

PSLV

Polar Satellite Launch Vehicle

User's Manual

ISSUE - 5

(Draft Version)

Indian Space Research Organisation



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PUBLICATION NOTICE

This publication is an update (Issue 5) of the document No:PSLV-VSSC-PM-65-87/3 (Issue 4) which was released in December, 1999 and reflects changes and additions to the information on vehicle performance, launch environment, payload interface requirements, launch facilities and services.

The current issue also includes details on PSLV Variants and provisions for Auxiliary payloads, mini and multiple-constellation type spacecrafts and their respective 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

Antrix ISRO Organisation & ISRO Organisation relevant to Launch Services

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ABBREVIATIONS

ACS	-	Auxiliary Control System
ASME	-	American Society of Mechanical Engineers
CDR	-	Critical Design Review
CG	-	Center of Gravity
CTR	-	Checkout Terminal Room
DA	-	Double Amplitude
DC	-	Direct Current
DLA	-	Dual Launch Adapter
DRSN	-	Down Range Station
DTO	-	Detailed Test Objectives
EB	-	Equipment Bay
EED	-	Electro Explosive Device
EGC	-	Engine Gimbals Control
EMC	-	Electro Magnetic Compatibility
FLSC	-	Flexible Linear Shaped Cord
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
INS	-	Inertial Navigation System
IRS	-	Indian Remote Sensing Spacecraft
ISRO	-	Indian Space research Organization
LEO	-	Low Earth Orbit
LO	-	Local Oscillator
LPSC	-	Liquid Propulsion Systems Center
MCC	-	Mission Control Centre
MIL-STD	-	Military Standard
MMH	-	Mono Methyl Hydrazine
MON	-	Mixed Oxides of Nitrogen
MST	-	Mobile Service tower
N	-	Newton
Nm	-	Newton Meter
N ₂ O 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	-	Polar Satellite Launch Vehicle with six S-9 strap on motors
PSLV-XL	-	Polar Satellite Launch Vehicle with six XL strap on motors
PSLV-CA	-	Polar Satellite Launch Vehicle core alone configuration
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
rad	-	Radian
RCS	-	Reaction Control system
RF	-	Radio Frequency
SDSC	-	Satish Dhawan Space Centre
SHAR	-	Sriharikota Launch Complex
SITVC	-	Secondary Injection Thrust Vector Control
SLP	-	Second Launch Pad
SP-1	-	Spacecraft Facility – 1
SP-2	-	Spacecraft Facility – 2
SP-3	-	Spacecraft 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
TTC	-	Telemetry, Tracking and Tele-command
UH-25	-	Unsymmetrical Di-methyl Hydrazine + 25% Hydrazine Hydride
VAB	-	vehicle assembly Bay
VMC	-	Valiamala Complex
VSSC	-	Vikram Sarabhai Space Centre

Section-1

INTRODUCTION

The Polar Satellite Launch Vehicle (PSLV) is a four-stage launch vehicle primarily designed and developed to orbit spacecrafts into Sun Synchronous Polar Orbit (SSPO). Subsequent to the three development flights PSLV was made operational in September 1997 by the launch of 1200 kg Indian Remote Sensing satellite, IRS-1D. During the development phase and in operational phase the payload capability has been improved and it stands at 1.6t for 630km SSPO and 1t for GTO launch. Today PSLV is also capable of launching multiple S/Cs either in dual launch mode using DLA or as auxiliary payloads in the EB. With the induction of higher variants in the strap on stages the pay load capability is being further increased.

PSLV can also perform launches to low inclinations from 20deg to near zero and 45deg to SSPO. ISRO is responsible for the overall mission consisting of mission analysis, vehicle integration & checkout, launch campaign & launch, ground support which includes telemetry & tracking support, preliminary orbit determination and post flight analysis.

The reliability of the vehicle has been proved by the successful launch of eight flights.

This manual brings out the details on vehicle interfaces and the requirements to be met by the spacecraft 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 & 1.2). The first and third stages use solid propellant while the second and fourth stages use liquid propellant.

PS1 stage

The first stage consists of a 2.8m diameter core motor (PS1) and six 1.0 m diameter motors (PSOMs) strapped onto the core. PS1 has a nominal propellant loading of 138 t while each PSOM contains 9t solid propellant. This stage is provided with Secondary Injection Thrust Vector Control (SITVC) for pitch and yaw control. Two swiveling 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 is used for pitch control during coast phase. One of the strap-on motors is provided with SITVC control to augment roll control of first stage.

Flexible Linear Shaped Cord (FLSC) system along with retro rockets is used for PS1 separation while ball and socket joint with spring thruster is employed to separate Strap on motors.

PS2 stage

The second stage (PS2) carries 40t of propellants (UH25 and N₂O₄) and has a pump fed high pressure engine with 800kN thrust. This stage has Engine Gimbal Control (EGC) for pitch and yaw and hot gas Reaction Control System for the roll. A clamp band separation system is used along with retro rockets for separation of PS2. Ullage rockets are used during the initial phase of PS2 thrusting to ensure positive acceleration.

PS3 stage

The third stage (PS3) has a twaron epoxy motor case with a propellant loading of 7t and a contoured and submerged nozzle. This stage has Flex Nozzle Control (FNC) for the pitch and yaw control during the thrust phase.. A Reaction Control System (RCS) with six thrusters is provided on PS4 for coast phase control of both PS3 and roll control during PS3 burn phase. A 'ball lock' mechanism separates the third stage from the ongoing vehicle.

PS4 stage

The fourth stage (PS4) with 2.5t propellant loading (MON and MMH) and is a twin engine configuration with each producing 7kN thrust. This stage (PS4) is controlled during thrust phase by gimbaling its two engines for pitch, yaw and roll. The Reaction Control System (RCS) used for PS3 coast phase control is used for coast phase control of PS4.

PS4 variant with 1.6t propellant loading (L1.6) is also being used for PSLV missions.

S/c separation is based on clamp band system.

PSLV EB

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 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 aluminum alloy Heat shield of 3.2 m diameter protects the spacecraft from hostile flight environment during the ascent phase. Description of the Heat shield and its separation system is given in Chapter-4.

Flight termination systems are provided onboard for strap on motors and for the first three stages which can be remotely activated through tele-command in case of any vehicle malfunction violating range safety constraints. Flight termination is provided in the fourth stage in case of a GTO or low inclination launch alone.

To enable launch of two medium class S/Cs with mass of around 1000kg each, a dual launch adapter is used.

Developments

Variant of PSOM with 12.5t propellant loading – PSOM XL is being ground tested for induction into flight. The induction of this motor would improve the payload capability by 150kg for a SSPO mission and 100kg for a GTO mission. This motor is expected to be ready by the middle of 2007. The vehicle configuration with PSOM XL is shown in Fig.

PSLV Telemetry

PSLV is equipped with the following transmission and reception system:

- Two telemetry systems in the EB in the frequency band 2237.5 & 2211 MHz. respectively with 5W transmitters. (Spurious output - 67dB with respect to carrier in 2200-2300 MHz band).
- A tele-command reception system with the antenna mounted in IS3/4 operation at 434 MHz.

Two radar transponder systems with receiver frequency 5.66 GHz, and transmitter frequencies in 5.5 -5.9 GHz. Band are available.

