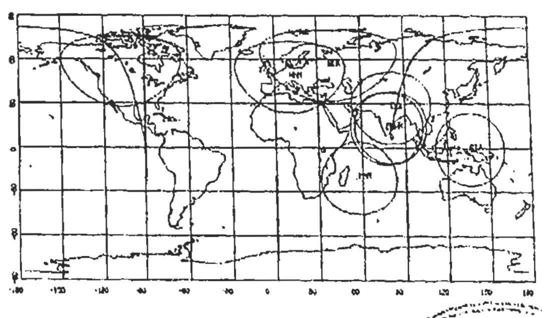


Fig 3. Reentry Trajectory Ground Trace

The orbit transfer and the final deboost maneuvers demand navigation system accuracy of 0.2 m/s and 90m (3σ) accuracy in velocity and position.

The navigation accuracy for the mission is achieved by using a realtime EKF in the feed forward configuration for data fusion of the strapdown INS and the GNSS. The major error sources are the uncertainty in initial state vector and attitude at initiation of navigation for the deboost. The propagation of navigation errors due to the above sources can not be corrected by an autonomous INS. The EKF corrects the INS navigation (position, velocity) errors due to all the error sources internal and external to the INS.

The design of the realtime EKF is the major challenge. Various novel techniques, realtime gain scheduling scheme, measurements based filter initialization scheme for fast convergence has been developed. Several error handling schemes for robustness viz loss of GNSS data, wild sample etc are designed, developed and validated.

Excellent navigation accuracy has been achieved using the Hybrid navigation EKF in the orbit transfer and final de-boost phases. In the orbit transfer phase, perigee accuracy of 15m was achieved. In the final de-boost, the reentry pill-box and final impact point has been achieved very accurately. The end to end design, development, validation and flight experience are presented.

2. NGC System Configuration.

Overall GNC system (Fig 4) consists of the following

- A precision INS and High dynamic GNSS receiver system.
- Mission management unit based on radiation-hardened processor.
- Fine sun sensors and Magnetometers for on orbit attitude determination and inertial attitude update, and Magnetic torquers and thrusters for attitude control.
- Eight numbers of 22 N thrusters for de-boosting and attitude control operations.
- Hybrid navigation algorithm with elaborate error handling logics and software.
- Robust guidance algorithm for deboost till reentry.
- On orbit gyro drift estimation and accelerometer bias estimation
- Orbit Determination (OD) based on S-band tracking and GNSS and Orbit propagation.
- Ground link for telemetry and Tele-command.
- Ground software for Mission planning, trajectory design and NGC system design.

The IMU cluster is configured with three linear servo accelerometers and two Dynamically Tuned Gyroscopes. The sensor outputs in a fine range is used for navigation and a coarse range is used for re-entry measurements. The IMU cluster temperature control has the provision for three precise set points 50, 60 & 70 °C. The IMU is evaluated by special low acceleration input tests in addition to standard calibration to meet the mission specific requirements. The digital outputs of the accelerometers and gyros are interfaced to the Mission Management Computer. The IMU package is shown in Figure 5 & 6. The sensors, accelerometers and gyros have been specially designed to operate in thermo vacuum condition for long duration and evaluated for their accuracy/stability under thermo vacuum.

