Polar Satellite Launch Vehicle User's Manual

(With Revised Environmental Levels)

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Data Control Sheet

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Proprietary Statement

Publication Notice

This publication is an updated version of the document No. PSLV-VSSC-PM-65-87/3 (Issue - 3) which was released in May 1996 and reflects changes and additions to the information on vehicle performance, launch environment, payload interface requirements, launch facilities and services.

Considerable improvement has been made in the payload capability of PSLV progressively through the three development flights and the two operational flights carried out so far. Provision has been made to carry two Auxiliary payloads up to a mass of 150 kg each as a piggy-back along with the primary satellite. The launch of PSLV-C2 in May 1999 successfully demonstrated this multiple satellite launch/deployment capability of PSLV.

The current issue carries a separate chapter dealing with the Launch provisions for Auxiliary Payloads and their interfaces.

This document will be revised periodically, to include changes, if any. Comments and suggestions on all aspects of this manual will be encouraged and may be communicated to:

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Foreword

The Polar Satellite Launch Vehicle (PSLV) Programme is carried out under the aegis of the Indian Space Research Organisation (ISRO), Department of Space, Government of India and under the management of Polar Satellite Launch Vehicle Project, Vikram Sarabhai Space Centre (VSSC).

Antrix Corporation Limited, the marketing wing of the Department of Space is directly responsible for marketing and providing Launch Services of PSLV to its customers. All commercial enquiries for launch services may be addressed to:

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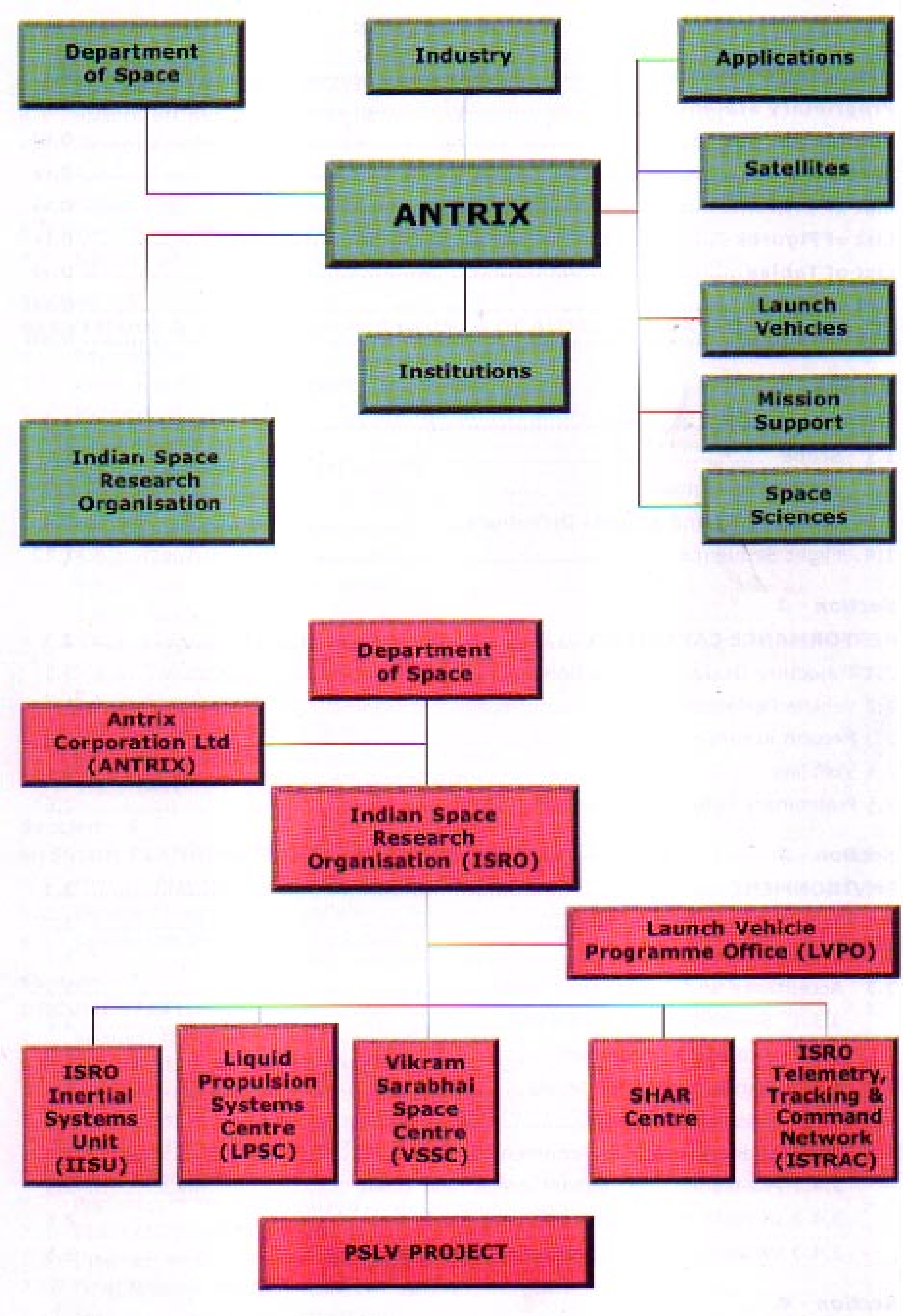

Plate 1 : Antrix-ISRO Organisation & ISRO Organisation relevant to Launch Services

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ABBREVIATIONS

ACS	-	Auxiliary Control System
Antrix	-	Antrix Corporation Limited
AP	-	Auxiliary Satellite / Payload or Micro Satellite
ASME	-	American Society of Mechanical Engineers
CDR	-	Critical Design Review
CG	-	Centre of Gravity
CTR	-	Checkout Terminal Room
DA	-	Double Amplitude
DC	-	Direct Current
DOS	-	Department of Space
DRSN	-	Down Range Station
DTO	-	Detailed Test Objectives
EB	-	Equipment Bay
EED	-	Electro Explosive Device
EGC	-	Engine Gimbal Control
EMC	-	Electro Magnetic Compatibility
FLSC	-	Flexible Linear Shaped Charge
FMA	-	Final Mission Analysis
FNC	-	Flex Nozzle Control
GTO	-	Geo-Synchronous Transfer Orbit
G/T	-	Figure of Merit (Ratio of Gain to System Temperature)
HTPB	-	Hydroxyl Terminated Poly Butadiene
IGS	-	Inertial Guidance System
IISU	-	ISRO Inertial Systems Unit
INS	-	Inertial Navigation System
IRS	-	Indian Remote Sensing Spacecraft
ISRO	-	Indian Space research Organisation
ISTRAC	-	ISRO Tracking, Telemetry and Command Network
LEO	-	Low Earth Orbit
LO	-	Local Oscillator
LPSC	-	Liquid Propulsion Systems Centre
LVPO	-	Launch Vehicle Programme Office
MAU	-	Mauritius Ground Station
MCC	-	Mission Control Centre
MIL-STD	-	Military Standard
MMH	-	Mono Methyl Hydrazine
MON	-	Mixed Oxides of Nitrogen
MST	-	Mobile Service tower
N_2O_4	-	Nitrogen Tetroxide
PCM	-	Pulse Code Modulation
PDR	-	Preliminary Design Review
POD	-	Preliminary Orbit Determination
PMA	-	Preliminary Mission Analysis
PSLV	-	Polar Satellite Launch Vehicle

PSLV (CA)	-	Polar Satellite Launch Vehicle Core Alone
PSOM	-	PSLV Strap-on Motor
PS1	-	PSLV First Stage
PS2	-	PSLV Second Stage
PS3	-	PSLV Third Stage
PS4	-	PSLV Fourth Stage
PSD	-	Power Spectral Density
QC/DC	-	Quick Connection / Disconnection
RCS	-	Reaction Control system
RF	-	Radio Frequency
SHAR	-	Sriharikota Launch Complex
SITVC	-	Secondary Injection Thrust Vector Control
SP-1	-	Spacecraft Preparation Facility – 1
SP-2	-	Spacecraft Preparation Facility – 2
SP-3	-	Spacecraft Preparation Facility – 3
SSPO	-	Sun-Synchronous Polar Orbit
STEC	-	Storage and Transport of Explosives Committee
TBC	-	To Be Confirmed
TBD	-	To Be Determined
TBI	-	To Be Identified / To Be Issued
TLV	-	Threshold Limit Value
TVM	-	Thiruvananthapuram Ground Station
TTC	-	Telemetry, Tracking and Telecommand
UDMH	-	Unsymmetrical Dimethyl Hydrazine
VEB	-	Vehicle Equipment Bay
VMC	-	Valiamala Complex
VSSC	-	Vikram Sarabhai Space Centre

Section 1

Introduction

Section - 1

INTRODUCTION

The Polar Satellite Launch Vehicle (PSLV) is a four-stage launch vehicle primarily designed to inject 1000 kg class spacecraft into a 900 km Sun-Synchronous Polar Orbit (SSPO) when launched at a nominal azimuth of 140 degree from Sriharikota (SHAR) located 80 km north of the city of Chennai (formerly Madras). The vehicle is designed and developed by Indian Space Research Organisation (ISRO) with the participation of Indian Industries and Institutions. The first developmental flight took place in September 1993 followed by two successful developmental flights in October 1994 and March 1996. The first operational flight carrying 1200 kg Indian Remote Sensing satellite (IRS-1D) was successfully conducted in September 1997. The second operational flight, PSLV-C2 conducted during May '99 successfully orbited in addition to the primary satellite, IRS-P4, two Auxiliary payloads also. The PSLV can also perform launches to Low Earth Orbits (LEO) as well as Geo-synchronous Transfer Orbits (GTO). ISRO is responsible for the launch and mission activities consisting of Mission planning, vehicle integration, checkout, launch activities, telemetry & tracking and preliminary orbit determination.

This document brings out the details of the vehicle interfaces and the requirements to be met by the spacecraft / User agency.

1.1 Scope

The scope of this document is

- to acquaint the prospective PSLV User with the vehicle and its performance capabilities.
- to define the Launch Vehicle / spacecraft interface requirements.

- to specify the documentation, integration requirements and procedures.
- to describe the Launch Site and tracking facilities, Range usage and operations including safety regulations.

1.2 Vehicle Description

PSLV is a four-stage vehicle (Fig.1.1). The first and third stage use solid propellant and the second and fourth stage use liquid propellant. The first stage consists of a 2.8 m diameter core motor (PS1) and six 1.0 m diameter motors (PSOMs) strapped onto the core. The motor case of PS1 is made of M250 grade maraging steel and it has a nominal propellant loading of 138 t. Each PSOM contains 9 t solid propellant in a high strength steel alloy motor case. The second stage (PS2) carries 40 t of propellants (UDMH and N_2O_4) in an aluminum alloy tank of 2.8m diameter and has a pump fed engine with 730 kN thrust. The third stage (PS3) has a kevlar epoxy motor case with a propellant loading of 7 t and a contoured and submerged nozzle. The fourth stage (PS4) with 2 t propellant loading (MON and MMH), has two 7 kN engines.

The Inertial Guidance System (IGS) in the Equipment Bay (EB) housed around the fourth stage propellant tank guides the vehicle till spacecraft injection. The closed loop guidance scheme resident in the on-board computer ensures the required accuracy in the injection conditions.

The three-axes attitude stabilisation of the vehicle is achieved by autonomous control systems provided in each stage. The first stage (PS1) is provided with Secondary Injection Thrust Vector Control (SITVC) for pitch and yaw control. Two swivelable Roll Control Thrusters (RCT) are used for roll control. During a short coast after the PS1 burnout, the RCT engines (of the roll control system) are used for both yaw and roll control. The Auxiliary Control System (ACS) consisting of 2 RCS thrusters are used for pitch control during coast phase. The second stage has Engine Gimbal Control (EGC) for pitch and yaw and hot gas Reaction Control System for the roll. The third stage has Flex Nozzle Control (FNC) for the pitch and yaw control during the thrust phase. The fourth stage (PS4) is controlled during thrust phase by gimballing its two engines for pitch, yaw and roll. A Reaction Control System (RCS) with six thrusters is provided on PS4 for coast phase control of both PS3 and PS4. This RCS provides roll control during PS3 burn phase also.

The vehicle is provided with instrumentation, PCM S-Band Telemetry systems and C-band Transponders for performance monitoring, tracking, range/flight safety and Preliminary Orbit Determination (POD).

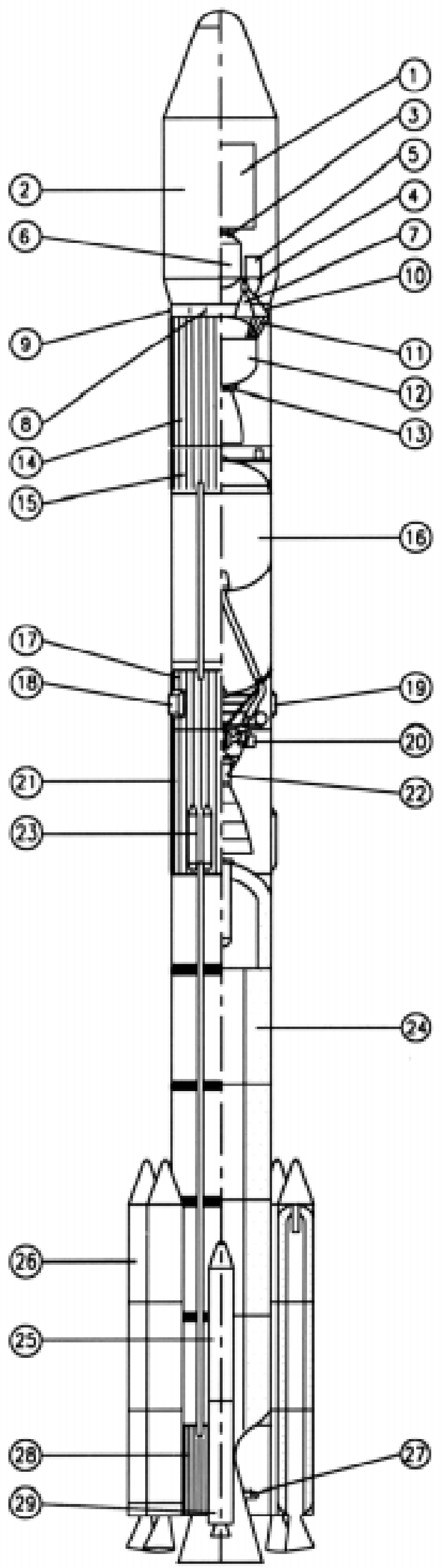
An aluminium alloy Heat shield of 3.2 m diameter protects the spacecraft from hostile flight environment during the ascent phase and jettisoned at the appropriate altitude.

The vehicle is provided with various stage separation systems such as Ball and socket joint with spring thruster for Strap-on. FLSC system for first stage, merman band for second stage and 'Ball lock' mechanism for third stage separation systems. Retro-rockets are used in PS1 and PS2 to ensure safe separation. Ullage rockets are provided in PS2 to ensure positive acceleration of the vehicle at PS2 engine ignition. Spacecraft separation is based on Merman band system.

Flight termination systems are provided onboard for strap-ons and for the first three stages which can be remotely activated through telecommand in case of any vehicle malfunction violating range safety constraints.

1.3 Vehicle Axes and Attitude Definitions

Vehicle axes and attitude definition for hardware assembly and mission analysis is shown in Fig.1.2 and the location of the PSOMs and the SITVC tanks with respect to the vehicle is shown in Fig. 1.3.



1. PAYLOAD
2. HEATSHIELD
3. PAYLOAD ADAPTOR
4. EQUIPMENT BAY
5. AUXILIARY PAYLOAD
6. FOURTH STAGE PROPELLANT TANK
7. FOURTH STAGE ENGINE (2)
8. ANTENNAE
9. REACTION CONTROL THRUSTER (6)
10. INTERSTAGE 3/4
11. THIRD STAGE ADAPTOR
12. THIRD STAGE MOTOR
13. FLEX NOZZLE CONTROL SYSTEM
14. INTERSTAGE 2/3U
15. INTERSTAGE 2/3L
16. SECOND STAGE PROPELLANT TANK
17. INTERSTAGE 1/2U
18. SECOND STAGE RETRO ROCKET (4)
19. ULLAGE ROCKET (4)
20. GIMBAL CONTROL SYSTEM
21. INTERSTAGE 1/2L
22. SECOND STAGE ENGINE
23. FIRST STAGE RETRO ROCKET (8)
24. FIRST STAGE MOTOR
25. SITVC INJECTANT TANK (2)
26. STRAP-ON MOTOR (6)
27. SITVC SYSTEM
28. CORE BASE SHROUD
29. ROLL CONTROL ENGINE (2)

Fig. 1.1 Polar Satellite Launch Vehicle

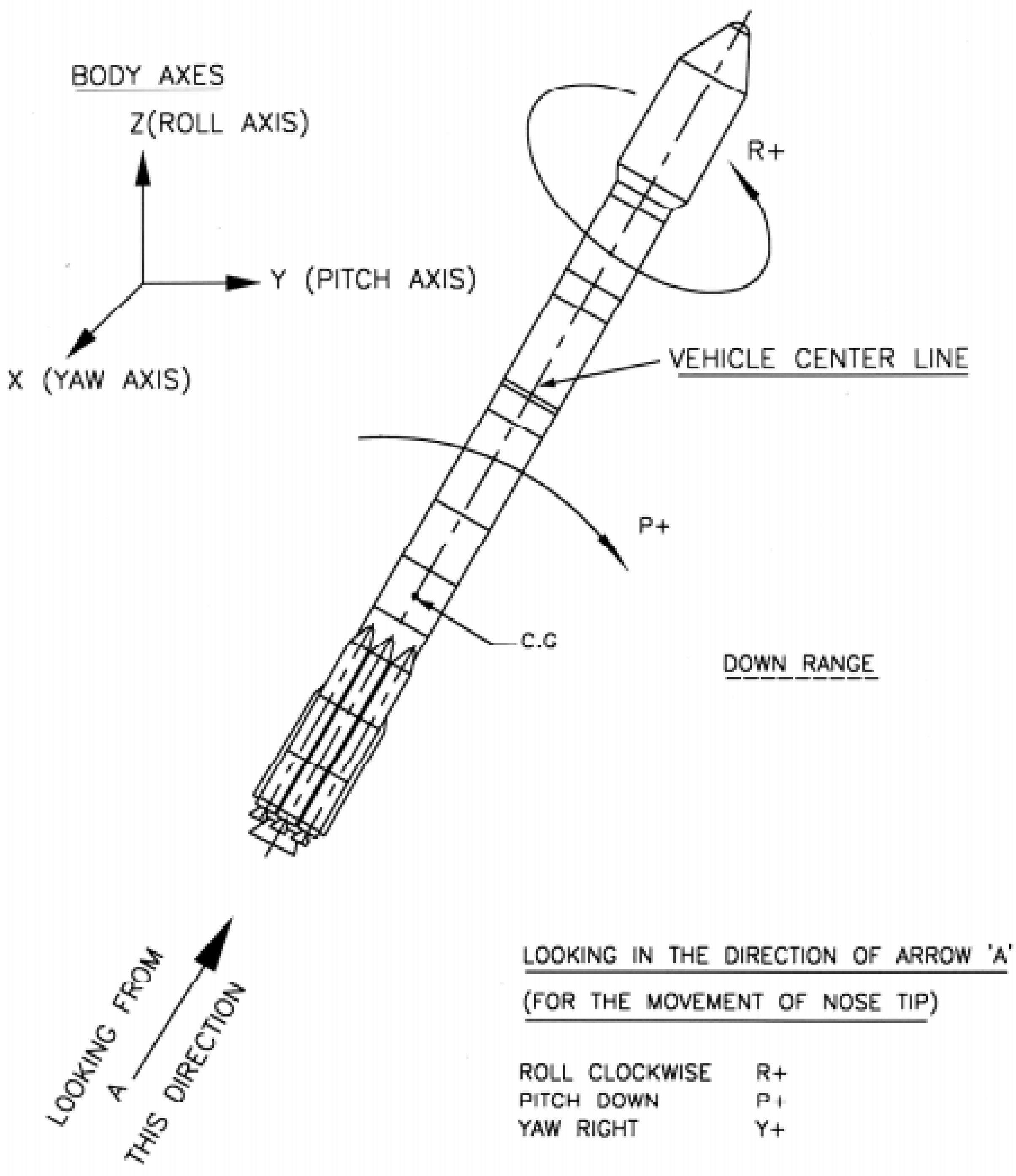


Fig. 1.2 PSLV Sign Convention

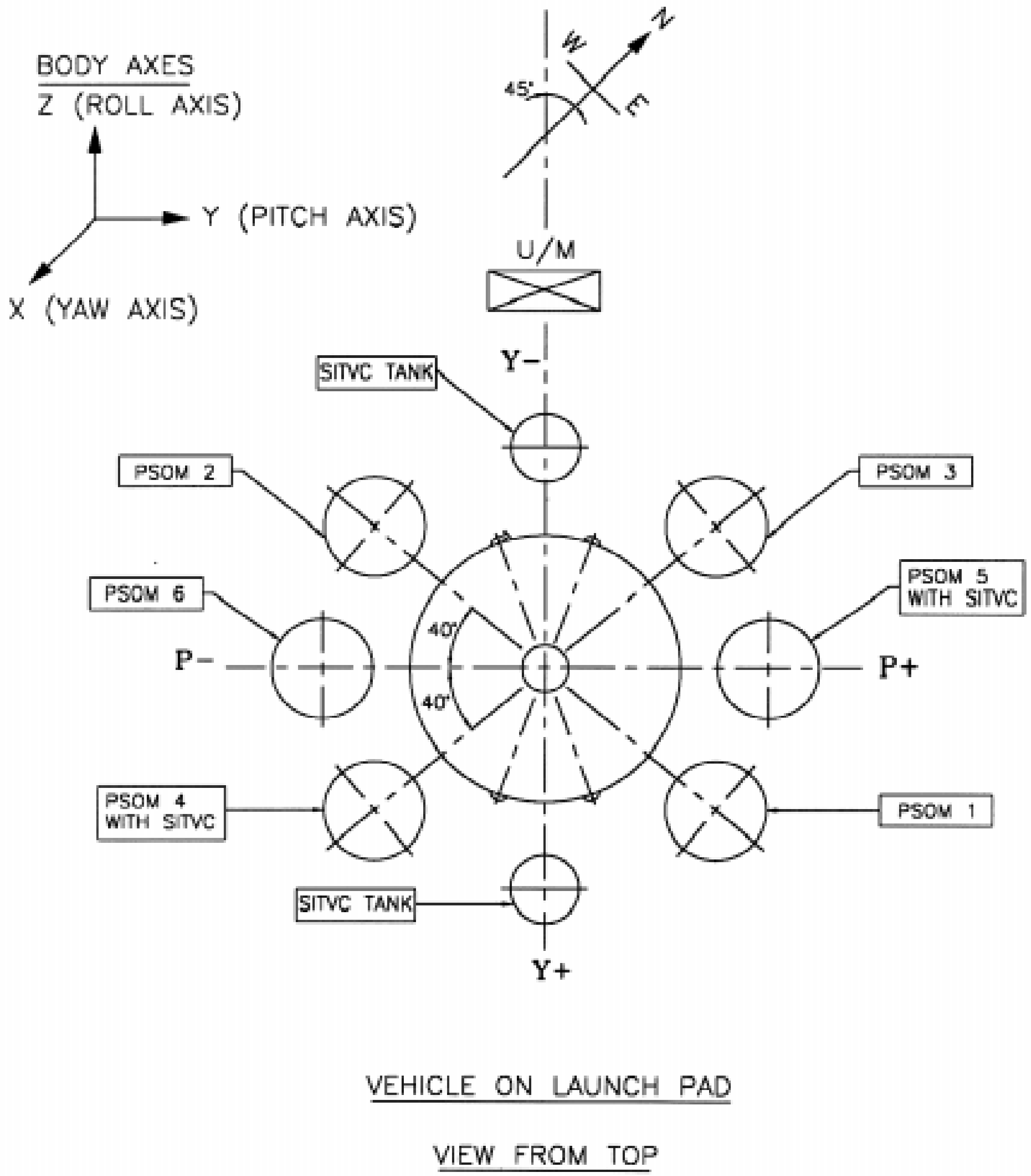


Fig. 1.3 Vehicle Orientation on Launch Pad

1.4 Flight Sequence

A typical flight sequence for the sun-synchronous polar launch of PSLV is pictorially shown in Fig. 1.4 and is enumerated in Table 1.1.

