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Modernized IRNSS Broadcast Ephemeris Parameters

M V Chandrasekhar D.Rajarajan G Satyanarayana Neetha Tirmal
S C Rathnakara A S Ganeshan
Space Navigation Group, ISRO Satellite Centre, Old Airport Road, Bangalore – 560017
* E-mail of the corresponding author: sekharmv@isac.gov.in

Abstract

India has successfully stepped into satellite Navigation system with the launch of its first three IRNSS satellites IRNSS 1A, 1B and 1C. IRNSS provides two types of services, Standard Posting Service (SPS), which is open for civilian use and the Restricted Service (RS), for authorized users. The system is set to change the facet of navigation, surveying, transportation, precision agriculture, disaster management and telecommunication in India. In any navigation system, broadcast navigation parameters are of paramount importance in arriving user position solution at user receiver end. IRNSS Navigation data is classified as primary and secondary Navigation parameters. Primary navigation data of a satellite principally represents its own orbit and onboard clock offset in the form of quasi-keplerian elements and clock coefficients (Bias, Drift and Drifts rate) respectively. Whereas secondary navigation parameters includes satellite almanac, ionosphere delay correction messages, differential corrections, Earth orientation parameters and IRNSS Time offset with respect to other GNSS. In existing IRNSS system satellite ephemeris of primary navigation parameters are broadcast in the form of 15 quasi-keplerian elements valid for a period of 2 hours or more. Spacecraft ephemeris which represents orbit in the form of 9 parameters, i.e., position, velocity and acceleration component of spacecraft in Cartesian coordinate system are chosen from Russian Global Navigation satellite system (Glonass) to improve Time to First Fix (TTFF) of IRNSS system with similar existing orbit accuracy. In addition, two models of user receiver orbit propagation algorithms with proposed ephemeris are briefed and their results are compared with standalone Glonass model. Generation of IRNSS ephemeris in Cartesian coordinate system and description of user receiver orbit propagation algorithms using new type of ephemeris to get user position solution is the scope of this paper..

Keywords: IRNSS, TTFF (Time to First Fix), Broadcast ephemeris

Disclaimer:

This paper does not declare/claim the proposed options are the actual design of new IRNSS service or modifications to existing IRNSS restricted service. The proposed options are part of analysis studies and are adaptable solely by the discretion of program management office of IRNSS, ISRO.

1. Introduction

Indian Regional Navigation Satellite System (IRNSS) envisages establishment of regional navigational satellite system using combination of 3GSOs and 4 IGSOs Satellites [1]. Indian Space Research Organization (ISRO) is realizing IRNSS by deploying IRNSS satellites in pre-defined orbits, till date two IRNSS satellites i.e., IRNSS-1A, 1B and 1C are in respective orbits and functioning successfully. IRNSS will provide basically two types of services: Standard Positioning Service (SPS) and Restricted Service (RS).Both services will be provided at two different frequencies one in the L5 band (1176.45MHz) and the other in S-band (2492.028). IRNSS is expected to provide navigation services over India and a region extending about 1500 km around India. IRNSS consists of three segments namely the space segment, the ground segment and the user segment.

The space segment consists of seven satellites with three satellites in GSO orbit and four satellites in IGSO orbit. The 3 GSOs will be located at 32.5° E, 83° E and 131.5° E and the 4 IGSOs have their equatorial longitude crossings at 55° E and 111.75° E (two in each plane) as shown in Figure 1. IRNSS satellites have two types of payloads, navigation and ranging payload. The navigation payload will transmit the ranging codes being generated onboard along with the navigation data uplink from the ground stations.

1.1 Space Segment

The space segment consists of seven satellites with three satellites in GSO orbit and four satellites in IGSO orbit. The 3 GSOs will be located at 32.5° E, 83° E and 131.5° E and the 4 IGSOs have their equatorial longitude crossings at 55° E and 111.75° E (two in each plane) as shown in Figure 1. IRNSS satellites have two types of payloads, navigation and ranging payload. The navigation payload will transmit the ranging codes being generated onboard along with the navigation data uplink from the ground stations.

1.2 Ground segment

The ground segment is responsible for navigation data generation, spacecraft control and maintenance of IRNSS constellation. It comprises of ISRO navigation center (INC) responsible for estimation of navigation parameters for IRNSS satellites, IRNSS Range and Integrity Monitoring Stations (IRIMS) for one way ranging, IRNSS

CDMA Ranging Stations for two way ranging, Laser ranging stations, IRNSS Spacecraft Control Facility Center for health monitoring, analysis and control of IRNSS satellites, IRNSS Network Timing centre (IRNWT) for precise time generation for IRNSS system and Data communication network.

