UNITED START

SPACE LAUNCH SYSTEM

START –1

USER'S HANDBOOK VOLUME I:

SPACECRAFT & LAUNCH VEHICLE INTERFACES

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Abbreviations

AFTS	–	autonomous flight termination system
ATB	–	assembly and test building
EGSE	–	electronic ground support equipment
GCS	–	guidance and control system
GPS	–	Global Positioning System
GRACS	–	gas-reaction attitude control system
HM	–	head module
ICBM	–	intercontinental ballistic missile
IS	–	interstage section
LV	–	launch vehicle
MIHT	–	Moscow Institute of Heat Technology
MS	–	measurement system
PBPS	–	post-boost propulsion system
PM	–	propulsion module
SC	–	spacecraft
SLS	–	Space Launch System
SPHGG	–	solid-propellant hot-gas generator
TLC	–	transport-and-launch canister
TEGSE	–	transponder electronic ground support equipment
UP	–	panels of umbilical plugs

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INTRODUCTION

The multipurpose transportable Start-1 Space Launch System is intended for injection of small spacecraft into low earth orbits.

The Start-1 Space Launch System (SLS) includes the Launch Vehicle (LV), equipment for transportation of LV and launch system elements, and launch processing equipment necessary for Launch Vehicle/Spacecraft integration, final preparation and launch.

The solid-propellant Start-1 LV was developed in the early 1990s under conversion of rocket technologies by a group of Russian enterprises leaded by Scientific and Technological Center "Complex-MIHT".

The Launch Vehicle and its systems, as well as ground launch equipment and processing equipment were developed using elements, components and technologies for missile systems from the SS-25 ICBM, which ensures very high reliability.

This Start-1 Launch Vehicle User's Handbook consists of two Volumes:

Volume 1: **Launch Vehicle & Spacecraft Interfaces**
Volume 2: **Pre-launch Preparation, Launch & Cosmodrome Operations**

The Users Handbook is administered by United Start Corporation out of Los Angeles, CA and Puskovie Uslugi out of Moscow Russia.

1. START-1 LAUNCH VEHICLE DESCRIPTION

1.1 General

Launch Vehicle performance data are as follows:

LV type		Solid-propellant vehicle
Number of stages		4
Lift-off weight		47 ton
Maximum diameter		1.8 meter
Length		22.7 meter
Launching mode		Cold starting from a transport-and-launching canister using solid-propellant hot gas generator
Launcher		• Autonomous launching system based on 7-axis undercarriage • Launch stand

1.2 Launch Vehicle Design and Spacecraft Accommodation

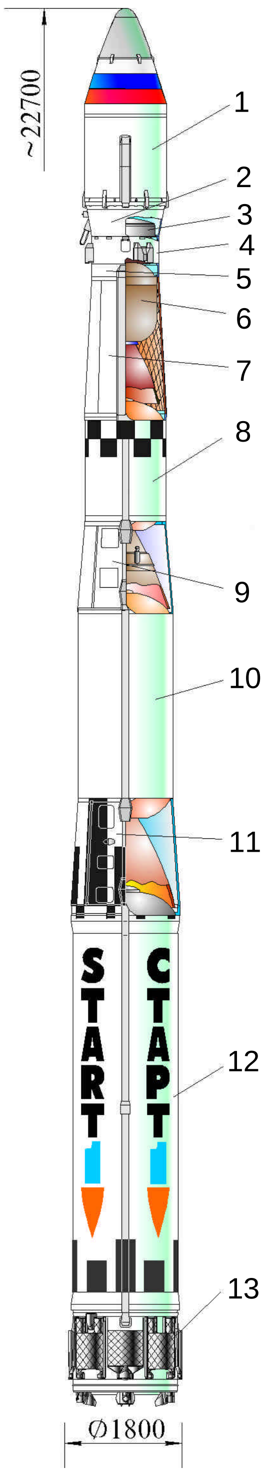
The Start-1 solid-propellant launch vehicle (Fig. 1-1 and Fig. 1-2) consists of four inline boost stages and a post-boost propulsion system.

The boost stages inject a spacecraft into a pre-determined orbit. The post-boost propulsion system provides accurate spacecraft placement in the orbit.

The boost stages are connected to one another by interstage sections (IS1, IS2, and IS3), and a post-boost propulsion system located inside the 3rd Interstage Section (IS3). The post boost propulsion system after its burnout is not separated from 4th stage.

LV upper section includes:

- Head module (HM) which includes spacecraft (SC), adapter and fairing
- Platform on which a sealed instrumentation compartment and set of guidance and control system equipment are installed
- Propulsion module (PM) in which the post-boost propulsion system and some units of guidance and control system (GCS) are installed
- Interstage section IS4.



1 – Head Module
2 – Platform
3 – Guidance and Control System Compartment
4 – Propulsion Module with Post-boost Propulsion System
5 – Interstage Section 4
6 – 4th Boost Stage
7 – Interstage Section 3
8 – 3rd Boost Stage
9 – Interstage Section 2
10 – 2nd Boost Stage
11 – Interstage Section 1
12 – 1st Boost Stage
13 – Aft Section with air vanes, fins and gas vanes

Fig. 1-1. Start-1 Launch Vehicle (Dimensions in mm)

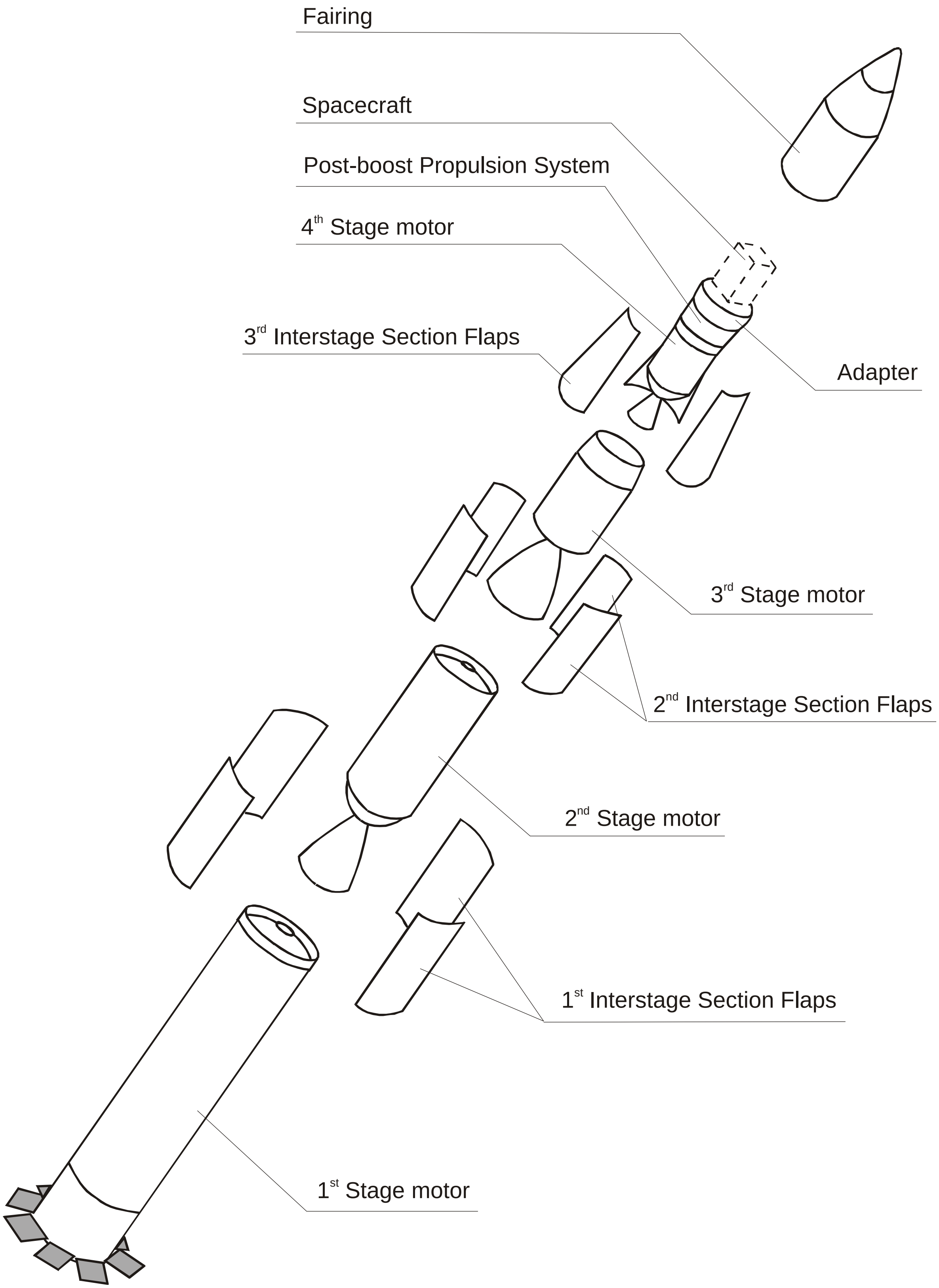


Fig.1-2. Start-1 LV exploded view

1.2.1 Boost Stages

The motor of each boost stage utilizes a composite solid propellant.

Various composite materials are used in boost stage design.

All boost stages are equipped with gas-dynamic controls in pitch, yaw and roll. 1st boost stage is equipped with both gas vanes and air vanes.

Gas-reaction attitude control system (GRACS) is located at the aft section of the 4th Boost Stage. The working medium of this system is pressurized nitrogen.

1.2.2 Post-boost Propulsion System and Propulsion Module

The post-boost propulsion system (PBPS) is provided to achieve required values of SC kinematic parameters at SC separation point in the final flight phase. The PBPS includes a solid-propellant gas generator, ducting, and three pairs of push-operated nozzle assemblies which are installed such that burnt products flow out in a direction opposite to the spacecraft location. These nozzle assemblies are placed outside the propulsion module and are sheltered by local protective covers.

On the side surface of the instrumentation compartment body there are two panels of umbilical plugs (UP-1) that are used to communicate onboard equipment (GCS and LV measuring system) with electronic ground support equipment (EGSE).

In case there are no hard requirements for orbit altitude insertion accuracy, the Start-1 LV without PBPS can be used to increase overall SC mass to be injected.

1.2.3 Sealed Instrumentation Compartment Onboard Guidance and Control System

The launch vehicle flight is controlled by autonomous inertial guidance system. The gyrostabilized platform with measuring instrumentation of GCS produces data on LV attitude and acceleration of center of mass. The onboard computer controls the flight according to the prescribed flight mission.

The main units of the GCS equipment are placed in a sealed instrumentation compartment that is installed inside the platform. Electrical connectors for onboard GCS instrument communication with LV units and EGSE are placed on rear bottom of the sealed instrumentation compartment. On side surface of the sealed instrumentation compartment there is an optically transparent window of the aiming system.

Also on the side surface of the instrumentation compartment body there is a panel with an umbilical plug (UP-2) that is used to communicate the SC onboard equipment with EGSE for SC maintenance.

On the front ring of the platform the attachment fittings for the head module mounting are placed.

1.2.4 Head Module, Payload Accommodation

The head module is an assembled unit that includes spacecraft, adapter, and fairing. It is mounted on the front end-face of the platform (see Fig. 3-1).

Head module design provides the environmental conditions required for the SC during ground operations and during ascent, up to fairing separation. The LV design and its operation mode in flight exclude SC surface contamination by exhaust products from boost stage pyrotechnical devices, retro boost motors, post-boost stage, and stage separation assemblies.

The dynamic envelope that bounds the available payload volume is shown in Fig. 1-3.

1.3 Transport-and-Launch Canister

The launch vehicle is transported and launched from a mobile transport-and-launch canister (TLC). The TLC protects LV from casual mechanical damages and along with systems of ground launch complex provides humidity-and-temperature conditions during all phases of LV operating. The LV is fixed in the TLC by sabots and seal sabots.

The TLC is placed on launch stand (Fig. 1-4 and Fig. 1-5) or mobile launcher based on undercarriage (Fig. 1-6 and Fig 1-7).

1.4 Flight Termination System

The Start-1 LV is equipped with an autonomous flight termination system (AFTS).

LV flight is terminated autonomously by the onboard GCS when LV angular error exceeds a programmable value on any of the axes. LV attitude analysis is performed by onboard computer using data obtained from gyro-stabilized platform attitude sensors. In case of computer failure the AFTS command is generated for extended angles at the closing of limit contacts placed on gyro-stabilized platform frames.

1.5 Telemetry and Trajectory Measurement Systems

The LV measurement system provides acquisition, transmission, and recording of data on operation of the LV units and systems during pre-launch preparations and LV flight as well as transmission of flight parameters, LV GCS operation and spacecraft separation.

The measurement system (MS) consists of:

- radio-telemetry equipment and its operation control instrumentation;
- autonomous power supply system
- antenna and feeder devices.

