VEGA C

USER'S MANUAL

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User's Manual
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Issued and approved by Arianespace

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Preface

This User's Manual provides essential data on the Vega C launch System.

This document contains the essential data which is necessary:

- To assess compatibility of a spacecraft and spacecraft mission with launch system,
- To constitute the general launch service provisions and specifications, and
- To initiate the preparation of all technical and operational documentation related to a launch of any spacecraft on the launch vehicle.

Inquiries concerning clarification or interpretation of this manual should be directed to the addresses listed below. Comments and suggestions on all aspects of this manual are encouraged and appreciated.

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This document will be revised periodically. In case of modification introduced after the present issue, the updated pages of the document will be provided on the Arianespace website www.arianespace.com before the next publication.

Foreword

Arianespace: the launch Service & Solutions company.

Focused on Customer needs

Arianespace is a commercial and engineering driven company providing complete, personalized launch services, covering the entire period from initial formulation of the project with the customer and its satellite manufacturer, up to the launch.

Through a family of powerful, reliable and flexible launch vehicles operated from the spaceport in French Guiana, Arianespace provides a complete range of lift capabilities.

Arianespace combines low risk and flight proven launch systems with financing, insurance and back-up services to craft tailor-made solutions for start-ups and established players.

With offices in the United States, Japan, Singapore and Europe, and our state-of-the-art launch facilities in French Guiana, Arianespace is committed to forging service packages that meet Customer's requirements.

An experienced and reliable company

Arianespace was established in 1980 as the world's first commercial space transportation company. With over 38 years of experience, Arianespace is the most trusted commercial launch services provider having signed more than 440 contracts, the industry record. Arianespace competitiveness is demonstrated by the market's largest order book that confirms the confidence of Arianespace worldwide customers. Arianespace has processing and launch experience with all commercial satellite platforms as well as with highly demanding scientific missions.

With its family of launch vehicles, Arianespace is the reference service providing: launches of any mass, to any orbit, at any time.

Configuration Control Sheet

Iss./Rev.	Date	Change description	Written by	Approved by
0 / 0	May 18	First Issue	V. Baudel	J. Thiery

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Acronyms, abbreviations and definitions

ω_p	Argument of perigee	
Ω	Ascending node	
Ω_D	Descending node	
a	Semi-major axis	
e	Eccentricity	
g	Gravity (9.81 m/s²)	
i	Inclination	
V_∞	Infinite velocity	
Z_a, h_a	Apogee altitude	
Z_p, h_p	Perigee altitude	
A		
ACS	**A**ttitude **C**ontrol **S**ystem	
ACU	Payload Deputy	**A**djoint **C**harge **U**tile
AE	**A**rian**e**space	
AMF	**A**pogee **M**otor **F**iring	
AIT	**A**ssembly, **I**ntegration and **T**esting	
ALOG		**A**djoint **Log**istique
AME		**A**djoint **Me**sures
AQB		**A**djoint **Q**ualité **B**ase
ARS	Satellite ground stations network Assistant	**A**djoint **R**éseau **S**tations sol Satellite
ASI	Italian Space Agency	**A**gence **S**patiale **I**talienne
AVUM+	**A**ttitude & **V**ernier **U**pper **M**odule **+**	
B		
BT POC	Combined operations readiness review	**B**ilan **T**echnique **P**lan d'**O**pérations **C**ombinées
C		
CAD	**C**omputer **A**ided **D**esign	
CCTV	**C**losed **C**ircuit **T**ele**v**ision network	
CCU	Payload Container	**C**ontainer **C**harge **U**tile
CDC	Mission control centre	**C**entre **d**e **C**ontrôle
CDL	Launch Centre	**C**entre **d**e **L**ancement
CFRP	**C**arbon **F**iber **R**einforced **P**lastic	
CEO	**C**hief **E**xecutive **O**fficer	Président Executif
CG/D	Range director	
CLA	**C**oupled **L**oads **A**nalysis	
CM	Mission Director	**C**hef de **M**ission
CMCU	Mast Payload Links Cabling Cabinet	**C**offret de **M**at **C**harge **U**tile
CNES	French National Space Agency	**C**entre **N**ational d'**E**tudes **S**patiales
COE	Electrical Umbilical Cable	**C**âble **O**mbilical **E**lectrique

	COEL	Launch Site Operations Manager	**C**hef des **O**pérations **E**nsemble de **L**ancement
	CoG	**C**enter **o**f **G**ravity	
	COP	**C**ombined **O**perations **P**lan	Plan d'Opérations Combinées
	COTE	**C**heck-**O**ut **T**erminal **E**quipment	
	CP	Program director	**C**hef de **P**rojet
	CPAP	Arianespace Production Project Manager	**C**hef de **P**rojet **A**rianespace **P**roduction
	CPS	Spacecraft project manager	**C**hef de **P**rojet **S**atellite
	CRAL	Post flight debriefing	**C**ompte **R**endu **A**près **L**ancement
	CRE	Operational Reporting Network	**C**ompte-**R**endu d'**E**tat
	CRSS	**C**lamp **R**ing **S**eparation **S**ystem	**C**ompte-**R**endu d'**E**tat
	CSG	Guiana Space Centre	**C**entre **S**patial **G**uyanais
	CT	Technical Center	**C**entre **T**echnique
	CTS	**C**SG **T**elephone **S**ystem	
	CU	Payload	**C**harge **U**tile
	CVCM	**C**ollected **V**olatile **C**ondensed **M**ass	
	CVI	Real Time Flight Evaluation	**C**ontrôle **V**isuel **I**mmédiat
D			
	DCI	Interface control document	**D**ocument de **C**ontrôle d'**I**nterface
	DDO	Range operations manager	**D**irecteur **d**es **O**pérations
	DEL	Flight synthesis report	**D**ossier d'**E**valuation du **L**ancement
	DLS	**D**ual **L**aunch **S**tructure	
	DMS	Spacecraft mission director	**D**irecteur de la **M**ission **S**atellite
	DOM	French overseas department	**D**épartement d'**O**utre-**M**er
	DUA	Application to use Arianespace's L/V	**D**emande d'**U**tilisation **A**rianespace
E			
	ECSS	**E**uropean **C**ooperation for **S**pace **S**tandardization	
	EGSE	**E**lectrical **G**round **S**upport **E**quipment	
	ELA	Ariane launch site	**E**nsemble de **L**ancement **A**riane
	ELS	Soyuz launch site	**E**nsemble de **L**ancement **S**oyuz
	ELV	ELV S.p.A. (**E**uropean **L**aunch **V**ehicle)	
	EM	**E**lectro**M**agnetic	
	EMC	**E**lectro**M**agnetic **C**ompatibility	
	EPCU	Payload preparation complex	**E**nsemble de **P**réparation **C**harge **U**tile
	EPDM		**E**thylène-**P**ropylène-**D**iène **M**onomère
	ESA	**E**uropean **S**pace **A**gency	
	ESR	**E**quipped **S**olid **R**ocket	
F			
	FC	FC **F**unctional **C**ommand	
	FM	**F**light **M**odel	
	FMA	**F**inal **M**ission **A**nalysis	
	FSA	Russian **F**ederal **S**pace **A**gency	

	FW	**F**ilament **W**ound	
G			
	GEO	**G**eosynchronous **E**quatorial **O**rbit	
	GMT	**G**reenwich **M**ean **T**ime	
	GH_2	**G**aseous hydrogen	
	GN_2	**G**aseous nitrogen	
	GO_2	**G**aseous oxygen	
	GRS	**G**eneral **R**ange **S**upport	
	GS	**G**round **S**upport	
	GSE	**G**round **S**upport **E**quipment	
	GTO	**G**eostationary **T**ransfer **O**rbit	
H			
	HEPA	**H**igh **E**fficiency **P**articulate **A**ir	
	HEO	**H**igh **E**lliptical **O**rbit	
	HPF	**H**azardous **P**rocessing **F**acility	
	HSS	**H**orizontal **S**eparation **S**ubsystem	
	HTPB	**H**ydroxyl-**T**erminated **P**oly**B**utadiene	
I			
	ICD	**I**nterface **C**ontrol **D**ocument	
	IO	Operational Intersite Intercom System	**I**ntercom **O**pérationelle
	IRD	**I**nterface **R**equirements **D**ocument	
	ISCU	Payload safety officer	**I**ngénieur **S**auvegarde **C**harge **U**tile
	ISDN	**I**ntegrated **S**ervices **D**igital **N**etwork	
	ISLA	Launch area safety officer	**I**ngénieur **S**auvegarde **L**ancement **A**rianespace
	ISP	**Sp**ecific **I**mpulse	
	ITAR	**I**nternational **T**raffic in **A**rms **R**egulations	
K			
	KRU	**K**ou**r**o**u**	
L			
	LAN	**L**ocal **A**rea **N**etwork	
	LBC	Check out equipment room	**L**ocal **B**anc de **C**ontrôle
	LCOM	**L**aunch **C**omplex **O**peration **M**anager	
	LCQM	**L**aunch **C**omplex & Customer **Q**uality **M**anager	
	LEO	**L**ow-**E**arth **O**rbit	
	LH_2	**L**iquid **H**ydrogen	
	LIA	Automatic inter link	**L**iaison **I**nter **A**utomatique
	LL	**L**eased **L**ines	
	LLPM	**L**ower **L**iquid **P**ropulsion **M**odule	
	LOX	**L**iquid **ox**ygen	
	LP	**L**aunch **P**ad	
	LSA	**L**aunch **S**ervice **A**greement	
	LSTO	**L**auncher **S**ystem **T**echnical **O**fficer	

L/V	**L**aunch **V**ehicle	
LVA	**L**aunch **V**ehicle **A**dapter	
LW	**L**aunch **W**indow	

M

MCC	**M**ission **C**ontrol **C**entre	
MCI	**M**ass**, C**enter of Gravity**, I**nertia	
MD	**M**ission **D**irector	
MEA	**M**ain **E**ngine **A**VUM+	
MEO	**M**edium-**E**arth **O**rbit	
MEOP	**M**aximum **E**xpected **O**perating **P**ressure	
MG	**M**obile **G**antry	
MGSE	**M**echanical **G**round **S**upport **E**quipment	
MLS	**M**ulti **L**aunch **S**ystem	
MTO	**M**edium-**T**ransfer **O**rbit	
MUA	Ariane user's manual	**M**anuel **U**tilisateur **A**riane
MULTIFOS		**MULTI**plex **F**ibres **O**ptiques **S**atellites

N

N/A	**N**ot **A**pplicable	
NEA	**N**on **E**xplosive **A**ctuators	
NRZ	**N**on **R**eturn to **Z**ero	
NTO	**N**itrogen **T**etr**O**xide	

O

OASPL	**O**verall **A**coustic **S**ound **P**ressure **L**evel	
OBC	**O**n **B**oard **C**omputer	
OCOE	**O**verall **C**heck **O**ut **E**quipment	

P

PABX	**P**rivate **A**utomatic **B**ranch e**X**change	
PAC	**P**ayload **A**ssembly **C**omposite	
PAF	**P**ayload **A**ttachment **F**itting	
PAS	**P**ayload **A**dapter **S**ystem	
PFCU	Payload access platform	**P**late-**F**orme **C**harge **U**tile
PFM	**P**roto-**F**light **M**odel	
PFRCS	Upper Composite Transport Platform	**P**late**F**orme **R**outière **C**omposite **S**upérieur
PLA	**P**ay**L**oad **A**dapter	
PLANET	**P**ayload **L**ocal **A**rea **N**etwork	
POC	Combined Operations Plan	**P**lan d'**O**pérations **C**ombinées
POE	Electrical Umbilical Plug	**P**rise **O**mbilicale **E**lectrique
POI	Interleaved spacecraft operations plan	**P**lan d'**O**pérations **I**mbriquées
POP	Pneumatic Umbilical Plug	**P**rise **O**mbilicale **P**neumatique
POS	Spacecraft operations plan	**P**lan d'**O**pérations **S**atellite
PPF	**P**ayload **P**reparation **F**acility	
PRS	**P**ayload **R**epeater **S**ystem	

ppm	**p**arts **p**er **m**illion	
PSD	**P**ower **S**pectral **D**ensity	
Q		
QA	**Q**uality **A**ssurance	
QSL	**Q**uasi-**S**tatic **L**oad	
QSM	**Q**uality **S**tatus **M**eeting	
QSP	**Q**uality **S**ystem **P**resentation	
QSR	**Q**uality **S**tatus **R**eport	
R		
RAAN	**R**ight **A**scension of the **A**scending **N**ode	
RACS	**R**oll and **A**ttitude **C**ontrol **S**ystem	
RAL	Launch readiness review	**R**evue d'**A**ptitude au **L**ancement
RAMF	Final mission analysis review	**R**evue d'**A**nalyse de **M**ission **F**inale
RAMP	Preliminary mission analysis review	**R**evue d'**A**nalyse de **M**ission **P**réliminaire
RAV	Launch vehicle flight readiness review	**R**evue d'**A**ptitude au **V**ol
RCU	Table Payload Links Interface Cabinet	**R**épartiteur **C**harge **U**tile
RF	**R**adio **F**requency	
RLOC		**R**esponsable **Loc**alisation
RMCU	Payload facilities manager	**R**esponsable des **M**oyens **C**harge **U**tile
RML	**R**evovered **M**ass **L**oss	
RMS	**R**oot **M**ean **S**quare	
ROMULUS	Multiservices operational network	**R**éseau **O**pérationnel **MUL**tiservice à **U**sage **S**patial
ROMS		**R**esponsable **O**ptronique et **M**oyens **S**pécialisés
RPS	Spacecraft preparation manager	**R**esponsable de la **P**réparation **S**atellite
RQLP	AE L/V Production Quality Manager	**R**esponsable **Q**ualité **L**anceur en **P**roduction
RS	Safety Manager	**R**esponsable **S**auvegarde
RSG	**G**round safety officer	**R**esponsable **S**auvegarde Sol
RSV	Flight safety officer	**R**esponsable **S**auvegarde **V**ol
RTEL	Telecommunication Manager	**R**esponsable **Tel**ecommunications
RTM	Telemesure Manager	**R**esponsable **T**ele**m**esure
RTW	**R**adio **T**ransparent **W**indow	
S		
S/C	**S**pace**c**raft	
SCOOP	**S**atellite **C**ampaign **O**rganization, **O**perations and **P**rocessing	
SCA	Attitude control system	**S**ystème de **C**ontrôle d'**A**ttitude
SG	General Specification	**S**pécification **G**énérale
SIW	**S**atellite **I**njection **W**indow	
SLV	Vega launch site	**S**ite de **L**ancement **V**ega
SOW	**S**tatement **of** **W**ork	
SPM	**S**olid **P**ropellant **M**otor	
SRM	**S**olid **R**ocket **M**otor	

	SRS	**S**hock **R**esponse **S**pectrum	
	SSO	**S**un-**S**ynchronous **O**rbit	
	STFO	Optic fibre transmission system	**S**ystème de **T**ransmission par **F**ibre **O**ptique
	STM	**S**tructural **T**est **M**odel	
T			
	TBC	**T**o **B**e **C**onfirmed	
	TBD	**T**o **B**e **D**efined	
	TC	**T**ele**c**ommand	
	TD	Countdown time	**T**emps **D**écompte
	TM	**T**ele**m**etry	
	TS	**T**elephone **S**ystem	
	TV	**T**ele**v**ision	
	TVC	**T**hrust **V**ector **C**ontrol	
U			
	UCIF	**U**pper **C**omposite **I**ntegration **F**acility	
	UCT	**U**pper **C**omposite **T**raveler	
	UDMH	**U**nsymmetrical **D**i**M**ethyl **H**ydrazine	
	ULPM	**U**pper **L**iquid **P**ropulsion **M**odule	
	UPCOM	**U**pper **P**art & Payload **C**ombined **O**perations **M**anager	
	UT	**U**niversal **T**ime	
V			
	VESPA	**Ve**ga **S**econdary **P**ayload **A**dapter	
	VESTA	**Ve**ga **S**hock **T**est **A**pparatus	
	VLAN	**V**irtual **L**ocal **A**rea **N**etwork	
	VS	**V**er**S**us	
	VSS	**V**ertical **S**eparation **S**ubsystem	
W			
	w.r.t.	**W**ith **Re**ference **T**o / **W**ith **R**espect **T**o	
Z			
	Z9	**Z**efiro **9**	
	Z40	**Z**efiro **40**	
	ZL	Launch pad	**Z**one de **L**ancement
	ZLV	Vega Launch pad	**Z**one de **L**ancement **V**ega
	ZLS	Soyuz Launch pad	**Z**one de **L**ancement **S**oyuz
	ZSE	Propellant storage area	**Z**one de **S**tockage d'**E**rgols
	ZSP	Pyrotechnic storage area	**Z**one de **S**tockage **P**yrotechnique

INTRODUCTION Chapter 1

1.1. PURPOSE OF THE USER'S MANUAL

This User's Manual is intended to provide basic information on Arianespace's launch services & solutions using the Vega C launch system operated from the Guiana Space Center.

The content encompasses:

- The Vega C launch vehicle description;
- Performance and launch vehicle mission;
- Environmental conditions imposed by the launch vehicle and corresponding requirements for spacecraft design and verification;
- Description of interfaces between spacecraft and launch vehicle;
- Payload processing and ground operations performed at the launch site;
- Mission integration and management, including support carried out throughout the duration of the launch contract.

Together with the Payload Preparation Complex Manual (EPCU User's Manual) and the Payload Safety Handbook, the Vega C User's Manual provides comprehensive information to assess the suitability of the Vega C launch vehicle and associated launch services to perform their mission, as well as to assess the spacecraft compatibility with the launch vehicle. For every mission, formal documentation is established in accordance with the procedures outlined in Chapter 7.

For more detailed information, the reader is encouraged to contact Arianespace.

1.2. ARIANESPACE LAUNCH SERVICES

To meet all Customers' requirements and to provide the highest quality of services, Arianespace proposes to Customers a fleet of launch vehicles: Ariane 5, Ariane 6, Soyuz, Vega and Vega C. Thanks to their complementarities, they cover all commercial and governmental missions' requirements, providing access to the different types of orbit from Low Earth Orbit (LEO) to Geostationary Transfer Orbit (GTO), and even to interplanetary destinations. This family approach provides Customers with a real flexibility to launch their spacecraft, and insure in a timely manner their planning for in-orbit delivery.

The Customer will appreciate the advantages and possibilities brought by the present synergy, using a unique high quality rated launch site, a common approach to the LV / spacecraft suitability and launch preparation, and the same quality standards for mission integration and management.

Arianespace offers to its Customers reliable and proven launch services that include:

- Exclusive marketing, sales and management of Ariane 5, Ariane 6, Soyuz, Vega and Vega C operations;
- Mission management and support that cover all aspects of launch activities and preparation from contract signature to launch;
- Systems engineering support and analysis;
- Procurement and verification of the launch vehicle and all associated hardware and equipment, including all adaptations required to meet Customer requirements;
- Ground facilities and support (GRS) for Customer activities at launch site;
- Combined operations at launch site, including launch vehicle and spacecraft integration and launch;
- Launcher telemetry and tracking ground station support and post-launch activities;
- Assistance and logistics support, which may include transportation and assistance with insurance, customs and export licenses;
- Quality and safety assurance activities;
- Insurance and financing services on a case-by-case basis.

The contractual commitments between the launch service provider and the Customer are defined in the **Launch Services Agreement (LSA)** with its **Statement of Work (SOW)** and its **Technical Specification**.

At the LSA signature, Arianespace provides the Customer with a project oriented management system, based on a single point of contact (the Program Director) for all launch service activities, in order to simplify and streamline the process, adequate configuration control for the interface documents and hardware, and transparency of the launch system to assess the mission progress and schedule control.

1.3. VEGA C LAUNCH VEHICLE – HISTORY

Vega

The Vega program (Vettore Europeo di Generazione Avanzata) has its origins back in the early 1990s, when studies were performed to investigate the possibility of complementing the Ariane family with a small launch vehicle using Ariane solid booster technology.

Vega began as a national Italian concept. BPD Difesa y Spazio in 1988 proposed a vehicle to the Italian Space Agency (ASI) to replace the retired US Scout launcher by a new one based on the Zefiro motor developed from the company's Ariane expertise.

After about ten years of definition and consolidation activities, the Italian Space Agency and Italian industry proposed Vega as a European project based on their know-how in solid propulsion inherited from development and production of Ariane 4 solid strap-on boosters (PAP) and components of the Ariane 5 solid strap-on boosters (EAP).

In April 1998, ESA's Council approved a Resolution authorizing pre-development activity.

As a result the present configuration was chosen with first stage that could serve also as an improved Ariane-5 strap-on.

The Vega program was approved by ESA Ariane Programme Board on 27-28 November 2000, and the project officially started on 15 December 2000 when seven countries subscribed to the Declaration.

Vega is operated starting in 2012 at the Guiana Space Center in French Guiana from rehabilitated launch pad ELA 1 that was originally used for Ariane 1 launch vehicle (taken benefit of the existing facilities).

AVIO S.p.A company is in charge of the Vega launcher development and production. The Vega launch system is developed for a launch rate up to four launches per year.

Vega C

Following the decisions taken during the December 2014 and December 2016 ESA Ministerial Councils, ESA and European industry are currently developing an upgraded and more powerful version of the Vega launcher: Vega C. The main objective is to increase the launch vehicle performance and increase the flexibility for multiple payloads missions.

The inaugural flight of the Vega C launch vehicle is scheduled in 2019.

1.4. LAUNCH VEHICLE DESCRIPTION

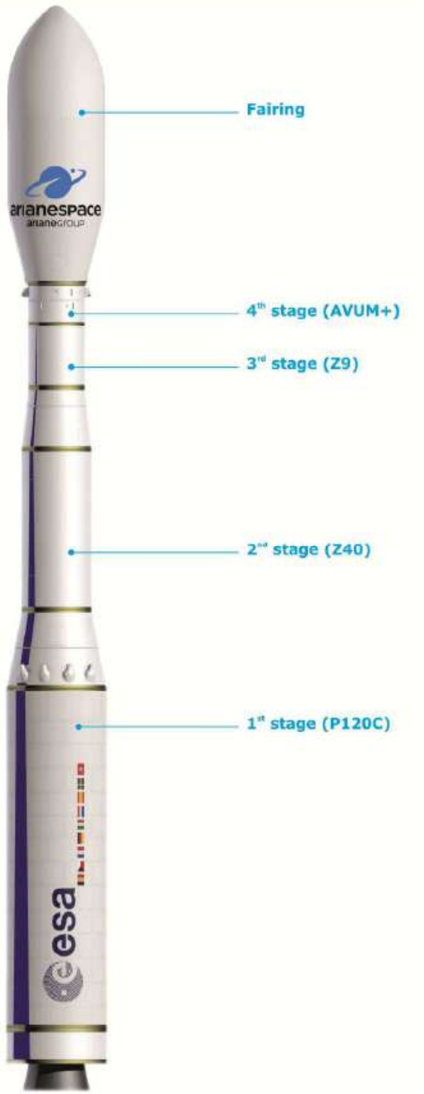
Arianespace offers a complete launch system including the vehicle, the launch facilities and the associated services.

1.4.1. Launch vehicle general data

The Vega LV consists primarily of the following components:

- A lower composite consisting of three solid propellant stages;
- A re-ignitable AVUM+ (Attitude and Vernier Upper Module) upper stage;
- A payload fairing; and
- Depending on the mission requirements, a variety of different adapters / dispensers / dual launch structure or carrying structures may be used;
- Carrying structure for mini, micro satellites and nanosats.

The Vega C configuration and relevant vehicle data are presented in Figure 1.4.1.a.



PAYLOAD FAIRING	
Diameter:	3.317 m
Length:	9.374 m
Mass:	860 kg
Structure:	Two halves - Sandwich panels CFRP sheets and aluminum honeycomb core
Separation:	Vertical separations by means of leak-proof pyrotechnical expanding tubes and horizontal separation by a clamp-band

PAYLOAD ADAPTERS, MULTIPLE LAUNCH STRUCTURE			
VAMPIRE 937		**VAMPIRE 1194**	
Height (mm):	2 596	**Height** (mm):	1 861
Mass (kg):	120 TBC	**Mass** (kg):	95 TBC
VESPA C – Short version		**VESPA C – Long version**	
Height (mm):	3 222 TBC	**Height** (mm):	4 552 TBC
Diameter (mm):	2 620 TBC	**Diameter** (mm):	2 620 TBC
Mass (kg):	390 TBC	**Mass** (kg):	455 TBC
SSMS			
Piggy-Back			
Ride-Share			

AVUM+ UPPER STAGE	
Size:	2.18-m diameter × 2.04-m height
Dry mass:	698 kg TBC
Propellant:	492 kg/248 kg of NTO/UDMH
Subsystems:	
Structure:	Aluminium cylindrical case with 4 Aluminium propellant tanks and supporting frame
Propulsion:	MEA (evolution of RD-869) – 1 chamber
- Thrust	2.45 kN – Vacuum
- Isp	315.8 s – Vacuum
- Feed system	Regulated pressure-fed 108 l (4.8 kg) GHe tank MEOP 328 barA
- Burn time/restart	Up to 612.5 s (max. cumulative firing time: 924.8 s) / up to 5 controlled or depletion burns
RACS:	Six 240 N hydrazine thrusters N_2H_4 ; 39 l (38.6 kg) N_2H_4 tank MEOP 26 barA
Avionics:	Inertial 3-axis platform, on-board computer, TM & RF systems, Power
Attitude control:	
- Pitch, yaw	*Main engine ±10 deg gimbaled nozzle → boosted phases* *Six RACS thrusters → ballistic phases*
- Roll	*Roll rate and attitude controlled by four of the six RACS thrusters*

	1st STAGE (P120C)	2nd STAGE (Z40)	3rd STAGE (Z9)
Size:	3.40-m diameter × 13.38-m length	2.40-m diameter × 8.07-m length	1.90-m diameter × 4.12-m length
Gross mass:	155 027 kg	40 477 kg	12 000 kg
Propellant:	141 634 kg of HTPB	36 239-kg of HTPB	10 567-kg of HTPB
Subsystems:			
Structure	Carbon-epoxy filament wound monolithic motor case protected by EPDM	Carbon-epoxy filament wound monolithic motor case protected by EPDM	Carbon-epoxy filament wound monolithic motor case protected by EPDM
Propulsion	P120 Solid Rocket Motor (SRM)	ZEFIRO 40 Solid Rocket Motor (SRM)	ZEFIRO 9 Solid Rocket Motor (SRM)
- Thrust	4 323 kN Max Vac thrust	1 304 kN Max Vac thrust	317 kN – Max Vac thrust
- Isp	279 s – Vac	293.5 s – Vac	295.9 s – Vac
- Burn time	135.7 s	92.9 s	119.6 s
Avionics		Actuators I/O electronics, power	Actuators I/O electronics, power
Attitude control:			
- Pitch, yaw	*Gimbaled ±5.9 deg nozzle with electro mechanical actuators*	*Gimbaled ±5.9 deg nozzle with electro mechanical actuators*	*Gimbaled ±6 deg nozzle with electro mechanical actuators*
- Roll	*Roll rate limited by four of the six RACS thrusters*	*Roll rate limited by four of the six RACS thrusters*	*Roll rate and attitude controlled by four of the six RACS thrusters*
Interstage:	**0/1 interstage:** Structure: Cylinder aluminum shell/inner stiffeners Housing: Actuators I/O electronics, power, safety/destruction subsystem **1/2 interstage:** Structure: Conical aluminum shell/inner stiffeners Housing: TVC local control equipment; safety/destruction subsystem		**2/3 interstage:** Structure: Composite grid structure Housing: TVC local control equipment; safety/destruction subsystem **3/AVUM+ interstage:** Structure: Aluminium cylinder with integral machined stringers Housing: TVC control equipment; safety/destruction subsystem, power distribution, RF and telemetry subsystems
Stage separation:	Linear cutting charge/Retro rocket thrusters		Linear cutting charge/springs Pyrotechnic tight expansible tube/springs

Figure 1.4.1.a – Launch vehicle general data

1.4.2. Launch configurations

The Vega C launch configurations presented in this User’s Manual are defined here below.

