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Development Strategy of Orbit Determination System for Korea's Lunar Mission: Lessons from ESA, JAXA, ISRO and CNSA's Experiences

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In this paper, a brief but essential development strategy for the lunar orbit determination system is discussed to prepare for the future Korea's lunar missions. Prior to the discussion of this preliminary development strategy, technical models of foreign agencies for the lunar orbit determination system, tracking networks to measure the orbit, and collaborative efforts to verify system performance are reviewed in detail with a short summary of their lunar mission history. Covered foreign agencies are European Space Agency, Japan Aerospace Exploration Agency, Indian Space Research Organization and China National Space Administration. Based on the lessons from their experiences, the preliminary development strategy for Korea's future lunar orbit determination system is discussed with regard to the core technical issues of dynamic modeling, numerical integration, measurement modeling, estimation method, measurement system as well as appropriate data formatting for the interoperability among foreign agencies. Although only the preliminary development strategy has been discussed through this work, the proposed strategy will aid the Korean astronautical society while on the development phase of the future Korea's own lunar orbit determination system. Also, it is expected that further detailed system requirements or technical development strategies could be designed or established based on the current discussions.

Keywords: Lunar mission, Orbit determination system, Orbit measurement system, Tracking network, CCSDS standard, ESA, ISRO, JAXA, CNSA

1. INTRODUCTION

Satellite orbit determination (OD) has evolved continuously over the past 50 years through researches conducted by worldwide astrodynamics specialists from industries, universities, and government organizations. In the early days of OD for earth orbiting satellites, for example, the Sputnik, the satellite was tracked mostly by visual observation and the accuracy of OD was only a few kilometers. In 1960s, the missile tracking technology used in the Army was transferred to the Air Force and employed for satellite tracking and OD. To improve the accuracy of satellite OD in 1960s, the mixed sets of observation data of optical and Doppler were used, and a positional uncertainty was about 500 m for a 1,200 km orbit. With the development of laser ranging system in the mid to late 1960s, the precision of the observation was improved to 5-10 m. Since 1970s, advances in laser, radio tracking and force modeling technology have improved orbit accuracies better than 5 cm in orbit altitude. Nowadays 3-D orbit precision accuracies are routinely about 2-5 cm range (Vetter 2007). The principal application of OD was limited to Earth orbiting satellites. However, Deep Space Network (DSN) enabled a very accurate angular measurement observations the and simultaneous observations of one satellite from two DSN sites in different continents made it possible to determine spacecraft motions beyond the Earth's gravitational attractions precisely. Currently, at least 18 major professional OD softwares are being used for various organizations, application areas, data types and program capabilities. In Table A1, organization

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specific OD programs and their major application areas are shown (Vetter 2007).

Korea Aerospace Research Institute (KARI) has been continuously operating an earth satellite since the first launch of Korea Multi-Purpose Satellite -1 (KOMPSAT-1) in 1999, and now expands interests to outer space. Korea plans to send up a lunar orbiter and a lander before the end of 2020, and also plans to explore the Mars, asteroids, and deep space. Therefore, Korean atronautical community performed numerous basic relevant studies and KARI is also performing pre-phase developments and researches for a lunar mission to be launched in next decades. The preliminary design studies comprises optimal transfer trajectory analysis, mapping orbit analysis, contact schedule analysis, link budget analysis, and the design analyses of a lander, a rover, and candidate payloads. Although numerous fascinating preliminary design studies were performed, they all have focused on system design point of views, nevertheless the development of OD software for a lunar mission is certainly another critical field that must be analyzed and studied in depth. In Korean astronautical community, researches about the OD of a satellite orbiting in the vicinity of planets other than the Earth have rarely been performed. Kim et al. (2004) performed measurement modeling studies for the Mars mission at the phase of Sun centered cruising, and none of the lunar mission OD related studies, neither design studies nor analysis results, have been actively done so far. As the future Korea's lunar mission is being shaped in detail, the specialized OD program for the lunar mission should be developed. The authors believe that it is desirable to develop the OD program for the future Korea's lunar missions on the basis of KARI's past heritages on satellite operations and development experiences. Therefore, a brief but essential development strategy for the future Korea's lunar orbit determination system is discussed which could be used as a guideline for a detailed OD program design in this work. Prior to the discussion of this preliminary development strategy, technical models of foreign agencies for the lunar orbit determination system, tracking networks to measure the orbit, and collaborative efforts to verify system performance are reviewed in detail with a short summary of their lunar mission history Based on the lessons obtained from their experiences, the preliminary development strategy mainly focused on technical issues is discussed. This work is intended to give numerous insights into system engineers who are willing to develop a future lunar or planetary OD program saving their time and efforts. Accordingly, it is expected that further detailed system requirements or technical development strategies could be designed or established based on the current discussions. In Section 2, details of other foreign agencies' previous OD program development experiences are described. Foreign agencies covered are European Space Agency (ESA), Japan Aerospace Exploration Agency (JAXA), Indian Space Research Organization (ISRO) and China National Space Administration (CNSA). In Section 3, discussions on technical issues (dynamic modeling, numerical integration, measurement modeling, estimation method, measurement systems as well as appropriate data formatting with regards to interoperability between foreign agencies) are made based on the common aspects discovered in Section 2 for the development of Korea's lunar OD program. Finally, conclusions are given in Section 4.

2. FOREIGN AGENCIES' EXPERIENCES

2.1 ESA's Experiences

2.1.1 Summary on Lunar Missions

Although many interesting planetary missions were performed by the ESA, Small Missions for Advanced Research in Technology -1 (SMART-1) was the first European spacecraft to travel to and orbit around the Moon. Launched on September 27, 2003, SMART-1 arrived in lunar orbit on November 15, 2004 and after conducting lunar orbit science operations, the mission ended on September 3, 2006 when the spacecraft impacted on the lunar surface in the Lacus Excellentiae region. The SMART-1 mission was aimed to test the solar electric propulsion and other deep-space technologies, while performing scientific observations of the Moon. Encouraged by the success of SMART-1 mission, a lunar lander is scheduled to launch in 2018 in accordance with the Europe's ambitions for lunar exploration. The lunar lander mission will demonstrate key European technologies and conduct science experiments which will be a forerunner of future human and robotic explorations of the Moon and the Mars (ESA 2014a).

2.1.2 Orbit Measurement System

The SMART-1 was tracked mainly with ESA tracking station network (ESTRACK). The ESTRACK comprises 10 stations dispersed in 7 countries: Kourou station in French Guiana, Maspalomas, Villafranca and Cebreros in Spain, Redu in Belgium, Santa Maria in Portugal, Kiruna in Sweden, Perth and New Norcia in Australia and Malargüe in Argentina. During routine operations, stations were remotely operated from the ESTRACK Control Center (ECC) at European Space

Operation Center (ESOC). All stations host 5.5, 13, 13.5 or 15 meter antennas except New Norcia, Cebreros and Malargüe. Those are equipped with 35 meter Deep Space Antennas (DSAs). New Norcia and Cebreros sites were completed in 2002 and 2005 while Malargüe began routine services in January 2013 (ESA 2014b). For the SMART-1 mission, OD has been routinely performed two or three times per week depending on the phase of the mission with two way S-band range and Doppler data. These data were available from the frequent passes of a number of ESA stations. Villafranca II had tracked most frequently followed by Perth, Maspalomas, Kourou, Villafranca I and New Norcia with meteorological data obtained at all stations. In addition, observations below 10 degree of elevation were excluded with standard deviations of 20 m for the range and 1 mm/s for the Doppler measurements (Mackenzie et al. 2004). Other than the SMART-1 mission, for the Mars EXpress mission (MEX), as an example, the primary ground station was the ESA at New Norica with 35 meter DSA. However, NASA DSN provided additional coverage mainly from the ground station in Madrid, but occasionally from Goldstone and Canberra (Han et al. 2004).

