TECHNICAL PRESENTATION-2

Navigation System For Closed Loop Guidance of Space Capsule Recovery Experiment (SRE) : Development & Flight Performance

B.C.Vidwani, **P. P. Mohanlal**,
A. Abdul Shukkoor, J. Krishnakumar, S. Syamala

ISRO Inertial Systems Unit, Trivandrum

Abstract

The Space Capsule Recovery Experiment (SRE) is the first Indian experiment in which a space vehicle is de-orbited and recovered from 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 the capsule into the earth's atmosphere to splash down at a pre-planned location, and recovery of the capsule from the sea. One of the major challenges in SRE is the Navigation Guidance Control (NGC) system. The core of the NGC system is a precision hybrid navigation system comprising of a inertial navigation system aided by onboard satellite navigation system. The key function of the NGC system is to achieve the pill- box at 100 km very precisely so that the impact point is achieved within limits. This paper presents the major design aspects and the flight performance of the navigation system of SRE.

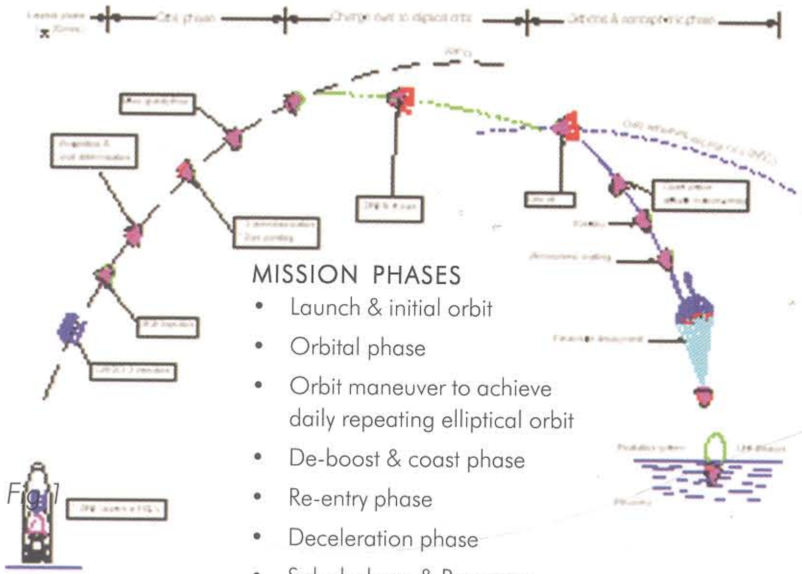
1. INTRODUCTION

The Space Capsule of SRE was launched on 10th January 2007 by PSLV-C7, the seventh operational flight of Polar Satellite Launch Vehicle of Indian Space Research Organization (ISRO) from the Shriharikota launch complex. After 12 days in orbit, it was de-boosted to splash down in the Indian waters off the coast of Shriharikota. Fig :1 shows the overall mission profile including REO and Deboost phase and Fig : 2 shows the final deboost ground trace and ground stations used for telemetry, Tele-command and tracking. The SRE Capsule itself is shown in Fig : 3. One of the major system required for this is a high precision navigation system which includes the necessary hardware and software. Precision orbital navigation is extremely demanding since the system has to meet stringent accuracy requirements even under likely large errors in initial state vector and attitude as the navigation has to be initiated during orbital motion.

The capsule has to be navigated to enter earth's sensible atmosphere (100 Km) with a precise position, velocity, Flight Path Angle (FPA) and angle of attack. If FPA is large the module in its atmospheric flight will experience large deceleration and heating which can lead to disintegration and burning up of the capsule from large structural and thermal loads, created by large aerodynamic uncertainties. If FPA is small, large dispersions in landing site location will be created by large aerodynamic uncertainties. Still, smaller FPA will result in capsule moving to another lower orbit, reaching earth at much later time with unpredictable landing location. A wrong orientation (angle of attack) of the capsule can destabilize the capsule causing tumbling and burning up.

Even though the autonomous Inertial Navigation System IMU (Inertial Measurement Unit) had very high accuracy, detailed studies showed that only Hybrid navigation (INS aided with SPS) could meet the mission accuracy even under large errors in initial state vector and attitude and INS errors. For this purpose, SPS aided navigation algorithm based on optimal Kalman filter was designed and developed.