At the launch pad, the vehicle axes are aligned to 135 degree azimuth. The first stage (PS1) along with four strap-on motors is ignited on the launch pad and the vehicle lifts off vertically. After 5s, the vehicle is rolled by 5 degrees to achieve the 140 degree launch azimuth. The vehicle starts pitching in the azimuth plane at T+7s. The remaining two strap-on motors are ignited at T+25s. The ground lit strap-on motors are separated at T+68s; and air lit strap-on motors at T+90s.

The PS1, PS2 and PS3 burnout are sensed during the flight from longitudinal acceleration measurements and subsequent events are sequenced. During PS2 burn phase, the Heat shield fairings are jettisoned, and Closed Loop Guidance (CLG) is initiated. After PS3 burnout there is a combined coasting for about 134s after which the separation of third stage takes place. Ignition of fourth stage is based on guidance, approximately 100s from the time of separation. The fourth stage ignition time is decided on-board by the CLG scheme. When the required velocity and altitude are attained, the fourth stage is shut off by onboard guidance command and the satellite is injected into the required orbit with a separation velocity of 0.8-1.2 m/s.

In case of multiple spacecraft mission, attitude manoeuvre and injection of spacecraft in appropriate orientation is carried out to avoid long term collision probabilities between spacecraft. Three-axes stabilisation and guidance are effective till completion of spacecraft separation.

Table 1.1 Typical Flight Sequence of Sun-Synchronous Polar Mission (1200 kg into 817 km Orbit**)		
Event	Time from PS1 igntiion(s)*	Altitude (km)*
Ignition of first stage	T+0.0	0.0
Ignition of four ground-lit PSOMs (1,2,3,& 4)	1.2	0.0
Ignition of two air-lit PSOMs (5 & 6)	25	2.5
Separation of ground lit PSOMs	68	24.0
Separation of air lit PSOMs	90	41.0
Separation of first stage	119	76.3
Ignition of second stage	119.2	76.5
Separation of Heat shield	163	128.0
Closed loop guidance initiation	168	123.0
Separation of second stage	286	277.0
Ignition of third stage	287	279.0
Separation of third stage	506	579.0
Ignition of fourth stage	635	701.0
Fourth stage cut-off	1040	826.5
Separation of satellite	1091	827.5

* For nominal vehicle performance

** Corresponding to Osculating Orbital element at 27 degree South latitude.

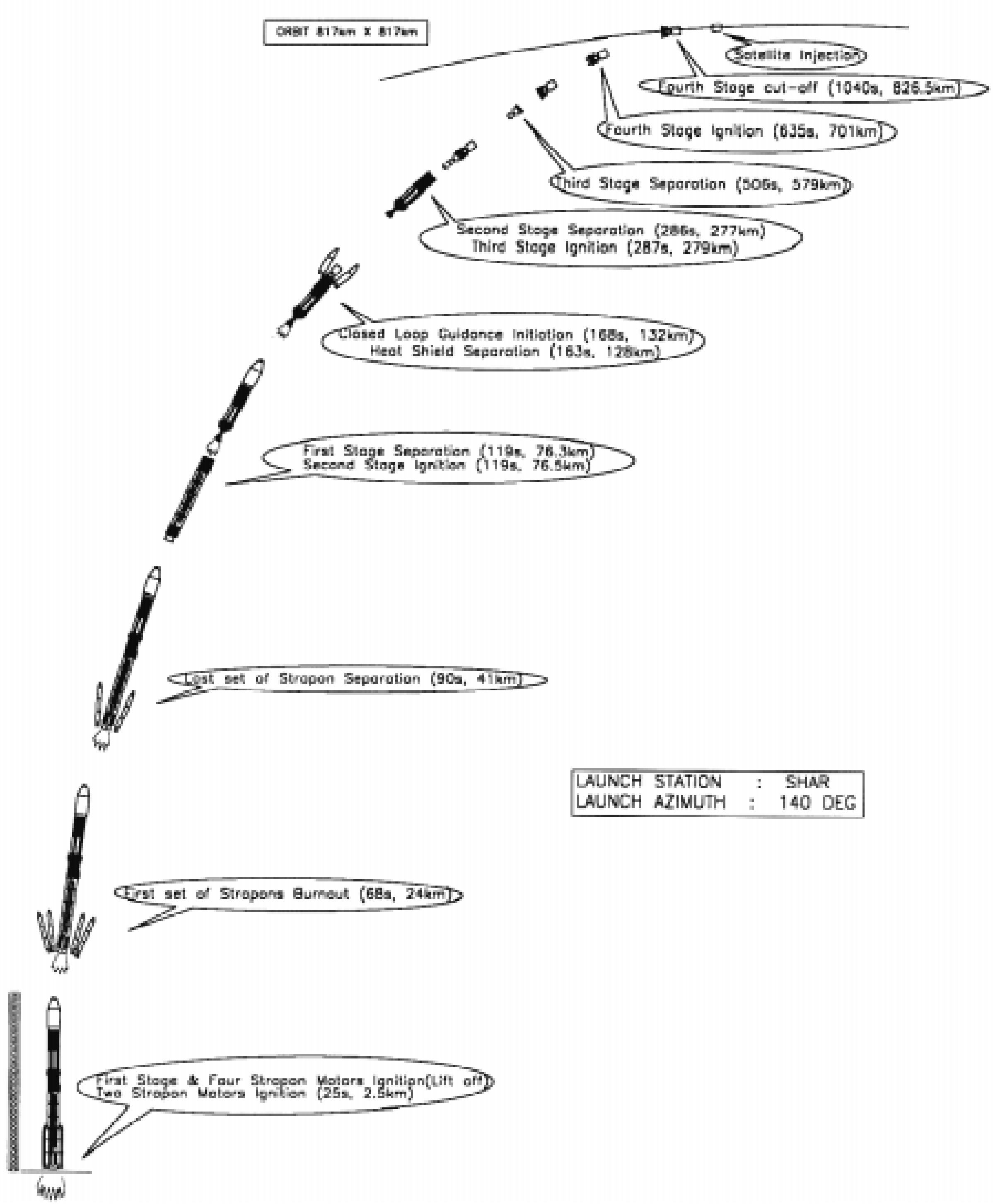


Fig. 1.4 Typical Flight Sequence for SSPO Launch

Section 2

Performance Capability

Section - 2

PERFORMANCE CAPABILITY

PSLV can perform three classes of missions:

- Sun-Synchronous Polar Circular Orbits (SSPO)
- Elliptical/Circular Low-Earth Orbits (LEO)
- Geo-Synchronous Transfer Orbits (GTO)

2.1 Trajectory Design Optimisation

ISRO designs mission trajectory for each PSLV flight to maximise payload performance while complying with the Satellite and Launch Vehicle constraints. The desired orbit parameters and constraints are the inputs to the above process along with appropriate mass, aerodynamic and thrust inputs.

One of the constraints for trajectory optimisation is the jettisoning of the heat shield fairings at such an attitude as to ensure that the free molecular heat flux on the Spacecraft is less than 1135 W/m^2.

After the trajectory optimisation, six-degree-of-freedom simulations are carried out to verify Vehicle attitude and control dynamics including stage separation dynamics.

2.2 Vehicle Performance

PSLV performance capability for different classes of missions with typical orbits is given in the following Table 2.1.

Mission	Inclination (deg)	Orbit (km x km)	Payload (kg)
SSPO	98.73	817 x 817	1450
GTO	18.0	200 X 36000	1100
PLANAR	52.0	400 X 400 to 1000 x 1000	3200 to 2250
ELLIPTICAL	49.3	400 X 10000	1350

The performance capability charts for different missions like Sun-Synchronous polar orbits, planar circular orbits, circular orbits with different inclinations and Elliptical orbits are given in Fig.2.1 to 2.3.

2.3 Mission Accuracy

PSLV normally injects the spacecraft in three-axes stabilised mode into the specified orbits. During the first stage thrust phase, the vehicle uses open-loop guidance. Closed loop guidance is initiated during the second stage regime and is active until command cut-off of the PS4 engines and injection of the spacecraft into orbit.

The Inertial Guidance System (IGS) used in the PSLV ensures the injection of the spacecraft in a typical 817 km SSPO within the error limits given below.

Error in the orbital parameters (3 s):

Apogee/Perigee : ± 35 km

Inclination : ± 0.2 deg.

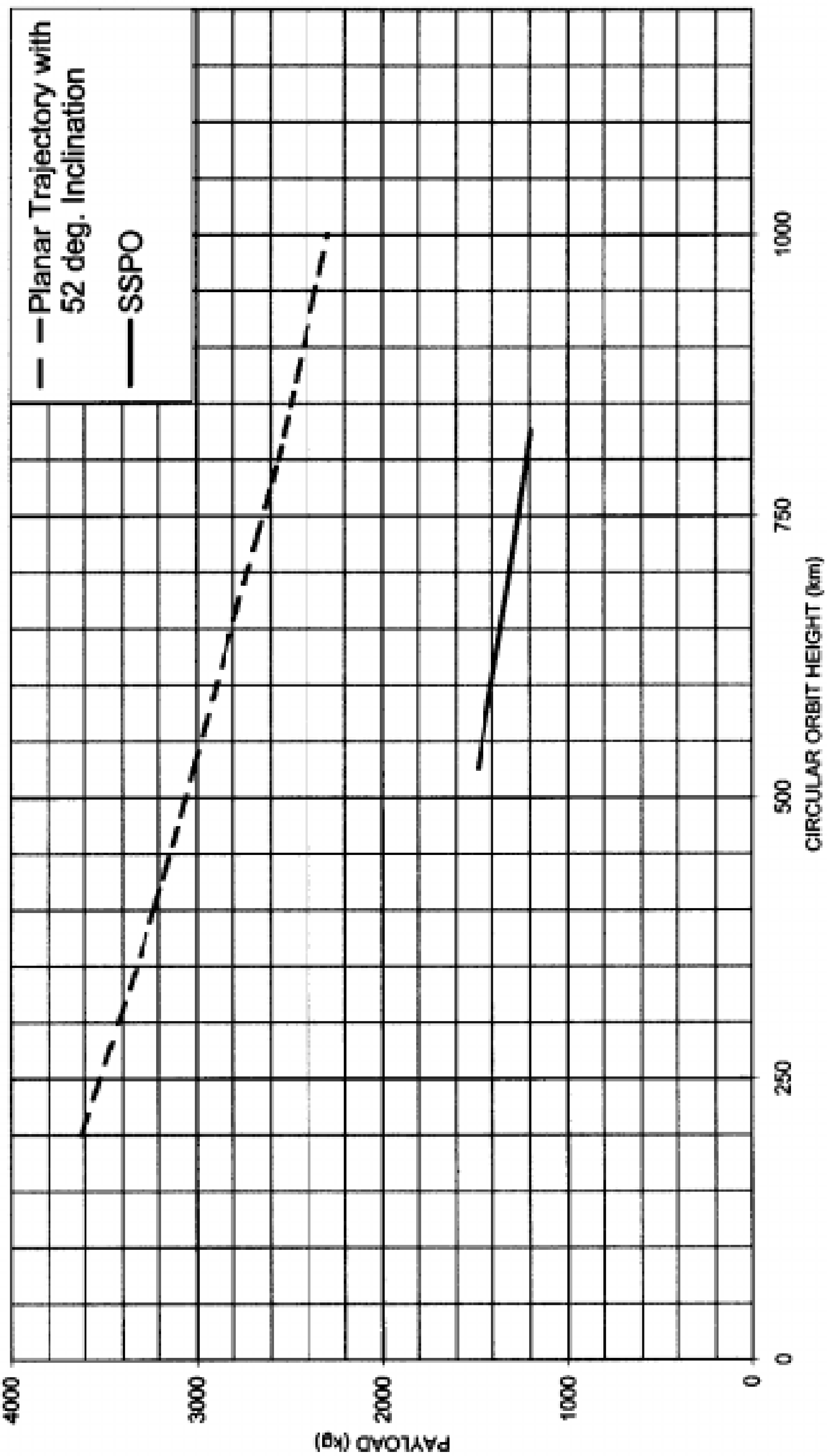


Fig. 2.1 Payload Capability for Planar & Polar Orbits

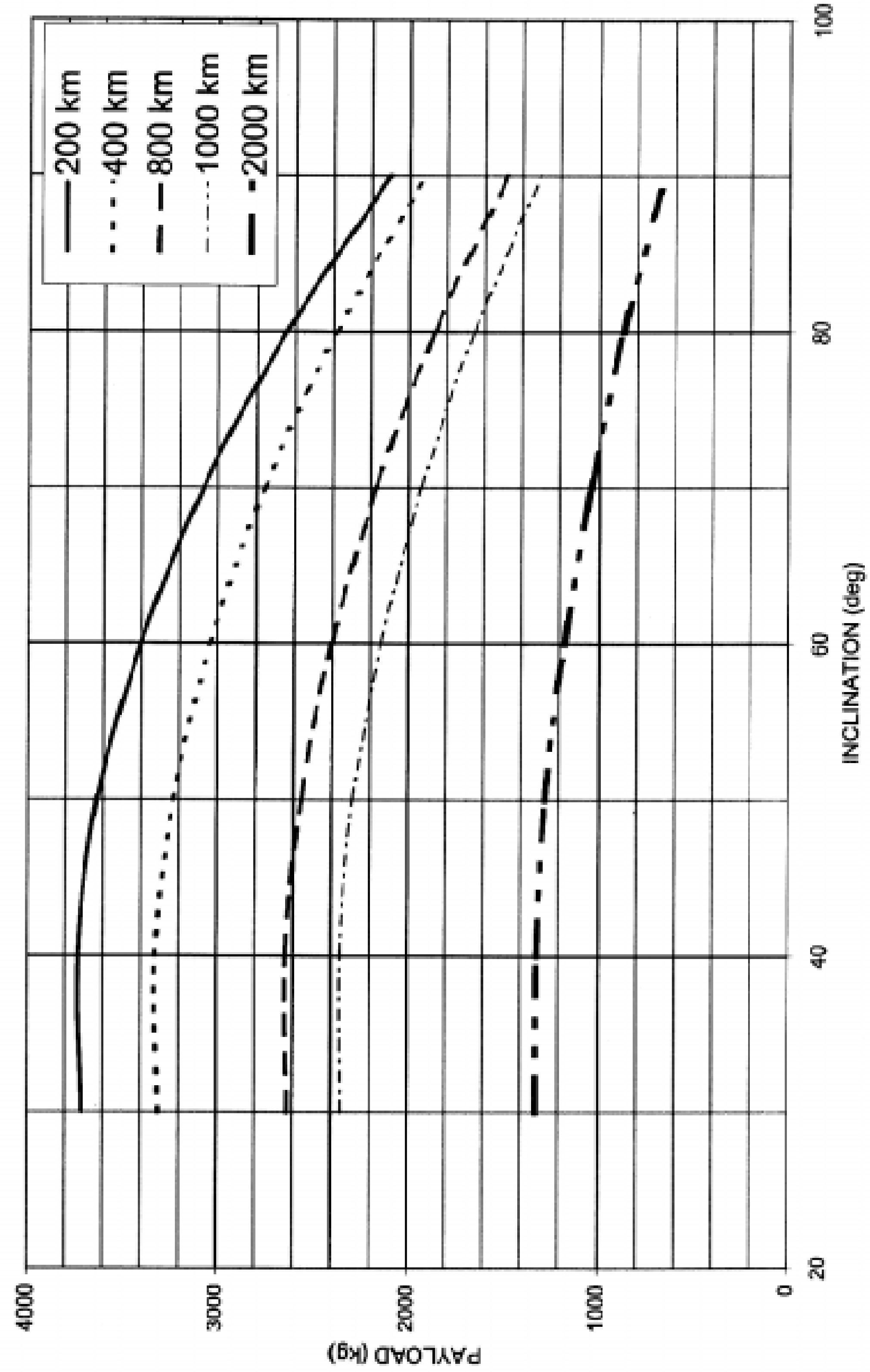


Fig. 2.2 Payload vs Orbital Inclination for Circular Orbits

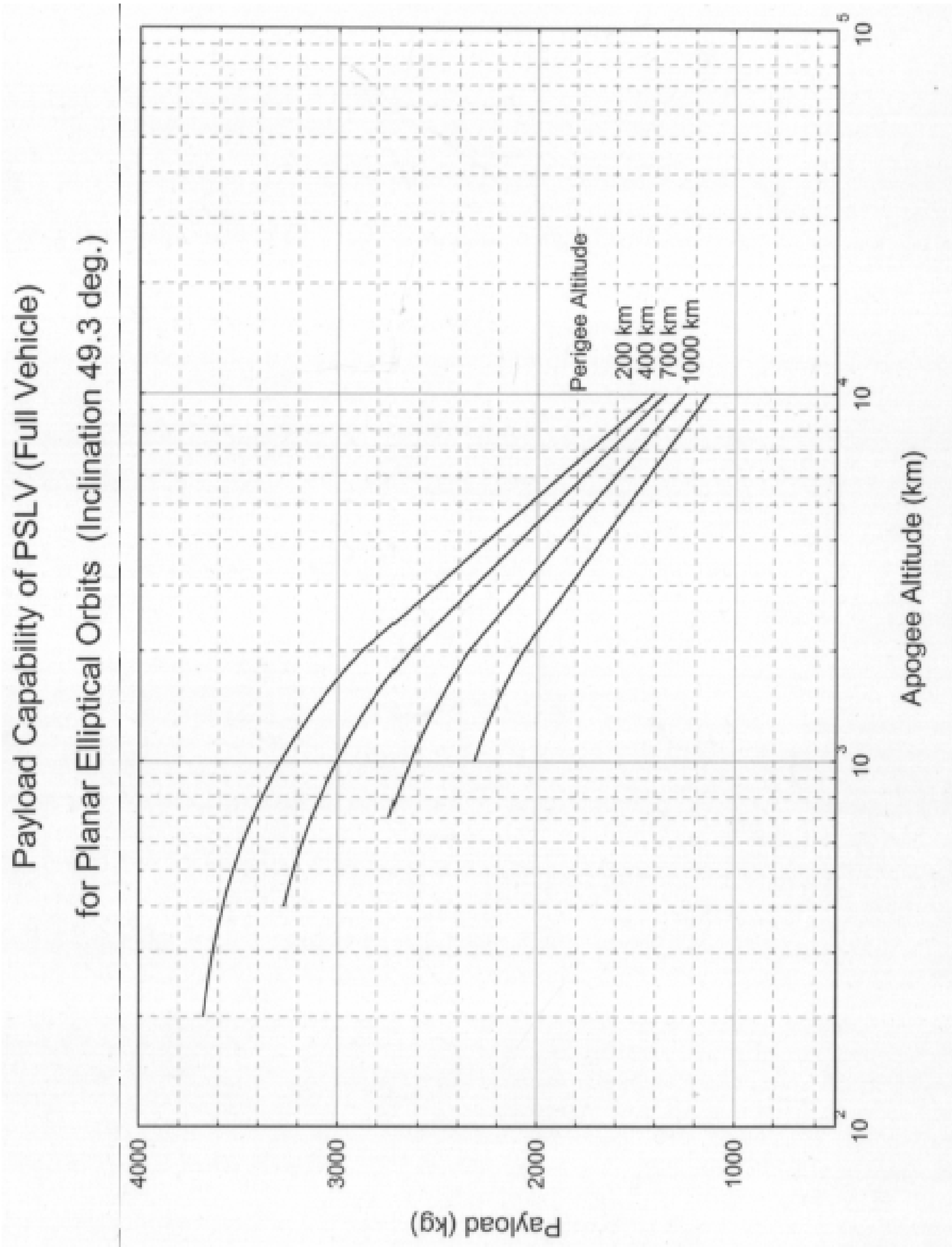


Fig. 2.3 Payload Capability for Elliptical Orbits

2.4 Visibility

Visibility of PSLV for a nominal SSPO launch from tracking stations located at SHAR, Thiruvananthapuram and Down Range Station (DRSN) at Mauritius are given in Fig. 2.4.

