PRECISE ORBIT COMPUTATION OF INDIA'S FIRST LUNAR MISSION CHANDRAYAAN-1 USING ACCELEROMETER AND TRACKING DATA DURING EARLY PHASE

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ABSTRACT

India's first Moon mission Chandrayaan-1 carrying eleven scientific instruments for the purpose of expanding scientific knowledge about the Moon was launched on 22nd October 2008 from Satish Dhawan Space Centre, Sriharikota, India by India's Polar Satellite Launch Vehicle PSLV-C11. Spacecraft was injected into transfer orbit of (254.4 X 22932.7) km with inclination of 17.9 deg at 2008-10-22-01-10-19-081 UT. The main objective of the mission is a simultaneous chemical, mineralogical and photo geologic mapping of the whole Moon with high spatial resolution using high resolution state of the art sensors. The spacecraft was put into Moon's polar, circular orbit of about (100 X100) km on 12th November 2008 by carrying out sequence of Earth and lunar bound maneuvers. The thrust cut off/burn duration of Chandrayaan-1 mission was controlled by accelerometers autonomously by monitoring the accelerometer data during the burn. Precise orbit determinations were carried out during each phase of the mission using tracking data collected from network of tracking stations configured for the mission. This paper describes Chandrayaan-1 Orbit Determination System (ODS) and its performance during initial phase of the mission using accelerometer and tracking data.

1. INTRODUCTION

India's first Moon mission Chandrayaan-1 carrying eleven scientific instruments for the purpose of expanding scientific knowledge about the Moon was launched on 22nd October 2008 from Satish Dhawan Space Centre, Sriharikota, India by India's Polar Satellite Launch Vehicle PSLV-C11. Spacecraft was injected into transfer orbit of (254.4 X 22932.7) km with inclination of 17.9 deg at 2008-10-22-01-10-19-081 UT. The main objective of the mission is a simultaneous chemical, mineralogical and photo geologic mapping of the whole Moon with high spatial resolution using high resolution state of the art sensors. The spacecraft was put into Moon's polar, circular orbit of about (100 X100) km on 12th November 2008 by carrying out sequence of five Earth bound maneuvers, one trajectory correction maneuver and four Lunar bound maneuvers. The thrust cut off/burn duration of Chandrayaan-1 mission was controlled by accelerometers autonomously by monitoring the accelerometer data during the burn. Precise orbit determinations were carried out during each phase of the mission using tracking data collected from network of tracking stations configured for the mission.

The orbit of the satellite had to be determined continuously at a brisk pace to a good degree of accuracy to meet the requirements of the mission operations. Tracking data was collected from NASA's DSN (Deep Space Network) as well as ISRO's DSN and non-DSN tracking stations. NASA's DSN stations namely Goldstone, Canberra, Madrid supported the mission during initial phase of the mission.

An accelerometer is an instrument that measures the acceleration of the case of the sensor due to external forces. Early accelerometers produced output that was directly related to acceleration; but modern sensors integrate the internally measured signal, to reduce noise, and the output is proportional to the change in velocity over the integration time. Chandrayaan-1 mission carries four advanced accelerometers for burn calibration. Software named as PROCAD (**Pr**ecise **O**rbit **C**omputation using **A**ccelerometer **D**ata) was developed to predict the satellite orbit in near real-time using onboard accelerometer measurements. PROCAD was made operational during all phases of Chandrayaan-1 by making use of the measurements of the accumulated velocity from the accelerometer gathered from telemetry. Chandrayaan-1 orbit was determined in real time even as the orbit maneuver was in progress. PROCAD gave the achieved orbit of Chandrayaan-1 immediately after the maneuver ended.

Comparison of the PROCAD orbit results with those determined with tracking data of range and accumulated Doppler measurements showed close association between the state vectors during all phases of Chandrayaan-1 missions. PROCAD showed a maximum position difference of 0.32 %. The PROCAD program used in the mission operations center for the Chandrayaan-1 mission is an effective validation tool for an onboard accelerometer.

Robust Orbit determination strategy, characterized by accurate solutions and fast-turnaround to minimize fuel penalties, resulting from delays in implementation of required maneuvers will play an important factor that contribute to maneuver targeting accuracy and low propellant consumption for the mission. Indian Space Research Organization's (ISRO) operational Orbit Determination Program [1] used for low earth missions was suitably updated and validated with simulated as well as live tracking data of Lunar Prospector mission before the launch of Chandrayaan-1 mission to cater the need aspect of meeting Lunar Mission. Software validation especially trajectory generation and achievable orbit determination accuracies during different phases of the mission are studied in detail [2].

This paper presents the methodology as well as the performance of Chandrayaan-1 orbit determination system using S-band tracking data as well as precise orbit computation using accelerometer data in near-real time during all initial phase orbit maneuvers.