1.3 User segment

The User segment mainly consists of: Single frequency IRNSS receiver capable of receiving signals at L5 or S band frequency, dual frequency IRNSS receiver capable of receiving both L5 and S band frequencies and receiver compatible to IRNSS and other GNSS signals. Each IRNSS satellite provides SPS and RS signals in L5 and S bands.

2. Modernised IRNSS Broadcast Ephemeris

In any navigation system, user receiver needs a minimum of four satellites signals in field of view to compute its position solution. User receiver acquires satellite signals in field of view to compute range and to decode navigation data of satellite. This navigation data is useful to compute satellite position at every instant from the time of signal acquisition. With the computed satellite position and corresponding range between satellite and receiver, the receiver arrives at its position solution in terms of x, y, z coordinates in Earth-Center-Earth-Fixed (ECEF) coordinate system.

In IRNSS the Master Frame is of 2400 symbols long made of four sub frames [1]. Each sub frame is 600 symbols long. Sub frames 1 and 2 transmit fixed primary navigation parameters valid for 2hours. Sub frames 3 and 4 transmit secondary navigation parameters in the form of messages. The master fra Primary navigation parameters include Satellite Ephemeris, Satellite clock correction parameters, Satellite & signal health status, User Range Accuracy and Total group delay. Existing IRNSS Satellite Ephemeris includes 15 parameter [1] listed as:

$\sqrt{A}$	Square root of Semi major axis of the orbit
e	Eccentricity
i_0	Inclination of the orbital plane
Ω_0	Long of Ascending Node
ω	Argument of perigee
M_0	Mean Anomaly
$\dot{\Omega}$	Rate of RAAN
Δn,	Mean Motion Difference
IDOT	Rate of Inclination angle
Cuc & Cus	Amplitude of Cosine and sine Harmonic Correction Terms to the Argument of Latitude
Cic & Cis	Amplitude of the Cosine and sine Harmonic Correction Terms to the Angle of Inclination
Crc & Crs	Amplitude of Cosine and sine Harmonic Correction Term to the Orbit Radius

me structure is shown in Figure 2.

This satellite ephemeris along with other primary navigation parameters requires two sub–frames to broadcast, which results in increase of Time to First Fix value (TTFF) in seconds of IRNSS receiver. To minimise TTFF value, an alternate satellite ephemeris representation is proposed to accommodate all primary navigation parameters in one sub-frame. The proposed satellite ephemeris is position, velocity and acceleration of satellite in Cartesian coordinate system$(x, y, z, \dot{x}, \dot{y}, \dot{z}, \ddot{x}, \ddot{y}, \ddot{z})$. The representation of broadcast ephemeris in Cartesian coordinate system was adopted from Glonass navigation satellite system of Russia, where satellite position, velocity and luni-sola accelerations components in x y z directions are broadcast parameters [2]. Direct adoption of above ephemeris and user receiver orbit propagator algorithm from Glonass has not given satisfactory results for IRNSS with one hour validity period, After a series of orbit accuracy studies, two orbit propagation models of user receiver algorithm developed for IRNSS with acceptable accuracy and validity period. All the three, satellite position computation models of user receiver are briefed as options 1, 2 and 3 in this paper.

In Option1 (Glonass model), ephemeris time, position, velocity and luni-solar acceleration of satellite in Cartesian coordinate system $(t_{oe}, x, y, z, \dot{x}, \dot{y}, \dot{z}, \ddot{x}_r, \ddot{y}_r, \ddot{z}_r)$ are broadcast as satellite ephemeris. Orbit propagation model in section IV is used to compute satellites position at every instant. This model holds for 30minutes of orbit propagation, the orbit error rise will be more beyond 30minutes of propagation in this model.
In oprtion2, only ephemeris time, position and velocity of satellite in Cartesian coordinate system $(t_{oe}, x, y, z, \dot{x}, \dot{y}, \dot{z})$ are broadcast as satellite ephemeris. In this model, the user receiver algorithm computes luni –solar acceleration of satellites at every instant and use in orbit propagation to compute satellite position. In this model 7 parameters will get broadcast as satellite ephemeris and can valid up to 1 hour with acceptable satellite position error.

In oprtion3, ephemeris time, position, velocity and residual acceleration of satellite in Cartesian coordinate system $(t_{oe}, x, y, z, \dot{x}, \dot{y}, \dot{z}, \ddot{x}_r, \ddot{y}_r, \ddot{z}_r)$ are broadcast as satellite ephemeris. In this model, the user receiver algorithm computes luni –solar acceleration of satellite at every instant and use in Orbit propagation along with residual accelerations from broadcast ephemeris to compute satellite position. In this model 10 parameters will get broadcast as satellite ephemeris and can valid up to 1 hour with acceptable satellite position error.