- **Single launch configuration**

  This launch configuration consists in only one spacecraft to be integrated on Vega C, using an off-the-shelf adapter.

  This adapter will be:

  - a “VAMPIRE 937”;
  - a “VAMPIRE 1194”.

Figure 1.4.2.a – Vega C single launch configuration

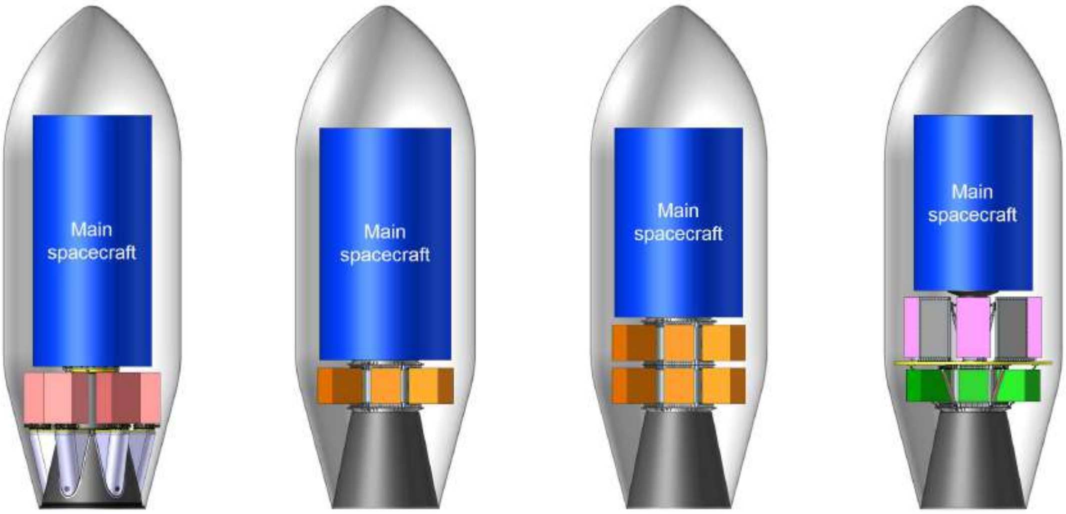
The use of the adapters PLA 937 VG and PLA 1194 VG on Vega C shall be evaluated on a case by case basis.

- **Launch configuration with main spacecraft and auxiliary passenger(s)**

  This launch configuration consists in one main spacecraft and several mini or micro satellites to be integrated on Vega C, using a multiple launch structure.

  This multiple launch structure will be:

  - a "VAMPIRE 937 with towers";
  - or a "SSMS" (Piggy-Back or Ride-Share configuration).



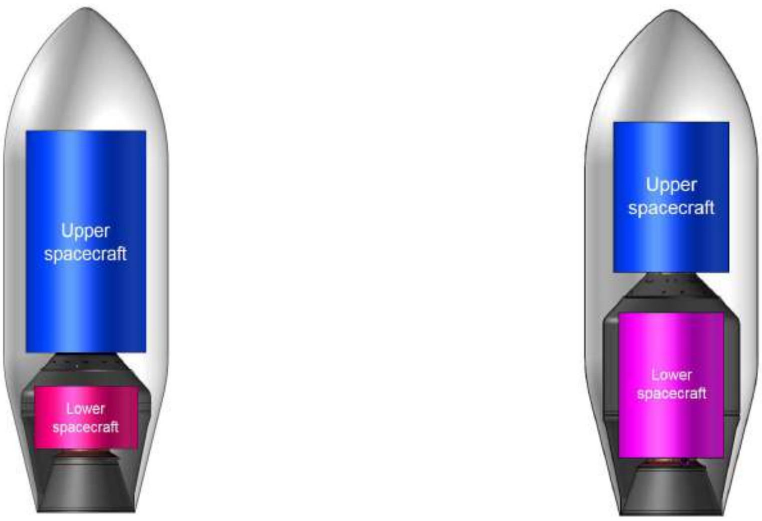
VAMPIRE 937 with towers | SSMS Piggy-Back (HEXA 1) | SSMS Piggy-Back (HEXA 2) | SSMS Ride-Share

Figure 1.4.2.b – Vega C launch configuration with main spacecraft and auxiliary passenger(s)

Specifications related to mini (200-400 kg), micro (50-200 kg) or nano (< 50 kg) satellites (illustrated in , , and in the Figure 1.4.2.b) will be part of the Auxiliary Passengers User's Manual.

- **Dual launch configuration**

  This launch configuration consists in one upper spacecraft and one lower spacecraft to be integrated on Vega C, using the VESPA C dual launch structure.



VESPA C – Short version VESPA C – Long version

Figure 1.4.2.c – Vega C dual launch configurations

The "VESPA C" dual launch structure proposed on Vega C is a "VESPA"-type structure with larger dimensions in order to take advantage of the larger volume available inside the Vega C fairing.

1.4.3. European spaceport and CSG facilities

Arianespace launch services are carried out at the Guiana Space Center (CSG) – European spaceport in operation since 1968 in French Guiana. The spaceport accommodates Ariane, Soyuz and Vega launch facilities (ELA, ELS and SLV respectively) with common Payload Preparation Complex (Ensemble de Preparation Charge Utile – EPCU) and launch support services.

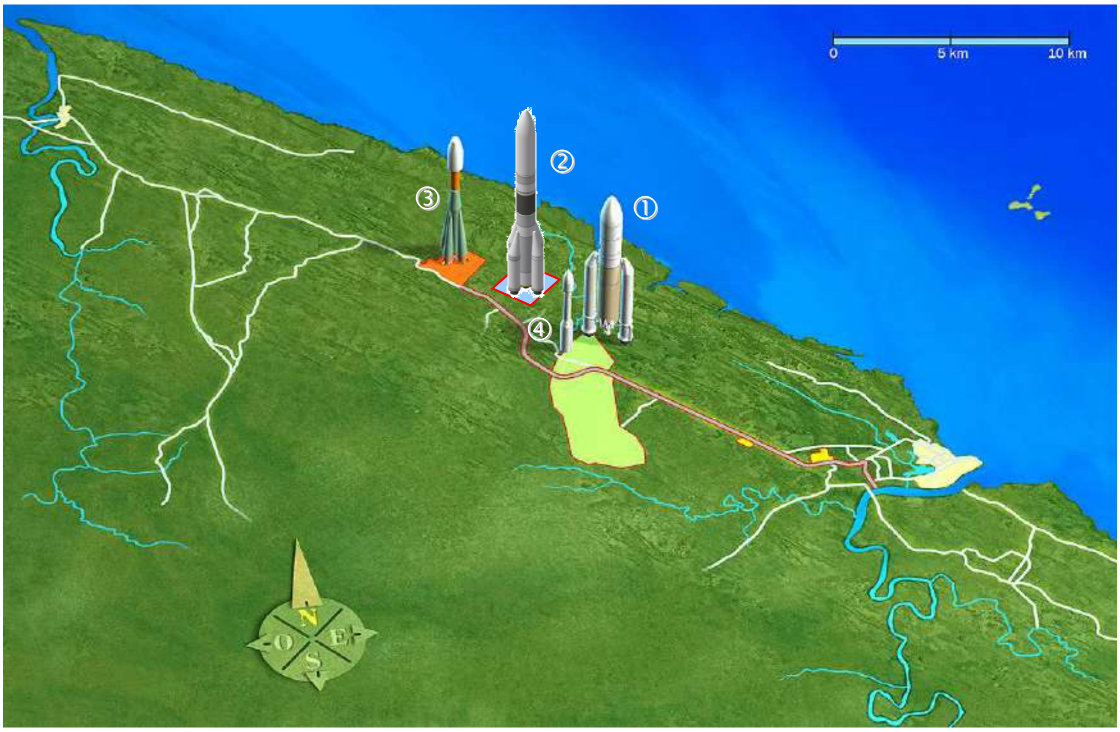
The CSG is governed under an agreement between France and the European Space Agency (ESA) that was extended to cover Soyuz and Vega installations. Day-to-day operations at CSG are managed by the French National Space Agency (Centre National d'Etudes Spatiales – CNES) on behalf of the European Space Agency (ESA). CNES provides range support to Arianespace, for spacecraft, launch vehicle preparation and launch.

The CSG provides state-of–the-art Payload Preparation Facilities (Ensemble de Preparation Charge Utile – EPCU) recognized as a high quality standard in space industry. The facilities are capable to process several satellites of different Customers in the same time, thanks to large clean rooms and supporting infrastructures. Designed for multiple launch capability and high launch rates, the EPCU capacity is sufficient to be shared by the Customers of all three launch vehicles.

The satellite / launch vehicle integration and launch are carried out at launch sites dedicated to the Ariane, Soyuz and Vega launch systems.

Vega and Vega C are operated from the Vega Launch Site (Site de Lancement Vega – SLV). The SLV is built on the ELA1 previously used for the Ariane 1 and Ariane 3 launches. SLV is located 1 km South-West of the Ariane 5 launch pad (ELA3) and provides the same quality of services for combined launch vehicle operations with spacecraft.

The moderate climate, the regular air and sea connection, accessible local transportation, and excellent accommodation facilities for business and for recreation – all those devoted to Customer's team and invest to the success of the launch mission.



①	Ariane 5 launch area	②	Ariane 6 launch area
③	Soyuz launch area	④	Vega & Vega C launch area

Figure 1.4.3.a – CSG overview

1.4.4. Launch service organization

Arianespace is organized to offer launch services based on a continuous interchange of information between a Spacecraft Interface Manager (Customer), and the Arianespace Program Director (Arianespace) who is appointed at the time of the Launch Services Agreement signature. As from that date, the Arianespace Program Director is responsible for the execution of the Launch Services Agreement.

For the preparation and execution of the Guiana operations, the Arianespace launch team is managed by a specially assigned Mission Director who will work directly with the Customer's operational team.

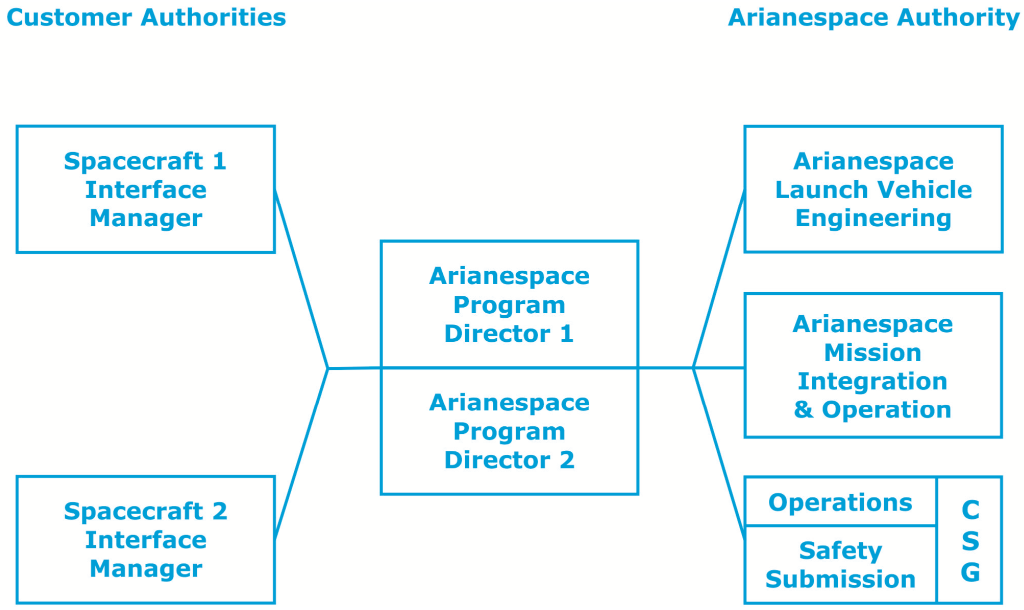


Figure 1.4.4.a – Principle of Customers / Arianespace relationship

1.5. CORPORATE ORGANIZATION

1.5.1. Arianespace

Arianespace is a French joint stock company which was incorporated on 26th March 1980 as the first commercial space transportation company.

In order to meet the market needs, Arianespace has established a worldwide presence: in Europe, with headquarters located at Evry near Paris, France; in North America with Arianespace Inc., its subsidiary in Washington D.C., and in the Pacific Region, with its representative office in Tokyo (Japan) and Arianespace Pte. Ltd., its subsidiary in Singapore.

Arianespace is the international leader in commercial launch services and today holds an important part of the world market for satellites launched to the geostationary transfer orbit. From its creation in 1980 up to mid 2018, Arianespace has performed 242 Ariane, 18 Soyuz and 11 Vega launches from the European spaceport. Arianespace signed contracts for more than 600 payloads with some 90 operators/Customers.

Arianespace provides each Customer a true end-to-end service, from manufacture of the launch vehicle to mission preparation at the Guiana Space Center and successful in-orbit delivery of payloads for a broad range of missions.

Arianespace as a unique commercial operator oversees the marketing and sales, production and operation at CSG of Ariane, Soyuz and Vega launch vehicles.

Figure 1.5.1.a – Arianespace worldwide

1.5.2. Partners

Arianespace is backed by shareholders that represent the best technical, financial and political resources of the European countries participating in the Ariane and Vega programs.

European Space Agency provides financing, technical and political support for Vega development and operation. The Vega program is financed by the following participating European states: Belgium, Italy, France, the Netherlands, Spain, Sweden and Switzerland. The ESA's technical supervision is provided in the same way as it was made for all Ariane family bringing the 30 years of the previous experience. The ESA and the participating states decisions provide the formal base for the Vega integration in European space transportation fleet and its access to the institutional market insuring long term prospects.

1.5.3. European space transportation system organization

Arianespace benefits from a simplified procurement organization that relies on AVIO S.p.A, prime supplier for Vega C.

Vega C launch operations are managed by Arianespace with the participation of the prime supplier and range support from CNES CSG.

Figure 1.5.3.a shows the launch vehicle procurement organization:

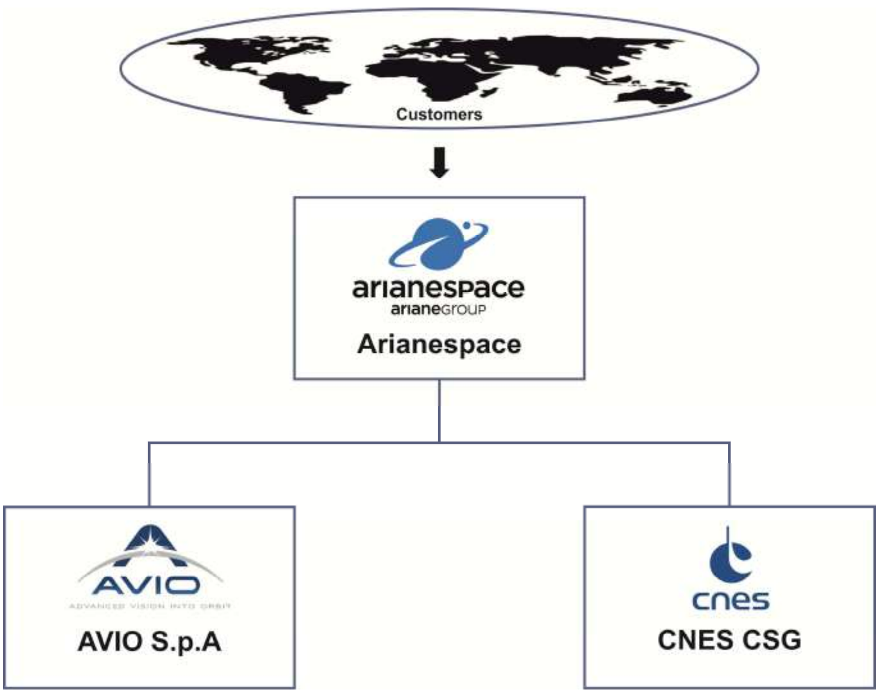


Figure 1.5.3.a – The launch vehicle procurement and range support organization

1.5.4. AVIO main suppliers

The AVIO S.p.A company, based in Colleferro, Italy, manages the Vega development and production. Its business relies on the experience gained by the shareholders in the field of the solid propulsion as suppliers of the Ariane 3, Ariane 4 and Ariane 5 boosters.

AVIO, as industrial prime contractor, is in charge of acceptance of the launcher's components and integration in French Guiana. As the launcher design authority, it will also participate in final preparations and launch operations

AVIO establishes close working relations with well-known European suppliers and partners.

Among them: Europropulsion, Airbus Defence & Space, SABCA, Dutch Space, Ruag Space, etc.

To illustrate the industrial experience concentrated behind the Vega C prime supplier, the Figure 1.5.4.a shows subcontractors and their responsibilities:

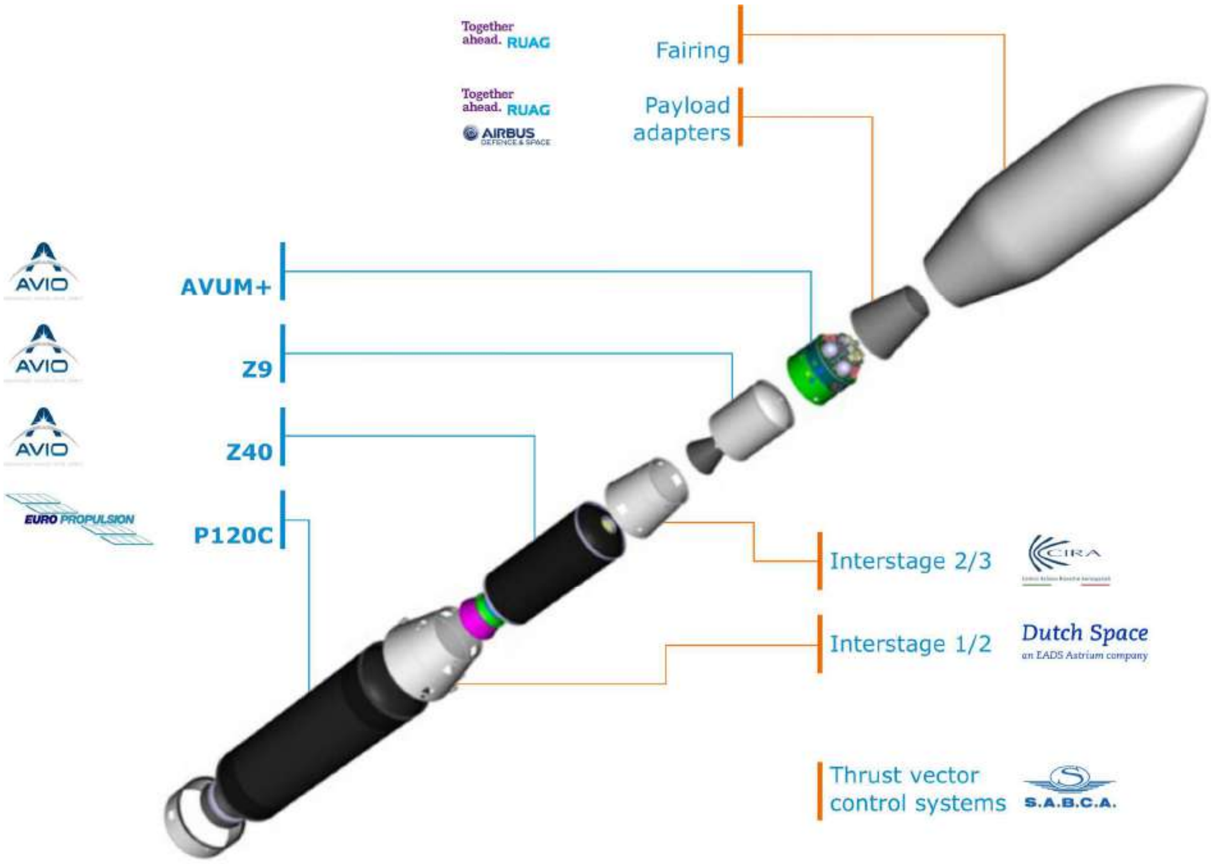


Figure 1.5.4.a – The Vega C subcontractors

PERFORMANCE AND LAUNCH MISSION Chapter 2

2.1. INTRODUCTION

This section provides the information necessary to make preliminary performance assessments for the Vega C launch vehicle. The following paragraphs present the launch vehicle reference performance, typical accuracy, attitude orientation and mission duration.

The provided data cover a wide range of missions from spacecraft delivery to injection into sun-synchronous orbits (SSO), to injection into low circular or equatorial elliptical orbit missions.

Performance data presented in this manual are typical ones and not fully tailored to the specificities of the Customer's mission.

2.2. PERFORMANCE DEFINITION

The performance figures given in this chapter are expressed in term of payload mass including:

- The separated mass of all embarked satellites;
- The mass of their adapters;
- The carrying structure mass if any (VESPA C or SSMS).

Available payload adapters and associated masses are presented in Appendix 5.

The payload adapter also ensures encapsulation of the AVUM+ upper stage and provides the electrical interface to the fairing.

Performance computations are based on the following main assumptions:

- Launch at the CSG (French Guiana) taking into account the relevant CSG safety rules. Nevertheless, the performance value may slightly vary for specific missions due to ground path and launch azimuth specific constraints. The Customer is requested to contact Arianespace for accurate data.
- Sufficient propellant reserve is assumed to reach the targeted orbit. The AVUM+'s fuel capacity is sufficient for transfer to a graveyard orbit or for a controlled re-entry in the Earth atmosphere, as required by regulation.
- Maximal aerothermal flux is less or equal to 1 135 W/m^2 at fairing jettisoning.

Data presented herein do not take into account additional equipment or services that may be requested.

2.3. TYPICAL MISSION PROFILE

A typical Vega C mission includes the following three phases:

- Phase I: Ascent phase of the P120C, Zefiro 40 (Z40), Zefiro 9 (Z9) and AVUM+ to reach the required orbit;
- Phase II: Ballistic phase with orbital maneuvers of the AVUM+ stage for payload delivery in the proper conditions;
- Phase III: AVUM+ orbit disposal maneuvers or deorbitation.

The AVUM+ upper stage is a restartable upper stage (up to 5 times) offering a great flexibility to servicing a wide range of orbits, and allowing delivering the payload to different orbits in case of shared launch.

The ascent AVUM+ phase typically consists of two burns to reach the targeted orbit: a first AVUM+ burn is used to reach an intermediate orbit, followed by a coast phase which duration depends of the targeted orbit, and a second AVUM+ burn to reach the final orbit. This is the typical mission profile for sun-synchronous orbit (SSO) and low earth orbit (LEO).

After spacecraft separation and following the time delay needed to provide a safe distance between the AVUM+ upper stage and the spacecraft, AVUM+ maneuvers intend to release spacecraft operational orbits or to trigger a controlled re-entry in the Earth's atmosphere. This can be carried out by an additional burn of the AVUM+ main engine. Parameters of the re-entry into the Earth's atmosphere will be chosen in accordance with CSG regulation and will be coordinated with the Customer during mission analysis.

The flight profile is optimized for each mission. Specific mission profiles can be analyzed on a mission-peculiar basis.

2.3.1. Ascent profile

The flight profile is optimized for each mission. It is based on the following flight phases and events:

- 1st stage flight (P120C) with initial vertical ascent, programmed pitch maneuver and a zero-incidence flight;
- 2nd stage (Z40) flight;
- 3rd stage (Z9) flight, fairing separation and injection into sub-orbital trajectory.

The typical Vega C ascent profiles and associated sequence of events are presented in the Table 2.3.1.a. A typical ground track and example of the flight parameters during the ascent profile are illustrated in paragraph 2.4.1 (sun-synchronous orbit) and paragraph 2.4.2 (equatorial elliptical orbit).

The fairing is released at the beginning of the Z9 flight phase when the aerothermal flux becomes lower or equal to 1 135 W/m^2.

	Time / H0 (s)	Altitude (km)	Rel. velocity (m/s)
• P120C ignition & lift-off	0	0	0
• P120C burn-out & separation, Z40 ignition	142	60	1 885
• Z40 burn-out & separation	245	121	4 555
• Z9 ignition	249	123	4 550
• Fairing jettisoning	254	126	4 600
• Z9 separation	417	190	7 564
• AVUM+ 1st ignition	448	199	7 553
• AVUM+ 1st cut-off	1 090	300	7 885
• AVUM+ 2nd ignition	3 151	619	7 533
• AVUM+ 2nd cut-off	3 287	623	7 631
• Spacecraft separation	3 427	626	7 627

Table 2.3.1.a - Typical ascent profile (two AVUM+ boosts mission profile)

2.3.2. AVUM+ upper stage phase

After 3rd stage (Z9) separation during the sub-orbital flight, the multiple AVUM+ burns are used to transfer the payload to a wide variety of intermediate or final orbits, providing the required plane changes and orbit raising.

Up to 5 burns may be provided by the AVUM+ to reach the targeted orbit(s) and ensure a re-entry of the AVUM+ upper stage.

2.3.3. AVUM+ deorbitation or orbit disposal maneuver

After spacecraft separation and following the time delay needed to provide a safe distance between the AVUM+ upper stage and the spacecraft, the AVUM+ typically conducts a deorbitation or orbit disposal maneuver by mean of one last burn. Parameters of the graveyard orbit or re-entry into Earth's atmosphere will be chosen in accordance with standard regulation on space debris and will be coordinated with the Customer during mission analysis.

