Space mission planning and operations

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Indian Space Research Organization successfully carried out more than 40 space missions, during the last four decades, in the area of space sciences, spacecraft technologies, space applications and launch vehicle technologies. Mission planning and mission design are the most important multi-disciplinary elements in realizing these objectives. The present article describes the methodology followed in mission planning and mission design for different types of space missions of ISRO, bringing out the trade-off and inter-disciplinary nature of these activities. Important aspects of mission planning in the lift-off phase, the atmospheric and exo-atmospheric phases of a launch vehicle are described. A number of conflicting mission constraints have to be addressed in designing an optimal and robust mission profile for the launch. In addition to reducing aerodynamic and structural loads, considerations related to flight safety and ground station visibility are important parameters. Special launch considerations are to be addressed in designing trajectories for lunar or planetary missions. Launch phase would extend up to injection of the spacecraft in the desired orbit. Thereupon, the spacecraft mission operations are divided into initial phase and operational phase. Initial orbital phase operations include, orbit raising and acquisition operations, and testing of the payloads. The operational phase of spacecraft mission covers, satellite health monitoring and maintenance, station keeping, special operations on the spacecraft, trend analysis and any contingency operations, all leading to gainful utilization of the satellite. A number of mission operations also have now become imperative for mitigating the detrimental effects of ever increasing space debris.

Keywords: Spacecraft operations, launch vehicle mission, mission constraints, lunar trajectory design, ground stations, control centre, space debris mitigation.

SPACE mission planning is a complex and multi-disciplinary activity, which spreads from definition of mission objectives to the end of mission life. The space missions can be broadly categorized into: Science and exploration, applications, technology development, and planetary missions. The objectives of a specific mission in any one of these categories depend upon the requirement. A typical set of mission objectives can be:

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Science and exploration

- Study of gamma ray radiations for exploration of universe.
- Multi wavelength studies of different types of cosmic sources

Applications

- Provision of satellite communication or TV broadcasting in a specific coverage area with the required satellite transmit and receive capabilities
- Remote sensing of the Earth either with optical or microwave sensors
- Provision of navigation services to the required precision and accuracy

Technology development

- Proving of technologies for complex future missions like lifting reentry vehicles, air breathing vehicles, or demonstration of re-entry thermal and recovery management.
- Evaluation of long duration performance of life support systems on the surface of moon or other planets

Planetary missions

- Orbiting of spacecraft in lunar or planetary orbits
- Operation of rovers on moon and planets
- Study of gaseous and magnetic properties of planets

The above set of specific mission objectives cover a broad range, and each of them demands a specific approach for mission planning and operations – like a lunar mission requires propulsion and trajectory capabilities to escape Earth's gravity, a planetary mission may need deep space communication capabilities to obtain the data from the satellite, etc. Mission objectives also call for a specific type of orbit, and the launch vehicle capability to inject spacecraft into those orbits. Low earth orbit (LEO), at an altitude between 400 and 800 km above earth, is used for the earth remote sensing missions. Geostationary orbit (GSO) at an altitude of 35,786 km from earth surface with 0° inclination (i.e. in the earth's equatorial plane) is used for the communication satellites. Most of the satellites providing navigation services are orbited as a constel-

Table 1. ISRO's space missions under various categories

Mission category	ISRO's satellites	Orbit	Launch vehicle used	Remarks
Technology demonstration	Aryabhatta	569 × 619 km	INTERCOSMOS, USSR	Technology/science
	Bhaskara 1	357 × 512 km	INTERCOSMOS, USSR	Technology/remote sensing
	RSD1	183 × 426 km	SLV 3	Technology/science
	Apple	GTO	Ariane 3	Technology/communication
Science	SROSS-C2	450 km circular	ASLV	Two payloads are scientific payloads
	GSAT-2	GSO	GSLV	
Applications – Remote sensing	IRS series, TES, Cartosat, Resourcesat, Oceansat	600–800 km Sun synchronous polar orbit	PSLV	Ground swath, resolution, and number of bands of imaging vary in different satellites
Applications – communication	INSAT series, GSAT series	Geosynchronous orbit (GSO)	Ariane (Europe) GSLV Mk-II	Number of transponders, frequency of operation, and ground coverage vary for different satellites
Applications – meteorology	INSAT-2E, INSAT-3A and Kalpana 1	Geosynchronous orbit (GSO)	Ariane (Europe) PSLV	
Planetary missions	Chandrayaan-1 (Planned in the immediate future)	Lunar orbit	PSLV	Earth escape from highly Elliptic Parking Orbit
Science missions	Astrosat-1 (Planned in the immediate future)	650 km circular orbit	PSLV	Astronautical observatory with monitoring in various bands and energy levels
Application navigation	Indian Regional Navigational Satellite System	GSO and GEO (with inclination)	PSLV/GSLV Mk-II	Planned during 11th Five Year Plan

lation in the orbits at altitudes of the order of 20,000 km from earth surface. The lunar and planetary missions require trajectories, which need to be optimized to the capability of the existing propulsion stages. Most of the satellites to image other planets are inserted into orbits around the intended planet at an altitude mostly determined by gravitational force and the density and composition of the atmosphere of the intended planet. As the range of these orbits greatly varies, the launch vehicles used to orbit these satellites also vary in their payload carrying capabilities.

A number of space missions have been carried out in the Indian space programme, and the missions so far, mostly covered the applications category, with a few missions also catering to the science category.

Indian space missions

The Indian space programme so far addressed experimental and operational missions in the fields of scientific satellites, earth imaging remote sensing satellites, communi cation satellites, and meteorology satellites. Missions in the near future also include lunar-orbiting satellite, regional navigational satellite system, and scientific missions to study some of the cosmic sources. So far, a total of 40 satellites of ISRO have been successfully orbited. These missions used both procured launches and indigenous launches. Twenty-one of these satellites were orbited using the launch vehicles of the erstwhile USSR, Europe, and USA. The remaining were launched and orbited successfully by the Indian launch vehicles. The different launch vehicles of ISRO – SLV-3, ASLV, PSLV and GSLV – had been used to achieve these spacecraft missions. A more powerful launch vehicle GSLV Mk-III is being developed to further enhance the indigenous capabilities and to orbit satellites with enhanced mission objectives. Some of the selected missions of ISRO, their orbits, and the launch vehicles used are given in Table 1, classifying them into their respective mission category.

Important aspects of mission planning

The space mission planning covers initial configuration studies, mission and trajectory design, pre-launch preparations of launch vehicle and satellite, and operations throughout the mission life. The spacecraft mission planning is aimed at maximizing the payload mass, maximizing the onboard power generation required to power the payloads and to maximize the on-orbit life of the spacecraft. The available technologies of subsystems, on-orbit reliability and the orbit injection capabilities of the launcher are the constraints or system drivers for the mission planning. The long duration planetary missions should take care of severe requirements on spacecraft sub-systems with respect to their thermal and leakage performances. The mass budget, power budget and operational modes of the spacecraft are optimized, while meeting the mission requirements within the overall constraints.

Some of the important phases of mission planning activity are:

- The pre-liftoff phase of launch vehicle and satellite, where all final values of the parameters are loaded into onboard computers.
- Studies, design and simulation of the most important atmospheric phase of the launching, namely, from launch vehicle lift-off to clearance of atmosphere (roughly 120 km altitude).
- Exo-atmospheric phase of launch vehicle trajectory, which includes orbital injection of the satellite.
- Satellite operations, which are divided into initial phase and operational phase. Initial orbital phase operations include evolution of orbits of multiple satellites injected in the same mission, orbit raising operations, and in-orbit testing of the payloads. The operational phase of spacecraft mission covers satellite monitoring and maintenance, special operations on the spacecraft, operation and utilization of the payloads.
- Passivation and disposal of final spent stages and the spacecraft at the end of their mission life to avoid space debris.

The design, development and launching of a launch vehicle involve strong interaction among aerodynamics, structures, thermodynamics, flight mechanics, navigation-guidance-control, propulsion, etc. The designs are validated through simulations and testing of the sub-systems on the ground. Pre-launch check out of the completely integrated launch vehicle constitutes the most important phase in mission planning. This pre-launch check out includes a number of operations – specifically related to propulsion, navigation and avionics systems – which have to be carried out only in the final phase before lift-off of the launch vehicle.

Launch vehicle performance in the atmospheric phase

A number of parameters influence launch vehicle ascent trajectory in the initial atmospheric phase. The aerodynamic loads on the launch vehicle vary as the dynamic pressure, which depends on the atmospheric density and the launch vehicle velocity, builds up. The aerodynamic coefficients, which determine the loads, are derived through wind tunnel tests and Computational Fluid Dynamic (CFD) simulations. The coefficients depend largely on the external configuration of the launch vehicle. After the lift-off, the vehicle steadily builds up velocity, crosses transonic region, goes through maximum dynamic pressure region, and finally clears the atmosphere. Further, if the vehicles have multiple motors that burn simultaneously, the jets may interact as they expand on entering higher altitudes. Such jet interactions can cause complex reverse flow and base heating. Hence the design of atmospheric phase of the launch vehicle trajectory is very important from aerodynamic, structural and thermal considerations.

Trajectory design involves finding out an optimum path, from the starting point (launch pad) to the final destination (the injection point). There can be an infinite number of paths joining the two points. One should select the best path that will consume minimum fuel for a given payload or will give maximum payload for a given propulsion system. While choosing the optimum path, there are a number of constraints. One has to abide by the international safety requirements relating to the normal planned impact of spent stages, and also relating to the impact of the stage in case of any malfunction. These are called Instantaneous Impact Point (IIP) constraints. There are also visibility constraints for the trajectory, especially all the critical events like, separation and ignition of stages, injection of the satellite which should be visible from tracking stations. For a rocket to achieve the mission, it is often necessary continuously to change the thrust orientation. This is known as the steering programme. These operations have to be done very gently, and also limiting the total angle of attack during the atmospheric phase. Large angles of attack produce large loads on the structure, leading to, in extreme cases, breaking up of the vehicle. Optimizing the trajectory therefore involves, choosing a path that satisfies all the above conditions and also maximizes the performance.

PYOPT (*P*itch and *Y*aw Trajectory *Opt*imization) software is used to design the steering programme and the trajectory performance[1]. In the scheme adopted in PYOPT, an attempt has been made to model some of the constraints in the system dynamics itself, while the rest are left for the optimization algorithm. This scheme is equivalent to making a transformation on the dynamics in order to make a highly constrained problem a much less constrained one, by allowing the system dynamics black box to address some of the constraints. Further it is assumed that the control variables are parameterized, so that the problem is to find a finite number of control parameters instead of a continuous control function.

Figure 1 shows the process of steering programme design using PYOPT. There are a number of mission constraints to be met for this optimal trajectory design. Some of the major constraints are the following:

- Vertical rise time selected to clear launcher and then pitch down manoeuvre starts

- Gravity turn starts at appropriate $Q\alpha$ values to maximize the payload
- Wind biasing adopted to reduce loads during atmospheric regime
- Stage separation dynamic pressure should be sufficiently small
- Stage impact point constraints to be satisfied
- IIP constraints to be satisfied
- Heat shield separation altitude should satisfy thermal constraints
- Trajectory dipping constraints of any low thrust stage to be satisfied
- Support orbit size constraints
- Tracking stations and visibility-related constraints
- Perigee and apogee altitudes and argument of perigee constraints to be satisfied.