2.5 Preliminary Orbit Determination (POD)

POD provides for the assessment of the Vehicle performance achieved in the flight and ascertains the fulfilment of the Launch Vehicle mission soon after launch. It also enables to estimate the azimuth and the elevation angles for acquisition of the Spacecraft from any designated ground station during subsequent orbits.

POD data will normally be made available within one hour after the injection of the spacecraft.

POD is carried out using the following data.

(i) Telemetered data of INS

(ii) Range, range rate, azimuth and elevation angles from Range and Range Rate System.

(iii) Range, azimuth and elevation angles from radar.

The first phase of POD will be carried out using INS data within 15 minutes of spacecraft separation. Accuracy of orbit prediction is ±35 km in altitude and ±0.2 degree in inclination. Second phase will be completed within one hour after launch with an accuracy of ±15 km in altitude and ±0.1 degree in inclination. This POD phase also ensures that the look angles are predicted within the required half beam width angle of the S-band telemetry antennae.

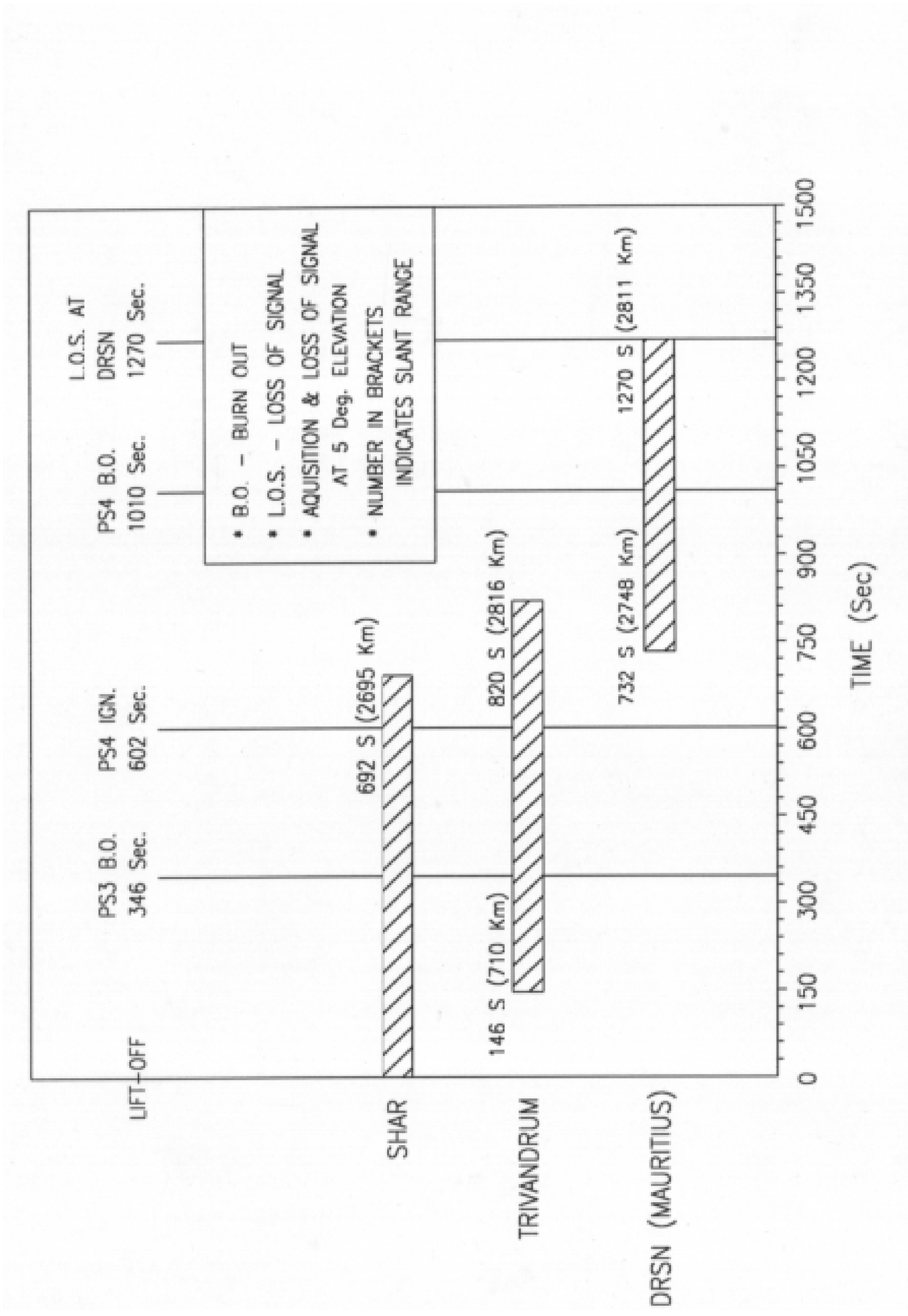


Fig. 2.4 Typical Visibility from Ground Stations for SSPO Launch (817 km)

Section 3

Environment

Section - 3

ENVIRONMENT

Presented in this section are the launch environmental conditions on spacecraft and related aspects.

3.1 Flight Dynamic Environment

During the powered launch phase, the spacecraft will be subjected to sinusoidal and random vibrations. Low frequency sinusoidal excitations are imposed on the spacecraft at various flight events such as ignition, burn-out and staging. Maximum acoustic excitation occurs during lift-off, transonic and maximum dynamic pressure conditions during the thrusting phase of the first stage.

To avoid dynamic coupling between the low frequency excitation and spacecraft modes, the stiffness of the spacecraft structure shall be designed to keep the fundamental frequencies above 35 Hz along the thrust axis and 20 Hz along the lateral axis, with base fixed at the plane separating the vehicle and the Satellite. For payloads which do not meet these criteria, it is necessary to co-ordinate the structural design closely with ISRO so that a coupled analysis can be performed to define loading conditions, and also to verify the design of interfaces. The flight loads given here are intended to provide the test input levels for the spacecraft.

3.2 Vehicle Acceleration

The maximum static and dynamic accelerations occurring at spacecraft interface during each stage are given in Table 3.1.

Based on the combined static and dynamic acceleration loads, the design levels shall be arrived at by taking ultimate load factor of 1.25 and a proof load factor of 1.1. These loads shall be used for design, analysis and structural qualification tests of the spacecraft.

Table-3.1
Maximum static and dynamic acceleration occuring at S/C interface during Flight
Longitudinal : 7 g / -2.5 g Lateral : +/- 1.5 g
Ultimate load factor : 1.25

3.3 Acceptance and Qualification Tests

This section outlines the flight acceptance and qualification environmental tests for the spacecraft.

It is intended that the flight acceptance test levels established will subject a spacecraft to an environment that will not be exceeded in flight 95% (2σ) of the time.

The tests prescribed here are generalised in order to encompass the different spacecraft configurations. For this reason, the spacecraft agency should critically evaluate its own specific requirements and develop detailed test specifications tailored to the particular spacecraft. After the completion of the dynamic qualification and flight acceptance tests, functional testing shall be conducted on the spacecraft, to ensure that all the systems have survived the testing without degradation.

3.3.1 Sinusoidal vibration levels

Table-3.2
Qualification and Acceptance Test Levels of Sinusoidal Vibration

	Frequency Range-Hz	Qualification Test level (Zero to peak)	Acceptance Test Level (Zero to peak)
Longitudinal Axis	5.0-11.5 11.5-20.0 20-25 25.0-33 33.0-35 35.0-100.0	6.75 mm (DA) 1.80 g 1.8-3.75 g 3.75 g 3.75-0.75 g 0.75	4.5 mm (DA) 1.2 g 1.2-2.5g 2.5 g 2.5-0.5 g 0.5g
Lateral Axis	5.0-7.0 7.0-30.0 30.0-100.0	6.75 mm (DA) 0.67 g 0.45 g	4.5 mm (DA) 0.45 g 0.30 g

3.3.2 Acoustic environment

The critical acoustic environments occur at lift-off due to jet noise and in transonic flight due to unsteady flow and boundary layer noise. The high frequency dynamic excitations in the payload area are generated mainly by acoustics. Acoustic blankets are provided on the inner surface of the heat shield at the cylindrical and boat tail regions to reduce the acoustic environment for the spacecraft. It is recommended that acoustic testing may be carried out for the spacecraft qualification and acceptance for closer simulation of in flight conditions and to avoid possibility of over/under testing of spacecraft equipment if tested through random vibration testing. The acoustic test levels are given in Table 3.3.

Table-3.3
Qualification and Acceptance Test Levels of Sound Pressure

Octave band centre Frequency (Hz)	Sound pressure level in dB	
	Qualification	Acceptance
31.5	128	124
63.0	130.5	126.5
125.0	134	130
250.0	140	136
500.0	144	140
1000.0	139	135
2000.0	132	128
4000.0	129	125
8000.0	126	122
Overall level in dB	147	143
Duration	2 minutes	1 minute

3.3.3 Random vibration levels

The random vibration test levels at payload interface are given in Table 3.4. These values are provided as an alternative method of testing the spacecraft in lieu of acoustic tests.

Table 3.4
Qualification and Acceptance Test Levels of Random Vibration

Frequency (Hz)	Qualification PSD g^2/Hz	Acceptance PSD g^2/Hz
20	0.002	0.001
110	0.002	0.001
250	0.034	0.015
1000	0.034	0.015
2000	0.009	0.004
Overall level	6.7 g (rms)	4.47 g (rms)
Duration	2 minutes/axis	1 minute

Note : The random levels established may be used to excite the main equipment of the spacecraft to levels generated in flight by acoustics. It shall be emphasised that excitation of external spacecraft surfaces of large area and low density such as solar panels and large antenna configurations cannot be duplicated by random vibration without over testing, and a separate evaluation of acoustic susceptibility of these components may be required.

3.3.4 Shock test levels

Spacecraft is subjected to shock environment at various events of flight like stage ignition, shut-off and stage separation. Of these the most significant shock will be felt due to the spacecraft separation.

The shock levels also depend on spacecraft mass and payload adapter construction. A typical payload separation shock spectrum is shown in Fig.3.1. It is recommended that an actual separation test be conducted on a representative payload adapter and to qualify the spacecraft.

3.4 Thermal and Climatic Environment

3.4.1 Pre-launch environment inside heat shield

After the spacecraft mounting and heat shield assembly, the satellite is cooled continuously during the entire pre-launch period. This is achieved by the use of dry cooled air injected through the cooling air duct into the heat shield at the following conditions.

Inlet temperature of air	:	Variable from 10° C to 15° C
Flow Rate	:	1800 kg/hour
Relative humidity	:	45 +/- 5 %
Cleanliness	:	10000 Class
Air circulation velocity	:	$<$ 2 m/s max.

The noise generated by the air cooling system does not exceed 90 dB as a general rule except for initial start up or shutdown conditions lasting for a few seconds.

3.4.2 In-flight thermal environment inside heat shield

The heat shield is given thermal protection to keep the temperature of the heat shield external surface during the flight at less than 120°C. The maximum heat flux density in the heat shield will be 1000 W/m^2.

3.4.3 Variation of static pressure inside Heat shield

Adequate venting has been provided in the heat shield to ensure that the difference in static pressure inside and outside the heat shield is less than 10 kPa. The variation of static pressure with time is given in Fig.3.2.

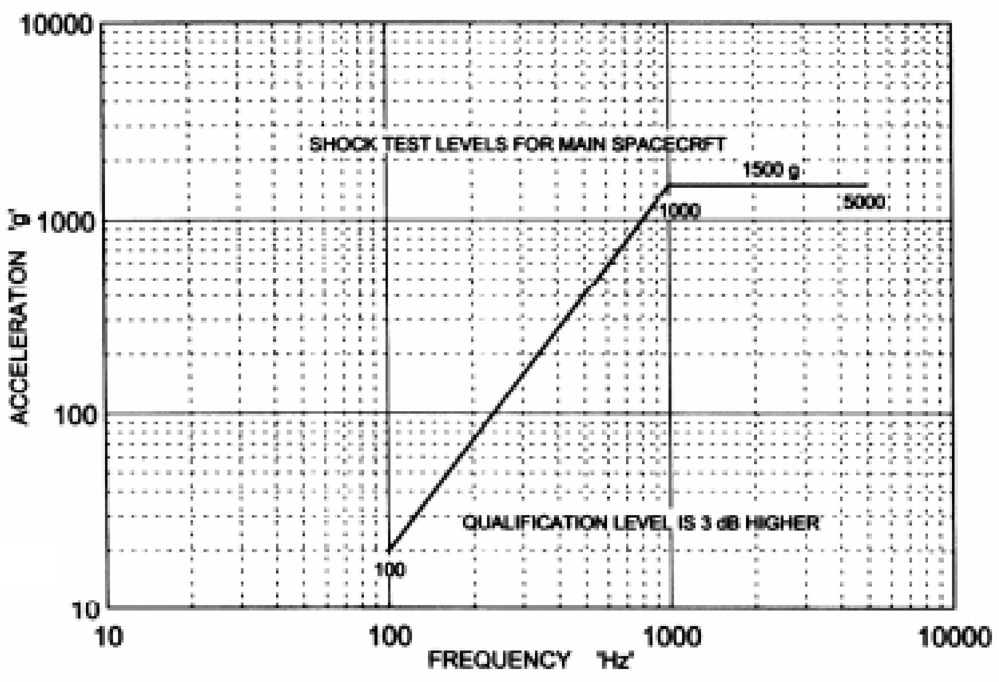


Fig. 3.1 Typical Payload Separation Shock Spectrum

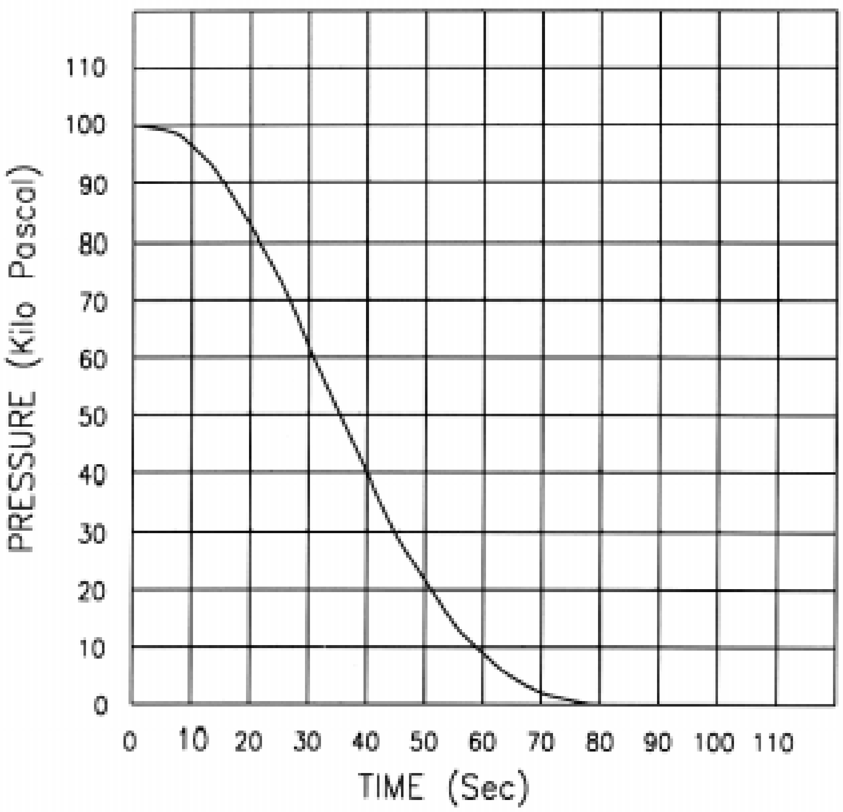


Fig. 3.2 Pressure Variation inside the Heat shield

Section 4

Payload Interfaces

Section - 4

PAYLOAD INTERFACES

This section gives detailed description of the heat shield fairing, the payload envelope and the mechanical and electrical interfaces of the launch vehicle with the spacecraft. The requirements and constraints herein specified are to be strictly observed. Waivers can be granted wherever feasible, but any relaxation of these requirements shall have to be approved by ISRO.

4.1 Heat Shield

The heat shield (Fig.4.1) is an all-aluminium structure fabricated in two halves. It consists of a spherical nose cap and a conical section at the forward end, a long cylindrical section and a short conical boat-tail. The conical sections are stiffened semi-monocoque structures and the cylindrical section is an integrally stiffened isogrid structure made up of 3 panels of 1.5 m height each.

Acoustic absorption blankets are provided within the cylindrical and boat tail portions of the heat shield. Particle contamination is controlled to provide a 10,000 class cleanliness. Outgassing from acoustic blankets meets the criteria of 1% maximum total mass loss and maximum of 0.1% volatile condensable matter.

The heat shield fairings are joined vertically through a contamination-free, linear piston cylinder separation and jettisoning mechanism (zip cord) running along the full length of the heat shield. A clamp band joint is employed for attaching the heat shield at its base to the vehicle.

The heat shield fairings are separated by the actuation of the clamp band joint at the base and the zip cord. The gas pressure generated by the mild detonating cord of the zip cord expands a rubber bellow, pushing the piston and cylinder apart after shearing the rivets holding the two halves. The force acting on the half shells pushes them laterally away from each other thus achieving the required jettisoning velocity. The bellow assembly retains the residual gases and prevents contamination of the spacecraft.

Removable access doors are provided on the heat shield to permit limited access to the spacecraft following heat shield installation. Standard and nominal size access doors are shown in Fig.4.1. Change in size and location of these doors shall be co-ordinated with ISRO in advance.

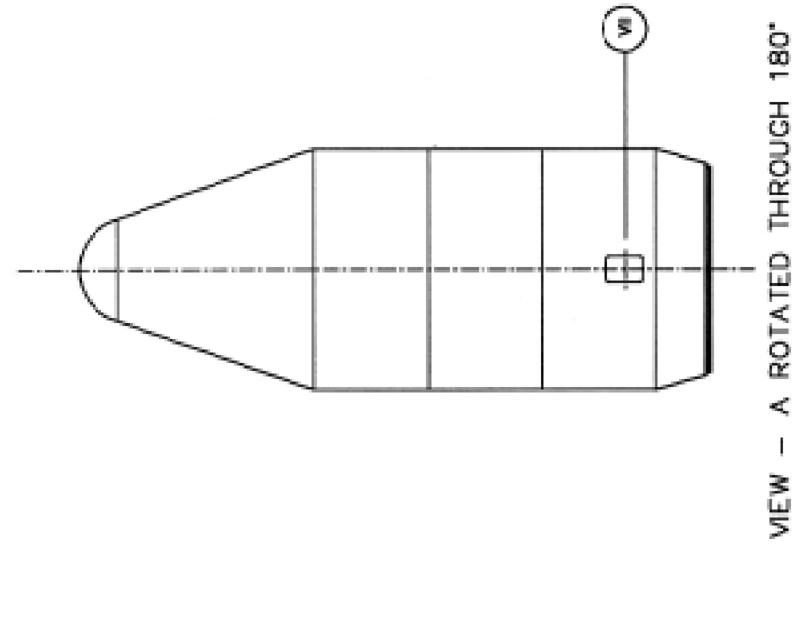
Radio Frequency (RF) transparent panels can be provided as options on access doors in the heat shield cylindrical section to provide for spacecraft telemetry transmission through the heat shield.

4.2 Payload Envelope

The payload envelope represents the maximum allowable spacecraft external boundary including manufacturing tolerances, spacecraft static and dynamic deflection during assembly and flight. The envelope also takes into account the allowances for vehicle/heat shield static and dynamic deflections, manufacturing tolerances and acoustic blanket thickness.

For a spacecraft, which does not meet the natural frequency criteria, or has protrusions that may extend outside the envelope shown, co-ordination is required with ISRO to define the appropriate envelopes, at the feasibility study phase. The necessary analyses are performed to estimate and ensure positive clearances, especially during transonic regime and during maximum dynamic pressure conditions.

The allowable payload envelope within the confines of the heat shield is shown in Fig. 4.2. The hatched area gives the usable area forward of the separation plane. It is also possible to use the space surrounding the payload adapter below the separation plane (cross hatched area) with some restrictions, should the need arise.



CUT OUT	HT OF CUT OUT CENTRE FROM NOSE TIP	SIZE (CLEAR OPENING)	PURPOSE
IV-A	3424	340 x 287	MAIN S/C-ACCESS
IV-B	2858	340 x 287	MAIN S/C-ACCESS
V-A	2773	242 x 230	MAIN S/C-ACCESS
V-B	2773	438 x 230	MAIN S/C-ACCESS
VII	1108	340 x 457	AUX. S/C ACCESS
IX	1108	340 x 457	AUX. S/C ACCESS

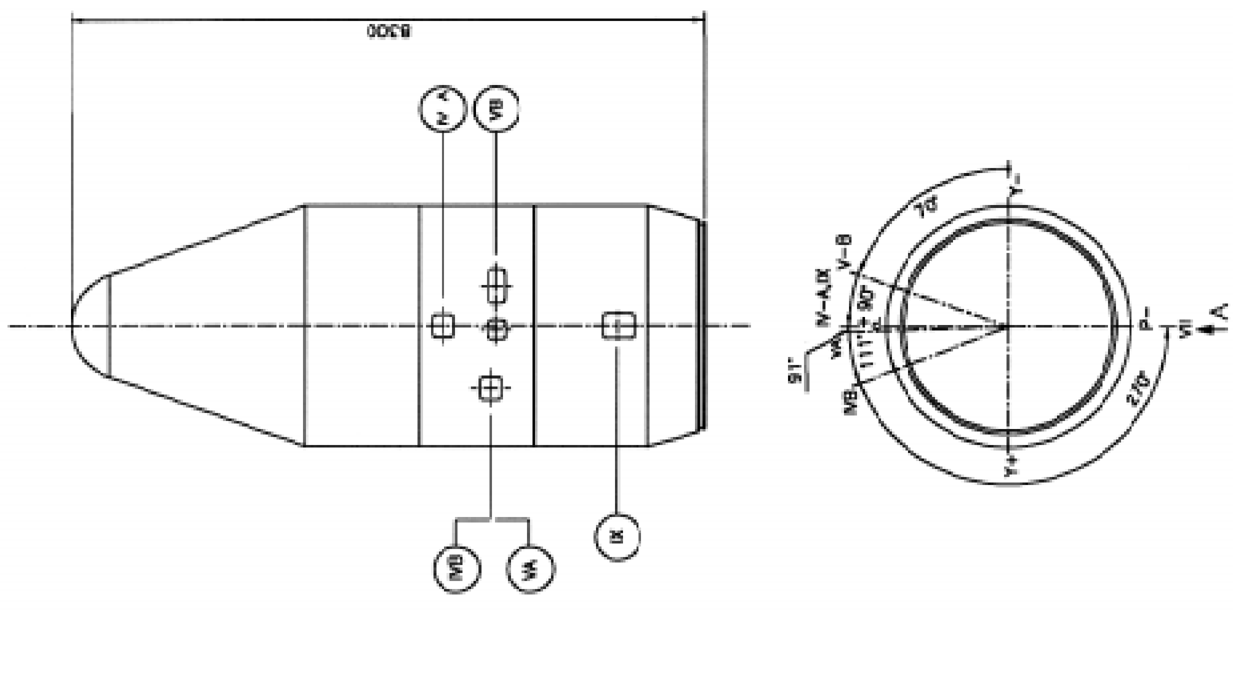


Fig. 4.1 Location of Cut-outs on Heat-shield for Payload Access

4.3 Payload Adapter

Payload adapter forms the interface between the fourth stage of PSLV and the spacecraft and carries the separation system at the forward end. It has a truncated conical structure with aluminium end rings and Carbon Fibre Reinforced Plastic (CFRP) panels.