2.4. GENERAL PERFORMANCE DATA

The earth observation, meteorological and scientific satellite will benefit from the Vega C capability to deliver them directly into the sun-synchronous orbits (SSO), polar circular orbits, or circular orbits with different inclination.

2.4.1. Sun-synchronous orbit (SSO) missions

The typical Vega C single launch mission includes an ascent profile with two AVUM+ burns as follows:

- A 1st AVUM+ burn for transfer to the intermediate orbit;
- A 2nd AVUM+ burn for orbit circularization;
- A 3rd AVUM+ burn for orbit disposal maneuver or deorbitation (not illustrated in the figures here below).

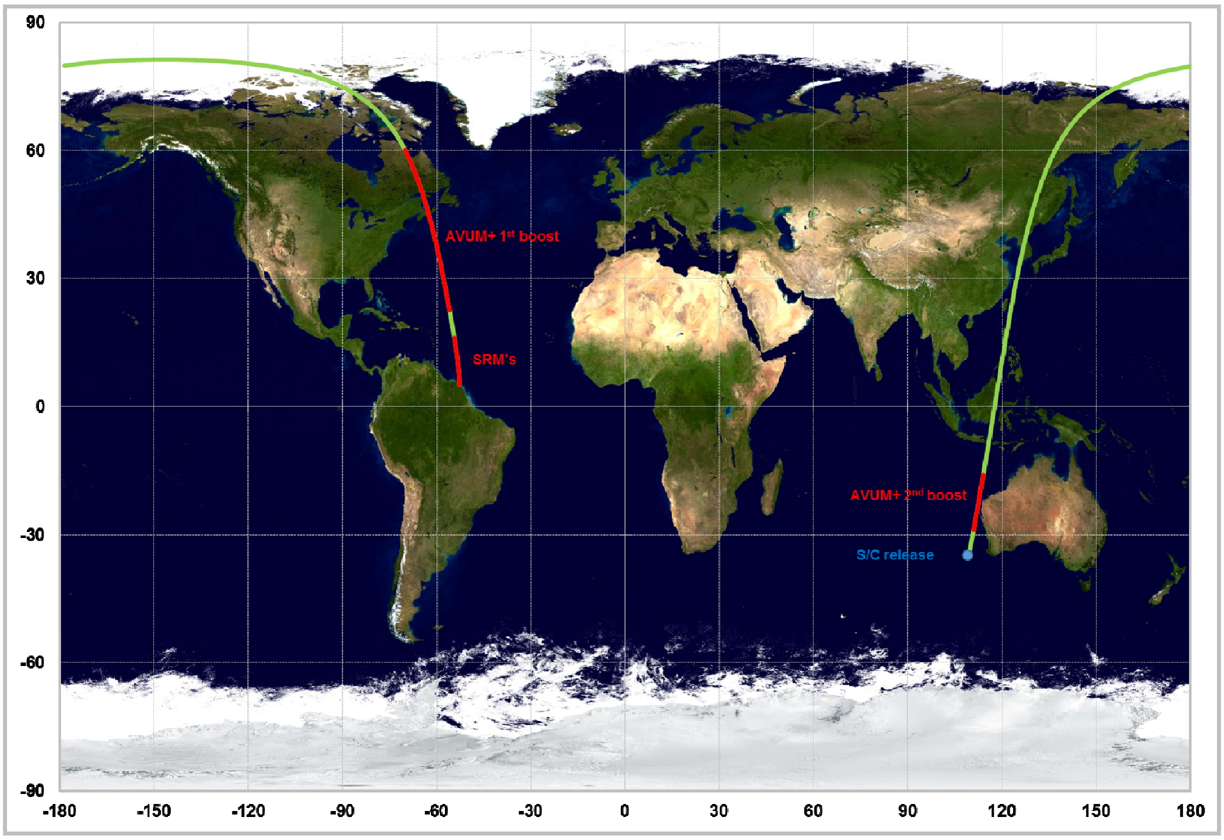


Figure 2.4.1.a – Typical ground path for the Vega C SSO single launch mission (two AVUM+ boosts mission profile)

Typical evolutions of altitude and relative velocity from lift-off till spacecraft separation are presented in Figure 2.4.1.b:

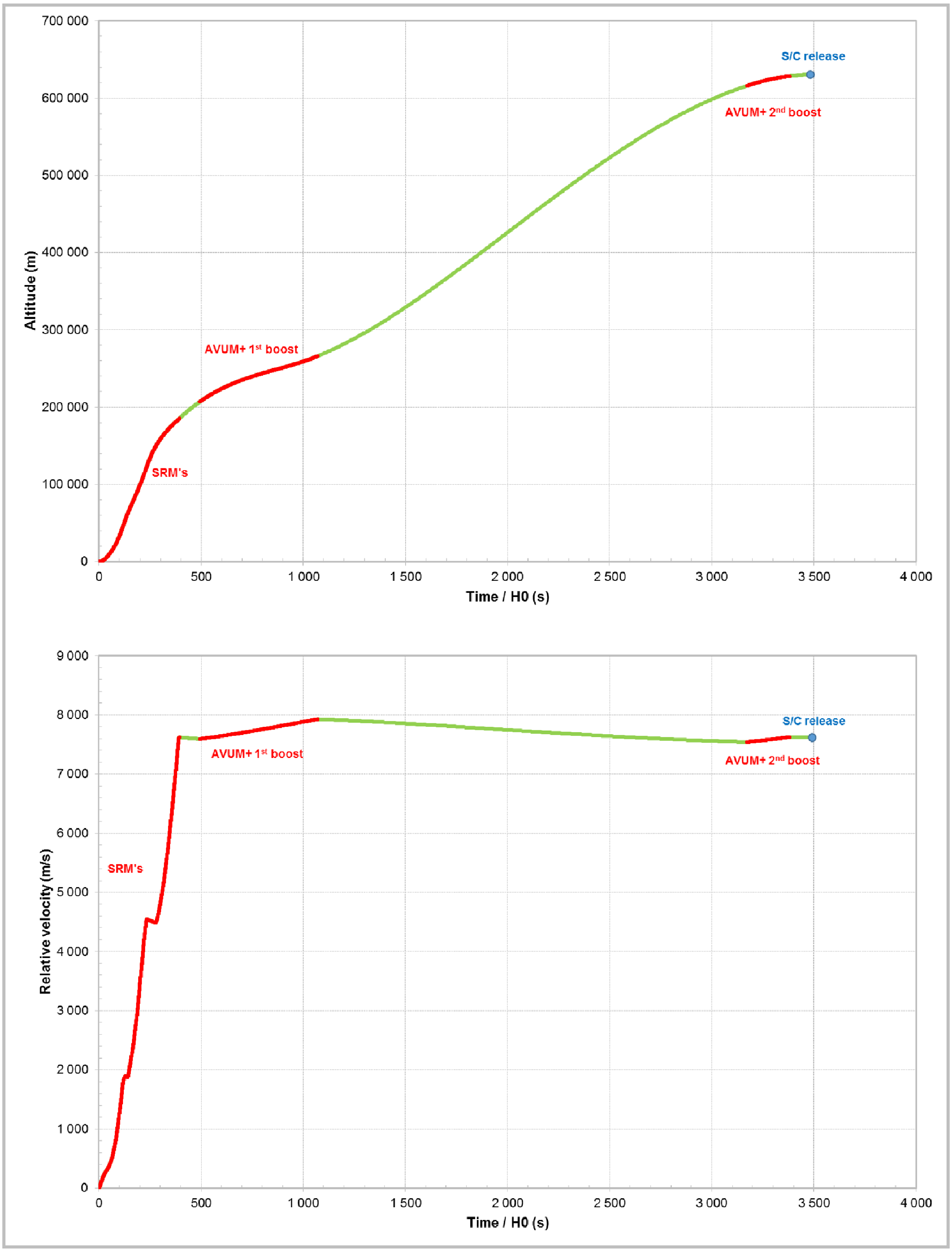


Figure 2.4.1.b – Typical altitude and relative velocity for a SSO single launch mission (two AVUM+ boosts mission profile)

The typical Vega C dual launch mission includes an ascent profile with four AVUM+ burns as follows:

- A 1st AVUM+ burn for transfer to the first intermediate orbit;
- A 2nd AVUM+ burn for orbit circularization;
- A 3rd AVUM+ burn for transfer to the second intermediate orbit;
- A 4th AVUM+ burn for orbit circularization;
- A 5th AVUM+ burn for orbit disposal maneuver or deorbitation (not illustrated in the figures here below).

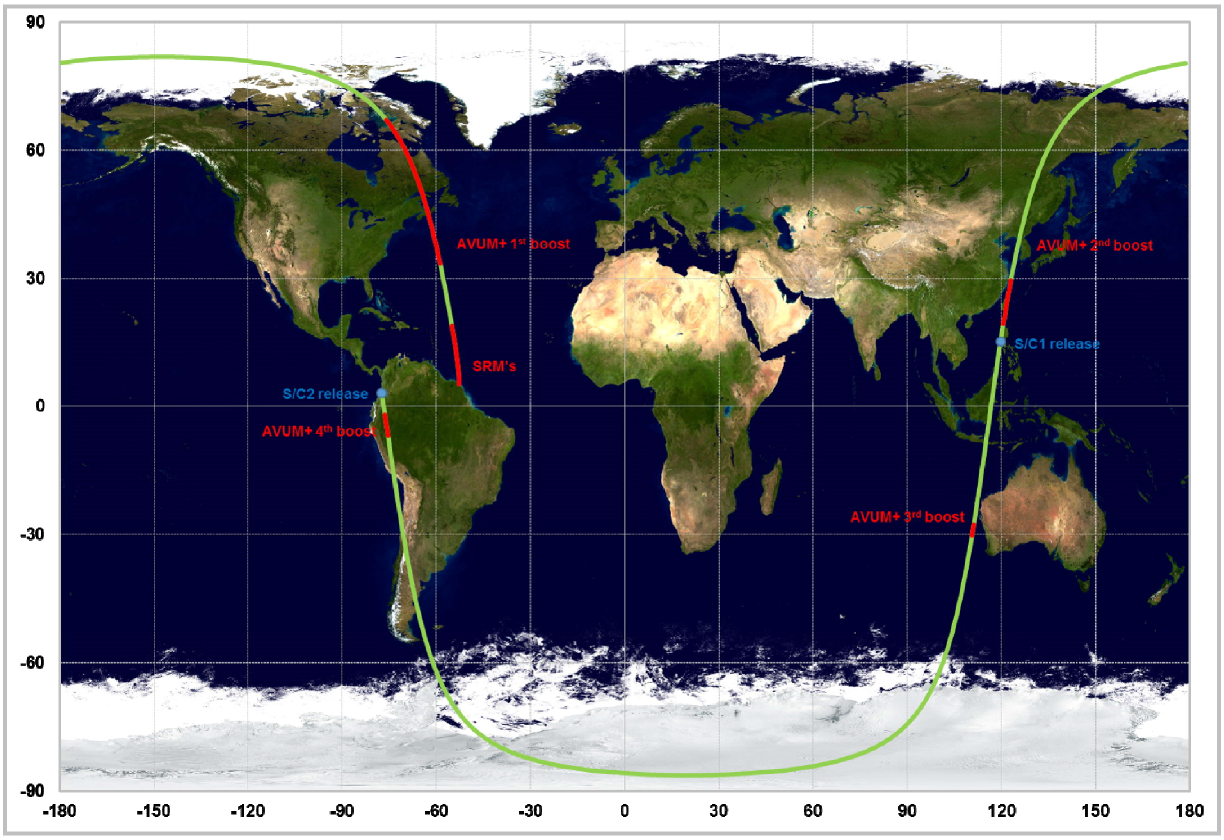


Figure 2.4.1.c – Typical ground path for the Vega C SSO dual launch mission (four AVUM+ boosts mission profile)

Typical evolutions of altitude and relative velocity from lift-off till spacecraft separation are presented in Figure 2.4.1.d:

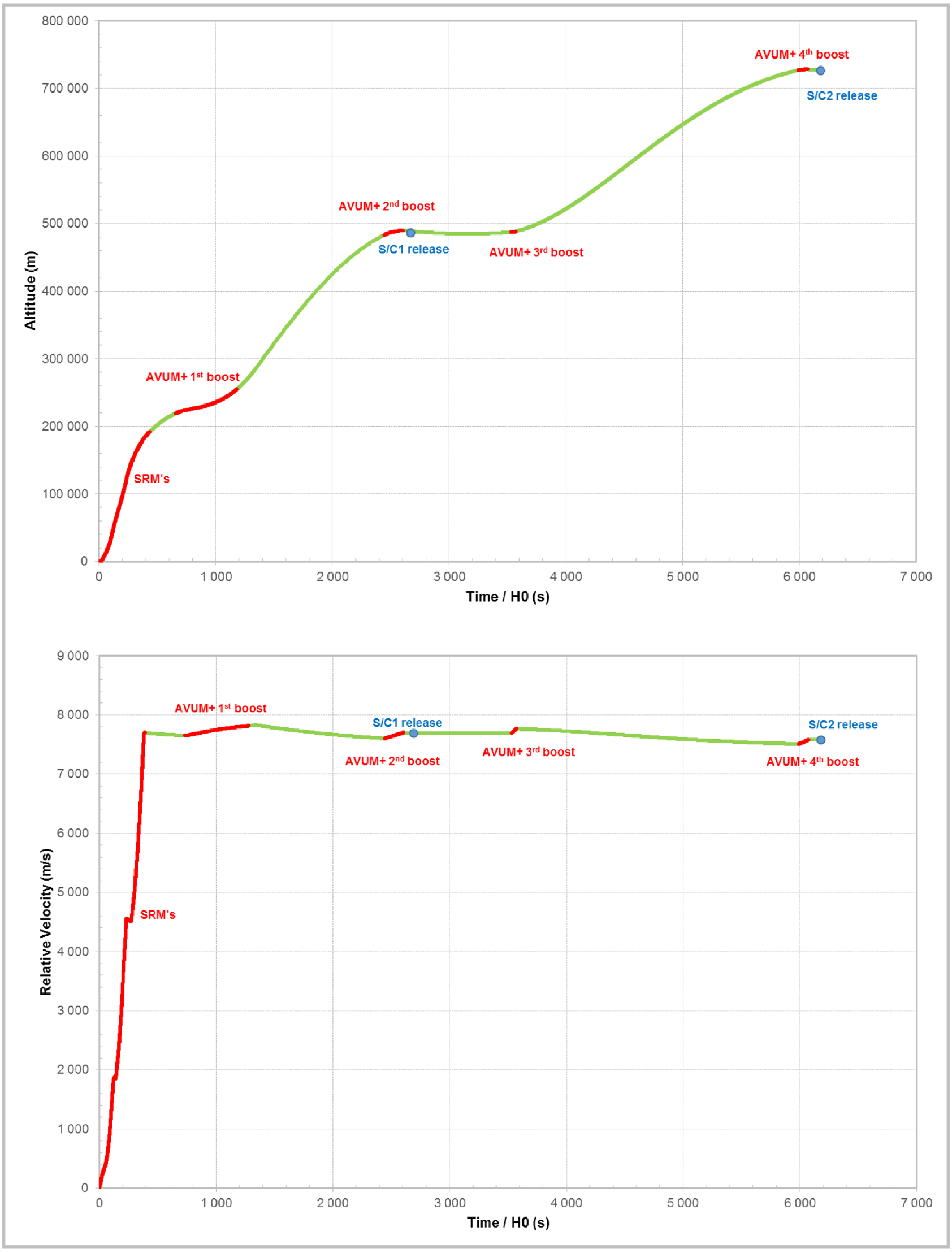


Figure 2.4.1.d – Typical altitude and relative velocity for a SSO dual launch mission (four AVUM+ boosts mission profile)

The Vega C launch vehicle performance data (including adapter) for sun-synchronous orbit (SSO) missions is presented in Figure 2.4.1.e as a function of altitude.

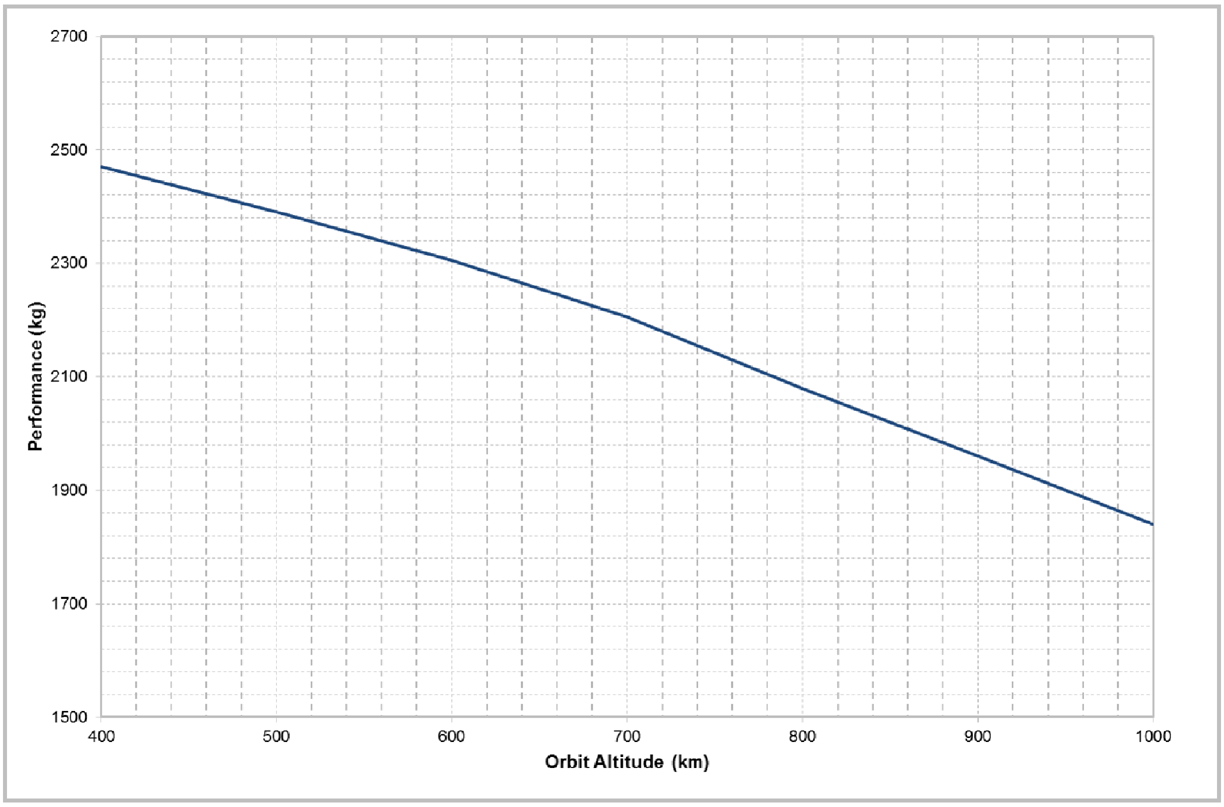


Figure 2.4.1.e – Performance for SSO orbits (two AVUM+ boosts mission)

2.4.2. Polar mission

The Vega C LV performance (including adapter) for polar orbit is presented in the Table 2.4.2.a here below.

Altitude (km)	Inclination (deg)	Performance (kg)
500	88	2 250

Table 2.4.2.a – Performance for polar orbit

2.4.3. Equatorial elliptical orbit missions

For equatorial elliptical orbits, the typical Vega C mission includes two AVUM+ burns.

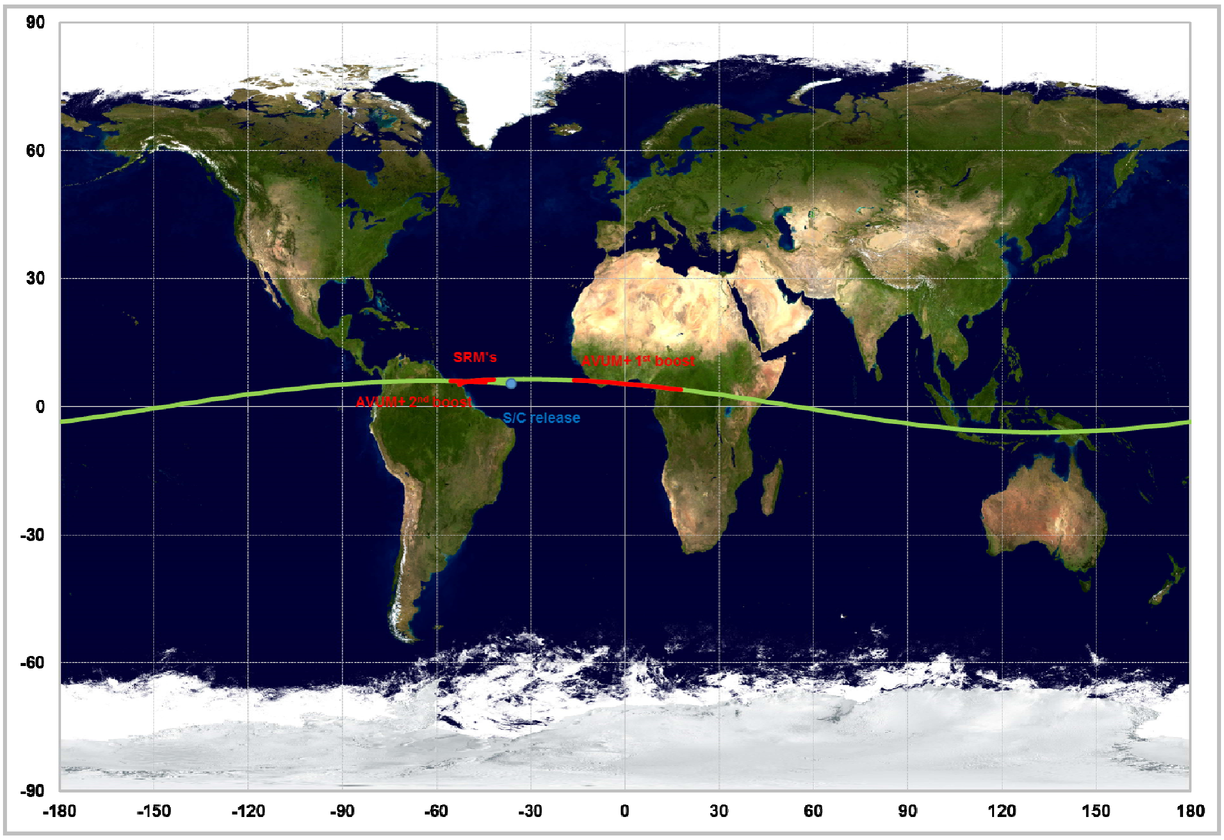


Figure 2.4.3.a – Typical ground path for the Vega C equatorial elliptical mission (two AVUM+ boosts mission profile)

Typical evolutions of altitude and relative velocity during the ascent are presented in Figure 2.4.3.b:

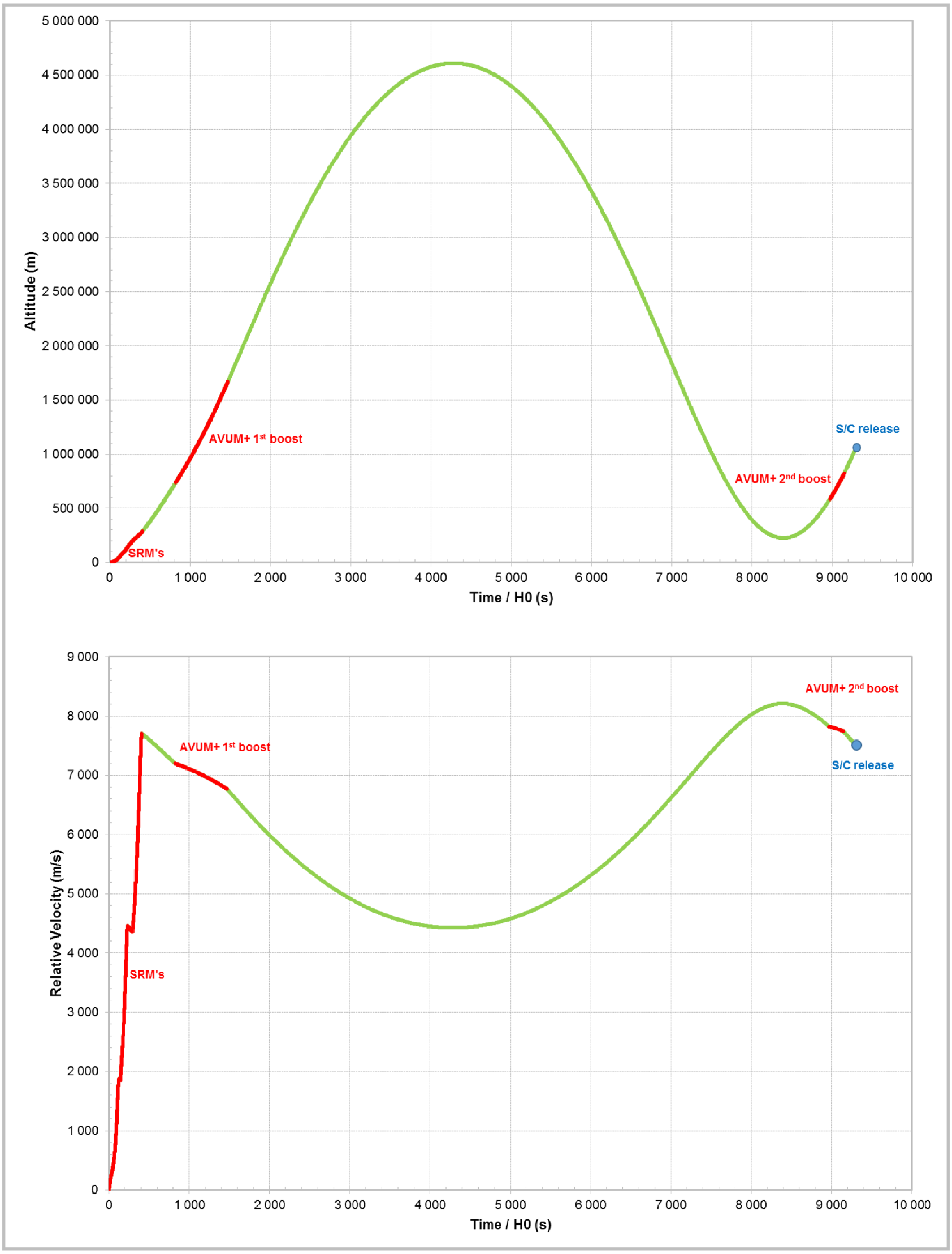


Figure 2.4.3.b – Typical altitude and relative velocity for the Vega C equatorial elliptical mission (two AVUM+ boosts mission profile)

LV performance data for typical equatorial elliptical mission:

Altitude of apogee,	Z_a	= 5 700 km
Altitude of perigee,	Z_p	= 250 km
Inclination,	i	= 6.0 deg

is 1 700 kg including adapter.