During initial phase of the mission, orbit solutions were carried out externally by JPL/NASA also. This paper includes comparison of OD results with that of external OD results during all phases of the mission. At each phase of the mission, the orbit was determined using tracking data obtained over varying periods. The orbit solutions obtained from short arc OD's are compared with those obtained using the longest arc OD of each stage of the initial phase of the mission. The achieved OD results along with consistency during each phase of the mission are demonstrated. ODS performance study from launch to beginning of Moon's mapping phase (17th November 2008) is described in detail. Subsequent sections give brief description of orbit determination system, PSLV-C11 vehicle performance, orbit solutions in each phase of the mission, OD consistency study and OD results comparison with external agency results and conclusions. Quick, accurate and consistent orbit determination solutions obtained during initial phase of the mission resulted in success of Chandrayaan-1 mission.

2. PHASES OF THE MISSION

The spacecraft was put into Moon's polar, circular orbit of about (102 X102) km on 12th November 2008 by carrying out sequence of five Earth bound maneuvers (EBNs), one trajectory correction (TCM) and four Lunar bound maneuvers (LBNs). The launch vehicle PSLV-C11 put the spacecraft into a geo transfer orbit on 22nd October 2008 at 2008-10-22-01-10-19-081 UT from where it was moved to an extended transfer orbits and then to a lunar transfer trajectory. Finally the spacecraft was placed in the lunar orbit (101.904 X 102.855) km on 12th November 2008 at 2008-11-12-13-04-52-797

UT. Following Table-1 gives the phases of Chandrayaan-1 mission for which orbit determination system analysis was carried out.

Table 1 Phases of Chandrayaan-1

Phase	Date	Orbit (km)
Launch	2008-10-22	(254.4 X22932.7)
EBN#1	2008-10-23	(299.2 X 37908.1)
EBN#2	2008-10-25	(336.6 X 74715.9)
EBN#3	2008-10-26	(347.9 X 165015.7)
EBN#4	2008-10-29	(459.6 X 266613.0)
EBN#5/LTT	2008-11-03	(972.8 X 379860.2)
TCM#1/LTT	2008-11-05	(815.1 X 378515.4)
LOI	2008-11-08	(507.9 X 7510.1)
LBN#1	2008-11-09	(200.6 X 7502.4)
LBN#2	2008-11-10	(183.0 X 255.2)
LBN#3	2008-11-11	(101.7 X 255.2)
LBN#4	2008-11-12	(101.9 X 102.8)

LTT : Lunar Transfer Trajectory; LOI: Lunar Orbit Insertion

3. ORBIT DETERMINATION USING ACCELEROMETER DATA

The maneuvers carried out during the initial phase of the Chandrayaan-1 mission were governed by the onboard accelerometer. The thrust stoppage during every maneuver was based on the accelerometer cut off. The accelerometer package consisted of 4 advanced accelerometers in tetrad configuration [3]. Data processing software in the package carried out velocity channel error compensation, velocity channel failure detection and isolation, and reconfiguration of sensor channel. The data processing software is part of the AOCS Bus Management Unit (BMU).

The data processing software does important functions. The minor cycle count (64 ms) is incremented for time reference. The major cycle count is 512 milliseconds. The sensor errors are compensated. The failure events are detected and isolated. The velocity increments along the satellite body axes are estimated. The estimated velocity increments are accumulated and these measurements are stored and transmitted to the ground stations.

The sensor error is compensated for scale factor, bias, and misalignments. Bias and scale factor are varying with temperature and the software has the capability to effect temperature compensation. Whenever there is a significant change in temperature a new scale factor and bias is computed at low periodicity. The compensation using the updated scale factor and bias is done at high periodicity. The bias in the accelerometer measurements is corrected for the day-to-day varying temperature effects. The operating temperature varies from 25^0 to 60^0 C. The temperature compensation is done by using a piecewise linear model.

3.1 Precise Orbit Computation using Accelerometer Data (PROCAD)

'PRecise Orbit Computation using Accelerometer Data' (PROCAD)' is an orbit determination program that was specifically developed for the Chandrayaan-1 mission. PROCAD makes use of the accumulated velocity measurements to determine the spacecraft orbit.
An initial state vector of the satellite in the inertial frame at a given epoch, Accumulated velocity of the satellite in the satellite body frame at equally distributed intervals of time as measured by the

accelerometer onboard and Quaternions at each measurement time for converting the velocity components from Body to Inertial frame are input to this program. The inertial state vector of the satellite at equally distributed intervals of trajectory integration time starting from the epoch is the output of this program.