2.1.3 Orbit Determination System

Since early 1980s, ESOC's system for interplanetary Orbit Determination (OD) program has been developed and set up. Initial staring point of interplanetary OD program was to support the Giotto mission, the ESA's first interplanetary mission, launched in 1985. Up until now, the OD system has proven to be reliable and robust through successful operations of numerous ESA's interplanetary spacecrafts (Budnik et al. 2004). Budnik et al. (2004) addressed that very long and remarkably challenging ways had to be travelled to reach the status. ESOC's OD software mainly consists of Multi-Satellite Support System (MSSS), Navigation Package for Earth Orbiting Satellites (NAPEOS), Portable ESOC Package for Synchronous Orbit Control (PEPSOC), Interplanetary Software Facility (IPSF) and Advanced Modular Facility for Interplanetary Navigation (AMFIN). With no doubt, the main purpose of all of these softwares is to ensure the safe navigation of a spacecraft (Budnik & Mackenzie 2009). Among these ESOC's OD software packages, NAPEOS is the navigation package for Earth orbiting spacecrafts and IPSF and AMFIN are used for interplanetary missions (Mackenzie & Budnik 2009). IPSF can be categorized into four main programs: spacecraft orbit determination program, comet and asteroid orbit determination program, relative orbit determination program and finally, orbit determination program for the spacecraft orbit around a comet. Spacecraft orbit determination program is the default OD program to determine the spacecraft orbit, and relative orbit determination program makes an improvement to the estimates of the states of a solar system body relative to the state of the spacecraft by using optical data including background stars (Budnik and Mackenize 2009). AMFIN is a provision for advanced concept of interplanetary OD program which is designed to be adaptable for future requirements; i.e., adding new dynamic/measurement models, adding uncertain parameters, allowing new measurements and spacecraft types. AMFIN's adaptability has worked well for MEX, Rosetta, SMART-1, Venus EXpress (VEX), Hershel and Planck Mission (Mackenzie & Budnik 2009). Basically, for the SMART-1 mission, the OD software based on the AMFIN libraries was used, and therefore the OD software used for the SMART-1 mission has much in common with the software used for the Rosetta and the MEX mission. In addition, the AMFIN software were rigorously cross-verified with respect to the JPL software through a series of tests (Mackenzie et al. 2004). For ESOC's interplanetary OD program, the trajectory of a spacecraft is propagated by numerical integration using a scheme attributed to Nordsieck (Nordsieck 1962). The method is known to be numerically very stable and utilizes multi-value, variable step size algorithm. For dynamical modeling, the ESOC used the same integration software but different dynamical models chosen appropriately were used for the Rosetta, the MEX and the SMART-1, respectively (Budnik et al. 2004). The dynamic model for the OD of SMART-l consisted of the followings. First, central potentials of the Earth, the Moon, the Sun and all planets were based on Jet Propulsion Laboratory (JPL) DE405 ephemerides and relativistic perturbations due to the Sun were modeled. Second, Joint Earth Gravity 3 (JGM3) was used for earth gravity field model, and NASA Goddard Lunar Gravity Model-2 (GLGM-2) was used for lunar gravity field with appropriate the degree and order of depending on the mission phases. Third, Solar Radiation Pressure (SRP) with a flat plate model was used with constant spacecraft mass but updated as required. Acceleration due to the motor burn was either treated as impulsive or having finite duration. For the SMART-1, Solar Electric Propulsion (SEP) maneuver was modeled. Finally, Wheel off Loading (WOL) maneuvers with finite duration were also modeled (Mackenzie et al. 2004).

For measurement modeling, current ESOC's interplanetary OD program can process two-way range and Doppler from ESA and DSN ground stations and Differential One-way Range (ΔDOR) from the DSN ground stations (Budnik et al. 2004). The typical random errors of ESA's deep space tracking system are about 1 m for ranging and those for the two-way range-rate are less than 0.1 mm/s. Nevertheless, the typical random errors described above indicate that the accuracy of

Table 1. General measurement accuracies for DSN with X-band signal (Budnik et al. 2004).

	Measurement Accuracy	Model Accuracy
2-way range (m)	1-2	0.1
2-way Doppler (mm/s)	0.1	0.01
Delta-DOR (nrad)	15	1.5

resulting OD may not be good enough for navigation during the critical stages of a mission, i.e., approaching a planet before landing, performing a swing-by or insertion into orbit (Madde et al. 2006). Therefore, to improve navigation accuracies, the concept of ΔDOR technique was established which was simple but very effective. Basic concept of ΔDOR is that the spacecraft is tracked simultaneously by two widely separated antennas to measure the time difference between the signals arriving at the stations. Then the signal delay due to numerous errors are corrected by tracking a quasar in a direction close to the spacecraft for calibration. The chosen quasar's direction is already known extremely accurately by astronomical measurements better than to a degree of 50 billionths typically (Madde et al. 2006). For the current ESA X-band tracking systems, the level of navigation accuracies is 0.1 mm/s for Doppler, 1 to 5 m for ranging and 6 to 15 nrad for ΔDOR, respectively (Iess et al. 2013). Generally, accurate measurement modeling requires extremely sophisticated mathematical model that could compute these measurement to an accuracy better than the actual measurement accuracy. The general requirement for the modeled observation should be approximately one order of magnitude better than the accuracy of the actual measurements. Assuming X-band signals, Budnik et al. (2004) addressed the general measurement accuracies for the DSN as shown in Table 1.

The measurement modelings made with the ESOC's interplanetary OD program were roughly divided into three main parts: (1) different time scales transformation, (2) the transformation of the ground station Earth-fixed coordinates into an inertial solar system barycentric system, and (3) the computation of the precision light time (Budnik et al. 2004). The authors hopefully expect that these partitioned models from the ESOC's system could be applicable to other OD programs aimed for interplanetary mission. Modeling in different time scale is essential as the time scale used in the ground station is Coordinated Universal Time (UTC) while the time scale used to propagate the orbits of a spacecraft and the solar system bodies is Barycentric Dynamical Time (TDB). It is well known that there exist two major transformation methods between TDB and UTC, the trigonometric formulation (Moyer 1971) and vector formulation (Moyer 2000), however, ESOC's system adapted new concept called "hybrid formulation" to minimize computation time as well as to be more accurate by about 4.2 μs which was neglected in vector formulation method (Budnik et al. 2004). By adapting "hybrid formulation" Moyer (2000) reported that the maximum difference between the vector and the hybrid formulation is 0.3 μs which is an order of magnitude smaller than the overal1 accuracy of the vector formulation implemented in the JPL system. The direct error on the modeling of different time scale could result in about 0.13 m/AU (Astronomical Unit) distance on 2-way range observable using the vector formulation which is the JPL model. In the ESOC's system the direct error caused due to the modeling of different time scale is reduced to 0.08 m/AU distance (Budnik et al. 2004). As it is well known that the interplanetary OD is extremely sensitive to the errors in the ground station positions, the transformation of the ground station Earth-fixed coordinates into an inertial solar system barycentric system should be as accurate as possible. For the ESOC's system, the station inertial position is measured with an accuracy of a few cm (obtained from GPS data and a local survey) to meet the required accuracy for modeling the measurements. The ESOC's system used International Terrestrial Reference Frame 2000 (ITRF2000) for the position of the ground station. And International Celestial Reference Frame aligned with the FK5 star catalogue at J2000 (ICRF FK5/J2000) was used for the spacecraft's inertial reference system (Budnik et al. 2004). During the transformation between ITRF2000 and ICRF FK5/J2000, well known effect of the Earth rotation, nutation and precession as well as polar motion were considered in the model, which is consistent with the International Astronomical Union 1980 (IAU 1980) theory of nutation. In addition, the ESOC's interplanetary OD program applied translations whose effects were expected to be significantly larger than 1 cm as follows: platetectonics, first order solid Earth tides, and modified Lorentz transformation (Budnik et al. 2004). To compute the precision light time, the following effects were taken into account in the ESOC's system: the Newtonian light time, the reduction of the coordinate velocity below the speed of light due to the gravity field of massive bodies (i.e., the Sun, Jupiter, Saturn, and the Earth etc.), the bending of the light path due to the gravity field of the Sun, the refraction of the signal when passing through the Earth's troposphere and ionosphere, the effects due to charged particles in the solar plasma, and instrumental delays in the spacecraft or the ground station (i.e., transponder delay, antenna mounting, station electronic delays). With precision light time, the range observables could be modeled easily in relation to station reception time. Finally, the ESOC's OD system used a statistical estimation method called Square Root Information Filter (SRIF) which is exactly equivalent to weighted least squares mathematically but numerically superior (Budnik et al. 2004). SRIF was implemented into the ESA's operational OD software by GMV© to cover a high level of efficiency, precision and reliability

Table 2. Considered parameter within the estimation process for the SMART-1 mission (Mackenzie et al. 2004).

Considered parameter	A priori standard deviations
Station location component uncertainty	10 cm
Range bias per station	20 m
Wet troposphere correction	4 cm
Dry troposphere correction	1 cm
Ionosphere correction	10 cm
Transponder delay	10 ns

Table 3. Observation modeling accuracies between ESOC and JPL with Stardust case (Budnik et al. 2004).