For other data, please contact Arianespace.

2.5. INJECTION ACCURACY

The accuracy is determined mainly by the performance of the AVUM+ navigation system. Conservative accuracy data depending on type of the mission are presented in Table 2.5.a. Mission-specific injection accuracy will be calculated as part of the mission analysis.

a Semi-major axis (km)	±15
e Eccentricity	±0.0012
i Inclination (deg)	±0.15
RAAN (deg)	±0.2

Table 2.5.a - Injection accuracy (± 3σ)

For other specific information, the Customer should contact Arianespace.

2.6. MISSION DURATION

Mission duration from lift-off until separation of the spacecraft on the final orbit depends on the specified orbital parameters and the ground station visibility conditions at spacecraft separation.

Typically, critical mission events including payload separation are carried out within the visibility of LV ground stations. This allows for the reception of near-real-time information on relevant flight events, orbital parameters on-board estimation and separation conditions.

The typical durations of various missions are presented in Table 2.6.a. Actual mission duration will be determined as part of the detailed mission analysis.

Mission (Altitude)	Ascent profile	Mission Duration (hh:mn)
SSO single launch	Ascent with coast phase	~ 01:00
SSO shared launch	Multiple AVUM+ burns	~ 01:00 (upper passenger) Up to ~ 02:00 (lower passenger or auxiliary passengers)
Equatorial elliptical mission	Ascent with coast phase	~ 02:35

Table 2.6.a - Typical mission duration (up to spacecraft separation)

2.7. LAUNCH WINDOW

The Vega C launch vehicle can be launched any day of the year, any time of the day respecting the specified lift-off time. The planned launch time is set with accuracy better than ±1 second, taking into account all potential dispersions in the launch sequencing and system start/ignition processes.

2.7.1. Launch window for single launch

For single launch, the launch window is defined taking into account the satellite mission requirements.

2.7.2. Launch window for multiple launch

For multiple launch, Arianespace will take into account the launch window requirements of each co-passenger to define a common launch time.

2.7.3. Process for launch window definition

The final launch window calculation will be based on actual orbit parameters.

The final launch window will be agreed upon by the Customer(s) and Arianespace at the Final Mission Analysis Review and no further modification shall be introduced without the agreement of each party.

2.8. SPACECRAFT ORIENTATION DURING ASCENT PHASE

During coast phases, the Roll and Attitude Control System (RACS) allows the AVUM+ to satisfy a variety of spacecraft requirements, including barbecue mode or sun-angle pointing constraints with an accuracy of ± 16°. On the contrary, during propulsive phases, the launch vehicle will determine the attitude position of spacecraft.

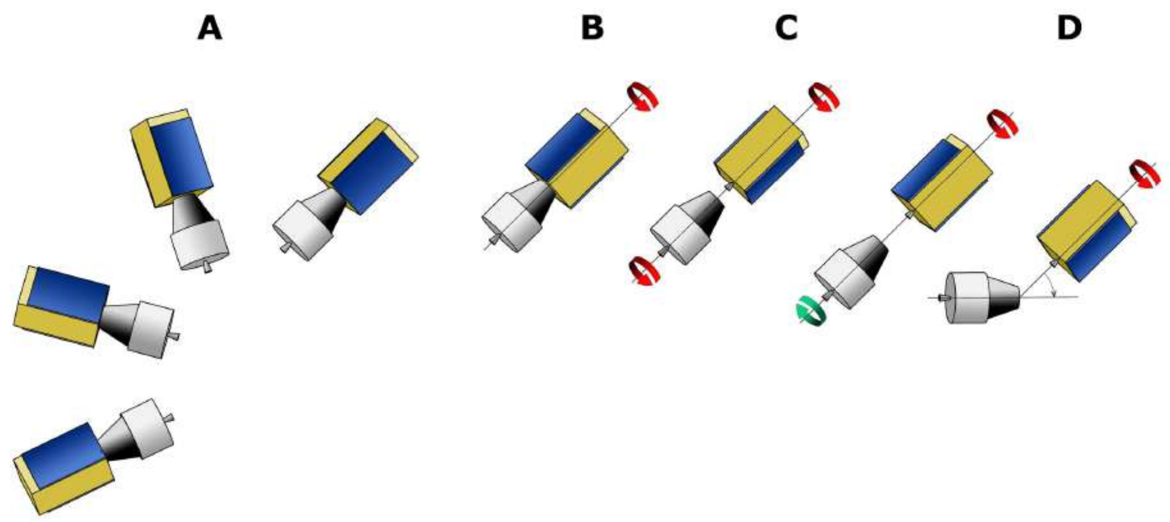
The best strategy to meet satellite and launch vehicle constraints will be defined with the Customer during the mission analysis process.

2.9. SEPARATION CONDIIONS

After injection into orbit, the AVUM+ Roll and Attitude Control System (RACS) is able to orient the upper composite to any desired attitude(s) and to perform separation(s) in various modes:

- 3-axis stabilization;
- Longitudinal spin.

Typical sequence of events is shown in Figure 2.9.a below.



A Orientation of composite (AVUM+ and payload) by Roll and Attitude Control System (RACS)
B Spin-up by RACS (if required by spacecraft)
C Spacecraft separation
D AVUM+ avoidance maneuver (spin down if necessary, attitude deviation by RACS and deorbitation)

Note: Spacecraft separation can also be accommodated under a 3-axis stabilized mode

Figure 2.9.a – Typical separation sequence for single launch

2.9.1. Orientation performance

The attitude at separation can be specified by the Customer in any direction in terms of:

- fixed orientation during the entire launch window;

or

- day variable orientation dependent on the sun position during the launch window.

For other specific satellite pointing, the Customer should contact Arianespace.

2.9.2. Separation mode and pointing accuracy

The actual pointing accuracy will result from the Mission Analysis (see Chapter 7 paragraph 7.4.2).

The following values cover Vega C compatible satellites as long as their balancing characteristics are in accordance with paragraph 4.2.3 (Chapter 4). They are given as satellite kinematic conditions at the end of separation and assume the adapter and separation system are supplied by Arianespace.

In case the adapter is provided by the spacecraft Authority, the Customer should contact Arianespace for launcher kinematic conditions just before separation.

Possible perturbations induced by spacecraft sloshing masses are not considered in the following values.

2.9.2.1 Three-axis stabilized mode

In case the maximum spacecraft static unbalance remains below 15 mm (see Chapter 4 paragraph 4.2.3), the typical 3σ pointing accuracies for a three-axis stabilized mode, and for 2.0 t class spacecraft, are without taking into account sloshing effect:

- Geometrical axis depointing ≤ 1.5 deg;
- Angular tip-off rates along longitudinal axis ≤ 1.75 deg/s;
- Angular tip-off rates along transversal axes ≤ 1.5 deg/s.

2.9.2.2 Spin stabilized mode

The AVUM+ RACS can provide a roll rate around the upper composite longitudinal axis up to 30 deg/s, clockwise or counterclockwise.

Although the spacecraft kinematic conditions just after separation are highly dependant on the actual spacecraft mass properties (including uncertainties) and the spin rate, the following values are typical results.

In case the maximum spacecraft static unbalance remains below 15 mm and its maximum dynamic unbalance remains below 1 deg (see Chapter 4 paragraph 4.2.3), the typical pointing accuracies for a 30 deg/s spin mode are:

- Spin rate accuracy ≤ 1.5 deg/s;
- Transverse angular tip-off rates ≤ 2 deg/s;
- Depointing of kinetic momentum vector, half angle ≤ 6 deg;
- Nutation, angle ≤ 5 deg.

2.9.2.3 Separation linear velocities and collision risk avoidance

The payload adapter's separation systems are designed to deliver a minimum relative velocity between spacecraft and upper stage of 0.5 m/s.

For each mission, Arianespace will verify that the distances between orbiting bodies are adequate to avoid any risk of collision.

For this analysis, the spacecraft is assumed to have a pure ballistic trajectory. Otherwise, in case some S/C maneuver occurs after separation, the Customer has to provide Arianespace with its orbit and attitude maneuver flight plan.

2.9.2.4 Multi-separation capabilities

The Vega C launch vehicle is also able to perform multiple separations with multiple launch carrying system (VESPA C or SSMS).

These structures are defined in Chapter 5.

In this case the kinematics conditions presented above will be defined through the dedicated separation analysis.

For more information, please contact Arianespace.

ENVIRONMENTAL CONDITIONS Chapter 3

3.1. INTRODUCTION

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

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-to-carrying structure (adapter, VESPA C, SSMS) interface.

For spacecraft fulfilling the design requirements specified in Chapter 4, the environmental conditions presented in the present chapter are applicable to:

- Single launch configuration, using an off-the-shelf adapter ("VAMPIRE 937" or "VAMPIRE 1194") as described in Annex 4a;
- Launch configuration with main spacecraft and auxiliary passenger(s), using a "VAMPIRE 937 with towers" or "SSMS" (Piggy-Back or Ride-Share configuration) multiple launch structure as described in Annex 4b, for spacecraft with a mass above 400 kg;
- Dual launch configuration, using a "VESPA C" dual launch structure as described in Annex 4c, for spacecraft with a mass above 400 kg.

For auxiliary passenger(s) with a mass below 400 kg, the environmental conditions are presented in the Auxiliary Passengers user's manual.

In case the adapter is provided by the spacecraft Authority and/or for other multiple launch configurations, the Customer should contact Arianespace.

3.2. GLOBAL MECHANICAL ENVIRONMENT

3.2.1. Quasi-static accelerations

During ground operations and flight, the spacecraft is subjected to static and dynamic loads. Such excitations may be of operational origin (e.g. transportation or mating), aerodynamic origin (e.g. wind and gusts or buffeting during transonic phase) or propulsion origin (e.g. longitudinal acceleration, thrust buildup or tail-off transients, or structure-propulsion coupling, etc.).

Figure 3.2.1.a illustrates a typical longitudinal static acceleration evolution overtime for the launch vehicle during its ascent flight. The highest longitudinal acceleration occurs just before the third-stage cutoff.

The highest lateral static acceleration occurs at maximum dynamic pressure and takes into account the effect of wind and gust encountered in this phase.

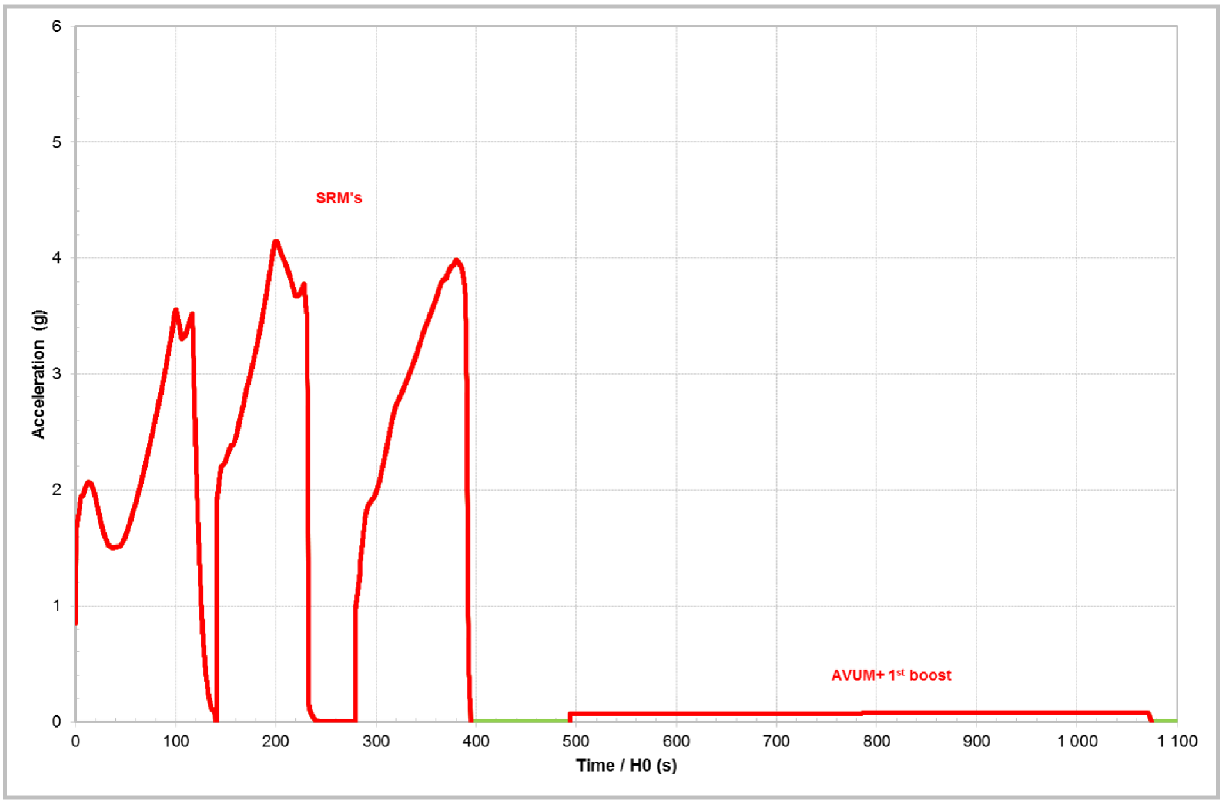


Figure 3.2.1.a – Typical longitudinal static acceleration for SSO mission

The associated loads at the spacecraft-to-carrying structure interface are defined by quasi-static loads (QSL), that apply at spacecraft center of gravity and that are the most severe combinations of dynamic and static accelerations that can be encountered by the spacecraft at any instant of the mission.

3.2.1.1. Single launch configuration

For a spacecraft in single launch configuration and complying with the stiffness requirements defined in Chapter 4 paragraph 4.2.3.4, the limit levels of quasi-static loads, to be taken into account for the design and dimensioning of the spacecraft primary structure, are defined in Table 3.2.1.1.a and illustrated in Figure 3.2.1.1.a.

Load event		QSL (g) (+ = tension ; - = compression)		
		Longitudinal		Lateral
		Compression	Tension	
1	Lift-off phase	-4.5	+3.0	±1.35
2	Flight with maximum dynamic pressure (Q_{max})	-4.0	+1.5	±0.9
3	1st stage flight with maximal acceleration and tail off	-5.0	+1.0	±0.7
4	2nd stage ignition and flight, 3rd stage ignition	-5.0	+3.0	±1.3
5	3rd stage maximal acceleration	$-\left(6.8-\frac{M^{(1)}}{1000}\right)-0.2$	N/A	±0.2
6	AVUM flight	-1.0	+0.5	±0.7

(1) M: mass [kg] of the spacecraft

Table 3.2.1.1.a – Design limit load factors for single launch configuration

Notes:

- The factors apply on spacecraft Center of Gravity.
- The 'minus' sign indicates compression along the longitudinal axis of the launch vehicle and the 'plus' sign tension.
- Lateral loads may act in any direction simultaneously with longitudinal loads.
- The gravity load is included.

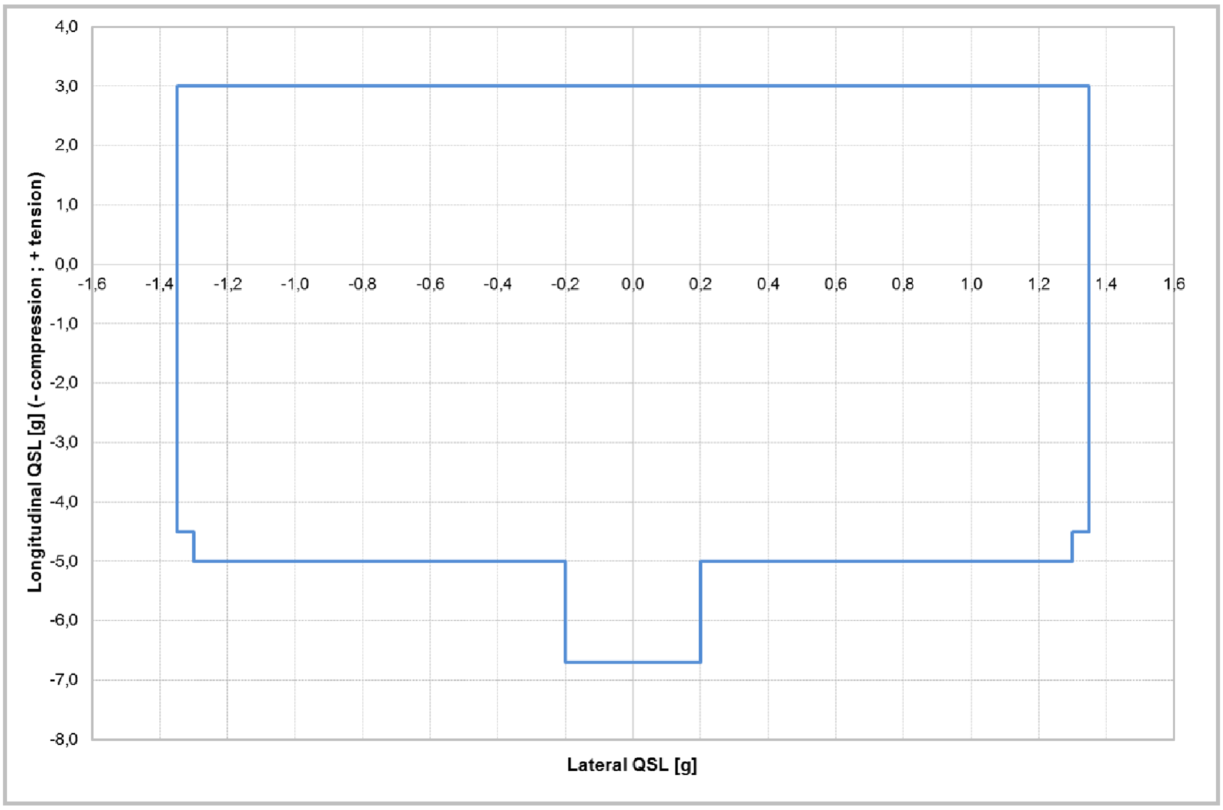


Figure 3.2.1.1.a – Spacecraft design limit load factors for single launch configuration

3.2.1.2. Launch configuration with main spacecraft and auxiliary passenger(s)

3.2.1.2.1. Using "VAMPIRE 937 with towers"

For a spacecraft integrated as main passenger on top of the "VAMPIRE 937 with towers", and complying with the stiffness requirements defined in Chapter 4 paragraph 4.2.3.4, the limit levels of quasi-static loads, to be taken into account for the design and dimensioning of the spacecraft primary structure, are defined in Chapter 3 paragraph 3.2.1.1.

3.2.1.2.2. Using "SSMS" multiple launch structure

To be issued later.

3.2.1.3. Dual launch configuration using "VESPA C"

For spacecraft with a mass above 400 kg and complying with the stiffness requirements defined in Chapter 4 paragraph 4.2.3.4, the limit levels of quasi-static loads, to be taken into account for the design and dimensioning of the spacecraft primary structure, are defined in Table 3.2.1.3.a and illustrated in Figure 3.2.1.3.a.

Load event		QSL (g) (+ = tension ; - = compression)		
		Longitudinal		Lateral
		Compression	Tension	
Upper spacecraft				
1	Max lateral	-5.5	+3.3	±2.5
2	Max compression / tension	-7.7	+3.3	±0.6
Lower spacecraft				
1	Max lateral	-5.5	+3.3	±2.0
2	Max compression / tension	-7.7	+3.3	±0.6

Notes:

- The factors apply on spacecraft Center of Gravity.
- The 'minus' sign indicates compression along the longitudinal axis of the launch vehicle and the 'plus' sign tension.
- Lateral loads may act in any direction simultaneously with longitudinal loads.
- The gravity load is included.

Table 3.2.1.3.a – Design limit load factors
for dual launch configuration and spacecraft mass above 400 kg

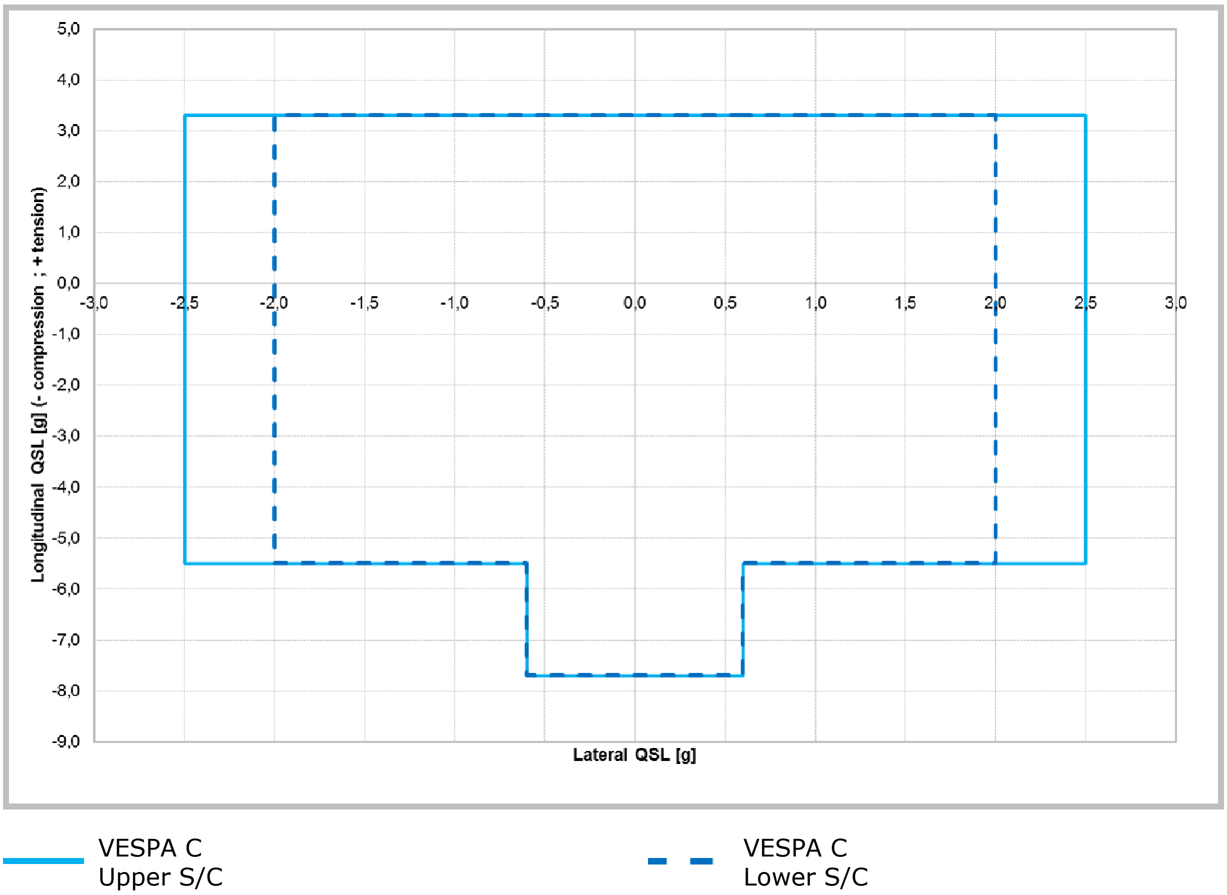


Figure 3.2.1.3.a – Spacecraft design limit load factors for upper and lower position in dual launch configuration and spacecraft mass above 400 kg

3.2.2. Line loads peaking

The geometrical discontinuities and differences in the local stiffness of the LV (stiffener, holes, stringers, etc.) and the non-uniform transmission of the LV's thrust at the spacecraft-to-carrying structure interface may produce local variations of the uniform line loads distribution.

The integral of these variations along the circumference is zero, and the line loads derived from the above QSL are not affected. The dimensioning of the lower part of the spacecraft shall however account for these variations which have to be added uniformly at the spacecraft-to-carrying structure interface to the mechanical line loads obtained for the various flight events.

Such local over line loads are specific of the carrying structure adapter. For off-the-shelf adapters, a value of 10% over the average line loads seen by the spacecraft is to be taken into account, with a minimum of 5 N/mm.

3.2.3. Handling loads during ground operations

During the encapsulation phase, the spacecraft is lifted and handled with its adapter: for this reason, the spacecraft and its handling equipment must be capable of supporting an additional mass of maximum 180 kg.

The crane characteristics, velocity and acceleration are defined in the EPCU User's Manual.

3.2.4. Sine-equivalent dynamics

Sinusoidal excitations affect the launch vehicle during its powered flight (mainly the atmospheric flight), as well as during some of the transient phases.

3.2.4.1. Single launch configuration

For a spacecraft in single launch configuration and complying with the stiffness requirements defined in Chapter 4 paragraph 4.2.3.4, the limit levels of sine-equivalent vibrations at spacecraft-to-adapter interface, to be taken into account for the design and dimensioning of the spacecraft, are defined in Table 3.2.4.1.a and illustrated in Figure 3.2.4.1.a.

