Phase C

Launch Environment

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RECORD OF REVISIONS

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1/0	17/07/2006	Initial Issue	Guillaume Roethlisberger
1/1	24/07/2006	Environmental Requirements Chapter	Guillaume Roethlisberger
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TERMS, DEFINITIONS AND ABBREVIATED TERMS

Definitions

Environment	Natural conditions and induced conditions that constrain the design definitions or operations for end products and their enabling products.
Launcher	Vehicle design to carry payloads into space NOTE The term "launch vehicle" is synonymous.

Abbreviated terms

ASAP 5	Ariane 5 Structure for Auxiliary Payload
C.o.M.	Center of Mass
FOS	Factor of Safety
GSC	Guiana Space Center
LV	Launch Vehicle
MOS	Margin of Safety
P-POD	Poly Picosatellite Orbital Deployer
PSLV	Polar Satellite Launch Vehicle
QSL	Quasi-Static Loads
SC	Spacecraft
SRS	Shock Response Spectrum
TBC	To be confirmed
TBD	To be defined
TLC	Transport and Launch Canister
TML	Total Mass Loss

1 INTRODUCTION

During the preparation for launch and then during the flight, the spacecraft is exposed to a variety of mechanical, thermal, and electromagnetic environments. This document provides a description of the environment that the spacecraft is intended to withstand.

In order to develop the design of the SwissCube, the launch loads as well as possible vibration frequency ranges were established based on the information given for the types of potential launch vehicles that are Vega, Ariane 5, ASAP 5, Soyuz, DNEPR and PSLV [1], [2], [3], [4], [5], [6], [7]. The NASA General Environmental Verification Specification (GEVS) document is also used [8] and defines a "worst-case" vibration profile which encompasses all major launch vehicles.

2 LAUNCH SYSTEMS

The launch process can severely constrain spacecraft design. Primary restrictions are the launch vehicle's lift capability and the environment to which it subjects the satellite during ascent. A launch system consists of a basic launch vehicle incorporating one or more stages and an infrastructure for ground support. It alters velocity to place the spacecraft in orbit, creates a severe ascent environment, and protects the spacecraft from its surrounding. Ultimately, it places the payload into the desired orbit with a functional spacecraft attitude.

2.1.1 Basic launch vehicle considerations

Space launch systems are unique form of transportation since they are the only systems that accelerate continuously throughout their performance envelope. Consequently, velocity is the fundamental measure of performance for launch systems. A launch system's ability to achieve orbital velocity comes primarily from its propulsion efficiency, with vehicle weight and drag acting against it.

2.2 Launch system selection process

The first step in the launch system selection process is to establish the mission needs and objectives, since they dictate the performance, trajectory, and the family of vehicles which can operate from suitable sites. A clear understanding of the real mission need is extremely important since it can dictate the launch strategy.

Another critical issue is whether the spacecraft will use a dedicated or shared launch system. The dedicated system may cost more, but it decreases the chance that a problem with another spacecraft will adversely affect the launch. On the other hand, shared launches are usually less expensive per spacecraft. Before deciding on a shared launch, we must consider the interaction between payloads. If we mount them serially, for example, we must analyze the probability that the upper payload will not deploy and thus interfere with the lower payload's deployment.

Once we establish the mission need, then we determine specific mission requirements. For low-Earth orbit missions, these usually consist of orbit attitude, inclination, and right ascension of the ascending node. In addition, estimated payload weight and dimensions become requirements to the launch system. A required launch date also may become a selection parameter as it affects schedules, and the vehicle and launch site availabilities.

Having established mission requirements and constraints, we must decide which launch-system configurations can deliver the spacecraft to its mission orbit. The launch systems selected during conceptual design should satisfy the mission's performance requirements and minimize program risk. We want to choose the launch systems early, so contractors for the spacecraft and launch systems can negotiate requirements early, as well. Doing so, decreases changes in design, cost and schedule downstream. Recent experiences show that we should design spacecraft to be compatible with several launch systems to increase launch probability, as well as to provide some leverage in negotiating launch cost.

Selecting a launch system depends on at least these criteria: the launch vehicle's performance capability to boost the necessary weight to the mission orbit, the required launch date versus vehicle availability, spacecraft-to-launch-vehicle compatibility, and of course, cost of the launch service. The launch system's performance capability must include factors such as performance margin, and a clear definition of weight and performance parameters like spacecraft dry weight, injected weight, boosted weight.

2.3 Determining the spacecraft design envelope and environments

Once we have identified several launch systems for our mission, we must determine the configuration of the interface between the launch system and the payload and understand the environment that the payload must withstand. We must consider this step early in mission design. The payload design must deal with launch environments and interfaces, whereas the launch system must support the payload during ascent.

In the spacecraft design, we must consider the payload environment for the time the payload leaves the vendor's facility until the spacecraft completes its mission. We need to pay attention to the predicted payload environments so we can protect the payload during ground transportation, aircraft take-off and landing, hoisting operation, launch and ascent.

Several static and dynamic loads affect the structure of the payload, adapters, and launch vehicle. These loads are either aerodynamic or they depend on acceleration and vibration. Aerodynamic loads are a function of the total pressure placed on the vehicle moving through the atmosphere. They consist of a static (ambient) pressure and a dynamic pressure. The relationship between altitude and velocity on the ascent trajectory determines these pressures. Payload fairings protect against dynamic pressure up to stated limits.

The acceleration loads, usually called load factors, experienced by the payload consist of static (steady state) and dynamic (vibration) loads. Note that we must consider axial and lateral values. The vibrational environments consist of launch-vehicle acceleration, aerodynamic drag and shear, acoustic pressures from the engines, and the mechanical response of the entire vehicle to these stimuli. We must design the payload and booster adapter to carry these loads.

To separate the launch vehicle from the spacecraft, or to deploy spacecraft components, we typically use pyrotechnic devices. Unfortunately, when activate they generate a shock load that transmits through the structure to the payload.