Parameter	Difference (ESOC-JPL)
(TDB-UTC) at reception and transmission time	-90 ns
Precision round-trip light time (TDB)	0.17 ns
ICRF station position at reception time	34.5 mm
ICRF spacecraft position at turn-around time	5.3 mm
ICRF station position at transmission time	35.4 mm
2-way range measurement	55.8 mm
2-way Doppler measurement	0.003 mm/s

(Mate & Fadrique 2000). For the SMART-1 mission, considered parameters within the estimation process are shown in Table 2 with a priori standard deviations.

2.1.4 Performance Verification and Validation

The ESOC's OD software was rigorously cross-verified with respect to the JPL software through a series of tests which were performed together with each agencies' Test & Validation Office (TVO). To validate the OD system's performance of two agencies', identical input data were used and then the results and the differences were compared. Since the JPL's OD program is mature and has been successfully applied for many interplanetary missions, JPL results were treated as a reference to validate the ESOC's system. For cross verifications, the appropriate scenarios of the Rosetta and the MEX mission have been chosen. The position differences of 90 days heliocentric cruise with 5 different force models through numerical propagation at the end of the arc between the ESOC and the JPL were found to be about 1 m in all 5 tests. In addition, for Mars-centric orbit propagation, 2 days Mars-centric orbit with 3 different force models showed the position differences of less than 2.2 mm at the end of the propagation arc (Budnik et al. 2004). To validate modeling accuracies of 2-way range and Doppler, actual measurements of Nozomi and Stardust spacecraft acquired at the DSN stations were used. According to the Nozomi's measurements, differences of 13.5 mm and 0.0005 mm/s in 2-way range and Doppler were achieved, respectively. For the Stardust case, the results are summarized in Table 3. Other than these modeling accuracies, ΔDOR modeling accuracies were about 1.2 nrad difference, and covariance mapping with targeting plane parameter were in good agreement between the ESOC and the JPL showing less than 10^{-4} in the B-plane quantities (Budnik et al. 2004).

2.2. JAXA's Experiences

2.2.1. Summary on Lunar Missions

The Kaguya mission, the code name of SELenological and Engineering Explorer (SELENE), is the first large lunar exploration mission of Japan. On September 14, 2007, the Kaguya spacecraft was launched by the H-IIA rocket. After several orbital correction maneuvers and lunar orbit insertion maneuvers, Kaguya was successfully inserted into the lunar orbit on October 4, 2007 (Ikeda et al. 2009) with payloads of 14 scientific instruments (Sasaki et al. 2008). As well known, the Kaguya mission is the first mission to obtain the tracking data on the far side of the moon by using the combination of the lunar orbiter and the data relay satellite. The Kaguya mission consisted with three satellites: the main orbiter Kaguya, the relay sub-satellite Okina, and the Very Long Baseline Interferometry (VLBI) sub-satellite Ouna. Kaguya was put into 100 km altitude circular orbit, Okina was put into 100×2400 km altitude elliptic orbit, and Ouna was put into 100×800 km altitude elliptic orbit, respectively. The inclination of these satellites were about 90 degrees (Ikeda et al. 2009). The nominal mission phase for the Kaguya mission was finished at the end of October 2008, and the extended mission phase started from the beginning of November 2008. At the end of extended mission phase, Kaguya made a controlled impact to the lunar surface on June 10, 2009 (Ikeda et al. 2009). For further future missions, JAXA plans the SELENE-2 mission which will include a lunar orbiter, a lander and a rover to be launched in 2017. The SELENE-3, a lunar sample return mission and an advanced lander for the future human mission to the Moon is also planned (Fujita 2012).

2.2.2. Orbit Measurement System

Although both JAXA and JPL applied flight dynamics for SELENE: orbit determination, prediction and real-time monitoring (Ogawa et al. 2008), the flight dynamics division of the JAXA played main roles (Ikeda et al. 2009). The SELENE mission operations have been mainly carried out at the SELENE Operation and Analysis Center (SOAC) at the Sagamihara campus of JAXA. The SELENE flight dynamics team was stationed both at SOAC and also at Tracking and Control Center (TACC) at the Tsukuba Space Center of JAXA (Ogawa et al. 2008). In the daily operation of SELENE mission, the JAXA's Ground Station Network (JGSN) and deep space centers were used for the Telemetry, Tracking and Command

(TT&C) operation. However, during the critical phase, the tracking data from the DSN were obtained and utilized (Ikeda et al. 2009). JAXA ground network equips 10-meter antennas at Katsuura (KTU1 and KTU2), Masuda (MSD1), Okinawa (OKN1 and OKN2), Perth (PRT1), Maspalomas (MSP1), Santiago (SNT1), a 64-meter antenna at the Usuda Deep Space Center (UDSC) and a 34-meter antenna at the Uchinoura Space Center (USC). The NASA DSN supporting SELENE has a 34-meter antennas at Goldstone, Canberra and Madrid and a 26-meter antennas at Canberra and Madrid (Ogawa et al. 2008). During the SELENE operation, the measurement data interface between JAXA and JPL was formatted using Consultative Committee for Space Data Systems (CCSDS) Orbit Ephemeris Message (OEM) which included data delivery time, ephemeris data space and covariance matrix map time (Ogawa et al. 2008). The 2-way S-band range and Doppler measurements were used for the orbit determination of Kaguya, Okina and Ouna in the daily operation. In addition, when the orbiter was behind the moon, satellite-to-satellite the 4-way Doppler measurement of Kaguya via Okina was obtained in order to improve the gravity model of the moon (Ikeda et al. 2009). This is the world's first to perform the direct observation of the gravity of lunar far-side and these 4-way Doppler data make considerable contributions to the improvements of the lunar gravity model (Namiki et al. 2009). To predict spacecrafts' ephemerides (Kaguya, Okina and Ouna) for the operation planning and the initial scientific analysis, OD were performed twice a week for the Kaguya spacecraft and once a week for the Okina and Ouna spacecraft, respectively. (Ikeda et al. 2009).

Table 4. Characteristics of the OD and propagation functions for SELENE mission (Ogawa et al. 2008).

Force Model	Gravity (Sun, Moon, all planets)
	Harmonization coefficients of the gravity of the Earth and the Moon (degree max=100, order max=100)
	Solar radiation (considering penumbra)
	Atmospheric drag (Earth)
Numerical integration	Predictor/corrector method by back points with variable step size and variable order
Time systems	Barycentric Dynamical Time (TDB)
Coordinate systems	Mean equator and equinox of 2000.0 (J2000.0)
Coordinate origin	Arbitrary celestial body (Sun, Moon, all planets)
Estimation method	Batch least square estimation by SRIF (Square Root Information Filter) with householder orthogonal transformation
Measurement models	2-way range and Doppler
	1-way Doppler
	Antenna angles (Azimuth and Elevation)
Estimation parameters	Orbital elements (Cartesian)
	Scale factor of air drag coefficient error
	Scale factor of solar radiation pressure coefficient error
	Observation data biases
	Ground station location
	Impulse maneuver
	Other Small force
Consider parameters	Observation data biases
	The gravitational constant of the moon Harmonization coefficients of the lunar gravity (degree max=100, order max=100)
	Scale factor of air drag coefficient error
	Scale factor of solar radiation pressure coefficients
	Ground station location
	Impulsive maneuver
	Other small forces

2.2.3. Orbit Determination System

Japan has begun developing OD software for a planetary mission at the early 1980s. The first software called ISaS Orbit determination Program (ISSOP) was developed to support the Sakigake mission which was launched in 1985 to explore the comet Halley. The ISSOP was then used for Hiten mission which was launched in 1990 for the experiments of swing-by, and it was modified in several areas to carry out the much more precise OD for the Nozomi mission (Yoshijawa et al. 2005). For the OD software for lunar missions, especially the SELENE, the flight dynamics section in National Space Development Agency of Japan (NASDA), the former organization of JAXA, has been studying a concept of the flight dynamics system since 1993 (Shinozakil et al. 2000). Their preliminary study included visibility analysis, coverage analysis, sensitivity analysis of lunar gravitational potential, orbit determination precision analysis, and covariance propagation analysis etc. Then the system requirements for their own OD system with Flight Dynamics Subsystem (FDS) were roughly derived. The outlines of the system components of FDS corresponding to the SELENE mission were as follows: (1) an orbit determination subsystem which was a group of functions about orbit determination including the observation data preprocessing, (2) a flight planning subsystem which was a group of functions related to the flight control parameter preparation, (3) a mission analysis subsystem which was a group of functions about the orbit determination precision and the mission analysis such as maneuver parameter error analysis, (4) a network support subsystem which was a group of functions related to the preparation of the flight dynamics information that is necessary for tracking station operation, (5) a SELENE characteristics information subsystem which was a group of functions related to the calculation of the characteristic information for SELENE, such as satellite/station event or solar angle, and finally, (6) a data management subsystem that was a group of functions related to the data

Table 5. OD models and assumptions used by the JAXA and the JPL for the SELENE mission (Haw et al. 2008).