Longitudinal direction			
Frequency Band (Hz)	1 – 5	5 – 35	35 – 110
Sine Amplitude (g)	0.4	0.8	1.0
Lateral direction			
Frequency Band (Hz)	1 – 5	5 – 30	30 – 110
Sine Amplitude (g)	0.4	0.8	0.5

Table 3.2.4.1.a – Sine-equivalent vibrations at spacecraft interface for single launch configuration

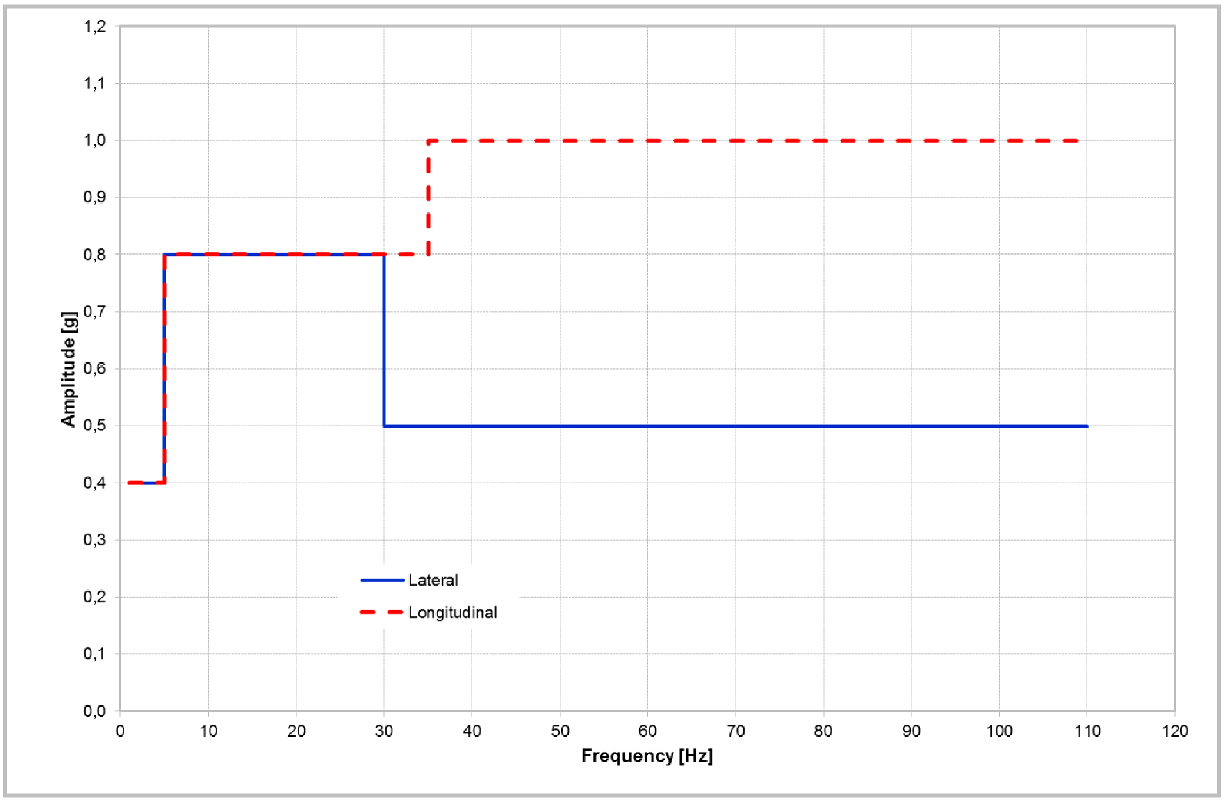


Figure 3.2.4.1.a – Sine-equivalent vibrations at spacecraft interface for single launch configuration

3.2.4.2. Launch configuration with main spacecraft and auxiliary passenger(s)

3.2.4.2.1. Using "VAMPIRE 937 with towers"

For a spacecraft integrated as main passenger on top of the VAMPIRE 937 with towers, and complying with the stiffness requirements defined in Chapter 4 paragraph 4.2.3.4, the limit levels of sine-equivalent vibrations at spacecraft-to-adapter interface, to be taken into account for the design and dimensioning of the spacecraft, are defined in Chapter 3 paragraph 3.2.4.1.

3.2.4.2.2. Using "SSMS" multiple launch structure

To be issued later.

3.2.4.3. Dual launch configuration using "VESPA C"

For spacecraft with a mass above 400 kg and complying with the stiffness requirements defined in Chapter 4 paragraph 4.2.3.4, the limit levels of sine-equivalent vibrations at spacecraft-to-VESPA interface, to be taken into account for the design and dimensioning of the spacecraft, are given in Table 3.2.4.3.a and illustrated in Figure 3.2.4.3.a.

Upper spacecraft			
Longitudinal direction			
Frequency Band (Hz)	0 – 5	5 – 35	35 – 110
Sine Amplitude (g)	0.4	0.8	1.0
Lateral direction			
Frequency Band (Hz)	0 – 5	5 – 30	35 – 110
Sine Amplitude (g)	0.4	0.8	0.5
Lower spacecraft			
Longitudinal direction			
Frequency Band (Hz)	0 – 5	5 – 35	35 – 110
Sine Amplitude (g)	0.4	0.8	1.0
Lateral direction			
Frequency Band (Hz)	0 – 5	5 – 30	35 – 110
Sine Amplitude (g)	0.4	0.8	0.5

Table 3.2.4.3.a – Sine-equivalent vibrations at spacecraft interface for dual launch configuration and spacecraft mass above 400 kg

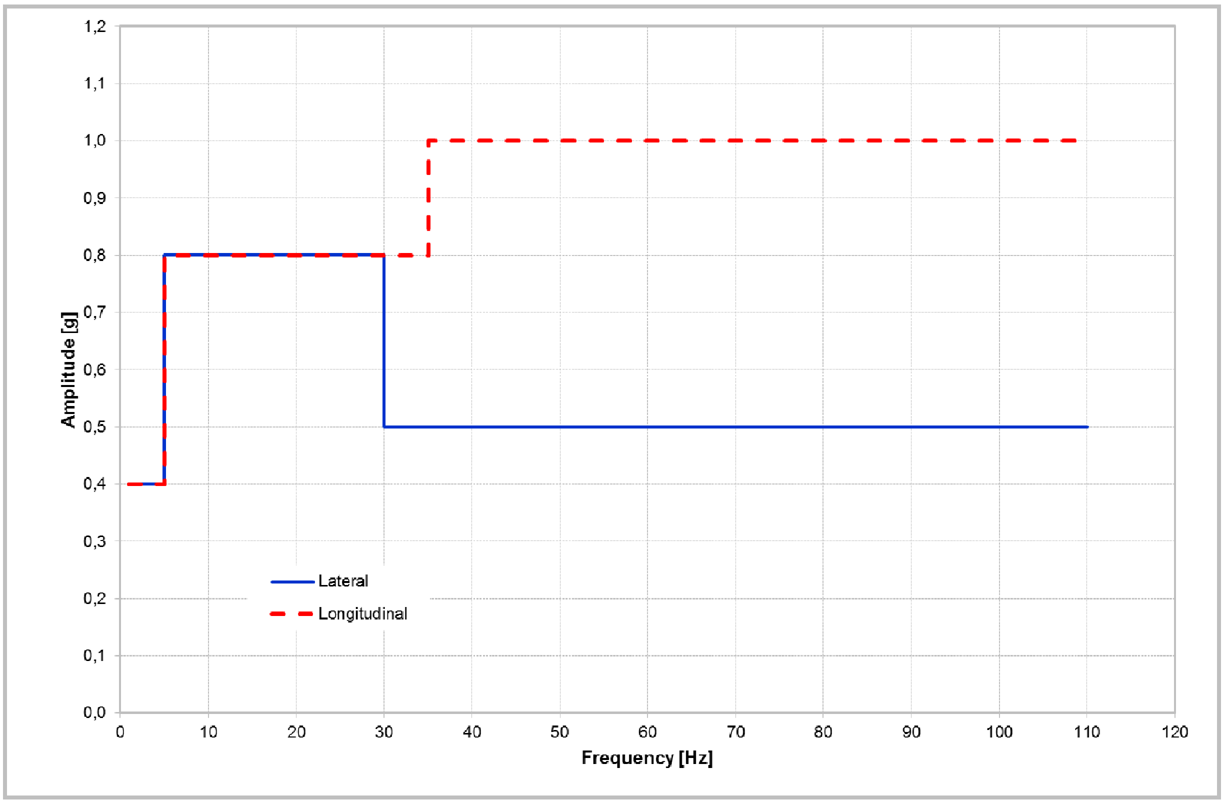


Figure 3.2.4.3.a – Sine-equivalent vibrations at spacecraft interface for upper and lower position in dual launch configuration and spacecraft mass above 400 kg

3.2.5. Random vibrations

For payload above 400 kg, the broadband vibrations are covered by the acoustic environment.

3.2.6. Acoustic vibrations

3.2.6.1. On Ground

On ground, acoustic pressure fluctuations under the fairing are generated by the venting system (refer to paragraph 3.4.2). The noise level generated in the vicinity of the venting system does not exceed 94 dB.

3.2.6.2. In Flight

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

The envelope spectrum of the noise induced inside the fairing during flight is shown in Table 3.2.6.2.a and in Figure 3.2.6.2.a.

It is assessed that the sound field under the fairing is diffuse.

Octave center frequency (Hz)	**Flight limit level (dB)** (reference: 0 dB = 2 x 10^{-5} Pa)	
	Lift-off (H0 → H0+3s)	Atmospheric phase (from H0+3s)
31.5	112	110
63	123	120
125	126	122
250	136	127
500	139	130
1000	127	127
2000	122	118
OASPL(1) (20 – 2828 Hz)	141.2	133.7
Duration	3 seconds	55 seconds

(1) *OASPL: Overall Acoustic Sound Pressure Level*

Table 3.2.6.2.a – Acoustic noise spectrum under the fairing in flight

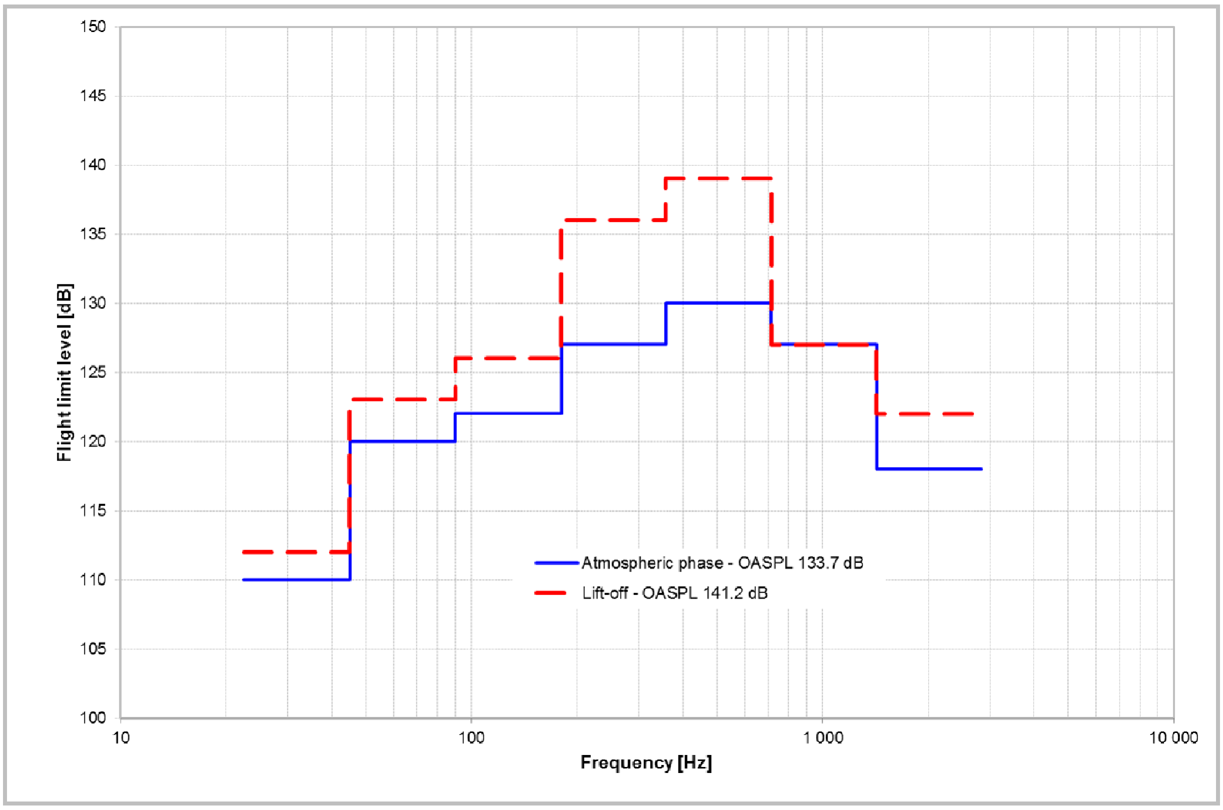


Figure 3.2.6.2.a – Acoustic noise spectrum under the fairing

3.2.7. Shocks

The spacecraft is subject to shock primarily during stages 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 presented in Tables 3.2.7.a & 3.2.7.b and Figure 3.2.7.a. These levels are applied simultaneously in axial and radial directions.

For Customers wishing to use their own adapter the acceptable envelope at the launch vehicle interface will be provided on a case-by-case basis.

Flight event	Frequency (Hz)	
	100 – 1 600	1 600 – 10 000
	SRS, Shock Response Spectra (Q = 10) (g)	
Fairing & stages separations	30 – 2 000	2 000

Table 3.2.7.a – Shock response spectrum for stages and fairing separations

Spacecraft adapter interface diameter	Frequency (Hz)	
	100 – 2 000	2 000 – 10 000
	SRS, Shock Response Spectra (Q = 10) (g)	
Ø 937, Ø 1194	20 – 2 000	2 000

Table 3.2.7.b – Envelope shock response spectrum for clamp-band release

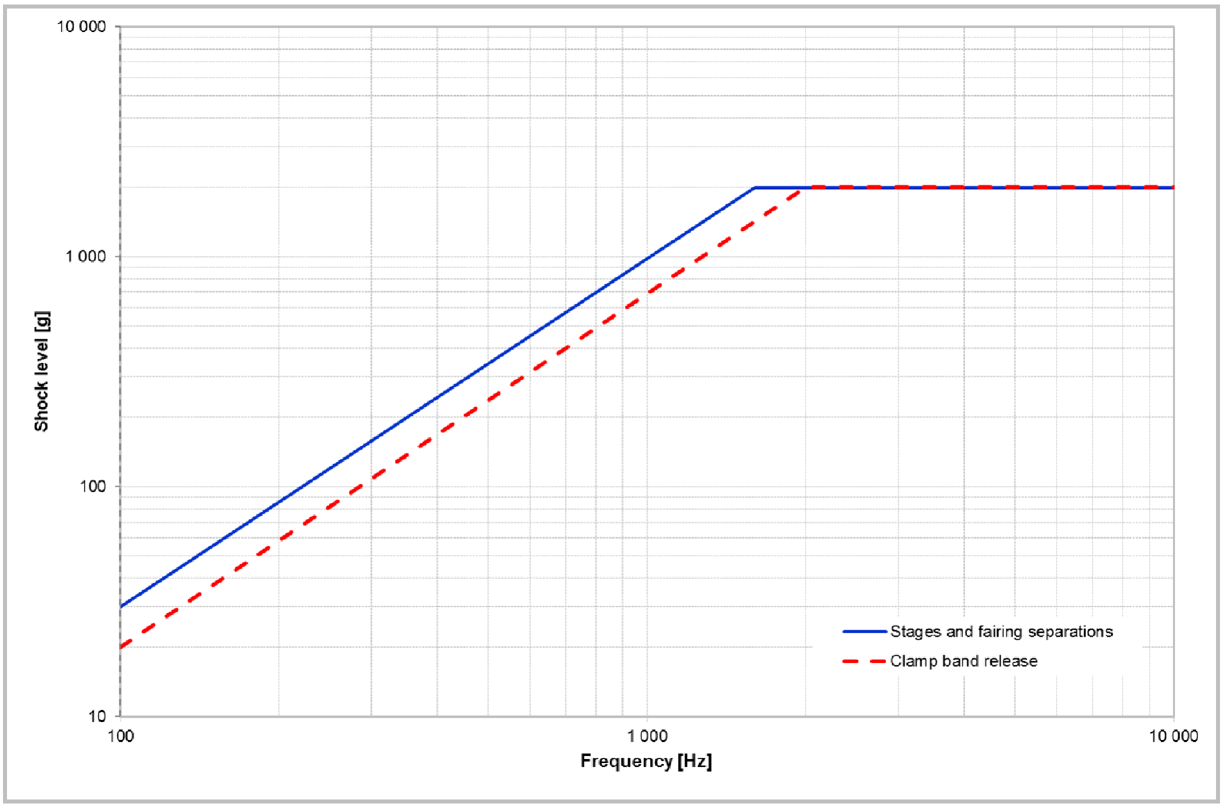


Figure 3.2.7.a – Envelope shock response spectra (SRS) for stages/fairing separations and clamp-band release at the spacecraft base (Q=10)

3.2.8. Static pressure under the fairing

3.2.8.1. On ground

After encapsulation, the average air velocity around the spacecraft due to the ventilation system is lower than 3.5 m/s. Locally, depending on spacecraft geometry, in close vicinity of fairing air inlets and outlets, this air velocity may be exceeded. In case of specific concern, please contact Arianespace.

3.2.8.2. In flight

The payload compartment is vented during the ascent phase through one-way vent doors insuring a low depressurization rate of the fairing compartment.

In flight, the depressurization rate inside the fairing can reach a maximum value of 35 mbar/s for a duration of 5 seconds. For the rest of the flight, the depressurization rate is ≤ 25 mbar/s.

3.3. LOCAL LOADS

The local loads which shall be considered for spacecraft sizing, on top of the global loads described in paragraph 3.2, are the followings:

- Payload adapter separation spring forces;
- Spacecraft umbilical connectors spring forces;
- Flatness effect at spacecraft-to-adapter interface;
- Pre-tension loads associated to the tightening of spacecraft-to-adapter separation subsystem;
- Thermo-elastic loads if applicable.

They will be specified in the Interface Control Document (DCI).

3.4. THERMAL ENVIRONMENT

3.4.1. Introduction

The thermal environment provided during spacecraft preparation and launch has to be considered during 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.

3.4.2. 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.

3.4.2.1. CSG facility environments

The typical thermal environment within the air-conditioned CSG facilities is kept around 23°C ± 2°C for temperature and 50% ± 10% for relative humidity.

More detailed values for each specific hall and buildings are presented in Chapter 6 and in the EPCU User's Manual.

3.4.2.2. Thermal conditions under the fairing

During the encapsulation phase and once mated on the launch vehicle, the spacecraft is protected by an air-conditioning system provided by the ventilation through pneumatic umbilical (see Figure 3.4.2.2.a).

Phase		Air conditioning system	Temperature [°C]	Relative humidity [%]	Air flow rate [Nm^3/h]	Duration
Launch preparation nominal sequence						
01	Transfer between EPCU building (CCU)	-	24 ±3°C	10% - 60 %	-	-
02	Operation in EPCU	EPCU air conditioning system	23 ±2°C	40% - 60 %	-	2 weeks max.
03	Payload Assembly Composite (PAC) transfer from EPCU to SLV	PFRCS air conditioning system	16 ±1°C	< 60 %	Up to 1 500 Nm^3/h ±10%	≈ 3h (D-8)
04	PAC hoisting to PFCU and positioning on AVUM spacers	Low flow rate to maintain a positive delta pressure under fairing	Ambient temperature	< 130 ppm	50 Nm^3/h ±10%	≈ 5h (D-8)
05	PAC stand-by on AVUM spacers and final mating on AVUM	Launch pad air conditioning system	13 < T° < 22°C ±1°C (1)	< 50 % (DP < 10°C)	500 … 2 000 Nm^3/h ±10%	≈ 5 days (D-8 → D-3)
06	PAC ventilation setup for launch	Low flow rate to maintain a positive delta pressure under fairing	Ambient temperature	DP < 10°C	Few mbars	≈ 4h (D-3)
07	Integrated launch vehicle stand-by and launch preparation	Launch pad air conditioning system	13 < T° < 22°C ±1°C (1)	< 50 % (DP < 10°C)	500 … 2 000 Nm^3/h ±10%	≈ 3 days (D-3 → H0)
Reported launch sequence						
08	Integrated launch vehicle stand-by	Launch pad air conditioning system	13 < T° < 22°C ±1°C (1)	< 50 % (DP < 10°C)	500 … 2 000 Nm^3/h ±10%	≈ 1 day

Notes:

(1) The ventilation temperature will be agreed on a case-by-case basis in order to fulfill the spacecraft heat dissipation.

The ventilation characteristics and settings will be such that no condensation shall occur inside the fairing cavity at any time during launch preparation.

The mobile gantry is removed at ≈ H0-3h and re-installed around the launch vehicle in case of reported launch at ≈ H0+1h30.

Table 3.4.2.2.a - Air conditioning under the fairing

To be issued

Figure 3.4.2.2.a – Configuration of the launch pad air-conditioning system

3.4.3. Thermal flight environment

3.4.3.1. Thermal conditions before fairing jettisoning

The average value of the thermal flux density radiated by the fairing during the ascent phase does not exceed 1 000 W/m^2 in the hottest area. A maximal value of 1 300 W/m^2 can be reached during a transient phase.

These figures do not take into account any effect induced by the spacecraft dissipated power.

3.4.3.2. Aerothermal flux and thermal conditions after fairing jettisoning

The nominal time for jettisoning the fairing is determined in order not to exceed a maximum instantaneous flux of 1 135 W/m^2 at 99%. This flux is calculated as a free molecular flow acting on a plane surface perpendicular to the velocity direction ($\frac{1}{2} \rho V^3$).

For dedicated launches, lower or higher flux exposures can be accommodated on request, as long as the necessary performance is maintained.

Solar radiation, 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.

In case of ascent profile with coast phase at daylight, the AVUM+ upper stage can spin the upper composite up to 2°/s in order to reduce the heat flux.

3.4.3.3. Other thermal fluxes

No thermal flux coming from 1st and 2nd stages need to be considered.

After fairing jettisoning, the maximal thermal impingement on the payload external surface due to the 3rd stage (Z9) engine firing is:

- lower than 1 500 W/m^2 on the payload plane "A" (see Figure 3.4.3.3.a) perpendicular to the launch vehicle longitudinal axis;
- lower than 600 W/m^2 on the payload surfaces "B" parallel to the launch vehicle longitudinal axis.

The maximum application time is 115 seconds.

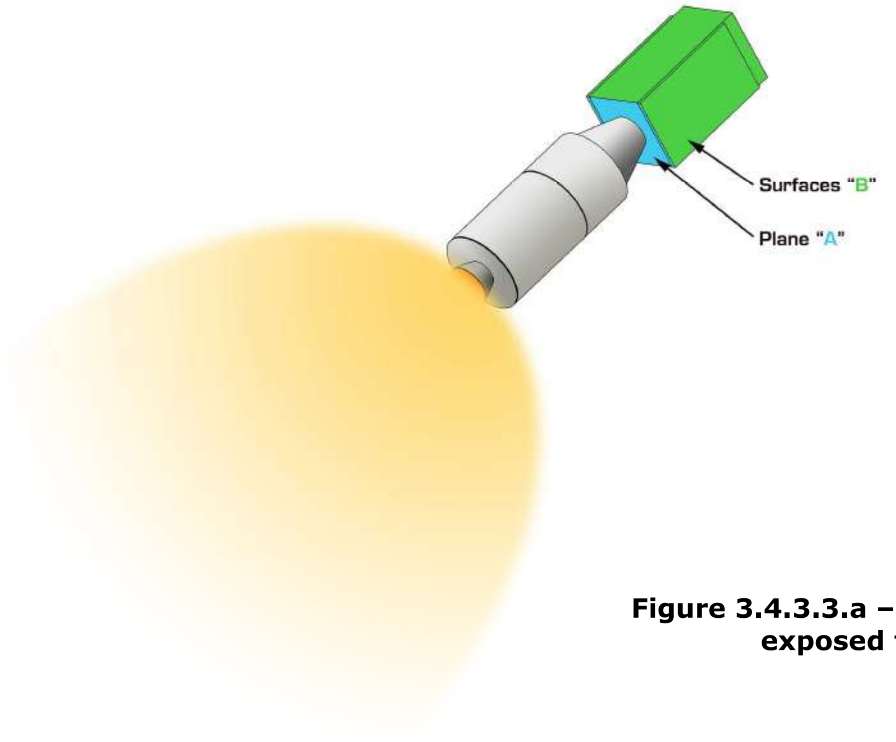


Figure 3.4.3.3.a – Spacecraft surfaces exposed to the 3rd stage (Z9) plume radiation

3.5. CLEANLINESS AND CONTAMINATION

3.5.1. Cleanliness

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 EPCU buildings (PPF, HPF and UCIF) in controlled Class 100,000 (or ISO 8) clean rooms. During transfer between buildings the spacecraft is transported in payload containers (CCU) with the cleanliness Class 100,000 (or ISO 8). 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 stage and fairing are verified based on the Visibly Clean Level 2 criteria, and cleaned if necessary.
- Once encapsulated, during transfer, hoisting or standby on the launch pad, the upper composite will be hermetically closed and an 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.

The cleanliness conditions are summarized in the Table 3.5.1.a, below:

S/C location	**Transfer between EPCU buildings**	**S/C in EPCU**		**Transfer between EPCU and SLV**		**S/C on LV**
	In CCU container	Not Encapsulated	Encapsulated (in EPCU)	Transfer on launch pad [(1)]	Hoisting [(1)]	Launch preparation [(1)]
Cleanliness class	ISO 8 (100,000)	ISO 8 (100,000)	ISO 7 (10,000)	ISO 7 (10,000)	ISO 7 (10,000)	ISO 7 (10,000)
Duration	~1 h 30	Several days	~1 day	~3 h	~5 h	8 days

(1) With the following filtration of air-conditioning system: standard HEPA H14 (DOP 0.3 µm).

Table 3.5.1.a - Cleanliness

3.5.2. Contamination

The organic and particle contaminations in facilities and under the fairing are controlled by contamination witness.

Plates are set up inside the buildings and inside the fairing from encapsulation until D-1. The LV systems are designed to preclude in-flight contamination of the spacecraft.

3.5.2.1. Particle contamination

Deposited particle contamination in the clean rooms

In accordance with ECSS-Q-ST-70-01C, the ISO 8 cleanliness level is equivalent to a deposited particle contamination of 1 925 ppm/week.

Deposited particle contamination on launcher items

Launcher equipment in the vicinity of a satellite will be cleaned in case the deposited particles contamination exceeds 4 000 ppm.

Prior to the encapsulation of the spacecraft, the cleanliness of the AVUM+ upper stage and the fairing is verified based on the Visibly Clean Level 2 criteria, and cleaned if necessary.

3.5.2.2. Organic contamination

Deposited Organic contamination in the clean rooms

The clean rooms and the surrounding environment of a satellite shall not generate deposited organic contamination exceeding 0.5 mg/m^2/week.

Deposited organic contamination on launcher items

Launcher equipments in the vicinity of a satellite will be cleaned in case deposited organic contamination exceeds 2 mg/m^2.

Deposited organic contamination from encapsulation to S/C separation

The maximum organic non-volatile deposit on satellite surfaces is lower than 4 mg/m^2 from encapsulation and until 4h00 after satellite separation, taking into account a maximum of 2 mg/m^2 due to out-gassing launcher materials and 2 mg/m^2 due to functioning of LV systems.

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

3.6. ELECTROMAGNETIC ENVIRONMENT

The LV and launch range RF systems and electronic equipments are generating electromagnetic fields that may interfere with satellite equipment and RF systems. The electromagnetic environment depends from the characteristics of the emitters and the configuration of their antennae.