The accumulated velocity in the satellite body frame is received as the measurement at equal time intervals. The spacecraft attitude is obtained from telemetry in the form of quaternions.

$$q \quad = \quad [\ q_0 \ q_1 \ q_2 \ q_3\ ] \tag{1}$$

The incremental velocity vector in body frame, $\mathbf{V_b}$, along the roll, pitch, and yaw axes at each time instant is obtained by differencing the accumulated velocities between consecutive measurement times:

$$V_b \quad = \quad [\ \Delta V_r \quad \Delta V_p \quad \Delta V_y\ ] \tag{2}$$

The incremental velocity vector in inertial frame, $\mathbf{V_i}$, is represented as:

$$V_i \quad = \quad [\ \Delta V_x \quad \Delta V_y \quad \Delta V_z\ ] \tag{3}$$

V_i is computed using the incremental velocity in body frame and the quaternions. The velocities in the body frame are transformed into velocities in inertial frame by using quaternion rotation operators.

The quaternion rotation operator L_q, associated with the quaternion q and applied to the velocity vector in the body frame V_b is defined as:

$$L_q\,(V_b) = q\, V_b\, q^*, \tag{4}$$

where, q^* is the complex conjugate of q.

$$V_i \quad = \quad L_q\,(V_b) \tag{5}$$

The rotation operator represents a rotation in the three-dimensional vector space with the axis of rotation given by the vector part of q, and the angle of rotation given by twice the angle associated with the quaternion q.

The velocity measurements were obtained from spacecraft telemetry at a sampling rate of 1.816 seconds and solid state recorder (SSR) at a sampling rate of 512 milliseconds

The satellite trajectory generation is based on solving the equations of motion through numerical integration. The orbit generator is based on Cowell's method. For integrating the second order differential equations of motion the method based on double integration, 'Gauss Jackson Merson Second Sum' method is used. The differential equation of motion of the orbit model includes the significant perturbing forces. Therefore, the predicted motion of the satellite is as close as possible to the true motion of the satellite.

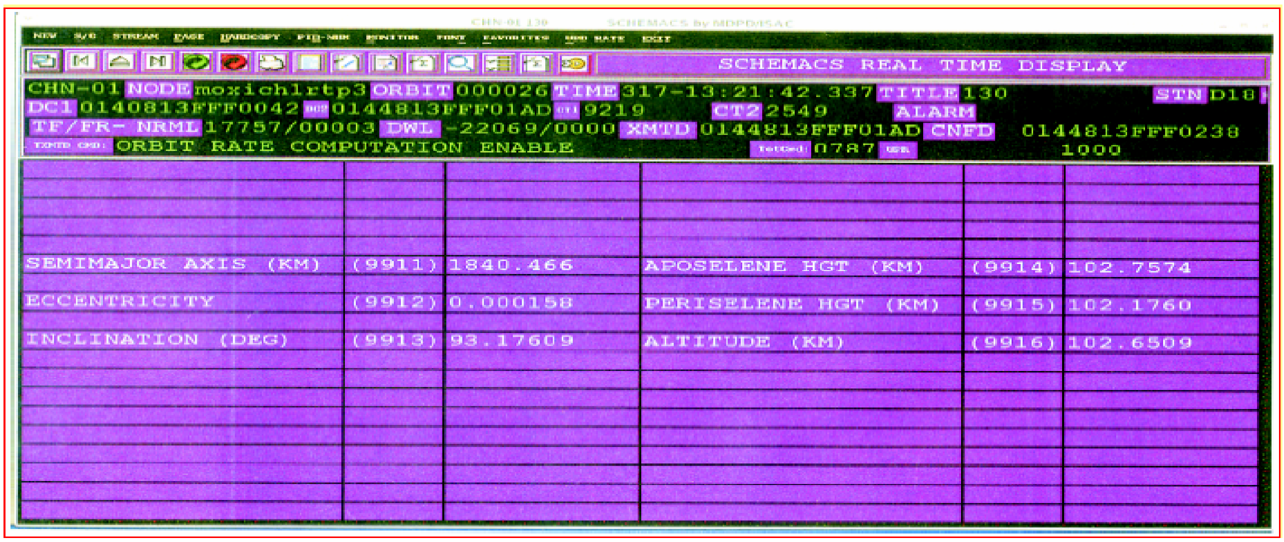
The dominant perturbing forces that affect the motion of the satellite are the central body perturbation (Earth/Moon), aerodynamic drag, third body perturbation (Moon/Earth, Sun and other planets), and solar radiation pressure. Since accelerometer measures all the non-gravitational forces acting on spacecraft, only the gravitational accelerations are considered in addition to the accelerations obtained

from accelerometers to find the satellite's true position. EGM96 and LP100 are the gravity models that were used to calculate the central body perturbations with respect to Earth and Moon in the Earth and Moon orbits of Chandrayaan-1 respectively. DE405 is the JPL planetary ephemeris file that was used to calculate the gravitational forces due to Sun, Moon, and the other planets, for satellite trajectory generation.
PROCAD has a feature to run 'live', for the purpose of real time orbit determination during the progress of a maneuver. Separate algorithms in PROCAD allow access from telemetry the velocities of the spacecraft measured at each time instant by the accelerometer, and the quaternions measured at the corresponding times, and instantaneously integrating them with the initial state vector. The state vector of the spacecraft realized at the end of the burn is instantaneously displayed, also as a telemetry page.