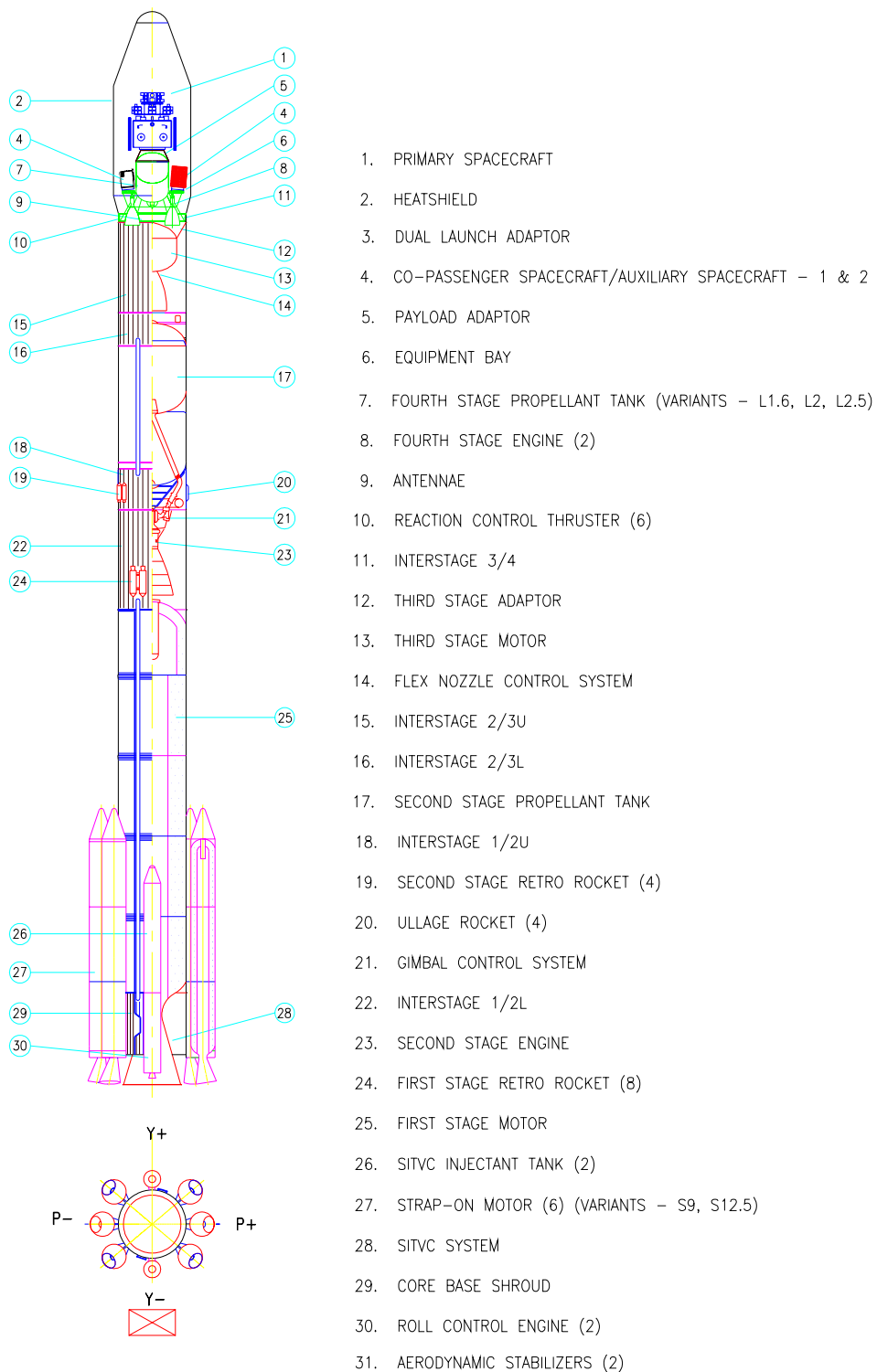


FIG 1.1 PSLV Configuration

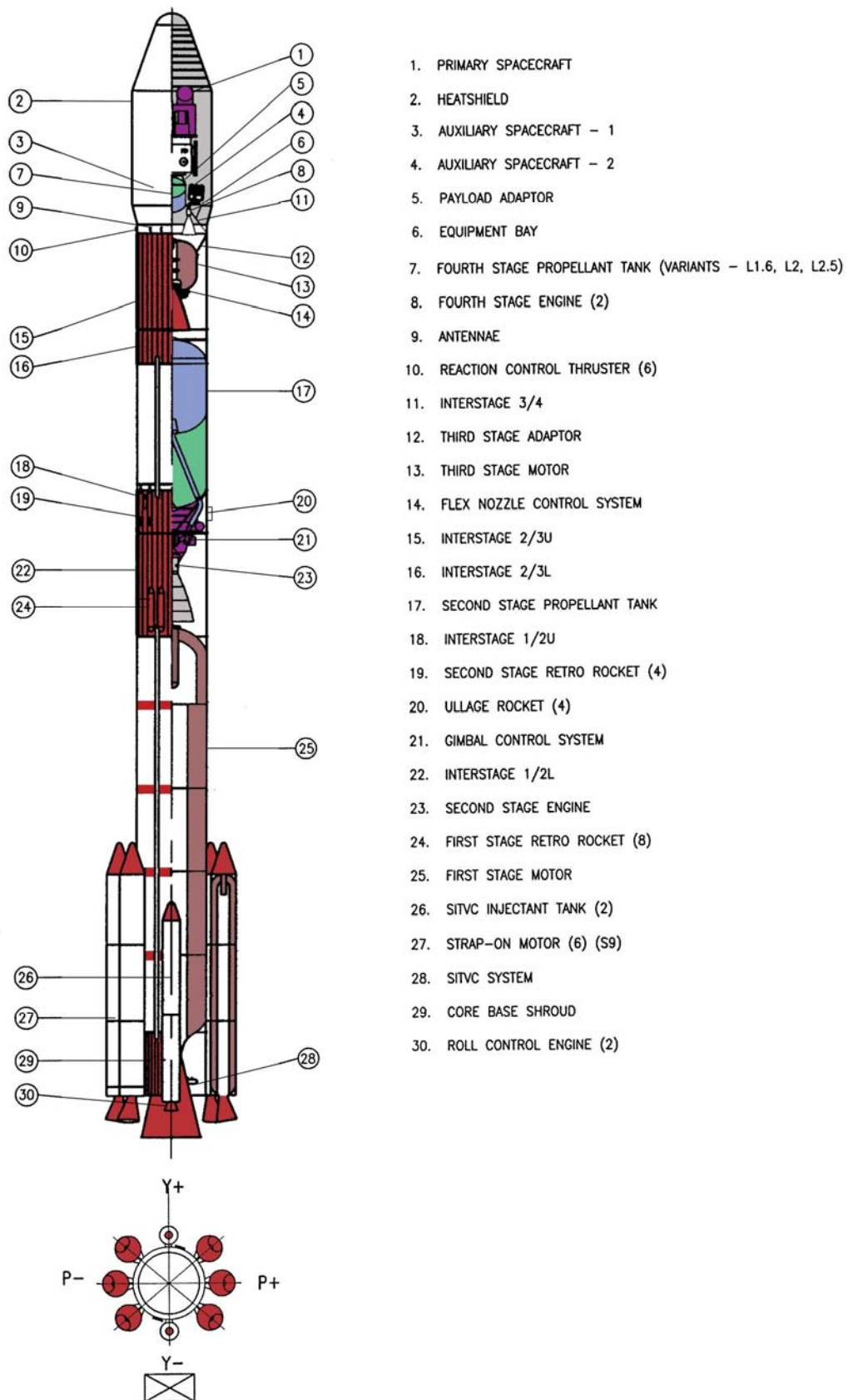


FIG 1.1 PSLV Configuration

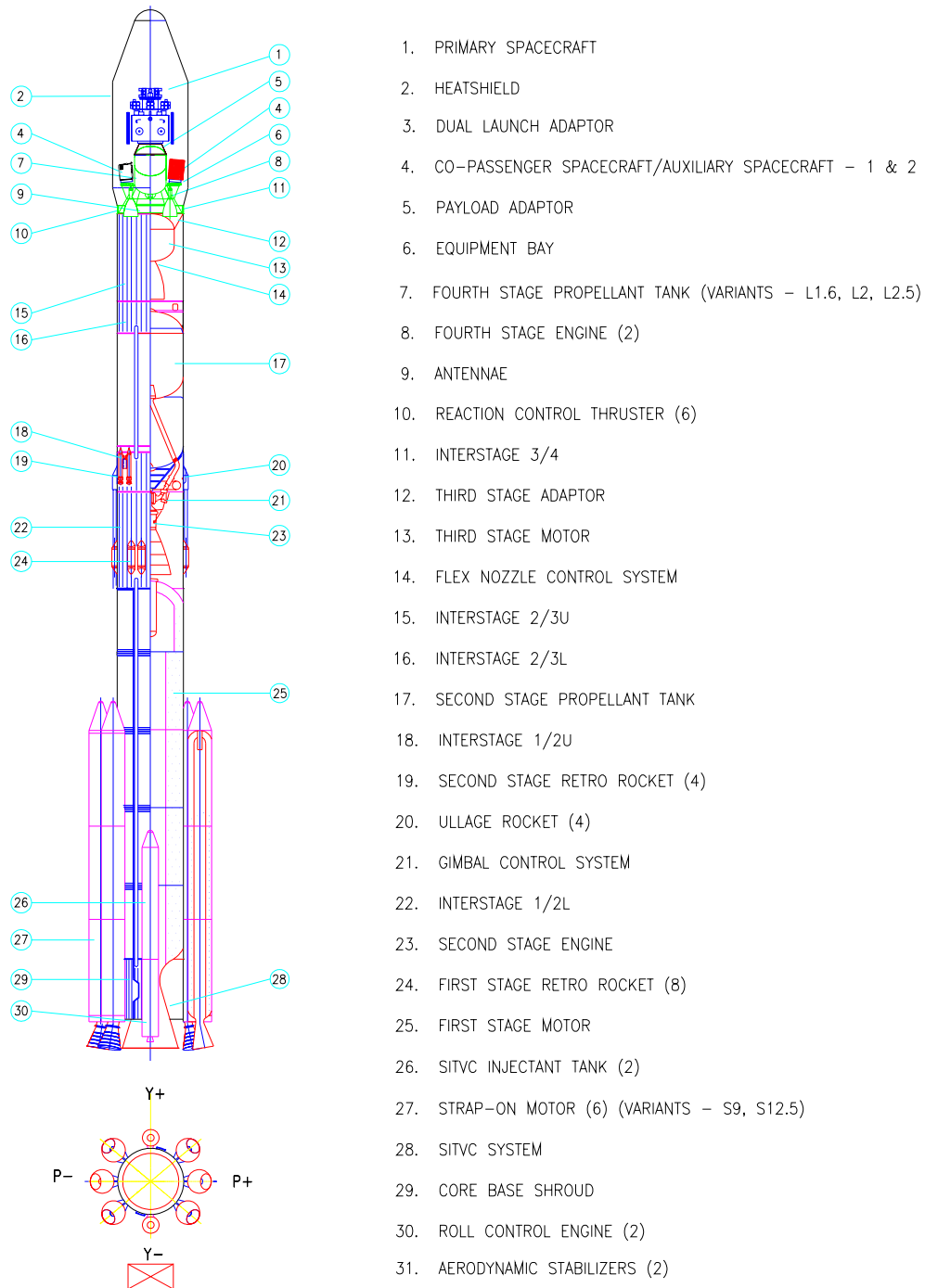


Fig 1.2 PSLV-XL Configuration

1. PRIMARY SPACECRAFT
2. HEATSHIELD
3. AUXILIARY SPACECRAFT - 1
4. AUXILIARY SPACECRAFT - 2
5. PAYLOAD ADAPTOR
6. EQUIPMENT BAY
7. FOURTH STAGE PROPELLANT TANK (VARIANTS - L1.6, L2, L2.5)
8. FOURTH STAGE ENGINE (2)
9. ANTENNAE
10. REACTION CONTROL THRUSTER (6)
11. INTERSTAGE 3/4
12. THIRD STAGE ADAPTOR
13. THIRD STAGE MOTOR
14. FLEX NOZZLE CONTROL SYSTEM
15. INTERSTAGE 2/3U
16. INTERSTAGE 2/3L
17. SECOND STAGE PROPELLANT TANK
18. INTERSTAGE 1/2U
19. SECOND STAGE RETRO ROCKET (4)
20. ULLAGE ROCKET (4)
21. GIMBAL CONTROL SYSTEM
22. INTERSTAGE 1/2L
23. SECOND STAGE ENGINE
24. FIRST STAGE RETRO ROCKET (8)
25. FIRST STAGE MOTOR
26. SITVC INJECTANT TANK (2)
27. STRAP-ON MOTOR (6) (S12.5)
28. SITVC SYSTEM
29. CORE BASE SHROUD
30. ROLL CONTROL ENGINE (2)