Visibility considerations from tracking stations

Visibility from the Down Range Station Network (DRSN) is required for real time monitoring of the flight, to get data from the vehicle and to issue commands from the ground, if required. The tracking station locations are to be planned in such a way that all the critical events like stage ignition, stage burnout, stage separation, strap-on separation, heat shield separation, spacecraft injection, etc. are visible at least from one of the tracking stations. Continuous coverage is desirable with possible exception during a long coasting phase, where some gap in visibility may be acceptable. Hence tracking stations are generally planned such that adequate overlap of coverage from successive stations exists. However, in case of no visibility, the trajectory should be designed in such a fashion that there are no critical events during that segment. This might have some implication on the mass of the payload that can be carried. For such regions, where visibility does not exist and no critical events would occur, data storage and delayed transmission will be planned. In case of trajectories, wherein critical events occur during some time in flight, where visibility from ground-based tracking stations is not available, ship-based tracking can be planned. During trajectory design the visibility parameters from a tracking station namely elevation, slant range, antenna azimuth, range rate and aspect angle are computed. Range information is used to design telemetry, tracking and command systems – both onboard and ground-based. Aspect angle is used to determine whether there could be any loss in signal due to rocket exhaust, coming in the path of the signal. Elevation angle of 5° is preferred during acquisition of signal (AOS) from a ground station and the antenna usually can track the vehicle up to 2° elevation angle. However, a minimum of 2° elevation angle from AOS to LOS generally gives satisfactory visibility. Sometimes, the topography of the land around the ground station may create certain blockages for visibility at low elevation angles. In such cases, visibility will be improved by positioning a mobile terminal at a vantage location or by redesigning the trajectory with possible implication on the payload and performance. Further, visibility coverage is planned for about 100–600 s after the spacecraft injection for monitoring the passivation and re-orientation manoeuvres of the separated final stage if any, and for Preliminary Orbit Determination (POD) of the spacecraft. Range, range rate, elevation and aspect angle information is useful for POD. Figure 2 gives typical visibility information

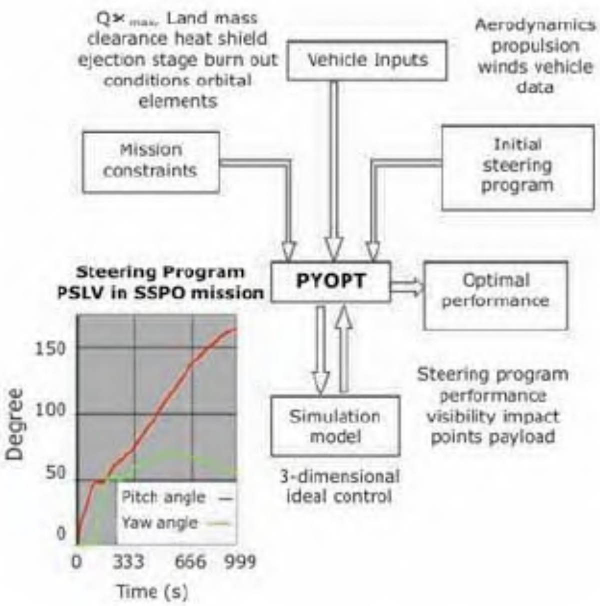

Figure 1. Steering program design using PYOPT.

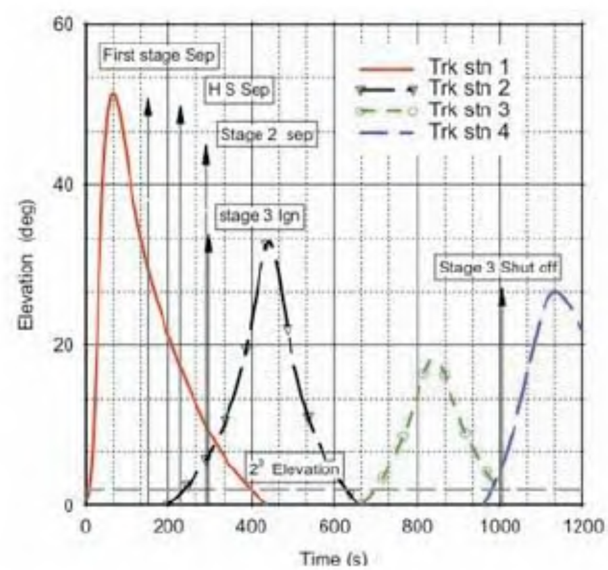

Figure 2. Typical visibility coverage for the ascent phase of launch vehicle flight.

in the form of flight time vs elevation angle, for a chosen set of ground stations.

The launch and injection orbits for lunar missions are different, compared to the missions of earth orbiting satellites. The direct ascents to moon from different parking orbits are possible and were also used earlier by USA and former USSR. The trajectory for the case of Indian Lunar mission is carefully selected, after a number of studies. Next section describes briefly the characteristics of transfer trajectory for a moon mission.

Targeting the moon

Going to moon is a natural continuation of the exploration of planet Earth. There is a continued and substantial scientific interest in the exploration of moon. Trajectory design and carrying out manoeuvres to achieve the desired lunar trajectory, minimizing the fuel requirement is an important aspect of mission planning. During its travel, lunar spacecraft is essentially subjected to the gravity fields of the Earth and the Moon. To generate the transfer trajectory characteristics for translunar injection (TLI), a three-body problem is to be solved, which is not amenable to closed-form solutions. Many approximate techniques and algorithms exist to generate the lunar transfer trajectory characteristics. They are based on point conic, patched conic or pseudo conic techniques and they provide quick data for preliminary mission design and analysis. Achieving a specified lunar parking orbit altitude and inclination accurately is the key to the success of a lunar mission. The translunar injection conditions/Earth parking orbit characteristics are to be chosen such that the resulting trajectory will end up with specified target conditions. In reality, the lunar craft will undergo perturbations, due to non-spherical gravity fields of the bodies. The transfer trajectory will deviate from its expected path and fail to achieve the target accurately, if these perturbations are not considered in the trajectory determination process. Mainly, the asphericity of the Earth causes these deviations in the neighbourhood of the Earth. The only known way to find the precise translunar injection characteristics accounting these perturbations is by search and by numerically simulating the trajectories. The methodologies of such precise lunar trajectory design are described in refs 2, 3.

Earth–moon distance is around 400,000 km. So the translunar orbit should have an apogee altitude of 400,000 km. Three different strategies are followed. The direct transfer is the traditional way to go to the moon. In this approach, the mission begins from a circular parking orbit around the Earth, usually with an orbital altitude in the range of 180–300 km. Then, with a substantial velocity addition of more than 3000 m/s, the spacecraft is placed in the Lunar Transfer Orbit, with the apogee at moon's distance. Duration for this transfer is around five days. Further, burning of the on-board liquid rocket motor is required at the moon, to enter a lunar orbit. All lunar missions from the 1960s to the 1980s used this traditional approach. The second way is to go to an elliptic parking orbit (EPO), say a GTO of 200 × 36,000 km and then use a spacecraft integrated liquid apogee motor (LAM) to raise the orbit to moon. Since, the energy of Geostationary Transfer Orbit (GTO) is considerably higher than that of a LEO, this approach leads to substantial savings on the trans-lunar insertion requirements. Hence, a LAM integrated with the lunar spacecraft can do this transfer; whereas, in the first approach, an additional stage will be required. A third and perhaps the most energy-conserving way is to use the weak stability boundary concept by going to the regions of Lagrange points in Earth–Sun system and return to the moon by a very small velocity addition. However, the disadvantages of the weak stability boundary method are that (a) the time taken to achieve lunar orbit is several months (instead of a few days in the other methods), (b) the spacecraft is required to traverse long distances of the order of four times the lunar distance, and this puts a very high burden on the communication power requirements. All these approaches are explored in the design of the lunar mission with our launch vehicles to arrive at the best possible solution for the Indian Moon Mission. The typical mission profile is given in Figure 3.

Spacecraft operations

The operations on satellite are taken over by the spacecraft operations team, after the satellite is injected into the orbit by the launch vehicle. The spacecraft operations are usually split into initial phase and operational phase. The initial phase operations cover orbit raising/changing of the orbits to completion of the testing of payload. The following sections cover the initial phase and operational phase of communication satellite operations.

Orbi-raising operations

The satellites meant for communication and broadcasting usually operate from geostationary orbits. Injection of the satellite directly into Geostationary Orbit (GSO) requires a lot of energy from the launch vehicle. Hence, launch vehicles usually orbit the satellites into an intermediate orbit called GTO. The orbit injection usually takes place near the perigee point.

Perigee altitude (nearest distance to the Earth) of the GTO is in the range of 180–650 km for different launch vehicles. Apogee altitude (the farthest distance to the Earth) is in the range of 36,000 km. Inclination depends on latitude of the launching base. For the launches of ISRO satellites, the inclination will be 7°, if launched from Kourou by Ariane Launch Vehicle or 19°, if launched from Sriharikota launching base using GSLV/PSLV.

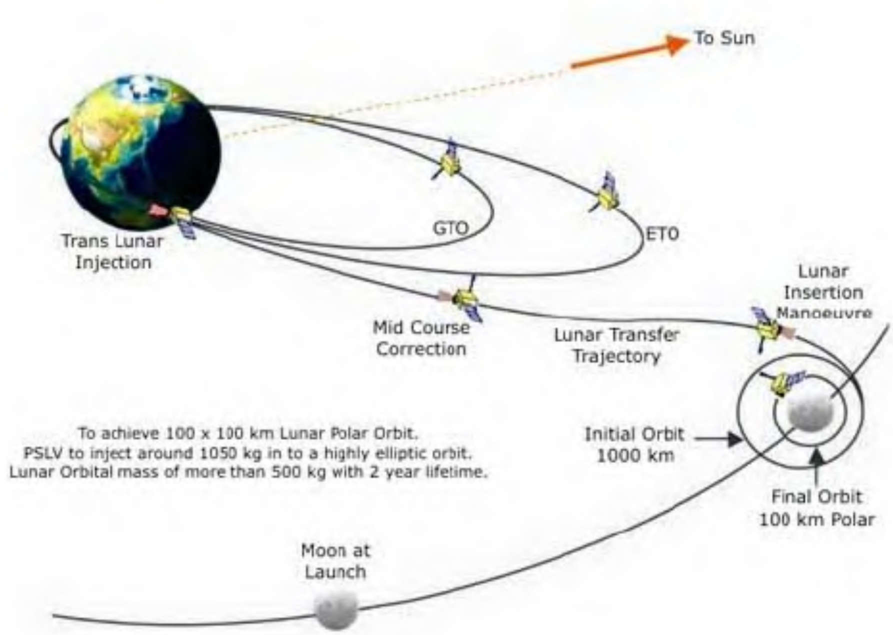

Figure 3. Typical lunar mission profile via highly elliptic transfer orbit.