3.2 Real Time Orbit Determination

The PROCAD program was put in loop with the telemetry. Even as the maneuver was progressing and the telemetry was receiving the accumulated velocity measurements from the onboard accelerometer and the quaternions, PROCAD was integrating the instantaneous velocities of Chandrayaan-1 with the initial state vector at the start of the burn. At the very instant the burn ended, the state vector at that instant was shown in the telemetry page. Table 2 gives the telemetry page demonstrated during LBN-4 when the spacecraft was put into Moon's polar, circular orbit of about (100 X100) km on 12th November 2008.

Table 2 Real-Time Orbit Display of Chandrayaan-1 during LBN-4



3.3 PROCAD Results

The accumulated velocity measurements were obtained either from the telemetry or from the SSR during every maneuver between the start and end time of the burn. The corresponding quaternions for each of the measurement times were also obtained from telemetry. The state vector of Chandrayaan-1 at the start of the burn is known. Using these inputs, the PROCAD program gave the state vector at the instant of burn end after numerical integration of the equation of motion.

The consolidated results of the differences between the PROCAD determined state vectors, ODP determined state vectors, and the post burn predicted state vectors, are shown in Table 3.

The state vectors determined by PROCAD at the end of each burn during initial phase of the mission are individually have been compared with the predicted state vectors at the end of the burn and the state vectors at the end of the burn determined by the Chandrayaan-1 operational Orbit Determination Program (ODP) using tracking data from the network of Earth stations configured for the mission.

Table 3. Position and Velocity Differences between CHANDRAYAAN-1 Orbit Solutions
{PROCAD (Vs) post-burn EXPECTED} & {PROCAD (Vs) post-burn TRACKING DATA DETERMINED}

Stage of the Mission	Position Difference				Velocity Difference			
	PROCAD (Vs) Post-Burn EXPECTED		PROCAD (Vs) Post-Burn DETERMINED		PROCAD (Vs) Post-Burn EXPECTED		PROCAD (Vs) Post-Burn DETERMINED	
	(km)	(%)	(km)	(%)	(m/s)	(%)	(m/s)	(%)
EBN-1	1.862	0.021	10.31	0.127	4.166	0.047	18.28	0.207
EBN-3	1.037	0.014	2.085	0.029	2.124	0.020	2.738	0.026
EBN-4	0.135	0.001	23.09	0.326	0.613	0.005	19.10	0.182
EBN-5	0.111	0.001	0.729	0.005	0.734	0.00	1.142	0.014
TCM-1	6.547	0.002	6.432	0.002	0.580	0.055	0.595	0.056
LOI	2.606	0.113	3.727	0.162	2.885	0.155	7.104	0.383
LBN-1	0.023	0.000	1.125	0.012	0.181	0.042	0.487	0.113
LBN-2	2.062	0.107	1.229	0.063	3.340	0.207	3.346	0.208
LBN-3	0.172	0.008	0.957	0.048	0.241	0.015	0.867	0.056
LBN-4	0.014	0.001	0.187	0.010	0.244	0.014	0.708	0.043

4. ORBIT DETERMINATION USING TRACKING DATA

A precise knowledge of orbital parameters is a prerequisite for the determination of the position of satellite at any given time. Mathematically orbit is specified by a set of six orbital parameters which completely describes the motion of the satellite within the specified accuracy over a period of time. Determination of the satellite orbit means the refinement of the initial orbit parameters. The problem of OD consists of comparing measurements taken on a satellite trajectory to a model representing that trajectory. The model is generally represented by a system of differential equations whose constants of integration define the satellite trajectory as a function of time. Thus, given a priori estimate of these constants of integration (state parameters), the problem is to update or correct the initial (a priori) state parameters as a function of measurements. The orbit determination system consists of tracking data system and data handling, dynamic models, computational techniques, which include trajectory generation and estimation technique.

The main computation process of the orbit determination system is trajectory generation, observation modeling and estimation. Trajectory generation is performed through numerical integration of differential equations of motion of satellite. The force model for artificial earth satellites normally includes earth's gravitation, aerodynamic drag, luni-solar gravitation and solar radiation pressure. Cowell's method is used for trajectory generation. Weighted least squares technique and iterative differential correction process is used to obtain the refined state. The method of integration of equations of motion and estimation are described in the following sections.