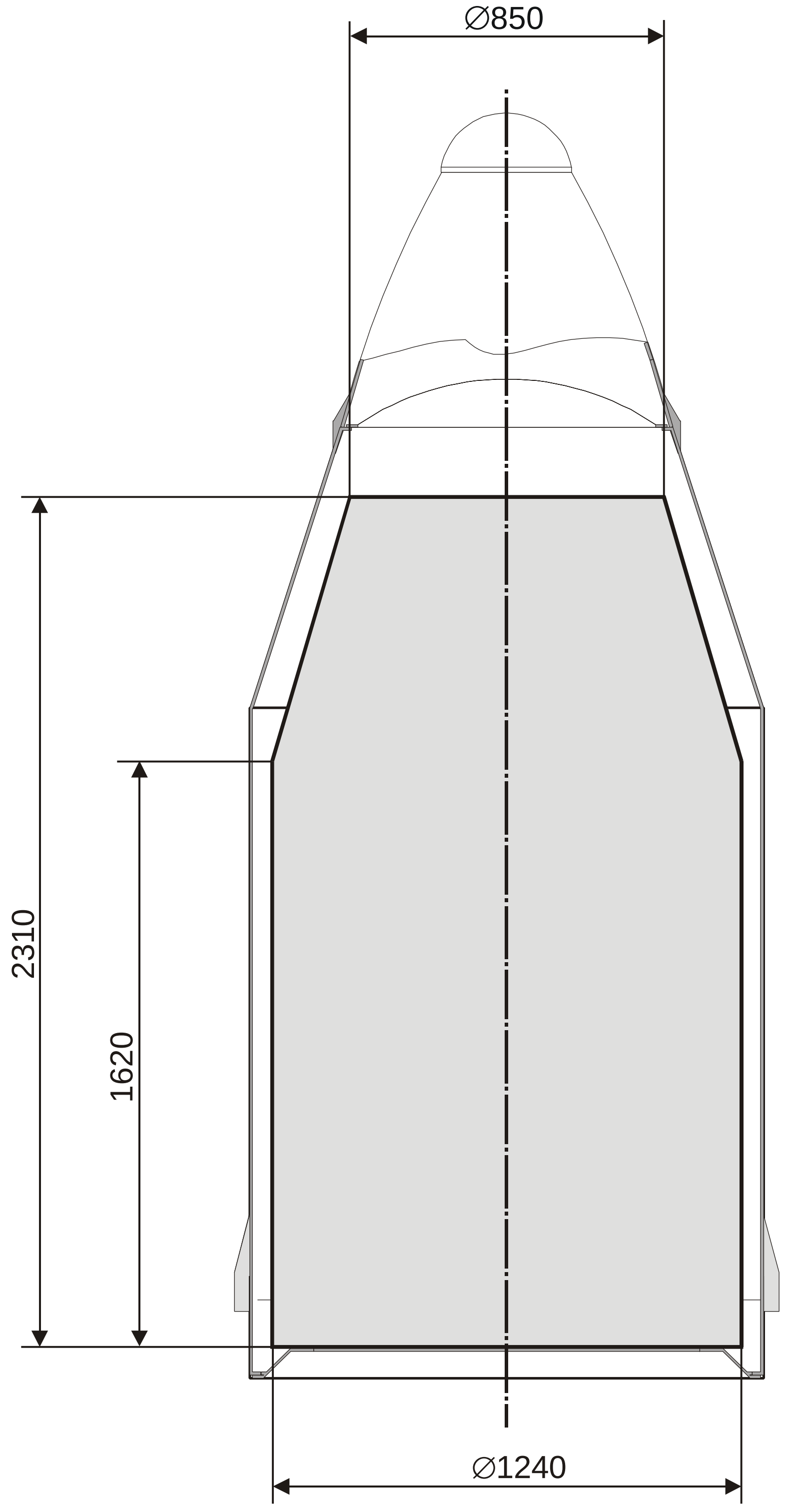


Fig. 1-3. Payload accommodation

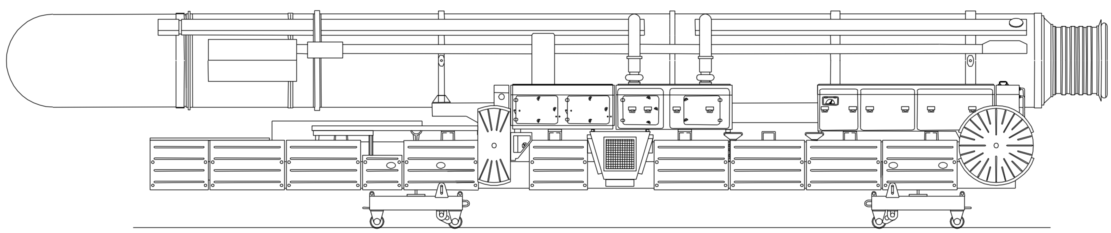


Fig. 1-4. Start-1 Launch Vehicle in TLC on Launch Stand

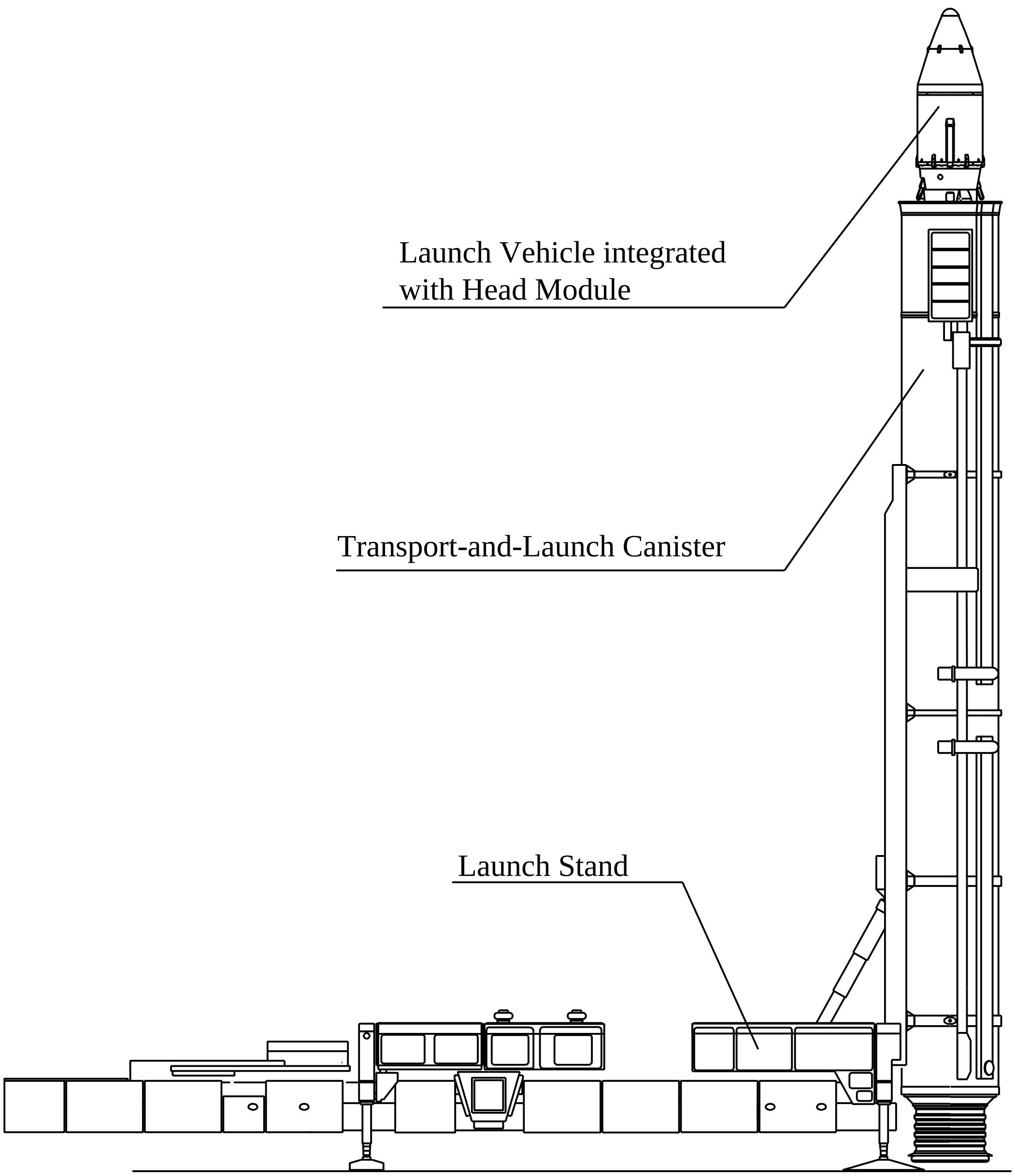


Fig. 1-5. Start-1 Launch Vehicle in launch position

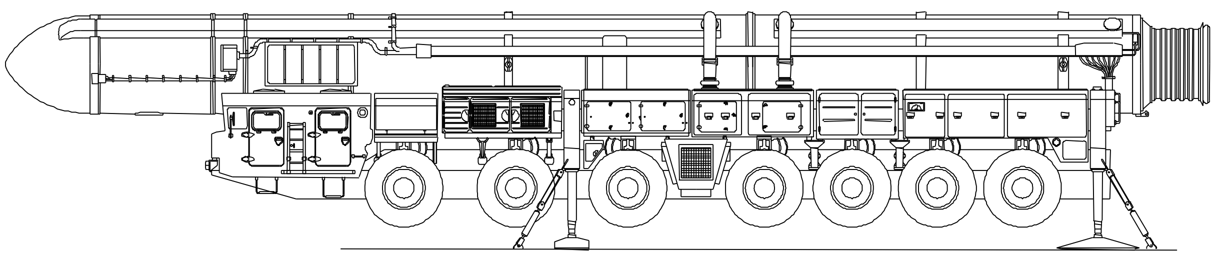


Fig. 1-6. Mobile launcher with Start-1 Launch Vehicle in pre-launch position

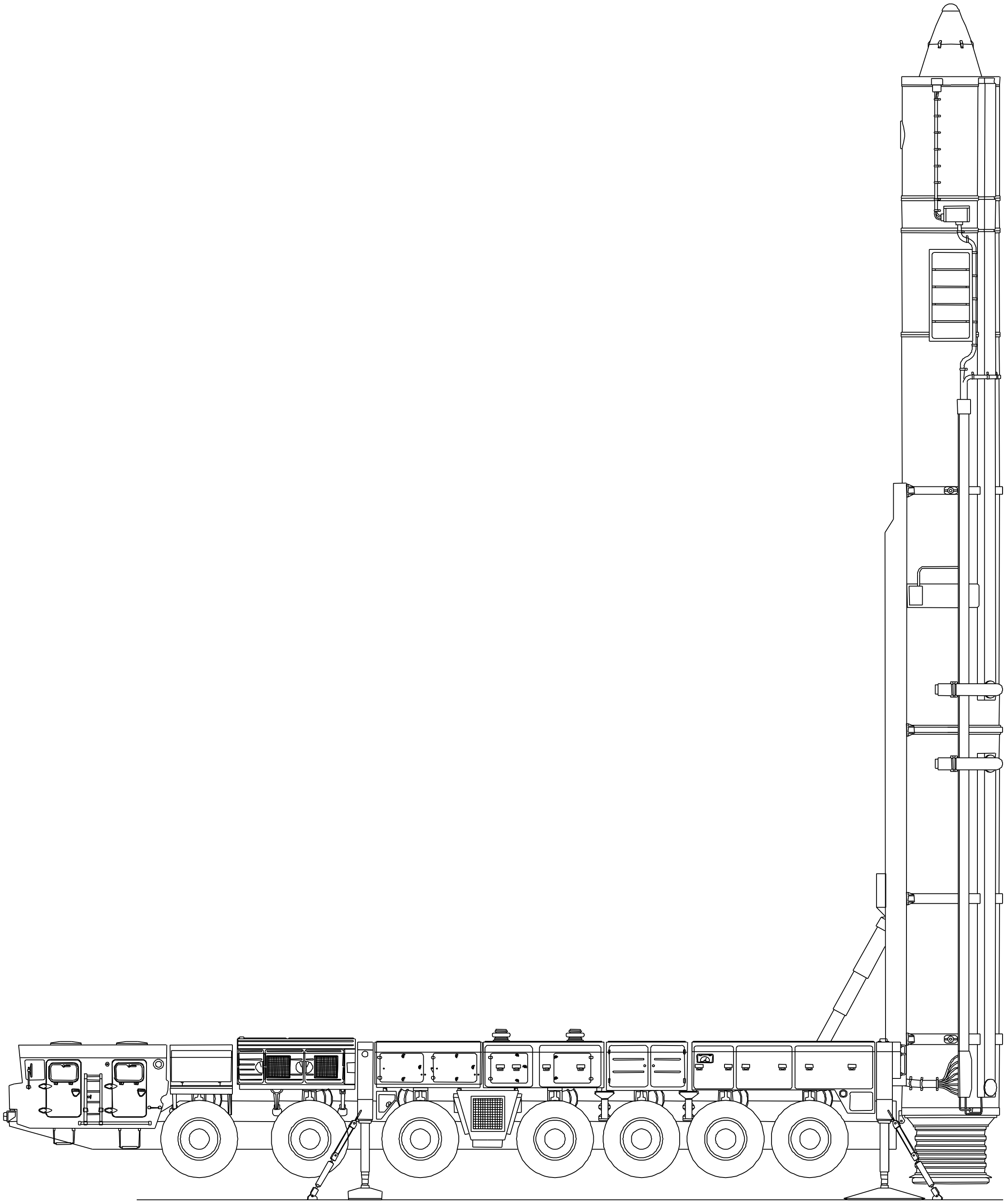


Fig. 1-7. Start-1 Launch Vehicle on mobile launcher in vertical position

Trajectory parameters are measured by onboard equipment providing communication with both GLONASS and GPS satellite navigation systems. The LV can be equipped with an onboard receiver-transmitter of trajectory measuring system as well.

Radio-telemetry system operates in the following modulation modes. Transmitter uses pulse-frequency modulation of carrier frequency. The first switching stage operates in pulse-amplitude modulation mode. The second switching stage operates in pulse-code modulation mode. Transmitter operates at a frequency of 203 MHz or 219 MHz. Transmitter power is 15-40 W.

The onboard receiver-transmitter operates in centimeter-wave band and transmitter power is about 2 W.

Spacecraft separation is monitored by LV systems via two contact sensors installed on the Adapter.

The LV onboard equipment does not provide SC telemetry downlink in flight for LV standard configuration.

As an option, it is possible to use the LV telemetry equipment for data transmission on SC functions. The number of transmitted parameters and sampling rate, electrical connections between LV and SC, parameters from sensors installed on SC, and other proposals associated with operating of the LV telemetry equipment to be used for downlink can be agreed upon by the spacecraft developer with STC "Complex-MIHT".

1.6 Launch Vehicle Reliability

The Start-1 LV reliability is supported through design and development work, theoretical investigation and calculations, manufacturing and technology activities conformable to International Standard ISO 9000-9004. These activities have been used and proven highly effective during testing, manufacturing and operation of the Start-1 ballastic missile-prototypes.

All key structural elements of the Start-1 LV have passed full ground tests including more than 30 firing tests for each type of motor.

Long term reliability of elements and systems of the Start-1 launch equipment has been proven by more than 60 flight tests of the SS-25 missile.

Reliability and validity of estimation methods used for LV reliability and also efficiency of reliability control had been confirmed by accumulated experience relating to the reliability estimation for ballastic missiles and more than 300 launches of the SS-20 and SS-25 missiles.

Serviceability of the Start-1 LV was proven by its successful demonstration launch from Plesetsk Cosmodrome on March 25, 1993 and its successful commercial launches from Svobodny Cosmodome on March 4, 1997, December 24, 1997, December 5, 2000, and February 20, 2001 (See Table 1-1).

Table 1-1
Commercial launches from Svobodny Cosmodrome

Launch date	Spacecraft	SC mass kg	Start-1 Launch Vehicle Injection accuracy								
			Mean orbit altitude			Orbit inclination, deg.			Orbital period		
			Predicted parameters (nominal)	Allowable deviations (2.7σ)	Factual deviations	Predicted parameters (nominal)	Allowable deviations (2.7σ)	Factual deviations	Predicted parameters (nominal)	Allowable deviations (2.7σ)	Factual deviatio
March 4, 1997	Zeya (Russia) Radio communications	87	490.4	±5	-1.4 / +0.9	97.27	±0.05	+0.01	5647.8	±2.5	-0.16
December 24, 1997	EarlyBird-1, EWI (USA) Earth observation	310	479.2	±5	+0.1	97.30	±0.05	+0.006	5642.7	±2.5	0.0
December 5, 2000	EROS-A1, ISI (Israel) Earth observation	247	493.9	±5	-0.1	97.33	±0.05	-0.001	5660.6	±2.5	-0.1
February 20, 2001	ODIN, SSC (Sweden) Scientific researches	240	611.5	±5	-0.5	97.83	±0.05	+0.01	5806.7	±2.5	-0.6

2 LAUNCH VEHICLE PERFORMANCE DATA

2.1 Coordinate System

The accepted coordinate system for the LV is shown in Fig. 2-1.

The X_1 axis is directed along the LV's longitudinal axis. X_1 positive direction coincides with flight direction.

The Y_1 axis lies in the I-III vertical plane and is directed upward along the III base line.

The Z_1 axis completes the right-handed coordinate system. It lies in the II-IV horizontal plane and its positive direction coincides with the IV base line.

2.2 Flight Sequence

Launch procedures are initiated at the point when the "fire" command is generated to the launch equipment. At this point the TLC with the LV is in a horizontal position on the launch stand (see Fig. 1-4) or on the autonomous launch stand (see Fig. 1-6).

Countdown cycle continues a few minutes. Then the TLC with LV is elevated into a vertical position. During elevation the spacecraft communications with ground support equipment are detached. After the TLC is in vertical position the LV mechanical links with TLC are detached and the solid-propellant hot gas generator is ignited to launch the LV from launch stand (Fig. 1-5) or mobile launcher (Fig. 1-7).

LV is ejected from TLC by pressure of a solid-propellant hot gas generator (SPHGG) system. The UP-1, which communicates ground support equipment with onboard GCS equipment and the LV measurement system, is detached via LV motion.

When the LV leaves the TLC the sabots and the seal sabot are dropped sequentially and the LV air vanes and stabilizers are opened.

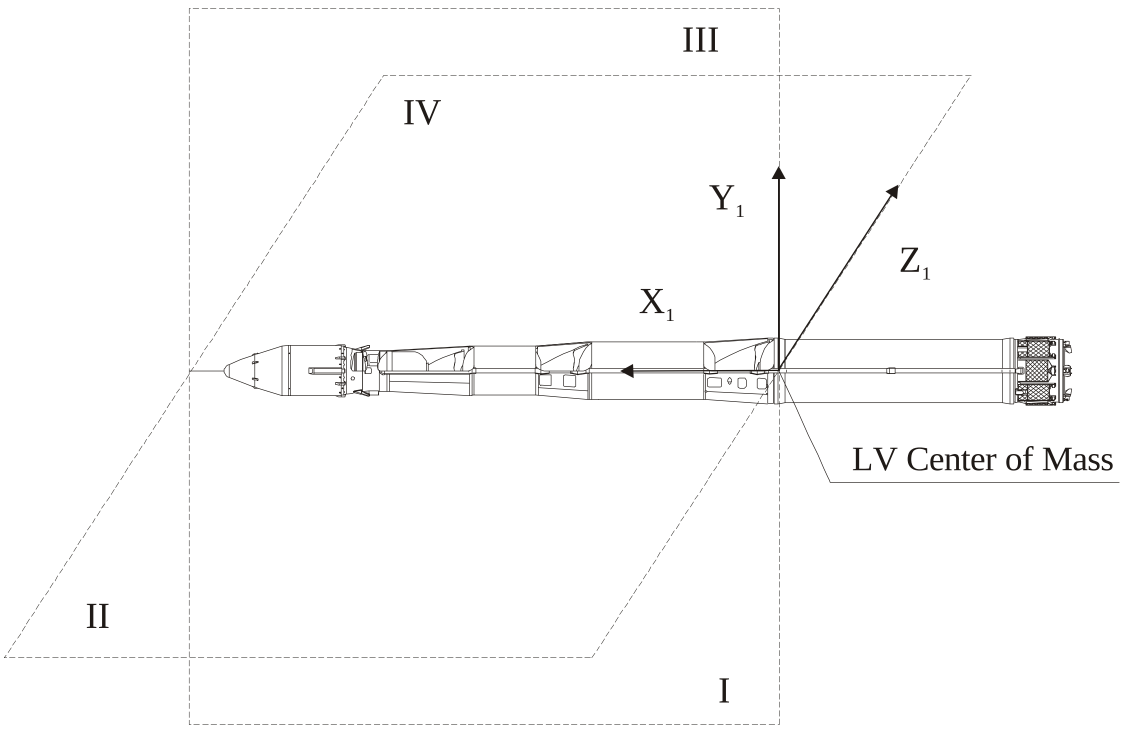
Following the LV withdrawal from TLC at a safe distance the 1st stage motor is fired.

Each boost stage operates about one minute until propellant burnout is complete.

Following the 1st stage motor burnout LV continues flight with inoperative motor at coast phase for 10-20 seconds. On finishing a prescribed coast phase the spent motor is separated and the 2nd stage motor is fired.

The interstage section connecting 1st stage motor with 2nd stage motor is released within 10-20 seconds after Stage 1/Stage 2 separation.

Stage 2/Stage 3 separation and 3rd stage motor firing are performed without a coast phase immediately after the 2nd stage motor burnout.



I, II, III, IV – Base coordinate planes

Fig. 2-1. LV Coordinate System

To place the spacecraft in an orbit of a prescribed altitude the LV flies with the inoperative 4th stage motor after separation of the spent 3rd stage motor. During the second (main) coast phase, which extends for a few hundred seconds, the stabilization and programmed orientation of LV are performed by gas-reaction attitude control system (GRACS).

During the main coast phase, LV lateral maneuver is performed to separate fairing using mechanical assemblies and then LV attitude (its longitudinal axis) is returned to the trajectory plane.

When the altitude is close to a prescribed SC orbit altitude the guidance and control system generates 4th stage motor ignition command.

At the end of the 4th stage motor firing the LV achieves an orbit close to the prescribed one. Following the 4th stage motor burnout the post-boost stage is ignited with a small delay (up to 5 seconds).

The post-boost propulsion system operates until burnout is complete.

By the end of the 4th stage motor burning the LV motion program during post-boost stage operation is formed with allowance for real values of LV kinematic parameters in such a way that the LV kinematic parameters conform to prescribed SC orbit with a required accuracy when the post-boost stage burnout is complete.

At the end of PBPS thrust decay the GRACS is activated again to control LV attitude.

The LV is turned so that the required SC orientation can be provided at the separation point.

The GCS generates a spacecraft separation command within about 375 seconds after the 4th stage motor is burnt out. Under that command the SC explosive fixing locks are opened and spacecraft is separated from the LV using spring assemblies.

Following the spacecraft separation the LV is turned in the SC orbit plane through 90° relative to the SC velocity vector and then LV is rotated by the GRACS.

2.3 Flight Performance

In Table 2-1 and Fig. 2-2, the injected payload mass dependence of the circular orbit altitude for different launch points and inclinations is given.

Table 2-1.

Injected payload mass dependence of the circular orbit altitude for different launch points and inclinations

Launch point		Altitude, km				
		1000	800	600	400	200
With PBS	i = 52°	204	295	395	505	632
	i = 90°	105	186	275	374	488
	i = 98° SSO	86	165	250	347	458
Without PBS	i = 98° SSO	150	250	350	450	550

An example of a trajectory for launch into sun-synchronous orbit at 481 km from the SVOBODNY Cosmodrome is shown in Fig. 2-3, and its respective burn sequence is shown in Fig. 2-4.

Fig. 2-5 presents the trajectory for the previous example without a post-boost stage. In this case after 4th stage motor thrust decay the LV GCS records zero thrust and activates GRACS which controls according to an algorithm developed for LV standard configuration. Spacecraft separation, as in LV standard configuration, is performed within about 375 seconds after the 4th stage motor burnout. The stated operation mode provides successful spacecraft injection with probability is no lower than for the PBPS configuration.

2.4 Spacecraft Injection Accuracy

The spacecraft injection accuracy is characterized by maximum deviations of orbit parameters at the SC separation point (with a confidence of 0.993) shown in Table 2-2.

Table 2-2, Deviations of spacecraft orbital parameters

Orbital parameter	Orbital parameter deviation	
	Standard LV configuration	LV configuration without PBPS
Orbit altitude, *km*		
– at injection point	±5	±5
– opposite to injection point	±5	±85
Orbit inclination, *arc. min*	±3	

2.5 Attitude, Linear and Angular Velocities of Spacecraft after Separation

Spacecraft's longitudinal axis orientation during separation from LV can be preset in both orbital coordinate system and inertial launch coordinate system. Attitude accuracy at the separation point is ±5°.

At separation point, the spacecraft is impacted to an additional linear velocity increment equal to 0.7-0.15 m/s ±10% depending on the spacecraft mass. Nominal value of this increment is taken into account when a command on spacecraft separation is generated. Considering spacecraft separation velocity spreads, the orbit injection errors do not exceed values given in Table 2-2.

After separation, the spacecraft angular velocities do not exceed (with confidence of 0.993) ±1.3 deg./s in pitch and yaw, and 1.0 deg./s in roll. (to be specified in view of spacecraft's inertial-mass properties).