Error Source	Estimated	1 sigma a priori uncertainty
Angle tracking data (deg)	No	0.02 (only first 2.5 hours after launch)
2-way Doppler data (mm/s)	No	0.25 (varied between 0.2~1.0)
Range (m)	No	2 (varied between 1~3)
Epoch position (km)	Yes	1000
Epoch velocity (km)	Yes	1
Range bias (m)	Yes	10 (estimated per station)
Angle bias (deg)	Yes	0.05
Ephemerides and GM	No	DE405
Station locations (cm)	No	3
Pole X, Y (cm)	No	10
Ionosphere –day, night (cm)	No	75,15
Troposhpere – wet, dry (cm)		4, 1
Solar pressure scale factor	Yes	0.5
Angular momentum desaturation event Del-V X,Y,Z (cm/s)	Yes	10,10,10
Non-gravitational acceleration (km/s^2)	Yes	3.0e10-11
Every orbit maneuvers' magnitude and locations (N,deg,deg)	Yes	Dependent to maneuvers
Lunar gravity	Yes	LP150Q

management such as system management (Shinozakil et al. 2000). After 5 years of preliminary study, a prototype system for OD was developed to process 2-way range and Doppler data in the earth-moon transfer orbit and the lunar orbit. The prototype program consisted of an orbit propagation program and an orbit determination program. The orbit propagation program used predictor-corrector method with variable order and variable step size integrator to integrate first/second-order differential equation. For the orbit determination program, the Square Root Information Filter (SRIF) with householder orthogonal transformation is applied to estimate the orbital elements and various parameters (Shinozakil et al. 2000). Finally, the JAXA developed a FDS dedicated to the operation of the SELENE mission. The trajectory and orbital maneuver planning sub-system was developed at SOAC, and a backup machine was also set up at TACC. The other subsystems, i.e., orbit determination, network support and operation management subsystems, were developed at TACC with the capability of remote operation from SOAC. Especially, the OD subsystem was developed based on the prototype OD system for interplanetary missions as discussed above (Ogawa et al. 2008). In Table 4, the characteristics of the OD and propagation functions for the SELENE mission is summarized.

2.2.4. Performance Verification and Validation

All JAXA OD solutions (ephemeris as well as covariance matrix) for the SELENE mission were verified by comparing those with the JPL solutions (Ogawa et al. 2008). To validate the OD solutions, JAXA and JPL took responsibility for trajectory predictions for the acquisition of the SELENE at the ground stations of their own organization. Since the DSN would acquire the SELENE tracking data first at the DSN's Madrid complex, about 1 hour after launch which is an extremely short interval compared to about 12 hours later for JAXA's initial acquisition of signals at its Usuda station, JAXA recruited JPL for the launch acquisition and the initial OD. In addition, major OD results were needed to be delivered to JAXA within 12 hours of launch (Haw et al. 2008). As numerous spacecraft maneuvers had been performed throughout the cruise phase and almost all of them were critical, JPL was asked subsequently to lead the OD task from the low Earth orbit to the low lunar orbit while the SELENE flight dynamics team performed a back-up OD during the same time span and they took the responsibilities for all maneuver designs (Haw et al. 2008). During orbital maneuvers, real-time Doppler monitoring was performed by JAXA using the UDSC 2-way Doppler data, while another monitoring was being performed by JPL using the DSN 3-way Doppler data, simultaneously (Ogawa et al. 2008). During the OD campaign by the JAXA and JPL, the JPL OD team were obliged to align the JPL's models as closely as possible with the JAXA models. Table 5 briefly summarizes the models employed in the OD by the JAXA and the JPL: spacecraft epoch states, a SRP bus model, momentum wheel de-saturation impulses, and stochastic non-gravitational accelerations (Haw et al. 2008).

The total support duration for JPL was ended when the spacecraft was in a stable, low lunar orbit, at the beginning

Table 6. LOI delivery uncertainty 3sigma (Haw et al. 2008).

	Requirement	Prediction (pre-launch nominal)	Actual delivery
Altitude (km)	99 ±10.0 (km)	98.9 ± 3.0	95.9 ± 1.2

of its science phase which took about 35 days. Over 35 days, JPL navigation team generated 21 OD solutions for the project and 16 independent, internal DSN solutions. Haw et al. (2008) addressed that the JAXA flight dynamics team participated in this works equally during the cruise and circulation phases, and they were fully competent practitioners of OD. Also, they emphasized that JPL's contribution to the mission was, mostly, an operation to manage and reduce risk. Indeed, the performance requirement levied by JAXA before launch was to deliver Kaguya with a perilune altitude error less than ±10 km in 3 sigma. Pre-launch analysis of JAXA indicated that this requirement would be satisfied by greater than a factor of three and finally, the JAXA flight dynamics team guided the spacecraft at the Moon better than expected by almost an order of magnitude as shown in Table 6 (Haw et al. 2008).

2.3. ISRO's Experiences

2.3.1. Summary on Lunar Missions

On October 22, 2008, India's first Moon mission Chandrayaan-1 was launched from Satish Dhawan Space Centre, Sriharikota, by India's Polar Satellite Launch Vehicle (PSLV) (Vighnesam et al. 2010a). Chandrayaan-1 have the payloads of scientific instruments for the purpose of expanding scientific knowledge about the Moon (Vighnesam et al. 2010c). For the purpose of precise OD, Lunar Laser Ranging Instrument (LLRI) was on board as one of the eleven scientific instruments carried by Chandrayaan-1 (Vighnesam et al. 2009a). The spacecraft was injected into a transfer orbit of 254.4×22,932.7 km with an inclination of 17.9 degree on October 22, 2008 at 01:10:19.081 UTC. The spacecraft was put into the moon's polar, circular orbit of about 100×100 km on November 12, 2008 by carrying out a sequence of five Earth Bound Maneuvers (EBM), a trajectory correction maneuver (TCM), a lunar orbit insertion (LOI) maneuver, and four lunar bound maneuvers (LBM) (Vighnesam et al. 2010c). India plans to launch Chandrayaan-2 around 2017 which will have an orbiter, a lander and a rover (ISRO 2014).

2.3.2. Orbit Measurement system

Precise OD for the Chandrayaan-1 mission for every mission phases were carried out using tracking data comprised of Range and accumulated Doppler measurements along with weather data which were collected from network of tracking stations especially configured for the mission. Tracking data were collected from NASA's DSN, Johns Hopkins University Applied Physics Laboratory (JHU/APL), ISRO's DSN and other non-DSN tracking stations. The tracking of the Launch phase, the Earth transfer orbit phase and the Lunar Transfer Trajectory (LTT) phase up to a slant range of 100,000 km was carried out by the existing 10, 11, 12 m dish antennas at ISRO's network stations. The DSN took over the tracking once the above limit of slant range is approached. During the normal operation phase, the Lunar orbiting phase, the Indian DSN (IDSN) station at Bangalore provided the tracking data (Vighnesam et al. 2006). During the normal mission phase, Chandrayaan-1 was S-band tone ranged from IDSN and JHU/APL ground stations (Vighnesam et al. 2010a). The tracking data acquired from the stations other than ISRO were transferred to the ISRO's Mission Operations Center (IMOC) for the purpose of OD, and appropriate format conversion was made (Vighnesam et al. 2010a) because the tacking data from the ISRO Telemetry and the Tracking and Command Network (ISTRAC) stations were in ISRO's 90-byte format and the data from DSN stations were in CCSDS Tracking Data Message (TDM) format (Vighnesam et al. 2010b). The CCSDS TDM was implemented for the first time in order to exchange ΔDOR between ESA/ESOC and NASA/JPL for NASA's Phoenix mission to Mars, and the second implementation was performed for the ISRO's Chandrayaan-1 mission to include the range data type and the transmit/receive frequency data types (Berry et al. 2009).