The acoustic environment is a function of the physical configuration of the launch vehicle, its acceleration time history, and the configuration of the propulsion system. The maximum dynamic environment is a consequence of the rate of acceleration and the aerodynamic smoothness of the launch vehicle shape. Solid-rocket boosters and first stage usually provoke a more severe environment, and the smaller the launch vehicle, the more stressed the payload. In general, the more rapidly the vehicle accelerates, the more severe the environment at a dynamic pressure. When the payload design is sensitive to the acoustic environment, it is common to add damping insulation.

Injection accuracy is also important to launch systems. Traditionally, the launch system and payload separate at a prescribed location and velocity. Launch-system vendors usually state an orbit-injection accuracy that depends primarily on the last stage of the vehicle.

3 ENVIRONMENTAL CONDITIONS

All environmental data given in the following paragraphs should be considered as limit loads, applying to the spacecraft. The related probability of these figures not being exceeded is 99 %.

Without special notice all environmental data are defined at the spacecraft base, i.e. at the adapter/spacecraft interface.

3.1 Frequency requirements

To prevent dynamic coupling with fundamental modes of the LV, the spacecraft should be designed with a structural stiffness which ensures that the following requirements are fulfilled, see Table 1. In that case the design limit load factors given in next paragraph are applicable.

The cantilevered fundamental mode frequencies of a spacecraft hard-mounted at the interface with an off-the shelf adapter must be:

Table 1 Stiffness criteria for the various LVs.

	In lateral axis	In longitudinal axis
Vega[1]	≥ 15 Hz	20 Hz ≤ F ≤ 45 Hz
Soyuz[2]	≥ 15 Hz	≥ 35 Hz
Dnepr	≥ 10 Hz	≥ 20 Hz
Ariane 5 (ASAP 5)	≥ 90 Hz	≥ 45 Hz
PSLV	≥ 90 Hz	≥ 45 Hz

[1] for spacecraft mass ≤ 2500 kg

[2] for spacecraft mass ≤ 5000 kg

3.2 Steady state acceleration

During flight, the spacecraft is subjected to static and dynamic loads. Such excitations may be of aerodynamic origin (e.g., wind, gusts, or buffeting at transonic velocity) or due to the propulsion systems (e.g., longitudinal acceleration, thrust build-up or tail-off transients, or structure-propulsion coupling, etc.). The highest longitudinal and lateral static accelerations for various launch vehicles are given in Table 2.

Table 2 Static accelerations induced on structure during launch.

Launch vehicle	Longitudinal acceleration (g)	Lateral acceleration (g)
Vega	5.5	0.9
Dnepr	7.8	0.8
Soyuz	4.3	0.4
Ariane 5 (ASAP5)	4.55	0.25
PSLV	7	1.5

3.3 The design load factors

The design load factors are represented by the quasi-static g-loads that are the more severe combinations of dynamic and steady-state accelerations that can be encountered at any instant of the mission (ground and flight operations).

The QSL reflects the line load (sometimes named mechanical fluxes, Φ) at the interface between the spacecraft and the adapter.

The flight limit of the QSL for a spacecraft launched on various LVs and complying with the previously described frequency requirements are given in Table 3.

Table 3 Quasi-Static Loads.

Launch vehicle	QSL [g] (+ = tension; - = compression)	
	Longitudinal acceleration (static + dynamic)	Lateral acceleration (static + dynamic)
Vega	-5.0 +3.0	± 0,9
Dnepr	-8.3 +1.0	± 0,8
Soyuz	-5.0 +1.5	± 1.8
Ariane 5 (ASAP5)	-7.55 +5.5	± 6
PSLV	-7.0 +2.5	± 2.5

3.4 Safety factors

Spacecraft qualification and acceptance test levels are determined by increasing the design load factors (the flight limit levels) — which have been previously presented — by the safety factors given in Table 4for the various LVs. The spacecraft must have positive margins of safety for yield and ultimate loads. The factors of safety (FOS) is a coefficient by which the design loads are multiplied in order to account for uncertainties in the statistical distribution of loads, uncertainties in structural analysis, manufacturing process, material properties and failure criteria.

Table 4 Test Factors, rate and duration.

SC Tests	Qualification		Protoflight		Acceptance	
	Factors	Duration/Rate	Factors	Duration/Rate	Factors	Duration/Rate
Static (SQL)	1.3 ultimate 1.1 yield	N/A	1.3 ultimate 1.1 yield	N/A	N/A	N/A
Sine vibrations	1.25	2 oct/mn	1.25	4 oct/mn	1.0	4 oct/mn
Random vibration	2.25	2 mn	2.25	1 mn	1.0	1 mn
Acoustics	1.41 (or +3dB)	2 mn	1.41 (or +3dB)	1 mn	1.0	1 mn
Shock	1.41 (or +3dB)	3 releases	1.41 (or +3dB)	1 release	N/A	N/A

3.5 Sine-equivalent dynamics

Sinusoidal excitations derived from motors pressure oscillations and controlled POGO effect [1]may affect the LV during its flight (mainly the atmospheric flight), as well as during some of the transient phases of the flight.

The envelope of the sinusoidal (or sine-equivalent) vibration levels does not exceed the values given in Figure 1 for the longitudinal case, and Figure 2 for the lateral case.

[1] The POGO vibrational instability effect is due to a coupling between the principal structure of a launcher, the so-called secondary structure, and the thrust. Thrust fluctuations generate longitudinal vibrations which in turn bring about pressure vibrations, leading again to thrust variations.