The rest of the paper is organized as follows. In section III, the full force orbit propagator algorithm for generation of modernized satellite ephemeris is presented. The user receiver propagator algorithms for three options 1,2 and 3 are briefed in section IV, validation of user receiver algorithm in section V, results and conclusions followed in Section VI and VII. The theoretical approach to full force model to which user algorithm is designed to closely represent for short duration is also provided.

3. Generation of Modernised Ephemeris

IRNSS ground segment comprises of 17 IRIMS stations, each station continuously tracks all IRNSS satellites in field of view to generate one–way range measurements. Range data of all IRNSS satellites from all IRIMS stations flow to IRNSS navigation software to process the data, estimate each satellite orbit and clock parameters at regular interval and to propagate the estimated parameters to generate navigation parameters. The proposed satellite ephemeris $(t_{oe}, x, y, z, \dot{x}, \dot{y}, \dot{z}, \ddot{x}_r, \ddot{y}_r, \ddot{z}_r)$ will be generated from the propagated orbits of each satellite.

IRNSS full force orbit propagator predicts the IRNSS satellite position, velocity, accelerations and the state transition matrix using numerical integration techniques. The equations of motion for a satellite moving under the attraction of a point mass planet without any other perturbations [3] is given in Equation-1

$$\frac{d^2r}{dt^2} = -\frac{GM}{r^3}\vec{r} \qquad (1)$$

Where

$\vec{r}$	=	Position vector from the center of the earth to the satellite
r	=	Distance between Earth center and satellite
G	=	Universal gravitational constant.
M	=	Mass of the Earth.

The two body orbit motion can be expressed by the conic solutions in closed form. The equations of two body motion and its solutions were derived through Newton's law of gravitation and Kepler's law of orbit motion under the assumption of point mass or mass with spherically symmetrical distribution.

Due to the presence of various perturbing forces the equations of motion can be used only as an approximation of the actual motion. Those perturbing forces include earth gravity harmonics (deviation from a perfect sphere), the lunisolar gravitational attractions, solar radiation pressure, and earth tides. The general form of equations of motion including perturbations can be expressed as in Equation 2.

$$\frac{d^2r}{dt^2} = -\frac{GM}{r^3}\vec{r} + a_p \qquad (2)$$

Here a_p is the resultant vector of all perturbing accelerations, and is given by,

$$a_p = a_{asp} + a_{sun} + a_{moon} + a_{srp} + a_{tide}$$

Where, $a_{asp}$is acceleration due to Earth's non-sphericity, a_{sun} is acceleration due to Sun's attraction, a_{moon} is acceleration due to Moon's attraction, $a_{srp}$is acceleration due to solar radiation pressure and a_{tide} is acceleration due to earth tides

3.1 Non –spherical Earth Effect

The acceleration on the satellite due to perturbed force field of non- spherical Earth can be computed by

$$a_{asp} = \frac{\vec{f}_{non-spherical\ Earth}}{m_{satellite}} = \frac{\partial U_{non-spherical}}{\partial u}$$

Where $U_{non-spherical}$ is, the perturbed potential function of the non-spherical earth in spherical coordinates [3][4][5][6], which need to transform into Cartesian ECI frame (u) before including to Eq. (2)

$$U_{non-spherical} = \frac{GM}{r}\left[\sum_{n=2}^{\infty}\sum_{m=0}^{n}\left(\frac{a_e}{r}\right)^n P_m^n\ (\sin\varphi)(\ C_m^n \cos m\lambda + S_m^n \sin m\lambda)\right]$$

r, φ, λ are spherical coordinates of satellite with respect to Earth ECEF frame, ae is the radius of the earth in equatorial plane, n, m are degree and order of the spherical harmonic expansion, P_m^n $(\sin\varphi)$ represents associated Legendre function,C_m^n , S_m^n are Spherical harmonic coefficients [3].

3.2 Lunar- Solar gravitational Effect

The acceleration on the satellite due to solar–lunar gravitational effect can be computed by

$$a_{sun} + a_{moon} = \frac{\vec{f}_{sun,moon}}{m_{satellite}}$$

Where the perturbed force field due to solar-lunar gravitational effect [2] [3] can be computed by

$$\vec{f}_{sun,moon} = -m_{satellite} \sum_i GM_i \left[\frac{\vec{r}-\vec{r}_{Mi}}{|\vec{r}-\vec{r}_{Mi}|^3} + \frac{\vec{r}_{Mi}}{|\vec{r}_{Mi}|^3}\right]$$

Where i is index of the sun (i = 1) and the moon (i =2), $\vec{r}$ and $\vec{r}_{Mi}$ are the geo – centric vectors of satellite and celestial bodies (sun and moon).