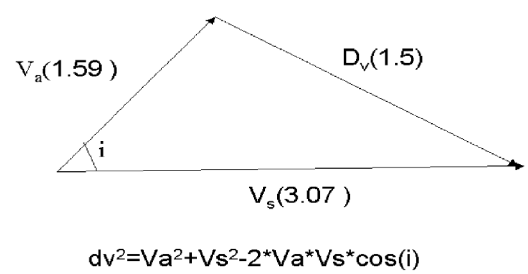

Figure 4. Vectorial representation of velocity to be added for realizing geostationary orbit from geostationary transfer orbit.

Satellite in GTO will have an orbital period of roughly ten-and-half hours, and hence will be drifting at high drift rate. The period of the satellite can be calculated if the semi-major axis (a) is known. The equation is[4]:

$$P = 2\Pi\,(a^3/\mu)^{1/2}\ \text{s}$$

a in km and μ is the universal gravitational constant in units of km^3/s^2.

The visibility from the Master Control Facility during GTO will not be continuous. Hence, a network of stations around the globe is necessary to have continuous telemetry data from the satellites.

In every revolution around the Earth, the satellites in GTO will be crossing Van Allen belts, which are in the altitude range of 5000–7000 km and 20,000–27,000 km above Earth. These belts contain charged particles and can cause charge/discharge of the satellite. To minimize these effects, in the nominal plan orbit-raising operations are to be planned in the first few days of launch.

The geostationary orbit is defined as a circular orbit at an altitude of 35,786 km above surface of the Earth and in the orbital plane of Earth's equator. This means, eccentricity e, and inclination i should be equal to 0. A satellite in such an orbit will have an orbital period of 23 h, 56 min, 04 s that is equal to one sidereal day of Earth. The satellite in GSO will move in exact synchronism with Earth, and appears stationary to the point below it on the Earth. In the typical GTO of 200 × 36,000 km, the velocity of the satellite at apogee will be roughly 1.6 km/s. The circular orbit velocity for GSO altitude of 35,786 km is 3.074 km/s. Hence velocity has to be added to the satellite at apogee point. This addition of velocity will raise the perigee to the required GSO altitude. The inclination of the orbit has to be decreased to 0, for the orbit to lie in the equatorial plane. This requires addition of velocity towards north direction at the nodal crossing point, when satellite in the inclined orbit is crossing equator from south to north. Usually the orbital injection parameters are so chosen with the node and apogee point to be very near. This will give an advantage in optimizing the velocity requirement to reduce the inclination. The orientation of the satellite at apogee, before addition of velocity by firing the apogee motor, is optimized, to raise the orbit and decrease inclination simultaneously. The velocity required to be added to attain GSO from GTO can be calculated[4] as is given in Figure 4.

The velocity is added in a few apogee manoeuvres, optimizing for minimum propellant consumption, minimal

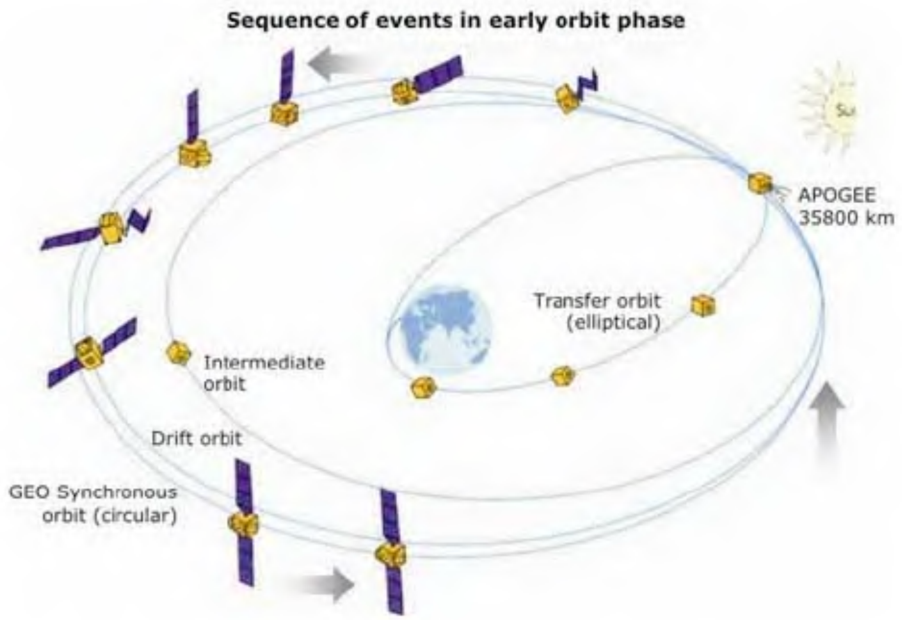

Figure 5. Geostationary transfer orbit to geostationary orbit raising and deployment operations.

attitude and velocity errors, and considering visibility constraints of ground stations. Major part of the inclination correction is carried out in the first apogee motor firing, where the satellite velocity at apogee is low.

The orbit-raising operations and other sequence of operations[5] like solar panel and antennae deployments are shown in Figure 5.

Station acquisition manoeuvres

After the final apogee manoeuvre of the satellite, the satellite will be in an orbit a few hundred kilometers below GSO. This orbit will have an orbital period of approximately 23 h 45 min, and with a longitudinal drift rate of about 2° per revolution. Hence, it will take a few days for the satellite to reach the designated orbital slot depending upon at what longitude final apogee manoeuvre is carried out. The orbit will be raised gradually from this drift orbit by adding small amounts of velocity (of the order of 4–5 m/s total) and the satellite will be finally positioned at its designated orbital slot. These manoeuvres are called station acquisition manoeuvres and they are carried out using small rockets located on the east and west faces of the satellite.

Payload testing

After the satellite reaches the designated orbital slot, the payload of the satellite, consisting of on-board transponders and antennae, is tested for various parameters. These important parameters are: Effective isotropic radiated power (EIRP), saturated flux density (SFD), cross pole isolation (XPD), on-board system G/T, noise floor spectrum, frequency response, third order intermodulation products, group delay, frequency translation; and antenna pattern measurements.

The typical earth station schematic to carry out these measurements are given in Figure 6. The Earth Station consists of uplink and downlink chains with sufficient redundancy. Highly accurate monopulse tracking antennae are deployed for this purpose[6].

The antenna pattern of the ground antenna should conform to the International Telecommunication Union's (ITU) recommendation[7] of ITU R 580–5, which is given in Figure 7. If the ground antenna pattern does not conform to this constraint, it will cause interference to the adjacent satellites, due to the power radiated through the side lobes of the pattern.

The satellite is brought into service after completion of initial phase operations, which includes payload testing. The satellite operations from this time up to end of mission life are covered under operational phase of the spacecraft mission.

On-orbit operations

On-orbit operations consist of the following:

- Reception of telemetry data and monitoring the health status of various subsystems through satellite telemetry parameters.
- Archival of telemetry data for trend analysis.

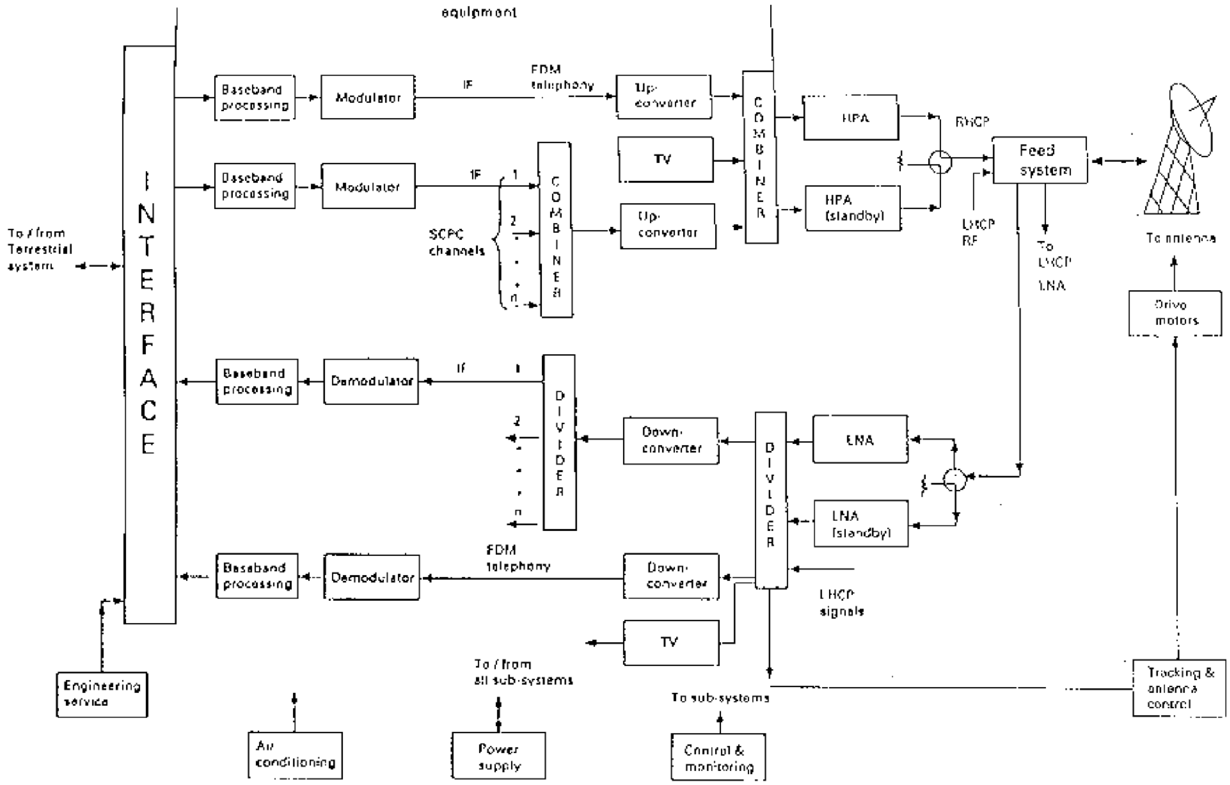

Figure 6. Configuration of earth station required for payload testing.

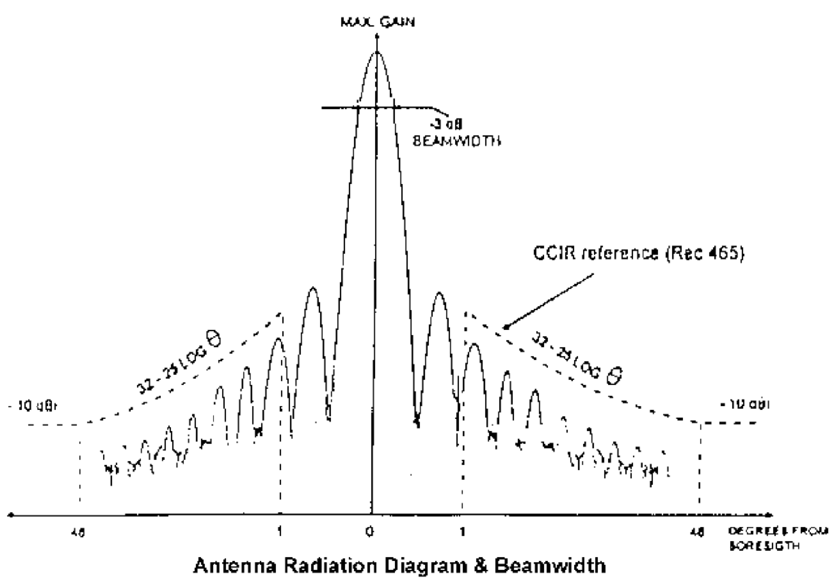

Figure 7. ITU recommended radiation pattern of ground antenna.