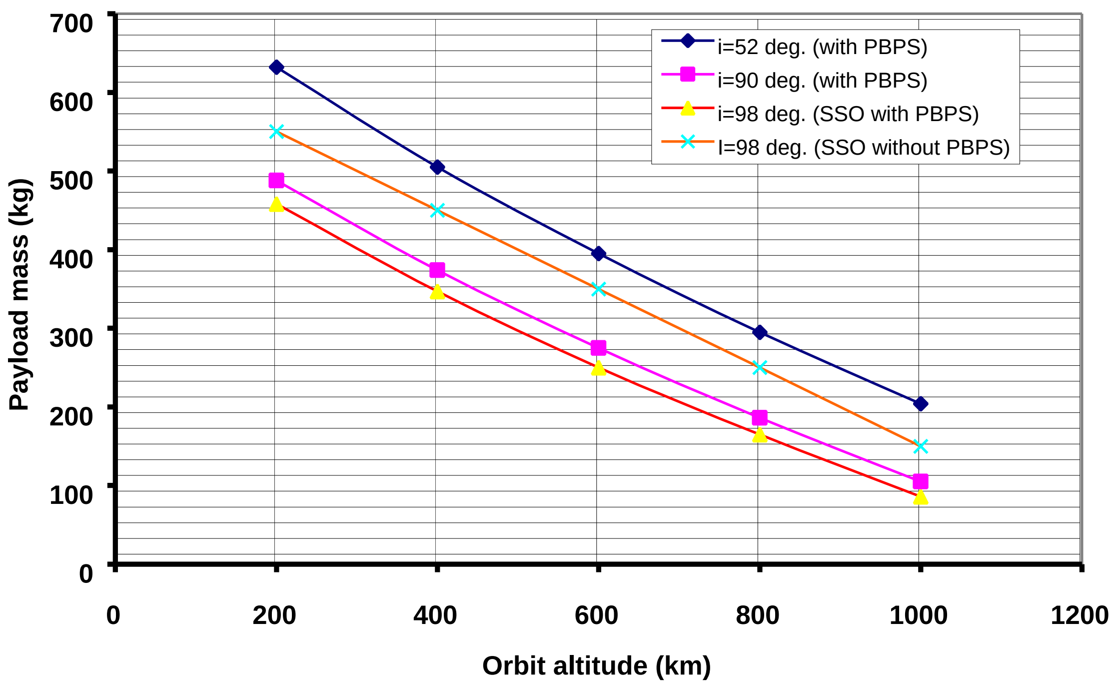


Fig. 2-2. The injected payload mass dependence of the circular orbit altitude. Launches from Svobodny Cosmodrome.

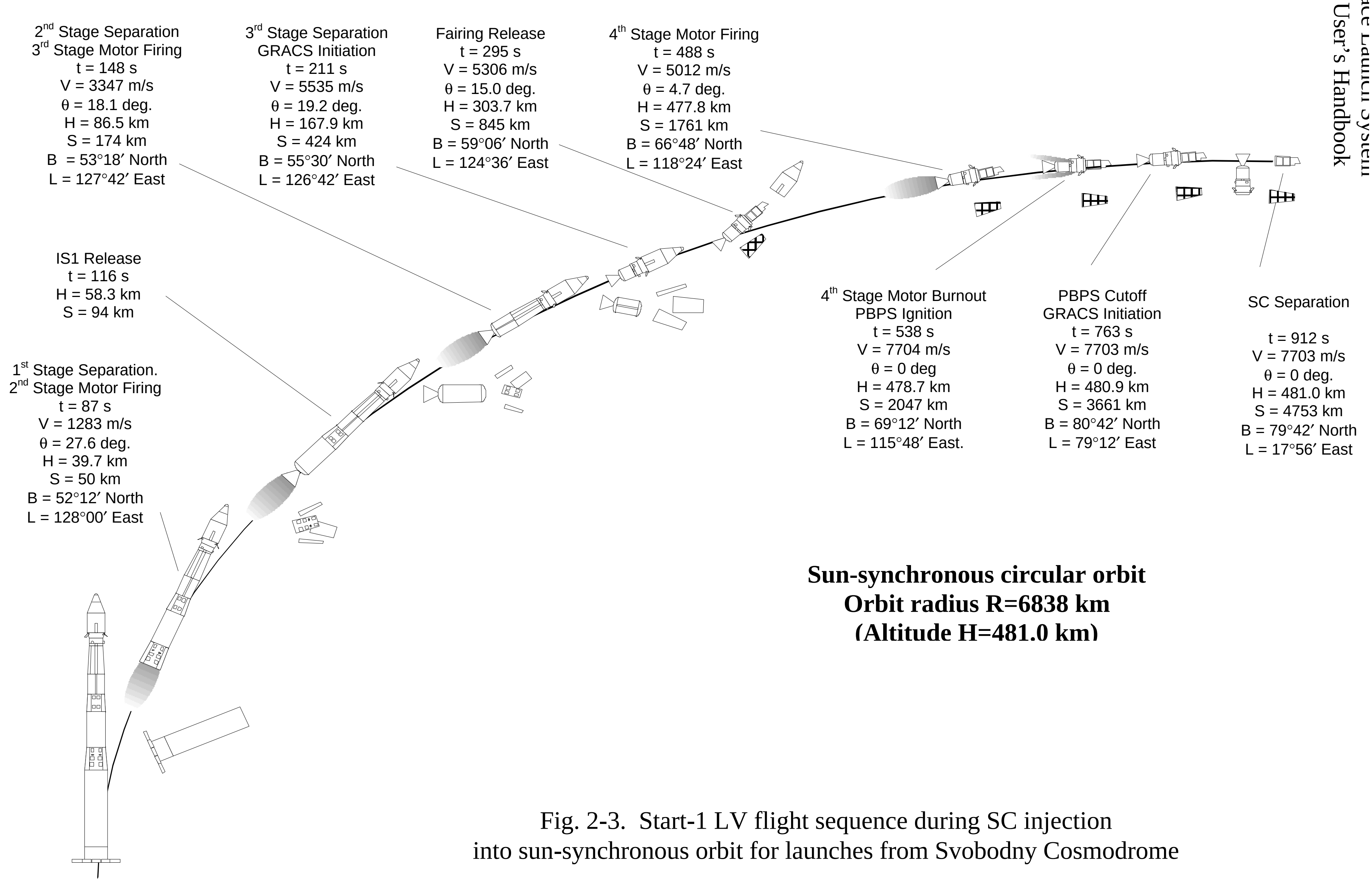


Fig. 2-3. Start-1 LV flight sequence during SC injection
into sun-synchronous orbit for launches from Svobodny Cosmodrome

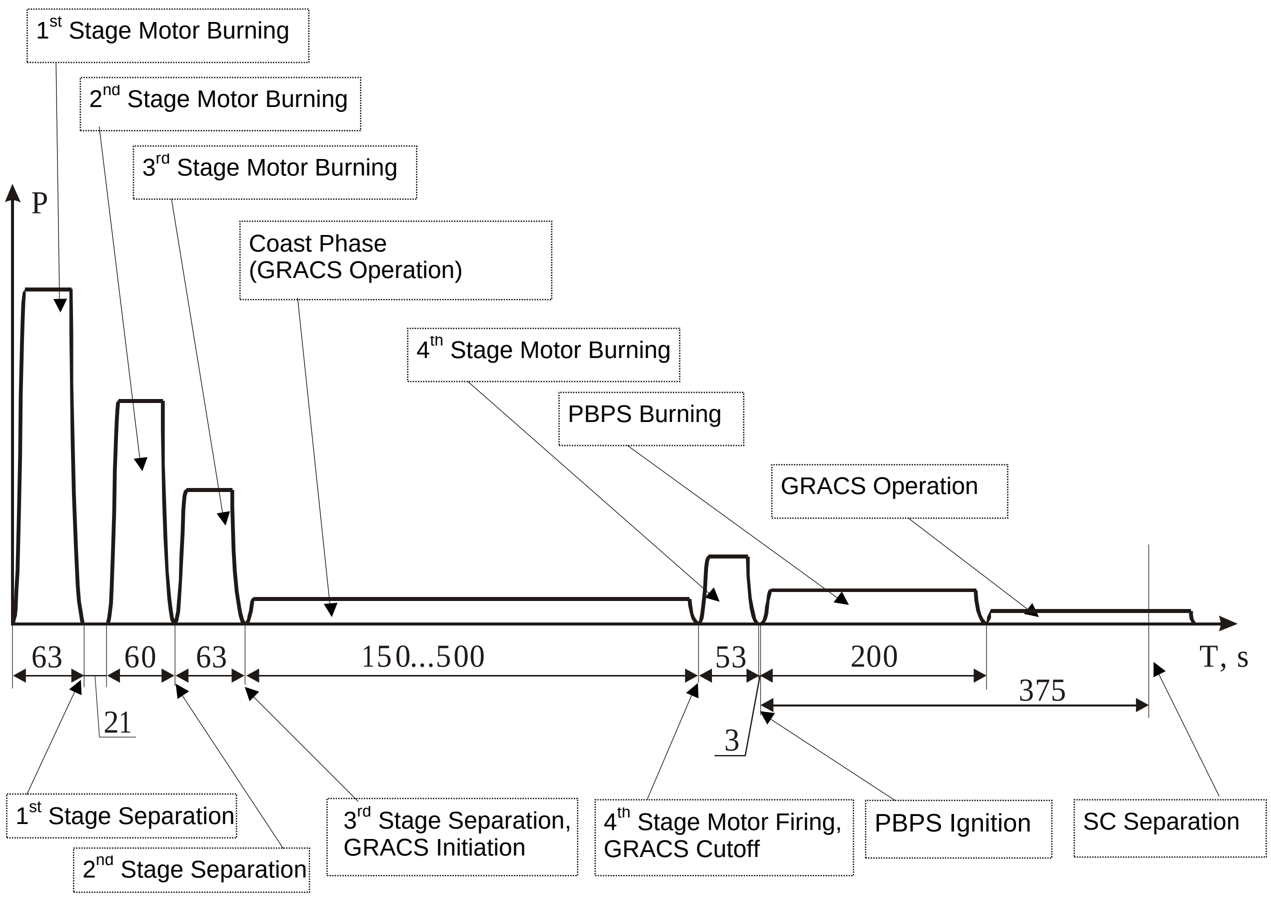


Fig. 2-4. Start-1 LV burn sequence during SC injection into
sun-synchronous orbit for launches from Svobodny Cosmodrome

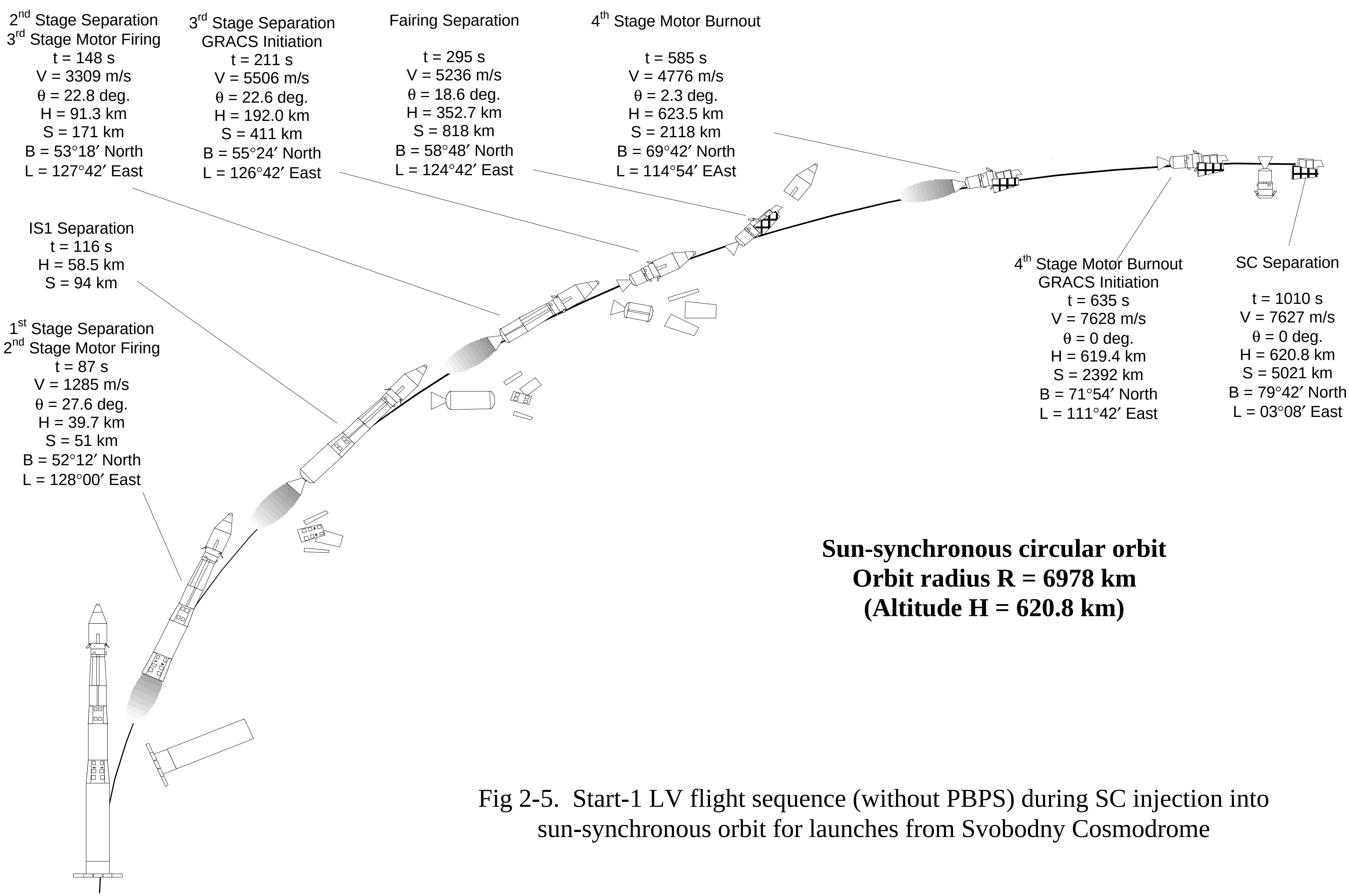


Fig 2-5. Start-1 LV flight sequence (without PBPS) during SC injection into sun-synchronous orbit for launches from Svobodny Cosmodrome

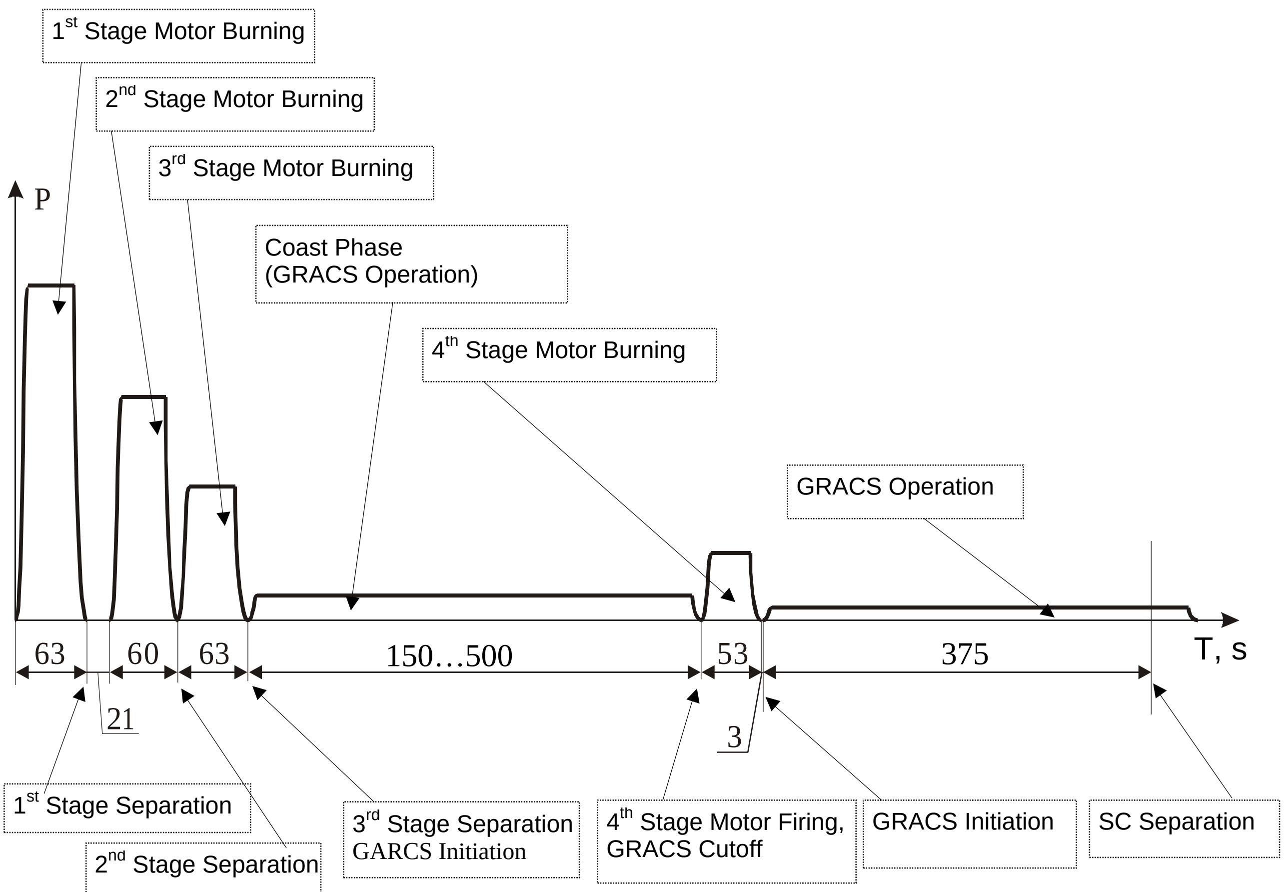


Fig 2-6. Start-1 LV burn sequence (without PBPS) during SC injection into sun-synchronous orbit for launches from Svobodny Cosmodrome

3 MECHANICAL AND ELECTRICAL INTERFACES BETWEEN LV AND SC

3.1 Mechanical Interface

3.1.1 Head Module

The head module (Fig. 3-1) is designed for:

- protection against atmospheric effects during pre-launch period, launching phase and in flight up to the fairing separation;
- maintaining the environment around the SC and providing environmental cleanliness specified by a Customer;
- maintaining a specified temperature inside the fairing;
- LV/SC mechanical and electrical interfaces;
- Providing a shockless fairing separation.

3.1.2 Adapter

The adapter (Fig. 3-2) provides the mechanical interface between LV and SC. The standard adapter has an aluminum alloy frame-type structure formed by two (front and rear) end rings joined to each other by a system of rods. To seal the head module the bottom is installed on the rear ring.

On the front ring of the standard adapter (Fig.3-3 and Fig. 3-4) the following devices are installed:

- three explosive locks for spacecraft attachment (Fig. 3-5);
- four separation spring assemblies for spacecraft separation (Fig. 3-5);
- two connectors;
- two sensors for spacecraft separation monitoring.

On the rear ring of the standard adapter the pressure equalization valves are installed so as to reduce effects upon the SC at fairing release. The pressure equalization valve design is a titanium-alloy case with a piston. The piston is moved by the pressure of gases produced by the explosive charges and opens the holes in the case which provides opening the interior of the head module to outside atmosphere. The gaseous combustion products are remained in the enclosed volume. The operation of these valves is controlled by two sensors which give a signal about the piston movement point.

In the interior of the adapter sensors are installed for measuring environment parameters inside the fairing and external effects. Number and type of these sensors to be agreed upon with a launch Customer. When SC is integrated with adapter, a maximum value of spring pusher pressing force is 106 ±1 *kgf*. Pusher stroke is 30 mm.