3.3 Solar Radiation Effect

The acceleration on the satellite due to solar radiation effect can be computed by

$$a_{srp} = \frac{\vec{f}_{solar}}{m_{satellite}}$$

Where the perturbed force field due to solar radiation pressure [3] [4] can be computed by

$$\vec{f}_{solar} = m_{satellite} \gamma P_s C_r r_{sun}^2 \frac{S}{m_{satellite}} \frac{\vec{r}-\vec{r}_{sun}}{|\vec{r}-\vec{r}_{sun}|^3}$$

Where γ is the shadow factor, P_s is the luminosity of the sun (4.5605X10-6 Newton /meter2), C_r is the surface reflectivity index of the satellite, $\vec{r}$ and $\vec{r}_{sun}$ are the geocentric vector of the satellite and the Sun, $\frac{S}{m_{satellite}}$ is the surface area to mass ratio of the satellite.

3.4 Earth Tidal Effect

The acceleration on the satellite due to Earth Tidal effect can be computed by

$$a_{tide} = \frac{\vec{f}_{tidal}}{m_{satellite}} = \frac{\partial U_{tidal}}{\partial u}$$

Where U_{tidal} is, the potential function of the earth tidal effect in spherical coordinates [3] [4], which need to transform into Cartesian ECI frame (u) before including into Equation

$$U_{tidal} = \sum_{j=1}^{2} \mu_j \sum_{n=2}^{\infty} k_n \left(\frac{a_e}{r}\right)^{n+1} \frac{{a_e}^n}{r_j^{n+1}} P_n(\cos z_j)$$

Here j is the index of the Moon (j = 1) and the Sun (j = 2), μ_j is the gravitational constant of body j, a_e is the geocentric distance of the Earth's surface, r_j is the geocentric distance of the body j, Pn(x) is Legendre function, z_j is the zenith distance of the body j, k_n is the Love number.

After computation of net acceleration of satellite due to all perturbation effects in (Earth Centred Inertial Frame) ECI, there are two general approaches to solve the equations of motion with perturbations in Eq. (2). One approach is through step-by-step numerical integration and other approach is through analytical expansion and integration of the equation of variations of orbit parameters. In this method the former approach is followed.

This model can be described as having two parts; first part is the modelling part which computes above mentioned perturbing accelerations and its partials acting upon a satellite apart from central body itself. A well detailed models for above orbit perturbation is available in standard text books [3][4][5][6].

While second part will integrate these acceleration and partials of acceleration equations computed by the first part using Cowell's method [4][7]. The Integration is performed in two stages. In First stage, Runge-Kutta fourth order method is used to give the initial values to start the integration and the second stage performs the actual integration process with above mentioned initial values by 12th order Adam-Bash forth predictor-corrector method.

This model propagates the pre –estimated state vector of each IRNSS satellite in the form of position, velocity and acceleration on each satellite to generate proposed satellite ephemeris i.e., satellite position, velocity and residual acceleration components of satellite in Cartesian coordinate system $(t_{oe}, x, y, z, \dot{x}, \dot{y}, \dot{z}, \ddot{x}_r, \ddot{y}_r, \ddot{z}_r)$ at an regular intervals. The satellite ephemeris corresponding to t_{oe} will be broadcast 15 minutes ahead of t_{oe} time to make the user receiver to propagate backward from the t_{oe}. The acceleration parameter broadcast as part of the ephemeris represents the un-modelled perturbation forces in the user algorithm as in section IV.

4. IRNSS User Receiver Orbit Propagation models

4.1 Option 1 orbit propagation model

The orbit of a satellite around the Earth is governed by the forces acting on that satellite. The primary force is due to the Earth's gravity field, the potential of Earth's gravity field can be written as Equation (3)

$$U = \frac{GM}{r} + U_{non-spherical} \quad (3)$$

U	=	Earth gravitational potential.
G	=	Universal gravitational constant.
M	=	Mass of the Earth.
r	=	Distance from the earth's geo-center to the center of mass of the satellite
$U_{non-spherical}$ Or R	=	perturbing function due to non-spherical part of earth gravitational potential

The perturbing or disturbing function due to non-spherical part R of gravitational potential in spherical harmonics system [3][4][7] is given in Equation 4

$$R = \frac{GM}{r}\sum_{n=2}^{\infty}\sum_{m=0}^{n}\left(\frac{a_e}{r}\right)^n P_m^n(\sin\varphi)(C_m^n \cos m\lambda + S_m^n \sin m\lambda) \quad (4)$$

Where

a_e	=	Earth's equatorial radius.
r, φ, λ	=	Earth-fixed polar coordinates (geocentric radius, latitude and longitude).
n, m	=	Degree and order of the spherical harmonic expansion.
P_m^n	=	Associated Legendre functions.
C_m^n, S_m^n	=	Spherical harmonic coefficients.