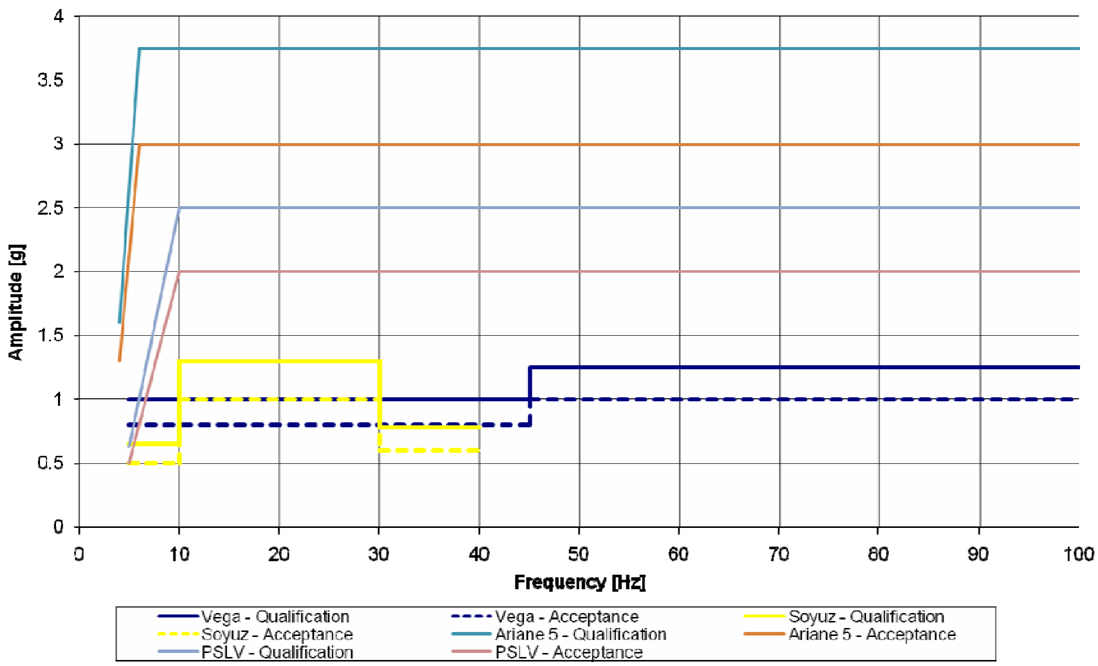

Figure 1 Amplitude of the sine excitations for the longitudinal direction.

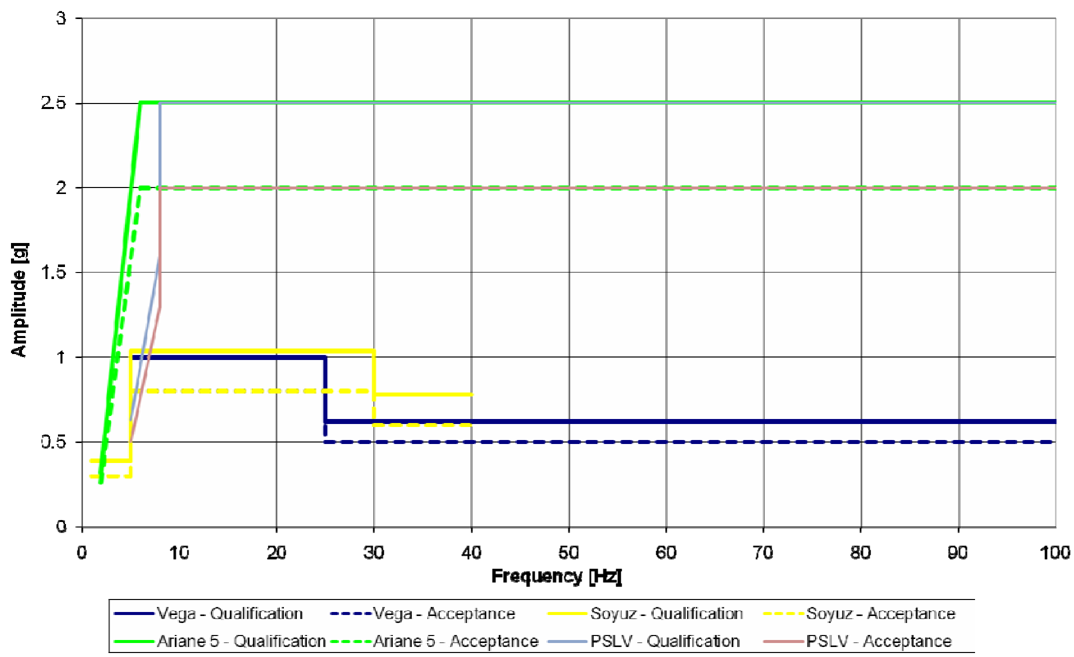

Figure 2 Amplitude of the sine excitations for the lateral direction.

3.5.1 Test levels and duration

According to the both previous figures, the sinusoidal vibration tests should have the levels described in Figure 3 and Table 5. Sinusoidal excitations shall be applied at the base of the mounting adapter, and shall be swept up through at a sweep rate of 2 octaves/min and 4 octaves/min for qualification and acceptance respectively.

Table 5 Sinusoidal vibration tests at qualification and acceptance levels.

Frequency [Hz]	2	4	6	6	100
Qualification [g]	0.32	1.6	3.75	3.75	3.75
Acceptance [g]	0.26	1.3	3	3	3