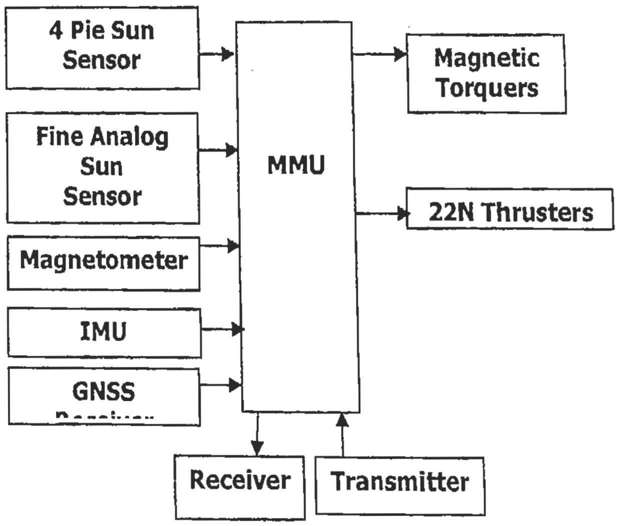


Fig 4. NGC System Configuration

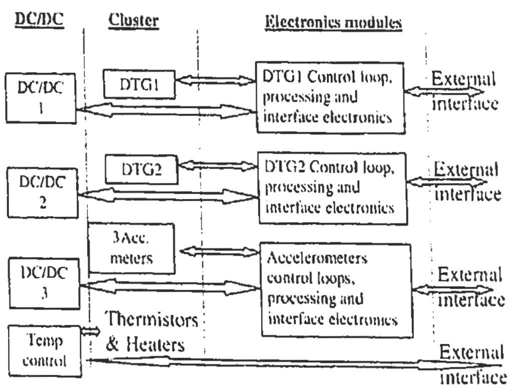


Fig 5 IMU Configuration

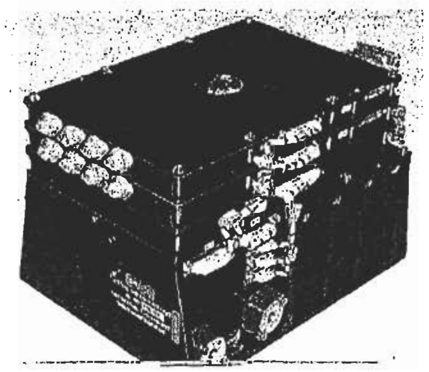


Fig 6 IMU package

3. The EKF Architecture and System Dynamics Model

An open loop feed forward configuration (Fig 7) is used. The filter estimate the INS error (position and velocity error) and corrects in the forward path.

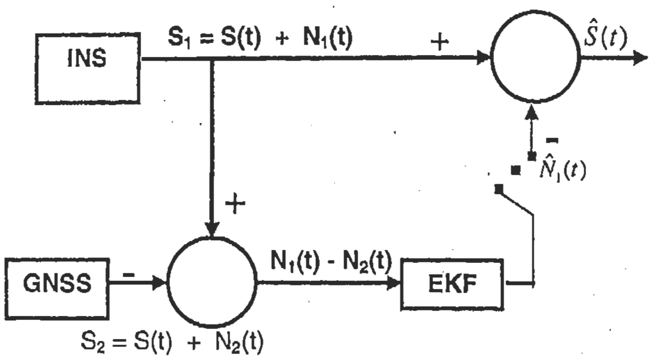


Fig 7. EKF Configuration : Open loop Feed Forward

The state variables are selected after considering the mission trajectory and velocity, position accuracy requirements so that the reentry pill box accuracy is met. The 12 state variables are

$X = [\ \delta rx, \delta ry, \delta rz, \delta vx, \delta vy, \delta vz, bx, by, bz, mx, my, mz\ ]^T$ (1)
$=$ [INS position error, INS velocity error, Accelerometer bias, attitude error]^T

The attitude error is modeled as three misalignment angles by which the computed inertial reference frame is misaligned from the true inertial frame,

$$C_b^{\hat{i}} = C_I^{\hat{i}}\, C_b^I = (I + \Omega m)\, C_b^i, \quad (2)$$

C_b^i is body to inertial DCM,

$$\Omega m = \begin{bmatrix} 0 & -mz & my \\ mz & 0 & -mx \\ -my & mx & 0 \end{bmatrix} \quad (3)$$

Computational frame misalignment

The system model for the 12 state EKF is

$$\dot{X} = AX + w \quad (4)$$

w is white Gaussian noise

Where

$$A = \begin{bmatrix} 0 & I_3 & 0 & 0 \\ J_G & 0 & C_b^I & A_I \\ 0 & 0 & 0 & 0 \\ 0 & 0 & 0 & 0 \end{bmatrix} \quad (5)$$

$$Acci = \begin{bmatrix} 0 & Azi & -Ayi \\ -Azi & 0 & Axi \\ Ayi & -Axi & 0 \end{bmatrix} \quad (6)$$

Sensed acceleration

$$J_G = \mu / r^3 \begin{bmatrix} -1+3x^2/r^2 & 3xy/r^2 & 3xz/r^2 \\ 3xy/r^2 & -1+3y^2/r^2 & 3yz/r^2 \\ 3xz/r^2 & 3xy/r^2 & -1+3z^2/r^2 \end{bmatrix} \quad (7)$$

The measurement is the 6 dimensional observation of the INS error obtained by differencing the INS and GNSS data.