Since Earth's gravitational potential is in first approximation rotationally symmetric, i.e. independent of λ, the zonal harmonics (m = 0 in Eq. (4)), which cause parts of the gravitational potential independent of λ, are much more significant to satellite motion than the tesseral (0 < m < n) and the sectorial (m = n) harmonics. It can therefore be assumed that the influence of tesseral and sectorial harmonics is insignificant over a short time span of orbit integration.

The other perturbing forces act on a satellite, such as the gravitational effect of the sun, moon and planets, and solar radiation pressure are small, although their effects do cumulate with time. The effects of lunar and solar gravity are the most dominant of these additional forces and can be great as 20% of the J2 term. With the exception of the J2 term, the perturbing forces are neglected in the model, with the lunar-solar effect being assumed to be constant. J2 is primarily representative of the earth's flattening and its value is about 400 times larger than next-largest value J3 (J2 = 1082.6300 x 10-6 and J3 = -2.5321531 x 10-6) [3]. Therefore, neglecting zonal terms greater than J2 and the total perturbing or disturbing function due to zonal harmonics [3] is given in Equation 5

$$R = -\frac{GMa_e^2}{r^3} J_2\left(\frac{3}{2}\frac{z^2}{r^2} - \frac{1}{2}\right) \quad (5)$$

Since z = rcosφ, Eq. (5) can be rearranged to Equation 6

$$R = -\frac{GMa_e^2}{r^3} J_2\left(\frac{3}{2}\cos^2\varphi - \frac{1}{2}\right) \quad (6)$$

The satellite acceleration in Cartesian coordinate's $\ddot{R}$ due to gravitational potential is given in Equation 7

$$\ddot{R} = \nabla V \quad (7)$$

And the individual Cartesian components of the acceleration vector $(\ddot{x}, \ddot{y}, \ddot{z})$ are computed as in Equation 8

$$\ddot{R}i = \frac{dV}{dRi} = \frac{\partial V}{\partial r}\frac{\partial r}{\partial Ri} + \frac{\partial V}{\partial \lambda}\frac{\partial \lambda}{\partial Ri} + \frac{\partial V}{\partial \varphi}\frac{\partial \varphi}{\partial Ri} \quad (8)$$

Where Ri, any one of the component of the satellite is position vector (x, y, z) and $\ddot{R}i$ are the corresponding acceleration components. r, φ, λ are as defined in Eq. (4)

$$r = (x^2 + y^2 + z^2)^{1/2}$$

As Eq. (6) is dependent on only r and φ, $\frac{\partial V}{\partial \lambda}$ is zero and the middle term in Eq. (8) disappears. The latitudes component, the last term, in equation (8) is smaller than the radial term and is neglected in model. Substituting Eq. (6) into Eq. (3) and evaluating the terms from Eq. (8) gives the following Equations 9a, 9b, 9c.

$$\frac{dV_x}{dt} = -\frac{GM}{r^3}x + \frac{3}{2}C_{20}\frac{GMa_e^2}{r^5}x\left(1 - \frac{5z^2}{r^2}\right) + \ddot{x}_{LS} \tag{9a}$$

$$\frac{dV_y}{dt} = -\frac{GM}{r^3}y + \frac{3}{2}C_{20}\frac{GMa_e^2}{r^5}y\left(1 - \frac{5z^2}{r^2}\right) + \ddot{y}_{LS} \tag{9b}$$

$$\frac{dVz}{dt} = -\frac{GM}{r^3}z + \frac{3}{2}C_{20}\frac{GMa_e^2}{r^5}z\left(3 - \frac{5z^2}{r^2}\right) + \ddot{Z}_{LS} \tag{9c}$$

Where $(\ddot{x}_{LS}, \ddot{y}_{LS}, \ddot{z}_{LS})$ represent the estimated Luni –solar accelerations of the satellite and the equality C20 = -J2 exists [3][4]. This orbital force model neglects minor forces such as non-zonal components of the Earth's gravity field and solar radiation pressure. This model is recommended for only a time period of up to 30 minutes either side of the broadcast ephemeris reference timet_{oe}. However, the integration time period can be extended beyond predefined 30mints with a relatively small degradation in satellite coordinate quality.

Eq. (9a, b, c) are valid only in an inertial system, since Newton's laws of motion are only valid in inertial systems. To determine the satellite coordinates in ECEF, one could integrate the satellite's equations of motion in the inertial system and then transform the obtained coordinates to the ECEF frame.

4.2 Option 2 orbit propagation model

In Option 2 model, Luni – Solar acceleration of satellites $(\ddot{x}_{LS}, \ddot{y}_{LS}, \ddot{z}_{LS})$ do not broadcast in satellite ephemeris. This model computes luni –solar accelerations on satellites at every instant using mathematical model of section III and use in equations (9a, 9b, 9c) of option 1 user receiver orbit propagation model. The sun and moon position for luni –solar mathematical model can be computed using Bretagnon and J.L.Simon Model [8] and Browns theory formula [7] respectively.