Payload adapter assembly (Figs 4.3 and 4.4) has a height of 500 mm and a diameter of mm at the aft end.

The forward end of the payload adapter has a 937B interface (15 deg semi angle and incorporates standard interface compatible with 937 V payload adapter of ARIANE-4/DELTA-2. The structure is capable of supporting a 1800 kg spacecraft with its CG 1.5 m above the Satellite separation plane. Maximum tolerable transverse offset of CG is 5 mm.

The above payload adapter is designed for the launch of single spacecraft. However, suitable payload adapters can be provided to meet the requirement for multiple spacecraft launch.

4.4 Spacecraft Interface Dimensional Requirements

Spacecraft interface dimensional requirements are defined in Fig.4.3 and 4.4.

4.5 Spacecraft Separation System

The spacecraft is mounted on the forward end ring of the payload adapter through a merman type clamp band at a mating diameter of 945.26 mm. The clamp band is tightened using tension bolts to ensure proper joint characteristics. A pair of bolt cutters is used in redundant mode as release devices. Four to twelve numbers of helical compression springs provide a separation velocity of 0.8 m/s to 1.2 m/s. The maximum interface load due to each spring is limited to 1000 N. The separation disturbance on the satellite is not more than 2 degree/s for the stipulated conditions of CG offset limits.

4.6 Electrical and RF Interfaces

4.6.1 Umbilical connector

A 61 pin snap-off connector (Deutsche type – DBAS-70-61-PN 059 on Vehicle side & DBAS-70-61-PS 059 on Satellite side) located at the payload separation plane serves the support functions of spacecraft while on launch pad. This connector is wired to a separate umbilical connector located on the external surface of the fourth stage. At the time of lift off, the corresponding mating connector on the umbilical tower gets disconnected automatically. The location of this connector is given in Fig. 4.4.

The lines from the 61 pin umbilical connector will be terminated at a junction box in the Checkout Terminal Room (CTR) adjacent to launch pad which also houses the spacecraft checkout equipment. The serialised checkout data along with 25 hard lines (data/command lines) are routed to Spacecraft Preparation (SP-1) facility. A typical cabling scheme for spacecraft checkout lines at SHAR is shown in Fig.4.5.

4.6.2 Balancing connector

Balancing connector is a 61 snap-off type of connector (Deutsche type) located in the separation plane between the vehicle and the spacecraft at a location 180° opposite. Two pairs of pins of this connector shall be used on the vehicle side for monitoring the spacecraft separation while the corresponding two pairs of pins on the spacecraft side shall be shorted. This connector also provides the balancing force during separation to reduce tip off.

4.7 Electro Magnetic Compatibility (EMC) Requirements

PSLV is equipped with the following transmission and reception systems to cater to data transmission, flight termination and tracking.

- ⇨ Two telemetry systems in the equipment bay operating in S-band
- ⇨ A telecommand reception system using two receivers in the third stage with antenna mounted in fourth stage, and operating in UHF band.

- ⇨ Two Radar transponder systems operating in C-band.

An electromagnetic compatibility analysis shall be carried out by PSLV in order to avoid undesirable interaction between the Vehicle and Spacecraft radiation during Checkout, Count Down and flight.

The Spacecraft agency shall submit the following details with respect to the spacecraft systems to enable such analysis.

- ⇨ Power and frequency of the On-Board Transmitters
- ⇨ Type of antennae, number of elements and antennae pattern.
- ⇨ Frequency and sensitivity of the On-Board receivers (including Local Oscillation frequencies).
- ⇨ Operation sequences during pre-launch and flight phases.

Refer to Annexure-2 for detailed spacecraft questionnaire

4.8 Spacecraft Grounding Requirements

Proper grounding between the various parts of the Spacecraft as well as between Vehicle and Spacecraft shall be ensured to avoid undesirable build up of static charage. The contact resistance between spacecraft and vehicle shall be less than 10 milli Ohms.

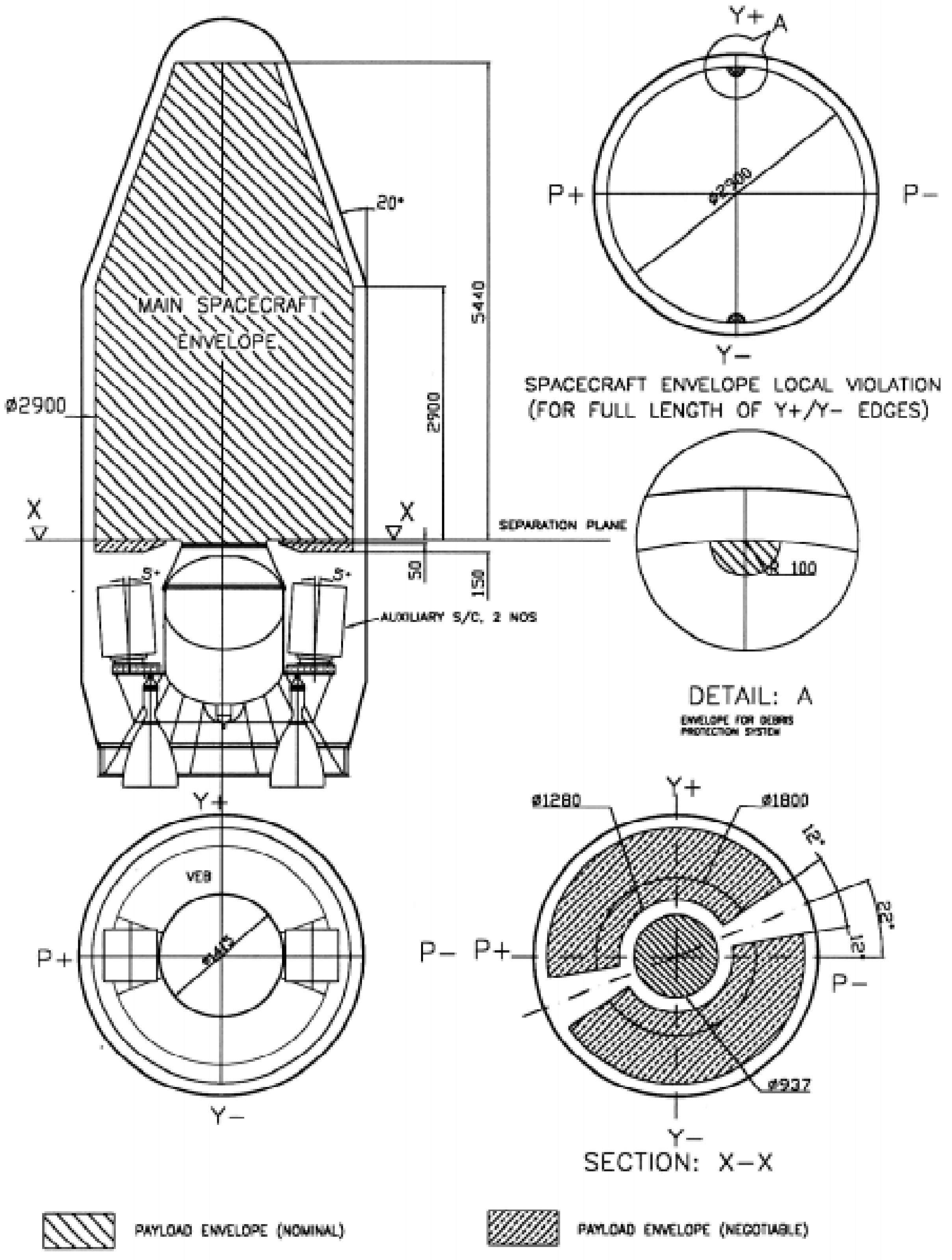


Fig. 4.2 Payload Envelope for PSLV

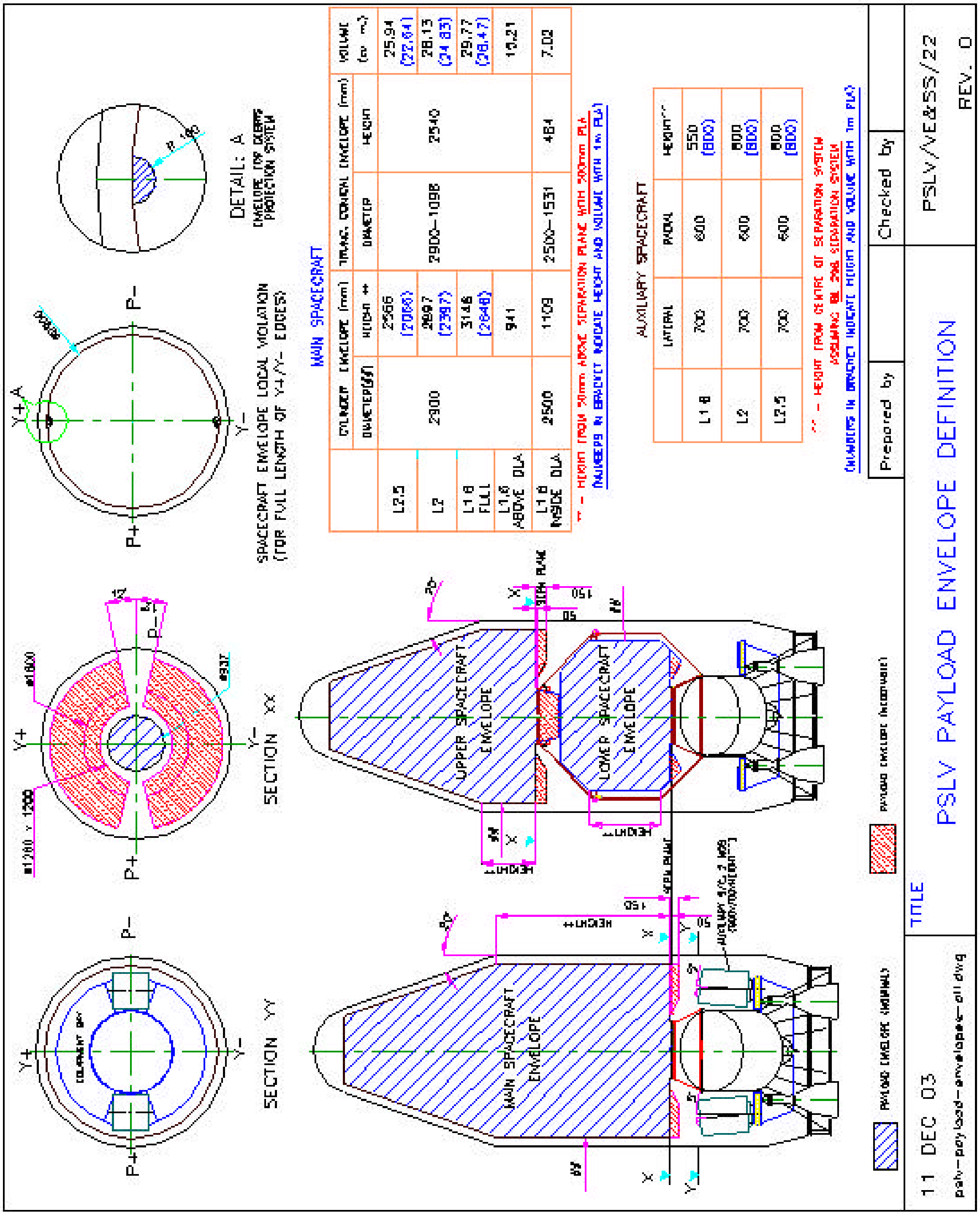


Fig. 4.2 A – Payload Envelope Options

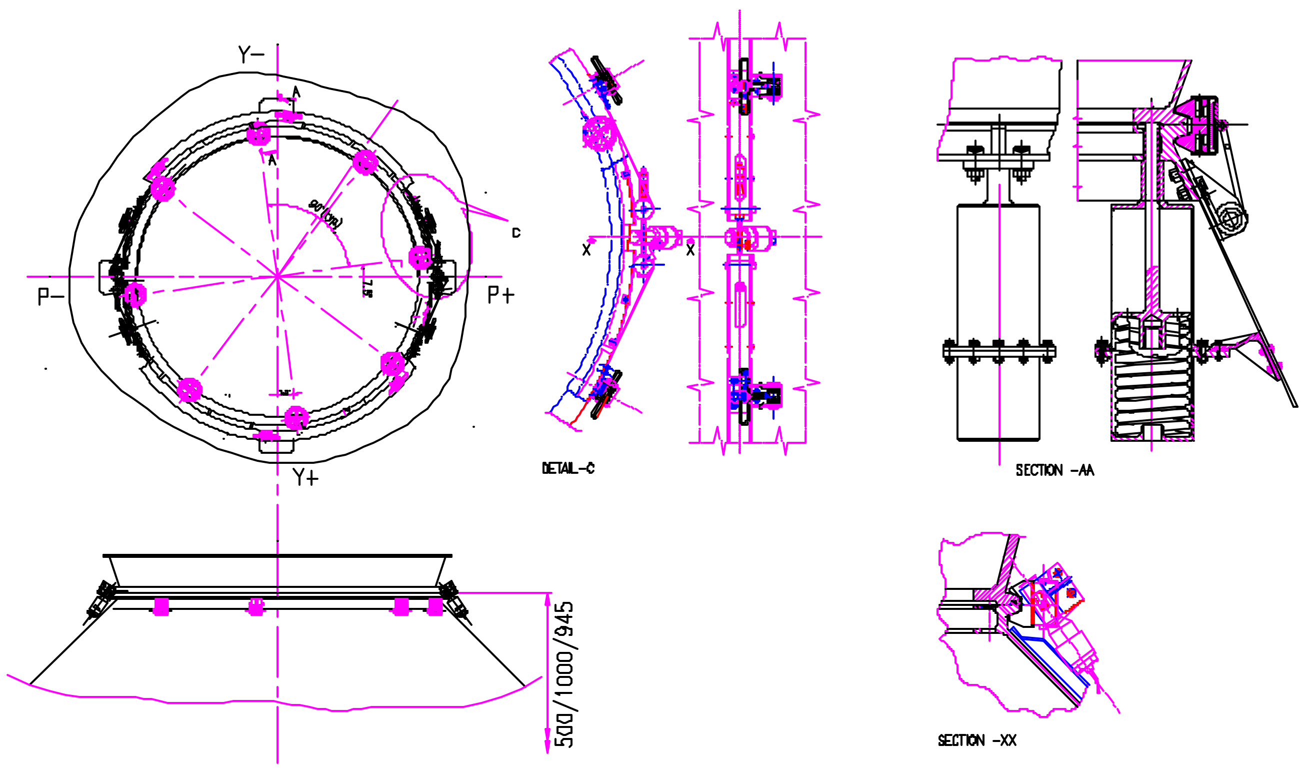


Fig 4.5 S/C separation system assembly (Typical)

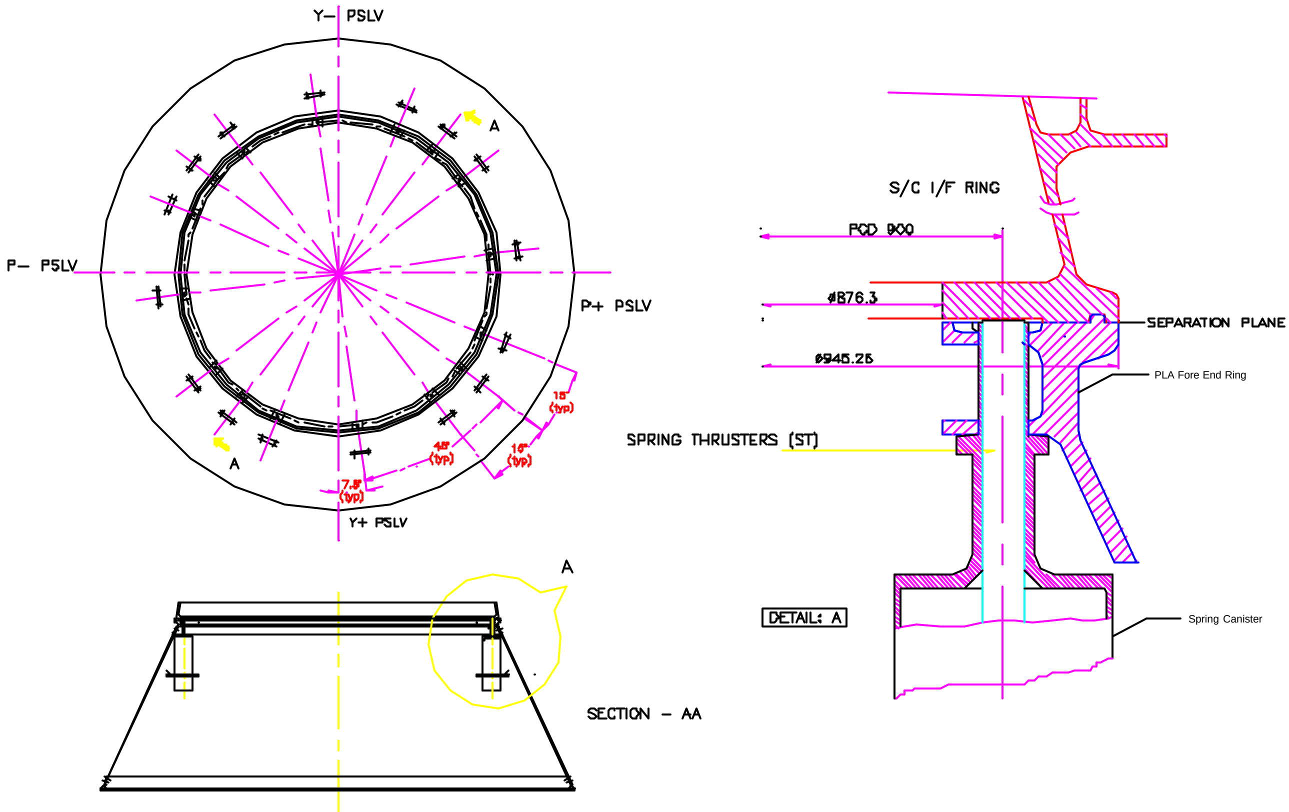


Fig. 4.4 – Satellite rear frame interface requirements

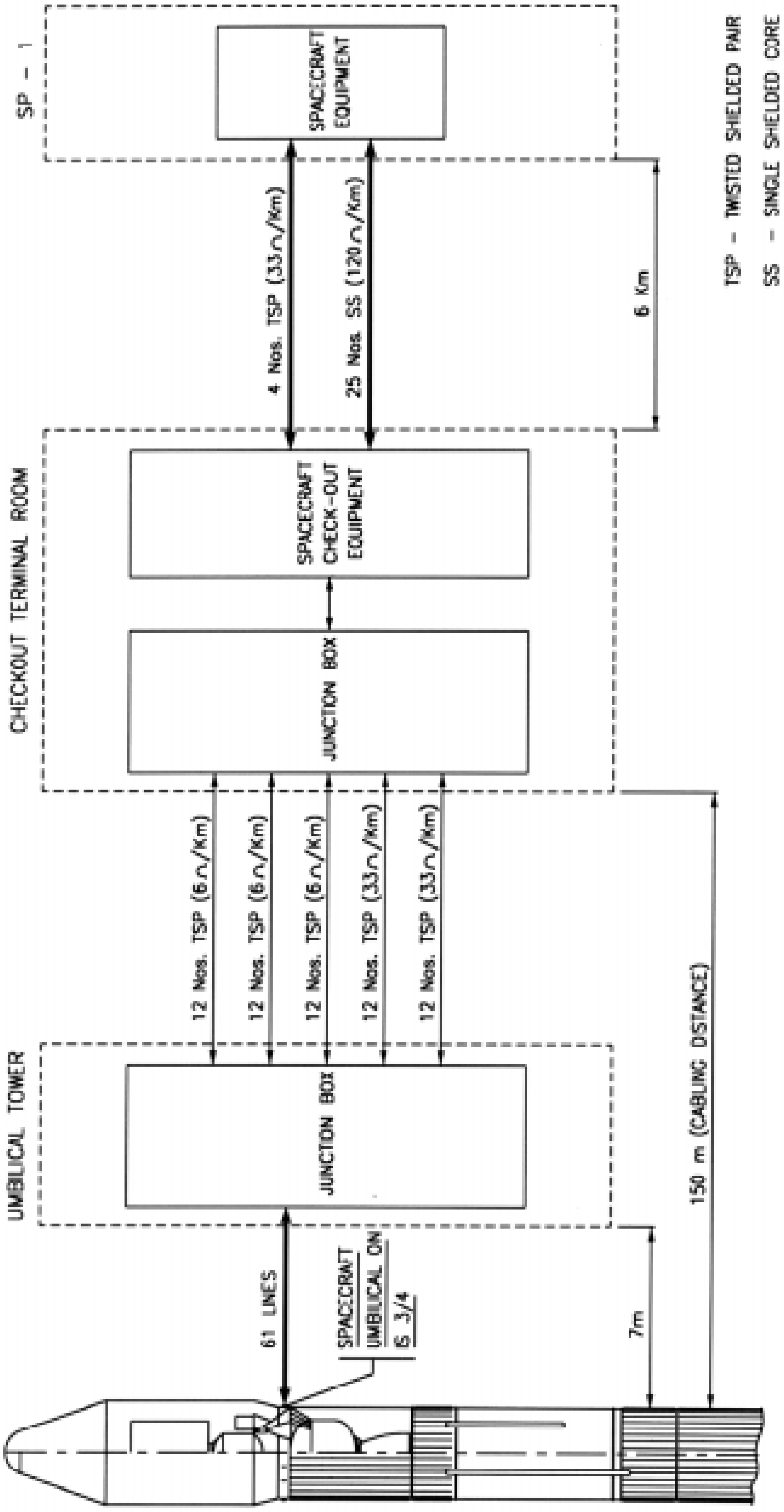


Fig. 4.5 Spacecraft Checkout Cabling Interface

Section 5

Provisions and Interfaces for Launch of Auxiliary Satellites

Section - 5

PROVISIONS & INTERFACES FOR LAUNCH OF AUXILIARY SATELLITES

The launch opportunity for the micro satellites, falling in the mass range of 50 to 150 kg normally has to be linked with that of a primary satellites since a dedicated launch may not exist and/or cannot be afforded. To cater to this type of requirement, provision is made in PSLV to accommodate and launch two Auxiliary Satellites in every flight in piggyback mode. The provision has been made on the Vehicle Equipment Bay (VEB) of PSLV. The Auxiliary Satellites will be offered the launch opportunity based on the availability of spare payload capability in any particular launch.