2.3.3. Orbit Determination System

To meet the OD of Chandrayaan-1, the ISRO's operational OD program for low earth orbit satellites and geo stationary satellites was updated and validated to be used for lunar missions (Vighnesam et al. 2006, Vighnesam, et al. 2010a). ISRO's main functional aspects of OD system were trajectory generation, observation modeling and estimation. The force models for ISRO's OD system included central body perturbation (Earth and Moon), aerodynamic drag, third-body perturbation (the Earth, the Moon, the Sun, and other planets), and solar radiation pressure (Vighnesam et al. 2010c). For the non-spherical harmonics of the Earth, the Earth Gravitational Model 1996 (EGM96) geo-potential model (70 by 70) was used, and for the atmospheric density computation, Mass Spectrometer Incoherent Scatter data 1990model (MSISE-90) was used. Through the JPL ephemeris, DE405, luni-solar gravitation attraction and solar radiation pressure were computed. When the spacecraft was under the influence of Moon, LP100K lunar gravity model (100 by 100) was additionally adapted. For trajectory propagation, the coupled nonlinear second order differential equations of motion were integrated numerically through Cowell's method with Gauss–Jackson–Merson's (GJM) 8th order method (Vighnesam et al. 2010a). For the measurement modeling in ISRO's OD system, appropriate corrections were made for Chandrayaan-1's tracking range and accumulated two-way Doppler data including spacecraft transponder delay, ground station delay and delay due to the earth's troposphere and ionosphere (Vighnesam et al. 2010b). Indeed, light time correction for range and Doppler measurements were also made during modeling the downlink and the uplink path (Vighnesam et al. 2010a). Finally, the optimal estimates of satellite states were obtained by the weighted least squares technique and the iterative differential correction process in ISRO's OD system (Vighnesam et al. 2010b).

In addition to the ground OD, the Chandrayaan-1 mission demonstrated the real-time orbit determination using four advanced accelerometers for burn calibration. A software called PROCAD (precise orbit computation using accelerometer data) was developed for the Chandrayaan-1 mission to determine the orbit using these accelerometer data. Mathematical models for PROCAD software was very similar to those of ground OD system. By PROCAD software, the Chandrayaan-1 orbit was determined in real time even as the orbit maneuver was in progress and determined orbit immediately after the maneuver ended (Vighnesam et al. 2010c). Vighnesam et al. (2010c) addressed that the real time OD gave a quicker awareness of the mission strategists about digressions that have major implications, like non-nominal injection of a satellite after launch, non-nominal progress of a maneuver, or the wrong orbit into which the satellite has cruised. Other than advanced accelerometers which availed the real-time orbit determination, Chandrayaan-1 carried LLRI. The LLRI was the instrument that made Chandrayaan-1 spacecraft to be capable of making lunar topography measurements with a resolution of less than 5 m. In addition, the LLRI based OD substituted the OD with tracking data in case of either non-availability of tracking data or non-availability of converged state tracking data (Vighnesam et al. 2009a). Vighnesam et al. (2009a) showed that the OD results between using the tracking data and the LLRI data were in very good agreements. For example, at lunar mapping phase, the minimum and maximum differences in semi-major axis were found to be about 7 and 45 m, respectively. For eccentricity, minimum of 0.18e-3 and maximum of 0.90e-5, and for inclination, minimum of about 0.0092 degree and maximum of about 0.2352 degree were achieved.

Table 7. Summary of comparison of OD solutions by the ISRO and the JPL (Vighnesam et al. 2009b).

Phase	Epoch(UTC)	Position difference (m)	Velocity difference (m/s)
Launch	2008-10-22 02-58-54-818	465	0.068
EBM-1	2008-10-23 08-00-00-000	58	0.010
EBM-2	2008-10-25 02-00-00-000	280	0.160
EBM-3	2008-10-26 02-08-54-817	126	0.029
EBM-4	2008-10-29 02-13-54-817	70	0.047
EBM-5	2008-11-03 23-43-54-817	74	0.015
LOI	2008-11-08 11-33-54-817	219	0.126
LBM-1	2008-11-09 14-38-54-817	361	0.017
LBM-2	2008-11-10 19-18-54-817	6	0.005
LBM-3	2008-11-11 23-58-54-817	80	0.056
LBM-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

2.3.4. Performance Verification and Validation

ISRO's Lunar Operational Orbit Determination Program (ILOODP) was validated with simulated and live tracking data of the Lunar Prospector (LP) mission and the SMART-1 data before the launch of Chandrayaan-1 mission to cater to the necessary aspect of meeting Chandrayaan-1 OD system requirements during other phases of the mission (Vighnesam et al. 2010a). LP mission's ephemeris was obtained from NASA's Goddard Space Flight Center (GSFC). One day orbit prediction difference between ILOODP ephemeris and GSFC's ephemeris for the LP mission were about 25 m and 4 cm/sec in position and velocity during Lunar Mapping (LM) phase, respectively (Vighensam et al. 2006). In addition, the determined orbit solutions for pre-lunar mapping phase were compared with the orbit solutions obtained from the JPL and almost identical tracking data were used for both orbit solutions. It was observed that differences in orbit solutions were relatively bigger during pre-lunar mapping phase as compared to lunar mapping phase and the maximum differences in position and velocity during pre-lunar mapping phase were within 500 m and 16 cm/s, respectively (Vighnesam et al. 2010a). Vighnesam et al. (2010b) addressed that accurate and consistent OD solutions through the ILOODP system using thrust modeling during lunar mapping phase enabled appropriate maintenance of Chandrayaan-1 till the end of the mission, and the achieved OD accuracy during lunar mapping phase was about 1 km in position which was within the mission requirement. In Table 7, the summary of orbit differences determined by the ISRO and the JPL for Chandrayaan-1 mission during the pre-lunar mapping phase and during the LM phase are shown.

2.4. CNSA's Experiences

2.4.1. Summary on Lunar Missions

China plans a three-phase Moon program: orbiting, landing, and returning from the Moon. In the first phase, a lunar orbiter, Chang'E-1 spacecraft was launched on October 24, 2007, from Xichang in the Sichuan province, China, using a Chang Zheng rocket (Liu et al. 2004). Chang'E-1 was the first lunar exploration mission of China. After three orbit transfer sequences, Chang'E-1 arrived at the Moon with an apolune altitude of 10,000 km and a perilune altitude of 2,000 km on November 5, 2007. After three lunar insertion maneuvers, Chang'E-1 was finally inserted into a near-polar, near-circular orbit with an orbital height of 200 km (Yan et al. 2010). The second lunar probe of China, Chang'E-2 was launched on October 1, 2010, and after nearly 5 days of trans-lunar journey, Chang'E-2 was captured by the Moon on October 6, 2010, and then successfully became a lunar satellite on a polar, near circular orbit with an altitude of approximately 100 km which is 100 km less than the Chang'E-1's orbital altitude (Li et al. 2012). The China's first lunar lander, Chang'E-3, was launched on December 1, 2013 and successfully soft-landed on the Moon on December 14, 2013 including its rover. China announced that their lunar exploration would be continued through Chang'E-4 to verify technologies for Chang'E-5, and Chang'E-5 is scheduled to be launched in 2017 for the first China's Earth return lunar mission.

2.4.2. Orbit Measurement system

Due to the project cost limit, China has no deep space network, and the operation of China's first lunar exploration project was undertaken by a Unified S-Band (USB) system which was mainly designed for manned space flights. Although the USB system had provided reliable TT&C services for Shenzhou-5 manned spaceship, it had only three antennas which had capabilities of receiving the signal transmitted from 400,000 km away (Liu et al. 2004). In addition, among three antennas, only one station could produce simultaneous range and Doppler data, which hardly reduced the effects of the ionosphere and solar plasmas (Liu et al. 2004). In order to meet the OD and prediction requirements of spacecraft and scientific data analysis and also in order to achieve an accuracy of 100 m for OD during the lunar orbiting phase (Yu et al. 2005), network of domestic VLBI antennas were used for the Chang'E-1 mission (Hu & Huang 2009). By combining the USB and the VLBI techniques for lunar capture and mission orbit insertion, the precision OD accuracy, especially of short-arc, was much better than that achieved with the USB alone (Huang et al. 2011). To combine USB and VLBI observables, two way range and range rate data were observed by USB and time delay and delay rate were observed by VLBI (Hu et al. 2005). For the ground tracking stations, the Chang'E-1 mission was tracked at Qingdao and Kashi USB TT&C stations by using two-way range and range rate with a 12 m antenna at the S-band (2.2 Ghz). Also, the spacecraft was simultaneously tracked by four VLBI stations in China at the X-band frequency (8.4 Ghz) with a maximum bandwidth of 16 MHz. The four VLBI stations were located at Shanghai, Beijing, Kunming and Urumuqi, respectively (Yan et al. 2009). In addition to these tracking stations, ESOC were contracted to support the tracking of Chang'E-1 (Billig et al. 2012). To make this collaboration, Beijing Aerospace Control Center (BACC), the China's lunar exploration project mission control center and ESOC started their own system consistency check nearly two years before the launch of ChangE-1. ESOC was to provide a support for Chang'E-1 based on CCSDS standards and therefore to provide system interoperability without modifying the BACC system and the ESOC system (Billig et al. 2012). CCSDS standard used by BACC and ESOC were as follows: Space Link Extension Return All Frames (SLE RAF) for telemetry, SLE Command Link Transmission Units (CLTU) for telecommanding, SLE OEM for orbit data, and SLE TDM for tracking data (Billig et al. 2012).