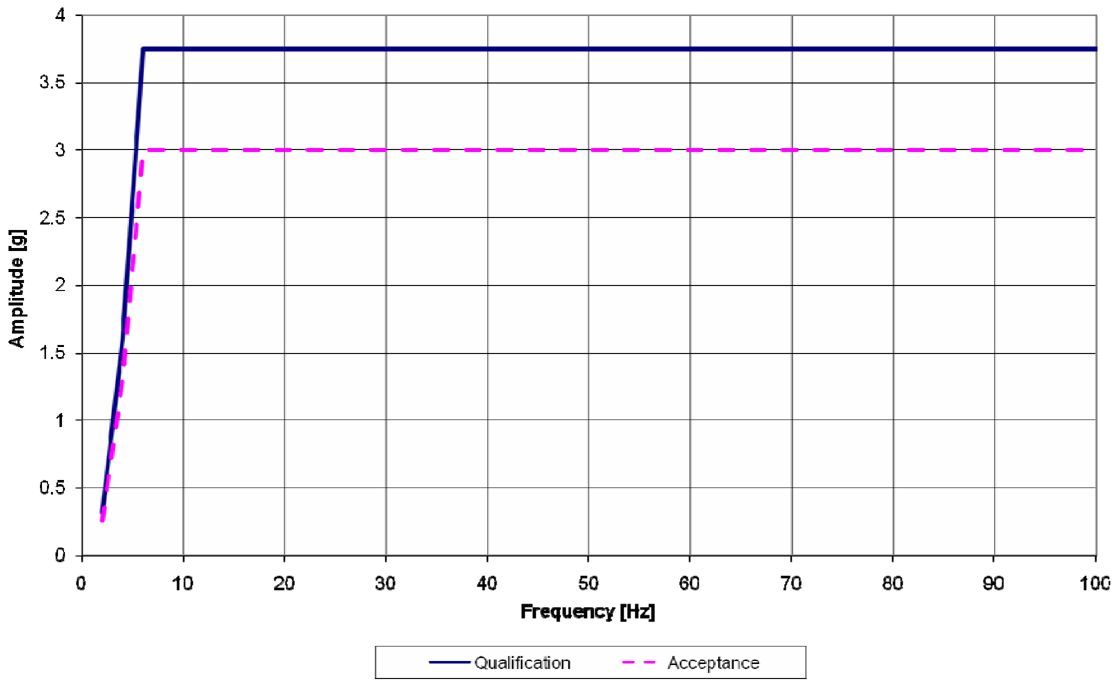

Figure 3 Sinusoidal vibration tests at qualification and acceptance levels.

3.5.1 Tolerances

The maximum tolerances for the different parameters are:

- Amplitude: ±10%
- Frequency: ±2% or 1Hz whichever is greater
- Sweep rate : ±5%

3.6 Random vibration

Random vibrations are generated by propulsion system operation and by the adjacent structure's vibro-acoustic response. Maximum excitation levels are obtained during the first-stage flight. Power spectral density (PSD) of various LV are given in Figure 4 and Figure 5.

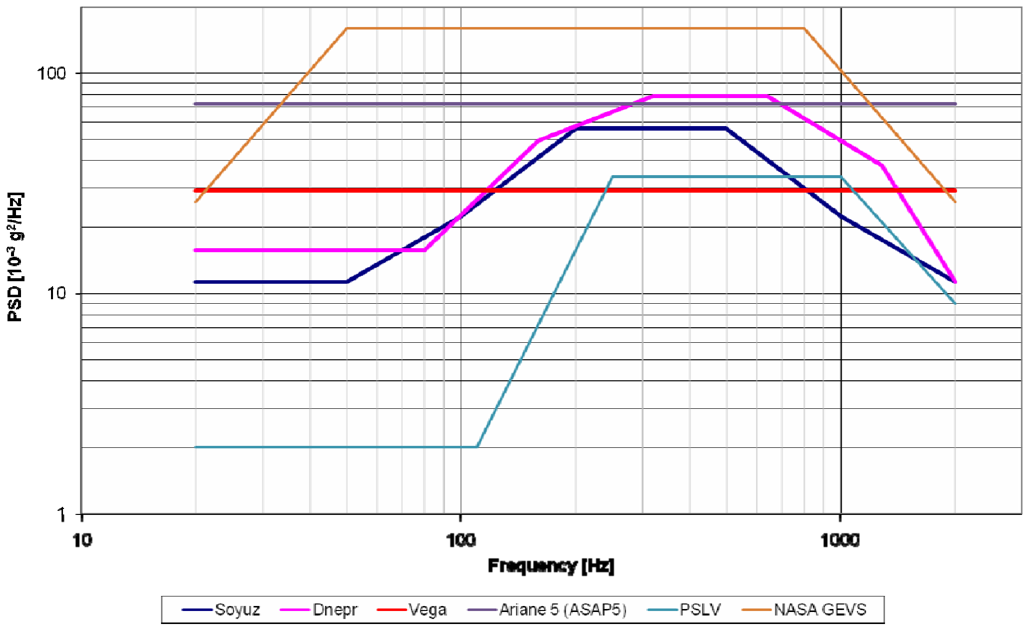

Figure 4 Random vibration at qualification level.

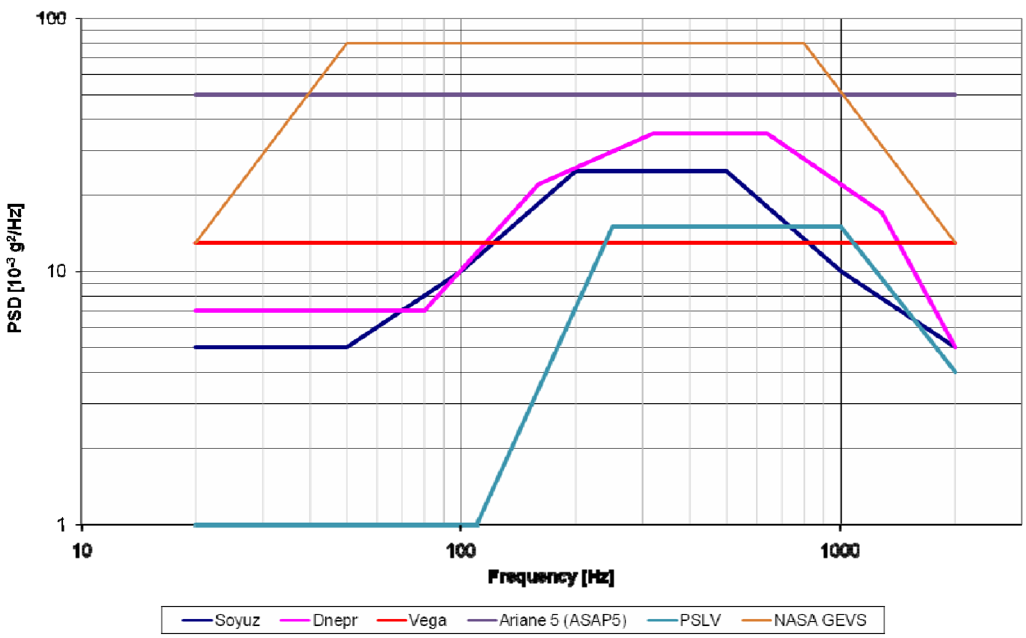

Figure 5 Random vibration at acceptance level.

3.6.1 Test levels and duration

According to the both previous figures, the random vibration tests should have the levels described in Figure 6 and Table 6. Random excitations shall be applied at the base of the mounting adapter, and applied in three mutually orthogonal directions, one being parallel to the thrust axis. The test duration shall be two minutes per axis for qualification and one minute per axis for acceptance.