Excellent navigation accuracy has been achieved using the Hybrid navigation for SRE both in DBM1 : REO (Repetitive Elliptic Orbit) and final Deboost phases. In the first phase (DBM1), perigee accuracy of 15m was achieved. In the final Deboost, based on the observed state vector at 100Km, the Down range dispersion is less than 1 Km from the preflight optimized impact point.



MISSION PHASES

- Launch & initial orbit
- Orbital phase
- Orbit maneuver to achieve daily repeating elliptical orbit
- De-boost & coast phase
- Re-entry phase
- Deceleration phase
- Splash down & Recovery

Fig 1

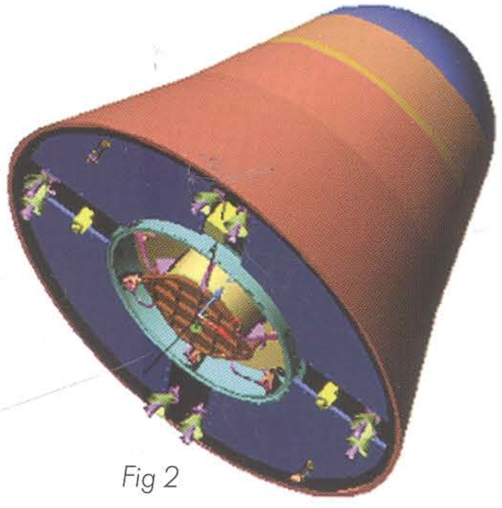


Fig 2

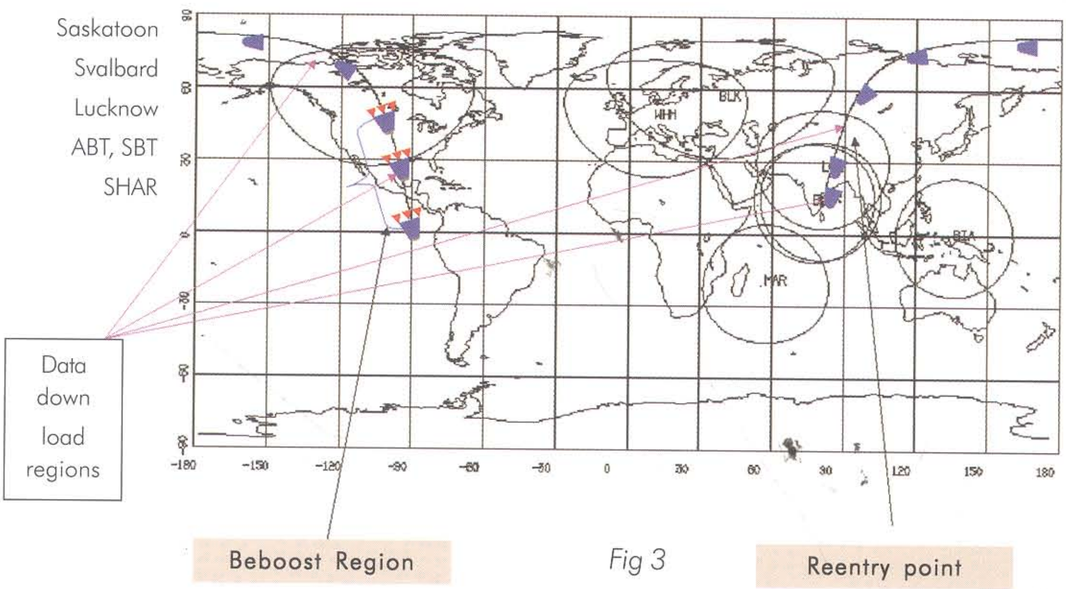


Fig 3

2. MISSION OBJECTIVES

To test and qualify the following reentry technologies under actual flight conditions.

- Validate the navigation, guidance and control algorithms and systems.
- Validation of the aero thermo structure and systems design and the material of the thermal protection system and aero thermo dynamic predictions.
- Assess the performance of the parachute and recovery systems and operations.
- Gain experience on carrying out microgravity experiments.
- Utilize the technologies developed for future Reusable Launch Vehicle and for crew return module design for Indian manned space missions.