- Capability to send commands to the satellite (usually to maintain the required satellite temperature conditions by heater operations, and also to configure the on-board payload as per user requirements).
- Ranging of the satellite for orbit determination.
- Periodic north south station keeping manoeuvres to maintain the orbital inclination within the limits ±0.1°.

Table 2. Typical activities of spacecraft health monitoring and control of communication spacecraft

Operation/action	No. of instances/parameters	Frequency of action	Remarks
Tracking	Two beacons	Continuous	Level fluctuation is alerted by computers
Telemetry monitoring	About 1000 parameters both analog and digital	Continuous	Rate of data reception is 1 kbps
Telemetry limit fixing for alarm indications	About 900 parameters	Once in a week	Database updated after trend monitoring
Telemetry limit fixing for critical alarm indications	About 200 parameters	Seasonal and for special operations	Database updated after trend monitoring
Tele-commanding	60 commands in a day (Heater related)	2 to 3 per hour	Latest spacecrafts have on-board automatic temperature controller
Attitude maintenance	10 commands	Data driven	Once in two weeks on need basis
Orbit adjustment commands	60 commands in a day	At proper time of orbit	Once in three weeks on need basis
Battery charge maintenance	15 commands in a day	Data driven	During eclipse season for 90 days in a year
Payload OBA setting requirement	2 to 3 commands	Data driven and specific use of transponder	As needed by user
Ranging	Multi-tone samples	At 10 min intervals for 48 hours and once in 10 days	Data pre-processing, Orbit determination and planning activities are done by a special team of engineers
Payload health monitoring	3 to 4 parameters per transponder	Continuous	Critical for payload use. Stringent limit monitoring to get immediate alert for corrective action
Pre-manoeuvre operations	100 commands	Special activity done once in 45 to 60 days	Monitored by sub-system engineer in real time
Special operations (Battery reconditioning)	30 commands	Special activity done prior to eclipse season	Monitored by sub-system engineer in real time

- Periodic east west manoeuvres to control the drift and to maintain the satellite at the designated longitude within ±0.1°.
- Management of pre-eclipse conditioning of batteries and management of loads during eclipse seasons of various satellites.
- Regular monitoring of the traffic and solving the problems faced by the users.
- Relocation of the satellites to other orbital slots, if required.
- Re-orbiting of the satellite to an altitude 300 km above GSO, at the end of the mission life.

Regular health monitoring and control

The health of the spacecraft is inferred through the regular reception of telemetry data from the concerned spacecraft. This data is analysed on a day-to-day basis to determine the status of the various sub-systems, and to decide on the necessary commanding operation. The telemetry data received over the entire life of the mission is archived and periodically analysed for long-term trend of the spacecraft performance.

Table 2 gives typical activities on the spacecraft in a day on an INSAT communication satellite of ISRO in GSO. As the GSO satellites are continuously visible from the spacecraft control centre of Master Control Facility (MCF), the reception and archival of the telemetry data needs huge computer memory. A big network of computers for storage and processing of the telemetry data and to carry out multiple spacecraft operations is in place at MCF. Very complex multi-mission data processing and analysis software, developed by ISRO Satellite Center (ISAC), and other analysis and diagnostics software developed by MCF reside in this computer network for operations.

Trend analysis of the data

The spacecraft control centres have unique opportunity to archive the data of any satellite throughout its mission life. This data is very valuable for detailed analysis and providing feedback to the spacecraft and payload design teams. The trend analysis of the data is also used during any contingency occurring on the spacecraft – specifically to analyse, if any gradual degradation of any sub-system had

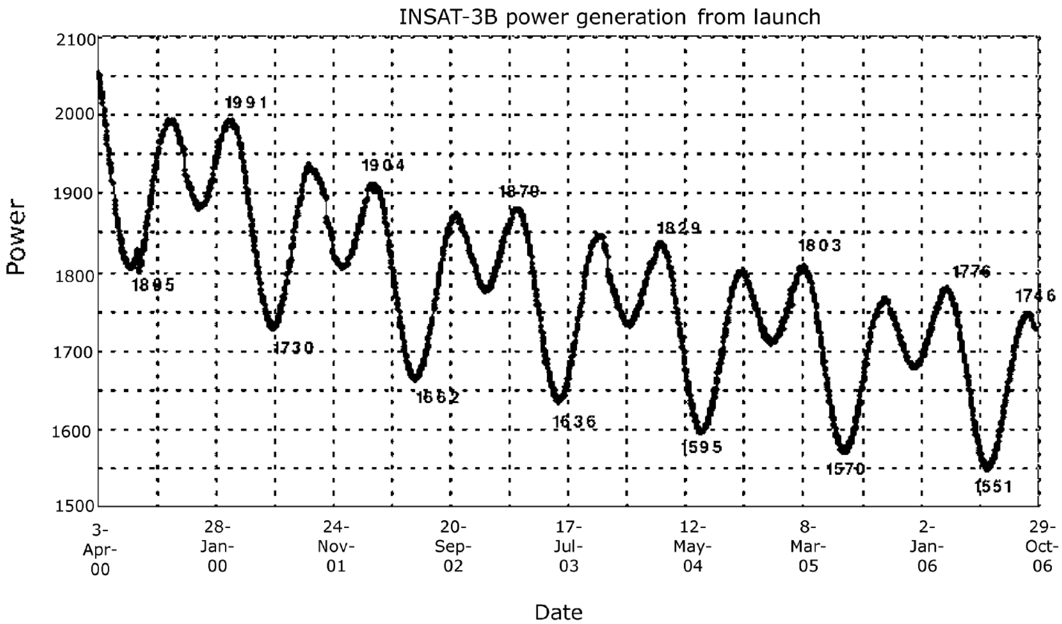

Figure 8 *a*. Trend of generated power of INSAT-3B.

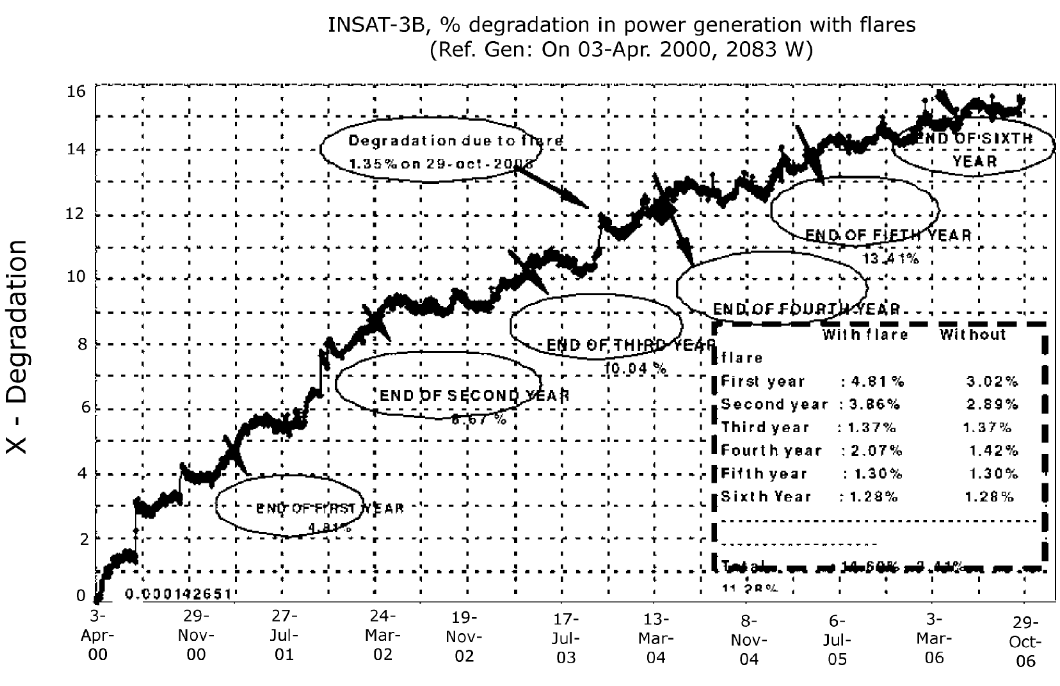

Figure 8 *b*. Trend of degradation in the generated power – INSAT 3B.

caused the contingency. Such a trend analysis of the archived telemetry data is an important and regular activity of MCF and ISTRAC of ISRO for GSO and LEO satellites respectively.

Two cases of trend analysis[8] are given below as typical examples:

Case 1: The electric power required is generated by the on-orbit spacecraft, using solar cells. Different types of solar cells offer different efficiencies and exhibit different long-term degradation effects. The sun's declination and the sun–earth distance influence the power generated. The solar cells gradually degrade in performance, due to the

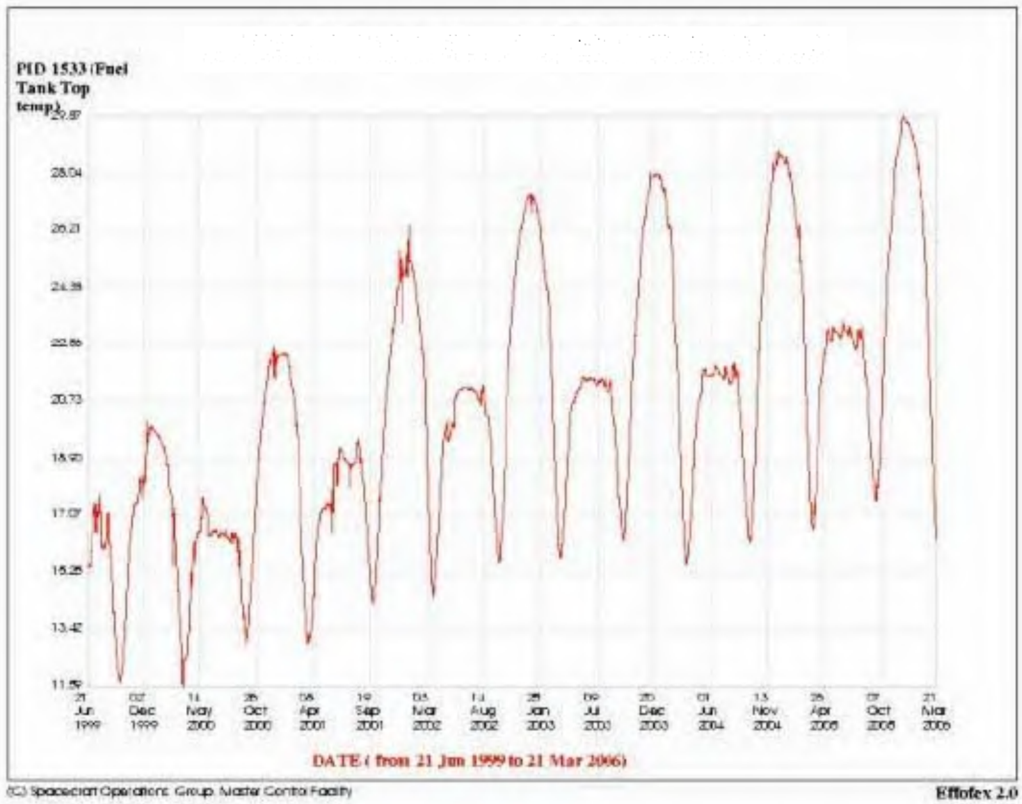

Figure 9. Trend of fuel tank temperature – INSAT 2E.

incidence of solar ultraviolet radiation. The solar flares cause sudden degradation. Figure 8 *a* gives the power generated during the orbital life of INSAT-3B, which uses silicon BSR solar cells in its solar panels. The power generation over the mission period is estimated using the load current and other related solar panel data telemetered. Figure 8 *b* gives the degradation in power generation from the launch time onwards, as percentage of initial reference power generated. Certain sudden degradations could be observed, which were due to big solar flares. The overall average degradation in the power generation, as derived from this analysis, over six years of mission life matches with the prediction.