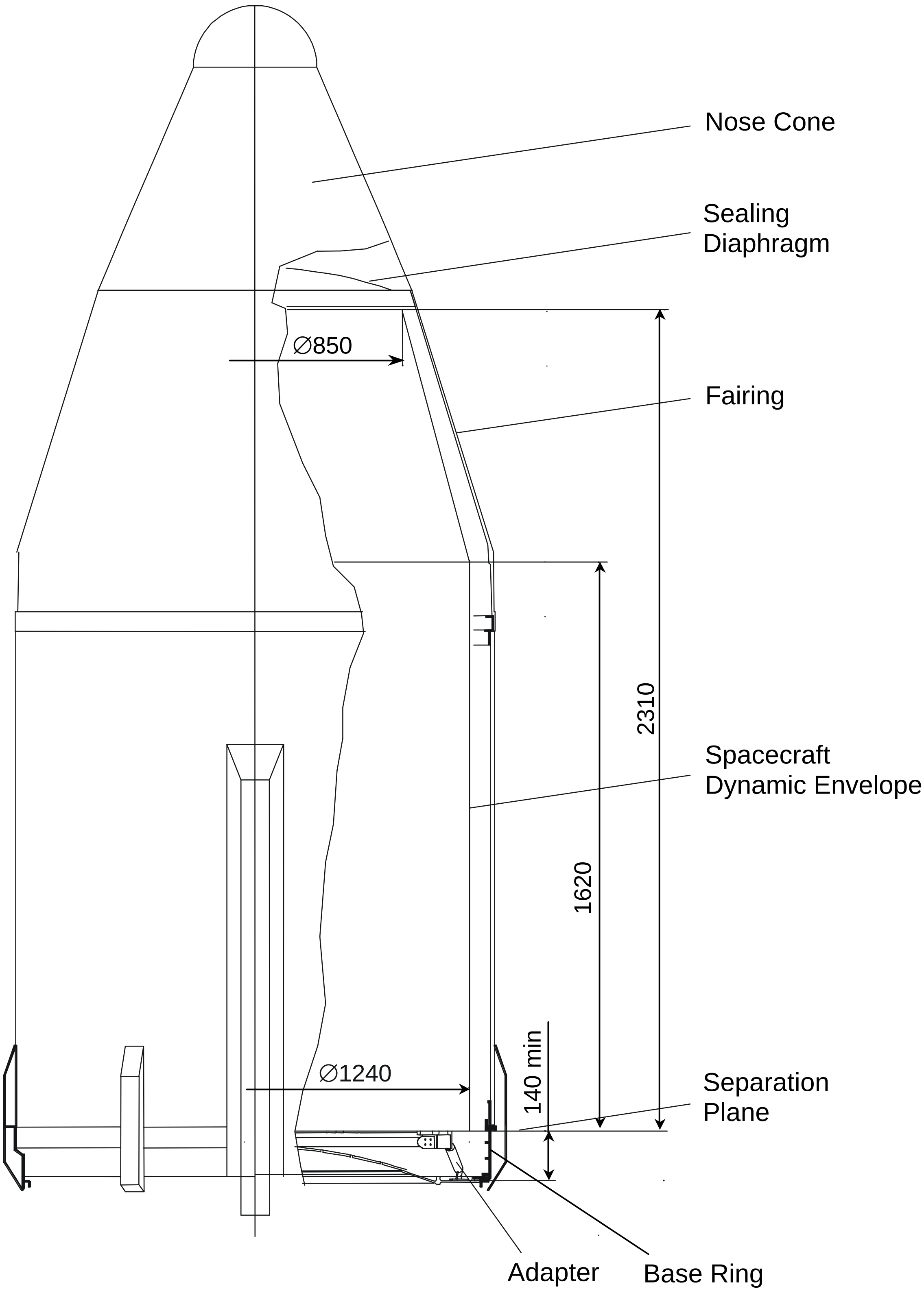


Fig. 3-1. Start-1 LV head module

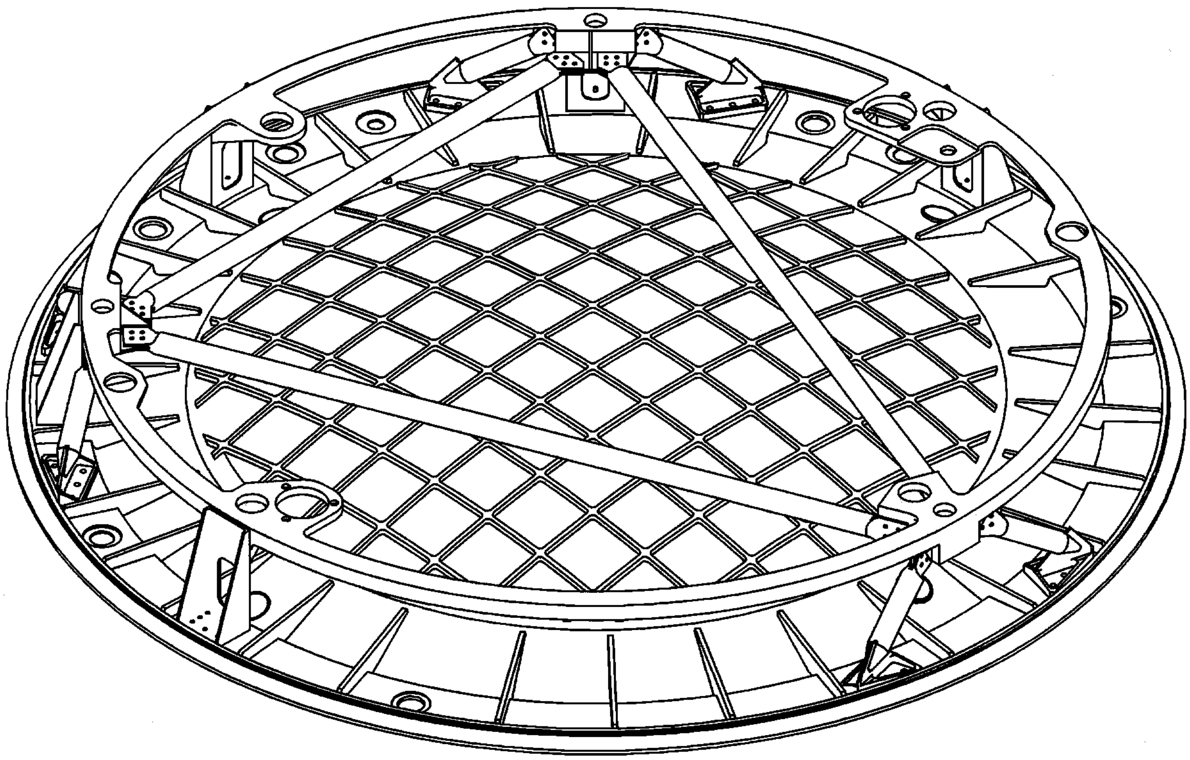


Fig. 3-2. Adapter design

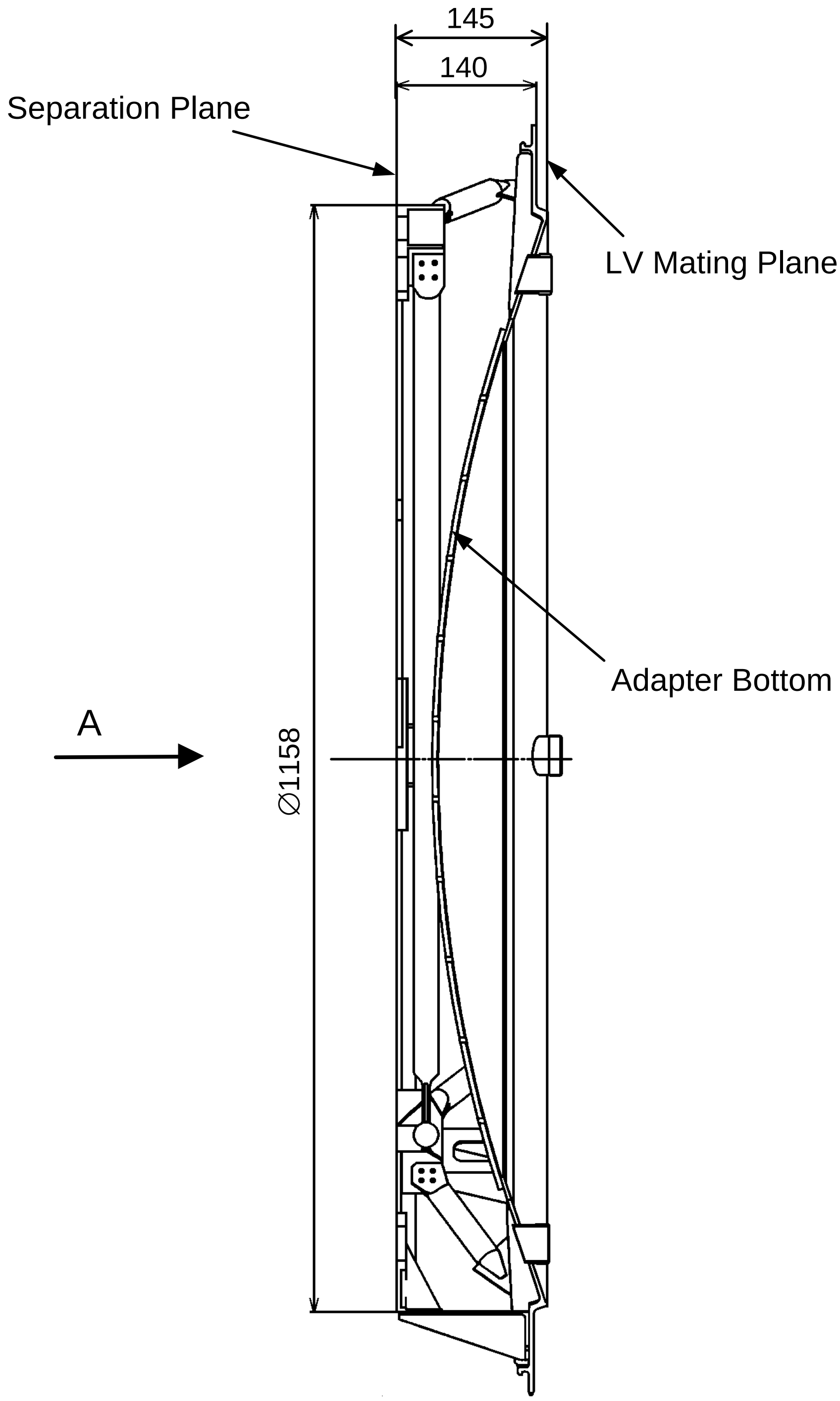


Fig. 3-3. Start-1 LV standard adapter

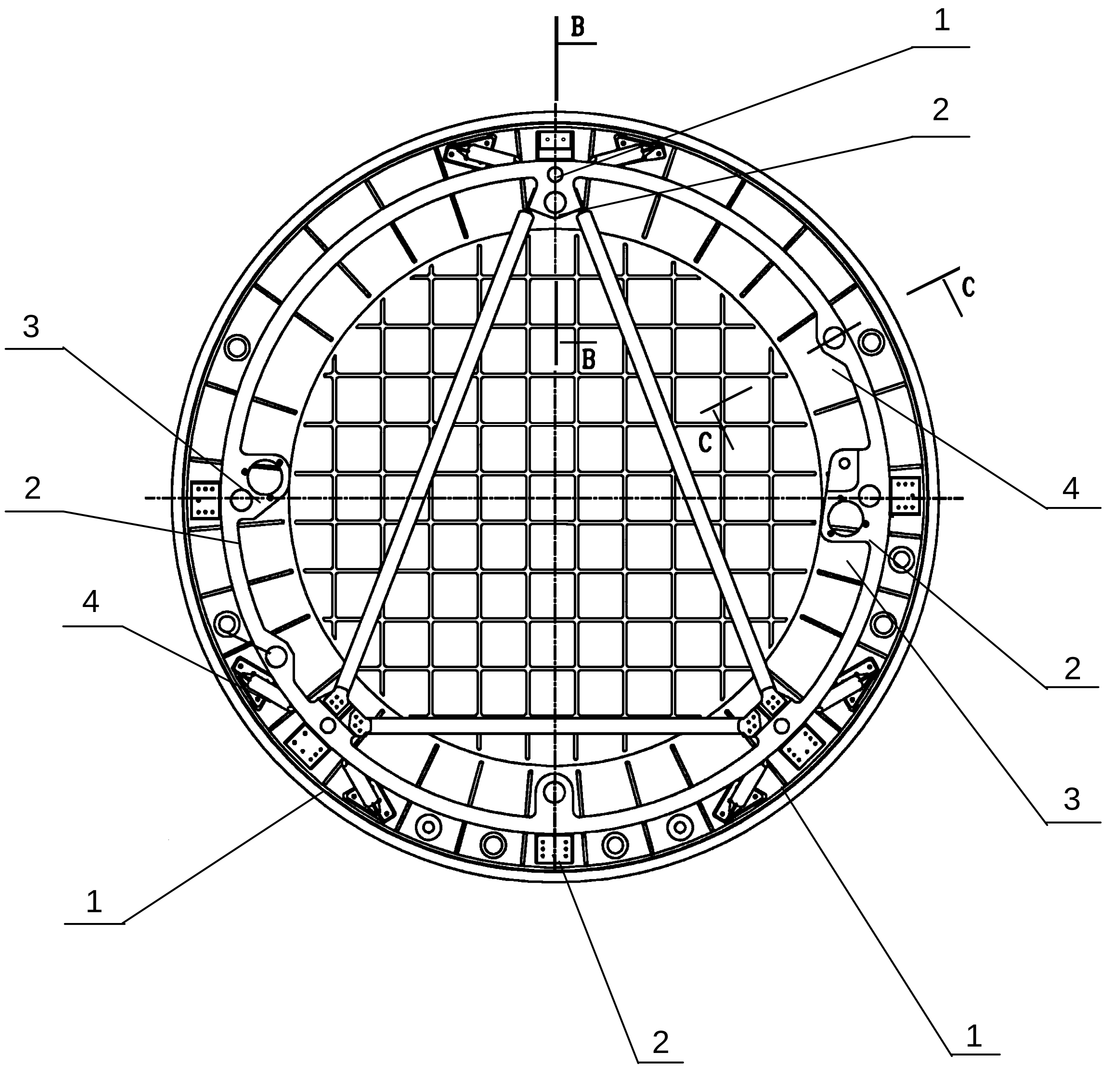
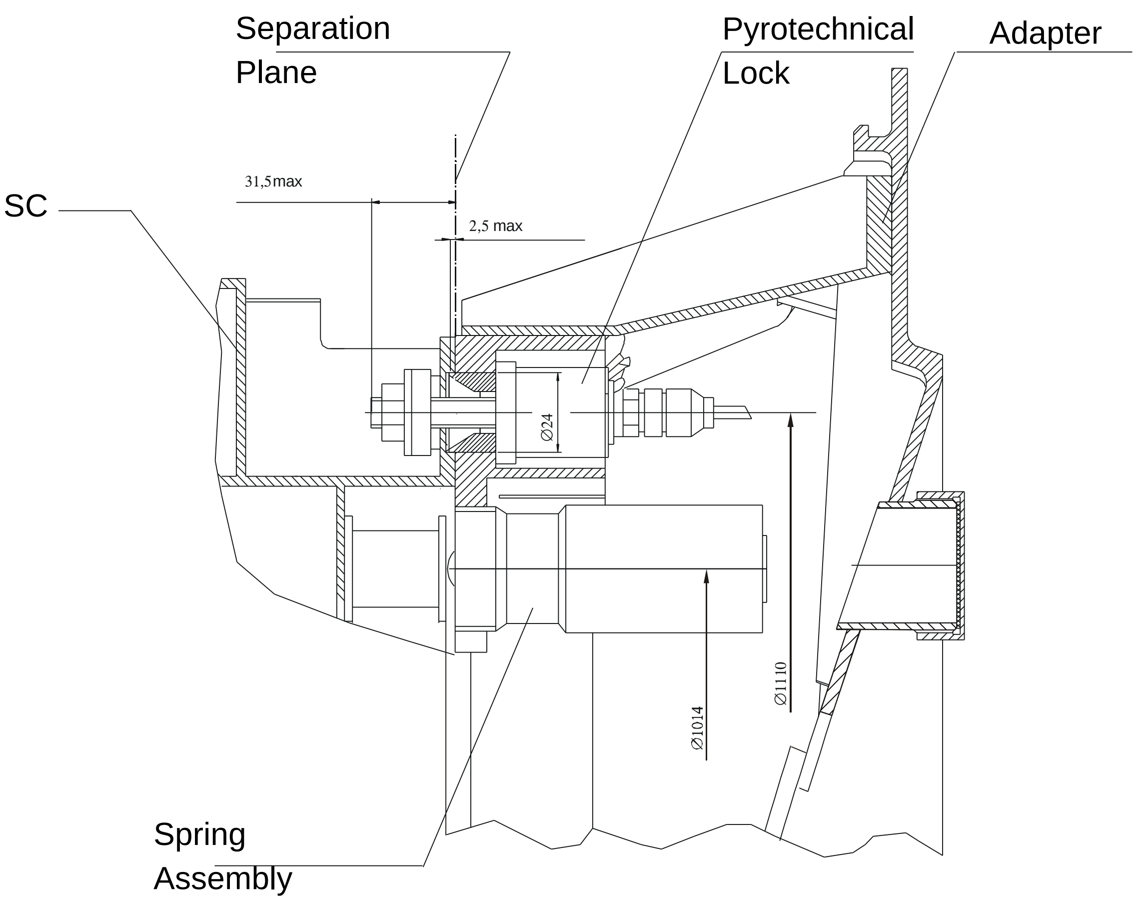


Fig. 3-4. Start-1 LV standard adapter. View A.

1. Pyrotechnical lock
2. Spring assemblies (Pushers)
3. Electrical connectors
4. Separation sensors

B – B



Position after separation

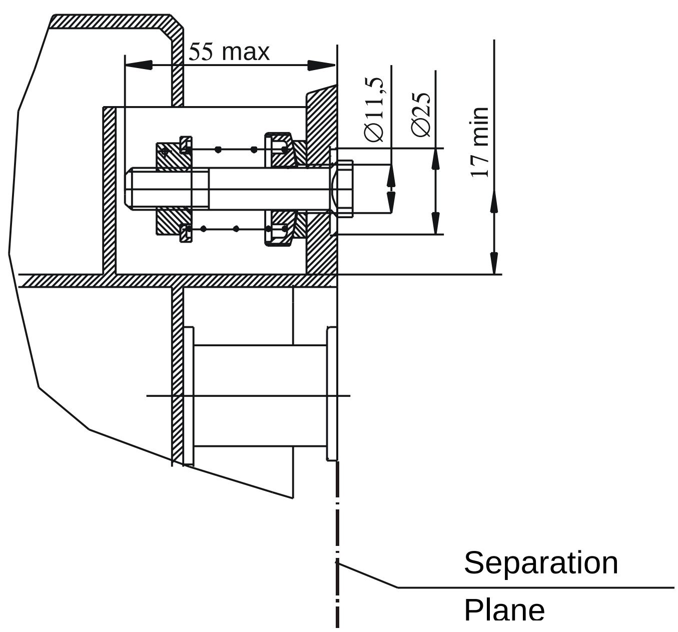


Fig. 3-5. Start-1 LV standard adapter. Cross Section B – B.