3. MAJOR MISSION EVENTS

Events		
S/C power on	:	23:23 UT (9.01.07)
S/C initialization	:	lift off – 4 hrs
IMU ON	:	23:53 UT (9.01.07)
Lift off	:	03:53 UT (10.01.07)
Space capsule injection in orbit	:	orbit achieved
Microgravity experiments	:	3 days
Ground track navigation	:	15.01.2007
Trial navigation(900 sec duration)	:	2:59:59:915 UT(18.01.07)
Preparations for first burn for REO	:	19 & 20 Jan 07 Reorientation for deboost orientation, Navigation start, Burn & CLG start -
Burn duration	:	133.823 sec
REO orbit achieved	:	639.159 x 485.488 km (planned 639.565 x 485.473 km)
Preparations for final burn	:	21 & 22 Jan 2007
Reorientation for deboost orientation	:	3:07:28:173(Deboost – 20 min)
Navigation start	:	3:27:28:173 UT (22.01.07)
Deboost burn & CLG start	:	Nav start + 180.224 sec
Burn end (558.045 sec burn)	:	Nav start + 738.269 sec
Attitude hold	:	180 sec from burn end
Reorientation for reentry	:	Nav start + 918.269 sec ~ 180 sec duration
Coasting	:	1470.555 sec
NGC termination (Re-entry)	:	Nav start + 2388.823 sec 04:7:16:99
RF Black out	:	04:08: 14 to 04:10:16
Drouge chute,Pilot chute, Main chute	:	04:13:18, 04:13:19, 04:14:09
SRE Impact	:	04:17:51
Impact point error due to NGC	:	1 km

4. NGC SYSTEM

Overall NGC system consists of the following

- A precision inertial navigation system and High dynamic SPS receiver system.
- Hybrid navigation algorithm with elaborate error handling logics and software.
- Mission management computer based on radiation-hardened processor.
- 8 nos. of 22 N thrusters for de-boosting and attitude control operations.
- Robust re-entry guidance algorithm and software for meeting large thrust perturbation of the thrusters and Telemetry, tracking and Tele-command systems.
- Fine sun sensors and magnetometers for on

orbit attitude determination and inertial attitude update and Magnetic torquers and thrusters for attitude control.

- On orbit gyro drift estimation and accelerometer bias estimation
- Orbit determination based on S-band tracking, uplink and Satellite navigation based OD and Orbit propagation for navigation systems initialization
- Mission planning and trajectory design and Guidance design software for the finalized mission plan.

5. NAVIGATION SYSTEM DESIGN AND DEVELOPMENT

Major design and developments for the navigation system includeIMU, High dynamic SPS receiver, Inertial navigation algorithm, Kalman filter design for hybrid navigation and optimization for onboard implementation and Flight software validation by SIP, OILS, HILS.

5.1 IMU

The IMU cluster is configured with three linear servo accelerometers and two mini-TDOF Dynamically Tuned Gyroscopes (DTG). The input axes of the three accelerometers are aligned along three mutually perpendicular axes (X, Y, Z). The maximum accelerations during deboost and re-entry is about 40 mg and -10g respectively. To achieve desired navigation accuracy two output ranges are switched depending on the phase of the trajectory. Fine range +1.5g range is selected up to re-entry and coarse range + 15g range is selected by RTD upon sensing 100km altitude to enable reentry deceleration measurement for monitoring purpose.

The DTGs have a overall dynamic range of +10°/s. however, the dynamic range of + 3°/s with fine resolution during orbital phase and Deboost to re-entry phase is selected to enhance navigation accuracy. The analog outputs have the dynamic range of +10°/s which is used for monitoring rates during reentry. The IMU cluster temperature control has the provision for three precise set points 50, 60 & 70 °C. Set point of 50°C was found to be adequate and used for the mission.

The IMU is evaluated by special low acceleration input tests in addition to standard calibration procedure. The digital outputs of the accelerometers and gyros are interfaced to the Mission Management Unit/ computer (MMU). The sensor compensation and navigation computations are carried out in the MMU. Fig 4, 5 & 6 show the axis convention, IMU package and configuration respectively.