This Section is intended to provide the users with technical information necessary to assess the compatibility of the Auxiliary Satellite with PSLV.

5.1 Payload Capability

PSLV can carry a maximum of two Auxiliary Satellites in every launch with a mass not exceeding 150 kg each. The above mass does not include the mass of separation system.

Dimensions:	600 mm x 700 mm x 850 mm (h) maximum. The requirement of appendages over and above the given dimensions need to be discussed on a case by case basis.	
Centre of Gravity constraints	Longitudinal	< 450 mm from mounting plane of the Spacecraft.
	Lateral	within ±5 mm with respect to geometric centre of the separation system

5.2 Orbit and Separation Sequence

PSLV can perform the launch of primary Satellites either into a Sun-synchronous Polar Orbits (SSPO) or a Low Earth Orbit (LEO) at a range of inclinations or into a Geosynchronous Transfer Orbit (GTO).

The orbit of the Auxiliary Satellites will be subordinated to the orbit requirement of the primary satellite in each launch.

The nominal separation sequence (Fig. 5.1) is as below:

(i) Shut-off of PSLV Fourth Stage after attaining Injection conditions

(ii) Separation of Primary Satellite

(iii) Collision Avoidance Manoeuvre for the Auxiliary Satellite-1

(iv) Separation of Auxiliary Satellite-1

(v) Collision Avoidance Manoeuvre for Auxiliary Satellite-2

(vi) Separation of Auxiliary Satellite-2

5.3 Mounting Provision

Auxiliary Satellites are mounted on VEB at two locations 180 deg. apart. VEB is an annular deck plate made of Aluminium honeycomb panels, and mounted around the propellant tank of the Fourth Stage. Two 40 deg. sector deck plates of 110 mm thickness are identified to accommodate the Auxiliary Satellites. Fig.5.2 shows the scheme of mounting for Auxiliary Satellites. The details on the interface details between the Auxiliary satellites and the separation system is shown in Fig. 5.3.

To provide safe clearance from the Vehicle during separation, the Auxiliary satellite is provided with 4 to 5 deg. tilt away from Vehicle longitudinal axis. This tilt is achieved by providing an interface ring placed between the deck plate of VEB and separation system. Fore-End ring of the separation system interfaces with the Auxiliary Satellite.

5.4 Separation System

ISRO offers qualified and flight proven separation systems based on Ball-Lock type, which can cater to the Auxiliary Satellites in the mass range of 50 and 150 kg.

5.4.1 Ball lock separation system

The separation system is based on "Ball Lock" mechanism. The configuration of the system in assembled condition is shown in Figures 5.4 - 5.6. The systems consist of an upper ring adapter (Satellite adapter) and a lower adapter (Vehicle adapter) locked together by means of steel balls. A retainer ring is used for locking and unlocking. Helical compression springs are positioned in between the rings to provide jettisoning energy to the Spacecraft as well as to aid the release of the steel balls. The system is unlocked by rotating the retainer ring using pyro thrusters by a small angle, which makes the holes on the retainer ring and the lower ring adapter to align. The balls are pushed into the retainer ring due to offset provided to its location within the assembly and also due to separation force of the springs. Two pyro thrusters are provided for redundancy. The balls and retainer ring are contained in the lower ring after separation. The system is compact and debris free and has low separation shock.

Three separation systems based on the ball lock mechanism have been qualified for use with 50 kg to 150 kg Class Auxiliary Satellites. The specifications are as below.

Table 5.1
Specifications for Separation System based on Ball Lock Mechanism

Class of Satellite	Designation	Mounting PCD (mm)	No. & size of bolts	Nominal Separation velocity (m/s)	Mass (kg)	Mass retained on the Satellite (kg)
50 kg	IBL-230	230	M6 x 6	1	3.5	1
150 kg	IBL-358	358	M6 x 12	1	5	1.5
120kg	IBL-298	298	M6 x 12	1	4.5	1.1

5.5 Electrical Interface

The electrical interface between the Auxiliary Satellites and PSLV consists of one umbilical link which is available for each Auxiliary Satellite for battery trickle charging and minimal checkout. The connector to be used is:

DBAS	74	12	0 SN	059	On Spacecraft side
DBAS	78	12	0 PN	059	On the Vehicle side

The umbilical link is extended from Vehicle to Checkout Terminal Room (CTR) and is accessible up to T0-60 Hrs for Checkout and battery trickle charging.

Out of 12 pins of the connector, Pin Nos. 3 and 4 are to be shorted on the Satellite side with a shorting loop. This connection on the Vehicle side will be used to monitor Satellite separation through Vehicle telemetry in flight.

The separation plane connector for the ball lock separation systems (both 230 and 358 dia systems) is located at the centre of the separation ring.

Refer Figure 5.7 for pin allocation of the electrical connector and wiring scheme.

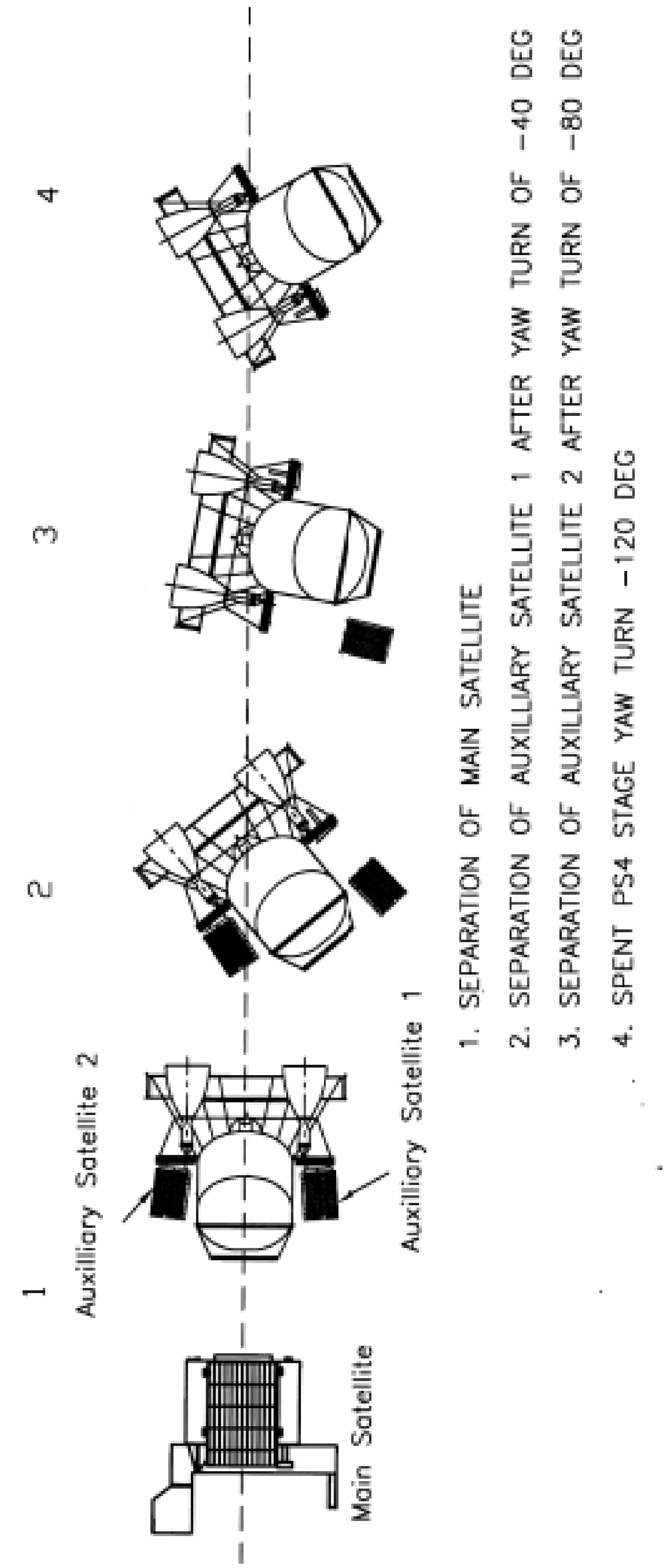


Fig. 5.1 Sequence of Satellites Separation

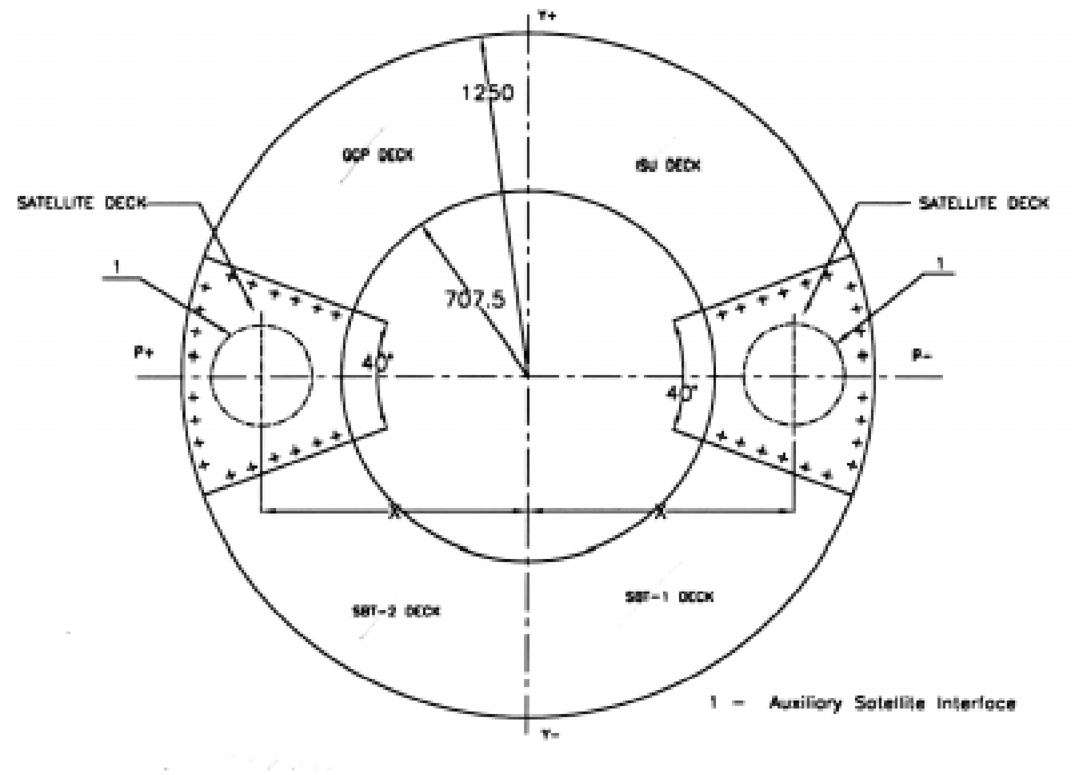


Fig 5.2 Scheme for mounting Auxiliary Satellites on EB

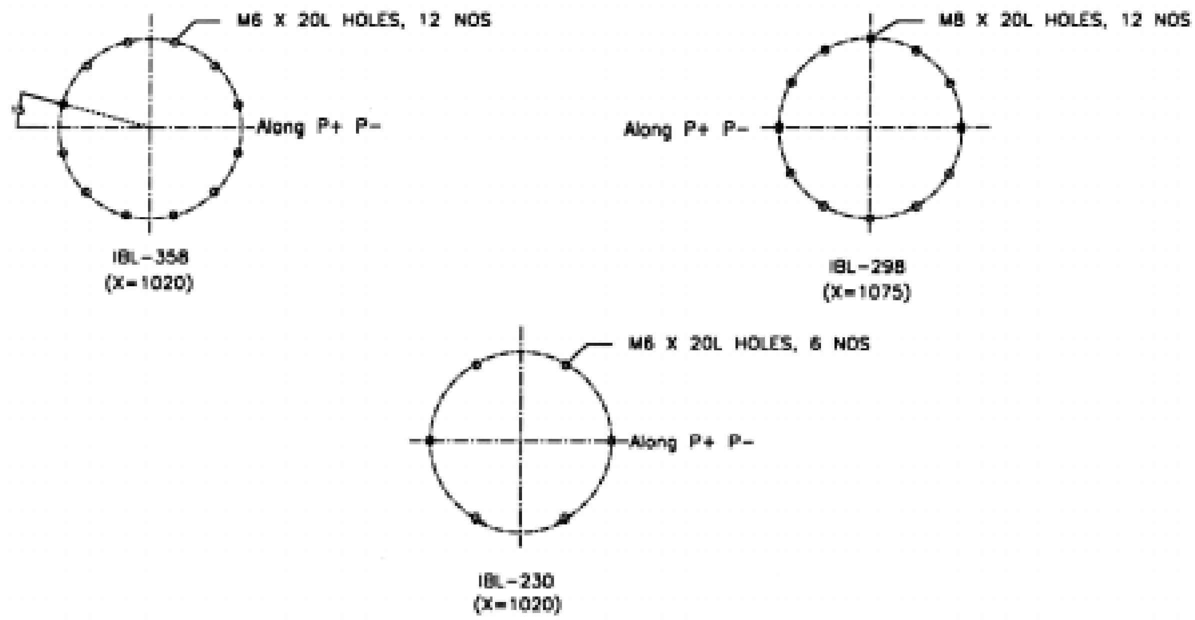


Fig 5.3 Interface definition between Auxiliary Satellite and the separation system