Table 6 Random vibration tests at qualification and acceptance levels.

acceptance	Frequency [Hz]	20	40	50	800	1000	2000	G_{rms}
	PSD [10^{-3} g^2/Hz]	50	50	80	80	50	50	11.5
qualification	Frequency [Hz]	20	35	50	800	1500	2000	G_{rms}
	PSD [10^{-3} g^2/Hz]	72.7	72.7	160	160	72.7	72.7	15.28

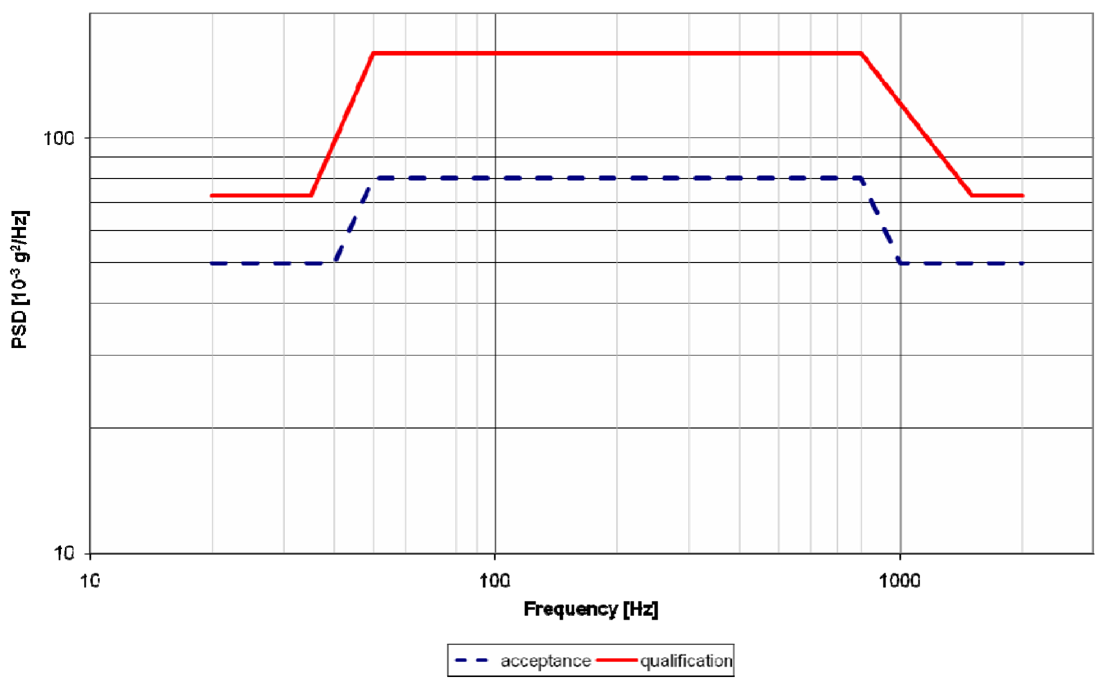

Figure 6 Random vibration tests at qualification and acceptance levels.

3.6.2 Tolerances

The maximum allowable tolerances for the different parameters are:

- Frequency: ±5% or 1Hz whichever is greater
- PSD from 20 Hz to 500 Hz (filter bandwidth 25Hz or less): -1/+3 dB for Qualification and -3/+1.5 dB for Acceptance

- PSD from 500Hz to 2000 Hz (filter bandwidth 50Hz or less): -1/+3dB for Qualification and -3/+1.5 dB for Acceptance
- Random overall G_{rms} : ±10%
- Test time : 0/+10%

3.7 Acoustic vibration

On ground, the noise level generated by the venting system does not exceed 94 dB. This value is for the case of the Ariane 5, Vega or Soyuz LV at the GSC and PSLV.

In flight, acoustic pressure fluctuations under the fairing are generated by engine operation (plume impingement on the pad during lift-off) and by unsteady aerodynamic phenomena during atmospheric flight (i.e., shock waves and turbulence inside the boundary layer), which are transmitted through the upper composite structures. Apart from lift-off and transonic/Q max flight, acoustic levels are substantially lower than the values indicated hereafter.

Table 7 Acoustic noise spectrum under the fairing (qualification level).

	Vega	Soyuz	Dnepr	Ariane 5	PSLV	**Worst case**
Octave Center Frequency [Hz]	Flight limit level [dB]					
31.5	127	128	125	132	128	**132**
63	132	135	132	134	130.5	**135**
125	138	137	135	139	134	**139**
250	135	139	134	143	140	**143**
500	134	137	132	138	144	**144**
1000	123	128	129	132	139	**139**
2000	103	124	126	128	132	**132**
4000			121		129	**129**
8000			115		126	**126**
OASPL	141.5	144	140	146	147	
Duration [sec]	120	120	35	120	120	**120**

Remark: reference 0 dB = $2 \cdot 10^{-5}$ Pa

OASPL: overall acoustic sound pressure level

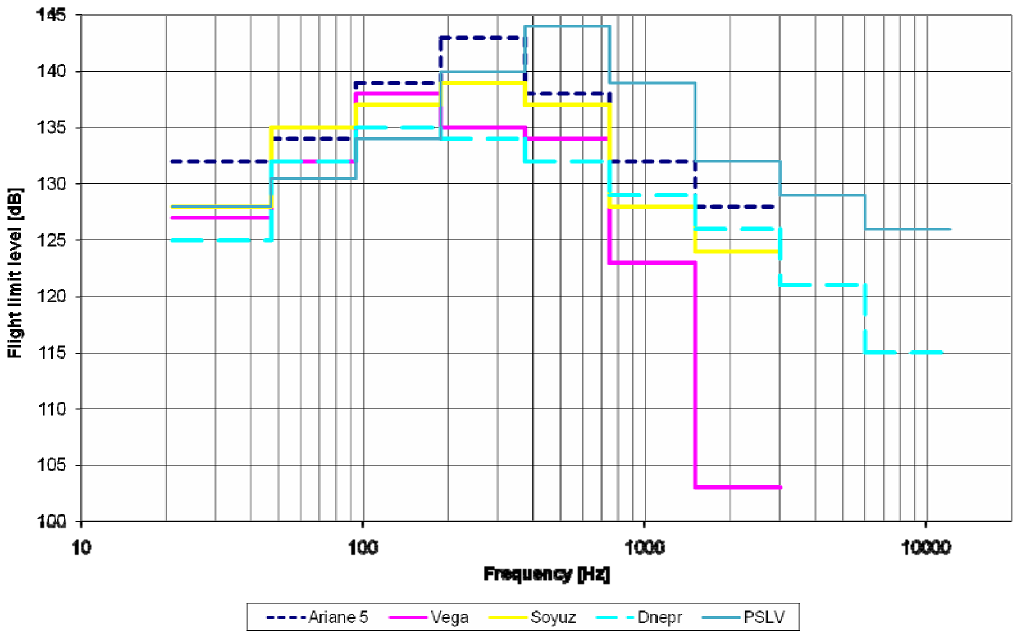

Figure 7 Acoustic noise spectrum for the various LV (qualification level).