3.6.1. LV and range RF systems

Launcher

The launch vehicle is equipped with the following transmission and reception systems:

- A telemetry system comprising one transmitter, coupled with one left-handed antenna and one right-handed antenna having an omnidirectional radiation pattern. This transmitter is located in the AVUM+ avionic module with its antennae fitted in the external section of the AVUM+ upper stage. The transmission frequency is in the 2 200 – 2 290 MHz band, and the transmitter power is 10 W. Allocated frequencies to the launch vehicle are 2 206.5, 2 218, 2 227, 2 249, 2 254.5, 2 267.5 and 2 284 MHz.
- A telecommand-destruct reception system, comprising two receivers operating in the 440 – 460 MHz band. Each receiver is coupled with a system of two antennae, located on the Z9 / AVUM+ interstage, having an omnidirectional pattern and no special polarization.
- A radar transponder system, comprising two identical transponders with a reception frequency of 5 690 MHz and transmission frequencies in the 5 400 – 5 900 MHz band. The minimum pulsed (0.8 μs) transmitting power of each transponder is 400 W peak. Each transponder is coupled with a system of two antennae, located on the Z9 / AVUM+ interstage, with an omnidirectional pattern and clockwise circular polarization.

Range

The ground radars, local communication network and other RF means generate an electromagnetic environment at the preparation facilities and launch pad, and together with LV emission constitute an integrated electromagnetic environment applied to the spacecraft. The EM data are based on the periodical EM site survey conducted at CSG.

3.6.2. The electromagnetic field

The intensity of the electrical field generated by spurious or intentional emissions from the launch vehicle and the range RF systems does not exceed the levels given in Figure 3.6.2.a. These levels are applicable for the complete cavity inside the fairing.

Actual levels will be the same or lower taking into account the attenuation effects due to the carrying structure configuration, or due to worst case assumptions taken into account in the computation.

Actual spacecraft compatibility with these emissions will be assessed during the preliminary and final EMC analyses.

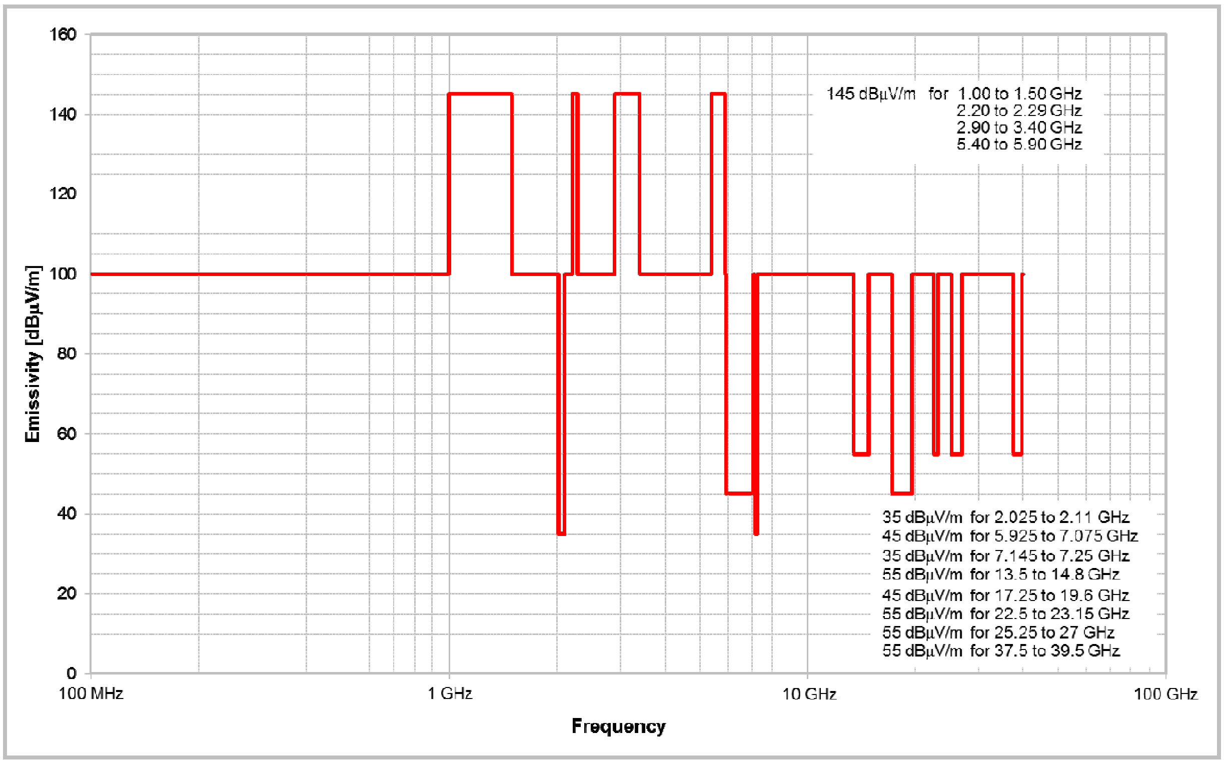


Figure 3.6.2.a – Spurious radiation and intentional emissions by launch vehicle and launch base – Narrow-band electrical field

3.7. ENVIRONMENT VERIFICATION

The Vega telemetry systems capture the low and high frequency data during the flight from the sensors installed on the fairing, upper stage and adapter and then relay these data to ground stations. These measurements are recorded and processed during post-launch analysis to derive the actual environment to which the spacecraft was submitted to during the launch. A synthesis of the results is provided to the Customer.

SPACECRAFT DESIGN AND VERIFICATION REQUIREMENTS — Chapter 4

4.1. INTRODUCTION

The design and dimensioning data and the verification requirements that shall be taken into account by any Customer intending to launch a spacecraft compatible with the Vega C launch vehicle are detailed in this chapter.

In case the adapter is provided by the spacecraft Authority, the Customer should contact Arianespace.

4.2. DESIGN REQUIREMENTS

4.2.1. Safety requirements

The Customer is required to design the spacecraft in conformity with the Payload Safety Handbook (CSG-NT-SBU-16687-CNES, Issue 1 Revision 1, May 2015).

4.2.2. Selection of spacecraft materials

The spacecraft materials must satisfy the following outgassing criteria:

- Recovered Mass Loss (RML) ≤ 1%;
- Collected Volatile Condensable Material (CVCM) ≤ 0.1%;

measured in accordance with the procedure ECSS-Q-ST-70-02C.

4.2.3. Spacecraft properties

4.2.3.1. Payload mass and CoG. limits

The spacecraft mass and C.o.G. position shall comply with a limitation for static moment applied on the spacecraft / adapter interface.

4.2.3.1.1. Single launch configuration

For a spacecraft in single launch configuration, the mass and CoG limits are presented in Figure 4.2.3.1.1.a in case of "VAMPIRE 937" and "VAMPIRE 1194".

For spacecraft with characteristics outside this domain, please contact Arianespace.

The use of the adapters PLA 937 VG and PLA 1194 VG on Vega C shall be evaluated on a case by case basis.

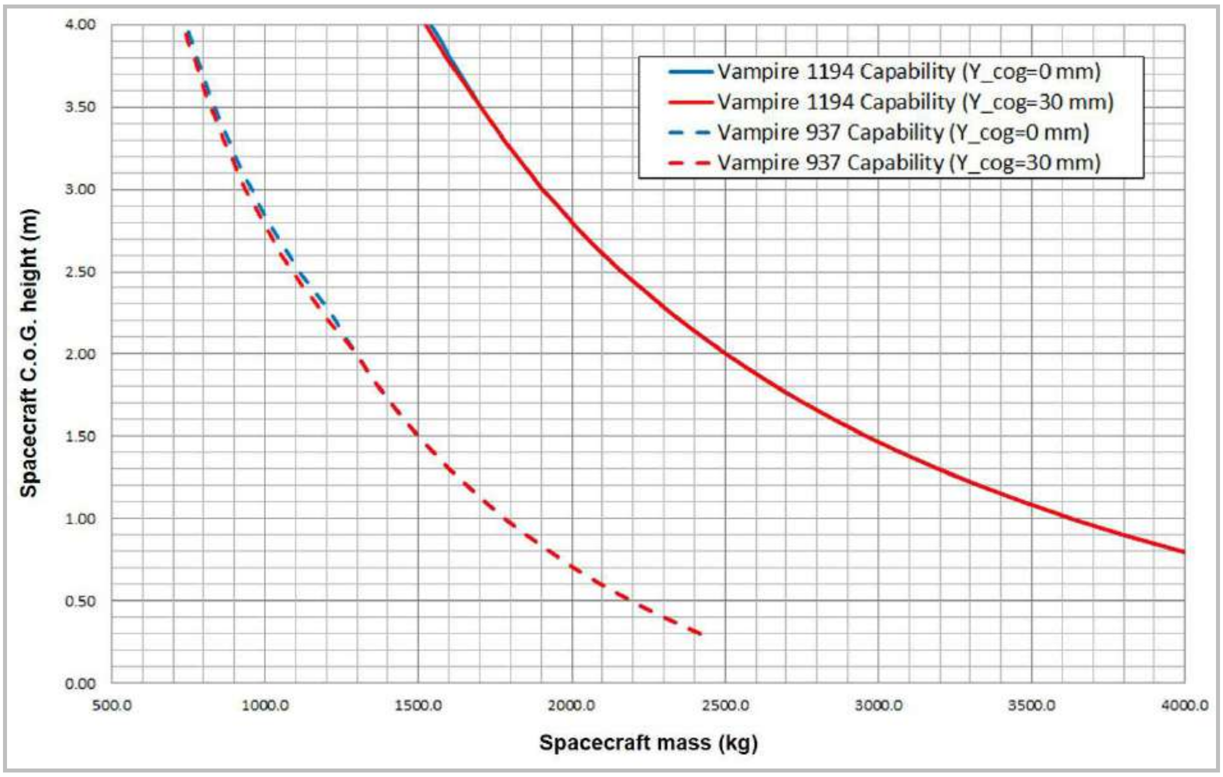


Figure 4.2.3.1.1.a – Limits of spacecraft C.o.G. position vs. spacecraft mass for single launch configuration in case of "VAMPIRE 937" or "VAMPIRE 1194"

4.2.3.1.2. Launch configuration with main spacecraft and auxiliary passenger(s)

4.2.3.1.2.1. Using "VAMPIRE 937 with towers"

For a spacecraft integrated as main passenger on top of the "VAMPIRE 937 with towers", the mass and CoG limits are defined in Chapter 4 paragraph 4.2.3.1.1.

For spacecraft with characteristics outside this domain, please contact Arianespace.

4.2.3.1.2.2. Using "SSMS" multiple launch structure

To be issued later.

4.2.3.1.3. Dual launch configuration using "VESPA C"

For dual launch configuration, the mass and CoG key features are defined in the Table 4.2.3.1.3.a.

	Upper spacecraft	Lower spacecraft
VESPA C	Max. CoG height = 1 500 mm @ 1 500 kg	Max. CoG height = 1 200 mm @ 1 000 kg

Table 4.2.3.1.3.a – Spacecraft mass and CoG key features for dual launch configuration

For spacecraft with characteristics exceeding this domain, please contact Arianespace.

4.2.3.2. Static unbalance

a) Spin-up spacecraft

The center of gravity of the spacecraft must stay within a distance d < 15 mm from the LV longitudinal axis.

b) Three-axis stabilized spacecraft

The center of gravity of the spacecraft must stay within a distance d < 30 mm from the LV longitudinal axis. A static unbalance above 15 mm may generate cinematic conditions after spacecraft separation greater than those described in Chapter 2 paragraph 2.9.2.1.

4.2.3.3. Dynamic unbalance

There is no predefined requirement for spacecraft dynamic balancing with respect to ensuring proper operation of the LV. However, these data have a direct effect on the spacecraft separation.

To ensure the separation conditions in spin-up mode described in the Chapter 2, the maximum spacecraft dynamic unbalance ε corresponding to the angle between the spacecraft longitudinal geometrical axis and the principal roll inertia axis shall be: $\varepsilon \leq 1$ degree.

4.2.3.4. 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. In that case, the design limit load factors in Chapter 3 paragraph 3.2.1 are applicable.

Lateral frequencies

The fundamental (primary) frequency in the lateral axis of a spacecraft cantilevered at the interface must be as follows, with an off-the-shelf adapter:

> 12 Hz

Longitudinal frequencies:

The fundamental (primary) frequency in the longitudinal axis of a spacecraft cantilevered at the interface must be as follows, with an off-the-shelf adapter:

> 20 Hz

No secondary mode should be lower than the first primary mode, apart from sloshing modes which have to be analysed on a case-by-case basis.

Nota:	Primary mode:	Mode associated with large effective masses (in practice, there are one or two primary modes in each direction).
	Secondary mode:	The mode which is not primary, i.e. with small effective mass.

4.2.3.5. Line loads peaking induced by spacecraft

The maximum value of the peaking line load induced by the spacecraft is allowed in local areas to be up to 10% over the maximum line loads induced by the dimensioning loads (deduced from QSL table in Chapter 3 paragraph 3.2.1).

4.2.3.6. Spacecraft RF emissions

To prevent the impact of spacecraft RF emission on the proper functioning of the LV electronic components and RF systems during ground operations and in flight:

- The spacecraft should be designed to respect the LV susceptibility levels given in Table 4.2.3.6.a and illustrated in Figure 4.2.3.6.a,
- The spacecraft must not overlap the frequency bands of the LV receivers.

The allocated frequencies to the Arianespace launch vehicles are in the S band 2 206.5, 2 218, 2 227, 2 249, 2 254.5, 2 267.5 and 2 284 MHz with a margin of 1 MHz and 2 805.5 MHz with a margin of 4 MHz; and in the C band, 5 745 and 5 790 MHz with a margin of 3 MHz.

The spacecraft transmission is allowed during ground operations. Authorisation of transmission during countdown, and/or flight phase and spacecraft separation will be considered on a case by case basis. In any case, no change of the spacecraft RF configuration (no frequency change, no power change) is allowed from H_0 – 1h30min until 20 s after separation.

During the launch vehicle flight until separation of the spacecraft (s) no uplink command signal can be sent to the spacecraft or generated by a spacecraft on-board system (sequencer, computer, etc.).

For multiple launch, in certain cases, a transmission time sharing plan may be set-up on Arianespace request.

Spurious radiations acceptable to launch vehicle	
Frequency range	**Level**
From 14 kHz to 420 MHz	100 dBµVm
From 420 MHz to 480 MHz	35 dBµV/m
From 480 MHz to 1 000 MHz	100 dBµV/m
From 1 000 MHz to 1 164 MHz	Linear interpolation from 100 to 109.86 dBµV/m
From 1 164 MHz to 1 300 MHz	100 dBµV/m
From 1 300 MHz to 1 555 MHz	Linear interpolation from 117.03 to 128.66 dBµV/m
From 1 555 MHz to 1 595 MHz	100 dBµV/m
From 1 595 MHz to 2 000 MHz	Linear interpolation from 130.31 to 145 dBµV/m
From 2 000 MHz to 5 450 MHz	145 dBµV/m
From 5 450 MHz to 5 825 MHz	70 dBµV/m
From 5 825 MHz to 40 GHz	145 dBµV/m

Table 4.2.3.6.a – Spurious radiations acceptable to launch vehicle Narrow-band electrical field measured at the AVUM+ / carrying structure interface

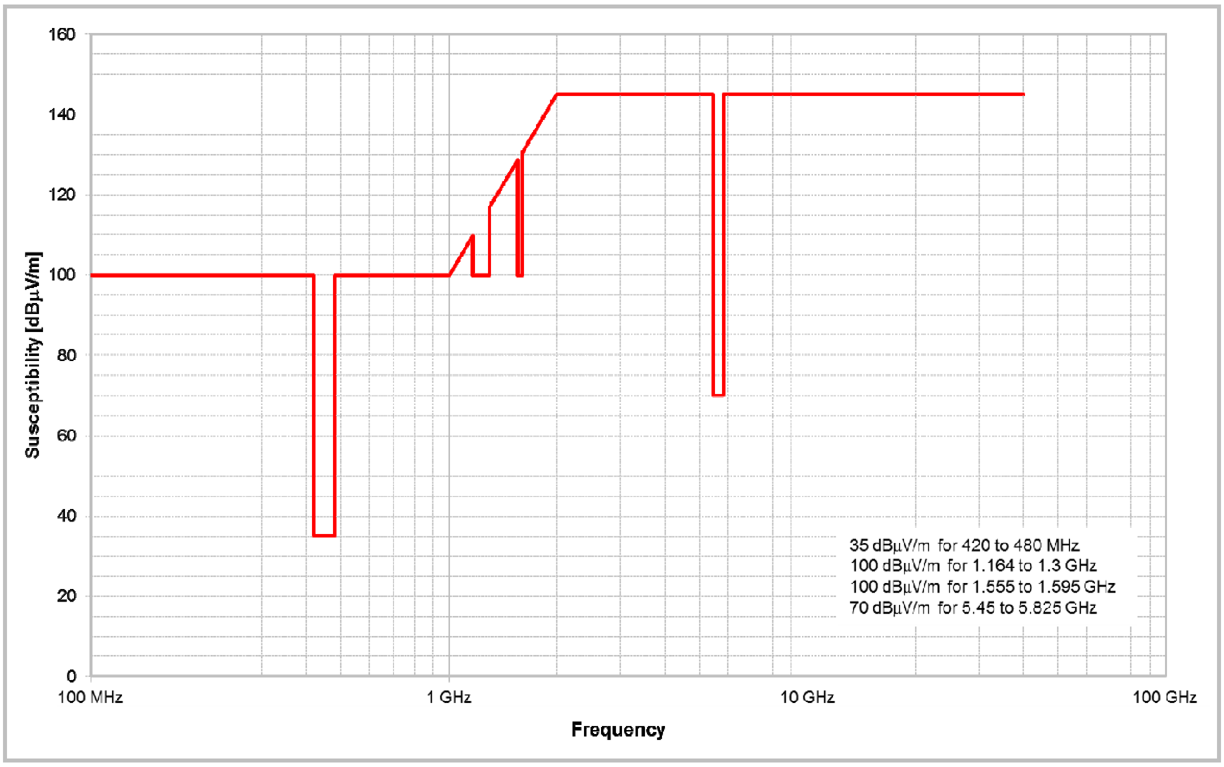


Figure 4.2.3.6.a – Spurious radiations acceptable to launch vehicle Narrow-band electrical field measured at the AVUM+ / carrying structure interface

4.3. SPACECRAFT COMPATIBILITY VERIFICATION REQUIREMENTS

4.3.1. Verification logic

The spacecraft authority shall demonstrate that the spacecraft structure and equipments are capable of withstanding the maximum expected launch vehicle ground and flight environments.

The spacecraft compatibility must be proven by means of adequate tests. The verification logic with respect to the satellite development program approach is shown in Table 4.3.1.a:

S/C development approach	Model	**Static**	**Sine vibration**	**Acoustic**	**Shock**
With Structural Test Model (STM)	STM	Qualification test	Qualification test	Qualification test	Shock test characterization and analysis
	FM1	By heritage from STM [1]	Protoflight test [2]	Protoflight test [2]	Shock test characterization and analysis or by heritage [1]
	Subsequent FM's [3]	By heritage from STM [1]	Acceptance test (optional) or Based on manufacturing control, quality process and analysis	Acceptance test	By heritage and analysis [1]
	For constellation	By heritage from STM [1]	Based on manufacturing control, quality process and analysis	Based on manufacturing control, quality process and analysis	By heritage and analysis [1]

Notes:

[1]: If qualification is claimed by heritage, the representativeness of the structural test model (STM) with respect to the actual flight unit must be demonstrated.

[2]: Protoflight approach means qualification levels and acceptance duration/sweep rate.

[3]: Subsequent FM: spacecraft identical to FM1 (same primary structure, major subsystems and appendages).

Table 4.3.1.a – Spacecraft verification logic

S/C development approach	Model	**Static**	**Sine vibration**	**Acoustic**	**Shock**
With ProtoFlight Model (PFM)	PFM = FM1	Qualification test or by heritage [(1)]	Protoflight test [(2)]	Protoflight test [(2)]	Shock test characterization and analysis or by heritage [(1)]
	Subsequent FM's [(3)]	By heritage [(1)]	Acceptance test (optional) or Based on manufacturing control, quality process and analysis	Acceptance test	By heritage and analysis [(1)]

Notes:

(1): If qualification is claimed by heritage, the representativeness of the structural test model (STM) with respect to the actual flight unit must be demonstrated.

(2): Protoflight approach means qualification levels and acceptance duration/sweep rate.

(3): Subsequent FM: spacecraft identical to FM1 (same primary structure, major subsystems and appendages).

Table 4.3.1.a continued – Spacecraft verification logic

The mechanical environmental test plan for spacecraft qualification and acceptance shall comply with the requirements presented hereafter and shall be reviewed by Arianespace prior to implementation of the first test.

The purpose of ground testing is to screen out unnoticed design flaws and/or inadvertent manufacturing and integration defects or anomalies. It is therefore important that the satellite be mechanically tested in flight-like configuration. In addition, should significant changes affect the tested specimen during subsequent AIT phase prior to spacecraft shipment to CSG, the need to re-perform some mechanical tests must be reassessed. If, despite of notable changes, complementary mechanical testing is not considered necessary by the Customer, this situation should be treated in the frame of a Request For Waiver, which justification shall demonstrate, in particular, the absence of risk for the launcher.

Also, it is suggested, that Customers will implement tests to verify the susceptibility of the spacecraft to the thermal and electromagnetic environment and will tune by this way the corresponding spacecraft models used for the mission analysis.

4.3.2. Safety factors

Spacecraft qualification and acceptance test levels are determined by increasing the design limit load factors presented in Chapter 3 paragraph 3.2 by the safety factors given in Table 4.3.2.a below. The spacecraft must have positive margins with these safety factors.

SC tests	Qualification [3]		Protoflight		Acceptance	
	Factors	Duration / Rate	Factors	Duration / Rate	Factors	Duration / Rate
Static (QSL)	1.25	N/A	1.25	N/A	N/A	N/A
Sine vibrations	1.25	2.0 oct./min [1]	1.25	4.0 oct./min [1]	1.0	4.0 oct./min [1]
Acoustics	+3 dB (or 2)	120 s	+3 dB (or 2)	60 s	1.0	60 s
Shock	+3 dB (or 1.41)	N/A [2]	+3 dB (or 1.41)	N/A [2]	N/A	

Notes:

(1): See paragraph 4.3.3.2.

(2): Number of tests to be defined in accordance with methodology for qualification (see paragraph 4.3.3.5.).

(3): If qualification is not demonstrated by test, it is reminded that a safety factor of 2 (margin ≥ 100%) is requested with respect to the design limit.

Table 4.3.2.a - Test factors, rate and duration

4.3.3. Spacecraft compatibility tests

4.3.3.1. Static tests

Static load tests (in the case of an STM approach) are performed by the Customer to confirm the design integrity of the primary structural elements of the spacecraft platform. Test loads are based on worst-case conditions, i.e. on events that induce the maximum mechanical line loads into the main structure, derived from the table of maximum QSLs (Chapter 3 paragraph 3.2.1) and taking into account the additional line loads peaking (Chapter 3 paragraph 3.2.2) and the local loads (Chapter 3 paragraph 3.3).

The qualification factors (paragraph 4.3.2) shall be considered.

4.3.3.2. Sinusoidal vibration tests

The objective of the sine vibration tests is to verify the spacecraft secondary structure dimensioning under the flight limit loads multiplied by the appropriate safety factors.

The spacecraft qualification test consists of one sweep through the specified frequency range and along each axis.

The qualification levels to be applied are derived from the flight limit amplitudes specified in Chapter 3 paragraph 3.2.4 and the safety factors defined in paragraph 4.3.2. They are presented in Table 4.3.3.2.a, Table 4.3.3.2.b and Table 4.3.3.2.c here below.

Sine	**Frequency range [Hz]**	**Qualification levels (0-peak) [g]**	**Protoflight levels (0-peak) [g]**	**Acceptance levels (0-peak) [g]**
Longitudinal	1 – 5 [(1)]	10 mm	10 mm	8 mm
	5 – 35	1.00	1.00	0.80
	35 – 110	1.25	1.25	1.00
Lateral	1 – 5 [(1)]	10 mm	10 mm	8 mm
	5 – 30	1.00	1.00	0.80
	30 – 110	0.625	0.625	0.50

Notes:

[(1)]: Pending on the potential limitations of the satellite manufacturer's test bench, the achievement of the qualification levels in the [0-5Hz] frequency range can be subject to negotiation in the frame of a request for waiver process, considering that the spacecraft does not present internal modes in that range.

Table 4.3.3.2.a – Sinusoidal vibration tests levels for single launch configuration

Sine	**Frequency range [Hz]**	**Qualification levels (0-peak) [g]**	**Protoflight levels (0-peak) [g]**	**Acceptance levels (0-peak) [g]**
Longitudinal	To be issued later.			
Lateral				

Notes:

(1): Pending on the potential limitations of the satellite manufacturer's test bench, the achievement of the qualification levels in the [0-5Hz] frequency range can be subject to negotiation in the frame of a request for waiver process, considering that the spacecraft does not present internal modes in that range.

Table 4.3.3.2.b – Sinusoidal vibration tests levels for SSMS main passenger

Sine	**Frequency range [Hz]**	**Qualification levels (0-peak) [g]**	**Protoflight levels (0-peak) [g]**	**Acceptance levels (0-peak) [g]**
Upper spacecraft				
Longitudinal	0 – 5 [(1)]	10 mm	10 mm	8 mm
	5 – 35	1.00	1.00	0.80
	35 – 110	1.25	1.25	1.00
Lateral	0 – 5 [(1)]	10 mm	10 mm	8 mm
	5 – 30	1.00	1.00	0.80
	30 – 110	0.625	0.625	0.50
Lower spacecraft				
Longitudinal	0 – 5 [(1)]	10 mm	10 mm	8 mm
	5 – 35	1.00	1.00	0.80
	35 – 110	1.25	1.25	1.00
Lateral	0 – 5 [(1)]	10 mm	10 mm	8 mm
	5 – 30	1.00	1.00	0.80
	30 – 110	0.625	0.625	0.50

Notes:

[(1)]: Pending on the potential limitations of the satellite manufacturer's test bench, the achievement of the qualification levels in the [0-5Hz] frequency range can be subject to negotiation in the frame of a request for waiver process, considering that the spacecraft does not present internal modes in that range.

Table 4.3.3.2.c – Sinusoidal vibration tests levels for dual launch configuration

A notching procedure may be agreed in the frame of a request for waiver, on the basis of the latest coupled loads analysis (CLA) available at the time of the tests to prevent excessive loading of the spacecraft structure. However it must not jeopardize the tests objective to demonstrate positive margins of safety with respect to the flight limit loads, while considering appropriate safety factor.

The acceptability of the sweep rate shall consider the dynamic characteristics of spacecraft secondary structures or appendages and the actual damping of the payload structure, in order to ensure proper solicitation of the whole spacecraft during the test.

4.3.3.3. Acoustic vibration tests

Acoustic testing is accomplished in a reverberant chamber. The volume of the chamber with respect to that of the spacecraft shall be sufficient so that the applied acoustic field is diffuse. The test measurements shall be performed at a minimum distance of 1 m from spacecraft.