4.3 Option 3 orbit propagation model

This model computes acceleration on satellites due to sun and moon $(\ddot{x}_{LS}, \ddot{y}_{LS}, \ddot{z}_{LS})$ as like in Option 2 and sum up with residual accelerations $(\ddot{x}_r, \ddot{y}_r, \ddot{z}_r)$ of broadcast ephemeris to use in equations (9a, 9b, 9c) of orbit propagation model of option 1 user receiver algorithm.

5. Validation of orbit accuracy with new ephemeris parameters

The proposed satellite ephemeris such as satellite position, velocity and acceleration components on satellite in Cartesian coordinate system such as $(x, y, z, \dot{x}, \dot{y}, \dot{z}, \ddot{x}, \ddot{y}, \ddot{z})$ are taken from IRNSS satellites full force orbit model output at 15th min and 45th minute of every hour in ECI frame as input to the user receiver algorithm for 30minutes validity period. For one hour validity period algorithms satellite ephemeris at every 30minutes is taken.

IRNSS satellite's position and velocity components in X Y Z directions are computed through user receiver algorithm at every instance (every second) in ECI frame of coordinate system, i.e., forward and backward propagation for half of validity period ahead and before the time of ephemerist_{oe}. The user receiver propagated orbit was compared with full force model orbit of IRNSS satellites. The results of option 2 and option3 models for a validity period of up to 1 hour indicates that satellite orbit position accuracy is achieved at meter level and satellite velocity accuracy at centi-meter/sec level which are satisfactory.

6. Results

User algorithm computed IRNSS satellites orbit accuracy with respect to full force orbit in the form of satellites position error RSS for 30minutes and one hour validity periods are shown in Figure 3 and 4 respectively. The exercise has been carried out using simulated data for all IRNSS satellites except IRNSS 1A, 1B and IC, for the case of IRNSS 1A, 1B and 1C real measurements were considered. The position error RSS of 30 minutes validity for option 1, 2 and 3 are lesser than 0.15meters, the accuracy of option 2 and 3 are better than option1. For one hour validity period also, option 2 and 3 models orbit error is comparatively lesser than option1 as shown in Figure 4.

7. Summary

The motivation and generation of modernized satellite ephemeris are discussed and the derivation of user receiver algorithm to propagate satellite orbit for different models with modernized satellite ephemeris covered. The results presented show the accuracy of modernized satellite ephemeris and its user algorithm for different models which are satisfactory and prime focus of the paper. TTFF of user receiver which was motivation for modernized satellite ephemeris will be improved with proposed satellite ephemeris which makes primary navigation parameters to fit in one sub-frame, which was also depend on signal broadcast rate and methodology. The modernised ephemeris calls for augmentation of the on-board memory of the existing navigation payload as number of broadcast sets required per day is increased due to the lesser validity period of the data.

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M.V. Chandrasekhar did Bachelor of Technology in Aeronautical engineering from Jawaharlal Nehru technological university, Hyderabad, India. He is working as Engineer in Advances Navigation studies section of space navigation group at ISRO Satellite centre since 2011. The author primarily involved in development of IRNSS measurement simulation software, software architecture design and development of IRNSS Navigation software. He was born in Madhira, Andhra Pradesh in 1987.

D.Rajarajan received his B.E in Aeronautical engineering from Anna University, Chennai,India. He is currently working as Engineer in space navigation group at the ISRO satellite centre since 2010. He has developed numerous software elements for ground segment of IRNSS. He is borrn in thanjavur, tamil nadu, in the year 1986. He held the post of design engineer with Taneja Aerospace and aviation limited before joing ISRO.