$Z = (\text{INS} - \text{GNSS})$
$Z = HX + n;\ \ Cov(n) = R,$ (8)
n is white Gaussian noise of GNSS error

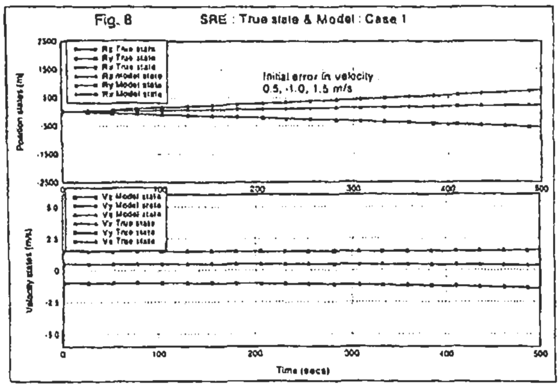
$$H = [I_6 \ \ 0_6] \quad (9)$$

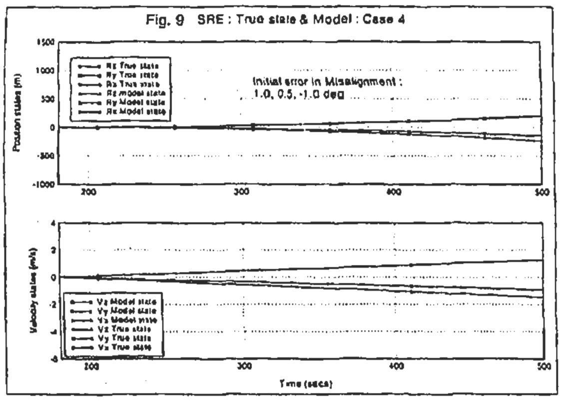
The discrete time state transition equation used for realization is

$X_{k+1} = \phi_k X_k + w,\ Cov(w) = Q,$ (10)
w : white Gaussian Noise
$Cov(X_0) = P_0$
$\phi_k = I + A_K\, dt + (A_K\, dt)^2/2 + (A_K\, dt)^3/6$ (11)

4. Model Validation

The 12 state discrete time INS error dynamics model (eqn. 10) is validated by simulation. A complete 45 state INS truth model is used as reference. The EKF states are propagated using the state transition model and is compared with the true states from the truth model. Typical comparison plots are shown in fig 8 & 9 and which prove the adequacy of the model.