The acoustic specification to be considered for spacecraft testing is defined considering:

- The acoustic limit levels described in Chapter 3 paragraph 3.2.6.2 (Table 3.2.6.2.a);
- The margin policy defined in Table 4.3.2.a;
- The transient nature of the maximum acoustic levels recorded during the first 3 seconds of the lift-off phase;
- A minimum level of 120 dB in all octave bands for workmanship demonstration purpose.

Octave center frequency [Hz]	Qualification levels [dB]	Protoflight levels [dB]	Acceptance levels [dB]	Test tolerance [dB]
31.5	123	123	120	-2 ; +4
63	126	126	123	-1 ; +3
125	129	129	126	-1 ; +3
250	136	136	133	-1 ; +3
500	139	139	136	-1 ; +3
1 000	130	130	127	-1 ; +3
2 000	125	125	120	-1 ; +3
OASPL[1] (20 – 2828 Hz)	141.7	141.7	138.7	-1 ; +3
Test duration	120 s	60 s	60 s	

(1) OASPL: Overall Acoustic Sound Pressure Level

Table 4.3.3.3.a – Acoustic vibration test levels

4.3.3.4. Shock qualification

The ability of the spacecraft to withstand the shock environment generated by the stages separation, the fairing jettisoning and the spacecraft separation shall follow a comprehensive process including tests and analysis.

➢ Launcher events (fairing/stages separation)

A shock test shall be performed in order to characterize the shock transmission inside the spacecraft and define the transfer functions between the spacecraft interface plane and the equipment base.

This test can be performed on the STM, PFM or on the first flight model, provided that the spacecraft configuration is representative of the flight model (structure, load paths, equipment presence and location,…). This test can be performed once, and the verification performed covers the spacecraft platform as far as no structural modification alters the validity of the analysis.

This qualification is obtained by comparing the component unit qualification levels to the equipment base levels experienced applying the interface shock spectrum specified in Chapter 3 paragraph 3.2.7, Table 3.2.7.a with the dedicated transfer function.

A minimum +3 dB margin has to be highlighted to validate the qualification (see Table 4.3.2.a). Note that each unit qualification status can be obtained from environmental qualification tests other than shock tests by using equivalent rules (e.g. from sine or random vibration tests).

A VEga Shock Test Apparatus (VESTA) generating a shock environment representative of the actual fairing separation event can be provided by Arianespace. Thanks to the representativeness of this test mean, the spacecraft qualification can be directly derived from the VESTA tests results, removing the uncertainties margins taken into consideration.

➢ Clamp-band release event

The demonstration of the spacecraft's ability to withstand the separation shock generated by the clamp-band release shall be based on one of the following methods:

Method Number One: Release drop test, extrapolation to specification, comparison to S/C sub-systems qualification:

A clamp-band release drop test is conducted with the tension of the band set at the **nominal tension at installation**. During this test, interface levels and equipment base levels are measured. This test can be performed on the STM, on the PFM or on the first flight model provided that the spacecraft structure close to the interface as well as the equipment locations and associated supports are equivalent to those of the flight model.

The release shocks generated at the spacecraft's interface and measured during the above-mentioned test are compared to the applicable shock specification (see Chapter 3 paragraph 3.2.7, Table 3.2.7.b). The ratio derived from the above comparison is then considered to extrapolate the measured equipment levels to the specification.

These extrapolated shock levels are then increased by a safety factor of +3 dB and are compared to the qualification status of each spacecraft subsystem and/or equipment. Note that each unit qualification status can be obtained from environmental qualification tests other than shock tests by using equivalent rules (e.g. from sine or random vibration tests).

<u>Method Number Two:</u> Release drop test with maximal tension, direct comparison to S/C sub-systems qualification:

A clamp-band release drop test is conducted with the tension of the band set as close as possible to its **maximum value during flight**. During this test, interface levels and equipment base levels are measured. This test can be performed on the STM, on the PFM or on the first flight model provided that the spacecraft structure close to the interface as well as the equipment locations and associated supports are equivalent to those of the flight model.

The induced shocks generated on spacecraft equipment measured during the above-mentioned test are then increased by:

- A +3 dB uncertainty margin aiming at deriving flight limit environment from the single test performed in flight-like configuration; [NB: In case two clamp-band release drop tests are performed, this +3 dB uncertainty margin can be removed but the maximum recorded value between the two tests has to be considered for each equipment.]
- A +3 dB safety factor aiming at defining the required minimum qualification levels, to be compared to the qualification status of each spacecraft subsystem and/or equipment.

These obtained shock levels are then compared to the qualification status of each spacecraft subsystem and/or equipment. Note that each unit qualification status can be obtained from environmental qualification tests other than shock tests by using equivalent rules (e.g. from sine or random vibration tests).

General nota: In case of recurring platform or spacecraft, the shock qualification can be based on heritage, pending that identical platform or spacecraft is already qualified to both launcher and clamp-band release event (for a tension identical or higher than the one targeted for the ongoing satellite).

SPACECRAFT INTERFACES Chapter 5

5.1. INTRODUCTION

The Vega C launch vehicle provides standard interfaces which fit most of spacecraft buses and satellites, and allows an easy switch between the launch vehicles of the European transportation fleet.

This chapter covers the definition of the spacecraft interfaces with the payload adapter, the fairing, the dual launch structure and the on-board and ground electrical equipment.

The spacecraft is mated to the launch vehicle through a dedicated structure called an adapter that provides mechanical interface, electrical harnesses routing and systems to assure the spacecraft separation. Off-the-shelf adapters (VAMPIRE 937 and VAMPIRE 1194), with separation interface diameter of 937 mm and 1194 mm are available.

For launch configuration with main spacecraft with auxiliary passenger(s), multiple launch structures so-called "VAMPIRE 937 with towers" or "SSMS" are proposed.

For dual launch configuration, the carrying structure so-called "VESPA C" is proposed, which house the lower passenger(s) and carries the upper passenger(s).

The payload fairing protects the spacecraft from the external environment during the flight as on the ground, providing at the same time specific access (if any) to the spacecraft during ground operations.

The electrical interface provides communication with the launch vehicle and the ground support equipment during all phases of spacecraft preparation, launch and flight.

The adapters/carrying structures and fairing accommodate also the telemetry sensors that are used to monitor the spacecraft flight environment.

These elements could be subject of mission specific adaptation, as necessary, to fit with the Customer requirements. Their respective compatibility with the spacecraft is managed through the Interface Control Document (DCI).

5.2. THE REFERENCE AXES

All definition and requirements shall be expressed in the same reference axis system to facilitate the interface configuration control and verification.

Figure 5.2.a shows the Vega C launch vehicle coordinate system which is the reference axis system.

The clocking of the spacecraft with regard to the launch vehicle axes is defined in the Interface Control Document (DCI) taking into account the spacecraft characteristics (volume, access needs, RF links, etc.).

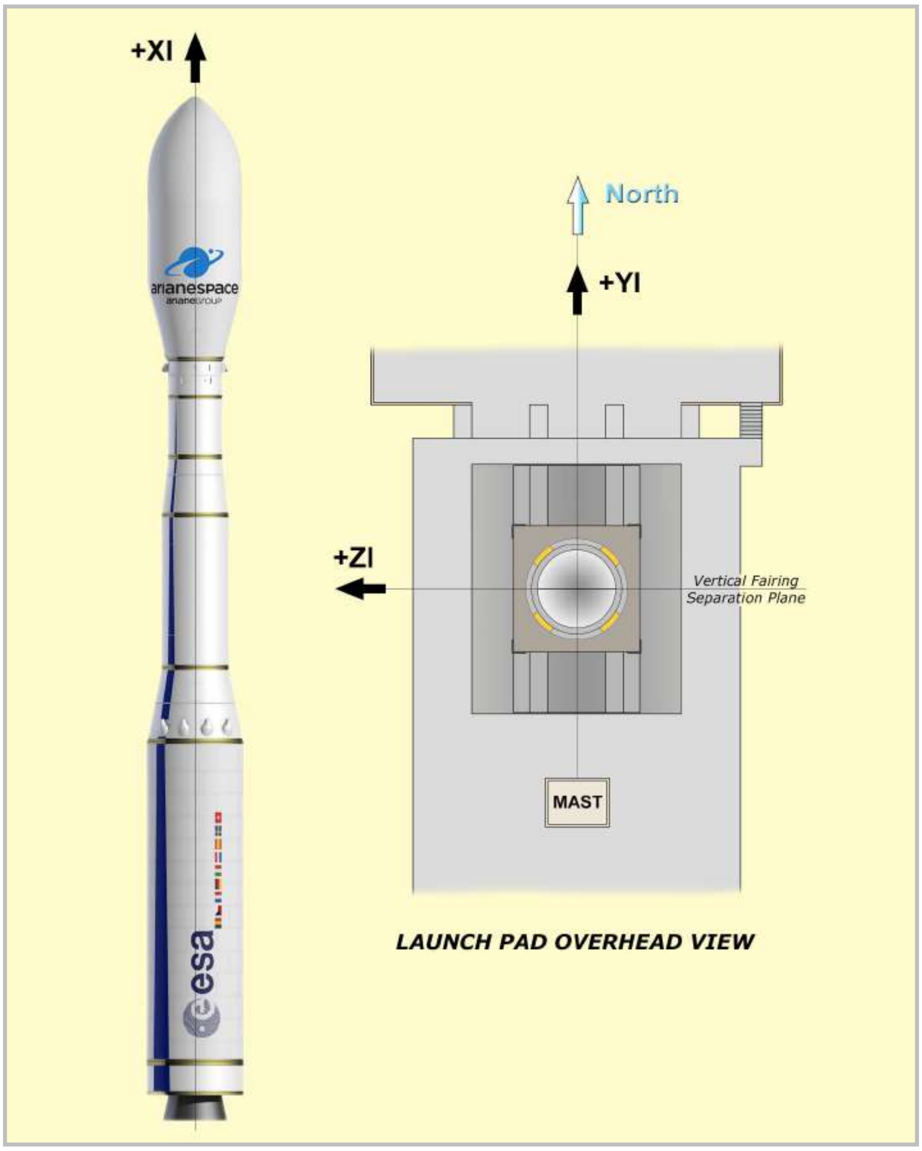


Figure 5.2.a – Vega C coordinate system

5.3. ENCAPSULATED SPACECRAFT INTERFACES

5.3.1. Payload usable volume definition

The payload usable volume is the area under the fairing, or the dual launch carrying structure, available to the spacecraft mated on the adapter. This volume constitutes the limits that the static dimensions of the spacecraft, including manufacturing tolerance, thermal protection installation, appendices, etc., shall not exceed.

It has been established having regard to the potential displacement of the spacecraft complying with frequency requirements described in Chapter 4.

Allowance has been made for manufacturing and assembly tolerances of the upper part (fairing, dual launch structure, adapter, upper stage), for all displacements of these structures under ground and flight loads, and for necessary clearance margin during dual launch structure separation.

In the event of local protrusions located slightly outside the above-mentioned envelope, Arianespace and the Customer can conduct a joint investigation in order to find the most suitable layout.

The usable volumes above the adapter for single launch configurations are defined in Figure 5.3.1.a and Figure 5.3.1.b.

The usable volumes for dual launch configurations are defined in Figure 5.3.4.2.b.

The usable volume definition for launch configurations with main spacecraft and auxiliary passenger(s) is presented in Annex 4b.

The allocated volume envelope in the vicinity of the adapter is described in the annexes dedicated to each off-the-shelf adapter.

Accessibility of the mating interface, separation system functional requirements and non-collision during separation are also considered for its definition.

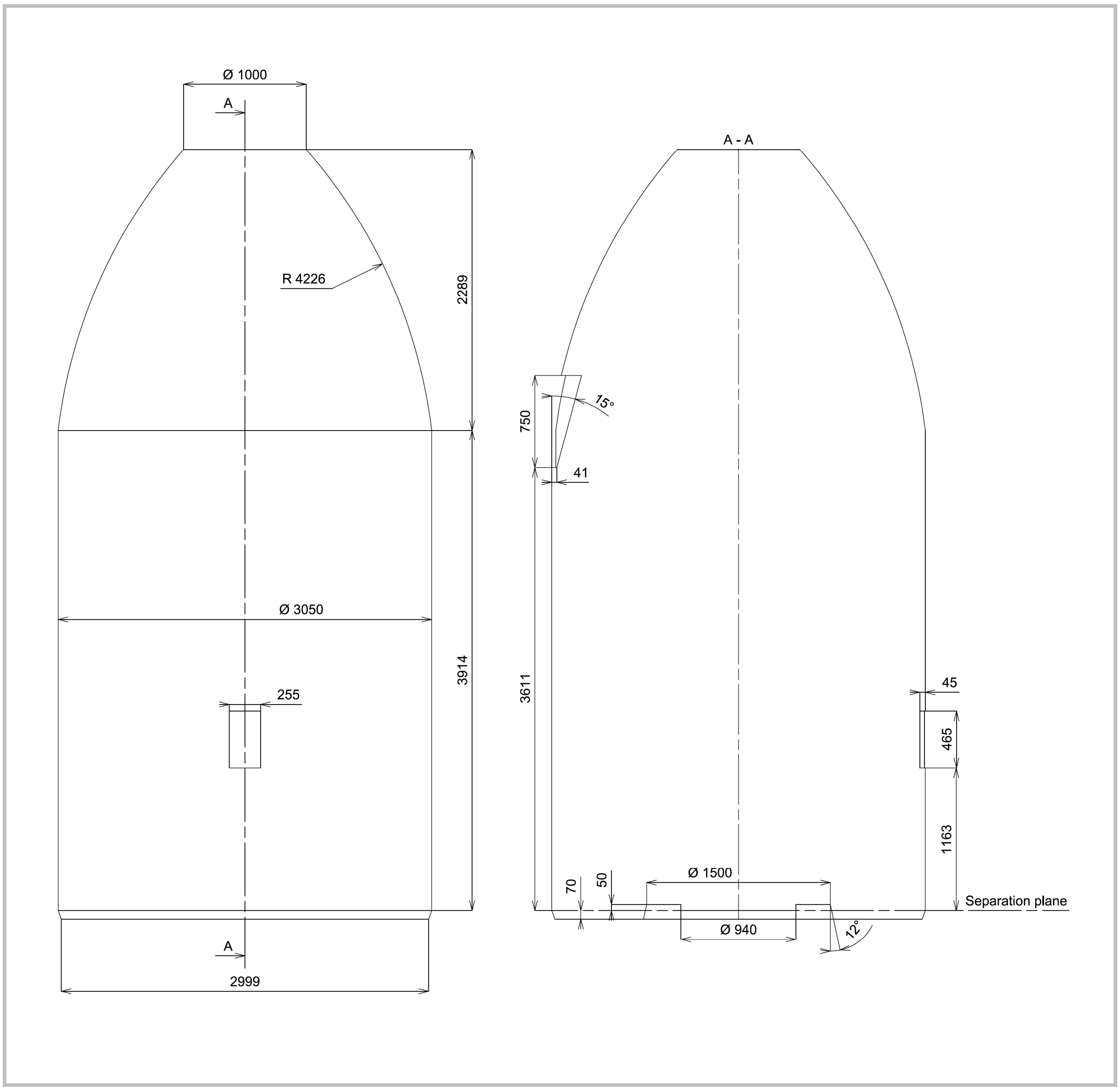


Figure 5.3.1.a – Usable volume inside Vega C fairing for single launch configuration with VAMPIRE 937 adapter

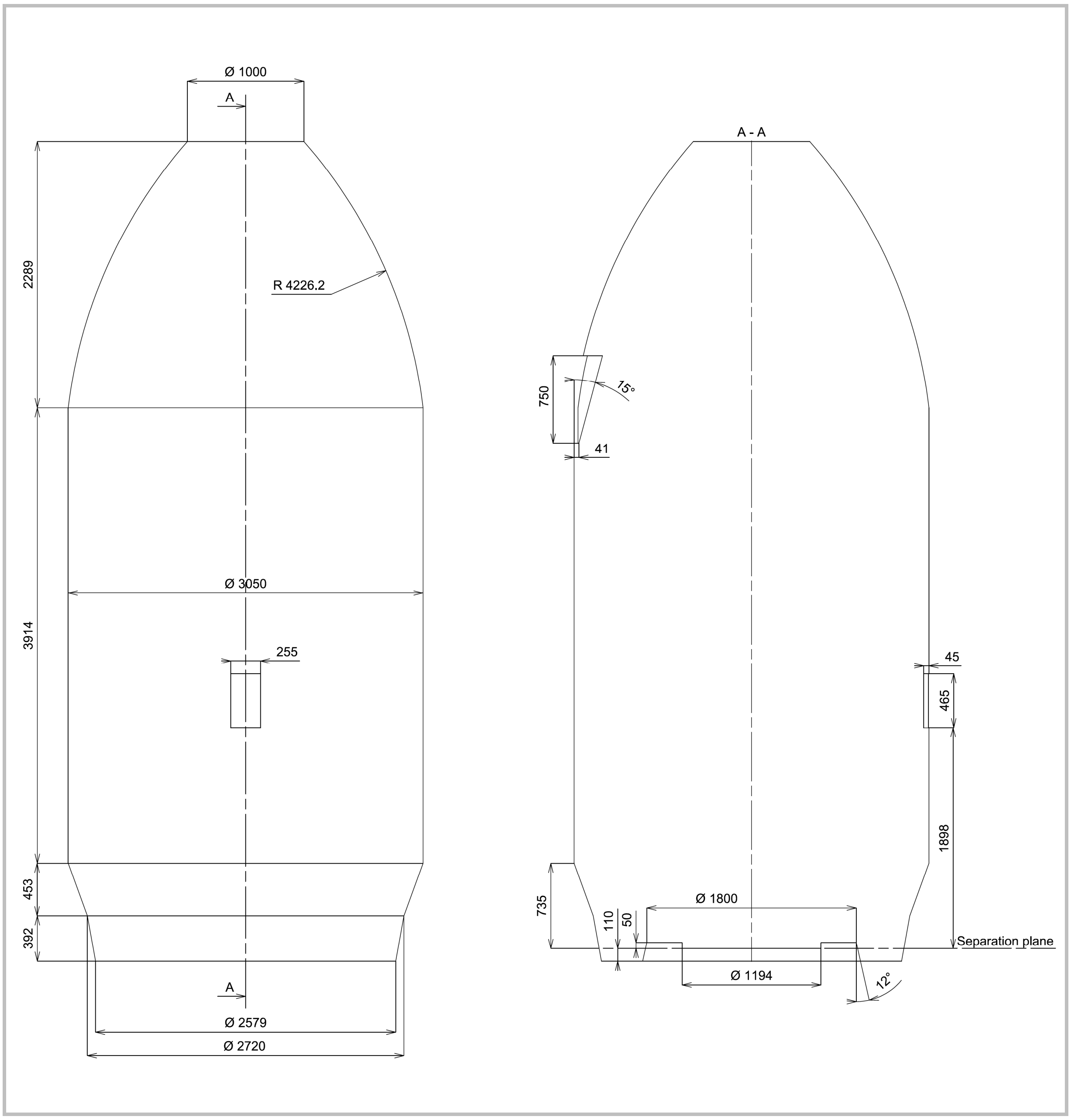


Figure 5.3.1.b – Usable volume inside Vega C fairing for single launch configuration with VAMPIRE 1194 adapter

5.3.2. Spacecraft accessibility

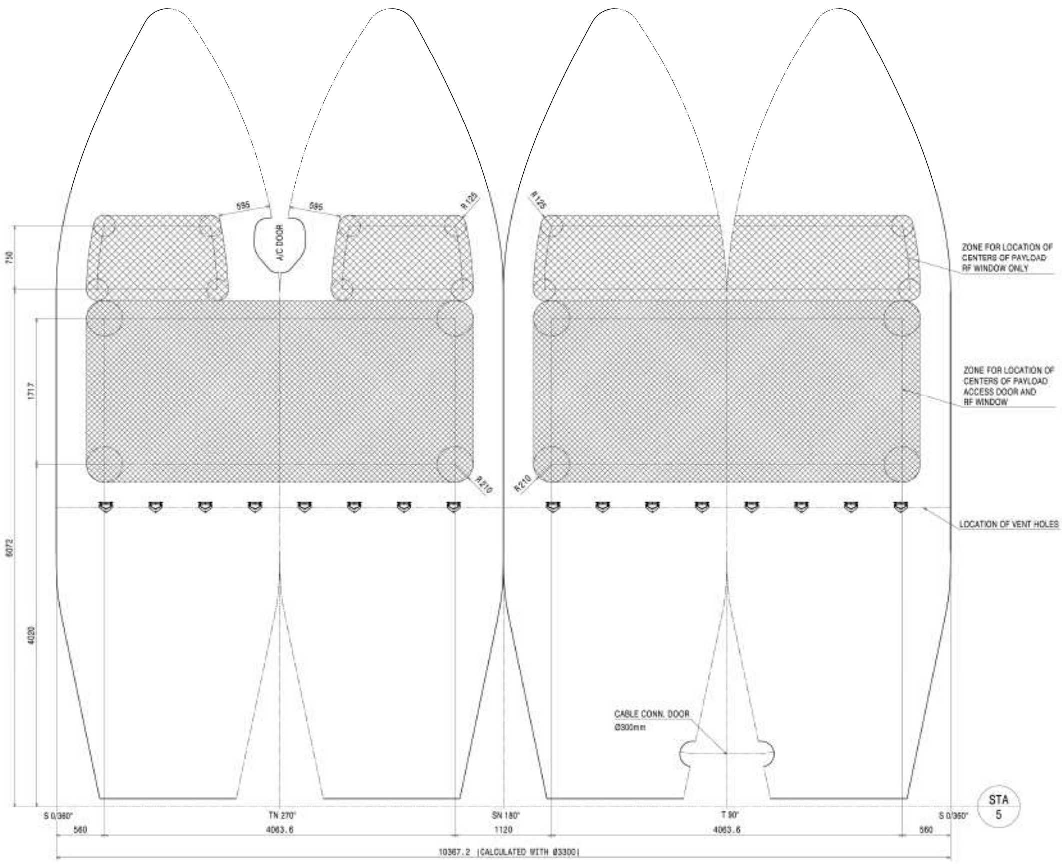
The encapsulated spacecraft can be accessible for direct operations up to D-1 through the access doors of the fairing structure. If access to specific areas of spacecraft is required, additional doors can be provided on a mission-specific basis. Doors shall be installed in the authorized areas.

During the operations, cleanliness in the fairing is ensured through overpressure until the hoisting of the upper composite inside the gantry. After final integration on the launch vehicle the fairing ventilation is activated (D-8).

Specific means can be provided (TBC) to ensure access from a protected area.

Similarly, if RF link through the fairing is required, radio-transparent windows can ensure RF link between spacecraft antenna and ground.

The access doors authorized areas and RF window possible locations are presented in Figure 5.3.2.a.



Access doors:	*∅ 420 mm* *1 door for each half-fairing (2 doors max.)*
RF window:	*∅ 250 mm* *Min distance between access door / RF window centers = 800mm*

Figure 5.3.2.a – Locations and dimensions of access doors and RF windows

5.3.3. Special on-fairing insignia

A special mission insignia based on Customers' supplied artwork can be placed by Arianespace on the cylindrical section of the Fairing.

The dimensions, colors, and location of each such insignia are subject to mutual agreement.

The artwork shall be supplied not later than 6 months before launch.

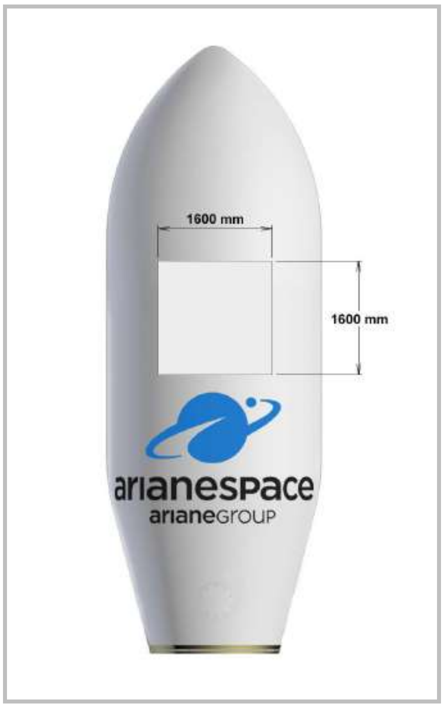


Figure 5.3.3.a – Location of Customers' logo

5.3.4. Payload compartment description

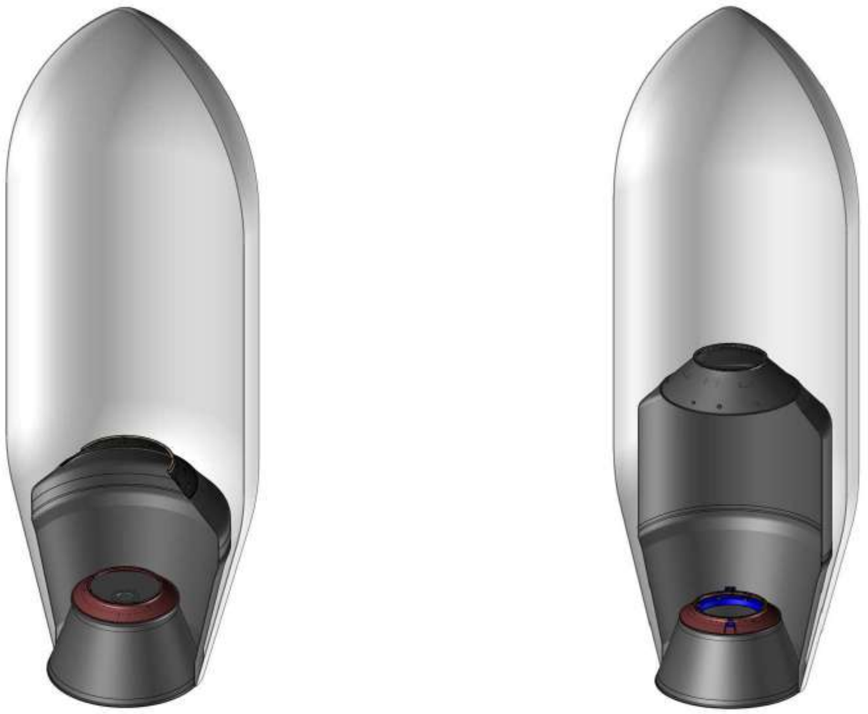
5.3.4.1. Nose fairing description

The fairing consists of a two-half-shell carbon-fiber reinforced plastic (CFRP) sandwich with aluminum honeycomb structure. The total thickness is approximately 20 mm.