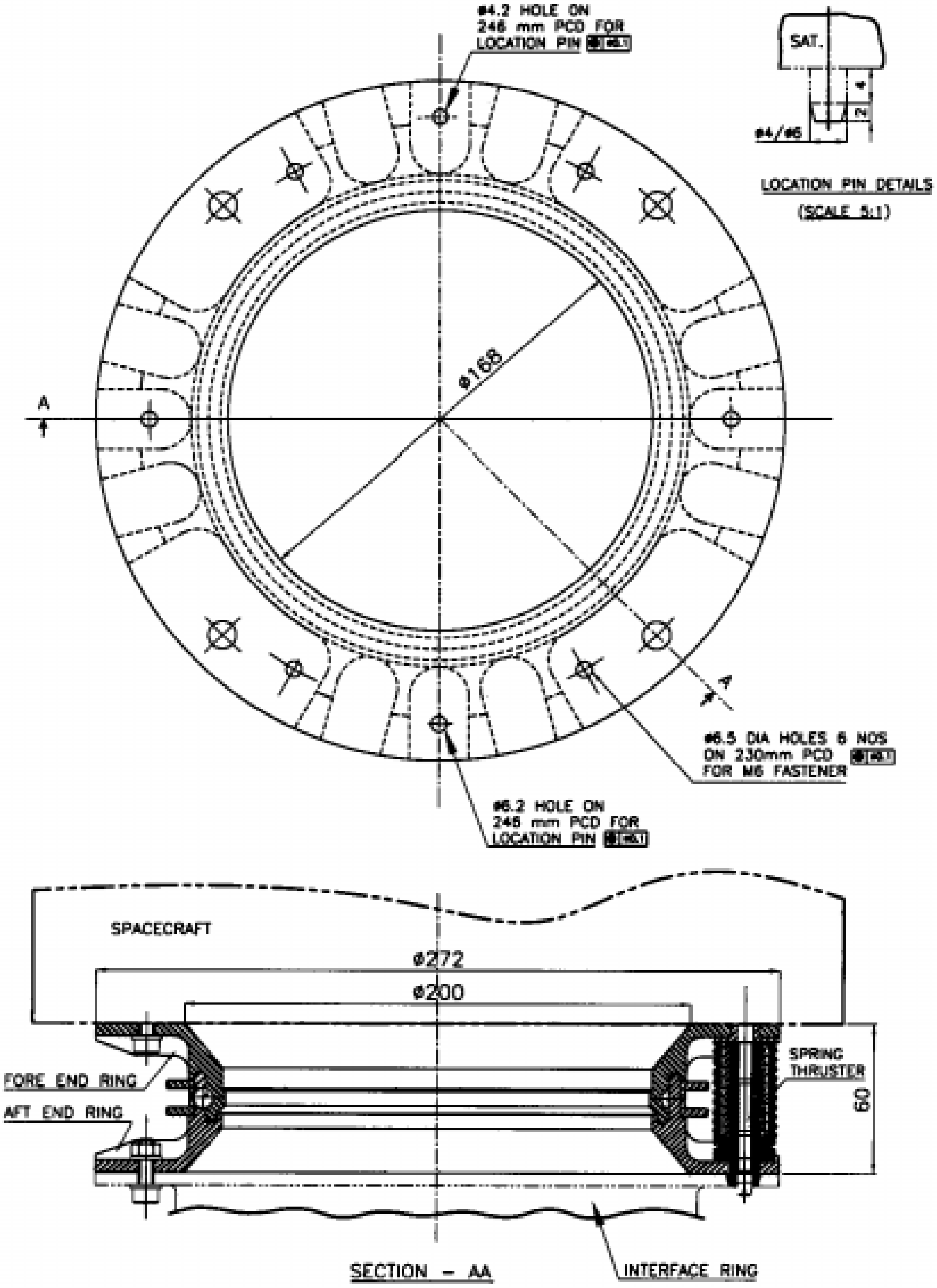


Fig: 5.4 Interface Definition for Auxiliary Satellite of 50 kg Class

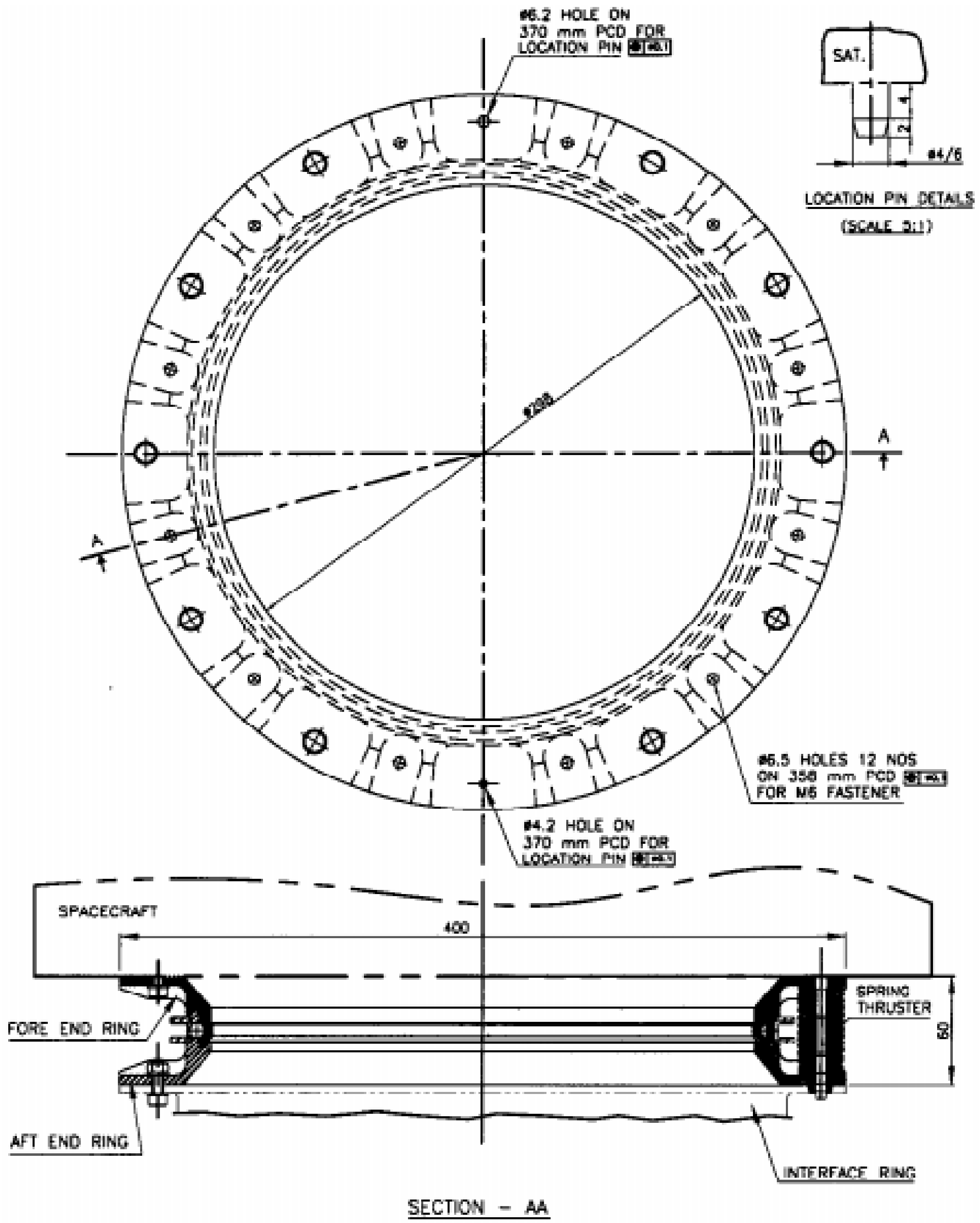


Fig: 5.5 Interface definition for Auxiliary Satellite of 150 kg Class

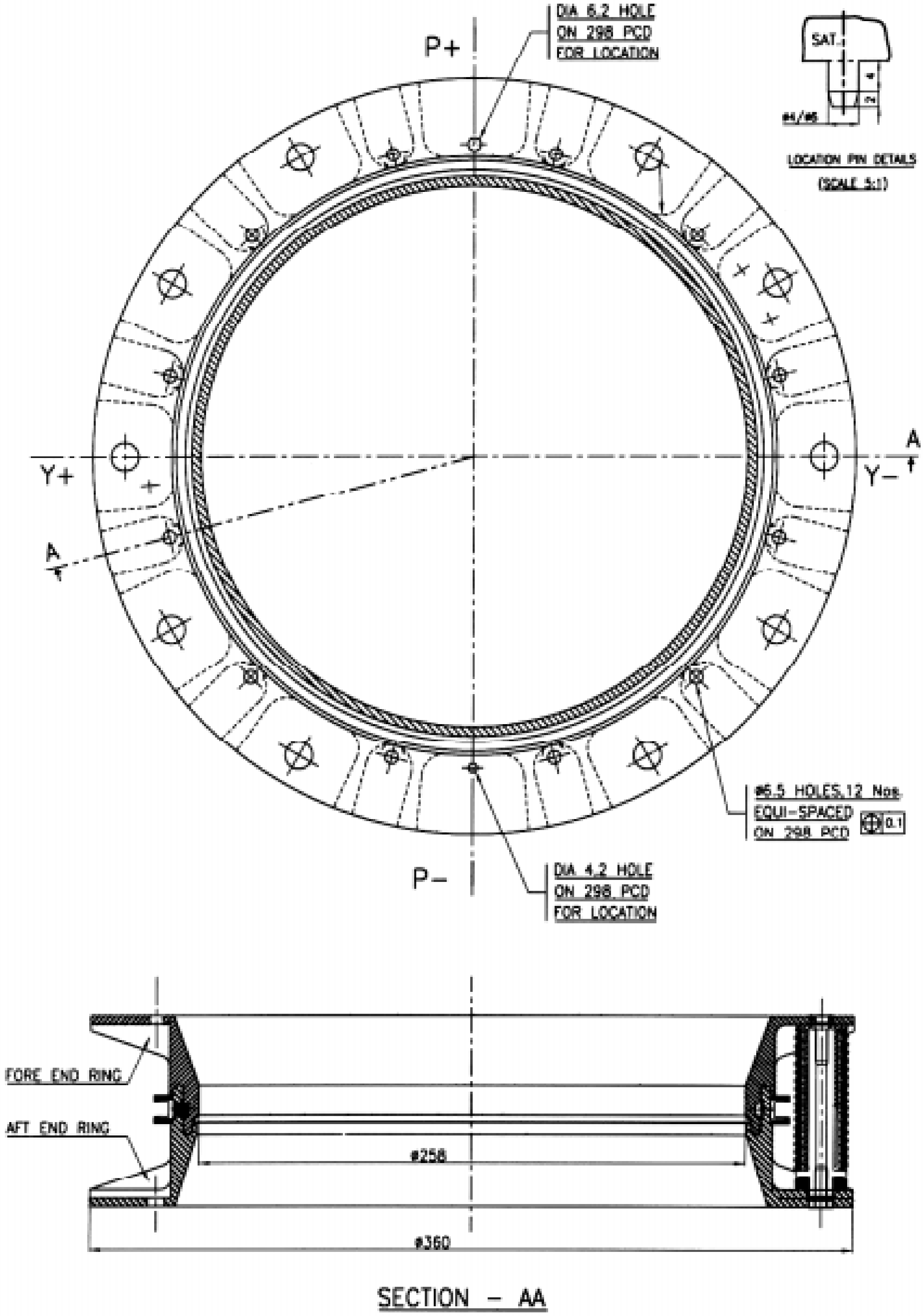


Fig. 5.6 Interface Definition for Auxiliary Satellites of 120 kg Class

5.6 Flight Environment Levels:

During the flight, the Auxiliary Satellite is subjected to both inertial and dynamic loads. The design of the primary structure of the Auxiliary Satellite shall be verified for compliance with the following levels.

5.6.1 Quasi static loads

Longitudinal acceleration (Static + Dynamic)	:	+ 7/-2.5 g
Lateral acceleration (Static + Dynamic)	:	± 6g

The design check shall be carried out by applying a load factor of 1.25 to the above levels. Lateral loads are to be considered acting simultaneously with the longitudinal loads and all loads act at CG of the Satellite.

5.6.2 Sine vibration test levels

Levels at the interface of the Spacecraft and the separation system are given below.

Table 5.2
Sine Vibration test levels

	Freq. Range (Hz)	Qualification level	Acceptance level
Longitudinal axis	4-10	10 mm (0 to Peak)	8 mm (0 to Peak)
	10-100	3.75 g	3 g
Lateral axis	2-8	10 mm	8 mm
	8-100	2.5 g	2 g
Sweep rate		2 Oct / min	4 Oct / min

5.6.3 Random vibration test levels

Table 5.3
Random Vibration test levels

Frequency	Qualification PSD (g^2/Hz)	Acceptanace PSD (g^2/Hz)
20	0.002	0.001
110	0.002	0.001
250	0.034	0.015
1000	0.034	0.015
2000	0.009	0.004
GRMS	6.7	4.47
Duration	2min/axis	1 min/axis

5.6.4 Frequency requirements

To avoid dynamic coupling between low frequency modes of the Vehicle and Spacecraft, the Auxiliary Satellite shall meet the following:

⇨ The fundamental frequency in longitudinal axis > 90 Hz

⇨ The fundamental frequency in lateral axis > 45 Hz.

These figures include the influence of the Satellite separation system.

5.6.5 Thermal environment

During pre-launch phase, the maximum power dissipation for each Auxiliary Satellite is not to exceed (TBD) W.

5.6.6 Shock environment

Auxiliary Satellites will be subjected to shock environment at various flight staging events like stage separation, Heat Shield separation and Auxiliary Satellite separation. A typical shock spectrum is given in Fig. 5.8, to which the Auxiliary Satellite has to be tested during qualification phases.

5.6.7 Powering during Launch

The Auxiliary Satellite shall not activate its on-board systems like telemetry /telecommand, either to receive or transmit signals during the total flight phase and up to a minimum period of 30 minutes (TBD) after injection.

5.6.8 Qualification and acceptance tests

The customer shall demonstrate that the Auxiliary Satellite endures the flight environment by conducting qualification and acceptance tests on the Satellite.

- Sinusoidal vibration, random vibration and shock tests are mandatory for qualification.

- Random vibration tests are mandatory for the acceptance.

- The test plan established for conducting the qualification and acceptance tests on the Auxiliary Satellite, and the test results shall be provided to ISRO.

5.6.9 Auxiliary Satellite/PSLV fit-check

A fit check for verifying mechanical and electrical compatibility with the defined Interface will be conducted with flight hardware at least 4 months before launch.

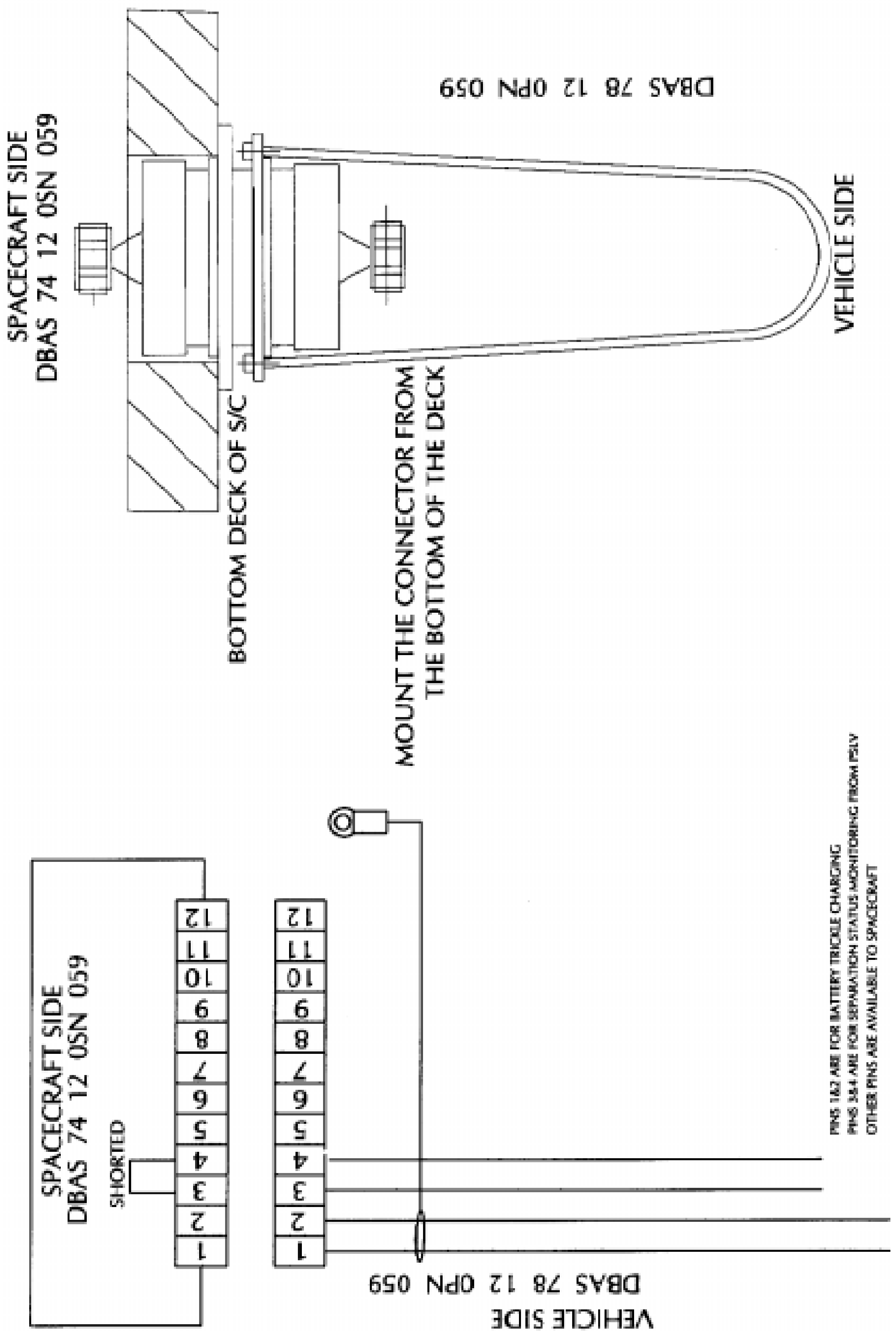


Fig: 5.7 Schematic of Connector Mounting and Cabling

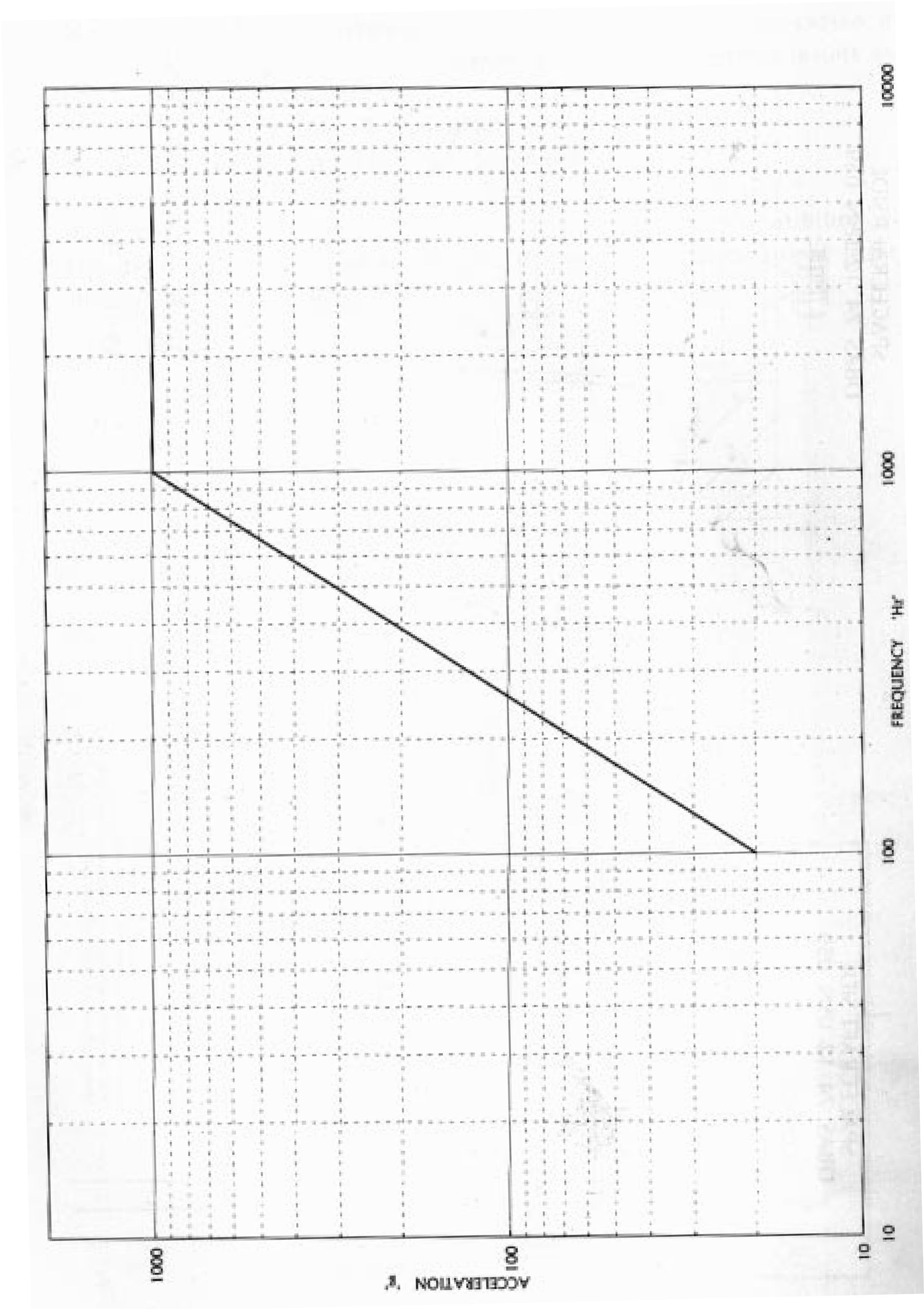


Fig. 5.8 Typical Shock Spectrum for Auxiliary Satellites

Section 6

Mission Planning and Integration Schedule

Section - 6

MISSION PLANNING & INTEGRATION SCHEDULE

6.1 Mission Planning

The mission planning for any Spacecraft mission starts approximately 24 months ahead of the scheduled launch date. The preliminary mission specifications and other requirements of the customer are studied by ISRO and at the end of mutual discussions, both technical and commercial, launch services contract is signed. An interface control document is generated covering various aspects of interfaces between Satellite and Vehicle and integration operation involving Spacecraft which is mutually agreed upon and signed. Preliminary mission analysis is carried out by ISRO on the various aspects of the mission covering Spacecraft separation sequence, collision avoidance manoeuvre, separation dynamics studies, long term as well as short term, coupled load analysis, electromagnetic compatibility studies etc.

Spacecraft mathematical model in the required format is provided by the Satellite Agency for ISRO to carry out coupled load analysis with the launch vehicle to ensure that requirements of structural dynamics are met and loads and environments are within acceptable limits. Match mate checks are made with the flight spacecraft and separation system at the appropriate time to preclude any mismatches. Final mission analysis is carried out by ISRO with realised/updated data on vehicle and Spacecraft to ensure that the mission objectives are met. The Spacecraft operation document at launch site/pad are prepared by Spacecraft agency based on which a combined Spacecraft and vehicle operations plan is generated for the launch operations.

The post flight analysis data, both quick look as well as detailed are provided to the Spacecraft agency by ISRO in the agreed format covering the preliminary orbit data for the Spacecraft and performance of the vehicle.

6.2 Vehicle Integration Schedule

The integration and launch of PSLV requires 60 days on the launch pad starting with the placement of the core base shroud and the nozzle end segment of the first stage motor on the launch pedestal. The PS1 segments are assembled on the pad vertically inside the Mobile Service Tower. The six strapon motors are attached after completing the first stage motor integration. This is followed by the two SITVC storage tanks and the RCS packages. The integrated second stage is transported to the pad in its special trailer and is assembled to the first stage. The combined module of fourth stage with vehicle equipment bay and third stage solid motor is then assembled followed by mating of Satellite. Assembly of heat shield fairings complete the vehicle integration. After checkout of the vehicle/satellite system at the designated time vehicle propellant filling is carried out. The Mobile Service Tower is retracted to its parking location at the start of the terminal count down operations. The final checkout and launch is carried out through an automatic launch sequence. The final operations are remotely controlled from the Launch Control Centre located approximately 6 km from the launch pad.

6.3 Spacecraft Integration Schedule

The spacecraft operation at launch site are carried out in dedicated facilities where checkout and propellant filling operations can be completed. The Satellite is moved to the Pad 10 days prior to launch. The spacecraft operations schedule is detailed in Fig 6.1.

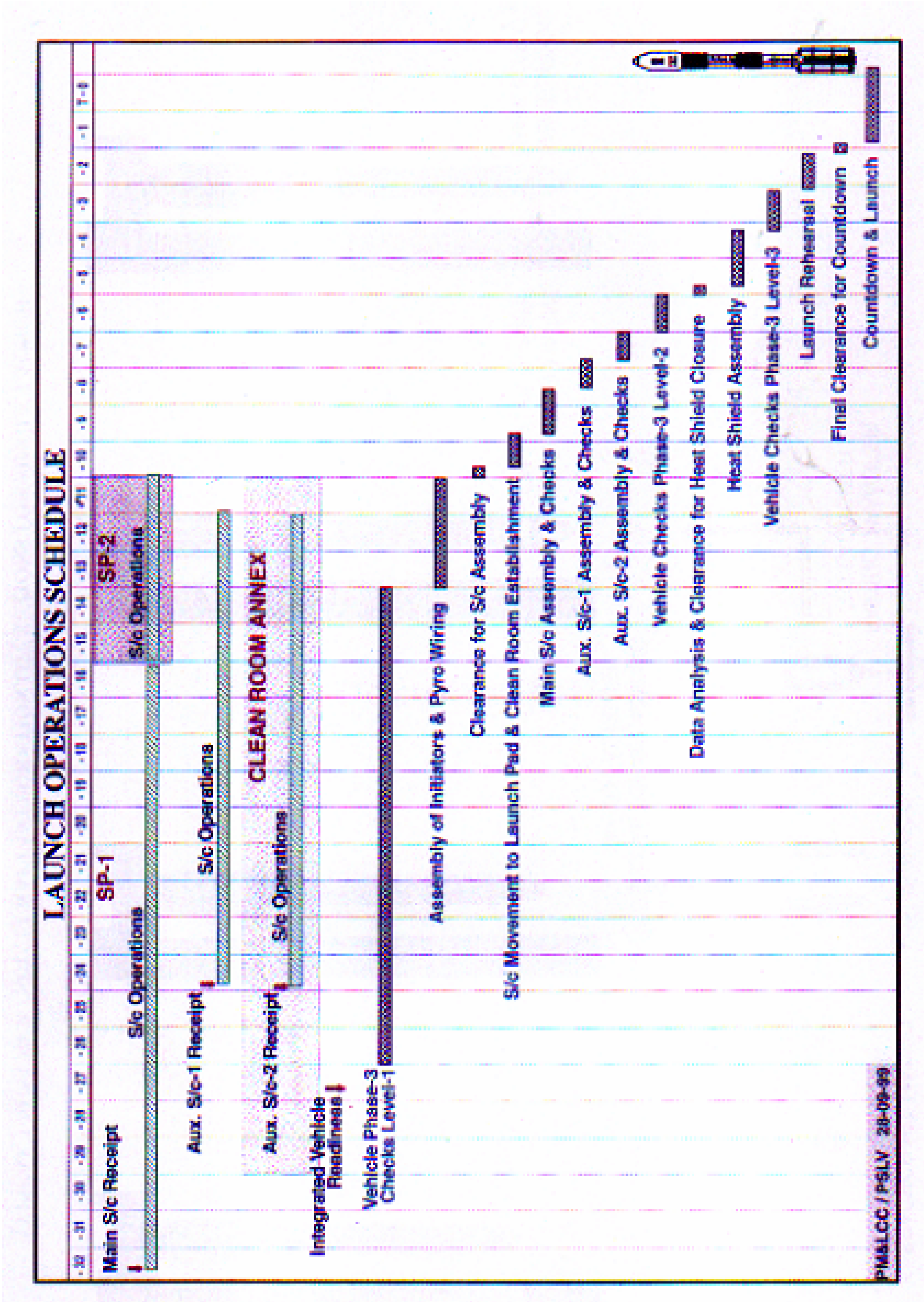


Plate 12: Launcher/Spacecraft Integration operations for single launch
Fig. 6.1 Launch Operations Schedule

Section 7

Documentation

Section - 7

DOCUMENTATION

Planning and execution of spacecraft mission requires timely preparation and submission of relevant documentation such as system description, mission requirements, safety data, interface details etc. The mission planning would start with the feasibility study by the customer for a launch on PSLV and continues till the post launch orbit data submission by ISRO.

This Section deals with broad guidelines for the preparation of the documents.

7.1 Request for Launch Services (RLS)

The customer is required to issue a Request for Launch Services to ISRO, based on which a proposal for Launch Services will be made. The RLS shall include the following information.

- Mission characteristics (orbit, and accuracies)
- Spacecraft data (Mass, dimensions, and interface provisions)
- Schedule requirement for launch (including spacecraft development status).

A typical format for defining the spacecraft requirements, interface details and preliminary safety data is given in Annexure-2.

7.2 Launch Services Agreement (LSA)

Based on the customer's RLS, ISRO will submit the proposal for launch service using PSLV. A Launch Services Agreement (LSA) shall be entered into by the customer and ISRO, after negotiating the details. LSA shall be binding on both the parties.

7.3 Interface Control Document (ICD)

All the interfaces between the spacecraft and the launch vehicle are included in the ICD. The document shall include mechanical, electrical, RF interfaces, mass and dimensional constraints, environmental test levels and schedules for activities.