ESA's ground tracking support for Chang'E-1 successfully started on November 1, 2007 at 03:35 UTC upon the first reception of telemetry signals through ESA's 35 m deep-space station at New Norcia. Two hours and 39 minutes later, the first telecommand to Chang'E-1 was transmitted via ESA's 15m station in Maspalomas, when the satellite was nearly 200,000 km away from the Maspalomas station. An hour later, the ESA station in Kourou, also successfully received the telemetry and the commands transmitted to Chang'E-1. Billig et al. (2012) addressed that these successful communications marked a major milestone because it was the first time a telecommand to a Chinese spacecraft was transmitted from an ESA station. In addition to the reception of telemetry and telecommands transmitted, the Maspalomas and the Kourou stations also measured ranging and Doppler which were used to determine the location and the direction of Chang'E-1 (Billig et al. 2012).

2.4.3. Orbit Determination System

Through numerous literature survey, it was found that the precision OD of Chang'E-1 was performed at the SHanghai Astronomical Observatory (SHAO). Also, it was concluded that the NASA/GSFC GEODYN II software was used in the Chang'E-1 precision orbit determination analysis (Hu et al. 2005, Hu & Huang 2009, Yan et al. 2009, Yan et al. 2010, Li et al. 2012, Yan et al. 2009, Yan et al. 2010). Hu et al. (2005) addressed that SHAO officially obtained user licenses for GEODYNE II software under a collaborative agreement with NASA. The dynamic models used to process the Chang'E-1 OD included Sun and Earth point-mass gravitation, Earth's oblateness effects, indirect oblateness effects due to Earth–Moon oblateness interactions, solar radiation pressure, lunar non-spherical gravitational perturbation, and relativity effects (the relativistic perturbative acceleration caused by the Sun, Earth and Jupiter on the Moon). In addition to these dynamic models, the maneuvers of reaction-wheel unloading and uploading were accommodated by estimating three-axis accelerations along the radial, along-track, and cross-track directions at the time of the maneuvers (Yan et al. 2010). For coordinate system of the locations of the tracking stations for data processing, ITRF2000 was adopted for Earth-fixed coordinate system. JPL's DE403 planetary ephemeris was used for the ephemerides of the Sun, the Earth and other planets. At the lunar orbiting phase, the inertial coordinate system that were used for orbit integration was the lunar-centered inertial coordinate system of J2000. Also, the lunar-fixed coordinate system was chosen to be consistent with the orientation parameters of the JPL DE403 planetary ephemeris (Yan et al. 2010). For the Chang'E-1 mission, the estimated parameters were the initial orbital elements and three-axis accelerations processed by Bayesian least squares method. Also, pass-dependent biases of the two-way range and range rate and VLBI delay and delay rates were estimated to account for any measurement modeling error, such as frequency offset, and the imprecise modeling of the troposphere and ionosphere (Yan et al. 2010).

Compared to the Chang'E-1 mission, the precision OD results of Chang'E-2 mission were significantly improved by the high quality of VLBI data (Li et al. 2012) although the same tracking, network and OD program (GEODYNE II) were used. In Table 8 and 9, OD and prediction strategy used for the lunar transfer and mapping phases for the Chang'E-2 mission is summarized. Compared to the Chang'E-1 mission, the Chang'E-2 mission considered more orders and degrees of the lunar non-spherical gravitations due to the lower altitude at the Moon (Wang et al. 2012). Wang et al. (2012) addressed that by utilizing both USB and VLBI, the precision OD accuracies of Chang'E-2 were found to be about 26 ~ 63 m in position, and about 0.02 ~ 0.06 m/s in velocity for 100×100 km orbit at the Moon. Moreover, for 100×15 km orbit around the moon, the accuracies of about 49 ~ 82 m in position and 0.04 ~ 0.07 m/s in velocity were achieved.

From the survey of many literatures, the authors believe that China performed the precision OD for Chang'E series missions with GEODYNE II software under a license agreement with NASA. However, it was discovered that SHAO developed its own precision OD software for future interplanetary missions. Cao et al. (2010) showed the precision OD results of MEX mission which was jointly tracked ESA and Chinese VLBI network. In their work, the tracking data of MEX mission were processed

Table 8. The OD and prediction strategy for the lunar transfer phase for Chang'E-2 (Wang et al. 2012).

Parameters	Strategy
The Earth's gravitational field	JGM-3
Solar radiation pressure	Cr=1.4
The third body perturbation	JPL DE403/LE403
The non-gravitational acceleration	Solving
Measurement data	The range and range rate of USB and the delay and delay rate of VLBI
Solution parameter	Orbital elements, Cr, bias of range, the non-gravitational acceleration term

Table 9. The OD and prediction strategy for the lunar mapping phase for Chang'E-2 (Wang et al. 2012).

Parameters	Strategy
The Earth's gravitational field	JGM-3
Solar radiation pressure	Cr=1.4
The third body perturbation	JPL DE403/LE403
The non-gravitational acceleration	Solving
Measurement data	The range and range rate of USB and the delay and delay rate of VLBI
Solution parameter	Orbital elements, Cr, bias of range, the non-gravitational acceleration term

and analyzed through the precision OD software written in the standard FORTRAN 77/90 program language, which was developed by SHAO to meet the China's Mars exploration project requirement. Cao et al. (2010) addressed that their own precision OD program was mainly divided into three modules: (1) numerical integration of orbit and state transition matrix, (2) measurement modeling and related derivatives, (3) statistical estimation of normal equations using least squares estimation. Current measurement modes which can be utilized in their system are 2-way, 3-way ranging, 1-way, 2-way, 3-way Doppler, and VLBI delay and their associated rate (Cao et al. 2010).

2.4.4. Performance Verification and Validation

During the verification and validation phase, CNSA mainly focused on achieving the precision OD accuracy using both measurements from USB and VLBI rather than precision OD system's own performances, since they had already used a performance verified GEODYNE II software. Before the launch of Chang'E-1 spacecraft, CNSA had jointly tracked the ESA SMART-1 with Chinese USB and VLBI network and verified the performances while SMART-1 was on an elliptical orbit around the moon with about 80 degree of inclination. In those stages, CNSA again used the GEODYNE II software to determine the orbit. Joint tracking was performed from May 29, 2006 to June, 2 2006. For the same time period, ESA tracking data from Vilspa and Perth station were also provided for the purpose of confirmation and validation (Hu & Huang 2009). Taking the orbit constructed by ESA as a reference, Hu & Huang (2009) addressed that the predicted Root Mean Square (RMS) position

Table 10. Forces models as well as estimated parameters adapted during the SMART-1 OD using CNSA's network (Hu & Huang 2009).

	Models and parameters
Reference coordinate system	Lunar-J2000
Lunar gravity	JGL165p1, 70 by 70
Ephemerides	JPL DE403/LE403
Solar radiation pressure	Fixed mass-area ratio
Wheel-off-loading	ESA provided
Relativistic effects	Schwarzschild
Initial orbit	ESA predicted
Estimated parameters	6 orbital elements + SRP factor + biases

error was about 250 m and the velocity error was about 15 cm/s by two agencies' precision OD strategy. In Table 10, force models as well as estimated parameters adapted during the SMART-1 OD using the CNSA's network are described.

3. DISCUSSIONS ON TECHNICAL ISSUES

In Section 2, it was found that there existed lots of common aspects on other agencies' development experiences for the planetary OD program. Therefore, the brief but essential development strategy focused on technical issues mainly for the Korea's lunar orbit determination system, is discussed through this section.