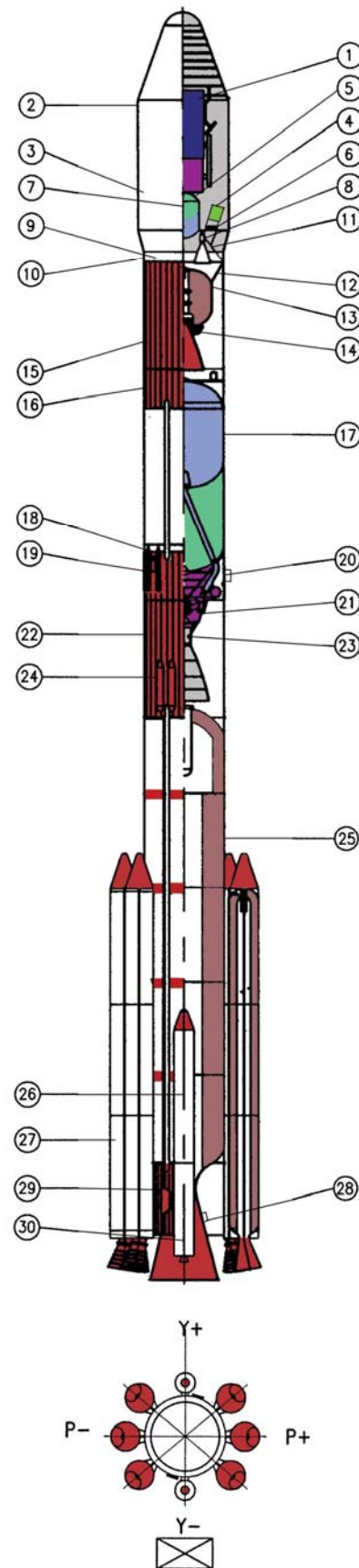


Fig 1.2 PSLV-XL Configuration

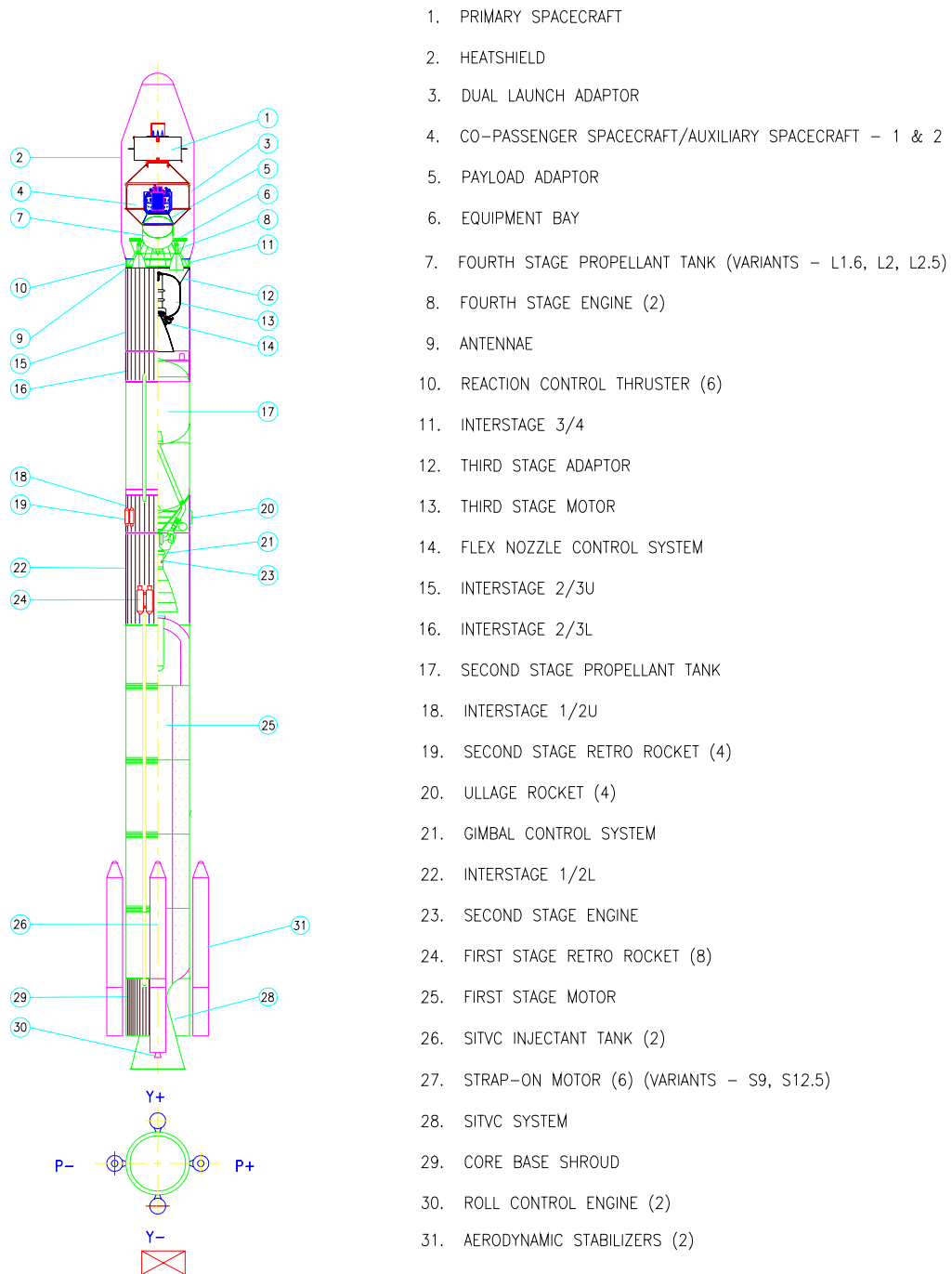


Fig 1.3 PSLV-CA Configuration

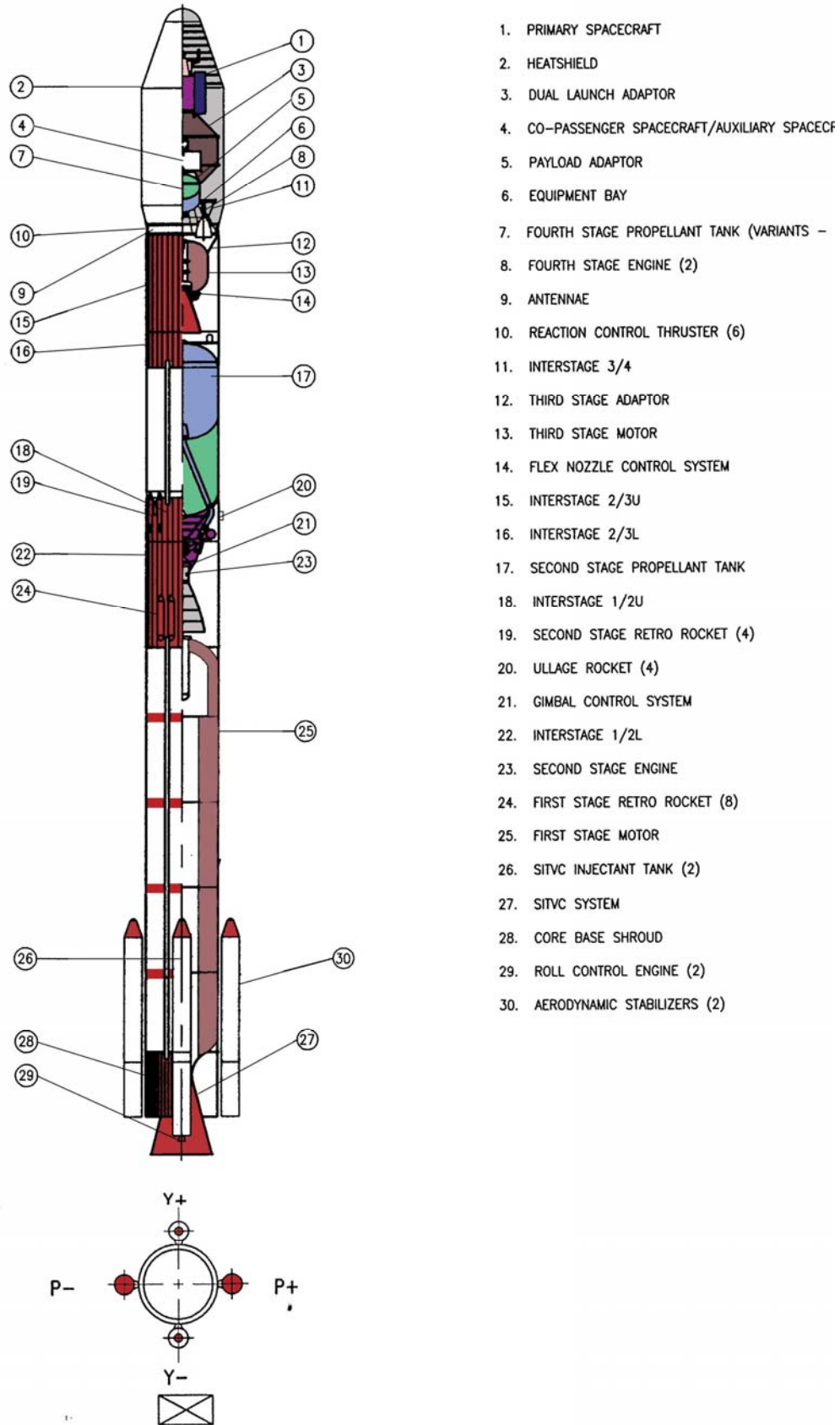


Fig 1.3 PSLV-CA Configuration

1.3 Vehicle Axes and Attitude Definitions

Vehicle sign convention for hardware assembly and mission analysis is shown in Fig.1.4 and the location of the PSOMs and the SITVC tanks with respect to the vehicle is shown in Fig. 1.5.

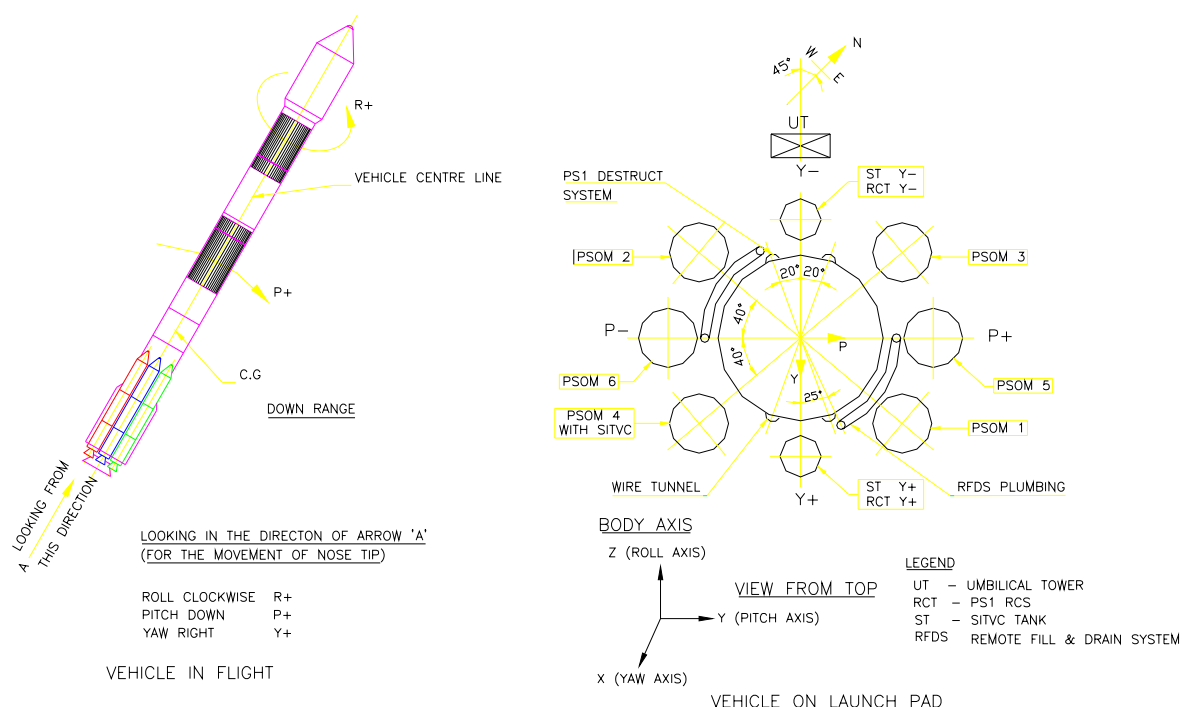
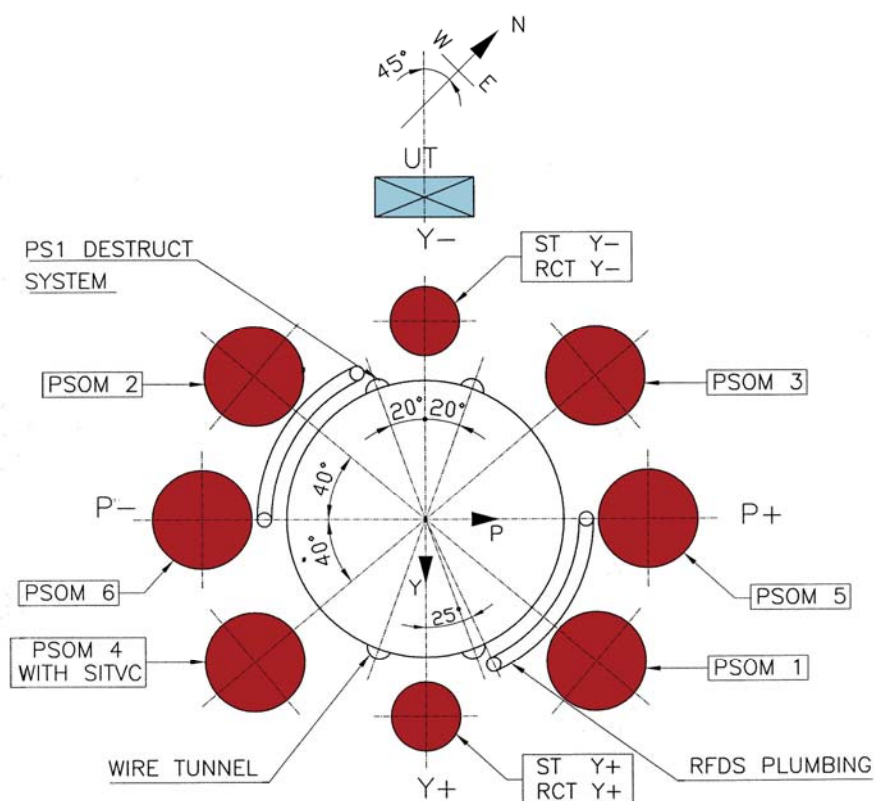


Fig. 1.4 PSLV sign convention



LEGEND

UT - UMBILICAL TOWER
 RCT - PS1 RCS
 ST - SITVC TANK
 RFDS - REMOTE FILL & DRAIN SYSTEM

VEHICLE ON LAUNCH PAD

1.4 Flight Sequence

A typical flight sequence for the sun-synchronous polar launch and geo-transfer launch of PSLV is pictorially shown in Fig. 1.6 and 1.7 res. and is enumerated in Table 1.1.

At the launch pad, the vehicle axes are aligned to 135 degree (SSPO) and 98 (GTO) azimuth. The first stage (PS1) along with four strap on motors is ignited on the launch pad and the vehicle lifts off vertically. After 5 seconds, the vehicle is rolled to achieve the required launch azimuth. The vehicle starts pitching in the azimuth plane at T+7 seconds. 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+90 secs.

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 after which the separation of third stage takes place. The fourth stage ignition time is decided on-board by CLG. When the required velocity and altitude are attained, the fourth stage is shut off by onboard guidance command and the spacecraft is injected into the required orbit with a separation velocity of 0.8m/s to 1.2m/s.