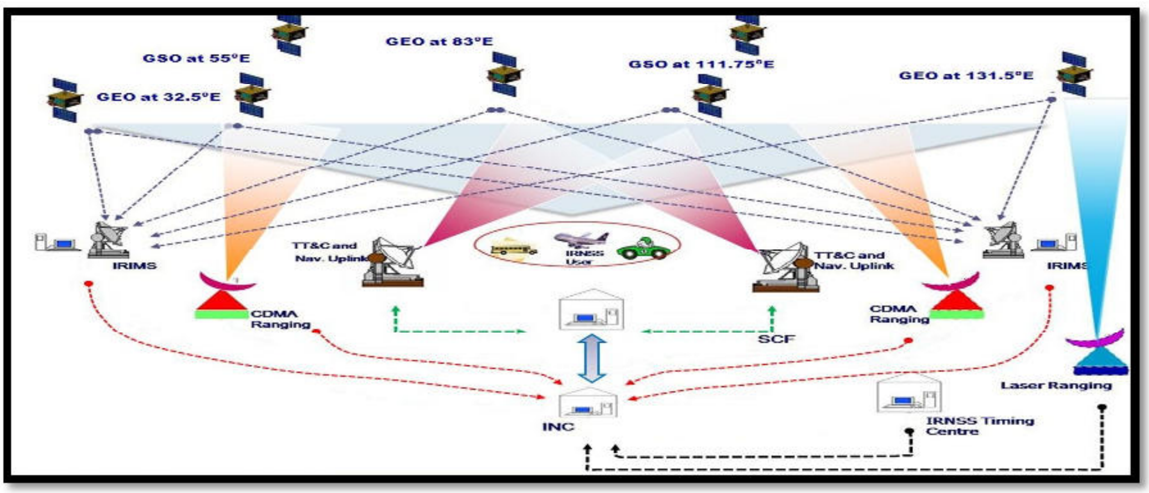

Figure 1: IRNSS system architecture

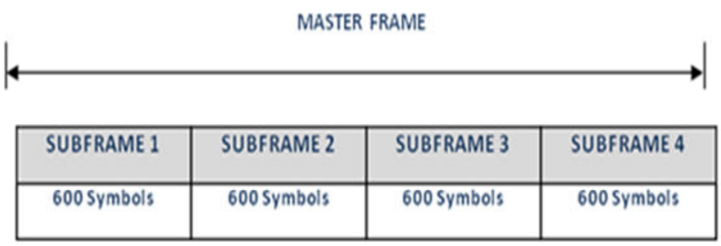

Figure 2: IRNSS master frame structure

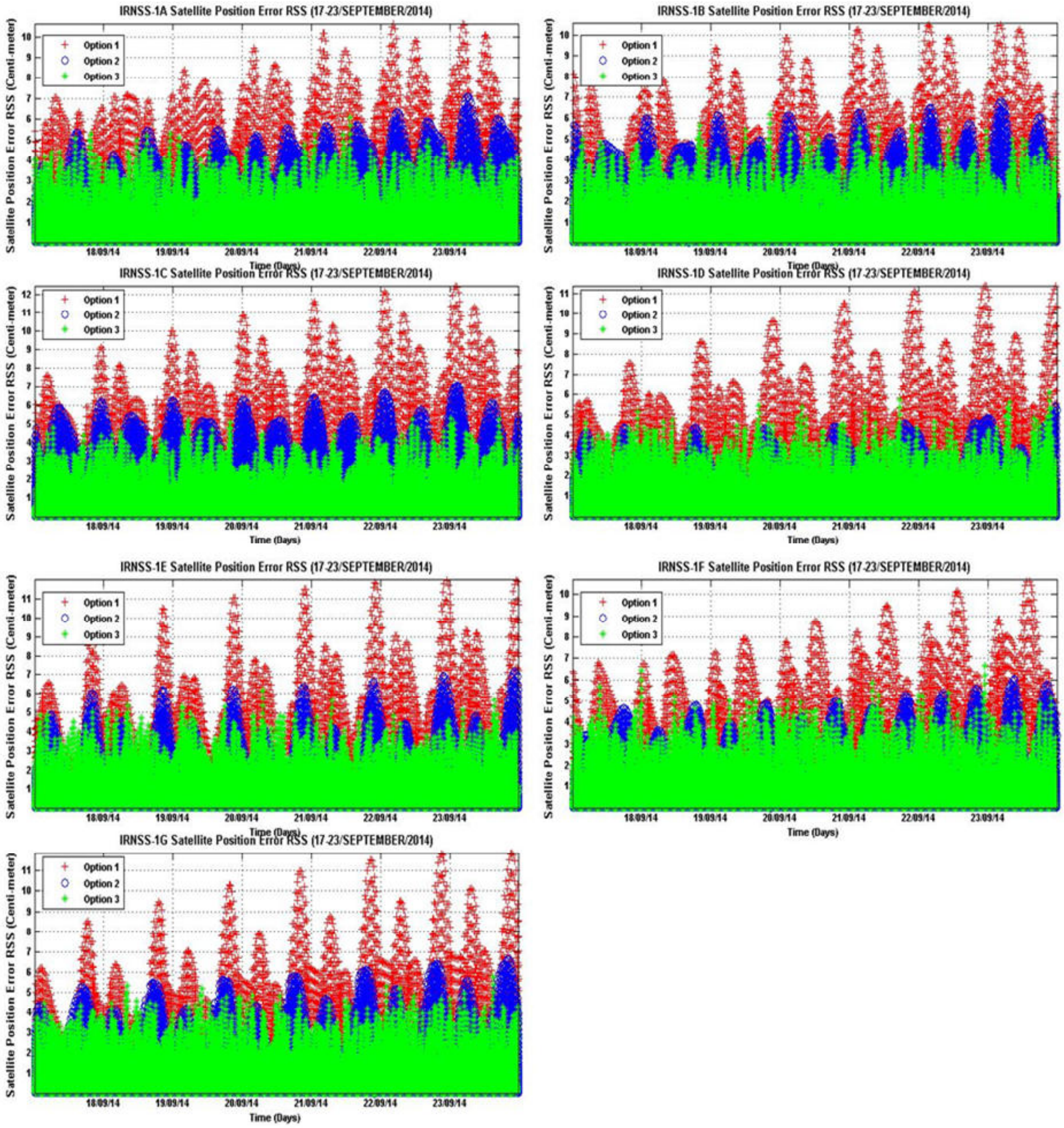

Figure 3: IRNSS satellites Position error RSS using models