3.1 Dynamic Modeling

One of the major facts that must be considered in dynamic modeling for a planetary mission is that the switching of integration center is required during the trajectory propagation, and thus, the definition of arbitrary planet centered coordinate frame is necessary. Switching condition may dependent on the magnitude of the major forces acting on a spacecraft which is again related to the location of the spacecraft with respect to the central body. For a lunar mission, the Moon centered Moon Mean Equator and IAU vector of epoch J2000 (M-MME2000) could be used for the lunar centered inertial coordinate system, and for a lunar body-fixed coordinate frame, the Moon Centered, Moon Mean Equator and Prime Meridian (M-MMEPM) could be adapted. For more details about lunar coordinate frame, readers may refer to works done by Song et al. (2010), and it should be noted that the orientation parameters for these lunar centered frame should be computed consistent with the JPL planetary ephemeris's orientation parameters. To model the Earth centered frame, it seems that the conventional ICRF FK5/J2000 would be appropriate as usual. Based on these reference frames, the accelerations due to the point masses of the Earth, the Moon, the Sun, and other plants bodies could be taken into account. Of course, precise planetary ephemerides, JPL's DE series, should be used which uses barycentric coordinates and TDB as reference coordinate origin and time scale. When the spacecraft is located near the Earth, atmospheric drag with appropriate model (MSISE-90 or Jacchia 1971 etc.) and gravity field of the Earth should be considered with JGM or EGM gravity model. While the spacecraft is under influence of the Moon, appropriate lunar gravity model such as GLGM or LP series also must be accounted with respect to M-MMEPM frame. It is noted that various degree and order of gravity field model may be considered depending on the mission phases and requirements. In addition to these force models, SRP must be included with a flat plate model with penumbra modeling. Most importantly, for a lunar mission, the relativistic perturbations of the Moon caused by the Sun, the Earth and Jupiter should be considered as in the models of the ESOC and the CNSA. Other than these forces, accelerations due to the maneuver burns (either impulsive or finite duration) and small forces due to the reaction-wheel unloading and uploading also should be modeled. For numerical integration of spacecraft's states done in TDB time scale, the coupled nonlinear second order differential equations of motion should be integrated with algorithm that provide variable step sizes and orders, i.e., Nordsieck, Adams-Cowell predictor-corrector, Gauss-Jackson and Runga-Kutta, etc., to guarantee a enough trajectory propagation accuracy.

3.2 Measurement Modeling and Estimation Method

There exist two major essential parts that must be solved as accurately as possible during the measurement modeling process. One is the time scale transformation between UTC and TDB, and the other is coordinate transformation between ITRF and ICRF. As the precise OD for planetary missions is extremely sensitive to errors in the ground station position and its associated time accuracy, the two transformations should be modeled precisely. And the ground station positions are usually measured in an accuracy of a few cm level. The method to transform UTC to TDB or vice versa is well treated in Moyer (2000) and one of the options would be a new concept augmented in ESOC's OD system called "hybrid formulation" for transformations between UTC and TDB. For the transformation of the ground station Earth-fixed coordinates, ITRF, into an inertial solar barycentric system, ICRF, those well known effects could be considered such as Earth's nutation, precession and polar motion based on the IAU theory of nutation with appropriate reduction released. Moreover, during the coordinate transformations, the small extra induced forces like platetectonics and solid Earth tides, etc. could be modeled to secure better accuracy. During the modeling of the uplink and downlink signal path, the media corrections should be made, i.e., the refractions due to the troposphere and the ionosphere, instrumental biases and delays (both in spacecrafts and ground stations) and the effects of charged particles of solar plasma. In addition, it is most important that light time corrections must be made as accurately as possible for the planetary OD which could be done by obtaining the light-time solutions for the observables of interest. It is well known that the light-time solutions for spacecrafts can be obtained both in the local geocentric frame of reference and in the Solar-System barycentric frame of reference. However, for a lunar mission, the Moon is not close enough to use the local geocentric frame of reference, and thus, the Solar-System barycentric frame of reference must be used (Moyer 2000). And the point is that the general requirement of the modeled observation should be approximately an order of magnitude better than the accuracy of the actual measurements for the planetary mission utilizing the DSN as already discussed in Subsection 2.1.3. With dynamic and measurement models established, we could apply an appropriate estimation method to determine the orbit or parameters of interest. Among numerous statistical estimation methods, it seems that SRIF would be a suitable filter for lunar missions, which ESOC, JAXA and ISRO have used, and it is exactly equivalent to the weighted least squares method with superior numerical performance.

3.3 Implementation of Data Format for OD System

In so far, neither detailed tacking network nor measure-ment system configurations for the future Korea's lunar mission has been determined. The authors believe that the configurations for measurement system, i.e., a measurement type, TT&C frequencies including antenna size, a station location, a co-operating foreign agencies' network, and detailed planned tracking schedules etc., would be determined when a detailed mission design study is completed. However, whatever the tracking network configurations are, it is obvious that the tracking of future Korea's lunar mission has to be collaborated with other foreign agencies. Therefore, the interoperability among the systems of each agencies should be considered at the early phase of system design and development, which will eventually reduce the efforts needed to verify and validate our own OD system. Based on other foreign agencies' experiences, adapting CCSDS standards seems to be the best way to maintain measurement data interface format since these formats have been used extensively for navigation purposes already including planetary mission, by NASA, ESA, JAXA, ISRO, and CNSA. Currently, the CCSDS Navigation Working Group has provided 4 standard formats that are related to flight dynamics: (1) Orbit Data Message (ODM), (2) Tracking Data Message (TDM), (3) Attitude Data Message (ADM) and (4) Navigation Data Message / XML Specification (Berry et al. 2009). As this paper focuses on the development of future Korea's own lunar OD program, only ODM and TDM are briefly discussed. The ODM recommended standard specifies three standard message formats for use in transferring spacecraft orbit information among space agencies and commercial or governmental spacecraft operators. Three standard message formats includes the Orbit Parameter Message (OPM), the Orbit Mean-Elements Message (OMM) and the Orbit Ephemeris Message (OEM). Among three standard message formats of ODM, OEM is suitable for data exchanges during lunar or other planetary missions as only the OEM supports DSN ephemeris data format that is applicable to non-traditional object such as planetary lander, rover, asteroids and comets (CCSDS, 2009). Example of OEM for Mars Global Surveyor is shown in Table A2 in appendix A. For more OEM details including file layout, header and metadata descriptions, the readers may refer to CCSDS ODM blue book (CCSDS, 2009) recently released. For tracking data interchange between agencies, the development of TDM had begun late in 2003 and was completed in late 2007. CCSDS TDM was implemented for the first time to exchange ΔDOR between ESA/ESOC and NASA/JPL for NASA's Phoenix mission to Mars, and the second implementation was performed for ISRO's Chandrayaan-1 mission to include the range data type and the transmit/receive frequency data types (Berry et al. 2009). TDM is designed for applications involving tracking data interchange between space data systems. Tracking data includes data types such as Doppler, transmit/receive frequencies, ranges, angles, ΔDOR, DORIS, PRARE, media correction, weather, etc. (CCSDS, 2007). The structure of TDM and contents including detailed header and metadata information are well described in CCSDS TDM blue book (CCSDS, 2007) with numerous examples of data messages including various Doppler observables and ΔDOR observables, which are all essential for a planetary mission. In Table A3, ΔDOR observable TDM example is shown. Recently, CCSDS released ΔDOR Raw Data Exchange Format (RDEF) which is a standard format for use in exchanging ΔDOR raw data among space agencies (CCSDS, 2013). When performing a ΔDOR measurement involving two (or more) agencies, raw ΔDOR data must be exchanged at least once between one of the agencies that has acquired the data and the agency that runs the correlation process and provides the results. Hence, it is noted that not only the ΔDOR data are exchanged during an interagency but also other data such as tracking data messages, including meteor data, and orbit ephemeris messages must be exchanged among agencies which again must be formatted in CCSDS standards for ease of cooperation.

3.4 Verification and Validation strategy

After the successful development (or at the development phase) of OD system for a lunar mission, verification and validation strategy must be established. As already discussed in Section 2, numerous agencies cross -verified their own system performance before routine operation. The authors believe that this verification phase is the most time consuming phase with the highest workload, indeed ESOC had spent almost 3 years to verify their OD system with the JPL. It seems that the processed data by the JPL or ESOC's OD system could be used as a reference for the validation purpose of Korea's future lunar OD system. Live tracking data of previous missions must be obtained, i.e., the LP or the SMART-1 mission, and should be also processed in our own OD system using identical inputs and mathematical models. If Korea's own ground station that has a capability to support a lunar mission is constructed before the main mission scheduled in 2020, joint tracking of some future missions with other agency would provide good opportunity to verify our entire ground system performance as JAXA and CNSA did. For validation accuracies, although they may be strongly dependent upon the mission requirements and also on the mission phases, it must be reminded that most of other agencies' OD solutions are consistent with each other by the accuracy of less than tens of m and cm/s levels.