Case 2: Figure 9 provides the temperature profile of fuel tank of INSAT-2E spacecraft, over the last seven and half years. The seasonal variation of the temperature is due to eclipse, as well as sun–earth distance variations. But the average temperature profile is an indication of variation of thermo-optical properties of the surfaces and variation in the thermal environment due to configuration changes. In this case, the spacecraft load has remained almost constant throughout the mission. The average temperature profile shows the effect of degradation of thermo optical properties of the external surfaces, and matches with the prediction.

North south station keeping manoeuvres

The gravity force from Sun, the Moon and the non-spherical shape of the earth and its mass distribution perturbs the orbit of the satellites in geostationary orbit. The constant evolution of inclination is one of the important effects of this perturbation. The Sun's gravity on average contributes a mean drift in the +X direction by 0.27° per year. The Moon causes a drift of the spacecraft inclination vector that varies between 0.48° per year and 0.68° per year. The contribution to the evaluation of inclination vector due to asymmetry of earth is relatively insignificant. Hence every GSO satellite will experience an inclination variation of 0.75°–0.95° per year depending on the year. A velocity of the order of 50 m/s is required to be added to the in-orbit spacecraft in a year to correct for this inclination change.

Whenever the inclination of the satellite orbit crosses 0.1°, a northward ΔV is added to the satellite velocity to correct the inclination. The schematic of addition of the

ΔV and its effect on the inclination of the orbit[9] are given in Figure 10. If inclination correction is done every time by 0.2°, the inclination correction manoeuvres have to be carried out once in roughly 70 days. More frequent manoeuvres will be called for if satellite or its sub-systems have any restrictions for continuous thruster firing durations.

The on-board propulsion system employed in the INSAT satellites is a unified bipropellant system utilizing MON-3 as oxidizer and MMH as the fuel. The liquid apogee motor used for orbit raising operations, and the bank of attitude and orbit manoeuvre thrusters are supplied with the propellants, from the same on-board propellants tanks.

The elliptic nature of earth and the gravity components due to the non-spherical shape creates a perturbation on GSO satellites on their longitude. This specific feature created two stable points at 75°E, and 105°W; and two unstable points at 1.5°W and 162°E in the equatorial plane. The stable points are at the ends of minor axis of ellipse, which is the real shape of the true equatorial orbit. The satellites at these stable points will stay without any drift in the longitude and GSO satellites at other longitudes will drift towards these stable points – the rate at which they drift depend upon their nominal longitude. The east–west station keeping manoeuvres are required to be carried out whenever the satellites, which are away from the stable points, drift towards stable points and cross the limit of ±0.1° limit on the orbital slot. The highest amount of correction required for east west manoeuvres is 1.8 m/s per year. The periodicity of E–W manoeuvre depends on the longitude of the satellite. In the case of ISRO's satellites, MCF carries out the longitude corrections by firing the thrusters on the east and west faces of the satellite in the pulsed mode everyday.

Eclipse operations

Sun appears as moving in a cyclic fashion with a maximum inclination of ±23.5° with respect to equatorial plane, when we observe it from earth. In this apparent motion, when Sun is in the range of ±8.7° with respect to equator, it casts the shadow of Earth on to the satellites, which are in GSO (on the other side of the earth)[4]. This causes an eclipse and during that period, the solar panels cannot generate the power required by the satellites. The declination and right ascension of Sun with respect to earth's equator are shown in Figure 11.

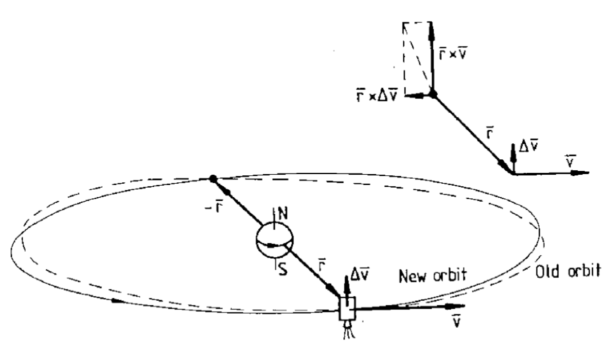

Figure 10. Change of the orbital plane by a north thrust.

Eclipse seasons last for 44 days each, twice in a year centred on 21 March and 21 September. The peak duration of the eclipse occurs on 21 March and 21 September for a duration of 72 min. During the eclipse, the on-board batteries have to supply the DC power required by the satellite bus and the payload. The batteries have to be discharged and recharged before the start of each eclipse season for their efficient operation. During the eclipse period, if the onboard batteries are not sized to maintain the full loads, loadshedding may have to be carried out. The eclipse happens on every day at midnight of Satellite Local Time (SLT). The entry into eclipse and exit from eclipse are also critical for the satellites as they experience large temperature changes at those times. Management of the satellites during eclipse is one of the important on-orbit operations. The Sun, Earth and GSO satellite geometry, which describes the eclipse, is given in Figure 12.

Traffic monitoring and space debris

The communication transponders are leased to the users according to their requirement. The traffic in these transponders is continuously monitored to see that the users do not exceed the power and bandwidth allotted. With the increased traffic on communication satellites, high sensitivities involved in the digital signal processing base band techniques, and due to reduced inter-satellite orbital separation, the users are experiencing a number of interference cases in satellite communications, nowadays. This is an emerging new technical field. ISRO developed good expertise in this area to identify interference problems and solve them to the satisfaction of the users.

When the satellites approach their end of missions, the space debris mitigation measures have to be implemented to minimize the debris in the most-used regions of the space, which are identified as zones to be protected. In fact, space debris mitigation cover all phases of space mission, including the launch phase to avoid generation of space debris during mission operations and also, due to left-over spent stages of the launchers. The activity in the mission planning and operations related to the space debris area are discussed later.

Ground station network and operations for IRS missions

In the case of RS missions, the satellite control, the ground station network is configured to provide adequate

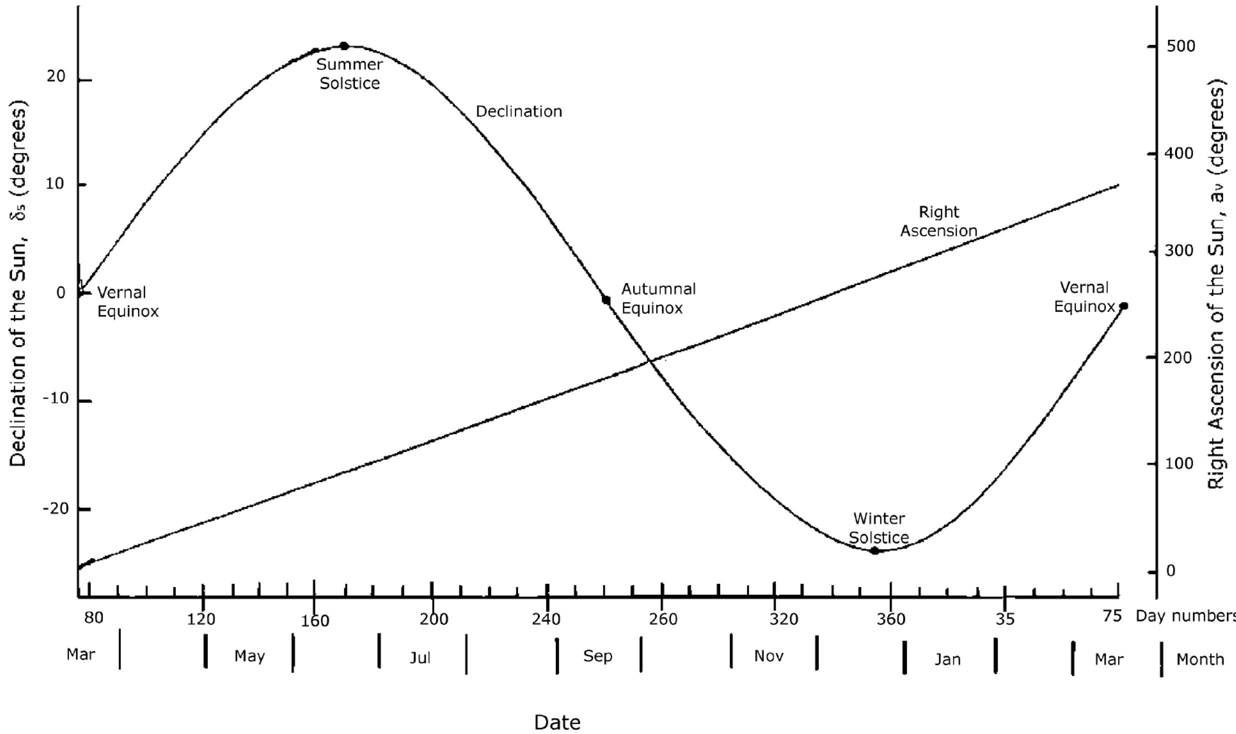

Figure 11. Solar declination and right ascension vs date.