3.7.1 Test levels and duration

According to the previous figure, the acoustic vibration tests should have the levels described in Figure 8. The test duration shall be two minutes for qualification and one minute for acceptance.

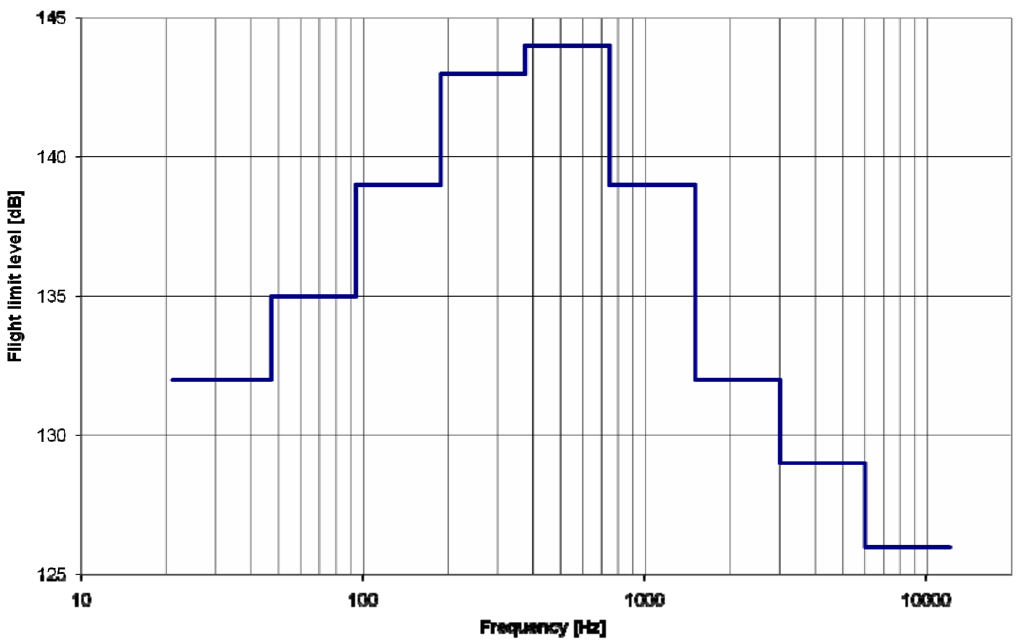

Figure 8 Acoustic noise spectrum, worst case at qualification level.

3.7.1 Tolerances

- SPL : ±3%
- OASPL : ±1.5%
- Test time : 0/+10%

3.8 Shocks

The spacecraft is subject to shock primarily during stage separations, fairing jettisoning, and actual spacecraft separation.

The envelope acceleration shock response spectrum (SRS) at the spacecraft base (computed with a Q-factor of 10) is showed on Figure 9. These levels are applied simultaneously in axial and radial directions.

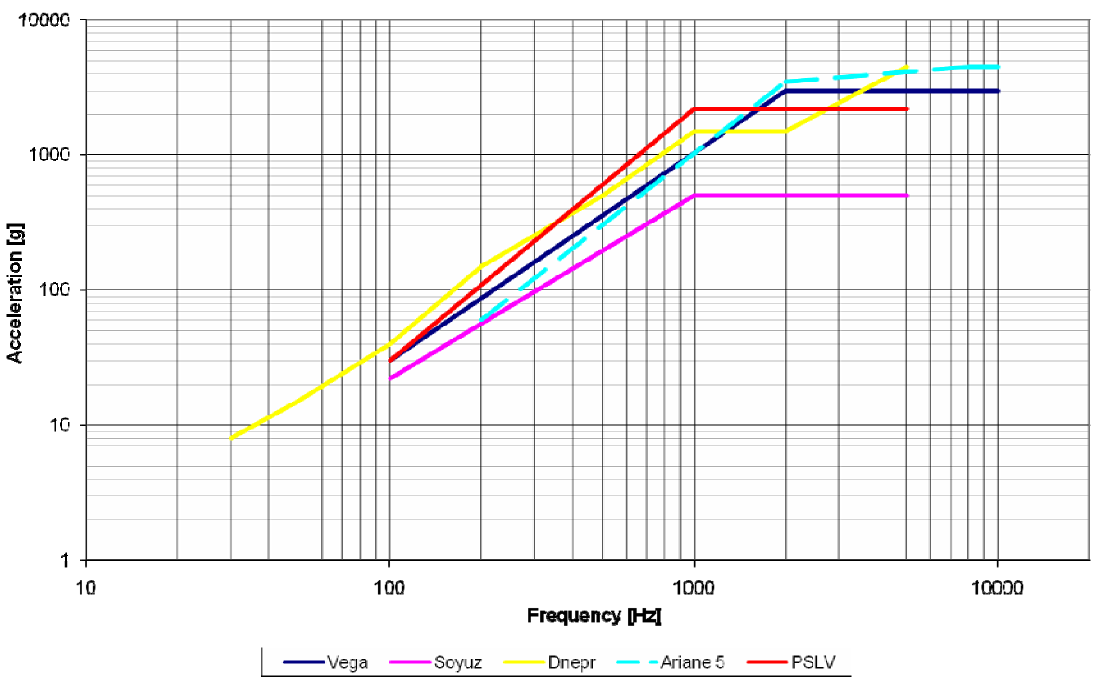

Figure 9 Envelope acceleration shock response spectrum (SRS).