ICD is the controlling document jointly prepared by the customer and ISRO. Any modification to the document can be made only with mutual consent.

7.4 Spacecraft Mathematical Dynamic Model

The mathematical dynamic model of the Spacecraft has to be supplied by the customer. A coupled load analysis between the Spacecraft and launch vehicle will be carried out by ISRO based on the above. The mathematical model has to be supplied in NASTRAN DMIG format and the specifications of the model are as follows.

- A model with interface nodes at the mounting locations.
- Restitution matrices to compute the dynamic displacement at the four extreme corners of the spacecraft and also for the critical locations where response is to be evaluated.
- Co-ordinate system as followed in the Interface Control Document.
- Units to be in SI.
- Model size to incorporate frequency upto 125Hz.

7.5 Spacecraft Environmental Test Document

The customer shall prepare the environmental test plan document compatible with the test specifications detailed in the ICD. The results of the qualification and acceptance tests shall be jointly reviewed by ISRO and the customer.

7.6 Safety Requirements

The customer shall prepare and submit to ISRO a document detailing all the aspects of the Spacecraft, which have bearing on the safety during pre-launch and launch phases. The details to be covered in the document are elaborated in Section-8.

7.7 Preliminary Mission Analysis

ISRO shall carry out mission studies such as flight sequencing and orbital dispersions, separation dynamics, coupled load analysis, injection attitude orientation requirements, Electromagnetic compatibility analysis, and the relative motion of the separated Satellites (in case of multiple Satellite launch). These results will be documented and presented to the customer by ISRO.

7.8 Spacecraft Operation Plan at Launch Site

The operations to be carried out on the Spacecraft at the launch site are to be documented and supplied to ISRO. This document shall cover transportation, spacecraft preparation, propellant filling, pre-launch checkout and the launch phases. The support requirements shall also be detailed in the document.

7.9 Interleaved Operations Plan

Based on the operations plan document of the customer, ISRO will finalise Interleaved Operations of the Vehicle, Range and the Spacecraft. In the case of multiple spacecraft launch, the combined operations of the spacecraft(s) shall also be finalised. ISRO will propose the interleaved operations document to the customers.

7.10 Final Mission Analysis

ISRO shall carry out detailed mission analysis with updated data on Vehicle and Spacecraft, launch time, ground environment etc. The performance and orbital parameters will be provided by ISRO to the customer.

7.11 Terminal Countdown Document

A document covering the operations towards the terminal phase of the count down shall be prepared by ISRO in consultation with customer (Spacecraft Agency), as well as Launch Vehicle and Range agencies. The document shall identify all the operations and go-ahead clearances required with respect to count down time.

The Terminal Count Down document and NO-GO criteria shall be discussed and coordinated with the customer on T-1 day.

7.12 Post Launch Orbit Confirmation

ISRO will provide the customer with Preliminary Orbit details based on On-Board INS data, and Vehicle tracking data. This information will be made available by T+1 Hr.

7.13 Post Flight Analysis

Preliminary and detailed analysis of flight will be carried out by ISRO. Relevant details of the flight analysis, including analysis by specialist teams wherever applicable, will be presented by ISRO to the customer.

Summary of the documentation requirements is given in Table 7.1.

Table-7.1
Documentation requirements

Sl.No.	Document	To be Prepared by	Schedule (C+Months)
1	Request for Launch Services (RLS)	Customer	
2	Launch Services Agreement (LSA)	Customer & ISRO	C_0
3	Interface Control Document (ICD)	Customer & ISRO	C+3
4	Mathematical Dynamic Model	Customer	C+6
5	Spacecraft Environmental Test document	Customer	C+9
6	Safety submission	Customer	C+9
7	Preliminary Mission Analysis Document	ISRO	C+10
8	Spacecraft Operations Plan at Launch Site	Customer	C+12
9	Interleaved Operations Document	ISRO	C+15
10	Final Mission Analysis Document	ISRO	C+20
11	Terminal Count Down Document	ISRO	D_0-1 day
12	Post Launch orbit confirmation	ISRO	T+1 Hr
13	Post Flight Analysis Document : Preliminary : Final	ISRO	D_0+5 days D_0+1 month

C_0 : Signing of Launch Services Agreement

D_0 : Day of launch

T : Launch Time

S. No.	Ref Item No.	Months	Activity
1	7.1		Request for Launch Services
2	7.2	C	Launch Services Agreement
3	7.3	C+3	Interface Control Document
4	7.4	C+6	Mathematical Dynamic Model
5	7.5	C+9	Spacecraft Environmental Test Document
6	7.6	C+9	Safety Submission
7	7.7	C+10	Preliminary Mission Analysis Document
8	7.8	C+12	Spacecraft Operations Plan at Launch Site
9	7.9	C+15	Interleaved Operations Document
10	7.10.	C+20	Final Mission Analysis Document
11	7.11	D-1 day	Terminal Count Down Document
12	7.12	T+1 hr.	Post Launch Orbit Confirmation
13	7.13	D+5 days & D+1 month	Post Flight Analysis Document

Launch: Pre launch | Post Launch

C | C+2 | C+4 | C+6 | C+8 | C+10 | C+12 | C+14 | C+16 | C+18 | C+20 | D | D+1

C - Signing of Launch Services Contract
D - Day of Launch
T - Time of Launch

Fig. 7.1 Documentation Requirements

Section 8

Safety Regulations

Section - 8

SAFETY REGULATIONS

This section lays down the regulations to be followed by payload/spacecraft development agency with respect to safety as per ISRO safety guidelines. The safety policy of ISRO is aimed at carrying out the planned pre-launch and countdown activities (of Satellite and Launch Vehicle) ensuring safety of personnel and facilities.

The responsibility of establishing the safety regulations for activities carried out at SHAR rests with ISRO.

It is mandatory for the spacecraft agency to strictly adhere to safety regulations as stipulated in the Range User's Manual. To ensure this, the following documentation requirements & review protocals should be compiled with by the User.

8.1 Documentation Requirements and Reviews

ISRO requires formal safety documentation containing detailed information on all the hazardous systems and associated operations and transportation. The approving authority for all hazardous operations, which take place at a particular location, rests with the respective safety authority at the location. Co-ordination of all safety clearances for vehicle preparation and integration activities will be obtained by ISRO. It is necessary that the spacecraft agency provide the safety requirements covering all the safety aspects discussed in the subsequent sections of this chapter normally at T-12 months. Further, a formal safety review shall be arranged between ISRO and the customer in which all documents on individual operations, combined integrated vehicle system preparations and pre-launch operations will be reviewed.

ISRO will organize reviews in phases and documents shall be submitted by the customer to cover the objectives of the reviews. The reviews and their objectives are as follows.

Phase-1 : Identify hazardous systems to enable ICD preparation.

Phase-2 : Review to define the controls and clearances for design from safety aspects after safety submission

Phase-3 : Review to verify hardware implementation and controls for safety at spacecraft delivery time.

The customer, at a minimum, shall generate the following information for Phase-1 review and inclusion in ICD.

- ✩ Identification of critical systems and operations with respect to safety
- ✩ Description of safety critical subsystems and their specifications

The Phase-1 safety review will discuss these reports in the light of the applicable safety regulations and will lay down the data requirements for the Phase-2 safety review.

At the end of all reviews, ISRO shall certify the compliance with the requirements and give clearance for integration of the spacecraft with the vehicle.

8.2 Hazardous Systems, Operations and Requirements

The safety requirements will be listed by ISRO and these shall be complied by the customer during the preparation and operation of the systems. The general categories of systems, which are considered hazardous, are listed below:

(i) Ordnance systems

(ii) Propulsion systems (liquid and solid)

(iii) Pressurised system

(iv) Non-ionizing radiation

(v) Ionizing radiation

(vi) Electrical systems

(vii) Hazardous materials (chemical products other than those used for propulsion) that are corrosive, toxic, asphyxiating, flammable, explosive or capable of electrostatic charging.

(viii) Electrical and mechanical ground support equipment used to support hazardous systems or used in hazardous environments.

Design of the systems shall meet the following general technical requirements:

(i) Critical hazards

- No single failure or operator error shall result in contingency or emergency procedures
- Functions must be controlled by alteast two independent inhibits.
- Clearances of operation shall be specifically listed with identification of authority.
- Monitoring of situations

(ii) Catastrophic hazards

- No combination of two failures or operator errors or strong Radio Frequency (RF) signals shall result in accidents leading to emergency or loss of life or damage to vehicle and ground facilities.
- Functions must be controlled by three independent inhibits whenever hazard potential exists and clearances of operations shall be specifically listed along with identification of authority.

8.2.1 Ordnance systems

Ordnance system design and test methods shall meet the requirements laid down by MIL STD-1576 "Electro-explosive subsystem safety "requirements and test methods for space systems".

Specifications

The electro-explosive devices (EEDs) used as initiating elements in ordnance sub-systems are classified by hazard level, i.e., either category "A" or "B". The definitions of category "A" and "B" are given below:

Category "A" – Electro-explosive devices (EED) of category "A" are those which, by the expenditure of their own energy or because they initiate a chain of events, may cause injury or death to people or damage to property.

Category "B" – Electro-explosive devices of category "B" are those which will not, by themselves or by initiating a chain of events, cause injury to people or damage to property.

Design requirements for electro-explosive devices include, among others:

a) Firing circuit shielding for 1A, 1W EEDs must provide a minimum of 40 dB attenuation from 150 KHz to 40 GHz. Waivers will be granted only on demonstration of survival in the range (ground and vehicle) environment such that the EEDs cannot receive more than 40 dB below its DC no-fire power in all modes of use.

b) The category of ordnance devices and their qualification requirements are to be finalized in consultation with ISRO.

c) Electrostatic sensitivity validation tests of all categories of electro-explosive devices are also required.

Documentation Requirement

(i) Complete EEDs statistical data including DC All-fire and No-fire data, RF sensitivity, electrostatic sensitivity test for all category "A" EEDs.

(ii) Detailed information on the construction and specifications of all category "A" EEDs and devices.

(iii) Firing circuitry schematic and description including length, location and spacing of all wires and description of shield termination and gaps.

(iv) Analysis showing that all category "A" EEDs will survive with sufficient margins in the specified environments.

8.2.2 Propulsion systems

Specification of propellants (Solid/Liquid) and other systems are to be provided to ISRO in the safety submission.

8.2.3 Pressurized systems

Specifications

- Functional or leak tests in shop areas: Vessels designed with a minimum burst pressure of four times maximum allowable operating pressure only are acceptable.
- The test pressure shall not exceed one-fourth of the minimum calculated design burst pressure.
- All pressurization to operating pressure require a stabilization time of five minutes prior to personnel exposure.
- A complete log of the pressurization hold time and number of pressurization cycles shall be maintained.
- For pressure vessels and tanks a minimum safety factor of two is required.
- Fittings and lines greater than 40 mm inside diameter shall have factor of safety of 4 for limit loads.
- Fittings and lines less than 40 mm inside diameter shall have factor of safety of 2.

Documentation Requirement

- Total design specifications which shall include service fluid, cycle life and pressure hold time.
- Stress analysis report including analysis based on fracture mechanics.
- Total inspection and quality assurance records.
- Test data which ensure that the minimum factor of safety has been met.
- Handling and operating procedures
- Description of related support equipment.

8.2.4 Non-ionizing radiation–RF emitters (30 KHz to 300 GHz)

Design Requirement

- Personnel are prohibited from performing work on energised equipment, if any part of the body could be exposed to a radiation of 10 mW/cm^2 or more.
- No liquid or gas fuel can be transported within the calculated maximum range of 5 W/cm^2 peak power density of RF radiating equipment
- RF silence is to be ensured for, during installation of electro-explosive devices and during other hazardous operations.
- All electro-explosive devices are to be in RF shielded units during transportation.

Documentation Requirement

- Transmitter power (peak and average), pulse width, pulse codes, pulse repetition frequency, operating frequency.
- Details on requirements of energisation of transmitter.
- Details on time at which electro-explosive devices are to be installed.

8.2.5 Electrical systems

- Refer RANGE USER'S MANUAL (To be reproduced)

8.2.6 Electrical and mechanical ground support equipment

- Refer Range USER'S MANUAL (To be reproduced)

8.3 Waivers

After a complete review of all safety requirements, the spacecraft agency shall determine if any waivers are necessary. A waiver is mandatory to any safety related requirement which cannot be met. The request for waivers shall be submitted to ISRO. ISRO will review the waiver request and decide on its acceptance.

Section 9

Launch Complex Facilities

Section - 9

LAUNCH COMPLEX FACILITIES

9.1 Launch Complex

PSLV is launched from Sriharikota Range (SHAR) located on the east cost of India, about 100 km North of Chennai (Madras). The final preparations of the vehicle and Spacecraft, integration, checkout and launching are carried out in this complex. Necessary launch support facilities are provided by SHAR Range for the vehicle and Spacecraft. A general layout of the launch complex and the facilities available are shown in Fig. 9.1. The geodetic latitude and longitude of SHAR range are: 13.2° N & 82° E. respectively

SHAR is well connected by rail and road to the nearest international airport at Chennai, which is approximately 100 km south of SHAR. Spacecraft transportation has to be by road in a suitable container and transportation trailer.

9.2 Facilities at Launch Complex

The following Satellite Integration Facilities are available at SHAR for spacecraft preparation.

★ Spacecraft preparation facility SP-1

★ Spacecraft preparation facility SP-2

★ Spacecraft preparation facility SP-3

Location of the facilities at SHAR is given in Fig.9.1

The spacecraft may undergo a detailed checkout in the Satellite building SP-1 before being shifted to SP-2 for propellant filling. This spacecraft is moved to the service structure at the launch pad (SP-3) around T-10 days.

9.2.1 SP-1 facility

SP-1 Facility is located approximately 6 km from launch pad and close to the Mission Control Centre (MCC). Fig. 9.2 gives the layout of the SP-1 facility. This facility is primarily meant for carrying out initial preparation and checkout of the satellite.

SP-1 has a 100,000 class clean room of dimensions 17x15x8m. The Satellite Checkout bay has a dimension of 12x6 m and is located adjacent to the clean room. The clean room has separate entries for equipment and for personnel, with proper interlocking arrangements. A brief specification of the clean room is given below.

Cleanliness	:	100,000
Temperature	:	21 to 23 degree C
Humidity	:	45%-55% RH
Lighting	:	300 lux minimum.

Single phase 230 V and 3 phase 440 V at 50 Hz supply with a number of outlets are provided. A small fitting shop, an electronics laboratory and a conference room are also provided. A sheltered area of 300 sq.m. adjacent to this building is available to unload the Satellite and equipment immediately on arrival at the site.

9.2.2 SP-2 facility

SP-2 facility is designed primarily for propellant filling operations. This facility includes a clean room with interlock arrangements for material entry. Size, cleanliness, temperature, humidity control and lighting provisions are same as those in SP1 facility.

The clean room has propellant loading area of 6m x 4m. Breathing outlets at appropriate intervals around the filling area and an emergency exit with an adjoining personnel shower provision are also provided at SP-2. To facilitate Satellite checkout during and after the filling operations, a room of dimensions 6mx8m for installation of checkout equipment is also provided. Fig.9.3 gives the layout of the SP-2 facility.

9.2.3 SP-3 facility

From the SP-2 facility, the Satellite will be moved to the SP-3 facility located at the top of the Mobile Service Tower (MST) for integrating it to the launch vehicle. The SP-3 is located at 41m level of MST. It has provision for storing an empty transportation container, and has a personnel entry airlock. The specifications for the SP-3 facility for cleanliness, temperature, humidity and lighting are same as SP-1 and SP-2 facilities. The launch vehicle – spacecraft interface height above the SP-3 platform level is approximately 2.5 meters. Suitable provisions exist to get close access to the Satellite and vehicle interface. At the base of the MST, a pressurization cart may be located in a room of the size 5mx5mx4m which can be used for pressurizing spacecraft control and propulsion systems. Location of SP-3 facility in the MST is shown in Fig.9.4 and its layout is given in Fig.9.5.

9.3 Logistics

SHAR Range is located in the Sriharikota island which houses the Launch Complex, tracking facilities, mission control centre and living accomodation for the operating and supporting staff. A well furnished guest house and hostel with all modern amenities like air-conditioning, telephone, running water cater to requirements of visitors including customers. Easy access to international telephone and telex/fax services is available to customers.

Chennai which is the nearest metropolitan city, is approximately 80 km from SHAR and is accessible by rail/road. Chennai is connected to important international destinations by air.

The temperature at SHAR varies between 25°C to 40°C during the year with humidity ranging from 75% to 90%.