5. Process Noise Estimation of 'Q'

(100 combinations) One set at a time

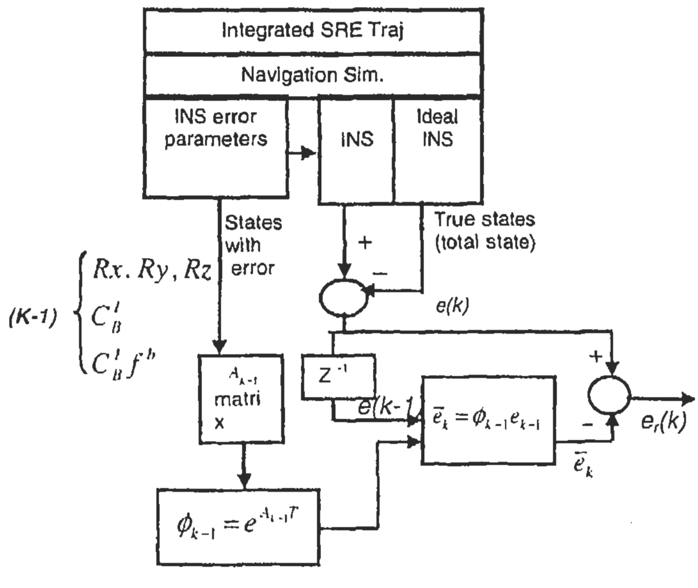


Fig 10.
Process Noise 'Q' estimation: Simulation

A simple and robust design using stored Kalman gains is adequate for this application instead of online gain as used in the launch vehicles. The process noise Q is estimated by an 'Monte Carlo' simulation (Fig. 10) by comparing the complete INS model and the state transition model. All the INS error parameters are perturbed in a Monte Carlo sense and the error per step in the state transition model of the EKF is obtained for the entire trajectory. The process noise is obtained by computing the ensemble statistics of the error per step of the model.

The Q is computed from 100 runs where all the modeled and unmodeled errors of the INS are perturbed. The selected Q is validated by 1500 Monte Carlo runs including off nominal mission scenarios.

6. Estimation of Measurement Noise R

The measurement noise 'R' is estimated based on

- Data collected from on orbit GNSS receiver and S-band tracking data of IRS satellites.
- RF simulations of SRE receiver for the mission trajectory and antenna pattern.
- Constant R is used 60m, 0.6 m/s (1σ)
- R is validated in simulation by varying noise level from 0.5 σ to 12 σ.

7. Filter Initialization and Selection of P_0

The filter states are initialized by averaging of few initial measurements. This give a very good estimate the initial state. However, the P_0 is fixed conservatively which result in higher initial gain for the measurement and hence faster convergence.

Observability Grammian and reachability Grammians were analyzed and found to have full rank. Because stochastic observability & reachability and boundedness of ϕk, Q,R are ensured, this guarantee that for large k the behavior of P_k is unique independent of P_0 [Frank Lewis, 1986]. This also guarantee uniform asymptotic stability of KF error system. ie, no divergence.

Observability Grammian:

Consider the following discrete time varying system

$$X_{K+1} = \phi_K X_K + w_K \qquad w_K \sim (0, Q_K)$$
$$Z_K = H_K X_K + v_K \qquad v_K \sim (0, R_K)$$

The observability Grammian for the above time varying system is $W(N,i) = \sum_{k=i}^{N-1} \phi^T(k,i)\ H_k^T\ R_k^{-1}\ H_k\, \phi(k,i)$

for finite N

Where $\phi(K,i)$ is the state transition matrix from initial time i to the instant k. In our case H and R are time invariant but ϕ is time varying.

The above observability Grammian has been computed for different phases of the mission trajectory and the Grammian has full rank 12

Reachability Grammian

A plant is uniformly completely reachable / stochastically reachable, if for every i the Reachability Grammian has full rank

$$\sum_{k=i}^{N-1} \phi(N,k+1)\, Q_k\, \phi^T(N,k+1) \quad N > i$$

This reachability grammian was computed for different phase of the mission and found to have full rank 12

8. Robustness of Filter Logics

- Enable/disable of aiding by ground command after assessing GNSS performance.
- INS data ensured to guidance under KF not started or stopped due to any error condition.
- 12 state KF : INS error states.
- No feed back to INS : position, velocity error states used to correct the INS data in forward path.
- KF start logic : averaging based on 5 measurement.
- Measurement update : GNSS data after validation.
- GNSS data latency correction using time tag & INS acceleration.
- Limit check on PDOP and innovation sequence for wild sample rejection.

GNSS Data Integrity Checks :

➢ **Level 1 (before KF)**

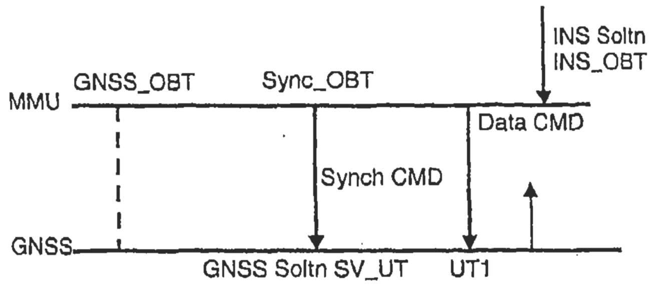
- Done by Data acq & preprocessing
- GNSS integrity/checksum check
- GNSS receiver (hardware) check
- Time synch validity check
- Availability of fresh solution
- Availability of solution in 3D mode
- GNSS PDOP checks (PDOP < 10)
- KF get synthesized status (data ready)
- State vector in ECI to KF