Separation of the nose fairing is achieved by means of two separation systems: a vertical one (VSS) which consists in a pyrotechnic cord, located at the level of the two half fairing joining plan; and a horizontal one (HSS) consisting in a clamp-band which connects the fairing to the AVUM+ upper stage.

5.3.4.2. Dual launch structure description (VESPA C)

The general characteristics of the VESPA C (Vega C Secondary Payload Adapter) dual launch structures are presented in Figure 5.3.4.2.a.



VESPA C – Short version		VESPA C – Long version
3 222 mm TBC	Total height	4 552 mm TBC
2 620 mm TBC	Max diameter	2 620 mm TBC
390 kg TBC	Total mass	455 kg TBC

Materials:
CFRP (automatic FP) and aluminum alloy

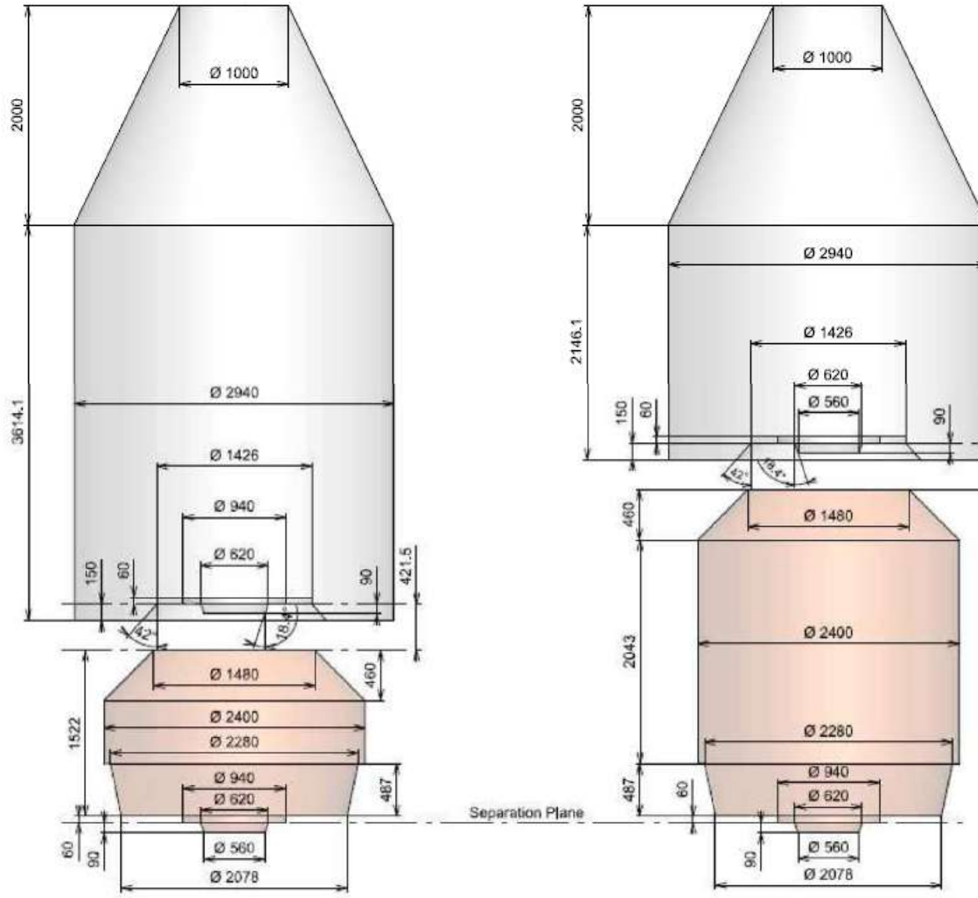
Jettisoning of the VESPA C upper part:

- Clamp-band with low shock separation system
- 8 springs (TBC)

Separation of the upper and lower satellites:
Clamp-band with low shock separation system

Figure 5.3.4.2.a – VESPA C dual launch structures

The usable volume offered for the upper spacecraft (on top of the VESPA dual launch structure) and for the lower spacecraft (inside the VESPA dual launch structure) are defined in the Figure 5.3.4.2.b.



Usable static volume available for the SC for the shorter VESPA possible.
Assumptions: PAF AR937 in lower position and interface 937mm in upper position but other interfaces (1194mm and plate) should be also possible.

Usable volume static available for the SC for the higher VESPA possible
(all height in between shorter VESPA and higher one should be also possible).
Assumption : PAF AR937 in lower position and interface 937mm in upper position but other interfaces (1194mm and plate) should be possible.

Figure 5.3.4.2.b – Usable volume inside Vega C fairing for dual launch configuration with VESPA C

5.4. MECHANICAL INTERFACES

Vega C offers a range of standard off-the-shelf adapters and their associated equipment, compatible with most of the spacecraft platforms, and derived from Arianespace extended family of adapters for Ariane and Soyuz launch systems.

In case of specific Customer's needs and requirements, dedicated adapter or dispenser can be developed.

All adapters are equipped with a payload separation system and brackets for electrical connectors.

The payload separation system is a clamp-band system consisting of a clamp-band set, release mechanism and separation springs.

The electrical connectors are mated on two brackets installed on the adapter and spacecraft side. On the spacecraft side, the umbilical connector's brackets must be stiff enough to prevent any deformation greater than 0.5 mm under the maximum force of the connector spring.

In case the dual launch structure (VESPA C) is used, the upper spacecraft is mated directly on the VESPA C which includes the adapter.

5.4.1. Standard Vega C adapters

The general characteristics of the off-the-shelf adapters and adaptation structures are presented in Table 5.4.1.a. A more detailed description is provided in the Annex 4.

Adapter		Description	Separation system
VAMPIRE 937		Height: 2 596 mm Max. mass: 120 kg (TBC)	Clamp-band ∅937 mm with low shock separation system
VAMPIRE 1194		Height: 1 861 mm Max. mass: 95 kg (TBC)	Clamp-band ∅1194 mm with low shock separation system

Table 5.4.1.a – Vega C standard adapters

The use of the adapters PLA 937 VG and PLA 1194 VG on Vega C shall be evaluated on a case by case basis.

Arianespace is open to any other separation device. Thanks to its experience, Arianespace has already developed or could develop different interfaces to fulfil any Customer need. In such specific case, Customer shall provide justifications and evidence about the reliability of the separation device.

5.4.2. Dispensers

Dispenser structures can be developed in the frame of dedicated programs.

5.4.3. Standardized carriers for small satellites

Vega C accommodates carriers for small satellites. Specifications related to mini (200-400 kg), micro (50-200 kg) or nano (< 50 kg) satellites will be part of the Auxiliary Passengers User's Manual.

Table 5.4.3.a – Vega C multiple launch structures for single launch configurations with auxiliary passenger(s)

5.5. ELECTRICAL AND RADIO-ELECTRICAL INTERFACES

The needs of communication with the spacecraft during the launch preparation and the flight require electrical and RF links between the spacecraft, the launch vehicle and the EGSE located at the launch pad and preparation facilities.

The electrical interface composition between spacecraft and the Vega C launcher is presented in Table 5.5.a.

All other data and communication network used for spacecraft preparation in the CSG facilities are described in Chapter 6.

The requirements for the spacecraft connector bracket stiffness are described in paragraph 5.4.

<table>
<tr><th>Service</th><th>Description</th><th>Lines definition</th><th>Provided as</th><th>I/F connectors (1)</th></tr>
<tr><td>Umbilical lines</td><td>Spacecraft TC/TM data transmission and battery charge</td><td>74 lines (2)
(see § 5.5.1)</td><td>Standard</td><td rowspan="5">2 × 37 pin
DBAS 70 37 0 SN
DBAS 70 37 0 SY

or

2 × 61 pin
DBAS 70 61 0 SN
DBAS 70 61 0 SY</td></tr>
<tr><td rowspan="5">LV to S/C services</td><td>Dry loop commands</td><td>(see § 5.5.2.1)</td><td>Optional</td></tr>
<tr><td>Electrical commands</td><td>(see § 5.5.2.2)</td><td>Optional</td></tr>
<tr><td>Spacecraft TM retransmission</td><td>(see § 5.5.2.3)</td><td>Optional</td></tr>
<tr><td>Additional power supply during flight</td><td>(see § 5.5.2.4)</td><td>Optional</td></tr>
<tr><td>Pyrotechnic command</td><td>(see § 5.5.2.5)</td><td>Optional</td><td>2 × 12 pin maximum</td></tr>
<tr><td>RF link</td><td>Spacecraft TC/TM data transmission</td><td>RF transparent window</td><td>Optional</td><td>N/A</td></tr>
</table>

(1) Arianespace will supply the Customer with the spacecraft side interface connectors compatible with equipment of the off-the-shelf adapters.

(2) Maximal capability can be extended on a case by case basis.

Table 5.5.a – Spacecraft to launch vehicle electrical and RF interfaces

Flight constraints

During the powered phase of the launch vehicle and up to separation of the spacecraft, no command signal can be sent to the spacecraft, or generated by a spacecraft on-board system (sequencer, computer, etc.). Orders can be sent by the LV, during ballistic phases. During this powered phase a waiver can be studied to make use of commands defined in this paragraph providing that the radio electrical environment is not affected.

After the powered phase and before the spacecraft separation, the commands defined in this paragraph can be provided to the spacecraft.

To command operations on the spacecraft after separation from the launch vehicle, microswitches or telecommand systems (after 20 s) can be used. Initiation of operations on the spacecraft after separation from the launch vehicle, by a payload on-board system programmed before lift-off, must be inhibited until physical separation.

The typical flight constraints are summarized in the Table 5.5.b:

H0-1h30min | Upper stage burn-out | Separation | Separation + 20s

Command	NO	NO	NO	YES
Spacecraft sequencer	NO	NO	YES	YES
LV orders	NO	YES	NO	NO

Table 5.5.b – Flight constraints for command signal to spacecraft

5.5.1. Spacecraft to EGSE umbilical lines

5.5.1.1. Lines definition

The spacecraft to the EGSE umbilical lines provide the following main functions:

- Data transmission and spacecraft monitoring;
- Powering of the spacecraft and charge of the spacecraft battery.

The umbilical lines passing through the umbilical connector are disconnected at lift-off.

A total of 100 pairs (i.e. 200 wires, one pair = 2 twisted wires) are available at launch system level, of which:

- 20 pairs (i.e. 40 wires) in 120 Ω impedance;
- 8 pairs (i.e. 16 wires) in 100 Ω impedance;
- 4 pairs (i.e. 8 wires) in 75 Ω impedance.

All other wires are in 50 Ω standard impedance.

These lines (200 wires) have to be shared between all the Customers.

5.5.1.2. Lines description

The LV to Launch Pad harness layout is defined in Figure 5.5.1.2.a.

The spacecraft EGSE(s) is (are) located in the launch pad basement. The spacecraft-to-launch pad room (LP room) wiring consists of permanent and customized sections.

The permanent sections have the same configuration for each launch, and consist of the lines between the LP room connectors and the umbilical connector to the AVUM+ / adapter interface. This segment is ~70 meters long.

The customized section is configured for each mission. It consists of:

1. The lines between the LP room connectors and the Customer COTE in the LP room. The Customer will provide the harness for this segment.
2. The adapter wiring harness.

TBD, to be issued later

Figure 5.5.1.2.a – Umbilical links between spacecraft mated on the launcher and its check-out Terminal Equipment

5.5.1.3. Lines composition and electrical characteristics

The characteristics of these umbilical links, between the connecting box in LP room (RCU) and the electrical umbilical plug POE (spacecraft/adapter interface), are:

- Resistance, for low impedance wire: $< 1.2\ \Omega$ (one way);
  for high impedance wire: $< 10\ \Omega$ (one way);
- Insulation > 5 MΩ under 500 Vdc (> 100 MΩ for on-board harness only).

The operating constraints are the following:

- Each wire shall not carry current in excess of 5 A;

  For higher currents or large number of power lines, please contact Arianespace.
- For all the lines, the voltage shall be less than 55 Vdc;
- No current shall circulate in the shielding.

The Customer shall design his spacecraft so that during the final preparation leading up to actual launch, the umbilical lines are carrying only low currents at the moment of lift-off, i.e. less than 100 mA – 50 V. Spacecraft power must be switched from external to internal, and ground power supply must be switched off before lift-off.

The harness length from COTE to spacecraft connectors is ~ 95 m, through by 4 LV + 3 GS interface connections (plus 4 terminal strips in the mast junction and connecting boxes).

5.5.2. Launch vehicle to spacecraft electrical functions

The launch vehicle can provide optional electrical functions used by the spacecraft during flight.

The execution of the different commands is monitored by the LV telemetry system.

Due to the spacecraft to launch vehicle interface, the Customer is required to protect the circuit against any overload or voltage overshoot induced by his circuits both at circuits switching and in the case of circuit degradation.

To protect spacecraft equipment a safety plug with a shunt on spacecraft side and a resistance on the LV side shall be installed in all cases.

5.5.2.1. Spacecraft separation monitoring (optional)

The spacecraft separation status indication can be provided by a dry loop strap on adapter side dedicated for the separation monitoring by satellite.

The main electrical characteristics of these straps are:

- Strap "closed": $R \leq 1\ \Omega$;
- Strap "open": $R \geq 100\ k\Omega$.

Note: As a standard, the spacecraft separation monitoring on LV side is provided by two redundant microswitches and transmitted by the LV telemetry system to LV ground segment.

5.5.2.2. Dry loop commands (optional)

This function can be used for spacecraft initiating sequence or status triggering. The information is sent through the opening or closing of a relay contact which is part of the AVUM+ electrical equipment (yes- or no-type information). 8 single commands (or 4 redundant commands) are available.

The main electrical characteristics are:

- Loop closed: $R \leq 10\ \Omega$;
- Loop open: $R \geq 100\ k\Omega$;
- Voltage: ≤ 32 V;
- Current: ≤ 0.5 A.

The insulation of the LV on-board circuit is $\geq 1M\Omega$ under 50 Vdc.

During flight, these dry loop commands are monitored by the LV telemetry system.

Protection: The Customer is required to protect the circuit against any overload or voltage overshoot induced by his circuits both at circuits switching and in the case of circuit degradation.

The Customer has to intercept the launcher command units in order to protect the spacecraft equipment and to allow the integration check-out by using a safety plug equipped with an open circuit on the spacecraft side and a short circuit on the LV side.

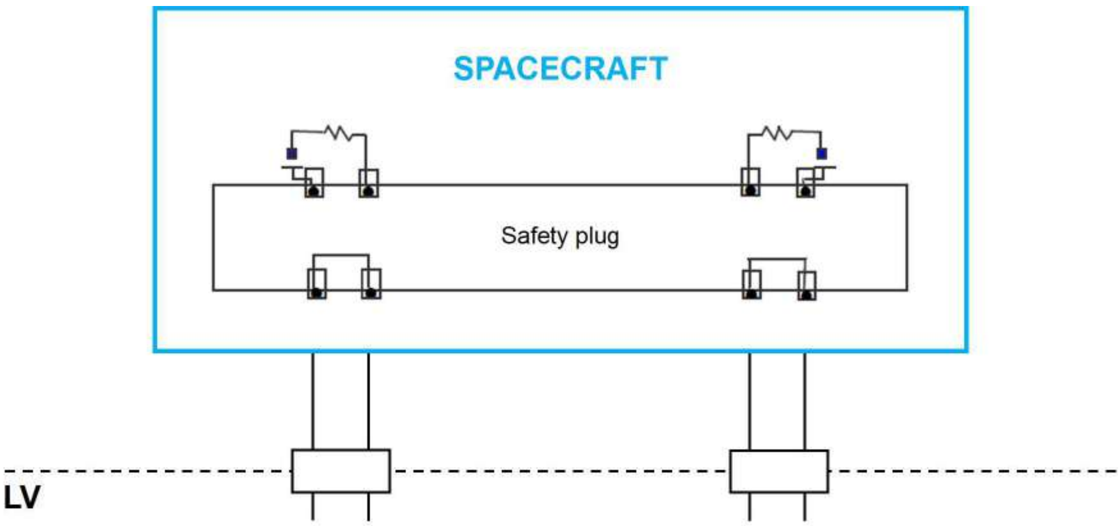


Figure 5.5.2.2.a – Typical principle of a dry loop commands diagram

5.5.2.3. Electrical commands (optional)

The launcher can send up to 4 dedicated single commands with the following main electrical characteristics:

- Input voltage: 28 V ± 4 V;
- Input current: ≤ 0.5 A.

During the flight, the commands are monitored through the LV telemetry system.

Protection: The Customer is required to protect the circuit against any overload or voltage overshoot induced by his circuits both at circuits switching and in the case of circuit degradation.

The Customer has to intercept the LV command units in order to protect the spacecraft equipment and to allow the integration check-out by using a safety plug equipped with an open circuit on the spacecraft side and a protection resistance (100 Ω ± 5%) on the LV side.

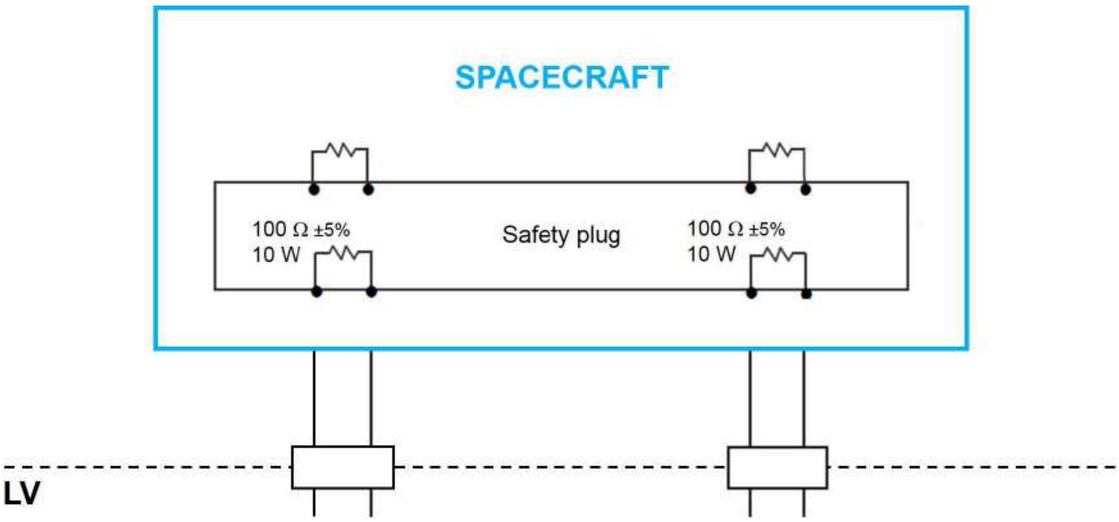


Figure 5.5.2.3.a – Typical principle of an electrical commands diagram

5.5.2.4. Spacecraft telemetry retransmission (optional)

In flight, transmission of spacecraft measurements by the LV telemetry system can be studied on a case by case basis. A Customer wishing to exercise such an option should contact Arianespace for interface characteristics. Signal data can only be provided after post flight telemetry exploitation.

5.5.2.5. Power supply to spacecraft (optional)

Additional power (three lines) can be supplied to the spacecraft as an optional service.

The main characteristics of these three lines are:

• 28 V power line:	Input voltage: Maximal output power:	28 V ± 4 V; 75 W;
• 12 V power line:	Input voltage: Maximal output power:	12 V ± 5%; 35 W;
• 5 V power line:	Input voltage: Maximal output power:	5 V ± 5%; 35 W;

The power output is equipped with protection device against overloads.

5.5.2.6. Pyrotechnic command (optional)

The avionic system has the capability to issue all nominal (N) and redundant (R) orders (one pyrofunction initiates simultaneously 2 squibs) to initiate adapter or dispenser separation systems.

The avionic system is compatible with 1 Ω resistance squibs.

The squib should be able to sustain a low current of 25 mA (with a cumulative time of 10 sec) for continuity test purpose

In addition to LV orders for spacecraft separation, other pyrotechnic commands could be generated to be used for spacecraft internal pyrotechnic system or in case where adapter with separation system is supplied by the Customer. The electrical diagram is presented in Figure 5.5.2.6.a.

The main electrical characteristics of a nominal pyro signal delivered by the launcher are:

• Minimal guaranteed current:	4.1 A;
• Maximal supplied current for one squib:	12 A;
• Typical current for standard cable and squib:	Between 5 and 7 A;
• Impulse duration:	20 msec ± 2.5 msec;
• Nominal battery voltage:	28 V ± 4 V;
• The redundant order:	The same – at the same time.

12 pyrofunctions maximum are available in total.

A maximum of two pyro functions (initiation of 2 × 2 (i.e. N+R) squibs) can be fired simultaneously.

The insulation between wires (open loop) and between wires and structure must be ≥ 1 MΩ under 10 Vdc.

These orders are supplied from dedicated battery and could be segregated from the umbilical links and other data links passing through dedicated connectors.

This pyrotechnic order is compatible with an initiator whose a resistance is equal to 1.05 Ω ± 0.15 Ω. The one-way circuit line resistance between the AVUM+ / adapter interface and the spacecraft initiator shall be adapted case by case. The Customer shall contact Arianespace for this specific interface characteristic definition.

To ensure safety during ground operations, two electrical barriers are closed before lift-off. During flight, the pyrotechnic orders are monitored through the LV telemetry system.

Protection: The Customer has to intercept the LV command circuits in order to protect the spacecraft equipment and to allow the integration check-out by using a safety plug equipment with a shunt on spacecraft side and a resistance of 2 kΩ ±1% (0.25W) on the LV side.

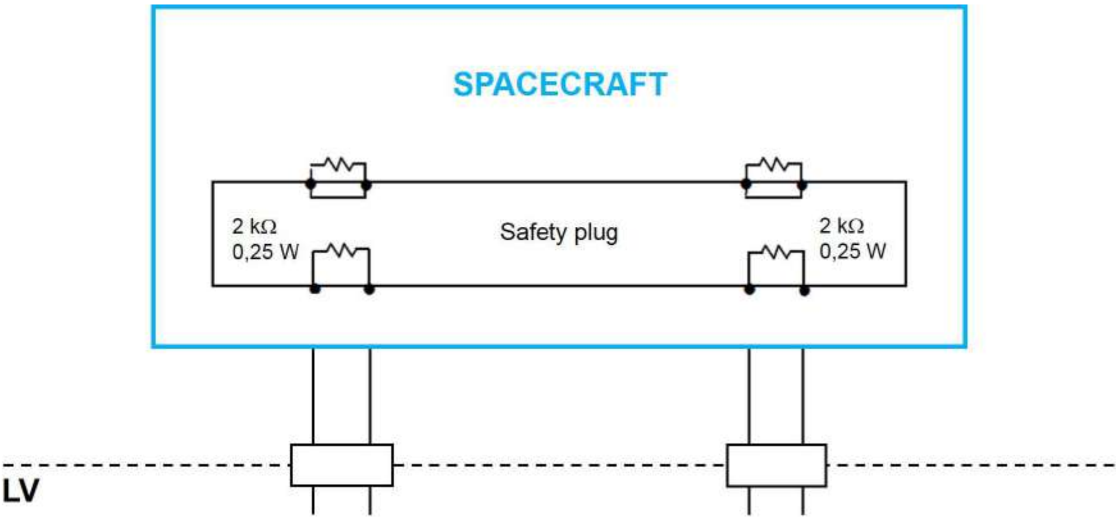


Figure 5.5.2.6.a – Typical principle of a pyrotechnic command diagram

5.5.2.7. Non explosive actuators (optional)

The Vega C launcher is able to initiate 10 non explosive actuators (NEA):

- Five with low voltage : 5 V ± 5% or 20 V ± 5%; Activation time ≤ 100 ms;
- Five with 28 V ± 4 V : Activation time ≤ 1 ms.

These ten NEA can be initiated simultaneously.

5.5.3. Electrical continuity interface

5.5.3.1. Bonding

The spacecraft is required to have an "Earth" reference point close to the separation plane, on which a continuity test can be done. The resistance between any metallic element of the spacecraft and a closest reference point on the structure shall be less than 10 mΩ for a current of 10 mA.

The spacecraft structure in contact with the LV (separation plane of the spacecraft rear frame or mating surface of a Customer's adapter) shall not have any treatment or protective process applied which creates a resistance greater than 10 mΩ for a current of 10 mA between spacecraft earth reference point and that of the LV adapter.

5.5.3.2. Shielding

In the on-board and ground harness, the shield is linked to the metallic structure (launcher and ground).

The ground shield is linked to the on-board shield at COE/POE interface, through the mechanical housing of the POE connector.

5.5.3.3. RF communication link between spacecraft and EGSE

A direct reception of RF emission from the spacecraft antenna can be provided as an optional service requiring additional radio-transparent window(s) on the fairing and additional hardware installation on the launch pad.

This option allows Customers to check the spacecraft RF transmission on the launch pad during countdown.

5.6. INTERFACE VERIFICATIONS

5.6.1. Prior to the launch campaign

Prior to the initiation of the launch campaign, a mechanical and electrical fit-check may be performed. Specific LV hardware for these tests is provided according to the clauses of the contract.

The objectives of this fit-check are to confirm that the satellite dimensional and mating parameters meet all relevant requirements as well as to verify operational accessibility to the interface and cable routing. It can be followed by a release or drop test.

This test is usually performed at the Customer's facilities, with a flight like adapter equipped with its separation system and electrical connectors provided by Arianespace. For a recurrent mission the mechanical fit-check can be performed at the beginning of the launch campaign, in the payload preparation facilities.

5.6.2. Pre-launch validation of the electrical interfaces

5.6.2.1. Definition

The electrical interface between satellite and launch vehicle is validated on each phase of the launch preparation where its configuration is changed or the harnesses are reconnected. These successive tests ensure the correct integration of the satellite with the launcher and help to pass the non reversible operations. There are two major configurations:

- Spacecraft mated to the adapter;
- Spacecraft with adapter mated to the launcher.

Depending on the test configuration, the flight hardware, the dedicated harness and/or the functional simulator will be used.

5.6.2.2. Spacecraft EGSE

The following Customer's EGSE will be used for the interface validation tests:

- OCOE, spacecraft test and monitoring equipment, permanently located in PPF Control rooms and linked with the spacecraft during preparation phases and launch even at other preparation facilities and launch pad;
- COTE, Specific front end Check-out Equipment, providing spacecraft monitoring and control, ground power supply and hazardous circuit's activation (SPM etc.).The COTE follows the spacecraft during preparation activity in PPF and HPF. During launch pad operation the COTE is installed in the launch pad room. The spacecraft COTE is linked to the OCOE by data lines to allow remote control.
- Set of ground cables for satellite electrical umbilical lines verification.

The installation interfaces as well as environmental characteristics for the COTE are described in Chapter 6.