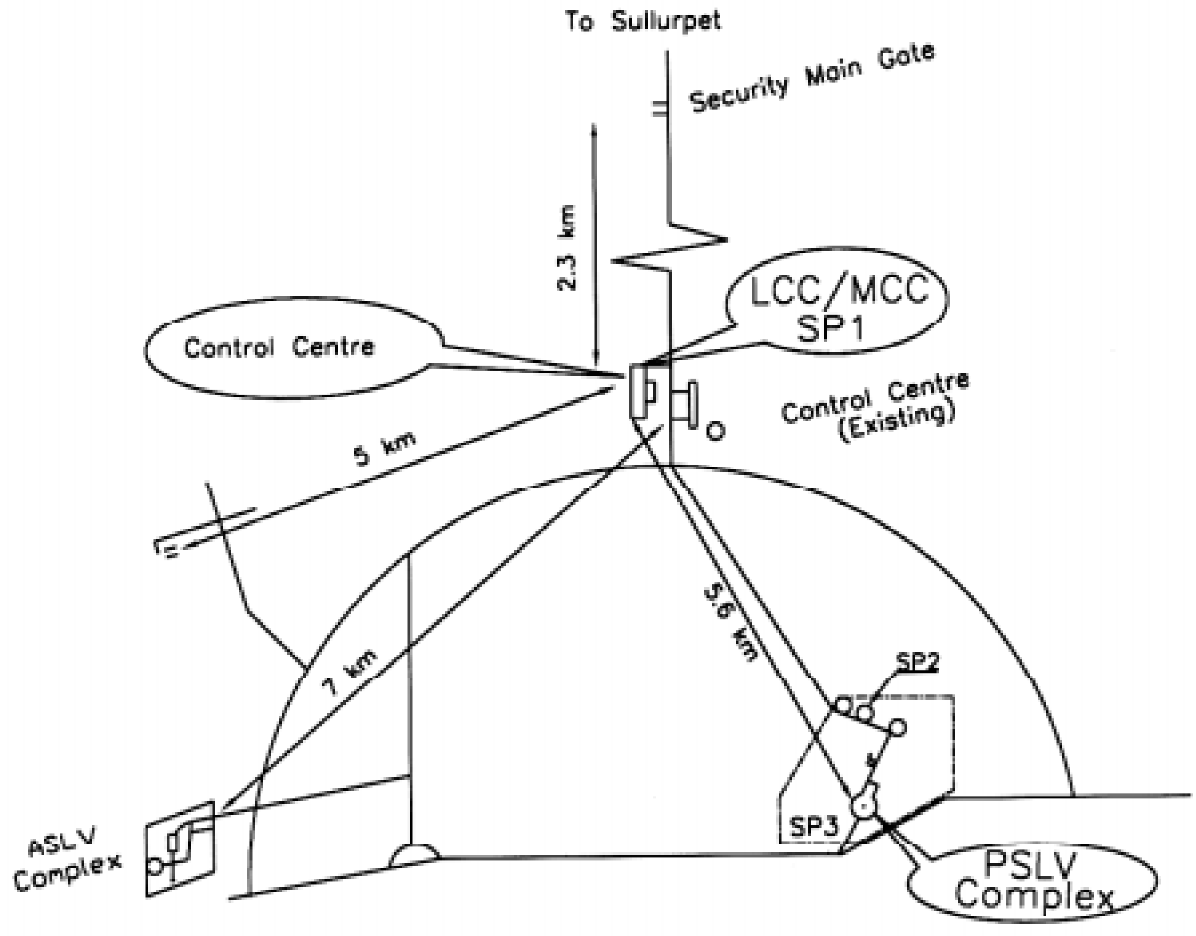


Fig. 9.1 Launch Complex Facilities

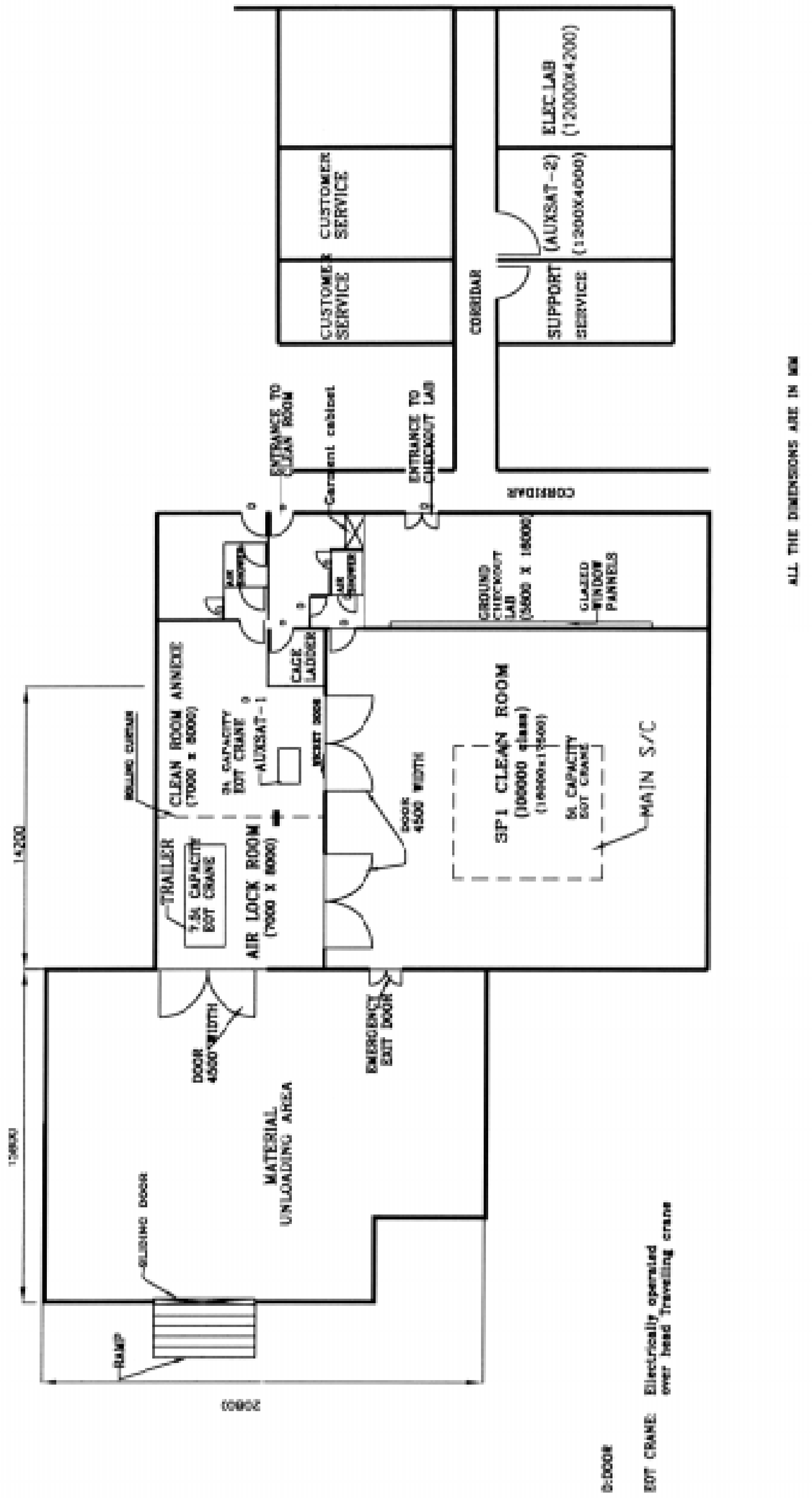


Fig. 9.2 SP-1 Facility

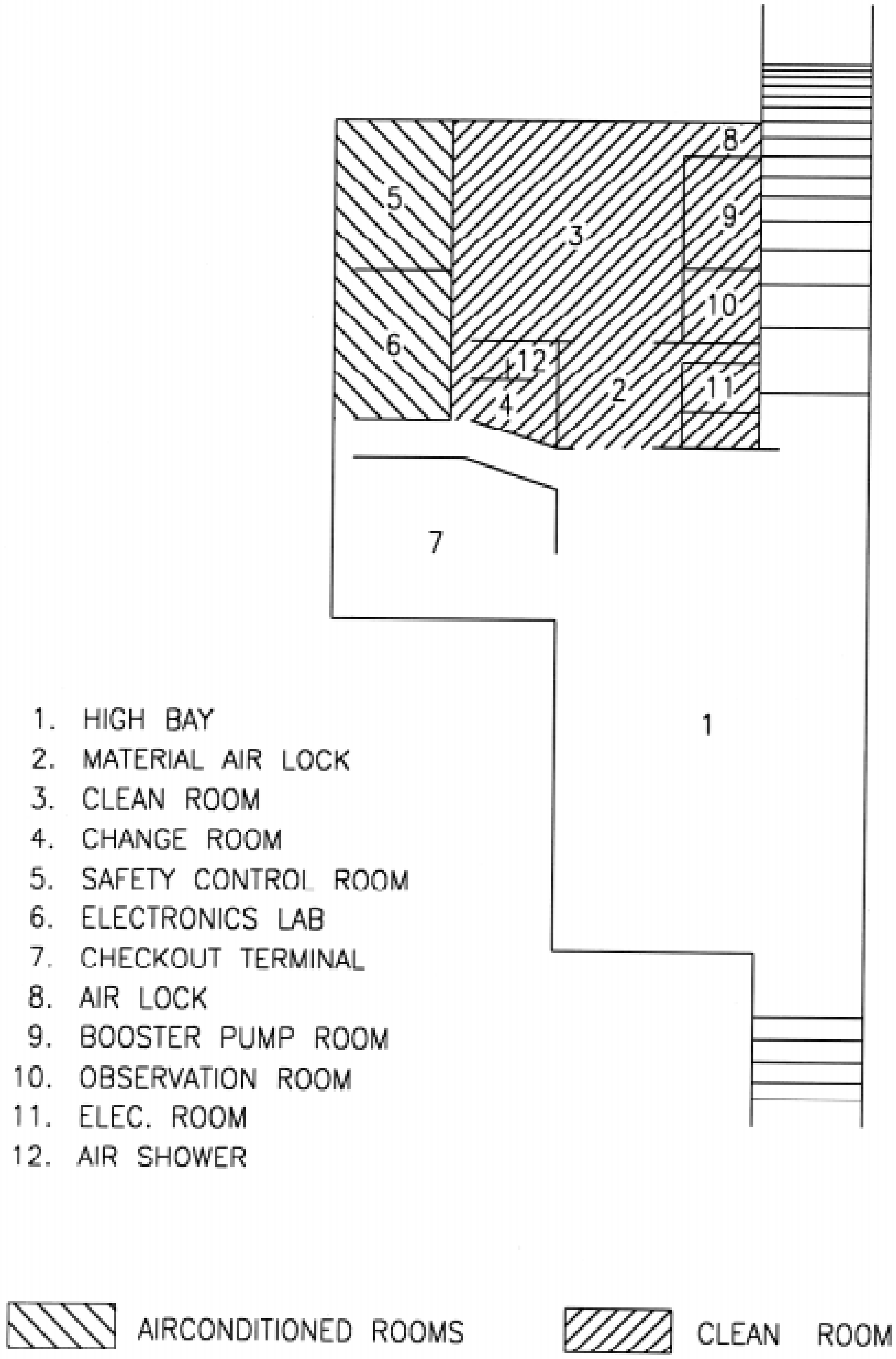


Fig. 9.3 SP-2 Facility

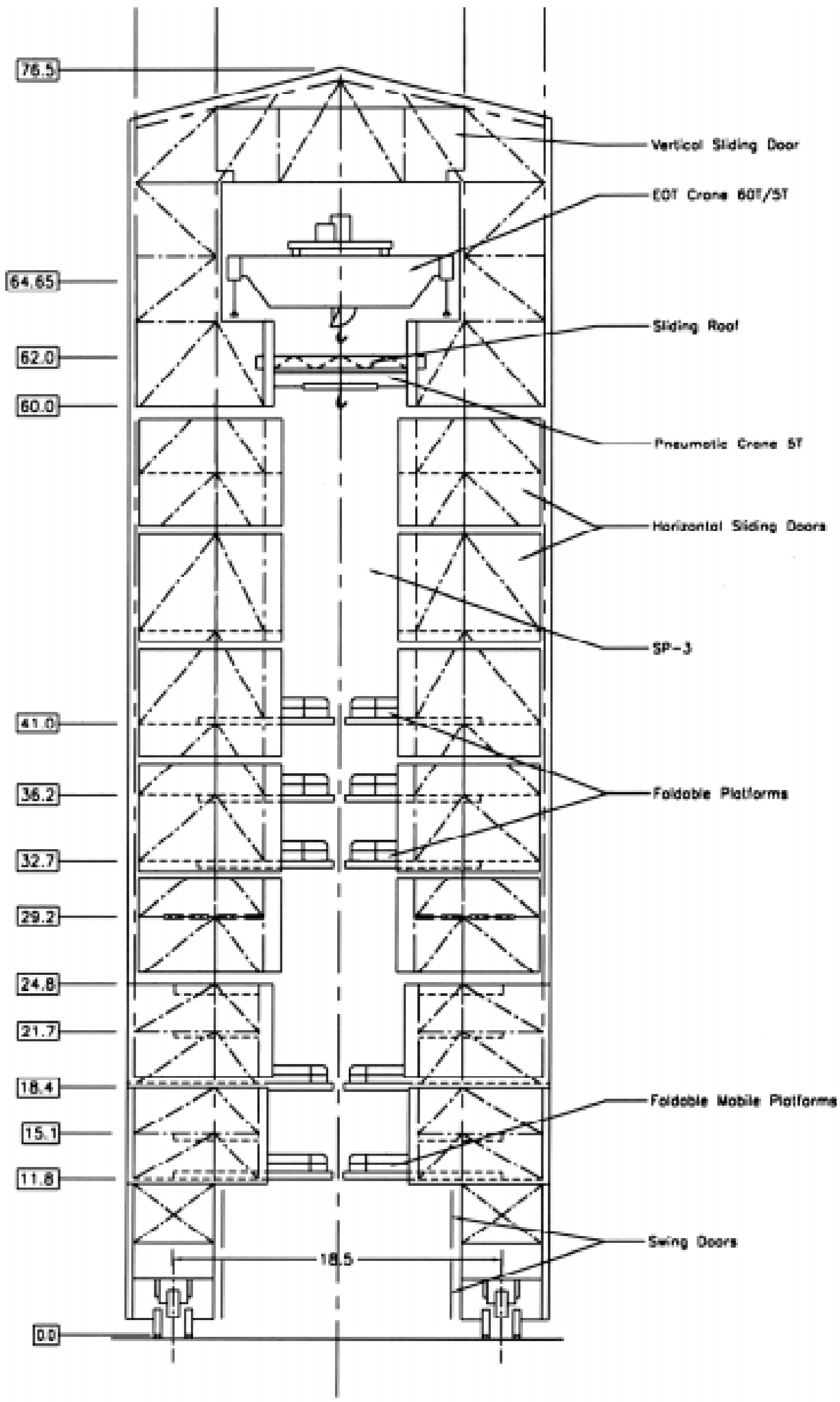


Fig. 9.4 Location of SP-3 Facility in MST

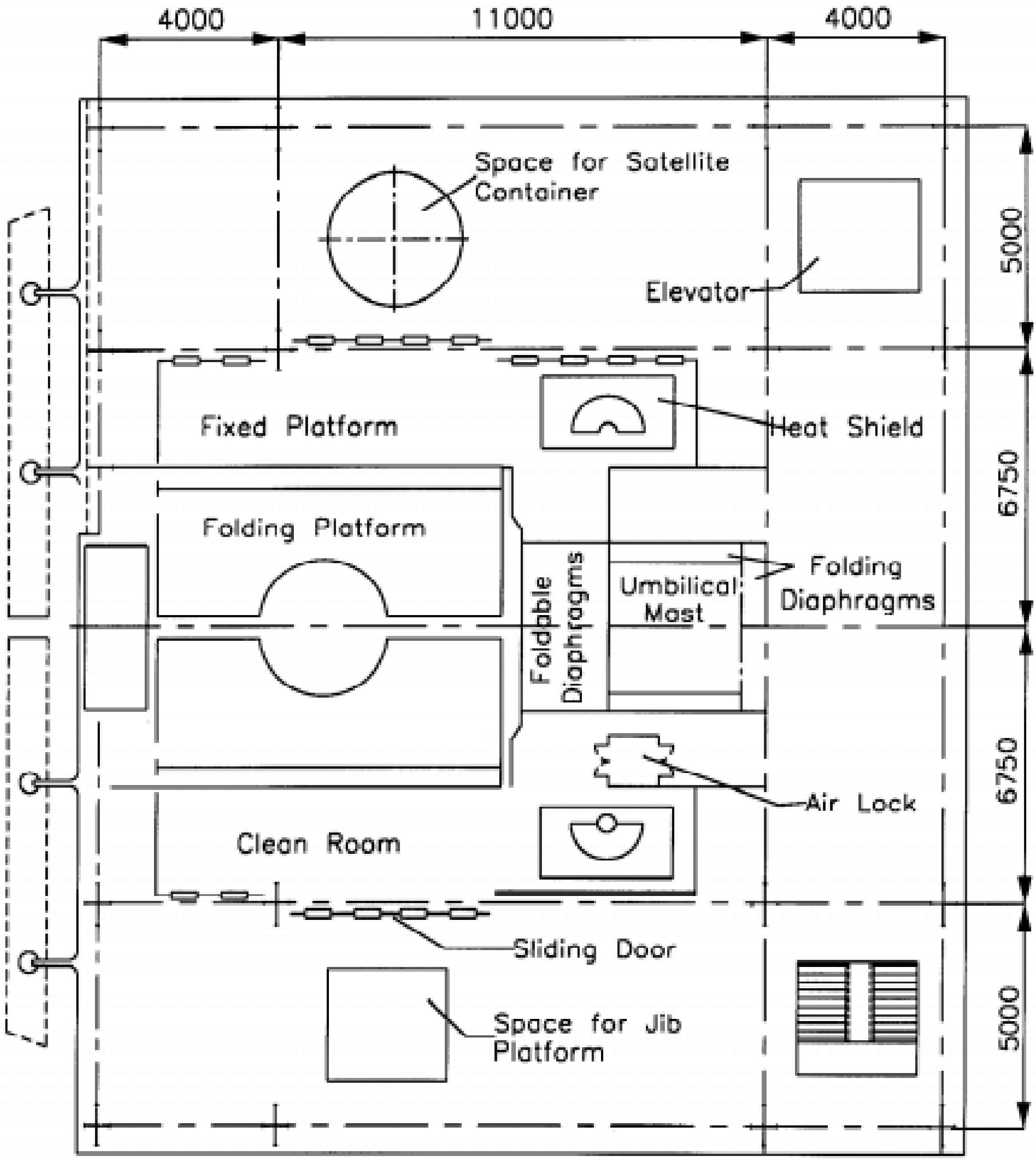


Fig. 9.5 SP-3 Facility (Layout)

Section 10

Telemetry, Tracking and Telecommand Support

Section - 10

TELEMETRY, TRACKING AND TELECOMMAND SUPPORT

Telemetry, Tracking and Telecommand (TTC) support for launch vehicles and spacecrafts are provided by ISRO Telemetry & Tracking Network (ISTRAC). TTC support for PSLV launch is derived from three Stations viz. Sriharikota (SHAR), Thiruvananthapuram (TVPM) and the Down Range Station (DRSN) at Mauritius (MAU) in the Indian Ocean.

During launch phase, SHAR, TVPM and MAU Ground Stations are configured to receive vehicle telemetry carriers. SHAR has Telemetry Tracking and Telecommand facilities and can cater to the launch phase upto approximately 680 s. The SHAR-I and SHAR-2 Stations work in redundant mode and provide the following functions for the launch.

- Acquisition of the two Telemetry carriers of the vehicle, data recording, pre-processing and transmission to mission computer for Range Safety and specialists' display consoles in real time.
- Ranging with Satellite transponder
- Acquisition of Satellite data, pre-processing and transmission to Satellite Control Centre, Bangalore.

MAU Station can track the vehicle from 656 s till 200 seconds after burnout of the fourth stage. TVPM Station has only telemetry facilities and provide the required space diversity for ensuring uninterrupted telemetry data. TTC support for spacecraft is provided by Bangalore, Lucknow and Mauritius telemetry stations. TTC support requirements for specific spacecraft mission have to be coordinated with ISRO by the Spacecraft agency.

Tracking is provided by two Precision Monopulse C-Band Radar (PCMC-1 & PCMC-2) located at SHAR in conjunction with two onboard C-Band Transponder and one PCMC-3 radar located at MAU. Real time data processing support is provided by ground stations for range safety & data display at the Mission Control Centre and for preliminary orbit determination. The data transmission between ground stations is totally carried out through dedicated satellite links.

The configuration of the three ground stations are given in Table 10.1 and the S-Band Telemetry facilities in Table 10.2.

Table 10.1
Ground Station Configuration

Station	Telemetry	Tracking Radar	Range & Range Rate System	Telecommand
SHAR-1	S-Band	C-Band	S-Band	UHF
SHAR-2	S-Band	C-Band	-	-
TVPM	S-Band	-	S-Band	-
MAU	S-Band	C-Band	S-Band	-

Table 10.2
S-Band Telemetry Facilities

Station	No. of carriers	Antenna dish Diameter (m)	G/T dB/C
SHAR-1	4	10	19.5
SHAR-2	3	10	19.5
TVPM	3	8	18
MAU	3	10	19.5

In addition to the above capabilities, all stations have the following features.

- A station processor, equipment for conducting simulation to check performance recording equipment and acquisition antenna.
- All frequencies are generated from station reference.
- Generator power supply in addition to commercial supply.

The stations are equipped with processors to perform the following functions.

- Control and monitoring of station equipment
- Telemetry data functioning
- Tracking data processing
- Command excitation
- Data communication.

The stations have one chain of extra equipment for receiving systems. The transmit chain has full hot standby. Frequency standard has hot redundancy and selection logic. Timing systems have redundancy and selection logic.

Annexure 2

Launch Request Format

Annexure - 2

LAUNCH REQUEST FORMAT

1 Introduction

S/C description and mission summary:

Shall include a 3 D view drawing of spacecraft in orbit, an exploded view and the coverage zones (if applicable)

1.1 Manufactured by :
Model :

1.2 Mass
- ☞ Total mass at launch :
- ☞ Mass of satellite in final orbit :

1.3 Dimensions
- ☞ Dimensions :
- ☞ Dimensions stowed for launch :
- ☞ Dimensions deployed on orbit :

1.4 Life Time :

1.5 Mission Summary :

	Telecommunications*
Purpose of the spacecraft and	Scientific*
description of the Payload	Radiolocalisation*
	Others
	(*) to be selected

1.6 Antennae : Omni antenna direction and location

1.7 Propulsion Sub-System(if applicable) :
Brief description : TBD
(liquid/solid, number of thrusters...)

1.8 Attitude Control : TypeTBD

1.9 Coverage Zones of the Satellite : TBD(with figure if applicable)

1.10 Electrical Power : Solar arrays description TBD
Beginning of life TBD W
End of life TBD W
Batteries description TBD

2 Mission Characteristics

Describe the sequence of events after separation until final orbit (main manoeuvres from separation until final orbit).

3 Spacecraft Description

3.1 Spacecraft Systems of Axes:

Shall include a sketch showing the Spacecraft system of axes, the axes are noted Xs, Ys, Zs and form a right handed set (s for spacecraft).

3.2 Spacecraft geometry in the flight configuration:

A drawing and a reproduceable copy of the overall spacecraft geometry in flight configuration is required. Detailed dimensional data will be provided for the parts of the S/C closest to the “static envelope” (antenna reflectors, deployment mechanisms, solar array panels, thermal protections...)

3.3 Mass, alignment, inertia (Nominal values & tolerances).

3.3.1 Mass, alignment, inertia details:

Element	Mass	CG Coordinates (mm)			Moment of Inertia Matrix (kg m^2)					
		X_G	Y_G	Z_G	I_{XX}	I_{YY}	I_{ZZ}	P_{XY}	P_{YZ}	P_{ZX}
Spacecraft										
Tolerances	(kg)	(mm)	(mm)	(mm)	%	%	%	Min. Max.	Min Max	Min Max

With Pxy = + xydm

Note: * CG coordinates are given in spacecraft axes with origin of the axes at the EB mounting plane.

* Inertia matrix is calculated in spacecraft axes with origin of the axes at the Center of Gravity.

3.3.2 Range of major/minor inertia axis ratio

3.3.3 Dynamic out of balance (if applicable).

Indicate the maximum dynamic out of balance in degrees.

3.4 Mechanical interfaces

Define adapter and its interface with the spacecraft

3.5 Electrical interfaces

3.5.1 Umbilical link

Define the characteristics for:

- ➢ battery trickle charging
- ➢ separation microswitch
- ➢ others

Indicate voltage and current during launch preparation.

LAUNCH PREPARATION

S/C Connector Pin allocation Number	Function	Max Voltage (V)	Max Current(mA)	Max Voltage OR Drop (ΔV)	Expected One way Resistance (Ω)

POE EXTRACTION (lift off)

Function	Max Current (mA)	Max. Voltage (V)

3.5.2 Umbilical link shielding definition.

3.5.3 Spacecraft earth potential reference point location.

3.6 Radio-electrical interfaces

3.6.1 Functional check-out prior and after integration with EB.

3.6.2 Antenna(e) diagrams and characteristics.

3.6.3 Satellite transmit and receive systems:

- description of spacecraft Telemetry and Telecommand systems
- description of payload telecommunications system.

3.6.4 System characteristics

- on board system.

For each TM & TC sources and systems used on the ground and during Launch, give the following:

UNIT DESIGNATION SOURCE		S1	S2	S3
Function				
Band				
Carrier Frequency, Fo (Mhz)				
Bandwidth centered around Fo :	-3 db			
60 db	-			
Carrier Modulation				
Type Index				
Carrier polarisation				
Local Oscillator Frequencies				
1^{st} intermediate frequency				
2^{nd} intermediate frequency				
EIRP, transmit (dbm)	MAX			
	NOM			
	MIN			
Field strength at	MAX			
	NOM			
	MIN			
Antenna:	Designation Location Gain Pattern			

3.6.5 Spacecraft transmission plan:

Source	Function	During Preparation On Launch Pad	After Ho-1h30 Until (TBDs) After Separation	In Orbit
S1			OFF*	
S2			OFF*	

*ON and standby mode for mini Auxiliary Payload

3.6 Environmental data

3.6.1 Fundamental modes (lateral, longitudinal) of spacecraft hardmounted at interface with PLA.

3.6.2 Thermal characteristics during launch preparation and boost phase including constraints.

3.6.3 Dissipated power during count down and boost phase.

3.6.4 Contamination characteristics and constraints:

Define the material selected and the out-gassing material

4 Operational Requirements

4.1 Provisional time schedule for operations at the range

4.2 Spacecraft preparation building

4.2.1 Main operations list and description

4.2.2 Power requirements

Indicate voltage, Amps, Nb phases, frequency, category (standard or no break).

4.2.3 Facility equipment requirements

4.2.4 RF and hardline links requirements.

4.2.5 Telecommunications requirements (Telephone,Facsimile, ...)

4.2.6 Miscellaneous

4.3 Transportation requirements

Give also dimension and weight of containers

4.4 Hazardous items storage requirements

Pyrotechnic devices description.

5 General

5.1 Estimate packing list (including heavier and larger container characteristics)

Indicate designation, number, size(LxWxH in mm) and mass(kg)

5.2 Technical support requirements

Workshop, instrument calibration, ...

5.3 Hotel and cars reservations

Give guidelines and policy for hotel and cars reservations during the campaign.

5.4 Miscellaneous services

6 Spacecraft Development Plan

7 Tests

7.1 Spacecraft test plan (vibration, acoustic, shocks, ...) and acceptance levels

Define the qualification policy, qualification (protoflight or qualification model).

7.2 Environmental test plan

Describe qualification and acceptance testing including levels and frequency ranges.

8 Definitions, Acronyms, Symbols, ...