3.8.1 Test levels and duration

According to the previous figure, the shock vibration tests should have the levels described in Figure 10 and Table 8. Shock excitations shall be applied at the base of the mounting adapter, and at least three shocks shall be imposed to meet the amplitude criteria in both directions on each of the three orthogonal axes.

If a suitable test environment can be generated to satisfy the amplitude requirement in all six axial directions by a single application, that test environment shall be imposed three times.

If an imposed shock meets the amplitude requirements in only one direction of a single axis, the shock test shall be conducted a total of 18 times in order to get three valid test amplitudes in both directions of each axis.

Table 8 Shock vibration tests at qualification level.

Frequency [Hz]	30	50	100	200	300	1000	1500	2000	5000	10000
Qualification [g]	8	15	40	150	250	2200	2200	3500	4500	4500

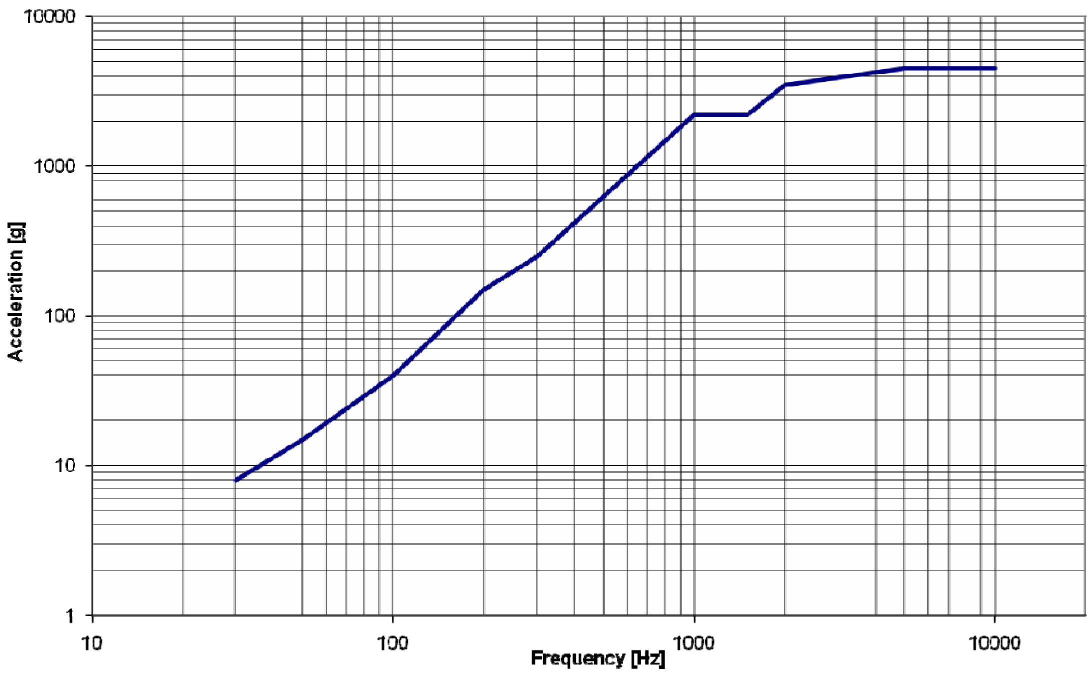

Figure 10 Shock vibration tests at qualification level.

3.8.1 Tolerances

The maximum allowable tolerances for the different parameters are:

- SRS (1/6 octave frequency): ±6dB (with 30% of the response spectrum centre frequency amplitudes greater than nominal test specifications)

3.9 Static pressure under the fairing

On ground after encapsulation, the air velocity around the spacecraft due to the ventilation system is lower than 2 m/sec (value experienced in front of the air inlet) for Ariane 5 and Vega, 5 m/sec for Soyuz. The velocity may locally exceed this value.

In flight, the payload compartment is vented during the ascent phase through vent holes insuring a low depressurization rate of the fairing compartment. For Vega, the depressurization rate under the fairing does not exceed 5,0 kPa/s (50 mbar/s).

For Soyuz, the depressurization rate does not exceed 2,0 kPa/s (20 mbar/s) for a sustained length of time. Locally at the time of maximum dynamic pressure, at ~ 50s, there is a short period of about 2 seconds when the depressurization rate can reach 3,5 kPa/s (35 mbar/s). The difference between the pressure under fairing and free-stream external static pressures, at the moment of the fairing jettisoning, is lower than 0,2 kPa (2 mbar).

For Dnepr, the maximum rate of in-flight pressure change inside the fairing envelope does not exceed 3,4 kPa/s (34 mbar/s), except for transonic phase of flight where a short term (2-3 seconds) increase up to 3,4 kPa/s (34 mbar/s) is possible.

And finally for Ariane 5, the depressurization rate does not exceed 2,0 kPa/s (20 mbar/s) for most time. Locally at the time of maximum dynamic pressure, at ~ 50s, there is a short period of less than 2 seconds when the depressurization rate can reach 4,5 kPa/s (45 mbar/s) in dual launch and 5,0 kPa/s (50 mbar/s) in single launch.

3.10 Thermal Environment

The thermal environment provided during spacecraft preparation and launch has to be considered during the following phases:

- Ground operations
  - The spacecraft preparation within the CSG facilities;
  - The upper composite and launch vehicle operations with spacecraft encapsulated inside the fairing
- Flight
  - Before fairing jettisoning;
  - After fairing jettisoning

3.10.1 Ground operations

The environment that the spacecraft experiences both during its preparation and once it is encapsulated under the fairing is controlled in terms of temperature, relative humidity, cleanliness, and contamination.

For the Vega, Soyuz and Ariane 5 LVs, the typical thermal environment within the air-conditioned GSC facilities is kept around 23°C ± 2°C for temperature and 55% ± 5% for relative humidity.

After encapsulation under the fairing, the environment around the spacecraft is ensured by the insulation capability of the fairing and by an air-conditioning system.

The fairing cavity is vented since encapsulation, including transfer of the upper composite, and the standby phase on the launch pad, up to the lift-off, except during short maintaining operation with the LV on the launch pad. Air-conditioning characteristics are described in Table 9.