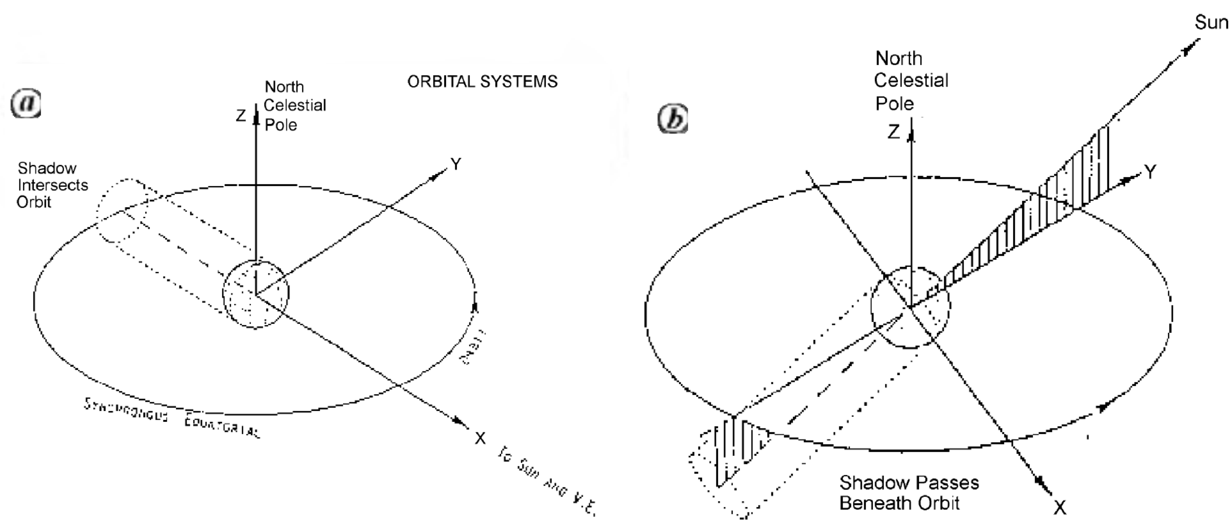

Figure 12. Representation of Earth's eclipse onto the GSO satellites. Geosynchronous orbit geometry ***a***, vernal equinox shadow; ***b***, summer solstice shadow.

coverage in the launch and early orbit phase as well as during the routine phase. The tracking network provides good quantity of range, range rate and angles data to provide orbit information both for prediction generation for satellite tracking and also for image data processing. ISTRAC, which is responsible for TTC and mission control activities, has a comprehensive network of S-band TTC ground stations at Bangalore, Lucknow, Mauritius, Biak (Indonesia), Bearslake (Russia) and Svalbard (Norway), which are used round-the-clock for satellite control operations. All stations are connected to the Satellite Control Centre (SCC) at Bangalore, by dedicated communication links. The satellite tracking ground stations operate under a three-layer architecture which enables remote operations from the network control centre at Bangalore through network control processors (at NCC), Monitor and Control Processors (MCP) and General Equipment Interfaces (GEI) at individual ground stations. In a multi-mission environment, the reconfiguration time of a ground station from one pass to another is reduced to less than 3 min.

Satellite control centre

SCC facilities are mission control, mission analysis and mission computers. Mission software for the purpose of spacecraft health monitoring and control; flight dynamics software for the purpose of orbit determination, orbit events generation, orbit manoeuvres, orbit scheduling and attitude determination run on mission computer network. The data from different ground stations, viz. telemetry, command, tracking, etc. are sent on wide area network by the station computers to SCC through a network monitoring and message passing software.

Payload data collection, processing and dissemination: (i) Payload data collection – The orbital height chosen enables a single data reception station in India located at Shadnagar (near Hyderabad) to perform the data reception and allied functions. A number of antennas at data recaption site enable data collection in multi-mission scenario. The visibility clashes over data reception stations are unavoidable as IRS constellation satellites operate in different altitudes and different local times. Thus, the payload data collection in multi-mission scenario is maximized by drawing up scheduling guidelines and mission rules.

(ii) Processing and dissemination – The payload data reception, level-0 processing, data products generation at various levels, browse processing, data quality evaluation, data archival and dissemination are carried out by National Remote Sensing Agency (NRSA), Hyderabad. More details on payload data processing and image generation are summarized below for the typical case of Cartosat-2 satellite.

(a) Level-0 processing – This is a scheduler-based automated operation which creates two files at the end of each payload data reception session. These two files together will help to produce basic images of standard scene size.

- Raw image data file is created after decryption, RS decoding, decompression and proper time stamping for every line of image data.
- Another file called ADIF (Auxiliary Data Information File) is created with information like satellite position in the orbit, its angular orientation (attitude) in the space while imaging using gyroscopic and earth sensor/star sensor measurements for the time of every image line and other information like sensor biases.

(b) Level-1 processing – This processing applies the radiometric calibration on each pixel data and creates a RAD corrected image. No attempt is made to improve the location accuracy at this level, so the location error may be higher up to 750 m.

(c) Level-2 processing – This level of processing uses improved orbit and attitude knowledge and also generates geo-referenced images in scene-based and area of interest (AOI) based modes. Location accuracy of these images is generally better than 100 m.

(d) Level-3 processing – This level of processing delivers high precision images with location accuracy better than 10 m using the knowledge of some ground control points (GCPs). This also delivers some value added products (images) with value addition like mosaicing of adjacent images seamlessly, merging of high resolution panchromatic images with low resolution colour (multispectral) images and stereo images.

(iii) Data quality evaluation: This is carried out periodically to estimate the performance deviation of the camera especially the detector elements (radiometric calibration) and the location errors in the image registration (geometrical calibration). Feedback is given to image processing and SCC for further correction.

Generic mission planning and ground operations system

The ground resolution obtainable from the imaging systems launched into space onboard IRS ranges from hundreds of metres to sub-metre. The generic mission planning aspects of IRS missions are as follows:

(i) The orbit selection and control for IRS missions: (a) Orbital height – For ensuring near constant illumination, throughout the mission life, in all seasons, with inferquent orbit maintenance corrections, and commensurate with the payload systems technology, particularly related to the trade-off between swath and resolution, IRS missions have been operated in the orbital heights ranging from 562 km to 904 km.

(b) Global coverage pattern – The IRS missions are essentially global in nature. The orbit chosen provides global coverage within a reasonable period. It is 2 days to 26 days for ocean and resource missions respectively. The tilt capability of the camera provided onboard enables a revisit period of 5 days or better for high resolution sys-

tems. A repetivity of 22 days to 26 days for medium resolution, cameras (~20 m), is found to be adequate for applications like crop monitoring. The coverage is enhanced by onboard data recorders, by which global data archive could be built up with proper recording sequences. Further, employing high latitude station like Svalbard, Norway which provides contact with spacecraft in every orbit, it is possible to maximize the remote sensing data return from the satellite.

(c) Phasing – The satellites in the same orbit are phased such that they arrive over the same data reception in a time-staggered manner, which enables data collection from many missions simultaneously.

(d) Selection of node – The node, either ascending or descending, in which imaging is carried out, has a bearing on the launch time, spacecraft systems design and ground stations network in the initial phase operations. The first three remote sensing satellites IRS-1A, IRS-1B and IRS-1C were launched from Beckoner, Russia. To take over the control of the satellite immediately after the launch over the TTC network of ISTRAC, a south bound orbit and hence the descending node was preferred. In the case of PSLV, launches happened towards southward in the daytime due to safety constraints, which also supported the descending node for local time reference. Finally, most of the satellites like SPOT followed the descending node for local time reference. Descending node reference was also favourable to maintain continuity to all the existing users. So, all the Indian remote sensing satellites image the Earth when they move from north to south, viz. at the descending node.

(e) Equatorial crossing time – The equatorial crossing time or local time, which determines, the sun-angle at which the imaging is carried out, the shadow lengths and the cloud cover is determined by the applications scientists. The resource application satellites like IRS-1A, IRS-1C and IRS-P6 (Resourcesat-1) have an equatorial crossing time of 10.30 a.m. from the descending node for better reflectance and cloud-free condition, while ocean applications satellites like IRS-P4 (Oceansat-1) will cross the equator at 12 noon to overcome the poor reflectivity of ocean water and cartography applications satellites have 9.30 a.m. as the equatorial crossing time for better shadow conditions. However radar satellites like RISAT do not depend on any particular local time. But 06:00 a.m. is selected as equatorial crossing time to minimize the eclipse duration and maximize the power generation, as RISAT requires more power. Astronomical satellites like ASTROSAT also do not depend on local time condition. So, near equatorial orbit is selected to avoid high-energy particles over poles and South Atlantic anomaly region.

(f) Frozen orbit – In order to minimize the scale variation in images due to altitude variation, frozen perigee is another feature in the final orbit achieved in IRS missions.

(g) Ground track maintenance – Due to atmospheric drag, solar wind pressure and luni-solar perturbation, the orbital parameters of IRS change and hence mission specifications. Due to change in the semi-major axis, the ground track shifts and therefore the coverage pattern changes which has to be held as a standard pattern to follow the assured path row pattern for the benefit of the user. Ground track maintenance is specified to be at ±1 km from the nominal path pattern. This is achieved by performing in-plane corrections to adjust the semi-major axis.

(h) Local time maintenance – Other effect of orbit parameters change is the decrease of inclination, which affects the equatorial crossing time. The out-of-plane correction, viz. inclination correction is performed once in approximately six months to maintain the local time within 5 min from the normal local time. Another strategy is to bias the inclination to a value such that the monotonically decreasing inclination takes care of decreasing local time and out-of-plane corrections are carried out once in a year or even once in two or more years. The inclination corrections are carried out in a set of well-planned short duration burns of the onboard thrusters.

(i) Orbit determination – The orbit determination system uses tracking data (range, range rate and angles) from S-band TTC stations, which perform sequential tone ranging using onboard coherent transponder. The orbit determination accuracy achieved with a set of TTC station data collected over a period of 48 h is about 80 m. The latest missions employ onboard satellite positioning system, which gives an accuracy of 10–20 m (1σ) in position.

(ii) Attitude consideration for IRS missions: The platform attitude pointing, determination, jitter and drift-rate are important aspects of any imaging mission.

(a) Pointing – The attitude pointing is governed by the onboard closed-loop attitude control system, with inputs from attitude sensors like earth sensors, sun sensors, star sensors and gyroscopes. The control loop actuates wheels, torquers and thrusters for ensuring the specified attitude control. The pointing accuracies in pitch, roll and yaw axes for the 3-axis stabilized IRS platform are ±0.05° (with star sensors for all axes) while it is ±0.15° (with earth sensors and gyroscopes).

(b) Drift rate – The specifications of internal distortion in the image dictate the drift rate acceptable on the platform. For IRS missions, employing different swaths for imaging and the scene sizes handled in ground processing, the drift rate specifications are 3×10^{-4} to 5.3×10^{-5} deg/s.

(c) Jitter – Various moving elements on the satellite platform cause shock-like disturbances, which are undesirable. The specification for jitter is 3×10^{-4} degrees.

(d) Attitude determination – The attitude determination on ground is carried out with the raw/onboard pre-processed star sensor data or earth sensor data and precision data from gyro. Periodic gyro calibration in-orbit, capability to uplink error matrices due to misalignments, etc., provide better attitude knowledge. Quaternion approach of representing attitude has been adopted in missions where processor based Attitude Orbit Control Electronics (AOCE) has been flown. The attitude determination accuracies achieved are better than ±0.01°.

(e) Yaw steering – Stereo imaging with multi cameras or mono imaging with single camera with staggered CCD arrays need yaw steering profile loaded onboard to compensate for the Earth rotation while handling images obtained from space.

Specific mission planning aspects of high resolution

As the technology evolved, the complexity of missions also increased, particularly for high-resolution system with more stringent overall performance specifications. New techniques of imaging, like spot imaging, paint brush imaging, stereo (along track and/or across track) imaging using step and stare techniques have been designed. Step and stare is a new technique employed on cartographic missions to achieve high-resolution imageries. Image data volume of a high-resolution satellite is much larger which limits the camera motion that goes out of sync with ground velocity of the satellite. When the imaging speed of the camera system is lower than the ground velocity of the orbiting satellite, angular motion of the camera is retarded through appropriate angular rates of the satellite body to match the ground velocity for getting contiguous coverage of an image strip. Before the start of imaging, satellite is tilted so as the camera is forward biased towards the beginning of the strip much ahead of the time and imaging is initiated until the camera goes for a reverse biasing through appropriate body tilt to cover a longer strip length. To obtain high-resolution images from IRS, the following tasks were planned and implemented successfully:

Payload programming: The step and stare imaging, with programmed attitude profile, selectable roll bias and selectable yaw bias, calls for complex attitude control software supported by an equally complicated programmable payload sequencer onboard the satellite. On ground an elaborate design of payload programming system is called for. The payload programming system is a software system having three components. All the three components interact back and forth for information transactions.