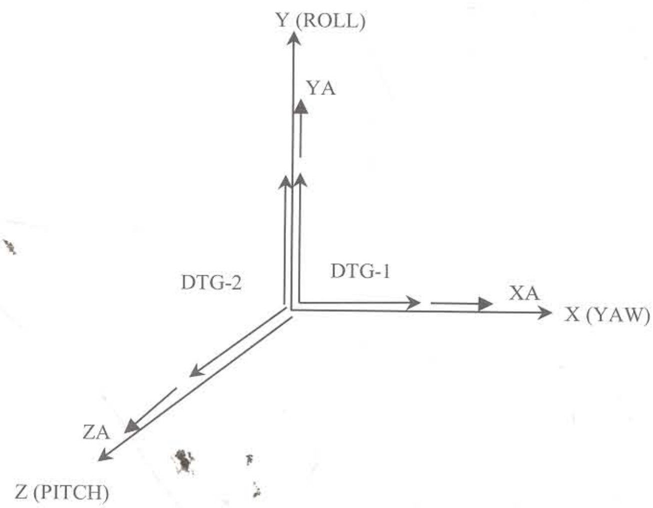


Fig 4

Fig 5

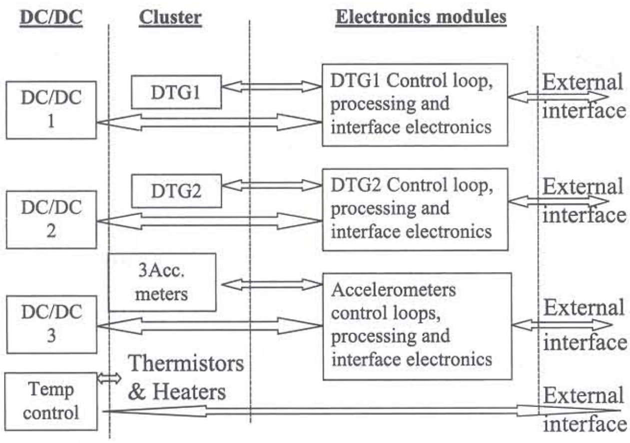


Fig 6

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.

5.2 High Dynamic SPS Receiver

An 8 - channel satellite navigation receiver with velocity range of 12 km/sec was developed and qualified [1]. The receiver was interfaced to MMU through MIL 1553 B interface. Two patch antennae, one for the orbit and Deboost phases and another for the reentry phase were used with switching. The receiver was tested extensively with the Deboost trajectory data (linear and angular) with measured antenna pattern using a RF satellite navigation simulator for visibility studies and navigation accuracy.

The RF simulation studies revealed that the spacecraft has to be re-oriented suitably, after the de-boost burn to ensure continued satellite visibility.

5.3 Kalman Filter Design for Hybrid Navigation

Even though the standalone inertial navigation system had high accuracy, the mission requirement specification could not be met with standalone INS, primarily due to uncertainities in initial state vector and attitude errors in addition to INS errors.

Detailed studies showed that Hybrid navigation (INS aided with satellite navigation) can meet the mission accuracy even under large errors in initial state vector, attitude and INS errors. For this purpose, a 12 state (error state) feed forward extended KF was designed and developed. The feed forward [2] configuration is shown in fig 7 below.

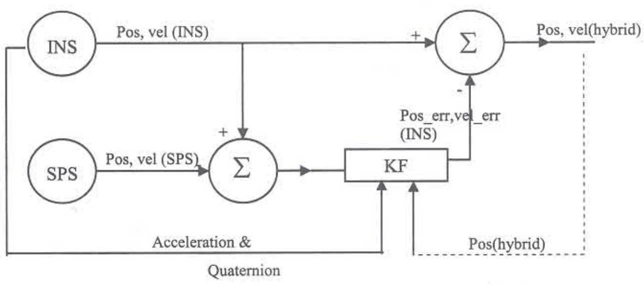


Fig 7

5.3.1 INS Error Model Validation

The 12 state error model for the INS was developed analytically and validated using a 6DOF trajectory simulator integrated with software model for INS. The full-fledged software model for INS and 6 DOF trajectory simulators are developed and validated for this purpose.

5.3.2 Estimation of Process Noise Covariance Matrix

Using the 6 DOF trajectory simulator integrated with software model for INS, the INS error parameters were perturbed in random combinations and by analyzing the ensembles of residual errors of the INS error model outputs in comparison with reference outputs, the process noise covariance, Q estimated for KF design.