The main force models used in ISRO's operational orbit determination system for Earth orbit satellites are a) Central Body perturbation - EGM96 geo-potential model (70 X 70), b) Aerodynamic Drag - MSISE 90 model for atmospheric density computation, c) Luni-Solar and other planets gravitation attraction - JPL ephemeris (jplde405 package) and d) Solar radiation pressure computation. For Lunar mission during GTO, Post EBNs, LOI and Lunar Orbit phases (Post LBNs) the force models are selectable, which are explained in the subsequent section.

4.1 Main features of Orbit determination System

The main features of orbit determination system are Tracking Data Processing, Trajectory Generation, Observation modeling and Orbit Estimation. The following subsections describe each of these features.

4.1.1 Tracking Data Processing

Raw tracking data is preprocessed through Tracking Data Preprocessing software (TDPP). In preprocessing, raw tracking data acquired from different stations is edited, smoothed, and corrected for atmospheric refraction effects using weather data information. The data is also corrected for onboard and ground system delays. After selecting the number of measurements required for OD, each type of data is assigned appropriate weight. Later, the processed data is stored in a file, in a suitable format for OD.

4.1.2 Method of integration for Trajectory Generation

The coupled nonlinear second order differential equations of motion are integrated numerically through Cowell's method. The chief advantage of special perturbation technique is its high accuracy. The perturbative acceleration acting on the satellite is modeled. Numerical integration methods are customarily classified into two parts namely single step and multi step methods. Multi step methods normally need a starter. The single step methods that are considered are those of Runge Kutta or RK-Gill etc. Multi step methods are "Adams" or its variants. Exclusively for solving second order equations, double integration method is the recommended procedure. These methods are stable. Gauss-Jackson-Merson's (GJM) 8^{th} order method is employed here [4].

4.1.3 Observation Modeling

Chandrayaan-1 tracking data measurements are range and Doppler. Theoretical measurements for orbit determination are computed from the predicted trajectory information described in section 4.1.2.

Range – Tone Ranging

Ranging is performed by measuring the travel time of a signal from the station to the spacecraft and back, via the onboard transponder. Ranging (tone ranging system) is done by phase modulation (PM) of the carrier signal by a series of tones of different wave lengths. The range measurement contains unknown integral multiples of the longest tone wave length, which must be resolved by ambiguity resolution. During the ranging, the equipment performs measurements for every 100 ms (integration time) and averaged to one single ranging measurement as one sample per second. Corrections must be applied for delay in signal travelling time through the onboard (spacecraft transponder delay), station electronics (ground station delay) and for delay due to the earth's troposphere and ionosphere.

The value of range is obtained from the two way round trip time interval. The interval say ΔT is provided in terms of nanoseconds to a resolution of one nanosecond. The range conversion is given by

Range (km):

$$\rho_i = c\Delta T_i /(2*10^9) \tag{6}$$

where, c is the velocity of light (km/sec), ΔT_i is round trip time interval. The time tag T_i corresponds to ρ_i is calculated from T_0 as $T_i = T_0 + i\delta T$, where T_0 is the 1st measurement tagged time in each record., ∂T is sampling interval.

Doppler/Range Rate

The range rate of spacecraft is determined by measuring the Doppler shift of a signal resulting from the relative motion between the station and s/c. Doppler is computed from accumulated phase data over the sampling interval time. The Pre-processor converts the raw Doppler data into range rate.

The range rate is computed from two-way Doppler is given below.

Range rate (km/sec)

$$\dot{\rho} = -c\Delta f /(2NF_T + \Delta f) \tag{7}$$

where, Δf is Doppler shift (Hz), F_T is transmitted frequency (Hz), c is speed of light (km/sec). N is the ratio of modulation by the transponder, given as 240/221.

4.1.3.1 Light Time Correction

Two-way radar and laser ranging comprises the signal uplink from the ground station to the satellite and the downlink from the satellite to the ground station. When two-way range measurements have been recorded at the ground station at time *t*, the signal has been received and transmitted back by the

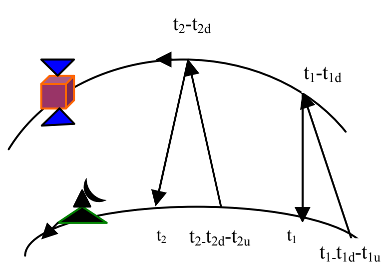


Fig.1 The motion of the satellite and ground station during the signal travel time for two-way Doppler measurement

satellite at $t\text{-}t_d$, where t_d is the downlink light travel time. The transmission time of the signal at the ground station is thus given by $t\text{-}t_d\text{-}t_u$, where t_u is the uplink light travel time. Two-way Doppler measurements are obtained from the integration of Doppler counts over a count time interval. The measurement range rate is modeled as the difference of the two-way ranges at the end t_2 and at the beginning t_1 of the count interval time. Thus four light time iterations are required for the modeling of a single range rate measurement as shown in Fig.1. The modeling of range measurements requires the iterative solution of two light-time equations for the downlink and the uplink path. The description of light time correction computation for range and Doppler/range rate measurements is described in [5].