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

Table 1.1
Flight Sequence for a typical SSO & GTO PSLV Mission

Event	SSPO (630km)		GTO (>180x36000km)	
	Time from PS1 ignition(s)*	Altitude (km)*	Time from PS1 ignition(s)*	Altitude (km)*
Ignition of first stage	T+0.0	0.0	T+0.0	0.02
Ignition of four PSOMs (1,2,3,& 4)	1.2	0.0	1.2	0.02
Ignition of two PSOMs (5 & 6)	25	2.5	25	2.5
Separation of four ground lit PSOMs	68	23.8	68	20.9
Separation of the two air lit PSOMs	90	42.9	90	35.2
Separation of first stage	112.9	69.1	111.5	60.7
Ignition of second stage	113.1	69.3	111.7	60.9
Separation of Heat shield	158.9	120.9	185	116.4
Separation of second stage	264.2	238.7	276.8	144.6
Ignition of third stage	265.4	240.1	278	144.7
Separation of third stage	521.5	508.8	530.7	158.1
Ignition of fourth stage	532	517.4	662.5	166.1
Fourth stage cut-off	944.3	633.5	1179.5	408.6
Separation of S/C / outer S/c	981.3	634.2	1251.5	524.9
Separation of DLA-U	1026.3	635	--	--
Separation of Inner S/C	1071.3	635.8	--	--

- * For nominal vehicle performance

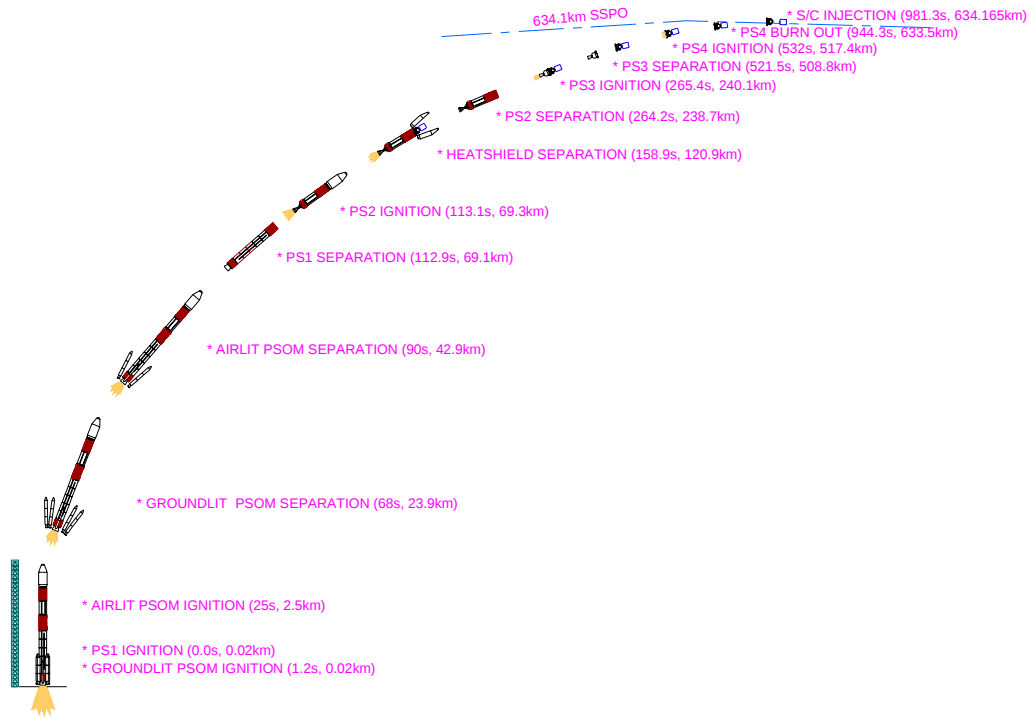


Fig. 1.6 Typical flight sequence for SSO launch

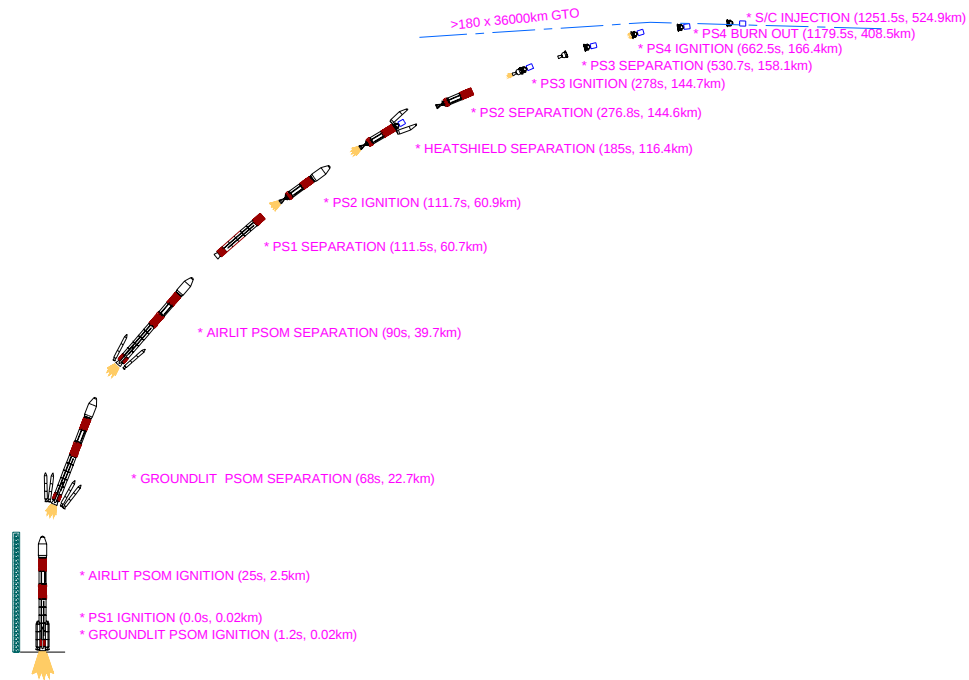
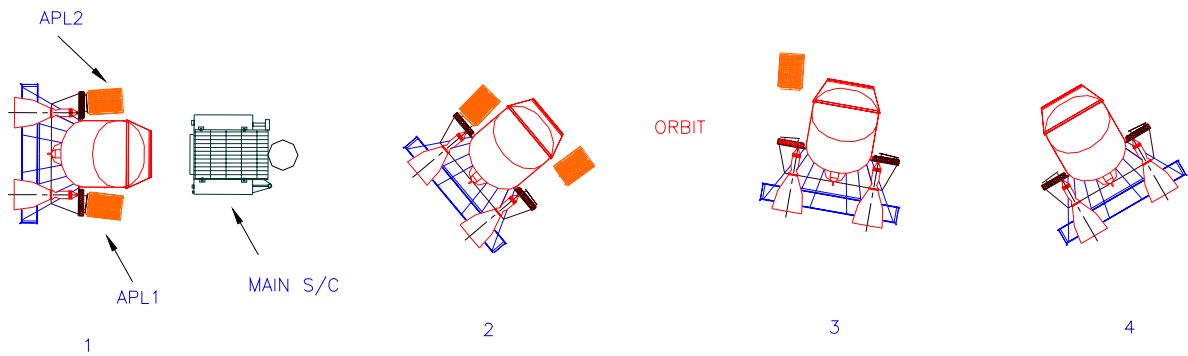


Fig. 1.7 Typical flight sequence for GTO launch



1. SEPARATION OF MAIN S/C
2. SEPARATION OF APL1 AFTER YAW TURN OF 40 deg BACKWARDS
3. SEPARATION OF APL2 AFTER YAW TURN OF 80 DEG BACKWARDS
4. YAW TURN OF 120 DEG BACKWARDS

Section:2**PERFORMANCE CAPABILITY**

PSLV can perform three classes of missions:

Sun-Synchronous Polar Circular Orbits

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 maximize payload performance while complying with the Spacecraft 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 at such an altitude as to ensure that the free molecular heat flux on the Spacecraft is less than 1135 W/m².

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 of full vehicle (PSLV-S9-L2.5) for different classes of missions with typical orbits is given in Table 2.1 and for core alone vehicle is given in Table 2.2

Table – 2.1 Payload capability of PSLV

Mission	Orbit(km)	Inclination (deg)	Payload (kg)
SSO	326	96.673	1734
	1006	99.48	1229
Planar (circular)	400	20 to 90	3760 to 2102
	2000	20 to 90	1567 to 789
elliptical	300 x 1000	17.8	3497
	300 x 10000	52	1622
GTO	300 x 36000	17.8	974

For capability of PSLV-XL add 150kg for SSPO and 100kg for GTO to Table 2.1

Table –2.2

Mission	Orbit (km)	Inclination (deg)	Payload (kg)	
			L2.5	L1.6
SSPO	526	97.5	1186	1104
	826	98.7	979	865
Planar (circular)	600	50,63	2106,2003	
	1000	50,63	1640,1515	
	400	50,63		2276,2114
	2000	50,63		871,780
Elliptical	300 x 1500	17.8		2212
	300 x 10000	17.8	1199	1031
GTO	300 x 36000	17.8	589	

Proper launch azimuth is selected to meet range safety constraints and mission requirement of inclination. Vehicle visibility exists for almost all the missions above an altitude of 500km

The performance capability charts of PSLV for different missions like Sun-Synchronous polar orbits, planar circular orbits, circular orbits with different inclinations and Elliptical orbits (Planar trajectories for 140 degree launch azimuth from SHAR is given in Fig.2.1 to 2.3.)

2.3 Mission Accuracy

PSLV is capable of injecting three-axis stabilized and guided spacecraft into the specified orbits. During the first stage thrust phase, the vehicle has 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 SSPO and GTO within the dispersion given in Table 2.3. The achieved injection accuracies for the previous launches are given in Annexure-