5.3.3 Measurement Noise Covariance Matrix

Based on the RF satellite simulation studies with the actual satellite navigation receiver, statistics of the satellite navigation errors were analyzed and the measurement noise covariance matrix, R determined for KF design.

The real time KF gain computation was implemented and computation time measured. Since single processor was used for all onboard tasks, the optimized real time computed gain were stored onboard and scheduled based on time and events

with available computational power. This novel scheduled gain scheme could meet the mission accuracy with 10 fold decrease in computation load.

5.3.4 KF Design Robustness Studies

The design was validated by the following robustness studies.

- Wild measurement noise (uniform distributed and bursts)
- Data loss (intermittent and continuous)
- Extreme levels of measurement noise (1/2 s to 12s).
- Gain sensitivity studies (+6dB)
- Q sensitivity (1/10 to 10)

5.3.5 Filter Initialization, Convergence and Error Handling Logics

KF initial states, initial state error covariance matrix (P_0) are very important with respect to optimality and faster convergence. A novel scheme of initializing filter states has been developed. Few consecutive differences between INS data and satellite navigation data are averaged for initializing the filter. The initial state error covariance P_0 is conservatively chosen to speed up filter convergence and optimality is ensured by controllability analysis [3]. Error handling logics are built in to detect and manage wild samples, data loss, poor PDOP etc. Observability studies [3, 4] and analysis carried out to ensure filter performance in prediction mode after convergence. A similar hybrid navigation system was successfully flight tested in aircraft sorties to prove the various error handling logics, data synchronization scheme and performance assessment.

5.4 Navigation Software

The navigation software is resident in the onboard computer : the Mission Management Unit (MMU). The overall configuration of the software is presented in fig 8. The software is partitioned to four major modules namely, Gyro and accelerometer processing and compensations, Quaternion propagation, Navigation computations, Aided navigation kalman filter.

The gyro and accelerometer processing is done continuously from power ON in the ground, navigation and aided navigation are done for the Deboost phases only.

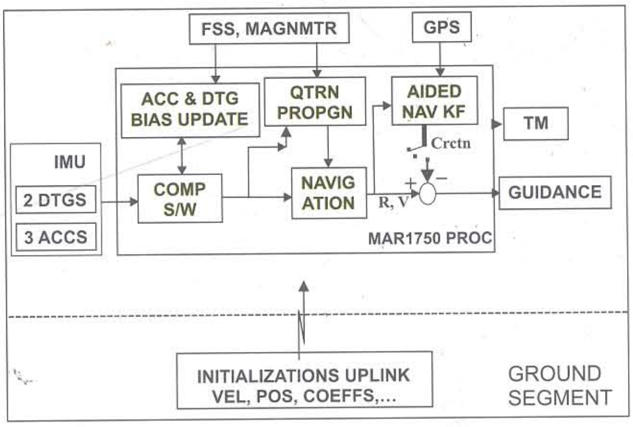


Fig 8

5.4.1 DTG and Accelerometer Processing

The sensor data is acquired by the MMU every 32 ms. The data consists of 3 incremental angles, 3 incremental velocity, 3 angular rates, 3 accelerations in body coordinates. The angular rates and accelerations are only for monitoring purpose. The angle increments are compensated for gyro errors, fixed drift, axis misalignments and linear acceleration sensitive errors. The error parameters are obtained from laboratory level characterization and calibration coefficients are up-linked to the computer. A second level of drift correction is also carried out based on estimation of gyro drifts onboard.

The velocity increments are compensated for accelerometer errors, bias and axis misalignments. The error parameters are obtained from the laboratory calibration. After reaching the orbit, the accelerometer biases are estimated and updated. The accelerometer data are checked for reasonableness before using for navigation.

5.4.2 Quaternion Propagation

The gyro angle increments are used to propagate the attitude quaternions every 32 ms. The injection attitude is up linked. The gyro quaternions are updated based on Sun-Mag attitude determination. The gyro quaternions are used for controlling the spacecraft attitude. The quaternions are used by the navigation computations also for inertial referencing of acceleration.

5.4.3 Inertial Navigation Computations

The inertial navigation processing is done in 512 ms periodicity. Computations are done in the ECI coordinates. The body mounted accelerometer measurements the gyro quaternions and earth's gravitational field model are used to obtain the velocity and position of the capsule. The oblate earth gravitational field model upto second harmonics (J2) are considered. The initial state vector for navigation is estimated based on orbit determination and orbit propagation and uplinked 6 orbit prior to deboost operation. The navigation computations start 3 minutes prior to deboost so as give time for the aided navigation kalman filter to convergence before deboost start.