- First component resides at the 'user end', to help the user to input the area of interest for image data acquisition during a particular period.
- Second component resides at NDC (National Data Centre, Hyderabad), which sequences the possible user requests that can be imaged on a particular day after accounting for clash resolution among various user requests, spacecraft and ground systems constraints, keeping track of unfulfilled requests because of cloud cover, data reception problems and constraints, requests clashes and maximizing data returns and sends the 'Requests Proposal' to SCC (Satellite Control Center), ISTRAC.
- Third component resides at SCC, which generates the required command sequence to carry out the camera operation in different orbits either for real time data collection or for recording the data in the onboard storage and dump the storage data over a data reception centre at a later time of the day.

Quantization: Careful coordination is given for quantizetion levels of imagery data on board and the levels to be transmitted to the ground.

Compression: The data compression (preferably loss less) is another aspect, which needs serious study. The type of compression adopted, trade-off between compression and no-compression, implementation of compression software onboard, number of compression tables to be managed onboard to take care of various terrains and seasons of imaging, decompression algorithms to be managed on ground, archival of compressed/decompressed data are some of the issues to be addressed. Data compression also allows the data transmission to take place within the allowed bandwidth of 375 MHz in X-band. JPEG algorithms with Huffman tables are employed in Cartosat data compression system.

Encryption: Data encryption is another feature, which denies data to unauthorized user. The encryption is selectable from ground and is periodically changed. Decryption key also rests with the ground data reception station 2048 keys for encryption is available in Cartosat system.

Encoding: Differential encoding (RS encoding) maintains the integrity of a large block of data against transmission errors using error correction techniques and randomization. This ensures binary data streams with equally distributed ones and zeros for continuous lock of bit synchronizers. These are some of the standard features adopted in Cartosat mission.

Transmission: Conventional antenna on a normal IRS platform subtends about ±65° from its position in orbit on earth. Any ground station within this field of view will be able to receive image data from the satellite. Such a conventional antenna is generally connected with a high-power Traveling Wave Tube Amplifier. Thus, for containing the data within a narrow beam the sub-metre resolution imagery from Cartosat is transmitted from a Phased Array Antenna (PAA) supported by a large number of low power transmitters. The narrow beam width of PAA enables high-resolution data reception by an intended ground station only. Single beam PAA is commonly used in Cartosat data transmission system. The onboard system has to have the precise orbit information and the ground station's co-ordinates to compute the look angles for the beam to point in the direction of the ground station. The orbit information could be obtained onboard by a satellite positioning system or the same could be supplied from the ground. Onboard ephemeris generation is based on truncated Fourier power series or alternatively is based on numerical integration of equation of motion. The beam angles generated with the orbit and ground station's co-ordinates needs to be integrated with the satellite control motions required for imaging operations and also with the payload sequence. The ground station co-ordinates could be uplinked to the spacecraft from ground in the form of data commands. Ground station inclusion and exclusion is also possible. Multiple switching of beam during a pass enables data transmission to many ground stations.

Storage: Since Cartosat missions depend on onboard resources quite heavily with complex software and attitude motions, it is essential that the data returns be maximized. Towards this, since each scene in high resolution takes only a few seconds of imaging time, it is normal to build the recording capability along with real time data transmission. Such an option enables collecting data onboard whenever PAA visibility is not favourable or for any other reason real time data reception is not possible. On some occasions, real time followed by playback of earlier recorded data enables better utilization of spacecraft visibility to the data reception station.

Accuracy: Cartosat missions depend on star sensors and gyros for meeting the stringent specifications of targeting accuracy, pointing and location accuracy. The contribution from orbit error is minimized by employing satellite positioning system onboard which locks onto GPS constellation to provide pseudo range and range-rate information which is used for orbit information generation onboard which in turn is used for attitude information generation, orbital events generation like eclipse entry/exit timings, ephemeris generation, PAA look angles generation and so on. The importance of gyros is crucial for meeting stringent mission specifications. Gyros misalignment and scale factors contribute significantly in the overall error budget. Hence, gyro calibration is an important activity to be carried out in the mission operations phase to determine the error matrices and uplink the same to the spacecraft. In order to contain the temperature variations to ensure stability, gyros have automatic temperature controllers, which enable gyro to remain within $\pm 1^{\circ}C$ from the nominal on-orbit temperature, which also could be selected from ground.

Payload calibration: In order to combat the defocusing/smearing effects in large-optics of high-resolution camera systems, the celestial bodies are scanned by the camera by imparting the requisite control motions to the platform. The point spread function determined by such star scans are used on ground to correct spread/smear effects while generating the data products. The other alternative is to provide the mechanism for incremental focusing adjustment, which will be incorporated in IRS missions in the next phase.

Radiometric response analysis of the CCD arrays is carried out once in a month by switching on the cameras in the eclipse and allowing it to observe varying intensity of LED light pattern.

More details of mission planning aspects for high-resolution satellites like Resourcesat-1 (IRS-P6), Cartosat-1 and Cartosat-2 are provided in refs 12–14.

Mission planning and operations

The first full fledged space astronomy satellite, Astrosat covers investigations of astrophysical objects, in visible, ultraviolet and X-rays region of the electromagnetic spectrum. A description of this satellite is given in the article by Agrawal *et al.* (this issue). The planned mission life for this satellite is five years. More details of mission planning, operation and data collection for Astrosat are provided in ref. 15.

Mission planning considerations for Astrosat

1. The mission consideration dictating the choice of the payloads and the spacecraft bus and the related details of the various subsystems and systems have been discussed in Shankara *et al.* (this issue). Further, the other aspects of the mission like all-up weight of the spacecraft, orbit and inclination are provided in Agrawal *et al.* (this issue), therefore it is not further elaborated here.

2. The science experiments are ON throughout the orbit and data gets recorded in the on-board solid-state recorder (SSR) with a capacity to record for four orbits. With low inclination, the satellite's science experiments spend a few minutes (2–14 min) in the South Atlantic Anomaly (SAA) region in about five orbits over a day for the chosen orbit. Whenever the satellite passes through the SAA region, the onboard automatic action switches off the high voltage systems of the experiments and switches ON after exiting from the SAA region.

3. The attitude pointing requirements take into account the technology available in inertial pointing with

star sensors and gyro. To meet the pointing capability requirement of visible and ultra-violet imaging telescopes, the control system will have a capability to achieve about 1.8 arcminute (rms). Also, the specified drift rate is 0.18 arcsec/s and the jitter specifications 0.3 arcsec (rms) over a time scale <32 msec.

4. Since the orbit chosen for Astrosat is inclined at 8° to the equator, it is planned to have one station at Bangalore to perform TTC and data reception functions. A station at Bangalore provides visibility in all orbits over a day for an average duration of 10 min. The station will be used for TTC functions in S-band and science data reception in X-band. The X-band data collection is above 5° elevation. Biak ground station will support TTC operations as back-up station to Bangalore. Port Blair and Mauritius ground stations will support the initial phase operations and serve as back-up stations in the routine phase.

5. The orbit determination accuracy using one ground station's tracking data is in the range of 1–2.2 km in position and 1–2.2 m/s in velocity. Since prediction errors over a day are 2.9 km in position and 0.5 m/s in velocity, it is planned to obtain tracking data from Biak ground station also in the routine phase to realize better accuracies.

6. Solar panels orientation is always held normal to the Sun by the drive mechanism, which is operated only when the inertial orientation of the spacecraft is changed to meet different observational requirements.

7. Since the orbit is not sun-synchronous, the Sun aspect angle and the Earth aspect angle are studied for their variations over a year. This is taken note in the onboard thermal design for different orientations to be supported in different seasons. Also, the Sun aspect angle constraint of each instrument is studied in advance for avoidance while operating them in-orbit. This constraint is built in the ground software meant of payload operations planning.

8. Unlike the single data transmission antenna onboard IRS, Astrosat should have two phased array antennas and at any orientation, one PAA at least should be visible to Bangalore station. Depending on the inertial orientation while making science observations, the favourable PAA may be selected and this can be predicted in advance for operations. If necessary, it should be possible to switch from one PAA to another during data collection. The PAA beam orientation software onboard Astrosat will be on lines similar to Cartosat missions. The onboard ephemeris generation will be based on onboard OD with the SPS data or by the Fourier Power Series (FPS) approach or by the numerical integration approach as followed in Cartosat mission. The ground station coordinates for data reception are uplinked to the satellite as data commands. So also, the orbit information is supplied to the satellite from ground through data commands.

9. The elaborate payloads operations planning is carried out off-line with two distinct activities, viz. proposal handling and observation sequence creation. The planning software will take note of all spacecraft and instruments operations constraints and Sun angle, Earth aspect angle, SAA region and eclipse information. The proposals for observations need evaluation, review and formal clearances for operations implementation.

10. The Indian Space Science Data Centre (ISSDC) along with Spacecraft Control Centre (SCC) enables science data processing, archival and dissemination. The system builders, software designers and operations crew will carry out all functions. SCC houses the operations support facilities for spacecraft health monitoring and control, flight dynamics system, scheduling and payload operations software, computer centre and associated communication links. The systems deployed for IRS and Chandrayaan-1 will be suitably configured to meet the Astrosat mission-specific needs.

Mission planning and operations of Chandrayaan-1 mission

Chandrayaan-1 mission aims at mapping of the Moon surface with various instruments to study the topography and chemical composition. The details of targeting the Moon and different phases of orbit raising to reach the desired orbit are given earlier. The list of payloads accommodated onboard Chandrayaan-1 is given in Agrawal *et al.* (this issue). Further other mission planning and operations aspects of Chandrayaan-1 are discussed below:

- The spacecraft main bus is configured to accommodate five Indian instruments and six foreign instruments. The Moon Impact Probe will be jettisoned from the main orbiter after attaining the desired lunar orbit with assured ground visibility at the required prime meridian.
- The data collection plan from each instrument follows the spacecraft power constraints, orientation requirements of individual instrument, data readout capabilities and constraints of ground stations.
- The science data return is maximized by having two data reception stations, one in Bangalore and the other in Maryland, USA, with the third station in Goldstone, USA filling the visibility gap.
- A comprehensive TTC Network (ISTRAC + external agencies) ensures almost continuous visibility of the spacecraft in all phases of the mission. These stations will also enable X-band systems checks en-route to Moon.
- The Indian Space Science Data Centre (ISSDC) facilitates the data processing, archival and dissemination functions. Data dissemination takes place through dedicated links and high-speed Internet links.
- The Indian Deep Space Network (IDSN) stations consisting of an 18 m-diameter antenna and a 32-m