Table 2.3 Orbit dispersions

	Error in the orbital parameters (3 Sigma)	
	SSPO	GTO
Apogee	±15 km	±675km

Perigee		
Arg of Perigee	-	± 0.2 deg
Inclination	± 0.1 deg	± 0.2 deg

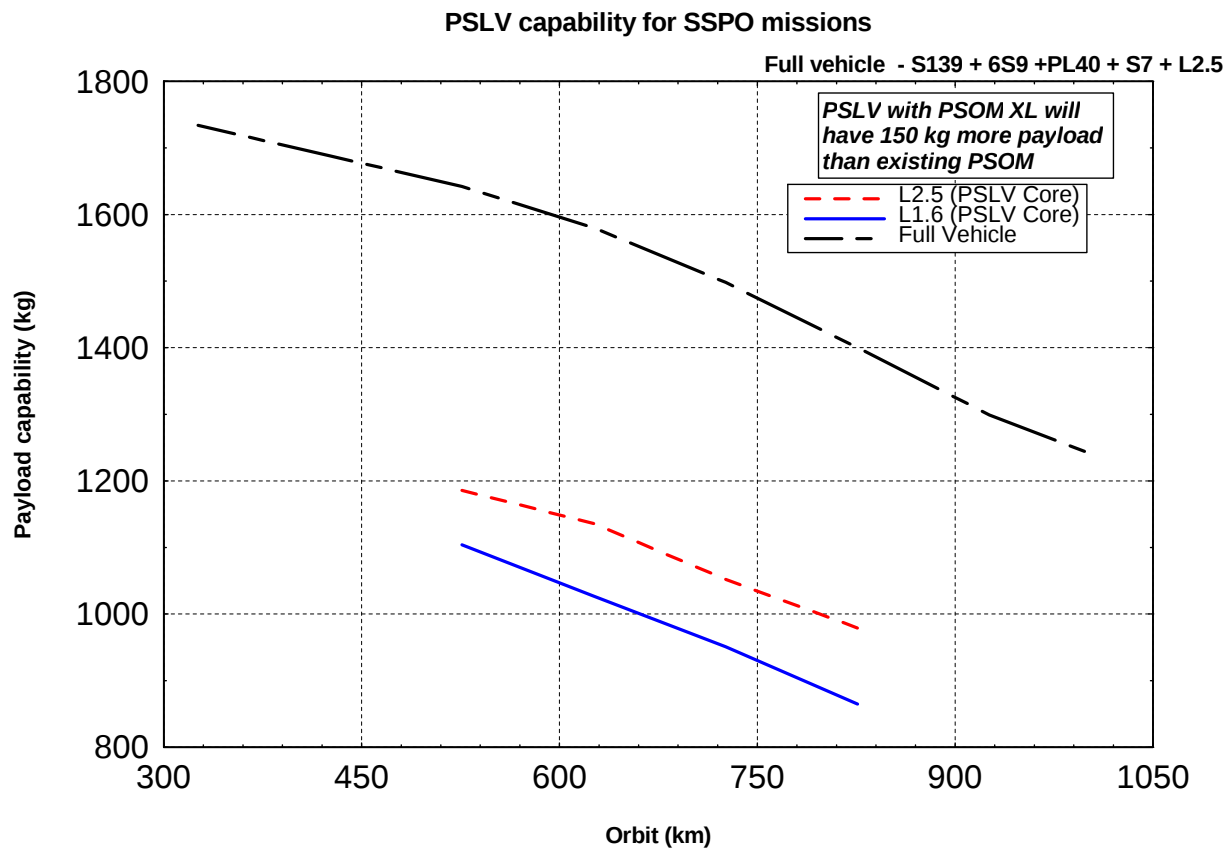


Fig. 2.1 PSLV payload capability for SSO launches

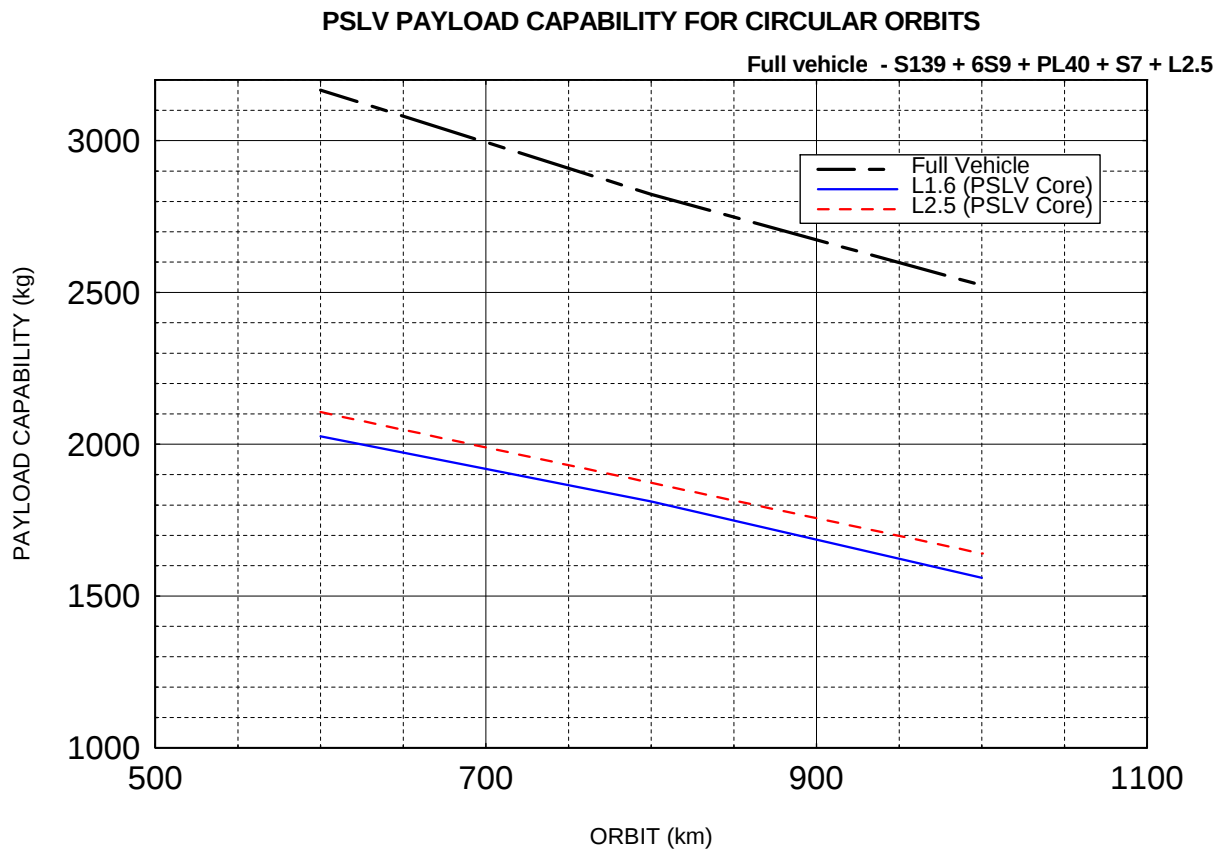


Fig. 2.2 PSLV payload capability for Circular orbit launches

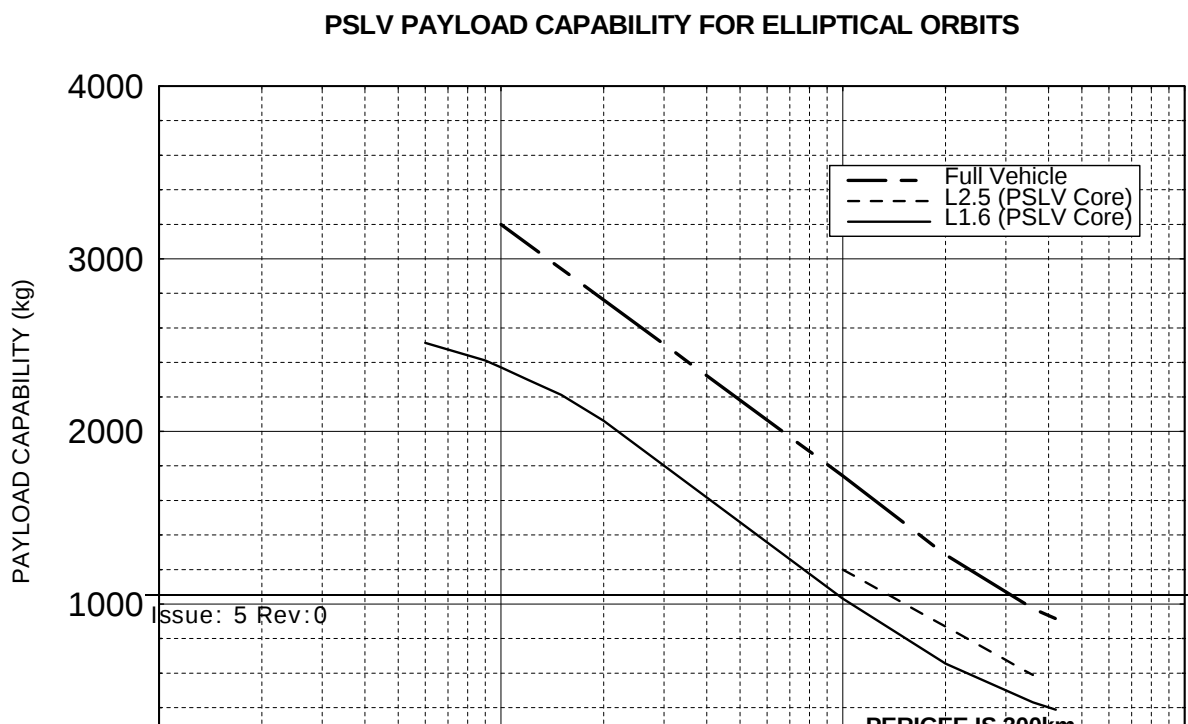


Fig. 2.3 PSLV payload capability for Elliptical orbit launches

2.4 Visibility

Visibility of PSLV for a nominal SSPO and GTO launch from tracking stations are given in Fig.2.4 and 2.5 res. In the event of any non-visible period a Data Storage Unit (DSU) will be used to acquire the data.

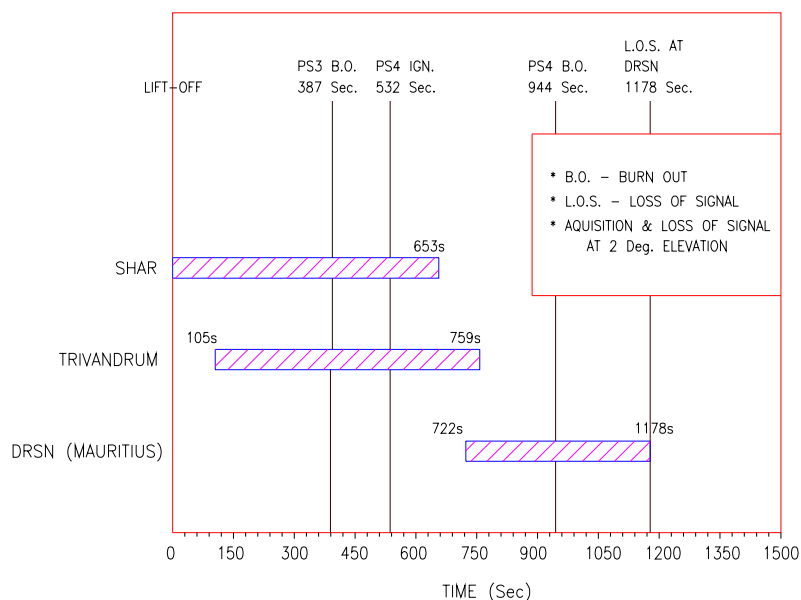


Fig. 2.4 Visibility chart for a typical PSLV SSO mission

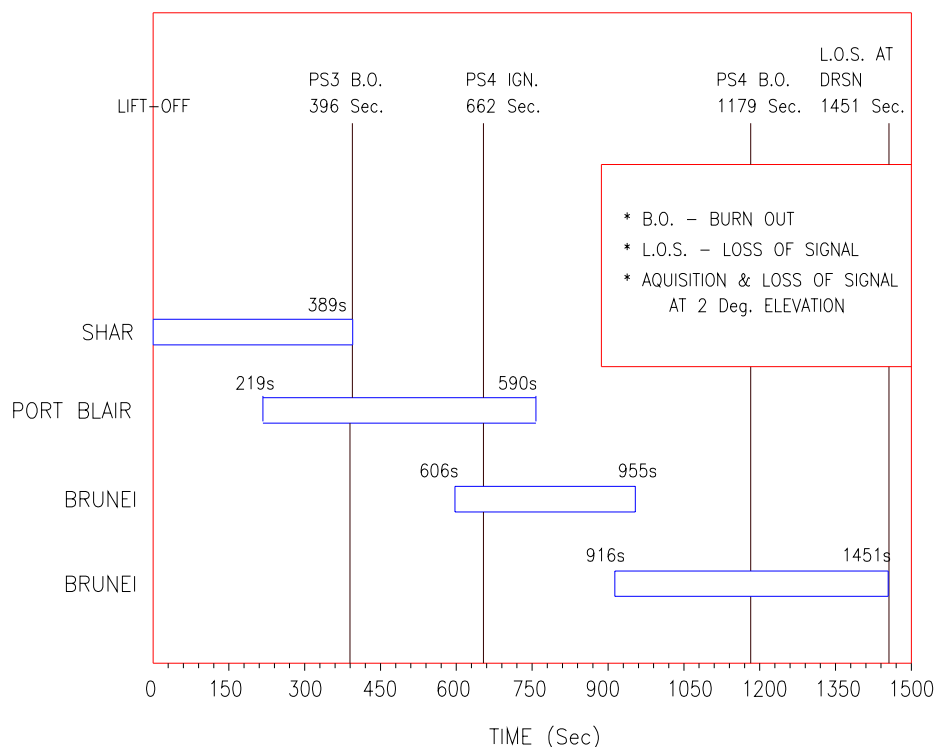


Fig. 2.4 Visibility chart for a typical PSLV GTO mission

2.5 Preliminary Orbit Determination (POD)

POD provides for the assessment of the Vehicle performance achieved in the flight and ascertains the fulfillment 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 is provided within one hour after the injection of the spacecraft.

POD is carried out using the following data.

Telemetry data of INS

Range, range rate, azimuth and elevation angles from Range and Range-Rate System.

Range, azimuth and elevation angles from radar.

Accuracy of orbit prediction is given in table 2.3. This POD phase also ensures that the look angles are predicted within the required half beam width angle of the S-band telemetry antennae.

Section 3

FLIGHT ENVIRONMENT LEVELS

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 and also POGO excitations during second stage thrusting phase. 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 preferably designed to produce fundamental frequencies as below

- > 40Hz along the longitudinal (thrust axis) direction
- > 18 Hz along the lateral axis

These frequencies are with base fixed at the S/C separation plane. For pay loads 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 spacecrafts.

3.2 Vehicle Acceleration

The maximum static and dynamic accelerations occurring at spacecraft interface during flight are as given below.

Longitudinal (axial)	:	7g compression and 2.5g tension
Lateral	:	$\pm 1.5g$
Proof Load Factor	:	1.1
Ultimate Load Factor	:	1.25

The design check shall be carried out for ultimate load. Lateral loads are to be considered acting simultaneously with the longitudinal loads and all loads act at CG of the Spacecraft.

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 would subject a spacecraft to an environment that will not be exceeded in flight 95% (2σ) of the time. The qualification test levels established are intended to assure, through reasonable over test, that the spacecraft, even with minor mass and design variations, can withstand the most severe dynamic environmental loads.

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.

The test levels specified below are applicable for all configurations of PSLV.

3.3.1 Sinusoidal vibration levels

The sine vibration levels are determined by PS2 shutdown transients, which cause the maximum responses at LV/SC interface. Propellant loading in second stage of PSLV is varied to achieve the desired transients viz. N depletion or command cut off.

N depletion causes higher response. However lower response levels on S/C can be achieved by adopting command cut off in PS2 with a marginal payload penalty.

The generalized test levels corresponding to N depletion and command cut off are in table 3.2 and table 3.3 respectively.

Coupled load analysis (CLA) will be carried out with vehicle FE model and the customer supplied S/C FE model. Sine test levels specific to the satellite will be finalized from the responses estimated from the CLA.