4.1.4 Estimation

Optimal estimate of satellite state is computed from a given vector of tracking measurements which depends on the instantaneous position and velocity of spacecraft. The estimated state is valid over the period during which tracking data is collected.

The general procedure for all definitive orbit computations is to set up some dynamical model of the orbit and use the observations to improve the orbit parameters of the model by the process of differential correction. Weighted Least Squares estimation process is applied. Observations are considered for some pre-selected time period and by differential correction of parameters of the model, the sum of the squares of the residual is minimized. The basic idea of least-squares estimation as applied to orbit determination is to find trajectory and the model parameters for which the square of the difference between the modeled observations and the actual measurements becomes as small as possible. In this estimation process of "Weighted Least Squares", it is necessary to compute partial derivatives of observations with respect to model parameters. Reference [6] gives description of estimation method adopted in this present study.

4.2 Chandrayaan-1 Orbit Force Model

The force model used for Chandrayaan-1 trajectory generation is described in Table 4.

Table 4 Force Model

Mission Phase	Earth's Gravity	Moon's Gravity	Sun's Gravity	Solar Radiation Pressure	Drag
Post launch / GTO	EGM–96 (50 X 50)	Point Mass	Point Mass	Yes	MSIS-90
Post EBN-1 to EBN-4	EGM–96 (50 X 50)	Point Mass	Point Mass	Yes	MSIS-90
Post EBN-5 to LOI	EGM–96 (50 X 50)	LP 100 (100 X 100)	Point Mass	Yes	MSIS-90
Post LOI to Lunar Orbits	EGM-96 (10 X 10)	LP 100 (100 X 100)	Point Mass	Yes	No

4.3 Tracking System

Physical measurements of the satellite may be from S-band ranging, C-band Tone Ranging, Burst tone ranging, Laser Ranging, Interferometer and Optical tracking systems, etc. The functional aspects of the above systems are beyond the scope of this paper. Chandrayaan-1 tracking system is S-band tone ranging from ISRO/ISTRAC's (ISRO Telemetry Tracking & Command Network) DSN and non-DSN, NASA/JPL (DSN) and APL (Applied PhysicalLaboratory, Maryland, USA) ground stations. The ranging system from ISTRAC is CORTEX. Data available for orbit determination are range and accumulated Doppler along with weather data. Tracking data format from ISTRAC stations is ISRO's 90-byte format. The tracking data from DSN stations is in CCSDS-TDM format.

4.4 Orbit Determination Performance

Orbit determination performance is studied by its quickness, reliability, consistency of results and by its achieved accuracy. To start with, 1st determined orbit (injected orbit) by mission operational OD software using long arc tracking data was compared with nominal/expected orbit and INS (Inertial Navigation System) determined orbit. Orbit determination consistency was studied with short arc tracking data OD results versus long arc tracking data OD results. Present study of orbit determination performance analysis is from start of launch to start of Moon mapping phase. Present study includes Chandrayaan-1 injected orbit determination, orbit determination consistency and comparison of OD results with other source.

4.4.1 Estimated injected orbit and its comparison

The first OD was carried out using 36 minutes of tracking data from Goldstone within ten minutes of tracking data availability. Chandrayaan-1 orbit determination results with short arc and long arc tracking data (about ten hours of tracking data) are given in Table 5. A good convergence was achieved and the realized orbit is (254.45 X 22932.74) Km. Table-6 gives 'realized orbit with long arc tracking data' comparison with nominal and INS measured orbit along with dispersions. The differences in perigee and apogee heights with respect to expected orbit are 1.47 km and 67.514 km respectively. The realized inclination is differed by 0.019 deg from the expected. Results indicate that the injected orbit is well within the vehicle quoted dispersions.

4.4.2 Orbit Determination consistency

A number of orbit determinations were carried out in each phase of the mission by including additional tracking data as and when available. Orbit determinations were carried out with short arc and long arc data durations. The determined orbit in each phase of the mission is shown in Table 7. The summary of Orbit determinations consistency is shown in Table 8. The maximum difference in perigee and apogee differences during all phases of the mission are within 2 km. The maximum inclination difference between short-arc OD versus long-arc OD is 0.109 deg during initial phase of the mission. These results indicate quick and reliable solutions for mission planning, in the sense of carrying out subsequent maneuvers.