4. CONCLUSIONS

In this work, a brief but essential development strategy for the future Korea's lunar orbit determination system was discussed. Prior to the discussion, other foreign agencies' (ESA, JAXA, ISRO and CNSA) previous development strategies as well as their efforts to develop the lunar orbit determination system were reviewed first based on currently available literatures. Through reviewing other agencies' development experiences, it was discovered that there existed lots of common aspects not only in detailed technical model (dynamic and measurement model with associated filter) used for the lunar orbit determination system but also in collaborative efforts to verify their own systems' performance. Based on the lessons from their experiences, the preliminary development strategy mainly focused on core technical issues was discussed including dynamic modeling, numerical integration, measurement modeling, estimation method, measurement system as well as appropriate data formatting with regard to interoperability among foreign agencies. In addition, although orbit determination accuracies may strongly dependent on mission requirements and also on mission phases, the accuracies of other agencies obtained in the course of validating the system as shown in this work could be used as reference guidelines for the future validation. Indeed, more profound and sustained researches are required in every technical issues discussed in this work. However, the discussions made through this work will certainly reduce the design effort for a development of the future Korea's lunar orbit determination software. Also, it is expected that further detailed system requirements or technical development strategies could be designed and established based on the current discussions.

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Appendix

Table A1. Organization specified OD programs and their major application areas (Vetter 2007).

Organization	Software program	Primary application
Aerospace Corporation/USAF	TRACE	Operational OD evaluation and covariance analysis
Charles Stark Draper Laboratory	DSST DGTDS	Precision semianalytical OD technique POD
APL	OIP/ODP	Transit Doppler post-processing OD used in the 1960s through the 1980s
MICROCOSM	MICROCOSM	Commercial software OD package of the NASA GEODYN program www.vmsi_microcosm.com
MIT/LL	DYNAMO	POD, specifically for HEO and GEO satellites
NASA/GSFC	GTDS RTOD	Operational OD for LEO, MEO, and GEO orbits (TDRSS) and lunar and interplanetary orbits Precision real-time OD for onboard spacecraft using Kalman filtering
NASA/GSFC	GEODYN II	POD for geodesy and geophysics
NASA/JPL	MIRAGE	Multiple satellite OD using GPS
NASA/JPL	DPTRAJ	Interplanetary OD
NASA/JPL	GIPSY/OASIS II (GOA)	POD of satellites using GPS, SLR, and DORIS observations
Navy/NSWC	OMNIS/EPICA	GPS precision orbits
Navy/NSWC	Special-K	Operational numerical OD program
Navy/NRL	OCEANS	Orbit studies, covariance analyses, and GPS orbits
SAO	DOI	Used in the early 1960s for OD of Baker-Nunn camera data and development of standard Earth gravity models
USAF/SPACECOM	MCS	GPS operational orbits
USAF/SPACECOM	SPADOC/ SPECTR	Operational numerical OD program used by Shreiver and Kirkland AFBs
USAF/SPACECOM	ASW	Workstation numerical OD program
University of Texas	UTOPIA, MSODP	Precision orbits using GPS, SLR, and DORIS observations

Table A2. OEM example of Mars Global Surveyor with no acceleration and covariance data (CCSDS 2009).

```
CCSDS_OEM_VERS = 2.0
CREATION_DATE = 1996-11-04T17:22:31
ORIGINATOR = NASA/JPL

META_START
OBJECT_NAME          = MARS GLOBAL SURVEYOR
OBJECT_ID            = 1996-062A
CENTER_NAME          = MARS BARYCENTER
REF_FRAME            = EME2000
TIME_SYSTEM          = UTC
START_TIME           = 1996-12-18T12:00:00.331
USEABLE_START_TIME = 1996-12-18T12:10:00.331
USEABLE_STOP_TIME = 1996-12-28T21:23:00.331
STOP_TIME            = 1996-12-28T21:28:00.331
INTERPOLATION        = HERMITE
INTERPOLATION_DEGREE = 7
META_STOP
COMMENT This file was produced by M.R. Somebody, MSOO NAV/JPL, 1996NOV 04. It is
COMMENT to be used for DSN scheduling purposes only.

1996-12-18T12:00:00.331 2789.619 -280.045 -1746.755 4.73372 -2.49586 -1.04195
1996-12-18T12:01:00.331 2783.419 -308.143 -1877.071 5.18604 -2.42124 -1.99608
1996-12-18T12:02:00.331 2776.033 -336.859 -2008.682 5.63678 -2.33951 -1.94687

< intervening data records omitted here >

1996-12-28T21:28:00.331 -3881.024 563.959 -682.773 -3.28827 -3.66735 1.63861


META_START
OBJECT_NAME          = MARS GLOBAL SURVEYOR
OBJECT_ID            = 1996-062A
CENTER_NAME          = MARS BARYCENTER
REF_FRAME            = EME2000
TIME_SYSTEM          = UTC
START_TIME           = 1996-12-28T21:29:07.267
USEABLE_START_TIME            = 1996-12-28T22:08:02.5
USEABLE_STOP_TIME = 1996-12-30T01:18:02.5
STOP_TIME            = 1996-12-30T01:28:02.267
INTERPOLATION        = HERMITE
INTERPOLATION_DEGREE = 7
META_STOP

COMMENT This block begins after trajectory correction maneuver TCM-3.

1996-12-28T21:29:07.267 -2432.166 -063.042 1742.754 7.33702 -3.495867 -1.041945
1996-12-28T21:59:02.267 -2445.234 -878.141 1873.073 1.86043 -3.421256 -0.996366
1996-12-28T22:00:02.267 -2458.079 -683.858 2007.684 6.36786 -3.339563 -0.946654

< intervening data records omitted here >

1996-12-30T01:28:02.267 2164.375 1115.811 -688.131 -3.53328 -2.88452 0.88535
```

Table A3. TDM example for ΔDOR observables (CCSDS 2007).

```
CCSDS_TDM_VERS = 1.0
COMMENT This TDM example contains Delta-DOR data.
COMMENT Quasar CTD 20 also known as J023752.4+284808 (ICRF), 0234+285 (IERS)
CREATION_DATE = 2005-178T21:45:00
ORIGINATOR = NASA/JPL
META_START
TIME_SYSTEM = UTC
START_TIME = 2004-136T15:42:00.0000
STOP_TIME = 2004-136T16:02:00.0000
PARTICIPANT_1 = VOYAGER1
PARTICIPANT_2 = DSS-55
PARTICIPANT_3 = DSS-25
MODE = SINGLE_DIFF
PATH_1 = 1,2
PATH_2 = 1,3
TRANSMIT_BAND = X
RECEIVE_BAND = X
TIMETAG_REF = RECEIVE
RANGE_MODE = ONE_WAY
RANGE_MODULUS = 1.674852710000000E+02
RECEIVE_DELAY_3 = 0.000077
DATA_QUALITY = VALIDATED
META_STOP

DATA_START
COMMENT Timetag is time of signal arrival at PARTICIPANT_2.
COMMENT Transmit frequency is spacecraft beacon a OWLT before receive time.
DOR = 2004-136T15:42:00.0000 -4.911896106591159E-03
DOR = 2004-136T16:02:00.0000 1.467382930436399E-02
TRANSMIT_FREQ_1 = 2004-136T14:42:00.0000 8.415123456E+09
DATA_STOP

META_START
TIME_SYSTEM = UTC
START_TIME = 2004-136T15:52:00.0000
STOP_TIME = 2004-136T15:52:00.0000
PARTICIPANT_1 = CTD 20
PARTICIPANT_2 = DSS-55
PARTICIPANT_3 = DSS-25
MODE = SINGLE_DIFF
PATH_1 = 1,2
PATH_2 = 1,3
TRANSMIT_BAND = X
RECEIVE_BAND = X
TIMETAG_REF = RECEIVE
RANGE_MODE = ONE_WAY
RANGE_MODULUS = 1.674852710000000E+02
RECEIVE_DELAY_3 = 0.000077
DATA_QUALITY = VALIDATED
META_STOP

DATA_START
COMMENT Timetag is time of signal arrival at PARTICIPANT_2.
COMMENT Transmit frequency is reference for 2-station interferometer.
VLBI_DELAY = 2004-136T15:52:00.0000 -1.911896106591159E-03
TRANSMIT_FREQ_1 = 2004-136T15:42:00.0000 8.415123000E+09
DATA_STOP

META_START
TIME_SYSTEM = UTC
PARTICIPANT_1 = DSS-55
PARTICIPANT_2 = DSS-25
DATA_QUALITY = VALIDATED
META_STOP

DATA_START
CLOCK_BIAS = 2004-136T15:41:00.0000 -4.59e-7
DATA_STOP
```