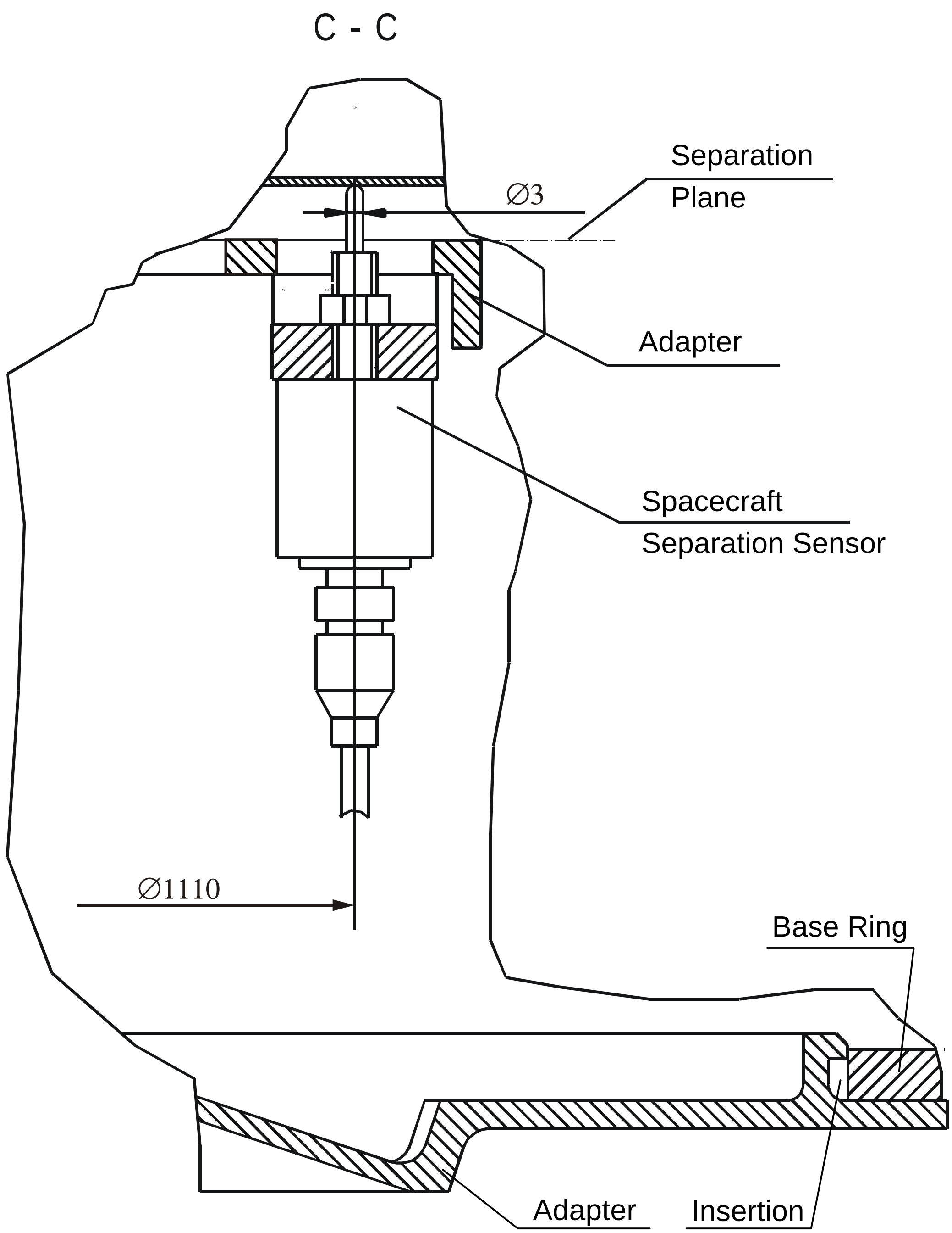


Fig. 3-6. Separation sensors.
Start-1 LV adapter. Cross Section C – C

3.1.3 Fairing

Fairing case (Fig. 3-1) consists of conical and cylindrical sections made of composite materials and it is attached to the support ring made of aluminum alloy. Fairing external surface is covered with a thermal-protective coating to reduce thermal effect on SC in flight. The fairing is RF isolated.

To seal the head module a spherical diaphragm is mounted on the front ring of the fairing. Fairing cap is attached to the same front ring. In order to eliminate contamination and gases in the space occupied by a spacecraft the internal surface of fairing is made of materials that have total mass losses of no more than 1% and total mass of gassing condensate of no more than 0.1%.

The fairing support ring provides fairing/adapter and HM/LV mechanical connections. Fairing case, released in flight, is attached to the support ring by six explosive bolts. To reduce impact action the explosive bolts are initiated in pairs at interval of about 0.1 second. After pyrotechnical devices are initiated the fairing is released by two spring assemblies. To protect the spacecraft from collision with the fairing the 650 mm guide rails for motion of fairing rollers are mounted on the support ring. To monitor fairing release the sensors are installed on these guide rails (two sensors on each of guide rails) which record the points in time of roller movement over guide rails.

The interior volume of the fairing is hermetically sealed. The sealing is provided by gaskets installed at the following places:

- between the front face of the front ring of the fairing and the spherical diaphragm;
- between the separated part and the base ring of the fairing;
- between the adapter and the base ring (see Fig. 3-6).

The degree of non-tightness is 5…7 mbar/h within the range of over-pressure of 10…60 mbar.

3.1.4 Spacecraft Access

For LV in standard configuration the access to spacecraft is terminated on finishing of head module integration. After that the monitoring of onboard spacecraft system and charging its batteries via EGSE of spacecraft are provided.

As agreed upon with a spacecraft developer, it is possible to remove the front part and the front diaphragm of fairing for access to the spacecraft in case there are no exacting requirements for SC environment parameters.

3.1.5 Optional Services

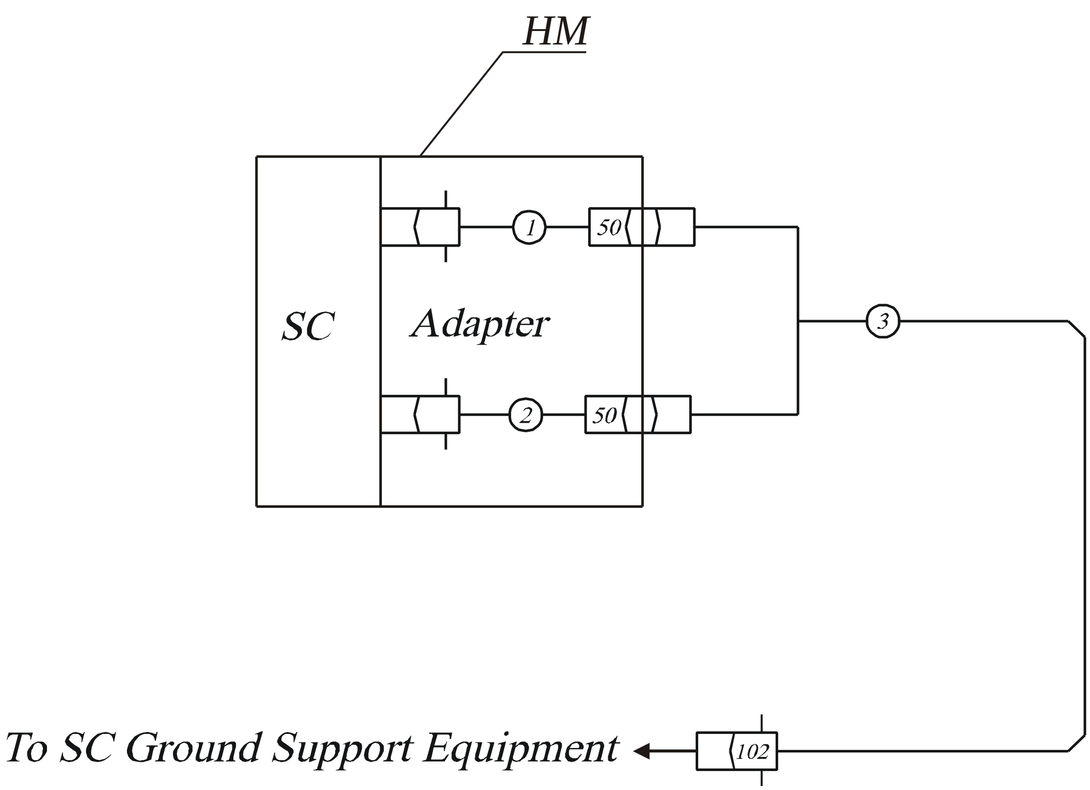
The following optional services can be offered as agreed upon with a Customer:

- the development of a unique adapter for a spacecraft;
- providing of gaseous environment in given composition (for example, clean dry nitrogen) for SC integrated with head module.
- access to the encapsulated spacecraft using a special doors on the fairing; in this case environmental parameters are not controlled.

3.2 Electrical Interface between Spacecraft and Launch Vehicle

3.2.1 Electrical Connections

Main variant of electrical connections between spacecraft and ground support equipment during pre-launch work for a spacecraft integrated with head module in the SC ATB is shown in Fig. 3-7.



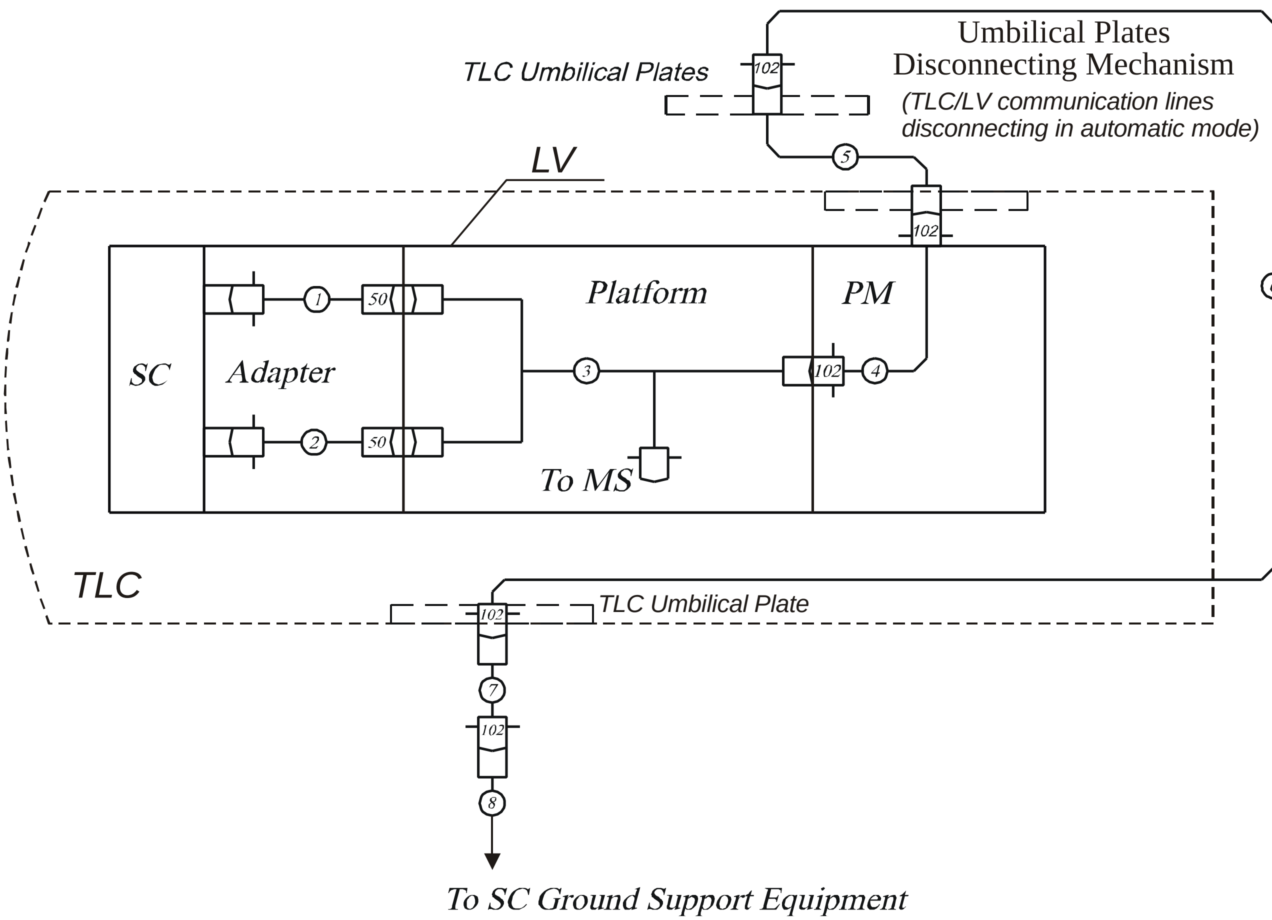
1,2 – adapter cables
3 – ground extension cable for communication with electronic ground support equipment

Fig. 3-7. Electrical connections between SC integrated with HM and SC ground support equipment.

Main variant of electrical connections between spacecraft and ground support equipment during pre-launch work for a spacecraft integrated with launch vehicle in the LV ATB and at Launch Site are shown in Fig. 3-8.

Electrical communication between SC and ground support equipment is realized via umbilical plate connectors and connector installed on TLC.

A specific variant of electrical connection between SC and ground support equipment to be defined on the basis of Customer's initial data and will take account the number of electrical lines and connectors, and their layout. Electrical parameters shall be additionally agreed upon with spacecraft and launch vehicle developers.



1-4 – LV cables; 5,6 – TLC cables;
7 – ground extension cable for communication with electronic ground support equipment

Fig. 3-8. Electrical connections between SC integrated with LV and SC ground support equipment

3.2.2 Spacecraft Grounding

During spacecraft integration with launch vehicle the spacecraft is grounded through the LV structure elements.

3.3 Information Interface between Spacecraft and Launch Vehicle

3.3.1 Standard Information Interface

Information interface between spacecraft and launch vehicle is not provided for LV in standard configuration.

3.3.2 Optional Services

If necessary, information interface can be provided through the extension cable line connected to LV guidance and control system.

3.4 Telemetry Interface between Spacecraft and Launch Vehicle

3.4.1 Spacecraft Ground Monitoring during Pre-launch Operations

Ground monitoring for SC systems through communication channels between spacecraft and SC EGSE is provided for LV standard configuration. After SC is installed inside the fairing the operations in radio-frequency emission mode via these communication channels is terminated.

3.4.2 Telemetry Monitoring at Injection Phase

SC telemetry monitoring during injection phase is not provided by LV standard configuration except for transmitting of information about SC separation from separation sensor installed on the adapter.

3.4.3 Optional Services

The following options can be provided as agreed upon with a Customer:

- telemetry data transmission through radio-frequency channel from launch system to a place where SC ground telemetry equipment is located.

- transmission of data about functioning of SC onboard systems during injection phase using LV measuring system. To perform this process the LV should be additionally equipped with telemetry switching devices and also the required number of data transmission channels should be provided.
- transmission of SC separation data that duplicates data from a standard sensor of SC separation (for example, breaking of jumpers at SC separation point).

In LV standard configuration the dynamic loads are measured by two sensors installed on adapter at the LV/SC interface.

As option, the installation of six sensors for measuring of dynamic loads along three axes can be developed. These sensors are installed on the adapter as viewed from SC.

4 SPACECRAFT ENVIRONMENTS

4.1 Flight Accelerations

During the LV flight, accelerations that spacecraft is subjected to do not exceed the values given in Table 4-1. Lateral accelerations act in any direction perpendicular to LV's longitudinal axis and simultaneously with longitudinal accelerations. Fig. 4-1 presents max. longitudinal acceleration values n_x dependence of the spacecraft mass during the 4th Stage motor burning.

Table 4-1
Start-1 LV flight accelerations

Flight phase		Accelerations	
		n_{x1}	n_{y1}, n_{z1}
1	TLC Erection and Launch	2.8	2.0
2	1st Stage Motor Burning	5.15	0.7
3	2nd Stage Motor Burning	6.5	0.6
4	3rd Stage Motor Burning	6.5	0.4
5	GRACS Operation	0	< 0.01
6	4th Stage Motor Burning	10.0	0.5
7	PBPS Burning	0.1	0.03

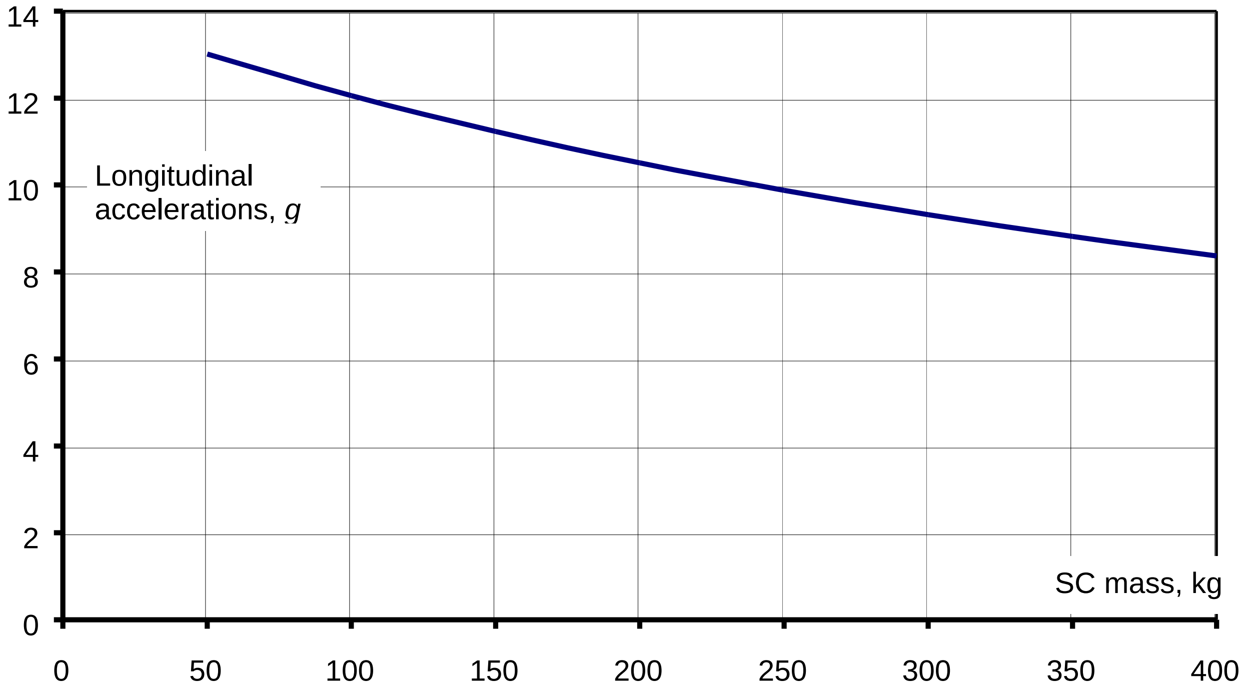


Fig. 4-1. The maximum longitudinal acceleration dependence of the SC mass for the 4th Stage

4.2 Spacecraft Angular Velocities & Accelerations in LV Flight

The spacecraft angular velocities do not exceed 25 deg./s along SC axes. SC angular accelerations during powered flight do not exceed 24.5 deg/s^2 in roll, 11 deg/s^2 in yaw, and 22 deg/s^2 in pitch.

4.3 Vibrations

During the LV flight, acoustic loads are the predominant type of a long dynamic effects acting on the payload. These loads are caused by pressure oscillations in the boundary layer of the approaching flow. Steady-state vibrations recorded at payload attachment places are secondary loads that are caused by acoustic noise effect on structural elements when the noise passes inside the fairing through its casing.

Vibro-shock processes are damping transient vibrations. Low-frequency vibro-shocks occur when launch vehicle leaves the TLC and Stage motors are fired. High-frequency vibrations are induced at initiation of pyrotechnical devices.

4.3.1 Acoustic Noise

Acoustic loads depend on change of dynamic pressure of approach flow.

During powered flight, the acoustical loads achieve a maximum value at transonic flight speed. Acoustic noise spectrum is presented in Table 4-2 and Fig. 4-2 (here 0 *dB* = $2 \cdot 10^{-5}$ *Pa*). A maximum value of the acoustic noise level is 138 *dB*, action time – 30 seconds.

At qualification testing for acoustic noise, load level specified in Table 4-2 is increased by 3 *dB*. Testing time is 90 seconds. The acceptance test modes are given in Table 4-2. Load time is 60 seconds. These values of test modes are minimum allowable for ground tests.