➢ **Level 2 : Validation before EKF usage**

- GNSS data loss handling(revert to INS if loss > 50s consec..)
- Rejection of GNSS wild sample [(measured - predicted) > limit], 12 sigma
- Stop aiding on persistence of any error (50s)
- KF prediction mode under data loss after KF convergence (496 sec).
- INS data correction start only after (100sec) initial convergence.
- Start of KF only when GNSS data is available (5 samples continuous)

9. GNSS Latency Measurements

The GNSS data latency is measured using time tags and a synch command. The measured latency, INS acceleration are used to project the GNSS data to the current instant of time before using by the filter.



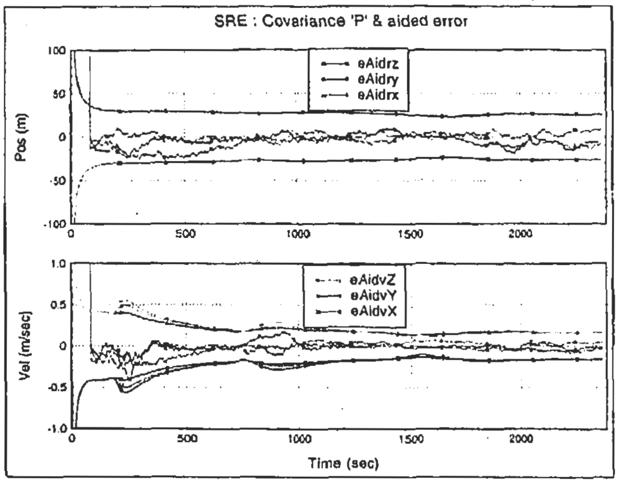
10. Kalman Gain Selection & Scheduling

- Stored gain with scheduling based on events and time slicing is selected as follows.
  12 gains : for thrusting phase
  1 gain : for attitude hold phase
  4 gains : reorientation/coast till reentry
- Separate gain schedule used for REO and Deboost phases.
- The gains are validated in off nominal cases and a Monte Carlo simulation.
- Monte Carlo : 1000 cases (integrated)
- Monte Carlo : 500 cases INS & GNSS (up to reentry)

10.1. Design Validation & Robustness Studies

- Low and High measurement noise
- GNSS data loss (continuous & intermittent)
- GNSS wild samples: multiple burst of noise

- GNSS latency: data synchronization
- Actual GNSS error obtained from RF simulation of Flight receiver
- Perturbation cases
    - Up to $\pm 12\sigma$ INS & GNSS errors
    - $\pm 32\%$ thrust perturbation
    - GNSS data loss
    - GNSS data with wild samples
    - Q perturbation
    - KF gain perturbation



11. Flight Performance & Conclusion

1. The velocity, position obtained in the mission are obtained by comparing with GNSS samples :

   Orbit transfer: <0.2 m/s, 10 m at thrust cut off
   Final deboost : <0.2 m/s, 60 m
2. The reenty pillbox errors are very low and the error translated in term of impact point error is <<1σ of the specifications.
3. The velocity, position error in the coast phase after the critical burn phase were slightly high due to sub optimality of gains. However, this has no concern to the mission accuracy.
4. The main reason of the above behaviors is due to the impracticability of finding a gain which is suitable for all seasons and launch time. It is possible to get rid of the problem to a large extend by changing the coordinate system to ECEF instead of ECI.
5. The GNSS Rx performed good during the critical phase, there was no data loss.
6. After the thrust cut off, during the reorientation maneuver the GNSS noise level and PDOP increased considerably. This may be taken care by adapting R based on PDOP or limit checking on innovation sequence.
7. The filter logics for robustness have worked very effectively, the filter initialization scheme, latency measurement scheme etc worked excellently.

12. References

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