Table 3.1			
Qualification and Acceptance Test Levels of Sinusoidal Vibration for PS2 command cutoff condition for s/c below 600kg (generalized)			
	Frequency Range-Hz	Qualification Test level (Zero to peak)	Acceptance Test level (Zero to peak)
Longitudinal Axis	5.0-11.5	6.75mm(DA)	4.5 mm (DA)
	12.0-38.0	1.5 g	1 g
	38.0-60	1.5-1.05 g	1.0 g -0.7g
	60.0-100.0	1.05 g	0.7 g
Lateral	5.0-7.0	6.75 mm (DA)	4.5 mm (DA)

Axis	7.0-30.0 30.0-100.0	0.67 g 0.45 g	0.45 g 0.30 g
Sweep rate		2 oct/min	4 oct/min

Table 3.2			
Qualification and Acceptance Test Levels of Sinusoidal Vibration for PS2 command cut off condition for s/c above 600 (generalized)			
	Frequency Range-Hz	Qualification Test level (Zero to peak)	Acceptance Test level (Zero to peak)
Longitudinal Axis	5.0-11.5	6.75 mm (DA)	4.5 mm (DA)
	12.0-38.0	1.8 g	1.2 g
	38.0-50.0	1.8-0.75 g	1.2-0.5g
	50.0-100.0	0.75 g	0.5 g
Lateral Axis	5.0-7.0	6.75 mm (DA)	4.5 mm (DA)
	7.0-30.0	0.67 g	0.45 g
	30.0-100.0	0.45 g	0.30 g
Sweep rate		2 oct/min	4 oct/min

The flight levels of vibration are the anticipated levels and are multiplied by a factor 1.5 for spacecraft qualification. This factor is intended to ensure a reasonable degree of margin and to provide for contingencies such as:

- Test tolerances (equipment and operation), and
- Difference between qualification and flight units (tolerance as well as minor development changes during development).

The sinusoidal qualification and acceptance test levels given in Table 3.1 to 3.2 are applicable at the base of the spacecraft.

3.3.2 Acoustic environment

The critical acoustic environments occur at lift-off due to jet noise and in transonic flight due to unsteady shocks 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 for PSLV core alone and full vehicle configuration are given in Table 3.3.

Table-3.3				
Qualification and Acceptance Test Levels of Sound Pressure for				
Octave band centre Frequency (Hz)	Sound pressure level in dB			
	PSLV -CA		PSLV	
	Acceptance	Qualification	Acceptance	Qualification
31.5	124	128	124	128
63.0	125	129	126.5	130.5
125.0	130	134	130	134
250.0	136	140	136	140
500.0	137	141	140	144
1000.0	132	136	135	139
2000.0	128	132	128	132
4000.0	125	129	125	129
8000.0	122	126	122	126
Overall level in dB	141	145	143	147
Duration	1 minute	2 minutes	1 minute	2 minutes

In a proto-flight concept, the flight model will be subject to testing at qualification levels and acceptance duration.

It may be noted that s/c external surfaces of large area and low density such as solar panel and large antenna configurations need to be evaluated for acoustic susceptibility.

3.3.3 Random vibration

The random vibration test levels at payload interface are given in Table 3.6

Table 3.6		
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	6.7g(RMS)	4.47 g(RMS)
Duration	2 minutes/axis	1 minute

Generally random test levels are specified in lieu of acoustic testing. However to determine the responses on the package mounting decks due to acoustics and launch vehicle borne vibration, the following testing scheme is recommended in addition to acoustic testing.

FE model will be generated corresponding to the levels as in table 3.6 and low level random of 0.5 gRMS input at s/c base. Satellite proto/mass model will be subject to 0.5 gRMS random vibration testing to validate the transfer function of FE model. Responses shall be estimated for full level random using the updated transfer function.

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 spacecraft to qualify the spacecraft.

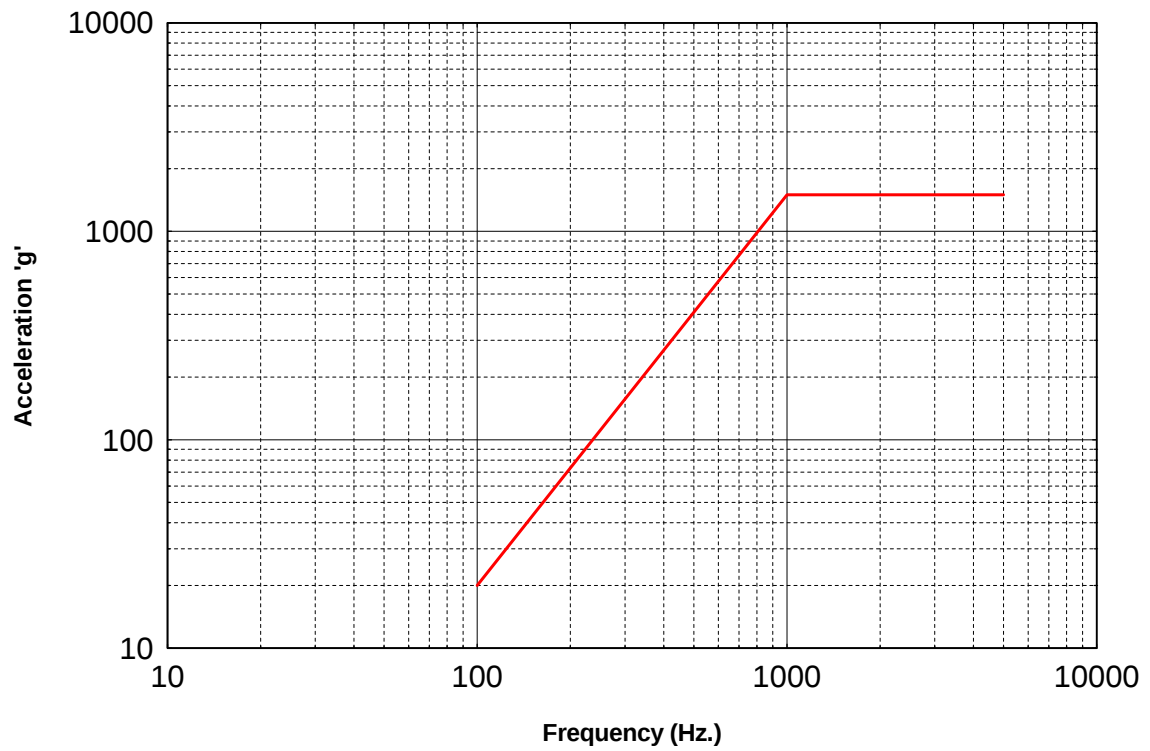


Fig. 3.1 Typical separation shock spectrum at S/C interface

Section : 4**PAYLOAD INTERFACES**

This section gives detailed description of the heat shield, 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 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 mono-coque structures and the cylindrical section is an integrally stiffened iso-grid 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. This gives an attenuation of acoustic levels.

The heat shield halves 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 is 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 integration. Fig. 4.1 gives the layout and details of access doors on HS. Any additional requirement needs to be informed in advance to study the feasibility of providing the cutouts.

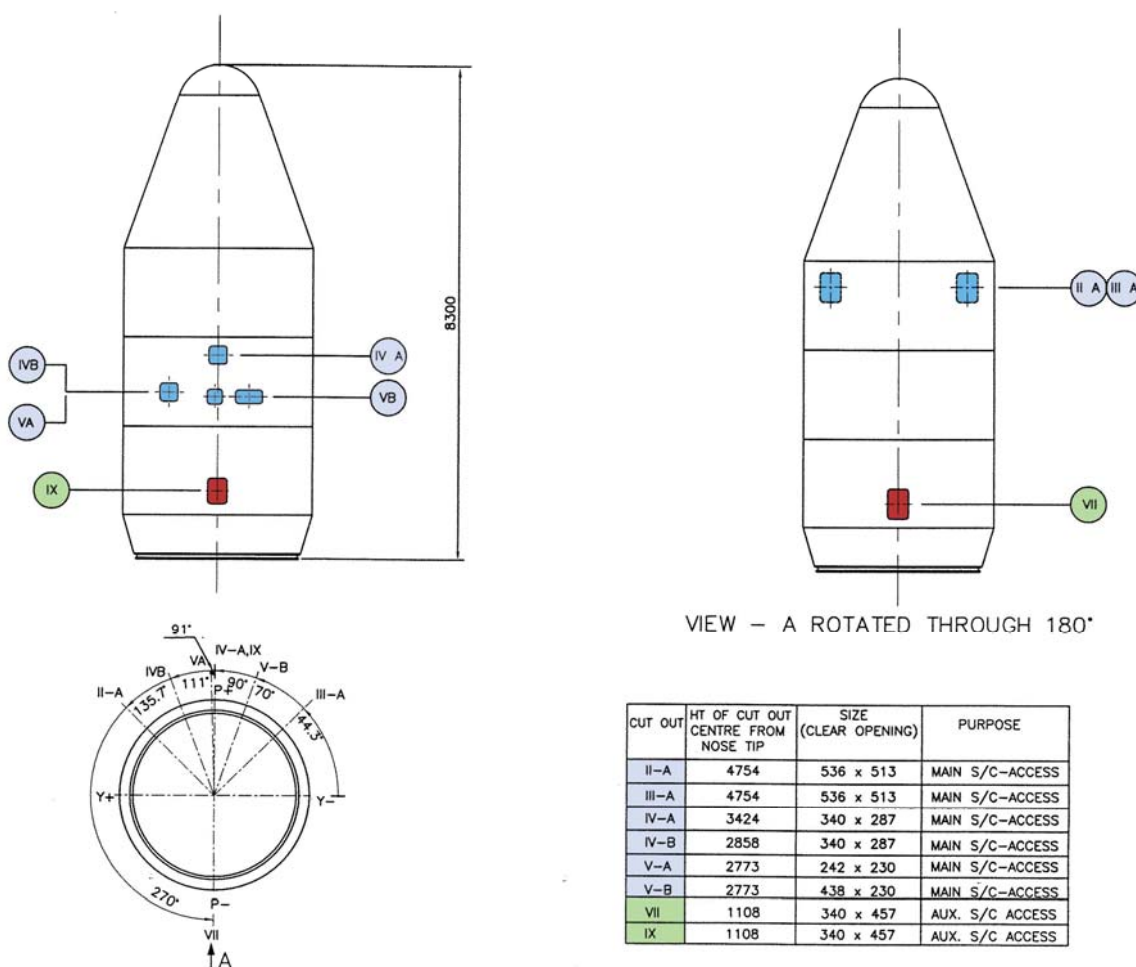


Fig. 4.1 Cutouts on HS for S/C access

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, coordination 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. It is also possible to use the space surrounding the payload adapter below the separation plane in consultation with ISRO.