of option 1, 2 and 3 for 17-23 September 2014 data with 30minutes validity Period

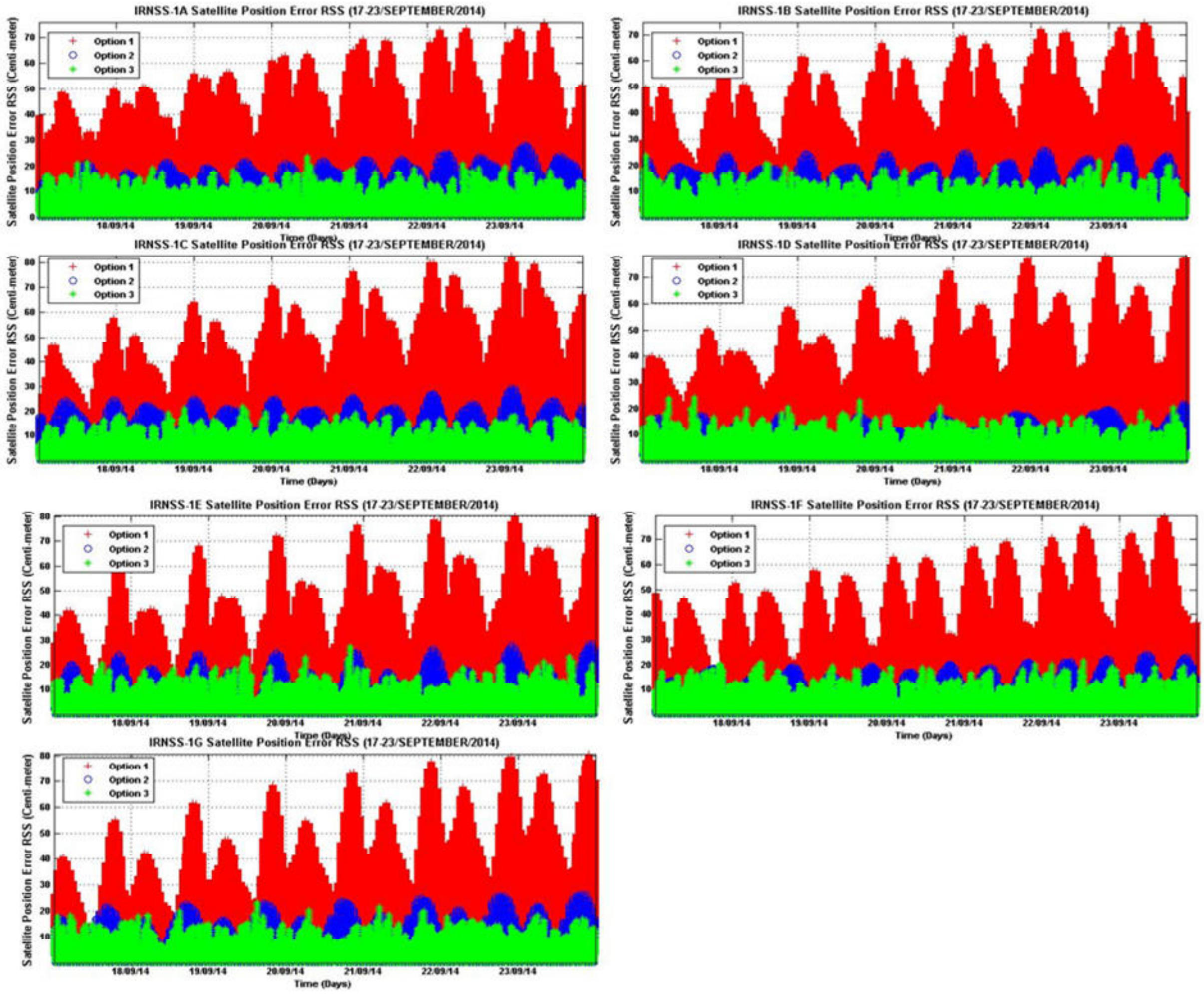

Figure 4: IRNSS satellites Position error RSS using models of option 1, 2 and 3 for 17-23 September 2014 data with one hour validity Period