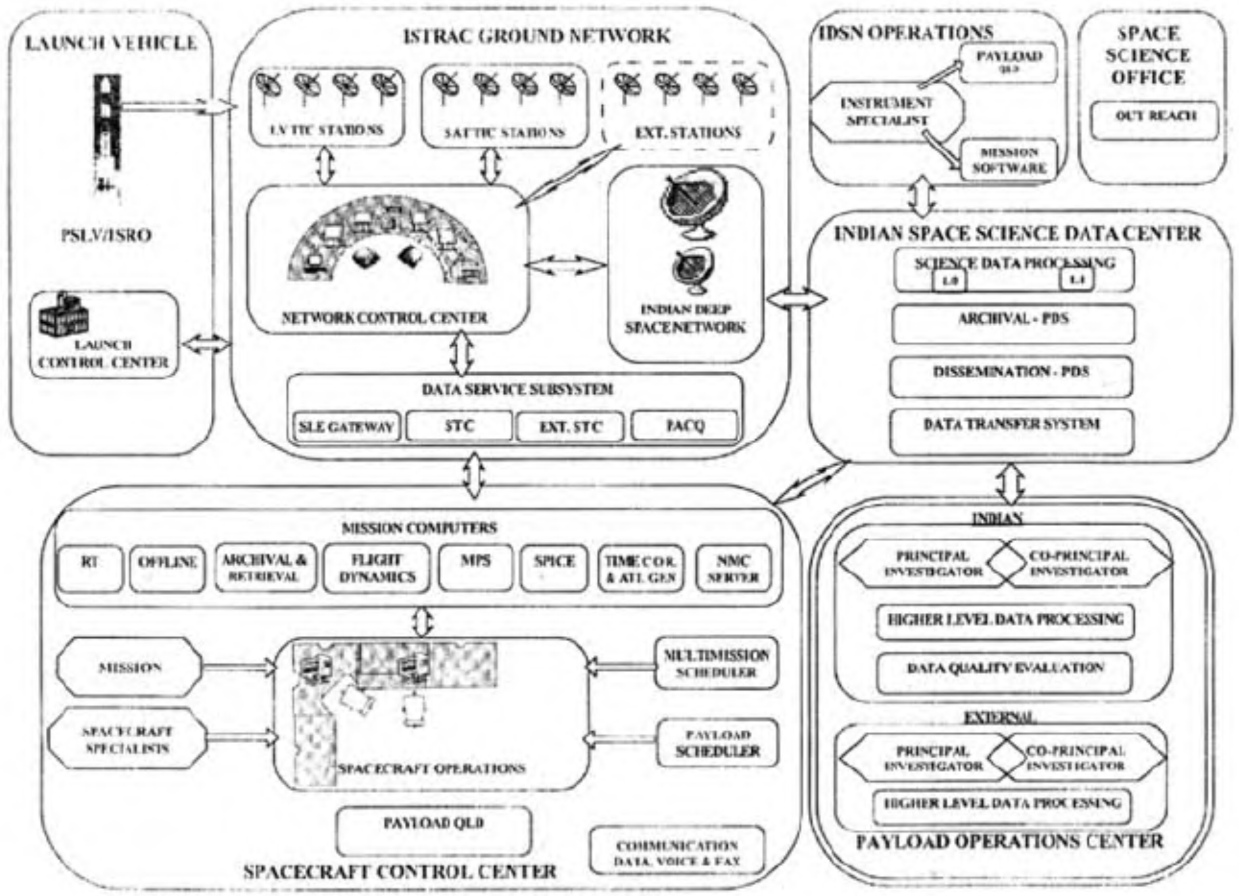

Figure 13. Ground segment organization for Chandrayaan.

diameter antenna serve the mission needs in terms of TTC and data functions.

- The launch window studies take into account the spacecraft constraints and the spacecraft – Moon geometry at the time of lunar orbit insertion. Two opportunities in a lunar month are available for launch. Any shift in launch date or launch time (2UT or 14UT) on a day has overall fuel penalty. This has been budgeted and about 6 kg of fuel allocated to account for delays in launch.
- Attitude steering is introduced during the orbit manoeuvres.
- The lunar orbit insertion is planned to be achieved with face-on geometry and ensuring visibility from Indian ground stations.
- The mission operations system inherits many features from the IRS and INSAT system. However, new ground stations, IDSN, ISSDC, payload operations centres and outreach activities are being planned for the first time.
- Detailed pre-launch simulations plans, development of software simulators, hardware simulators and integrating the entire mission operations system with completion of test and evaluation of all hardware software and procedures are new challenges to ISRO.
- The entire ground segment organization encompassing all aspects of the mission is given in Figure 13.
- The noon/midnight orbit, when the Sun vector is parallel to orbit plane, is chosen for imaging operations with an allowed excursion of ±30° of Sun aspect angle. One imaging season will be for a duration of two months.
- In order to generate adequate power throughout the mission life, two strategies are adopted: (a) the panel is canted at an angle of 30°, with respect to positive pitch in pitch-yaw plane, (b) solar panel is flipped by 180º, about pitch axis at dawn/dusk orbits and to give 180º yaw rotation at noon/midnight orbits.
- Orbit maintenance is required to maintain the altitude above the Moon surface and is executed once in 14 days during face-on orbit conditions. Onboard sequencer takes care of manoeuvre preparation, execution and normalization.
- Star sensor used for gyro updates when occulted by Earth or Sun is deselected in favour of non-occulted redundant star sensor.

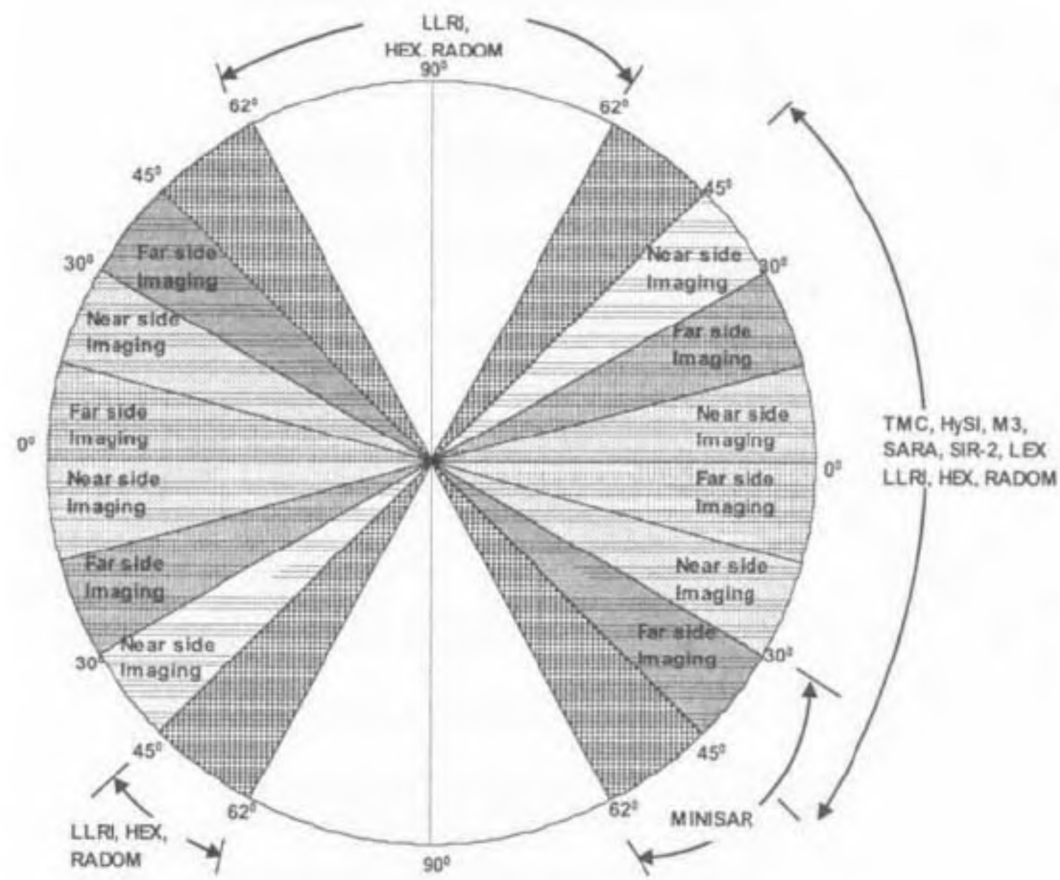

Figure 14. Payload operation profile.

- The payload operation sequence is depicted in Figure 14. The science data collection from all instruments, with the planned ground stations network, which would provide global coverage of the Moon's surface is expected to be completed within the mission life of two years. A payload scheduler implemented at SCC will enable the operations of various instruments as dictated by the constraints of main platform (power, thermal, TTC, antenna, storage, etc.), ground stations, clash scenario and priorities.

More details of mission planning, operation and data collection for Chandrayaan-1 are provided in refs 16 and 17.

Space debris mitigation in mission planning and mission design

Over a period of time, international space initiatives have left behind a plethora of space objects that no longer serve any useful functions but pose risk to space operations. Thus, space debris becomes an important subject for all space-faring nations. The space debris issues in ISRO missions have been addressed in the design and operational phases of its launch vehicle and satellite programmes. As a member of the UN Committee on the Peaceful Uses of Outer Space (UNCOPUOS), and through ISRO's membership in the Inter-Agency Space Debris Coordination Committee (IADC), India is contributing significantly to the international efforts and activities in the field of space debris. The space debris policy and mitigation measures followed in ISRO are documented in refs 10 and 11.

In the design of PSLV final stage, which uses earth-storable liquid propellants, a propellant venting system has been designed. ISRO's launch vehicle, GSLV, also employs passivation of the cryogenic upper stage at the end of its useful mission. In the operational phase, the last stage of PSLV has been passivated beginning with PSLV-C4, which was successfully launched in September 2002. With the implementation of this passivation, the possibility of on-orbit fragmentation has been minimized in all the future flights of PSLV. India's launch vehicles, PSLV and GSLV, and the satellites IRS, INSAT and GSAT series are designed in such a way that no operational debris is created in the launch and deployment phases of the mission.

At the end of mission, the GEO satellites are planned to be re-orbited in accordance with the IADC guidelines. The ISRO's communication satellites in GSO are designed with adequate propellant margins for re-orbiting to a higher orbit at the end of their useful life. The strategy is implemented on a case-by-case basis, consistent with national service requirements, like re-orbiting of INSAT 2B, 2C and 2DT at the end of their useful orbital life. The propulsion systems, by design, are built as integrated systems with the spacecraft bus and payload, and are not separated in orbit. The liquid propulsion systems plumes do not contain any solid particles. Also, the batteries are safe to prevent in-orbit explosion.

The analysis of close approaches of space debris with active ISRO spacecraft is carried out on a routine basis at the operational centres. ISRO developed the models and software to predict the close approach of any of the debris to the functional satellites. The software are being regularly used during the control and management of the orbiting spacecraft, and are specially useful, during the relocation of the geostationary satellites from one orbital slot to another orbital slot. The space debris proximity analysis software is regularly used for planning the launch window. In the area of analytical modelling related to fragmentation, a number of approaches are developed to study the evolution of break-up fragments.

Conclusions

The mission planning and mission design cycles for launch vehicle and spacecraft missions are described in the background of the experience gained by ISRO in these multi-disciplinary activities over the past four decades. While presenting the elements of mission planning and operations, the article also highlights the intrinsic complexity and enormity of the tasks covered. Future launch vehicle and spacecraft missions, like reusable hypersonic lifting re-entry space transportation systems, or missions to other planets and asteroids, are generally governed by the same generic principles of mission design and operations; but they may also require further specific developments to address their unique mission requirements.

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