4.4.3 Achieved orbit determination accuracy

Orbit determination accuracy was analyzed with theoretical definition and by comparing orbits achieved by other source. The orbit accuracy is measured as the difference between two consecutive definitive ephemerides over the period of overlapping tracking data. It is the method of "Difference in Position" [7]. Present study carried out for OD accuracy is based on comparing ISRO OD solutions with solutions achieved by other source.

4.4.3.1 Comparison of Orbit Solutions

A study was made to assess the orbit determination accuracy by comparing orbit solutions obtained from JPL during initial phase of the mission. Tracking data used is almost identical for both orbit solutions. Summary of determined orbits comparison during pre lunar mapping phase and during lunar mapping phase is shown in Table 9. These results indicate that the maximum difference in position is within 500m.

Table 5 Chandrayaan-1 injected Orbit with short arc and long arc tracking data

OD Description	Tracking data duration 36 minutes of data	 10.28 hrs of data
Epoch	2008 10 22 01 10 19 081	2008 20 22 01 10 19 081
Tracking data start	2008 10 22 02 53 00 000	2008 10 22 02 53 00 000
Tracking data end	2008 10 22 03 29 00 000	2008 10 22 13 09 50 000
No. of measurements	21 (range); 217 (Doppler)	338 (range); 3239 (Doppler)
Tracking station	Goldstone	Canberra and Goldstone
Spacecraft separation	2008 10 22 01 10 19 081	2008 10 22 01 10 19 081
Tracking data availability	2008 10 22 03 50 00 000	2008 10 22 13 39 00 000
OD solution availability	2008 10 22 04 00 00 000	2008 10 22 13 49 00 000
OD Solution Definitive State (km, km/s, EME J2000)		
X, Y and Z	-6815.348, -867.612, -511.567	-6812.371, -870.606, -516.818
XD, YD and ZD	-0.252706, -9.200956, -2.968840	-0.250933, -9.203888, -2.96618
X, Y , Z , XD, YD and ZD are State Parameters		

Table 6 Chandrayaan-1 Realized Injected Orbit
Comparison with Nominal and INS Measured Orbit Parameters (PSLV-C11)

Parameters	Realized	INS	Expected	Difference (Re-INS)	Difference (Re-Ex)
Semimajor axis (km)	17971.737	17937.758	17938.716	33.979	33.022
Eccentricity	0.6309430	0.6301530	0.6301817	0.00079	0.000761
Inclination (deg)	17.9112250	17.894000	17.891655	0.017225	0.019570
Perigee height (km)	254.457	256.091	255.928	-1.634	-1.470
Apogee height (km)	22932.741	22863.151	22865.227	69.59	67.514

Quoted Dispersions (1σ): ÄPh : : 5km, ΔAh : 675km, Δi : 0.2 deg

Table 7 At a glance: Chandrayaan-1 Determined Orbits in Each Phase

Phase	Date	a (km)	e	i (deg)	Ph (km)	Ah (km)	Period (hr)
Launch	2008-10-22	17971.737	0.630942	17.911224	254.457	22932.741	6.660
EBN#1	2008-10-23	25481.828	0.737956	17.897671	299.212	37908.168	11.245
EBN#2	2008-10-25	43904.411	0.847059	17.971973	336.624	74715.921	25.431
EBN#3	2008-10-26	89059.982	0.924477	17.975949	347.939	165015.748	73.474
EBN#4	2008-10-29	139914.481	0.951128	18.043095	459.623	266613.064	144.678
EBN#5	2008-11-03	196794.686	0.962646	18.261291	972.898	379860.202	241.340
TCM#1	2008-11-05	196040.462	0.963321	18.098487	815.179	378515.480	239.953
LOI	2008-11-08	5747.071	0.609197	92.881109	507.972	7510.169	10.860
LBN#1	2008-11-09	5589.564	0.653159	92.719579	200.690	7502.439	10.417
LBN#2	2008-11-10	1957.189	0.018446	92.967628	183.086	255.292	2.158
LBN#3	2008-11-11	1916.479	0.040051	93.021453	101.723	255.236	2.091
LBN#4	2008-11-12	1840.379	0.000258	93.200827	101.904	102.855	1.968