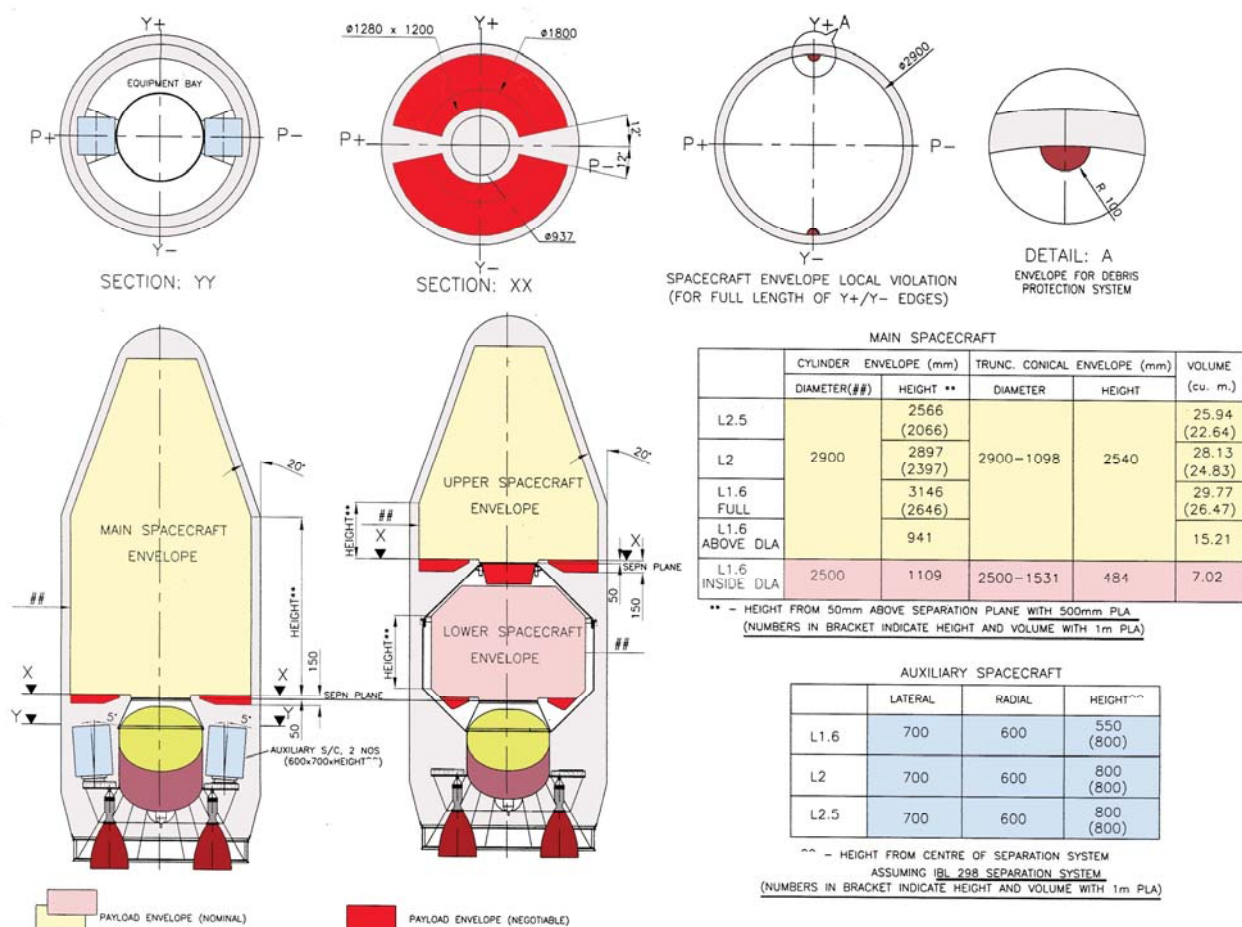


Fig. 4.2 PSLV payload envelope

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. The aft end interfaces with the vehicle at 1370PCD. It has a truncated conical structure with aluminium end rings and carbon Fibre Reinforced Plastic (CFRP) panels.

Two versions of Payload adapter assembly are available viz. (Figs 4.3) heights of 500/1000mm.

The forward end of the payload adapter has 937B interface (15 deg semi wedge angle) and incorporates standard interface similar to payload adapter of ARIANE-4/DELTA-2. The structure is capable of supporting up to 1800kg spacecraft with CG at about 1.5m above the Spacecraft separation plane. Maximum tolerable transverse offset of CG is ± 5 mm.

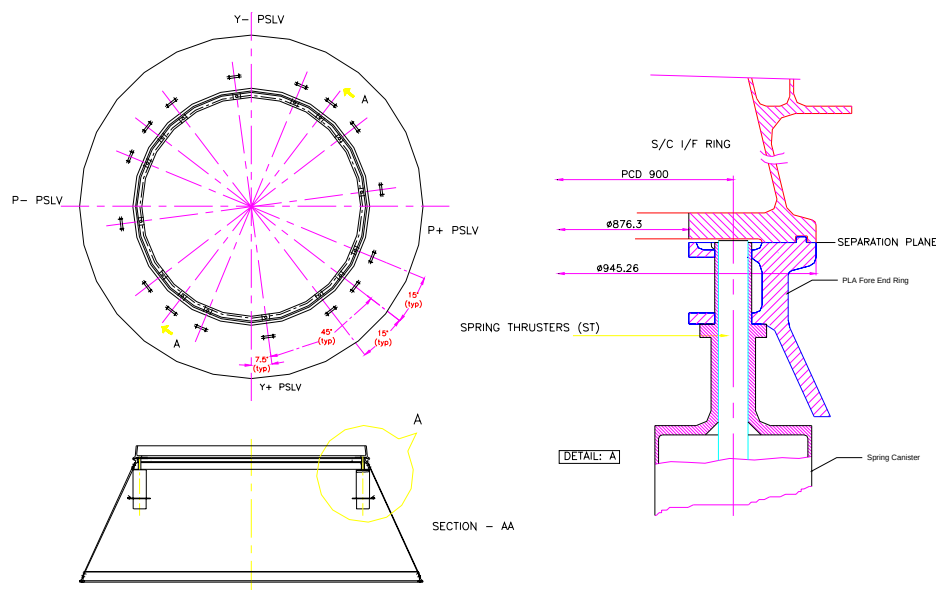


Fig. 4.3 PLA configuration and interface

4.4 Dual launch Adapter

The dual launch adapter is a CFRP structure that enables mounting of two S/Cs of relatively lower mass and smaller envelope. DLA is made of three parts DLA-L, DLA-M and DLA-U. DLA-L interfaces with the PS4 tank while DLA-U interfaces with the upper S/C. The DLA-U has 937B interface similar to PLA and is shown in Fig. 4.4. DLA-M / DLA-U has a clamp band interface which enables separation of DLA-U after upper S/C separation. Inside the DLA the inner S/C will be mounted on the PLA that is assembled to the PS4 tank and DLA-L. Cut outs are available on the DLA for RF checks, accessibility and cooling purpose.

The DLA is capable of supporting 1000kg S/C with a CG of 750mm above separation plane. The maximum tolerable lateral CG offset shall be $\pm 5\text{mm}$.

DLA-UPPER



DLA-MIDDLE



DLA-LOWER



Fig. 4.4 DLA configuration and interface

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 945.26mm. The clamp band is tightened to ensure proper joint characteristics. A pair of bolt cutters is used in redundant mode as release devices. Four to sixteen numbers of helical compression springs provide a separation velocity of 0.8 m/s to 12 m/s. The maximum interface load due to each spring is limited to 1000 N. The separation disturbance is not more than ± 2 degree/s for the stipulated conditions of CG offset limits. The separation disturbance also depends on the S/C MI properties. The separation system is same for both the PLA and DLA and is shown in Fig. 4.5.

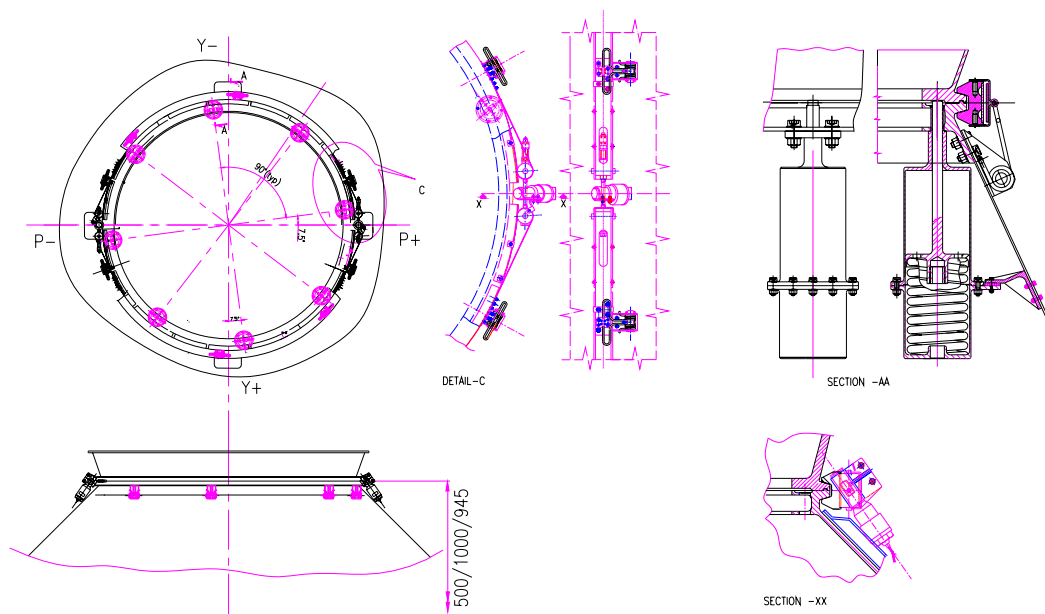


Fig. 4.5 S/C separation system assembly (Typical)

4.6 Electrical and RF Interfaces

A snap-off connector 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 in 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 shall be outside the PLA/DLA and shall comply with the stay out zone (Fig. 4.6) requirement of separation system. The 27 pin connector will be used in case of a dual launch while 61 pin connector will be used in case of single launch. The details about the connector are in Table 4.1. Additional lines in case of a dual launch can be provided after studying requirement of both S/Cs.

Table 4.1 Details of umbilical connector

Type of launch		Umbilical connector - UC01	Umbilical Connector - UC02
Single S/C launch	Connector type	DEUTSCH type	DEUTSCH type
	Designation – Vehicle	DBAS-78-61-0PN	DBAS-78-61-0PN
	Designation – S/C	DBAS-70-61-0SN	DBAS-70-61-0SN
	No. of pins	61	61
	Lines available to S/C	61	
	Pins for separation monitoring on S/C side	1,2	60,61
Dual launch	Connector type	DEUTSCH type	DEUTSCH type
	Designation – Vehicle	DBAS-78-27-0PN	DBAS-78-27-0PN
	Designation – S/C	DBAS-70-27-0SN	DBAS-70-27-0SN
	No. of pins	27	27
	Lines available to S/C	50	
	Pins for separation monitoring on S/C side	1,2	26,27

The lines from the 61 pin umbilical connector will be terminated at the junction box in the Checkout Terminal Room (CTR) adjacent to launch Fig. 4.6 Stay out zones for PLA/DLA separation system

pad which also houses the spacecraft checkout equipments. The serial 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.. 4.7.

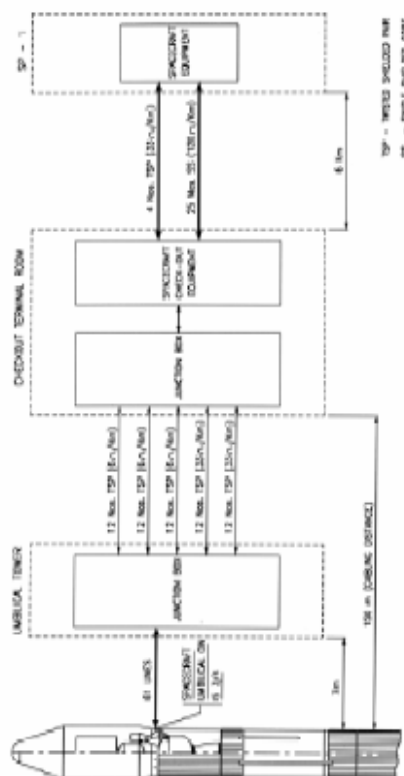


Fig. 4.7 S/C checkout cabling configuration at MST/FLP

4.7 Electro Magnetic Compatibility (EMC) Requirements

PSLV is equipped with transmission and reception systems to cater to data transmission flight termination and tracking as mentioned earlier.

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.

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 voltage. The contact resistance between spacecraft and vehicle shall be less than 10 milli Ohms.

Section-5

PROVISIONS & INTERFACES FOR LAUNCH OF AUXILIARY SPACECRAFTS

Micro Satellites in the weight range of 50 to 150kg require launch opportunities and in many cases a dedicated launch cannot be afforded. To cater to this type of requirement, provisions are made in PSLV to launch the micro satellites on piggy back mode. These will be designated as auxiliary S/Cs and will be offered 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 Spacecraft with PSLV.

5.1 Payload Capability

PSLV can carry a maximum of two auxiliary Spacecrafts with a nominal mass of 150kg (or less) each. The above mass does not include the mass of separation system.

Dimensions: 600 mm x 700*mm x 850mm (h) maximum. The requirement of appendages over and above the given dimensions need to be discussed on a case by case basis.

Center of Gravity: Longitudinal < 450 mm from mounting constraints plane of the Spacecraft.

Lateral ± 5 mm

* - Parallel to vehicle yaw axis

5.2 Orbit and Separation Sequence

PSLV can cater to launching the primary S/Cs into Sun Synchronous Orbits (SSO) or Low Earth Orbits (LEO).

The orbit of the auxiliary S/Cs will be subordinated to the orbit requirements of the primary S/Cs in each launch.

The nominal separation sequence is as below:

- (i) Shut-off of PSLV Fourth Stage after attaining Injection conditions.
- (ii) Separation of primary S/C
- (iii) Collision Avoidance Maneuver for the auxiliary S/C-1.
- (iv) Separation of auxiliary S/C-1.
- (v) Collision Avoidance Maneuver for auxiliary S/C-2.

(vi) Separation of auxiliary S/C-2.

5.3 Mounting Provision

Auxiliary Spacecrafts 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 Spacecrafts. Fig.5.1 gives the scheme of mounting auxiliary Spacecrafts.