During tests, the spacecraft should be held by hanger to exclude effect of camera walls and floor on the SC behavior.

Table 4-2

Frequency, *Hz*	Spectral component, *dB*
20 - 200	100
200 - 230	103
230 - 310	107
310 - 400	111
400 - 500	114
500 - 630	117
630 - 800	124
800 - 900	126
900 - 1000	129
1000 - 1050	128
1050 - 1250	125
1250 - 1600	122
1600 - 2000	118
Total:	138

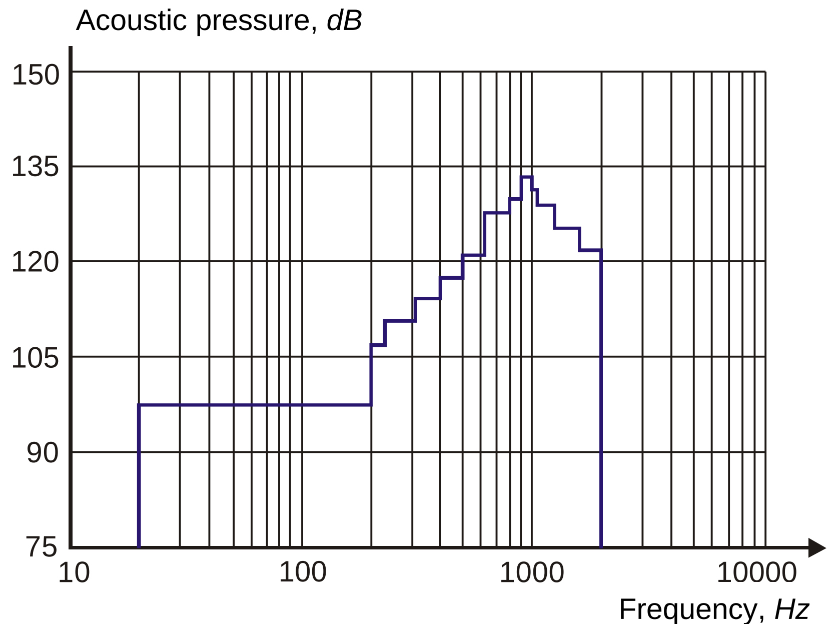


Fig. 4-2. Acoustic noise spectrum

4.3.2 Steady-state Vibrations

Steady-state vibrations are response of dynamic system consisting of a spacecraft and adapter to acoustic pressure effect. The sum value of the steady-state vibrations that are registered on the adapter, does not exceed 5*g* rmsv. Steady-state vibration frequency content depends on SC resonance characteristics and it can be verified after consultations with the spacecraft developer.

During autonomous tests of SC units for the steady-state vibrations, the following modes presented in Tables 4-3 and 4-4 are accepted. The steady-state vibrations are random process. In the Table 4-3 and Fig. 4-3, the values of power spectral density (PSD) are given. These data are used to determine a mode of SC qualification testing. Vibration level should be controlled at places where SC is attached to the LV adapter. SC acceptance tests are conducted using testing modes presented in Table 4-4 and Fig. 4-4. The vibrations specified in Table 4-4 are accepted the same for all three axes. Test time is equal to 90 seconds for qualification testing and 60 seconds for acceptance testing.

As the steady-state vibrations are induced by acoustic noise acting on SC, random vibrations testing is allowed to conduct only during autonomous tests for units and instruments of the spacecraft.

Table 4-3

Frequency, *Hz*	Power spectral density, g^2/Hz
20 - 200	0.00003 – 0.0003
200 - 250	0.0003 – 0.00045
250 - 315	0.00045 – 0.0011
315 - 400	0.0011 – 0.0018
400 - 500	0.0018 – 0.0036
500 - 630	0.0036 – 0.0065
630 - 800	0.0065 – 0.017
800 - 900	0.017 0.025
900 - 1000	0.025 – 0.03
1000 - 1050	0.03 – 0.028
1050 - 1250	0.028 – 0.025
1250 – 1600	0.025 – 0.005
1600 - 2000	0.005 – 0.0035
Total:	4.36*g* rmsv

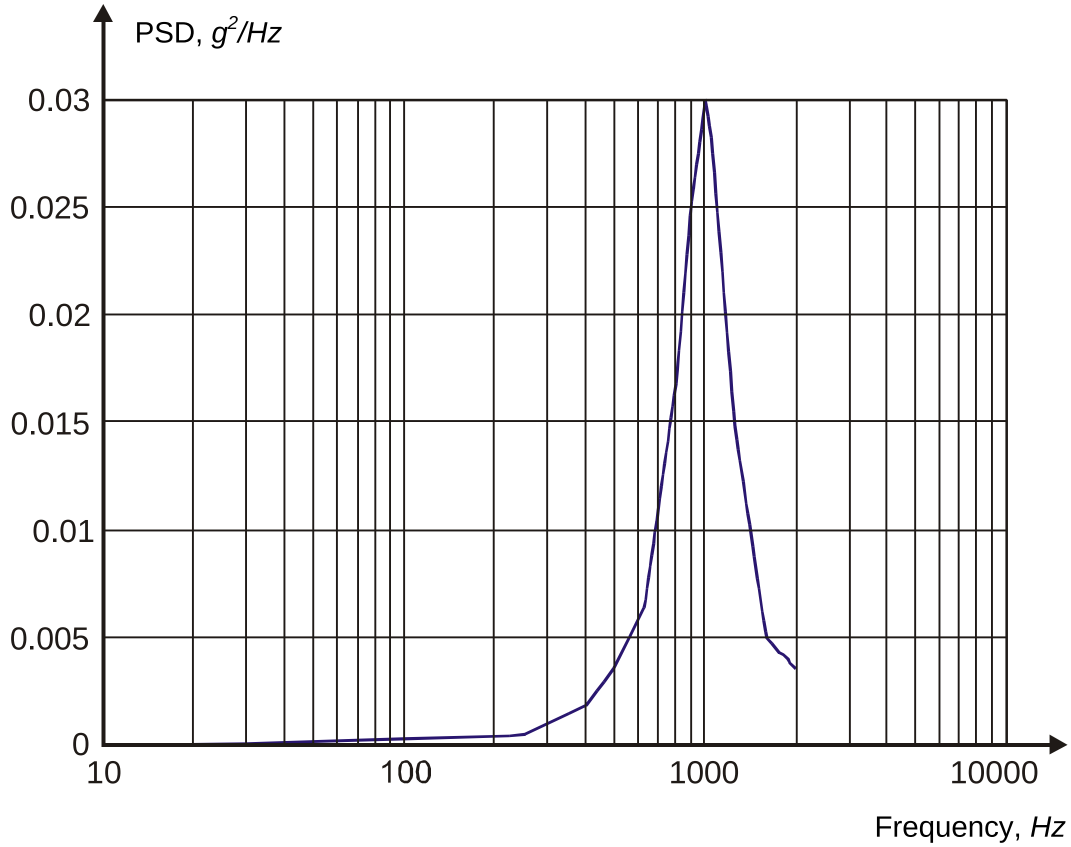


Fig. 4-3. Power spectral density for qualification testing

Table 4-4

Frequency, *Hz*	Power spectral density, g^2/Hz
20 - 25	0.001
25 – 31.5	0.0018
31.5 - 40	0.0023
40 - 50	0.003
50 - 63	0.004
63 - 80	0.0055
80 - 1000	0.0075
1000 - 1250	0.0058
1250 - 1600	0.0039
1600 - 2000	0.0026
Total	3.34 g rmv

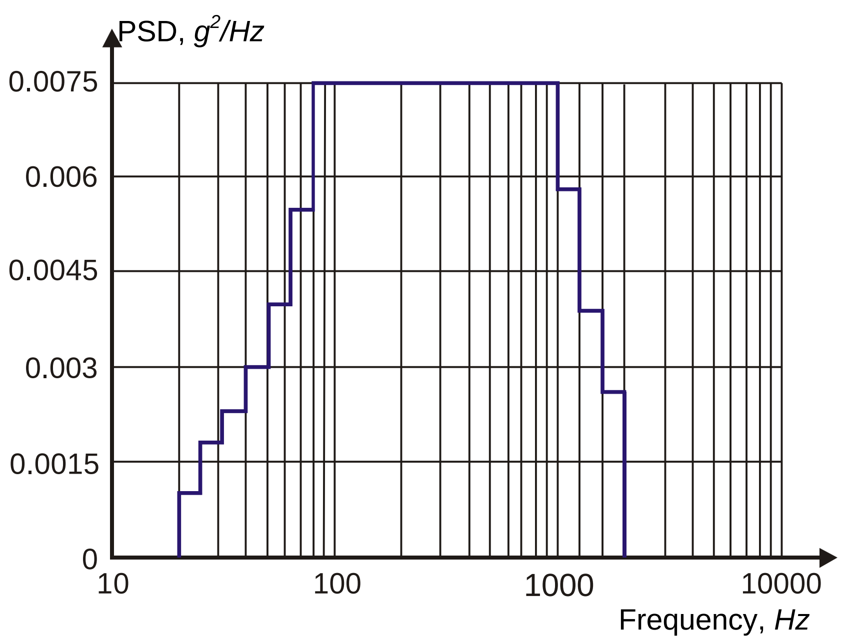


Fig. 4-4. Power spectral density for acceptance testing

4.3.3 Vibro-Shock Loads

Figures 4-5 through 4-9 present a time chart of low-frequency vibro-shock accelerations at the adapter ring along the LV axis when the LV leaves the TLC and Stage motors are fired. Values of maximum accelerations along two other axes do not exceed 30% from the given ones.

The curves for these accelerations are plotted from assumption that the SC low natural longitudinal frequency is 55 *Hz*. More accurate curves can be computed with consideration for SC finite-element model. Total number of low-frequency vibration shocks is 5.

Maximum high-frequency vibration shocks are induced by initiation of pyrotechnical devices used to separate the 3rd stage motor and to release the fairing. Parameters of high-frequency vibration-shocks are characterized by shock spectrum shown in Fig. 4.10. Maximum amplitude of high-frequency vibro-shocks reaches 200 *g*, and in this case the energy is within the frequency band of 600…1200 *Hz*. During tests, it is allowed to replace vibro-shock actions with triangle impulse with 200 *g* amplitude at duration of 0.5…1.0 *ms* for high-frequency shocks, and for low-frequency shocks these impulse parameters correspondingly are 2…10 *g* and 10…50 *ms*. (linear dependence of the amplitude on duration). The total number of vibro-shocks is the same as indicated above. In specific case, if natural frequencies of instruments do not exceed 300 *Hz*, during tests for high-frequency vibro-shocks it is allowed to increase the duration of impulse (as agreed upon with LV developer), which replaces high-frequency vibro-shock, from 1 *ms* to 5…6 *ms* with retaining the impulse energy.

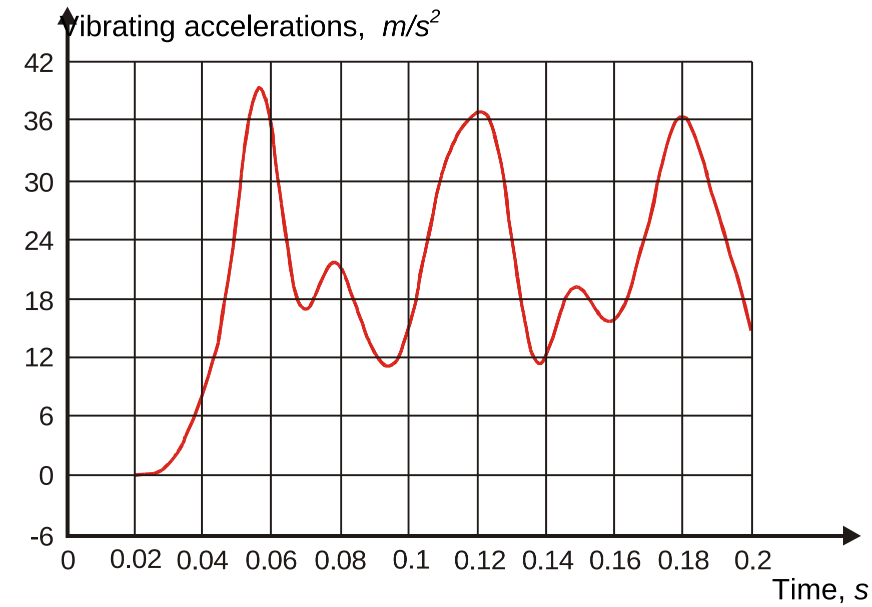


Fig. 4-5. Launch Vehicle injection from TLC

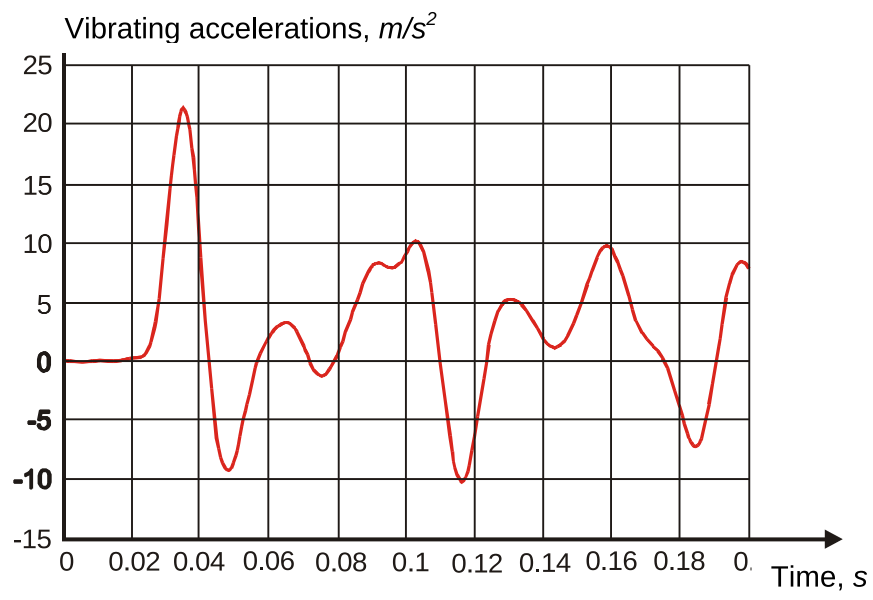


Fig. 4-6. 1st Stage motor firing

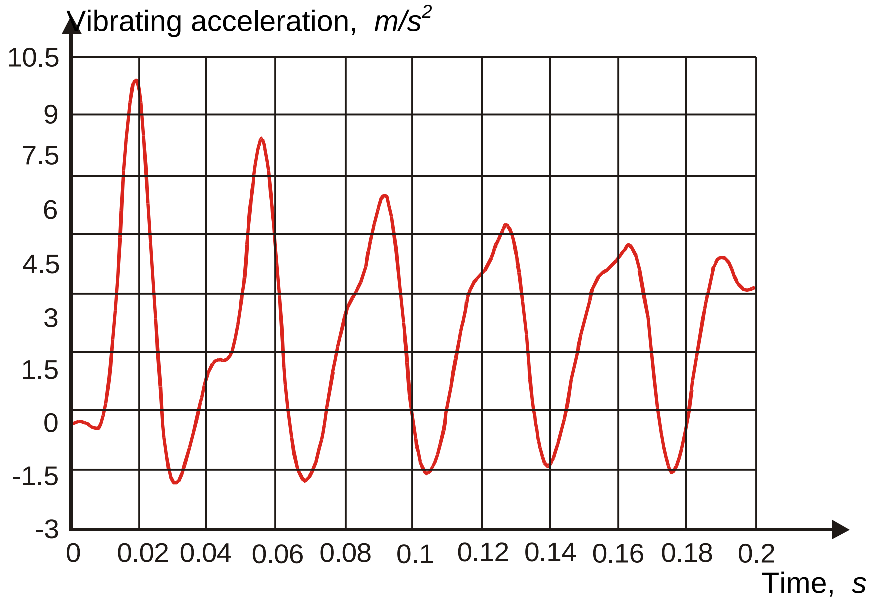


Fig. 4-7. 2nd Stage motor firing

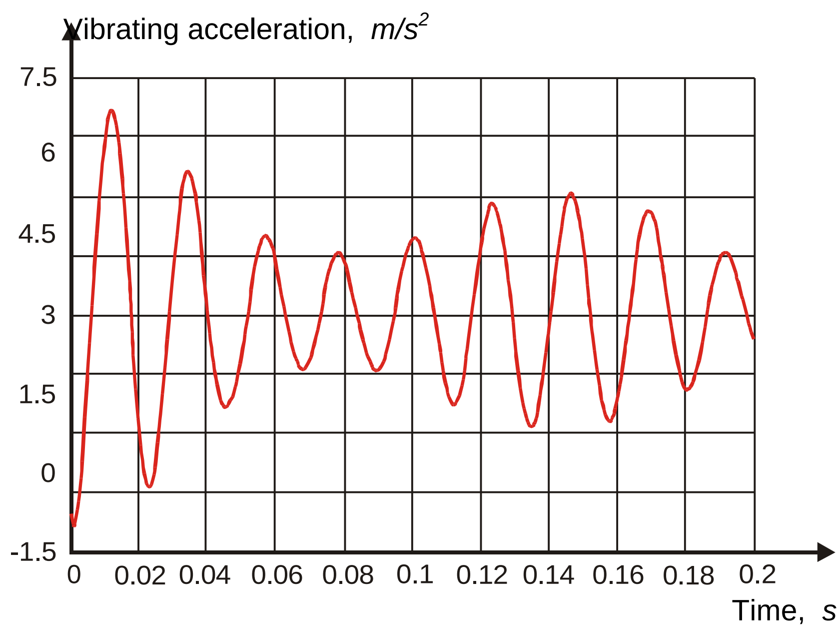


Fig. 4-8. 3rd Stage motor firing

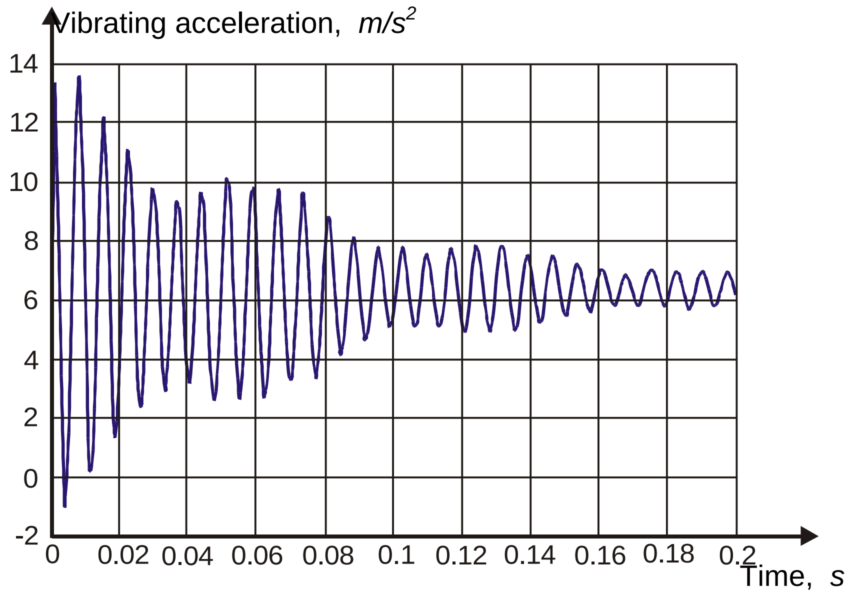


Fig. 4-9. 4th Stage motor firing.