Table 9 Thermal environment on ground (for Vega).

S/C location	Transfer between EPCU buildings	S/C in EPCU		Transfer from EPCU to SLV	S/C on L/V
	In CCU container	Not encapsulated	Encapsulated	Encapsulated	On launch pad
Hygrometry level	55% ± 5%	55% ± 5%	55% ± 5%	≤ 20%	≤ 20%
Temperature	≤25°C	≤25°C	Air input temperature of injected air is adjustable between 11°C and 25°C (Accuracy: ± 1°C) Outlet temperature of air ≤25°C for payload radiating less than 200 W	15°C min	≥ 11°C

For information, in the EPCU buildings 998 mbar ≤ P_{atm} ≤ 1023 mbar

3.10.2 Flight environment

Before fairing jettisoning, the thermal flux density radiated by the fairing does not exceed at any point 1000 W/m^2 for Vega and Ariane 5, and 800 W/m^2 for Soyuz. For the complete estimation of the thermal environment under the fairing the spacecraft dissipated power shall be taken into account.

After fairing jettisoning, the nominal time for jettisoning the fairing is determined in order not to exceed the aerothermal flux of 1135 W/ m^2 for Vega, Soyuz and Ariane 5. Typically the aerothermal flux varies from 1135 W/ m^2 to less than 200 W/ m^2 within 20 seconds after the fairing jettisoning, as presented in Figure 11.

Solar-radiation flux, albedo and terrestrial infrared radiation and conductive exchange with LV must be added to this aerothermal flux. While calculating the incident flux on spacecraft, account must be taken of the altitude of the launch vehicle, its orientation, the position of the sun with respect to the launch vehicle, and the orientation of the considered spacecraft surfaces.

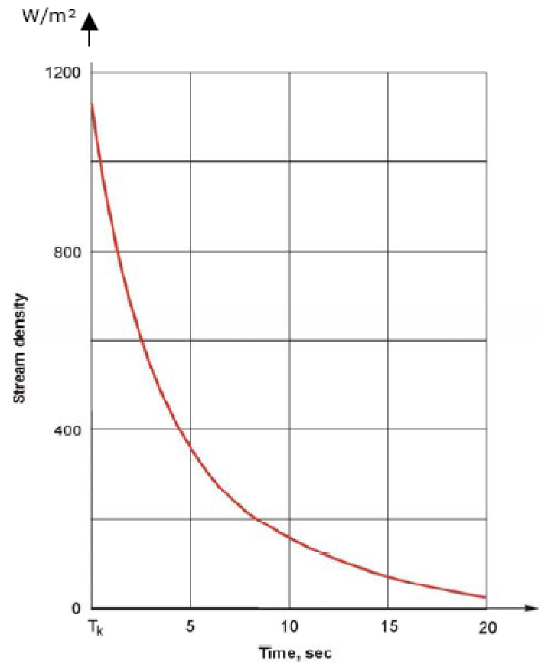

Figure 11 Typical Aerothermal Flux Decay after fairing jettisoning for Vega, Soyuz and Ariane 5.

3.11 Cleanliness and contamination

3.11.1 Cleanliness

At the GSC (for Vega, Soyuz and Ariane 5), the following standard practices ensure that spacecraft cleanliness conditions are met:

- A clean environment is provided during production, test, and delivery of all upper-composite components (upper stage, interstage section, fairing, and adapter) to prevent contamination and accumulation of dust. The LV materials are selected not to generate significant organic deposit during all ground phases of the launch preparation.
- All spacecraft operations are carried out in payload preparation complex buildings in controlled Class 100’000 clean rooms. During transfer between buildings the spacecraft is transported in payload containers with the cleanliness Class 100’000. All handling equipment is clean room compatible, and it is cleaned and inspected before its entry in the facilities.
- Prior to the encapsulation of the spacecraft, the cleanliness of the upper stages and fairing are verified based on the Visibly Clean Level 2 criteria, and cleaned if necessary.
- Once encapsulated and during transfer and standby on the launch pad, the upper composite will be hermetically sealed and a Class 10’000 air-conditioning of the fairing will be provided.
- On the launch pad access can be provided to the payload. The gantry not being air-conditioned cleanliness level is ensured by the fairing overpressure.

3.11.2 Contamination

The following values are coming from the User's Manual of Vega, Soyuz and Ariane 5.

During all spacecraft ground activities from spacecraft delivery to launch site and up to lift-off, the maximum organic non-volatile deposit on the spacecraft surface will not exceed 2 mg/ m^2/week. The organic contamination in facilities and under the fairing is controlled.

The non-volatile organic deposit on the spacecraft surface generated by the materials outgassing does not exceed 2 mg/m^2 (4 mg/m^2 in the case of Ariane 5 LV).

The LV systems are designed to preclude in-flight contamination of the spacecraft. The LVs pyrotechnic devices used by the LV for fairing jettison and spacecraft separation are leak proof and do not leads to any satellite contamination.

The non-volatile organic deposit generated by the attitude control thrusters plume on the adjacent spacecraft surfaces is does not exceed 2 mg/m^2.

The non-volatile organic contamination generated during ground operations and flight is cumulative.

3.11.3 Selection of spacecraft materials

Arianespace gives the following outgassing criteria that the spacecraft materials must satisfy:

- Total Mass Loss (TML) ≤ 1 %
- Collected Volatile Condensable Material (CVCM) ≤ 0.1 %

measured in accordance with the procedure "ECSS-Q-70-02A".

4 REFERENCES

[1] VEGA User's Manual, Issue 3/Revision 0, March 2006.

[2] DNEPR User's Guide, Issue 2, November 2001

[3] Soyuz User's Manual from the GSC, Issue *Draft*, January 2006.

[4] Ariane 5 User's Manual, Issue 4 Revision 0, November 2004.

[5] DNEPR Safety Compliance Requirements, Version 1.0, CalPoly, August 2004.

[6] PSLV User's Manual, Issue 5, April 2007.

[7] ASAP5 User's Manual, Issue 1, May 2000.

[8] NASA GEVS, Rev A, June 1996.