To provide safe clearance from the Vehicle during separation, the auxiliary S/C is provided with 5 deg. tilt away from Vehicle longitudinal axis (Roll axis of vehicle). This tilt is achieved by providing an interface ring placed between the deck plate of VEB and separation system.

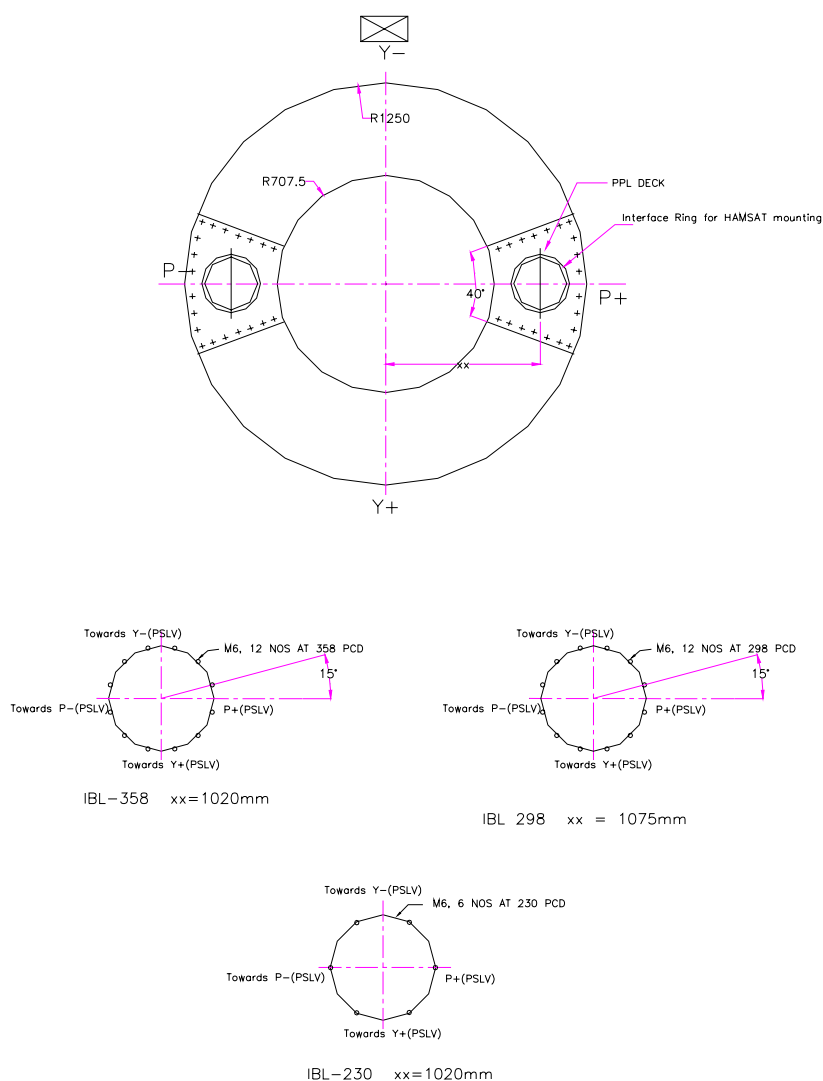


Fig. 5.1 Mounting provision for auxiliary S/C

5.4 Separation System

ISRO offers qualified ball lock separation system, which can cater to the auxiliary S/Cs of mass up to 150Kg.

The configuration of the separation system consist of an upper ring adapter (Spacecraft 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 the necessary 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 by a small angle using pyro thrusters, 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 (Fig. 5.2) based on the ball lock mechanism have been qualified for use with 50, 100 and 150kg Class auxiliary S/Cs. The specifications are as below.

Table 5.1						
Specifications for Separation System based on Ball Lock Mechanism						
Class of Spacecraft	Designation	Mounting PCD (mm)	No. & size of bolts	Nominal Separation velocity (m/s)	Mass (kg)	Mass retained on the Satellite (kg)
50 kg	1BL-230	230	M6 x 6	1	5	1
150kg	1BL-298	298	M6 x 12	1	15	1.2
150kg	1BL-358	358	M6 x 12	1	8	1.5

5.5 Electrical Interface

The electrical interface between the auxiliary S/C and PSLV consists of one umbilical link, which is available for each auxiliary S/C for battery trickle charging and minimal checkout.

The connector to be used is:

DBAS 74 12 05N 059 On Spacecraft side

DBAS 78 12 0PN 059 On the Vehicle side

The connector for mounting on the Spacecraft side is provided by ISRO.

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

Out of 12 pins of the connector, Pin Nos 1 and 2 are to be shorted on the S/C side with a shorting loop. This connection on the Vehicle side will be used to monitor S/C separation through Vehicle telemetry in flight.

The separation plane connector shall be located outside the separation system interface for easiness of assembly.

5.6 Flight Environment Levels:

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

5.6.1 Quasi static loads

Longitudinal acceleration (Static+Dynamic) :	7g compression and 2.5g tension
Lateral acceleration (Static+Dynamic)	: $\pm 2.5g$ at LV/SC interface (separation system Forward ring to SC interface)
Proof load factor	: 1.1
Ultimate load factor	: 1.25

The design check shall be carried out for ultimate load. Lateral loads are to be considered acting simultaneously with the longitudinal loads and all loads act at CG of the Spacecraft.

5.6.2 Qualification and acceptance tests

The customer will demonstrate that the auxiliary Spacecraft meets the flight environment by conducting the following qualification and acceptance tests on the Spacecraft.

Qualification tests: Sinusoidal vibration, random vibration and shock tests

Acceptance tests : Random vibration test

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

5.6.3 Sine vibration test levels

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

Table 5.2			
Sine Vibration test levels			
	Freq. Range (Hz)	Qualification level	Acceptance level
Longitudinal axis	5.0 – 10.0	12.5mm (0 to peak)	10mm (0 to peak)
	10-100	2.5g	2.0g
Lateral axis	5-8	12.5mm (0 to peak)	10mm (0 to peak)
	8 – 100	2.5g	2.0g
Sweep rate		2 Oct / min	4 Oct / min

5.6.3 Random vibration test levels

Table 5.3		
Random Vibration test levels		
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
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 S/C shall meet the following:

- The fundamental frequency in longitudinal axis > 90Hz
- The fundamental frequency in lateral axis > 45Hz

These figures include the influence of the S/C separation system.

5.6.5 Thermal environment

During pre launch phase, the maximum power dissipation for each auxiliary S/C is not to exceed (TBD) Watts.

5.6.6 Shock environment

Auxiliary S/Cs will be subjected to shock environment at various flight staging events like stage separation, Heat Shield separation and auxiliary S/C separation. A typical shock spectrum is given in fig. 5.3, to which the auxiliary S/C has to be tested during qualification phases.

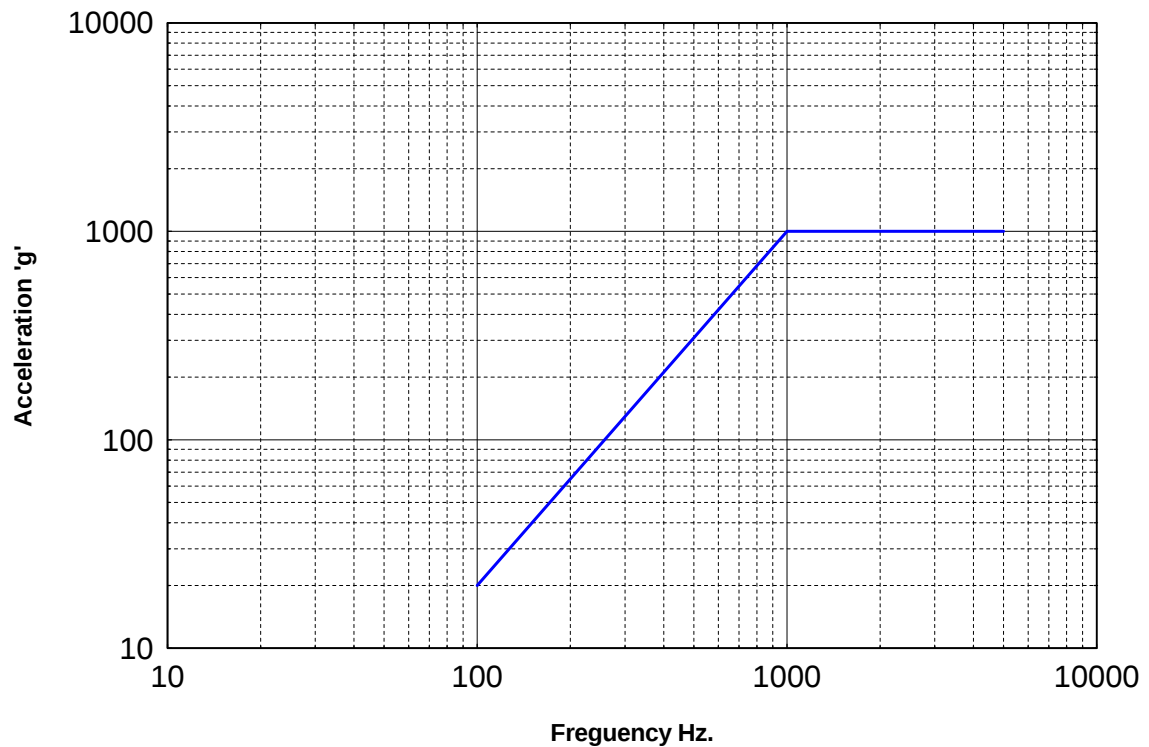


Fig. 5.3 Typical separation shock spectrum for Auxiliary S/C

5.6.8 Auxiliary spacecraft / 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.

5.7 Electromagnetic Compatibility:

Details of RF systems on board the auxiliary S/Cs shall be provided to ISRO for Electromagnetic Compatibility analysis. The details shall include:

- Carrier frequency, bandwidth, power (EIRP), antenna gain and location for Transmitters.
- Carrier frequency, bandwidth, sensitivity, frequencies and power of local oscillators, antenna gain and location for Receivers.
- Plan/sequence of operation of transmitters / receivers / transceivers during pre-launch phase (after assembly to the Vehicle).

The RF transmissions to and from the auxiliary S/Cs after they are assembled on the Launch Vehicle are subject to control by the ISRO Mission managers based on Electromagnetic Compatibility restrictions, and Safety restrictions.

5.8 Constraints

The auxiliary S/C customers are subject to the following technical and managerial constraints:

General:

- (a) The auxiliary S/C has to be compatible with the Primary S/C and should be transparent to ISRO.
- (b) The customer shall provide a representative dummy (with exact simulation of mass, Centre of Gravity and mounting interfaces) at the beginning of the launch campaign. The mass dummy will be launched in case of non-availability of auxiliary S/C on time.

Launch Phase:

- (c) The auxiliary S/C has to be ready for combined operations and assembly to the Vehicle at least 10 days before launch. The transportation and pre launch operations at Launch complex are to be planned accordingly.
- (d) The auxiliary S/C has to be totally inert without any radio transmissions during count down, launch and until the specified time after separation.

ISRO will define the time (after the auxiliary S/C separation) at which the S/C can start radio transmissions based on Electromagnetic Compatibility analysis.

(e) The separation of the auxiliary S/C will be provided under 3 axis-stabilized mode. The attitude of the auxiliary S/C at separation will be defined by ISRO. The customer cannot ordinarily insist on a preferred orientation.

5.9 Safety Requirement:

The customer shall provide detailed information on all subsystems/operations which are having bearing on either ground safety or flight safety. A typical list of such subsystems shall include:

- ✚ Pressure vessels
- ✚ Pyro and explosive devices
- ✚ Batteries
- ✚ Propellants and hazardous operating fluids.
- ✚ Ionising radiations and high voltages generated for operations.

The decisions based on safety analysis shall be final during auxiliary S/C preparation at Launch Centre, and during count down phases.

5.10 Documentation

The execution of the Launch Services agreement will be regulated technically through Interface Control Document (ICD). The ICD will be prepared by ISRO taking into account the following.

- ✚ The technical requirements of the auxiliary S/C.
- ✚ The interfaces with Launch Vehicle.
- ✚ Facilities at Launch Centre and the S/C preparation requirements
- ✚ Assembly procedures during integration.

The ICD will be reviewed and jointly approved by the customer and ISRO. The ICD shall be the controlling technical document in all phases of the execution of Launch Services agreement.

The customer and ISRO will discuss and agree to the documents to be generated and exchanged at various phases of the contract by both the parties.