5.4.4 Software Validation

The well-established software engineering practices are followed to develop high quality flight software. The flight software is developed ADA language. The software has been validated by unit testing, static integration testing in software emulation/hardware emulation. Dynamic input testing by simulated input profile (SIP) in offline and MMU level. Testing with IMU with software resident in checkout and static and dynamic input to IMU.

Major software developments are carried out to realize closed loop mission trajectory simulation. This simulated the mission dynamics, propulsion, IMU, SPS, navigation guidance, control and various perturbations. This is the major tool for design of the mission and NGC system and perturbation and Monte Carlo analysis for design and validation of the NGC software and systems. The integrated software is tested in closed loop with OILS (onboard processor in loop simulation), HILS (hardware in loop simulation) where IMU is mounted on angular motion simulator and acceleration and SPS by digital simulation.

6. Flight Performance

- For IMU in the orbital condition, accelerometer stability of $<50\mu g$ and gyro stability of the order of 0.1deg/hr achieved
- Hybrid navigation accuracy of 20m (position) and 0.25 m/s (velocity) were achieved

DBM1 (REO) performance achieved is tabulated below.

	Apogee (km)	*Perigee (km)*	*Inclination (deg)*
Target	639.565	485.473	93.939
Achieved	639.159	485.488	93.937
Error	-0.406	0.015	-0.002

DBM2 (final De-boost) performance

	Down range (Km)	*Cross range (Km)*
Actual error due to NGC (Hybrid Nav)	1	0.34
Hybrid Nav pre-flight MC (3s)	35	4.1
Standalone INS pre-flight MC (3s)	81	4.9

7. Conclusions

- The hybrid navigation system performance was excellent and hence resulted in very precise impact point as planned.
- The mission has met all the intended objectives in the maiden experiment itself.

8. References (Books)

1. Pratap Misra & Per Enge, "Global Psitioning System", Ganga Yamuna Press, 2002.
2. Bradford W. Parkinson, James J. Spilker Jr; "Global positioning system : Theory and Applications", Vol I & II, Vol 164, progress in Astronautics and Aeronautics, A1AA, 1996.
3. Frank L Lewis, "Optimal Estimation: with an introduction to stochastic control theory" John Wiley and sons, 1986.
4. Robert Grover Brown & Patrick Y.C. Hwang, "Introduction to Random Signals and Applied Kalman Filtering", John Wiley & Sons, 1997.

P P Mohanlal

- **B.E** degree, Electronics & Communication, 1976, Madras University (PSG College of Technology, Coimbatore)
- **M-Tech** degree, Applied Electronics & Instrumentation, Kerala University, First class, Distinction with first rank.
- **PhD** degree, Kerala University, 2005 **"Development of Fuzzy Models for Stability based optimal control and Observer Design for nonlinear systems"**
- **Senior Member of IEEE** for the last 11 years.
- **Life Member, System Society of India**
- **Joined VSSC, March 1977**, in Test & Evaluation Division, (**Developed Test methods, test systems for testing electronic components for space applications**)
- 1991 to 1993 : Quality assurance tasks of Launch Vehicle Avionic Systems for PSLV project.
- **1995-2001 : Control, guidance design group developing Kalman filtering methods for various applications**
- **2002 onwards, Head of Navigation Software and Simulation Division of ISRO Inertial Systems Unit**
- Guided 13 M Tech thesis of Kerala & Bharathiar Universities.
- Guided 15 B Tech projects of Kerala, CUSAT & Calicut universities.
- Research Interests include:
  - Intelligent control methods for nonlinear systems (Neural, fuzzy and neuro-fuzzy methods).
  - Optimal filtering and control
  - DSP & Digital filter design using feedback neural network.
  - Hybrid Inertial Navigation Systems
- Currently, **member of Post Graduate board studies for Computer Science of the University of Kerala**
- **Research Publications -12** (International Journal & IEEE Conference Papers).
- Books (1)**"Neuro –Fuzzy Control"**, Narosa Publishing House, New Delhi, 1998.