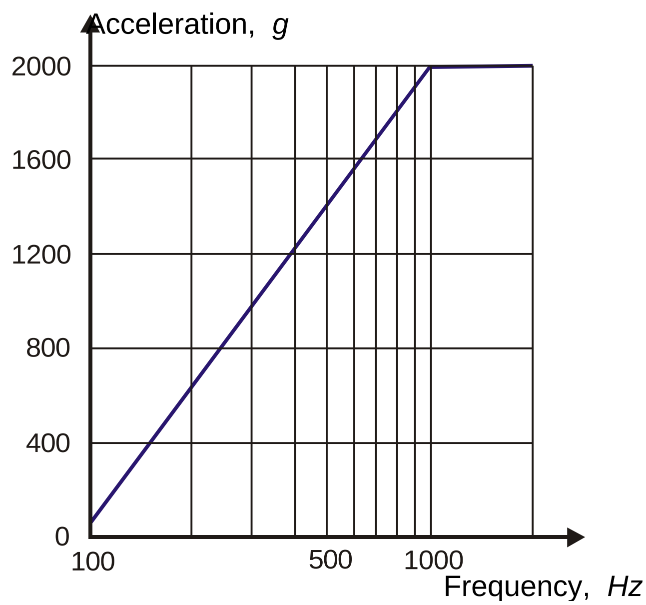


Fig. 4-10. Shock spectrum

4.4 Spacecraft Thermal Environments

4.4.1 General

The initial temperature and humidity conditions of a spacecraft are determined by its location area according to the SC working process flow diagram (refer to Section 6) and they are related to the following phases of pre-launch preparation and launch:

- SC transportation to the SC ATB
- SC preparation in the SC ATB
- Integration with launch vehicle in the LV ATB
- Transportation of SC integrated with LV to Launch Site
- Pre-launch preparation and launch at Launch Site
- SC injection phase

4.4.2 Optional Services

The following optional services can be provided as agreed upon with a Customer:

- Maintenance of environmental temperature inside TLC at head module area within the range that is allowed by operating conditions of LV systems to provide more favourable thermal environment for SC;
- Cold nitrogen purge to maintain a required thermal environment for SC systems with excessive heat when they are put in operating condition at SC ATB and launch site (for example, during SC battery trickle charging);
- Control of environment temperature inside the head module and inside the TLC in the HM location area during pre-launch preparation.

The nitrogen cooling system is used for supplying dry gas nitrogen with controlled parameters (temperature, flow rate).

4.4.3 Temperature inside Fairing at Injection Phase

Temperature change of fairing internal surface during flight up to fairing separation point is shown in Fig. 4-11.

Emissivity of the fairing internal surface is equal to 7.

Convective heat transfer does not virtually affect the SC thermal environment inside fairing as by the time of the beginning of fairing internal surface warming the pressure around SC drops up to values less that 300 *Pa*.

Approximate relation of the radiation heating flow effects on the SC as a function of LV flight time is presented in Fig. 4-12.

Start-1 LV fairing is jettisoned at speed no more than 5500 *m/s* at altitude of more than 150 *km*.

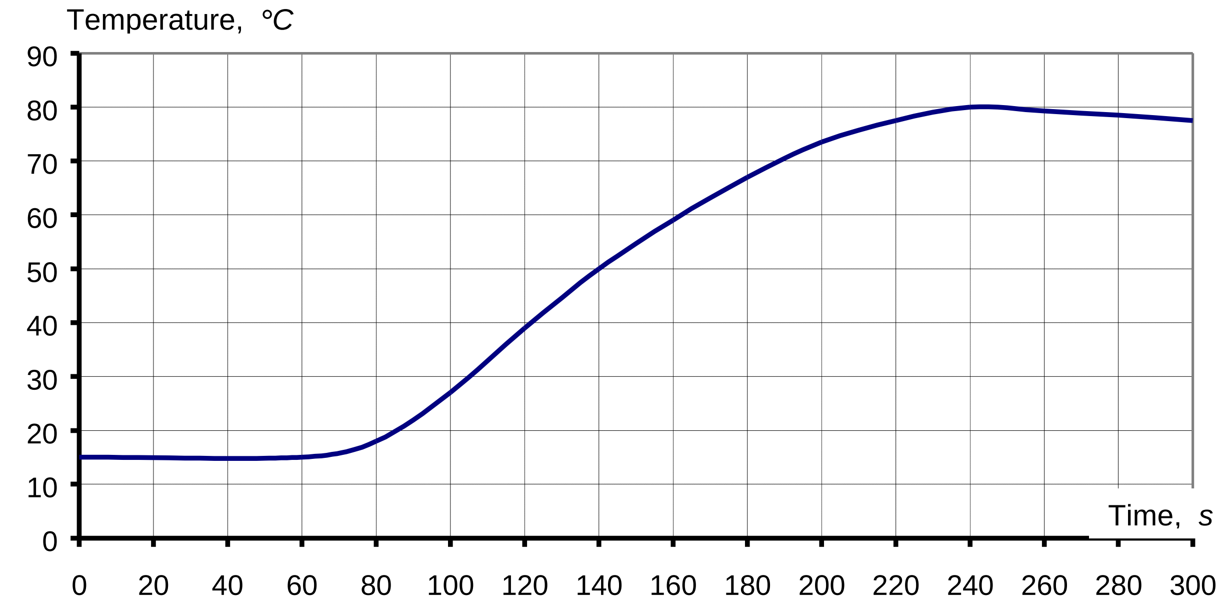


Fig. 4-11. The fairing internal surface temperature versus the LV flight time

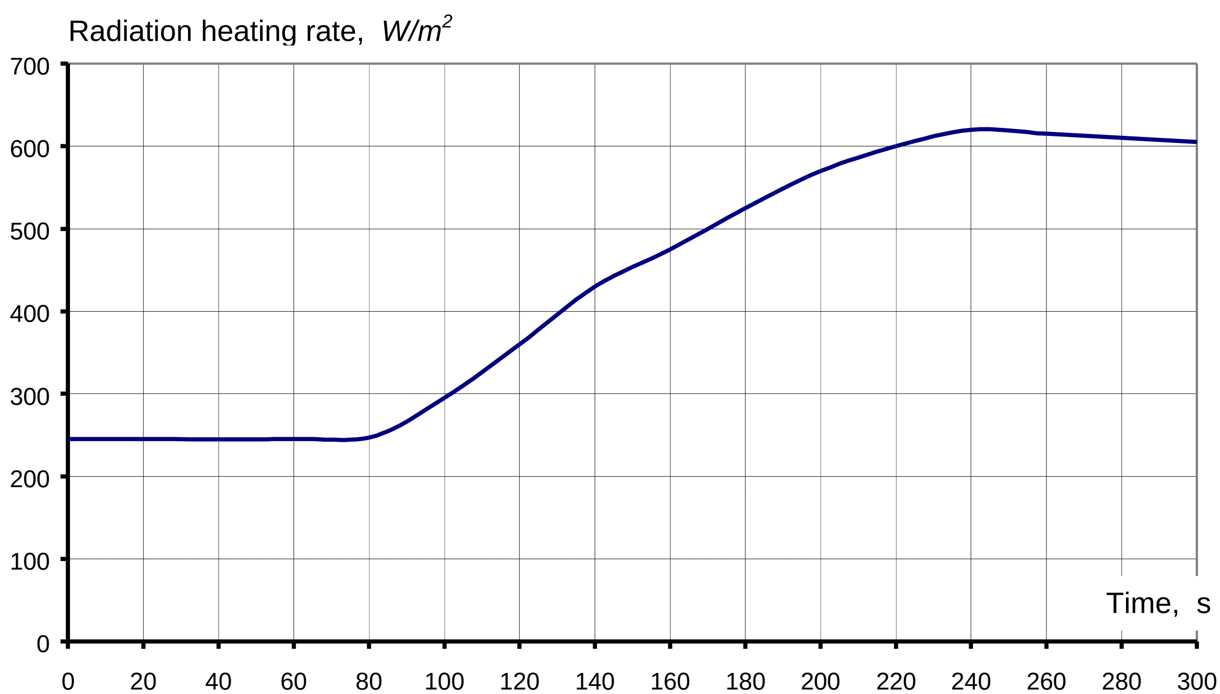


Fig. 4-12. The radiation heating flow from the fairing internal surface versus the flight time.

4.5 Pressure Change inside Fairing in Flight

In order to level the pressure inside head module with external pressure by the time of fairing jettison, two equalizing valves are mounted in the adapter bottom. These valves are initiated by pyrotechnical devices installed in the casing of the valves. The mentioned pyrotechnical devices are ignited by GCS command at the same time when the first sabot is jettisoned at the initial LV flight phase.

Pressure profile inside fairing in flight is related to mission program and available fairing volume. Pressure drop inside the fairing in flight is characterized by the following approximate data (refer to Fig. 4-13:

- Max. overpressure inside the fairing – 0.2 kgf/cm^2
- Gas flow rate at SC surface – no more than 0.5 *m/s*
- Pressure inside fairing by the time of fairing jettison – no more than 0.0004 kgf/cm^2

To prevent thermal impact on SC and atmosphere impact effect the fairing is jettisoned at altitude of no more than 150 *km*.

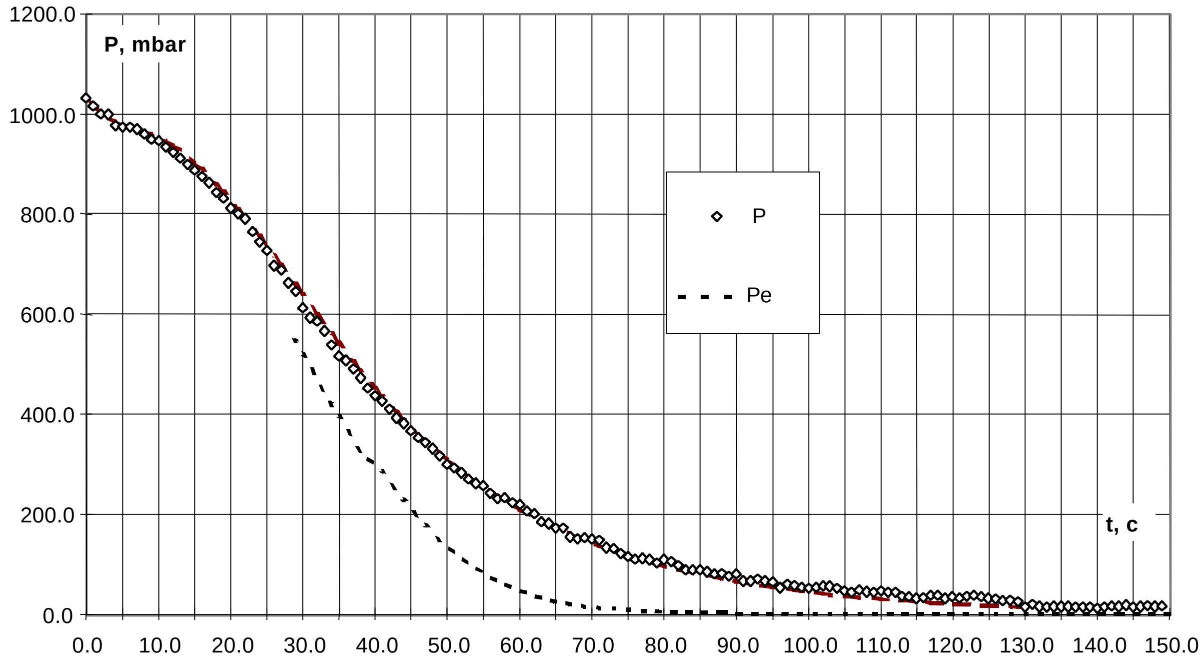


Fig. 4-13. Pressure profile inside the fairing in flight

Po – pressure inside the fairing (experimental data)
Ph – atmosphere pressure at flight altitude

4.6 Environment Cleanliness Requirements

4.6.1 Cleanliness Environment during Pre-launch Preparation

SC environment parameters during all ground operations at the Cosmodrome comply with cleanliness class 10000 to 100000 of the U.S. Federal Standard FED-STD-209F. These parameters are provided by air conditioning and filtration of clean rooms of the SC ATB.

At the Cosmodrome, the fulfillment of requirements for hardware surface cleanliness are supervised by Customer jointly with Provider using required means and materials.

4.6.2 Cleanliness Environment at Injection Phase

Up to activation of pyrotechnical valves the head module is a sealed. Before fairing jettison, after the pyrotechnical valves have been operated, the overpressure inside HM relative to atmosphere pressure at current flight altitude is retained (refer to Fig. 4-13) that prevents from ingress of contamination to the HM inside.

After the fairing jettison, the SC contamination by exhaust products of the 4th Stage motor and PBPS is excluded using the following operations:

- the 4th Stage motor and PBPS are operated in propulsive mode up to propellant burnout;
- in the flight between PBPS burnout and SC separation the SC Attitude and stabilization are performed by GRACS is operated by pressurized nitrogen;
- the SC separation is performed spring assemblies not earlier that 375 seconds after the 4th Stage motor burnout and not earlier 90 seconds after PBPS burnout;
- during SC separation when pyrotechnical devices have been operated the derbies and combustion products are remained inside of these devices;
- the separation assembly part remaining on SC contents only metal elements (stainless, carbon spring, and alloyed steels) that meet the outgassing requirements;
- after SC separation, the LV's longitudinal axis is turned out into plane perpendicular to the velocity vector and is rotated in this plane. This ensures perpendicularity of the motor nozzle to the SC motion and excludes contamination of SC by gas exhaust from LV motor.

4.7 Radio-frequency and Electromagnetic Environment at Pre-launch Preparation and Launch

4.7.1 Spacecraft Radio-frequency and Electromagnetic Environment during Pre-launch Preparation

During pre-launch preparation, the radio-frequency and electromagnetic fields induced by LV onboard radio systems, SLS ground equipment and Cosmodrome equipment affect a spacecraft.

LV onboard telemetry system transmitter.

LV onboard telemetry system operates in pulse-frequency modulation of carrier frequency using pulse-amplitude and pulse-code modulation.

Carrier frequency	203.3 *MHz*
Pulse duration	1.56 *ms*
Pulse-recurrence rate	320 *kHz*
Maximum emissive power	up to 40 *W*
Antenna and feeder device losses	1.5 *dB*
Attenuation in the second and subsequent harmonics	no less than 60 *dB*
Antenna gain	2.61 *dB*
Polarization	linear
Cutoff time	30 *s* after SC separation

Onboard telemetry system antenna location is shown in Fig. 4-14. Normalized pattern of onboard telemetry system antenna is given in Table 4-5. The angles in this pattern are measured in the LV coordinate system as shown in Fig. 4-15.

LV onboard transponder.
Launch vehicle can be equipped with transponder that is a part of trajectory measuring system and radiates pulse radio signals.

Carrier frequency	2860 *MHz*
Pulse duration	1 *ms*
Pulse-recurrence rate	up to 3.5 *kHz*
Maximum emissive power	up to 300 *W*
Antenna and feeder device losses	1.5 *dB*
Attenuation in the second and subsequent harmonics	no less than 60 *dB*
Antenna gain	5.1 *dB*
Polarization	linear
Cutoff time	at the time of SC separation

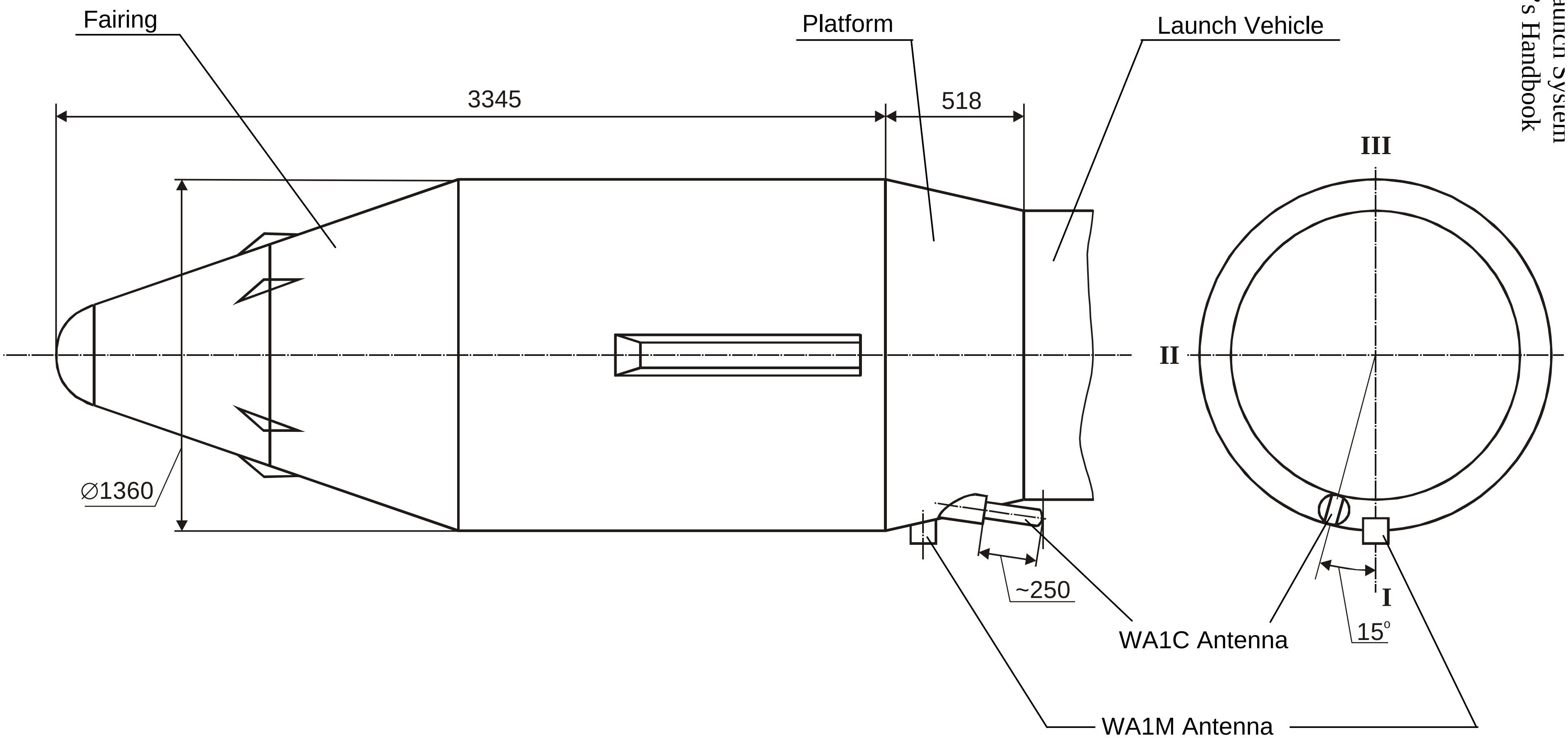


Fig.4-14. Antenna location on LV body

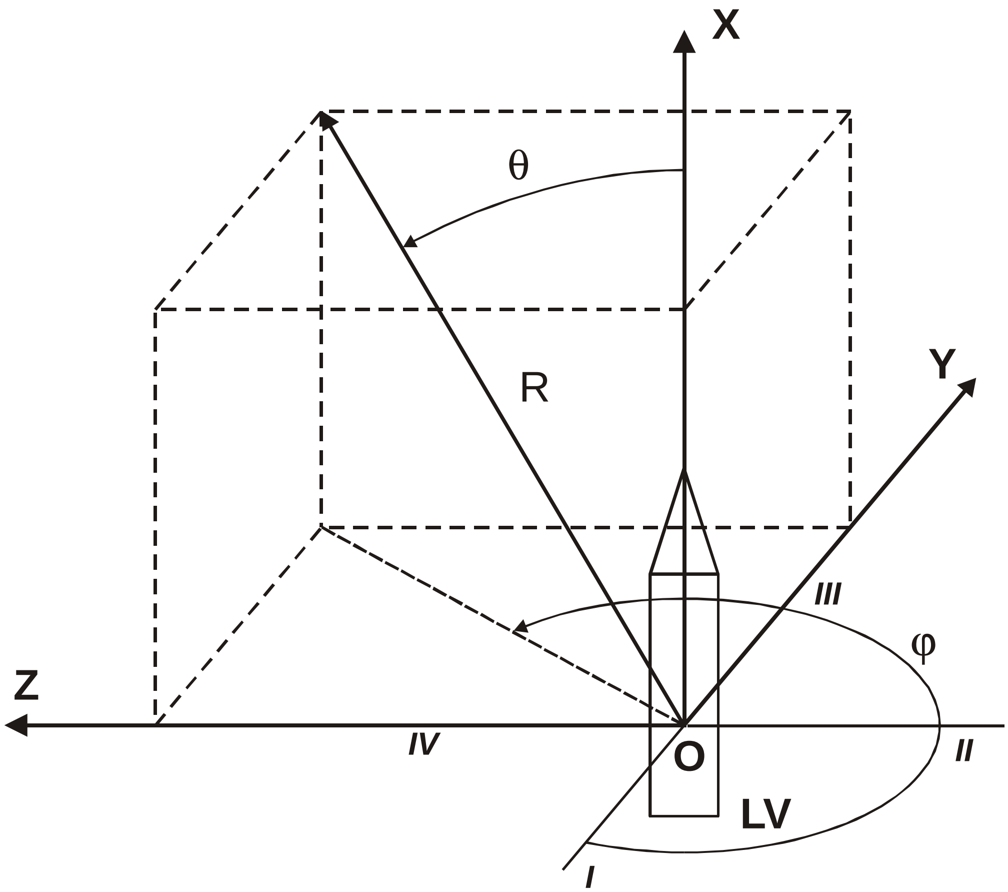


Fig. 4-15. Measuring of normalized antenna pattern angles

LV onboard transponder antenna Location is shown in Fig. 4-14. Normalized pattern of onboard transponder antenna is given in Table 4-6. The angles in this pattern are measured in the LV coordinate system as shown in Fig. 4-15.