Table 8 Summary of Orbit Determination Consistency

Phase/Orbit (km)	Short Arc Vs Long Arc	ΔP_h (km)	ΔA_h (km)	Δi (deg)
Launch (254X22932)	4 h 25 min Vs 10 h 16 min	0.000	0.079	0.003
EBN#1 (299X37908)	4 h 41 min Vs 28 h 37 min	0.040	0.266	0.002
EBN#2 (336X74715)	3 h 28 min Vs 10 h 08 min	0.285	0.287	0.002
EBN#3 (347X165015)	3 h 47 min Vs 37 h 27 min	0.074	0.481	0.001
EBN#4 (459X266613)	4 h 15 min Vs 37 h 15 min	0.222	0.439	0.001
EBN#5 (972X379860)	3 h 38 min Vs 28 h 36 min	0.641	0.623	0.002
LOI (508 X 7510)	2 h 23 min Vs 25 h 53 min	0.119	1.155	0.022
LBN #1 (200.7 X 7502)	2 h 54 min Vs 11 h 44 min	1.836	0.044	0.002
LBN #2 (183.0 X 255.2)	1 h 11 min Vs 09 h 16 min	0.181	0.086	0.009
LBN #3 (101.7 X 255.2)	1 h 27 min Vs 23 h 59 min	0.094	0.215	0.109
LBN #4 (101.9 X 102.8)	0 h 57 min Vs 08 h 03 min	0.005	0.001	0.005

Table 9 Summary of Comparison of OD Solutions with external agency

Phase/ Epoch (U.T)	Position Difference (m)	Velocity Difference (m/s)
Launch: 2008-10-22-02-58-54-818	465	0.068
EBN-1 : 2008-10-23-08-00-00-000	58	0.010
EBN-2 :2008-10-25-02-00-00-000	280	0.160
EBN-3 :2008-10-26-02-08-54-817	126	0.029
EBN-4 :2008-10-29-02-13-54-817	70	0.047
EBN-5 :2008-11-03-23-43-54-817	74	0.015
LOI :2008-11-08-11-33-54-817	219	0.126
LBN-1 :2008-11-09-14-38-54-817	361	0.017
LBN-2 :2008-11-10-19-18-54-817	6	0.005
LBN-3 :2008-11-11-23-58-54-817	80	0.056
LBN-4 :2008-11-12-13-08-54-817	10	0.015
LM :2008-11-13-09-58-54-817	62	0.080
LM :2008-11-17-20-48-54-817	149	0.073
LM :2008-11-19-00-08-54-817	144	0.123

EBN: Earth Burns; LOI: Lunar Orbit Insertion; LBN: Lunar Burns; LM: Lunar Mapping

5. CONCLUSION

5.1 Orbit Solutions using accelerometer data

Real time orbit determination was demonstrated live for the first time in ISRO's history of all of its satellite missions. The Chandrayaan-1 Mission gave this opportunity. PROCAD program was vindicated as it gave orbit solutions to a high degree of accuracy. The PROCAD orbit determination result is available to the mission managers at the very instant of the burn end. The orbit determination result using tracking data is known much later after the burn end, after enough tracking data is collected. Since the PROCAD orbit results compared so well with the tracking data orbit results, PROCAD proved it's utility to be, as in the LunarBurn-4 of the Chandrayaan-1 mission, an instantaneous provider of the achieved orbit after a maneuver. Comparison of the PROCAD orbit results with those determined with tracking data using 'Batch Weighted Least Squares Estimation', showed close association between the state vectors. PROCAD showed a maximum position difference of 0.32 % while the minimum position difference was as less as 0.0002 %. The maximum velocity difference was 0.38 % while the minimum velocity difference was as less as 0.006 %. The PROCAD program used in the Mission Operations Center at ISTRAC for the Chandrayaan-1 Mission is an effective validation tool for an onboard accelerometer.

5.2 <u>Orbit Solutions using tracking data</u>

ISRO's operational Orbit Determination Program (ODP) for low earth missions was suitably updated and validated with simulated as well as live tracking data of Lunar Prospector (LP) and SMART-1 before the launch of Chandrayaan-1 to cater the need aspect of Chandrayaan-1. Performance of Chandrayaan-1 orbit determination system during launch and initial phase of the mission is presented. Performance analysis includes tracking data quality analysis, orbit determination consistency and comparison of OD results with other source. PSLV-C11/Chandrayaan-1 injected orbit was determined and compared with nominal orbit to study the performance of PSLV-C11.The differences in perigee and apogee heights with respect to expected orbit are 1.47 km and 67.514 km respectively. The realized inclination is differed by 0.019 deg from the expected. Results indicate that the injected orbit is well within the vehicle quoted dispersions. Orbit Determination consistency was studied. The orbit solutions obtained from short arc OD's are compared with those obtained using the longest arc OD of each stage of the initial phase of the mission. The maximum difference in perigee and apogee differences during all phases of the mission is within 2 km. The maximum inclination difference between short-arc OD versus long-arc OD is 0.109 deg. This analysis indicates the determined orbit solutions are quick and reliable for mission planning, in the sense of carrying out subsequent maneuvers. A study was carried out on comparison of OD results with that of external agency OD results during all phases of the mission and concluded that the orbit solutions do match well. Quick, accurate and consistent orbit determination solutions obtained during early phase resulted in success of Chandrayaan-1 mission.

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