Transmitters of ground radars.

Carrier frequency	2725 *MHz*
Pulse duration	1.56 *ms*
Code distance between pulses	3-6 *ms*
Pulse-recurrence rate	625 *Hz*
Maximum emissive power	5000 *kW*
Transmitting antenna gain	35 *dB*
Antenna and feeder device losses	1.0 *dB*

Transponder electronic ground support equipment.

Transponder electronic ground support equipment (TEGSE) is a radar simulator and it emits radio signals with parameters closed to radar ones. The transponder electronic ground support equipment is used only during LV preparation phase at the LV ATB when spacecraft is inside the HM and is covered by the fairing. This provides reduction of electromagnetic effect on SC from TEGSE no less than 30 *dB*.

Maximum emissive power	10 *W*
Transmitting antenna gain	5 *dB*

4.7.2 Radio-frequency and Electromagnetic Environment in Flight

At injection phase the spacecraft is subjected to radio-frequency and electromagnetic effects from the mentioned above LV radio systems: telemetry system and trajectory measuring system transponder.

Table 4-5
Normalized pattern of onboard telemetry system antenna

Angle φ, deg.	Angle θ, deg.																		
	0	10	20	30	40	50	60	70	80	90	100	110	120	130	140	150	160	170	180
0	.34	.84	.92	.86	.74	.65	.63	.84	.79	.77	.80	.69	.42	.39	.41	.60	.65	.48	.36
10	.34	.83	.92	.86	.73	.73	.63	.83	.88	.72	.72	.52	.33	.35	.41	.58	.70	.57	.41
20	.34	.85	.87	.83	.71	.63	.68	.88	.85	.73	.71	.53	.35	.40	.43	.60	.72	.59	.46
30	.34	.84	.92	.83	.66	.58	.62	.78	.74	.59	.57	.35	.28	.35	.42	.58	.69	.63	.50
40	.34	.77	.85	.78	.66	.59	.64	.77	.73	.58	.58	.36	.25	.30	.35	.50	.66	.62	.54
50	.34	.68	.69	.79	.80	.56	.58	.71	.75	.67	.70	.58	.36	.34	.31	.39	.51	.56	.52
60	.34	.57	.78	.74	.61	.59	.63	.72	.76	.71	.73	.69	.50	.36	.31	.37	.42	.44	.46
70	.34	.48	.67	.69	.54	.52	.60	.65	.71	.69	.77	.76	.57	.43	.36	.44	.66	.42	.41
80	.34	.47	.63	.63	.54	.56	.60	.63	.65	.67	.73	.75	.63	.46	.40	.45	.53	.48	.46
90	.34	.48	.59	.59	.52	.50	.56	.58	.58	.65	.67	.69	.63	.53	.49	.51	.54	.48	.50
100	.34	.40	.51	.54	.50	.47	.57	.60	.56	.59	.63	.64	.61	.54	.52	.53	.52	.52	.53
110	.34	.34	.45	.49	.47	.40	.51	.58	.53	.55	.62	.64	.61	.54	.53	.52	.52	.52	.60
120	.34	.26	.30	.35	.33	.27	.32	.42	.43	.46	.54	.63	.63	.61	.55	.51	.51	.56	.59
130	.34	.24	.26	.28	.28	.24	.19	.24	.24	.23	.39	.53	.59	.58	.60	.63	.60	.61	.61
140	.34	.27	.25	.25	.27	.25	.16	.17	.17	.16	.31	.61	.52	.54	.66	.69	.66	.66	.62
150	.34	.26	.20	.19	.24	.21	.14	.23	.27	.21	.25	.42	.53	.58	.77	.88	.77	.68	.58
160	.34	.37	.17	.13	.21	.22	.14	.32	.43	.39	.39	.51	.55	.67	.80	.88	.77	.65	.58
170	.34	.30	.12	.16	.21	.14	.16	.44	.49	.47	.48	.61	.70	.60	.74	.87	.75	.65	.62
180	.34	.40	.12	.09	.22	.21	.13	.39	.57	.46	.52	.63	.82	.67	.57	.70	.69	.60	.58
190	.34	.33	.05	.11	.21	.10	.14	.45	.40	.28	.39	.53	.73	.62	.52	.55	.54	.50	.54
200	.34	.40	.07	.06	.20	.19	.06	.30	.27	.11	.21	.30	.61	.59	.50	.56	.48	.46	.49
210	.34	.32	.05	.05	.23	.14	.08	.30	.21	.11	.20	.32	.63	.60	.56	.59	.48	.43	.46
220	.34	.46	.14	.03	.12	.14	.04	.28	.40	.30	.36	.42	.64	.59	.61	.70	.60	.43	.41
230	.34	.33	.09	.04	.10	.07	.06	.37	.47	.45	.45	.57	.71	.61	.66	.76	.59	.41	.38
240	.34	.33	.16	.20	.12	.06	.09	.44	.59	.52	.50	.62	.66	.58	.71	.87	.76	.48	.42
250	.34	.28	.20	.17	.17	.09	.07	.31	.51	.43	.39	.60	.67	.66	.83	.92	.85	.59	.48
260	.34	.30	.25	.24	.29	.16	.27	.25	.35	.23	.32	.61	.69	.75	.90	.96	.88	.70	.53
270	.34	.25	.26	.32	.33	.27	.21	.20	.21	.21	.41	.65	.73	.85	.94	.90	.82	.68	.55
280	.34	.24	.31	.39	.42	.30	.23	.25	.25	.44	.71	.77	.83	.92	.84	.73	.66	.55	.45
290	.34	.26	.38	.49	.52	.47	.47	.44	.54	.81	.96	.92	.85	.81	.66	.49	.48	.47	.45
300	.34	.29	.41	.60	.61	.61	.49	.50	.62	.95	1.0	.90	.90	.88	.64	.48	.47	.52	.47
310	.34	.33	.46	.58	.64	.62	.54	.61	.77	.94	.90	.82	.82	.65	.34	.43	.32	.40	.43
320	.34	.46	.60	.71	.75	.67	.58	.62	.75	.93	.63	.56	.58	.39	.34	.31	.41	.34	.37
330	.34	.49	.64	.69	.71	.65	.60	.64	.71	.63	.50	.43	.37	.29	.35	.40	.46	.27	.31
340	.34	.60	.70	.72	.66	.56	.49	.59	.70	.56	.49	.50	.38	.27	.41	.34	.39	.31	.30
350	.34	.79	.88	.87	.73	.60	.55	.76	.77	.57	.59	.51	.60	.32	.27	.35	.39	.28	.32
360	.34	.84	.92	.86	.74	.65	.63	.84	.79	.77	.80	.69	.42	.39	.41	.60	65	.48	.36

Table 4-6
Normalized pattern of onboard transponder antenna

Angle φ, deg.	Angle θ, deg.																		
	0	10	20	30	40	50	60	70	80	90	100	110	120	130	140	150	160	170	180
0	.14	.08	.22	.61	.60	.14	.03	.34	.11	.12	.30	.27	.20	.41	.97	.97	.08	.26	.14
10	.14	.16	.36	.58	.49	.21	.25	.32	.14	.20	.22	.29	.13	.35	.89	.63	.13	.18	.13
20	.14	.17	.38	.40	.45	.17	.11	.24	.14	.22	.45	.35	.37	.95	1.0	.46	.11	.14	.13
30	.14	.12	.25	.26	.35	.33	.27	.12	.12	.15	.15	.22	.65	.71	.49	.14	.14	.16	.09
40	.14	.11	.21	.25	.28	.39	.38	.35	.29	.17	.31	.53	.45	.35	.15	.08	.21	.10	.07
50	.14	.12	.18	.21	.25	.34	.36	.61	.59	.39	.47	.61	.50	.38	.14	.13	.18	.15	.07
60	.14	.17	.19	.25	.24	.37	.30	.44	.51	.45	.49	.61	.54	.27	.09	.13	.21	.04	.06
70	.14	.17	.20	.06	.26	.17	.11	.17	.13	.26	.35	.46	.46	.14	.13	.16	.21	.09	.07
80	.14	.18	.17	.12	.15	.18	.21	.13	.10	.01	.04	.16	.09	.07	.14	.11	.15	.11	.07
90	.14	.16	.19	.14	.18	.21	.34	.33	.33	.15	.05	.05	.07	.06	.13	.11	.11	.13	.10
100	.14	.16	.16	.12	.16	.14	.26	.35	.37	.26	.14	.06	.08	.08	.07	.05	.08	.13	.10
110	.14	.16	.16	.11	.12	.11	.17	.30	.27	.16	.09	.10	.09	.11	.06	.10	.07	.13	.12
120	.14	.16	.15	.09	.21	.10	.11	.22	.12	.11	.06	.11	.11	.05	.04	.04	.04	.12	.10
130	.14	.00	.00	.00	.00	.00	.00	.00	.00	.00	.00	.00	.00	.00	.00	.00	.00	.00	.00
140	.14	.00	.00	.00	.00	.00	.00	.00	.00	.00	.00	.00	.00	.00	.00	.00	.00	.00	.00
150	.14	.00	.00	.00	.00	.00	.00	.00	.00	.00	.00	.00	.00	.00	.00	.00	.00	.00	.00
160	.14	.00	.00	.00	.00	.00	.00	.00	.00	.00	.00	.00	.00	.00	.00	.00	.00	.00	.00
170	.14	.00	.00	.00	.00	.00	.00	.00	.00	.00	.00	.00	.00	.00	.00	.00	.00	.00	.00
180	.14	.00	.00	.00	.00	.00	.00	.00	.00	.00	.00	.00	.00	.00	.00	.00	.00	.00	.00
190	.14	.00	.00	.00	.00	.00	.00	.00	.00	.00	.00	.00	.00	.00	.00	.00	.00	.00	.00
200	.14	.00	.00	.00	.00	.00	.00	.00	.00	.00	.00	.00	.00	.00	.00	.00	.00	.00	.00
210	.14	.00	.00	.00	.00	.00	.00	.00	.00	.00	.00	.00	.00	.00	.00	.00	.00	.00	.00
220	.14	.00	.00	.00	.00	.00	.00	.00	.00	.00	.00	.00	.00	.00	.00	.00	.00	.00	.01
230	.14	.00	.00	.00	.00	.00	.00	.00	.00	.00	.00	.00	.00	.00	.00	.00	.04	.08	.09
240	.14	.06	.19	.05	.07	.12	.03	.03	.04	.22	.40	.37	.27	.22	.28	.04	.09	.04	.08
250	.14	.16	.10	.10	.04	.07	.07	.05	.04	.13	.65	.22	.22	.08	.12	.05	.04	.03	.04
260	.14	.13	.14	.09	.11	.12	.21	.18	.32	.22	.86	.66	.35	.23	.12	.25	.11	.06	.17
270	.14	.15	.18	.11	.15	.33	.22	.14	.12	.11	.07	.07	.10	.19	.03	.17	.16	.04	.10
280	.14	.13	.13	.10	.33	.29	.14	.03	.05	.19	.08	.09	.08	.12	.05	.07	.24	.06	.04
290	.14	.15	.15	.27	.39	.33	.09	.12	.33	.99	.97	.65	.25	.25	.06	.13	.38	.44	.16
300	.14	.25	.14	.23	.33	.32	.26	.25	.56	.99	.99	1.0	.81	.38	.19	.04	.13	.11	.13
310	.14	.20	.09	.15	.33	.52	.54	.41	.32	.25	.37	.64	.69	.97	.65	.08	.33	.10	.11
310	.14	.09	.19	.27	.27	.30	.19	.14	.05	.15	.15	.16	.25	.97	.97	.56	.10	.25	.14
330	.14	.09	.17	.26	.25	.22	.10	.16	.16	.21	.35	.11	.13	.63	.70	.41	.06	.22	.13
340	.14	.07	.15	.26	.24	.18	.07	.11	.21	.03	.15	.09	.05	.32	.67	.53	.08	.19	.10
350	.14	.06	.23	.44	.45	.13	.09	.25	.06	.15	.18	.28	.07	.36	.83	.91	.07	.22	.10
360	.14	.08	.22	.61	.60	.14	.03	.34	.11	.12	.30	.27	.20	.41	.97	.97	.08	.26	.14

5 PAYLOAD LIMITATIONS

5.1 Center of Mass Limitations

After separation the spacecraft angular velocities do not exceed (with confidence of 0.993) ±1.3 deg./s in pitch and yaw, and 1.0 deg./s in roll (refer to Section 2.5) provided that maximum deviations of the SC center of mass in lateral plane (static imbalance) is within limit ±4 mm.

If the static imbalance increases, the angular velocities increase after SC separation. Particular values of the angular velocities to be specified according to presented SC mass and inertial properties and SC center of mass position.

5.2 LV Angular Position Limitations at SC Separation Point

Roll maneuvering at the SC separation phase is not provided for LV standard configuration. At the SC separation time the LV's longitudinal axis makes an angle ***TBD*** (by a SC developer) with the velocity vector. Also a SC developer should specify the relative orientation of SC axes and LV axes after SC integration with LV taking into account loads acting during LV erection before launch (refer to Chapter 3, Mechanical Interface).

5.3 Spacecraft Structure Rigidity

To avoid dynamic interaction between LV and SC in low-frequency range the SC should meet the following rigidity requirements:

- at the points where SC is rigidly attached to the LV, the lower frequency in lateral plane should be more than 15 *Hz*.
- at the points where SC is rigidly attached to the LV, the lower frequency in longitudinal direction along the LV axis should be more than 50 *Hz*.

5.4 Electromagnetic Compatibility Limitations

Launch vehicle, systems and units of the Start-1 SLS and a spacecraft should meet the requirements to electromagnetic environment induced by onboard and ground radio systems. Effects of radio systems used for LV and SLS are given in Section 4.7.

A spacecraft developer should issue to LV developer the initial data on all radio systems used during launch campaign, including:

- onboard radio systems;
- electronic ground support equipment;
- communications (satellite communication system, radio intercommunication system, wireless telephones etc.);

Initial data should include:

- operating frequency range;
- antenna pattern;
- transmitter power
- operating mode (direct radio measurement, operation in cable network);
- radiation power attenuation coefficient for devices operating without direct radio radiation.

These initial data should be presented for 1 year prior to the beginning of launch campaign at Cosmodrome

On the basis of initial data issued by SC developer, LV developer should conduct analysis for electromagnetic compatibility.

5.5 Safety Limitations

SC developer shall issue initial data on hazardous operations and procedures, hazardous materials used during launch campaign, and requirements for hazardous work organization.

LV developer shall conduct jointly with Cosmodrome the analysis of the presented initial data that is used as a base to issue instructions on safety measures during pre-launch and launch operations for each launch campaign.

6 DOCUMENTATION

Documents to be issued during preparation and launch campaign are presented in Table 6-1.

Table 6-1
List of documents to be issued

Document	Responsible Party
1. Start-1 Launch Vehicle and Spacecraft User's Handbook (adapted User's Handbook)	Provider
2. Statement of Work (schedule of SC pre-flight operations using Start-1 LV)	Customer
3. Spacecraft/Start-1 LV Interface Control Document (ICD)	Provider and Customer
4. Mission Analysis (calculations associated with a spacecraft launch using the Start-1 LV; two versions – preliminary and final releases)	Contractor
5. Spacecraft and Start-1 LV Mechanical Interface Drawings	Provider
6. Diagrams of electrical and telemetry interfaces for SC, Start-1 LV, and Cosmodrome	Provider and Customer
7. Spacecraft and Launch Vehicle Integration Process Flow Diagrams	Provider and Customer
8. Schedule of Spacecraft Hardware Delivery	Customer
9. Work Schedule of Spacecraft Preparation and Launch using Start-1 LV	Provider and Customer
10. Security and Safety Plan for Pre-flight Operations and SC launch from the Svobodny Cosmodrome aboard the Start-1 LV	Provider
11. Information on SC state vector at the separation time	Provider
12. Final Report on Spacecraft Launch Results	Provider

7 MISSION ANALYSIS

7.1 General

Mission analysis is prepared to estimate the possibility to perform launch program and to determine expected trajectory parameters.

Mission analysis is conducted to make sure that the program purposes are achieved and spacecraft can be injected into prescribed orbit with designed parameters of angular motion.

Program analysis is conducted in two phases:

- Preliminary analysis in order to define the spacecraft compatibility with the Start-1 LV;
- Final analysis in order to develop a mission program and to determine the expected trajectory parameters.

Mission analysis includes the following Sections:

- Coupled Dynamic Analysis
- Flight Sequence
- Trajectory Parameters and Injection Accuracy
- Fairing Separation Dynamics
- Spacecraft Separation Dynamics
- SC and LV relative motion in the first orbit after separation

7.2 Requirements for Spacecraft Mathematical Model for Coupled Dynamic Analysis

The SC mathematical model for calculations using the finite-element method should be presented by SC developer as physical (if degree of freedom does not exceed 10000) or condensed model. The initial data should includes:

– Mass, Stiffness and Damping matrices,
– requirements to damping;
– information on frequency range where the SC model is representative.

The model must describe the full three-dimensional behavior of the spacecraft, and values of the initial data shall be expressed according to International metric System. In case of the condensed method, the inverse transformation matrices shall be presented for acceleration, displacement and loads. The calculations are performed using NASTRAN software package. Kind of model and format of results shall be previously agreed upon with LV developer.